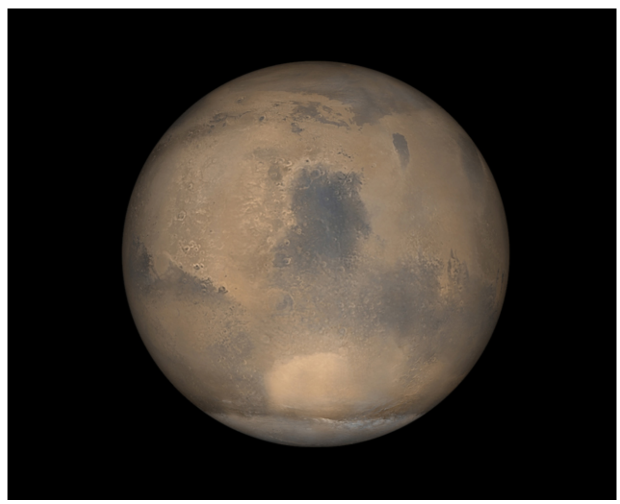
16.89J / ESD.352J Space Systems Engineering
Spring 2007

PARADIGM SHIFT IN DESIGN FOR NASA'S NEW EXPLORATION INITIATIVE



16.89 Graduate Design Class
Space Systems Engineering

Massachusetts Institute of Technology
May 12, 2004

16.89 Team Members

Students

Sophie Adenot
Julie Arnold
Ryan Boas
David Broniatowski
Sandro Catanzaro
Jessica Edmonds
Alexa Figgess
Rikin Gandhi
Chris Hynes
Dan Kwon
Andrew Long
Jose Lopez-Urdiales
Devon Manz
Bill Nadir
Geoffrey Reber
Matt Richards
Matt Silver
Ben Solish
Christine Taylor

Staff

Professor Jeff Hoffman
Professor Ed Crawley
Professor Oli de Weck

Table of Contents

List of Figures

List of Tables

Abstract

On January 14, 2004, President George W. Bush presented the nation with a bold new initiative to "explore space and extend a human presence across our solar system…using existing programs and personnel…one mission, one voyage, one landing at a time." (Bush, 2004) NASA was charged with the task of developing a sustainable and affordable human space exploration program with the initial objective of returning a human presence to the Moon by the year 2020. The directive thus raises two broad engineering questions: First, what is the purpose of an exploration system, and how one evaluates its performance. Second, how does one architect a sustainable space exploration system? The following report makes the case that the primary purpose of an exploration system is the delivery of knowledge to the stakeholders, and that the design should be evaluated with respect to knowledge.

1. Introduction

On January 14, 2004 President George W. Bush presented the nation with a new vision for space. The National Aeronautics and Space Administration (NASA) will develop a sustainable human space exploration program taking humans back to the Moon by 2020, and eventually to Mars and beyond (Bush, 2004). The vision, and plan that goes with it, calls for the completion of the ISS, the retirement of the Space Shuttle by 2010, and the development of a new Crew Exploration Vehicle (CEV). Bush's vision provides a bold push towards mankind's traversing of the solar system. The following report, representing the culmination of MIT's 2004 spring 16.89 graduate design class, presents a design methodology and conceptual tools to facilitate the achievement of this vision. It addresses two critical questions facing the space community: What is sustainability in the context of space systems? How can sustainability be provided for during conceptual design? The following report addresses these questions. In doing so, it demonstrates that an exploration program is by definition a *knowledge acquisition and transfer system*, and it presents a process by which one may design for sustainability.

The goal of exploration is knowledge

While the motivation behind exploration has varied throughout history, the primary function of any "exploration system" has been to discover the unknown, to gain knowledge. Some of the more common ways to gain knowledge have been through the use of visual, electrical, or physical transportation of information. A simple example of a space knowledge transfer system is the human eye. The human eye gathers knowledge in the form of light. Several hundred years ago mankind developed the telescope in a hope to improve upon the amount of knowledge delivered to the eye through the discovery of magnification. The magnification of objects resulted in a higher order of knowledge resolution and consequently more information about space was discovered.

More recently mankind has sent satellites and drones into the solar system, with sensors that can gather information unattainable by the human eye alone. Information gathered by these systems is sent back to Earth through the use of electrical transmissions where it is turn into knowledge. A number of characteristics increases the "knowledge resolution" of these satellites and drones compared to telescopes, including: Shorter distance between optics and target, physical contact, sample return, in-situ analysis, etc. It is noteworthy that order to achieve this higher of knowledge resolution, mankind had move beyond light as the sole transfer-mechanism, to in-situ measurement and mass transport. Future exploration systems must necessarily follow this trend, exploiting the duality between mass and knowledge transfer, with one critical improvement--humans will provide degree of knowledge resolution previously unimaginable with satellites, drones, and telescopes alone.

No matter the form of the space exploration system (human eye, telescope, robotic probe, or human contact), the end product of the exploration system is knowledge. Currently, the majority of the work being completed on NASA's new initiative is directed towards a new exploration vehicle. The class believes that any new space vehicle developed by NASA must be designed with an understanding that it will be but one tool in system whose ultimate function is to gather and transfer knowledge in space and on Earth.

To say that an exploration system must deliver knowledge to achieve its goal is to recognize that while mass transport enables exploration, the ultimate success of an expedition depends on the acquisition, communication, and synthesis of visual imagery, scientific data, and human experience to key stakeholders. This suggests revaluing traditional space system characteristics and trades to account for the demands of knowledge acquisition and delivery. Further, in order to make clear decisions about system capabilities and mission goals, attributes of knowledge must be categorized and valued in accordance with stakeholder needs. System designers must have a firm grasp of the knowledge delivery process, and establish how it will occur at each point in the system's lifecycle.

Sustainability in the Design Process

Before knowledge can be incorporated into system valuation and trades, however, there must be a clear understanding of what is a sustainable space system and how can this can be addressed during conceptual design? Current space system design methods are not geared towards enhancing "sustainability." Traditionally, they have focused on developing requirements, conducting trades based on assumptions about the future, and then optimizing the system with regard to some metric. Results are commonly single point designs optimized for single missions.

While such methods have proven adequate for low-frequency missions, they rely on assumptions about an uncertain future. A design that is optimal at one point in time may become less optimal in the future. Due to the expected duration of the new exploration initiative, major investments should note be made based on unverified assumptions. The new exploration system should be designed so that it can respond to changes in the future. The approach to design described in this report addresses this problem. Using an iterative process, and emerging system valuation tools, it creates a rigorous development strategy which is flexible and robust to environmental changes.

Chapter two proposes a definition of sustainability. Drawing from recent scholarship and historical examples, it argues that sustainable exploration programs must first and foremost have the capability to manage various kinds of uncertainty, including policy, budgetary, technical, and logistical changes. Conceptual designs must provide system operators with the ability to anticipate and capitalize on emerging opportunities and positive feedback loops while simultaneously adapting to changing value-structures and external circumstances.

Properties that enable sustainability have been termed flexibility, extensibility, robustness, and commonality. Much recent scholarship has addressed the need to rigorously value these system properties for the purposes of design. Generally, these properties translate to formal architectural attributes, such as modularity and platforming, as well as operational attributes such as staged deployment and spiral development. Chapter three defines these terms in the context of space systems, and presents methods for their formalization in system architecture.

There are two ways in which flexibility and extensibility are introduced and evaluated: mathematical evaluation methods and architecture design considerations. The mathematical evaluation methods used are based upon decision analysis, real options theory, and scenario planning. The architectural design considerations are commonality, scalable systems, and modularity. Both methods evaluate a given system based on the resulting value of knowledge delivered by the system. Notice that the system is not evaluated on cost or mass, but on knowledge, which is the primary purpose of an exploration system.

A major aspect of this study involves identifying a process to combine these properties and methods can be systematically incorporated into system design. Part of the solution involves creating a strategy, rather than a point design, that can react to change. Chapter 4 presents an example strategy, or "baseline," which was conceived through an iterative process of design, needs mapping, and synthesis of sub-strategies. Sub-strategies consist of small, medium, and large Moon and Mars expeditions, each designed with principles of extensibility such as commonality and staged deployment. Individually, these missions are rough "point-designs." However, major architectural decisions in each reflect anticipation of gradually increasing mission scale, and eventual transit to Mars.

After completing the sub-strategies, areas of functional commonality and uniqueness can be anticipated across the system, and architectural forms refined appropriately. The resulting forms and operations can then be synthesized into an integrated life-cycle strategy, with options for reacting to uncertainty. The following schematic illustrates the design process used:

In developing the integrated baseline, commonality trades at the formal and operational level become necessary. Chapter 5 details such trade studies and their results.

Once the final version of the baseline strategy and associated trades has been developed, more rigorous tools may be applied to determine when, and under which circumstances different design options become valuable. For example, the decision to transit through the Earth-Moon Lagrangian Libration Point 1 (EM-L1) while en route to the Moon may not be optimal for a single mission to the lunar equatorial region. However, if the frequency of non-equatorial lunar missions is sufficiently high, the option of utilizing EM-L1 becomes increasingly valuable. Tools, including modified forms of real options valuation, can inform trade studies of this sort, resulting in up-front design decisions that drastically reduce life-cycle cost and increase system flexibility.

Chapter 6 introduces such tools and methods. Scenario planning is applied to the integrated strategy to examine how the system can react to environmental changes. Adjustments are then suggested, based upon the baseline's reaction to the scenarios. Decision analysis and Real Options analysis techniques are also used to determine at what point time-critical decisions should be made in the execution of the baseline strategy, and which investments should be made now to allow for the option of adapting to future uncertainty.

2. Intro to Sustainability

What exactly is a sustainable exploration program? In one sense, the answer is rather straightforward. To "sustain" means literally: to maintain in existence, to provide for, to support from below (Dictionary.com website). At the programmatic level, an exploration system will be maintained in existence so long as it is funded, and it will be funded provided it meets the needs of key stakeholders, members of Congress, the Administration, and ultimately the American people. Realistically, however, system designers must recognize that these needs themselves will change. A multi-year, multi-billion dollar program in the US Government must expect to face a great deal of uncertainty with respect to objectives, budget allocations, and technical performance.

In order for an exploration system to be sustainable, then, it must be able to operate in an environment of considerable uncertainty throughout its life-cycle. Designing for sustainability implies identifying sources of uncertainty and managing them through up-front system attributes. Various terms have been used to describe such system attributes, including: flexibility, robustness, and extensibility.

While a large complex system must react to changing environments in order to be sustainable, technological aspects of systems can themselves impact the environment. Once in development and operation, a multi-billion-dollar system will mediate political interests, organizational decisions, and technical alternatives, creating potential sources of stability and positive feedback-loops, as well as sources of uncertainty. Early decisions that create high switching costs or large infrastructure sites, can "lock-in" architectural configurations and influence the objectives and development path of later systems (Klein, 2000). A sustainable design will be one in which, to the greatest extent possible, the dynamics behind political, technical, and financial sources of stability support, rather than hinder, system development and operations.

The following chapter identifies three kinds of sustainability, and relates these to formal system attributes. It reviews current thinking about flexibility and extensibility, and their relation to architectural form. The chapter concludes with a historical investigation of Antarctic exploration, drawing lessons for the sustainability of exploration programs.

2.1 Elements of Sustainability

It is increasingly evident that large, complex, technological systems cannot be conceived independently from the political, economic, and organizational environment in which they operate. While at a technical level, exploration is dependant on continuous and reliable logistical support, at a programmatic level, political and organizational factors greatly affect sustainability. With space activities in particular, motivations and objectives can change rapidly compared to system life-cycles, increasing the impact of political and organizational issues on system development and use. A sustainable space exploration system will successfully mediate and react to political, organizational, and technical uncertainty, and also exploit, to the extent possible, sources of "stability" that arise from the interaction of these factors.

2.1.1 Policy Sustainability

Policy uncertainty can take the form of changes in objectives or the regulatory environment in which a system must operate. It stems from the dynamic nature of the US government, and the need for space systems to suit both national/strategic and political/tactical interests. Government programs are re-assessed on a yearly basis in terms of national priorities and, in some cases, performance. Changes in the political and geopolitical environment can alter the perception of the value of exploration activities. An important aspect of policy sustainability is thus the ability to maintain relevance, and continue operations, in the face of shifting objectives and regulatory environments.

To take one example, while the decision to build the Space Station Freedom was motivated largely by Cold War concerns, the fall of the Berlin Wall transformed the ailing project into a symbol for international peace and cooperation (Wikipedia, 2004). To the extent possible, system designers should consider the implications of such changes for system operation. If a policy decision to focus on Mars rather than the Moon is likely in the near term, current designs should be extensible to both objectives. Similarly, if international cooperation is based on uncertain agreements, alternatives to international participation on the critical path of development should be available.

2.1.2 Budgetary Sustainability

Shifting political priorities also create changes in funding. During its years of development and operations, a programs budget may oscillate unpredictably. Figure 1 illustrates how NASA's budget fluctuates over time.

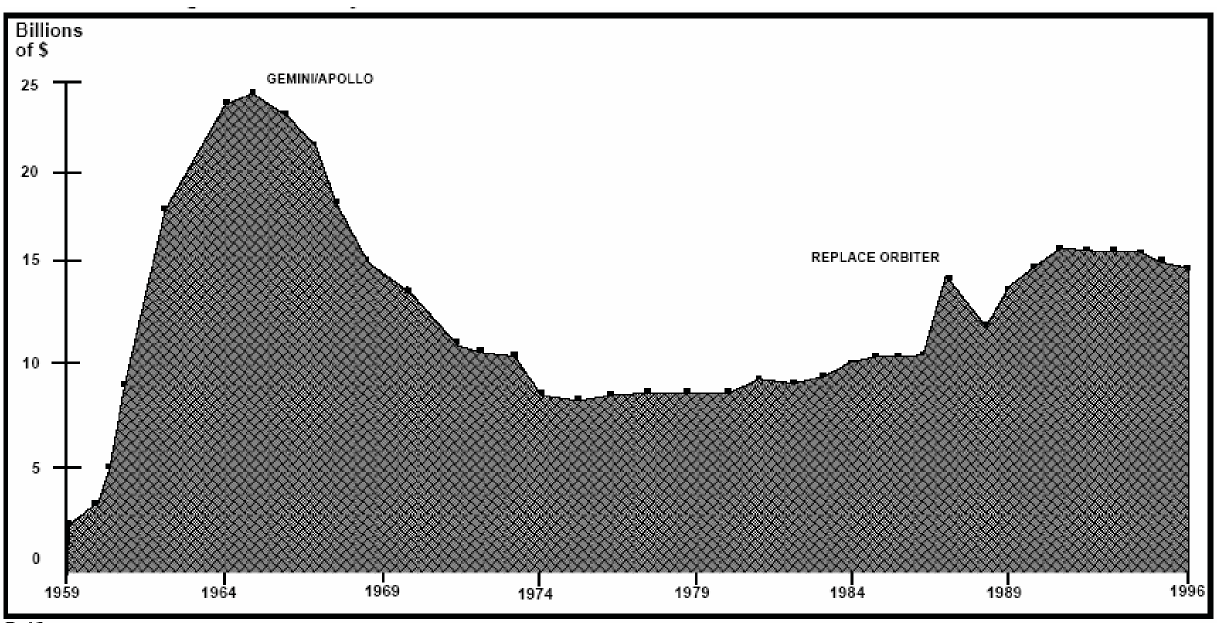


Figure 1: NASA budgetary fluctuations in 1996 dollars (courtesy http://history.nasa.gov)

A flexible system will maintain exploration capability even in the face of budgetary fluctuations, whether through changes in schedule, scale of operations, or by other means.

2.1.3 Organizational Sustainability

Recent scholarship has investigated the relationship between organizational structure and technical design. Charles Perrow (1984) has characterized socio-technical systems in terms of their dynamics and complexity, drawing conclusions for system safety and reliability. He defines space systems as highly coupled, nonlinear, and complex. Organizational structure and technical complexity can impact system reliability by creating “quite erroneous worlds in [the] minds” of system operators and managers. (Perrow, 1984)

Diane (1996) Vaughan furthers this understanding, suggesting that “the microscopic world of daily decisions” can create almost imperceptible changes in organizational culture over time, with important consequences for safety. Her term, the “normalization of deviance,” encompasses the way in which expectations can change and aberrations become accepted, through continual exposure to anomalies. Organizational structure, which impacts daily decisions, plays an important role in system performance and reliability, and thus sustainability.

A space system will be sustainable from an organizational perspective, then, if the technological system and management structure are designed together to minimize organizational drift and normalization of deviance.

2.1.4 Technical Sustainability

Technical sustainability refers to system performance, reliability, and the potential infusion of new technologies. An exploration system must support and maintain human and robotic activity at various fronts of exploration, and incorporate technological advances to continuously improve system performance without major operational changes. Further, any highly complex system is likely to fail at some point during its life cycle. A sustainable system will be one that is robust to failures, both small and large.

An important factor related to technical sustainability is risk tolerance. Risk tolerance can be divided into three main areas:

1. Development risk: during design, test integration of architecture components
2. Planning risk: willingness to exploit more or less of known system margins while planning an exploration mission
3. Operations risk: willingness to take risk during operations.

By definition, risk-free exploration does not exist. System designers must balance the risk associated with architectural form, schedule, and operations, in order to achieve system objectives. Risk tolerance can change throughout a system life cycle, and thus change how a given system is operated.

2.2 Sustainable Exploration Systems – Dynamics

While each of the three domains above impacts the development and operations of complex systems in different ways, they are closely interrelated. The dynamic relationship between the three has important ramifications for sustainability. The relationship between these three broad domains is shown in Figure 2.

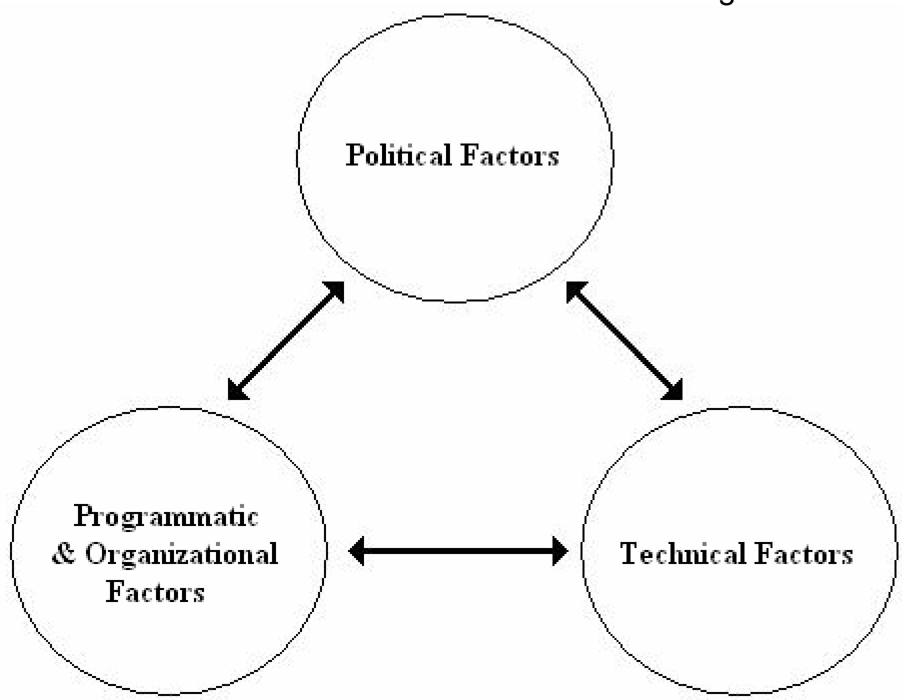


Figure 2: Interaction of political, organizational, and technical factors

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The Columbia Shuttle Accident Report (CAIB) repeatedly stresses the adverse affects that broader issues such as indecisive national leadership, increasingly stretched budgets, and continued mischaracterization of Shuttle capabilities have had on NASA's organizational and safety culture.

Conversely, Hans Klein has suggested that the characteristics of a technological system and development program can facilitate or impede coalition politics, thereby reducing or exacerbating conflicts between politics and program administration (Klein, 2000). Technology and politics are linked when program administrators translate political forces into design requirements. Further, once developed, a given system architecture together with its supporting facilities can become "locked-in" and perpetuated through later designs. The space shuttle, for example, made use of facilities designed partly as the result of short-term political wrangling conducted during the Apollo era (p. 319).

Annalisa Weigel and Daniel Hastings have similarly investigated the interrelation between technical design and political change (2003). Weigel and Hastings stress that space transportation infrastructures are affected as much by political considerations as technical problems. It is thus imperative to understand the coupling of both domains if a system is to operate successfully in the "politico-technical" arena. Weigel presents a framework to understand how policy directives couple with technical parameters. Figure 3 is an "influence diagram" used to illustrate such coupling.

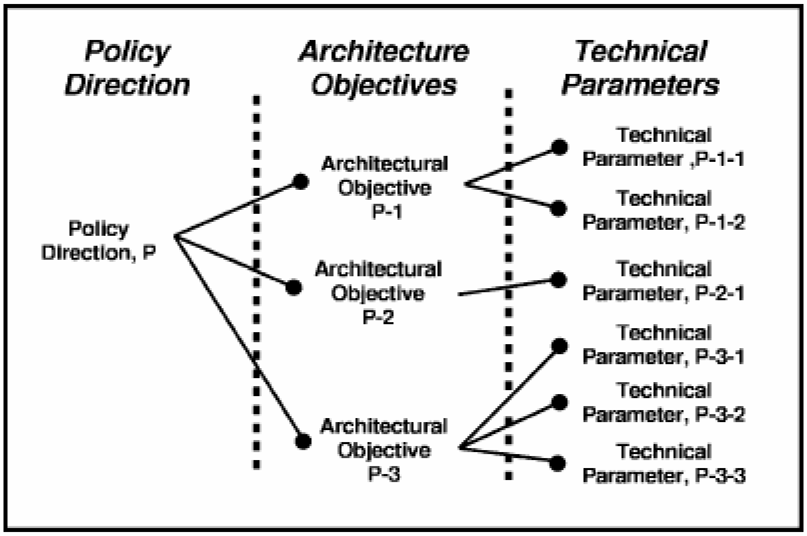


Figure 3: Translating policy parameter affects into the technical domain: an influence diagram (courtesy, Weigel and Hastings, 2003)

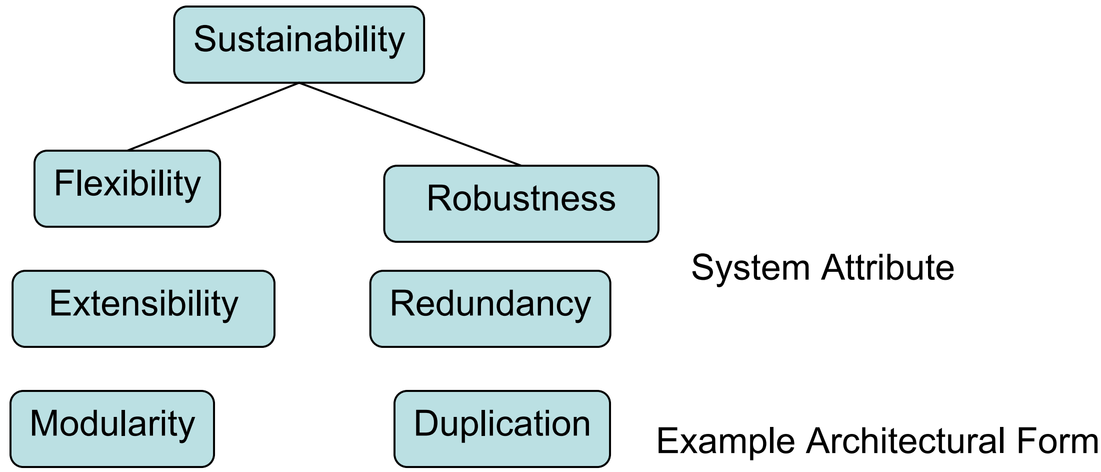
At a different level, as a later section of this chapter notes, the interplay between news, politics, and technical development was an important factor in the evolution of Antarctic exploration. In this respect, designing for sustainability implies understanding how various design decisions can lead to organizational and political dynamics that may improve or impeded the flexibility of the system.

2.3 Sustainability, Flexibility, Robustness

A sustainable system will have attributes that allow it to cope with, or mediate, various forms of uncertainty throughout its life-cycle. Many terms have been used to define characteristics which give systems these properties. They include flexibility, robustness, and extensibility. But what are the relationships between these terms?

In many ways this is simply a question of definition. Flexibility can be defined as the ability of a system to change or be used differently than intended after it is initially fielded. Flexibility can be intentional, but is often unintentional such as in the case of the B-52 or the use of the LM as a "life boat". The speed with which a system reacts to change is a measure of *agility*. Extensibility is a particular kind of flexibility. Conversely, *robustness* is the property of a system that allows it to be insensitive to change. A system is robust if it continues to deliver value in changing circumstances.

All of these “ilities” are enabled by attributes of architectural form. The follow schematic illustrates how the various concepts relate to each other:



2.4 Extensibility – An Enabler of Sustainability

Extensibility is defined as “the property that new elements can be added to a system in such a way as to alter the value delivered.” (Crawley, 2003) Designing systems to be extensible drives life cycle cost down through anticipating future goal and environmental changes and then translating this understanding into upfront system design actions aimed at minimizing overall life-cycle cost. Extensibility addresses both known and unknown future changes, with expected payback being variable, based on the certainty and magnitude of the anticipated change, along with the cost associated with making the system extensible.

Designing systems for extensibility requires a fundamental shift in the way design decisions are made, a shift from near optimal fulfillment of immediate requirements at minimal cost, to minimizing life cycle cost, maximizing life-cycle performance, etc. In other words, an extensible design will not be the highest performing design when compared to a point design optimized for a given set of capabilities- a penalty is placed on ultimate system performance in order to increase life-cycle value. An extensible design will not be the lowest cost design under the same conditions either. The advantages of an extensible design are only realized in the context of multiple generations of the system. New metrics must be implemented for valuing the benefits of extensibility. In addition, a culture shift must occur from near term to longer-term expectations of success.

The large investment associated with complex systems dictates the need for an evolutionary growth path, although not all elements of the system undergo the same degree of change. Therefore, it is important to invest “extensibility dollars” only where needed. Investing in extensibility provides an option for future change. As an example, an in-space crewed exploration vehicle could be designed for extensibility in terms of number of crew supported and days of support through decoupling of living quarters with the command and control portion of the spacecraft. While the initial need may be support of a four-person crew for two weeks, this need may extend to support of six people for nine months. Clearly, using the same vehicle for both missions would unduly penalize the shorter mission while design of two separate vehicles would result in high costs associated with development of redundant functions such as the command and control functions. Separating the habitat functions from command functions through creation of two modules and a common interface, for instance, would enable the habitation portion of the spacecraft to be easily modified. If the change is executed, the implementation of the change is expected to cost less than if the option had not been put into place. If not executed, the extensibility feature represents wasted resources in terms of the expense to implement, reproduce and support the unused feature.

Several concepts overlap almost directly with extensibility- staged deployment, and spiral/incremental development. Staged deployment seeks to match demand and supply through scaled rollout of a system. Expenses are delayed until a later date, reducing the net present value of the expense and increasing the certainty of the need, at the time of the expense. De Weck et al. (2004) describe staged deployment as a potential alternative to full deployment of the Iridium communications satellite network, with the potential benefit being lower investment in order to start operations. Additional satellites could have been deployed as demand increased. While Iridium was ultimately displaced from most of the expected market due to widespread cellular coverage, the lost investment could have been significantly reduced.

Like staged deployment, spiral development (Figure 4) is also an incremental method of deploying new systems and their capabilities in a flexible manner. Initial capabilities are selected based on prioritized goals, enabling quick deployment of high priority capabilities. Additional iterations of the process focus on deploying lower priority capabilities and addressing newly discovered needs. The result is quick deployment of primary capabilities combined with risk reduction through decision delay that enables incorporation of current technology into new stages and shifts in strategy as needs become clearer (time advances).

Image removed due to copyright restrictions.
Boehm, Barry W. "A Spiral Model of Software Development and Enhancement." *Computer* 21, no. 5. (May, 1988): pp. 61-72.

Figure 4: Boehm's model of spiral development (picture from Boehm, 1988)

2.4.1 Reasons for Extensibility

Extensibility reduces overall life-cycle cost and/or increases life-cycle performance through a number of difference paths. Several are listed below, along with brief descriptions.

2.4.1.1 Management of Technology Obsolescence

As the lifetime of a system grows, the rate of change of technology is increasingly mismatched with the scale of system replacements. Within a system, different modules have different rates of technological change. Charles Fine (1998) uses the term “clockspeed” to describe the rate of change and to highlight the differences between rates of change. Extensible systems allow for management of technological change within the system. As an example, consider a vehicle, such as a spacecraft. While structural technology may undergo significant improvements on the timescale of a decade or more; control system components, especially the electronic elements such as logic chips, undergo significant change on an annual basis. Designing a system to accommodate varying clockspeeds enables the design to evolve over time. One method for accommodating technological change is through grouping components with similar rates of change into modules, therefore, enabling easy replacement of the module, with minimal impact to other areas. Ease of change leads to the ability to keep a system modernized.

2.4.1.2 Risk Mitigation

Delaying decisions improves the likelihood of making a correct decision. While delay can cripple a program if not handled properly, the result of effective use of delay is confidence in decision-making.

2.4.1.3 Policy Fluctuation Robustness

Extensibility is beneficial in the face of the uncertainties produced by the policy domain, and the resulting budget fluctuations. The potential for a change in President occurs once every four years, a timescale much shorter than that of an exploration program. Given the mismatch in timescales, it is critical that achievement of intermediate milestones provides lasting value, a foundation for future work.

2.4.2 Describing Extensibility

Methods are needed for describing what an extensible system is and how the extensibility is achieved. Ultimately, the metrics and descriptions must be quantifiable to enable trades to be made between designs and design options. Which system is more extensible? How extensible is the system?

One view of the evolution of a system over time is a consideration of the relationship between available capabilities and required capabilities; in other words, a type of supply and demand curve. Figure 5 provides a notional view of this concept. The system needs over time are represented as a continuous curve. While the system needs curve may in fact be discrete, the aim here is to highlight the high degree of changing need in relation to the ability of the system to change. The design points represent the available capability levels. From a system performance standpoint, the ideal available capability would be a direct overlay over the needs curve. While the ideal curve cannot be reached due to practical considerations such as the cost of each change (engineering, deployment, etc.), the ideal curve can be approached through the creation of an extensible architecture. This view is closely related to previous work in the area of staged deployment. (de Weck *et al.*, 2004).

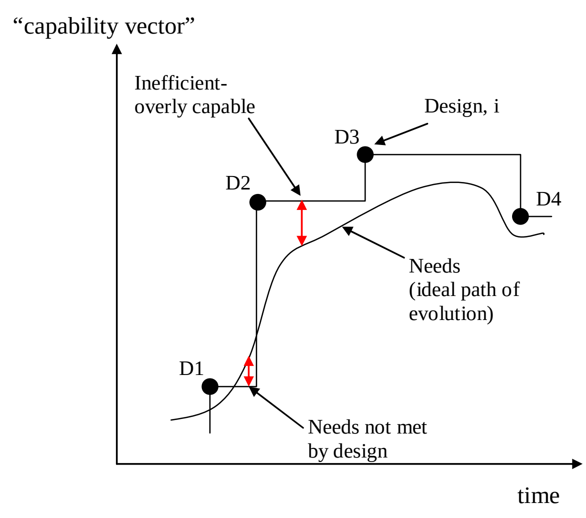


Figure 5: Change in system need and capability over time

The relationship between the supply and demand “curves” is an important one. As Figure 5 illustrates, a system that is overly capable is inefficient. More dollars and time have been spent on unneeded functionality, at the given point in time. The reverse situation means that the system is not meeting needs, also a problem. As an example, consider the transition from Design 2 (D2) to Design 3 (D3). Before the transition, needs aren't met by capabilities, while after the transition, the system is over-designed, as would be expected immediately after an improvement. Also note the transition from D3 to D4. While this transition was not required to meet new capabilities, since needs have actually decreased, the change was made in order to maintain design efficiency.

In order to analyze the evolution of a system over time, a well-defined method of describing change is needed. This void can be filled by a series of operators, such as those defined by Baldwin and Clark (2000):

- Splitting (into two or more modules)
- Substituting- replace module with a different one
- Augmenting (adding a module)
- Excluding- removal of a module from the system
- Inverting- creation of new design rules
- Porting- use module in another system

The above operators can be used to perform all module-level operations. As was mentioned in the previous section, it is critical to realize that evolution is synonymous with adaptation or change, not addition. Continuous adaptation to changing conditions may mean eliminating functionality that is no longer needed; an operation accomplished with the exclusion operator. As a simple example of the use of operators, consider the creation of a launch vehicle. The augmentation operator is used to add strap-on boosters for heavy lift capability, while the substitution operator could be used to express the change of a launch fairing.

2.4.3 Principles Supporting Extensibility

Four key principles support extensibility- modularity, ideality/simplicity, independence, and integrability. These principles were originally linked to "flexibility" by Schulz and Fricke (1999) and are briefly summarized below.

2.4.3.1 Modularity

The first principle supporting extensibility is modularity, defined by Baldwin and Clark (2000) as:

"A module is a unit whose structural elements are powerfully connected among themselves and relatively weakly connected to elements in other units. Clearly there are degrees of connection, thus there are gradations of modularity." (p. 63)

The principle of modularity enables complex problems to be broken down through a hierarchical structure. Changes internal to a module are isolated at the module boundaries, limiting the cascading impacts of a required change. Expense is reduced in development, test, hardware exchange, etc. Changes made to a modular system can be described in terms of the modular operators described in the previous section.

2.4.3.2 Ideality/Simplicity

Ideality is defined by Schulz and Fricke (1999) as the ratio between useful and harmful/undesired effects, a notion of design efficiency (pp. 1.A.2-4, as an additional reference, see Suh, 2001.) This principle highlights the importance of the ongoing culling of unneeded functionality as a system evolves over time. Failure to do so increases system complexity unnecessarily, eventually making total replacement of the system a more effective option than change.

2.4.3.3 Independence

The independence axiom derives from the independence axiom in axiomatic design (Suh, 2001). Each function is satisfied by a different design parameter. Creating a decoupled design, in terms of functionality, produces a design that is more easily managed over time.

2.4.3.4 Integrability

Integrability relates to the degree to which a system's interfaces are open, or flexible. Compatibility between elements is a critical enabler of flexibility. As an example, consider a docking interface on the space station. This interface would ideally be common across all future spacecraft, ensuring full compatibility. As an additional example outside the aerospace industry, consider the USB interface standard now used by many electronic peripheral devices such as keyboards, computer mice, flash memory cards, etc. The use of dedicated interfaces for each one of these devices would be highly inefficient, especially given the fact that only a small subset of the devices is needed at any one time.

2.4.4 Extensibility Summary

The concept of extensibility is critical to the creation of a sustainable exploration system. Extensibility must be an integral part of the exploration strategy to ensure that forward progress serves as a continually growing exploration foundation, even in light of policy direction changes. The concepts of extensibility are woven into the baseline missions and example conceptual designs within this report.

2.5 Historical Comparison: Antarctic Exploration

The history of Antarctic exploration provides valuable lessons for space system designers. From its inception Antarctic exploration and science shared many attributes and constraints with current space activities. Both, for example, have been highly dependant upon technological advances, including the need for complex logistics and cutting-edge life-support capabilities. Months of isolation during Antarctic expeditions present psychological hardships similar to those anticipated in extended Moon and Mars missions. More generally, Antarctic exploration, like space activities, has brought science into close involvement with politics. The following section examines how these factors affected some aspects of the development of Antarctic exploration and science, and draws lessons for space exploration programs.

2.5.1 Technology and Logistics:

> "More than any other, Antarctic science is dependant on logistics, on the ability to place and maintain a scientist and his equipment in the right place at the right time. Expeditions to Antarctica up to 1925 depended on techniques of transport, communication, survival, which remained largely unchanged for 100 years…. after 1925 the development of mechanized transport, the airplane, radio and technology based on better understanding of human physiology, were to make access to the Antarctic, travel within it and survival in its hostile environment, much less difficult." (Beck 1986, p.131).

The above quote summarizes well the disjointed nature of Antarctic exploration. Rather than follow a steady, continuous path of progress, the pace of discovery on the continent advanced through steps and jumps. Importantly, these advances in capability often resulted from the congruence multiple technologies, rather than any single technical development. Each jump offered great advances in knowledge returned per expedition, a situation that should be anticipated and exploited in the design of space exploration systems.

Most significant of these advances involves a shift from what has been termed the "Heroic" age to the Modern age of Antarctic exploration. The Heroic age is roughly delineated as the period from 1895 to the dawn of the First World War in 1915 (Walton, 1987). It marked a dramatic shift in capability from the previous era because of the use of liquid fuel, however, due to the still rather primitive methods of transport and "life support," expeditions during this period often brought extreme hardships. National prestige, sovereignty, and personal fame—not science—motivated exploration during this period.

The Modern age begins roughly with the American expedition lead by Richard Evelyn Byrd from 1928-1930. It is characterized by the comprehensive use of airplane travel, electric communication, mechanized transport, and thus continuous logistical support (Fogg, 1992). Most of these technologies had existed for some time, and had been tested and refined through previous expeditions. Byrd's expedition was the first to coordinate them systematically, increasing the amount of data collected by orders of magnitude. The following table summarizes the major technical advances that enabled this shift, as well as the impact on exploration capability and knowledge return. Systematic use is defined as use in everyday operations, as opposed to sporadic use and testing.

Technology	Introduction for Exploration	Systematic Use	Mission/Logistics Impact	Initial Knowledge Return Impact	Space-based equivalent
Radio Communication	1911	1929 (Byrd)	Coordination, safety	Immediate news of success increased public interest	Satellite Communications
Combustion Engine (land travel)	1907 (Shackleton)	1933 (Byrd)	Outdoor activity and travel in harsher conditions	Distribution of heavy seismic equipment	Rover
Airplane	1929	1928 (Byrd)	Pre-positioning for extended expeditions; Aerial photography	1 field season of land-based observation per hour (4000 square miles)	UAV's, Pre-positioning technology
Ice Breakers	---	post-WWII	increased access, extended access	More feasible permanent base	cyclers

Implications can be drawn from these examples for space exploration. Advances fall into rough classes of technologies with analogues in space systems. Combustion engines, which enabled the equivalent of surface rovers, had a great impact on the kinds of fieldwork that could be executed. Their introduction created the possibility of distributed use of heavy equipment for seismic operations. Their impact on mission logistics, however, was minimal at first.

The airplane and the radio had dramatic affects on knowledge return and mission logistics. The Byrd expedition was the first to fly over the pole. In doing so, he took over 1600 pictures covering 150,000 square miles, or the equivalent of 37.5 field seasons worth of observations using previous methods (Walton, 1984). He also discovered two Mountain ranges. The airplane also allowed for the possibility of pre-positioning and logistical support for inland bases.

Soon after flying over the pole, Byrd was able to communicate the accomplishment. His successful flight was beamed via radio immediately back to the United States, and this greatly increased US interest in Antarctica (Fogg, 1992).

An interesting feature of the progression of technological development is the lag between testing and systematic use. Radio communication and the combustion engine were tested with little impact in many expeditions before the Byrd expedition.

Interestingly, life support capabilities advance much more gradually than logistics technology. Man learned to live the extreme environment gradually, over several hundred years, with advances coming more through trial and error than scientific or technological breakthrough. (Fogg, 1992)

In many ways, NASA's current task is to transition space activities from a heroic to a modern age. While national prestige and public attention will continue to play important roles in space activities, the time has come for more systematic and knowledge return. The history of Antarctic exploration demonstrates that when this occurs, as in the case with the first Byrd expedition, public attention and government funding are likely to increase rather that decrease. The next section examines this dynamic of science and politics.

2.5.2 Politics and Technology

Antarctic exploration requires support at the national level. Thus, as one author notes, "Antarctic scientists have often been used as political instruments and it would be unrealistic for them to think that their work can be isolated from the spheres of interest of economics, law, and politics."(Klein 2000, p.319) The motivations behind various stages of Antarctic exploration are extraordinary in their similarity to space activities. They include included: prestige of geographical discovery, information and experience for navigation and commerce, and sovereignty. While science always played an important role during expeditions, and is now the single most important product of exploration, it is important to note that the underlying motivation for countries to invest in Antarctic travel has almost always been the "maximization of influence" rather than knowledge (Lee, personal communication).

Territorial issues became increasingly important at the transition from the Heroic to Modern age of Antarctic exploration. From 1908 until the signing of the Antarctic treaty in 1961, international tension rose and fell as countries made varied and conflicting claims to sovereignty. The following events in particular were important to this dynamic.

1908 and again in 1907 Britain issued formal territorial claim
1923 British claim the Roth Dependency
1924 French claim Adelie land
1933 Australia makes claim
~1939 Norway claims Dronning Maud Land

While the motivations behind these claims were complex and interrelated, the World Wars and advances in technological capability were certainly central factors. As with space activities during the Apollo Era, international interest, enabled by technological advances, fueled funding for exploration.

Byrd's expeditions are a particularly interesting example of this kind of feedback loop in the US. As mentioned above, Byrd was the first to systematically incorporate modern logistical technology in his 1928 expedition. This mission and second following it were funded privately. Their success captured the public attention, increasing US popular interest in Antarctic exploration. (Fogg, 1984). At the same time increasing territorial claims and impending war on the European sharpened political and military perception of the strategic value of access Antarctica. The result was that Byrd's third expedition, in 1939, was funded publicly and had the attention of President Roosevelt himself. In a letter to Byrd in 1939, Roosevelt explicitly stated the confluence of interests that lead to public funding:

> "The most important thing is to prove (a) that human beings can permanently occupy a portion of Continent winter and summer; (b) that it is well worth a small annual appropriation to maintain such permanent bases because of their growing value for four purposes—national defense of the Western Hemisphere, radio, meteorology, and minerals. Each of these is of approximately equal importance as far as we know." (Fogg, 1984, p.162)

Following the Second World War, international interest in Antarctica increased together with improved access. Antarctica exploration was facilitated by the use of ships designed specifically for working in ice, including modern ice-breakers. (Walton, 1987) In the tense environment of the Cold War, the ability to access Antarctica, much as with space, was itself justification for doing so. As is often the case, science was the veil behind which these interests developed. One state department official, Henry Dater, makes clear how these issues were interrelated in a letter he wrote in 1959:

> "Because of its position of leadership in the Free World, it is evident that the United States could not now withdraw from the Antarctic…national prestige has been committed…. our capacity for sustaining and leading an international endeavor there that will benefit all mankind is being watched not only by those nations with us in the Antarctic but also by noncommitted nations everywhere. Antarctic simply cannot be separated from the global matrix. Science is the shield behind which these activities are carried out." (Beck, 1986 p. 64)

While this view is a product of the geopolitical context, it illustrates how various factors can coalesce to form a sustainable program from a political perspective. The Byrd expeditions from before WWII had demonstrated American technical superiority in exploration and proven that modern technologies could be used to improve access to the continent. After the war, politicians and diplomats began to view exploration as an important symbol for global cooperation and competition, and were committed to continuing operations. Once implicated, national prestige and technical capability became intermingled, heightening the perception of value of continuing exploration.

Conclusions – Exploration and Sustainability

An important lesson that the history of Antarctic exploration provides for space exploration system designers involves the interplay between news, knowledge, technology, and funding. While Arctic exploration progressed slowly for decades, it was marked by distinct stages of increasing capability and increased interest. As the Byrd expedition illustrates, quite often advances in logistical and knowledge acquisition and transfer capability translate to increased political interest and funding. The spread of news creates public interest, while increased knowledge and logistical capability creates military interest. Both can generate funding for further expeditions, thus creating a positive feedback loop of discovery and technological development. Figure 6 illustrates the salient aspects of the feedback loop, which enabled the Byrd expeditions.

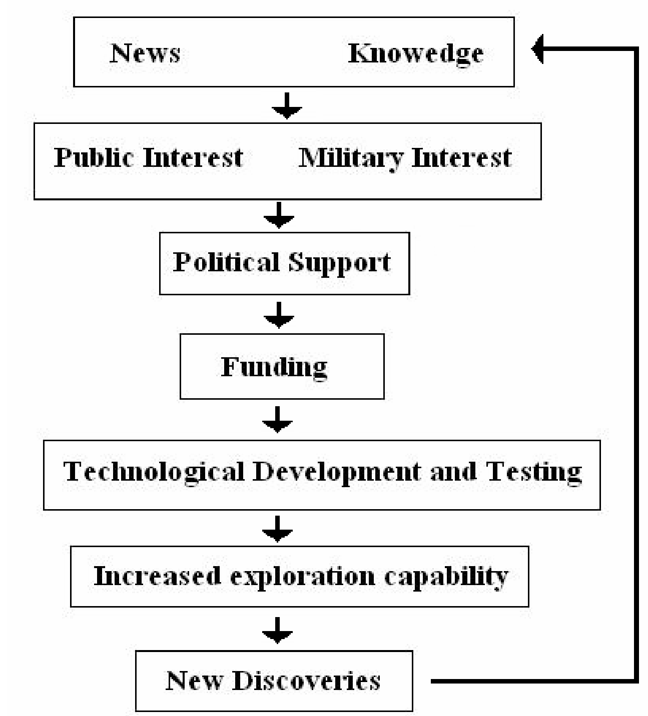


Figure 6: Positive feedback loop for exploration

Of course the real dynamics behind such a process are complex and varied. Byrd's expedition occurred at a time when international interest in Antarctica was increasing for many reasons. Still, these reasons are at least enabled, if not intimately connected with increasing logistical capability and knowledge creation. Such dynamics are worth investigating for the sake of creating successful exploration systems in the future.

2.6 Designing for Sustainability: A Process

MIT's 2004 spring class in Space Systems Design investigated the design of extensible space system architectures. A central difficulty in this task was the shear complexity of the problem, and the lack of an established methodology to design system architectures. An important result of the investigations was thus the methods developed to approach the problem, and the process by which "sustainability" could become central to design decisions. The end result was an iterative and holistic approach to the problem, which will hopefully inform future space systems architecture projects.

It should be stressed that not every aspect of the process described was completed rigorously during the semester. Rather, the process represents a way to integrate the lessons learned and eventually create a systematic architectural design. Of course every element of this process did not proceed in clear and neat steps. Most of the steps were iterative within themselves, and individual elements were re-worked as

The underlying goal of the design process was to develop an integrated strategy that could quantify how the system reacted to changes in the environment. Rather than create a point design to accomplish a Moon or Mars expedition, the class wanted to demonstrate that various scenarios could be anticipated and addressed during conceptual design and, as importantly, that the elements designed to address these scenarios (which would likely make the system sub-optimal from a point-design perspective) could be justified quantitatively. A strategy includes various scales of Moon and Mars missions, robotic scout missions, and considers the program changes such as budget cuts and regulatory constraints.

Figure 7 illustrates the five step process arrived at to create the strategy. An important goal was the establishment of common operations and across manned Moon, Mars and potentially asteroid missions, as well as through stages of missions at each body. Common elements defined baseline architecture forms and operations, from which options could be created to address specific missions and changing scenarios.

The first three steps in the process identify common forms and functions needed to explore the Moon, Mars and other destinations. Two teams conceived of staged Moon and Mars missions, and created matrices with functional requirements for each stage. With these functional requirements, a simple Venn diagram captures the relationship of requirements between the Moon and Mars. An interesting feature of this part of the process involves the ability to identify how formal elements can be extracted from functional requirements based on commonality between Moon and Mars needs at various levels. “Options” can be created to supplement the core needs, based on requirements outside of the intersection of the circles.

Functional Commonality Mapping thus revises the forms created to enable various Missions. The two teams must then return to the mission storylines and establish how and whether mission objectives can still be met with the revised forms, and alter staged missions accordingly. This iterative process can continue until a satisfactory level of refinement is achieved.

It was found that this iterative part of the process reveals key trades that need to be made with respect to commonality and architecture operations. Based on our designs, trades on issues such lander design, rover design, aerobraking capability, and operational capability processes such as the use of the Earth-Moon Lagrangian points, could not be solved by commonality mapping alone. The next step of the process is thus to evaluate the key trades revealed by the first three steps of the process.

In order to create a flexible strategy, however, it was important to evaluate these trades with consideration for the value of flexibility and robustness, not just optimality. Tool such as real-options, multi-attribute utility theory, and decision analysis, can be used to carry out the trades while preserving system flexibility, thus creating a rigorous development strategy and architecture.

Chapter 6 addresses how these tools can be used to evaluate strategic and technical options. The strategy includes staged deployment of Moon and Mars missions, with development options forming branches from the baseline mission. Ideally the aspects of the system designed early in the strategy will minimize the need for redesign if new directions in the strategy are taken.

As noted, the full strategy was not generated during this design course. Instead, various aspects of the process were addressed and tools were conceived to facilitate their design in later studies.

3. Knowledge Delivery: The Core of Exploration

3.1 Explanation of the view

An extensible space exploration infrastructure may be modeled as a mass transportation system, but also as a knowledge delivery system, since mankind is sending robotic and human explorers to space for the purpose of exploring and returning knowledge about the Moon, Mars and Beyond.

To justify knowledge as the deliverable to the stakeholders one must investigate why knowledge is the deliverable and who the stakeholders are. To answer the first question, one must first understand why do humans explore. To summarize, the three main reasons are

1. To expand the knowledge of our surroundings
2. To improve the technological leadership of the United States
3. To inspire interest in science and technology

Knowledge is the product of the exploration process. The knowledge of our surroundings is closely tied to science. Technological leadership is knowledge delivered to the technologist and explorers. The third point is that inspiration in science and technology is the knowledge delivered to public and commercial enterprises. In other words, the knowledge gained by the space exploration system is the value-added delivery to the beneficiaries or stakeholders. Therefore, to ensure the maximum value delivery, one may model the space infrastructure as a knowledge delivery system. Knowledge returned may be categorized as scientific knowledge, resource related knowledge, technical knowledge, and planning related knowledge. To build up the argument, first one must understand the value delivery to the scientists, which is diagramed in Figure 8. To understand the value identification, the goal of the space infrastructure is to increase the quantity and depth of scientific knowledge of the solar system by sustainably and successfully exploring the solar system, specifically the Earth, Moon, Mars, and Asteroids (EMMA) using an affordable and extensible human and robotic exploration system for the immediate benefit of the scientific community.

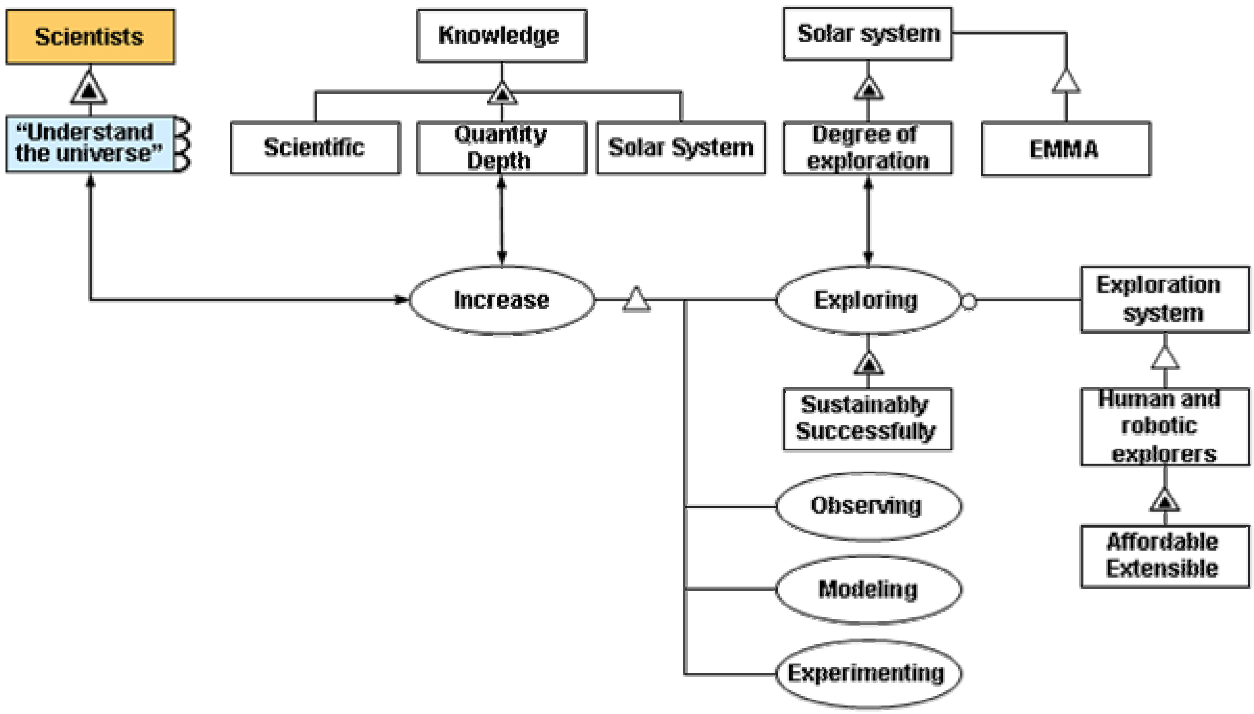


Figure 8: Value delivery to scientists diagram

The value delivered to the technologists and explorers is an increase in the quantity and depth of resource and planning related knowledge of the solar system by sustainably and successfully exploring the solar system, specifically the EMMA using an affordable and extensible human and robotic exploration system, and the previously gained resource. The value delivery can also be seen in Figure 9.

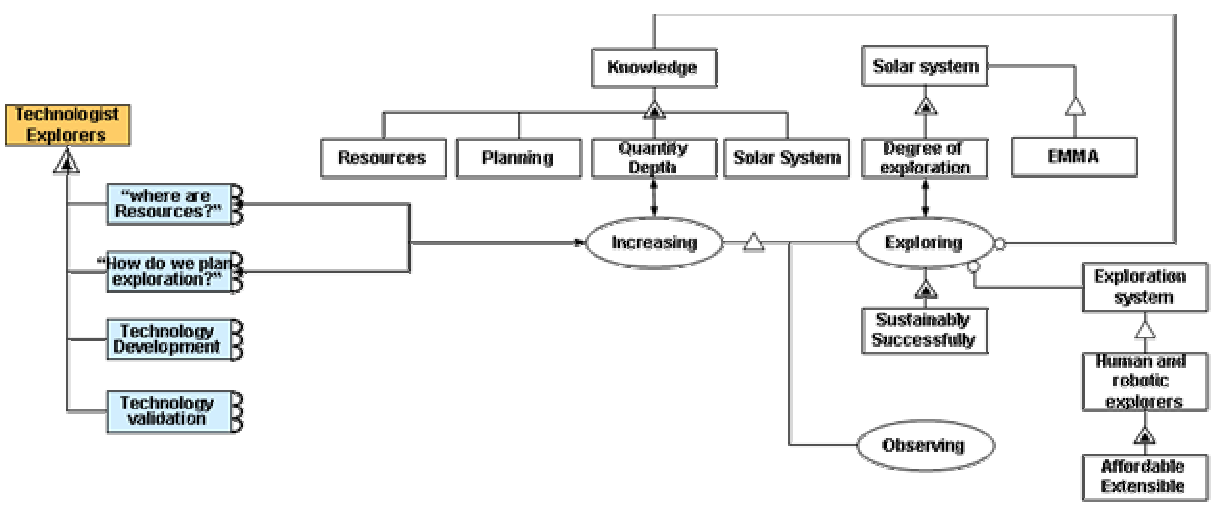


Figure 9: Value delivery to technologist/explorers diagram

In addition to the scientists and technologists/explorers, knowledge may be returned for the benefit of the United States public and mankind. NASA and the US government, international partners and commercial enterprises may derive additional knowledge benefit. The full objective process methodology map is shown in Figure 10.

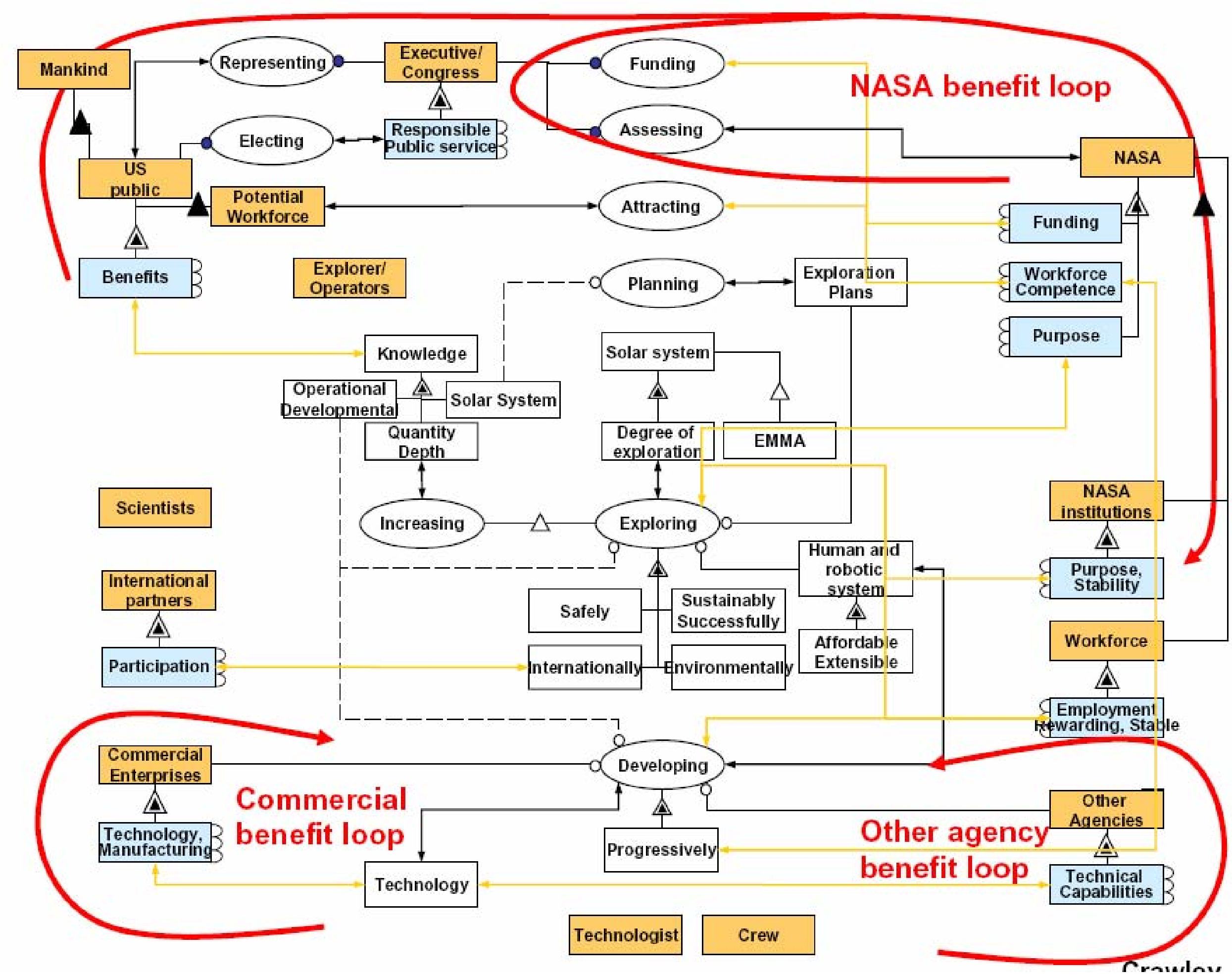


Figure 10: Knowledge delivery system OPM (Crawley, 2004)

3.2 Types of Knowledge

There are five main types of knowledge: Scientific-, Resource-, Technical-, Operational-, and Experience-related knowledge as seen in Figure 11.

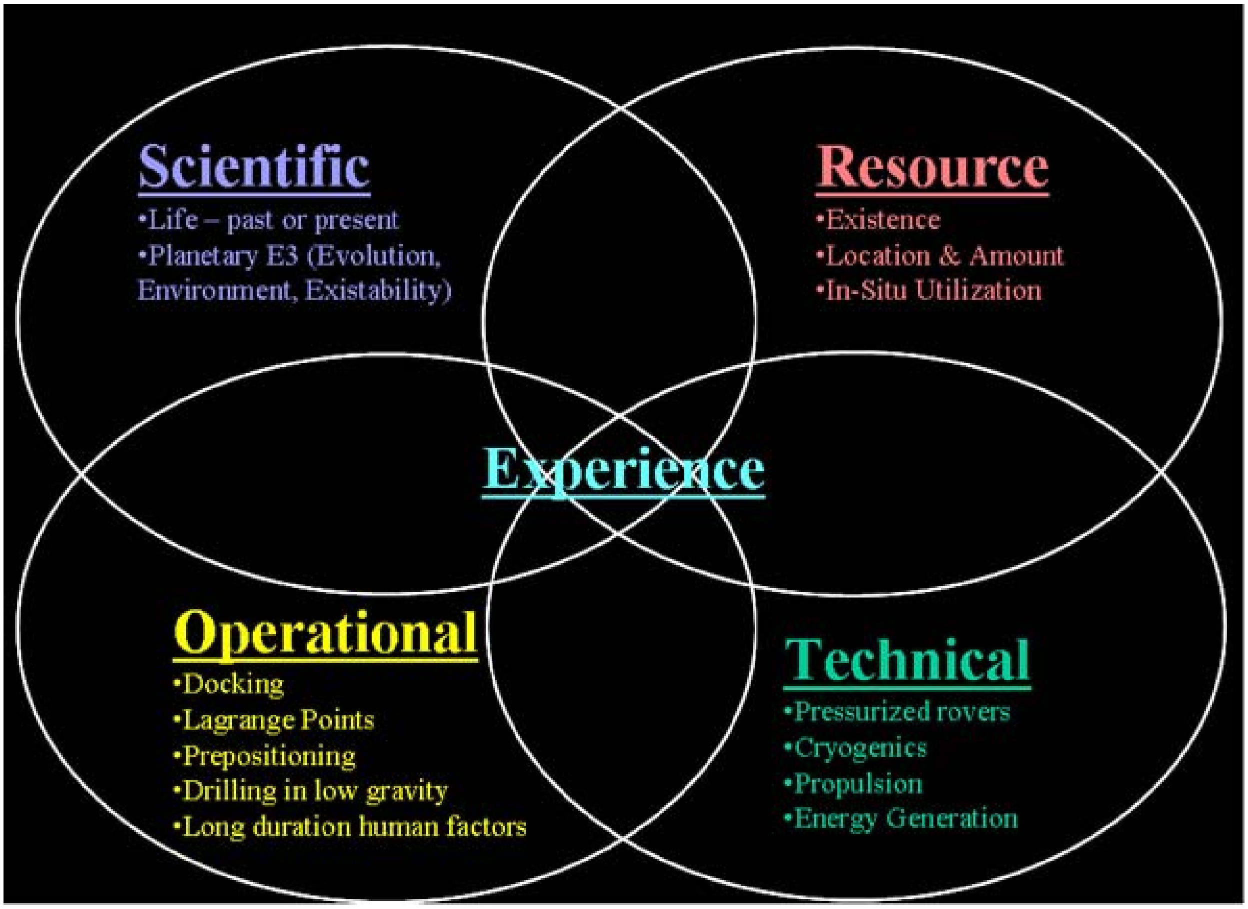


Figure 11: Five types of knowledge

3.2.1 Scientific Knowledge

Scientific knowledge can be generalized as the search for the existence of life and Planetary E^3 (the characterization of Evolution, Environment, and Existability of a planet or any celestial body). The existence of past or present life drives the search for resources such as water and other biomarkers. Evolution is mainly concerned with understanding the geology of a planet while Environment is the climate characterization. Existability is an assessment of biological potential, or how benign or hostile a planet is to human settlement.

One way to quantify scientific knowledge is through keeping track of the number of scientific publications resulting from the exploration effort. This is "an accepted measure of scientific productivity" and can be easily tracked using databases such as the NASA Astrophysics Data System (ADS) (Green, 2004). An example of this is seen in Figure 12, which captures the number of papers published as a function of the publication year for the Hubble Space Telescope. Using a numeric quantity, such as the number of publications, it is possible to make comparisons between different exploratory missions. It is then possible to understand when a diminishing amount of knowledge is returned and when it may be beneficial to gracefully retire an exploration mission. For example, if as in Figure 12, the number of papers per year were to steadily decrease for several consecutive years, the exploratory phase of the mission would be approaching its end. The final result of the knowledge publication graph might resemble a Gaussian distribution, where a mission is retired after it reaches a certain point in the distribution. Other possibilities for measuring scientific knowledge include news articles, press releases, website hits, educational television programs, PhD dissertations, or proposals.

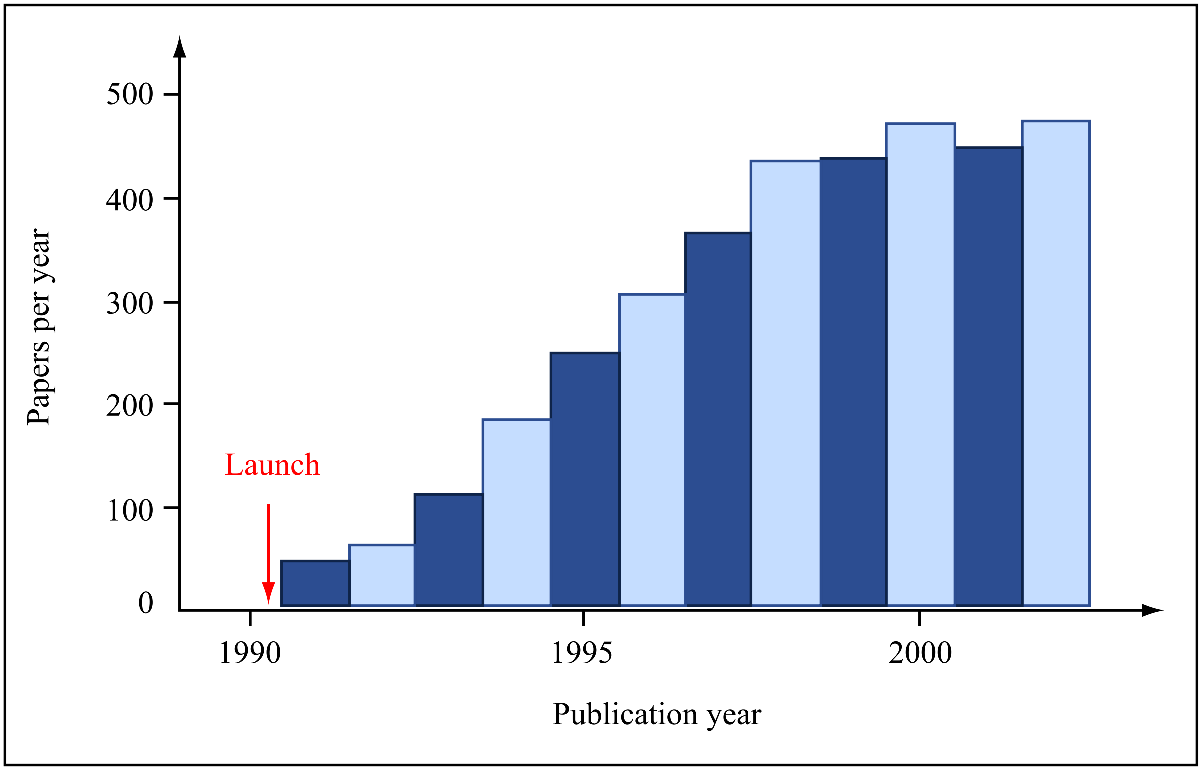


Image by MIT OpenCourseWare. Adapted from Beckwith, 2003.

Figure 12: Example of the quantity scientific knowledge from Hubble (Beckwith, 2003)

3.2.2 Resource Knowledge

Resource knowledge is defined by the existence, location, and amount of planetary resources that can be utilized by human explorers. These indigenous resources are necessary to build and maintain an extensible space infrastructure. Possible indigenous resources include water, Oxygen, Hydrogen, Ores/major metals, Nitrogen, and energy sources such as fusion materials. These resources may be obtained using the following three-step strategy:

1. Existence. The first step to is to determine the existence of the resource, most likely using robotic explorers such as orbiters. First, implied existence of the resource is obtained by knowledge carriers, which transmit passive bits. The next step is to obtain direct proof of the resource's existence, either by transmitting bits or by transporting atoms.

2. Location and Amount. The second step is to determine the global amount of a resource, possibly using an orbiter or rover. An unmanned rover is beneficial for reconnaissance of biohazardous and toxic regions. As the resolution of resource knowledge about the specific resource locations and amounts increases with exploration, a point is reached when a human mission may begin to extract the resource. This point would occur when the resource location accuracy at least meets the landing accuracy plus the travel distance of a human mission.

3. In-situ Utilization. The final step is to begin in-situ resource utilization for exploration needs, such as propellant, building materials, and energy. A lander can achieve basic in-situ knowledge, but full exploitation will likely occur with a human mission. Some of the issues with in-situ utilization are related to the degree of

manipulation needed. For example possible water ice on the Moon may need a heating, purification, and extraction process before it is useable.

3.2.3 Technical Knowledge

Technical knowledge is the assessment of the engineering abilities associated with the space transportation system similar to the NASA Technology Readiness Levels (TRL). The space transportation system will slowly attempt to integrate various new technologies into the existing infrastructure. The level of working ability for each technology is the technical knowledge delivered. An example is the development of in-situ resource technology, where currently designs exist at various conceptual levels. As the system is developed, in-situ resources can be utilized. The degree of success delivered, measured in cycle efficiency, total power consumption, and resource produced by the in-situ technology is the technical knowledge. Technical knowledge gained will affect the evolution of the space transportation system. It will help determine how missions grow, which will be discussed in later sections.

3.2.4 Operational Knowledge

Operational knowledge is the capability of performing activities related to the space transportation system. An example of operational knowledge obtained during the Apollo program is lunar orbit rendezvous, or docking. The technology for docking existed and the procedure for it was known, but not until it was successfully accomplished was there a large amount of operational knowledge gained concerning docking. Other examples include operational knowledge gained from Lagrange point maneuvers, pre-positioning, drilling in low gravity environments, and long duration human factors issues. An interesting point about operational knowledge is that unlike the previous types of knowledge, a good deal can be gained from failures. For example, during Apollo 13, operational knowledge was gained when the Lunar Module was used as a “life boat” and various components were also creatively utilized to ensure crew survivability. It is uses of a system beyond their intended designs, which can lead to operational knowledge. Therefore operational knowledge can be gained by understanding the flexibility of a system.

3.2.5 Experience Knowledge

The human experience can also be a type of knowledge, because there is a unique gain that is achieved outside of data or physical returns. It is may be thought of as a combination of the four other types of knowledge. A human presence can gain knowledge that is different from any robotic explorer or remote sensor due to its rapid cognitive thinking and senses. This idea is very similar to the notion of experience as a knowledge carrier, which is outlined in Section 2.3.3.

3.3 Carriers of Knowledge

Carriers of knowledge are divided into three main categories, bits, atoms, and human experience.

3.3.1 Bits

Bits carry knowledge in the form of the data. There are two types of bits, passive bits and active bits. Passive bits are defined by data obtained without interacting with the observed environment, such as taking a picture. Active bits involve interacting with the environment such as by taking a measurement and transmitting the measurement data back.

3.3.2 Atoms

Atoms are the physical samples that carry knowledge about an exploration excursion. These samples carry two forms of knowledge: implied discoveries and direct proof discoveries. An implied discovery is knowledge that is gained by observation or measurement of a sample, which leads to an implicit discovery; for example, a weathered rock exhibiting the past existence of water by erosion patterns. A direct proof discovery is the knowledge carried by hard evidence of a phenomenon, for example, a Mars rock with a pocket of water carries proof of Martian water by direct observation of the specimen.

3.3.3 The Human Experience

The human experience of exploration has the ability to carry the greatest amount of knowledge. While robotic explorers could be the prime means of bringing back bits and atoms, they are most effective for large amounts of systematic returns. In contrast, there are three human physiological traits that provide an optimal combination for returning knowledge:

1. The human brain. Capable of instantaneous programming, the human brain is a “qualitative supercomputer” (Schmitt, personal communication). It can react to field experience and training and adds a high degree of flexibility
2. Eyes. The human eyes have high mobility, dynamic range, and quick three-dimensional integration, especially in the 10 – 15 meter range (Schmitt, personal communication).
3. Hands. Perhaps the most underutilized human tool, but if their dexterity can be used to their full potential they can greatly increase the human exploration ability. For example, hands posses the capability of returning detailed tactile feedback, etc.

An example of the benefit of the human experience can be seen in the NASA Opportunity Rover on Mars. Throughout its mission, it has returned knowledge by observation and interaction with the environment (bits), but it has sent back even more questions about Mars. These questions could have been answered immediately by a human field geologist present on Mars, due to his/her unique ability to analyze the environment with his/her experience, physiological tools, and basic scientific instruments, such as a hammer (Schmitt, personal communication).

Robotic and human explorers have different degrees of time and spatial processing abilities, as seen in Figure 13. Time processing ability is meant by how an explorer is able to take in and understand the value of interesting exploration targets. Spatial processing is defined as the ability to understand the value of exploration targets in a global resolution and also a high-resolution sense. Figure 13 shows three types of robotic explorers, penetrators, orbiters, and rovers. Penetrators are geologic instruments that are embedded in the ground and have no mobility. An example of a penetrator is the Deep Space 2 probes. Penetrators cannot move and only have spatial resolution of their immediate surrounding, and rely on their instruments to record data as it comes to them. They passively gather data and have limited range to interact with the environment and collect additional data. In addition, penetrators are not reprogrammable (yet), once they land, they execute their specified tasks. Therefore they are shown to have low time and spatial processing abilities. Orbiters have a large global resolution, however they are unable to achieve high resolution of specific targets (yet) or look at a target from multiple unique angles. An example of an orbiter's limitation is that, it would have a difficult time looking inside a cavern. Rovers are shown with greater spatial processing ability because they are able to look at targets from multiple viewpoints and with a high resolution. They cannot achieve the global scale resolution of an orbiter, however with increased mobility and presence rovers can attempt to create a larger global picture with high resolution. Rovers are also shown with higher time processing ability because they can be flexible to their environment. They can take a picture of their surroundings, and then be commanded to move to locations they seem the most interesting. In contrast, an orbiter can only explore targets that are in its orbit's coverage region. Finally, the human field geologist equipped with a rover and tools such as a microscope has the highest amount of time processing ability due to his training, experience, and brain. Equipping the human explorer with high mobility (rovers) and microscopes, will give him the ability to have global resolution and high specific resolution of targets. Therefore, a human explorer is the optimal combination of time processing ability and spatial processing ability. This forms an argument for exploration by humans in place of robotic systems.

