Cyclical Visits to Mars via Astronaut Hotels

Phase I Final Report

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Abstract

Global Aerospace Corporation is developing a revolutionary concept for an overall interplanetary rapid transit system architecture for human transportation between Earth and Mars which supports a sustained Mars base of 20 people circa 2035. This innovative design architecture relies upon the use of small, highly autonomous, solar-electric-propelled space ships, we dub *Astrotels* for **astro**naut ho**tels** and hyperbolic rendezvous between them and the planetary transport hubs using even smaller, fast-transfer, aeroassist vehicles we shall call *Taxis*. Astrotels operating in cyclic orbits between Earth, Mars and the Moon and Taxis operating on rendezvous trajectories between Astrotels and transport hubs or *Spaceports* will enable low-cost, low-energy, frequent and short duration trips between these bodies. This proposed effort provides a vision of a far off future which establishes a context for near-term technology advance, systems studies, robotic Mars missions and human spaceflight. In this fashion Global Aerospace Corporation assists the NASA Enterprise for Human Exploration and Development of Space (HEDS) in all four of its goals, namely (1) preparing to conduct human missions of exploration to planetary and other bodies in the solar system, (2) expanding scientific knowledge (3) providing safe and affordable access to space, and (4) establishing a human presence in space. Key elements of this innovative, new concept are the use of:

- Five month human flights between Earth and Mars on cyclic orbits,
- Small, highly autonomous human transport vehicles or *Astrotels*,
  - In cyclic orbits between Earth and Mars
  - Solar Electric Propulsion for orbit corrections
  - Untended for more than 20 out of 26 months
  - No artificial gravity
- Fast-transfer, aeroassist vehicles, or *Taxis*, between Spaceports and the cycling Astrotels,
- Low energy, long flight-time orbits and unmanned vehicles for the transport of cargo,
- i*n situ* resources for propulsion and life support
- Environmentally safe, propulsion/power technology

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Table of Contents

List of Figures

List of Tables

1 Introduction

In this section we summarize the Mars transportation architecture concept, its objectives, and what makes it revolutionary; describe the potential significance of the concept to NASA and the World, and discuss past Mars studies and current relevant technology and activities.

1.1 Cyclical Visits to Mars via Astronaut Hotels

In 1985 the National Commission on Space (NCOS) published their plans for the future of space exploration, which included support to a sustained Mars base [1] (References for Section 1 are in Section 1.6). The NCOS plan assumed the existence of a sustained Mars base of 20 humans circa 2035, which required significant support in the form of crew replacement and cargo. The NCOS Mars base was supported by the use of large (>460 metric tonnes [mt]) interplanetary space ships for transporting humans and their material back and forth between the planets originally conceived by William Hollister at MIT in 1967 [2]. In addition, an entire support infrastructure was envisioned that includes human, cargo and propellant transfer vehicles, transport hubs and propellant manufacturing plants [3].

The new innovative Mars transportation system architecture concept being developed by Global Aerospace Corporation uses small, highly autonomous, solar-electric-propelled space ships, we dub *Astrotels* for **astro**naut ho**tels**, for transporting humans to and from Earth and Mars on cyclic orbits between these planets. Human transfer between planetary Spaceports and Astrotels is by means of hyperbolic rendezvous trajectories using new, even smaller, fast-transfer, aeroassist vehicles called *Taxis*. Figure 1-1 illustrates one concept for an Astrotel along with a Taxi docked at one end.

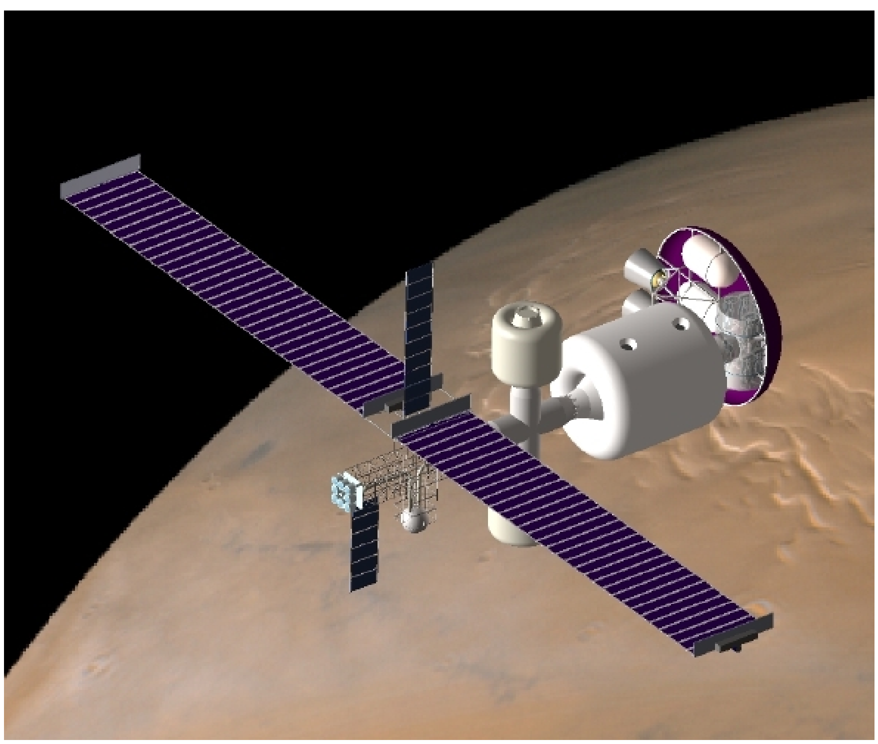


Figure 1-1 Astrotel and Taxi Concepts

These basic systems combined with other elements of the Mars transportation infrastructure and a new analysis of the celestial mechanics and aeroassist options will enable low life cycle costs, low–energy, frequent and short duration trips between these bodies. Figure 1-2 illustrates a schematic of the overall concept for regular human visits to Mars via an Astrotel concept that uses cyclic interplanetary orbits. The innovative design architecture being developed by Global Aerospace Corporation departs from the concepts in the mid-1980s in several fundamental ways, and these are described below.

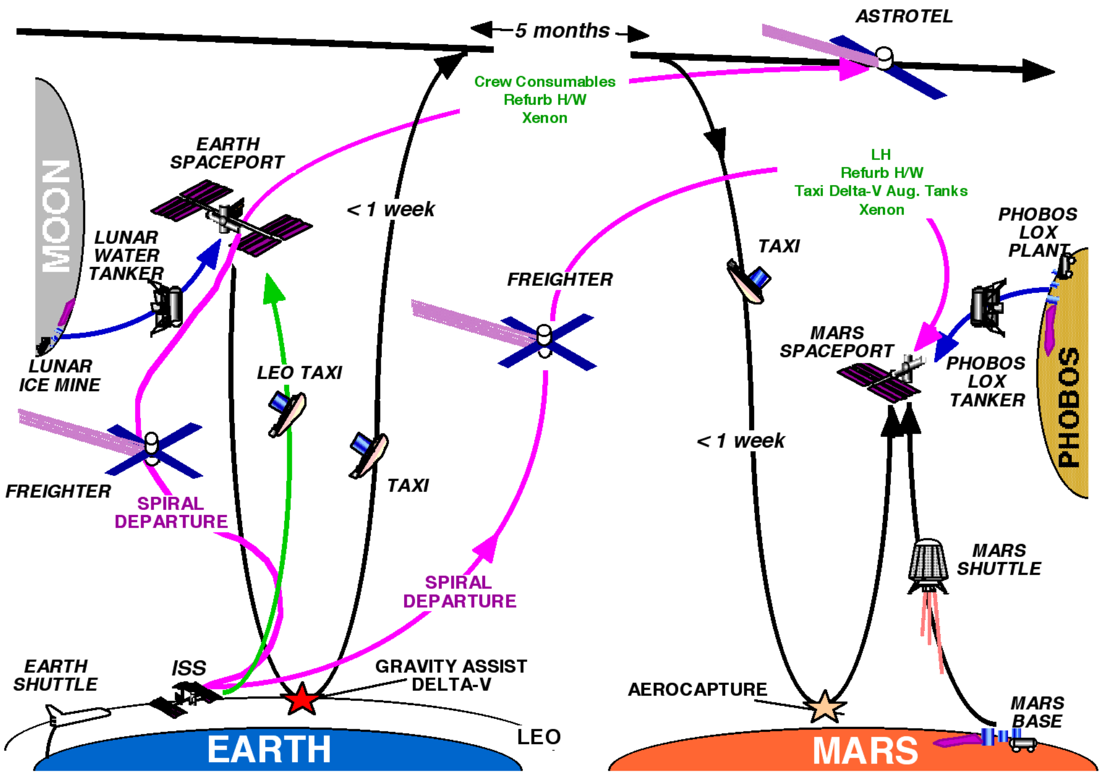


Figure 1-2 Mars Transportation Architecture Schematic

Our visions for a Mars base circa 2035, its transportation support concept, and its elements are discussed later.

1.2 What Makes This Concept Revolutionary?

First, this new transportation systems architecture assumes the use of highly autonomous on–board systems to a) reduce the number of crew and b) their occupation time of the transfer space ships to only five months in interplanetary space. Recent experience with untended space flight on the Russian Mir and the construction phase of the International Space Station make it clear that crew are not essential to maintain support systems. Reducing the size of crew and reducing the duration of their time spent in space reduces the size of the space vehicle and its complexity and the amount of logistics supporting the daily needs of the crew. In addition, by eliminating crew on long flight legs, we eliminate the need for additional Taxis for return to Spaceports thus reducing the number required by one half. Because these Taxis are not carried on these long trajectory legs, Astrotel propulsion requirements are therefore reduced.

Second, in previous plans, a means to generate artificial gravity was required due to the lengthy crew stay time (up to 2 1/7 years). The Mir experience, Russian (one year) and US (Shannon Lucid's 6 month flight), indicates 6 months of zero-g are clearly tolerable. When transit times are reduced to no more than 5 months, artificial gravity is no longer necessary thus reducing mass, complexity and risk.

Third, in past planning, conventional propulsion has been envisioned for the crew transport space ships using a Taxi's rockets. We are proposing instead to use solar electric propulsion for the periodic course corrections that are required for Astrotels (major corrections will generally occur during untended periods). Utilization of low-thrust Solar-powered Ion Propulsion reduces propellant mass requirements by a factor of 9. The cost in propellant mass for conventional chemical propulsion for course corrections for the large 460-mt vehicle over 15 years is more than 173 mt (more than twice our entire proposed Astrotel vehicle!). If we combine the interplanetary vehicle size reductions with SEP, the total reduction in propellant required for the Astrotel in 15 years is less by a factor of ***sixty***! This reduction has a tremendous mass and cost multiplying effect since all this propellant must also be mined, manufactured and stored, transported to the Spaceport and injected onto high-energy trajectories required for rendezvous with the Astrotels. See Table 1-1 for a comparison of several NCOS and preliminary NIAC study results including propellant requirements. As they are developed, evolutionary improvements in propulsion technologies will further reduce propellant requirements, but they probably will not change the fundamental architecture explored in this study.

Table 1-1 Comparison of NCOS and NIAC Study Results

Item	NCOS Study	NIAC Study	NIAC Improvement Factor
Cyclic Transport Vehicle Size, mt	460	70	7
Total 15-year Propellant and Consumables, mt	34,335	2,011	17
Lunar LOX Production Rate, kg/day	4,014	73	55
Phobos LOX Production Rate, kg/day	1066	189	6
Primary Power Generation Mode	Nuclear	Solar	--

Finally, in previous planning, all cargo except certain propellants needed at Mars, went via the same large crewed interplanetary space ship. The implication was that a lot of propulsive energy was being expended on hardware and supplies that could take a lot longer to get to Mars without detrimentally effecting the operation of the base.

All of these departures from the plans originally envisioned by the NCOS result in significant reductions in mass requirements and therefore they have enormous implications to overall energy requirements of a Earth-to-Mars transportation system. Reduced energy requirements impact the design of other elements of the transportation infrastructure and the cost of their development and operations. Since this new concept for support of a future Mars base results in a significant reduction in operations cost over previous concepts, a Mars base could be much the closer to reality. In fact, elements of this concept could be implemented at the very beginning of Mars exploration insuring that the first humans to Mars begin the permanent inhabitation of this our nearest, most hospitable neighbor.

The key elements of the overall Earth-to-Mars interplanetary rapid transit infrastructure in support of a permanent Mars base are listed below:

- Cycler orbits between Earth and Mars that enable fast, frequent transfers between these planets
- Small, human transport space ships, or Astrotels, on cycling orbits between planets,
- Orbital Spaceports at the planets
- Very small, fast, hyperbolic transfer vehicles, or Taxis, between Spaceports and Astrotels.
- Propellant and life support *in situ* resource manufacturing plants
- Cargo vehicles that utilize low-energy, long-flighttime orbits to transport propellant and low value cargo to and from planets
- Shuttles to and from Spaceports and planetary surfaces

1.3 Concept Objectives

The objectives of this concept are to provide:

- Low-cost, frequent access to Mars for scientists and explorers,
- Systems concepts that could be utilized in expedition phases of Mars missions, and
- A framework and context for future technology advance and robotic mission exploration.

The primary objective of this concept is to provide low-cost, frequent access to Mars for scientists and explorers by means of cyclic visits to and from Mars using new concepts for interplanetary transport vehicles. Such a concept will have significant implications on our ability to understand one of Earth's nearest neighbors and our preparedness for future visits to other planetary bodies.

The concepts envisioned by this systems architecture have a potential role to play in the expedition phase of Mars exploration. The application of these orbit and systems concepts in the expedition phase of Mars exploration may serve to reduce overall mission development costs and improve overall mission reliability and safety. Once launched into cycling orbits Astrotels can orbit indefinitely as long as they are periodically maintained, improved and supplied with orbit correction propellants. In addition, the result of embracing such a mission concept early in an expedition phase means that a permanent inhabitation phase of Mars is all the more closer.

Finally this interplanetary rapid transit concept provides a framework and context for future technology advance and robotic mission exploration. If one can envision an optimized interplanetary transportation systems architecture, then one can take steps today that will enable it. These steps could include establishing key technology goals to insure technology advance meets the future need. Other steps include embarking on robotic pathfinder missions to explore Mars, Phobos and the Moon and to search for *in situ* resources that are useful in any transportation systems architecture. For example, there is the high potential for the existence of water on the Moon, within Phobos and at the Martian North Pole. It is clear that the existence of water, or even just hydrogen, could have a dramatic impact on future plans and technology development for Mars exploration. Water broken down into its component molecular states of oxygen and hydrogen is rocket propellant. Hydrogen could be combined directly with oxygen for propulsion as with the current Space Shuttle. Or hydrogen could be combined with carbon to make methane, a more easily stored form of chemical energy. Past robotic missions have not unambiguously resolved the issue of water at any of these bodies listed above. Unfortunately, there are also no planned missions to resolve the uncertainties at this time. A concept for an Earth-to-Mars transportation system could generate the interest and excitement necessary to get such missions off the ground.

1.4 Potential Significance to NASA

NASA's Enterprise for Human Exploration and Development of Space (HEDS) has four goals addressed by this revolutionary concept, namely:

1. Preparing to conduct human exploration missions to planetary and other bodies in the solar system,
2. Expanding scientific knowledge,
3. Providing safe and affordable access to space, and
4. Establishing a human presence in space.

The proposed concept supports the first HEDS goal by providing a means to expand human exploration to Mars and by providing a transportation architecture that could be put in use to explore other planetary bodies, potentially near-Earth and Main Belt asteroids. The second HEDS goal is supported by this concept by enabling frequent, short visits to Mars by scientists. Opportunities for extended direct and teleoperated field science (e.g. geology) by scientists at Mars will swiftly expand scientific knowledge of the planet and increase our understanding of its similarities and differences with our own planet. This transportation architecture supports the third HEDS goal by offering transport to and from Mars at an expected very low life cycle cost. High life cycle costs will limit Mars exploration by Apollo-like expeditions. If life cycle costs can be significantly reduced, permanent exploration and inhabitation of Mars can be argued as being cost effective. This concept very clearly supports the fourth HEDS goal by contributing to the establishment of a permanent human presence on the planet Mars. Finally, this concept could also provide future direction to NASA regarding flight system technology development that could set the stage for Mars expeditions in the future.

This concept also supports an important goal of NASA's Space Science Enterprise (SSE), which is to "*pursue space science programs that enable and are enabled by future human exploration*". The mission and system architecture concept proposed by Global Aerospace Corporation assists NASA's SSE in their planning of future robotic exploration missions necessary to establish the key resource utilization technologies. In addition, there is the potential use of autonomous

Astrotels in exploration of the main belt asteroids that needs to be explored. Small robotic vehicles could be deployed from each Astrotel for main-belt asteroid exploration when the Astrotels are at the farthest from the Sun at about 2 AU or alternatively they could be deployed at Mars approach where they could use gravity assist to raise perihelion. Such a concept could enable a series of very low-cost, main-belt asteroid, sample return missions.

1.5 Past Mars Studies and Current Relevant Technology and Activities

Early studies of human missions to Mars primarily focused upon short duration human expeditions of the Apollo type [7-11]. Since that time, new concepts have been identified and developed which may become factors in the design of a future transportation system, including resonant orbits between planets, transportation hubs at libration points, the use of aeroassist concepts, tethers, and *in situ* resource utilization (ISRU). In addition, much more is now known about the effects of the space environment upon humans especially with the experience of the long duration U. S. and Russian missions to Mir.

In the mid-1980s, the philosophical approach to future manned Mars transportation evolved from Apollo-like expeditions to the consideration of a future Mars transportation architecture in support of establishment of a sustained Mars base [1, 12, 13]. Since that time many ideas have been developed that may become factors in this transportation architecture. The implication of such an approach could mean that the first people to visit Mars would begin an era of permanent inhabitation of Mars.

Current "Reference Mission" plans [14] are a hybrid of the Apollo expedition and sustained base approaches. Due to a desire to minimize overall costs, to reduce implementation time and to have a reasonably reliable expedition to Mars, a three-mission set has been studied. Such an approach however does not establish a sustained, permanent presence on Mars but instead results in deployment of significant infrastructure on Mars for future potential use. In this fashion, such missions could lay the groundwork for a sustained base. Providing a pathway to establishment of permanent activities on Mars is desirable in order to avoid a hiatus of human Mars missions like that following the Apollo Moon missions that has now lasted more than thirty years.

Current NASA planning for Mars missions includes the use of nuclear thermal rockets (NTR) for human flight to Mars. These NTRs are based on nuclear rocket technologies developed to the point of ground test firings in the 1960s [15]. One reason to use the highly efficient NTR is to reduce flighttime and thus the debilitating aspects of prolonged weightlessness. Another rationale for NTRs is the reduction in fuel mass required due to the higher specific impulse of the NTR engine as compared to conventional chemical propulsion. NTRs were shelved in the 1970s because they were expensive to develop, had environmental concerns, were potentially restricted by space treaties, and they were really not necessary as the Apollo program wound down.

In the "Reference" expedition-class mission currently under study by NASA, the mission is broken up into multiple elements that eventually rendezvous in orbit and on the surface of Mars. This strategy eliminates the undesirable assembly of major system components in Earth orbit prior to injection to Mars. Some of the low-value cargo elements, such as surface systems, Earth return and Mars ascent vehicles, go to Mars on minimum energy transfers similar to the approach we have proposed here.

After successful emplacement of key infrastructure in Mars orbit and on the surface, crews are sent using fast-transfer, NTR-propelled vehicles that use short flighttime, high-energy trajectories. An early, untended flight includes a Mars ascent vehicle, nuclear reactor, liquid hydrogen and a propellant production plant. Over 30 mt of propellant components (O_2 and CH_4) are manufactured from the hydrogen brought from Earth using the CO_2 in the atmosphere. An advantage of this propellant production mode is its autonomous nature and the absence of any required mining processes. Three crewed flights are planned, each preceded by slower cargo flights. Such a plan is attractive from the standpoint of delivering infrastructure and redundant system components to Mars before crews arrive. In addition, having the return propellant manufactured at the surface for the return flights results in a considerable savings of Earth launch and Mars injected mass.

Much progress has been made in the last 5 years and much work is obviously still required to develop an expedition concept for human Mars exploration. We list only a few concerns with expedition mode approaches to Mars exploration as used by NASA and other Mars Mission planners. First, there is a requirement for an expensive, new class of space rocket. Second, these approaches assume the launch and use of nuclear reactors in general for propulsion, propellant production and power. These reactors must be built and fueled on Earth with all the usual environmental and political baggage that they entail. Finally, we note the lack of a plan for sustaining transportation features after the three expeditions.