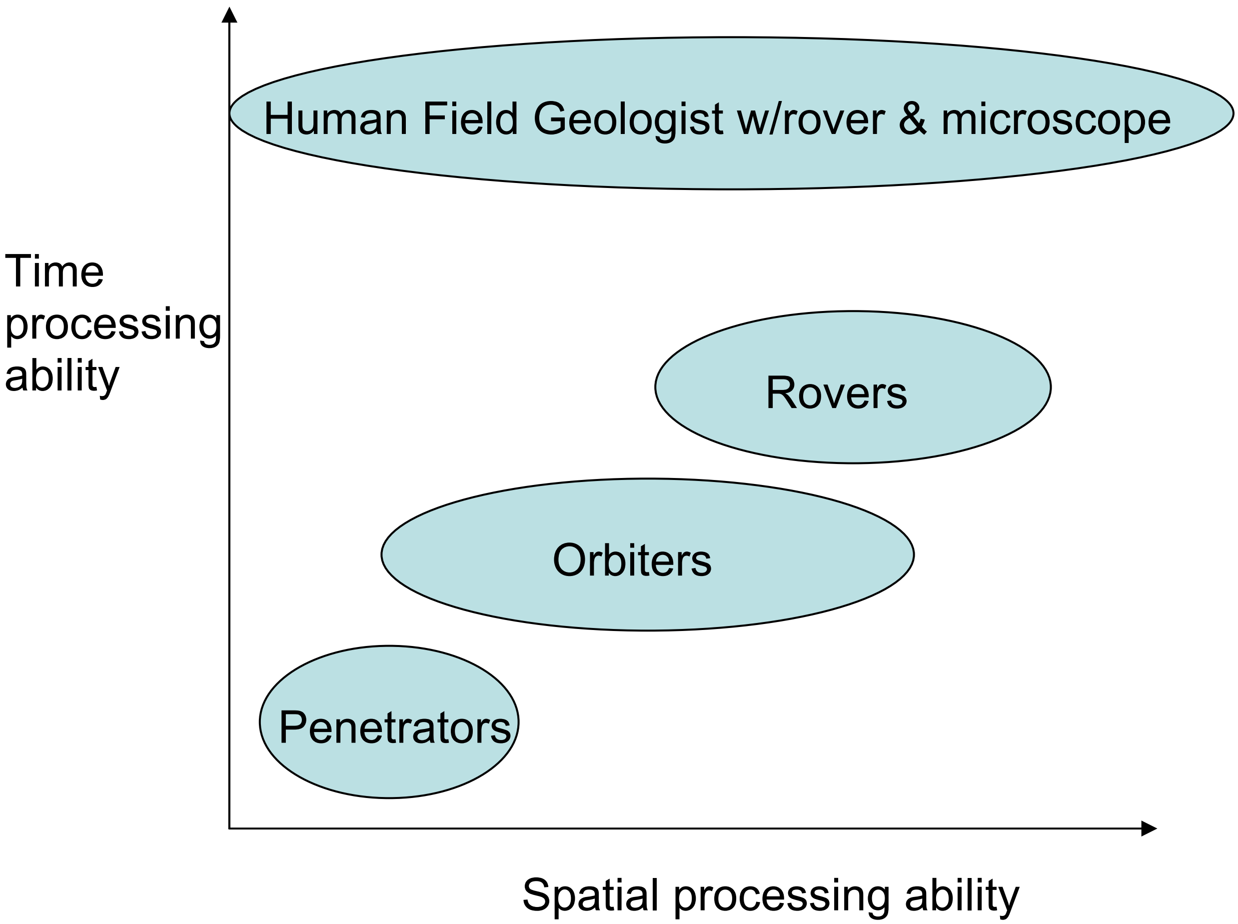


Figure 13: Time and spatial synergy for robotic and human explorers

Figure 14 illustrates a summary of the knowledge carriers and how they are related by their degree of interaction with the environment, and the quantity of that specific knowledge carrier that mankind has accumulated. Passive bits are represented by pictures of planets and the galaxy and currently carry the most knowledge. In decreasing amounts of quantity are active bits represented by graphs of Mars Seismic activity from Voyager, followed by pictures of Mars rocks from the Opportunity rovers. A Moon rock represents sample return, which is solely from the Moon. Finally, the human experience has the highest degree of interaction with an exploration environment; however, it is limited to the Apollo excursions. The yellow curve illustrates the utility of the knowledge carriers for our current state of exploration. If exploration is to be successful in returning larger amount of knowledge, the red curve illustrates possible outcomes of an extensible space architecture.

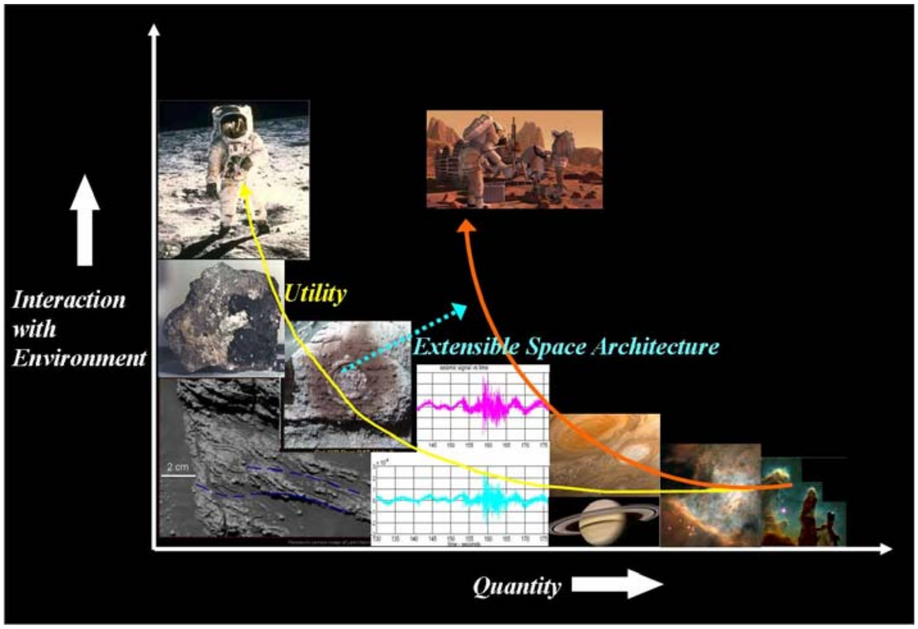


Figure 14: Carriers of knowledge

3.4 Knowledge vs. News

One challenge for the knowledge delivery system is to understand the difference between knowledge and news. To first order, news is the unique knowledge on a generalized subject. For example, the discovery of an extrasolar planet is news; however, discovery of the nth extrasolar planet is not news to the public. News is the knowledge that immediately appeals to the public. A notional graph of news versus exploration milestones can be seen in Figure 15. Shown are theoretical news values for Apollo and future milestones. The diagram shows the notion that a new milestone, such as the first Apollo mission will have a high news value, but there is a decay in news as the Apollo missions progress, shown by the decaying black line. If there is a new unique milestone, such as a human Moon return or a 1st Mars Human Landing, it is possible that there will be a large increase in the news they generate. As with Apollo, these events will be followed by a decrease in news value, since the 2nd and 3rd human Moon return and 2nd and 3rd Mars human landing will not be new milestones. The notional diagram exhibits this high frequency decay. Overall, there is a low frequency decay in news value of the entire exploration system. Thus if the media does follow this trend, it is unfavorable for sustainability.

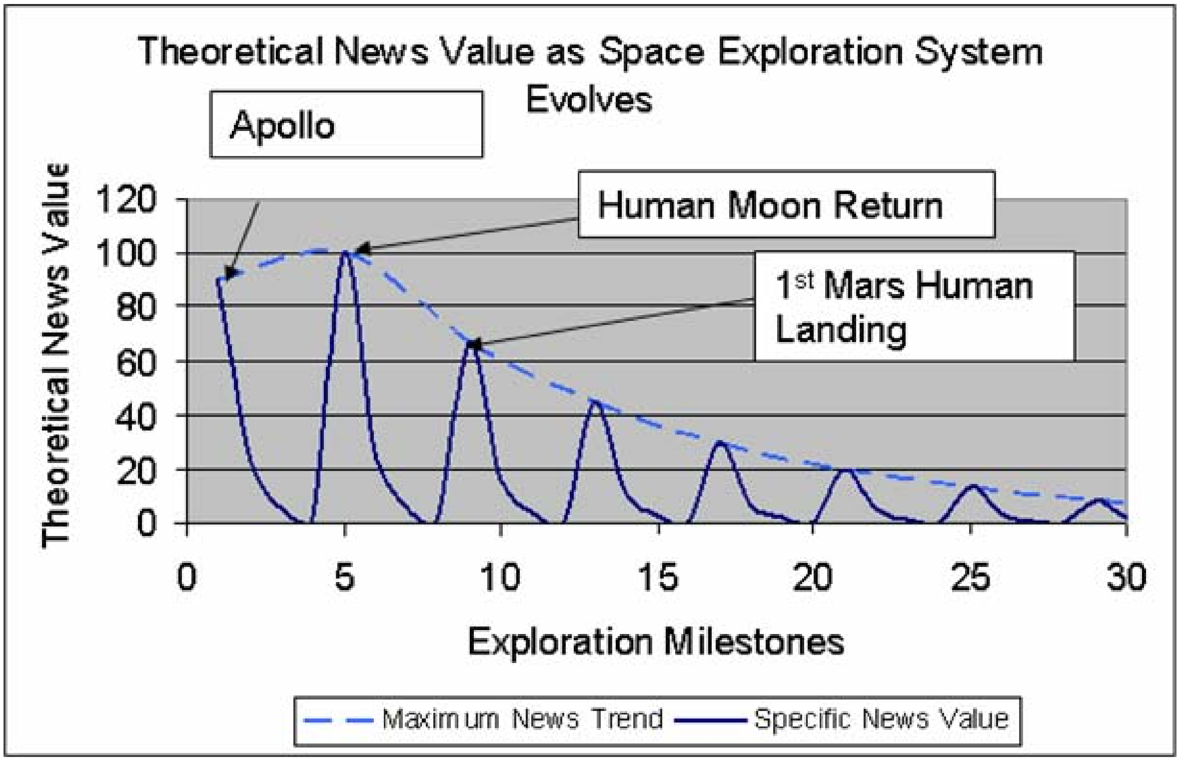


Figure 15: Theoretical news value as the space exploration system evolves

An important distinction may be drawn between public interest and media interest. When the media loses interest in a subject, the public tends to lose awareness of it. The media interest is also one of the main processes of continuing education that the knowledge delivery system uses. For the system to be most effective, the public interest could be coupled with education, but decoupled from the media. A gradual increase in public interest is necessary to create a knowledge distribution system that is independent of media. A challenge to this solution is that many politicians, who advocate for funding, tend to have some of the strongest associations with the media. A breakthrough in separating public interest from media interest can occur when personal connections with the space transportation architecture are developed. For example, when settlements, be they permanent or semi-permanent, exist outside of the Earth, many people on Earth will have personal connections with those on the Moon or Mars generating interest that is independent of the media. A breakthrough can occur when there is commercial interest in the Moon and Mars.

It is important that the knowledge delivery system does not rely too heavily on the media. The media loves success, the first time, but in general it looks for disaster (Schmitt, personal communication). A good example may be found in the Apollo missions. The media coverage of Apollo 8 and 11 was huge, since these missions achieved historic firsts. Coverage was also large for Apollo 13 because of its challenges, and then Apollo 14 since it was the first after a disaster. However the later Apollo missions did not experience such significant media coverage.

3.5 Knowledge Delivery Process Map

The knowledge delivery process can be summarized by the CDIO phrase/process with an added S at the end. These letters stand for Conceive, Design, Implement, Operate, and Science. During the Conceive stage the mission goals, requirements and trades are identified, then during the Design stage the goals and requirements are used to create an comprehensive design of the mission and all the elements that are required to make it successful. The Implementation stage consists of the building of the elements designed in the previous stage. During the operate stage the mission will collect the data that will eventually be turned into knowledge during the Science stage. Of course there is some overlap in the stages, but for the most part each stage acts as its own step in the knowledge delivery process. Each mission in an extensible exploration system must follow this pattern, which should begin to repeat at about the time that previous mission has reached the O stage. The estimated relative times for each stage for robotic, human Mars and Moon missions are given in Table 1.

Table 1: Knowledge delivery process

Knowledge process Timeline								
Mission 2				C	D	I	O	S
Mission 1	C	D	I	O	S			
Robotic Missions	1x	3x	1x	5x	nx			
Human Moon Mission	1x	3-4x	1x	0.1-0.5x	mx	where m<<n		
		(launches)						
Human Mars Mission	1x	4-5x	1x	3x (0.1x)	mx			

In the above Table 1, n stands for a constant amount of time that should be between six months and one year. The amount of time that it takes to evaluate and handle raw science data is less on a human mission than on a robotic one. After the first round of missions, the relative times should change slightly, especially if there is any form of reusability added into the system.

3.6 Knowledge Delivery Time

The Knowledge Delivery process is summarized in Figure 16, which shows the overall knowledge delivery cycle and introduces the concept of Knowledge Delivery Time (KDT). There is some time difference between the beginning exploration phase and the knowledge delivery. The primary mission can directly lead to mission delivery shown by the dotted line in Figure 16, or knowledge processing on Earth, after the primary mission on Earth can then lead to knowledge delivery. After knowledge is delivered, the exploration cycle begins again with another conceive, design, Implement, and operate processes.

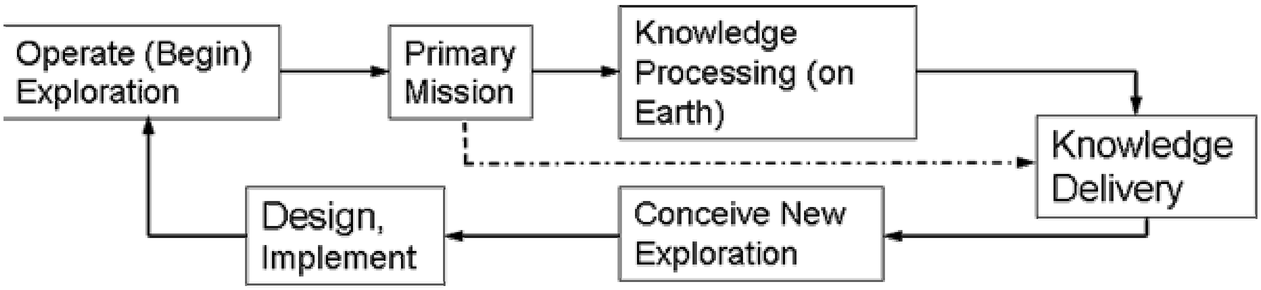


Figure 16: Knowledge delivery cycle

Knowledge delivery time is defined as the time between collection of knowledge and its delivery. It is usually not instantaneous. Two case scenarios, illustrated in Figure 17, can be used to understand the concept of knowledge delivery time.

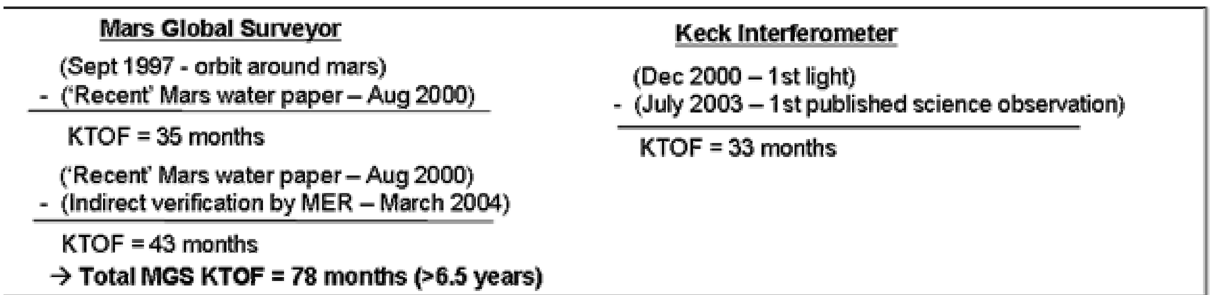


Figure 17: Knowledge delivery time examples

For Mars Global Surveyor (MGS), there is a time difference between orbital insertion, which represents when exploration and the collection of knowledge begins, to the delivery of a significant amount of knowledge, in the form of a journal paper discussing the existence of water on Mars. Pictures from Mars from MGS, which are passive bits knowledge carriers, were used for the discovery (Malin, 2000). The KDT from initial exploration to knowledge return was approximately 35 months. The next exploration of Mars with the objective of determining recent Mars water was the Spirit and Opportunity rovers. Knowledge delivered by the rovers was the indirect verification of recent Mars water by pictures (passive bits) and by drilling (active bits). The knowledge delivery time between the two results of the two knowledge carriers, passive and active bits, is approximately 43 months. Thus, the total KDT is about six and a half years. The Keck Interferometer also exhibits a KDT in a range similar to that used by the Mars rovers; approximately 33 months between the start of exploration when first light is achieved and its first published science observation (Keck website, 2004).

The following examples of KDT come from robotic missions. Therefore, future robotic explorers could have similar knowledge delivery times. The benefit of human exploration is that it has the ability to decrease and even eliminate the knowledge delivery time. The human exploration experience can process and interact with the environment rapidly and return knowledge with minimal delay. For example, the Apollo explorers could immediately determine that lunar regolith (to first order) was mainly composed of inert dust and rock fragments verifying knowledge from photos taken over many years.

3.7 Drivers of Knowledge

Different aspects of the mission such as crew size, experience, excursion time, exploration time, mobility, range, and instrumentation affect knowledge. All of these with the exception of instrumentation will be modeled. The reason instrumentation is not modeled here is that it varies by mission depending on the specific science objectives of that mission. For example, a mission focused on geology will have very different instruments than a mission focused on climatology. Instrumentation is further discussed in Appendix 9.5. During the later Apollo missions approximately one third of the total time spent on the Moon surface was spent during an excursion (Table 2). The earlier Apollo missions did not have as high of an excursion time for two possible reasons. The first is the lack of experience. Later Apollo missions were able to gain experience with surface operations on both the Earth and Moon from the first Apollo mission. The other reason is because the first few Apollo missions did not have science as their primary mission objective. Only Apollo 15-17 had "extensive scientific investigation" of Moon as a primary mission purpose whereas Apollo 11's primary mission was a manned lunar landing demonstration (NASA website, 2004). The primary mission for Apollo 12-14 was precision piloted landing and systematic lunar exploration. Thus, experience had an effect on the excursion time for lunar missions, and had a maximum of 30% of the total lunar stay time for a mission.

Table 2: Apollo mission details (NASA website, 2004)

	kg	duration (hrs)	outside LM[min]	max d from LM (m)	%outside
Apollo 11	21.6	22	152	61	11.5
Apollo 12	34.3	31	465	411	25
Apollo 14	42.3	33	563	1454	28.4
Apollo 15	77.3	67	1115	5020	27.7
Apollo 16	95.7	71	1214	4600	28.5
Apollo 17	110.5	75	1324	7629	29.4

Each excursion had a predetermined plan, however there were times when independent exploration was allowed and carried out by the astronauts. An example of independent exploration results was the orange pyroclastic glass discovered in Shorty Crater by Apollo 17. This exploration was not dictated by ground. When the astronauts were exploring in the area, they had 30 minutes of rapid assessment and gathering before the mission controllers were even aware of the events. Independent exploration allows a human to fully utilize his/her training, experience and senses to return knowledge, either as samples, pictures, observations, technology used, or operational procedures.

Using the number of crew, excursion time, exploration time, and mobility, the coverage area during exploration can be determined. Assuming an experienced crew, the excursion time can be maximized as 30% of the mission surface time, which is similar to the Apollo missions. Exploration is defined by examining the surrounding area within 10 meters of the astronaut, since this is the range for which the human eye has an optimal ability to determine unique aspects of the surroundings (Schmitt, personal communications). It is assumed that some portion of an excursion is spent performing this exploration process. For this knowledge model, 30% of the excursion time was spent as this independent exploration time. This could vary a great deal on an excursion by excursion basis, but it was approximated based on personal communications with Jack Schmitt. There are three types of mobility, a walking pace (or gait) while traversing without exploring, a slower exploration pace, and a rover speed. The walking pace is based on the design speed of an Apollo astronaut and determines the maximum traveling distance per day from the starting point, presumably a lunar module . The exploration pace is estimated as four times slower than the gait because the human is more carefully analyzing the environment and perhaps taking measurements or pictures (NASA Headquarters website, 2004). The rover pace is based on the Apollo Lunar Rover (Apollomaniacs, 2004) and is capable of expanding the maximum traveling distance per day. These parameters are summarized in Table 3 and are used to determine how much coverage per day can be accomplished.

Coverage is defined as the area traversed at an exploration pace and can be used to quantify how much knowledge potential is gained. As more area is explored, a greater amount of knowledge is potentially gained, either from science or resource data, or technical and operational procedures. Since exploration pace is slower than the maximum speed by astronauts, either by rover or by walking, there is a certain amount of coverage than can be achieved per day. The number of crew available will directly affect this coverage, which is shown in Figure 18. There is a clear direct relationship between the number of crew and the maximum exploration coverage achievable per day. The coverage achieved while walking can either be completed by increasing the number of days on the surface, or by increasing the number of crew. If walking is the fastest mode of transportation, 100% coverage can be achieved rather easily, after which the knowledge potential is maximized in that particular landing site. Using a rover increases mobility, and in this case, the maximum area that may be traversed in a day dramatically increases, decreasing the percent area covered per day. To increase the exploration coverage requires either a longer stay than the non-rover case or a larger crew.

Table 3: Knowledge drivers model parameters

outside LM fraction	0.3
exploration time	0.3
traveling pace (gait)(m/hr)	3600
exploration pace(m/hr)	900
rover pace (m/hr)	14000

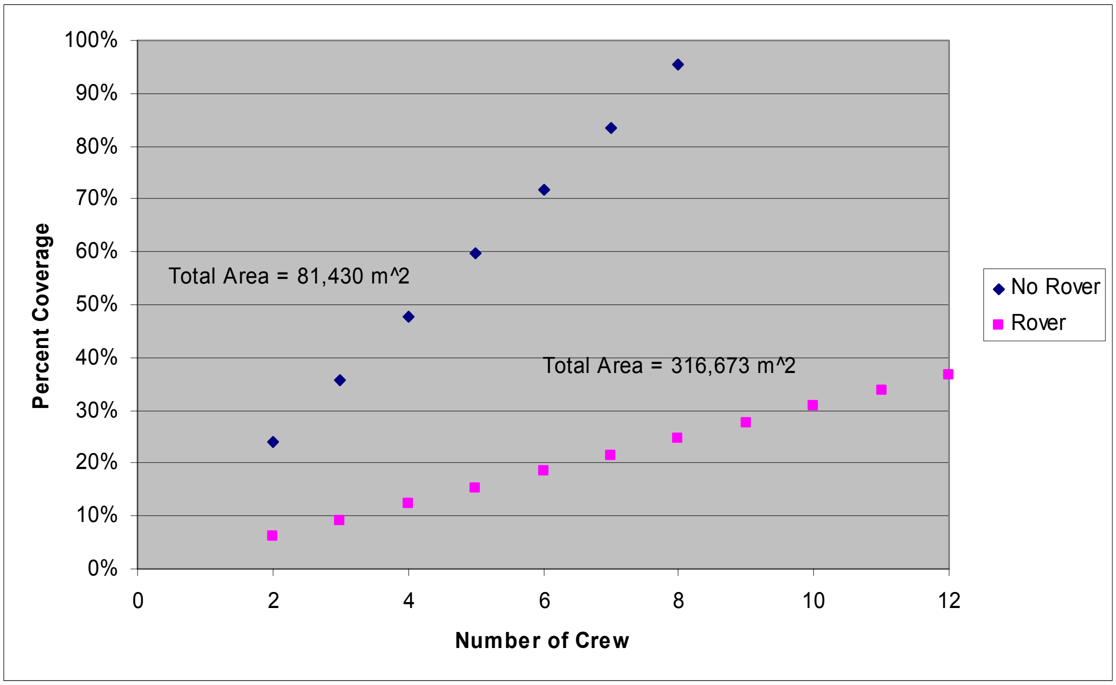


Figure 18: Knowledge potential: maximum exploration coverage per day versus number of crew

This model can also incorporate faster speed rovers. Introduction of a pressurized rover that allows astronauts to stay outside of the base can further expand knowledge potential as seen in Figure 19. By creating a remote base, travel can be further expanded from the remote location, thus expanding the knowledge potential from exploration.

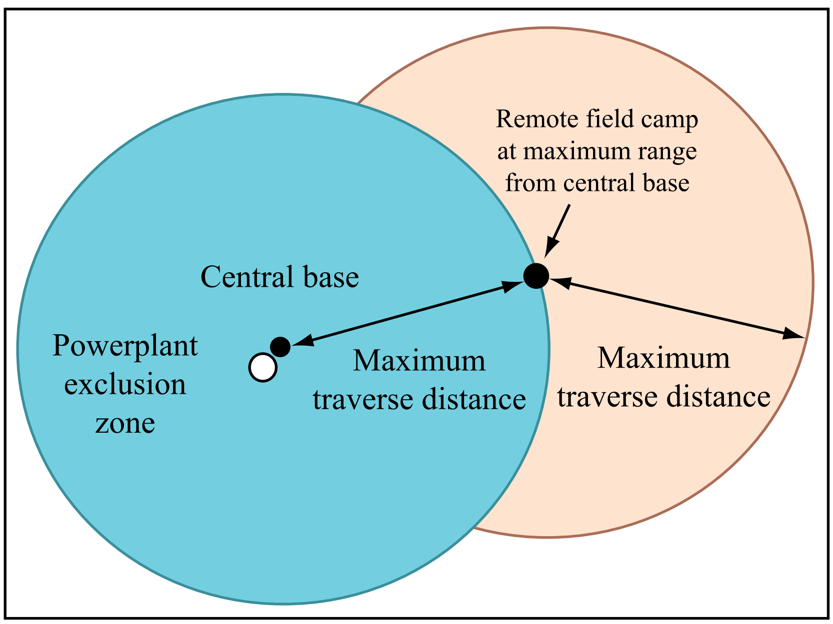


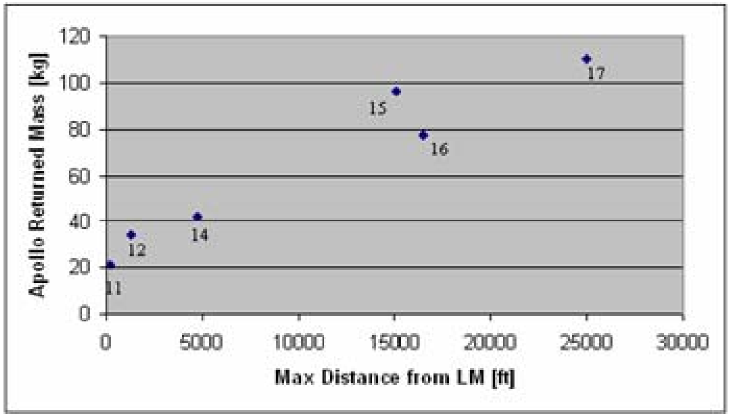
Image by MIT OpenCourseWare. Adapted from Hoffman, 1998.

Figure 19: Expanding the exploration potential using a remote base (Hoffman, 1998)

Another possible method of increasing knowledge potential is to increase the amount of independent exploration time per excursion. In fact, as missions proceed, a greater amount of independent and flexible time for the astronauts should be encouraged, especially as communication delays increase for farther missions (Schmitt, personal communication). In conclusion, the knowledge potential of a mission can be predicted by the exploration coverage, which is affected by the number of crew, experience, excursion time, exploration time, and mobility.

3.8 Knowledge Drivers: Apollo Case Study

The Apollo missions offer a good case study for how knowledge return is affected by knowledge drivers. In this case, knowledge returned is quantified by the mass of samples returned from the Moon. If Moon rocks are carefully chosen, increasing amounts of samples gathered should return increasing amounts of knowledge (to a first order), most likely scientific or resource related. The amount of knowledge can also be driven by the exploration time and the distance traveled. Figure 20 illustrates the returned mass as a function of the maximum distance from the Lunar Module and as a function of the time spent outside of the Lunar Module (data from NASA website, 2004). Clearly there is a direct relationship, however there are other inherent drivers that are not as easily captured. They are human experience, timing, and new technology. Apollo 11 lacked experience and thus had a limited exploration time and did not traverse much distance, as shown by the point closest to the origin in each plot. As the missions progressed in time, experience increased, resulting in longer and further exploration. Each of the graphs illustrate a large jump in the mass of samples, and therefore the knowledge, returned, coinciding with Apollo 14 to Apollo 15 because Apollo 15 was the first mission to include a rover. Figure 21 illustrates a graph of the cost of the knowledge returned and the table showing the percent increase of cost and mass returned of one Apollo mission relative to the previous. It is important to note that the cost shown here does not include development costs. The largest percent increase in knowledge coincides with the largest percent increase in cost, which occurred in Apollo 15 because of the introduction of the rover. This is a clear example of how infusing new technology into an existing architecture can result in an increase of knowledge returned. Apollo 16 and 17 continued using the rover and experienced a much higher sample return than the non-rover missions. With added human experience from previous rover-enable missions, they were also able to traverse further and explore for longer periods of time.



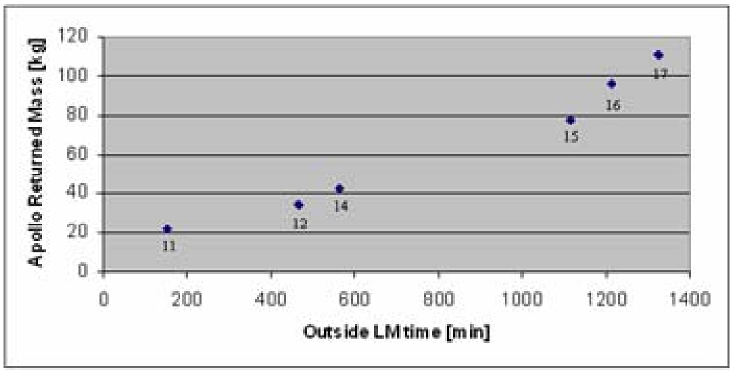
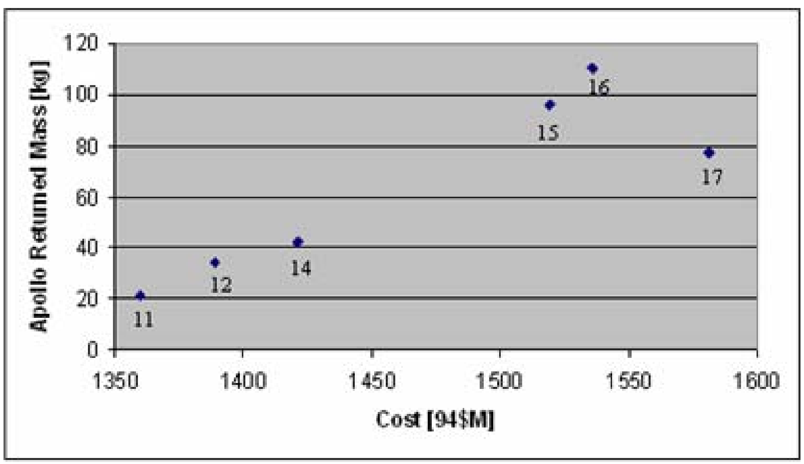


Figure 20: Apollo knowledge drivers



	Samples Returned (kg)	cost (94$M)	%kg inc from previous	%cost inc from previous
Apollo 11	21.6	1360		
Apollo 12	34.3	1389	59	2.1
Apollo 14	42.3	1421	23	2.3
Apollo 15	77.3	1581	83	11.3
Apollo 16	95.7	1519	24	-4
Apollo 17	110.5	1536	15	1
	381.7			

Figure 21: Apollo cost trends

3.9 Knowledge Summary

The design of a sustainable space infrastructure will use knowledge as the deliverable as a metric for its design. The next sections will discuss how different missions gain various types and levels of knowledge. Knowledge will affect mission characteristics such as landing locations, surface mobility, and mission lifetime. The knowledge model developed in this section was not directly used in the mass transportation architecture, but it would be beneficial for future research to incorporate the knowledge drivers into the mass architecture in a more integrated fashion.

4. Baseline Mission Designs

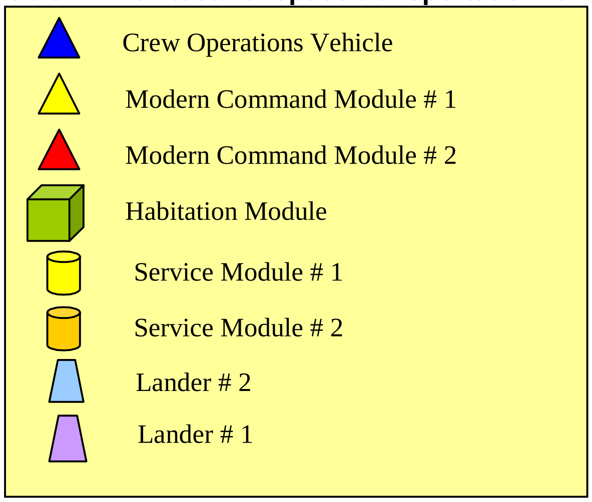
Having identified knowledge as the key value-added deliverable, and examined how kind and quality of knowledge vary with mission profile and technology, the problem remains how to integrate this understanding into a space system conceptual design that takes into account sustainability. This chapter describes the first step in the process described in chapter two to create sustainable exploration systems: designing staged missions to the Moon and Mars. Staged missions are defined as Short, Medium, and Extended Stay, and were designed with the goal of maximizing commonality through each stage. Later steps in the design process, described in chapters five and six, help determine functional and formal commonality *across* Moon and Mars missions.

It is important to note that once commonality across Moon and Mars missions has been identified and exploited through changing mission forms and operations, the individual staged Moon and Mars mission must be revisited and altered according to new capabilities and requirements. Steps one through three of the design process are thus iterative. This chapter represents the culmination of one iteration of this process, and thus presents considerable formal and operational commonality between staged Moon and staged Mars missions. For this reason, it begins with a discussion of the forms that will be used for *both* the Moon and Mars staged missions.

4.1 Brief Description of Formal Elements

The *Crew Operations Vehicle (COV)* is functionally similar to the Apollo Command Module, capable of transporting a crew of three and supporting the crew for a short duration mission. The *Habitation Module (HM)* is an extensible habitable volume, made up of multiple modular sections. The Habitation Module can sustain life for long duration missions. When COV and HM modules dock, they form the *Crew Exploration System (CES)*. The *Service Module (SM)* is capable of providing propulsion for transiting the crew from Earth to destination or destination to Earth. Service Module #1 is the engine for the trip to the destination while Service Module #2 is the engine for the return trip. In combination with the COV and HM, this module is defined as the *Moon/Mars Transfer Vehicle (MTV)*. The *Mars Landers (ML)* or the *Lunar Landers (LL)* are functionally similar to the Apollo type Lander, although have slightly different forms for Moon and Mars missions, and capable of transporting three crewmembers from orbit to the surface and back into orbit. Capable of providing accommodations for three crew members for launch into LEO and descent back to Earth, the *Modern Command Module (MCM)* is functionally similar to the COV. Two MCMs are needed for missions with crew sizes of six. These modules are summarized in Table 4.

Table 4: Architectural space transportation forms



4.2 Moon

4.2.1 Introduction

The lunar baseline mission includes Short, Medium, and Extended missions. The objective of Short Stay Lunar Missions (SSLM) is to demonstrate the basic technology for lunar missions. The SSLM is equivalent to the first “return to the Moon” mission described in President Bush’s New Vision for Space Exploration Program. Approximately two SSLMs are suggested, with a frequency of two per year, as dictated by Earth launch considerations aimed at maximizing launch cost efficiency. Refer to Section 6.4.2 for more information about launch considerations.

The objectives of the Medium Stay Lunar Missions (MSLM) are to acquire scientific knowledge and assess the value of potential locations for Extended Stay Lunar Missions (ESLM). Approximately five MSLMs are suggested with at least one landing on the far side of the Moon or a lunar pole. A frequency of two MSLMs a year is suggested based on Earth launch considerations. The MSLM is based on the “stepping stone” approach mentioned in President Bush’s New Vision for Space Exploration Program to steadily increase our mission complexity while expanding our reach out into the solar system.

Finally, the Extended Stay Lunar Missions (ESLM) will involve a six-month surface stay and will include a semi-permanent base. This is a continuation of this “stepping stone” approach. These missions are to serve as a testbed for future Mars missions and provide a platform for long-term lunar-based science investigations. Approximately two ESLMs are suggested on the far side of the Moon and/or a lunar pole to carried out at a frequency of approximately 1.5 missions per year.

4.2.2 Literature Review

Although the baseline mission architectures presented in this paper are not optimized, previous lunar mission architecture studies were reviewed to inform architectural decisions for the lunar baseline mission presented in Section 4.2.4.

Houbolt (1961) extensively studies many combinations of mission architectures, concluding that Lunar Orbit Rendezvous (LOR) is the fastest and most reliable architecture. Houbolt also shows that this method of going to the Moon greatly reduces the amount of mass required to be launched from Earth. Houbolt outlines requirements for power, instrumentation, life support, and navigation, as well as launch masses for different mission architectures. These include a direct to the Moon concept, Earth orbit rendezvous, and the use of different fuels and launch vehicles. He considers small, medium, and large landers, and also suggests the possibility of using two small landers, because "this combination has a rescue capability not possessed by direct or other forms of lunar landing missions" (Houbolt, 1961, p.13). Trade-off studies and calculations include trajectory options, errors in guidance, abort options at different stages of the mission, possible fuels, size of landers and return vehicles, and the mass required in Earth and lunar orbit for several architecture options. Notably, Houbolt's report includes sections on safety and reliability, development program scheduling, and facility needs.

Eckart (1999) qualitatively describes the advantages and disadvantages of six lunar mission architectures. The first architecture, a Direct One-Way Mission, is particularly beneficial for cargo missions with expendable transfer and landing stages. A slight variation of this mission, the One Way Mission with Lunar Orbit Staging is beneficial for both cargo missions and crewed missions, assuming a return vehicle for the crew is positioned on the lunar surface. The third architecture, an Apollo-type Mission is useful for architectures involving crew exchange and re-supply from one vehicle to another. One variant of the Apollo-type Missions uses the Earth-Moon L1 (EM-L1) or EM-L2 as a staging point rather than lunar orbit; however, Eckart posited that this architecture yields lower performance and longer transit times than the Apollo-type Missions. The fifth architecture described uses a Lunar Transfer Vehicle (LTV) and a Lunar Excursion Vehicle (LEV). In this case, the LTV transfers the crew from and Earth orbit space station to a lunar parking orbit. The LEV transfers the crew from lunar orbit to lunar surface and back to lunar orbit. The LTV then returns the crew from lunar orbit back to the Earth orbit space station. This mission utilizes reusable stages and LEO infrastructure, and restricts trans-lunar and trans-Earth injection opportunities. Finally, the last architecture is a variation of the LTV/LEV architecture using EM-L1 or EM-L2 as a staging point rather than lunar orbit. Eckart asserts that use of either of these libration points lowers performance and increases transit time.

Condon and Wilson (2004) describe mission architectures similar to those put forth by Eckart in quantitative detail. Condon and Wilson analyze ten different mission profiles grouped into three architecture types: lunar surface rendezvous (LSR), lunar orbit rendezvous (LOR), and libration point rendezvous (LPR). They compare the mission profiles in terms of ΔV with the constraint that the crew should not be required to wait longer than three times the period of the lunar phasing and rendezvous orbit to initiate a lunar departure. Condon and Wilson conclude that for a sustained, ambitious program of lunar exploration requiring global access to the lunar surface, a stay time greater than twenty-eight days, and a capability to abort at any time, LPR costs less ΔV than LOR missions. However, LSR missions require the lowest overall ΔV, because they do not require plane changes in lunar proximity or the additional ΔV associated with a stopover in EM-L1.

Joosent (2001) analyzes different space system architectures in the Earth's Neighborhood, defined as its gravitational sphere of influence with a radius of 1.5 million km. Joosent specifically discusses the benefits of using the EM-L1 point as staging area for reaching high latitude lunar landing sites. The author also explores different physical architecture designs and elements, using current space transportation and infrastructure elements (in particular the Space Shuttle and the International Space Station). For instance, he analyzes the advantages of using the ISS as a LEO staging facility, to decouple in time that the complex launch choreography that a long stay Moon mission will require. Joosent suggests that a single "gateway" located in the EM-L1, will centralize all human deep space operations by providing accessibility to the Moon, the Earth-Sun Lagrange points, and to Mars transfer orbits. He mentions that an "Omega" Space Station at the EM-L1 is a likely complement of the "Alpha" ISS already in place.

Engineers since before Apollo have theorized and analyzed different architectural possibilities for human Moon missions. Houbolt did not consider pre-positioning, because propulsion technologies such as electric propulsion were not feasible at that time. Current authors have the option of using existing infrastructure not available prior to Apollo as well as advancements in launch technologies, navigation and communication tools, and other technologies. As the nation's space program has matured, the technologies available to space architects have increased substantially. The number of combinations of possible elements has increased to thousands, and has made the process of architecture selection a question of rigorous mathematics, as well as space mission "common sense." This literature review summarizes necessary considerations for choosing a lunar baseline mission architecture and relevant trade studies.

4.2.3 Requirements and Assumptions

The requirements for the Moon missions are threefold. First, these missions must demonstrate the capability to transport humans safely from the Earth to the Moon. This includes launch, transfer, rendezvous, and landing. Second, these missions must demonstrate the capability to support humans in terms of life support, communication, and in-space and ground operations. Third, these missions must serve as a technology testbed for future Mars Missions and expand our knowledge of the Moon.

The purpose of the Short Stay Lunar Mission is to demonstrate the mission capability of going to the Moon. The purpose of the medium stay lunar mission is acquisition of scientific data and knowledge, and the purpose of the Extended Stay Lunar Mission is to demonstrate technologies for a mission to Mars including long duration habitation technologies.

In addition, it should be mentioned that a crew size of three is used on the Short and Medium missions to the Moon. This crew size was chosen because it is a smaller scale than the Mars missions, which involve crews of six. The rationale for using crew sizes of six is discussed later in Section 4.3.2.1. The Extended+ lunar mission has a crew of six to gain experience with such a large crew size in preparation for missions to Mars.

The baseline mission assumptions include:

- Pre-positioned modules will transit to the staging location using electric propulsion.
- Manned mission segments will use cryogenic chemical propulsion.
- Technology will be developed to store cryogenic chemical fuel for long durations without significant boil off.
- Radiation and low-gravity countermeasures will be developed by the time Extended Stay Lunar Missions are performed.
- Advanced EVA spacesuits are developed for medium and long duration missions.
- The ability to land humans and cargo on the far side of the Moon is developed in time for medium stay lunar missions.
- The ability to separately land humans and cargo within walking distance on the lunar surface will be developed prior to the Extended Stay Lunar Missions.

4.2.4 Operational View of Lunar Baseline Missions

4.2.4.1 Short Stay Mission

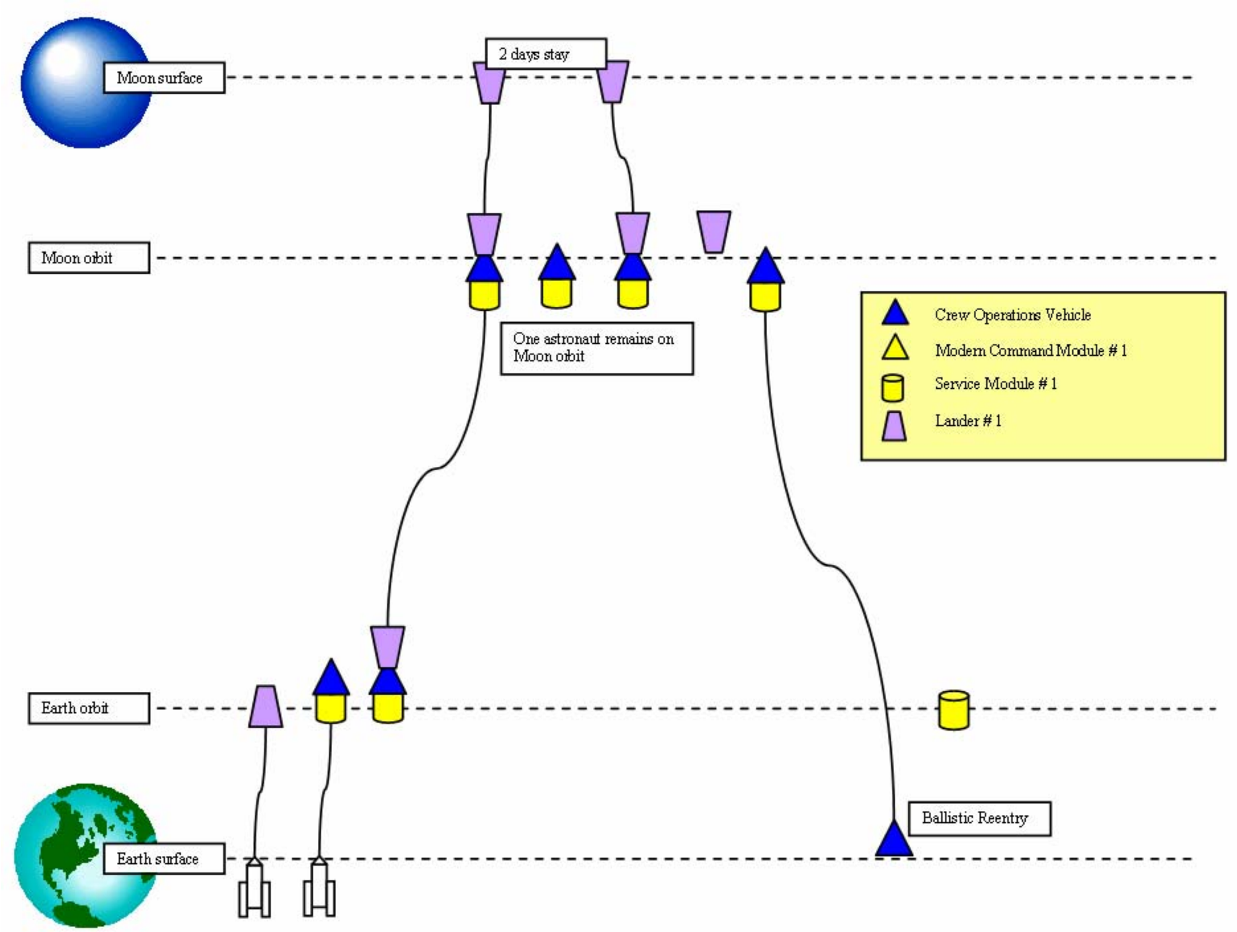


Figure 22: Operational view of Short Stay Lunar Mission

Short Stay Lunar Missions have a crew of three astronauts and two of these astronauts spend approximately two days on the lunar surface.

A Crew Operations Vehicle (COV) containing three astronauts is launched into low Earth orbit on a man-rated launch vehicle such as a man-rated heavy EELV. A Lunar Lander (LL) is launched into LEO separately using an STS-derived launch vehicle. Launch vehicles are chosen based on mass estimates documented in Appendix 9.3. The COV and LL dock in LEO and transit to lunar orbit together using cryogenic chemical propellant. Once in lunar orbit, two crewmembers transfer to the LL, undock from the COV and descend to an equatorial landing site on the near side of the Moon using cryogenic chemical propellant. One crewmember remains in the COV in lunar orbit.

The astronauts on the lunar surface will live in the LL for approximately two days and explore the landing site on foot; EVA will have minimal science capabilities since the purpose of this mission is to be a basic technology demonstration.

Upon the conclusion of the surface stay, the two astronauts ascend to lunar orbit in the LL using cryogenic chemical propellant and dock with the COV. One person is left in the COV as a safety measure for the basic technology demonstration; in case the LL fails to dock with the COV, the astronaut in the COV can manually maneuver to dock with the LL. Then, the astronauts transfer to the COV, undock with the LL, and initiate the return trip using cryogenic chemical propellant. The COV performs a ballistic re-entry, returning the astronauts to Earth.

4.2.4.2 Medium Stay Mission

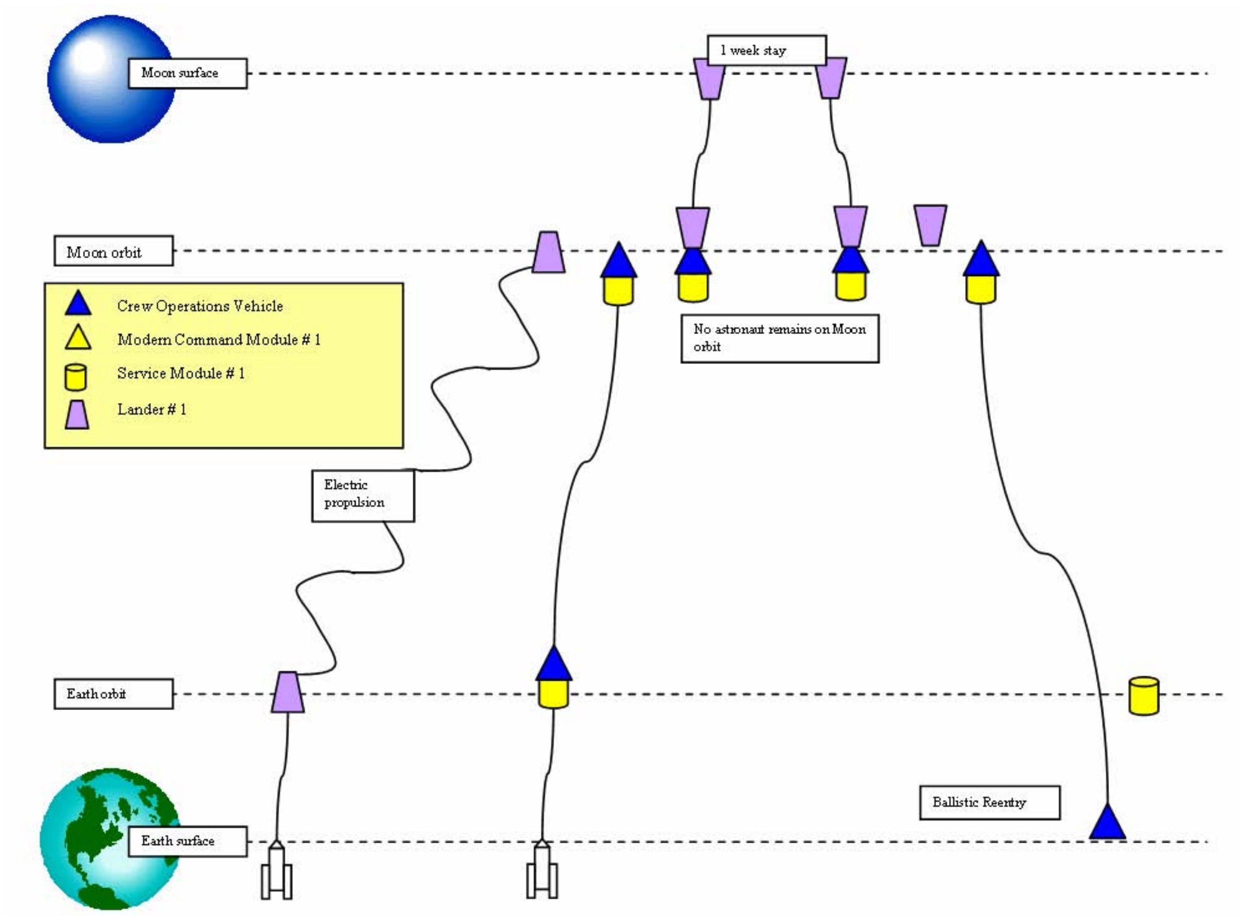


Figure 23: Operational view of Medium Stay Lunar Mission

Medium Stay Lunar Missions have a crew of three astronauts and use the same spacecraft forms as the Short Stay Lunar Missions. However, there are three differences between the Short and Medium Stay Missions: the LL is pre-positioned in lunar orbit using electric propulsion, all astronauts transfer to the LL to descend to the lunar surface, and the astronauts stay on the lunar surface for one week.

The justification for pre-positioning the Lunar Lander in lunar orbit before the arrival of the crew is to test the technology of pre-positioning essential mission cargo utilizing electric propulsion technology. While not providing a major mass savings for missions to the Moon, the capability of pre-positioning will allow for dramatic mass savings for missions to Mars. This is one of the ways the Moon can be used as a testbed for future missions to Mars.

First, a LL is launched into LEO alone using an STS-derived launch vehicle. Electric propulsion is then used to pre-position the LL in lunar orbit. Later, a COV containing the three astronauts is launched into low Earth orbit using an EELV (Delta IV Heavy). The COV transits to lunar orbit together using cryogenic chemical propellant. Once in lunar orbit, the COV docks with the pre-positioned LL, the three crew members transfer to the LL, undock from the COV and descend to non-equatorial landing sites on the near side of the Moon using cryogenic chemical propellant. No crewmembers remain in the COV in lunar orbit; it is assumed LL ascent was proven to be reliable during the Short Stay Lunar Missions.

The astronauts on the lunar surface will live in the LL for approximately one week and explore the landing site using an "open-air" rover to aid mobility within walking distance from the LL; EVA will have high science capabilities including research in some of the areas outlined in Section 3.2.1: Scientific Knowledge.

Upon the conclusion of the surface stay, the three astronauts ascend to lunar orbit in the LL using cryogenic chemical propellant and dock with the COV. The astronauts transfer the COV, undock from the LL, and initiate the return trip using cryogenic chemical propellant. The COV performs a ballistic re-entry, returning the astronauts to Earth.

4.2.4.3 Long Stay Lunar Mission

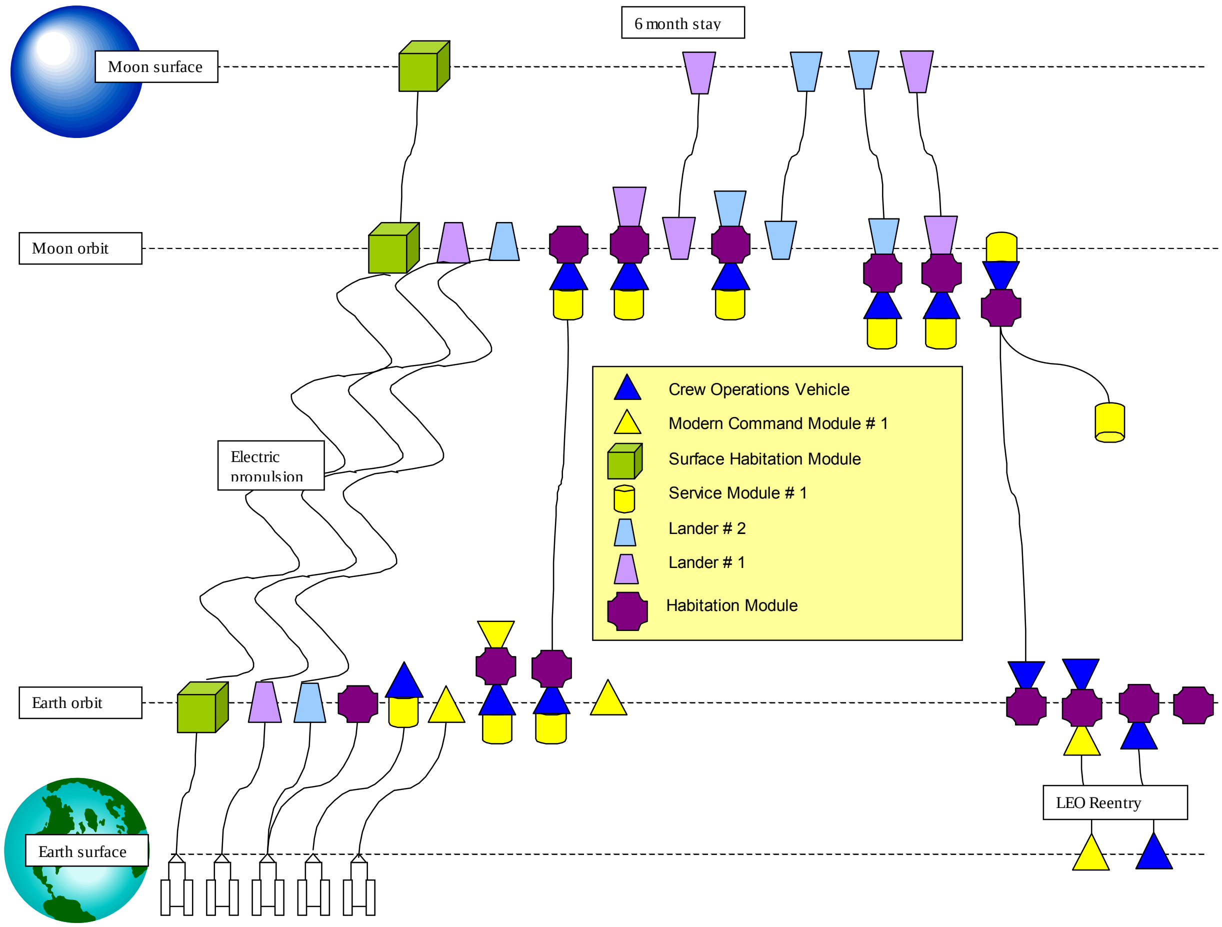


Figure 24: Operational view of Extended Stay Lunar Mission

Extended Stay Lunar Missions have a crew of six astronauts and require a pre-positioned surface habitat for the crew to live in for up to six months. A crew of six is used for this mission since future missions to Mars will have crew sizes of six.

It is important to mention that the increased complexity of the Extended Stay Lunar mission is essential for two major reasons. First, the increased complexity results from the multitude of technologies being tested in preparation for missions to Mars. Second, the increased complexity is used to return even greater amounts of knowledge back to Earth. This is made possible from a crew living on the Moon for an extended period of time in a location that is highly valuable from a knowledge point of view.

First, a surface habitation module (SHM) is launched into LEO possibly using two STS-derived launch vehicles. The SHM is then pre-positioned on the lunar surface, using electric propulsion for the transit to lunar orbit, and cryogenic chemical propulsion for the descent. Second, two Lunar Landers are launched into LEO separately using STS-derived launch vehicles. Electric propulsion is then used to pre-position the Lunar Landers in lunar orbit. One module of the habitation module (HM) is launched into LEO along with one of the Lunar Landers. Third, a COV containing three astronauts is launched into LEO using a man-rated launch vehicle such as a man-rated EELV. The same launch vehicle also contains a Modern Command Module (MCM). Each of these vehicles contains three astronauts. The COV docks with the HM and MCM, the crew of the MCM transfers to the HM, and the MCM undocks.

The docked COV and HM then transit to lunar orbit using cryogenic chemical propellant. In lunar orbit, the pre-positioned LL1 docks with the COV and HM, the three crewmembers transfer to the LL1, the LL1 undocks from the COV and descends to the pre-positioned SHM on the far side or pole of the Moon using cryogenic chemical propellant. Likewise, the second pre-positioned Lander, LL2 then docks with the COV and HM and transfers the crew to the SHM. No crewmembers remain in the COV in lunar orbit.

The astronauts on the lunar surface will transfer from the Lunar Landers to the SHM for a surface stay of approximately six months. The semi-permanent base allows for extensive science capabilities, possibly including but not limited to Moon-based observatories, greenhouse technology demonstrations for closed-loop life support, and nuclear power production. A habitable, pressurized rover for overnight field trips will aid surface mobility.

At the end of the surface stay the six astronauts ascend to lunar orbit in the two Lunar Landers. Each Lander individually docks with the COV and HM. After each docking, three astronauts transfer to the COV and then the COV and Landers undock. Finally, the astronauts initiate the return trip to Earth in the COV using cryogenic chemical propellant. The COV and HM use aerobraking to establish Earth orbit and then dock with the MCM. Three astronauts transfer to the MCM to return to Earth. The other three astronauts remain in the COV, undock from the HM, and return to Earth.

4.2.5 Commonality within Moon Missions

The lunar baseline missions were designed with a stepping-stone approach; complexity is added to missions in stages. For example, the SSLM requires one COV and one LL. While the MSLM requires the same forms as the SSLM, complexity is added in two ways: first, the LL is pre-positioned. Second the LL remains unmanned in lunar orbit. The ESLM requires another increase in complexity to prepare for future Mars missions. For these missions, three astronauts travel to LEO in the COV as in previous missions, however three other astronauts travel into LEO in a new form, the MCM. The transit to the Moon is carried out with the COV and another new form, the HM. Finally, for this mission, two LL are pre-positioned rather than one.

4.2.6 Discussion of Lunar Baseline Missions

4.2.6.1 Technologies to be Demonstrated for Future Mars Missions

This section describes Mars technology demonstrations carried out on Extended Stay Lunar Missions. The short and medium stay missions are not intended for technology demonstration, although by necessity certain technologies will be demonstrated on these missions.

4.2.6.1.1 Surface Operations

Lunar Lander (LL)

The Lander technologies that must be demonstrated on the Moon for the future Mars missions are:

- **Slow descent engines.** The Mars mission will initially use aerobraking and a parachute to decelerate to the surface. Once near the surface, descent engines are used to touch down; descent engines will be demonstrated on the Moon.
- **Ascent stages.** The Mars Lander will use a staged ascent, and the Lunar Lander will not; nevertheless, the "launch pad" technology used for ascent is similar for both Moon and Mars missions.
- **Reduced gravity.** The gravity constraints are different for the two planets; the Moon has 1/6-G and Mars has 3/8-G. The Moon will be used to demonstrate reduced gravity landings, but due to the gravity disparity, the Moon can only serve as a partial testbed.
- **Life support.** Both Landers will use the same life support systems.
- **Ability to land unmanned.** For rescue capabilities, an unmanned orbiting Lander should be able to land, unmanned, and rescue crewmembers on the lunar or Martian surface. This is not a direct requirement for the Martian missions, but may be used in future Mars missions. The capability is deemed important enough for the lunar missions alone, that it will be demonstrated for Mars, even if the unmanned Lander is ultimately not used for the Mars missions.

Surface Habitat Module (SHM)

The SHM technologies to be demonstrated on the Moon for the future Mars missions are:

- **Life support.** The technology for extended life support necessary for Mars surface operations will be demonstrated with a lunar surface habitat.
- **Pre-positioning.** The surface habitats will be pre-positioned on the lunar and Martian surfaces. They will need to achieve high accuracy in landing location and be able to land in a specified orientation.
- **Surface manipulation, docking.** If more than one surface habitat module is used, crewmembers will need to be able to connect the pre-positioned surface modules together once they reach the surface.
- **Communications.** The lunar communication network will use Ka-band communication and is mostly extensible to a Martian communication network. The primary difference between the two networks is the size of the antenna placed on the Moon and on Mars. The lunar communication network may also be used as a backup communication infrastructure for Mars missions in the event a Mars-Earth link is severed.

Rovers

A pressurized version of the rover will be used for Extended Stay missions to the Moon as well as Extended and Extended+ Mars missions. The rover technologies that must be demonstrated on the Moon for future Mars missions are:

- **Range.** The rover will have a round trip traveling distance capability of up to 500 km, as there is no constraint requiring that the astronauts be capable of walking back to the Surface Habitat. A round trip traveling distance of 500 km was determined by assuming a 400 hour sortie driving at a speed of 10 km/hr for three hours per day.
- **Habitability.** The rover that will be demonstrated on the Moon for Mars is a "habitable rover." It will have the capability of sustaining humans for up to 400 hours on a geologic traverse. The interior of the rover is a "shirtsleeve environment:" geologist-astronauts will be able to work outside the rover during the day and sleep in the rover without wearing their spacesuits at night.
- **Science capabilities.** The purpose of the extended geologic traverses will be to do field work. It is necessary that the rovers have basic tools to extract samples from the surface; however, analysis will be done in the SHM and upon return to Earth.

Other technology demonstrations:

- **Spacesuits.** It is highly desirable to develop a spacesuit that allows for increased mobility and locomotion to decrease the human energy output required for performing surface EVA tasks. This increases the ability of crewmembers to perform scientific analysis of the surface. This new spacesuit should have a longer useful life than the Apollo suit and should use advanced techniques in materials, gas pressure versus mechanical counter-pressure, etc. In order to increase useful life, the suit needs to address the problem of dust faced during Apollo. This problem with dust is also a problem for bringing the spacesuits in and out of the SHM; it may be necessary to devise a cleaning system for both the exterior and the interior of the suit. Small parts of the suit may be changed out for the different planets, but as much as possible the suit should be tested on the Moon for Mars, and capable of operations on both planets.
- **Tools.** All missions will require simple tools for sample collection. Missions may include robotic "helpers" for sample collection that may or may not interact with the crew. For extended missions on the Moon and Mars, resource extraction is desirable, and the missions may require industrial-like machinery that should be demonstrated on the Moon before being used on Mars.
- **Closed-loop life support.** Before an extended Mars mission is launched using closed-loop life support, it is necessary to demonstrate such capabilities on the Moon. These capabilities include the use of greenhouses, resource extraction, the manufacture of breathable air, and other capabilities. While the resources at the two destinations are clearly different, the capability of sustaining human life without external support should be demonstrated.

- **Aerobraking.** The lunar mission will demonstrate the technology of aerobraking into Earth's atmosphere. The modules that will aerobrake for the lunar mission are the COV and one HM module docked together.

4.2.6.1.2 Other Operations

Pre-positioning

Successful pre-positioning must be accomplished during lunar missions before a Mars mission is performed.

The components that must be pre-positioned in LEO are:
- Habitation Module (HM)
- Modern Command Module 1 (MCM1)

The components that must be pre-positioned in lunar or Martian orbit are:
- Landers (LL1 and/or LL2)

The component that must be pre-positioned on the lunar or Martian surface is:
- Surface Habitation Module (SHM – one or more)

Docking

Successful docking must be accomplished during lunar missions before a mission to Mars. Components that must dock for both lunar and martial missions are:
- HM and COV
- HM/COV and MCM1
- COV/HM and LL1, COV/HM and LL2

These components will need to be designed considering the requirements needed for docking in orbit. Maneuvering propulsion systems as well as guidance, navigation, and control capabilities may be required on board these modules to facilitate on-orbit docking.

Unmanned Orbiter

Both the lunar and Martian missions will involve an unmanned orbiting craft around the destination. For example, in the Extended lunar mission, the docked HM and COV will dock with the Lunar Lander, to transfer the crew, then the HM an COV must be able to orbit, unmanned, for six months or longer before the crew ascends and docks to transfer for the return trip. The capabilities of orbiting unmanned, and maintaining life support capabilities for the return trip, must be demonstrated.

4.2.7 Scientific and Resource Knowledge

The Moon is a natural laboratory for studying planetary and geologic processes, described more fully in Appendix 9.6. Among many others, we wish to explore and expand our knowledge in the areas of:

1. **Volcanism.** It is useful to study history of lunar basalts from the beginning of the Moon's volcanism to the most recent basalts found on the surface, to help determine the end of lunar volcanism.
2. **Volatiles,** including sources of water ice for possible in-situ resource production.
3. **KREEP basalts.** Discovered during the Apollo missions, the strange basaltic material, KREEP (Potassium, Rare Earth Elements, and Phosphorus) is the result of thermal differentiation of rare earth elements from liquid magma during cooling.
4. **The poles.** The poles may contain water ice; they are also interesting due to their consistent shade in certain places and small temperature variations.
5. **Stratigraphy.** Certain locations will help the study of crustal processes, allowing sampling of the most ancient materials.
6. **Seismology.** A complex seismic network (Neal *et al.*, 2003) would allow study of the lunar interior.
7. **Helium-3.** Resource extraction with power implications.

In addition to the scientific and resource knowledge potential of the Moon, it can also be used as testbed for future technologies such as in-situ propellant production (ISPP) or as the location of an astronomical observatory.

4.2.8 Knowledge Delivery Infrastructure

The knowledge delivery infrastructure will consist of two parts: the delivery of data in the form of bits and the delivery of samples from the planets surface. This section is specifically about the delivery of knowledge in the form of bits. This delivery of bits is referred to as the communication delivery system.

The same communication radio frequency has been selected for all lunar and Martian missions in order to provide an easily extensible system. The radio frequency that each of these missions will use is Ka-Band or 32 gigahertz. This frequency was selected because it can support a high data rate with comparably lower power than all lower frequency bands, and because the Deep Space Network (DSN) ground infrastructure will support it by the year 2007, while other higher frequency bands are not supported by the DSN. There is some concern about weather interference especially when communicating with Mars, however a Martian sand storm would prevent an X-band communication as it would a Ka-band communication, the differences would mainly lie in the moderate weather such as a cloudy day, or light dust storm in which case the Ka-bands data rate would be decreased.

For the Short Stay Lunar Missions, a direct link can be set up between the Lunar Lander and one DSN station. This would allow constant communication between Earth and the Moon throughout the mission. The data rate required for this mission would be approximately 0.07 megabits/sec and would require 0.01 Watts of constant power per transmission forty minutes. After the mission is completed the communication equipment that placed on the Moon will be left there for two reasons. First, if a future mission decides to return to the same location, placing new equipment at the destination will not be necessary. Second, in the unlikely case that mission communication equipment fails, the crew may have the option of traveling in a rover to a previous landing site to use the equipment left at that site.

For the Medium stay missions, the infrastructure is essentially the same as the Short mission. The main difference, however, is that the medium mission requires higher data rates and transmitting power. The transmission data rate will be 0.7 megabits/sec with a required constant power of 0.1W over the forty minutes of transmission time

The long stay missions require the ability to communicate between the far side of the Moon and Earth. The astronauts will communicate through one of four possible ways. For the first option, a communications relay satellite placed at the L4 point in the Earth Moon system. This infrastructure allows for a constant communication stream between the Earth and the Moon. Unfortunately, this option would only allow for communication for over 900km on the Moon's far side at its center or 30 degrees off the far side facing the satellite. The next option is to set up a relay satellite in low lunar orbit. This infrastructure is capable of covering most of the Moon, but a significant time delay for far side communications. The third option and involves placing a satellite in an orbit around the EM-L2 point. The coverage area provided by this communications satellite entirely covers the far side of the Moon and maintains nearly continuous communication with Earth. The shortcoming of this infrastructure lies in the difficulty of establishing orbit around EM-L2. The transmission data rate for the long stay missions will be 3.5 megabits/sec with a required constant power of 0.5W over the forty minute transmission time for near-side operations and a required constant power of 20W for far-side operations over the same transmission time. Please note that the equations used to determine these numbers are shown in Section 9.5.2.

4.3 Mars Baselines

4.3.1 Literature Review – A Brief History of Mars Mission Designs

Humans have dreamed of travel to other planets for centuries. In particular, Mars has been the focus of much interest. Arguably the first actual mission design for Mars was presented in 1952 by rocket engineer Dr. Wernher von Braun (History of Humans to Mars website, 2004). The plan was immense in scale and involved a fleet of interplanetary spaceships carrying large crews to Mars. Because of the corresponding cost requirement, von Braun's design it did not become reality. Although the Space Race between USA and the USSR fuelled interest in space travel, it did not provide enough motivation to propel such an expensive plan forward.

In 1989 political interest in Mars travel was revived with President Bush's call for a Space Exploration Initiative, and it resulted in a study known as "The 90 Day Report." This report resulted in a projected mission to Mars cost estimate of $450 billion. The mission design required assembly in orbit of a 1000 tonne spacecraft as well as a large orbiting facility to enable this assembly (History of Humans to Mars website, 2004). Again the high cost ruled out this humans-to-Mars initiative, however, at the same time, a radically low cost architecture was being designed.

The Mars Direct plan, developed by Robert Zubrin and David Baker at Martin Marietta Astronautics Company was a revolutionary paradigm shift that has had a significant impact on Mars mission design. The focus of this plan was to “live off the land” as much as possible, using the Martian atmosphere and soil to provide resources and in particular, using the atmosphere to enable in-situ propellant production (ISPP) of methane and oxygen for the return journey. The resulting mass required in LEO dropped dramatically, as did the projected cost that was reduced to approximately twenty to thirty billion dollars (Zubrin, 1996).

The Mars Direct approach was well received at NASA under a new administration and NASA performed its own study and corresponding mission design in the late 1990s. This study was known as the Design Reference Mission (DRM) (Hoffman, 1997). This study relied on Zubrin’s ideas including direct travel to the Martian surface and ISPP, although it was scaled up in some respects, having a crew size of 6 instead of 4. The cost estimate of such a mission was fifty billion dollars. The significant difference was a two-stage departure from Mars as opposed to the direct departure planned by Zubrin. The DRM had a Mars Ascent Vehicle (MAV) that rose from the surface of Mars via ISPP to rendezvous with an Earth Return Vehicle (ERV) for the trip back to Earth. The trans-Earth injection was predicated on the use of conventional fuel.

Other agencies and companies have proposed Mars mission architectures in recent years. One ESA scheme designed for the Aurora program attempts to extend Lunar activities towards a Mars mission design (Aurora, 2002). This architecture involves an interplanetary crew transport vehicle that travels between LEO and LMO, two Mars Excursion Modules that serve as ascent/descent vehicles between LMO and the Martian surface, and a Space Shuttle-like spacecraft that delivers the crew to the transport vehicle. This design encapsulates redundancy in that it provides a backup ascent/descent vehicle as well as a backup interplanetary transport vehicle pre-positioned in LMO and can be used for the crew’s journey home in case of failure of their primary vehicle.

EADS Space Transportation has outlined a number of options for short and extended-stay missions (Ransom, 2003) with short missions using Mars Orbit Rendezvous (MOR) and landing vehicles for crew ascent from and descent to the surface and with longer missions using ISPP and an architecture similar to NASA’s DRM. For the short stay missions, the main options are: pre-positioning, the use of ISPP to fuel ascent vehicles, and dual Lander architectures that allow the exploration of more than one site. Longer stay options include larger crew sizes, and fully established infrastructure.

Global Aerospace Corporation presents an interesting approach to infrastructure on Mars (Nock, 2001) with the use of “Astrotels” cycling between Earth and Mars, and “taxis” operating on rendezvous trajectories between these Astrotels and transport hubs such as orbiting facilities. This concept makes use of highly autonomous on-board systems to control the operations of the vehicles when they are not crewed. As well, it details the potential of harvesting LOX from the Moon as well as Phobos and using these resources to fuel interplanetary transfer.

Two main points emerge from this discussion. One is that ISPP is a relevant enabling technology with a good potential for decreasing costs and allowing Mars missions to be performed more frequently. A second point is the sense that there is not a single correct architecture. It is important to include options for scalability and extensibility from current capabilities to reduce cost and maintain flexibility in mission design; however, this can be accomplished in a number of different ways, each with its own benefits and penalties.

4.3.2 Mars Baseline

4.3.2.1 Introduction

The following Mars mission architectures are predicated on the idea of an evolution of mission scale and complexity starting with a rationale for using Phobos as a preliminary step. This methodology is closely aligned with the message presented in President Bush's New Vision for Space Exploration Program document. In addition, the mission architectures defined are reliant on successful demonstration of certain technologies and operations on the Moon. As our operations knowledge increases, utilization of complex technologies and other mission enhancements becomes feasible and these enhancements are incorporated into the mission profiles. The aim is that this evolution of missions in the space exploration system architecture will provide a means to create a sustainable transportation network and infrastructure for travel to Mars as well as to develop the capacity for missions to future destinations.

It is important at this point to mention how the crew size used in the following mission profiles was determined. A manned mission to Mars will require trip times well in excess of any mission thus far. This extended time in relative seclusion from all other personal contact suggests the need for a large crew for psychological considerations. However, this need must be balanced with mass requirements for each additional crewmember. During the mission, each astronaut will be required to perform a number of functions during the mission due to long communications delays require a higher level of autonomy. There appear to be five relevant technical fields required for exploration: mechanical engineer, electric engineer, geologist, life scientist, and physician. In addition to the primary specialty, each crewmember would need to be cross-trained in another mission critical field and be responsible for a variety of support tasks during the mission. However, with only a crew of five, a single loss of a crewmember, even temporarily in the event of sickness, could jeopardize the mission. Thus a crew of six is recommended. (Hoffman, 1997)

The plan for launching the required equipment needed for missions to Mars into LEO is mentioned in this section. In order to achieve the enormous mass required in LEO for a manned Mars mission, a number of launches must occur. For a Short Stay mission, a number of elements are pre-positioned in Mars orbit. These elements are transported using electric propulsion and thus must be launched approximately two years prior to crew departure. Using the STS-derived cargo launcher, three launches are required to place the propulsion for Earth return, the Martian Landers and the surface habitat in

LEO. Prior to the crew launch, the Habitation Module, which houses the crew during transit to and from Mars, is launched unmanned into LEO using two heavy lift cargo launch vehicles, such as an STS-derived launch vehicle. Finally, the crew travels to LEO in two separate vehicles: the Modern Command Module and the Crew Operations Vehicle. Both of these components can be launched by a single man-rated launch vehicle such as a man-rated heavy EELV. Thus, for a short stay mission, a total of five STS-derived cargo launches and a single man-rated EELV are required.

For an Extended Stay mission, an even larger mass is required to be pre-positioned at Mars due to a prolonged surface stay. Two years prior to crew departure, the pre-positioned elements using electric propulsion are launched via four STS-derived cargo launchers. Prior to launch, the Habitation Module is launched in two separate components on two STS-derived cargo launchers. Finally, the crew is launched in the crew operations vehicle and modern command module, using a single man-rated launch vehicle. Thus, a total of six STS-derived cargo launchers and a single man-rated launch vehicle are required.

4.3.2.2 Assumptions

Each mission defined below has some common assumptions built into the design. These assumptions are detailed here to emphasize commonality and avoid unnecessary repetition. For each mission to Mars a crew size of 6 and the use of chemical propulsion for crew transfer have been assumed. Electrical propulsion is used for pre-positioned elements such as cargo and Landers.

Since this report focuses on a strategy to achieve sustainability, certain specific mission details are not explicitly stated. These details are assumed. They include such items as radiation shielding and various features of the life support system such as air-regeneration facilities.

The general mission architecture is a Mars orbit rendezvous (MOR). The primary benefit of MOR is the decoupling of in-space transportation from descent and landing operations. MOR also allows flexibility with regards to the timing of landing, and this may prove to be important in the case of bad weather on Mars. Furthermore, because MOR involves landing from orbit as opposed to directly from the trans-Mars injection, landing precision can be improved. This allows architectures to include a pre-positioned surface component, which corresponds to a reduction in propellant mass.

4.3.2.3 Why Visit a Martian Moon as a Preliminary Step

The Martian moons, Phobos (Fear) and Deimos (Terror), were discovered in 1877 and have since been the subject of long-range observations made by Earth-based telescopes as well as by spacecraft traveling to Mars. Aside from their inherent scientific interest, these two Martian satellites could play a significant role in the context of Mars exploration and in the extensibility of NASA's space exploration initiative.

Figure 25: Phobos

In constructing a Mars exploration strategy, one of the principal objectives is to expand human knowledge and thus engage the public. For this reason, it is anticlimactic to send humans on a voyage of over a year in duration only to enter Mars orbit before returning to Earth. However, landing on Mars, which is likely to be achieved via aerobraking and rendezvous or docking operations, is one of the most dangerous parts of the mission. To mediate the risks of landing and surface habitation and obviate the requirement of a human mission to Martian orbit, it is suggested that the preliminary Mars mission have a surface component on one of the Martian satellites, Phobos or Deimos.

New technology must be tested as much as possible before the first manned mission to Mars, and this will be accomplished primarily by using the Moon as a testbed. Nevertheless, due to the inherent differences between lunar and Martian exploration missions, the first human mission to Mars will still be, in some respects, a venture into the unknown. A preliminary mission to Phobos or Deimos would allow NASA to decouple the test of the different components of the proposed space transportation system. Humans would be sent on a Martian trajectory and would therefore prove the capabilities of the transportation system and vehicles without having to perform the dangerous task of landing on Mars. Instead, the landing maneuver would be limited to a "docking" procedure in the micro gravity (1/1000g) environment of Phobos or Deimos, and a surface stay in a location without atmosphere similar to that of the Earth's Moon.

A preliminary mission to Phobos or Deimos would also lend itself to extensibility because it could be a stepping stone goal on the way to other exploration initiatives. A human mission to one of the Martian satellites would allow a telerobotic presence on Mars whereby the crew could control rovers on the Martian surface and respond to events of interest, taking advantage of the minimal communications delay. This could also aid in landing site certification. In addition, because these two moons may be captured asteroids or at least very similar in composition, a mission to these bodies could prepare NASA for a future asteroid rendezvous mission while taking advantage of the moons' predictable orbits (as determined precisely by the 1988 Soviet Phobos mission (Brat, 2001). Looking further ahead, a mission to one of the Martian moons may help to provide NASA with some of the operational knowledge necessary for future exploration of the moons of Jupiter.

Recent initiatives by the Planetary and Space Sciences Research Institute at the Open University in the UK, and past attempts by the Russian space program underscore the interest in Phobos and Deimos, both from a planetary science and evolution point of view and from a resources perspective. Scientific objectives include: distinguishing between origins models of the moons, establishing links between the moons and known asteroid types, and studying the mutual effects Mars and its moons have had on each other (Ball, 2004). There is also an interest in the Martian moons from a resources point of view, and it has been speculated that the moons may have frozen volatiles such as water ice in their interiors that could be exploited for in-situ propellant production (Ball, 2004).

Finally, a successful mission to one of the Martian moons would build public confidence and spark interest in the space exploration problem. This would enable NASA to gain support for subsequent missions to Mars.

4.3.2.4 Phobos Mission Design

Phobos has been chosen as the de facto destination between the two Martian satellites since it is larger and closer to Mars. A mission to Phobos would require remote sensing capabilities to survey the surface for potential landing sites during initial flybys. A landing capability would also be required to allow surface exploration, and dedicated scientific payloads would be used to collect samples and perform geological testing.

Since a crew would be sent to Phobos, a high priority would be to minimize transfer time and hence the trans-Mars injection would probably follow the opposition-class profile, making use of a Venus flyby. The trajectory to Phobos can be broken into seven sections as follows with various propulsive requirements (Brat, 2001):

- Vehicle launch and trans-Mars injection (TMI)
- Mid-course corrections and plane change in Earth-to-Mars trajectory
- Capture at Mars placing the vehicle in a near-Phobos "walking orbit"
- Rendezvous (circularization) with Phobos
- Station keeping at Phobos
- Lander descent and roving operations
- Attitude control during all of the above phases

Of these maneuvers, only the TMI and capture at Mars would have significant ΔV requirements, on the order of 4 km/s and 2.7 km/s, respectively. Energy requirements could be reduced via aerobraking by approximately 1.2 km/s (Brat, 2001), but this would add to mission complexity and could be maintained as an option if a test of aerobraking is required. The return trajectory requires a trans-Earth injection and ballistic entry at Earth's atmosphere.

4.3.2.5 Short-Stay Mission

The short-stay mission is the shortest Mars mission possible in terms of total mission duration; it is composed of approximately 600 days transit time and 60 days surface stay (Walberg, 1993). The crew travels to Mars via an opposition class free-return trajectory with a Venus fly-by in the Mars Transfer Vehicle (MTV), which is composed of a Habitation Module (HM) and a Crew Operations Vehicle (COV). Upon arrival at Mars, the MTV aerocaptures into Martian orbit and performs a rendezvous with two pre-positioned Mars landing vehicles (ML1 and ML2). Three crewmembers descend to the Martian surface in each Lander. This allows flexible timing for each landing, with the second being contingent on the success of the first. The landing is achieved using a heat shield for atmospheric entry after which parachutes are deployed to slow the spacecraft. The final stage of the landing is a powered touchdown that gives the crew as much control as possible over the landing so as to minimize risk of damage to the landing vehicle.

The crew remains on the surface for approximately 60 days. During this time, the crew lives in a pre-positioned surface habitat that could be extended by an inflatable module if more volume is required. At the end of the surface stay, the crew returns to Mars orbit in the two landing modules and docks with the MTV. The MTV docks with the pre-positioned return propellant module (SM2) and executes a trans-Earth injection maneuver. Entry back into low Earth orbit (LEO) is achieved via aerocapture, and the MTV docks with the Modern Command Modules (MCM) allowing half of the crew to transfer into this Earth re-entry vehicle. The other half of the crew returns to Earth's surface in the COV. An operational overview of the short-stay mission to Mars is provided in Figure 26.

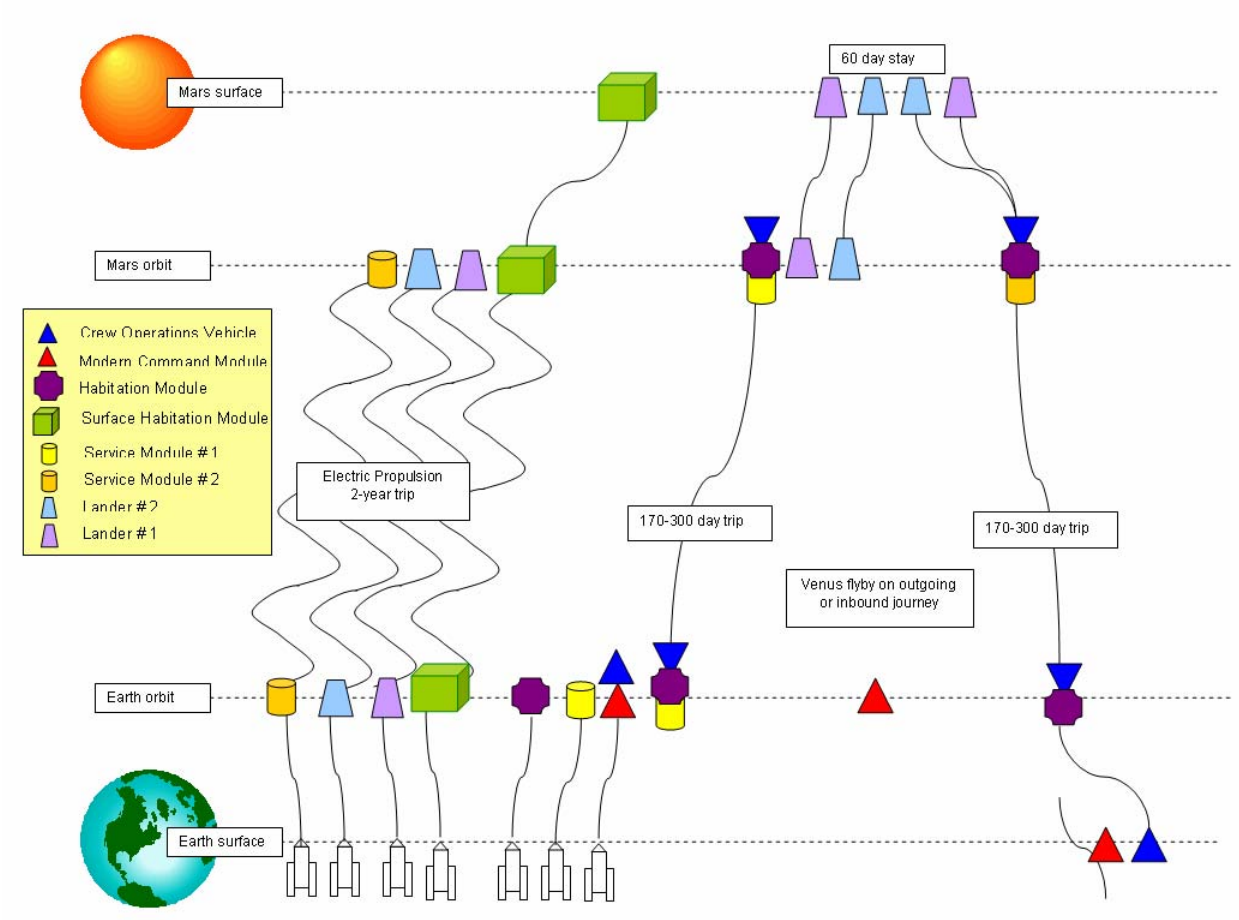


Figure 26: Short stay mission to Mars

During the surface stay, the crew will explore the Martian surface within close range of their landing site using EVA suits in conjunction with an un-pressurized rover. They will perform activities such as searching for water and life, collecting samples, imaging the surface, and recording weather phenomena. Requirements will include life support facilities in the surface habitat, radiation shielding, and consumables. An adequate communications network will also be required for this mission as discussed in further detail in Section 4.3.4.

To facilitate the short-stay mission, pre-positioning of mission elements is employed. In addition to the pre-positioned Martian Landers, surface habitat, and return fuel; surface equipment, such as an un-pressurized rover and scientific payloads will be pre-positioned on the surface. Pre-positioning requires a farsighted approach to mission planning, but this should be well within the scope of the space transportation system outlined here, and as discussed in Section 6.4.3.1.4.1, the benefits of pre-positioning justify the added constraints on mission timeline. Because the pre-positioned elements will make use of electric propulsion, they will have to be sent at the launch window approximately two years prior to that of the crew departure.

The description above outlines the basic structure of the short-stay mission; however, there are several options available for such a mission that would improve prospects of extensibility by testing enabling technologies for a subsequent mission. The first, most important option is to begin verifying and testing an in-situ propellant production (ISPP) plant. An ISPP plant would allow for fuel production on the Martian surface as described in further detail in Section 6.4.3.3.2.1. This option would require a pre-positioned propellant production module, a small nuclear power plant, and a hydrogen fuel stock for a Sabatier process. During the surface stay, the crew could check the functionality of the ISPP system and determine if it is reliable for propellant production. In addition, the propellant could be used to augment power for surface systems, such as the rover vehicle. This option would facilitate an extensible and sustainable infrastructure for further Mars missions.

Another option is to provide extended surface mobility. This could be accomplished by providing one or more remote-controlled rovers to assist in surface exploration. These rovers could act as “scouts” and be dispatched to areas of potential interest to determine which site holds the most interest for a follow-up visit by the crew.

4.3.2.6 Extended-Stay Mission

The extended-stay mission offers the advantage of a longer surface stay with only a small increase in the total mission duration over the short-stay mission. It is also less complex from a trajectory perspective since it does not require a Venus flyby to provide the required ΔV. Instead, for this mission, the crew travels to Mars in the MTV via a fast transfer conjunction class trajectory. In most respects the mission architecture is taken directly from the short-stay mission. Assuming the short-stay mission precedes the extended-stay, the required operational knowledge to carry out the mission should be almost complete. An operational view is presented in Figure 27.

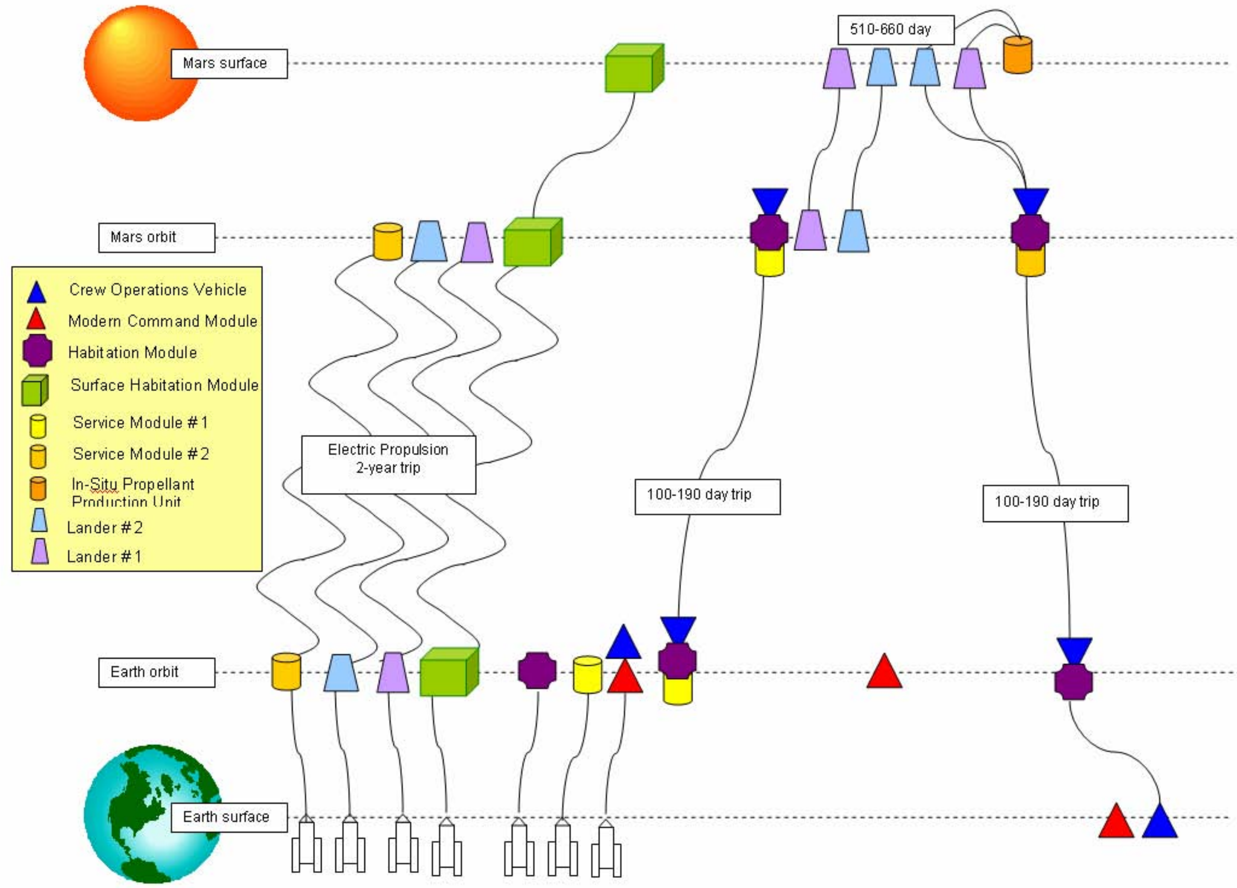


Figure 27: Extended stay mission to Mars

One of the main differences between the extended and short-stay missions is the ascent to Mars orbit in the landing modules. Assuming the option of ISPP during a short-stay mission is employed, and successfully demonstrated, the ascent propellant for the Landers will be provided in-situ using the Sabatier process. Details about the Sabatier process are available in Section 6.4.3.3.2.1. This will change the mission requirements slightly in that an ISPP unit consisting of hydrogen feedstock, a chemical plant and a nuclear power plant will have to be sent to Mars two years prior to the departure of the crew, and it will necessarily have autonomous communications ability to relay its status to ground control centers on Earth. This way, the crew will be assured that the propellant required for their ascent from Mars has been generated prior to their departure. In addition, since the Sabatier process produces methane/oxygen fuel, the landing module will have to be able to dock with the required propellant tanks and engine. All of these components may have been pre-positioned prior to crew departure or may be attached to the landing module itself. Using ISPP to fuel the Landers' ascent will help to reduce the mass of pre-positioned cargo and thus save mass or enable the transportation of other science payloads. A further advantage of using ISPP is that the fuel produced may also be used to power combustion engine rovers and possibly life support systems as well. Finally, the functional test of ISPP at this stage will act as a stepping-stone towards the eventual goal of using ISPP to fuel the entire return journey to Earth.

Another principal distinction between extended and short-stay missions is the length of surface stay. For an extended-stay, the crew surface habitation module will need to be considerably larger than that of a short-stay mission. To this end, an inflatable module or an additional habitation module will be pre-positioned

Besides the obvious extensive life support requirements for a mission of this duration, the crew will require an extended means of exploration equipment for this mission. Along with EVA suits, a transport vehicle, open or pressurized, will be pre-positioned, to enhance surface mobility. An open, un-pressurized, rover is limited to ranges of 10km, such that the crew is always within walking distance of the surface habitat. However, this safety requirement highly constrains the amount of exploration that can be accomplished during this long duration mission. Therefore, although not a requirement, a pressurized rover capable of ranges on the order of 500km is a recommended option for the long-stay mission. This will allow the crew to explore a large area, searching for water and life, collecting samples to return to Earth, and taking various measurements.

In addition to the large-scale physical exploration of the Martian surface, the crew will have the opportunity to conduct more advanced scientific experiments including experiments that may require a longer duration, such as small-scale agriculture development. The construction of an inflatable greenhouse prototype is one option for the extended-stay mission. This could supplement the crew's food supply for both the surface stay and Earth return trip.

Another important option to consider would be to include a pressurized surface transportation vehicle to satisfy the surface mobility requirement. This pressurized vehicle, with a mass of approximately 6000kg, would allow for a "shirtsleeve" environment for up to three astronauts on sorties on the order of 500 km and 400 hours duration. For safety reasons, two pressurized rovers would have to be sent to Mars to ensure rescue capability. It is likely that these vehicles would be sent separately from the crew via a direct launch in conjunction with other cargo such as extra food supplies or a power plant.

Further options that would facilitate movement towards a semi-permanent infrastructure include the development of ISPP to a high enough level of production and reliability, such that it would be used to supply the return fuel for the MTV in subsequent missions. Also, a drilling device that could be transported as pre-positioned cargo to tap into subsurface aquifers if it was determined in the short stay or robotic precursor missions that underground water may exist. Since surface life support systems require a substantial power supply, a nuclear power plant could be pre-positioned. Finally, a closed loop life-support system with bioregenerative components could be tested. Although this technology would not be implemented until its reliability is ensured, such a technology would help a self-sustained presence on Mars become more feasible.

A comparison of the ΔV requirements for the Short and Extended Stay missions to Mars is shown in the following tables. Table 5 shows the ΔV requirements for the Short and

Extended Stay missions to Mars assuming aerobraking and parachutes are not used. Table 6, on the other hand, assumes parachutes can be used in the missions.

Table 5: ΔV requirements assuming parachutes and aerobraking not used

Orbital Maneuvers	Short Mission ΔV (m/s)	Extended Mission ΔV (m/s)
Trans-Mars Injection	4098	4217
Mars Orbit Insertion	3278	3050
Mars Surface Descent	741	741
Mars Surface Ascent	4140	4140
Trans-Earth Injection	1415	2180
Earth Orbit Insertion	2774	5485
Total ΔV (m/s)	16446	19813

Table 6: ΔV requirements assuming parachutes used

Orbital Maneuvers	Short Mission ΔV (m/s)	Extended Mission ΔV (m/s)
Trans-Mars Injection	4098	4217
Mars Orbit Insertion	3278	3050
Mars Surface Descent	111	111
Mars Surface Ascent	4140	4140
Trans-Earth Injection	1415	2180
Earth Orbit Insertion	2774	5485
Total ΔV (m/s)	15816	19183

If it aerobraking is assumed for use in one of the Mars missions, the ΔV numbers in Table 6 for the stages of the mission in which aerobraking is used, such as Earth orbit insertion, are ignored for propulsive engine burn requirements. Instead, the initial mass required for the aerobraking maneuver is assumed to be 15% of the payload mass (Walberg, 1993).

4.3.2.7 Extended-Stay Mission with Infrastructure

The final mission class is the extended-stay mission with the development of infrastructure. The idea behind this architecture is that if, after previous short and extended-stay missions, Mars remains an interesting destination either from a science or operations perspective. Another possibility is that Mars can serve as a testing ground for the next exploration target. Subsequent Mars missions will then develop infrastructure to facilitate surface stays and exploration as well as to minimize mass that has to be transported from Earth.

In this case, the aim of the mission is to use in-situ resources as much as possible. These resources are used to provide return fuel, generate power, develop sustainable agriculture, and enable closed loop life support. Missions to Mars will take place more frequently, possibly with one crew traveling at every launch opportunity

The initial architecture will follow the proven MOR scheme for a long-stay conjunction class mission, but assuming previous attempts at ISPP generation and fuelled ascent have been successful, it is likely that the architecture design will become more similar to Mars Direct (Zubrin, 1996). The eventual outcome of this transition would be that the MTV would travel directly to the surface of Mars without orbital rendezvous and ascend from the Martian surface using ISPP fuel directly into a trans-Earth injection. The result of a move towards this architecture would be a significant reduction in IMLEO.

For these extended-stay missions, pressurized transport vehicles will be pre-positioned allowing the crew to have significant surface mobility on the order of 500km. As previously mentioned, two pressurized rovers will be necessary to provide a rescue capability. The crew will conduct science experiments as described above. In addition, the scientific payloads will be chosen to explore areas that have been proven to be the most interesting in previous missions as well as any other new areas of interest. The ECLSS will be designed to achieve as close to 100% closure as possible, and the crew will derive most of their power from ISPP. Agricultural facilities such as inflatable greenhouses will be installed to provide or supplement the crew's food supply. The crew habitat will take the form of multiple inflatable modules as well as pre-positioned Habitat modules sent direct from Earth.

The extended-stay mission with infrastructure will provide a testbed for further exploration technology development, and it is also possible that the ISPP facilities will allow Mars to serve as a way station for vehicles traveling to more remote destinations.

4.3.3 Commonality

The four missions described above, from a preliminary mission to Phobos through to a mission for an extended stay on the Martian surface, are designed for logical evolution from one to the next. With this in mind, there is significant commonality between the missions. In all cases, one mission provides a testbed for the technology that is required for subsequent missions. A Phobos mission presents the opportunity to test the in-space transportation system that will be used to reach Mars during the short-stay mission. Furthermore, the short-stay mission allows the test of aerobraking, landing, surface habitation, and in-situ propellant production that will all be used during the extended-stay mission. Finally, during the extended stay mission, more advanced technologies such as agriculture can be tested in preparation for the next mission with infrastructure. The missions to Mars as described above would also be useful preparation for further destinations such as asteroids or Jovian moons.

4.3.4 Knowledge Delivery Infrastructure

For the short-stay sized Mars missions a direct link can be set up between the Mars Lander and one of the Earth's Deep Space Network (DSN) stations. This would allow semi-frequent communication between Earth and Mars throughout the entire mission. The data rate required for this mission would be 1 gigabit/day and would require 8 watts of power per transmission with a transmission data rate of 0.04 megabits/sec. After the mission is completed the communication equipment that was landed on Mars will be left there for two reasons, one if a future mission decides to use that spot as a landing or settlement site then they won't have to bring their own equipment, and in the unlikely case that another future mission communication equipment fails the crew will have the option of traveling in a rover to the old site and using its equipment.

The extended-stay missions require the ability to communicate with much greater data rate and thus it might be necessary to create a relay satellite around Mars. There are two realistic options for the location of this satellite. The satellite could be placed in a Geostationary Martian orbit (GMO) around the landing site; the advantage of a GMO satellite is that it increases the time that the astronauts can communicate with the Earth, the disadvantage is that it can only really be set up for one portion of the planet. The other option is to position a satellite at the Earth-Mars L1 point thus decreasing the power required to send large communication streams to the Earth; unfortunately this would not add any extra time that the mission could communicate with the ground. As with the Moon missions there is an option in the case of emergencies to communicate with the Moon and use it as a relay station. The daily data rate for a large sized mission would be 10 gigabits/day, the transmission data rate would be 0.4 megabits/sec and the transmission power required would be approximately 80 W. Please note that the equations used to determine the numbers used in this section are in Section 9.5.2.

4.4. Transport

The purpose of this section is to highlight the commonality between the Moon and Mars missions within the framework of a baseline set of forms that will each perform certain functions in order to complete a baseline mission. The forms used to accomplish a baseline mission are described below.

4.4.1. Selection of Forms

The baseline mission suggests forms for specific functions. The purpose of developing a baseline mission is twofold. Firstly, with forms specified, the missions to the Moon and Mars can define functional requirements that are desired of each form. Secondly, specification of forms creates a framework for comparing missions and evaluating the attributes of each mission.

The method of selecting this baseline mission was based on the first draft of requirements for a Mars mission. A Mars mission was selected as a starting point and analyzed considering the impact of each mission decision on the extensibility of this architecture from a Moon mission. The architecture was designed considering reconfigurability, adaptability and extensibility of a Moon mission to a Mars mission. Each decision required specification of the baseline architecture and evaluation based on its influence on a Short, Extended or Extended+, Mars or Moon mission. Once each form was selected, specific details of the functions for each form were specified for the Mars and Moon missions.

4.4.2. Summary of Baseline Forms

A baseline mission was developed to create a complete list of forms required to accomplish the Mars and Moon missions. The baseline mission includes assumptions based on current technology limitations, safety concerns, and policy requirements to a lesser extent. These assumptions influence the technology of each form, but not the functional requirements of the forms. The focus of the baseline mission was not to develop another detailed description and analysis of the forms required for an extensible Mars mission. Instead, a framework is created that provides a method of comparing the functional requirements for each of the Mars and Moon missions. Details of the forms are provided in Section 6.4.3.1.

A schematic representation of the baseline mission is shown visually in
Figure 28 and can be summarized as,

- Pre-position in LEO (HM, SM1)
- Pre-position in Destination orbit (SM2, ML1, ML2) (LL1, LL2 for Moon mission)
- Launch MCM and COV into LEO on man-rated launcher.
- Sequentially dock MCM and COV to HM (undock MCM and leave in LEO)
- Dock SM1 to COV/HM (This combination is known as the MTV)
- Transit to destination, undock SM1
- Burn into Lunar orbit or aerocapture into Mars orbit (mission specific)
- ML1&ML2 (LL1&LL2 for Moon mission) sequentially dock and undock with MTV
- Crew ascends to destination surface and ascends to orbit
- ML1&ML2 (LL1&LL2 for Moon mission) dock and transfer crew to MTV (ML1&ML2 stay in orbit)
- Dock SM2 to MTV
- Transit to Earth, undock SM2
- Aerocapture at Earth
- Dock MCM to HM, transferring crew of three
- Remaining crew of three enter COV
- COV & MCM Earth EDL

Many decisions regarding sustainability and extensibility were made in determining this baseline mission. As such, a number of trade studies were considered and presented in later sections of this report (see Chapter 6).

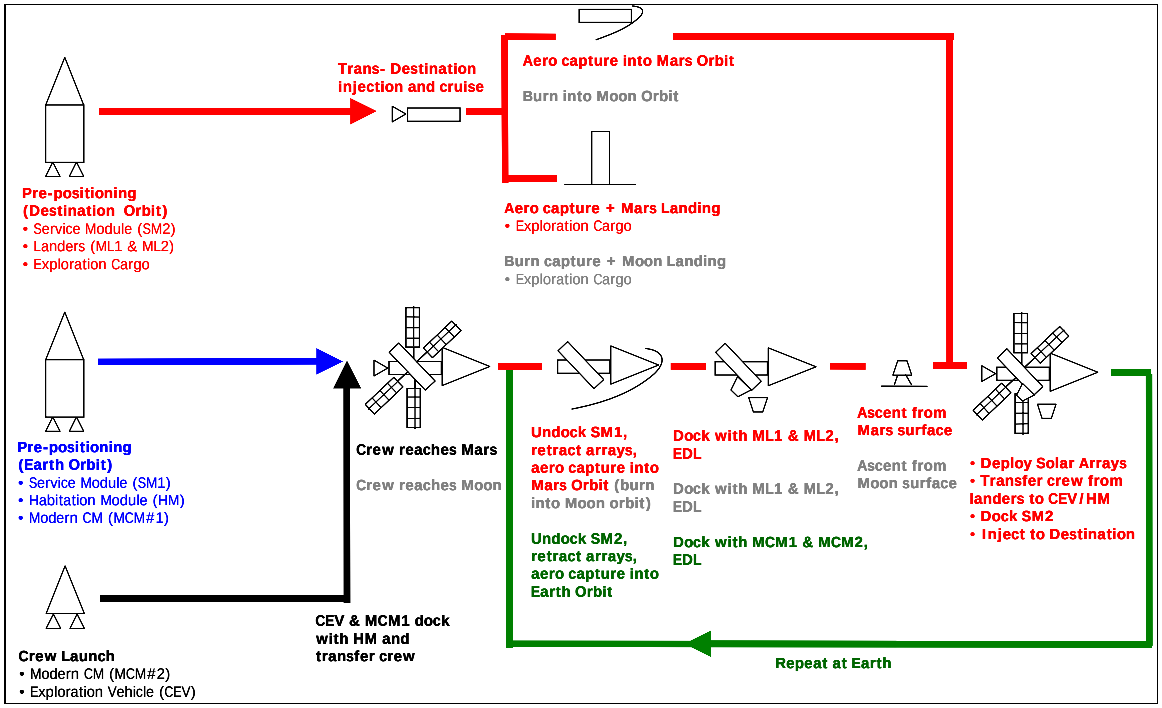


Figure 28: Schematic representation of the Moon and Mars Baseline missions

4.4.2.1. Pre-position in Mars or Moon Orbit

Pre-positioning crew habitation modules, fuel for return transport, Landers, and return capsules allow the mass of the module transporting the crew to the Moon or Mars to be reduced. Although pre-positioning modules reduces the injected mass required for a mission, additional difficulties are adopted. These difficulties include on-orbit docking capability and increased complexity in pre-positioned module functionality. Safety concerns are also raised because pre-positioning return fuel for the first mission to Mars will require a high level of confidence in the technology. As such, carrying return fuel for the first Mars mission increases confidence while providing a method of developing a stockpile of fuel, which could be used on Mars or as a safety measure for future Mars missions. A detailed description and justification for pre-positioning is presented in Section 6.4.3.1.4.

Based on the assumption of pre-positioning mass in Earth and the destination orbit, the following modules are pre-positioned in Martian orbit and correspondingly for Lunar orbit for the Moon mission:

1. *Service Module #2 (SM2)* – As an assumed function, this module must be capable of providing propulsion for transiting the crew from Martian (or lunar) orbit to Earth orbit.
2. *Two Lunar or Mars Landers (ML1 & ML2 or LL1 & LL2)* – Two identical Apollo type Landers (slightly different forms for Moon and Mars), each capable of transporting three crewmembers from orbit to the surface. Functionally, only

minimal redesign is required for Lunar landing capabilities (see Section 6.4.3.1.6.). As such, the Moon can be a “true testbed” for the Lander technology. An option also exists to re-use the Landers for the Moon mission because there is no heat shield requirement for a lunar landing (see Section 6.4.3.1.6.) Both Landers must have propulsive capabilities in addition to the requirements for pre-positioning. This extra propulsion redundancy allows either the Lander to dock with the HM or the HM to dock with the Lander in Mars orbit (or Lunar orbit). Although there is increased mass associated with two three crew Landers as compared to one six crew Lander,

- This technology is modular and extensible to a Mars mission (6 crewmembers) from a Moon mission (3 crewmembers).
- Increased reliability and flexibility observed because if one Lander were deemed unusable, the second Lander could still perform the desired mission objectives.

A detailed trade study that discusses the above can be found in Section 6.4.3.1.6.2.

4.4.2.2. Pre-position Cargo on the surface of Mars or the Moon

Similarly, pre-positioning all of the necessary cargo for Mars and Moon surface habitation, on the surface, in a separate mission, reduces the payload mass that must be launched from Earth or LEO. For surface habitation, if the environments were deemed similar in design requirements, an option may be to use a duplicate of the habitation module on the surface of Mars. Details of the surface habitation modules are given in Appendix 9.2.

4.4.2.3. Pre-position in Earth Orbit

4.4.2.2.1. Non human-rated launches

The following items will also be pre-positioned in LEO.

1. *Habitation Module (HM)* – This module will be launched in pieces and assembled in LEO, allowing the overall volume to not be limited by the minimum launch volume requirements. Since there are many modules, none, one or many can be used for each mission, depending on the mission requirements for duration and crew size. This module must have propulsion capabilities to perform docking maneuvers in both Earth and Mars (or Lunar) orbit.

2. *Service Module #1 (SM1)* – This module must be capable of providing propulsion for transiting the crew from Earth orbit to Martian orbit or Lunar Orbit (Note that SM2 was used for (Mars or Moon) to Earth transit). This module will require minimal redesign to be truly extended for use on a Mars mission (the same interface and platform could be used).

Since no crews are launched into LEO during the pre-positioning phase, all launches can take place on non-man-rated launchers. This serves to increase launch flexibility and lower overall cost.

4.4.2.2.2. Human-rated Launches

By using a pre-positioning approach, only a single man-rated launch is required to deliver the six crewmembers (or three crewmembers for smaller Moon missions) to the Habitation Module. The launched modules include:

1. *Modern Command Module (MCM)* – The Modern Command Module is discussed in Section 6.4.3.1.1.1. This module will dock with the Habitation Module and return crew back to Earth from LEO at the end of the mission. This module is also capable of docking and delivering the crew of three to the Habitation Module (HM). Note that this module will not travel to the destination with the Habitation module, but only serves as a means of transporting the crew to and from Earth (first and final phases of the mission). Since some of the defined Moon missions specify a crew of three instead of a crew of 6, a three-crewmember transfer vehicle was chosen to reduce the mass launched on a human rated launcher for the three crewmember Moon missions. This was also done to ensure that the Earth to LEO and LEO to Earth transfer vehicle was not over designed for a three-crewmember Moon mission. This can be done because the mission scales via the number of Habitation modules (see Section 6.4.3.1.3.) used and not the size of the LEO crew transport vehicle. A detailed crew vehicle scaling analysis and justification for this decision are presented in Appendix 9.1.

2. *Crew Operations Vehicle (COV)* – This module docks and delivers the remaining three crewmembers to the Habitation Module in the same manner as MCM. However, this module does not remain in LEO during the mission. Instead, this module travels with or without the HM to the destination. Functionally, this vehicle must provide all the necessary functions of a crew transport module (i.e. docking capabilities, GNC, radiation protection, thermal control, ECLSS, attitude control, communication equipment, etc.). The reasons for choosing an additional module that travels with the HM are twofold:
    - Having a COV separates the functions for crew transport, allowing the mission to scale down to transporting a crew in only the COV to the Moon for a Short Moon mission, to scaling up to transporting a crew in the COV & HM to Mars for the Extended Mars mission.
    - Having the smallest mass and volume re-entering Earth at the end of the mission is beneficial to reducing the mass penalty of the mission. This is further described in Section 6.4.3.1.1.1.

All modules considered common interfaces and serve specific functions. Details of the launch logistics, timing and forms for both the human rated and non-human rated launches are presented in Section 6.4.2.2.

4.4.2.4. Earth Orbit, Injection and Transfer

Now that all of the modules have been pre-positioned, SM1 docks with the HM in Earth orbit. To perform this docking, it is necessary that SM1 and HM both have propulsion capabilities for added redundancy. After this docking is complete, the COV docks with the HM and SM1. These three modules comprise the entire crewed vehicle that travel to Mars or the Moon, known as the Moon/Mars Transfer Vehicle (MTV) (see Figure 29).

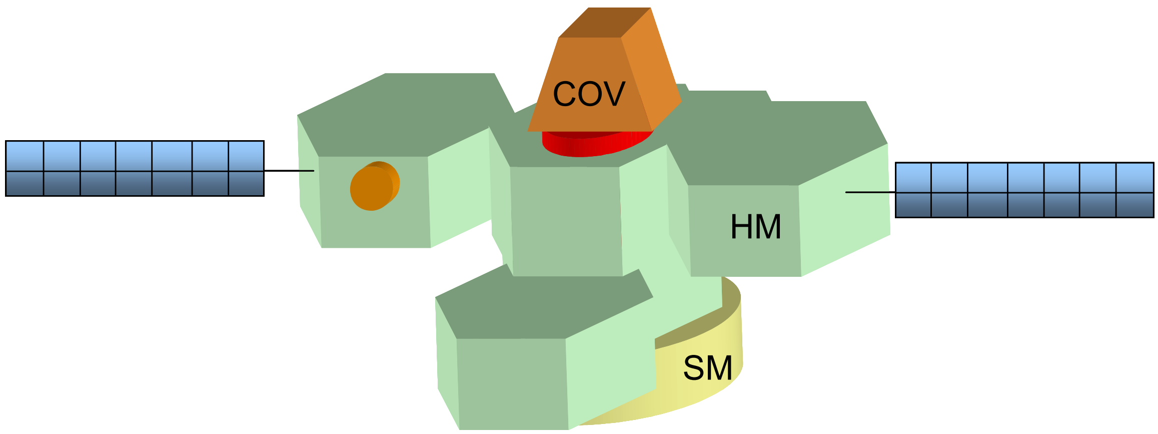


Figure 29: Mars/Moon Transfer Vehicle (MTV)

4.4.2.5. Entering Destination Orbit

The MTV (COV/HM/SM1) attains Martian orbit by aerocapture. This requires the use of the COV heat shield and additional protection for the HM in contact with the atmosphere. It is assumed that the shielding could be modular (i.e. detachable) to ensure HM extensibility remains intact. Prior to entering Martian or Lunar orbit, SM1 undocks from the MTV, reducing the aerocapture mass (at Mars) or the mass of fuel required to enter Lunar Orbit.

4.4.2.6. Descending and Ascending

Two pre-positioned Landers dock sequentially (ML1 & ML2 or LL1 & LL2), requiring the HM to have only one docking interface for the Landers. Three crewmembers transfer into each Lander, and then descend to the surface. In a similar manner to the Apollo Missions, the Landers ascend to Mars/Lunar orbit. Details of the Lander are presented in Section 6.4.3.1.6.

4.4.2.7. Orbit, Injection and Transfer to Earth Orbit

Once the Landers dock sequentially with the MTV, a second Service Module (SM2) consisting of the necessary propulsion to return the crew to Earth docks with the MTV in orbit. These three modules (COV/HM/SM2) comprise the entire vehicle that travels back to Earth. The vehicle is identical to the MTV that transited from Earth to Mars (or Earth to Moon) with the exception of SM2 now being used in place of SM1. Note that the Landers do not return to Earth with the crew for the Mars mission, but could possibly be reused for the Moon mission.

4.4.2.8. Returning to Earth

The MTV attains Earth orbit by aerocapture. For the Mars mission, the same MTV heat shield used for aerocapture at Mars is used at Earth. While in Earth orbit, the Modern Command Module (MCM) docks with the HM and the crews of three enter their respective modules (COV or MCM). MCM and COV undock from HM and reenter. If the MTV were capable of direct entry, an aerocapture maneuver would be eliminated and an argument could be made for MTV re-usability. However, at this stage it is difficult to foresee whether the entire Habitation Module could handle the heat load of re-entry, especially after experiencing the heat load associated with aerocapture at both Mars and Earth. For a Moon mission an argument could be made for MTV reusability because aerocapture is not required to enter lunar orbit.

5. Commonality Across Missions

5.1 Introduction

NASA's current direction is return to exploring the Moon, then explore Mars and beyond. It is with this in mind that we are proposing a high-level set of causally connected baseline missions. These missions are aimed at a continued expansion outward into the solar system, with no single ultimate destination. The experience and knowledge gained from one mission will be put to use on the following missions, thereby enabling future exploration. To recognize this fact is to understand that commonality must play a critical role in all extensible space exploration system. It is by searching for commonality between aspects in the forms and functions of these missions that they may be integrated to generate one consistent over-arching program plan.

5.2. Commonality

Each of the proposed missions for the Moon and Mars contains a number of requirements. Clearly, the requirements of each module (or form) vary from mission to mission, but the objective of an extensible set of mission architectures is to utilize as many functions as is feasible for each form. Deciding the "as is feasible" is difficult, and as such, the following method of comparison aims to illustrate the functional requirements of each form that are not clearly demonstrated in the baseline mission.

The requirements were specified for three Moon missions: Short, Medium, and Extended. These were also specified for the four Mars Missions: Phobos, Short, Extended, and Extended+. Basic forms were selected and the functions were discussed for each form. Each mission was considered independently in the analysis below. For example, if an Extended+ Mars mission requires that the COV be equipped with an aeroshield, while the other three Mars missions do not, the Mars set of missions are assumed to require an aeroshield, however these situations are indicated in the table located below the Venn diagrams in the following figures.

This method of analysis allows functional traits of a form to be easily evaluated and compared with the other required functions in terms of importance across the entire set of mission objectives. If a form does cannot perform a specific function, a decision must be made as to whether or not extending the functionality of a form to include other functions is justified or whether an additional form should be developed to serve the functional requirements flow down from the Mars and Moon mission objectives.

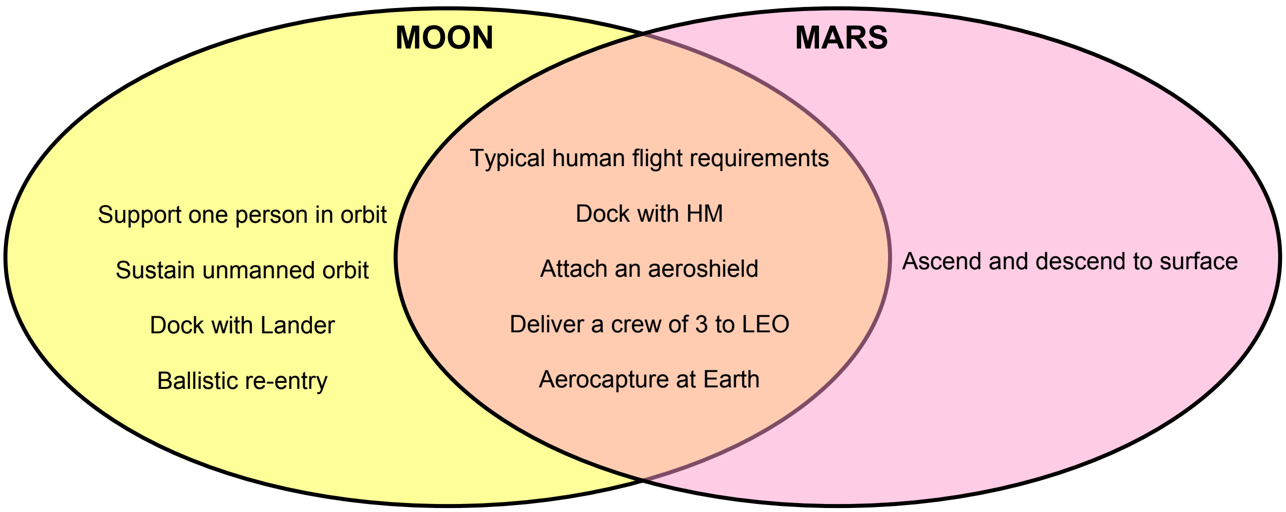
5.2.1 Form/Function Mapping

The following sections will present an analysis of the form/function matrix by means of Venn diagrams. This is a helpful tool in deciding what functions each form should be able to perform, depending on the needs for specific Moon/Mars missions.

Although the shared portion of the Venn diagram is critical to both Mars and Moon missions, it is the regions on the right and left of the Venn diagram that impact individual missions within the respective Mars and Moon mission frameworks. Therefore, considering each of the functions in these areas of the Venn diagram allows the mission designer to decide when a new form capable of accomplishing the functions not included in the “overlap” region of the Venn diagram should be added to the network of modules.

5.2.1.1. Crew Operations Vehicle

The Crew Operations Vehicle (COV) is part of the crew exploration system. It is designed for transporting a crew of three from Earth to the destination and back, with or without the help of the Habitable Module (HM). Specific details of the COV were discussed in Section 4.1.1, but from the Form/Function Matrix results (see Appendix 9.2.1) a Venn diagram of functional requirements for the COV was determined and shown in Figure 30.



	Moon			Mars			
Crew Operations Vehicle (COV)	**Short**	**Medium**	**Long**	**Phobos**	**Short**	**Long**	**Long+**
Dock with HM	-	-	X	X	X	X	X
Communications Equipment	X	X	X	X	X	X	X
Attitude Control[F]	X	X	X	X	X	X	X
Aeroshield Attachment[G]	-	-	X	-	X	X	X
Ascend and Descend to Surface[H]	-	-	-	-	-	-	X
Life Support for Crew of 3	X	X	-	X	X	X	X
Deliver a Crew of 3 to LEO	X	X	X	X	X	X	X
Life Support for 2-3 weeks	X	X	X	X	X	X	X
Ballistic Earth re-entry	X	X	-	-	-	-	-
Aerocapture at Earth	-	-	X	X	X	X	X
Dock with Lander (manual)	X	X	X	-	-	-	-
Dock with Lander (autonomous)	-	X	X	-	-	-	-
Support one person in orbit	X	-	-	-	-	-	-
Sustain itself in unmanned orbit for extended periods	-	-	X	-	-	-	-

[F] Required for docking and rendezvous

[G] For long+ mission, heat shield is required for the COV to descend to the surface

[H] The COV would descend to the surface and provide habitat along with the SHM

Figure 30: Functional requirements for a Crew Operations Vehicle

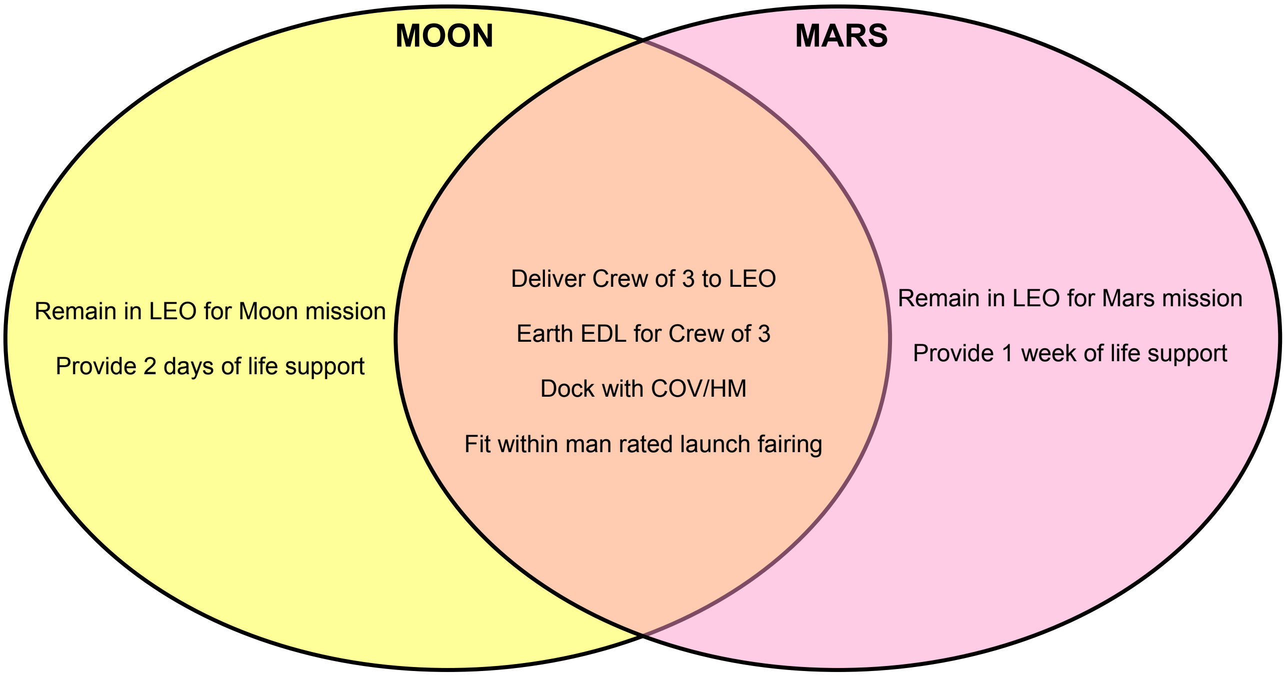
Considering Figure 30, the COV must be capable of providing attitude control and communications for a crew of three, as well as additional human space flight requirements for both the Moon and Mars missions. Based on this diagram, additional forms must be created in addition to the base form to accomplish the required additional functions.

Since it is beneficial to have a minimum number of forms in addition to an extensible network of forms, it is important that some of the functions required of a COV be captured by the baseline COV. For example, docking with the HM reduces the number of functions required of a COV because the HM can provide some of the functions listed for the COV. Since the functions listed in Figure 30 include the unique, additionally complex Extended+ Mars mission, it is difficult to truly evaluate how extensible the COV should be. For example, the Extended+ mission is the only mission that requires the COV be capable of ascending and descending to the surface of Mars. This single functional requirement is essential for only one of the seven missions. Furthermore, the particular mission that requires this added ability is well in the future. Thus, not only is the mission less likely to be carried out as planned, but the expected knowledge returns from this mission would diminish considerably upon discounting to present value. Considering the environment of uncertainty surrounding activities that are so far removed in time, this function should not be captured by the COV unless it could be added at low cost. An additional or alternative crew vehicle form might suffice in the future.

For the Short Moon missions, the COV is required to support one person in orbit. This extends the period of time that the COV must provide life support. For the Extended Moon mission, the COV is required to sustain unmanned orbit. This situation indicates that either an additional form should be added to the network or the additional functions be adopted by the COV. This important decision is made easier when all of the low-level functions can be considered together. The COV will be required to re-enter Earth ballistically for the Short and Medium Moon missions. This delays the requirement of Earth aerocapture technology until the Extended Moon mission, allowing the Moon missions to be a “true testbed.”

5.2.1.2. Modern Command Module

The Modern Command Module (MCM) is the form specified for the Earth launch module in the initial phase of the mission and Earth EDL in the final phase of the mission. The MCM resembles an Apollo-style command module, capable of transporting crew from the Earth’s surface to orbit and back to Earth from orbit, but without the required capability of transit to the Moon. Specific details of this module were discussed in Section 4.4.2.2.2, but from the Form/Function Matrix results (see Appendix 9.2.1) a Venn diagram of functional requirements for the MCM was determined and shown in Figure 31.



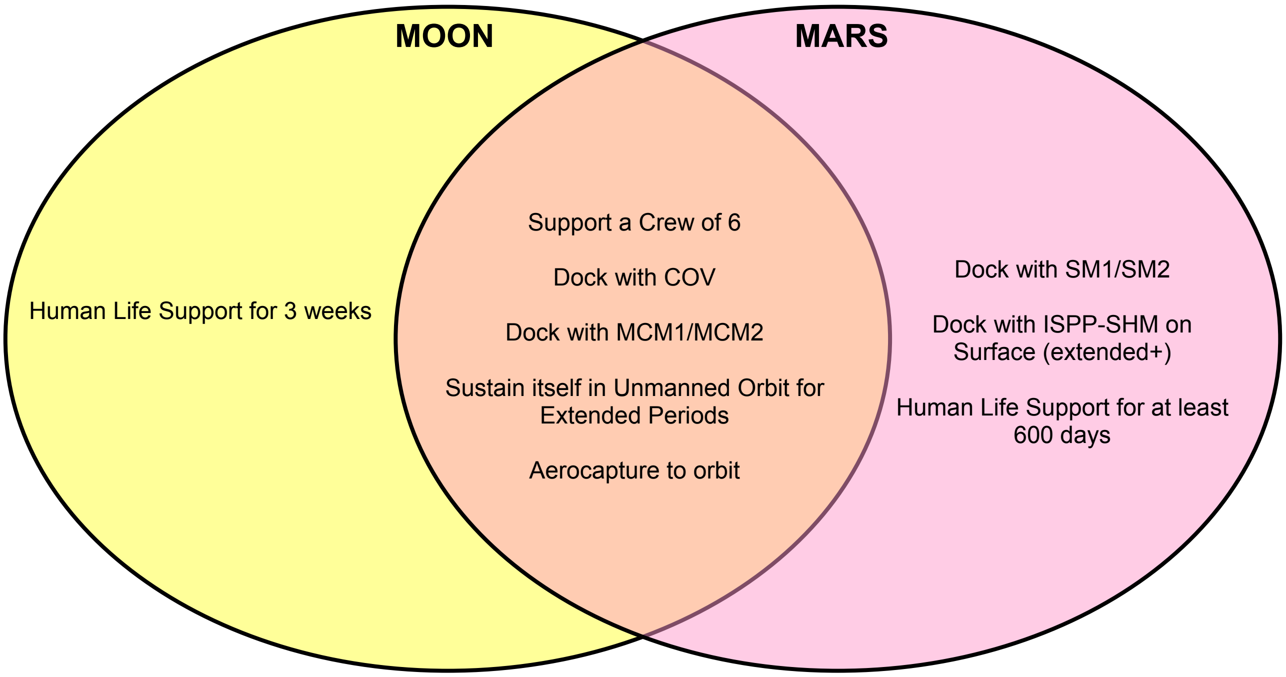
	Moon			*Mars*			
Modern Command Module (MCM)	**Short**	**Medium**	**Long**	**Phobos**	**Short**	**Extended**	**Extended+**
Deliver crew of 3 to LEO	-	-	X	X	X	X	X
Earth EDL for a Crew of 3	-	-	X	X	X	X	X
Remain in LEO for Mars mission	-	-	-	X	X	X	X
Remain in LEO for Moon mission	-	-	X	-	-	-	-
Dock with COV/HM	-	-	X	X	X	X	X
Fits within fairing for man-rated launcher	-	-	X	X	X	X	X
Life support for 2 days	-	-	X	-	-	-	-
Life support for 1 week	-	-	-	X	X	X	X

Figure 31: Functional requirements for a Modern Command Module

It is evident that the requirements for Earth launch are similar for all of the missions and independent of the mission objective. This indicates that the MCM is an extensible architectural element for these functions. The only specific requirements that differ between the Mars and Moon missions are the number of days of life support and the number of crew that must be transported. The number of days of life support was estimated based on the expected assembly time in orbit. Providing additional life support for the longest possible duration in the MCM is not a difficult function to incorporate in the baseline MCM. Rather than developing two functionally equivalent modules such as a three-person and a six-person module, many benefits exist when two identical forms (COV is a similar to the MCM) are used in place of a different form to perform the same list of functions. This premise was discussed in Section 6.4.3.1.1.1.

5.2.1.3. Habitation Module

A Habitation Module (HM), as the name indicates, is a module that supports human life on long duration missions. While the transit durations for the Moon missions are short enough to not require a Habitation Module, the module could still be used to test the technology for future missions to Mars. Specific details of the HM were discussed in Section 4.4.2.2.1, but from the Form/Function Matrix results (see Appendix 9.2.1), descriptions of its desired functions are given in Figure 32.



	Moon			*Mars*			
Habitation Module (HM)	**Short**	**Medium**	**Long**	**Phobos**	**Short**	**Extended**	**Extended+**
Support a Crew of 6	-	-	X	X	X	X	X
Human Life Support for 3 weeks	-	-	X	-	-	-	-
Human Life Support for 360 days[A]	-	-	-	-	-	X	X
Human Life Support for 600 days[B]	-	-	-	X	X	-	-
Aerocapture to Orbit	-	-	X	-	X	X	X
Dock with COV	-	-	X	X	X	X	-
Dock with SM1/SM2	-	-	-	X	X	X	-
Dock with MCM1/MCM2	-	-	X	X	X	X	-
Dock with Landers	-	-	X	X	X	X	-
Dock with ISPP-SHM on Surface[C]	-	-	-	-	-	-	X
Sustain itself in Unmanned Orbit for Extended Periods	-	-	X	X	X	X	X

[A, B] Duration of transit varies depending on the year of departure from Earth

[C] Similar to Mars Direct Architecture, COV goes direct to Mars surface, returns via ISPP

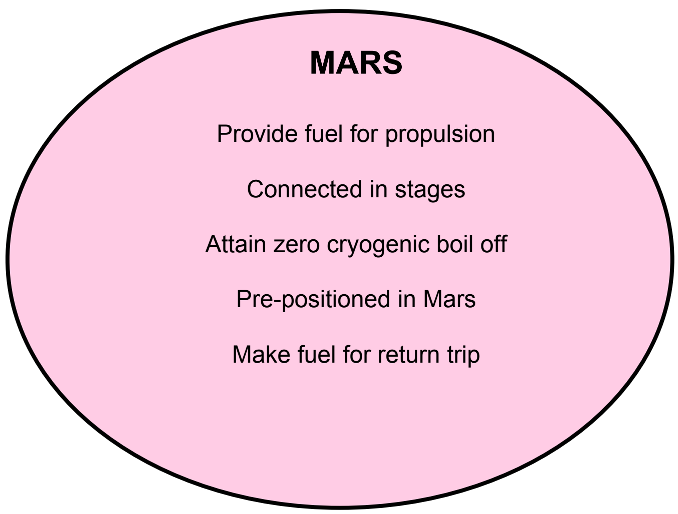
Figure 32: Functional requirements for a Habitation Module

An interesting observation is the number of different forms that must have docking capabilities with the HM. Since the HM module will be docking with the COV, SM1, SM2, and MCM at separate times during the missions, the design of all these elements should require an identical interface. Such a similarity allows for more flexibility and adaptability for new strategic decisions. For example, if the timeline of a mission changes and, for example, docking with SM2 occurs before docking with the Landers, the system can still function adequately during logistic constraints because of the common interface. An option also exists to use a portion of the modular HM to support a crew of 6 on a Moon mission. This allows the functional requirements of a COV supporting a crew of 6 to be removed. For the Moon mission, an engine burn is required to attain lunar orbit. However, a heat shield is still required for the HM because an aerocapture maneuver will take place when the HM re-enters LEO on the Extended Moon mission.

It is important to mention that the HM has significantly different life support requirements for the Extended and Extended+ missions compared to the Short and Phobos missions. This is a result of different trajectories being used for those missions.

5.2.1.4. Crew Service Module

The Crew Service Module (SM) is the form that provides fuel to transport the crew traveling in either the COV for a Moon mission or MTV (COV/HM) for a Mars mission. This module could be functionally compared to the Apollo Service Module. Specific details of this module were presented in Section 4.4.2.2.1, but from the Form/Function Matrix results (see Appendix 9.2.1) a Venn diagram of functional requirements for the SM was determined and shown in Figure 33.



	Moon			*Mars*			
Crew Service Module (SM)	**Short**	**Medium**	**Long**	**Phobos**	**Short**	**Extended**	**Extended+**
Dock with COV/HM	-	-	-	X	X	X	-
Ability to be connected in stages	-	-	-	X	X	X	X
Insulation for zero cryogenic boiloff	-	-	-	X	X	X	X
Ability to be prepositioned in Mars orbit	-	-	-	X	X	X	X
Make fuel for return trip[D]	-	-	-	-	-	-	X
Ability to dock with Hab/COV/Lander[E]	-	-	-	X	X	X	X

[D] Direct architecture - H2 feedstock sent ahead of time, fuel made and stored for return trip, fuel connects to the HM/COV for the return to Earth

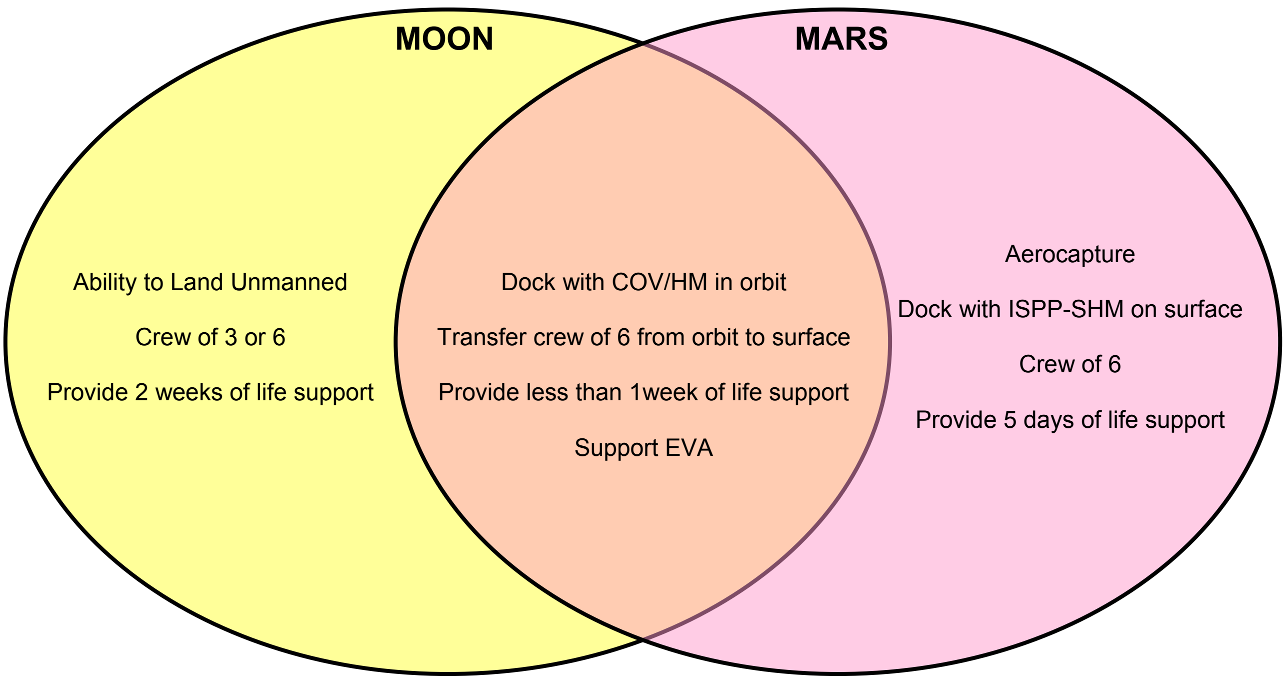
[E] Moon - Docking with HM is only necessary if pre-positioning is used

Figure 33: Functional requirements for a Crew Service Module

It is evident that the pre-positioning requirement given by a Mars mission is not a critical technology for a Moon mission; however, the Moon could use pre-positioning to test the technology. As such, two similar versions of an SM could be designed, one designed for pre-positioning use and the other for conventional propulsion. The first Moon mission would not require the technology and would thus use the conventional propulsion Service Module. However, later Moon missions could use pre-positioning technology using electric propulsion as a stepping-stone for the future Mars mission requirements. Furthermore, the amount of fuel required for a trans-lunar injection is significantly different than the amount of fuel required for a trans-Martian injection. Taking this into consideration may require that the Moon SM and the Mars SM have slightly different forms. The extensibility value of using similar structures is in the manufacturing savings resulting in the use of legacy hardware.

5.2.1.5. Moon and Mars Landers

A Mars Lander (ML) or Lunar Lander (LL) share similar functionality. This similarity in functionality is exploited to incorporate commonality among the various Lander designs. Specific details of the Landers are discussed in Section 6.4.3.1.6. The Lander was based on an Apollo style Lander, capable of transporting a crew of three from orbit to the surface and back to orbit. From the Form/Function Matrix results (Appendix 9.2.1) a Venn diagram of functional requirements for the Lander was determined and shown in Figure 34.



Lander [I]	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Dock with COV/HM in orbit	X	X	X	X	X	X	-
Dock with ISPP-SHM on surface	-	-	-	-	-	X	-
Ability to transfer crew of 6 from orbit to surface and surface to orbit	-	-	X	X	X	X	-
Ability to transfer crew of 3 from orbit to surface and surface to orbit	X	X	-	-	-	-	-
Support EVA	X	X	X	X	X	-	-
Life support for 3 crewmembers	X	X	-	-	-	-	-
Life support for 6 crewmembers	-	-	X	X	X	X	-
Life support for at least 2 days	X	X	X	-	-	-	-
Life support for at least 5 days	-	X	-	-	X	X	-
Life support for at least 1-2 weeks	-	X	-	X	-	-	-
Ability to Land Unmanned	-	-	X	-	-	-	-

[I] For Long+ mission, assume Direct architecture, lander not required

Figure 34: Functional requirements for a Moon/Mars Lander

Consider Figure 34. For comparison purposes, it is clear that common functions are shared. When common functions exist, extensibility will benefit the overall group of missions to the Moon and Mars. Considering Figure 34, the Lander must dock with the COV or the COV/HM in both lunar and Martian orbit. As well, the Lander must deliver a crew of 6 to the surface for all of the Mars missions and some of the Moon missions. If two identical Landers are chosen instead of a single, larger Lander, the impact of this decision can be observed by evaluating whether or not the new option satisfies the functional requirements. If all of the functions are deemed satisfied, only then was the impact of the decision not critical. As can be expected, a wide range of requirements are made for the Landers, but many of these requirements are specified by only one of the seven missions, making it difficult to justify changing the baseline form. Indeed, the Landers are a mission critical piece of hardware and must be highly reliable. Therefore, when considering extensibility of such a device, it may be beneficial to target the Lander design for the most difficult landing mission, thereby ensuring a robust, if over-designed, form for the other missions. This has the effect of increasing net reliability while still maintaining an extensible form. The idea of designing a non-optimal form now such that it may be optimal when used in a different manner or location stands as one of the cornerstones of extensibility.

5.2.2 Form Conclusions

When this method of comparison of using Venn diagrams and form/function matrices was developed, it became apparent that many functions were required of each form. At this stage, the Venn diagrams do not capture the extreme detail required of an extensible Moon/Mars mission architecture. However, this technique provides a tool for designing an extensible transportation system. Although many simplifying assumptions were required to analyze a transportation system in this framework, the advantage of this method is seen when considering the impacts of a decision to not have a form perform a certain function. This method allows these decisions to be traced. The diagram highlights the functions that were not captured by the baseline mission architecture. Thus, pains must be taken to integrate these functionalities into the eventual design for the missions that require them. In doing so, it is important to recognize a fundamental engineering tension that exists between optimality and extensibility. A form that is designed purely with optimality in mind is restricted to the point design for which it was originally conceived. This makes the creative use and extension of such technology difficult. On the other hand, a form designed with only extensibility in mind will become "spread thin," and unable to perform the functions required of it at certain stages or missions. Thus, a compromise must be made between these two extremes.

When designing for extensibility to missions involving high degrees of uncertainty, care must be taken to ensure that the current mission does not become so overburdened with extraneous requirements that it is prohibitively expensive to function as planned. For a "point design" system, it is a possibility that some functionality designed into the system may never be used. In addition, this "point design" system may need to be entirely re-designed before future missions take place. However, when designing for extensibility for near-term missions, the addition of an extra function now which will likely be needed in future missions may decrease the cost of developing and testing that functionality in the future, thus enabling further exploration in the long term and increasing operational knowledge.

5.3 Integrated Baseline

The Presidential Directive on the Moon, Mars and Beyond clearly sets the Moon as the first goal for today's astronauts. The Moon is intended to be used as a testing ground for missions to Mars and Beyond. In the absence of existing infrastructure on the Moon, SSLMs may be expected to occur in the near future. The primary purpose of these missions is to serve as lunar "scouts," which will search primarily for location information, perhaps regarding possible resources that may be exploited on the Moon. They possess the positive attributes of being relatively low in cost and have the potential for high knowledge return given that they land in unexplored locations. Prior to the first SSLM, unmanned robotic probes may be sent to a number of promising landing sites. Similarly, lunar satellites could be constructed to allow for continuous radio contact with the far side of the Moon, if deemed necessary for communications with these probes. SSLMs will occur, each time in a different location, until a decision is made to study particular locations in more depth and for a longer period of time. Future Short Moon missions may include testing of preliminary rover hardware, testing of new space-suit concepts, and gradual extensions of the life-support capabilities of possibly up to 1-2 weeks. Once sufficient experience with lunar operations has been established through the SSLMs, MSLM missions may be launched.