The concept developed in this NIAC Phase I effort involves none of these undesirable features of past studies and has a number of new attractive features that are discussed in the following sections.

1.6 References

1. National Commission on Space, *Pioneering the Space Frontier*. Bantam (1986).
2. W. M. Hollister, Castles in space. *Astronautica Acta*. (1967).
3. K. Nock and A. Friedlander, Elements of a Mars Transportation System, *Acta Astronautica*, Vol. 15, No. 6/7, pp. 505-522, 1987
4. E. E. Aldrin, Cyclic trajectory concepts. SAIC presentation to the Interplanetary Rapid Transit Study Meeting, Jet Propulsion Laboratory (1985).
5. A. L. Friedlander, J. C. Niehoff, D. V. Byrnes and J. M. Longuski, Circulating transportation orbits between Earth and Mars. AIAA Paper No. 86-2009-CP, AIAAIAAS Astrodynamics Conference, Williamsburg, Va (1986).
6. S. Hoffman, A. Friedlander and K. Nock, Transportation Performance Comparison for a Sustained Manned Mars Base, AIAA Paper No. 86-2016-CP, AIAA/AAS Astrodynamics Conference, Williamsburg, Va (1986).
7. Von Braun, *The Mars Project*. University of Illinois Press, Urbana, Illinois (1953).
8. E. Stuhlinger and J. C. King, Design requirements for a fleet of electrically propelled vehicles for a manned Mars expedition. In Progress in Astronautics and Aeronautics, Vol. 9, pp. 647-664 (1963).
9. L. Jones and W. V. McRae, Manned Mars landing and return mission study, Vol. 1. North American Aviation, Inc, SID 64619-1 (1964).

10. Anon., Alternative mission modes study. TRW Systems Group, Vol. I, II, NAS: 8-18056 (1967).
11. G. S. Canetti, Definition of experimental tests for a manned Mars excursion module, Vol. 1-3. North American Rockwell Corp., NASA CR 65911-65913 (1968).
12. P. J. Boston, The Case for Mars, Vol. 57. Science and Technology Series, American Astronautical Society (1984).
13. M. B. Duke and P. W. Keaton (eds.), Manned Mars missions--a working group report. Report No. NASA M001 (1986).
14. S. Hoffman and D. Kaplan (eds.), Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team, http://spaceflight.nasa.gov/mars/reference/hem/hem1.html.
15. E. Stuhlinger, Study of a NERVA-electric manned Mars vehicle. Presented at AIAA Stepping Stones to Mars Meeting, Baltimore, MD (1966).

2 Concept Architecture Development Summary

2.1 Summary of Phase I Tasks

Phase I of the Cyclical Visits to Mars via Astronaut Hotels Concept Development included the following tasks as originally planned and described.

2.1.1 Task 1 Conceptual Design Requirements and Assumptions

Key conceptual design requirements of a Earth-to-Mars transportation system architecture for sustained support and logistics to a Mars base circa 2035 will be identified and defined.

2.1.2 Task 2 Celestial Mechanics

Advanced celestial mechanics and aeroassist dynamics technologies that can help to achieve low energy, low cost, frequent and short flighttime missions to and from Mars will be reviewed, explored, evaluated, and optimized. Propulsion requirements for key elements of human transport will be established.

2.1.3 Task 3 Conceptual Transportation System Architecture Design

A conceptual, integrated Mars transportation system architecture will be developed, primarily focusing on the key elements of the Astrotel and Taxi, that are required to support a Mars base circa 2035. The impact on architecture design will be assessed for the use of *in situ* resources for propulsion.

2.1.4 Task 4 Transportation System Costs

Costs for a human Mars transportation system relying on the new Astrotel and Taxi concepts will be estimated. Relative cost estimates will be used to compare concept options.

2.1.5 Task 5 Planning and Reporting

A detailed Phase I plan will be generated. Monthly status reports and a final report shall be written. We shall participate in and present a report at the NIAC Fellows Conference in the Washington, D.C. area in the Summer of 2000.

2.2 Summary of Work Accomplished

This section provides a concise summary of the work accomplished during the Phase I effort. A more detailed description follows in later sections.

2.2.1 Task 1 Conceptual Design Requirements and Assumptions

Key conceptual design requirements of an Earth-to-Mars transportation system architecture for sustained support and logistics to a Mars base circa 2035 were identified and defined. These requirements and assumptions are discussed in Section 3.

2.2.2 Task 2 Celestial Mechanics

The focus of the Phase I effort was on the optimization of the Aldrin Cycler concept, though alternative options were explored including the use of the so called Semi-cycler orbits. The Aldrin Cycler concept was optimized to achieve an optimal balance of low energy injection, regular and frequent transportation opportunities to and from Mars, reasonably short transit times for crew health and safety, low propellant budgets for orbit maintenance maneuvers, and generally lower cost transportation. Delta-V requirements were generated for the Astrotels and the Taxis over a 15-year period. Low thrust propulsion requirements for the Astrotel maneuvers and for the reusable SEP Cargo Freighters transfers to Astrotels and Spaceports were optimized and generated. Aeroassist concepts for the Taxi and Mars Shuttle vehicle were explored and developed.

2.2.3 Task 3 Conceptual Transportation System Architecture Design

A first order conceptual, integrated Mars transportation system architecture was developed, focusing on the key elements of the Astrotel, Taxi, Mars Shuttle, Cargo Freighters and the Lunar Water Tanker, which are all required to support a Mars base. *In situ* resource utilization options were identified, assessed and systems developed. The impact of transportation systems requirements on *in situ* resource utilization systems requirements and designs were developed and integrated into the overall system architecture model. A Mars Astrotel Model Architecture (MAMA) design model was developed that uses vehicle, resource and surface systems design models to compute the refurbishment hardware, propellant and consumables requirements throughout the transportation nodes including LEO, the Moon, Spaceports, Phobos and Mars. Vehicle configuration design concepts were developed for the Astrotel and Taxi.

2.2.4 Task 4 Transportation System Costs

A MAMA System Integrator (SI) has been developed that integrates the architecture system elements with cost assumptions to compute life-cycle costs including, advanced technology development, development launch and operations cost. Preliminary life cycle costs have been estimated for the Mars transportation system architecture studied in Phase I.

2.2.5 Task 5 Planning and Reporting

A detailed Phase I plan was generated. Five monthly status reports and a final report were written. We participated in and presented a report at the NIAC Fellows Conference at NASA GSFC in June of 2000.

3 Key Conceptual Design Requirements

This section contains a brief description of system definitions, key concept assumptions and preliminary requirements for a transportation system for astronauts between Earth and Mars. These conceptual design requirements are levied on the design of the operational system. This section serves to provide conceptual design requirements for use in the preliminary design phase of Phase 1 of the NIAC Astrotel study. A revised version of this section, taking benefit of the understanding of user requirements and the conceptual design details developed during Phase I, will be developed during the Phase II effort.

3.1 Definitions

3.1.1 Mars Base

The Mars Base is a permanent human surface station and research center for scientific study of Mars and its environs. It consists of human habitats and life support systems; warehouses; science laboratories; any resource mining, storage and distribution facilities; shuttle landing, operations, refueling, refurbishment and launch facilities; and human and robotic surface mobility systems and their refurbishment facilities.

3.1.2 Astrotels

Astrotels is a contraction of the words **Astro**naut Ho**tels**. An Astrotel is a crew habitat for fast human trips between Mars and Earth.

3.1.3 Spaceport

A Spaceport is a crewed habitat, warehouse and refurbishment facility in orbit in the vicinity of Mars or the Earth/Moon system. Spaceports are collection and way points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels.

3.1.4 Taxi

A Taxi is a small, self-contained, crewed, aeroassist spaceship for fast human transfers between Spaceports and Astrotels. Taxis employ aeroassist technology within a planetary atmosphere to reduce orbit energy thus facilitating orbit capture. Taxis utilize propulsion systems to escape planetary Spaceports, to rendezvous with Astrotels, to depart Astrotels and to rendezvous with Spaceports.

3.1.5 Shuttle

Shuttles are crewed aerospace vehicles for human and time-critical cargo travel between Spaceports and space stations or planetary surfaces. The characteristics of these vehicles may be quite different for Earth and Mars application.

3.1.6 Lunar and Phobos Propellant Tankers

These uncrewed vehicles are required to transport raw material (Lunar Water Tanker or Phobos LOX Tanker) to Spaceports where the materials are processed into propellants and/or the propellants are stored.

3.1.7 Freighter

Freighters are uncrewed cargo vehicles for transporting bulk materials between planets, Astrotels and Spaceports. Characteristics of Freighters may vary however they are expected to be made from the same modular building blocks. Freighters are reusable.

3.1.8 Propellant Augmentation Tanks (PATs)

Propellant augmentation tanks (PATs) provide additional propulsion capability to vehicles when delta-V requirements exceed the base vehicle capability. For example, when Taxi delta-V requirements exceed the Taxis capability (about 3.4 km/s) for Mars departures, PATs are added to the Taxi. PATs are not planned to be reusable. In some cases, the very large Mars delta-Vs, PATs must include additional rocket engines in order to reduce burn time and thus gravity losses.

3.1.9 *In situ* Resource Production Plants

These include planetary and orbital facilities where propellants are created from indigenous materials, e.g. oxygen, water, etc. or are processed e.g. electrolyzed, liquefied and stored. Potential locations for such plants include the Martian surface, Phobos and the Moon. Potential locations for processing plants include the Earth or Mars Spaceports.

3.2 Transportation System Assumptions and Requirements

There are both assumptions and requirements that govern this Phase I conceptual design study of an Earth/Mars transportation system. In Phase II, we will test several of these assumptions by comparing the performance of competing options.

3.2.1 Assumptions

A number of assumptions have been made that form the basis for the conceptual design study. These assumptions include the basic timeframe of Mars Base operations, number of people in a Mars Base, the technology horizon for the study, cargo transport assumptions, Spaceport locations, and propulsion and power systems.

3.2.1.1 Timeframe of Mars Base

The timeframe for a sustained Mars Base is assumed to be circa 2035.

3.2.1.2 Mars Base

It is assumed that all food and life support consumables required at Mars Base are grown and processed at or near the base.

3.2.1.3 Mars Base Contingent

It is assumed that the full Mars base crew is 20 human beings and many robots. Half the crew is assumed rotated to Earth on each opportunity. This means that there are times when there are only 10 people on Mars for short periods (2-4 months).

3.2.1.4 Lunar Base

It is assumed that a sustained Lunar Base exists for its own scientific rationale. The Mars transportation requirements imposed on a Lunar Base are therefore only incremental in nature.

3.2.1.5 Technology Horizon

For the purpose of this study, the technology horizon will be 2010. All technology is assumed to be at NASA Technology Readiness Level (TRL) equal to 9 by 2010. This means that this study will not employ technologies if they are not projected to be at TRL-9 by the end of 2010. This early date, as compared to the 2035 date for a sustained Mars base, insures that the systems and architecture concepts developed in this study could be utilized for near-term missions to Mars.

Typically, technology advances despite plans or the lack thereof. It is therefore desirable to have a system architecture that is robust to reasonable technology advance, which means that when technology evolves, the system architecture can take advantage of that advance without wholesale alteration of the system architecture.

3.2.1.6 State of Scientific Knowledge

It is assumed that the currently envisioned Mars robotic science exploration program will be carried out through the return of one or more samples from Mars by 2012. These missions will provide fundamental data for Mars surface mission planning and propellant production.

3.2.1.7 Space Station

A Space Station in LEO is a key transportation node of any Mars transportation infrastructure. It is the place where Earth cargo will be collected for transport to Mars and the Astrotels and where crew will be transferred between LEO and the Earth Spaceport.

3.2.1.8 Robotics and Automation

Extensive use of robotics and automation are assumed throughout the entire Mars transportation system architecture. Activities such as equipment monitoring and fault protection, cargo handling, refueling, periodic machine maintenance, *in situ* resource system operation and many others are carried out by robots and other automated machines.

3.2.1.9 Cargo Transport

Bulk cargo such as propellants, PATs, refurbishment hardware, and other non-time critical cargo are sent to Mars, Planetary Spaceports, and Astrotels by means of Cargo Freighters that fly low thrust trajectories to and from their targets. It is assumed that cargo does not need to be transported from Mars to the vicinity of the Earth. Off-loading of cargo is assumed to be carried out robotically, thus not requiring human crews to be inhabiting the target vehicle at the time of arrival of Freighters.

3.2.1.10 Spaceports

3.2.1.10.1 Locations

For the purposes of the Phase I conceptual design study; (a) the Earth Spaceport is located at the Earth/Moon L1 point and (b) the Mars Spaceport is located in a Phobos-like orbit in the near vicinity of Phobos. These locations are chosen for the Phase I study because the delta-V requirements to and from them from the Earth or Mars are reasonable compared to other potential locations and they are close, energetically, to potential propellant resources.

3.2.1.10.2 Crew

The Mars Spaceport is assumed not to have a permanent human crew. The Earth Spaceport may be crewed only if required to support activities in Earth-Moon space such as the Lunar Base. Temporary crews on their way to or from Mars will carry out maintenance.

3.2.1.10.3 Propulsion Systems

Spaceport station-keeping propulsion is assumed to be a xenon ion solar electric propulsion (SEP) system identical to but smaller than the Astrotel propulsion system.

3.2.1.11 Astrotel

3.2.1.11.1 Size

The basic volume of the Astrotel is assumed to be the same as the Mars Reference Mission Mars Hab Module [14], which is based on the TransHab that has an inflated volume of 340 m^3.

3.2.1.11.2 Propulsion Systems

Astrotel propulsion is assumed to be a xenon ion solar electric propulsion (SEP) system.

3.2.2 Requirements

Requirements on the various Mars transportation infrastructure are delineated below. Where a “TBD” is used, estimated requirements are shown in brackets “[estimated]”.

3.2.2.1 Interplanetary Cyclic Orbits

3.2.2.1.1 Minimum Planetary Flyby Altitude

The minimum planetary flyby altitude for the conceptual design studies shall be 200 km.

3.2.2.1.2 Orbit Correction Delta-Vs

Minor orbit correction delta-Vs shall be performed in the vicinity of planetary flybys in order to achieve precise flyby targeting and to correct navigation errors after the flyby.

3.2.2.1.3 Maximum Allowable Human Transfer Time Between Planets

The maximum transfer time between Earth and Mars shall be less than 6 months. It is desirable to minimize transfer times (5-6 months).

3.2.2.2 Spaceports

Spaceports are collection points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels. Spaceports shall support crew needs during the crew stopovers between interplanetary Taxis and planetary Shuttles. Spaceports perform delta-V maneuvers required for station-keeping, provide protection to the crew from solar flare, or solar proton events (SPE), and cosmic ray radiation, provide electrical power to its subsystems, are highly autonomous, and are capable of refurbishment and autonomous resupply.

3.2.2.2.1 Vehicle Type

There shall be a Mars and Earth Spaceport vehicle type, which shall have a high degree of subsystem commonality. In addition, Spaceports shall have high subsystem commonality with the Astrotel vehicle.

3.2.2.2.2 Crew Complement

The Spaceport shall generally not be tended between crew rotation phasing. During tended phases it shall accommodate a crew of 10 for periods of time up to 10 days.

3.2.2.2.3 Crew Support

Crew support shall include life support; minimal dormitory, kitchen, health, and recreation facilities; interplanetary and local communications; and computational capability.

3.2.2.2.4 Propulsion

The propulsion subsystem shall be capable of carrying out all station-keeping delta-Vs.

3.2.2.2.5 Radiation Protection

Radiation sensors and radiation shielding to a TBD level shall be required to protect the crew against natural radiation including galactic cosmic rays (GCR) and solar proton events (SPE). Radiation shielding can be incrementally increased over time in order to improve the shielded environment.

3.2.2.2.6 Modularity

The power, propulsion and cargo hold systems shall have maximum commonality with similar systems on the Astrotel and Freighters. In other words, these subsystems shall be designed in modular sizes that can be used on Freighters, Spaceports and Astrotels.

3.2.2.2.7 Power System

The power system shall support all crew power requirements of TBD [10-30] kW, continuous, plus propulsion power requirements. Renewable energy storage capability shall provide emergency minimum crew power requirements for a period of TBD [8] hours.

3.2.2.2.8 System Interfaces

The Spaceport shall be capable of autonomous docking with the Taxis, Freighters, and planetary Shuttles.

3.2.2.2.9 Refurbishment, Repair and Upgrade

The Spaceport shall have facilities for refurbishment, upgrade and repair of high-maintenance hardware contained within its systems or in systems carried by any other vehicle that interfaces with it.

3.2.2.2.10 Resupply

Resupply of Spaceports shall be by means of Freighters.

3.2.2.2.11 Autonomy

The Spaceport shall be highly autonomous to reduce crew workload during crewed mission phases, maintain needed subsystems during uncrewed mission phases and enable autonomous resupply that may occur during untended phases.

3.2.2.3 Astrotels

The Astrotel shall support crew needs during the short transit between Earth and Mars, perform delta-V maneuvers, provide protection to the crew from solar flare, or solar proton events (SPE), and cosmic ray radiation, provide electrical power to its subsystems, be highly autonomous, and be capable of refurbishment and autonomous resupply.

3.2.2.3.1 Vehicle Types

There shall be only one vehicle type.

3.2.2.3.2 Crew Complement

The maximum crew size for the Astrotel shall be 10 people.

3.2.2.3.3 Crew Support

Crew support shall include life support, dormitory, kitchen, health, recreation, interplanetary and local communications, and computational capability.

3.2.2.3.4 Propulsion

The propulsion subsystem shall be capable of carrying out all navigation trajectory corrections and major orbit shaping delta-Vs.

3.2.2.3.5 Radiation Protection

Radiation sensors and radiation shielding to a TBD level shall be required to protect the crew against natural radiation including galactic cosmic rays (GCR) and solar proton events (SPE).

3.2.2.3.6 Modularity

The power, propulsion and cargo hold systems shall have maximum commonality with similar systems on the Freighters and Spaceports. In other words, these subsystems shall be designed in modular sizes that can be used on Freighters, Spaceports and Astrotels.

3.2.2.3.7 Power System

The power system shall support all crew power requirements of TBD [10-30] kW, continuous, plus propulsion power requirement of 160 kW. Renewable energy storage capability shall provide emergency minimum crew power requirements for a period of TBD [8] hours.

3.2.2.3.8 System Interfaces

The Astrotel must be capable of autonomous docking with the Taxi and the Earth-to-Astrotel Cargo Freighter.

3.2.2.3.9 Refurbishment, Repair and Upgrade (RRU)

Astrotels shall be capable of supporting RRU of high-maintenance subsystems contained within its systems or in systems carried by any other vehicle that interfaces with it. Major RRU will be carried out during crew tended periods.

3.2.2.3.10 Resupply

The Astrotel shall be capable of autonomous resupply by Freighters of bulk cargo and propellants.

3.2.2.3.11 Autonomy

The Astrotel shall be highly autonomous to reduce crew workload during crewed mission phases, maintain needed subsystems during uncrewed mission phases and enable autonomous resupply that may occur during untended phases.

3.2.2.4 Taxis

Taxis shall support crew needs during the very short transit (<10 days) between Spaceports and Astrotels. In addition, Taxis must perform delta-V maneuvers, perform aeroassist maneuvers within planetary atmospheres, navigate autonomously during all maneuvers, provide protection to the crew from solar flare, or solar proton events (SPE), and provide electrical power to its subsystems.

3.2.2.4.1 Vehicle Types

There shall be one basic vehicle type for Earth and Mars application. The propulsion systems for the basic Taxi vehicle shall be sized on Earth departure delta-V requirements (up to 3.4 km/s).

3.2.2.4.2 Crew Complement

The maximum crew size for the Taxis shall be 10 people for flight duration of less than TBD [7] days.

3.2.2.4.3 Crew Support

Life support, sleeping areas, food, and communications shall be provided.

3.2.2.4.4 Propulsion

The propulsion subsystem shall be capable of carrying out all navigation trajectory corrections and major orbit shaping delta-Vs up to 3.4 km/s. When delta-Vs are greater than 3.4 km/s, as with several Mars departures, PATs shall be used and staged during the departure maneuver. Taxis shall be required to maintain space storable propellants at cryogenic temperatures at Spaceports and Astrotels utilizing external power supplies.

3.2.2.4.5 Aeroassist

The Taxis shall be capable of aeroassist orbit capture at Earth and Mars.

3.2.2.4.6 Radiation Protection

Radiation sensors and radiation shielding to a TBD level shall be required to protect the crew against natural radiation including major SPEs.