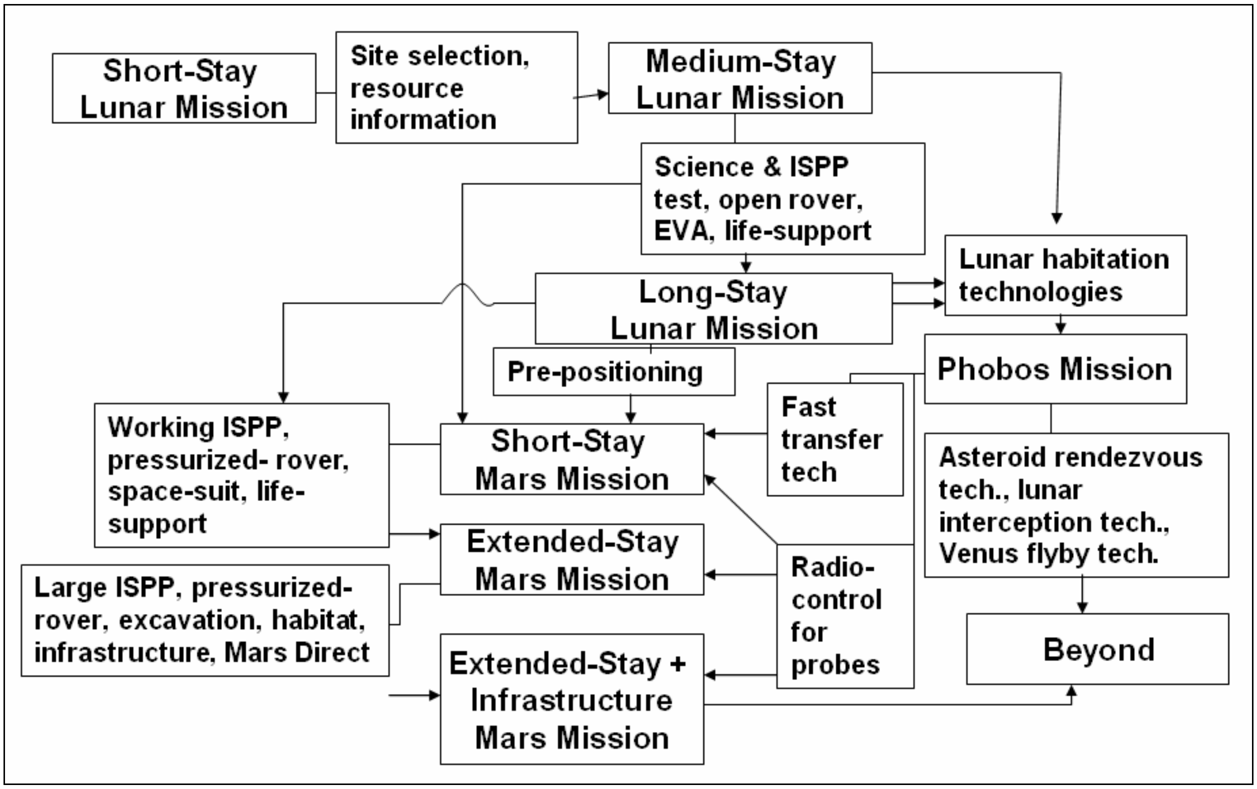


Figure 35: Flow diagram describing elements of extensibility in integrated baseline

The primary purpose of MSLMs is to generate scientific knowledge and to establish non-permanent infrastructure on the Moon. These missions will be aimed at scientific exploration and resource evaluation of promising sites found during the Short Stay Lunar Missions. These missions are larger in scale than the SSLMs, and they possess the ability to carry more equipment. As such, astronauts participating in these missions will be tasked with operating larger scale scientific apparatus and the scientific precursors to the first in-situ propellant production facilities. These will be small scale at first, serving primarily as technology demonstrators, but may be scaled up in future missions to allow for some basic functionality. Included with Medium Moon missions will be an unpressurized rover, which is designed to carry astronauts to locations beyond their operational walking radius. These rovers may initially be tested on SSLMs, and will probably initially be restricted to traveling distances that astronauts can safely walk back from. With sufficient experience gained by using this rover system, the range of the rovers may be extended. MSLMs will occur in a limited range of locations and multiple missions may be sent to the same location. These missions will occur until a primary site is chosen for a future semi-permanent lunar base. MSLMs are still limited by the fact that astronauts may only stay on the lunar surface for limited periods of time and the fact that most mass utilized will come from Earth. In general, these missions will not have the capability to survive throughout the lunar night, although later MSLMs may wish to gear some activities towards this kind of sustenance.

The next step beyond the MSLMs is dependant on the degree to which in-situ lunar resources may be used. If these resources are present, it may be desirable to conduct a series of Extended Stay Lunar Missions (ESLMs) designed to set up a semi-permanent lunar base and to generate the capability to search the far side of the Moon in more detail. This mission would be primarily aimed at allowing more humans to live on the lunar surface for increasingly long amounts of time. This report assumes that such resources are present and usable. In the event that in-situ resource production is not directly feasible, such a long-term base on the Moon would require a significant supply-chain from the Earth. The requirement that this supply chain imposes is not directly in line with overall mission sustainability. Thus, if in-situ resources are not available, the ESLMs may not be launched. Instead, more MSLMs may be carried out to prepare for an eventual Martian Short Stay Mission.

The primary purpose of ESLMs is for this architectural design to align with the President's declaration that more humans will remain on the Moon for increasing lengths of time. These missions will be aimed at establishing a semi-permanent habitat with at least six astronauts on the lunar surface where astronauts may gain experience in living in non-Earth environments for long periods of time. Since these missions will be aimed at the construction of a habitat, some pre-positioning will take place by necessity through resupply from unmanned probes and the use of cargo from previous Moon missions. Therefore, this suite of missions will allow for the buildup of accurate pre-positioning operational knowledge. If the targeting is not accurate on the first few attempts, rovers from the MSLMs will be available to allow the astronauts to reach the landing site. In the worst-case scenario, they may return home as is done in a MSLM. Astronauts participating in these missions will be tasked with operating larger-scale in-situ propellant production facilities. The eventual goal will be to create a largely self-sustained semi-permanent base. Some potential capabilities to be added include a rescue vehicle stored in an easily accessible location so that astronauts may escape to Earth in case of unforeseen circumstances. Included with ESLMs will be habitat modules in which astronauts may live for increasing lengths of time. ESLMs will initially occur only in one location. These missions will likely occur until the first Mars mission is launched, at which point the semi-permanent base may be turned over to international partners or the commercial sector for further development. These missions will have the capability to survive through the lunar night, and will therefore require an independent source of power to be used during the lunar night. The next step beyond the ESLMs is the Martian Short Stay set of missions. These will be the first time humans will travel beyond Earth's gravitational sphere of influence with respect to the Earth-Moon system.

The primary purpose of the Martian Short Stay is to demonstrate the ability of mankind to survive on the surface of Mars. These missions nominally require pre-positioning of cargo on the Martian surface, although the first such mission may not utilize this capability simply because of the complexity of pre-positioning maneuvers. If pre-positioning has already been successfully tested during Moon missions, the technology used may be used on missions to Mars. Prior to the first Short Stay mission, unmanned robotic probes may be sent to a number of promising landing sites. These probes may also be used to practice the accuracy of pre-positioning technology. Like an ESLM, Mars missions utilize habitat modules. In the event that no in-situ resources exist to be exploited on the Moon, the habitat capability must be developed for this mission with no in-situ resources utilization knowledge to be extended from a previous mission. Like the MSLM, Short Stay missions will possess unpressurized rovers that may be used to explore over a relatively large range. They will also be used to test and verify in-situ resource production and utilization facilities for use on Mars.

Short Stay missions will occur, usually in different locations, until a decision is made to study particular locations in more depth and for a longer period of time. Although Short Stay missions will likely occur more than once, the mass and energy requirements to perform these missions will moderate the number of Short Stay missions compared to the number of Short Stay Lunar Missions. Rather, upon finding an ideal long-term site, Martian exploration may continue with longer term, shorter-transfer missions to minimize the effect of microgravity and to take advantage of the resources expected to be found on Mars. The major limitations of the Short Mars missions are imposed by the mass required to support life on Mars and in transit for such a long time. Any opportunity to reduce mass discovered during lunar mission operations would likely be implemented in this mission. For example, if in-situ propellant could be produced on the Moon, it could significantly reduce the mass required to get to Mars, even if that propellant required a detour to the Moon. Similarly, weight concerns impose an upper limit on the number of samples that may be brought back to Earth. Future Short Moon missions may include testing of longer-term habitation facilities, testing of new space-suit concepts, and alternative propulsion and in-situ propellant production concepts. Once sufficient experience with Martian operations has been established, Extended Stay, and Extended Stay + Infrastructure missions may occur.

The primary purpose of Extended Stay and Extended Stay + Infrastructure missions is to demonstrate the ability of mankind to survive on the surface of Mars for an increased duration. In the case of Extended Stay + Infrastructure missions, humans will establish a semi-permanent infrastructure on Mars to be used for science, operations research, or as a testbed for the next destination. These missions nominally require pre-positioning of cargo on the Martian surface, and therefore require the performance of a successful Short Stay Mission to ensure that pre-positioning technology is adequately developed. Prior to the first Extended Stay mission, Short Stay missions will have identified promising resource excavation sites, and resource processing activities. One of the purposes of an Extended Stay mission is to take advantage of these capabilities for refueling, for life-support and agriculture, and to explore Mars in a more comprehensive manner over a longer period of time than is possible with a Short Stay mission. An Extended Stay + Infrastructure mission will have the capability to be self-sustained based upon in-situ resource production, thus reducing mass in LEO as much as possible. To this end, the transit characteristics of an Extended Stay + Infrastructure mission will evolve from a MOR class mission to a Mars Direct mission. If in-situ production is not available, Extended Stay + Infrastructure mission will likely not occur because of the difficulties in maintaining a Martian supply chain from Earth. Like an ESLM mission, all Mars missions utilize surface habitation modules. Extended Stay and

Extended Stay + Infrastructure missions will take advantage of knowledge gained from habitat technology used for the Short Stay missions. Extended Stay and Extended Stay + Infrastructure missions will possess upgraded rovers that may be used to explore a larger surface area. These rovers may be pressurized in the Extended Stay missions and will almost certainly be pressurized in the Extended Stay + Infrastructure missions. This represents extensibility from the ESLM. Extended Stay + Infrastructure missions will make use of inflatable structures and other innovative semi-permanent construction materials in the establishment of a Martian base. Extended Stay and Extended Stay+ missions will occur confined to valuable locations. These locations will often be dictated by those that are richest in exploitable resources and those that promise to yield the greatest knowledge returns.

Upon successful completion of Extended+ missions, NASA will have gained significant experience in the area of manned space exploration beyond LEO. This experience will help further exploration throughout the solar system. A mission to Phobos is just one way to start this expansion. The primary purpose of Phobos missions is to demonstrate extensibility on multiple levels. Beyond the Martian system, there are three places that NASA may choose to explore. These include:

- The inner solar-system
- The asteroid belt
- The moons of the outer solar system

For reasons previously mentioned, a pre-Short Stay mission to Phobos is an ideal technology demonstrator for the destinations listed above. Phobos missions may be conducted at any time within the baseline framework. The suggested time for the mission is before Short Stay Mars missions, when resources are not yet heavily invested in a semi-permanent Martian base. This will give NASA operation experience in navigating to Mars orbit, similar to how Apollo 8 achieved lunar orbit before the landing of Apollo 11.

Although this report only focuses on the Moon and on Mars as locations to be explored, the new NASA vision unequivocally states that the program does not end there. Exploration is an activity that will never cease and the potential to educate and to inspire from this exploration will never run dry. Although the next location to be visited by human astronauts is not certain, there is no shortage of secrets to be unlocked and mysteries to be discovered. It is for this reason that the President’s vision is entitled Moon, Mars and Beyond, for Beyond is the ultimate destination of mankind.

6. Analysis and Trade Studies

6.1 Introduction

The ultimate goal of the design process is to create an architecture that is flexible and robust in the face of change. Having identified sources of commonality between the Moon and Mars, and translated this commonality to operational and formal attributes, the next step is to create a flexible architecture. Final decisions regarding the architecture will depend on key trades identified during commonality mapping. Because we hope to create an architecture using a long-term view of the system life-cycle, however, tools that capture the value of flexibility and robustness will be needed.

This section presents three possible tools for such analysis:

- Analytic-Deliberative Process, through the Analytic-Hierarchy Process
- Multiatribute Utility theory, implemented through Decision Analysis
- Real Options Analysis

It then reviews architectural trades that we have identified as critical to sustainable exploration, and architectural commonality.

6.2 Decision Analysis Using Multiattribute Utility Theory

One defining attribute of a sustainable system is a long expected life-cycle. Thus, predicting the circumstances under which the system will operate throughout its life cycle becomes difficult as uncertainty increases with time. Such systems must also incorporate subsystems that operate throughout the architectural domain, from the mechanical to the political, to the commercial. Again, significant uncertainty is present in the system's operating environment. As a result, the system must be prepared to adapt to unexpected situations without significantly reducing the system's operational utility. Space systems must balance budget constraints and risk. The difficulty of maintaining a delicate balance has resulted in a low mission success rates.

This section suggests a way to choose a flexible architecture that will adapt to different scenarios, thereby helping the system to accommodate to changing environmental conditions without significantly compromising performance. Two approaches are available; either a closed "best design" that attempts to take into account each and every possible change, or a strategy that will change and evolve to accommodate the unforeseen. The former option is restricted to current projections of future events, whereas the latter option is dynamic and adaptable in the face of uncertainty. This report proposes the latter approach as a way to adapt to changing environmental conditions. That is, a baseline option is chosen now, while preserving the widest possible options for the future. Decisions, which would otherwise have to be made at the outset, are delayed such that, when the final choice is required, it is made in an environment of decreased uncertainty.

The bulk of this report focuses on the creation of a baseline strategy to go to the Moon, to Mars and beyond. This baseline was designed with a set of implicit assumptions, regarding the state of the system's operating environment through time. Ideally, the baseline is the strategy that is most likely to succeed given present knowledge of future events.

An alternative set of extreme environmental changes that would impact the baseline design (either positively or negatively) was identified through brainstorming. These changes constitute scenario descriptions. A set of responses to those scenario descriptions was then developed. These responses constitute alternative paths that may be taken in designing the system. Using this framework, one may also think of the baseline design as a response to the most likely scenario. The scenarios also provide a means to identify a possible set of architectural trades or options. These critical trade decisions are analyzed further. In doing this selection, the amplitude of the field to be explored was severely cropped in an effort to perform a somewhat deeper analysis on these interesting features of the trade space. If all of these decisions were to be made at the outset, based only on current understanding of future events, there is a high likelihood that these decisions would not be the best choices in the future. There is almost always more information that becomes available through time that has the effect of changing the environment under which the system must operate. Thus, it is beneficial to delay decisions to allow for a better, more informed decision to be made later.

All potential architectures can be fully described by a vector of the different decisions that have to be taken in order to implement it. The meaning of the word architecture in this case is not restricted to the physical form of the objects. A space architecture includes forms, transit points, budget, policy, etc. It is possible to apply Utility Theory to analyze each of these vectors in the context of scenarios and associated trades.

In doing so, one must first explore how the present baseline reacts to a change in the operating environment, and how appropriate decisions taken at points throughout the system's life-cycle could buy some insurance against negative scenarios, or increased payoffs in case of positives ones. An investment increase will reduce the expected utility, while potentially reducing risk by neutralizing negative outcomes. Similarly, one could increase the payoffs by taking real advantage of optimistic outcomes. This analysis is based on Real Options theory. The problem of acquiring utility is complex; however, and in its complexity lays its value. The utility is not an absolute number; rather it is a scaled numerical representation that reflects a synthesis of the opinions of a group of stakeholders around an issue. The tool proposed to make this synthesis is called the Analytical Deliberative Process. This tool is a formal framework that helps a group of people to discuss a set of opposing measures and to synthesize their potentially contradictory opinions. In doing so, utility values may be determined such that the analysis can proceed. However, one important point should be clarified. This utility number is an artificial creation. It is not a "silver bullet". Rather, it expresses a set of opinions, many of which may be subject to change through time. Computers are often used to calculate these values based upon subjective input, and it is therefore a temptation to treat these utilities as incontrovertible data, although they are subjective assessments of individuals' opinions.

By assessing the different performance metrics that each of the architecture vectors presents, it would be possible to get a measure of the net utility that each architecture holds for the stakeholders. In order to select the architecture that will provide the highest expected utility, a method using Decision Analysis is proposed.

As time passes, some decision points are encountered. Similarly, there are points in time in which the architecture's operation depends on some element of chance. Some of these chance points may have been predicted through previous analyses, and thus the baseline would remain unchanged. Still, others may not have been anticipated and an out-of-baseline approach might be necessary. It is crucial to understand each of these possible branching points, and to study them through a conceptual scenario analysis, such as the one described above, in order to be able to apply the real options approach. The approach provides a thoughtful way to decide where, when, and how much to pay for risk reduction and benefit magnification.

6.2.1 Tools

In order to proceed, three tools are proposed:

- Analytic-Deliberative Process, through the Analytic-Hierarchy Process
- Multiattribute Utility theory, implemented through Decision Analysis
- Real Options Analysis

Both tools help to understand how to deal with a complex set of options and requirements, which are sometimes in conflict.

6.2.1.1 The Analytic-Deliberative Process

This is a rigorous and replicable method that provides a protocol, under which a community of "experts" may arrive at an answer to a factual question. This concept of a consultative body of experts, which assigns these utility values, is important, since it allows for the incorporation of viewpoints ranging from members of the technical community to members of the political community to the public at large.

The process uses the following steps:

1. Decompose the problem into a **hierarchy**:
   All the different requirements have to be sorted into a hierarchy.
   In order to calculate the value of a certain approach, several higher-level Impact categories are devised. They may or may not be independent, in the sense that a change in any of them may affect others. Each of these impact categories includes several objectives that use performance metrics in order to be measured. Some of these performance metrics will be formulas, and will have exact values, whereas others will be more subjective, and be measured in ranges. In each case, a number of utility levels are chosen. This yields a tree-like structure in which the main trunk depicts the utility value.

2. Experts rate the importance of the different levels and state their preferences. They are asked to make **pairwise comparisons**, starting from the utility levels on the lower branches of the tree.

3. In order to synthesize the results, these previous pairwise comparisons are used to fill the so-called Ranking matrices. The eigenvalues of these matrices are the weights that the experts have chosen for the different branches that feed into the utility values.

4. To evaluate the **consistency** of a judgment, a confidence index is calculated. This index assesses the coherency with which the experts have made their judgments. This index helps to build confidence on the coherence of the opinions expressed.

The same approach is used to generate weights for the different branches of the tree: performance measures feed an objective, objectives feed an impact category and, finally, impact categories feed into the utility value.

Using these methods, experts may determine how to rate the different approaches. Utility values may be calculated using some metric, such as mass to be launched to LEO, or a subjective measure, such as the desired level of extensibility through a constructed range scale.

6.2.1.2 Decision Analysis Theory

Decision Analysis Theory is a structured way to rank the decision options available to the decision maker. It enumerates the immediate and later choices available, characterizes uncertainty, quantifies their desirability, and provides rules to help the decision maker to choose the “best” alternative.

Choices to be made, and chances to be taken are organized in a tree-like hierarchy, with the immediate choice at the trunk of the tree, and the posterior ones following in some sort of order (e.g. chronologically).

At each position where a chance event is encountered, there is a chance node, and at each position where a decision has to be taken, a decision node is assigned.

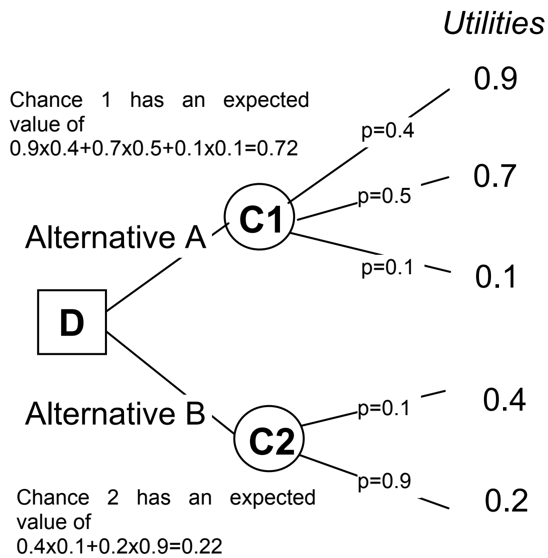


Figure 36: Decision analysis tree.
Decision nodes are drawn as squares, and chance nodes are drawn as circles. In this example, the decision maker will choose Alternative A because it expected utility is 0.72, which is higher than the utility of Alternative B (0.22).

In order to evaluate the desirability of each option, a certain utility is assigned to each final outcome of the tree as shown in Figure 36. This utility function could be an absolute number, but the previously explained Analytic Deliberative process can be used to assign these utilities. Similarly, at each chance point, probabilities are assigned to each possible outcome. These probabilities could either be assigned by a complete assessment of the probabilities, or by asking experts about the likelihood of the different events and applying the Analytic Deliberative process once again. Taking into account these utilities and probabilities, it is possible to travel backwards in the tree, and thus to calculate expected utilities at each chance node.

6.2.2.1 An Example Employing the Analytic-Deliberative Process

A short example follows, using a decision node that is defined by the option of using EM-L1 as a transit point on the way to the Moon. This example also considers the probability of having water on the lunar South Pole. The utility of including the fuel necessary to transit through EM-L1, and then to construct a lunar base, is the calculated. The numbers and opinions expressed in this example are not rigorous. Rather, they have been used to show how a method of this kind could be used, even when exhaustive analysis and complete studies of the involved probabilities have not been done. The primary purpose of this example is to show the mechanics involved in the decision-making process.

The following impact categories have been arbitrarily chosen:

1. Cost
2. Schedule fulfillment

3. Long Term projection of the architecture.

In order to measure the above impacts, the following objectives have to be fulfilled

- Cost
    - P1 = The utility gained from reducing mass,
    - P2 = The utility gained from reducing schedule time
- Schedule fulfillment
    - P3 = The utility gained from launches going according to schedule
- Longer term projections
    - P4 = Probability of maintaining a permanent base,
    - P5 = Probability of producing and shipping fuel

Using the AH process matrix, and pairwise comparisons, experts have arrived at the following conclusions:

P1 Minimum weight
A LEO weight between 40MT to 60MT is most desirable with a utility of 0.5
A LEO weight between 60MT to 250MT is less desirable with a utility of 0.4
A LEO weight between 150MT to 600MT is least desirable with a utility of 0.1

P2 Shorter Schedule
A schedule between 3 and 8 years is the most desirable with a utility of 0.75
A schedule longer than 8 years is less desirable with a utility of 0.25

P3 Schedule fulfillment
95% of launches on time is most desirable with a utility of 0.6
50% of launches on time is less desirable with a utility of 0.3
Less 50% of launches on time is least desirable with a utility of 0.1

P4 Probability of maintaining a permanent base
An 80% probability of maintaining a permanent base has a utility of 0.8
Less than 80% probability of maintaining a permanent base has a utility of 0.2

P5 Probability of producing and shipping fuel
An 80% probability of producing and shipping fuel has a utility of 0.85
Less than 80% probability of producing and shipping fuel has a utility of 0.15

Using the same method, experts then decide that the weights between factors P1 and P2 should get a utility for the cost. These are 0.7 and 0.3 respectively. Similarly they have found that their AH process gives a weight of 0.8 to P4 and 0.2 to P5

Therefore we have now

Cost utility = 0.7 P1 + 0.3 P2
Schedule fulfillment utility = P3
Longer term projections utility = 0.8 P4 + 0.2 P5

Next, experts are asked again to evaluate their pairwise preferences between these three impact categories. They return values that result in the following weights:

- Cost = 0.5
- Schedule fulfillment = 0.2
- Longer term projections = 0.3

The following value formula may therefore be derived:

Total Utility = 0.5 x (0.7 P1 + 0.3 P2) + 0.2 x P3 + 0.3 x (0.8 P4 + 0.2 P5)

Total Utility = 0.35 P1 + 0.15 P2 + 0.20 P3 + 0.24 P4 + 0.06 P5

$$\text{Total Utility} = \sum_i a_i \cdot P_i$$

It is important to note that while it would be possible to obtain firm data for some of these performance measures, the lack of unequivocal data does not hinder the analysis. In any case, the result will be a mix of opinions and engineering calculus that reflect the experts' best knowledge of the situation. This concludes the Analytic-Deliberative Process.

6.2.2.2 Decision Analysis

To analyze the Decision Analysis Tree, one must enumerate the chances and events that will occur, in chronological order:

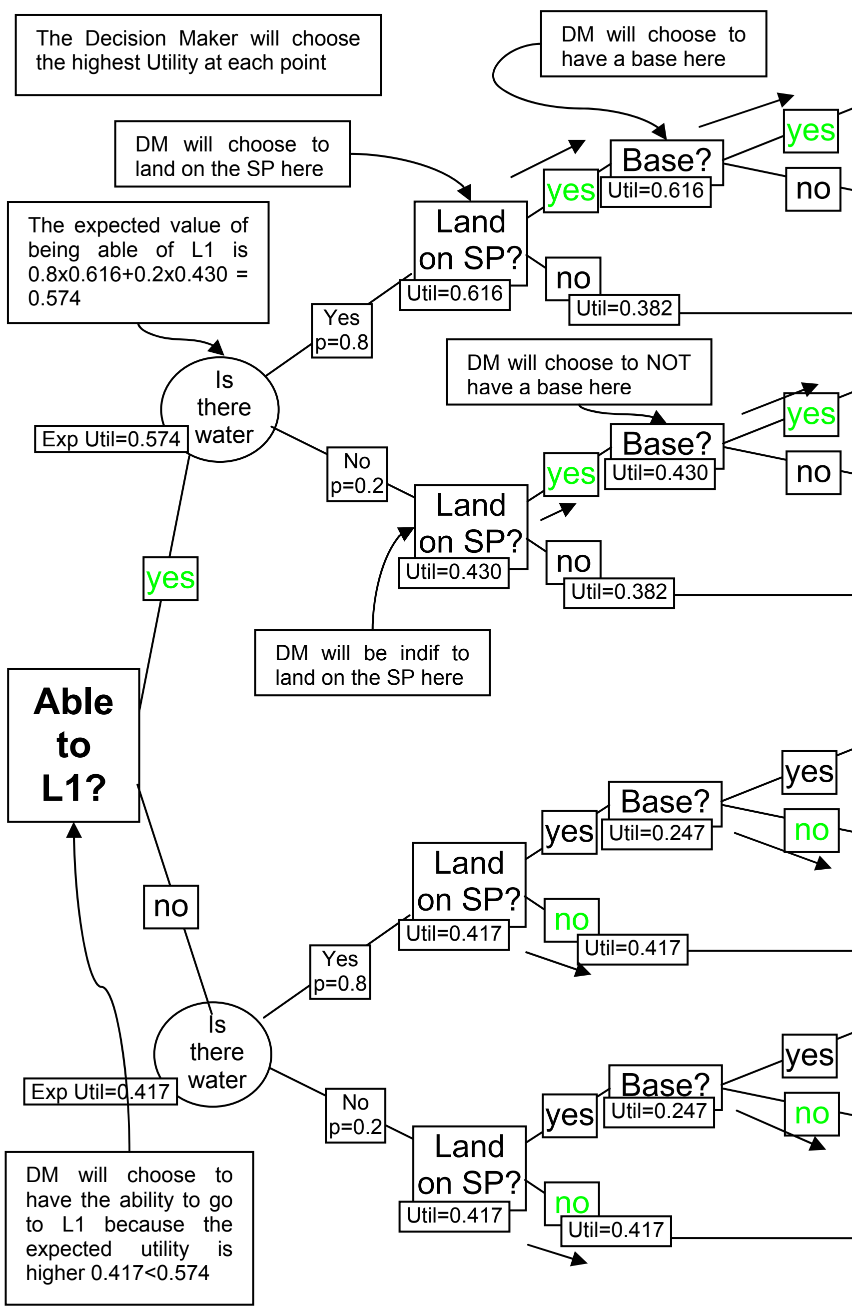
First Event
Decision: Should an architecture to travel to the Moon include L1 as a transit point? This will require an extra 10% of propulsion capability that will not be used in case the mission is not directed to pass through at L1. This implies an extra 3% of IMLEO, regardless of whether the capability is used or not.

Second Event
Chance: Is there water in the South Pole? In this case, the experts are asked to rate the probabilities, result in an 80% probability of the presence of at least 100MT of water in the lunar South Pole.

Third Event
Decision: Should a mission be sent to the lunar South Pole? If this is done from L1, the cost is trivial. If this has to be done from an equatorial Moon orbit this is implies an expensive burn to change orbital planes, and an increase of 200% in IMLEO.

Fourth Event
Decision: Should a permanent base be established at the lunar South Pole?

The decisions and chances described above may be graphically represented in Figure 37.



P1	P2	P3	P4	P5	Util
0.40	0.75	0.60	0.80	0.85	**0.616**
0.40	0.75	0.60	0.00	0.15	**0.382**
0.40	0.75	0.60	0.00	0.15	**0.382**
0.40	0.75	0.60	0.20	0.15	**0.430**
0.40	0.75	0.60	0.00	0.15	**0.382**
0.40	0.75	0.60	0.00	0.15	**0.382**
0.10	0.25	0.10	0.20	0.15	**0.150**
0.40	0.25	0.30	0.00	0.15	**0.247**
0.50	0.75	0.60	0.00	0.15	**0.417**
0.10	0.25	0.10	0.20	0.15	**0.150**
0.40	0.25	0.30	0.00	0.15	**0.247**

Figure 37: Graphical representation of decisions and chances for the example to decide whether to have the capability to go to L1

Thus, the above tree may be used to determine what decisions need be made at what points. This method is not a "silver bullet" solution. Rather, it proposes a formal framework for analysis, argue about options, and quantify the sensitivity to factors. A diverse set of variables also helps to take into account the sometimes-contradictory opinion of a group of stakeholders.

6.3 Real Options Analysis

Real Options is a method to value flexibility in system design. It evaluates the costs of the enabling decisions taken today to perform certain functions in the future. Since these decisions are enablers, they are a necessary condition. In addition, a trigger must act in the future to decide whether to exercise the option or not. In some sense, creating a real option is buying insurance. Thus, a price is paid, whether or not this option is exercised. Once this price is paid, one has the right, but not the obligation, to exercise the option.

In the example explained above, the option bought will be the ability to use L1. Its benefits are realizable only if a decision is made to land on the South Pole. To assess the cost of the option, the following exercise may be executed.

First, compare the utilities using the Decision Analysis tree, assuming that the mission path never lands on the South Pole. As shown in Table 7, the expected utilities are 0.382 for having the ability to go to L1 and 0.417 for not having that ability. In this case, the choice of L1 ability is not ideal, since the polar opportunity is never exploited.

Table 7: Expected utilities from the Decision Analysis tree for the L1 capability decision

	Lands on South Pole	Forbids South Pole
Able to go to L1	0.574	0.382
Not able to go to L1	0.417	0.417

On the other hand, when a decision is made to go to the South Pole, a utility of 0.417 with L1 capability is traded with a potential utility of 0.574 without L1 capability.

Given the decision is not made to go to L1, the cost of the real option is therefore the difference between the utility that the option is not taken (0.417) and the utility of taking the option erroneously (0.382). Thus the cost of the option is a utility of 0.035. Similarly, the potential benefit of taking this option is 0.157. In this conceptual situation, the potential reward is significantly larger than the potential cost, indicating that a decision should be made to utilize L1.

6.3.1 Example: L1 Options

Another way to frame the decision of using L1 to access the Moon involves using the notion of “expected mass” rather than utility. The following example uses real options thinking to value the benefit of creating a system that has the option to use L1 to explore the Moon.

Background and assumptions:
An exploration system starting in LEO is assumed to be composed of both a COV and a Lunar Lander. Two operational architectures are considered:

1.) LOR: The COV and Lander enter lunar orbit. The Lander descends and ascends to and from lunar orbit.
2.) L1: The COV enters L1 orbit, the Lander descends and ascends to the Moon from L1.

If continuous access to the poles is desired together with continuous free return (a likely need if a lunar base is to be built at the poles), then the system architectures present two possibilities:

1.) Use of L1, with a Lander that can descend and ascend to and from L1
2.) Use of LOR, with a plane change burn once in lunar equatorial orbit

The plane-change burn at the Moon requires considerable ΔV and makes the L1 options more appealing.

Under what circumstances would it be beneficial to have the option access L1? Framed as a real option, one can examine the mass savings and penalties from LEO for using L1 compared to the base case in which the equator is accessed. Decision analysis is used here, although more complicated modeling techniques could also be used to increase the accuracy of the valuation.

If a mission to the Moon targets the equatorial region, it is clear that lunar orbit is the best location from which to descend. A mission to the lunar equatorial region, employing a Lander from L1 requires about 11% more Δv than from lunar orbit. If lunar pole access is required, however, this architecture demands a plane change in lunar orbit, resulting in extra mass in LEO compared to the base case. Conversely, if the pole does not need to be accessed and the Lander is equipped with L1 capability, the architecture requires extra mass in LEO compared to the base case.

Assuming that mass in LEO is a surrogate for cost, we can calculate the expected mass of a mission to the Moon, based on the probability of a decision to access the poles.

Expected Mass (EM):

EM = (Mass in LEO for Pole Access)*P + (Mass in LEO for no Pole Access)*(1-P)

where *P* is equal to the probability that the pole will be accessed for a given mission.

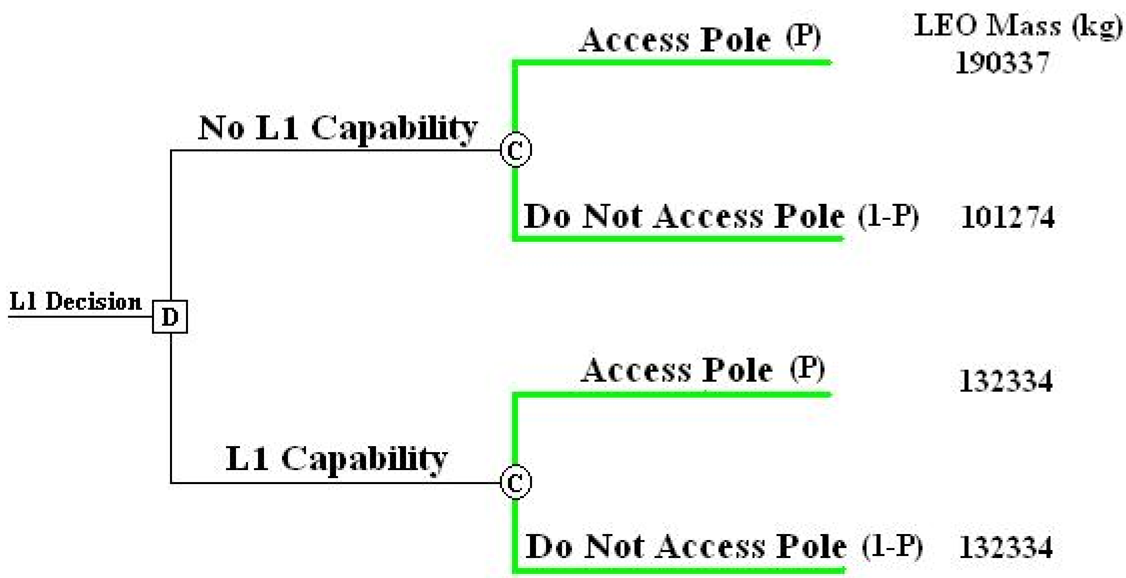


Figure 38: Decision tree for L1 capability example

Then, sensitivity analysis can be used to determine where the value of the option, exceeds that of the base case. The sensitivity analysis reveals that the option to use L1 becomes more valuable (less costly) than the base-case if there is more than a 30% chance that the poles will need to be accessed during the system's life-cycle.

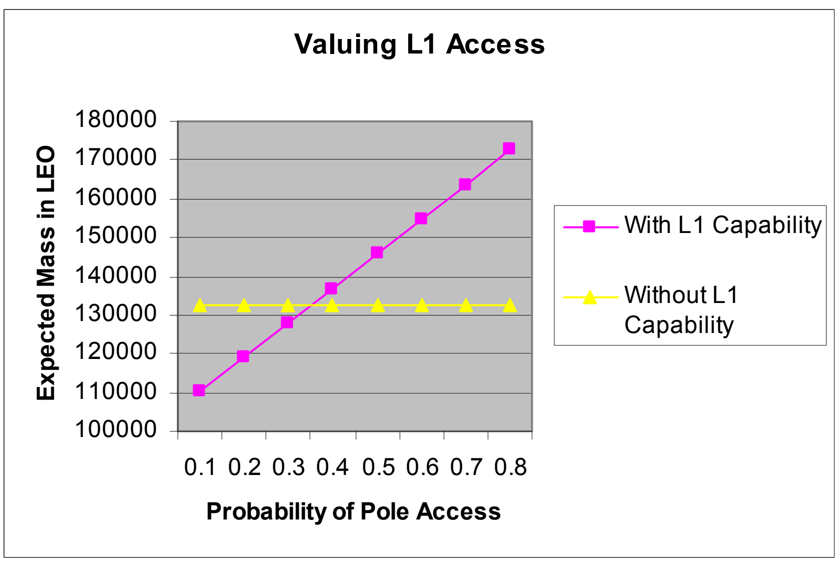


Figure 39: Value of L1 capability

Of course, a number of other factors will affect the decision-making process. The number of missions to each location is an important factor in determining mass savings. Also, this analysis does not consider the fact that mass savings, like cost, might need to be "discounted" over time.

Still, options valuation presents a powerful method to quantitatively justify a decision that is currently sub-optimal, but may increase in optimality as circumstances change.

By framing the L1 decision as a real option, system architects can design flexibility into the architecture and produce a more sustainable system.

6.3.2 Example: Staged vs. Cycler Transportation System Design

Real options can be used to evaluate commonality in the design of the transportation system for LEO to beyond. The two major transportation architectural designs for the transfer to the Moon and Mars are a staged and a cycler system. Traditional design methodologies would evaluate both systems, but would eventually choose only one design for the transportation system. Instead of following a traditional approach, this section will describe how the design of the space transportation system could be evaluated as a real option and how commonality can exist between the designs of transportation systems with different destinations.

A staged system is similar to the design of Apollo where stages were used to get from LEO to lunar orbit and then from lunar orbit back to Earth. In the staged architecture, each stage is discarded once it has been used; the use of stages in the transportation design maximizes the efficiency of the rocket equation because the design eliminates any dead weight that would have to be carried throughout the mission.

A cycler architecture differs from the staged architecture in that a cycler architecture has no stages, consequently carrying around dead weight. One stage provides the Δv for the entire mission. Where a staged design would have two or more sets of engines and fuel tanks, the cycler has only one set of engines and fuel tanks. The other main difference is that the cycler is reusable after it re-enters Earth orbit. A common example of a cycler is the modern automobile. Between destinations the physical form of a car doesn't change except for fuel. From a high-level perspective, the cycler design has the advantage over the staged design because it can be reused from mission to mission, while the expendable, staged architecture would have to be rebuilt for each mission.

In order to determine which design approach to take in the development of the transportation system, a sample Mars mission was used as a baseline and the minimum required mass in LEO was calculated. The characteristics of the sample Mars mission can be found in Table 8. The total mass required at LEO for the staged mission is roughly one-fourth of the cycler architecture mass. This result is partially due to the fact that the cycler has to carry around additional dead weight on the return trip that requires significant additional fuel for the return trip, which in turn leads to the need for increased fuel for the outbound trip. The most likely main reason for the large mass difference is because the staged architecture is not required to reestablish Earth orbit on the return trip and therefore the staged design does not have to carry the additional fuel mass to perform a re-orbit burn. Because of the reusable nature of the cycler, the vehicle must perform an additional burn in order to reestablish Earth orbit. The additional burn to reestablish Earth orbit is about 5 km/s and results in a significant mass increase in the amount of fuel required by the cycler.

Table 8: Staged vs. Cycler transportation vehicle design

***All masses in Kg**	**Staged Design**	**Cycler Design**

Mass COV	5,700	5,700
Mass Habitation Module	55,000	55,000
Cargo Mass to Destination	39,180	39,180
Cargo Mass Returned	9,180	9,180
Mass of Fuel for Stage 1	176,267	>10,000,000
Mass of Stage 1	488,000	>10,000,000
Mass of Fuel for Stage 2	138,138	>10,000,000
Mass of Stage 2	170,394	>10,000,000
Total Initial LEO Mass	740,000	>>10,000,000

Since the required burn to reestablish Earth orbit is a significant constraint on the design of the cycler, one must consider whether the burn at the Earth is truly required. The cycler is required to re-enter Earth orbit, but the cycler is not required to perform a burn in order to reestablish Earth orbit. It could be possible for the cycler to perform some form of aero-braking in order to minimize or possibly eliminate the need for a burn to establish Earth orbit.

After reevaluating the required mass at LEO for the cycler assuming the use of aerobraking, the staged architecture is preferred over the cycler architecture. However, the mass required at LEO for the cycler architecture has been reduced by a factor of three. The use of aerobraking did not change the preference of the staged over the cycler architecture, although it significantly improved the required initial cycler mass LEO. The results of the case in which aerobraking was performed can be found in Table 9.

Table 9: Staged vs. Cycler design comparison with aerobraking

** All masses in Kg*	**Staged Design**	**Cycler Design**
Mass COV	5,700	5,700
Mass Habitation Module	55,000	55,000
Cargo Mass to Destination	39,180	39,180
Cargo Mass Returned	9,180	9,180
Mass of Fuel for Stage 1	176,267	722,000
Mass of Stage 1	488,000	794,000
Mass of Fuel for Stage 2	138,138	281,000
Mass of Stage 2	170,394	309,000
Total Initial LEO Mass	740,000	1,202,422

Perhaps, the requirement of a burn was not the deciding factor in the mass difference. Instead, one might consider the inefficiencies of the mass fraction. How would the required mass at LEO change if the return fuel could be pre-positioned at the Moon or Mars? In order to compare both transportation designs on an equal level, pre-positioning must be applied to both the staged and cycler transportation designs. In the case of pre-positioning for the staged architecture, the transfer vehicle would only be required to carry the first stage on the outgoing leg. It could then drop off the first stage and pick-up the second stage for the return flight home. In the case of the cycler, the fuel for the return flight home could be provided at the final destination, but instead of dropping a stage like the staged architecture, the cycler would simply refuel using the pre-positioned fuel provided at the destination. The design of the cycler would require that the fuel stage be sized accordingly to accommodate the leg of the trip which required the largest fuel mass. The design choice would result in either the inbound or outbound trip with a sub-optimal use of the mass fraction equation. It turns out that the extra structural mass is insignificant when compared to the mass of the entire system.

The concept for re-fueling the cycler opens the idea for the development of in-situ propellant production. At a top level, it is conceivable that some form of cycler system could also be used in the design of in-situ propellant production delivery. Using a cycler for the delivery of in-situ produced fuel would be a tremendous advantage because it would require only one delivery vehicle be developed, as opposed to multiple vehicles in a staged system. However, in this case in-situ propellant production was not considered in the design and return fuel was pre-positioned similar to the staged system.

After recalculating the design for the staged and cycler designs assuming pre-positioning of return fuel, the preferred architecture again is the staged architecture. The total mass required for the staged architecture went from 740,000 kg to 464,000 kg and the mass required for the cycler architecture went from >>10,000,000 to 6,000,000 kg. The results can be seen in Table 10.

Table 10: Staged vs. Cycler design comparison with the pre-positioning of return fuel

All masses in Kg	**Staged Design**	**Cycler Design**
Mass COV	5,700	5,700
Mass Habitation Module	55,000	55,000
Cargo Mass to Destination	39,180	39,180
Cargo Mass Returned	9,180	9,180
Mass of Fuel for Stage 1	176,000	1,500,000
Mass of Stage 1	193,000	1,650,000
Mass of Fuel for Stage 2	138,000	3,000,000
Pre-positioned Mass of Stage 2 (LEO)	170,000	4,200,000
Total Initial LEO Mass	464,000	6,000,000

The final trade to consider is the situation in which both aerobraking and pre-positioning of return fuel are used. In this case, the staged and cycler architectures combine architectural elements of the pre-positioning and aerobraking cases. The results were different from the previous cases: the cycler architecture is the preferred architecture when the number of mission exceeds two. The mass required for the staged architecture went from 740,000 kg to 464,000 kg, while the mass for the cycler architecture went from 1,200,000 kg to 472,000 kg for the first mission and 359,000 kg for each additional mission. Here the benefits of the reusable nature of the cycler dominate the design of the transportation architecture. These results are shown in Table 11 and the total mass in LEO per number of missions are plotted in Figure 40.

Table 11: Staged vs. Cycler design comparison with aerobraking and pre-position return fuel

*** All masses in Kg**	**Staged Design**	**Cycler Design**
Mass COV	5,700	5,700
Mass Habitation Module	55,000	55,000
Cargo Mass to Destination	39,180	39,180
Cargo Mass Returned	9,180	9,180
Mass of Fuel for Stage 1	176,000	176,000
Mass of Stage 1	193,000	193,000
Pre-positioned Mass of Fuel for Stage 2	138,000	144,000
Pre-positioned Mass of Stage 2 (LEO)	170,000	178,000
Total Initial LEO Mass	464,000	472,000
Additional Mass needed for next mission	464,000	354,000
Total LEO mass for 2 Missions	928,000	826,000

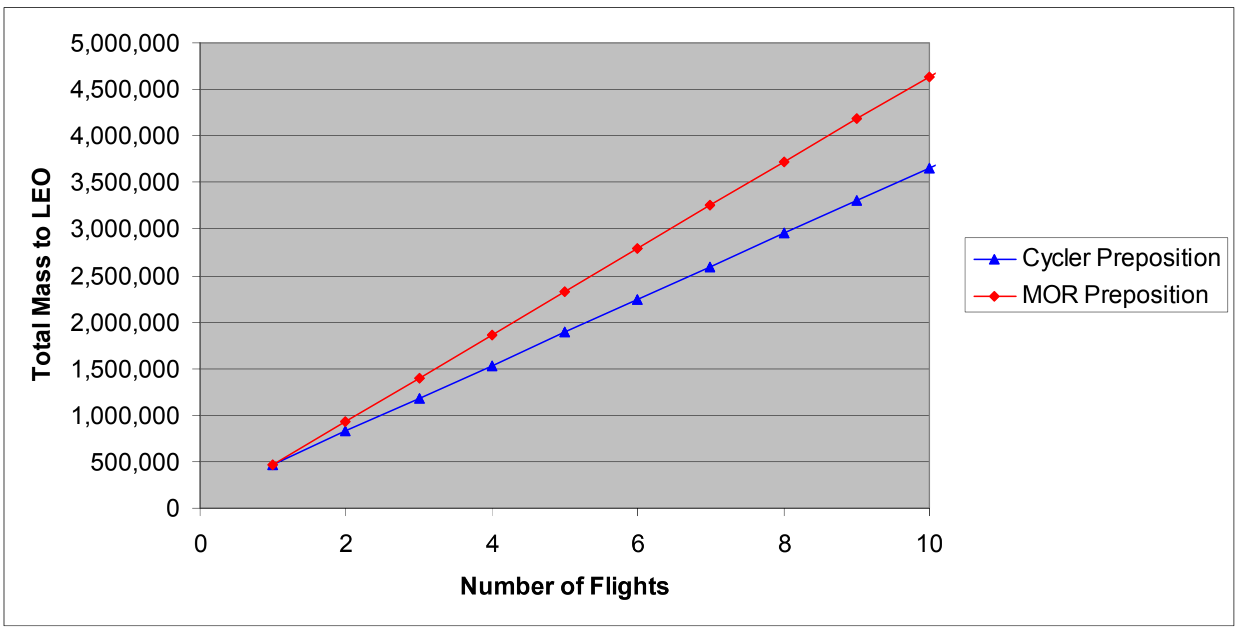


Figure 40: Total Cycler and Staged transportation systems LEO mass per number of flights assuming aerobraking and pre-positioning of return fuel

6.3.1.1 Commonality

When comparing the transportation designs for Mars to the design for the Moon, it can be assumed that the resulting trends for Mars are the same for the Moon. Coincidently, when the Δ*v* requirement for a Moon mission and the Δ*v* requirement for a Mars mission, assuming pre-positioning and aerobraking are compared the total required Δ*V*s are almost identical (~8km/s round-trip). Therefore, if pre-positioning and aerobraking are used in the design of the transportation system, the transportation system design for a

Mars and Moon mission are almost identical. It is conceivable that one vehicle could be designed such that it could provide the transportation for both a Moon and Mars mission.

6.3.1.2 Real Options application

Now that the results of these four trades have been evaluated, should the transportation system be designed as a cycler or as a staged system? The answer is that the transportation system should at first be designed as a staged system, as expected. However, at the same time that the staged design is being developed or used, research into aerobraking and pre-positioning should be examined. If at any time it is discovered that either pre-positioning or aerobraking is unlikely, then the design of the transportation system will remain as a staged system. In the event that both pre-positioning and aerobraking have been found to be feasible, NASA should then and only then switch to a cycler transportation architecture. Therefore, in order to have the ability to switch between architectures, NASA needs to develop a staged architecture that has common elements that could be used in the development for the cycler system. This commonality could be accomplished through a modular design for the habitation module, COV, and transportation, most likely propulsion, system.

The design choice of building a staged system with common elements to a cycler system is an example of a real option. NASA only needs to develop a staged system, with commonality in mind, and research aerobraking and pre-positioning in order to have the capability for a cycler system. Only once both pre-positioning and aerobraking have been found to be feasible should NASA make the decision to spend the resources to switch the design of the transportation system to a cycler.

6.4 Trades

6.4.1 Introduction

The theories and tools about decisions analysis, which were presented in the previous section, were not used for all the trades we considered. This part of chapter 6 presents the trades that were considered for Earth to LEO options (launch site, Earth launch system for both human and cargo, crew escape system and entry, descent and landing), followed by trades for In-Space transportation (crew exploration vehicle, rover, habitation module, pre-positioning, planetary landing systems, crew module scaling, Moon options, Mars options). This voluminous chapter reflects all the background studies that were performed during this class to support our decisions for the baseline mission.

6.4.2. Earth-to-LEO Options

6.4.2.1. Launch Site

The policy that is more likely to be applied for this human exploration program is that all critical launches should be made from US territory. Taking into account that all our architectures benefit from a launch from low latitudes Kennedy Space Center remains the most attractive option. KSC has both EELV and Shuttle launch pads, which minimizes the need for new ground infrastructure. It should be noted thought that there is room for international cooperation and launches in the realm of backup or non-critical tasks. In this respect it should be pointed out that it would be very beneficial to use an APAS-89 docking mechanism. This system was developed for the Apollo-Soyuz program and is the one STS orbiter uses now to dock with the ISS. It is also the same system used by the Shenzhou, this would leave open the possibility of launching crew in that vehicle.

6.4.2.2. Earth Launch Systems

Directed by the new policy, independent launch systems were considered for the crew and the heavy cargo needed for human exploration missions. Also following the new policy, only expendable launchers will be considered.

6.4.2.2.1. Human Launches

To transfer humans from the surface of the Earth to a destination in LEO is a capability that, due to the early retirement of the shuttle, has to be reacquired in the US. Since the time between the retirement of the shuttle and the new human rated vehicles lifting off must be minimized, we have decided to consider adaptations of current launchers. These adaptations should not involve re-qualifying every piece of hardware, but instead adding some redundancy to the avionics and, most importantly, focusing on a very safe launch escape system. Many lessons can be learned here from the Chinese space program and how they human-rated the Long March vehicle.

A strong reason why humans should be launched separately from the cargo is that the US may want to have the capability of transferring crew and docking to the International Space Station (ISS). To do this using a large cargo launcher would be complicated and expensive. This extensible set of forms that are used to complete the Moon/Mars mission does not require the use of the ISS.

The Crew Operations Vehicle (COV) and functionally-similar Modern Command Module (MCM) are required to transport a total of six people in some configuration or grouping, and must have rendezvous and docking capabilities in LEO. It is a reasonable to assume 20 metric tons will be required in LEO. The two launchers that can soon be adapted to launch such a human mission are the EELVs. The Delta-IV provides the added advantage because it uses the RS68 engine, which can be used to replace the SSME on an STS-derived heavy launch vehicle for cargo. As much work as possible from the Orbital Space Plane concepts that used capsules instead of winged vehicles shall be reused in this design. Therefore, the launcher of choice for humans is a human-rated version of the Delta IV.

6.4.2.2.2. Cargo Launches

The decisions regarding the launch of cargo for the Mars and Moon missions have a very important impact on the overall cost and feasibility of the Exploration Program. We argue that an STS-based heavy launcher should be developed and employed for this mission.

Since cost is a bounded variable in this program it makes sense to include some cost estimates in the overall evaluation of the different architectural options. One of the main choices that face the program is the decision to develop a heavy launch capability. It has been argued that, if using modularity and extensibility, the mission's hardware can be broken down into parts, of about 20 metric tons, that are manageable by the heavy version of the Delta IV (an EELV launcher). We will now compare this option to an STS-derived heavy launch architecture.

A typical payload in LEO for a small lunar human exploration mission is 118 metric tons.

To obtain the number of Delta IV Heavy (DIV-H) launches that would be needed we cannot just divide 118 by 20. A "penalty factor" must be applied that accounts for the extra mass stemming from the rendezvous and docking systems as well as the less structurally efficient geometry. This factor has been chosen to be 1.3. This gives roughly 6 DIV-H launches.

6.4.2.2.2.1. Cost Estimation

All cost figures were corrected for inflation into FY14 dollars using the Consumer Price Index and the prognosis of the Office of Management and Budget and the Congressional Budget Office. The cost of a DIV-H launch in FY99 is $170 million. Correcting for inflation in FY14, the total cost of the six launches would be $1.4 billion.

This figure can be benchmarked with the cost of the launch of a Saturn V rocket which was $431 million FY67, which makes almost $3 billion FY14.

The cost estimates per launch of a Shuttle C were estimated to be $85 million FY85, which is $182 million FY14. This valuation, as is the case with programs that do not get to the stage of operation, may be incorrect by as much as a factor of four. Therefore, we will assume that the cost of a STS launch is approximately the cost of a shuttle flight and the more reliable values that are given for a launch of the Space Shuttle will be used. This value is highly dependent on the flight rate, so again caution should be exercised. Assuming a flight rate of 6 per year, the value that is commonly accepted for a shuttle flight is $245 million FY88, which is $477 million FY14. For a flight rate required for a crew of six, the STS-derived is probably too ambitious. If we assume a flight rate of 4 per year, that is a Moon trip every three months, then a single flight would be about $715 million FY14. It should be noted that this value is substantially less than the value obtained for the launch using 6 DIV-Hs. A flight rate of 2 missions per year gives a break even in the cost making. From a cost point of view, both architectures are equally attractive at a price tag of $1.4 billion FY14. Naturally the same argument applies even more strongly to the case of Mars missions.

Another advantage of the STS architecture over the DIV-H is that, since all the hardware for a small Moon mission is launched at one time, automatic rendezvous and docking capabilities are not a critical new technology to be developed for the lunar missions.

This capability of automatic rendezvous and docking will be necessary when human exploration missions to Mars will be attempted. The mass budget in LEO required for even the simplest human exploration mission to Mars are in the range of 200-600 metric tons. This can be reasonably done with 2-6 STS based launches per mission, which is a flight rate of Mars missions of roughly 1 per year.

Therefore, for an Apollo-class Moon mission, one STS would launch most of the mass and a roughly 20 tons capsule, launched separately, would carry the humans to dock with the rest of the mass.

Among several architectures for an STS-derived vehicle the one that seems most attractive is an external tank two SRBs and three disposable 3RS68 as well as a newly developed J2 class upper stage. Such a vehicle can deliver roughly 100 metric tons in LEO. A detailed explanation and performance curves can be found in Appendix 9.1.3.

It has been argued that a new launch system will be better than an STS-based design because most of the problems with the STS are not a consequence of the Orbiter's design, but are rather related the parts that would be kept in any STS-derived. For instance, note the problems with the O-rings in the SRBs and the foam in the ET. To provide a complete answer to that question falls beyond the scope of this study, however the results of this study can be used as a baseline to know what a new design should consider in terms of cost and performance as compared to an existing STS based system.

6.4.2.3. Crew Escape Systems

Human spaceflight escape systems have been developed for on-pad abort and boost phase emergencies. Once in-orbit, the escape mechanism for the crew is the same as the normal re-entry sequence into the Earth's atmosphere. Throughout the history of human spaceflight, ejection seats and escape towers have been developed to provide this additional layer of safety to the crew. However, the option of including such systems must be traded against significant mass penalties (Nuttall, 1971).

6.4.2.3.1. Legacy / Proposed Systems

The first manned orbiter, Vostok I, had an ejection seat escape device. In addition to providing an escape mechanism for the cosmonaut on the pad and during the boost phase, this ejection seat functioned as the normal means of landing after post-orbital descent.

The first US manned orbiter, Mercury, included an escape tower. This tower was attached to the top of the Mercury capsule and consisted of a solid rocket motor. In an on-pad or boost phase emergency, the rocket motor would fire, separating the manned capsule from the booster. A parachute was deployed after the rocket firing, lowering the capsule to the ocean as in a normal post-orbital descent.

The US Gemini escape system also utilized ejection seats just as the Vostok I. This ejection system was flight-tested up to 20,000 ft and Mach 1.75. The decision to use ejection seats instead of the escape tower incorporated into the design of Mercury was driven by the hundreds of kilograms in mass savings. Unlike the Vostok I, the dual rocket-powered ejection seats were only used in landing emergencies (on-pad, pre-orbital ascent, post-orbital descent over land). Normally, the Gemini capsule was lowered to the ocean by a large parachute.

As in Mercury, the Apollo Launch Escape System (LES) utilized an escape tower. Providing an emergency escape capability to the crew from the on-pad launch sequence to the end of second-stage ignition, the LES engines weighed 5,500 pounds with a total structural mass in excess of 9,000lb. Its maximum operational parameters were 320,000 feet and a Mach number of 8.0. The LES consisted of three solid-propellant rocket motors. After the firings of explosive bolts to separate the command module from the service module in an escape sequence, the launch escape motor would pull the 11,000 pound command module to safety using a 155,000 pound thrust solid rocket (Townsend, 1974). The tower-jettison motor was employed to separate the escape tower from the command module prior to parachute deployment (Lee, 1971).

As for Apollo, the Soyuz Emergency Escape System (EES) utilizes an escape tower of solid-propellant rocket motors to pull the crew capsule to safety in an emergency during on-pad launch operations of the boost phase. The EES is operational throughout all phases of powered flight trajectory prior to orbital insertion (Kolesov, 1969).

A seated tractor rocket escape system was proposed for the STS in the wake of the Challenger accident. The tractor system is lighter and less voluminous than an equivalent ejection seat system, however, aerodynamic "blow-back" causes unsuccessful extraction at altitudes above 15,000 feet (Ondler, 1989).

Utilizing Lockheed Martin's Pad Abort Demonstration (PAD) platform consisting of sensors and mannequins in a simulated crew cabin to measure accelerations and motions generated, NASA will conduct seven integrated PAD test flights during 2005-2006 to test an escape tower system of four 50,000-pound thrust RS-88 rocket engines. These tests aim to trade various propulsion systems; parachute deployment, vehicle configurations, and landing techniques for a future tower escape system (Orbital Space Plane/Crew Exploration Vehicle).

6.4.2.3.2. Safety vs. Mass Penalty Trade for COV Tower Escape

Ejection and tractor rocket seats are lighter than tower escape systems by an order of magnitude—weighing hundreds of pounds instead of thousands. However, because the tower escape system is jettisoned during ascent while ejection and tractor seats are carried in the service module throughout the mission, LEO payload mass reductions for tower escape systems are approximately only two to five times greater than ejection and tractor seats.

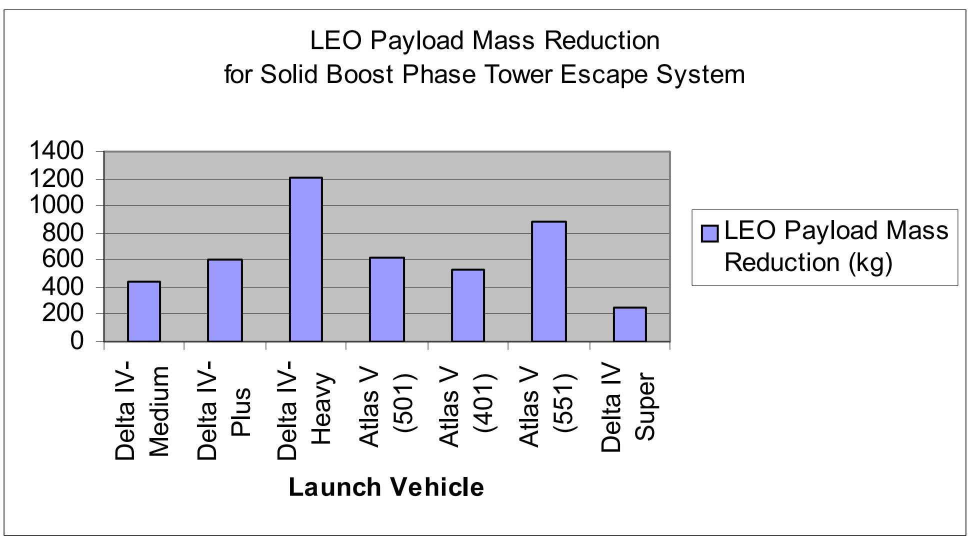


Figure 41: Minimum LEO payload mass penalty for EELV tower escape

Figure 41 displays the lost LEO payload mass when a 5,500lb (2495kg) tower escape system is added to various EELV designs. (A tower escape system of this mass is a minimum estimate for systems capable of saving service modules in the 6,000kg range). It is assumed in these calculations that the escape system is jettisoned with the first stage.

For Delta-IV vehicles, the average impact is ~6% reduction in payload to LEO. Specifically, a Delta-IV designed to launch 6,760kg would have the payload mass reduced by 441kg, a Delta-IV designed to launch 9,070kg reduced by 610kg, and a Delta-IV designed to launch 20,500kg by 1,210kg. The Atlas-V vehicles have an average 5% reduction in payload mass to LEO. For Atlas-V launchers designed to launch 10,300kg, 12,500kg, and 20,520kg, the payload mass reductions are 621kg, 530kg, and 880kg, respectively. For an EELV-derived heavier lift vehicle capable of placing 50,000kg in LEO, adding the escape system would have a much lower effect—reducing the payload capability by only 251kg (or about 0.5%).

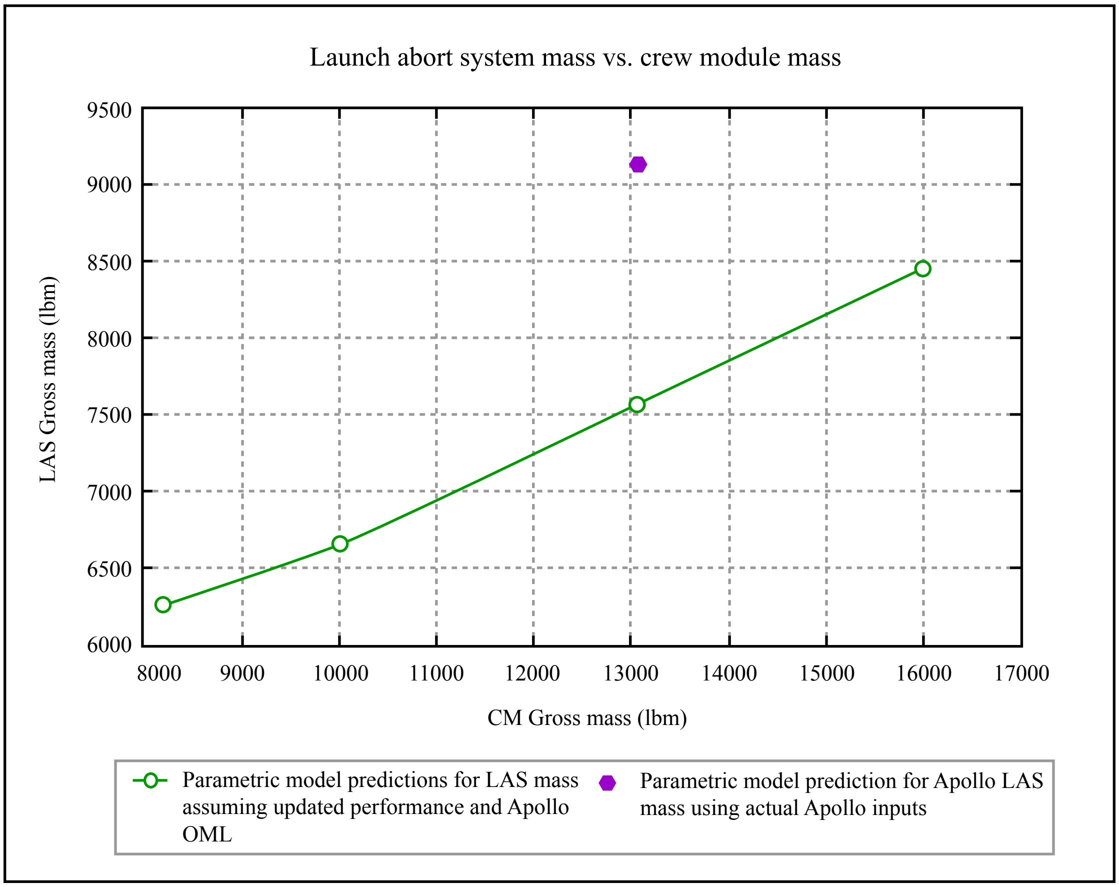


Image by MIT OpenCourseWare. Adapted from Orbital Science Corp.

Figure 42: Launch escape mass as a function of crew module mass (Source: Orbital Science Corp.)

Figure 42 displays how tower escape system mass scales parametrically with the crew module mass according to a model developed by Orbital Sciences Corporation. With escape system ranging from 6,000-8,500lb, the 5,500lb escape system mass selected to calculate the LEO payload mass reductions for EELV architectures (see Figure 41) is clearly on the low-end of the scale.

6.4.2.4. Earth Entry, Descent, and Landing (EDL)

6.4.2.4.1. Vehicle Shape adapted from (Larson, 1999)

To appreciate the available options for entry vehicle shapes, one must first determine the criteria affecting selection. Decreasing development cost has been identified as a major constraint in the project. Another highly important requirement is minimizing mass. A subjective criterion to represent minimum mass for given vehicle's shape is volumetric efficiency. A third objective might be to limit the peak deceleration forces on the crew. Clearly, there are other criteria affecting the shape of the entry vehicle, but these have been identified as the most important. Next, the relative importance of the identified criteria is determined:

Minimum development cost		0.5
Volumetric efficiency		0.3
Peak entry deceleration		0.2
	Total	1.0

Each option is pair-wise compared in Figure 43. That is, each option in the vertical column is compared to each option across the row. If the first option is estimated to be less expensive than the second, then a "1" is placed in the box. If it appears more expensive, then a "-1" is placed in the box. If no significant difference can be determined, then the pair is assigned a "0". The sum at the end of the row is the relative score for that option. Similarly, the volumetric efficiency is subjectively assessed by how effectively each shape can contain a roughly cylindrical pressure vessel for the crew and equipment. Peak deceleration is compared by assuming shapes with higher lift-to-drag ratios maintain lower peak deceleration forces on the crew during entry.

Soyuz, Apollo, and heat shield with afterbody all have the same development cost, whereas the lifting body will have the highest development cost. The biconic most closely resembles a cylinder. Soyuz and Apollo are almost as efficient as the biconic, but their blunt conic shape is more conical. The heat shield and afterbody shape is cylindrical, but the cylindrical diameter is smaller than the heat shield, so some volume is wasted. The lifting body sacrifices volumetric efficiency in the interest of streamlining. The comparison method used for assessing peak entry deceleration indicates the lifting body would have the lowest and the Soyuz would have the highest deceleration.

Second Option / First Option	Minimum Development Cost						Volumetric Efficiency						Entry Deceleration					
	Soyuz	Apollo	Heatshield & Afterbody	Biconic	Lifting Body	Score	Soyuz	Apollo	Heatshield & Afterbody	Biconic	Lifting Body	Score	Soyuz	Apollo	Heatshield & Afterbody	Biconic	Lifting Body	Score
Soyuz		0	0	1	1	2		0	1	-1	1	1		-1	-1	-1	-1	-4
Apollo	0		0	1	1	2	0		1	-1	1	1	1		0	-1	-1	-1
Heatshield & Afterbody	0	0		1	1	2	-1	-1		-1	1	-2	1	0		-1	-1	-1
Biconic	-1	-1	-1		1	-2	1	1	1		1	4	1	1	1		-1	2
Lifting Body	-1	-1	-1	-1		-4	-1	-1	-1	-1		-4	1	1	1	1		4

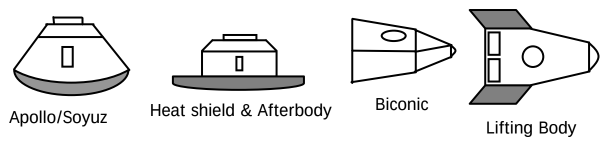


Figure 43: Entry vehicle shape pair-wise option comparison

The selection criteria weightings generate an overall score, which is mapped onto a number line in Figure 44. Notice that the criteria-weighting factors directly influence the final rankings. Importantly, these rankings are subjective assessments that should not suggest an “optimal” option. Experience and intuition might confound the option space. For example, a winged body might be too difficult to equip with thermal protection for high-speed lunar or Martian returns. For the purposes of this report, an Apollo-class entry vehicle, termed the Modern Command Module (MCM) shall be used for Earth return.

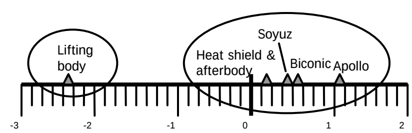


Figure 44: Comparison scale for entry vehicle

6.4.2.4.2. Descent and Landing

Descent and landing is the flight phase designed to reduce the horizontal and vertical velocities to a desired value for surface touchdown. The thick atmosphere of the Earth allows a spacecraft to follow the aeroentry phase with an inflatable, parachute, or parafoil deceleration all the way to the surface. The Apollo Command Module used parachutes to an ocean splashdown, and the Russian Soyuz capsule rides a parachute until a retrorocket fires just before a land-based touchdown.

To achieve mission objectives, atmospheric entry is constrained by three fundamental requirements: deceleration, heating, and accuracy. Although a vehicle’s structure and payload limit maximum deceleration, we must consider the requirements of a human-rated exploration system. Well-conditioned humans can withstand a maximum of about 12 Earth *g*’s for a short time. A system designed for de-conditioned crew must produce less than 3.5-5 Earth *g*’s (Hale, 1994).

Friction between the speeding entry and atmosphere generates heating that must be dissipated during the few minutes of atmospheric entry. The thermal protection system must withstand the total heating and the peak-heat rate encountered during entry.

A third important mission requirement is accuracy. The spacecraft's capability to maintain a predetermined trajectory depends on its inertial navigation systems and available ground support.

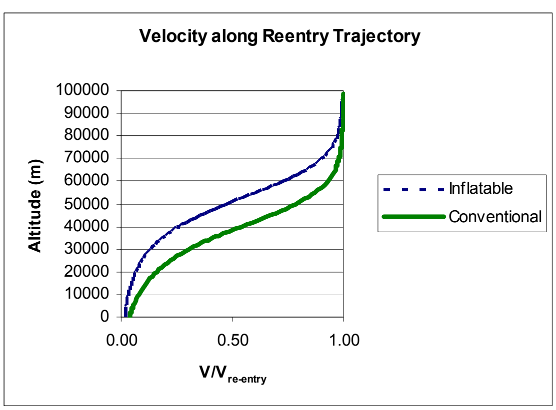
Details of the calculations can be found in Appendix 9.2.2.7.

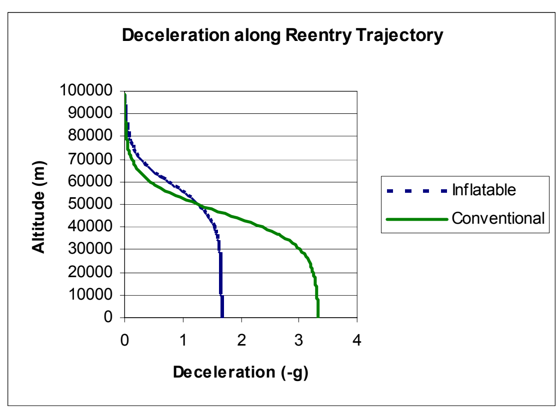
6.4.2.4.2.1. Inflatable Alternatives

As an alternative to the heavy ablative heat shield systems, researchers have refocused studies on inflatable technologies such as the ballute and Inflatable Braking Device (IBD). The IBD is an aerodynamically shaped cone to increase the surface area of the entry vehicle. The increase in surface directly affects the ballistic coefficient, β, of the vehicle, thereby decreasing the maximum heat and deceleration loads during entry.

A Primary IBD is inflated just before reaching the atmospheric interface. Once the maximum deceleration, pressure, and heat flux are passed, a Second IBD inflates to replace the parachute system at the appropriate altitude. A Third IBD may also be inflated to increase the size of the vehicle to achieve the required terminal velocity. Depending on the design, the landing system can either be one of the conventional landing systems or can be replaced by one of the stages of the IBD that cushions the impact.

EDL systems based on the conventional approach benefit from a strong and proven heritage, but depend on the use of a heavy heat shield and a dedicated landing system for re-entry. To offer a unique perspective on such a system's purported benefits of inflatable over conventional EDL systems, a parametric comparison study was performed on for an Apollo-class Earth return vehicle. Graphical results are shown in Figure 45. Because of the infancy of research in inflatable technologies, sizing was scaled from a proposed post-Beagle2, robotic mission to Mars.





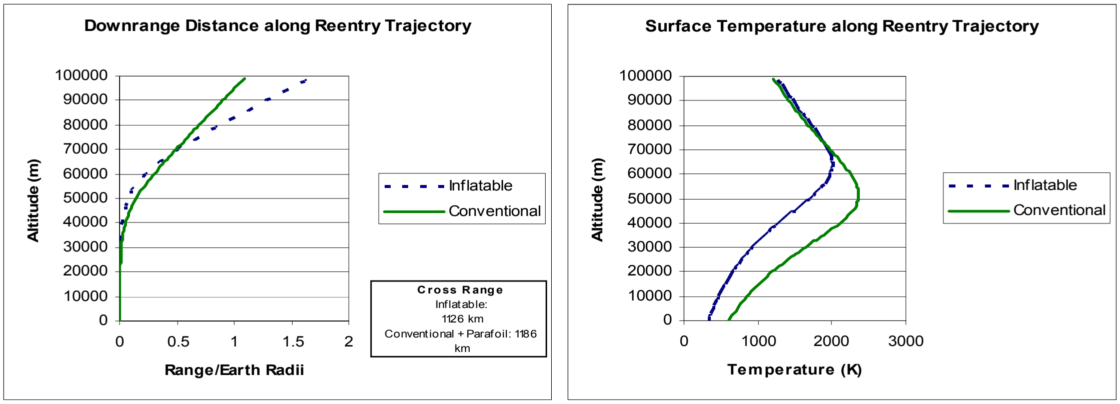


Figure 45: Parametric comparison of inflatable versus conventional Earth re-entry technology

6.4.2.4.2.2. Earth EDL Architecture Discussion

To subjectively assess the trade space of possible descent and landing system combinations, a pair-wise option comparison was performed in Figure 46. The methodology used in this study follows that of Section 6.4.2.4.1. The selective criteria and relative weighting chosen for this comparison were:

Minimum mass	0.40
Minimum development	0.25
Maximum cross-range	0.25
Minimum peak deceleration	0.10
Total	1.00

The mass, cross-range, and peak deceleration of the descent and landing systems were calculated using the methods previously stated. The minimum development cost was subjectively assessed based on current technology readiness level (NASA/TRL).

Minimum Mass — First Option \ Second Option	Inflatable	Infl. + Parafoil	Infl. + Retrorocket	Conv. + Parachute	Conv. + Parafoil	Conv. + Retrorocket	Score
Inflatable		1	1	1	1	1	5
Infl. + Parafoil	-1		1	-1	1	1	1
Infl. + Retrorocket	-1	-1		-1	-1	1	-3
Conv. + Parachute	-1	1	1		1	1	3
Conv. + Parafoil	-1	-1	1	-1		1	-1
Conv. + Retrorocket	-1	-1	-1	-1	-1		-5

Minimum Development — First Option \ Second Option	Inflatable	Infl. + Prafoil	Infl. + Retrorocket	Conv. + Parachute	Conv. + Parafoil	Conv. + Retrorocket	Score
Inflatable		1	1	-1	0	0	1
Infl. + Parafoil	-1		0	-1	-1	-1	-4
Infl. + Retrorocket	-1	0		-1	-1	-1	-4
Conv. + Parachute	1	1	1		1	1	5
Conv. + Parafoil	0	1	1	-1		0	1
Conv. + Retrorocket	0	1	1	-1	0		1

Minimum Deceleration — First Option \ Second Option	Inflatable	Infl. + Prafoil	Infl. + Retrorocket	Conv. + Parachute	Conv. + Parafoil	Conv. + Retrorocket	Score
Inflatable		0	-1	1	1	0	1
Infl. + Parafoil	0		-1	1	1	0	1
Infl. + Retrorocket	1	1		1	1	1	5
Conv. + Parachute	-1	-1	-1		0	-1	-4
Conv. + Parafoil	-1	-1	-1	0		-1	-4
Conv. + Retrorocket	0	0	-1	1	1		1

Maximum Cross-Range — First Option \ Second Option	Inflatable	Infl. + Prafoil	Infl. + Retrorocket	Conv. + Parachute	Conv. + Parafoil	Conv. + Retrorocket	Score
Inflatable		-1	0	1	0	1	1
Infl. + Parafoil	1		1	1	1	1	5
Infl. + Retrorocket	0	-1		1	0	1	1
Conv. + Parachute	-1	-1	-1		-1	0	-4
Conv. + Parafoil	0	-1	0	1		0	0
Conv. + Retrorocket	-1	-1	-1	0	-1		-4

Figure 46: EDL pair-wise option comparison

The analytical calculations assume the use of an Apollo-class Earth re-entry capsule. The conventional systems rely on an ablative SLA561V heat shield. The EDL systems that include inflatable devices substitute parachutes with the Third IBD. The conventional system that includes retrorockets for touchdown deceleration also includes the release of drogue parachutes. Notice that the purely inflatable system has the least mass, the conventional system with parachutes require the least development time, the inflatable system with retrorocket decelerators produces minimum peak deceleration, and the inflatable system with parafoil technology has the maximum cross-range capability.

The options are subjectively ranked, using the selection criteria weightings listed above. Notice that the ranking of the entry and descent architectures follows the distribution of the system mass. This ranking should be used only as a qualitative estimation that is inherently dependent on the relative selection criteria weightings chosen.

Table 12: EDL option ranking and system mass for an Apollo-class Earth re-entry vehicle

Reentry Arch.	Total Score	Rank	Mass (kg)
Inflatable	2.6	1	825
Conv. + Parachute	1.05	2	1049
Infl. + Parafoil	0.75	3	1050
Conv. + Parafoil	-0.55	4	1104
Infl. + Retrorocket	-1.45	5	1235
Conv. + Retrorocket	-2.65	6	1249

The parametric comparison of Apollo-class inflatable and conventional Earth EDL systems yielded interesting results. The inflatable system had 15% to 20% less mass, 40% to 45% less maximum deceleration, 20% to 25% less average surface temperature than a conventional heat shield system with a cross-range capability that is roughly equal to a parafoil system. Additionally, the inflatable device can serve as an air-cushion for ground-based landings or as a flotation device for water-based landings. Although the inflatable systems seem promising, they are untested. Consequently, the system's cost, reliability, and safety are difficult to estimate. A test flight in February 2000 from Babakin Space Center was only partially successful. As a result, the design of the return vehicle should be modular. Instead of integrating the heat shield into the spacecraft body, like the Apollo Command Module, the ablative heat shield should be designed as a module of the return vehicle. When inflatable or other next-generation technology has been proven for human spaceflight, the conventional system could be replaced with minimal cost. Furthermore, a modular ablative heat shield might support the construction a reusable return vehicle design.

6.4.2.5. Landing Site

The chosen EDL architecture directly influences the choice of landing site. The Apollo Command Module landed in the water to reduce the touchdown impact of its unpowered descent. Similarly, Soyuz fires a small solid-motor thruster just before touching down on land. In addition, uncertain atmospheric density, navigation errors, and unanticipated winds can push an uncontrollable vehicle, such as a spacecraft on parachutes, away from its intended landing location. The Apollo Command Module landed in the South Pacific Ocean to accommodate its large landing footprint. Contrastingly, the steerable parafoil used by the X-38 lifting body permits a smaller landing area and a touchdown on land.

There are roughly three recovery possibilities: land, sea, and lake/coastal. Recovery operations on land can be relatively fast and inexpensive by utilizing existing infrastructure. Land-based touchdowns require a sink rate below 7.5 m/s, whereas water landings can sustain velocities of about 9.5 m/s. Because land-based landing g-loads can be 2 to 3 times higher than water-based landings, a crushable nose or inflatable air-cushion is required. To distribute the impact forces over the entire lower surface of the spacecraft, a self-leveling honeycomb might be used to plastically deform to absorb the shock of the landing. Possible materials for the honeycomb might include lightweight metal alloys, carbon-carbon composites, and high density styrene polymers. This design would provide significant mass savings over conventional landing mechanisms because a composite honeycomb weighs a fraction of aluminum or steel.

Sea-based landings tolerate higher impact velocities, reducing the mass needed to decelerate the entry vehicle. The mass needed for flotation bags might partially offset this benefit, unless an inflatable landing system is used. Water also provides immediate cooling of the overheated spacecraft. The Apollo program demonstrated that recovery operations at sea can be costly, and can be adversely affected by poor weather.

Sea and lake/coastal-based landings share similar properties, except that lake or coastal-based landings have lesser infrastructure costs. A lake-based landing could use existing Coast Guard recovery capabilities, instead of deploying a large Naval Carrier Battle Group. Possible landing sites for such a landing might be the Gulf of Mexico or the Great Lakes. A lake/coastal-based landing requires a re-entry system with a large cross-range capability to maintain a precise trajectory (inflatable or parafoil). An inflatable system might provide both flotation capabilities for such a landing and air-cushioning for an inadvertent land-based touchdown. Because of its cost benefits and advantageous qualities, the chosen EDL architecture (see Section 6.4.2.4.) would be well-suited for a lake/coastal-based landing site. When inflatable technology is validated and replaces the conventional heat shield, it will also replace the separate flotation device.

6.4.3. In-Space Options

A number of trades were examined in determining the forms for the extensible Moon/Mars mission architecture. The space transportation system is a network of modules that was developed from the trades described below. It was assumed that the space transportation system does not require the use of the International Space Station (ISS) as an assembly or return point. This was done to ensure that NASA can divest itself from the ISS and STS to meet the Space Exploration goals within budgetary and political constraints.

6.4.3.1. Transportation Modules

6.4.3.1.1. Mass Transportation Vehicle

The Mass Transportation Vehicle (MTV), as presented in the baseline mission, is made of the Crew Operations Vehicle (COV) and the Habitation Module (HM). It is part of the

Crew Exploration System (CES), which consists of all the forms necessary to support manned exploration of the solar system.

The different phases that the MTV is required to perform is in space transportation. The first trade study, which led us to dissociate the Earth to LEO transportation from the In Space transportation is described below and shown in Figure 47.

- From Earth to LEO and back to Earth
- In-space travel: LEO to another destination in space

6.4.3.1.1.1. MTV Trade Study

For this initial trade, we considered only small range exploration (up to 8 crew and 40 days), which excluded Mars exploration. The results and lessons learned from this short study led us to the choice of separating the forms as much as possible. Most notably, it will be highlighted that separating the function of in-space and Earth to LEO transportation is a beneficial choice. In this initial study, the CES (Crew Exploration System) performs the following: it goes from Earth's surface, travels in orbit or further (but middle range) and comes back to the Earth surface's surface at the end of the mission.

The basic functions that are required are listed below:
- Support and shelter the crew during launch
- Provide crew escape in case of a launch emergency
- Provide a habitat for the crew during in-space transportation
- Provide energy to displace the crew module during in-space phases
- Perform landing on a selected site

For performing these functions, we have studied two forms: service module and crew module (called HM here) – the names are internal to this trade study.

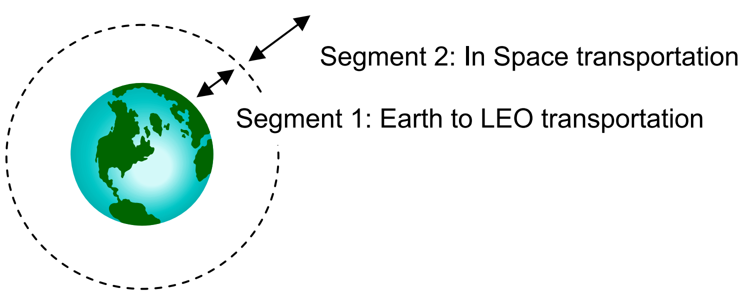


Figure 47: Mission segmentation

An interactive model was developed to determine the mass of various components of a MTV based on the number of crew and the mission duration. The model is described in Appendix 9.1.1.

A strong inter-level dependence exists between technologies used for the various functions to be performed. For example, the mass of the re-entry system depends on the volume of the capsule, on the type of deceleration device, etc. Figure 48 shows the decision tree for key technologies/options that have been traded. Links between levels represent preferable or feasible options. For example, the lake/coastal landing site option requires a re-entry system with a large cross-range capability for precisely landing into a small body of water (inflatable or parafoil).

Three main categories of HM have been identified, as shown on the figure below:

- *Combined* (one unique form which transports the crew for all the sections; Earth to LEO, In-Space and Landing)
- *Separate* (two different forms; one performs In-Space transport and the other performs Earth to LEO and landing)
- *Flexible* (the same core transports the crew, but minor modifications are made so that it is able to land or perform In-Space transportation)

For each category, the vehicle could be expendable or reusable (see Figure 49).

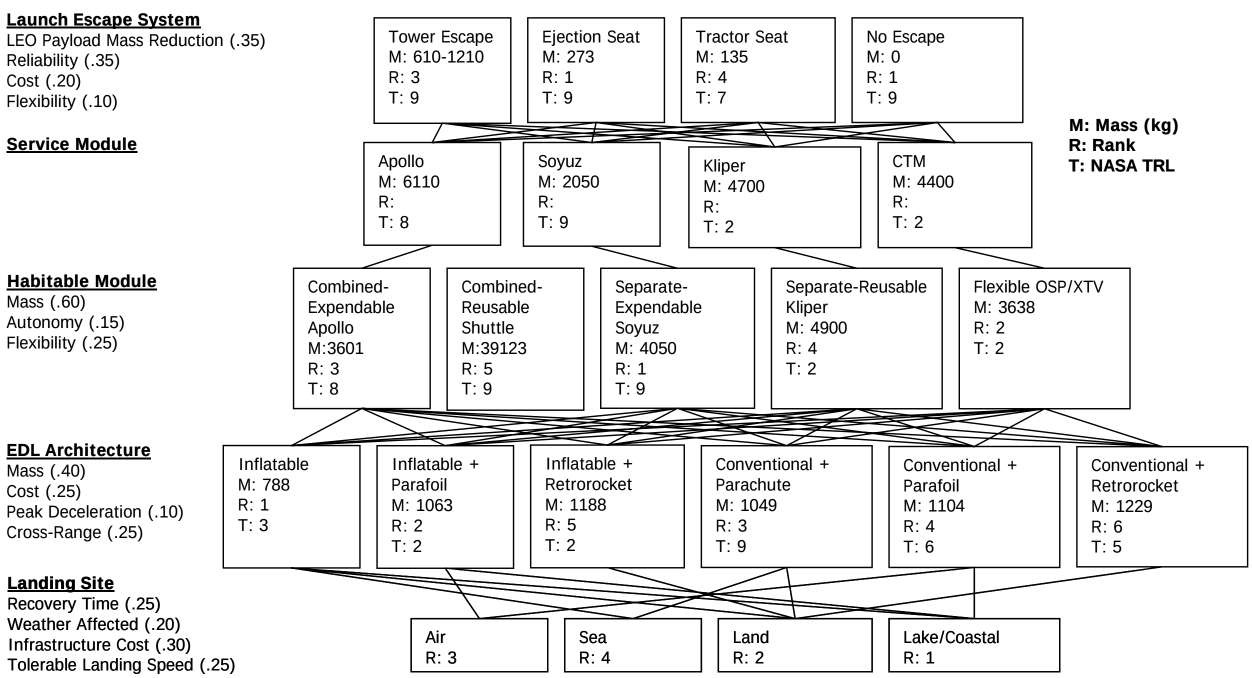


Figure 48: Elements of the MTV, assuming a crew of three for a ten-day mission

	Combined		Separate		Flexible
	Expendable	Reusable	Expendable	Reusable	Expendable
Reentry vehicle	Apollo CM	Shuttle	Soyuz DM	Kliper DM	OSP
Orbital Vehicle	Apollo CM	Shuttle	Soyuz OM	Kliper OM	XTV

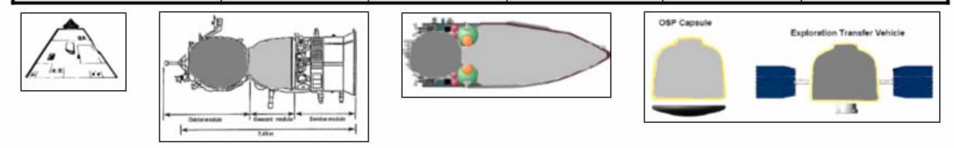


Figure 49: Classification of existing crew transport modules

To assist in decision-making, three metrics were used: mass, TRL, rank. The Technology Readiness Level (TRL) methodology is a NASA metric based on a nine-stage process ranging from the basic principle being observed and reported (#1) to flight proven through successful mission operations (#9). The rank-measurement enabled a normalized comparison across elements. For example, the launch escape system shouldn't be chosen for the same reason as the EDL elements. Each element of the CES (each row in the network) has its own criteria. Ranking also enables comparison between each option in a row with different metrics, while trying to assess each option as objectively as possible for trading different measures of performance/priority. Ranking also allows weighing metrics by priority. Depending on the priority, you can weigh the metric so that the final ranking reflects priorities.

Three of the many combinations were considered in detail:

- ***Modern Apollo CM - MCM -*** Tower Escape, Modern CM and SM, Conventional Re-entry, and Sea Recovery.
- ***Improved Soyuz - IS (Best Rank) -*** Ejection Seat, Soyuz DM & OM & SM, Inflatable Re-entry, and Lake Recovery.
- ***New Type - XTV (Lowest TRL) -*** Ejection Seat, OASIS CTV & CTM, Inflatable Re-entry, and Lake Recovery.

The mass of these configurations is shown in Figure 50.

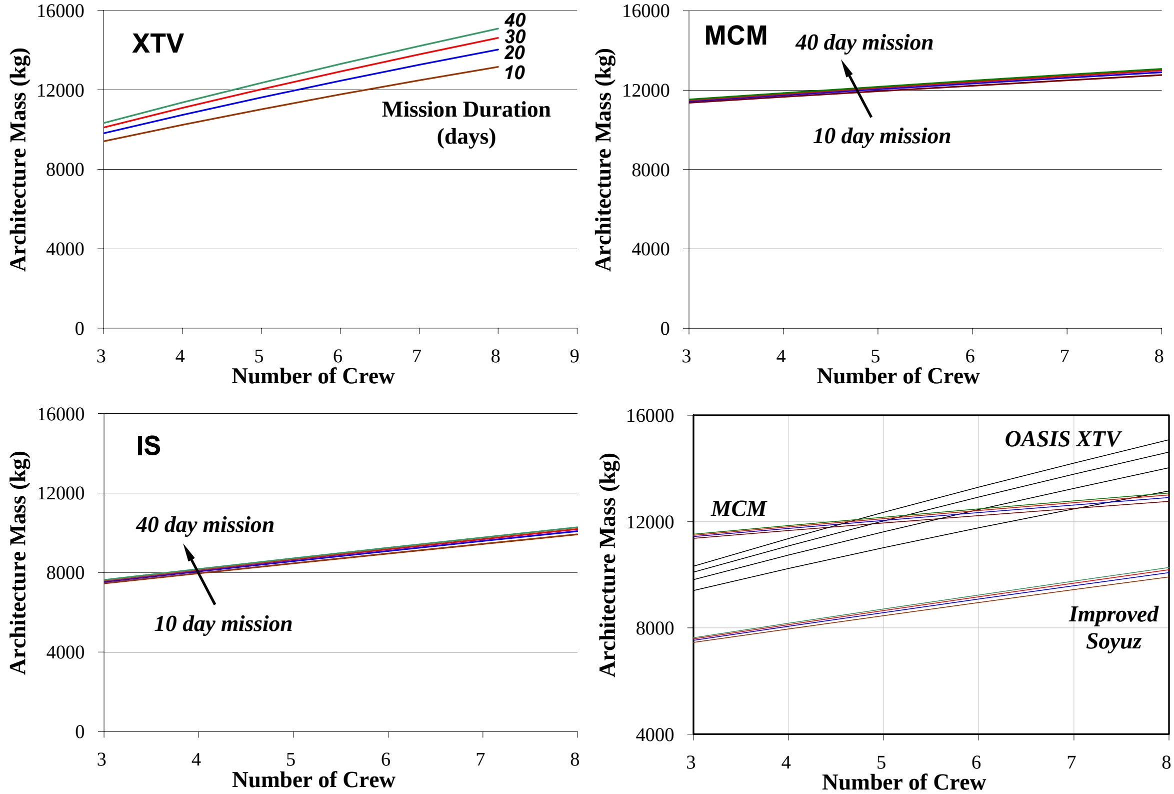


Figure 50: Configuration masses (10-day to 40-day missions)

The Improved Soyuz (IS) architecture is a separate-expendable type of Habitable Module. It is comprised of all of the best-ranked components for the launch escape system, habitable volume, service module, re-entry system and landing site. This architecture has the Ejection Seat Launch Escape System, which is ranked as the best launch escape system based on the metrics of mass penalty, reliability, cost and weight. As can be expected, if the relative weights are altered, the final launch escape system ranking may change. The separate expendable Soyuz crew module had the best overall ranking in the following categories (minimum launch mass, minimum development cost, autonomy and flexibility). Similarly, the best-ranked technique of re-entry was inflatable re-entry. This was based upon the minimum mass, minimum development cost, minimum deceleration and maximum cross-range. The method of landing that was best ranked was a water landing. This was based upon the minimum recovery time, least weather affected, minimum infrastructure cost and maximum landing speed.

Scaling was performed for both the launch escape system and the crew module. Based on the analysis, which proposes that mass of the vehicle is a function of both the mission duration and number of crew, it is clear that the length of the mission from Earth to LEO does not greatly affect the overall CES mass. It should be noted that the mission durations highlighted here are completed arbitrary and were chosen to illustrate that the required increases in structure and CES component mass will not be the primary factor affecting mass increases.

The “Modern” Apollo uses all of the same methods of re-entry and transportation modules as the original Apollo, however a structural analysis was performed, which determined a new vehicle mass based on modern materials. For this architecture, the COV mass is greater, but it is still moderately dependent on the mission duration.

Using the same methods of launch escape and re-entry as the Soyuz based architecture discussed earlier, the Oasis XTV-CTM combination was chosen as a third architecture to present because of its lowest TRL, but also second-best rank after the Improved Soyuz. For this architecture, the mission duration had a greater influence on the overall architecture mass compared to the other architecture. The vehicle structure comprises a much greater proportion of the overall mass than in the case of the “Modern Apollo”. Since the mission duration is related to the habitable volume and the external vehicle surface area scales the structural mass, the overall mass is more greatly affected for this case. Even though this configuration had the lowest TRL, which could indicate the use of advanced or modern technologies, other configurations had lower masses.

A summary of the three configurations is shown in Figure 51.

	Improved Soyuz ***Best Rank***	**Modern Apollo CM**	***Lowest TRL***
Total Mass (kg)	7,177	11,369	9,133
Total Rank [4;19]	4	13	5
Average TRL	7.5	8.5	4

Figure 51: Three COV configurations for launch from Earth to LEO

The mass of the COV was approximated as 5708kg from the Model described in Appendix 9.1.1. It was assumed that the EDL mass could be neglected and the additional mass required to aerobrake at Mars was 15% greater than the COV mass. Similarly, the additional mass required to aerobrake at Earth was 6% greater than the COV mass (Larson, 1999).

6.4.3.1.1.2. Baseline Transportation Form Selection

The main lesson learned from this trade study was that **the forms used for Segment 1** (Earth to LEO and back to Earth) **and Segment 2** (In-Space transportation) **should be separate and expendable**. Such a separation leads to a high rank and low mass within the framework of this trade study. A vehicle that performs all the functions at once (such as the Shuttle) is sub-optimal and leads to additional mass, especially when it is reusable.

6.4.3.1.1.2.1. Transportation Form for Segment 1

The conclusions highlighted in the previous Section guided the from selection process for Segment 1 (Earth to LEO and back to Earth) of the baseline mission. The paragraph below explains why the Modern Command Module (MCM) form was selected for the baseline mission.

When determining the type of COV to use for launch and re-entry, the mass of the three configurations without their respective Service Modules were determined. From this analysis, the Modern Apollo Command Module was observed to have the lowest mass (5,200kg) and was selected as the form for Earth launch at the start of the mission and Earth EDL at the end of the mission. Since this vehicle has approximately three times less the habitable volume per person as compared to the OASIS XTV, this may indicate that separating the function of crew habitation and re-entry is beneficial to overall mission mass reduction.

6.4.3.1.1.2.2. Transportation Form for Segment 2

For Segment 2 (In-Space transportation), a modular approach was taken to ensure increased commonality between the forms required to complete a Moon and Mars exploration mission.

6.4.3.1.2. Rovers to Support Planetary Surface Operations

Surface exploration of the Moon and Mars will require a diverse array of robotic capabilities. Mobility systems such as rovers are critical to achieving scientific missions and accomplishing a variety of operational requirements. Use of rovers will increase effectiveness and safety while reducing costs. Tasks to be performed include instrument deployment, soil manipulation, and human transportation.

- Instrument deployment (<200kg)
    - Understand local geological context of landing site
        - High Resolution Multispectral Imaging (required)
        - Microscope (required)
        - Visible Short Wave Infrared Imaging Spectroscopy
        - Thermal Infrared Imaging Spectroscopy
        - Laser Induced Breakdown Spectroscopy
        - Raman Spectroscopy
    - Understand current climate conditions of landing site
        - Pressure
        - Temperature
        - Wind speed at multiple heights
        - Water vapor abundance
            - Tunable Diode Laser
            - Mass Spectrometer
            - Hygrometers
    - Access material for close investigation
        - Deploy instruments to surface, robotic arm

        - Access fresh interiors of rocks
            - Drilling
            - Abrading
            - Coring
            - Thin sections
        - Collect soils

- Soil manipulation (<500kg)
    - In-situ resource utilization
    - Burying nuclear power units
    - Building bunkers to protect astronauts from solar radiation
    - Preparing foundations for bases

- Human Transportation (~1000kg for open and ~6000kg for pressurized)
    - Explore geological sites
    - Move to emergency shelter
    - Reach other landed modules
        - Collect supplies
        - Reach pre-positioned habitat
    - Aid in construction of lunar base
        - Deploy power systems
        - Lay cables
        - Erect antennas, radiators, and photovoltaic panels

Four categories of rovers exist. Automated, autonomous rovers are equipped with artificial intelligence for hazard avoidance and are capable of collecting and communicating scientific data. Autonomous rovers on precursor missions may identify potential landing sites for human missions and refine Lunar/Martian scientific goals. Remote controlled rovers may be utilized by human operators on Mars for missions of varying duration. Unpressurized rovers can be driven by astronauts but should only support exploration within walking distance of living quarters. The Apollo Lunar Rover Vehicle and Soviet Lunakhod are legacy unpressurized rover systems (Arno, 1999). Pressurized rovers possess a self-sustained life support system and enable extended excursions for astronauts in a shirtsleeve environment. The functional requirements of the Rovers for the various missions are shown in Table 13.

Table 13: Rover functional requirements

Category	Unmanned Precursor	Short-Stay	Medium-Stay	Extended-Stay
Automated, Autonomous	**x**			
Remote Controlled		**x**	**x**	**x**
Unpressurized		**x**	**x**	
Pressurized				**x**

6.4.3.1.2.1. Modular Architecture for Unmanned Rovers

To support these operations, modular rover architecture is proposed whereby rover for instrument deployment and soil manipulation tasks can be assembled from an inventory of modules to accomplish a specific task. This inventory of modules includes actuated joints, links, end-effectors, sensors, and mobility units. Initial configuration and reconfiguration can be done autonomously or by an astronaut (Farritor, 2000). A modular architecture for robotic surface operations may represent a new paradigm in NASA robot design but it does not rely on developing new technologies.

Mission Scenario → Requirements → Design Specifications → Modular Inventory

For our design space—multiple missions to the Moon and Mars—a modular rover architecture was selected for a variety of reasons:

1) **Efficiency**: for missions requiring a wide variety of tasks a single modular system is superior to creating a dedicated robot for each task, additionally, packaging modules on launch vehicle may be more efficient than packaging assembled robots in terms of mass and volume
2) **Adaptability**: modules enable construction of novel robots, including robots for tasks that are not foreseen
3) **Reliability**: failed modules can be replaced, different configurations can potentially accomplish the same task, Mars missions should place a premium on this reliability
4) **Extensibility**: fits spiral development model of increasing capability over exploration program life
5) **Cost**: standardized modules will limit non-recurring research and development

6.4.3.1.2.2. Human Mobility

A variety of design architectures are possible with open and pressurized rovers to transport astronauts on the Lunar and Martian surface. For mobility, tracks, screw drives, legs, rockets, and balloons are all available, although wheels offer the greatest overall performance when considering energy, ground pressure, ground clearance, reliability, and human factor requirements. For rover structure and pressure shell, an inner pressure shell of aluminum alloy and an outer shell of aluminum and carbon/graphite epoxy offer a strong baseline design. The communications system must maintain contact with all manned rovers at all times for navigation, scientific investigations, and safety. As a legacy system with proven reliability, hydrogen-oxygen fuel cells are an ideal power source. For life support, an open system is recommended given the relatively short excursions and high mass penalty (~1000kg) to recover consumables (Arno, 1999).

An open, unpressurized rover is limited to sorties of 10km for safety considerations (within walking distance of their surface habitat). These vehicles will typically support a crew of two. Therefore, although not a requirement, a pressurized rover capable of sorties ranging from 50-500km and 12-400 hours duration is a recommended option for extended-stay missions (Arno, 1999). The pressurized rover will have an airlock to support EVA. Three astronauts will crew each pressurized rover during regular sorties in order to enable two-astronaut EVA with a third remaining in the vehicle. For safety considerations, it is recommended to have a pressurized rover in reserve to conduct rescue missions of astronauts beyond walking distance of the main base.

6.4.3.1.3. Habitation Module

The Habitation Module (HM) will sustain human life for an extended-duration Mars mission. This module will be launched in two pieces and assembled in LEO, allowing the overall volume to not be limited by the minimum launch volume requirements (see Figure 52). Since there are two modules, none, one or both can be used for each mission, depending on the mission requirements for duration and crew size. This module must have propulsion capabilities to perform docking maneuvers in both Earth and Mars (or Lunar) orbit.

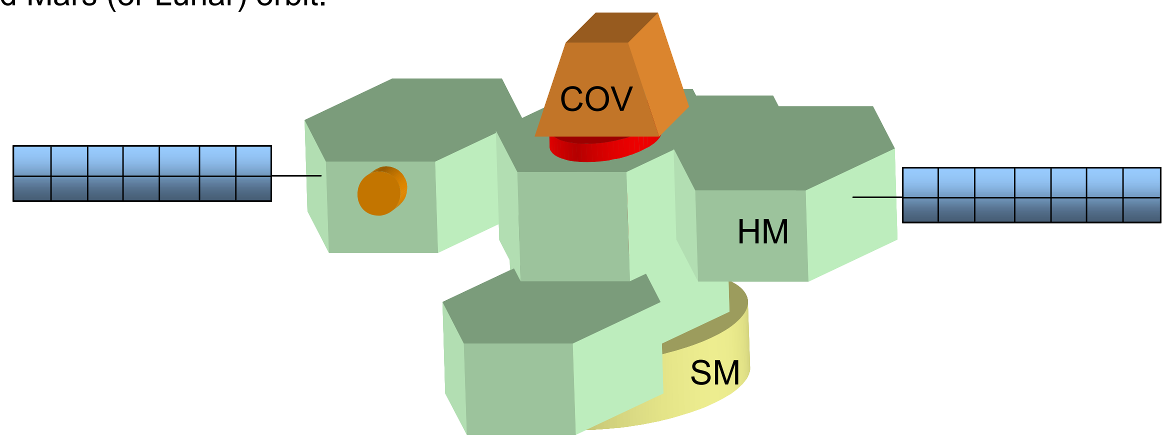


Figure 52: Mars/Moon Transfer Vehicle (MTV)

Following the Mars study performed by (Larson, 1999), the mass of the HM was calculated as ~55,000kg for a crew of six, depending on a number of critical factors (mission duration, type of radiation protection, life support, supplies, aeroshield and power requirements).

The HM is composed of separable modules that promote significant modular spacecraft design flexibility. Six of these modules are combined in two groups of three and platform, forming one large volume required for a Mars mission. Based on Larson (1999), it was assumed that a habitable volume of 20m^3 per person was required for a 6 crew, 6-month mission. For this analysis, 30m^3 was specified per person. It was also assumed by Larson (1999) that 33% of the total volume was assumed to be habitable. Therefore, a total volume of 540m^3 could be created by 6 octahedrons as shown in Figure 52, each with a 5.6m diameter. As shown in Figure 53, this volume agreed well with other pressurized spacecraft volumes.

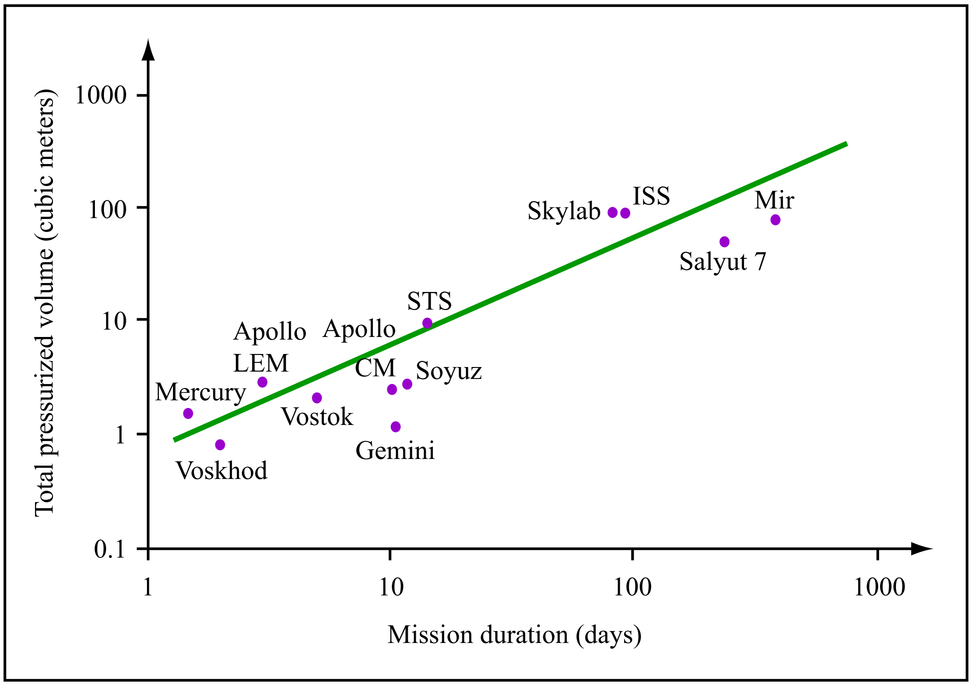


Image by MIT OpenCourseWare. Adpated from Kennedy, 2002.

Figure 53: Historical space habitat pressurized volume (Kennedy, 2002)

Based on Larson (1999), an estimate for spacecraft mass (based on current technology) could be made:

$$m = 592(NDV)^{0.346}, \tag{1}$$

where m is the total vehicle mass, N is the number of crew, D is the mission duration in days and V is the total spacecraft volume in m^3. Based on the missions in question, the results compared well with the calculated 55,000 kg mass for the Habitation Module.

A summary of the module masses is given in Table 14.

Table 14: Baseline module masses

Mission Module Masses (kg)		
COV	Crew Operations Vehicle	5,700
HM	Habitation Module	55,000
SM1	Service Module # 1	-
SM2	Service Module # 2	-
MCM	Modern Command Module # 1	5,200
ML1	Mars Lander # 1	15,000
ML2	Mars Lander # 2	15,000
LL1	Lunar Lander # 1	10,000
LL2	Lunar Lander # 2	10,000

6.4.3.1.4. Pre-positioning

6.4.3.1.4.1. Pre-positioning Overview

Pre-positioning items needed for a space exploration mission is the act of sending hardware or any other required cargo to its respective destination in advance of the "main" portion of the exploration mission. In the case of this space exploration architecture, the "main" portion of the mission is the launch through the landing of the human crew.

Non-time critical components with lifetimes of appropriate length that are not required for the crew to have available during transfer to their destination are good candidates for pre-positioning.

The pre-positioning of mission components will likely be accomplished with an efficient propulsion system technology such as electric propulsion. Once this propulsion technology is successfully demonstrated for large masses, electric propulsion will then be used to send more exploration equipment. However, it should be mentioned that until the ability of the crew to live off re-supply provided by electric or other efficient propulsion systems, or "live off the land" has been demonstrated, the crew should carry all of the material with them needed for safe return.

6.4.3.1.4.2. Pre-positioning Benefits

6.4.3.1.4.2.1. Mass Reduction

One major reason to pre-position items for a space exploration mission is to take advantage of being able to transport this cargo using a more efficient propulsion system than would be used otherwise. This results in a reduced overall mass of the pre-positioned module. For the purposes of this project, it is assumed that this propulsion system will be a form of electric propulsion.

Electric propulsion, while more efficient, has much less thrust than engines using chemical propellant. This requires a longer time of flight to get the cargo to its destination. This necessitates that the cargo being pre-positioned is not time critical.

This increased efficiency of electric propulsion over chemical is due to a significantly increased specific impulse, I_{sp}. The famous "rocket equation," shown below, is used to exemplify why this is beneficial.

$$\Delta V = I_{sp} g_0 \ln\left(\frac{m_i}{m_f}\right) \tag{1}$$

where ΔV is the change in velocity provided by the engine burn, g_0 is acceleration due to gravity on Earth, m_i is initial mass, and m_f is the final mass after the engine burn.

Solving for the final mass:

$$m_i = m_f e^{\frac{\Delta V}{I_{sp} g_0}} \tag{2}$$

This shows that an increase in I_{sp} results in a decrease of initial total mass required. This is the resultant benefit of a more efficient propulsion system. The relaxed time of flight of the pre-positioned mission phase, compared to the crewed segment, makes this advantage possible.

The initial mass benefit due to pre-positioning for this project can be seen in Table 15 and Table 16 for missions to the Moon and Mars, respectively.

Table 15: Mass benefit using pre-positioning for a Medium Moon mission

Total Mass in LEO Without Pre-positioning (kg)	Total Mass in LEO With Pre-positioning (kg)
101,000	80,000

Table 16: Mass benefit using pre-positioning for an Extended Mars mission

Components Pre-positioned	Total Mass in LEO (kg)
SH	745,000
SH, Landers	463,000
SH, Landers, Earth-return fuel	379,000

For missions to the Moon, Table 15 shows a significant mass savings due to pre-positioning non-crewed mission components at the destination. For missions to Mars, Table 16 shows an increasing benefit as more mission components are pre-positioned. The use of efficient propulsion systems such as electric propulsion combined with a relaxed time of flight requirement allow for such a mass savings.

6.4.3.1.4.2.2. Launch Vehicle Selection

A more subtle benefit for pre-positioning non-time critical mission components is the ability to use less expensive, non-human-rated launch vehicles to launch these pre-positioned components. The launch vehicle used to launch the crew will likely be a new or partially new heavy-lift launch vehicle design. The human-rated launch vehicle is likely to cost more per launch than launch vehicles such as the Evolved Expendable Launch Vehicles.

For example, EELVs could be used to launch “packages” to be pre-positioned in advance of the main crewed portion of the mission as opposed to launching portions of this pre-positioned cargo with the crew on a more expensive heavy-lift launch vehicle.

In addition, if enough mission components are pre-positioned using launch vehicles such as EELVs, a reduction of the payload mass requirement for this new man-rated launch vehicle could be realized.

6.4.3.1.4.2.3. Risk Reduction

An inherent advantage with pre-positioning is the reduction of mission risk. This risk reduction is possible for two main reasons. First, mission planners on Earth would know in advance of the launch of the crew if the pre-positioned components were successfully deployed in their desired locations. Second, mission planners would also know if these components are functioning properly before the launch of the crew.

If a pre-positioned mission critical component is found to not be functioning properly before the crew launches on their mission, the mission planners have several options to solve the problem. One option is to try fixing the problem via communication with the malfunctioning component and delay the launch of the crew if required. Another option is to launch a replacement component either using chemical or electric propulsion depending on the mission schedule. Finally, a replacement component could be sent with the crew when they launch as scheduled.

Mission planners do not have these same options if such components are not pre-positioned. If everything is sent to the destination at the same time and a mission critical component malfunctions, the only options available are to rely on a backup component or fix the malfunctioning component. This requires that more redundancy be incorporated into the design of the components used in the mission. This may result in increased overall mission mass, which translates to increased cost.

Finally, the risk of launching many mission components using existing launch vehicle technology such as EELVs will likely be less than sending the same cargo using a new or partially new heavy-lift launch vehicle.

6.4.3.1.4.3. Pre-positioning Penalties

Some drawbacks exist for incorporating pre-positioning into a space exploration mission design. First, pre-positioning may require an increase in the number of required Earth launches. This may increase the risk of a launch failure. Second, the designs of the pre-positioned components will need to accommodate increased component lifetime requirements to account for an increased time of flight and the time between the arrival of the component(s) at the destination and the arrival of the crew. A final penalty to pre-positioning mission components is the increased time required of mission control personnel for monitoring and control of pre-positioned components. These personnel will likely need to monitor these components for an extended period of time before the launch of the crew. If everything were launched at the same time, less time would be required for monitoring mission components.

6.4.3.1.5. Crew Module Scaling

Varying the mission duration and the number of crew greatly affects the overall mass of the habitable crew vehicle. It was necessary to develop a method of accurately scaling the habitable crew vehicle to predict the required launch to LEO crew vehicle mass.

By comparing the masses of various crew modules (Gemini, Mercury, Apollo CM, OASIS CTV and Soyuz), while holding the habitable volume constant, the various architectures were compared. By answering the questions shown in Figure 54, a new vehicle mass can be obtained.

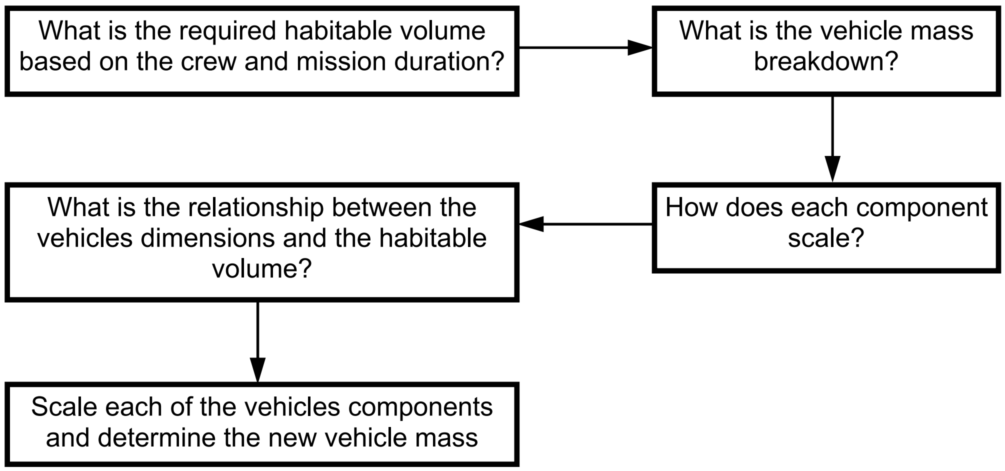


Figure 54: Flowchart of scaling analysis

1. An equation was developed for the habitable volume as a function of the number of crew and mission duration. Based on these inputs, the volume was calculated.
2. The mass breakdown of the vehicle components (i.e. structure, avionics, communications equipment, etc) is well documented for existing crew vehicles and was determined for the Apollo CM and OASIS CEV (see Appendix 9.1.2.)
3. Predict how each of the vehicle components scale (i.e. external surface area, number of crew, etc)
4. Based on the known vehicle geometries, when increasing the habitable volume of a vehicle, maintain the relative sizes of specific vehicle dimensions.
5. Scale each component and re-calculate the vehicle mass.

It should be noted that the heat shield mass was determined in a separate analysis and was not considered in this analysis. Using modern materials and structural analysis on the Apollo CM, the mass might be reduced. For comparison, the materials used for the OASIS CEV were used for the Modern Command Module (MCM) and Crew Operation Vehicle (COV) structures. From a detailed analysis described in Appendix 9.1.2., Figure 55 was obtained.

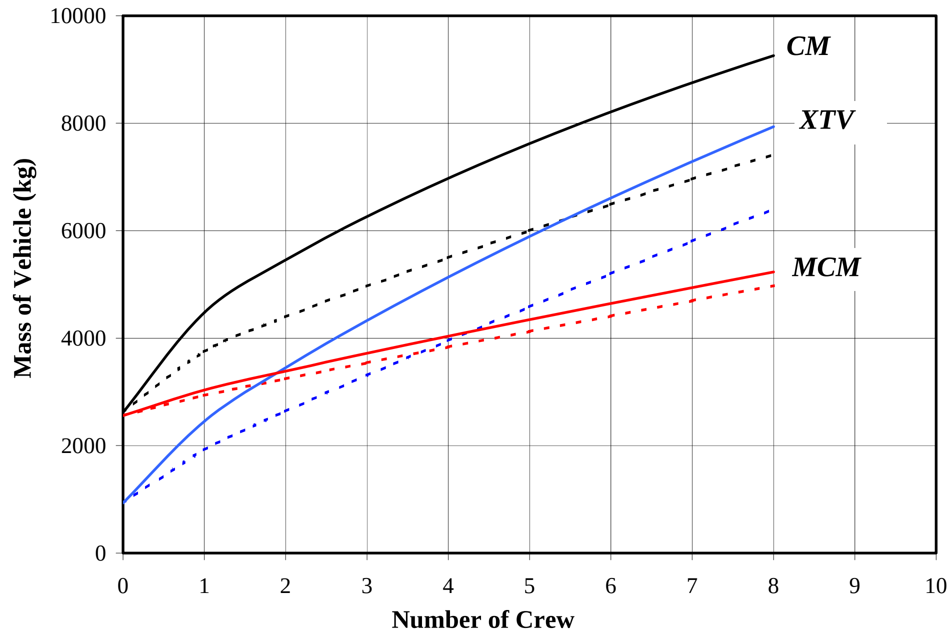


Figure 55: Vehicle mass scaling (broken line: 3-day mission, solid line: 30-day mission)

The masses shown in Figure 55 do not include a heat shield because a comparison can be made between functionally equivalent forms. Details of heat shield sizing are described in Appendix 9.2.2.7.2.

As the mission duration increases, the habitable volume increases, causing the vehicle surface area to increase correspondingly, to maintain the vehicle proportions. As explained earlier, when the surface area increases, all of the mass breakdown components that scale with surface area increase. Since the majority of the mass breakdown components that were scaled for the Apollo CM scaled with the vehicle surface area, there was a significant overall vehicle mass dependence on the mission duration and/or crew. When the mass per surface area was reduced (as was the case for the Modern Command Module analysis), the influence of mission duration increased the overall mass of the Modern Command Module only slightly (see Figure 55).

6.4.3.1.6. Moon and Mars Entry, Descent, Landing, and Ascent (EDLA)

Comic strip removed due to copyright restrictions.

Figure 56: The reality of designing an EDL system (Amend, 2004)

EDLA phases and options for lunar and Martian missions, including Earth return, are summarized in Figure 57. The connected boxes identify a possible human mission scenario. Although aerocapture has never been employed on a human mission, it should substantially reduce the total required vehicle mass (see Section 6.4.3.3.4). Human missions require drag- and lift-modulated entry at Mars and Earth to lower the peak acceleration, augment the trajectory precision, and increase control.

Flow chart image removed due to copyright restrictions.
Eckart, P. "The Lunar Base Handbook: An Introduction to Lunar Base Design, Development, and Operations." *Space Technology Series*. Edited by W.J. Larson. New York, NY: McGraw-Hill, 1999.

Figure 57: Trade space for EDLA missions (Larson, 1999)

Mission phases that are common for both Moon and Mars EDLA are highlighted in Figure 57. In general, the thin atmosphere of Mars does not allow parachutes to slow a spacecraft sufficiently for a surface landing. Still, use of a parachute system does reduce the Δv requirements for landing. A typical Martian EDL system might include a conventional heat shield, parachutes, and a final retrorocket decelerator. The thermal protection and parachute systems can be modeled using the equations of motion, as discussed in 5.1.4.2. The airless Moon necessitates an all-propulsive descent and landing. Thus, both Moon and Mars missions share a powered descent phase to varying degrees. Additionally, both missions require a propulsive ascent return stage to launch off the planetary surface. The missions also have a common Earth re-entry phase that includes aerocapture, a parachute and parafoil system, and a final lake/coastal-based touchdown (see Section 6.4.2.5.).

6.4.3.1.6.1. Lander requirements and commonalities

The propulsive Δv requirements for the phases of lunar and Martian EDLA are summarized in Table 17.

Table 17: Propulsive Δ*v* requirements for Martian and lunar EDLA

Δ*v* [km/s]	Mars	Moon
De-orbit	0.111	0.019
Descent and Landing	0.630	1.862
Ascent and Rendezvous	4.140	1.834

The Martian de-orbit Δ*v* involves transferring from a 500 km circular orbit to a 500 km × 20 km Mars transfer orbit. The initial descent phase entails aeromanuevering and parachute deployment that does not require any propulsive Δ*v*. The powered descent phase includes ± 4.5 km lateral translation capability for dispersion accommodation and landing target site redesignation. The lunar de-orbit maneuver is from a 100 km circular orbit to a 100 km × 17.5 km transfer orbit. The powered descent and landing phase includes initiation, braking, pitch-up/throttle-down, and vertical descent to surface. The Martian and lunar ascent/rendezvous Δ*v* include boost and circularization at 500 km and 100 km altitudes, respectively. To build a sustainable EDLA architecture for the variably sized Mars and Moon missions, the Lander's functionality requirements for each mission type are integrated in Table 18. The commonality with the Lander required for the Earth return portion of each mission is shown in the figure as well. This architecture design has deliberately chosen Lander designs such that significant portions of the designs are common amongst all missions.

Table 18: Integrated Lunar and Martian Lander functionality requirements

Landers (LL/ML)	Moon			Mars			
	Short	Medium	Extended	Phobos	Short	Extended	Extended+
Dock with COV/HM in Orbit	X	X	X	-	X	X	-
Dock with ISPP-SHM on Surface	-	-	-	-	-	X	-
Transfer crew of 6 from Orbit to Surface and Back	-	-	X	-	X	X	-
Transfer crew of 3 from Orbit to Surface and Back	X	X	-	-	-	-	-
Support EVA	X	X	X	-	X	-	-
Life support for 3 crew members	X	X	-	-	-	-	-
Life support for 6 crew members	-	-	X	-	X	X	-
Life support for 2 days	X	X	X	-	-	-	-
Life support for 5 days	-	X	-	-	X	X	-
Life support for 2 weeks	-	X	-	-	-	-	-
Aeromanuevering	-	-	-	-	X	X	-

The commonality shown between Lander designs in Table 18 allows for the leveraging of at least a portion of engineering design, manufacturing, and testing costs across all missions.

Although significant commonality between Lander designs has been purposely designed for this space exploration architecture, several differences between Lander designs do exist. These differences are due to the significant differences between the landing environments on the Moon and Mars. The differences in ΔV requirements and Lunar and Martian atmospheres play major roles in determining the final Lander designs for each destination. Schematics for Earth return, Lunar, and Martian Landers are shown below in Figure 58, Figure 59, and Figure 60.

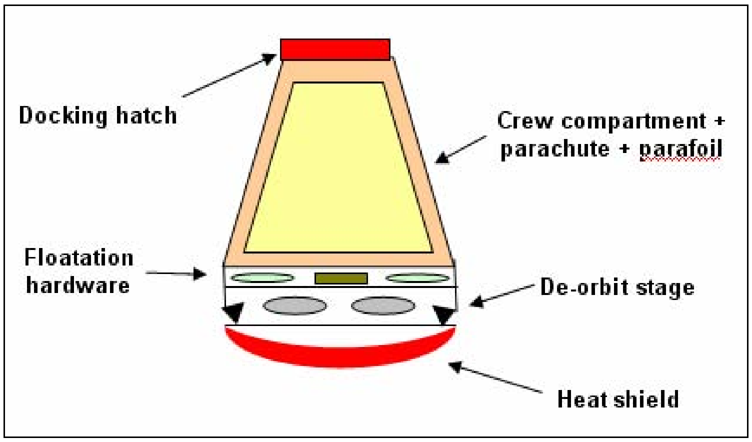


Figure 58: Earth return capsule design

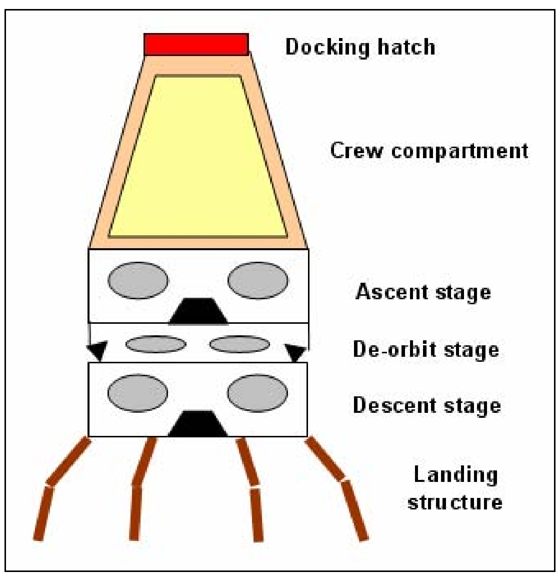


Figure 59: Lunar Lander design

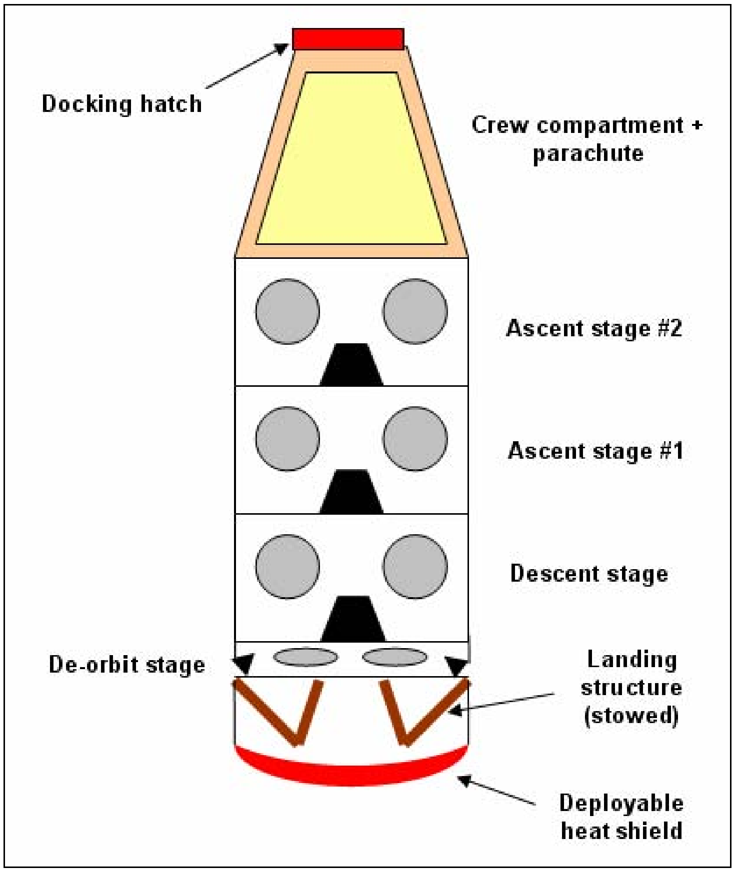


Figure 60: Martian Lander design

The three Lander designs can be seen to have some identical components, some similar components, and some differences. For example, the crew capsule on top of each of the Landers is identical. The engines for the various Lander designs are similar to each other. The actual engine masses can be seen in Table 19 in the next section. The differences among the Landers are few. A major difference is the requirement for floatation devices on the Earth return capsule. This is not required for the other two Lander designs. In addition, the Earth return Lander will require a parafoil while the other two Landers do not. Finally, the Martian Lander has increased complexity due to the fact that it has a deployable heat shield and landing structure. This is not required on the other Landers.

6.4.3.1.6.2. Three-person versus six-person Lander trade study

A major decision made during the Lander design for this project was to determine how many crewmembers the Landers should be designed to accommodate. If missions with varying numbers of crew members are going to be done, perhaps it would be better to have greater numbers of Landers that each accommodate fewer crew members. On the other hand, if most or all missions will be done with the same crew size, it may be beneficial to simply design the crew compartment to be common for all missions.

Since three and six-person crew sizes are being considered for this project, a trade study was proposed to weigh the benefits and drawbacks from designing a three-person

Lander versus a six-person Lander. For missions with a crew of six, two three-person-sized Landers would be required to transport the crew instead of one Lander to accommodate the entire crew.

Table 19 below is a comparison of the Lander component masses required for Earth return, lunar, and Martian Lander designs for three and six-person crews. An important assumption made was that the Landers have life support systems capable of supporting the crew for several days.

Table 19: Three and six-person Lander component mass comparison

	Earth Return		Lunar Lander		Martian Lander	
	3-person vehicle masses (kg)	6-person vehicle masses (kg)	3-person vehicle masses (kg)	6-person vehicle masses (kg)	3-person vehicle masses (kg)	6-person vehicle masses (kg)
Crew capsule	3617	4281	3541	4411	3541	4411
Parachute	225	450			410	425
Parafoil	275	300				
Heat shield	438	573			950	986
Landing structure			40	55	50	65
De-orbit stage	95	124	50	62	436	535
Descent stage			3942	4915	2049	2553
Ascent Stage 1			2319	2889	3430	4272
Ascent Stage 2					4067	5066
Total mass	***4650***	***5728***	***9893***	***12332***	***14932***	***18318***
Total mass/crew	***1550***	***955***	***3298***	***2055***	***4977***	***3053***

The last row in Table 19 makes the best argument for simply designing one six-person Lander if crews of only six are to be used for all missions. This is because the mass per crew for six-person-sized Landers is significantly lower than that of the three-person Landers. However, if a crew of three were to use a six-person Lander, this benefit would be lost. In fact, a significant mass penalty would result from a crew of three using a six-person-sized Lander.

A benefit from a crew of less than six people using a six-person Lander would be the option of using the extra internal volume to bring additional life support and other equipment for either enhancing mission capabilities or simply providing extra redundancy to provide additional safety margin for the crew.

For a six-person crew, a major benefit for using two three-person Landers as opposed to using one six-person Lander is the ability for a portion of the crew to survive and complete their mission even if there is an accident in which a major Lander failure results in the loss of that portion of the crew. If a crew of six is landing in one Lander and the Lander fails and the entire crew is lost, that would effectively end the mission and it would be deemed a failure. Having two Landers adds redundancy to mission success by removing a single-point failure. If a remaining crew of three were able to successfully complete their mission and return home, the mission will most likely be deemed a success.

6.4.3.1.6.3. Autonomous Landing Systems

Many of the lunar and Martian mission architectures are dependent on the pre-positioning of cargo on the planetary surface. Indeed, a major objective is to separate cargo from crew as much as practicable for reasons of safety and cost-benefit. For the crewed phases of the mission, the Landers have been assumed to be partially controllable by the astronauts. For instance, on Apollo 11 Neil Armstrong took manual control of the powered descent system when he noticed the target landing site was in a boulder field. Unfortunately, unmanned cargo Landers cannot rely on such human situational awareness and adaptability. The autonomous landing systems for these Landers will require a high degree of accuracy to position cargo modules in close proximity, while avoiding terrain hazards and other modules. NASA/JPL Mars Exploration Rover's Spirit successfully landed within its 77 km long footprint at a distance of only 9 km from its dead-center target. NASA has renewed focus on autonomous descent and terminal guidance. As part of NASA's New Millennium Program, the "Smart" landing technology capabilities roadmap is summarized in Figure 12. The roadmap matches well with the time-scale of developing the Moon and Mars exploration system.

Potential Mission Timeline	"Smart" Landing Capability Needs
2009-2010	**2009-2010**
Mars Science Laboratory Lunar South Pole/Aitken Basin Sample Return	Landing accuracy <6 km (Mars), 0.1-1 km (Moon) 100 m maneuvering to avoid hazardous slopes/rocks
2012-2013	**2012-2013**
Comet/Asteroid Surface Sample Return Venus In-Situ Explorer	Landing accuracy <0.1 km (small body), 10-100 km (Venus) 100-200 m maneuvering to avoid small body terrain hazards
2014-2015	**2014-2015**
Mars Sample Return	Landing accuracy 1-3 km 100-300 m maneuvering to avoid all hazardous terrain features
2020+	**2020+**
Europa Lander Titan Explorer Mars and Lunar Robotic Outposts Human Exploration Missions	Landing accuracy <0.1 km (airless bodies, Mars), 10-100 km (Titan) 100-500+ m maneuverability to avoid all hazardous terrain (airless bodies, Mars)

Figure 61: NASA's missions and "smart" landing technologies roadmap (Thurman, 2003)

6.4.3.1.7. On-orbit Assembly

On-orbit assembly is a difficult task in more ways than one. Choosing a launch vehicle, selecting the timeline and determining the launch logistics for assembly are three difficulties of using this method of module assembly.

According to (Larson, 1999), four considerations are important when designing a launch sequence,

1. *Basic subsystem functions* – The space elements must have vital systems working at all times (attitude and orbit control, electrical power, thermal control, communications etc.)
2. *Configurational aspects* – All orbital stages are space elements and need an analysis of their expected functional requirements
3. *Launch system restrictions* – There are limits in terms of payload mass and volume, and launch rates.

4. *Contingency considerations* – Ensure launch time slips or complete payload loss will not be critical to the space element. Failure mode analysis and contingency plans have to be established assuming the failure or loss of each item.

From a transportation perspective, it was assumed that the space transportation system architecture does not require the use of the International Space Station (ISS) as an assembly or return point. This was done to ensure that NASA could divest itself from the ISS and STS to meet the Space Exploration goals within the future budget cap. When pre-positioning becomes paramount to a mission architecture, the assumption of vehicle on-orbit assembly is critical to mission success.

On-orbit assembly prior to crew arrival increases the safety of the mission. Once the transfer vehicle is assembled and the crew is delivered to orbit, there will be some on-orbit "check-out" functions that the crew will need to accomplish to complete assembly, verify functionality, and prepare for injection to the destination. This may require a few days during which the crew would be exposed to a high radiation environment. Launching a crew to a fully assembled vehicle ensures that radiation exposure is minimized. This also ensures that the vehicle has been correctly assembled and checked over prior to crew arrival. If any problems occur during assembly, the vehicle is more readily accessible, and if problems occur during check-out the crew can return home easily.

Spacecraft platforming, reconfigurability, extensibility, and assembly are discussed elsewhere in the Report.

6.4.3.2 Moon Options

6.4.3.2.1 Trajectories

The lunar baseline mission architectures were evaluated for two trajectories: one using lunar orbit at a staging point, as discussed in the lunar baseline mission description, and one using the Earth-Moon L1 (EM-L1) rather than lunar orbit as a staging point. The different trajectories were compared in terms of total mission mass in LEO, and the results are shown in the figure below.

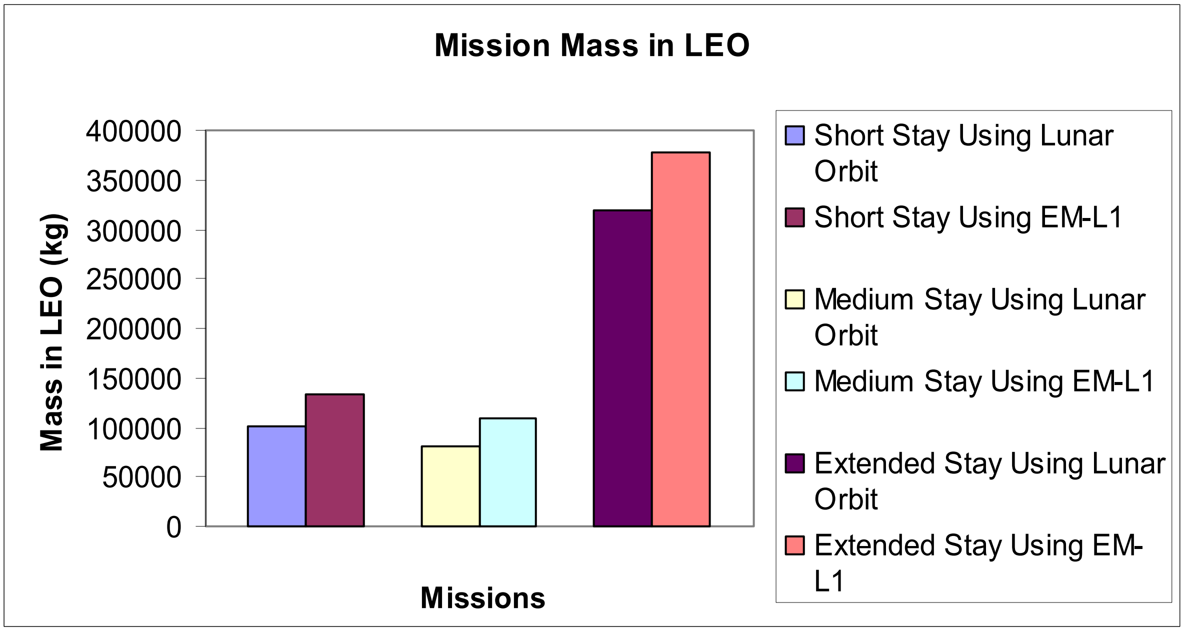


Figure 62: Comparison of Mass in LEO for Different Missions

For the purpose of comparison, all missions segments use the minimum Δv trajectories, employing a Hohmann transfer where appropriate. Also, calculations assume a lunar parking orbit of 100km where applicable, and include additional Δv required to establish and leave EM-L1 halo orbits where applicable. The appropriate parameters to reproduce this calculation are summarized in Appendix 9.3.

The results shown for architectures using lunar orbit apply for lunar missions to any latitude landing site if a free-return trajectory constraint is not imposed. They also apply to lunar equatorial landing sites if a free-return trajectory is imposed. The results shown for architectures making use of EM-L1 apply for missions to any latitude landing site, regardless of free-return trajectory constraints. Because the use of a trajectory utilizing the EM-L1 consistently results in a greater mass in LEO, the baseline mission uses lunar orbit as a staging area. However, this analysis does not include consideration such as accessibility to high latitude landing sites when free-return trajectory or scheduling constraints are imposed. These issues are dealt with separately in Section 6.4.3.2.3.

6.4.3.2.2 Landing Sites

A list of suggested landing sites is shown in Table 20.

Table 20: Suggested landing sites

Landing Site	~Latitude	~Longitude	Scientific Motivation	Suggested by:
Lichtenberg Basalts	32°N	68°W	Volcanism	Robinson, personal communication
Aristarchus Plateau	24°N	48°W	Volatiles, stratigraphy	Robinson, Taylor, and Schmitt, personal communications, and Ryder *et al.* 1989
Alphonsus	13°S	3°W	Volatiles	Robinson, personal communication, and Ryder *et al.* 1989
Sulpicius Gallus	20°N	20°N	Volatiles	Robinson, personal communication
South Pole Aitken Basin	18°S (crater) 25°S (massifs)	172°E (crater) 155°E(massifs)	Volatiles, poles, stratigraphy	Robinson, Taylor, and Schmitt, personal communications, Pieters *et al.* 2003, Ryder *et al.* 1989,
Apenine Bench Formation	20° N	0°	KREEP	Robinson, personal communication, and Ryder *et al*. 1989.
Tsiolkosky Crater	21°S	129°E	Stratigraphy	Taylor and Schmitt, personal communications, and Ryder *et al* 1989.
Mare Tranquilitatis	0°	25°E	He-3	Schmitt, personal communication

6.4.3.2.3 Lunar Orbit vs. EM-L1

There are three primary factors influencing whether to use lunar orbit or the Earth-Moon L1 point in a lunar mission architecture: choice of landing sites, free-return option, and scheduling.

If lunar missions are targeting sites within plus or minus five degrees of the equator, an architecture utilizing a lunar equatorial orbit provides the option of a free-return trajectory. In addition, it offers the ability to descend from and ascent to lunar orbit every two hours.

Lunar missions that do not target equatorial sites can achieve intermediate inclination and polar lunar orbits by making minor targeting maneuvers early in the trajectory. This targeting requires a negligible amount of Δv, however it removes the possibility of a free-return trajectory. Making use of intermediate inclination and polar orbits also introduces scheduling constraints. For example, from an intermediate lunar orbit, a spacecraft can only descend, ascent, or enter a return to Earth trajectory every 27 days. From a polar orbit, a spacecraft can descend to or ascent from an intermediate or equatorial landing site every 14 days, and can descend to or ascend from a polar landing site every two hours. An opportunity to enter a return to Earth trajectory from polar orbit occurs every 14 days (Larson, 2002).

If a lunar mission is targeting a non-equatorial landing site and a free return trajectory is deemed necessary, two possible solutions exist. The first solution is to make use of the Earth-Moon L1 (EM-L1). The second solution is to enter a lunar equatorial orbit and then initiate a propulsive maneuver to change orbital planes.

The EM-L1 provides access to all lunar landing sites with the option of a free-return trajectory if no burn is initiated at the EM-L1, and the continuous ability to descend from, ascend to, and enter a return to Earth trajectory using the EM-L1. However, an architecture utilizing the EM-L1 requires an increase in total mission ΔV of approximately 11% (plus or minus 2% depending on the trajectory used) and four extra propulsive burns as compared to an architecture using lunar orbit.

The second option, entering a lunar equatorial orbit and then initiating a propulsive maneuver to change orbital planes, is expensive in terms of total mission Δv. If the lunar mission is targeting a landing site no more than 39° from the lunar equator, then the total increase in Δv is less than that required to use the EM-L1. However, if the landing site is greater than 39° from the lunar equator, using the EM-L1 is less expensive in terms of Δv.

The advantages of using lunar orbit are clear for missions targeting equatorial sites, but the best option is not clear for missions targeting non-equatorial landing sites. Unfortunately, as described in the discussion of landing sites, many interesting sites are not located along the equator. Thus, mission planners must weigh the importance of a free-return trajectory and scheduling constraints in terms of increased ΔV to determine the appropriate architecture.

While total mission Δv provides a general means of comparing lunar orbit and EM-L1, mission architecture can play a significant role in determining the influence this metric has on the total mission mass in LEO. For example, Figure 63 shows the total mission mass in LEO assuming a free-return trajectory requirement for the manned segments of a mission to a lunar pole. While EM-L1 may be beneficial in terms of total mission Δv if the mission is targeting a polar landing site such as the South Pole Aitken Basin, the total mission mass in LEO may be smaller for a mission using lunar orbit with a plane change if a certain mission architecture is utilized. In this case, only short stay missions save a significant amount of mass in LEO by using EM-L1. Medium and extended stay lunar missions targeting a pole require approximately the same amount of mass in LEO whether they use EM-L1 or carry out a plane change in lunar orbit. These results occur because short-stay missions do not use pre-positioning, while medium and extended stay missions do. The use of pre-positioning in these architectures reduces the amount of crewed module mass that must undergo a plane change in lunar orbit. Also, these results occur only for the baseline lunar missions described in Section 4.2 analyzed using the specific mission parameters found in Appendix 9.3.

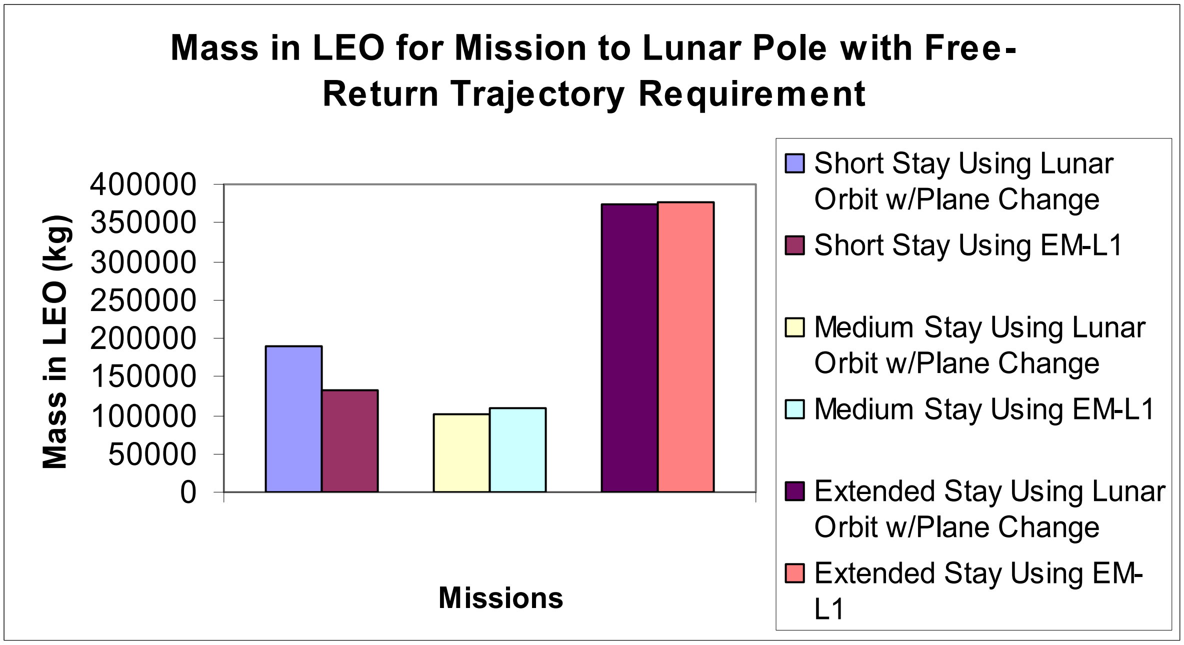


Figure 63: Mass in LEO for mission to lunar pole with free-return trajectory requirement

6.4.3.2.4 Reusability of the Lunar Lander

The proximity of the Earth and Moon, and the likelihood of multiple lunar missions to serve as test beds for future Mars missions raise the question: what benefit, if any, can be accrued through use of a reusable Lunar Lander?

A trade study was performed to compare a non-reusable Lander to a reusable Lander in terms of cumulative mass in LEO required to pre-position and ready one three-person Lander in lunar orbit. The results are shown in the figure below.