3.2.2.4.7 Power System

Renewable energy storage capacity will be provided that will be sufficient to provide crew and support systems requirements throughout the cruise period to and from Astrotels and Spaceports.

3.2.2.4.8 Major System Interfaces

The Taxi must be capable of autonomous docking with the Astrotel and the Earth and Mars Spaceports. Taxis must also interface with the PATs when delta-V requirements are beyond the basic Taxi capability.

3.2.2.4.9 Refurbishment, Repair and Upgrade (RRU)

Taxis shall be capable of supporting RRU of high-maintenance subsystems contained within its systems. Major RRU of Taxis shall occur at the Earth Spaceports. Only minor RRU shall be carried out at Astrotels and the Mars Spaceport.

3.2.2.4.10 Resupply

Resupply of Taxis shall be carried out at Spaceports and Astrotels.

3.2.2.4.11 Autonomy

The Taxi shall be highly autonomous to reduce crew workload during crewed mission phases and to carry out robotic docking.

3.2.2.5 Cargo Freighters

Cargo freighters are uncrewed transporters of cargo. They use slow, low-thrust trajectories and therefore require long trip times.

3.2.2.5.1 Vehicle Types

There shall be two vehicle types. One vehicle shall be dedicated to transport cargo from LEO to the Earth Spaceport and to the Astrotel while the other vehicle shall be designed to transport cargo from LEO to the Mars Spaceport.

3.2.2.5.2 Crew Complement

Cargo Freighters shall not be crewed.

3.2.2.5.3 Reusability

Both Cargo Freighters shall be reusable after RRU upon return to LEO at Earth.

3.2.2.5.4 Propulsion

The propulsion subsystem shall be capable of carrying out all major orbit shaping delta-Vs and navigation trajectory corrections.

3.2.2.5.5 Modularity

The power, propulsion and cargo hold systems shall have maximum commonality with similar systems on the Astrotel. In other words, these subsystems shall be designed in modular sizes that can be used on Freighters, Spaceports and Astrotels.

3.2.2.5.6 Power System

The power system shall support propulsion power requirement of TBD [300] kW. Renewable energy storage capability shall provide emergency minimum power requirements for a period of TBD [3] hours.

3.2.2.5.7 System Interfaces

Freighters must be capable of autonomous docking and cargo transfer with Astrotels or Spaceports.

3.2.2.5.8 Refurbishment, Repair and Upgrade (RRU)

RRU will be carried out at the LEO Space Station upon return to Earth.

3.2.2.5.9 Cargo Loading

Freighters shall be loaded with cargo at or near the LEO Space Station (Astrotel/Taxi consumables, Xenon, and RRU hardware) and at the Earth Spaceport (Hydrogen for Mars).

3.2.2.5.10 Autonomy

Freighters shall be highly autonomous in order to carry out continuous propulsion low thrust maneuvers and navigation for hundreds of days and enable autonomous cargo transfer at their destinations.

3.2.2.6 LEO-to-Spaceport Shuttle

The LEO Shuttle is required to transport crew from the LEO Space Station to the L1 Spaceport and return to LEO using aerocapture. This vehicle shall be almost identical to the Taxi. Propellant tanks will be about the same as Taxis since the delta-V from LEO to L-1 is about the same as the delta-V for departing the Earth Spaceport and rendezvousing with the Astrotel.

3.2.2.7 Lunar Water Tanker

The Lunar Water Tanker shall be a reusable vehicle that can be fueled either on the Moon or at the Earth Spaceport. Lunar Water Tanker shall transport water from the lunar surface to the Earth Spaceport and returns empty to the lunar surface.

3.2.2.7.1 Crew Complement

The Lunar Water Tanker shall not be crewed.

3.2.2.7.2 Reusability

The Lunar Water Tanker shall be reusable after RRU upon return to the Earth Spaceport.

3.2.2.7.3 Propulsion

The propulsion subsystem shall be capable of carrying out all major orbit shaping delta-Vs and navigation trajectory corrections.

3.2.2.7.4 Modularity

Power and propulsion systems shall have maximum commonality with similar systems on the Taxis and Shuttles.

3.2.2.7.5 System Interfaces

Lunar Water Tanker must be capable of autonomous docking and cargo transfer with the Earth Spaceport.

3.2.2.7.6 Refurbishment, Repair and Upgrade (RRU)

Major RRU will be carried out at the Earth Spaceport.

3.2.2.7.7 Autonomy

The Lunar Water Tanker shall be highly autonomous in order to carry out water loading, propulsion maneuvers and autonomous water transfer at the Earth Spaceport.

3.2.2.8 Mars Shuttle

The Mars Shuttle shall support crew needs during the very short transit (<2 days) between the Mars Base and the Mars Spaceport. In addition, the Mars Shuttle must perform delta-V maneuvers, perform aero-entry and landing maneuvers within the Martian atmosphere, navigate autonomously during all maneuvers, provide electrical power to its subsystems and carry RRU cargo from the Mars Spaceport to the Mars Base.

3.2.2.8.1 Crew Complement

The maximum crew size for the Taxis shall be 10 people for flight duration of less than TBD [2] days.

3.2.2.8.2 Crew Support

Life support, sleeping areas, food, and communications shall be provided.

3.2.2.8.3 Propulsion

The propulsion subsystem shall be capable of carrying out all navigation trajectory corrections and major orbit shaping delta-Vs up to 5.1 km/s, which is required for the transfer from the Mars Base to the Mars Spaceport. The Mars Shuttle shall be required to maintain space storable propellants at cryogenic temperatures at the Mars Base utilizing external power supplies.

3.2.2.8.4 Aeroassist

The Mars Shuttle shall be capable of aero-entry at Mars.

3.2.2.8.5 Power System

Renewable energy storage capacity will be provided that will be sufficient to provide crew and support systems requirements throughout the cruise period to and from Mars Base to the Mars Spaceport.

3.2.2.8.6 Major System Interfaces

The Mars Shuttle must be capable of autonomous docking with the Mars Spaceport. In addition, the Mars Shuttle shall be interfaced directly into the Mars Base propellant manufacturing plant and act as an element of its storage system.

3.2.2.8.7 Refurbishment, Repair and Upgrade (RRU)

Major RRU of the Mars Shuttle shall occur at the Mars Spaceport. Only minor RRU shall be carried out at the Mars Base.

3.2.2.8.8 Autonomy

The Mars Shuttle shall be highly autonomous to reduce crew workload during crewed mission phases and to carry out robotic docking.

3.2.2.9 Other Vehicles

Examples of other vehicles for which detail requirements have yet to be developed, include:

a) LEO Space Station

b) Earth Surface to LEO Space Station Shuttle

c) Heavy Lift, Low-cost Launch Vehicle

d) Phobos Oxygen Tanker

3.2.2.10 Propellant and Resources Plants

3.2.2.10.1 Mars Surface Plant

The Mars surface propellant plant shall produce all propellants required for the Mars Shuttle to reach the Mars Spaceport.

3.2.2.10.2 Phobos Plant

The Phobos Plant shall produce sufficient LOX for operation of the Taxi and the return of the Mars Shuttle to the Mars surface.

3.2.2.10.3 Mars Spaceport

The Mars Spaceport shall have facilities to receive and store LOX and LH propellants.

3.2.2.10.4 Lunar Water Mine

The Lunar Water Mine shall excavate and extract sufficient water from Lunar soils to operate the Lunar Water Tanker, Taxis and the LEO Shuttle vehicles and to produce LH for transport to the Mars Spaceport.

3.2.2.11 Propellant Augmentation Tanks (PATs)

The Propellant Augmentation Tanks (PATs) enable staging of the Taxi vehicles in order to accommodate natural celestial mechanic variations in Mars-to-Astrotel orbit injection velocities over a 15-year cycle of operation. PATs shall be designed to be fully integrated with the basic Taxi vehicle design. PATs shall be designed to be jettisoned after the propellants within them are used. It is possible that these tanks could be used for storage of propellants at the Spaceport.

4 Advanced Technologies, Systems and Methodologies

4.1 Introduction

A number of advanced technologies, systems and methods have been incorporated in the Mars transportation architecture developed during Phase I. These include the use of cyclic orbits between Earth and Mars, solar ion propulsion, radiation protection concepts, recent crew habitation module developments, aero-assist technology, and advanced space and surface solar photovoltaic power systems. These are all discussed in detail in the following sections.

4.2 Celestial Mechanics

4.2.1 Cyclic Orbits

Cycler orbits are resonant or near resonant trajectories between celestial bodies. Cyclers can be designed to enable sustained human interplanetary transportation through regular encounters with Earth and the target planet or between Earth and the Moon. Several interplanetary cycler orbit concepts have been developed over the last two decades to support sustained Mars operations. Cycler orbits include Aldrin Cyclers (or Up and Down Escalator Orbits), VISIT (Versatile International Station for Interplanetary Transport) orbits, which were both developed in the 1980s [Section 1, Ref. 4, 5, 6], and Semi-cyclers, which were developed recently by Aldrin and Byrnes and also Bishop [Bishop, et. al., "Earth-Mars Transportation Opportunities: Promising Options for Interplanetary Transportation", Paper AAS 00-255, presented at AAS Astrodynamics Conference, March 2000].

4.2.1.1 Aldrin Cyclers

The Aldrin Cycler orbits have a period that is approximately equal to the Earth-Mars synodic period (26 months) and, when the line of apsides is rotated by gravity assist methods (average of about 51.4° each orbit), will enable Earth-to-Mars and Mars-to-Earth transfers every 26 months. Aldrin Cycler orbits come in two types, an Up Escalator and a Down Escalator orbit. The Up Escalator has the fast transfer occurring on the Earth to Mars leg while the Down Escalator is just the reverse. Figure 4-1 illustrates both orbit transfer geometries. Figure 4-2 illustrates an example 15-year propagation of the outbound Aldrin Cycler. When two Astrotels are used, an Aldrin Cycler provides relatively short transit times (~5 months) and regular transit opportunities. However, the planetary encounters occur at high relative velocities and typically, impose harsher requirements on the Taxi craft than other cyclers. Also, as illustrated in Figure 4-2, the Aldrin Cycler requires a substantial orbit correction on 3 out of 7 orbits to maintain the proper orbit orientation. Shown are the impulsive or ballistic delta-Vs that would be required to maintain the proper orbit alignment. These corrections are required because of limitations of flyby altitude during the gravity assist. In the case shown a 200-km flyby altitude constraint has been imposed. Mathematically correct, but impractical subsurface flybys eliminate the need for corrections.

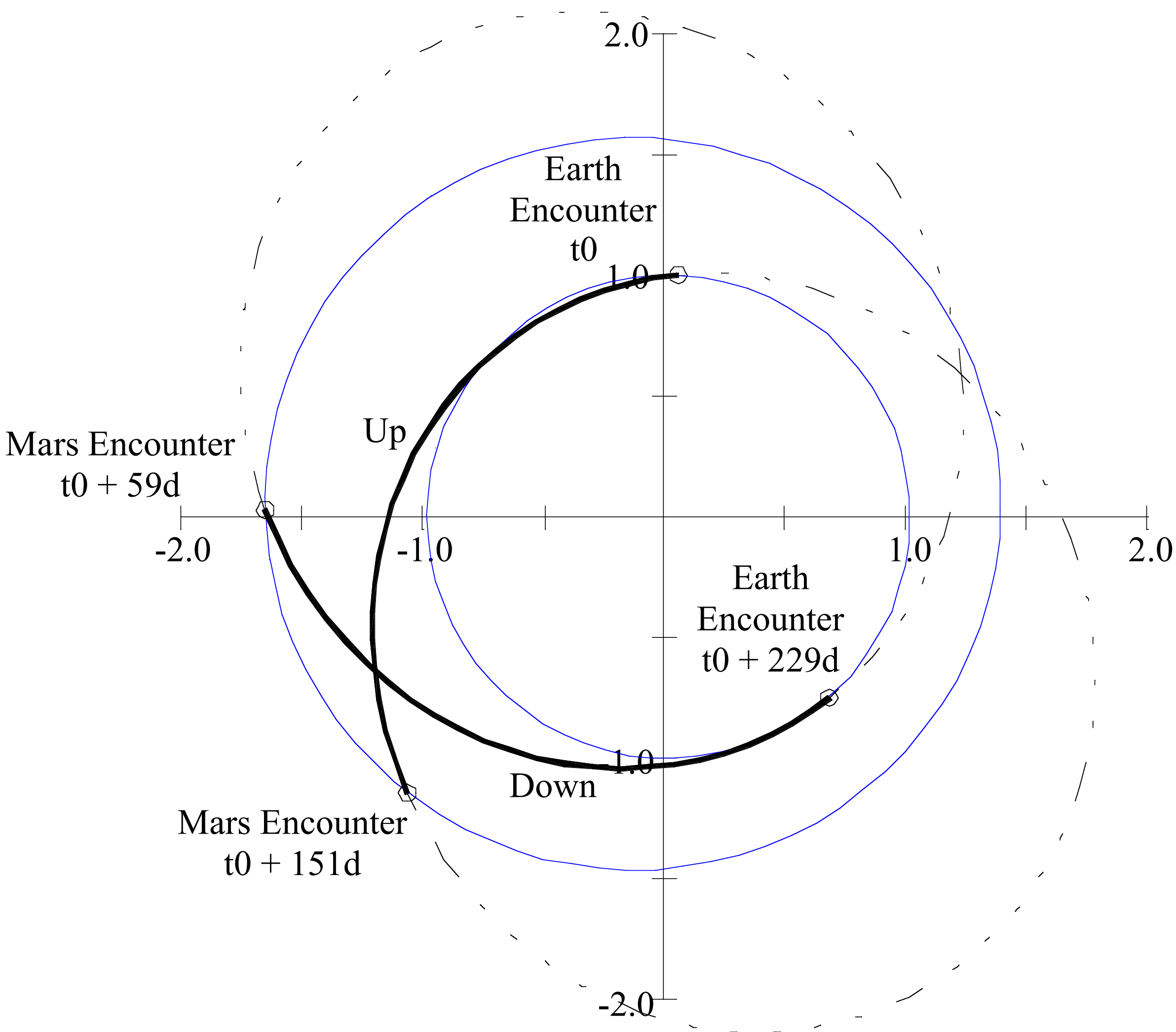


Figure 4-1 Aldrin Up/Down Cyclers

Cycler concepts have an advantage of providing an orbital address for locating large crew transport vehicles. However, because the VISIT orbits can require human flight duration considerably longer than 6 months, they were not considered further in our concept. Because of limited time and resources in the Phase I study recent work on semi-cyclers (with Mars stopover orbits) will be discussed briefly. Earth/Moon cycler orbits also exist but have never been integrated into an interplanetary cycler concept (this is a proposed topic of Phase II). The advantages of Earth/Moon cycler orbits are great because such an orbit between Earth and the Moon could be an attractive location for the Earth Spaceport.

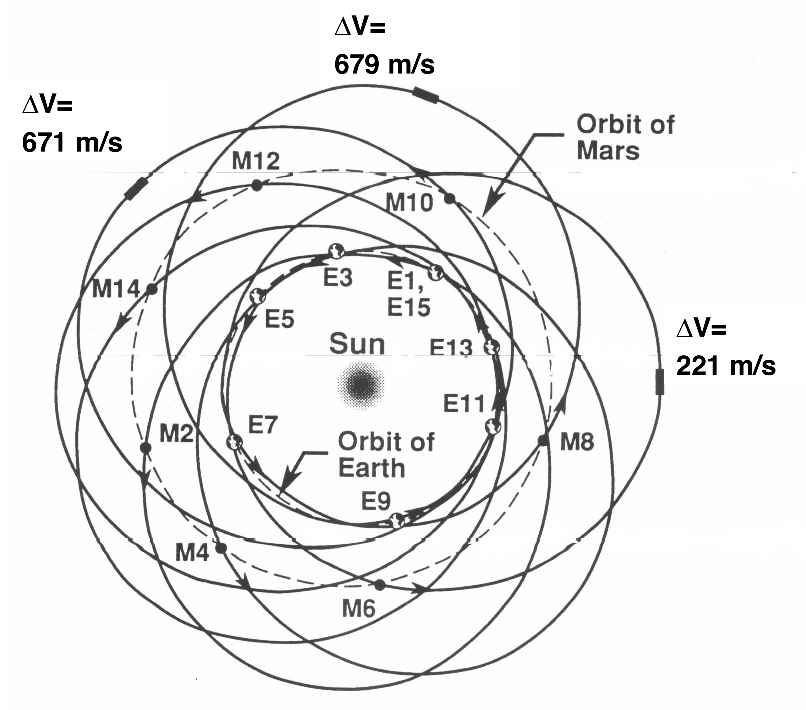


Figure 4-2 15-year sequence of Aldrin Up Cycler Orbits with Impulsive Delta-Vs

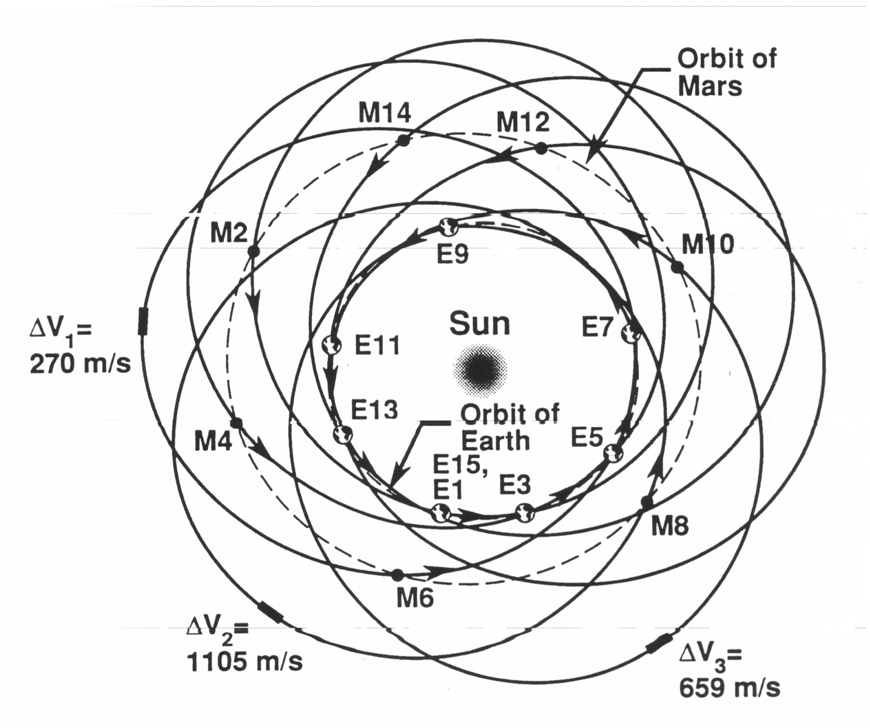


Figure 4-3 15-year sequence of Aldrin Down Cycler Orbits with Impulsive Delta-Vs

4.2.1.2 Semi-cyclers

Work by Aldrin and Byrnes and also Bishop has focused on a "semi-cycler" concept that has repeated Earth swingbys and near-Mars rendezvous. Semi-cyclers include 4 or 5 Earth flybys and cycle duration of 52 or 78 months (2 or 3 synodic periods) but with stopovers at Mars instead of the free return flyby trajectories back to Earth. These new concepts have similar advantages as the original Aldrin cycler concept, including the 5-month trips to Mars and somewhat lower flyby velocities, but they come with disadvantages including more required cycling spaceships or Astrotels, as well as additional propulsion requirements to insert into and depart from a Mars stopover orbit. Some of these disadvantages may be offset by the reduction of Taxi delta-V requirements at Mars.

In the case of the 78-month semi-cycler, 3 Astrotels are required to maintain a continuous flow of Earth-to-Mars or Mars-to-Earth transits every 26 months. These Astrotels do not stopover at Earth in the sense of entering Earth orbit, but rather take 5 consecutive Earth gravity-assist swingbys (each 12 months apart and typically at 15,000 km perigee altitude) in order to time-phase their next insertion unto a favorable Earth-to-Mars trajectory. Mars stopover orbits of about 17-month duration are necessary to time-phase the returns to Earth. The "loose" orbit capture and departure delta-V's can be implemented with solar electric propulsion at a reasonably efficient propellant mass fraction between 5% and 10% depending on the encounter opportunity. Still, the basic Aldrin Up/Down cyclers do not require stopovers and have much lower propulsion requirements to maintain the continuous cycling pattern. As for the taxi vehicles that would transport crew to and from these Astrotels, the semi-cycler concept imposes somewhat higher propulsion requirements (on average) at Earth than does the basic Aldrin cycler, but significantly lower requirements at Mars.