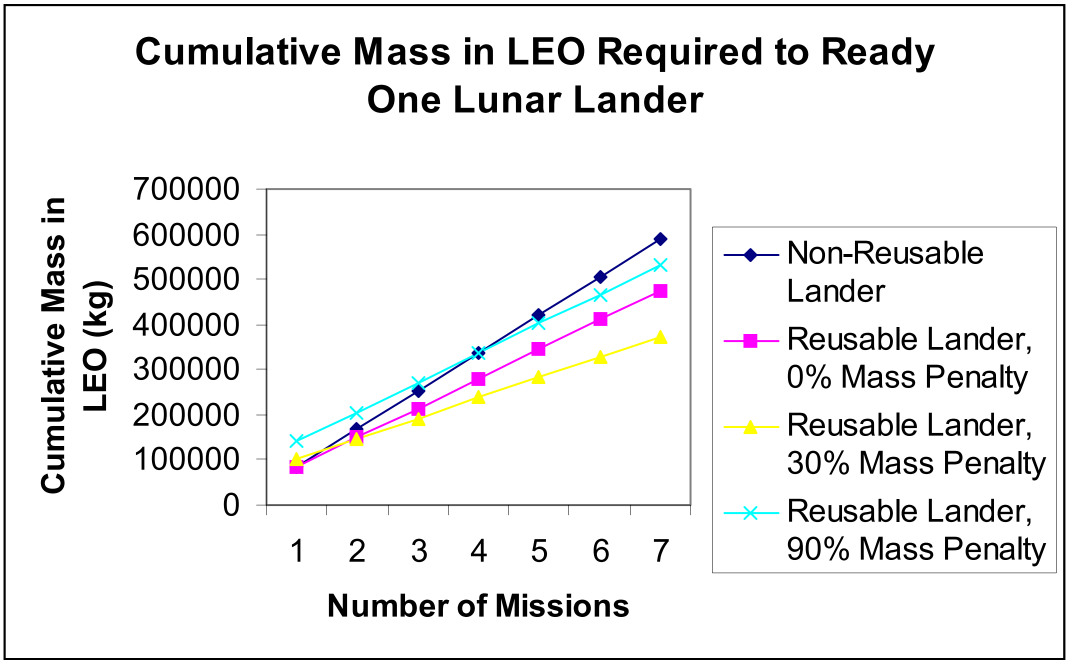


Figure 64: Comparison of a non-reusable and reusable Lunar Lander

The mass in LEO for a non-reusable Lander was calculated assuming electric propulsion to pre-position one Lunar Lander with enough chemical fuel to descend to and ascend from the lunar surface. The mass in LEO for a reusable Lander was calculated assuming electric propulsion to pre-position one wet Lunar Lander for the first mission. Mass in LEO of subsequent missions was calculated assuming electric propulsion to transport fuel for the pre-positioned Lander's descent and ascent. Because a reusable Lander will likely have a greater mass than a non-reusable Lander, a mass penalty was included in the study. For example, a 30% mass penalty means the calculations shows that an expendable Lunar Lander has a mass 30% greater than the non-reusable Lander mass.

Figure 64 shows that if a reusable Lander can be designed with only a 30% mass penalty, then a reusable Lander is beneficial after only two uses. However, if the mass of a reusable Lander is nearly double the mass of a non-reusable one, then five missions are necessary before benefit is accrued.

While Figure 64 suggests that a reusable Lunar Lander provides benefits in terms of mass in LEO, this study does not include the mass in LEO for tools and extra equipment to service the Lunar Lander when necessary. Also, mass in LEO as a comparison metric does not capture all cost associated with designing and building a reusable spacecraft; there are additional development and operational costs not captured in this study.

6.4.3.3 Mars

A human mission to Mars is an enormous undertaking that poses significant design constraints. There are many factors that have influenced the baseline mission and many more that will emerge. In order to understand these design constraints, a deeper investigation into the factors that influence the trajectory and the mission must be conducted.

6.4.3.3.1 The Mars Environment and Design Effects

The Mars environment will directly affect the design of any mission to the planet. Therefore, it is necessary to examine the environmental attributes and determine how they impact the design.

6.4.3.3.1.1 Environmental Factors

Martian seasonal change can drastically affect in weather and the working environment. The northern hemispheres' spring is 94 sols long and autumn is 143 sols. The Martian day, or sol, is 24 hours and 39.6 minutes long. Gravity on Mars will affect all activities on the surface as well as the health and well-being of the crew. The gravity on Mars is approximately 3.758 m/s^2, which is slightly more than one-third of Earth's surface gravity.

Landing site selection will depend on a number of factors; altitude and site characteristics will be paramount. Altitudes range from +27 km on Olympus Mons to –4 km in Hellas Basin. The southern atmosphere is at an overall higher elevation +4km than the northern plains –2 km. There are three distinct regions on the planet: cratered terrain, the volcanic provinces of Tharsis and Elysium, and the lowland plains.

Three types of planetary surface material exist: rocks, regolith, and fines. Mars' surface material can shield against radiation and micro-meteoroids. Thus, about 0.5 m of Martian surface materials should be enough to stop the primary dose from solar particle events, although you could use several meters to prevent more radiation.

Previous excursions to Mars, such as VL-1, VL-2, and Sojourner spacecraft exceeded operation lifetime. Thus, planners should not expect long-term chemical degradation if the system is properly designed.

Atmospheric conditions will have the biggest impact on mission design. Weather and atmospheric composition will drive all surface operations. Wind speeds vary by season and are lowest during the summer (around 2 m/s – 7 m/s), and reach speeds around 5m/s – 10m/s during autumn. Global dust storms, which can reach speeds of 30 m/s, occur in the southern spring and summer (Hamilton, 2001). The atmosphere is composed of: 95.32% carbon dioxide (CO_2), 2.7% nitrogen (N_2), 1.6% argon (Ar), 0.13% oxygen (O_2), 0.03% water (H_2O), and 0.00025% neon (Ne).

The severity of the Martian temperature will have a major effect on both mission design and planetary surface operations. The average recorded temperature on Mars is -63° C (-81° F) with a maximum temperature of 20° C (68° F) and a minimum of -140° C (-220° F). Barometric pressure varies at each landing site on a semiannual basis. Carbon dioxide, the major constituent of the atmosphere, freezes out to form an immense polar cap, alternately at each pole. The carbon dioxide forms a great cover of snow and then evaporates again with the coming of spring in each hemisphere. When the southern cap was largest, the mean daily pressure observed by Viking Lander 1 was as low as 6.8 millibars; at other times of the year it was as high as 9.0 millibars. The pressures at the Viking Lander 2 site were between 7.3 and 10.8 millibars. In comparison, the average pressure of the Earth is 1000 millibars. Thus, the comparatively low barometric pressures will pose serious design constraints and limitations.

6.4.3.3.1.2 Design Guidelines

The Mars environment will direct the design of any mission to the planet. Thus, having examined some of the environmental factors, it is necessary to understand how they impact the mission design. .

Examining the environmental conditions given above we can infer some general conclusions. The soil can support Landers, stations, and rovers, however footpad and wheel dimensions must be sized according to load. Structures can be anchored in the soil for additional stability against seismic events, wind, etc. In addition, the soil can be used to provide some level of radiation and environmental shielding. The effective shielding against radiation is 0.5 m to 3 m. The effective shielding against micrometeoroids and orbital debris is 0.5 m.

The design of equipment must account for the harsh environment and atmospheric conditions on Mars. The physical effect and chemical effect of soil and dust on mechanical and electrical systems is unknown, however mechanical devices will need lubrication and sealing.

Using solar energy to provide a power supply is advantageous on Mars. Mars has longer days and no significant eclipses. The surface can be used as an electromagnetic ground. In addition, the local geology contains usable quantities of critical resources such as CO_2 and water from the atmosphere (Larson, 2003).

6.4.3.3.1.3 Landing Site Selection

There are certain criteria for landing site selection that is dictated by both the mission objectives and the planet's environment. The latitude should be between 30-60^{0}S for power constraints and poleward of 30^0 for near surface water. Positioning the landing site near fluvial activity, where subsurface water may be expected (i.e. gully locations), sites of past seismic activity. The elevation should be at a maximum of 1.3 km above the Mars' datum. The site should be a smooth flat plain, relatively devoid of large obstacles and the landing ellipse must fit within the flat plateau region.

The following is a list of potential sites that satisfy the above criteria:

• Dao Vallis (33°S, 267°W)
• Gorgonum Chaos (37°S, 168°W)
• Nirgal Valles (30°S, 39°W)
• Elysium Planitia (37°N, 252°W)

• Newton Crater (41°S, 160°W)

6.4.3.3.2 Enabling Technologies

Although a human mission to Mars may be accomplished without developing new technologies, to develop a sustainable initiative, it will be quite beneficial to investigate some or all of these technologies. Some of these technologies will allow for a lower mass to be launched from Earth, while others will allow for a semi-permanent infrastructure on Mars to remain relatively self-sufficient. A review of some of the important technologies follows.

6.4.3.3.2.1 ISPP

In-situ propellant production (ISPP) promises to be a viable new technology that will have a significant effect on missions to Mars and the corresponding mission architectures. The Mars Direct plan developed primarily by Robert Zubrin in the late 1990's introduced a realistic ISPP scheme, which makes use of a readily accessible resource: the Martian atmosphere (Zubrin, 1996). ISPP would allow propellant to be produced on Mars for the ascent to Mars orbit or even for the return journey to Earth, thus dramatically reducing the required IMLEO. Before the Mars Direct proposal, mission plans estimated the IMLEO to be on the order of 1500 metric tonnes or ten times that of the Apollo missions. By utilizing ISPP, the IMLEO estimate is reduced to 250 tonnes for Mars Direct and 450 tonnes for a slightly scaled up version, Mars Semi-Direct, with a crew of 6, as presented in NASA's 1997 Design Reference Mission (Heidmann, 2003).

The ISPP method, which produces methane/oxygen (I_{sp} ~370s), as described by Zubrin, is still the most widely recognised possibility for Mars. This method entails sending hydrogen feedstock, a small nuclear power plant, and a chemical processing plant to Mars in advance of the human crew. The ISPP plant is autonomously set up to begin fuel production.

The production of fuel is performed by the Sabatier process, which converts carbon dioxide from the Martian atmosphere into methane and oxygen via the following reaction:

$$4H_2 + CO_2 \rightarrow CH_4 + 2H_2O.$$

The resulting water is electrolysed into hydrogen and oxygen. The oxygen is stored and the hydrogen is recycled back into the Sabatier reaction. This series of reactions provides a mass leverage (produced fuel mass to imported fuel mass ratio) of 12, which can be increased further by simply improving the methane/oxygen molar ratio to that required for propellant, by utilizing a Reverse Water-Gas Shift (RWGS) to generate oxygen as follows:

$$CO_2 + H_2 \rightarrow CO + H_2O.$$

Using this process, the mass leverage is increased to 20 (Heidmann, 2003). Yet another option is to use RWGS to produce methanol that has a lower I_{sp} than the methane propellant but uses a more efficient process and does not require cryogenic storage. Since methanol-based propellants have a relatively simple engine design, such a fuel may be very useful for surface power such as rover fuel cells (Heidmann, 2003).

The development of ISPP technologies requires several obstacles to be surmounted. Considerations of the Martian environment, such as dust and reduced gravity will affect the functioning of the chemical processing plant and nuclear power plant. The design of these components will have to be carefully implemented such that these factors are taken into consideration. Secondly, ISPP will rely heavily on autonomous systems. A spacecraft will have to survive the hazards of launch and interplanetary travel, land successfully, deploy its various components, and begin to produce power even before the chemical processing plant can begin to function. Furthermore, autonomous verification and communication of the status of propellant production will be required to validate the functioning of ISPP. The above obstacles suggest that the first human mission to Mars not rely on ISPP, but instead verify and test the technology such that future missions can benefit from this technology.

6.4.3.3.2.2 ISRU

In-situ resource utilization (ISRU) has the potential advantage of producing a significant savings in both mass and cost for Mars missions. Concerns about developing in-situ resources include technological readiness of many processes, their safety and reliability, and their sometimes serious effects on mission design. However, with the idea of developing a sustainable exploration, some examples of potentially advantageous ISRU are given below.

First, the use of solar radiation for power generation could provide a sustainable surface infrastructure with a low cost, reliable power supply. Also, the low gravity and near-vacuum environment is good for most material processing. To this end, carbon dioxide components can be used not only for life support, but for the creation of plastics, which allows for the return to commercial beneficiaries. Nitrogen from the atmosphere will be added to the air in the life support systems and as well as the soil in a greenhouse. The soil itself can be used for radiation shielding, and can be melted and sintered for the purpose of construction. Any metal extracted from the soil can be used in construction (Larson, 2003). In addition, water ice permafrost can be extracted from the soil to be used in propellant as well as life support. These in-situ resources can be utilized to enable a low cost, semi-permanent infrastructure on Mars.

6.4.3.3.2.3 Closed-Loop Life Support Systems

To support a long-duration mission, it will be necessary to recycle resources from the life support system. The closure-level of a life support system is the percentage of waste products that are recovered as useful resources. A high closure level requires less re-supply but may also add cost for technology development, increase power requirements, and increase complexity.

A life support system must provide resources for meals, hygiene, medical activities, and science experiments. The primary consumables required for astronauts are water, oxygen, and food. Four major life support functions exist: 1) managing the atmosphere to maintain nitrogen-oxygen pressures, provide a comfortable temperature and humidity, and remove contaminants such as carbon dioxide "bubbles," 2) distributing water to the crew and processing waste water, 3) treating waste and recycling consumables, 4) produce and process food. For missions longer than a few weeks, regenerable technologies will be utilized to remove CO_2 and recover wastewater. For missions on the order of several months, it will be necessary to also employ oxygen-regeneration technologies.

In general, food, water, and air represent most of the mass of a life support system. Closed-loop life support technologies (also known as advanced life support or ALS) can significantly reduce water and clothing masses, where the clothing mass is reduced at the expense of an aqueous laundry. This is a viable trade for long-duration missions (Hanford, 2003). Food mass is difficult to decrease without the addition of a biochamber component, and this is currently not very practical for transfer vehicles due to the large volume and power consumption rate required to grow plants. The results of a NASA study devoted to measuring the potential utility of ALS development show that in terms of Equivalent System Mass (ESM) – a total mass metric that incorporates such factors as power consumption and volume – ALS techniques are most beneficial for reducing the mass of a surface habitat module. Here it is estimated that ALS would result in mass savings of approximately 31% over life support technologies used on board the ISS (Hanford, 2003).

One element of the trade space for closed-loop life support systems is whether or not to employ such a system for the pressurized rovers that astronauts will use to travel hundreds of kilometers on the surface of the Moon and Mars. Pressurized rovers provide a shirtsleeve environment with a breathable atmosphere—oxygen is provided and CO_2 and water vapor are removed. It is possible to reclaim all CO_2 and water vapor for recycling. However, the mass and power requirements of such a system may exceed the mass of the consumables saved on any one excursion or even multiple trips. Given that roughly 1000 kg of equipment is required to recover 1 kg of nitrogen and oxygen per crewmember per day, nearly 50 week-long surface trips by three astronauts would be required to gain mass savings from a closed life support system in a pressurized rover. Whether utilized in rovers, or just in the stationary surface habitats, closed life support systems allow for a semi-permanent infrastructure to be sustained.

6.4.3.3.2.4 Nuclear Propulsion

Nuclear Propulsion is a relatively well-developed area of research that can provide significant mass savings in LEO. The safety concerns of launching a nuclear powered engine must be balanced with the benefits of having a high specific impulse form of propulsion.

Similar to electric propulsion, nuclear propulsion can provide constant thrust, unlike chemical propulsion. However, the thrust is significantly higher than electric propulsion and allows for competitive transit times. This decreases the major concern involved with the use of electric propulsion, which is the significant transit times during the outward spiral through the Van Allen belts. A nuclear propulsion engine can provide a specific impulse on the order of 960 sec (Walberg, 1993).

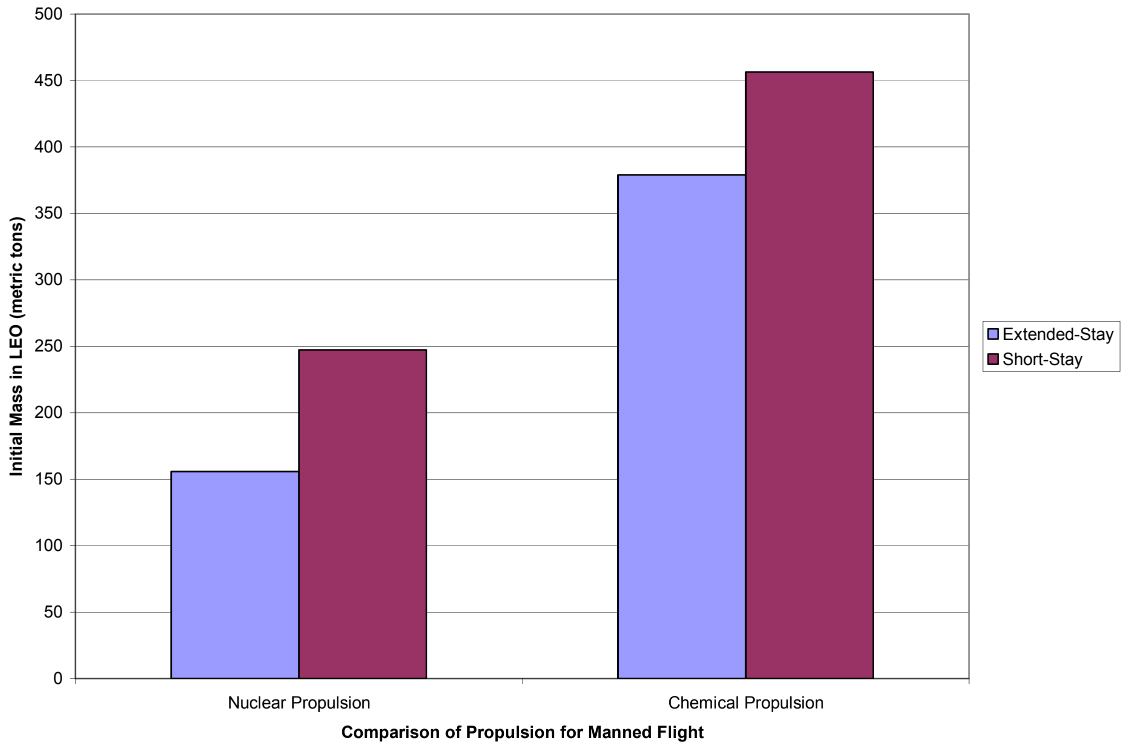


Figure 65: Comparison of nuclear propulsion to chemical propulsion for baseline trajectories

Assuming that a nuclear propulsion engine is employed for the human travel to and from Mars and that pre-positioning is still provided by electric propulsion (since flight times are less of a concern), Figure 65 shows that a significant decrease in initial mass in LEO exists when nuclear propulsion is utilized. Thus, to develop a truly efficient and sustainable transportation system to Mars, and beyond, nuclear propulsion will provide a low mission mass cost, and therefore, warrants further consideration (Walberg, 1993).

6.4.3.3.2.5 Power for Planetary Surface Operations

Power requirements for a Mars mission are projected to be between 25 kW-100 kW for an initial surface habitat, 10 kW for a piloted rover, and 160 kW for a habitat that process in-situ resources. Any sustainable exploration architecture must meet these requirements in an economical manner. Power systems on Mars face a variety of challenges: 12.3 hour nights, variations in the day/night cycle by season and latitude, and low temperatures (especially, during the night and winter). Atmospheric dust poses a threat to solar arrays on long-duration missions. In such cases, scrubbers may be needed to limit degradation. The primary challenge to power systems on the surface of the Moon is the 354 hour night. Other design issues arise from lunar dust that is produced from normal surface operations, structural issues in a one-sixth Earth g environment, and daytime temperatures, which can exceed 100˚C (Landis, 1999).

The primary electrical power source on the surface of the Moon and Mars may come from a combination of solar cells, batteries, radioisotope thermoelectric generators (RTG), or nuclear reactors. Photovoltaic power sources provide a long-term source of power at an estimated degradation rate. For relatively short manned missions in the past, energy storage systems have proved sufficient to power habitual modules (e.g., Mercury, Gemini, Apollo, STS). Nuclear power sources provide a long-term source of power and are appropriate for missions operating with little sunlight (e.g., polar regions of Mars, lunar nights).

The principal metric for evaluating solar and nuclear power sources is specific power (watts per kilogram). Photovoltaic specific power ranges from 25-250 W/kg while RTG specific power ranges from 5-20 W/kg. For a Mars mission requiring 100 kW of power, this translates to 400-4000 kg of solar arrays or 5000-20000 kg of RTG. However, nuclear power outperforms solar cells in terms of radiation hardness, stability, maneuverability, shadowing sensitivity, and obstruction of view. In fact, the US has flown 44 RTGs and one nuclear reactor to provide power for 25 different space systems since 1961. Advocates of nuclear power observe that all US nuclear power sources flown have met their specified requirements and allowed many extended missions to be performed.

Nuclear power offers the greatest flexibility for human solar system exploration because of its utility in virtually any space environment. Given the likelihood of extended Martian missions and the construction of a lunar base that will require electricity during the long lunar nights, photovoltaic sources are not optimal.

One option that would meet initial power requirements and provide plenty of room for extensibility is the deployment of a nuclear reactor. NASA is developing a nuclear reactor called the SP-100, which provides 825 kW at a specific power of 41 with a lifetime working at full power of around 7 years. Baseline versions of the SP-100 provide 100 kW with a specific power of 30 (3000 kg), 50% less mass than the best RTG projections.

With nuclear reactors, it is necessary to shield the crew from radiation. This shielding makes up a significant fraction of the power systems overall mass. Lunar or Martian soil may be employed to reduce the shielding materials transported from Earth. Launch pad safety is of utmost importance when dealing with nuclear reactors and it may be necessary to launch the reactor in an un-powered state to minimize radiation. Of course, obtaining permits and completing environmental impact statements for nuclear reactors add cost and may delay program schedules.

For redundancy and safety, it is optimal to have a back-up power system. Solar arrays could accomplish this task. Although they would not provide all of the power necessary for the entire spacecraft, they would provide enough electricity for emergency life support.

6.4.3.3.2.6 Agriculture on Mars

In space, the more material you discard, the more you have to bring with you, thus the longer the trip, the more Herculean the logistics requirements become. Human missions to Mars will most certainly have one main design characteristic: a closed-loop life support system that includes plants and microorganisms (bio-regenerative). The most likely way to grow plants on Mars is the design of a greenhouse.

Significant attention must be paid to the rather extreme conditions of Mars. These conditions include the soil chemistry, the gravity field, radiation, and temperature. Special kind of glass that can allow in light by filter harmful radiation will be required. An alternative to this would be to utilize solar arrays to provide power for the lights inside the greenhouse and to generate the temperatures necessary. In addition, extra nitrogen, along with various other chemicals will be added to the regolith (Cowing, 2002).

Along with providing crew with food, the plants would provide a secondary use as natural carbon scrubbers for breathable air. This technology is highly capable and is easily testable on Earth and on the precursor missions to the Moon. Due to infrastructure cost, this technology would be more useful for the medium and extended missions, and thus will increase in importance as we transition to a semi-permanent infrastructure.

6.4.3.3.3 Mission Architectures

In order to develop a baseline mission scenario, both the architecture of the mission and the choice of trajectory must be considered. When referring to the mission architecture, we are investigating the sequencing of events and the main form to function relationships that will guide the mission design. To compare different mission architectures a nominal trajectory is assumed. For an accurate comparison, all specific impulses are assumed to be 425 sec, and all engines have a structural factor of 0.1. In addition, all pre-positioning is performed assuming electric propulsion with a specific power of 150 W/kg, an efficiency of 0.7 and a specific impulse of 3200 sec.

Figure 66, shows the comparison between different mission architectures, assuming a short-stay mission. For each mission architecture, all maneuvers at Mars are powered, and a direct entry at Earth is assumed. In addition, all missions assume a surface habitat is pre-positioned. The following is a brief description of each mission according to the labels given in Figure 66. A NOVA mission refers to a direct entry at Mars without orbit insertion. A Mars orbit rendezvous (MOR) is an Apollo-class architecture in which separate Landers are carried from Earth. The transfer vehicle enters Mars orbit and the crew uses the Landers to travel to and return from the Martian surface. The MOR2 and MOR3 architectures have a similar design, but allow for the pre-positioning of the Landers and pre-positioning of the Landers and return fuel, respectively. Figure 66 confirms that a Mars orbit rendezvous, with pre-positioning of the Landers, surface habitat, and return fuel is the most mass efficient way to perform the transfer.

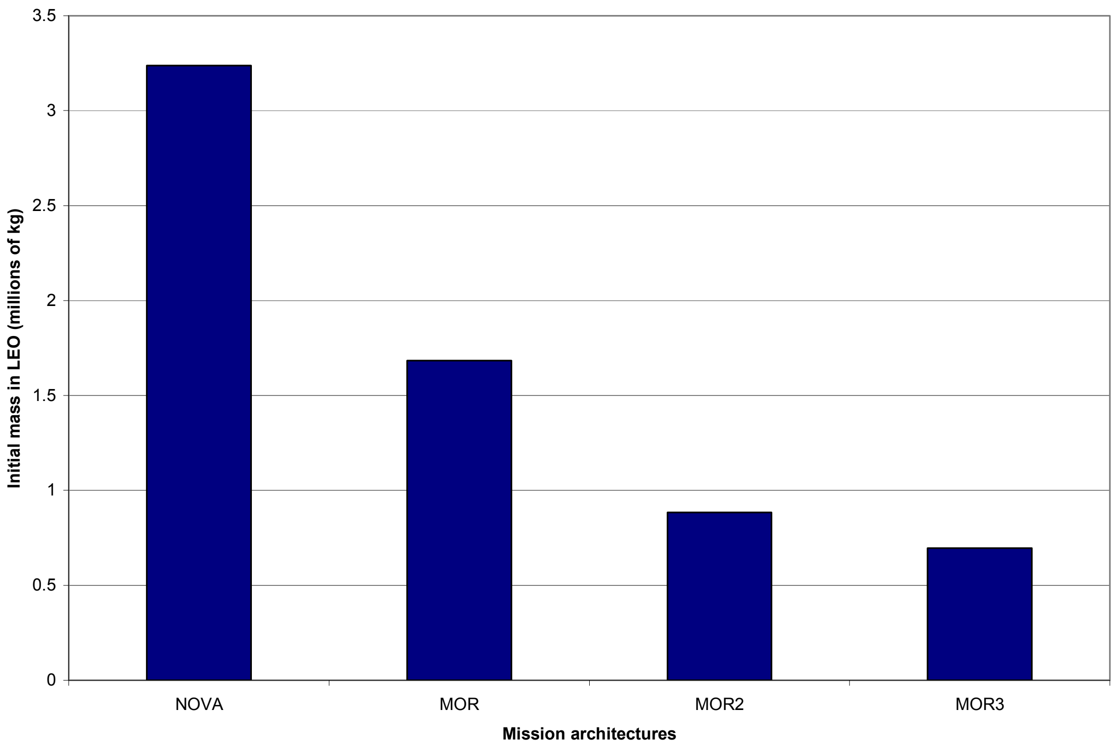


Figure 66: Initial Mass in LEO for Various Mission Architectures

MOR3 was chosen as the baseline architecture to decrease the IMLEO. This allows for a more robust design that can carry greater payload and produce a more sustainable transportation system architecture. However, since MOR2 has only a slightly higher IMLEO, if through testing on the Moon, or for just safety concerns, pre-positioning of the return fuel is eliminated for the first few Mars missions, then MOR2 could easily replace the baseline architecture with few changes to the rest of the baseline mission design. However, for a sustainable architecture, pre-positioning of the return fuel will provide a significant benefit as the missions progress, and therefore is chosen as the baseline design.

6.4.3.3.4 Mars Trajectories

Having shown that a Mars orbit rendezvous with pre-positioning of both the Landers and the fuel is a mass-efficient architecture, different trajectories were evaluated. In order to compare trajectories, a baseline architecture was assumed for each comparison. For each trajectory, the Mars Transfer Vehicle (MTV), with crew, departs from LEO, transfers to Mars, and injects into a 1-sol Martian orbit. The crew transfers into two pre-positioned landing vehicles and descends to the Martian surface. Once on the surface, the crew will live in a separate habitation, which has been pre-positioned with the relevant consumables for the surface stay. At the end of the surface stay, the landing vehicles ascend and dock with the transfer vehicle. In addition, the transfer vehicle docks with the return fuel. The crew returns to Earth in the transfer vehicle, leaving behind the landing vehicles in Martian orbit. Upon Earth return, the transfer vehicle injects into Earth orbit and the crew transfers into two Earth return capsules. In order to accurately compare different trajectories, the specific impulses of all chemical engines are assumed to be 425 seconds, the structural factor is assumed to be 0.1, and the payload to initial mass factor for aerobraking is assumed to be 0.15. All pre-positioning is accomplished by an electric propulsion engine with the same characteristics as listed in Section 6.4.3.1.4.

Figure 67 compares an Opposition-class mission with and without a Venus fly-by maneuver during one direction of transit. In addition, both types of missions are compared with and without utilizing aerobraking. As can be seen in Figure 67, without a Venus fly-by, the opposition class mission, even employing aerobraking at both Earth and Mars is prohibitive, for the above described mission. The addition of the Venus fly-by maneuver lowers the required initial mass in LEO (IMLEO) to a reasonable value, especially with aerobraking, for only a small increase in time of flight. Other issues remain in question for this maneuver, such as radiation exposure due to the closer pass to the Sun. However, from a strictly mass related standpoint, a short-stay mission is not possible without a Venus fly-by.

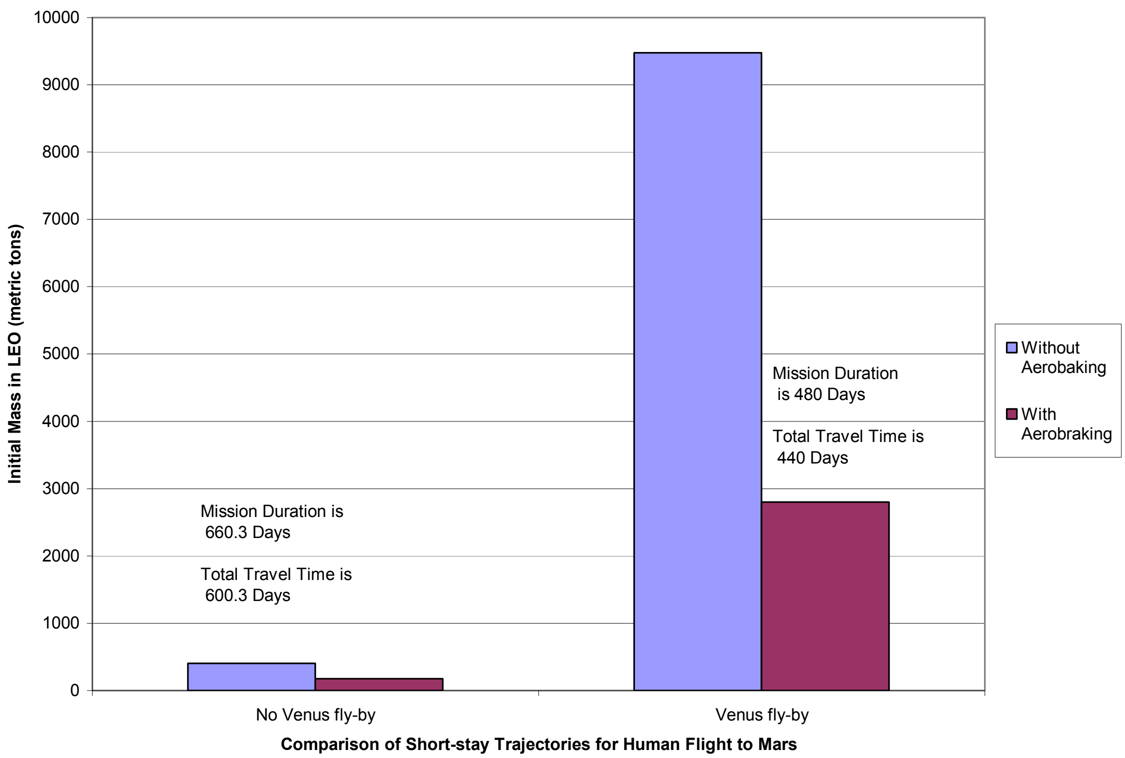


Figure 67: Comparison of Opposition-class mission with and without a Venus fly-by

Figure 68 displays a comparison between different trajectories for an extended-stay mission. As we can see in Figure 68, aerobraking significantly reduces IMLEO for each mission. An all-propulsive maneuver yields a prohibitively high IMLEO for a fast transfer; however employing aerobraking makes this class of missions feasible.

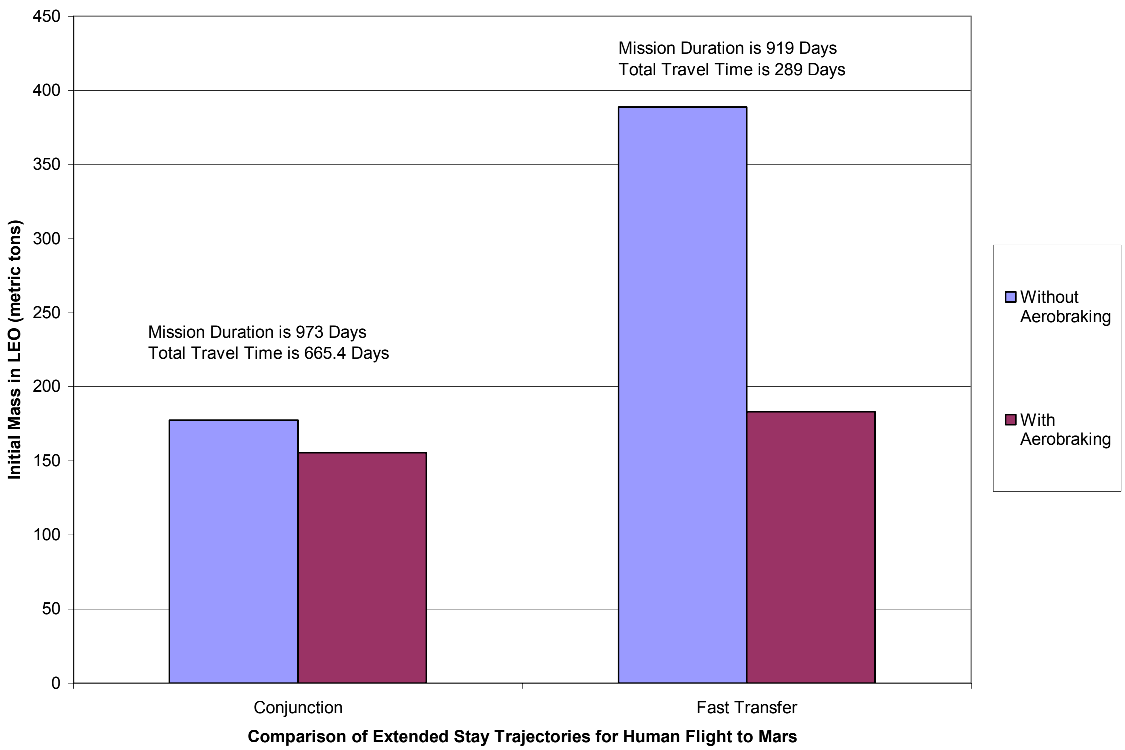


Figure 68: Comparison of Conjunction-class missions

According to Figure 68, the time of flight for the fast transfer is only slightly shorter than that of the regular conjunction class mission, and requires a greater IMLEO. However, the travel time is significantly shorter for a fast transfer then for a conjunction class mission. Since the travel time is important for other considerations, such as zero gravity and radiation exposure during flight, and since when aerobraking is employed, the initial mass in LEO is only slightly higher than that of a regular conjunction class mission, the fast transfer is chosen as the trajectory for an extended stay mission.

Figure 69, summarizes the IMLEO requirements for the feasible mission designs: Opposition-class trajectory with Venus fly-by, and fast transfer. Once again, we see that aerobraking yields a significant reduction in initial mass in LEO. If we refer to Figure 67 and Figure 68, we notice a large difference in the amount of IMLEO for a manned Mars mission. However, if we include the IMLEO for all pre-positioned elements, as in Figure 69, once aerobraking is employed, the IMLEO is virtually the same for both a short and extended stay mission. In addition, the use of aerobraking and parachutes, followed by a powered touchdown for the Landers, reduces the amount of fuel that they must use, and therefore further reduces the initial mass in LEO.

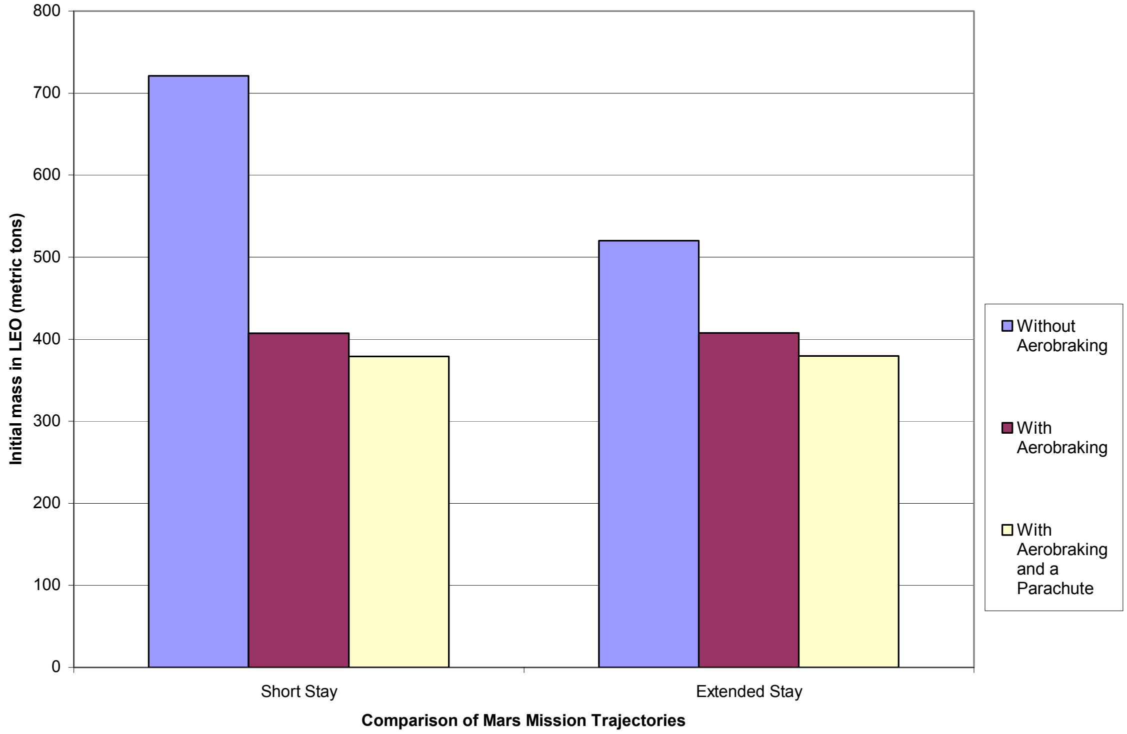


Figure 69: Comparison of Mars trajectories

By analyzing the Mars trajectories, we have concluded that a Mars orbit rendezvous (MOR) with pre-positioning is essential for reduced mass in LEO and is thus the baseline architecture for all other comparisons. For a short-stay mission, a fly-by of Venus is essential for a reasonable IMLEO for a human Mars mission. Since this maneuver is complicated and potentially exposes the crew to significant increases in radiation dosage, as well as other complications, the human factors elements must be studied in greater depth. By examining the Mars trajectories for an extended stay, the increase in IMLEO for a fast transfer is abated by the significantly lower travel time, as compared to a conjunction class mission. In addition, the reduced microgravity and radiation exposure makes this trajectory desirable. Aerobraking has been shown to significantly reduce the initial mass in LEO for every type of trajectory and is essential for a manned mission to Mars. When the short-stay and extended-stay trajectories are compared, including all pre-positioned elements, we notice that the initial masses in LEO are comparable, when aerobraking is employed. IMLEO is further reduced when a parachute and powered touchdown are utilized.

7. Scenarios

7.1 Introduction

Scenario planning is a method that may be used to determine the degree to which a given system responds to changes in the environment. By proposing scenarios, many of them examples of the extreme cases, and evaluating the ways in which the system would change or would fail to change, critical contingencies may be planned into the final design. This section outlines a set of seven scenarios and the anticipated responses to them.

7.2 Reasons for scenario-based planning

It is important to recognize that each design, including the baseline presented in this report, carries with it implicit assumptions about the state of the present and future environment. These assumptions about the environment constitute a de facto scenario in which the system is designed to operate nominally. Due the high degree of uncertainty, and associated high probability of change surrounding these assumptions, it is necessary that the system designed be able to adapt to changing environmental factors. Seven extreme changes in the system's operating environment were selected as scenarios against which the baseline system's performance could be evaluated. Scenario planning is primarily used in this fashion to identify architectural options and trades. In doing so, the system may be made adaptable or robust in the face changing environmental conditions. In this section, the baseline strategy was evaluated using each of the following scenarios. Where possible, options were exercised that allowed for significant adaptability or robustness to the constraints imposed by the scenario's parameters. The performance of the baseline strategy upon the use of these options serves to demonstrate the degree to which this strategy is sustainable and extensible in the face of drastic change.

7.3 Scenarios

7.3.1 Space Race II

7.3.1.1 Description

A foreign power successfully demonstrates a mission to the Moon similar in style to the Apollo-11 mission of 1969. Upon successful completion of their mission, the foreign power announces its intention to establish a permanent colony on the Moon and to land a human crew on Mars within the decade. The United States public perceives that its leadership in space exploration is in danger. In response to this concern, Congress triples NASA's budget to approximately 4% of the total US budget. NASA is instructed to reach Mars and to demonstrate colonization capabilities on the Moon before the foreign power as the top priority.

7.3.1.2 Response

With such a rapid influx of money with the requirement of exploring Mars and colonizing the Moon within a decade, NASA's best chance for success would be realized by following the Apollo model: develop a simple mission statement and optimize a point design. Technology development would be kept to a minimum to meet schedule constraints. In this case, the current prioritization of cost over schedule and performance would change: schedule would be fixed, performance would be optimized, and cost would be variable.

The mission outline is as follows: to meet schedule requirements, a 70-80-ton launch vehicle will be developed utilizing components of the STS. Money will be poured into man-rating this new vehicle, and existing STS construction plants and launch facilities will be utilized. Funding for the development and operation of the International Space Station would also increase to accelerate research into countermeasures to the adverse impact of microgravity on human physiology. Because the mission to Mars is a priority, all hardware developed (landers, crew transport, rovers) will be optimized for Mars, not the Moon, and thus will most likely be over-designed for the lunar missions.

To demonstrate colonization capabilities on the Moon, construction of a lunar surface base would have to commence within ten years. Specifically, *getting to* the Moon, and *demonstrating technologies* are a priority, but not science; therefore, the medium-sized missions will be sacrificed (see section 4.2.2) in favor of a few short missions and a long mission (lunar base) as soon as possible, establishing semi-permanent human presence. In order to credibly demonstrate a sustained human presence capability, the base will support developing in-situ resources with the ultimate goal of becoming self-sufficient. Rather than selecting a fixed construction site before detailed investigation of the lunar surface, one possibility is for the initial surface bases to be pressurized rovers on the order of 10,000 kg with habitation, laboratory, EVA airlock, and re-supply elements. In this case, short-stays on the order of two weeks enabling exploration of the lunar surface would begin the colonization program; with infrastructure build-up occurring once a site has been selected for a permanent base. Fuel cells would power the initial pressurized rovers; a nuclear reactor would be the primary source of power for a permanent, fixed base. Science operations not in support of in-situ resource utilization and other colonization technologies will be kept to a minimum, as these are not imperative demonstrations for a semi-permanent Moon base or for a Mars mission.

For landing humans on Mars and bringing them safely back to Earth, a short stay mission minimizing duration and consisting of 600 days of transit time and 60 days of surface operations via an opposition class free-return trajectory with a Venus fly-by would serve as a baseline goal (see section 4.2.4). To meet schedule requirements of successful completion within a decade, the short stay Mars mission would have to be launched within eight years. Given this short time window, all chemical propellant will be used for propulsion, and no pre-positioning will be used.

Although the schedule is accelerated rapidly in this scenario to accomplish two specific objectives within a decade, it is important to note that our overall exploration architecture remains unchanged: a sustainable human presence in outer space. Missions in this scenario flow from existing Moon and Mars baselines with infrastructure increasing capabilities over time.

7.3.1.3 Associated Trades and Options

This scenario illustrates a choice that must be made with regards to how extensible the final system should be. A typical engineering situation will require a trade-off between extensibility and optimality. In the case of a drastic shift in U.S. policy, it may be required to sacrifice some of the options that gear the baseline strategy towards extensibility so that one particular goal can be met as soon as possible. In this case, a point-design would be ideal, and all of the features geared towards creating a space exploration system that can adapt through time would simply be a hindrance on overall system performance. On the other hand, if no such policy goal is set, and sustainability remains the direction of choice, extensibility will dominate over optimality in many situations. An example of this trade-off is the degree to which time should be spent on the Moon in training for Martian missions. A maximally extensible mission might perform a large number of lunar tests before finally deciding to send humans to the Martian environment, where rescue will be significantly more difficult. On the other hand, a mission that is optimized for a quick Martian landing will spend less time training technologies which are unnecessary to the primary goal of landing a human on Mars, such as rover transportation and lunar ISPP.

7.3.2 Launch System Failure

7.3.2.1 Description

NASA's primary launch vehicle is destroyed during operation due to a technical failure, killing the entire crew compliment. All usage of that particular vehicle ceases until the problem can be isolated and fixed, a process that may take as long as five years. American astronauts are involved in space exploration activities during the catastrophe, thus requiring that they find another method to leave and return to Earth as soon as possible.

7.3.2.2 Response

A launch system failure is a terrible situation. Due to the danger inherent in launch activities, the risk of something like this happening is always present. Much can be done in advance to reduce the impact that such an accident would have on the overall exploration mission, for example, launch the crew and cargo separately. Design of lunar transfer and Martian transfer vehicles to employ a docking mechanism and an orbit that allows for foreign vehicles to dock on it will ensure that in the case of a launch system failure, NASA has the option of turning to international partners for support. If these cautious measures are taken in advance, the impact on the exploration agenda can be minimized.

If the launch failure involves only the heavy cargo vehicle it will be a setback, but only a minor one. In this case the vehicle is unlikely to be grounded for more than a year. However, if the decision where made to use the same vehicle for humans and heavy cargo and that vehicle failed catastrophically, then the consequences would be very serious and the leadership of the United States of America in human space flight would be very vulnerable.

7.3.2.3 Associated Trades and Options

One way to incorporate robustness against this scenario would be to create a redundant second launch vehicle design. There are two ways that this could be done:

The first way involves heavy cooperation with international partners who already have human launch capability. Essentially, the United States government would encourage countries to create a second human-rated heavy-lift launch vehicle, which could be capable of replacing the American vehicle in case of a disaster. As a foreign policy tool, this approach would internationalize the space exploration program and would allow for another country to share in the financial burdens. This has benefits in that it may save long-term costs and in that it could potentially improving foreign relations. Furthermore, America would be less likely to pull out of the space exploration theater if another country were actively participating. This eventuality is highly unlikely however, since, in the event of a major American launch system failure, the other power would retain de facto leadership in the space exploration theater. Furthermore, the U.S. would probably have to supply some initial monetary investment in the foreign space vehicle and/or some initial economic incentive to the country that would be providing that vehicle. Finally, there is no guarantee that a foreign country would follow through with their promise to create a launch vehicle, and no way to enforce that they do follow through. These considerations make this option a high-risk and high-initial-cost investment.

The second way involves creating a competitive structure within American business. In this situation, the U.S. would contract the creation of two separate and different launch vehicles, each from a different company. Each of these launch vehicles would compete for a use, on a per-mission or a per-program basis. Although the initial investment in this option is very high as well, there are significant benefits to be realized. Primarily, in the event of a launch vehicle disaster, a second launch vehicle would be available to continue the program. Furthermore, the competitive structure of this system would encourage each of the two companies to provide their top assurances that their own vehicle out-performs its competitor(s) in every respect (safety, reliability, mass, etc.). This would contribute greatly to overall sustainability, and would overcome the limitations imposed by this scenario. Furthermore, this approach has added extensibility benefits, since each launch vehicle provider would be competing for the next mission or the next program. This would encourage launch vehicle providers to design their vehicles in an extensible fashion such that, following the completion of one mission, their product would require minimal redesign before the next mission. There are, however, major cost-related drawbacks to this option. NASA's budget is limited, and the initial cost for this type of venture may be expected to be at least twice as high as the initial cost of designing just one launch vehicle. Since the market is limited to just one buyer, namely NASA, companies would not be as likely to commit to this proposition since it would, for them, represent a very high investment with a high-risk of small or no payoff. This structure could only be maintained through one of two methods, namely continual subsidies, which would put unparalleled strain on NASA's budget, or the opening of the market to other buyers (e.g. other countries and private individuals), which has strong policy implications (e.g., technology transfer, maintenance of space leadership, etc.) that would have first have to be addressed.

7.3.3 Dawn of the Nuclear Propulsion Age

7.3.3.1 Description

Nuclear propulsion technology emerges as a viable replacement to chemical propulsion. The technology is more efficient, can generate a higher-specific impulse, and has a higher amount of total thrust at liftoff. While it is initially very expensive and has not yet been flight-tested, it is expected to be approved for flight within 2 years. The sensitivity of nuclear technology prevents its development in cooperation with foreign nations and its use on foreign launch vehicles. If mishandles it may cause catastrophic failure. Although extensive testing suggests that the technology is highly reliable, the public is wary given that significant damage may result if it is misused.

7.3.3.2 Response

The use of nuclear propulsion would benefit a space exploration system program in several ways. First, since nuclear propulsion has a higher specific impulse than conventional chemical propulsion systems, initial mass required in low Earth orbit would be reduced for each mission using nuclear propulsion. This benefit would have a large impact on the mission design. One benefit would be an increase the amount of non-propulsion system mass in the launch vehicle. This increased payload efficiency would allow for potentially more redundancy or scientific hardware to be launched for the same launch cost. On the other hand, this reduction in propulsion system mass could also be a cost savings measure since it could reduce overall payload mass and in-turn reduce launch costs.

In addition, if high-thrust nuclear propulsion systems become available, a significant reduction in time of flight from Earth to the desired destination would occur. This would provide several benefits. First, it would minimize the exposure of the crew to microgravity by reducing their transit time. Second, this would allow the crew to spend more time at the destination than they would if they used a chemical propulsion system to transfer to the destination.

If the use of nuclear propulsion is realized and these cost reduction and human factors benefits become clear, nuclear propulsion technology will likely be incorporated into the space exploration system architecture. The incorporation of this technology depends on what stage the space exploration program is at when nuclear propulsion technology becomes viable. If nuclear propulsion becomes viable during the design phase of the program, it is likely that the initial, short missions to the Moon will still use chemical propulsion but the larger-scale missions to the Moon would test nuclear propulsion technology in preparation for its use on missions to Mars.

If nuclear propulsion technology becomes viable when the first manned missions to Mars are being launched, it is likely the technology will not be used for the first Martian mission. However, nuclear propulsion technology may still be incorporated into the space exploration architecture. For example, a precursor mission to the Moon could be used to test the propulsion technology. This would once again utilize the Moon as a "test bed" for missions to Mars. Once the technology has been successfully tested, the "Extended Stay" or "Extended Stay + Infrastructure" Mars missions could be launched using nuclear propulsion.

In conclusion, if the potential cost savings from incorporating advanced nuclear propulsion technology can be realized, the space exploration program may be able to increase mission frequency or enhance program sustainability. The reduction in the cost of launching each mission may allow for more missions to be flown for the same cost. Alternatively, a reduction in program costs would make the program more sustainable by reducing the portion of the total NASA budget consumed by the space exploration program. Politicians may be more willing to fund the program if they see more value for reduced costs.

7.3.3.3 Associated Trades and Options

Although the initial cost of designing nuclear propulsion modules for use in a space exploration program may be high, the cost of incorporating the new propulsion technology into the exploration program should not be large if the propulsion architecture is modular. This will allow propulsion modules to be exchanged as long as common interfaces are used. This is another exploitation of the advantages of designing an extensible space exploration system by using modular components. Generally, modularity is more difficult and more expensive to design into a system. If, on the other hand, the propulsion systems are designed such that they are "built-in", incorporation of the benefits of this new technology may be even more expensive.

7.3.4 Asteroid Strike

7.3.4.1 Description

A Near Earth Asteroid impacts the Earth's atmosphere, exploding harmlessly over the ocean, causing noticeable changes in weather patterns (e.g., tsunamis, storms, etc.). Scientists unanimously agree that if the asteroid had exploded over a populated area, significant death would have resulted. The US government allocates approximately 4% of the total yearly US budget between NASA and the DoD for the development of an early warning system, and to explore the possibilities of destroying or diverting asteroids on Earth impact trajectories.

7.3.4.2 Response

7.3.4.2.1 Background on Asteroids

It is estimated that asteroid impacts occur approximately once every century, with the most recent such event taking place in Tunguska forest, Siberia in 1908. Small asteroids on the order of one meter in diameter burn up in the atmosphere with impact energy equivalent to tens of kilotons of TNT (Rabinowitz, 1998). The threshold size for asteroids capable of global disaster is believed to be ½ to 1 km, and the impact frequency for these asteroids are once every 1000 centuries on average (Rabinowitz, 1998). It is hypothesized that there are between 1000 and 2000 Earth-approaching asteroids larger than 1 km, however only approximately 100 have been discovered so far (Rabinowitz, 1998).

7.3.4.2.2 State of the art

Current programs for asteroid (also called Near Earth Objects, NEOs) detection and cataloging involve observations at optical telescopes worldwide including efforts by MIT's Lincoln Near Earth Asteroid Research (LINEAR) Project, JPL, and SPACEWATCH at the University of Arizona. Large telescopes and time exposures are usually required in the search for asteroids since they have absolute magnitudes ranging from approximately 17 to less than 24 (Smith, 2001). Significant progress has been made in detection, and in the year 2000, the rate of Near Earth asteroid discovery reached almost one per day from about one per month in 1990 (Smith, 2001). In addition, NASA has designed and carried out a successful asteroid rendezvous mission, the NEAR Shoemaker, which was launched on Feb. 17, 1996 and became the first spacecraft to orbit an asteroid on Feb. 14, 2000 (Brand, 2001). Almost a year later, the spacecraft executed a controlled landing on the surface of Eros. This mission has provided many images of the target asteroid as well as operational experience with asteroid interception.

7.3.4.2.3 Plan of Action and its Effect on the Exploration Initiative

In the case that NASA were charged with developing an asteroid detection and deflection system, three concerns would be of primary importance, namely: characterization of asteroids, development of early warning capabilities, and procedures for interception and deflection

It is likely that a good part of NASA's budget increase would be used to fund programs such as MIT's LINEAR project and the requisite telescopes for detection and cataloging. This would aid in asteroid characterization, as well as a reduction in uncertainty of ephemeris data that would assist in the development of an early warning program.

Furthermore, a series of small missions akin to the NEAR project to orbit and rendezvous with asteroids could be initiated. It is very unlikely that these missions would be manned since the time scale would be too short to develop the redundancy and safety assurances necessary. Also, it would be much easier to launch these missions if the mass could be kept relatively low – not having a crew would mean that life support systems and a return to Earth (and the required propellant) would be unnecessary. This way, the mass could be kept to a level were the probe could be launched on a common launch vehicle such as a Delta IV (Smith, 2001). These missions would aid in the characterization of asteroids as well as the development of operational knowledge required for interception and deflection. Deflection schemes could also be tested on such missions.

The increase in NASA's budget would have a beneficial effect on the exploration initiative as well as the asteroid protection scheme. Budgets for certain enabling technologies such as nuclear propulsion would be likely to increase since in the case of a short warning time, it would be necessary to reach the dangerous asteroid as quickly as possible. Nuclear propulsion would facilitate a mission to Mars or the Moon by significantly reducing IMLEO. The exploration program may also be affected in that its principal destinations would be modified. Along with the series of unmanned missions to NEOs, a manned mission to Phobos would be a practical alternative: both as a precursor Mars mission, and as a possible asteroid characterization mission (since Phobos is thought to be a captured asteroid). This would be possible since the ephemeris data for Phobos is much more certain than that of most NEOs and therefore sending humans to Phobos would be less risky than a similar asteroid mission.

The shift of focus towards asteroids and away from exploration for its own sake would have a significant effect on the exploration program. It is possible that planned lunar or Mars missions would be postponed while effort is diverted into smaller asteroid rendezvous missions. On the other hand, given the increased total budget, NASA may be able to expand its workforce and maintain the exploration program at full strength with minor changes such as emphasis on particular enabling technologies or destinations.

7.3.4.3 Associated Trades and Options

A decision on whether or not to land on Phobos, and thus develop knowledge and technology that may be applied toward future asteroid operations, represents the option for this scenario. On the one hand, Phobos may not be compelling to the public compared to the allure of Mars. Landing on Phobos may be viewed by many as a waste of resources that could otherwise be diverted to a Mars exploration program. On the other hand, if asteroid missions become a priority, Phobos would provide a testing ground for these operations.

7.3.5 Lunar Water World

7.3.5.1 Description

An American expedition to the Moon discovers reserves of resources at the Lunar Poles, allowing for the large-scale extraction of hydrogen, oxygen, and water ice. Potential rates of extraction and production could sustain a lunar colony of 30 people indefinitely. International interest rises, and coalition of developing and space-faring nations proposes the development of a permanently manned international lunar base.

7.3.5.2 Response

The American Expedition discovers the water resources in year 2010. This event results in an increased interest in the Moon, and the modification of previous plans, installing a permanent base by year 2016. International partners participate in this effort, and the base is used largely for science and a deeper exploration of the lunar satellite.

This deeper exploration leads to the discovery and valuation of additional resources, which would include additional underground water resources, He3 repositories and Rare Earth mining. The fact that water can be cheaply dissociated using solar energy brings an important economic value to this discovery.

The fact that some of the cargo for the Mars and beyond missions will be shipped from the Earth and some of it will be shipped from the Moon under this scenario increases the importance of an EM-L1 node in Space Exploration beyond the EM neighborhood, and propitiates a case for a Space Station parked at this point (Kent, 2001).

Using present day technologies, the following table shows the advantage gained in sending one kilogram of propellant to the L1 point, which is the most likely storage point for propellant to be sent out of the Earth Moon Neighborhood.

	Mass at Planet surface	Mass at planet orbit	Mass at L1
Departing from Moon Pole	2050 kg	1200 kg	1000 kg
Departing from Earth Equator and using electric propulsion	64300 kg	2250 kg	1000 kg

This advantage allows increasing travels outside of the Earth neighborhood to be easier to afford, and at the same time gives the Moon station a revenue case that helps to sustain the costs of the scientific base.

Additional water resources and increased Moon travel frequency prompt market forces to intervene, and make feasible the establishment of a tourist's hotel on the Moon, and eventually a stable civil population.

By year 2022, freight to the Moon is handled by private corporations, while NASA only operates the Moon scientific base, and focuses its exploration efforts on Mars and beyond. Fuel production also is handled by private corporations, which have already invested heavily in the Hydrogen Industry on Earth.

There is therefore an increased interest on the Moon that leads, to some degree, to its trivialization. The exploration quest continues, but beyond the Earth neighborhood, helped by the fact that fuel is easier and less costly to obtain, and that two lower escape velocity nodes are present on the system: the Moon itself, with a stable population, and the L1 point as a supply node that holds deposits of fuel and supplies.

It is arguable that the additional interest, and resources that the Moon base will require, could reduce the investment rate for the Mars exploration on the short term. On the other hand, the trivialization of space travels that this scenario could imply will allow a cheaper and faster exploration of the remaining solar system assets on the longer term.

7.3.5.3 Associated Trades and Options

The primary option associated with this scenario is the use of the EM-L1 point. If the ability to transit through this point is originally included in the lunar transit architecture it may be easier to reach the lunar poles without having to do an energy-expensive plane change in lunar orbit. If, on the other hand, a decision is made not to transit to the lunar poles, this extra option is not utilized, and constitutes a deviation from an efficient design. A similar option to be considered is the value of creating a space station of some sort at the L1 point. Doing so may allow for refueling to occur on the way to the Moon, Mars and other celestial bodies. On the other hand, stations require maintenance. One possible way to get around this constraint would be to create a station that is somewhat autonomous. In this situation, the station would not require a constant human presence, although such a temporary presence may be desirable for other reasons, such as regular routine maintenance and microgravity-related research.

7.3.6 Little Green Martian Cells

7.3.6.1 Description

Following discoveries of microbial fossils on Mars, unmanned probes find strong evidence of one-celled life currently existing in the Martian subsurface soil. The Public's interest is piqued and NASA receives a 5% budget increase to hasten Mars exploration efforts. Some groups on Earth protest the government's decision, stating that the Martian biosphere should not be contaminated by human presence.

7.3.6.2 Response

With the discovery of possible life, the Moon missions would be de-emphasized, and the timelines for Mars would be moved up. It is likely that more precursor-manned and unmanned mission would be sent to Mars to further investigate the life phenomenon. The budget increase would be used to fund these efforts.

There are currently a significant number of missions geared towards investigating lunar resources and towards building up a lunar infrastructure/habitation knowledge base. Since one of the purposes of the lunar missions is to serve as technology test beds for subsequent missions to Mars, lunar missions may be cut down in size, with emphasis placed on those missions and technology demonstrations that are deemed critical for enabling human life on Mars. Since the Lunar missions are to serve as testbeds many of the systems used for Martian explorations, the exploration system is highly flexible to this change.

Since most of the lunar missions and tests are scheduled to occur in the timeframe of the next 20 years, the introduction of this scenario would significantly reduce this schedule. This development would have the effect of shifting timelines for Mars missions almost 20 years ahead. This would have the effect of making the projection of a Mars short-stay mission more feasible since it relies heavily on current technology and requires relatively little technology testing.

To placate concerns on Earth about contaminating the Martian biosphere as well as protecting any human crew from contamination, precursor missions such as the Phobos and Deimos missions and more robotic missions could increase in value. Landing site certification would become paramount, so as to ensure that the environment is not destroyed by astronaut landing activities. An ideal landing site would be close to the signs of life, but the landing site itself should be chosen so as to will not irreversibly disturb the local organisms. Further unmanned investigation would give mission planners a better idea of any contamination risk.

7.3.6.3 Associated Trades and Options

One trade that may be drawn from this scenario is a decision on the degree to which lunar missions will focus on activities that are not Mars-related, such as exploration of the lunar poles and of other sites of scientific interest. The fundamental decision that must be made is one of the science agenda versus the exploration agenda. On one extreme, every square meter of the Moon could be mapped and cataloged in a search for scientifically interesting phenomena that are associated with the lunar surface. On the other extreme is a situation in which NASA simply lands on the Moon as a technology demonstration before going directly to Mars in the name of exploration. A similar question may be asked of the degree to which life on Mars is studied. On the one extreme, it may warrant such in-depth analysis that humanity does not go beyond Mars for decades. On the other extreme, the presence of life may simply be confirmed for its news value before NASA proceeds to other locales. This scenario highlights a situation in which an option to explore the Moon was sidelined in favor of the option to explore the possibility of Martian life. In evaluating this choice, a decision must be made so as to ensure that the knowledge, which is most important to the key stakeholders is delivered.

7.3.7 Budget Catastrophe

7.3.7.1 Description

Motivated by election-year debates, a nationwide referendum reveals that the 60% of the population of the United States prefers to divert their tax dollars away from NASA towards programs such as inner-city development and veterans' hospitals. Congress cuts NASA's by 25% and restricts NASA's activity to education, remote sensing and Earth observation.

*7.3.7.*2 Response

A situation of this magnitude essentially puts a moratorium on all space exploration activities. If such an eventuality were to occur, the only way exploration activities would be reinstated requires direct intervention of the President or of Congress. In either case, this would probably have to be motivated by public demand or by outside political pressure (e.g., for foreign policy reasons). Although outside political concerns are beyond the control of NASA in any situation, the degree to which NASA could rekindle the public's interest in exploration is directly related to the knowledge that has already been gained. If there were significant knowledge available, NASA, which is responsible for educating the public on its space exploration activities, would be able to present this body of knowledge. Like most learning, this education would probably raise at least as many questions as it does answer. Thus, the natural curiosity and inquisitiveness of many people would contribute to public support for re-instatement of the space exploration initiative. Therefore, the worst-case scenario would involve this situation occurring immediately, before any new knowledge has been gained. In this situation, very little could be done by NASA to overcome the public's disinterest, particularly since NASA, like any other government agency, does not advertise. As NASA begins to collect more knowledge and to return more results and information to the public, it becomes less likely that the exploration initiative will be cut by public demand. In any case, NASA, as a government agency, does not advertise or push a specific political agenda. Thus any movement for space exploration would have to be independently initiated. Therefore, even in the very unlikely event that public support were to drop dramatically after a momentous act (such as the first human landing on Mars), there would be little NASA could do about it directly. It is therefore incumbent upon NASA to design a knowledge delivery system in such a way that it would prevent this scenario from occurring. One sure-fire way to do this is to keep the public's interest high. This could be accomplished through a regular series of reports and outreach activities in which NASA discusses its most recent accomplishments. The four year long election cycle provides an ideal length of time during which NASA may set a series of major milestones, which would be designed to engage the public and to maintain interest in space exploration. For example, at the end of one four-year cycle, a CEV prototype could be flown and docked with the ISS. Four years later, the first lunar CEV could land on the Moon. Each of these major milestones could be punctuated with a series of smaller yearly milestones. At the end of the fist year, for example, the CEV concept would be revealed to the public. The year after, an unmanned mockup would be flown. The year after that, test flights would occur, culminating in the ISS docking at the end of the fourth year. Not only would this approach constantly engage the public, thereby promoting sustainability, but also it would foster regular technical progress as NASA proceeded with the exploration agenda milestone by milestone, thus also promoting extensibility.

7.3.7.3 Associated Trades and Options

The primary trade that must be analyzed for this scenario is the degree to which NASA should concentrate on public education, inspiration and awareness campaigns. Keeping the public constantly informed is a difficult and time-consuming process. Furthermore, it opens the doors for criticism from the public. On the other hand, it may be argued that public criticism would improve NASA's operations in the long run, if taken constructively. Furthermore, NASA is ultimately supported by the direction of the President, who is, in turn, supported by a mandate from the public. If the public is allowed to lose interest in NASA, it is likely that funding will be cut, as the public's priorities shift away from space exploration and towards other, more immediate concerns. Ultimately, the American people are the primary beneficiaries of NASA's efforts and some of the prime recipients of the knowledge gained through exploration programs. If the public decides that exploration is not worth the associated cost in tax dollars, the program will be cut.

8. Conclusions

In conclusion, the following two recommendations are made to NASA:

- NASA needs to develop its own rigorous design methodology to incorporate sustainability into all levels of the space exploration architecture.
- NASA needs to view knowledge as the product of its exploration system and to ensure that the design and operation of this system is guided by the need to acquire, transfer, and process knowledge.

Future systems must be designed with sustainability in mind, ensuring maximal life cycle value (benefit at cost), as opposed to the traditional point design approach that optimizes missions based on a fixed set of requirements. An initial design framework has been presented as an example of proactively designing sustainable attributes into the exploration system. While NASA can certainly improve the process, the key message is that sustainability is not accidental; it must be actively pursued, and the short-term costs associated with designing for sustainability must be accepted in order to reap the long-term benefits.

This paper lays the foundation of a methodology for designing sustainability into space systems. It must also be stressed that a system must be sustainable throughout time. This requires that any potential design method must have the ability to be reevaluated throughout time, so that the design has the ability to react to uncertainties in the future.

Tools such as form/function mapping, commonality mapping, scenario planning along with formalized decision analysis, such as utility analysis and real options, were described and demonstrated in the proposed design process. These tools are only provided as examples of structured methods for complex system design, offering the potential for proper valuation of nontraditional system attributes such as sustainability.

The methodology and tools described in this paper support the design of NASA's new exploration system. As has been mentioned previously, the goal of this system is to deliver knowledge to all stakeholders. This goal must be kept continually in mind if the space exploration program is to be successful and therefore, sustainable.

It is the authors' hope that the readers take away a vision that current design methods are not sufficient to meet the goals of the new space exploration initiative and that a new design methodology must be developed. The new methodology accounts for sustainability and evaluates designs based on knowledge. While the proposed design method is certainly not the only solution, it is intended to be a starting point for further improvement, and ultimately, a catalyst that enables NASA and the nation to move into the next era of space exploration in a sustained and consistent fashion.

9. Appendices

9.1 Earth to Low Earth Orbit

9.1.1 CEV Model

The interface for the CEV model is shown in Figure 70.

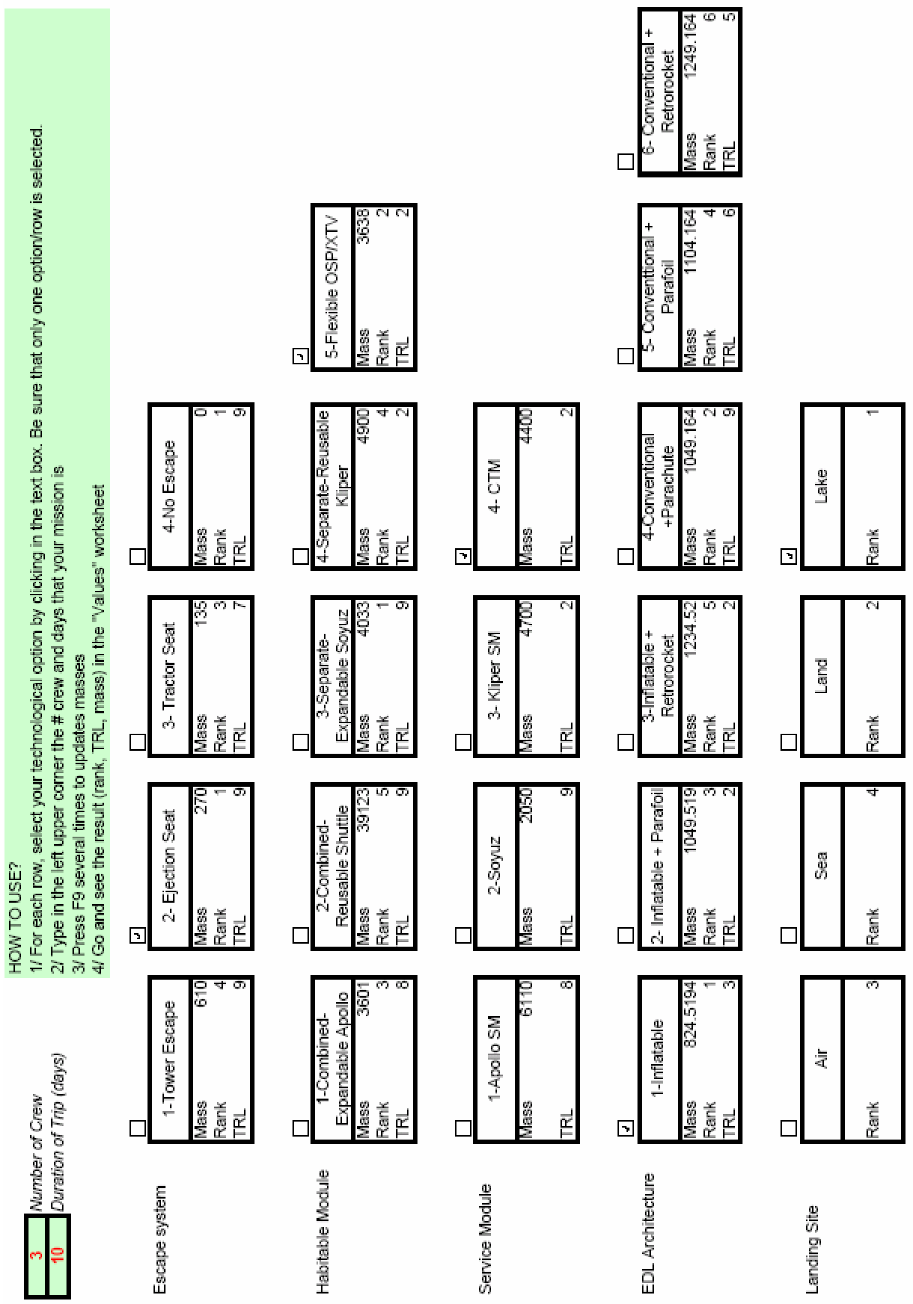


Figure 70: Interface used for the Excel CEV model

Figure 70 presents the interface used for the model. It is a view from the "choices" worksheet in the "architecture" Excel file. The user is required to enter the number of crewmembers and duration of the mission, and also to make choices for each of the five options:

- escape system
- habitable module
- service module
- EDL architecture
- landing site

All of the options cannot be linked together, so all of the possible combinations are shown in Figure 71. This architecture Excel file is linked to other Excel worksheets that are responsible for updating the masses for the option chosen. These are:

- "CES history" which give numbers for some the service module, and for some habitable modules
- "escape system" which give numbers for the escape system
- "re-entry_landing" which calculates the masses required to land the corresponding crew modules.
- "scaling" which computes the scaled masses of the options: 1-combined-expendable-Apollo, and 5-Flexible-OSP/XTV.

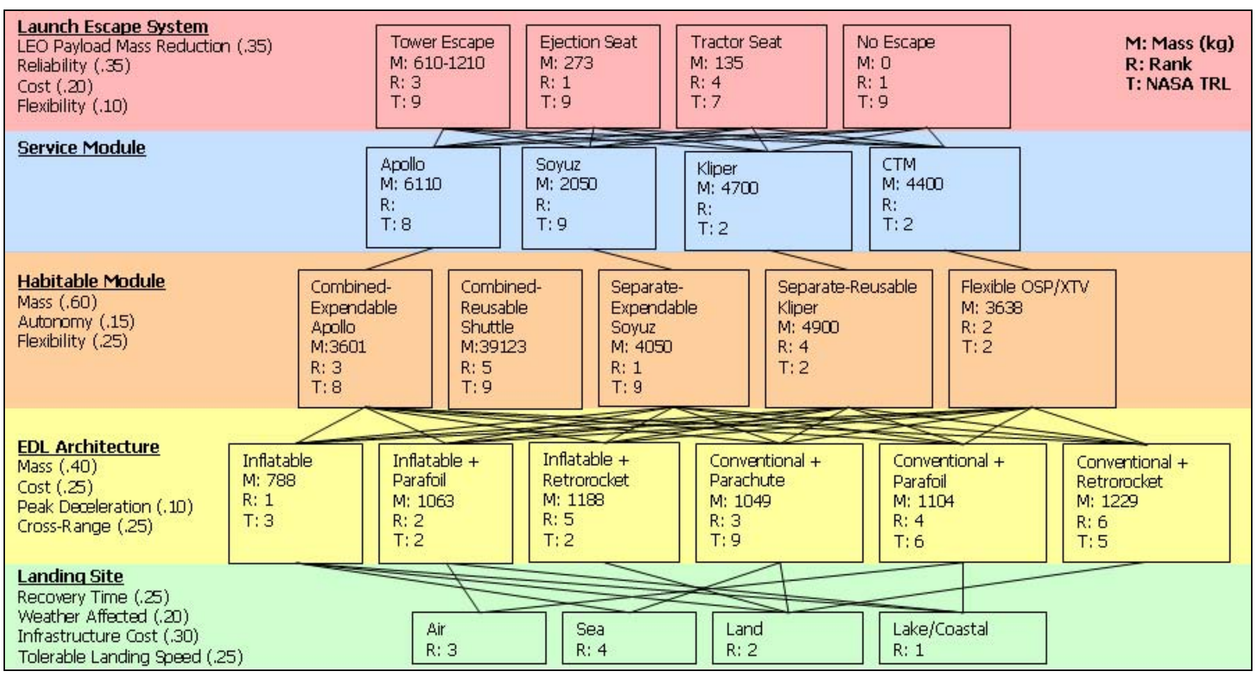


Figure 71: Linking possibilities among CEV options and ranking criteria and weights

9.1.1.1 Mass Calculation

For the service module, mass numbers were found from studies or real systems. These dry masses were not scaled depending on the number of crew and days of the mission.

For the habitable module, the Combined-Reusable-Shuttle, the Separate-Expandable-Soyuz and Separate-Reusable-Kliper configurations, were linearly scaled given a number of crewmembers. The duration of the mission was not included in the scaling.

For the Combined-Expandable-Apollo and the Flexible-OSP/XTV configurations, a detailed scaling was performed, including the number of crew and duration of the mission.

The reason for not scaling all the habitable modules configurations is that we lacked detailed mass breakdown for them.

All the existing spacecraft masses (and the breakdown and other spacecraft information) were found in one of the following references: (Wade, 2004) or (Zak, 2004) or (Braeunig, 2001).

For each system, a rank was determined according to the technique described in Section 6.4.2.4. It was a way to try to evaluate systems with the few available data while being as objective as we could. The criteria for evaluating each system (and their corresponding weights) are described on the left of
Figure 71.

9.1.2. Crew Module Scaling

A Report was written by the Orbital Aggregation & Space Infrastructure Systems (OASIS) titled, *The Revolutionary Aerospace Systems Concepts Preliminary Architecture and Operations Analysis Report (2002)*. This Report aimed to "identify synergistic opportunities and concepts among human exploration initiatives and space commercialization activities while taking into account technology assumptions and mission viability in an Orbital Aggregation & Space Infrastructure Systems (OASIS) framework." This Report provided detailed information about a proposed crew exploration vehicle and the component mass breakdown. The methods of analysis were explained as well as engineering design details of the structural components and hardware. Additional resources were used to augment the scaling analysis, including a NASA Report titled, "JSC Lunar Transfer Vehicle (LTV) design concept, *Crew Transfer Vehicle Element Conceptual Design Report*, EX15-01-094."

9.1.2.1. OASIS Crew Transfer Vehicle

For LEO crew transfer, the Crew Transfer Vehicle (CTV) was described by OASIS (2002) and summarized below. This module is designed for short sleeve environment transport from LEO to the Lunar Gateway and back, and to transfer crews between the ISS to any other crewed orbiting infrastructure (see Figure 72).

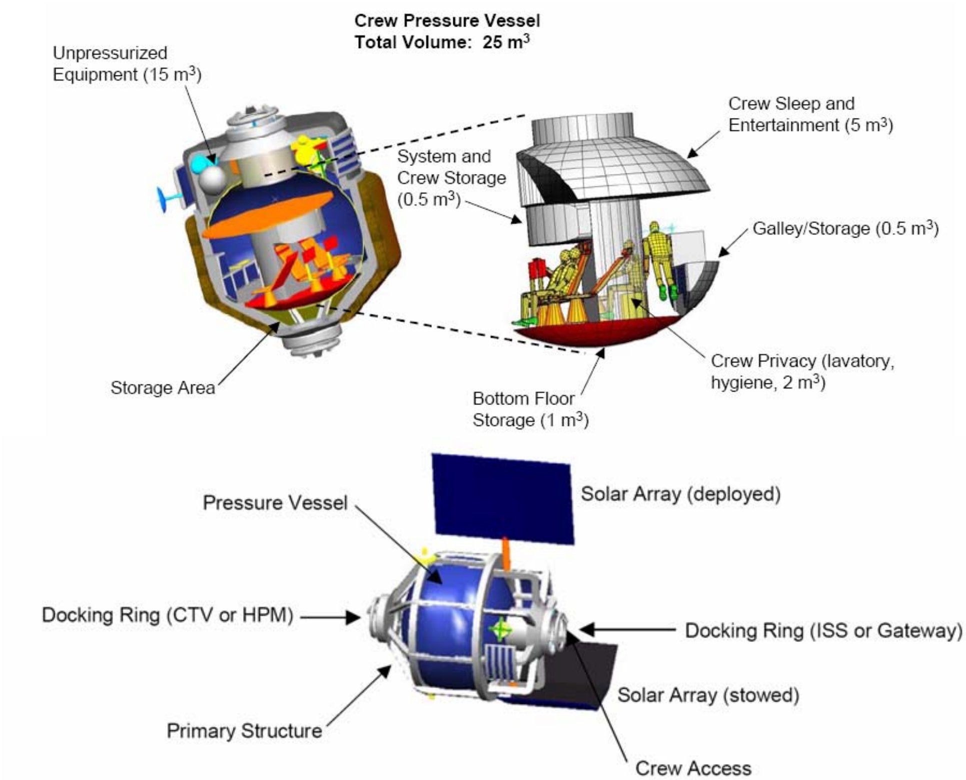


Figure 72: OASIS CTV Internal Layout

System requirements and mass properties have been derived from other OASIS elements as appropriate. CTV-unique system requirements and mass properties (e.g., for human habitability systems) have been derived from the NASA JSC Lunar Transfer Vehicle (LTV) design concept (*Crew Transfer Vehicle Element Conceptual Design Report*, EX15-01-094).

Vehicle specifics include,

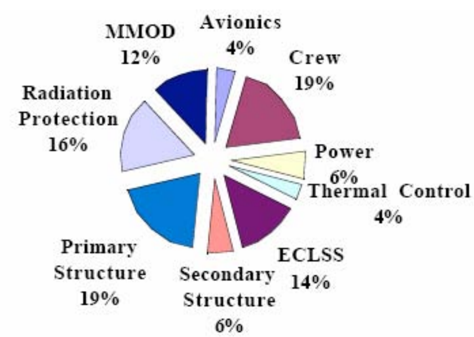
- Normal mission duration of 4.5 days for transfer from the ISS to the Lunar Gateway (9 day total transfer time from ISS to the Lunar Gateway and back ISS).
- Systems shall be sized for a 22-day extended contingency mission.
- Crew of four (deemed sufficient for operational requirements and mission science).
- Vehicle remains at ISS and is designed to travel to Moon L1 and return crew to ISS with a one-time return to the surface of the Earth in an emergency situation.
- Internal volume shall be sufficient to meet NASA minimal habitable threshold requirements of 4.25 m^3/person for a 22-day mission (NASA-STD-3000, Man-Systems Integration Standards).

- Systems shall meet all other human habitability and life support design requirements specified in NASA-STD-3000.
- Designed for launch by a Shuttle-class launch vehicle.

A preliminary estimate of CTV system mass is provided in given in Table 21 based on derivations from the HPM and the NASA JSC Lunar Transfer Vehicle. Note that the heat shield analysis was considered elsewhere.

Table 21: CTV mass estimation *(OASIS, 2001)*

System	*Source*	*Mass Estimate*	
Avionics	LTV Update 8-8-01	200	A
Crew Gear	LTV Update 8-8-01	672	
Crew Weight	LTV Update 8-8-01	332	
Power	HPM Derived	293	B
Thermal Control	LTV Update 8-8-01	217	
ECLSS	LTV Update 8-8-01	734	C
Radiation Protection	LTV Update 8-8-01	851	D
Pressure Vessel	LTV Update 8-8-01	213	E
Docking to HPM	HPM Derived	235	F
Docking Hatch	HPM Derived	272	
Structure	HPM Derived	338	G
MMOD	HPM Derived	624	H
Secondary Structure (20%)	HPM Derived	294	
	Total Mass (kg)	5275	



[A] Duplicate Hardware provided by HPM and CTM
[B] Used HPM Derived System (3.3 kW)
[C] 10% mass reduction for no O2 fuel cells or no contingency EVA capacity
[D] 10% mass reduction for integrating Radiation Protection system with MMOD & primary structure
[E] 10% mass reduction for resizing pressure vessel and using advanced materials
[F] Hatch replaced
[G] Resized CTV length and used advanced materials
[H] 10% mass reduction for integrating Radiation Protection system with MMOD

9.1.2.2. Habitable Volume Analysis

From NASA Standards (8.6.2.1 Mission Duration Design Considerations), the duration of the mission has an overall effect on the required envelope geometry. Increasing mission duration requires a greater physical envelope to accommodate mission tasks and personal needs. Crew accommodation needs are additive, so the total required habitable volume per crewmember increases with mission duration. Guidelines for determining the amount of habitable volume per crewmember for varying mission durations are shown in Figure 73.

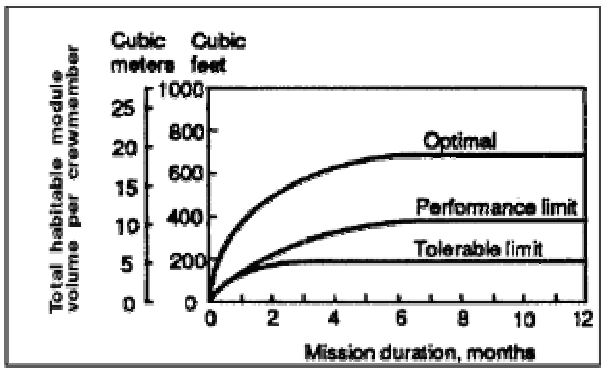


Figure 73: NASA Habitable Volume Standard 8.6.2.1

Initially, it was thought that his standard would provide a reasonable approximation of the total volume required for a given mission duration. However, it was thought that as the number of crew increases, the habitable volume per crewmember should decrease. Therefore, Figure 74 was assumed to better approximate the habitable volume requirements for a given crew, for a given mission duration.

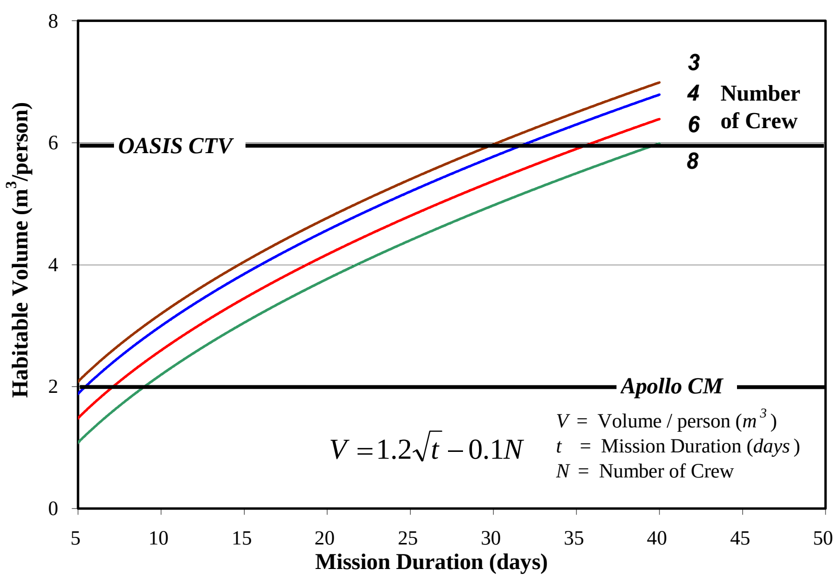


Figure 74: Habitable volume for various crew sizes as a function of mission duration

The habitable volume per person for the Apollo Command Module and the OASIS CTV are shown as a reference. By comparing the masses of various crew modules (Gemini, Mercury, Apollo CM, OASIS CTV and Soyuz), while holding the habitable volume constant, the various architectures were compared.

9.1.2.3. Mass Breakdown Scaling Relationships for OASIS CEV

Based on the mass breakdown described in the OASIS Report, each component needed to be considered independently. Based on the type of component, its scaling relationship was chosen accordingly.

The following components of the mass breakdown were assumed to be independent of the number of crew:

1. Avionics,
2. Docking to HPM,
3. 2nd Docking Interface, and
4. Power.

The following components of the mass breakdown were assumed to scale in direct proportion with the number of crew:

1. Crew Gear,
2. Crew Weight, and
3. Environmental Control and Life Support System (ECLSS).

The following components of the mass breakdown were assumed to scale in direct proportion with the external vehicle surface area:

1. Radiation Protection,
2. Structure,
3. MMOD, and
4. Solar Arrays.

The following components of the mass breakdown were assumed to scale in direct proportion with the habitable volume:

1. Pressure Vessel mass,
2. Thermal Control.

A more detailed discussion of scaling is provided for specific components of the mass breakdown.

9.1.2.3.1. Power

The mass of the Power Systems was derived from the HPM, which consisted of,

- *Propellant Management System* (Zero Boil-off cryogenic cooling system, plumbing, instrumentation, data acquisition) – 2777 W
- *Guidance, Navigation and Control* (scanner, Inertial Measurement Unit - IMU) – 40 W
- *Communications and Tracking* (flight computers, transponders, solid state recorders) – 103 W
- *Thermal Control System* (Heaters, Adhesives, Controllers, Thermostats, Temperature sensors) – 155 W
- *Electrical Power System* (Power Distribution System) – 505 W

Therefore, a total average power level of 3.580 kW was predicted. When scaling these values it was assumed that the cryogenic cooling system (the largest power load for the propellant management system - PMS) was independent of the number of crew. However, the power load will be significant when in-space propellant requirements are specified later in the project. The Guidance, Navigation & Control (GN&C) was assumed to be independent of the number of crew. This was also assumed for Communications & Tracking System (C&T) and the Electrical Power System (EPS).

As the number of crew increases, the pressurized vessel volume increases proportionally (according to NASA-STD-3000, Man-Systems Integration Standards). Based on conduction heat transfer, the increased surface area permits increased heat loss to the exterior of the vehicle. Since the heater power load is ~5% of the total heat load, the increased mass associated with a larger heating unit would only slightly increase the total mass of the vehicle (on the order of less than 1%).

9.1.2.3.2. Radiation Protection

A radiation protection layer is added to the vehicle exterior and as such, its mass is directly proportional to the exterior surface area of the vehicle. The exterior surface area was calculated for the base case (4 crew XTV).

According to Figure 75, the exterior was modeled as the following,

$$V = \frac{\pi}{4}E^2 A + \frac{\pi}{4}F^2 B + \frac{\pi}{12}C\left(F^2 + D^2 + FD\right)$$

Since many of the mass contributions are a function of the vehicle surface area it was critical to calculate the approximate exterior surface area. Since docking hatches and other exterior attachments are located on either end of the vehicle it was assumed that only the main cylindrical volume and the lower conical volume contributed to the exterior surface area. Therefore,

$$A \approx \pi FB + \frac{\pi(F + D)}{2}\sqrt{C^2 + \left(\frac{\pi F - \pi D}{2}\right)^2}\ .$$

The mass of the crew of four XTV, and the mass of the radiation protection material was scaled per external surface area. Based on the new vehicle geometries required for a larger number of crew the new radiation protection material mass could be predicted.

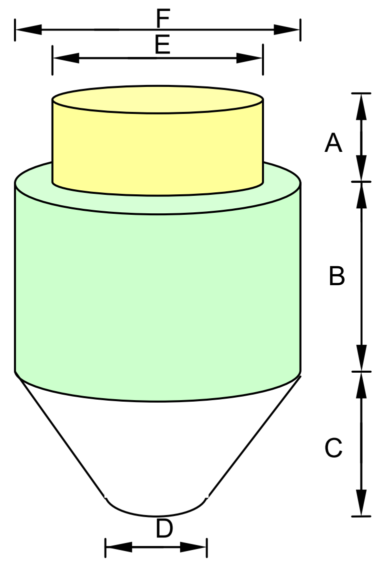


Figure 75: XTV scaling model

9.1.2.3.3. Pressure Vessel

A surface area relationship was also assumed for the pressure vessel. However, for the pressure vessel, the surface area of the spherical vessel was used and scaled based on the habitable volume of the vehicle.

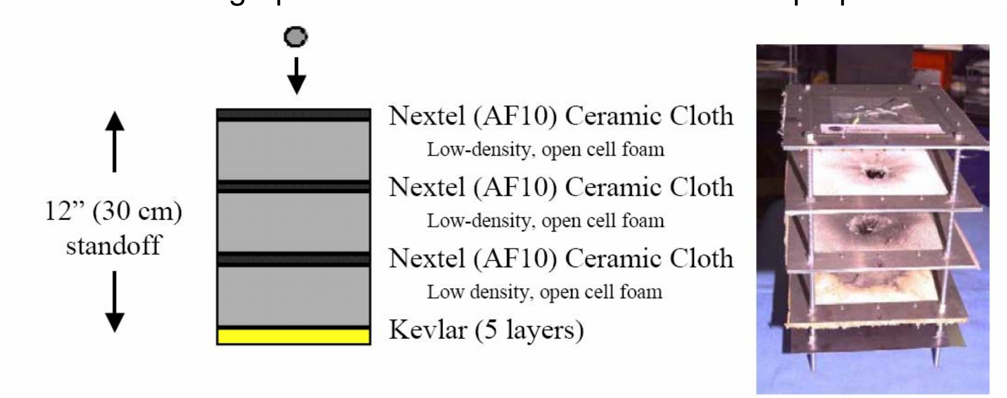
9.1.2.3.4. Structure

The materials proposed for the Hybrid Propellant Module (HPM) were described by OASIS (2002) and are summarized below. The materials for this module are similar to the materials proposed for the Crew Transfer Vehicle (CTV). The HPM structural system meets the requirements of NASA Standard 5001, Structural Design and Test Factors of Safety for Space flight Hardware. This module can withstand the launch loads from a Shuttle-class RLV or an augmented Delta IV-Heavy ELV.

To protect this module and its contents from impacts due to micrometeoroids and orbital debris a *Micrometeoroid and Orbital Debris Protection* (MMOD) exterior shield was proposed. The designed shield is capable of withstanding an impact with no penetration from a 4mm diameter aluminum projectile with an impact velocity of 7km/s (ISS orbital velocity).

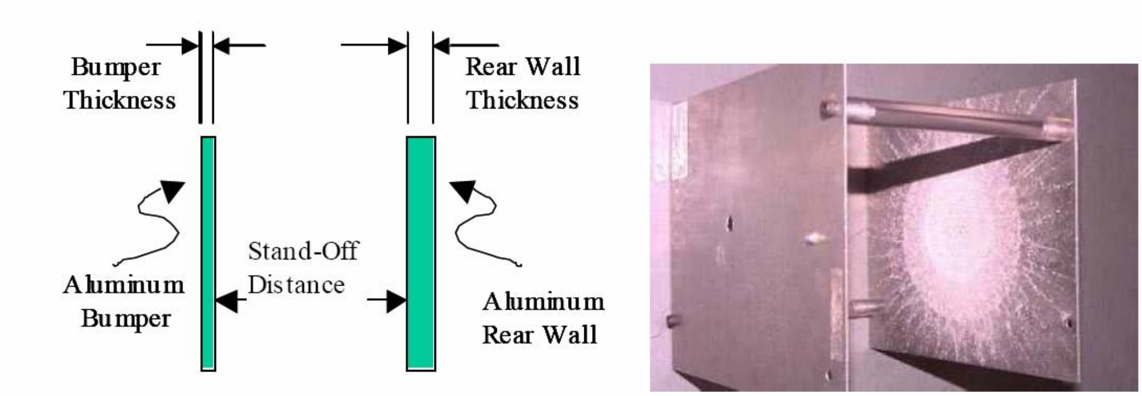
The longerons (axial members) are made from a magnesium metal matrix with long fiber carbon strands. This composite has a structural I beam cross section (S20cm-

15cm), which facilitates the attachment of the MMOD. Both the upper and lower sections contain these types of members. For the upper section, the skin surrounding the shield is made of five layers of Kevlar fabric and epoxy composite. This serves as a stiffener to the main longeron and ring base structure. Space is left for layers of radiation protection of insulation for thermal protection. This section includes three layers of Nextel ceramic cloth that provides thermal protection (see Figure 76). The 30cm exterior thickness contains alternating layers of low density, open cell foam. This foam is a carbon-based graphite material with excellent thermal properties.



(Courtesy of OASIS, 2002)
Figure 76: HPM upper section material

For the HPM lower section, the exterior is slightly different. A Whipple type shield was chosen for the lower section MMOD shielding and thermal protection (see Figure 77). The shield is made of syntactic aluminum metal foam to minimize the material density, while maintaining sufficient strength.



(Courtesy of OASIS, 2002)
Figure 77: HPM lower section material

Tapered longerons were used for the internal support structure of the CEV. This was done to ensure the 4-G load (HPM thrusting) could be sustained. The upper half was untapered to maximize volume in the unpressurized area. It was also expected that the maximum loading during docking would be light.

The CTV MMOD shield design is conceptually similar to the HPM MMOD shield. This section incorporates an expandable multi-shock design, which is deployed on-orbit.

Non-expandable syntactic aluminum foam is used on the upper section to avoid potential complications with shield deployment around externally mounted systems (including solar arrays and radiators).

The number of longerons supporting the XTV structure was 8 (crew of 4). Therefore, it was assumed that the number of members was directly proportional to the exterior surface area. To a first order approximation, buckling scales well with surface area, provided additional ring supports could be added where necessary. This provides only a minor mass penalty and since the structural load per circumferential length will remain the same, this is a good assumption. It was assumed that the ring structure attached to each member also scales with the surface area. This is a reasonable assumption, as the support members constitute a greater portion of the total structural mass (assuming the same material is used for vertical members and ring base structure).

9.1.2.3.5. Micrometeoroid and Orbital Debris (MMOD)

This exterior protective structure was assumed to scale with the exterior vehicle surface area in a similar manner as the Radiation Protection layer.

9.1.2.3.6. Secondary Structure (20%)

Similar to Radiation Protection and the MMOD, the Secondary Structure was assumed to scale with the exterior vehicle surface area.

9.1.2.3.7. Solar Arrays

Assumed to scale with the exterior vehicle surface area.

9.1.2.3.8. Thermal Control

At this stage of the analysis it was assumed that the Thermal Control mass scaled with the volume of the pressurized vessel. This seems reasonable as the pressurized volume scales with the number of crew (NASA-STD-3000 90).

Each of the components was scaled according to the parameters specified above. The original vehicle proportions were maintained when the size was increased. Details of the vehicle proportions can be found in the Section 9.1.1.

9.1.2.4. Mass Breakdown Scaling Relationships for Apollo Command Module

A similar analysis was performed for the Apollo Command Module (CM) as was performed for the OASIS CEV.

The command module was approximated by a conical structure. The pressurized inner shell was fabricated from aluminum honeycomb panels and separated from the heat-resistant outer shell by a micro-quartz fiber insulator.

The external surface area (see Figure 78) was calculated as,

$$Area \approx \frac{\pi}{4} A^2 + \frac{\pi A}{2} \sqrt{\frac{A^2}{4} + B^2} \; .$$

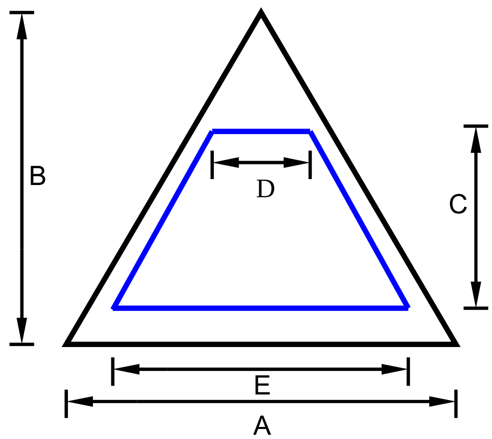


Figure 78: Apollo CM schematic

The mass breakdown of the Apollo CM is shown in Table 22.

Table 22: Apollo CM mass breakdown (*http://www.astronautix.com/craft/apolocsm.htm*)

System	***Mass Estimate***
Structure	1985
Reaction Control System	400
Recovery Equipment	245
Navigation Equipment	505
Telemetry Equipment	200
Electrical Equipment	700
Communications Systems	100
Crew Seats & Provisions	550
Crew Mass	216
Miscellaneous Contingency	200
Environmental Control System	200
Propellant	95
Total Mass (kg)	**5396**

9.1.3 Elements of the Heavy Cargo Shuttle Derived Vehicles Study

9.1.3.1 Useful Definitions

- A5G: Ariane 5 "Generique". It is the reference European commercial launcher.
- ET: External Tank. It is the external tank where de LH2 and LO2 are stored in the shuttle.
- I_{sp}: Specific impulse. It is a measure of the performance of an engine-propellant combination. For a given system, it increases with altitude and is maximum at vacuum. It is measured in seconds. If you have a kg of propellant and you burn it to produce 1 kg of thrust, the I_{sp} is the number of seconds it lasts.
- SRB: Solid Rocket Booster. Has a lot of thrust but low I_{sp}, good for lifting off.
- J2: high energy LH2 LO2 upper stage from the Saturn 5 third stage.
- SL: Sea level
- P/L: Payload
- LEO: Low Earth Orbit, an orbit between 150 to 1000 km over the Earth surface. Unless otherwise stated, it is assumed to be of 280 km.
- Mission delta-V: The sum of the ideal delta-V needed for a particular mission plus all the other velocity losses.
- SSME: Space Shuttle Main Engine, it uses liquid H2 and liquid O2. It has a high I_{sp} but it is expensive and complicated.
- RS68: Main engine of the Delta IV common core, it uses liquid H2 and liquid O2, it has a lower I_{sp} than SSME and it is not human-rated, but it is cheaper and simpler.
- STS: Space Transportation System. It is the whole system commonly known as shuttle.
- Pod: Canister where the payload is stored in an STS-derived heavy launch vehicle.

9.1.3.2. Assumptions

- I_{sp} of the SRB is constant and average between SL and vacuum.
- Pod weights 10000kg.
- I_{sp} of SSME and RS68 is equal to the SL value while the SRBs burn and equal to the vacuum value afterwards.
- Massflow is constant for each engine.
- SSME can be throttled to 109% of the nominal thrust at lift off.
- The I_{sp} during parallel burn is the weighted average using the massflow rate as the weight.
- Inline and piggyback configurations of STS-derived heavy launch vehicles are considered equivalent.
- The gravity, turning, and aerodynamic losses are the same for all STS-derived vehicles.
- The payload mass to LEO assumes a total delta-v (including all losses) of 9,400 m/s, the mass to ISS assumes total delta-v of 9,600 m/s, and the mass to escape assumes delta-v of 11,600 m/s

9.1.3.3. Missions with Minimal Hardware Development

No high-energy second stage would be developed or revived.

9.1.3.3.1. Launch Sequence

- Parallel burn of the SSMEs (or RS68s) and SRBs for 123 seconds
- SRB separation.
- SSMEs (or RS68) continue to burn until they ran out of propellant.
- SSMEs (or RS68) and ET separate
- Payload separates from the Pod.

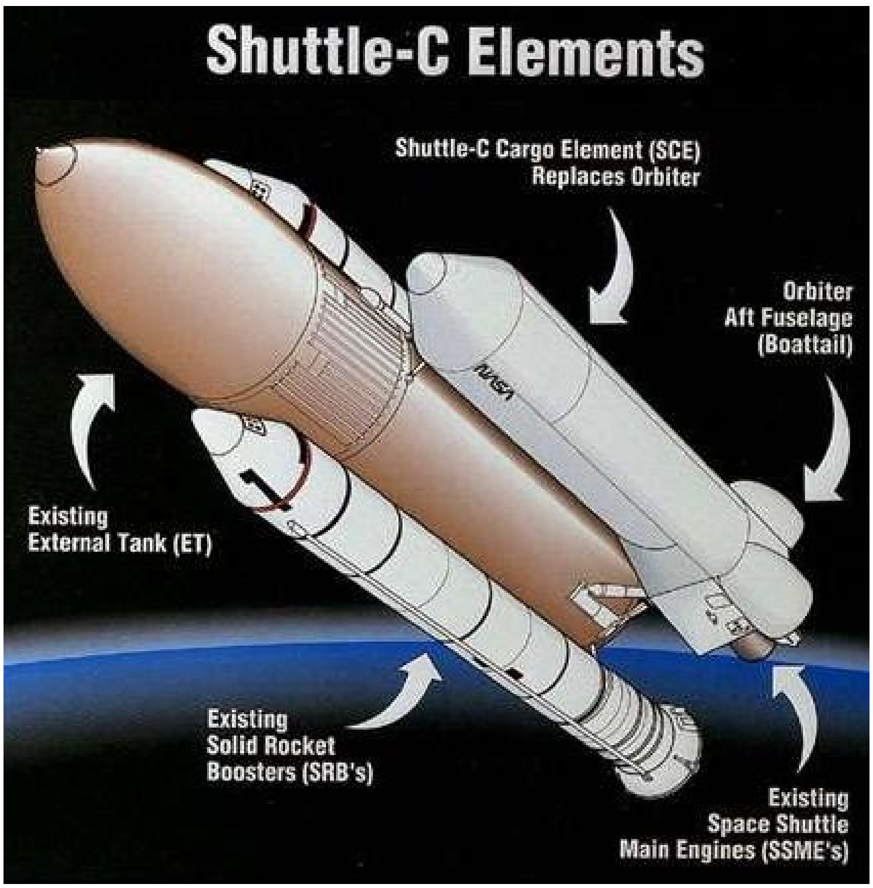


Figure 79: Shuttle-C elements (Source: NASA)

9.1.3.3.2. Performance Curves

The curves show total mission delta-V versus P/L mass for various launchers.

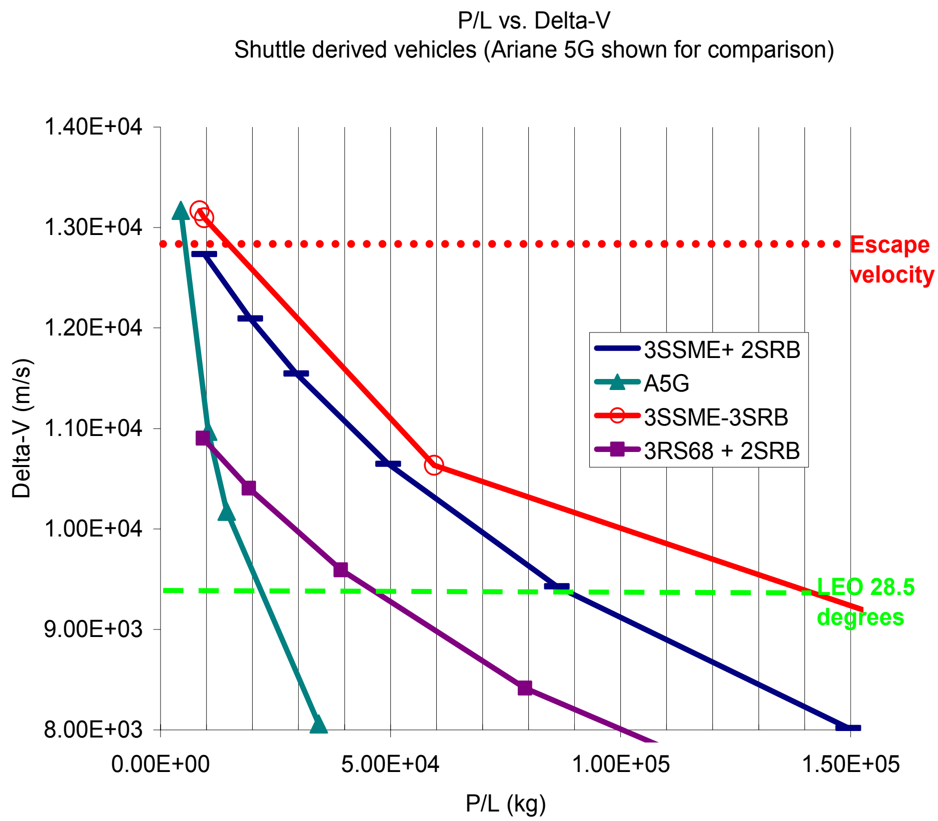


Figure 80: Performance curves

9.1.3.4. Missions with Some Hardware Development

A high-energy second stage of the J2-class, similar to the third stage of the Saturn five would be developed or revived.

9.1.3.4.1. Launch Sequence

- Parallel burn of the SSMEs (or RS68s) and SRBs for 123 seconds
- SRB separation.
- SSMEs (or RS68) continue to burn until they ran out of propellant.
- SSMEs (or RS68) and ET separate
- J-2 class stage burn.
- Payload separates from the Pod and J-2 class upper stage.

9.1.3.4.2. Performance Curves

The curves show total mission delta-V versus P/L mass for various launchers.

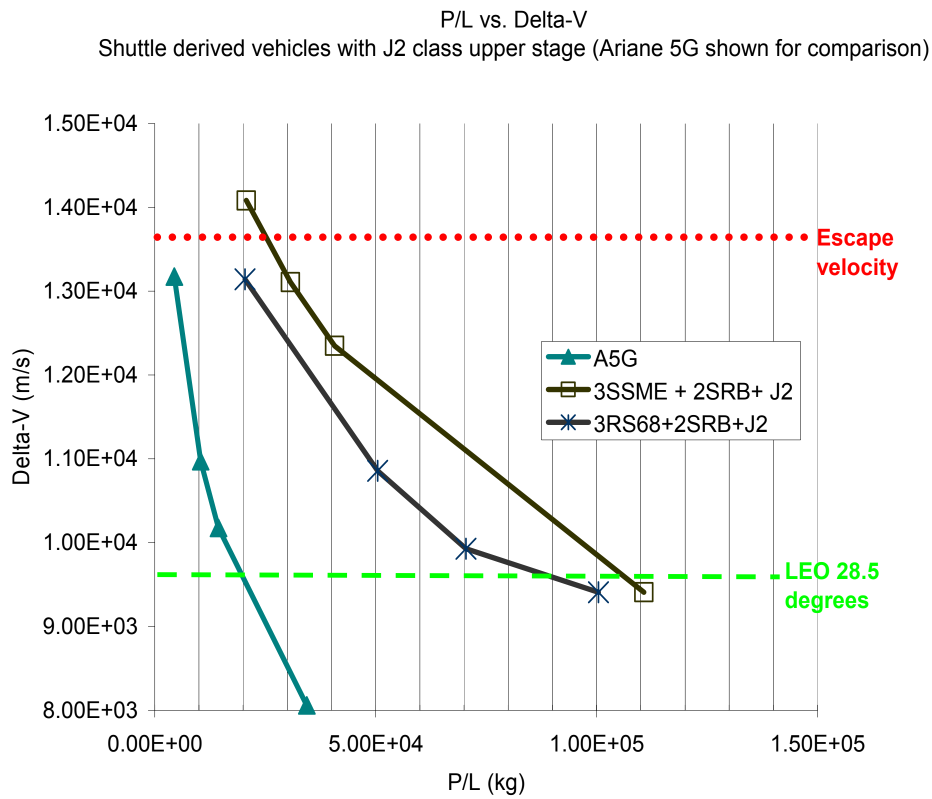


Figure 81: Performance curves

9.1.3.5. Estimates of the Mass in LEO for Exploration Missions

Table 23: Mass requirements in LEO (ISU SSP Report 99')

Project - Mission	Payload Mass Demand (tons)
90 day study - Moon mission	60-100 tons (A. Cohen, 1989)
90 day study - Mars mission	140 tons
Synthesis group on America's space exploration	150 - 250 tons (T.P. Stafford, 1991)
Alternative infrastructure study from General Dynamics	98 - 150 tons (General Dynamics, 1993)
NASA Mars reference human mission. Version 1.0	240 tons (S. Hoffman, 1997)
NASA Mars reference human mission. Version 3.0	80 tons

9.1.3.6. Conclusions

Table 24 summarizes the calculations made on the various STS derived options.

Table 24: Various STS-derived options

	Launchers with a newly developed high energy J2-class upper stage		Launchers with existing components		
	3SSME + 2SRB+ J2	3RS68+2SRB+J2	3SSME+ 2SRB	3SSME + 3SRB	3RS68 + 2SRB
LEO (407km)	110694	100424	86124	100813	49118
P/L to ISS	103915	79151	80176	93299	39225
P/L to Escape	50425	30694	Unpractical range		
TRL	5	4	6	4	5

The most attractive combination for heavy cargo launch is 3RS68, 2SRB and a J2 class upper stage.

- It has 1 engine out capability.
- It uses RS68 which are not human rated and therefore cheap.
- It matches nicely with the NASA Mars reference human mission v. 3.0

9.1.4. EELV assessment

The Evolved Expendable Launch Vehicle program (EELV) is the backbone of the US defense related launch complex. It includes two families of launchers the Delta IV by Boeing and the Atlas V by Lockheed Martin. Although commercially uncompetitive the Delta IV and the Atlas V have the monopoly of the medium heavy defense related payloads in the USA. For this strategic reason, the continuity of both programs is almost guaranteed. The capacity of production is about 10 a year for each vehicle. Existing EELV designs are modular and the Atlas V and Delta IV families have similar performance envelopes, although the Delta IV has a higher specific impulse because instead of using kerosene it burns hydrogen in the fist stage.

In the framework of the new human exploration initiative there are two potential uses of EELV launchers, one is cargo and the other one is human transportation.

Concerning cargo there are two approaches that can be based solely on EELV technology. One is to use the Delta IV Heavy to launch the heavy exploration payloads in pieces of about 20000 kg. This approach has been studied in Section 6.4.2.2.2. The other approach is to develop a new heavy launcher based on Delta IV technology. One such concept could be a Delta-IV with five common booster cores (instead of the three used on the Delta IV Heavy) and a J2 class upper stage. We will refer to it as Delta –IV Super. The Delta IV is assembled horizontally, that would not be possible for the Delta-IV Super, and substantial changes in the way that the launcher is assembled would be required. According to our calculations this option promises payload capabilities that are slightly inferior to those with an STS based architecture. A Delta-IV super would require new infrastructure and more development work than an STS based launcher to arrive at a lower performance. From our simplified analysis it seems that it is a less mature and technically inferior option than to use the current EELV fleet or an STS derived.

Concerning human transportation it should be first noted that the EELVs were not conceived to send humans, but classified payloads and commercial telecommunications satellites instead. The fact that the vehicles are not man rated does not mean that it cannot be done very effectively; as an example: the most family of vehicles that has ever been used to launch humans, the Soyuz (originally the R7 missile), was conceived to launch thermonuclear warheads.

The launch pads would have to be modified to allow for the access and escape of astronauts. This would however be less costly than designing new launch pads from scratch.

Table 25 shows some characteristics that have been calculated for EELV vehicles.

Table 25: Various STS-derived options

	Delta-IV Medium	Atlas-V (552)	Delta-IV Heavy	Delta-IV Super
G.L.O.W. (no payload)	241,160	533,749	677,220	1,209,493
Payload to LEO (407km)	10,081	11,446	18,531	51,599
Payload to ISS	9,308	10,392	17,000	47,724
Payload to Escape	3,873	3,566	7,162	20,508
Tech. Readiness Level	9	9	8	6
Reliability	94%	87%	94%	90%

- All masses in kilograms
- Payload mass calculated assumes no fairing and no escape system
- Reliability is estimated from current flight history and the flight history of the vehicle generation immediately prior to EELV (e.g. Delta-II and Atlas-II Centaur)

9.1.5 Solid Rocket Booster derived launcher considerations

It has been argued that a single Solid Rocket Booster (SRB) is a viable alternative to launch humans. In this section we will evaluate the attractiveness of using SRBs to launch crews.

The concept has some advantages. First, the vehicle components are already human rated since they are used on the STS. Second there has been only one SRB failure of in 113 flights. That is a 99.5% reliability, which is similar to that of the Soyuz. Another advantage is that it supports the business of ATK Thiokol, a critical supplier of the strategic nuclear forces of the USA.

It should be noted though that the reliability of an SRB as a part of an STS stack cannot be simply assumed to be preserved in an SRB based launch system.

This concept would require new launch pads and ground infrastructure,

A concern that is often raised up is the environmental effect of the SRB plumes. Due to the low flight rate (in any case less than 12 a year) that we expect for the program, we consider that effect to be minor.

A more serious concern is the safety of the use and storage of stages that are constantly fully loaded with explosive. An explosion of an SRB in the VAB would be very damaging and it is a scenario that, although unlikely would be catastrophic. Naturally this applies to the STS derived launcher because it uses SRBs too.

A study of the capabilities of various combinations of the SRB with different high energy cryogenic upper stages has been performed and is summarized in Table 26:

Table 26: Various combinations

	SRB+					
	Launchers with a newly developed high energy J2-class upper stage			Launchers with existing components		
	Saturn V - 2nd Stage (S-II)	Saturn V - 3rd Stage (S-IVB)	Beal BA-2 2nd Stage	Delta IV Heavy (development)	Ariane 5 - 2nd Stage	Ariane 5 - Cryogenic 2nd Stage
G.L.O.W. (no payload)	1,197,757	825,779	895,979	737,689	719,479	734,479
Payload to LEO (407km)	~104000	~33400	~23500	~12450	~860	~11100
Payload to ISS	~98600	~30600	~21200	~11560	~545	~10000
Payload to Escape	~54600	~14500	~5100	~4900	ot feasable	~4680
Tech. Readiness Level	6	6	6	7	7	7
Reliability	0.99558	0.99558	0.9956	0.9956	0.99558	0.9956

All masses shown in the table are in kilograms. The payload mass has been calculated assuming no fairing and no escape system. It should be pointed out that SRB with S-II combination involves a rather unusual geometry and mass distribution, the diameter of the second stage being 2.75 (10.2/3.71) times than that of the SRB. The aerodynamic, structural and control problems that such a configuration would have are at first sight very large however its assessment is beyond the scope of this report.

An important concern using this vehicle is the peak acceleration. Since the SRB has a very high thrust to weight ratio a very heavy mass would have to be launched every time that an SRB based vehicle is used to launch humans. To keep the STS requirement of less than 3 g is not necessary since, in the context of the new exploration missions, it will not be claimed that almost anybody can use the system, as was the case on the shuttle. 5 g is a more reasonable threshold. The minimum payload in LEO compatible with an acceleration of 6g has been calculated to be 47000 kg for a 5 g limit, 74500 kg for a 4 g limit and 116,000 kg for a 3g limit. Comparing these minimum masses to the payload masses with the various upper stages show that the only option that would have an acceleration of less than 5 g would be the SRB plus S-II stage. As has already being commented that option has problems in its geometrical configuration. This problem would not be easily dealt with just by designing a new upper stage with a different geometry. Actually, the S-II is quite a slender body. The physical reason can be traced back to the different in density of the propellants used in the SRB and the upper stages. Due to the very low density of liquid hydrogen, cryogenic stages occupy a very large volume. If the diameter of an upper stage where reduced by a 30% the diameter of the SRB then, to preserve the volume, the length of the stage would have to be increased by a 57% making then the second stage longer than the SRB.

This consideration renders the use of a single SRB to launch astronauts troublesome.

9.1.6 Penalty of 1kg

The effect on the final P/L mass of 1kg of inert mass placed on a stage has been evaluated for a typical LEO mission for both an A5G and an STS-derived that used SSME and a J2-class upper stage.

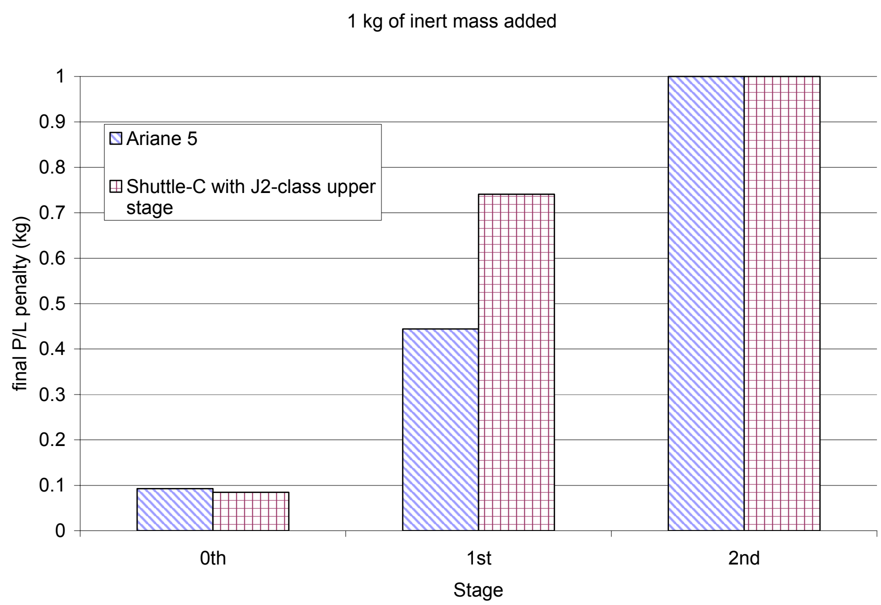


Figure 82: Ariane V and STS-Derived

9.1.7 STS derived assembly platform

For missions requiring on orbit assembly, such as large Moon missions or any Mars missions, it may be useful to have an assembly platform with a robotic arm. Using STS legacy hardware, this capability could be achieved rather easily. To be able to launch such a platform the STS Based has to be side mounted. It should be launched on a low inclination LEO. External Tank Corporation studied the development of a space station using the external tank as living quarters and a modified orbiter such as the one that would be needed for this assembly platform. The cost was estimated to be $3 billion FY92.

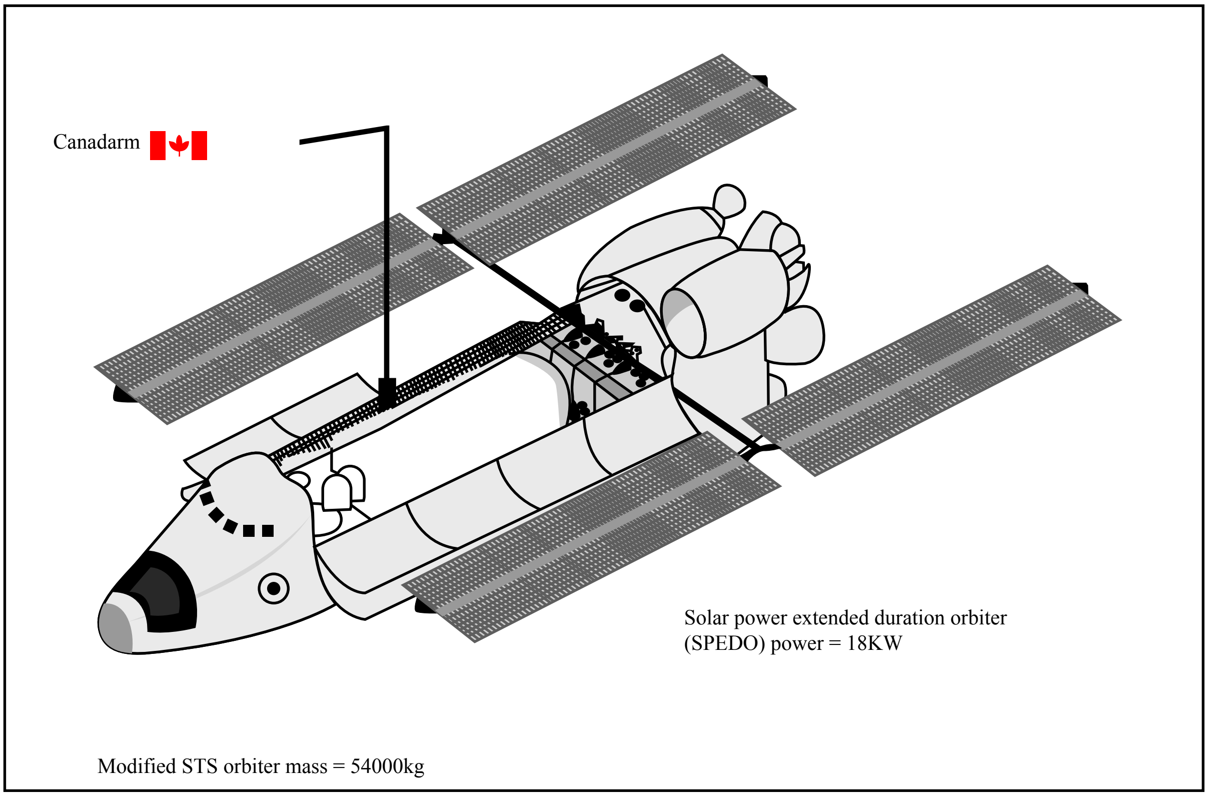


Image by MIT OpenCourseWare.

Figure 83: STS derived assembly platform

9.1.8 LabView tool for evaluating launch capabilities

9.1.8.1 Introduction

The tool described in this Appendix was developed in LabView, and provided a way to evaluate launch capabilities that could be considered for crew or cargo launch. Two types of analysis can be done: a combinatorial evaluation of the possible architectures and an evaluation of a single architecture selected by the user.

9.1.8.2. Evaluation of a single architecture

Figure 84 shows a view of the graphical user interface. The following paragraph describes how to use this simulation tool, and the results it generates for the evaluation of a single user-defined architecture.

- Step 1: The user enters the parameters available for each option:
    - masses, TRL and rank for the EDL technology, the habitable module and the launch escape system
    - **mass capability of lift for the first stages (4th and 5th rows of** Figure 84)
- Step 2: The user selects one of each option (knowing that all the combinations between rows are not always possible) with the help of the rectangular select button on the right.

Step 3: The user runs the simulation to obtain the results, which are displayed in the bottom - right corner of Figure 84.

The program outputs include,

- The average TRL for the selected architecture as well as the lowest TRL among the selected options of this architecture;
- The mass margins for launch capabilities for ISS and 28.8 deg destinations, as well as the mass margin for the escape system. The mass margin is defined as the extra mass that the launcher could launch in addition of the crew module. Note that in this part of the study, no care was taken of separating human-rated and cargo launches;
- The rank of the crew module architecture;
- The reliability of the launch system in percentage.

Figure 84: GUI interface for the LabView combination tool

9.1.8.3. Combinatorial evaluation of the possible architectures

The advantage of using LabView was also in that it enabled evaluation of all the combinations for each option of CEV (+ EDL + crew escape system) and launcher.

Figure 85 shows the result of such a simulation. LabView evaluated, for 999 different architectures, the mass margin to ISS in kg. The mass margin is defined as the extra mass that the launcher could launch in addition of the crew module. For some architecture, the mass margin is negative, which means that the selected launcher couldn't lift the selected crew module. On this graph, too, some gaps are presented (red arrows) and the corresponding launch technologies, which enabled launching the mass (3RS68+2SRB, Delta IV super, 3SSME+2SRB).

For the final decision of a launch system, not only the masses (given by this model), but also cost and feasibility issues should be looked at, these are not included in this model.

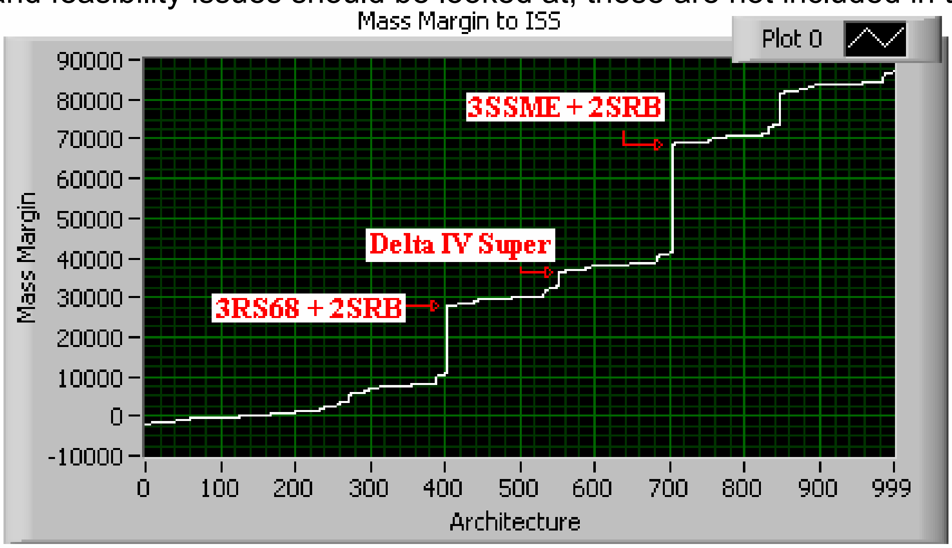


Figure 85: Mass margin to ISS for 999 options of launch + CEV configurations

9.2 Space Transportation

9.2.1 Form/Function Matrix

Shown in Table 27 are the detailed requirements for the Mars and Moon missions. These requirements are discussed in greater detail in their respective Report chapters.

Table 27: Form/Function matrix

	Moon			*Mars*			
Habitation Module (HM)	**Short**	**Medium**	**Extended**	**Phobos**	**Short**	**Extended**	**Extended+**
Support a Crew of 6	-	-	X	X	X	X	X
Human Life Support for 3 weeks	-	-	X	-	-	-	-
Human Life Support for 360 days[A]	-	-	-	-	-	X	X
Human Life Support for 600 days[B]	-	-	-	X	X	-	-
Aerocapture to Orbit	-	-	X	-	X	X	X
Dock with COV	-	-	X	X	X	X	-
Dock with SM1/SM2	-	-	-	X	X	X	-
Dock with MCM1/MCM2	-	-	X	X	X	X	-
Dock with Landers	-	-	X	X	X	X	-
Dock with ISPP-SHM on Surface[C]	-	-	-	-	-	-	X
Sustain itself in Unmanned Orbit for Extended Periods	-	-	X	X	X	X	X

[A, B] Duration of transit varies depending on the year of departure from Earth

[C] Similar to Mars Direct Architecture, COV goes direct to Mars surface, returns via ISPP

	Moon			*Mars*			
Crew Service Module (SM)	**Short**	**Medium**	**Extended**	**Phobos**	**Short**	**Extended**	**Extended+**
Dock with COV/HM	-	-	-	X	X	X	-
Ability to be connected in stages	-	-	-	X	X	X	X
Insulation for zero cryogenic boiloff	-	-	-	X	X	X	X
Ability to be prepositioned in Mars orbit	-	-	-	X	X	X	X
Make fuel for return trip[D]	-	-	-	-	-	-	X
Ability to dock with Hab/COV/Lander[E]	-	-	-	X	X	X	X

[D] Direct architecture - H2 feedstock sent ahead of time, fuel made and stored for return trip, fuel connects to the HM/COV for the return to Earth

[E] Moon - Docking with HM is only necessary if pre-positioning is used

	Moon			*Mars*			
Crew Operations Vehicle (COV)	**Short**	**Medium**	**Extended**	**Phobos**	**Short**	**Extended**	**Extended+**
Dock with HM	-	-	X	X	X	X	X
Communications Equipment	X	X	X	X	X	X	X
Attitude Control[F]	X	X	X	X	X	X	X
Aeroshield Attachment[G]	-	-	X	-	X	X	X
Ascend and Descend to Surface[H]	-	-	-	-	-	-	X
Life Support for Crew of 3	X	X	-	X	X	X	X
Deliver a Crew of 3 to LEO	X	X	X	X	X	X	X
Life Support for 2-3 weeks	X	X	X	X	X	X	X
Ballistic Earth re-entry	X	X	X	X	X	X	X
Aerocapture at Earth	-	-	X	X	X	X	X
Dock with Lander (manual)	X	X	X	-	-	-	-
Dock with Lander (autonomous)	-	X	X	-	-	-	-
Support one person in orbit	X	-	-	-	-	-	-
Sustain itself in unmanned orbit for extended periods	-	-	X	-	-	-	-

[F] Required for docking and rendezvous

[G] For long+ mission, heat shield is required for the COV to descend to the surface

[H] The COV would descend to the surface and provide habitat along with the SHM

Lander [I]	Moon			Mars			
	Short	Medium	Extended	Phobos	Short	Extended	Extended+
Dock with COV/HM in orbit	X	X	X	-	X	X	-
Dock with ISPP-SHM on surface	-	-	-	-	-	X	-
Ability to transfer crew of 6 from orbit to surface and surface to orbit	-	-	X	-	X	X	-
Ability to transfer crew of 3 from orbit to surface and surface to orbit	X	X	-	-	-	-	-
Support EVA	X	X	X	-	X	-	-
Life support for 3 crewmembers	X	X	-	-	-	-	-
Life support for 6 crewmembers	-	-	X	-	X	X	-
Life support for at least 2 days	X	X	X	-	-	-	-
Life support for at least 5 days	-	X	-	-	X	X	-
Life support for at least 2 weeks	-	X	-	-	-	-	-
Ability to land unmanned	-	-	X	-	-	-	-

[I] For Long+ mission, assume Direct architecture, lander not required

Modern Command Module (MCM)	Moon			Mars			
	Short	Medium	Extended	Phobos	Short	Extended	Extended+
Deliver crew of 3 to LEO	-	-	X	X	X	X	X
Earth EDL for a Crew of 3	-	-	X	X	X	X	X
Remain in LEO for Mars mission	-	-	-	X	X	X	X
Remain in LEO for Moon mission	-	-	X	-	-	-	-
Dock with COV/HM	-	-	X	X	X	X	X
Fits within fairing for man-rated launcher	-	-	X	X	X	X	X
Life support for 2 days	-	-	X	-	-	-	-
Life support for 1 week	-	-	-	X	X	X	X

Surface Habitation Module (SHM)	Moon			Mars			
	Short	Medium	Extended	Phobos	Short	Extended	Extended+
Human Life Support for 60 days	-	-	-	-	X	-	-
Human Life Support for 180 days	-	-	X	-	-	-	-
Human Life Support for 600 days	-	-	-	-	-	X	X
Countermeasure capability for reduced gravity environment	-	-	X	-	-	X	X
Long-duration hygiene & supply needs	-	-	X	-	-	X	X
Radiation protection	-	-	X	X	X	X	X
Medical Capabilities (mini-hospital)	-	-	X	-	-	-	-
Attach to Inflatable Habitation Module	-	-	-	-	X	X	X
Closed loop life support	-	-	X	-	-	-	X
Capability to work on far side of moon	-	X	X	-	-	-	-
Capability to work in sustained periods of darkness	-	-	X	-	-	-	-

Surface Mobility	Moon			Mars			
	Short	Medium	Extended	Phobos	Short	Extended	Extended+
Pressurized habitable rover	-	-	X	-	-	X	X
Open unpressurized rover	-	X	-	-	X	-	-
Support EVA	X	X	X	X	X	X	X
Reach important geological sites	-	X	X	-	X	X	X
Construction capability	-	-	X	-	-	-	X
Excavate soil for radiation shielding	-	-	X	-	-	-	X
Manipulator arm	-	-	-	-	-	X	X
Drill up to 10 m	-	-	-	-	-	-	X
200 kg Science Payload Carrying Capacity	-	-	-	-	X	-	-
1000 kg Science Payload Carrying Capacity	-	-	-	-	-	X	X
Mobility - Walking Distance from Base	-	X	-	-	-	-	-
Mobility - 10 km from base	-	-	-	-	X	-	-
Mobility - 200 km round trip	-	-	-	-	-	-	-
Mobility - 500 km round trip	-	-	X	-	-	X	X
Closed life support system (recover CO2 and water)	-	-	-	-	-	-	X

Habitation Module (HM)	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Support a Crew of 6	-	-	X	X	X	X	X
Human Life Support for 3 weeks	-	-	X	-	-	-	-
Human Life Support for 360 days[A]	-	-	-	-	-	X	X
Human Life Support for 600 days[B]	-	-	-	X	X	-	-
Aerocapture to Orbit	-	-	X	-	X	X	X
Dock with COV	-	-	X	X	X	X	-
Dock with SM1/SM2	-	-	-	X	X	X	-
Dock with MCM1/MCM2	-	-	X	X	X	X	-
Dock with Landers	-	-	X	X	X	X	-
Dock with ISPP-SHM on Surface[C]	-	-	-	-	-	-	X
Sustain itself in Unmanned Orbit for Extended Periods	-	-	X	X	X	X	X

[A, B] Duration of transit varies depending on the year of departure from Earth

[C] Similar to Mars Direct Architecture, COV goes direct to Mars surface, returns via ISPP

Crew Service Module (SM)	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Dock with COV/HM	-	-	-	X	X	X	-
Ability to be connected in stages	-	-	-	X	X	X	X
Insulation for zero cryogenic boiloff	-	-	-	X	X	X	X
Ability to be prepositioned in Mars orbit	-	-	-	X	X	X	X
Make fuel for return trip[D]	-	-	-	-	-	-	X
Ability to dock with Hab/COV/Lander[E]	-	-	-	X	X	X	X

[D] Direct architecture - H2 feedstock sent ahead of time, fuel made and stored for return trip, fuel connects to the HM/COV for the return to Earth

[E] Moon - Docking with HM is only necessary if pre-positioning is used

Crew Operations Vehicle (COV)	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Dock with HM	-	-	X	X	X	X	X
Communications Equipment	X	X	X	X	X	X	X
Attitude Control[F]	X	X	X	X	X	X	X
Aeroshield Attachment[G]	-	-	X	-	X	X	X
Ascend and Descend to Surface[H]	-	-	-	-	-	-	X
Life Support for Crew of 3	X	X	-	X	X	X	X
Deliver a Crew of 3 to LEO	X	X	X	X	X	X	X
Life Support for 2-3 weeks	X	X	X	X	X	X	X
Ballistic Earth re-entry	X	X	-	-	-	-	-
Aerocapture at Earth	-	-	X	X	X	X	X
Dock with Lander (manual)	X	X	X	-	-	-	-
Dock with Lander (autonomous)	-	X	X	-	-	-	-
Support one person in orbit	X	-	-	-	-	-	-
Sustain itself in unmanned orbit for extended periods	-	-	X	-	-	-	-

[F] Required for docking and rendezvous

[G] For long+ mission, heat shield is required for the COV to descend to the surface

[H] The COV would descend to the surface and provide habitat along with the SHM

Lander [I]	Moon Short	Moon Medium	Moon Long	Mars Phobos	Mars Short	Mars Extended	Mars Extended+
Dock with COV/HM in orbit	X	X	X	X	X	X	-
Dock with ISPP-SHM on surface	-	-	-	-	-	X	-
Ability to transfer crew of 6 from orbit to surface and surface to orbit	-	-	X	X	X	X	-
Ability to transfer crew of 3 from orbit to surface and surface to orbit	X	X	-	-	-	-	-
Support EVA	X	X	X	X	X	-	-
Life support for 3 crewmembers	X	X	-	-	-	-	-
Life support for 6 crewmembers	-	-	X	X	X	X	-
Life support for at least 2 days	X	X	X	-	-	-	-
Life support for at least 5 days	-	X	-	-	X	X	-
Life support for at least 1-2 weeks	-	X	-	X	-	-	-
Ability to Land Unmanned	-	-	X	-	-	-	-

[I] For Long+ mission, assume Direct architecture, lander not required

Modern Command Module (MCM)	Moon Short	Moon Medium	Moon Long	Mars Phobos	Mars Short	Mars Extended	Mars Extended+
Deliver crew of 3 to LEO	-	-	X	X	X	X	X
Earth EDL for a Crew of 3	-	-	X	X	X	X	X
Remain in LEO for Mars mission	-	-	-	X	X	X	X
Remain in LEO for Moon mission	-	-	X	-	-	-	-
Dock with COV/HM	-	-	X	X	X	X	X
Fits within fairing for man-rated launcher	-	-	X	X	X	X	X
Life support for 2 days	-	-	X	-	-	-	-
Life support for 1 week	-	-	-	X	X	X	X

Surface Habitation Module (SHM)	Moon Short	Moon Medium	Moon Long	Mars Phobos	Mars Short	Mars Extended	Mars Extended+
Human Life Support for 60 days	-	-	-	-	X	-	-
Human Life Support for 180 days	-	-	X	-	-	-	-
Human Life Support for 600 days	-	-	-	-	-	X	X
Countermeasure capability for reduced gravity environment	-	-	X	-	-	X	X
Long-duration hygiene & supply needs	-	-	X	-	-	X	X
Radiation protection	-	-	X	X	X	X	X
Medical Capabilities (mini-hospital)	-	-	X	-	-	-	-
Attach to Inflatable Habitation Module	-	-	-	-	X	X	X
Closed loop life support	-	-	X	-	-	-	X
Capability to work on far side of moon	-	X	X	-	-	-	-
Capability to work in sustained periods of darkness	-	-	X	-	-	-	-

Surface Mobility	Moon Short	Moon Medium	Moon Long	Mars Phobos	Mars Short	Mars Extended	Mars Extended+
Pressurized habitable rover	-	-	X	-	-	X	X
Open unpressurized rover	-	X	-	-	X	-	-
Support EVA	X	X	X	X	X	X	X
Reach important geological sites	-	X	X	-	X	X	X
Construction capability	-	-	X	-	-	-	X
Excavate soil for radiation shielding	-	-	X	-	-	-	X
Manipulator arm	-	-	-	-	-	X	X
Drill up to 10 m	-	-	-	-	-	-	X
200 kg Science Payload Carrying Capacity	-	-	-	-	X	-	-
1000 kg Science Payload Carrying Capacity	-	-	-	-	-	X	X
Mobility - Walking Distance from Base	-	X	-	-	-	-	-
Mobility - 10 km from base	-	-	-	-	X	-	-
Mobility - 200 km round trip	-	-	-	-	-	-	-
Mobility - 500 km round trip	-	-	X	-	-	X	X
Closed life support system (recover CO2 and water)	-	-	-	-	-	-	X

Cargo Module (CM)	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Dock with SM using Electric Propulsion	-	-	-	-	X	X	X
Contain Lander	-	X	X	X	X	X	X
Contain Rover	-	X	X	-	-	X	X
Contain Surface Habitation Module	-	-	X	-	-	X	X
Contain science payload, ISPP equipment, food	-	-	X	-	X	X	X

Cargo Service Module (CSM)	Moon			Mars			
	Short	Medium	Long	Phobos	Short	Extended	Extended+
Electric Propulsion[J]	-	X	X	-	X	X	X

[J] Only necessary if pre-positioning is used

9.2.2 Habitation Module

9.2.2.1 Crew Size and Composition

Mission duration is critical when determining the size of the crew compartment. As well as the size of the vehicle, the number of crew is an important decision based on group dynamics, science/research requirements and vehicle design limitations. Human factor requirements such as habitable volume, crew health & safety, food & waste management, and thermal & power requirements dictate much of the vehicle design and the structural aspects of the vehicle are determined by these requirements.

A long duration mission, such as one to Mars and back poses many new challenges that have not been the focus of earlier human exploration initiatives like Apollo. The size and composition of the crew is an extremely important factor based on psychological and sociological aspects of such a mission. Important factors to consider are summarized from an earlier MIT study in *16.851 Satellite Engineering* (2003). Large crews tend to have lower levels of deviance and conflict and this tends to decline with increasing mission duration. Also, heterogeneous crews have lower rates of deviance and conflict (Dudley-Rowley, 2002). This same investigation indicated that the least dysfunction of any crew studies was a crew of nine people. Since a short duration mission will have the crew anticipating their return in the short term, long-term group dynamics are less of an issue.

Gender, ethnic and cultural make-up is an important factor for long mission durations. More heterogeneous crews begin a mission with some deviance, conflict and dysfunction, but this tends to decrease as the mission progresses. The opposite is observed for a homogeneous crew, whereby deviance, conflict and dysfunction tend to be initially less than a heterogeneous crew, but increase as the mission progresses (Dudley-Rowley, 2002).

As was discussed in an earlier section, the interior “free” space for a crew is important and should be sufficiently large for long-term mission durations. This results in increased performance and mental health.

9.2.2.2. Life Support Systems

Equipment necessary to keep the crew alive must be extremely reliable and robust to external perturbation. This equipment involves a galley & food system, waste collection system, personal hygiene, clothing, recreational equipment, housekeeping, operational supplies, maintenance, sleep provisions, and health care.

The pressurized spacecraft must have the appropriate temperature, pressure, and atmosphere, as well as control over disturbances from living organisms. According to Larson (1999), open loop life support systems (carry food, water and oxygen on board) are reliable, but are limited by mission duration (cost and volume occupation). An open loop system processes waste products and recovers useful resources. However, the disadvantage are; development costs, power requirements, reliability and maintainability (Larson, 1999).

The functions of a life support system involve managing atmosphere (pressure, temperature, humidity, removal of contaminants, composition and ventilation), water (provide, monitor, process and store for hygiene and drinking), waste (collect, process, store) and food (provide, store and prepare) (Larson, 1999).

9.2.2.2.1. ECLSS Atmosphere Management

Deciding which technology to select depends largely on the characteristics of the crew size, mission duration and mission location. For long-duration missions, hygiene water will likely dominate design decisions on sizing the ECLSS (Larson, 1999). An atmosphere management system suggested by HSMAD consists of **4BMS** (4-bed molecular sieve), **TCCS**, and **Sabatier P/C** atmosphere management system (see Table 28).

Table 28: ECLSS atmosphere management

	4BMS	*TCCS*	*Sabatier*
Number of Crew	3	4	3
Mass (kg)	90	80	114
Volume (m^3)	0.45	0.60	0.21
TRL	6	5	7

A flow chart of the processes used to manage the atmosphere is shown in Figure 86.

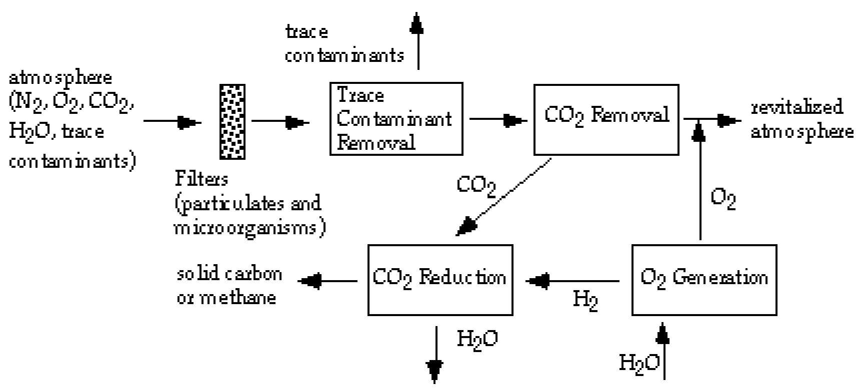


Figure 86: Atmospheric control and supply (Wieland, 1999)

A triple redundant system was selected, which consisted of the three systems listed earlier. The resultant mass was multiplied by a factor of three for redundancy (Wieland, 1999). This results in an overall mass and volume per crewmember (CM) of 255 kg/CM and 1 m^3, respectively.

9.2.2.2.2. ECLSS Water Management

A P/C water management system of vapor compression distillation (VCD) was presented in the Mars mission design example in HSMAD. A flow chart of the water recovery and management is shown in Figure 87.

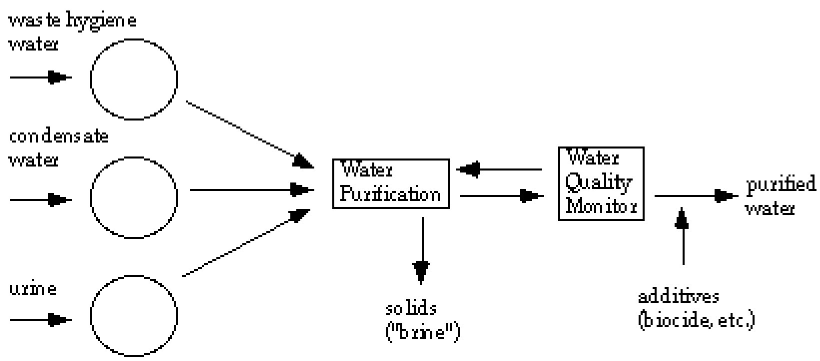


Figure 87: Water recovery and management (Wieland, 1999)

Two water management systems were selected, bringing the total mass and volume to 50 kg/CM and a volume of $0.2m^3$/CM (Wieland, 1999). The ECLSS atmosphere and water management systems were predicted for various crew sizes. The method of scaling was similar to the method described earlier in the Report, in which the mass of the ECLSS was scaled as being directly proportional to the number of crew (see Figure 88).

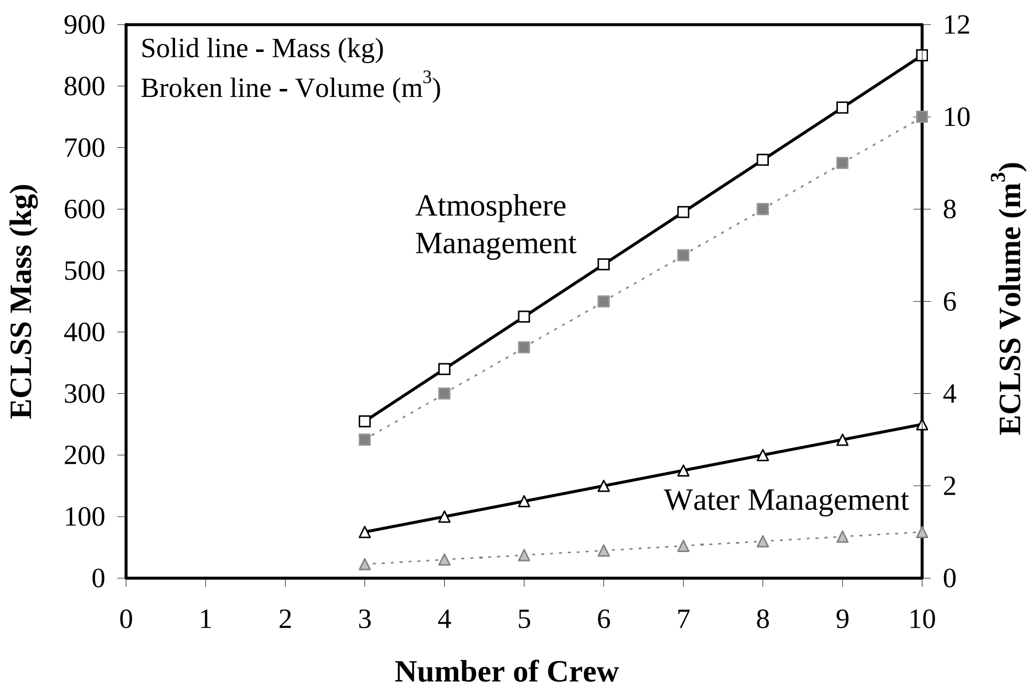


Figure 88: Mass and volume of ECLSS atmosphere and water management systems

9.2.2.3. Radiation Protection

For missions more distant than geosynchronous Earth orbit, the Earth's magnetic field does not provide protection and radiation from the Sun, especially during solar storms (during a Mars mission), and galactic cosmic radiation (Wieland, 1999). Since a crew will leave the Earth's atmosphere on a mission from Earth to Mars, radiation protection from high-energy sun particles is required. Background space radiation, such as galactic cosmic rays (GCR) may also influence crew health during the mission to Mars.