4.2.2 Low-Thrust Trajectory Analysis

Low-thrust trajectories utilizing mass-efficient solar electric propulsion are applied to the cycler concept architecture in three areas: (1) midcourse correction of the Astrotel orbits; (2) round-trip cargo freighters to resupply the Astrotel vehicles; and (3) round-trip cargo freighters to resupply the infrastructure at Mars. Numerical results of this analysis are presented in Section 7.2.2 of this report. The method of analysis and key assumptions used to generate the performance results are briefly described below.

SAIC's version of the Chebytop computer code (originally developed by Boeing in the early 1970s) is used to calculate optimal trajectory solutions with normalized, parametric mass performance data. The program's basis of a 2-body central force field and a direct method of approximate optimization with Chebychev polynomials is generally regarded as accurate enough (within a few percent) for purposes of preliminary mission analysis. SAIC's program called CHEBY2 also allows appending planetocentric escape and capture spirals to the optimized heliocentric trajectory segments of the mission. Round-trip cargo delivery missions are assumed to begin and end in LEO at 1000-km altitude where the transport vehicle is serviced for multiple reuse. The delivery terminal at Mars is the spaceport at Phobos orbit distance. All escape and capture spirals are the circular orbit type and match up with the heliocentric trajectories at parabolic energy conditions ($C_3 = 0$).

The SEP system is assumed to operate at constant specific impulse while thrusting, with coast periods allowed as needed to maximize payload mass (or minimize propellant mass for a fixed payload). Thrust magnitude is proportional to the power input level, which varies with distance from the Sun as determined from a typical solar array power function. Mass performance data are obtained parametrically over a range of specific impulse and initial power-to-mass ratios in order to allow scaling to any desired specifications on payload or vehicle size. Results presented in Section 7.2.2 assume a specific mass of 8 kg/kWe for the combined power and propulsion subsystems of an advanced technology SEP vehicle, and a tankage and reserve fraction of 15% of the nominally required propellant load. Although parametric results were generated over a specific impulse range of 2000-to-5000 seconds to cover the possibility of different types of electric thrusters, the final selection focused on an ion thruster operating at 5000 s and an overall power efficiency of 69%; this choice favored propellant mass efficiency for Astrotel orbit corrections as well as cargo resupply missions.

4.3 Propulsion

4.3.1 LOX/LH Propulsion

Many options exist for the design basis for the LOX/LH engines used in the Mars Transportation architecture. For the purpose of the Phase I study, however, a 7:1 mixture ratio LOX/LH, Pratt & Whitney (P&W) RL-series propulsion system has been assumed. The RL10 engine is an Expander Power Cycle engine, meaning that the Hydrogen is circulated through the nozzle to pre-heat it and provide high-pressure gas power for the fuel and oxidizer turbo-pumps. For typical LOX/LH engines the mixture ratio is 5 or 6:1. Today's Space Shuttle main engines are set at a 6:1 mixture ratio while the P&W RL10, used on Centaur, actually varies its mixture ratio between 5:1 and 6:1 in flight to ensure near simultaneous fuel and oxidizer depletion. The implications of the 7:1 mixture ratio are 1) reduced the hydrogen volume required and thus reduced vehicle size and 2) somewhat reduced thrust level. Nominal parameters of the RL10B-2 engine at 5-6:1 mixture ratios are a thrust of 110.1 kN (24,750 lbs_f), I_{sp} of 464 s, propellant burn rate of 24.4 kg/s and a mass of about 277 kg. Thrust and burn rate of this engine operating at a 7:1 mixture ratio is estimated at 66.7 kN (15,000 lbs_f) and 14.8 kg/s. A nice feature of this engine is its extendable nozzle, which enables it to fit in a volume about half of its deployed state (its length goes from about 4.1 m to 2.1 m long). The following is a drawing of the RL10B-2 engine.

A higher thrust engine is desirable for the Mars transportation architecture studied in Phase I. P&W is considering the development of an advanced RL-series engine called an RL60, having 60,000 lbs. of thrust, roughly three times higher than the RL10. For the purpose of the Mars transportation architecture, a higher thrust engine than the existing RL10 is desired in order to reduce finite burn and gravity losses due to the large, long delta-V burns at Mars. The RL60 engine is about the same form factor as the RL10, the increase performance coming from running the chamber pressure about three times higher.

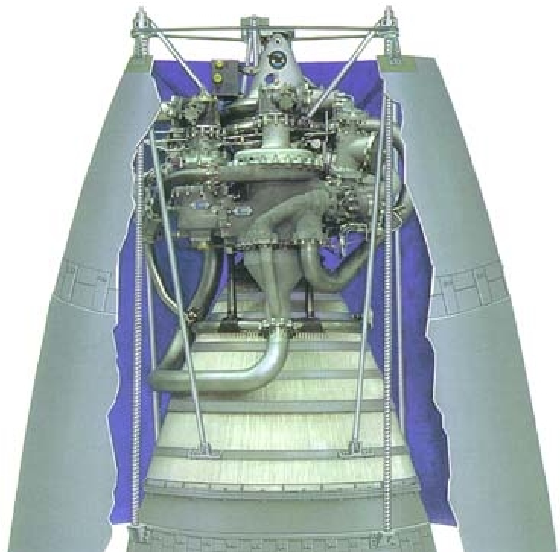


Figure 4-4 Pratt & Whitney RL10B-2 Engine

Considerable engine technology development may be required to enable several hours of reliable operation in remote locations, however, the shuttle main engines were originally designed to operate for about 8 hours over about 55 starts without major overhaul. An RL10 type engine is limited in cyclic life by the number of starts (and to a lesser degree by shutdowns). Typically this limits the engine to a few 10's of firings (e.g. 10-30). For firing duration, the engine for normal expendable space use is limited to ~0.5 to 1.5 hours accumulative firing time. The specific wear issues relate generally to rotating machinery items such as bearings, seals, etc. For a multi-year mission requirement, there are additional considerations that need to be addressed (e.g. slow flowing of static seals, slow degradation of material properties specifically used in the engine due to factors such as radiation).

For the purpose of the Phase I study we assume a LOX/LH engine based on the new RL60 engine with a derating to account for the higher mixture ratio the projected longer burn and longer life requirements. The assumed an engine with the following characteristics:

Table 4-1 LOX/LH Engine Characteristics

Parameter	Value
Mixture Ratio	7:1
Thrust	266.9 kN (60,000 lbs_f)
Specific Impulse	460 s
Propellant Burn Rate	59.2 kg/s
Engine Mass	500 kg

4.3.2 Solar-Powered Ion Propulsion

After more than 40 years of development by NASA an operational solar-powered ion propulsion system (IPS) is now changing the speed of an interplanetary spacecraft on its way to a rendezvous with P/Comet Borrelly in 2001. The spacecraft is JPL's Deep Space 1 (DS1) launched in October of 1998. DS1's IPS consists of a throttling, single 30-cm diameter, 2.5 kW input Xenon ion thruster operating at an exhaust velocity of about 30.4 km/s (I_{sp} of 3100 s) capable of thrust levels from about 21-92 mN. The following figure illustrates the key components of the DS1 IPS.

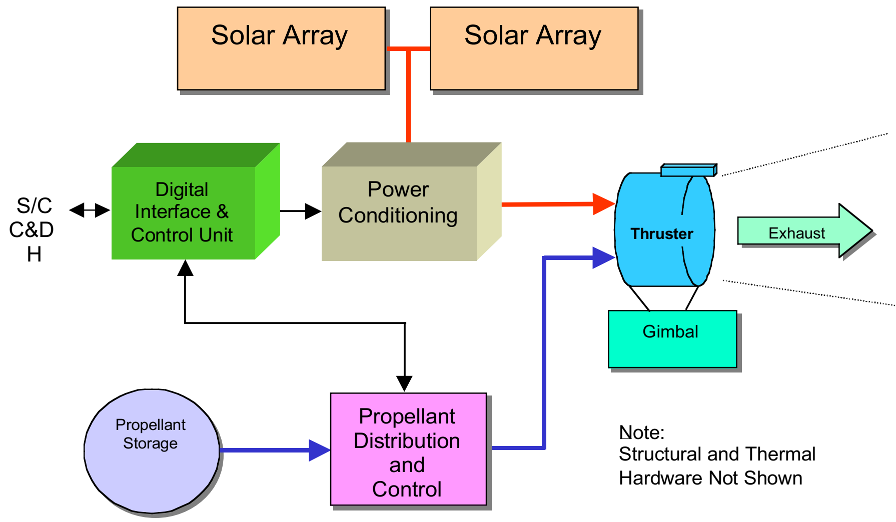


Figure 4-5 Simplified Block Diagram of an IPS (courtesy J. Brophy, JPL)

An ion propulsion system (IPS) converts solar generated electrical energy to momentum of positively charged molecules or ions (since the 1980s noble gases, like Xenon, have replaced Mercury for ion thrusters). This conversion is accomplished by first ionizing suitable atoms by electron bombardment and then accelerating them in the desired direction by using two electrically charged grids. The magnitude of the applied voltage and the charge-to-mass ratio of the ions determines the exhaust velocity. The momentum of these ions reacts against a spacecraft propelling it in the opposite direction. An IPS can be extremely efficient if sufficient solar power is available and a long time is allowed for making velocity changes. The reason for this efficiency can be seen in the "rocket equation". The rocket equation is, $\Delta V = V_e * \ln(m_i/m_f)$, where V_e is the exhaust velocity of the thruster and m_i and m_f are the initial and final mass of the spacecraft. In the rocket equation the delta-V is directly proportional to the exhaust velocity of the rocket engine. If the initial spacecraft mass is only 10% greater than the final mass, (meaning $m_i = 1.1 m_f$), the delta-V capability of the DS1 IPS is nearly 3 km/s. Given the same final mass but a LOX/LH engine with a V_e of 4.5 km/s, more than 90% of the final mass is required to achieve the same delta-V ($m_i = 1.9 m_f$). It is this efficiency that makes an IPS very attractive for making the occasional course corrections required of the Astrotels, providing station-keeping forces for the Spaceports and providing the primary motive force behind the interplanetary cargo freighters delivering consumables, propellants and refurbishment hardware to Spaceports and Astrotels.

4.3.2.1 Status and Plans

The DS1 IPS is obviously a resounding success but considerable technology advance is still needed by the Mars transportation systems under study in this NIAC Phase I effort. Fortunately, there is ongoing NASA research into IPS technology that is moving in the proper direction. The following figure illustrates one possible evolution of IPS technology development.

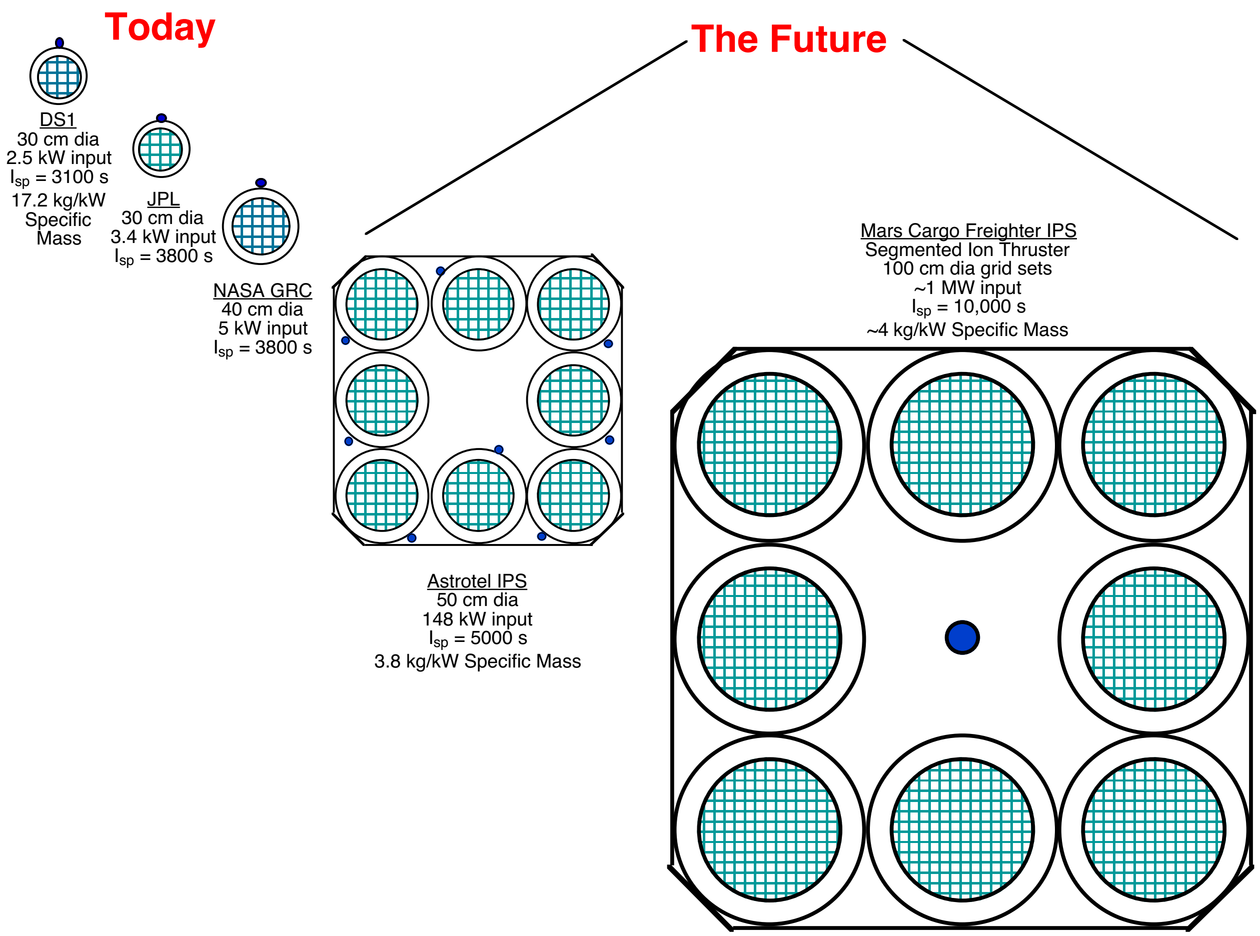


Figure 4-6 Example Evolution of IPS Technology

JPL is working on a modest upgrade of the DS1 thruster which increases the input power to the Power Processing Unit (PPU) by about 35% to 3.4 kW at the same time increasing the specific impulse by about 23% to 3800 s. This thruster design is slated for several possible missions including a comet nucleus sample return. NASA Glen Research Center is working on a 40-cm diameter thruster that also would operate at a specific impulse of 3800 s.

Conventional single engine designs are limited as higher powers are processed and exhaust velocities are increased because of the difficulty in making the accelerator system (thruster span-to-gap ratios, accelerating voltage constraints, and current handling capability). In order to maintain constant power density across a thruster the grid separation must remain constant. Current thrusters have an engine-span-to-grid-gap ratio of about 500 (a 30-cm diameter engine could have its high voltage grids separated by a gap of only 0.6 mm depending on voltage across). As the desire to process more power grows, the engine diameter grows. Assuming practical limits to the electric field between the grids the span-to-gap ratio can eventually grow beyond the state-of-the-art.

Because of the need to process much larger levels of power over larger area thrusters and because of the practical limits to span-to-gap ratios, multiple grid sets are attractive. Multiple grid sets, along with their smaller individual ion source components, electrically connect several grid sets together so that they simulate a larger diameter thruster yet still retaining the desired span-to-gap ratio for each engine segment. Additionally, only one neutralizer is required for the multiple grid set and the smaller individual ion source chambers reduce complexity and other plasma problems. Multiple grid sets per engine can significantly increase the beam current per engine. Such Segmented Ion Thruster (SIT) designs include individual propellant ionization chambers for each engine segment and only one neutralizer for the set of two or more thrusters. Multiple aperture grid ion propulsion is assumed for the Astrotel and Cargo Freighter propulsion systems because of the advantages discussed above.

4.3.2.2 Astrotel IPS

There are several system options for the Astrotel IPS depending on technology advance.

4.3.2.2.1 Upgrade DS1 Engine Technology

Upgrade of DS1 single engine technology from a 30-cm diameter, 2.5 kW, 3100 s specific impulse, and 19.2 kg/kW specific mass system to a 50-cm diameter, 17.2 kW input, 5000 s specific impulse, and 3.8 kg/kW specific mass system. This appears to be a modest improvement in technology, especially since 30-cm diameter engines have already been run at 20 kW input power about 15 years ago, though not with the lifetime capability required for the Astrotel system. Given the requirements of the Astrotel IPS, J. Brophy at JPL generated a model of this engine. The model inputs and resultant performance are presented in the following tables.

Table 4-2 Astrotel Ion Engine Performance Input Data

ION ENGINE PERFORMANCE INPUT DATA	
Parameter	**Value**
Engine Type	RING CUSP
Propellant (AMU)	131.3
Beam Diameter (cm)	50
Specific Impuse (s)	5000
Max. Span-to-Gap Ratio	500
Minimum Grid Gap (mm)	0.6
Max. E-Field (V/mm)	2600
Max. Disch. Current (A)	100
Max. Beam Current (A)	6.75
Maximum R-Ratio	0.9
Minimum R-Ratio	0.55
Perveance Coef. Xe x10E9	2.48
Perveance Exponent	1.5
Screen Grid Tranparency	0.75
Beam Flatness Parameter	0.6
Divergence Thrust Loss	0.98
Double Ion Ratio	0.1
Discharge Voltage (V)	28
Disch. Chmbr Prop. Eff.	0.92
Discharge Loss (eV/ion)	180
Keeper Current (A)	0
Keeper Voltage (V)	4
Coupling Voltage (V)	15
Neut. Keeper Current (A)	2
Neut. Keeper Voltage (V)	15
Neut. Flow Fraction	0.05

Table 4-3 Model-Estimated Performance of Astrotel Ion Engine

CALCULATED ION ENGINE PERFOMANCE	
Parameter	**Value**
Thrust (N)	0.515
Engine Input Power (kW)	17.23
Total Engine Efficiency	0.733
Thrust-to-Power Ratio (mM/kW)	29.91
Beam Voltage (V)	2353
Total Voltage (V)	2615
Net-to-Total Voltage Ratio	0.9
Beam Current (A)	6.75
Discharge Current (A)	43.38
Grid Gap (mm)	1.006
Actual Span-to-Gap Ratio	497.2
Screen Hole Diameter (mm)	3.35
Effective Acceleration Length (mm)	1.95
Maximum Beam Current Density (mA/cm2)	5.73
Average Beam Current Density (mA/cm2)	3.44
Double Ion Thrust Loss Factor	0.97
Total Propellant Efficiency	0.87
Total Propellant Flow Rate (g/s)	0.01052

The overall ion propulsion system mass breakdown based on the upgrade of the DS1 IPS is shown in the following table.

Table 4-4 Astrotel IPS Mass Breakdown

Conventional Approach with Redundancy -- XFS				
Item	**QTY**	**Unit Mass, kg**	**Total Mass, kg**	**Comments**
Engine	8	18.40	147.2	
PPU	8	36.40	291	
DCIU	2	3.00	6	1 required plus one spare
Regulator	2	0.45	0.90	
Service Valve - HP	1	0.01	0.01	
Service Valve - LP	17	0.01	0.17	
Pressure Transducer	2	0.25	0.50	
Latch Valve - HP	2	0.10	0.20	8.71
Latch Valve - LP	8	0.10	0.80	
Filter	1	0.13	0.13	
Var. Reg. With Flow Meter	24	0.15	3.60	
Tubing	1	2.00	2.00	
Fittings	1	0.40	0.40	
Gimbal	8	5.52	44.16	One for each thruster
Misc. Thermal (5% of dry mass)	1	24.86	24.86	
Cabling (5% of dry mass)	1	24.86	24.86	
Structure (4% of dry mass)	1	21.88	21.88	
Total			***569***	

The summary assumptions and description for this option for the Astrotel IPS is shown in the following table.