The radiation dose limit is used to predict the hull thickness. Typically, spacecraft radiation wall thickness is determined by the wall thickness that does not permit the radiation dose limit. As such, a radiation dose of 1 Gy requires an aluminum hull thickness of 1.5cm (Larson, 1999).

Distinctions were presented between the design of the crew habitat for the interplanetary space travel and ones for the Mars surface (Cohen, 1996). This is known as the "Being There Versus Getting There" philosophy that argues that interplanetary and surface capabilities are fundamentally so different that it is not possible to optimize them within a single set of habitation elements. Cohen indicates that "radiation shielding is the most overlooked feature of proposed interplanetary vehicles" and that "NASA and space industry mission planners consistently underestimate the radiation hazards on a trip to Mars, particularly from GCRs and thus minimize the shielding to protect against this exposure." Cohen indicates that the shielding requirements form radiation hazards in interplanetary space indicates the need for substantial omni directional shielding on the order of 30g/cm^2.

It may be possible to extract shielding for the Mars surface habitat from the Martian surface, which indicates that the mass penalty of carrying sufficient shielding for the Martian surface habitat from Earth is unnecessary (Cohen, 1996). By having a small "solar storm shelter" in the crew transport vehicle, the overall radiation protection mass would be reduced, however this comes at the expense of close crew quarters, which as discussed earlier, affects group dynamics. From multiple standpoints, providing radiation protection for the entire crew vehicle is critical to the success of the mission.

Since a sphere has a minimum ratio of surface area to volume for any solid, as closely approximating this shape reduces the overall radiation shielding mass. According to Cohen, the shielding can either be solid, as in the form of solid aluminum gore panel or liquid, as in water to pump into interior perimeter tanks. Cohen also indicates that "whatever the shielding, it makes no sense to waste the effort, cost and energy that put it in LEO by landing it on Mars as part of a multi-purpose habitat."

In ISS orbit, roughly half the radiation dosage comes from trapped protons and half from Galactic Cosmic Rays (GCR). As well, the flux of low-energy GCRs is *inversely proportional* to solar activity and Solar Particle Events (SPEs) are mostly low-energy protons (30-120 MeV) and are more common at solar maximum. Since SPEs vary so much in size, it's hard to design a spacecraft that will be totally immune to their effects (Larson, 1999).

9.2.2.4. Power Systems

This information was summarized from Larson (1999).

A spacecraft needs continuous housekeeping power to operate and support guidance, navigation, and attitude control, telemetry downlink, active thermal control, the computer system, and crew needs – circulating atmosphere, lighting, cabin heat, etc. Consider the following housekeeping power requirements:

- Apollo Command Module – less than 2 kW
- Skylab space station – about 4 kW
- International Space Station – about 36 kW
- For a Mars mission, the baseline (or continuous) power needed is estimated 30 kW-50kW for transfer vehicles.

9.2.2.4.1. Earth and lunar orbit constraints

The power system must withstand thermal cycling because of the short 35-minute orbits. Solar power systems should be able to store energy and account for radiation in low-Earth orbits.

9.2.2.4.2. Solar Photovoltaics (PV)

PV arrays consist of solar cells with transparent covers to protect the cells from radiation. Arrays may mount directly on the spacecraft's body without deployment or pointing. This option limits the array area and may compromise thermal control. The power produced from a given area of solar array from the power-conversion efficiency is:

$$P = \Phi_{Sun} \eta F_p A \cos(\theta)$$

where, $\Phi_{Sun} = [1368/d^2]$ is the flux of sunlight at a distance d (in astronomical units) from the Sun, η the conversion efficiency that is dependent on the solar cell type chosen (for silicon cells used on International Space Station about $\eta=.145$), F_p the fraction of the array actually covered by solar cells (85-90%), and θ the angle of the array normal to the Sun.

The solar array *blanket mass* is the mass of the solar cells, including cover glass and interconnects, and the substrate on which the cells are mounted but does not include the structural mass, array orientation motors, or the deployment and packaging mass. Because of the difference between the "standard" efficiency and actual delivered to the user, the load power output is about 33% of the array's output in nominal sunlit conditions. Typical blanket mass per unit area is 1.7 kg/m^2 for silicon. The blanket mass is typically about 55% of the array's total mass m_{array}. The drive mechanism's estimated mass m_{drive} is a function of the array's mass m_{array} (in kilograms) from

$m_{drive} = |-.014m_{array} + 20.6|*m_{array}/100.$

The *launch volume* depends on the technology chosen and the packaging. The rough estimate for the total packaged volume for a solar array is *$0.05m^3$ per m^2 of the array's total area.*

9.2.2.4.3. Solar Dynamic and Nuclear Systems

Solar thermal systems (known as solar dynamic, or SD) are predicted to have lower specific mass and cost for high power (>100 kW_e) applications. Nuclear systems may be preferable for long eclipse, high-power missions such as large lunar base; and missions requiring low, long-term continuous power such as missions with low sunlight intensity. All thermal-conversion (SD and nuclear) require radiators to reject the waste heat, which account for much of the system's mass and area.

The baseline SP-100 reactor can produce 100 kW_e of electrical power with a specific power of 30 W_e/kg and a 7 year life-span. The shielding requirements to protect the crew and payload from the radiation produced by a nuclear power system can be a large fraction of the power system's mass. A reactor system is typically not radioactive until energized. The reactor should be launched from Earth un-powered to allow handling on the pad.

9.2.2.4.4. Energy Storage

One primary option is the hydrogen-oxygen fuel cell, which reacts hydrogen with oxygen to produce electricity and water. Fuel cells produce high energy per unit reaction mass, but are less compact and more complex (requiring storage of hydrogen and oxygen) than other batteries.

The cell's mass is proportional to the power level required, and the mass of the reactants is directly proportional to the power required times the mission length. For example, the Shuttle's system has three primary fuel cells that provide a total of 14 kW_e. Each fuel cell has a mass of 91 kg. Together, the three cells use about 150 kg of hydrogen and oxygen per day.

9.2.2.5. Thermal Control

This information was summarized from Larson (1999).

The thermal control system must maintain comfortable and a uniform temperature distribution for the crew, other systems and equipment. As well, the thermal protection system provides the first line of defense by shielding from extreme heat sources and sinks (sun and deep space, respectively)

Active thermal control implies movement of mass, information or energy. This is usually some type of loop to circulate fluids. This allows convective heat transfer to augment conduction and radiation. Passive thermal control involves conduction through the spacecraft and radiation from its surface to dissipate heat and keep temperatures relatively low. Geometric design and layout, insulation, heaters, heat pipes, and louvers are common passive techniques that aid in maintaining all parts of the spacecraft at acceptable temperatures.

9.2.2.5.1. Design of Thermal Control System

Depending on the mission, the mission phases each contribute unique thermal requirements. After considering a particular mission, examine the phases (*Table 16-2,* Larson, 1999) and complete the following design iteration (*Table 16-1,* Larson, 1999).

1. Determine mission requirements - sum heat loads of electrical equipment (typically half of total heat load for transfer vehicles. Assumed to be 100% of electrical energy becomes thermal energy), metabolic (human production), walls (gains or leaks), other.
2. Establish temperature requirements for crew and cargo
3. Select the thermal protection system (determine materials)
4. Decide which systems can have passive control
5. Size heat transport loops and heat sinks with an appropriate fluid maximum temperature
6. Determine the architecture to connect all actively cooled heat sources to the heat sinks.
7. Calculate size and capacity of other components
8. Estimate mass, power, volume
9. Iterate

Various mission phases are described in HSMAD (*Table 16-2, HSMAD*).

1. **Pre-launch** – Most vehicle systems require cooling
2. **LEO** – Avoid sun and Earth view to radiators. May include docking with another craft.
3. **Earth entry** – Significant heat gains through walls
4. **Lunar Transit** – Environment may get cold
5. **Lunar Surface** – Hot during day, cold at night (surface dust)
6. **Mars Transit –** Environment may be cold (Microgravity)
7. **Mars Surface** – Surface dust, winds, pressure limited expendable heat sinks.

Typical heat rejection values of space radiators are given in Table 16-4 (Larson, 1999). Note that some of the rejection systems are not suitable for long-term missions because significant degradation will take place over time.

Thermal control fundamentals include convection, conduction and radiation. These three heat transfer processes are used in some form to control the temperature of the spacecraft interior. Table 16-7 (Larson, 1999) discusses the masses, power and volume of the components of a thermal control system.

The Moon temperatures range from 100K to 400K because there is no moderating atmosphere. The radiators on the Moon can be as low as 3K (facing deep space) and as high as 325K (vertical at the equator at noon). The interplanetary thermal environment is generally cold, dominated by solar radiation and deep space. In this environment, simple radiators, facing away from the Sun, are very effective.

Radiant heat exchange dominates the Martian thermal environment, but a thin CO_2 environment helps transfer heat through conduction and convection. High velocity winds stir up huge dust storms that block radiation. Surface temperature of Mars can range from 130K to 300K.

9.2.2.5.2. Entry Heating

Radiative and absorptive systems are the two basic external types used to dissipate entry heating. Radiative typically dissipates 80-90% of the heating. Absorptive systems absorb the heat through heat sinks, ablation, or transpiration. *Ablation* is a self-regulating transfer of heat and mass in which material absorbs entry heating then degrades. *Transpiration* means injecting a fluid through the skin of the vehicle into the boundary layer to provide cooling.

Radiative thermal protection systems are limited to about 1370ºC. This is a passive system and usually does not involve mass loss or shape change. Absorptive systems absorb heat through a phase change, chemical change, temperature rise, or convective or transpiration cooling. These systems are typically more complex and weigh more than radiative systems. If the material burns up, we can use the system only once. However, absorptive systems can handle the high heating rates from velocities needed for planetary travel or missile entry.

To solve the surface temperature of the thermal protection concept for a range of entry velocities between 6 and 14 km/s,

$$T_w = \left(\frac{q_s}{\sigma \varepsilon} \right)^{1/4} . \quad (1)$$

T_w = wall temperature (K),
q_s = surface heat flux (W/m^2)
σ = Stefan-Boltzmann constant = 5.67×10^{-8} W/m^{2}K^4, and
ε = surface emissivity at wavelength mix corresponding to temperature, T_w.

Re-entry heat gain can be obtained by assuming a re-entry surface temperature and an insulation conductance,

$$q_{re-entry} = kA\Delta T . \quad (2)$$

q = re-entry heat gain (kW/m^2),
kA = insulation conductance (W/m^{2}K), and

ΔT = temperature difference between interior and exterior surfaces (K).

9.2.2.5.3. Heat Sinks for Short Duration Missions

For short-duration missions, heat sinks can be used as an active method of thermal control. Since fluids like water or ammonia absorb a tremendous amount of heat during phase change, and the space environment is conducive too this (near vacuum), the approximate mass of expendable heat sink required for this is,

$$M_e = \frac{QD}{h_{fg}}. \tag{3}$$

M_e = heat sink mass (kg),
Q = required heat rejection rate (kW)
D = mission duration (s), and
h_{fg} = latent heat of vaporization (kJ/kg).

Table 16-3 (Larson, 1999) should be consulted for general guidelines for using expendable heat sinks instead of radiators, and Table 16-10 (Larson, 1999) for the latent heats of vaporization for commonly used fluids.

Thermal capacitors are short duration heat sinks. Using the same principle of evaporative cooling, elements can be "cold-soaked", which when heated, will cool the element. This technique incurs only a very small mass penalty (only the weight of the fluid soaking the element), however a complex analysis is required to model this transient heat transfer.

9.2.2.5.4. Internal Thermal Control System

This thermal control system gathers all of the heat loads from within the pressurized volume (cabin or module), including those collected directly from equipment through heat exchangers and cold plates.

The internal thermal control system must cool the cabin air heat exchanger to control humidity and temperature (see ECLSS Report). A coolant capable of removing the heat from the cabin must be selected. An important consideration is the crew's safety should the coolant leak. Water is often selected, however it is critical that temperatures do not fall below freezing. It may also be necessary to insulate plumbing, cold plates, and other surfaces that operate below the air's dew point temperature.

9.2.2.5.5. Thermal Control Components

1. Electric heaters are used in cold-biased equipment, controlled by thermostats.

2. Space radiators are heat exchangers on outer surface that radiates heat into space.
3. Cold plates are structural mounts using convective transfer for electrical equipment.
4. Doublers are passive aluminum plates that increase the heat exchange surface area.
5. Phase change devices are used to generate heat in short bursts.
6. Louvers shield radiator surfaces to moderate heat flow to space.

Additional considerations for human space flight can be found in Larson (1999).

9.2.2.6. Attitude Determination and Control System (ADCS)

9.2.2.6.1. Introduction

ADCS stabilizes the vehicle and orients it in desired directions during the mission despite the disturbance torques acting on it. ADCS determines the vehicle's attitude using sensors and controls it using actuators. ADCS is a major spacecraft system, and its requirements often drive the overall S/C design. Components are cumbersome, massive and power consuming. Table 29 shows a summary of the process to design ADCS system on a spacecraft. This table was inspired from Larson (1999).

Table 29: Design process of ADCS

STEP	INPUTS	OUTPUTS
1. Define control modes and derive the corresponding requirements	Mission profile Type of insertion for launch vehicle	List of different control modes during mission Requirements and constraints
2. Select type of spacecraft control by attitude-control mode	Orbit, pointing direction Disturbance environment	How to stabilize and control: - three-axis - spinning - gravity gradient
3. Quantify disturbance environment	Spacecraft geometry Orbit Solar/magnetic models Mission profile	Values for forces from - gravity gradient - magnetic, aerodynamic and solar pressure - internal disturbances Effects of powered flight on control (center of mass, cm offsets, slosh)
4. Select and size hardware for the ADCS	Spacecraft geometry, pointing accuracy and direction, orbit conditions, mission requirements, life time, slew rates	- Sensor suite: Earth, sun, inertial or other sensing devices - Control actuators (reaction wheels, thrusters or magnetic torquers) - Data-processing electronics
5. Define algorithms for determination and control		

Note: In Table 29, steps 1 and 3 can supposedly be found in the literature for each type of mission. But for each chunk to be designed, work has to be done on steps 2 and 4 (shown in gray). Especially the output of step 4 determines the hardware and corresponding masses that should be taken onboard the spacecraft.

9.2.2.6.2. Control modes and methods

Table removed due to copyright restrictions.
Eckart, P. "The Lunar Base Handbook: An Introduction to Lunar Base Design, Development, and Operations." *Space Technology Series*. Edited by W.J. Larson. New York, NY: McGraw-Hill, 1999.

Figure 89: Attitude control modes, from Larson (1999)

Figure 89 shows typical attitude control mode in which spacecraft have to be in. The method to control these different modes depends on 1/ the mode and 2/ the type and amount of disturbance.

Disturbance torques can be cyclic (which average to zero over an orbit) or secular (which do not average to zero over an orbit). These disturbances can be controlled passively, i.e. without moving parts (by taking advantage of vehicle's inertia or favorable disturbances), or actively. For active control, the S/C senses the attitude motion and applies control torques to counter it.

9.2.2.6.3. Hardware for ADCS

The hardware to account for ADCS includes actuators, sensors and computers (+electronic wiring). Table 30 gives a description of control methods and the hardware associated. It was inspired by de Weck (2001) and Larson (1999).

Table 30: Description of actuators, inspired by de Weck (2001) and Larson (1999)

	Type	Pointing options	Attitude maneuverability	Lifetime limits	Additional hardware required
Passive attitude control	Gravity gradient	Earth's local vertical only	Very limited	None	Libration damper: eddy current, hysteresis rods No torquers
	Passive magnetic	North/south only	Very limited	None	?
	Pure spin stabilization	Inertially fixed any direction Re-point with precession maneuvers	High propellant usage to move stiff momentum vector	Thruster propellant	Nutation damper Torquers to control precession (spin axis drift) magnetically or with jets
	Dual spin stabilization	Limited only by articulation on de-spun platform	Momentum vector same as above De-spun platform constrained by its own geometry	Thruster propellant De-spin bearings	
Active attitude control	Reaction wheels RW (most common actuator)	No attitude constraint	Rates limited by available momentum and low torques	Propellant (if applies) Bearing life, motors	External torque required for momentum dumping
	Control moment gyros	No attitude constraints	Rates limited by available momentum Double gimbal CMG has limited torques	Propellant (if applies) Bearing life, motors	
	Magnetic torquers (to de-saturate RW)	Depends on the Earth magnetic field	Harmful influence on star trackers		
	Thrusters (to de-saturate RW)	No attitude constraints		Propellant	

9.2.2.6.4. Example masses for different missions

9.2.2.6.4.1. Crew vehicles ADCS

Table 31 shows the masses of reaction control system for some crew vehicle, these were obtained from mass breakdown from Braeunig (2001) and other NASA document for the XTV.

Table 31: ADCS masses for some crew vehicles

	Mercury	Gemini	Apollo	XTV (avionics…)
RCS mass	40	133	400	200
Total mass	1118	1982	5806	5760
Percentage	3.6 %	6.7 %	6.9 %	3.4 %

9.2.2.6.4.2. Communication Satellite ADCS

For a LEO communication satellite, according to Springmann (2003), typically the ACDS mass is 7% of the dry mass of the satellite, as shown on
Table 32.

Table 32: ADCS mass of communications satellite, from Springmann (2003)

Subsystem	% M_{dry} (Std. Dev.)
Payload	27 (4)
Attitude Control	7 (2)
Electrical Power	32 (5)
Propulsion	4 (1)
Structure	21 (3)
Thermal	4 (2)
Tracking and Command	5 (2)
TOTAL	100

9.2.2.6.4.3. Apollo Lander ADCS

According to Gavin (2003),
Table 33 shows the masses for the LM configurations (in kg):

Table 33: Apollo lander ADCS

	Ascent stage	Descent stage	Total
Reaction control system	110	0	110
Dry mass	2154	2783	4937
Percentage of dry mass	5.1 %	0 %	2.2 %
Propellant for RCS	275	0	275

Image removed due to copyright restrictions.

Figure 90: Apollo lander mass breakdown, from Gavin (2003)

9.2.2.6.5. Conclusions

As a first approximation, the percentage of the ADCS of the total dry mass is 3 to 7 %.

9.2.2.7. Entry, Descent and Landing (EDL)

9.2.2.7.1. Deceleration

Analytical solutions to the equations of motion provide estimates for preliminary mission and vehicle design for atmospheric entry. The peak aerodynamic loads and heat rate can be used to estimate crew's acceleration exposure and the required thermal protection system. For example, consider the equations of motion for ballistic re-entry. The peak aerodynamic load (G_{max}), g_e's, at v is 0.607(v_e) is:

$$G_{\max} = \frac{v_e^2 \sin(\gamma_e)}{2 e g_e H_s} \tag{3}$$

where v_e is the atmospheric entry speed (km/s), γ_e is the flight-path angle (radians), e is 2.71828, g_e is the gravitational constant (9.81 m/s^2), and H_s is the planet atmosphere's density scale height (km). This quantity can be determined from a table of atmospheric density (ρ) and altitude (H) by forming an exponential fit between the altitudes of interest: $H_s = -(H_2 - H_1) / \ln(\rho_2/\rho_1)$.

9.2.2.7.2. Heating

The keystone to a conventional EDL system is its heat shield. The heat shield is usually built from a structure of aluminum honeycomb and CFRP skins covered with an ablative material that absorbs the heat of the entry and keeps the payload at an appropriate temperature. The ablator most used on the Shuttle External Tank, Mars Pathfinder, and the Mars Exploration Rovers programs was SLA561V (Allouis, 2003). This material, based on a mixture of cork wood, binder and tiny silica glass spheres, has a density, ρ_m, of 264 kg/m^3 and an effective heat of ablation, Q_m, of 5.41×10^7 J/kg.

The maximum heating rate for an entry profile is evaluated at the stagnation point from a Sutton-Grave correlation:

$$\dot{q}_s = k\sqrt{\frac{\rho_\infty}{r_n}}(v_e)^{3.233} \tag{4}$$

where r_n is the nose radius (m), v_e is the entry velocity (m/s), ρ_∞ is the atmospheric density, and k is 2.84×10^{-5}. From Equation 2, it is apparent that the bigger the nose radius, the lower the heat rates. The maximum heat rate for a lifting body is given by:

$$q_{\max} = 1.06 \times 10^{-4}\sqrt{\frac{\beta g_e}{(L/D)r_n}}v_e^2 \tag{5}$$

where β is the ballistic coefficient ($m / C_D S$), kg/m^2, where m is the mass of the vehicle, C_D is the hypersonic drag coefficient, and (L / D) is the hypersonic lift-to-drag ratio. Notice that higher ballistic coefficient values result in higher heat and deceleration loads.

The total heat load, Q, of the mission is derived from the integration of the heat rates (Equation 3) over the heat peak during entry. The ablator on the heat shield must be thick enough to keep the back face of the heat shield below a threshold temperature even after the ablation process that takes place during entry. The radiative equilibrium temperature is given by $q_s = \varepsilon\sigma(T_s)^4$ where ε is the black-body emissivity and σ is the Stefan-Boltzmann constant (5.67×10^{-8} W/m^{2}K^4). The minimum thickness of ablator can be estimated from the heat transfer formula at constant wall temperature:

$$\frac{T(x,t) - T_s}{T_i - T_s} = erf\left(\frac{\delta}{2\sqrt{\alpha t}}\right) \tag{6}$$

where α the diffusivity of the ablator is defined by the thermal conductivity (n), the density (ρ), and the specific heat (C_p):

$$\alpha = \frac{n}{\rho C_p} \tag{7}$$

The thickness ablated during entry is given by:

$$\Delta\delta = \frac{q}{\rho_m Q_m} \quad \textbf{(8)}$$

where q is the heat per unit area on the heat shield, ρ_m is the density of the material, and Q_m the effective heat of ablation of the material. The minimum thickness for the heat shield is therefore $\delta+\Delta\delta$, but a safety coefficient is of 1.5 is usually applied to δ.

9.2.2.7.3. Accuracy

The Earth's thick atmosphere allows a spacecraft to perform an un-powered descent and landing after an aerodynamically controlled entry. The design of parachute and parafoil systems is typically an iterative process. Still, we can use the main design driver – the terminal velocity of the probe – to determine some estimates. Assuming the parachute has a circular cross-section, the parachute's diameter for a given Lander mass (m_L) can be calculated from:

$$Drag = \rho\, v^2 / 2 \times C_D \times A = m_L g \quad \textbf{(9)}$$

where ρ is the atmospheric density at parachute release, v is the desired terminal velocity, C_D is the drag coefficient (parachute and vehicle), m_L is the mass of Lander at engine ignition, g is the planet's acceleration due to gravity. Both the Apollo and Russian Soyuz capsules used parachutes to get terminal velocities of about 9 m/s and 7 m/s, respectively.

The X-38 lifting body used a steerable parafoil that slowed vertical landing speeds (about 5 m/s), increased cross-range capability, and allowed for a gliding touchdown on land. Parafoils allow for improved hypersonic range and cross-range capability, as well as decreased entry acceleration and thermal loads without the complexity of horizontal landings. An Apollo-class vehicle, like the Modern Command Module (MCM), designed with this technology would no longer require a water landing and recovery. Furthermore, by using a flare of the parafoil system, we can reduce touchdown loads without the use of decelerating retrorockets. The following equation estimates a vehicle's straight-line range while on a parafoil, assuming the flight path has a small radial acceleration and rate of change:

$$R = R_0 \times \frac{L}{D} \times \ln\left(\frac{1 + H_i / R_0}{1 + H_f / R_0}\right) \approx (H_i - H_f)\frac{L}{D}$$

where R_0 is the planet's radius near the landing zone, (L/D) is the parafoil's lift-to-drag ratio, and H_i and H_f are the vehicle's initial and final altitude above the ground.

As suggested by the equations of motion, a spacecraft must be aerodynamic to perform EDL. The external shape is critical for interactions with the significant atmospheres of Earth and Mars. The lift-to-drag ratio (L/D) is a convenient way to express a vehicle's ability to maneuver in the atmosphere. In addition, the vehicle's mass distribution and location of the vehicle's center-of-mass is fundamental to its controllability.

9.3 Parameters for Calculating Lunar Mission Mass in LEO

ΔV Table for Lunar Missions Using Lunar Orbit	
Mission Segment	*ΔV (m/s)*
To Moon Transit	3100
To Moon Orbit (100km)	800
To Moon Surface	1870
To Moon Orbit (100km)	1870
To Earth Transit	800

Table 34: ΔV table for lunar missions using lunar orbit

ΔV Table for Lunar Missions Using EM-L1	
Mission Segment	*ΔV (m/s)*
Transit to EM-L1	3100
To EM-L1 Orbit	600
To Moon Transit	150
To Moon Orbit (100km)	600
To Moon Surface	1870
To Moon Orbit (100km)	1870
Transit to EM-L1	600
To EM-L1 Orbit	150
Transit to Earth	600

Table 35: ΔV table for lunar missions using EM-L1

Payload Masses	
Module	*Dry Mass (kg)*
Crew Operations Vehicle (COV)	5700
Modern Command Module (MCM)	5200
Lunar Lander (LL)	10000
One Octahedron of Habitation Module (HM)	9167
Surface Habitation Module (SHM)	38150

Table 36: Lunar payload masses

Other Parameters	
Propulsion	*I_{sp} (s)*
Cryogenic Chemical Propulsion	425
Electric Propulsion	3200
Other	
Structural Factor	0.1
Boil-off	0

Table 37: Other lunar mission parameters

9.4 Mars Initial Mass in LEO Calculations

In order to compare different architectures and different trajectories, to select a baseline mission design, estimates of the initial mass in LEO were presented. In this section, a verification of the payload masses is given, followed by the relevant equations and a detailed calculation. All assumptions are restated for convenient reference.

9.4.1 Verification of initial mass in LEO estimates

A manned mission to Mars will require a significant mass to be launched from Earth in order to provide the required delta V (ΔV) capacity and life support necessary for the transit and surface stay. The mass estimates derived in this report are compared to those a paper by Walberg (1993).

Walberg's paper reviews four mission classes (opposition, conjunction, conjunction with a fast transfer, and split-sprint mission) with three different scenarios (all propulsive, aerobraking, and nuclear propulsion) and gives IMLEO estimates for each. After describing each mission class and scenario, I will compare the mass estimates for validation against the reference values for each mission scenario.

9.4.1.2 Description of Mission Classes

The following is a short description of each mission class, as presented in Walberg (1993).

9.4.1.2.1 Short-Stay Opposition

The first mission type investigated is an opposition class mission, with a Venus swing-by, as displayed in Figure 91. An opposition class mission is one where the alignment for one leg of the transfer is not optimal, but allows for relatively short planetary stays, between 30 and 60 days. Performing a swing-by at Venus allows for a major reduction in the required ΔV for a relatively small increase in time of flight. Comparing the ΔV's given for missions not employing a Venus swing-by Larson (1999), to those using a swing-by Walberg (1993), and assuming that a direct entry is performed at Earth for each, the average ΔV savings is on the order of 8.3 km/s, reducing the average mission ΔV from between 16-23 km/s to between 8-12 km/s. The total increase in flight time required to perform a Venus swing-by is between 50 to 100 days (depending on a given year). Other considerations, such as a close pass the sun must be taken into account when selecting a mission design, but for the purposes of analyzing initial mass, these considerations are neglected.

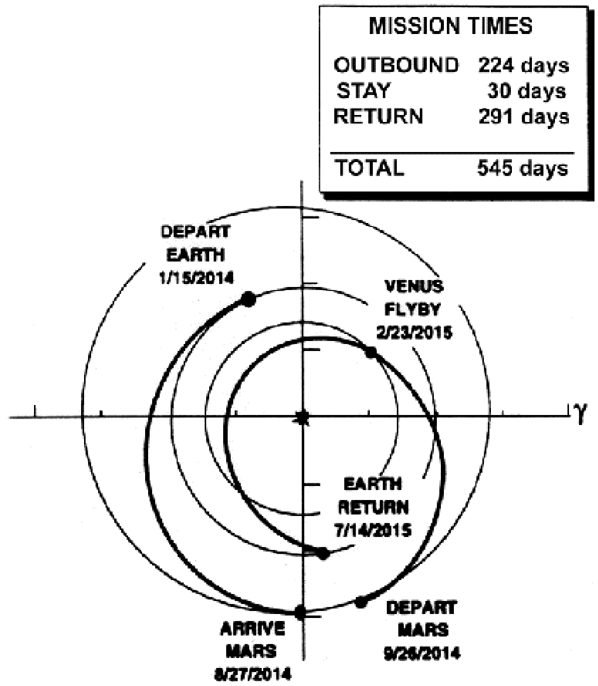


Figure 91: Diagram of opposition class mission with a Venus fly-by (NASA DRM website)

9.4.1.2.2 Long-Stay Conjunction

The second type of mission is a conjunction class mission where a trade in surface time on Mars is made in favor of a more optimum flight trajectory, as displayed in Figure 92. Thus, this class of mission requires stays on Mars on the order of 300 to 500 days. However the ΔV required is significantly reduced (between 5.2 km/s and 6.9 km/s), assuming a direct Earth entry. In addition, the decrease of ΔV required to enter both an Earth orbit and a Mars orbit are reduced which better facilitates aerobraking.

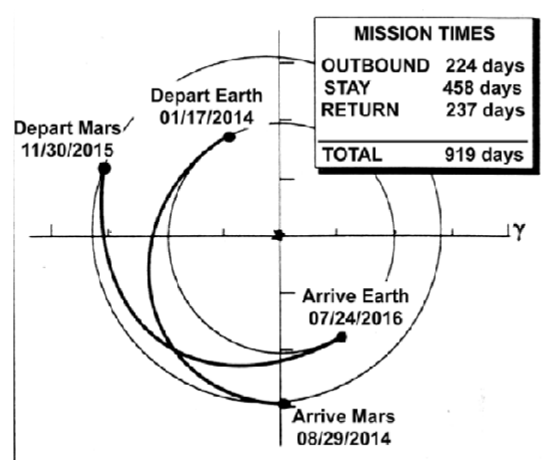


Figure 92: Diagram of conjunction class mission (NASA DRM website)

9.4.1.2.3 Fast-Transfer Conjunction

The third type of mission is a fast-transfer conjunction-class mission, as displayed in Figure 93. This type of mission increases the required ΔV to between 8 and 10 km/s, assuming direct entry at Earth, but decreases the transit times to recorded zero-g levels (between 100 to 200 days per direction). The stay times at Mars are increased slightly (approximately 50 days).

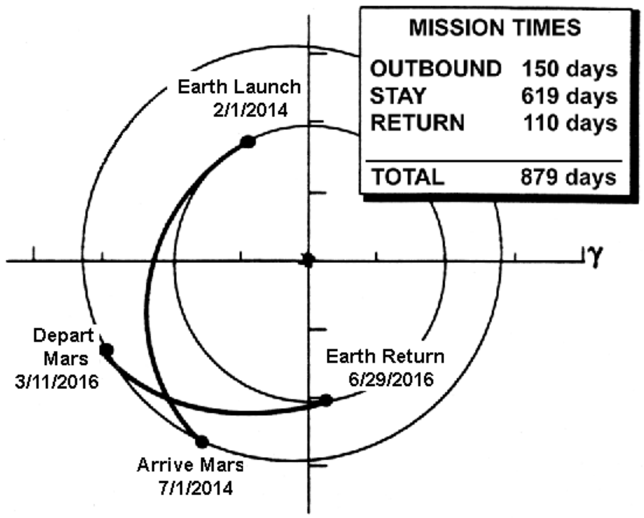


Figure 93: Diagram of fast-transfer conjunction class mission (NASA DRM website)

9.4.1.2.4 Split-Sprint Mission

The fourth type of mission is a split-sprint mission. The basic idea is to pre-position the cargo, including return supplies and return propellant using a fuel-efficient conjunction class mission. The crew is then transported via an opposition-class (outbound) and fast-transfer (inbound) mission, such that the overall mission duration is reduced to around 440 days, with a 30-day surface stay. Although the ΔV's are on the order presented above for opposition-class and fast-transfer missions, the amount of payload on such a mission is significantly reduced.

Table 38 : Mission class overview

Mission Architecture	Transfer TOF (days)	Surface Stay (days)	Total ΔV (km/s)
Opposition w/ Venus Swing-by	470-750	60	8-12
Conjunction	400-700	300-500	5-7
Fast-transfer	200-400	500-650	8-10
Split-sprint	410	30	12-18

9.4.1.3 Description of Scenarios

Having overviewed the four missions classes described above, Walberg detailed the required IMLEO for each mission using three scenarios. The first scenario is an all-propulsive maneuver including propulsive orbit insertion at both Earth and Mars. The second scenario uses aerobraking at both Earth and Mars to reduce propellant requirements. The third is the use of nuclear propulsion. For chemical propulsion, the specific impulse is 480 sec, and a structure to propellant ratio of 0.1 is assumed. For aerobraking, the structure mass is assumed to be 15% of the payload mass. For nuclear propulsion, a specific impulse of 960 sec. is assumed. In addition, a 5% gravity loss is assumed for all propulsive maneuvers.

9.4.1.3.1 Mass Payload Estimates

The general trajectory as outlined by Walberg includes five maneuvers: a trans-Mars injection from a 500 km circular Earth orbit, a mid-course correction, insertion into a 1-sol Mars orbit, a trans-Earth injection, and an insertion into a 500 km circular Earth orbit. Thus the mission architecture is similar to that of Apollo, using a Mars orbit rendezvous. All propulsive maneuvers to the surface of Mars and return to Mars orbit are included in the payload mass for the Lander. Walberg lists payload masses for the habitation module, landing module, and Earth return capsule. The habitation module is the crew living quarters during transit to and from Mars. The landing module includes crew living quarters for the surface stay as well as the propulsion for these maneuvers. The Earth return capsule is similar to that of an Apollo style mission and houses the crew during Earth orbit insertion and return to Earth.

The payload masses defined by Walberg are taken from Freeman et al. (Freeman, 1990) and are assumed to be in agreement with the NASA 90-Day study (NASA, 1989). However, the assumption of crew size is not stated in either Walberg (1993) or Freeman (1990), and although, the NASA 90-Day study assumes a crew of four for a Mars mission, it does not outline any solid numbers for payload masses. In addition, Walberg lists two different masses for the habitation module. A larger mass is given for opposition and conjunction class missions, due to the increased transit times. However, there is no difference in the Lander mass despite different surface stay times and no verification of the reduced mass estimates is provided.

The payload masses used in this paper are derived from a number of sources, but loosely reference equations and estimates provided in Larson (1999). These estimates assume a crew of 6, do not account for mission duration and assume a surface habitat is provided for all missions, such that the Lander mass is independent of surface stay requirements. Table 39: Comparison of opposition class mass estimates with Walberg through Table 41 list the payload masses for each mission class defined by Walberg, for presumably a crew of 4, and the current mass estimates used for a crew of 6. The Lander wet masses for the current estimates were calculated by applying the rocket equation, using ΔV's obtained from Larson (1999), and the vehicle assumptions stated above.

Table 39: Comparison of opposition class mass estimates with Walberg

A	Walberg mass (t)	Current mass estimates (t)
Crew size	4	6
Habitation	61	60.7
Lander (dry mass)	Unknown	30*
Lander (wet mass)	76	53.5
Earth return vehicle	7.8	9.2

* Assumes surface life-support and habitation requirements are pre-positioned

Table 40: Comparison of conjunction class mass estimates with Walberg

Module	Walberg mass (t)	Current mass estimates (t)
Crew size	4	6
Habitation	61	60.7
Lander (dry mass)	Unknown	30*
Lander (wet mass)	76	53.5
Earth return vehicle	7.8	9.2

Table 41: Comparison of fast-transfer mass estimates with Walberg

Module	Walberg mass (t)	Current mass estimates (t)
Crew size	4	6
Habitation	46	60.7
Lander (dry mass)	Unknown	30*
Lander (wet mass)	76	53.5
Earth return vehicle	7.8	9.2

9.4.1.3.2 Comparison of Initial Mass in LEO Estimates

Using the payload masses listed in Table 39 through Table 41, and ΔV's listed in Walberg, his results are verified with the rocket equation calculations used. Under the same mission assumptions as those provided in Walberg, we can compute the IMLEO for the current estimates and these results are listed in Table 42.

If we compare the IMLEO for each mission scenario, we notice a few trends. The mass estimates are higher than Walberg, but these estimates take into account a crew of 6, which would seem to validate the results. If we compare each mission scenario with and without aerobraking, it becomes obvious that aerobraking yields a significant benefit for each mission. Nuclear propulsion yields a significant mass savings for each mission class, which would indicate that this is an area of technology that may be worth developing.

Table 42: Comparison of IMLEO estimates with Walberg

Mission Scenarios	Walberg IMLEO (t)	Current 16.89 mass IMLEO (t)
Opposition	1268	1505
Opposition with Aerobraking	593	806
Opposition with Nuclear Propulsion	409	493
Conjunction	597	715
Conjunction with Aerobraking	500	599
Conjunction with Nuclear Propulsion	285	345
Fast Transfer	1440	1888
Fast Transfer with Aerobraking	599	806
Fast Transfer with Nuclear Propulsion	392	527

9.4.1.4 Conclusion

The payload estimates provided by Walberg can be reproduced using the rocket equation calculations defined, which validates the model used in this paper. The current mass estimates seem to provide a reasonable approximation of the IMLEO estimates since they are only slightly higher than those provided by Walberg, but are meant to accommodate a crew of 6 instead of a crew of 4.

Having analyzed the different IMLEO estimates for comparison with each other and with the different mass scenarios, it has become obvious that aerobraking is required to achieve reasonable IMLEO estimates for chemical propulsion. In addition, nuclear propulsion has been shown to yield an additional benefit, and may warrant further development.

9.4.2 Example Calculation of Initial mass in LEO

The initial estimates of the mass in LEO derived by applying the rocket equation in succession to the payload masses. The rocket equation for n stages is simply

$$\frac{m_p}{m_i} = \prod_{i=1}^{n}(1+\alpha_i)\exp\left(\frac{-\Delta V_i}{I_{spi}g_0}\right) - \alpha_i$$

where α_i is the structure factor. By applying this formula from the final payload mass delivered back to Earth, we can determine the initial mass in LEO for a chemical burn. If aerobraking is employed, the formula for calculating the initial mass for that stage is simplified to

$$m_i = (1+\gamma) m_p$$

where γ is the aerobraking factor, which is set to 0.15 for this analysis. If electric propulsion is used to determine the initial mass for a pre-positioned element, the equation used is

$$m_i = m_p \left(\frac{\exp \frac{-\Delta V}{I_{sp} g_0}}{\frac{\left(1 - \left(\exp\left(\frac{-\Delta V}{I_{sp} g_0}\right) - 1\right)\left(I_{sp} g_0\right)^2\right)}{2SP\eta t}} \right)$$

where SP is the specific power, γ is the efficiency, and t is the time of flight.

Table 43 shows an example calculation. In this example, we list the payload masses and the ΔV's necessary to perform each maneuver. The module names are the payload masses for different maneuvers, the functions are different transfer maneuvers required, and the specific impulse is assumed to be 425 sec. In this example, we assume that the return propulsion, Landers with fuel, and habitation module is pre-positioned. The total mass in LEO is the sum of all the components.

Table 43: Example calculation

Mars Orbit Rendezvous								
Module Names	**CODE**	**mass (kg)**	**TOF(days)**					
Habitat Module	HAB	40000	600.3					
Crew Operations Vehicle	COV	4600	60					
Lander	LAND	15000	0					
Modern Command Mod.	MCM	9200						
Functions	**CODE**	**DV (m/s)**	**TOF(days)**					
Trans Mars Injection	TMI	4098	264	days				
Mars Orbit Prop	MO	3278	0					
Mars Surface Descent	MSD	741	0					
Mars Re-orbit Prop	MO2	4140	0					
Trans Earth Injection	TEI	1415	176	days				
LEO	LEO	2774						
	Habitate		30	days				
Process	**Function**	**Payload**	**alpha**	**DV (m/s)**	**ISP (sec)**	**mp (kg)**	**mi (kg)**	**mp/mi**
Establish Earth Orbit	LEO	HAB+COV	0.1	2774	425	44600	99863.15	0.446611
Trans-Earth Injection	TEI	HAB+COV	0.1	1415	425	99863.15	149057.7	0.669963
Return to Mars Orbit	MO2	LM	0.1	4140	425	30000	104396.8	0.287365
Land on surface	MSD	LM	0.1	741	425	104396.8	128478.1	0.812564
Establish Mars Orbit	MO	LM+HAB+COV	0.1	3278	425	44600	116940.5	0.381391
Trans-Mars Injection	TMI	LM+HAB+COV	0.1	4098	425	116940.5	401183.9	0.291489
Preposition Lander								
Ion Engine Characteristics								
Specfic Power	150	W/kg						
efficiency	0.7							
Propultion time	51840000	sec (600 days)						
Process	**Object**	**Payload**	**alpha**	**DV (m/s)**	**ISP (sec)**	**mp (kg)**	**mi (kg)**	**mp/mi**
TMI and Orbit	EPprop	Return propulsion	0.1	7376	3200	104457.7	135399.8	0.771476
Process	**Object**	**Payload**	**alpha**	**DV (m/s)**	**ISP (sec)**	**mp (kg)**	**mi (kg)**	**mp/mi**
TMI and Orbit	EPprop	Landers	0.1	7376	3200	128478.1	166535.6	0.771476
Process	**Object**	**Payload**	**alpha**	**DV (m/s)**	**ISP (sec)**	**mp (kg)**	**mi (kg)**	**mp/mi**
TMI and Orbit	EPprop	Surface Habitat	0.1	8117	3200	13348	17765.34	0.751351
TOTAL Mass from LEO	**720884.577**	**kg**						

9.5 Knowledge Transport Calculations and Architecture

9.5.1 Architecture

The knowledge delivery infrastructure will consist of two parts the delivery of data in the form of bits and the delivery of samples from the planets surface. This section will largely deal with the delivery knowledge in the form of bits and thus will in this section be referred to as communication delivery system.

For every mission size the same communication radio frequency has been selected in order to provide an easily extensible system. The radio frequency that each of these missions will use is Ka-Band or 32 gigahertz. This frequency was selected because it can support a high data rate with comparably lower power than all lower frequency bands, and because the DSN ground infrastructure will support it by the year 2007, while other higher frequency bands are not supported by the DSN. There is some concern about weather interference especially when communicating with Mars, however a Martian sand storm would prevent a X-band communication as it would a Ka-band communication, the differences would mainly lie in the moderate weather such as a cloudy day, or light dust storm in which case the Ka-bands data rate would be decreased.

For the small sized lunar missions a direct link can be set up between the Moon Lander and one of the Earth's DSN stations. This would allow constant communication between Earth and the Moon throughout the entire mission. The data rate required for this mission would be 1 gigabit/day would require 0.01 Watts of power per transmission with a transmission data rate of 0.07 megabits/sec. After the mission is completed the communication equipment that was landed on the Moon will be left there for two reasons, one if a future mission decides to use that spot as a landing or settlement site then they won't have to bring their own equipment and in the unlikely case that another future mission communication equipment fails the crew will have the option of traveling in a rover to the old site and using its equipment.

For the medium sized missions the infrastructure is essentially the same as the small mission except that it will require a higher daily data rate and transmitting power.
Daily data rate: 10 gigabits/day
Transmission data rate: 0.7 megabits/sec
Power required: 0.1 Watts

For the Large sized missions require the ability to communicate between the far side of the planet and Earth. The astronauts will communicate through one of four possible ways. For the first option an comm. relay satellite could be placed at the L4 point in the Earth Moon system, this is good because it allows for a constant communication stream between the Earth and the Moon, unfortunately this option would only allow for communication for the first 900km onto the Moon's far side. The next option is to set up a relay satellite in a Low Lunar orbit that has the advantage of covering most of the Moon, but the disadvantage of a large time delay between far side communications. The third option is probably the best and involves setting up a satellite in orbit around the Earth-Moon L2 (EM-L2) point thus covering the far side of the Moon in it's entirety and can keep in almost constant communication with Earth, the drawback to this architecture is that though orbiting the L2 point is technically and theoretically feasible it is untried and less stable than placing the satellite at a Lagrangian point. The last option is more for emergencies sake than anything else, it is possible for the dark side of the Moon to use an orbiter around Mars or a Martian settlement as a relay to Earth, the main problem with this is that it would require a large amount of power for a very small data rate, and would only be feasible at certain windows when Mars is visible to the far side of the Moon. The daily data rate for a large sized mission would be 50 gigabits/day, the transmission data rate would be 3.5 megabits/sec and the transmission power required would be 0.5 W.

All of these missions will have the capacity in some manner to point their antennas at Mars and send or receive communications from or to future Mars missions at low data rate.

For the Mars missions only a small a large size mission will be considered. For the small sized Mars missions a direct link can be set up between the Mars Lander and one the Earth's DSN stations. This would allow semi-frequent communication between Earth and Mars throughout the entire mission. The data rate required for this mission would be 1 gigabits/day and would require 8 Watts of power per transmission with a transmission data rate of .035 megabits/sec. After the mission is completed the communication equipment that was landed on Mars will be left there for two reasons, one if a future mission decides to use that spot as a landing or settlement site then they won't have to bring their own equipment and in the unlikely case that another future mission communication equipment fails the crew will have the option of traveling in a rover to the old site and using its equipment.

The Large sized missions require the ability to communicate with much greater data rate and thus it might be necessary to create a relay satellite around Mars. There are two realistic options for the location of this sat. The satellite could be placed in a Geostationary Martian orbit around the landing site, the advantage of a GMO satellite is that it increases the time that the astronauts can communicate with the Earth, the disadvantage is that it can only really be set up for one portion of the planet. The other option is to position a satellite at the Earth-Mars L1 point thus decreasing the power required to send large communication streams to the Earth, unfortunately this would not add any extra time that the mission could communicate with the ground. As with the Moon missions there is an option in the case of emergencies to communicate with the Moon and use it as a relay station. The daily data rate for a large sized mission would be 10 gigabits/day, the transmission data rate would be 0.35 megabits/sec and the transmission power required would be 8 W.

Image removed due to copyright restrictions.

Figure 94: Communication Architecture

9.5.2 Calculations

Link Budget:

$$P = E/N+10*LOG(k)+10*LOG(T)+10*LOG(R)-L_l-L_h-L_{it}-L_w-L_p-G_t-L_s-L_a-L_o-G_r$$

Where:
P = power
E/N = signal to noise ratio
  Derived from the required bit error rate (BER) and type of coding.
k = Boltzman's constant
T = antenna noise temperature
  Provided by the ground stations
R = data rate
  Estimated for different sized missions
L_l = transition station line loss
  Estimated
L_h = hot body noise loss
  Estimated
L_{it} = Ionospheric & Tropospheric loss
  Estimated
L_w = weather losses
  Estimated from DSN 810-005

L_p = polarization mismatch loss

Estimated

L_s = space loss = $10*\log((\lambda/(4\pi S))^2)$

Where S is the distance between the transmitter and the receiver and λ is the wavelength

L_a = receiving station line loss

Estimated

L_o = other losses

Estimated

G_t = transmitting antenna gain = $10*\log(\eta((\pi*D)/\lambda)^2)$

Where D is the diameter of the antenna, λ is the wavelength, and η is the antenna efficiency.

G_r = receiving antenna gain = $10*\log(\eta((\pi*D)/\lambda)^2)$

Where D is the diameter of the antenna, λ is the wavelength, and η is the antenna efficiency.

9.5.3 Optical Communication Trades

The case for the use of optical communication in an extensible exploration program is not as strong as the current knowledge of Ka-band or X-band communication. Optical communication has several advantages over the more common radio communication channels, these being a much higher data rate for the same amount of mass, power and volume, all very important in the design of a space mission. Unfortunately there are also several devastating drawbacks to optical communication, in particular the serious losses due to atmospheric interferences and its high pointing requirements, making it next to impossible to use optical communication over any distance greater than from the Earth to the Moon. These drawbacks while serious can be overcome in some cases by the aforementioned advantages, however, in an extensible exploration program such as this one that is being proposed, it is far more important to have a common communication system for the entire program, thus when the need arises to extend to the next exploration site there is already a communication network in place to help relay transmissions. This is not to say that optical communication should not be used in all space missions, on the contrary there are many situations where the use of optical communication would benefit the mission substantially, however in this case it is wiser to use a more established form of radio communication.

9.5.4 Mars Science Details (Knowledge)

One of the main interests of Mars is the search for water. Possible locations for water are the polar caps, subsurface ice, gullies, stream lined islands, rampart craters, outflow channels, and layered terrain (PSSS 2003). There are two scientific methods of determining water on Mars, geologically or by studying the climate and their objectives and approach are summarized below.

Follow the Water		Objectives	Approach
	Geology	• Search for liquid water aquifer • Characterize crustal structure • Characterize seismicity	• Active seismic refraction and ground penetrating radar • Long-term passive seismic monitoring of mars quakes and impacts
	Climate	•Characterize the atmospheric boundary layer •Constrain global climate models •Search for minor organic constituents	• Long-term monitoring of temperature, pressure, wind velocity, solar flux, and humidity • Infrared spectrometry

Figure 95 Mars Science Objectives (PSSS, 2003)

Some of main questions in Mars geology include understanding planetary origin and evolution by determining the core and mantle size and composition and mapping the current and past tectonic activity (PSSS 2003). Additional knowledge questions can also be found using Mars Field Geology, Biology, and Paleontology Workshop (Harvey 1998).

9.5.5 Additional Knowledge Materials (background)

A great deal of research has been done on the instruments needed to gather scientific and resource related knowledge on the Moon and Mars. An important first step before sending robotic explorers is to understand what current information exists. An example of a database of Moon and Mars constituents and their locations can be seen in Table 1. Future robotic missions can add to the resolution of location and occurrence until it is necessary to send a human mission.

Table 44: Moon resources - preliminary findings (Taylor, 2001)

Table 1. Geological occurrence of important commodities on Earth and possible locations on the Moon and Mars.

Type of Occurrence	Main Commodities	Moon	Mars
Mafic igneous rocks	Cr, Pt-group, Ni, Ti	Central peaks of craters; crater rim deposits	Central peaks and ejecta of craters in highlands
Pegmatites	Ta, Nb, Be, Li	No	Possibly in highlands
Hydrothermal	F, Au, Ag, W, Cd, Bi, Hg	No	Volcanic systems, ancient highlands, large impact craters
Volcanic	Ag, Pt-grp, Zn, Hg, Pb, Cu	Pyroclastic deposits; near vent deposits in maria	Volcanoes, volcanic plains, possible pyroclastic deposits
Chemical sedimentation	Mn, Co, K, Fe, S, P	No	Ancient lake beds or hypothetical northern ocean; hematite deposit in Sinus Meridiani
Mechanical sedimentation	Th, Sn, Au	No	Outflow channels
Weathering	Al, Ni	No	Oldest surfaces with deepest soils, if there are soils
Solution-remobilization-deposition	K, Pb, U, V	No	Ground water systems; possibly in surface regolith
Space weathering	None	Solar wind gases in regolith; microscopic metallic iron	Organic compounds from micrometeorites

A method of determining science and resource related knowledge is through the use of a geophysical network. There are several methods of achieving this, outlined by the Lunar Exploration Science Working Group and the 2003 JPL Planetary Science Summer School (LExSWG 1995, PSSS 2003). Both studies included using penetrators and soft Landers. A summary of different deployment methods and their advantages and disadvantages are seen in Table 45. A challenge of creating geophysical nets is aligning the instrument and achieving a large global access for a long duration. Current planning tends to focus on penetrators and soft landings, which can be accomplished by robotic missions such as the Mars Net Landers. However human missions have two advantages over robotic geology missions. They are the experience knowledge gained by a human mission and a more optimal aligning and positioning of geologic instruments. The goals of the space transportation system are to eliminate the current disadvantages listed such as the high expense and global access.

Table 45: Methods of creating geophysical networks (LExSWG, 1995)

Deployment Method	Advantages	Disadvantages
Penetrators	Inexpensive Global access Good coupling of seismometer to surface	Lifetime short Survivability Instrument alignment difficult
Soft Landers	Surface emplacement feasible Long-lived power sources	Several costly spacecraft needed Coring required Alignment difficult
Rovers	Surface emplacement feasible Moderate expense Long-lived power sources	Long-transverse needed Coring required Global access
Humans	Ease of emplacement Ease of alignment Possible repair of instruments Long-lived power sources	Expensive Life-support needed Protection from radiation Global access

Past research on the capabilities needed to varying amount of knowledge returned can also be seen in "Geoscience" (1988). Outlined in Table 46 are three levels of knowledge, correlating with small, medium, and large, for a Moon mission and their respective instruments with focus on geology. Future Moon and Mars architectures should also have detailed instrumentation and the information gathered levels for climatology and resource related knowledge.

Table 46: Knowledge levels and instrumentation for a moon mission (Geoscience, 1988)

Capability	Instruments	Information	Purpose
Basic	Stereo microscope with VIMS Crude chemical analyzer	Mineral identification Grain size Chemical Classification	Classification of rock types
Intermediate	Automatic thin-section machine Reflected light microscope with VIMS Improved chemical analyzer X-ray diffractometer Ferromagnetic resonance spectrometer	Above plus: Texture Bulk composition Soil Maturity Magnetic Properties	Improved classification Identification of unusual samples Identification of soils rich in gases Study of Moon's magnetic field
Advanced	Above plus: Automated SEM with energy-dispersive X-ray analyzer	Above plus: Mineral composition and zoning Microtextures	Above plus: Even better rock classification Identify samples unusual in subtle ways

9.6 Lunar Landing Sites

For this design study we do not need a thorough understanding of the geologic value of the Moon (or Mars), but we do need to know what landing sites will be sought by the scientists, so that we can design our missions accordingly. Unlike early Apollo missions (Figure 96), we should not be so constrained to the equator. Based in recommendations from some planetary geologists as well as landing site selection papers, we recommend the capability for orbits to at least +/- 30° latitude, and maintain the capability to land on the far side. This should satisfy most of the suggestions for landing sites.

The following is an elementary and somewhat oversimplified summary of some of the scientific motivations for going to the Moon, and associated landing sites.
Figure 97 and Figure 98 show the sites (sites **[A]** through **[G],** plus Apollo 11 site in Figure 96), and images of each site are also shown in the figures section.

Volcanism. When did volcanism end on the Moon? The *Lichtenberg Basalts* (68°W, 32°N, **[A],** suggested by Robinson, personal communication) are potentially the Moon's youngest basalts; they may be between 1 and 2.5 billion years old. This area would give us a view of what basalts looked like from the beginning of the Moon's volcanism (~3 billion years) to the formation of this area.

Lunar volatiles. *Aristarchus Plateau* (48°W, 24°N, **[B]**, suggested by Robinson and Taylor, personal communications) is a complex pyroclastic center with a rich volcanic history. Samples are likely to be a diverse suite of the magma source, and it would be possible to determine spectral reflectance/composition from this previously unsampled material. *Alphonsus* (3°W, 13°S, **[C]**) and *Sulpicius Gallus* (20°N, 12°E, **[D]**, both suggested by Robinson, personal communication) are two other regions with pyroclastic materials that would hint at the history of lunar volatiles (which are necessary for volcanic explosions). The *South Pole Aitken Basin* (crater is 172°E, 18°S, **[E]**, massifs 155°E, 25°S) is also a good place to study lunar volatiles (including water), as mentioned in The Poles section below.

KREEP basalts. Apollos 12, 14, and 15 found a strange basaltic material they nicknamed KREEP (potassium, Rare Earth Elements, and Phosphorus). Rare earth elements are extracted from liquid magma when other elements in the magma thermally differentiate and cool into a crystalline structure; the rare earth elements do not fit into this crystalline structure, so they form abnormally high concentrations, nicknamed KREEP. This finding introduced the concept that lunar maria did not form simultaneously, but over hundreds of millions of years. The *Apennine Bench Formation* (mountain range centered at 0°, 20°, **[F]**, Robinson, personal communication) would allow us to sample this mysterious material and learn more about the thermal differentiation of the ancient magma and maria formation. *Aristarchus Crater* (48°W, 24°N, **[B]**, Taylor and Schmitt, personal communications) is also likely to contain high concentrations of KREEP, and would also allow us study of cratering processes and crustal stratigraphy.

The poles. The *South Pole Aitken Basin* **[E]** (suggested by Robinson, Taylor, and Schmitt, personal communications, and Spudis 2000) is one of the most sought after landing sites. Its temperature variations are considerably less than elsewhere on the planet, and the -230° C temperatures in the shadowed regions may contain 4 billion year old volatiles (water – Harland, 1999). It is the oldest impact basin on the Moon, but the exact age is unknown. If we land near the massifs, we could "sample impact melts and do geophysical measurements to study the structure of basin massifs to understand how they formed" (Taylor, personal communication - a massif is basically a mountain range; faults and folds in the Moon's crust). The composition of the South Pole Aitken Basin is currently not known, but it is known that interior of the basin has not been later filled or covered by foreign material (Pieters *et al*., 2003).

Stratigraphy. As mentioned above, *Aristarchus Crater* **[B]** and the *South Pole Aitken Basin* **[E]** are useful for studying crustal processes. The crater *Tsiolkosky* (129°E, 21°S, **[G]**, suggested by Taylor and Schmitt, personal communications) has a central peak, which may be a part of the original lunar crust. It may be a "great place to study the nature of cumulate anorthosites" (Taylor, personal communication; anorthosites among the most ancient rocks on the Moon) as well as crustal stratigraphy.

Seismology. The nature of the lunar interior is still somewhat ambiguous. Neal *et al.* (2003) suggest a lunar seismic network (see Figure 99), including a minimum of 8 seismometers deployed around the Moon. These could be deployed with unmanned or manned missions, but could certainly involve international cooperation. Seismometers would allow testing of the hypothesis that the Moon was formed from a magma ocean in its early stages.

Other questions (Ryder *et al.,* 1989). The origin of the Moon may be better studied if early thermal differentiation were better understood (see KREEP discussion above). Lunar mare basalts should be better studied to understand not only the thermal history of the Moon, but also the depth of the ancient magma oceans, and their detailed compositions. Lunar stratigraphy may help us to understand the impact history of the Moon – when were the greatest periods of bombardment? (This has implications for the history of Earth, because if there was heavy bombardment on the Moon, there also was on Earth.) Lunar regolith (loose soil), which sits on the top layer of the Moon, contains the history of billions of years of solar wind and flares. Finally, usable lunar resources (such as water and Helium-3) need to be explored for future manufacturing plausibility. Landing sites for this purpose may include *Mare Tranquilitatis* (Apollo 11 site) and *Aristarchus* **[B]** (both suggested by Schmitt, personal communication).

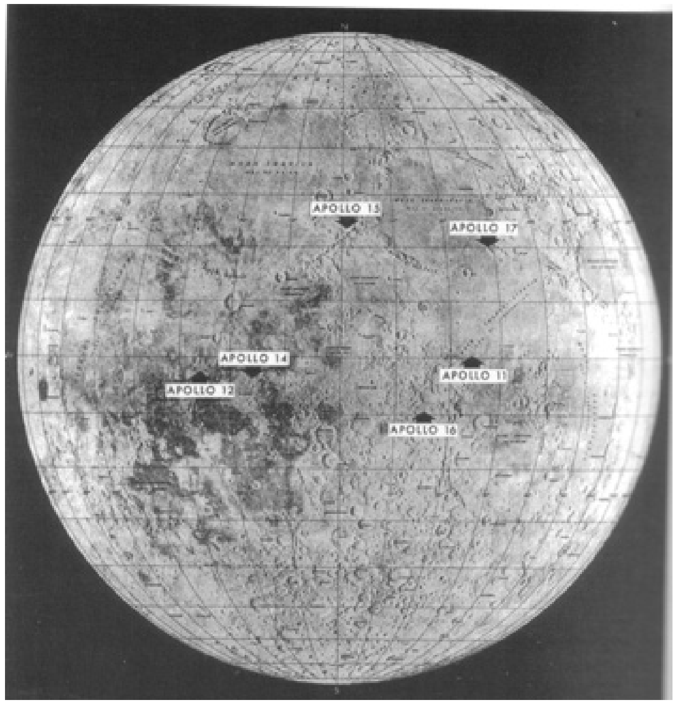


Figure 96: Apollo landing sites. Near side of the Moon, center (0, 0).

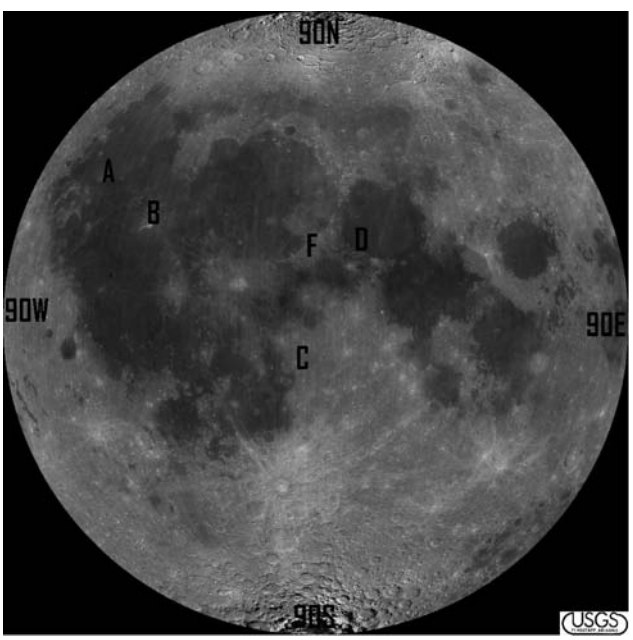


Figure 97: Near side of Moon.
Landing sites are numbered according to text of this report. Center (0, 0).

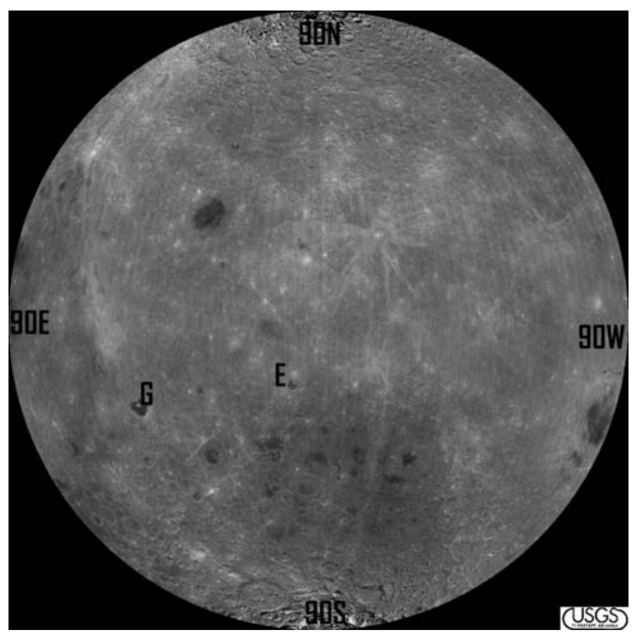
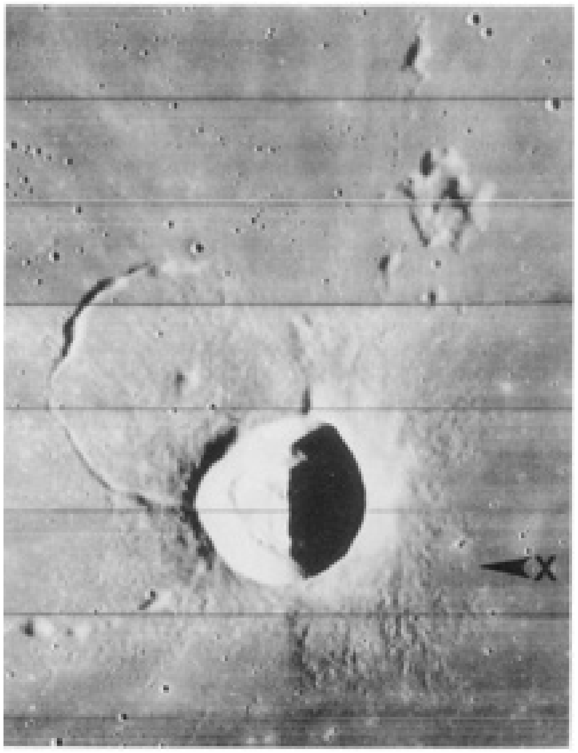
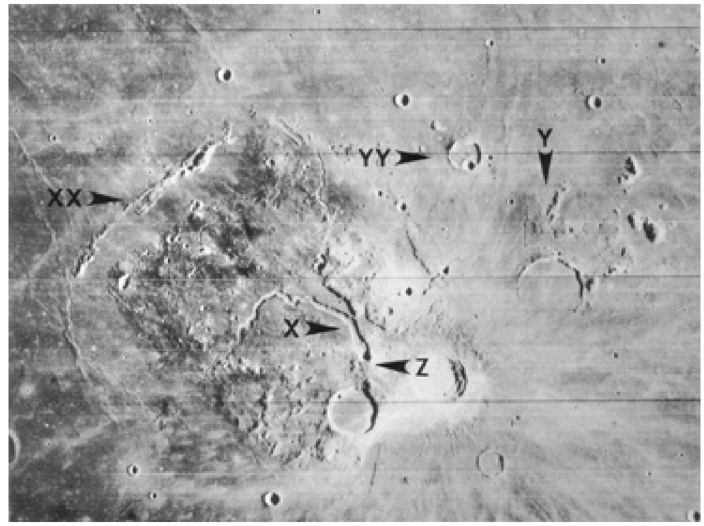


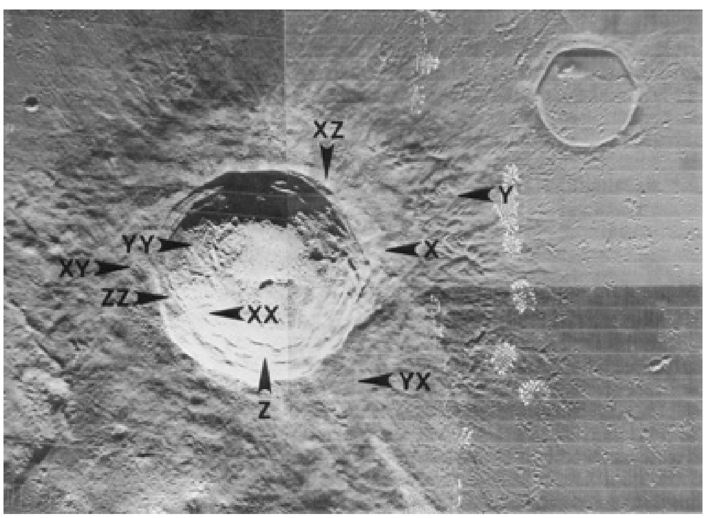
Figure 98: Far side of the Moon.
Landing sites are numbered according to the text of this report. Center (0, 0).



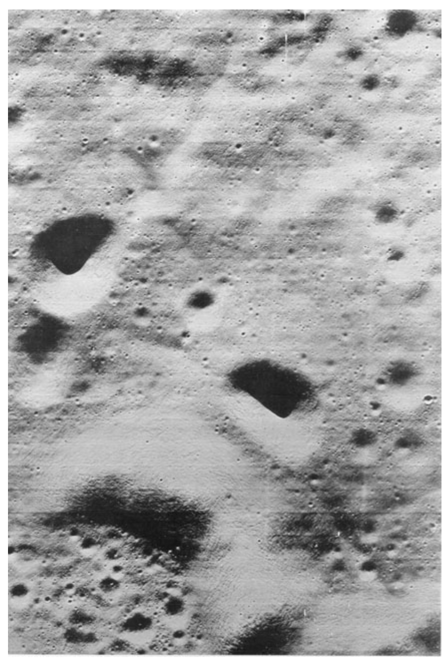
Site A. Lichtenberg Basalts.



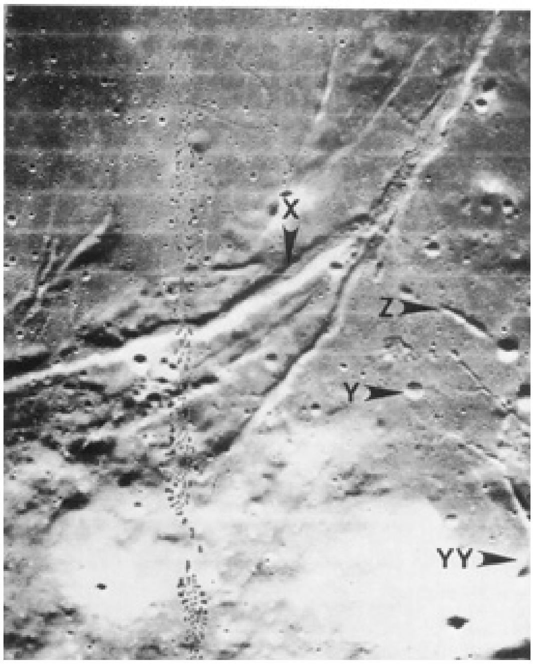
Site B. Aristarchus Plateau.

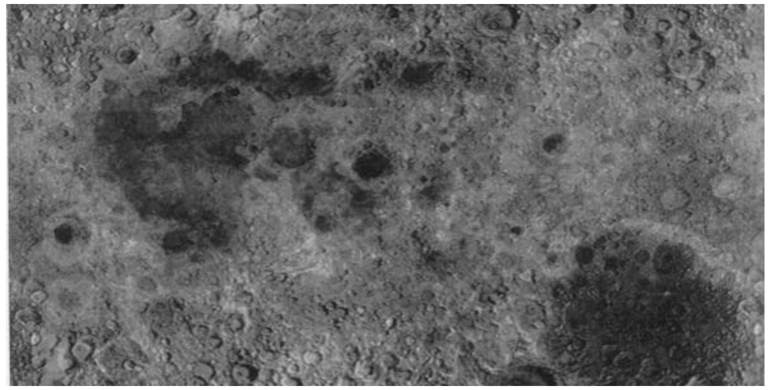


Site B. Aristarchus Crater.



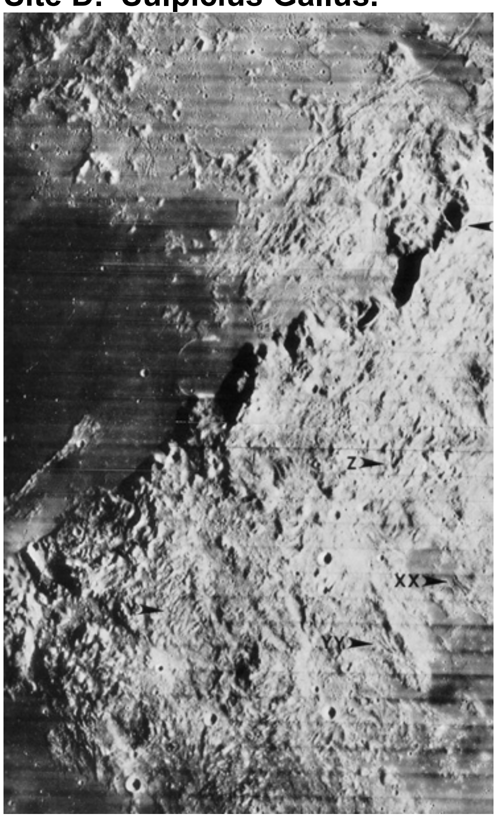
Site C. Alphonsus.

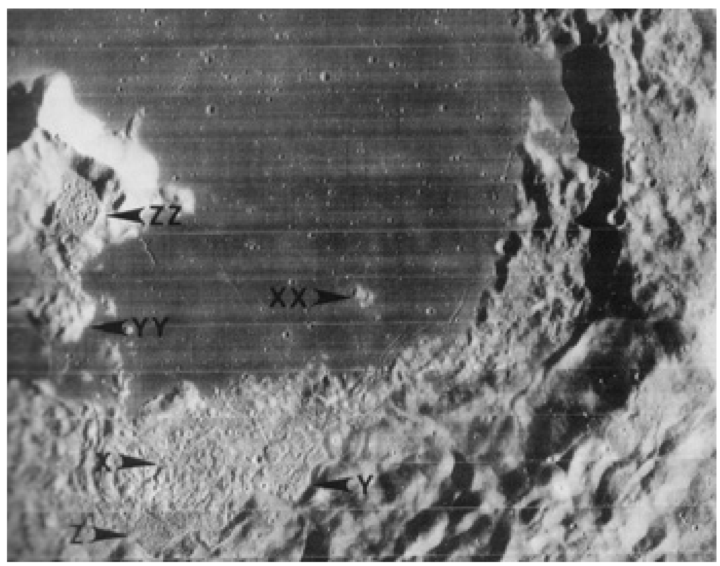




Site E. South Pole Aitken Basin.

Site D. Sulpicius Gallus.





Site G. Tsiolkoksy Crater (southwest edge shown; peak is on left side of picture).

Site F. Apennine Mountains.

Note: All images are from the NASA lunar orbiter, gathered from Schultz 1972, except Site E, which is from Harland 1999.

Image removed due to copyright restrictions.

Figure 99: Figure 1 from Neal *et al.* 2003. A lunar seismic network is proposed to study the Moon's interior.

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