Table 4-5 Summary Assumptions and Description for Astrotel IPS

Assumptions		
1 150 kW System Input Power		
2 2800 kg total propellant processed		
3 Xenon propellant		
4 Gridded ion engines		
5 Specific Impulse = 5000 s		
PROPULSION MODULE SUMMARY		Notes
Number of Engines	8	
Number of Operating Engines	8	
IPS Thrust (N)	4.12	
IPS Input Power (kW)	148.2	
Engine Input Power (kW)	17.23	
Engine Thrust (N)	0.515	
Engine Unit Mass (kg)	18.4	
Gimbal Mass (kg)	5.5	
Propulsion Module Cabling Mass (kg)	24.9	
Xenon Feed System (kg)	8.71	
Xenon Tank Mass (kg)	51.3	Assumes a tank sized for 1027 kg of xenon
Propulsion Module Structure (kg)	21.88	
Propulsion Module Dry Mass (kg)	**272**	Not including PPUs
Propulsion Module Specific Mass (kg/kW)	1.8	
POWER PROCESSING SUBSYSTEM SUMMARY		
Number of PPUs	8	
PPU Specific Mass (kg/kW)	1.96	
PPU Unit Mass (kg)	36.4	
PPU Efficiency	0.93	
Radiator Area per PPU (m2)	3.1	
Total PPU Mass (kg)	**291**	
Total PPU Radiator Area (m2)	24.8	Radiator mass not included
Number of DCIUs	2	
DCIU Unit Mass (kg)	3	
Total IPS Mass (kg)	***569***	
Total IPS Specific Mass (kg/kW)	***3.84***	Note the total NSTAR IPS specific mass is 19.2 kg/kW

The Xenon tank was sized for about one third the Xenon required over a 15-year cycle of operations. Note that the system specific mass is 3.8 kg/kW, a factor of over 4-times improvement over the current DS1 IPS. Such an improvement is projected to be possible in the 2010 timeframe.

4.3.2.2.2 Alternative Astrotel IPS

Instead of having eight individual engines, an eight-set segmented ion thruster (SIT) could be employed. This approach might offer some simplification and hardware part reduction since there would only be one neutralizer and one high voltage power supply.

If the 100-cm diameter grid set operating at 125 kW input could be developed, a throttled 2 grid-set SIT could be employed for the Astrotel IPS and an eight grid-set SIT could be used for the Mars Cargo Freighter.

4.3.2.3 Cargo Freighter IPS

For the low thrust trajectory analysis and the system definition of the Astrotel and Mars Cargo Freighter vehicles we assumed a propulsion specific mass of 4 kg/kW, consistent with the estimated performance in 2010. In addition, we are also assuming a power system specific mass of 4 kg/kW, for a total power and propulsion system specific mass of about 8 kg/kW. The Astrotel and Mars Cargo Freighter input power requirements are expected to be much larger than the Astrotel requirements between 300-900 kW. This large size will likely require SIT configurations. For the purpose of the low thrust trajectory analysis, a specific impulse of only 5000 s was assumed. If a specific impulse of 10,000 s becomes a reality for a SIT system of this size, significant improvement in performance will occur. The details of these designs are deferred to Phase II.

4.3.2.4 References

1. Brophy, J., “Near-Term, 100 kW-Class Ion Engines”, AIAA Paper # 91-3566, AIAA/NASA/OAI Conference on Advanced SEI Technologies, Cleveland, OH, September 1991.

2. Brophy, J., “Ion Propulsion System Design for the Comet Nucleus Sample Return Mission”, AIAA Paper # 2000-3414, 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Huntsville, AL, July 2000.

3. Polk, J., et. al., “In-Flight Performance of the NSTAR Ion Propulsion System on the Deep Space One Mission”, IEEE paper, 2000.

4.4 Radiation Protection

Interplanetary radiation includes high energy Galactic Cosmic Radiation (GCR), which is very difficult to protect against, and solar flare particle events (SPEs), which consist mostly of high-energy protons. GCR is continuous while SPEs only occur during major solar storms. SPEs can last from minutes to hours. It is necessary to provide protection against SPEs because this radiation can be quite harmful and can cause death for unprotected humans. A major SPE, had it occurred with Apollo astronauts on the moon, would likely have killed them. The effect of GCR is expected to result in a small increased risk of cancer over the crew times usually considered. The following figure describes one model for the shielding required of various materials in order to reduce cell damage (transformation) by a particular amount for a one year exposure (in NRC report on Radiation Hazards to Crews of Interplanetary Missions, 1996).

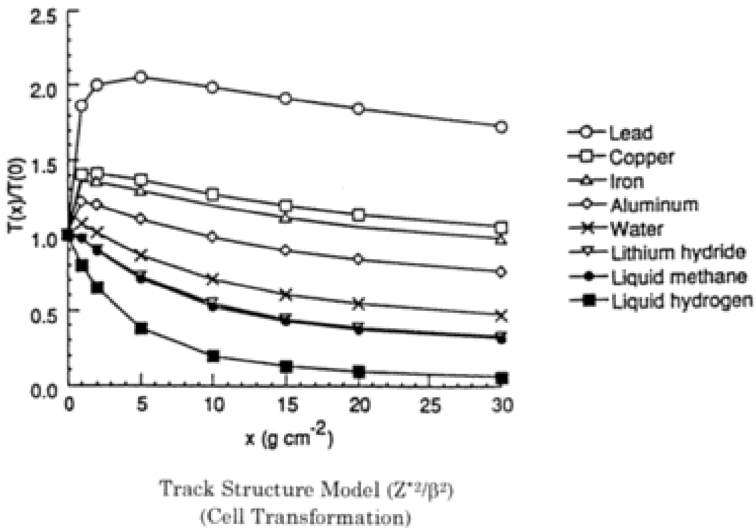


Figure 4-7 Required Shielding of Various Materials

The effect of high Z materials for shielding against radiation is counter-intuitive. Lead shielding actually increases the dose of damaging radiation as compared to no shielding at all. This increase is due to more particles being generated as the result of collisions in the shielding material. Very thick layers of lead are required for any protection at all. In discussions with SAIC personnel at JSC, current reference mission planning assumes the use of available water and water-bearing food stuffs in the transfer vehicle cargo for protection. In the future, there may also be consideration of an onion skin approach to shielding materials, which could be added over time. Substantially shielding of crew sleeping quarters, where crews will spend a significant amount of their time, can significantly reduce the overall GCR dose.

Liquid hydrogen is a very good shielding material for SPEs, a 30-cm thickness reducing the cell damage by an order of magnitude below unprotected cells. Of course water is easier to store at room temperature, but it only reduces cell damage by 50% with the same thickness of shield. It is clear that an SPE storm shelter of some kind will be required on the Astrotel and the Taxi vehicles because the protection of the entire vehicle will likely be prohibitively massive. The best way to protect the crew against GCR dose is to limit flight times in interplanetary transit to the shortest practical values.

4.5 TransHab Crew Module

4.5.1 Introduction

The TransHab Module is an inflatable home in space being developed for the International Space Station (ISS). We assume a modified TransHab Module is used for the Astrotel crew habitation. In addition, it has been considered as a habitability module for future human missions to Mars and as a possible hotel for tourists to visit in Earth orbit. As currently designed the TransHab is a home to a crew of 6 astronauts on board the space station, which includes sleeping compartments, food preparation and eating facilities, windows, exercise gym and food storage areas. The concept for TransHab originated in 1997 at NASA JSC during studies of future Mars missions. Development has continued and has included vacuum chamber testing in late 1998. See http://spaceflight.nasa.gov/station/assembly/elements/transhab for more information.

4.5.2 TransHab Design

The TransHab is an inflatable structure so as to allow it to be launched in a smaller stowed volume and inflated once on orbit. This approach enables a much larger volume per crew than otherwise would be available from rigid structures that can be launched into space. The 30-cm thick shell is composed of several layers of differing materials to provide the maximum protection to the crew form orbital debris and meteoroid impacts. The multi-layer meteoroid and orbital debris (MOD) design facilitates particle break up before penetration of the envelope. The shell consists of include several sheets of meteoroid and orbital debris shielding, a Kevlar restraint layer and several layers of redundant pressure-retaining bladders. The length of the TransHab is 11 m including pressurized air-lock and equipment tunnel and its diameter after inflation is 8.2 m. The total enclosed volume is about 340 m^3. At launch, as designed for the ISS, it is 13.2 mt. The central core of the TransHab is a lightweight structure made from carbon-fiber composite materials. This structure provides the base for the three floors and several compartments. A central tunnel provides access between the three floors of crew space and the pressurized air-lock to the docking port. Extendable floors are unfolded and erected after shell inflation. In the ISS design, an integral water storage volume surrounds the crew sleeping quarters to provide protection from high-energy charged particles. A pressurized docking cone is also included on one end of the ISS design. The following figure illustrates the ISS TransHab with crew.

Figure 4-8 TransHab Design Cutaway (courtesy NASA/JSC)

At the very top of this figure is the pressurized tunnel. Next down is Level 3 which is the Crew Health Care floor that contains exercise equipment, exercise area and soft material stowage space. Level 2, next below, is the crew quarters and environmental controls and life support equipment areas. The crew quarters are individual spaces surrounding the central access tunnel. Surrounding the crew quarters is an integral water tank that provides additional protection to the crew against high-energy particle impacts while they are sleeping. Level 1 is where the galley and wardroom are located, which includes food storage, preparation, eating and clean up facilities. At the very bottom of the TransHab, as seen in this view, is the unpressurized "tunnel" where the inflation systems and tanks are contained.

4.5.3 Adaptations for Astrotel and Spaceport Application

Several adaptations will be required for use of the TransHab design for the Astrotel and Spaceport concepts. Some of the more obvious modifications will likely include:

- Expansion of crew quarters from 6 to 10. This may require reducing the currently available space or carving out new space from the health or galley areas.
- Expanding the radiation protection volume to include most of the volume occupied by the crew. This could include an internal bladder inside the current inside dimension of the TransHab or it could include filling one or more of the current outer shell bladders or MOD volumes with protective material such as water or polyethylene.
- Replacement of the unpressurized "tunnel" with another docking port for attaching pressurized cargo bays. Once the system is inflated, this equipment is not necessary.
- Inclusion of a command and control area, perhaps replacing storage areas, to provide space for electronics and interplanetary communications systems.

4.6 Aeroassist Technology

4.6.1 Introduction

There are four aero-assist technologies of current interest to mission designers: aerocapture, aerobraking, aero-gravity assist, and ballute aero-assist. A general reference for the first three is Ref. 1. Aerocapture is where a vehicle is in the configuration of an entry body and enters a planetary atmosphere at high speed and loses velocity in one atmospheric pass, exiting at a reduced speed consistent with a low orbit. A required sequel is a burn at the first apoapsis to raise the periapsis. The aero-maneuver in the atmosphere is controlled by a predictor-corrector closed algorithm loop, and the conventional method is to roll a lifting vehicle around the velocity vector to control lift while maintaining drag. Aerocapture often begins with direct approach of the vehicle at hyperbolic speed, but can also be approach from a long period orbit, as in the return of the Apollo spacecraft from the Moon. A closely related maneuver with a similar control loop is accurate landing, where the descent path is made to follow a predetermined path by comparing the measured flight parameters with stored nominal parameters. Examples of aerocapture/controlled descent are the Apollo flights from 1969 on, the AMOOS studies by Boeing in the seventies, the Viking entries from orbit at Mars in 1976, and the future reentry of the Stardust comet sample return capsule to land in Utah. A reference for the guidance principles is Ref. 2.

Aerobraking is the name given to a gradual orbit decrease due to repeated passes through a planetary atmosphere. Here the spacecraft need not have a heat shield, but may have one to allow deeper incursions in the atmosphere, thus expediting the orbit decay. The first example was a period of several months decay in Earth orbit of the Atmospheric Explorer in 1972. An early detailed analysis of pressure, heating and orbit sequence was given in Ref. 3. Other examples are Earth atmosphere passes of the Muses A spacecraft in 1993, with a heat shield, and the aerobraking of the Magellan spacecraft at Venus in 1994, without a heat shield, and of the Mars orbiter at Mars in 1997.

Aero-gravity Assist is the name given to the maneuver where a high lift/drag vehicle flies through a planetary atmosphere at hyperbolic speed, using downward lift to balance centrifugal force, and emerges in a direction aligned with the heliocentric velocity of the planet. Thus the vehicle gains energy relative to the Sun, although it loses energy due to aerodynamic drag. The vehicle envisioned for this role is the waverider [4 and 5].

Ballute Aerocapture has appeared in the last few years as a promising new method of aerocapture in a planetary atmosphere. In this method a large inflatable light-weight drag device is inflated behind an orbiter prior to entry. It is so large that the convective and radiative heating rates are low enough to be radiated at temperatures not exceeding about 500 C, which some ballute materials, e.g. Kapton, can withstand. The balloon, pressurized to at least the peak stagnation pressure during the aeropass, is encased in a net of stronger material, like PBO, to take the substantial g-load. The delta-V speed loss is measured by onboard accelerometers, and the ballute is released when enough delta-V has been lost. The orbiter alone incurs relatively little speed loss in the remaining part of the aeropass, so that the exit speed can be achieved with acceptable accuracy. The method has the advantage that the orbiter need not have a heat shield other than MLI, and is not encased in an entry vehicle, considerably easing the design. The low mass of the ballute facilitates future planetary missions such as a Venus Sample Return, a Saturn Rings Mission, a Titan Entry Probe, and a Neptune Orbiter. The ballute can also be used for a lander, and has some interesting possibilities. For example, during a direct entry into the Mars atmosphere a ballute could first release an orbiter and then land a lander. The orbiter would be available for telecommunication virtually at once. As for entry body aerocapture, a periapsis raise maneuver is required at the first apoapsis.

4.6.2 Status and Current Plans

These aeroassist technologies were placed on a NASA Aero-Assist Roadmap in January 1997, ballute aerocapture being added in April 1999, and are in line for funding as new technology money becomes available. The NASA centers have assigned priorities favoring conventional technologies and flow field and computing aspects of technology development. Several NASA centers have developed algorithms for aerocapture, and at one time it was proposed to aerocapture the 2005 Mars orbiter. Perhaps because of the failure of the two Mars 98 spacecraft, that plan was set aside, but a recent proposal by JPL is for a demonstration secondary payload at Mars on the 2005 vehicle, the demonstration being a closed-loop remote autonomous aerocapture maneuver. However, the mainstream thought at JPL, the customer for interplanetary missions in the short term, is a ballute flight test in the Earth's atmosphere, because of the many future missions that become feasible only if the ballute passes a flight test. It is likely that both a ballute test and a general rigid body aerocapture closed-loop test will be implemented at Earth in the next few years, especially if an Earth test can be designed to simulate aerocapture given the uncertainties for Neptune, Titan, Saturn, etc. For example, the solid Earth entry vehicle would be given a deliberate error in the expected trajectory, and the correction maneuver developed on board would try to correct the trajectory to meet the desired exit speed. The vehicle would be recovered if possible to provide information on ablation etc. The ballute entry vehicle would be given an exit speed and set on an entry path, and the ballute would be released by onboard guidance seeking to meet the required exit speed. The ballute orbiter would also be recovered if possible.

The next step proposed for Aero-Gravity Assist is to develop software to deal onboard with measured decelerations to decide when the vehicle should exit the atmosphere in order to rendezvous with the next planet. This will require onboard computing capability and star tracking so that the vehicle exits close enough to a speed and direction computed to be about correct for the next rendezvous, even if this differs from nominal. The point is that only small corrections can be implemented, large corrections requiring too much propellant.

Aerobraking is routinely considered for orbit shaping by mission designers and will be implemented periodically if a spacecraft needs to wait in orbit for conditions to change, for example. Otherwise the time taken for aerobraking, usually a few months, is a deterrent in respect of increased cost and a delay in the spacecraft reaching operational readiness.

Studies of ballutes in the late 1960s for Viking and later in the 1970s and 1980s assumed the ballute was deployed late in the entry phase. The result of late deployment was the requirement for a very robust and heavy ballute, therefore interest in ballutes waned due to their lack of performance advantage. Recent studies at JPL and elsewhere have assumed a mode where the ballute is deployed prior to entry, which significantly reduces the forces on the envelope during the deployment process. NASA Ames and Langley Research Centers have confirmed aeroassist trajectory analysis and heating rates of the new ballute deployment mode. Of course, until flight tests are performed, it is appropriate to be skeptical until the packaging and deployment of suitable thickness envelopes and materials have been developed and their performance confirmed. JPL is leading a ballute development program, focused on robotic missions, that favors studies of flow and heating parameters and concurrent development of materials, packaging, deployment and eventual flight testing. The NASA New Millennium Program is considering flight test possibilities including deployment tests on sounding rockets, Delta and Ariane launchers, as secondary payloads, or on a dedicated launch of a small launch vehicle.

4.6.3 Taxi Aeroassist Design

Aerocapture is planned for use at both at Earth and at Mars. The entry speed at Earth is modest, relative to the NASA Stardust Mission sample return vehicle for example, and the delta-V to be lost is consistent with a relatively short-duration aerocapture flight. At Mars, the entry speed is much larger than the exit speed desired, so that the aerocapture vehicle has to cruise around the planet for a relatively long period. It is shown below that a vehicle with relatively high lift-to-drag L/D ratio is required at the start of the cruise in order to supply the required centripetal acceleration and to stay under a total g-load of about 5. The vehicle described in Ref. 6 has an L/D of 0.63, which is not enough. This vehicle is known as an elliptical raked cone. A similar vehicle reported in Ref. 7 has higher L/D (over 1) at higher angle of attack, so that it is likely that the Ref. 6 vehicle can generate higher L/D than reported. It is planned to investigate aero-assist and supplementary techniques that may be used to minimize the peak g-load, including an engine burn prior to entry to reduce the entry speed, and use of a ballute early in the entry to generate high aerodynamic drag at high altitude. In the later sections we examine the effects of changing the mass and area of the entry vehicle and of the lift-up maneuver prior to the aero-cruise phase. We also look at using an engine burn at the start of cruise, to augment the lift force and to reduce the total g-load.

The present plan at Mars is for the vehicle to enter in the maximum drag attitude to reduce the speed as much as possible during the descent. As the vehicle will have to fly at the maximum downward lift attitude in the cruise it will have to change attitude as it approaches the cruise altitude, in fact varying the angle of attack to keep the total g-load below the agreed value, say 5 g. At the critical start of aero-cruise it may be necessary to augment the downward lift with an engine burn, and it is shown below that an engine burn can also reduce the horizontal deceleration and thus reduce the total g-load. The cruise altitude is determined by the maximum lift, and the mass and area of the vehicle.

A preliminary examination of the 15-year set of the Earth-Mars Aldrin Up/Down Cyclers indicates a maximum entry speed of 12.7 km/s at Earth, with only a small variation with date. At Mars the maximum entry speed, V_e is at most 12.5 km/s, and is considerably less for some dates. At Earth the exit speed for an entry with constant upward lift is about 10.8 km/s, only about 1.7 km/s less than the typical entry speed. This low aerodynamic delta-V implies that there may be only a short aero-cruise period required at Earth where lift is used to keep constant altitude while drag reduces the speed.

On the other hand, at Mars, preliminary examination indicates that there will generally be an extensive aero-cruise phase in the atmosphere. A gradually decreasing downward lift is used to maintain constant altitude during this phase, until at a speed of about 5.5 km/s upward lift is commanded in order to exit with about the desired speed of 4.5 km/s. For Earth, the entry angle at 125-km altitude is about -6 deg, while it is about -10 deg at the same altitude for Mars. The aeroassist vehicle design being considered for the Taxi vehicle is described in Ref. 6 as the Aerobraking Orbit Transfer Vehicle (AOTV). The AOTV was sized for a smaller payload than is being considered for the current Taxi. The AOTV has a C_d of about 1.6 and a C_l of 0.48, i.e., L/D of 0.3 at angle of attack (AOA) zero deg, and has maximum L/D of about 0.63 at AOA -20 deg (C_d = 0.95 and C_l = 0.6). For the vehicle in Ref. 6, with diameter 12 m and mass 13,200 kg (dry mass 6,800 kg + fuel) the m/C_dA is 72.9 kg/m^2 at an angle-of-attack (AOA) of zero and 123 kg/m^2 at AOA -20 deg. At AOA +20 deg the C_d is about 1.75 (m/C_dA = 60 kg/m^2), and L/D = 0 and there is no lift. The Taxi has an entry mass of about 16,000 kg, and the assumed diameter is also 12 m. A larger aeroshell may be necessary depending on configuration studies in an attempt to place all permanent propellant tanks behind the aeroshell.

4.6.3.1 Mars Entry Strategies

A number of aero-assist strategies exist that can reduce the crew g-loads during the aerocapture maneuver. These strategies include 1) the baseline of descending into the atmosphere at a shallow angle and using drag to reduce velocity before reaching aero-cruise altitude, 2) a steeper entry angle to bleed off more speed prior to aero-cruise, 3) the use of a ballute device at entry, 4) the reduction of velocity by propulsive means before entry, and 5) propulsive thrusting in the velocity direction (which is very counter-intuitive) during the initial aero-cruise phase.

4.6.3.1.1 Baseline

The baseline Mars entry strategy is to descend into the atmosphere at a shallow angle (~-10°) and use maximum drag to reduce the speed as much as possible before reaching the cruise altitude. The initial cruise demands flight at maximum L/D, and the transition from maximum C_d to maximum L/D has to be implemented without exceeding the acceptable g-load.

To generate high lift the vehicle must also have high drag, and the cruise altitude is where the total g-load due to both drag and lift is acceptable, generally about 50-km altitude. If the maximum L/D is not enough the engine must be burned briefly to provide the required vertical force. In a short time the aero lift will suffice alone.

Thereafter the strategy is to cruise with downward lift until reaching the point where upward lift is again used to exit at the required orbital speed (about 4.0-4.5 km/s). The resulting aerodynamic delta-V is large, about 8 km/s for the maximum entry speed of 12.5 km/s. The critical point is to stay under the maximum acceleration allowed, say g-loads of 5 to 6. For example, if we level off at $V_{horizontal}$ = 11.9 km/s (for V_e = 12.5 km/s) and have a drag deceleration of 4 g, the downward acceleration (DA) required to stay level is DA = V^2/R - 3.71 m/s (Mars g_0). Taking R as 3397 km and adding 50 km or 3,447,000 m, we have DA = 38.37-3.71 = 34.66 m/s^2, or about 3.53 g. With drag at 4.0 g and required lift at 3.53 g, the total deceleration is 5.34 g, and the L/D needed is 0.88, which the current Ref. 6 vehicle design will not provide. Thus, there are two issues: staying under about 5 g total, and generating enough downward lift.

It may be that some of the high V_e cases at Mars will require drag augmentation and/or a higher allowable g-load on the crew for that opportunity. Table 4-6 was generated to examine what maximum g and L/D are required. An altitude of 50 km is assumed, and the $V_{horizontal}$ /V_e is an empirical fraction of V_e, the entry speed. It is assumed that the vehicle guidance arranges the entry angle and initial lift to level off at a speed and altitude consistent with keeping L/D and g_{drag} at the values shown.

Table 4-6 Launch Year Variations of Aeroassist Parameters vs. g-Load and L/D

Year	2012	2014	2016	2018	2020	2023	2025
V_{inf}, km/s	10.14	11.45	11.49	8.91	5.68	7.14	8.16
V_e, km/s	11.28	12.47	12.50	10.18	7.53	8.68	9.53
$V_{horizontal}/V_e$	0.930	0.952	0.952	0.920	0.895	0.910	0.920
$V_{horizontal}$, km/s	10.49	11.85	11.90	9.37	6.74	7.90	8.77
Required g_{lift}	2.87	3.77	3.81	2.22	0.96	1.46	1.90
Allowed g_{drag}, For 5g total limit	4.09	3.28	3.24	4.48	4.91	4.78	4.63
L/D needed	0.70	1.15	1.18	0.50	0.20	0.30	0.41
Allowed g_{drag}, For 6 g total limit	5.27	4.67	4.63	5.57	5.92	5.82	5.69
L/D needed	0.55	0.81	0.82	0.40	0.16	0.25	0.33
Allowed g_{drag}, For 7 g total limit	6.38	5.90	5.87	6.62	6.93	6.85	6.74
L/D needed	0.44	0.63	0.64	0.34	0.14	0.21	0.28

It can be seen that L/D for the 5-g limit is above the 0.63 potential of the current Taxi vehicle design for the dates 2012, 14 and 16. (1.15 and 1.18 are needed for 14 and 16). For the 6-g total limit cases 14 and 16 need too much L/D (about 0.8), and for a 7-g limit the current Taxi vehicle design would just cover all cases. A question one might ask is: can the $V_{horizontal}$ be reduced in some way, e.g. by changing the entry path or by using a ballute. This comes down to reducing the speed without reaching a lower altitude, and two strategies seem possible: a) a steeper entry with zero lift to lose speed at full allowed g initially, then trim to less C_d and some upward C_l to level off at a speed as low as possible; and b) deploying a ballute in the early entry where the dynamic pressure is low and the drag of the vehicle alone is low. The following figure illustrates the typical aerocapture maneuver at Mars.

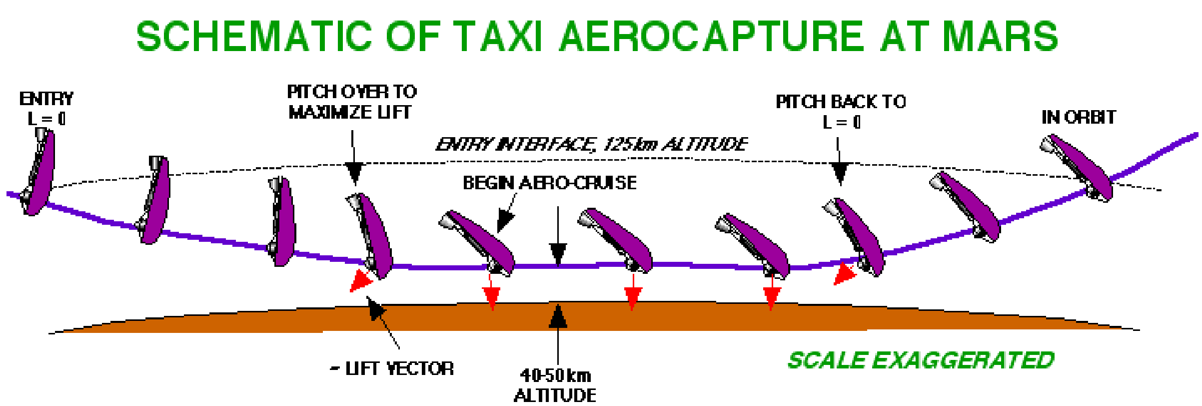


Figure 4-9 Taxi Aerocapture Profile at Mars

The aerocapture parameters for a typical baseline case are presented in the following figures. The case shown illustrates g_{lift} and g_{drag} decreasing with time, for aero-cruise at a near constant altitude. During the high drag entry the angle of attack (AOA) is at +20° (C_D=1.8, L/D=0). For the aero-cruise, the AOA is –20° (C_D=0.95, L/D=0.63). Up until 72 s there is no lift on the vehicle. At 72 s into the entry the active control system begins to pitch the vehicle over to begin lifting. At this time the total g-load again rises to the 5-g limit. Rolling the Taxi vehicle about the velocity vector alters the vertical lift component thus controlling altitude. Another option, not shown, is to perform aero-cruise at a constant dynamic pressure, implemented by a slow descent as the speed decreases. This option gives a higher g-load and heating rate but for a shorter time.

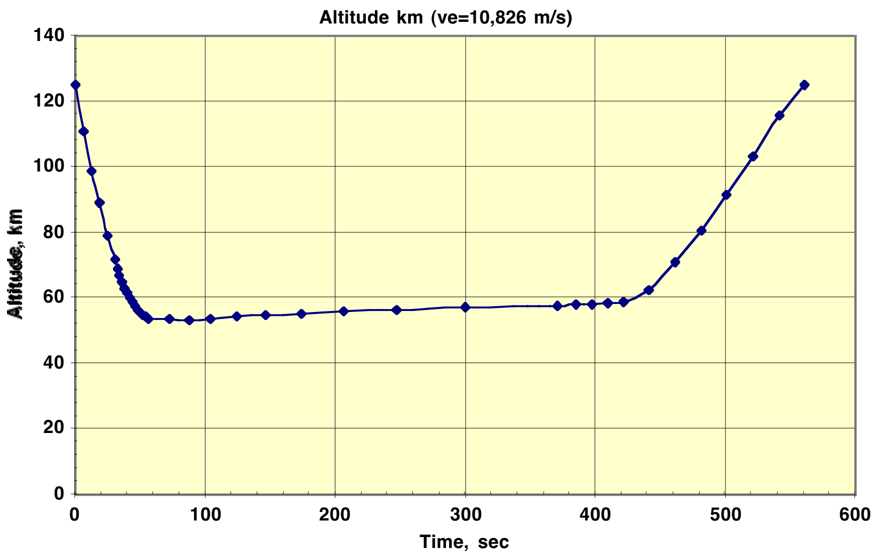


Figure 4-10 Taxi Altitude vs. Time

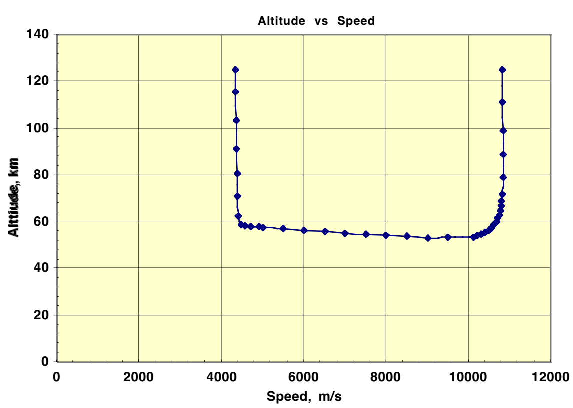


Figure 4-11 Baseline Aerocapture Speed vs. Altitude

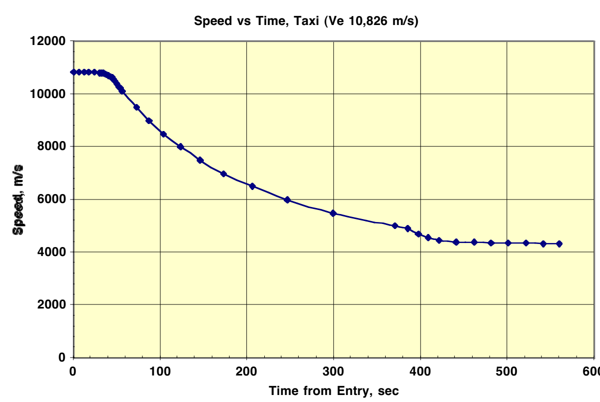


Figure 4-12 Baseline Aerocapture Speed vs. Time

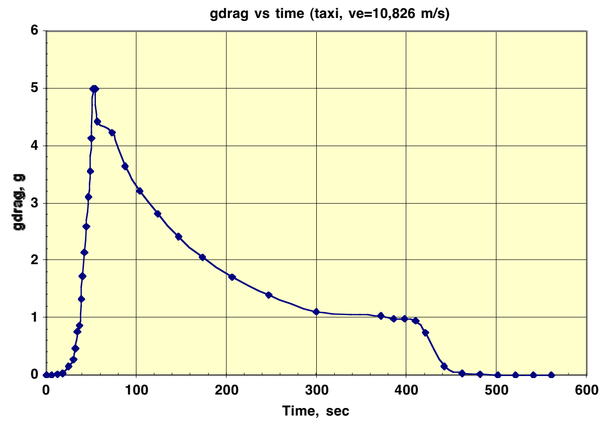


Figure 4-13 Baseline Aerocapture g_{drag} vs. Time

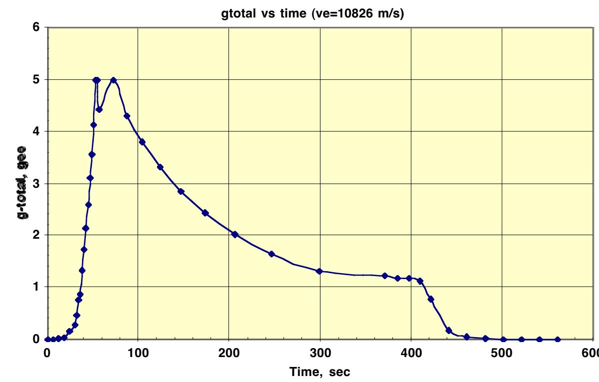


Figure 4-15 Baseline Aerocapture Total g-load vs. Time

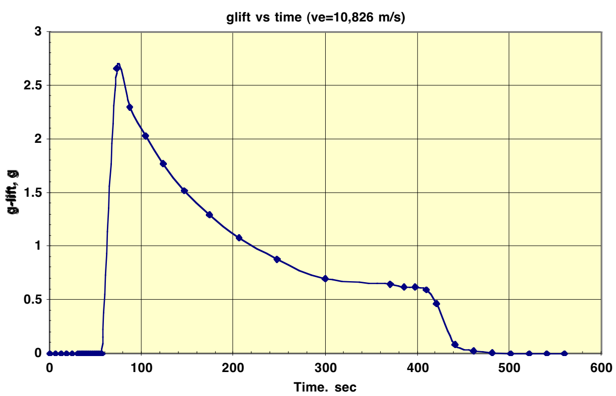


Figure 4-14 Figure 4-10 Baseline Aerocapture g_{lift} vs. Time

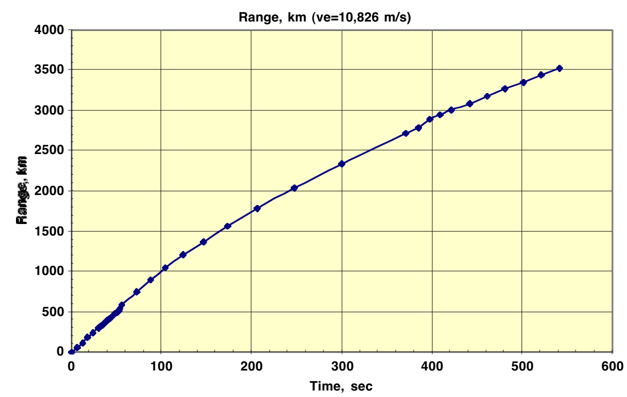


Figure 4-16 Baseline Aerocapture Range vs. Time

4.6.3.1.2 Steeper Entry

Examination of some entry trajectories into Mars at V_e = 12.502 km/s, with ballistic coefficient, B= m/C_dA of 30 and 60 kg/m^2 indicates that entering at a slightly steeper angle (than the nominal 10 deg) can remove about 700 m/s prior to arriving at about the 5-g deceleration level.

4.6.3.1.3 Ballute Drag Augmentation at Entry

A ballute alone could be used to remove almost all necessary delta-V however such a ballute would be excessively massive and result in excessive g-loads for human crews; greater than 17.

Alternatively, using a smaller ballute on the inward path could generate over 1000 m/s more delta-V prior to the point of level flight. The characteristics of such a ballute design are initial 50-m radius, a peak temperature about 500°C, a ballute envelope mass of about 100 kg and an envelope structure (a strong net) mass of about 350 kg. A pressure relief valve responding to the g level could partially deflate the ballute, reducing its cross-section area, to maintain the g-load below 5 or 6. Ballutes of this kind have not been flight-tested but NASA is presently considering a development program leading to a flight test

4.6.3.1.4 Propulsive Delta-V

4.6.3.1.4.1 Propulsive Pre-entry Delta-V

Here we examine how a pre-entry delta-V of 1.0 km/s could improve the 2016 opportunity shown in the Table 4-6 for 5, 6 and 7 g. We compute the g_{lift} needed, the allowed peak g_{drag}, and the corresponding L/D, for the new lower initial horizontal aero-cruise speed of 10.9 km/s (11.9 –1.0). For this situation, the peak initial cruise total g-load of 6 enables all the Mars entry cases with the current Taxi vehicle design.

Table 4-7 Peak Allowable g-load Due to Drag

Total g-load Allowed	g_{lift}	Peak g_{drag} Allowed	Required L/D
5.0	3.13	3.90	0.80
6.0	3.13	5.12	0.61
7.0	3.13	6.26	0.50

4.6.3.1.4.2 Propulsive Thrusting During Aero-Cruise

One promising new and innovative scheme to reduce crew g-load is to fly a little higher, reducing g_{drag}, and to use propulsive thrust to augment the limited lift of the vehicle during the critical few seconds at the start of aero-cruise. With a Mars entry velocity of 12.5 km/s, the start of aero-cruise is at a velocity of about 11.9 km/s. The allowed g_{lift} to keep $g < 5$ g is 2.67. Since the required g_{lift} is 3.81, a propulsive thrust level of 1.14 g will augment the lift of the vehicle and allow it to continue aero-cruise and still keep the resultant total crew g-load below 5 gs. From observing the placement of the engines on the vehicle of Ref. 1, it seems that the nozzle axes would be about 45 deg to the velocity vector, and to generate vertical force would also generate a horizontal force opposing drag. This has the unexpected effect of reducing the total g-load. For example in the above case a propulsive vertical force of 1.14 g would give the required 3.81 of g_{lift} and would reduce g_{drag} by 1.14 g, resulting in 3.81 vertical and 3.10, horizontal, giving a total g-load of 4.91 g. There will evidently be an optimum value and direction of the engine thrust to meet the cruise conditions and give minimum total g-load.

The total burn delta-V to augment cruise lift was evaluated for a 45 deg case of entry at 12.5 km/s, starting cruise at 11.8 km/s. The delta-V was 358 m/s, compared with a burn of 1674 m/s just prior to entry, or 1858 m/s at "infinity" to reduce the speed enough to perform cruise with only lift. The vehicle of Ref. 6 is derived from a set of "canoe" type entry vehicles evaluated for a range of cone angles and rake angles. Data published [7] for a vehicle close to that of Ref. 6 indicate that higher L/D, up to 0.9, is generated at greater angle of attack, and one can infer that the Ref. 6 vehicle has also this capability.

The following figures display the key parameters for the case where we are thrusting at an angle of 45° to the velocity vector and in the same general direction.

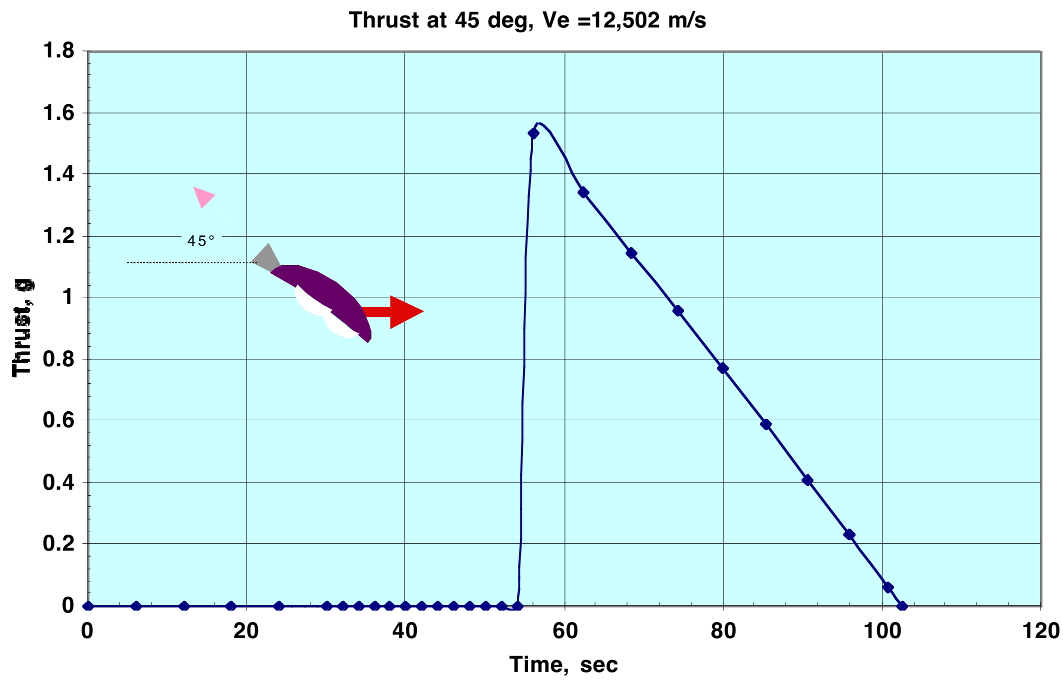


Figure 4-17 Propulsive Thrust vs. Time

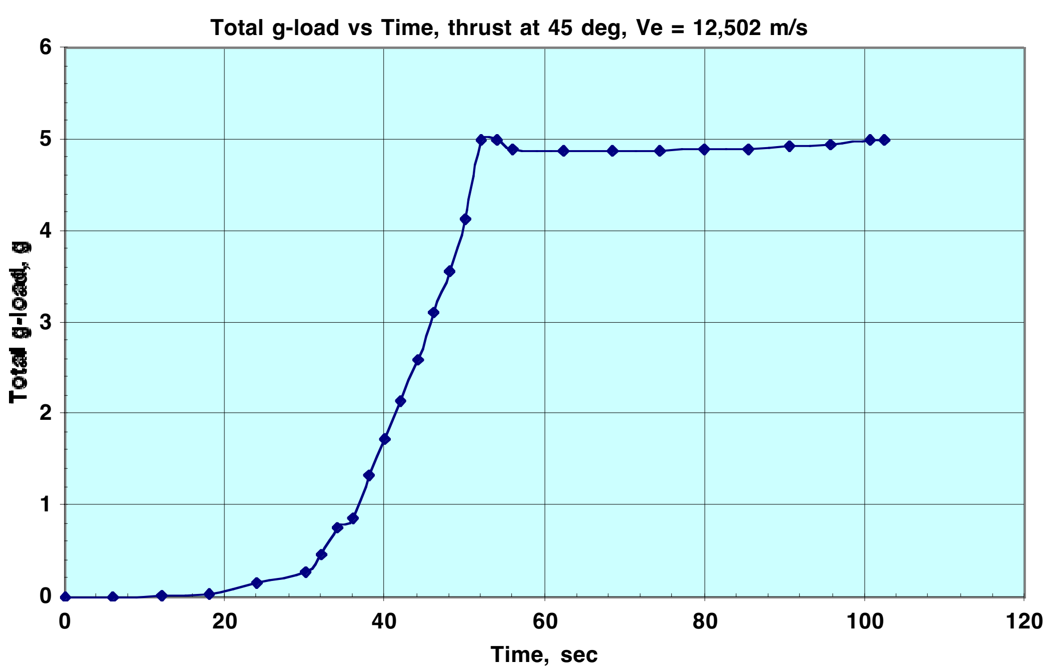


Figure 4-18 Total g-load vs. Time for Propulsive Thrusting Case

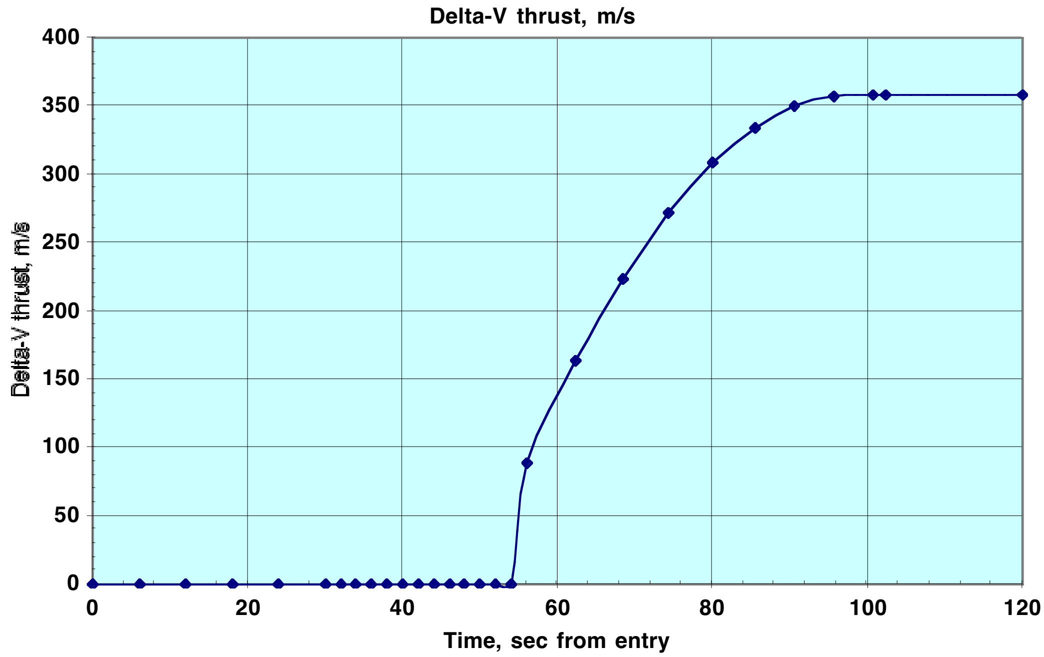


Figure 4-19 Integrated Delta-V vs. Time for Propulsive Thrusting Case

4.6.3.2 Earth Entry

For Earth entry the required L/D is much less because the planet radius, R is larger. The peak entry speed at Earth is 12.693 km/s at altitude 125 km and the inertial entry angle is -6.0 deg. For a vehicle with L/D of +0.3 and B = m/C_dA of 60 kg/m^2, the vehicle levels off at about altitude 74 km, experiencing a peak deceleration of 5.25 g due to drag and 1.58 g due to lift for a total of 5.48 g. The Taxi vehicle exits with a speed of 10.620 km/s, which is about the correct exit speed, so that a cruise at constant altitude is not needed. We note that entry at a somewhat smaller angle gives a larger exit speed if the vehicle does not "cruise" with downward lift for a short period. For example the exit speed with no aero-cruise and entry angle -5.9 deg is 10.834 km/s, so that a short "cruise" to lose about 214 m/s is still required. As with Mars, it is likely a ballute alone could give the correct exit speed at Earth, thus avoiding the need for a heat shield, but it is probable that the peak g-load would be larger than 5. There is no doubt that by using more lift one could achieve the exit speed with less peak g-load, and as at Mars, an engine burn could be used to reduce the peak g-load.

4.6.3.3 Entry Heating

As far as heating is concerned, Earth entry parameters drive the design of the Taxi vehicle. The entry heating for a 12-m diameter entry body at Earth and Mars has been computed. The convective stagnation point value for Earth (speed 12.693 km/s maximum) at -6.0° entry angle has a maximum of about 64 W/cm^2. For Earth, the entry speed does not vary much with the date, whereas for Mars only the 2014 and 2016 cases have comparable entry speed to Earth. The radiative heating for Earth was estimated from the data of Ref. 8.

The radiative peak heating for a reference 1-m radius sphere is about 46 W/cm^2, and the scaling is about radius to power 0.6, giving a value of about 135 W/cm^2 for the 12 m diameter vehicle, so that the unblocked peak heating will be about 200 W/cm^2, corresponding to a peak radiative equilibrium temperature of about 2570 K, with an emissivity of 0.8. This appears to be below the maximum temperature of the new high-temperature ceramic Thermal Protection System (TPS) materials that have been developed, so that one would not expect significant ablation to occur. Accordingly the TPS should withstand repeated entry cycles without having a significant repair or replacement activity.

4.6.3.4 Heat Shield Mass

The TPS and its support structure for the vehicle design concept in Ref. 6 were quoted as having a total of about 16.7% of the entry mass, for an entry speed of 10.3 km/s and a m/C_dA of 74 kg/m^2. The heat shield of the Apollo vehicle had a mass of about 13% of the entry mass, for an entry speed of 10.8 km/s. A recent review of the Apollo case indicates that there was a greater than needed margin on the TPS thickness. With a combination of lightweight, high temperature tile material and low-mass insulating material the Taxi vehicle heat shield mass is estimated to be 15% of the entry mass.

4.6.3.5 Taxi Aeroassist Design Conclusions

The vehicle described in Ref. 6 seems appropriate for the Taxi, with the caveat that for Mars entry at the high speeds of 2012, 2014 and 2016, the vehicle will encounter g-loads that are higher than 5 (but only for a short time) and this could be lowered by using a ballute at entry or an engine burn at the start of cruise. For the other four opportunities in Table 4-6 in the section above, the peak g-load is less than 5 and the L/D of the vehicle is less than 0.5, both within the Taxi vehicle capability. The 2012 case is within the Taxi vehicle design using only vehicle lift and drag if one accepts a little more than a peak of 5.0 g-load. For the 2014 and 2016 cases, a peak g-load of about 7 is required if the Taxi vehicle design remains constant. If the vehicle has a higher L/D than published in Ref. 6 the 2014 and 2016 Mars speeds could be accommodated, and the use of an engine burn at an appropriate angle would reduce the peak g-load substantially, and a backup engine to ensure crew safety in controlling the cruise is desirable. Employing a ballute for the initial entry reduces peak g-loads, but it must have an autonomously controlled deflation system to keep peak g-loads <5 g at all times. Alternatively, a cluster of small ballutes could be released in sequence during the initial entry. A heat shield composed of a high-temperature radiating outer layer with a high-temperature light-weight insulator substrate seems likely to perform the entry without ablation or the need for significant emplacement or repair of tiles, etc. A mass of about 15% for the heat shield is estimated.

4.6.3.6 Taxi Aeroassist Design References

1. Walberg, G.D."A Review of Aeroassisted Orbit Transfer", AIAA Paper 82-1378, August, 1982.

2. Braun, R. D. and R.W. Powell, ”A Predictor-Corrector Guidance Algorithm for Use in High-Energy Aerobraking System Studies”, AIAA Paper 91-0058, January, 1991.

3. McRonald, A. D. and K. T. Nock, ”Atmospheric Braking to Circularize an Elliptic Venus Orbit”, Paper at the AAS/AIAA Astrodynamics Conf. at Jackson, Wyoming, September 1977.

4. McRonald, A. D. and J. A. Randolph “Hypersonic Maneuvering to Provide Planetary Gravity Assist”, J. Spacecraft and Rockets, Vol. 29, No.2, 223-232, March-April, 1992.

5. McRonald, A. D., J. E. Randolph, M. J. Lewis , J. Longuski, et al., “From LEO to the Planets Using Waveriders”, AIAA Paper 99-4803, November 1999.

6. Scott, C. D., et al., "Design Study of an Integrated Aerobraking Orbital Transfer Vehicle”, NASA TM 58264, March 1985.

7. Mayo, E. E., et al., "Newtonian Aerodynamics for Blunted Raked-Off Circular Cones and Raked-Off Elliptical Cones", NASA TN D-2624, May 1965.

8. Taubert, M. and Sutton, K., “Stagnation Point Radiative Heating Relations for Earth and Mars Entries”, J. Spacecraft and Rockets, Vol. 28, No. 1, p 40-42, January-February 1991.

4.6.4 Mars Shuttle Entry

The atmospheric entry of the Mars Shuttle has been analyzed and it has been determined that the design originally proposed by the NCOS study (Section 1 Ref 1) was robust enough, given current technology, to allow direct entry into the Martian atmosphere. Direct entry eliminates a modest delta-V required to first circularize the vehicle prior to entry targeting and thus reduces Phobos propellant production requirements and its storage at the Mars Spaceport. More study of the design of this vehicle is planned in Phase II.

4.7 Power Systems

4.7.1 Introduction

Two power conversion system choices were initially examined, namely nuclear and photovoltaic arrays. Factors considered in power generation technology selection are life cycle cost, specific power (power generated divided by generation and storage mass), modularity, and safety (space operations and manufacturing). Because solar photovoltaic power generation appears very attractive due to the projected very low cost and mass, it was selected for study in Phase I. This initial selection will be tested in Phase II. Energy storage options are also considered.

4.7.2 Solar Photovoltaic Power Generation

There are two very different photovoltaic powered missions involved in this report – deep space and planetary or satellite surface operation. The near term technology applicable to both these missions will be discussed first and then projected to technologies likely to be available in 2010.

For the projection of solar array technology to 2010, cells are expected to have improved efficiency and structures and optical systems should become lighter.

4.7.2.1 Deep Space Solar Arrays

Deep space missions are those which take the spacecraft out of Earth orbit to orbits between Earth and the Moon, the L1 point and past Mars. Since Mars is not too distant from the sun, all of these missions can be handled with essentially the same power generation system technology. The size of power generation systems for the Mars transportation architecture ranges from about 12 kW to nearly 1 MW.

4.7.2.1.1 Mechanical and Optical Technologies

The near term technology was reviewed with regard to demonstrated performance at least at the solar array module level, cost and possibility of improvement. Two aspects were studied including lightweight deployable and concentrator arrays.

Lightweight photovoltaic energy conversion systems, or solar arrays, have been used on most space missions. However, very large solar arrays (100 kW) are not common so there is little experience applicable to this study. The analysis presented in this section is based upon projection of past development onto present demonstration technologies. Lightweight, deployable, photovoltaic arrays have been demonstrated for space missions ["Advanced Photovoltaic Solar Array Design," TRW Report No. 46810-6004-UT-00, 3 November 1986] and are being incorporated into a number of programs. Dependent upon the technology selected, large (in excess of 2kW), high performance arrays cost about \$850/W. Using current costs, a 160 kW array, as suggested for the Astrotel, could cost 136 million dollars.

A solar cell concentrator approach, which should reduce array cost, uses fewer of the expensive elements i.e. the solar cells. A 15 times (15X) concentrator array uses roughly one-twelfth the number of solar cells. Concentrator arrays have now been space qualified [P. A. Jones et al., “The SCARLET Light Concentrating Solar Array,” 25th IEEE-PVSC, 1996] on the DS-1 spacecraft. The SCARLET array has achieved over 200 W/m^2 areal power density and 45 W/kg specific power. Figure 4-20 illustrates the SCARLET array in flight.

A combination of lightweight array and concentrator technology is in the demonstration phase [M. J. O’Neill, “The Stretched Lens Ultralight Concentrator Array,” 28th IEEE-PVSC, 2000]. The stretched lens array (SLA) has been incorporated into a deployable, flexible-blanket planar space array concept called Aurora by AEC-ABLE. Figure 4-21 illustrates a stretched lens Array Prototype. Figure 4-22 illustrates a deployment concept for the SLA.

Aurora components have already demonstrated a cell efficiency of 30% and a lens efficiency of 92%. Operational efficiency at beginning of life (BOL) is expected to be 22% or about 300 W/m^2 areal power density. This corresponds to a near term expectation of 170 W/kg BOL specific power at the deployed wing level. The combination of SCARLET and SLA technology potentially provides a 50% increase in areal power density and almost a 300% increase in specific power.

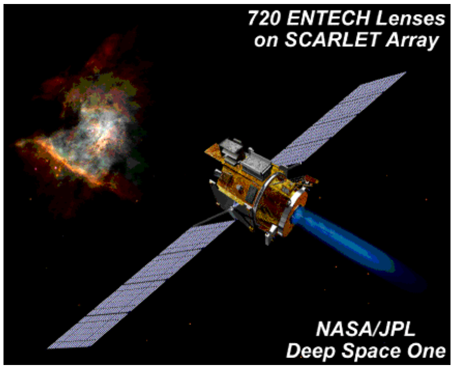


Figure 4-20 Picture of SCARLET Array on DS1 Spacecraft (Courtesy ENTECH)

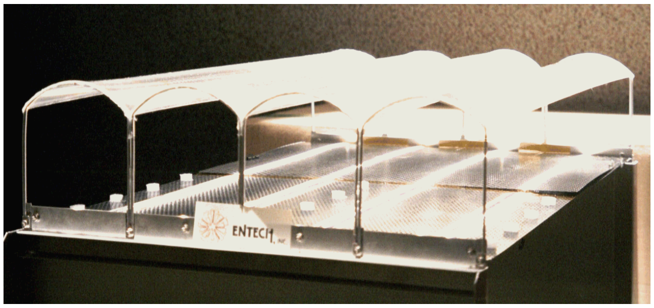


Figure 4-21 Stretched Lens Array Module Prototype (Courtesy ENTECH)

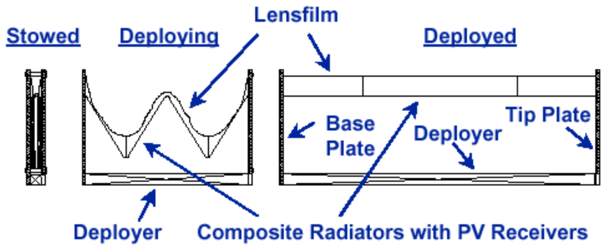


Figure 4-22 SLA Deployment Concept (Courtesy ENTECH)

Table 4-8 Aurora Array Performance

Beginning of Life (BOL) Performance Parameters	
Average Cell Efficiency at 8 Suns and Room Temperature (Demonstrated)	30%
Average Cell Efficiency at 80C (GEO Operational Temperature)	26%
Lens Efficiency (Demonstrated)	92%
Cell-to-Panel Packing Factor	95%
Wiring/Mismatch Factor	95%
Operational Array Efficiency (Product of Last Four Values)	**22%**
Areal Power (W/sq.m.)	**296**
Areal Mass (kg/sq.m.)	**1.74**
Specific Power (W/kg)	**170**

4.7.2.1.2 Solar Cell Technology

JX Crystals projects 32-35% cell efficiency in 10 years. This is a conservative estimate since they have already achieved 30% with non-optimized cells [L. Fraas et al., "30% Efficient InGaP/GaAs/GaSb Cell-Interconnected-Circuits for Line-Focus Concentrator Arrays," 28th IEE-PVSC, 2000]. The following figure shows a typical stacked cell set. The top picture is a completed InGaP/GaAs/GaSb circuit, the middle is a circuit with GaSb IR cells and the bottom is the substrate with metal traces. These overlay each other to form the integrated cell.

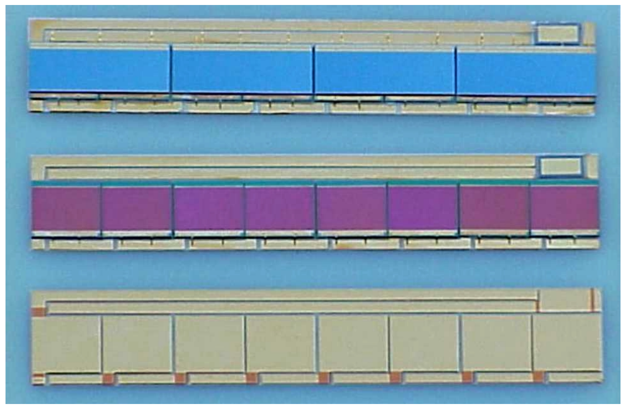


Figure 4-23 Example of a Stacked Cell Set (Courtesy JX Crystals)

Mechanically stacked cells have been assumed for the Phase I study due to the present uncertainty over the feasibility of 4 junction photovoltaic cells [P. Iles, "Future of Photovoltaics for Space Applications," Progress in PV Research and Apps. 8, 39-51, 2000]. Other selection criteria were the lower projected cost of the mechanically stacked cells and the ability to electrically connect the stacked cells to take advantage of the larger currents produced by the bottom cells.

4.7.2.1.3 Integrated Solar Array Design and Costs

Discussions with AEC-ABLE, ENTECH and JX Crystals indicate that there is a high likelihood of significant mass reduction to achieve 600 W/m^2 and 340 W/kg in 10 years. For the purposes of the Phase I study we have derated these numbers to 450 W/m^2 and 250 W/kg. An issue to be discussed is the cost of arrays. Two elements of array cost are the cost of the cell itself and the cost of the mechanical/optical systems.

Cost of a mechanically stacked cell can be assumed to be slightly higher than the cost of a triple junction cell. For simplicity we will assume a factor of 1.2 times the cost of a triple junction cell or about $300/W in year 2000 dollars. The mechanically stacked cell consists of an epitaxially formed double junction (InGaP on GaAs) cell stacked on top of a diffused single junction gallium antimonide (GaSb) cell. The double junction cell is slightly less expensive than a triple junction cell and the diffused GaSb cell is much less expensive than an epitaxial cell. Each cell has two wiring connections. The upper cells are wired in parallel since they are high voltage, low current cells. The lower cells are wired in series since they are low voltage, high current cells. There will be a different number of each type of cell in order to match voltages. At the end of each voltage balanced module the two wire strings can be connected. This module then is a natural size for application of a stretched lens 15X optical concentrator and attendant thermal radiator. Due to optical losses and packing considerations, the final concentrator array can be assumed to achieve about a 12X reduction in required cell aperture area. This gives a 12X factor for reduction of cell costs along with attendant stringing costs. The reduction in stringing costs is improved since it can be done with rugged automated wire bonding machines rather than fussy cell bonding machines. The final cost savings on cells would be only a factor of 10 since the stacked cells are more expensive. The expense of bonding all of the strings onto a module in a series of large areas is reduced to that of mechanical assembly and wiring the separate modules together. There may be a cost savings here but it is hard to quantify at this time. Cost of the stretched lens concentrator and thermal radiator is less than that of an equivalent area of solar cells. The total area of array required is reduced by a factor of 25/30 which is the ratio of the operating efficiencies at GEO of a triple junction cell to that of the stacked cell. The net result of all of these changes is to produce a final cost per watt of an Aurora type space array of about $700/W. Additional savings might be realized from the large size of the array but this can not be easily quantified especially 10 years into the future.

4.7.2.2 Planetary Surface Solar Arrays

Surface operations requiring solar array power are contemplated on the Moon, Mars, and Phobos. These locations can be serviced with essentially the same technology – at least as a first approximation. The types of solar arrays that can be placed on a planetary surface include a) a rigid structural space or terrestrial solar array modules, which can be oriented at a fixed angle or pointed in one or two axes to track the sun or b) a low-cost, lightweight, flexible, thin-film solar array that can be laid out on the surface over large areas. In Phase I we have selected the low-cost approach without having completed all the trades. The reason is that these arrays are potentially very low mass and low cost. In Phase II will test this assumption by comparing the overall cost of solar array types including all cost elements (including array and emplacement costs).

An additional consideration for surface array operation on natural satellites or planets might be the inclusion of a robotic dust removal system. On the surface of Mars or other bodies with an atmosphere a vacuum cleaner approach might be used. On satellites without an atmosphere some sort of electrostatic or other systems may need to be developed.

4.7.2.2.1 Near Term Technology

Planetary (Mars) and satellite (Moon and Phobos) surfaces provide a fixed surface for mounting a photovoltaic array and thus an opportunity to reduce mass by eliminating most structure. Due to planet rotation, the solar angle on the array changes continuously and is almost never normal to the surface of the cells. Thus the size of the array must account for the varying solar angles. In addition, operations (including ISRU) duty cycles and capacities must be designed to respond to the varying solar energy input. At the Lunar poles the Sun is almost always at the horizon. A suitable location for a large surface mounted array must be found that will allow sufficient solar illumination. The Lunar South Pole has a mountainous region where an array could be positioned at an angle of 40° to the horizontal. Significant oblique solar illumination will require several times the array area as opposed to when the sun is normal to the array.

A United Solar press release on their web site [http://www.ovonic.com/unitedsolar/uninews] claims that their thin-film triple-junction amorphous silicon (see following figure) modules on the MIR space station have a specific power greater than 500 W/kg (2 kg/kW).

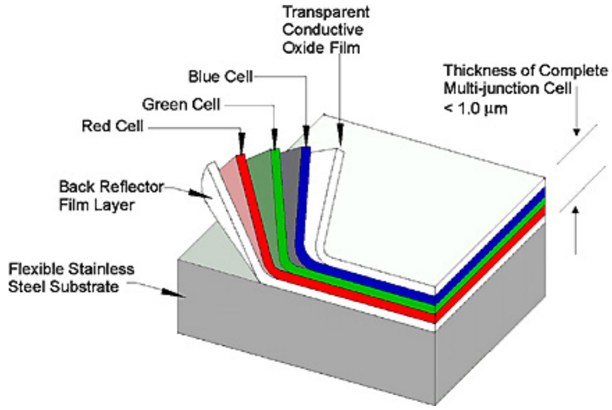


Figure 4-24 Triple Junction a-Si Cell

The United Solar modules were a special run with the cells deposited on a thin (between 0.5 and 1 mil thick) stainless steel substrate rather than the standard 5mil substrate. Neither areal power or conversion efficiency is noted in the press release. If the stainless steel substrate were 0.8 mil thick and no other mass is assumed then the conversion efficiency can be calculated at 5.9%. This low efficiency imposes a large supporting structure mass penalty on a spacecraft but not on a fixed ground installation.

Iowa Thin Film Technologies (ITFT) is producing single-junction amorphous silicon modules on a 5-mil thick polymer (Kapton) substrate. The ITFT modules operate at 5% efficiency and have been made on special order on 2-mil thick Kapton. These special modules had a specific power of about 650 W/kg (1.54 kg/kW).

4.7.2.2.2 2010 Technology

Thin film photovoltaic cell systems of amorphous silicon, cadmium telluride, copper indium diselenide (CIS), and copper indium gallium diselenide (CIGS) are all being developed for application on flexible substrates like thin polymer films. One near-term technology – triple junction amorphous silicon solar cells - has achieved 10% efficiency for long lifetime and their future is very promising. When compared with conventional crystalline solar cells, thin film solar cells have several advantages for planet surface operations application. Advantages include: physical flexibility afforded by their thin-film construction, ability to be fabricated into monolithic cell strings and good resistance to radiation.

NASA has studied advanced thin-film, flexible solar array systems for Lunar and Mars surface applications utilizing amorphous silicon solar cells for flexible solar array blankets [Colozza, A. J., *Design and Optimization of a Self-deploying PV Tent Array*, NASA CR 187119, June 1991]. The areal density of these planetary solar array systems was stated to be about 20 g/m^2. This represents an array using a polymer substrate less than 1-mil (25 um) in thickness and has little or no allowance for wiring and probably no allowance for any deployment or attachment hardware. A more realistic estimate would be about 4 times higher mass or 80 g/m^2 for a 2-mil substrate, wiring and minimal hardware. CIGS technology is very likely to produce AM0 (air mass zero – space) cells on polymer substrates with about 14% conversion efficiency in 10 years. Using the 80 g/m^2 and 14% figures gives a calculated specific power of 2400 W/kg (0.42 kg/kW). This value needs to be derated to account for packing factor and operating temperature. Using estimates of 0.9 for packing factor and 0.9 for temperature derating yields a more reasonable specific power of 1920 W/kg (or about 0.52 kg/kW).

A design value of 1 kg/kW is appropriate for this study, which includes stakes, tie downs, robotic dusters and other necessary surface deployment and maintenance hardware. A much more conservative specific power of 250 W/kg (4 kg/kW) has been assumed in the calculations of surface power mass requirements since this technology projection of about 1 kg/kW was available late in the study.

4.7.3 Energy Storage

Energy storage requirements are in two categories, namely the need for power for small, non-solar-powered spacecraft like Taxis and power storage for solar-powered space and surface systems during nighttime or eclipse operation.

Since photovoltaic energy conversion requires sunlight, the "fixed" base solar-powered space (Astrotels and Spaceports) and surface (resources plants) systems must have an energy storage system to handle nighttime or eclipse power needs. In most surface system cases, limiting major operations to the daytime hours minimizes nighttime energy requirements. Nighttime requirements are generally keep-alive power in order to keep electronics from failing at cold temperatures.

In the case of power for small spacecraft like Taxis, small, high energy density energy storage systems are needed. In addition, they must be either rechargeable or refillable at a transportation node.

4.7.3.1 Energy Storage Options

The types of energy storage that can produce electricity directly are fuel cells, ultra capacitors, and batteries. For larger systems flywheels sometime can also become competitive in energy density with some types of batteries. Flywheel storage has not been widely used and has a serious limitation with friction losses when the storage period is long term. High capacity capacitors still do not have an adequate power density for space use. This reduces the energy storage options to fuel cells and secondary batteries.

4.7.3.1.1 Fuel Cells

Fuel cells using hydrogen and oxygen are used in the space shuttle and are a proven technology. Standard fuel cell energy conversion systems require pressurized tanks for both fuel and oxidizer. Estimated specific power for a fuel cell is 700-1000 Wh/kg [Nesmith, W., DOE HQ, Personal communications, October 1997]. New technology options include a regenerative fuel cell that contains both a fuel cell and an electrolysis capability in the same unit. Regenerative fuel cells for energy storage have been investigated for application to solar-powered aircraft at energy densities about 450 Wh/kg including tanks [see http://www-atp.llnl.gov/str/Mitlit.html]. Oxygen and hydrogen for fuel cell operation can be generated and stored at a surface based facility to refuel the cells periodically. Space energy storage might benefit from the use of regenerative fuel cells where the hydrogen and oxygen are reacted to form water when power is needed. Later when surplus power is available the water can be broken down by electrolysis into hydrogen and oxygen.

4.7.3.1.2 Secondary Batteries

The types of secondary batteries that should be considered are nickel-hydrogen, nickel-metal hydride, lithium-ion, and lithium-ion polymer. These presently have the approximate energy densities (Wh/kg) shown (cell size and packaging dependant):

Table 4-9 Battery energy densities, Whr/kg

Nickel-hydrogen	60
Nickel-metal hydride	80
Lithium-ion	100
Lithium-ion polymer	130

Nickel-hydrogen has been included because of its continuing outstanding performance in space. Information is available that shows performances of 60,000 cycles at 60% depth-of discharge (DOD). However, the nickel-hydrogen battery basic energy density really has not improved in some time. Some improvements have been obtained in smaller batteries with common (2 cells per pressure vessels) and single pressure vessels (all cells in one).

Nickel-metal hydride promises the volume of nickel-cadmium with an energy density exceeding nickel-hydrogen. Larger cell sizes are becoming available.

Lithium-ion batteries are being made in larger sizes and they may even exceed the 100 Wh/kg given. Battery charging has been somewhat critical with successful methods using bypass circuits across each cell to prevent overcharge on a cell level. A concern is that high cycle life has not really been demonstrated. Additional data shows a loss of capacity with cycling that seems to level out after about 500 cycles.

Lithium-ion polymer is really a lithium-ion battery with a different kind of separator and packaging. The comments on lithium-ion charging and capacity loss also apply to the lithium-ion polymer battery.

4.7.3.2 Space Mission Energy Storage Selection

At this time, the lithium-ion polymer appears to be the best candidate for space Astrotel use because of projected improvements in energy density and the low cycle life requirement. For Astrotel space missions the number of cycles is generally low since batteries would essentially be standby power or used during eclipses by a planet or natural satellite.

The emerging lithium-ion polymer technology is generally referred to as lithium polymer. This battery was developed and patented by Bellcore. The polymer separator material is made by

Valence Technology under a Bellcore patent. Valence and at least two other companies are making cells using this separator material. The other companies are Alliant (part of Power Sources Company) and Ultralife. The cells are thin and somewhat flexible and have an energy density around 130 Wh/kg.

At present, Valence is making four cell sizes for commercial evaluation; the maximum capacity being 3 Ah. Custom cell sizes can be made by varying the cell dimensions and thickness. The maximum thickness seems to be about 10 mm.

Alliant recently was making two standard cell configurations and varying the thickness to get desired capacities. The standard sizes are 4 by 4 inches and 5 by 7 inches. In the 5 by 7 size, a 10 mm thickness would result in a capacity of about 12 Ah. Alliant can manufacture custom dimensions and shapes.

Ultralife, several months ago, was developing two small sizes, a 600 mAh and a 750 mAh for cell phone applications. They call their unit a solid polymer rechargeable battery, but it is the same type as the others.

Table 4-10 Typical Lithium-Ion Polymer Cell Characteristics

Energy density:	125-140 Wh/kg
Operating/storage temperature range	-20°C to + 60°C
Charge conditions	C/2 max. to 4.2V max. (0 to 40°C)
Discharge voltage	4.0 to 3.25 V (3.7typ @ 25°C)
Typical discharge rate	(C/2) [C – cell capacity]

For the 2010 technology horizon assumed, available cells of at least 25 Ah and energy density of at least 200 Wh/kg should be available.

4.7.3.3 Mission Energy Storage Requirements

The following table describes the energy storage requirements and assumptions for various systems of the Mars transportation architecture.

Table 4-11 Mars Transportation System Energy Storage

System	Power kW	Duration days	Energy kWhr	Storage Media	Energy Density kWhr/kg	Mass kg
Taxi	10	10	2400	NRFC LOX/LH	0.7-1.0	2400-3430
Mars Shuttle	10	2	480	NRFC LOX/LH	0.7-1.0	480-690
Astrotel	10-30	0.33	80-240	Li Ion Polymer	0.2	400-1200
Surface (Mars)	1.0	0.75	180	Li Ion Polymer	0.2	900
Surface (Mars)	1.0	0.75	180	NRFC LOX/LH	0.7-1.0	180-260

NRFC – non-regenerative fuel cells

5 Planetary Resource Utilization Systems

5.1 Introduction

The use of planetary resources significantly reduces the material that needs to be brought up through the gravity well of the Earth and delivered to a planetary transportation node. The energy required for transportation of propellant is proportional to the square of the velocity change that it must undergo. For example, the energy required conveying propellant from the Moon to L-1 is approximately 1/30th of that required from the Earth's surface to L-1 and requires a much simpler spacecraft.

Cost savings, in part, result from the reduction in number of launch vehicles needed to deliver these materials and propellants to LEO. In addition, there is the cost of delivering these cargoes to the Earth or Mars Spaceports from LEO. There is a further reduction in transportation costs due to the elimination of transportation elements themselves and their refurbishment mass, which must all be launched to LEO. The saving of transportation energy, mass and cost must be balanced against the cost and mass of propellant production systems on Mars, Phobos and the Moon and the cost of the transportation systems for getting propellants from the Moon to L-1. For an *in situ* resource production scheme to be successful, these costs must be less than the transportation of terrestrial materials by a large enough amount to overcome the inertia inherent in the status quo.

Generally, propellant production systems are sought that produce hundreds or thousands of times the mass of the production hardware in the useful lifetime of the production plant. *In-situ* propellant production scenarios work best when the propellant is produced at a single location over a long period of time. In that case, the production rates can be low, allowing small production systems.

5.2 *In situ* Resource Options

Table 5-1 describes options for *in situ* resource utilization at or near the main depot locations based on current knowledge of possible occurrences of the resources and preliminary estimates of the processes needed to recover them. Preliminary spreadsheet models were constructed for the most promising options (marked with *) that modeled excavation, extraction, water electrolysis, liquefaction, storage and transportation approaches to help select the baseline for the current study.

Table 5-1 *In situ* Resource Options

	Mars Surface	Phobos	Moon/L-1
1	*Water from Permafrost (ground ice)	H_2/O_2 from Regolith	***Water from Moon; H_2/O_2 produced at L-1**
2	***Extract bound water from the Regolith**	***O_2 from carbonaceous Regolith; H_2 from Earth or L-1**	*H_2/O_2 from Regolith used to send O_2 to L-1; H_2 from Earth to L-1.
3	*H_2 from Earth; O_2 from Mars' Atmosphere	O_2 from non-carbonaceous Regolith; H_2 from Earth or L-1	

A brief description of each option is given below. The options given in bold type are those selected for the baseline.

5.2.1 Mars Surface Options

There are several potential sources of water on Mars. These include ground ice, subterranean liquid water or brine, atmospheric water and adsorbed or bound water in regolith (soil). In addition, if hydrogen is brought from Earth, water can be produced by reaction of hydrogen with CO_2 from the Martian atmosphere. Extraction of water from ice, liquid water and the atmosphere will likely place constraints on the location of the water production facility. The most feasible processes seem to be extraction from ground ice (permafrost), the regolith and producing water by reaction of terrestrial (or lunar) hydrogen and Martian atmosphere.

5.2.1.1 Ground Ice

At locations greater than 30° from the equator, ground ice may be present close to the surface and could be mined. In some places, liquid water may be found well below the surface by geophysical exploration and drilling techniques. Ice-bearing material would be excavated and heated to ~100° C in a closed container to extract water. The water would be condensed, then electrolyzed, and the liquid hydrogen and oxygen stored in tanks on the Shuttle vehicle for its next trip to the Mars Spaceport. This option probably requires the least energy, but would have to be performed at higher latitude or even polar locations, where seasonal variations of solar energy are large and nuclear power systems would probably have to be used.

5.2.1.2 Regolith

Viking demonstrated that the surface regolith on Mars contains 1-2% water by weight, released at 500°C. This is consistent with the occurrence of 5-10% of clays or other hydrated minerals in the regolith. The true distribution of water in the regolith is currently unknown, and places where larger concentrations of water exist may be located. The mineralogical form is not known, but could be easily determined on an upcoming robotic science mission to Mars. The process of extracting water from the regolith requires excavating fine material and heating it in a closed reactor. A model was constructed for these processes, making reasonable assumptions for the mass of excavation, extraction and power systems. The data from this model was used in developing mass and power requirements and costs of regolith water production. Research on excavation and extraction systems is currently underway at the Colorado School of Mines (Muff et. al., 2000). Regolith would be excavated from drift or dune deposits and heated in a closed chamber to extract water. The water would be condensed, then electrolyzed, and the liquid hydrogen and oxygen stored in tanks on the Shuttle vehicle for its next trip to the Mars Spaceport.

The principal competitor to this process is to bring hydrogen from Earth to the surface of Mars and react it with CO_2 from the Martian atmosphere to produce water. The selection of this or the regolith water extraction process will depend on other parts of the system, primarily the power system. If energy is inexpensive, regolith water extraction is favored.

Regolith water extraction has been selected as the baseline in Phase I because of the availability of regolith everywhere on the planet. For the purposes of the Phase I study, 1% water in the regolith is assumed.

5.2.1.3 Atmosphere

One of several possible techniques could extract oxygen from the atmosphere, such as solid state CO_2 electrolysis with Zirconia cells. Hydrogen would have to be brought from Earth. This approach may be preferred if space transportation costs of hydrogen from Earth are low, but the current cost of transporting hydrogen favors obtaining water from the regolith.

5.2.2 Phobos Options

Production of propellant at Phobos could have significant importance to human exploration of Mars. Provision of propellant for humans to leave the vicinity of Mars and return to Earth is the most important use, though later on Phobos might become a node for human journeys to main belt asteroids. The propellant combination of most interest would be cryogenic H_2 and O_2, though CH_4 and O_2 might be useful if the same propellants were being used for transportation from the surface of Mars.

Little is known about the composition of Phobos. In the late 1980's it was thought by many that Phobos might be a captured asteroid with carbonaceous chondrite composition, because it has a very low albedo and a low bulk density. Phobos (and Deimos) appears to be very similar to many main belt asteroids. However, the Russian Phobos-II mission (1988) was unable to detect significant water absorption bands in Phobos' IR reflectance spectrum. Water is a characteristic component of many carbonaceous chondrites and water absorption bands can be observed in the IR reflectance spectra of some main belt asteroids. The low density of Phobos may be due to repeated fragmentation and accumulation of debris enhanced by Phobos' orbital position around Mars. It is thought that even complete fragmentation could result in re-accretion of Phobos' material into a loosely bound and underdense aggregate. It was initially thought that the low density indicated the presence of significant ice in the deep interior; however, the likelihood of Phobos' disruption and re-aggregation suggests that this is not likely. It is possible that Phobos is of carbonaceous chondrite composition and that its regolith, formed by meteorite and micrometeorite bombardment of its surface, has driven off the water. Or, it may be a water-poor carbonaceous chondrite, of which several exist in terrestrial meteorite collections. These meteorites tend to be denser than other carbonaceous chondrites and would therefore require Phobos to be physically less compacted to achieve its observed low density. The upper few centimeters of the surface of Phobos is known to consist of relatively fine-grained regolith, with few large blocks; however, Thomas (1992) argues that Phobos may have regolith up to 100 m thick in places. More information is needed before the chemical and physical properties of the regolith of Phobos can be ascertained.

Four possibilities exist for the provision of H2 and O2 propellants at Phobos: (1) transport everything from Earth, or the Earth's Moon; (2) Produce oxygen on Phobos, from the silicates that are bound to exist there, and bring hydrogen from Earth or Earth's Moon; (3) Produce both hydrogen and oxygen from Phobos, if its composition is carbonaceous chondrite; or (4) bring propellants from the surface of Mars. The last proposition seems least useful, because the energy required in order to bring material from the surface of Mars to Phobos is comparable to that for bringing material from LEO. If the Moon can be used as a source of propellant in the Astrotel scenario, lunar hydrogen can be trucked to Phobos. Alternatively, if Phobos is hydrous, it may be the optimum location for propellant production at Mars and at L-1.

The options for propellant production at Phobos are identified in the following table along with the manner in which the propellants might be extracted.

Table 5-2 Phobos Propellant Production Options

Source Material	Possible Water Concen-tration	Means of Extraction	Propellant Strategy
Water-bearing Carbonaceous Chondritic, covered with 1 meter of water-poor regolith	10%-20%	Removal of surface material; Pyrolysis to ~500°C	Produce H2/O2 at Phobos
Water-poor Carbonaceous Chondritic	<1%	Carbothermal reduction of silicates to produce oxygen using indigenous Carbon	Produce O2 at Phobos, bring H2 from Moon or Earth
Water-poor and Carbon-poor dark chondritic	<1%	Carbothermal reduction of silicates to produce oxygen; Carbon brought from Earth	Produce O2 at Phobos, bring H2 from Moon or Earth

It is assumed that at least the uppermost meter of Phobos is relatively fine-grained regolith that is readily excavated. In the case of water-poor regolith overlying a water-rich subsurface, it may be necessary to remove significant amounts of regolith before water-rich material can be recovered. Working in the very low gravity field of Phobos will present some interesting practical problems in excavation, hauling and manipulating granular materials, but these should not require larger equipment masses or power systems than the equivalent systems on the surface of Mars.

5.2.2.1 Water-bearing Carbonaceous Chondrite

The assumption is made that Phobos consists of water-bearing carbonaceous chondrite-like material (~10% water) that exists in the regolith near its surface. The extraction of water from carbonaceous chondrite materials would be quite similar to the extraction of water from the Martian regolith, as in both cases the water is most probably in the form of hydrous silicates (clay minerals) from which water is driven off by heating to 500° C. In the case of carbonaceous chondrites, significant amounts of water may be recoverable by heating at lower temperatures, though it is difficult to tell from studies of carbonaceous meteorites on Earth, which typically have significant amounts of water adsorbed by exposure to the Earth's atmosphere. The carbonaceous chondrite material has approximately 10 times the water content assumed for Martian soil. Therefore, the reactors and the energy requirements are much smaller. For the production of equivalent amounts of propellant, however, the water electrolysis and hydrogen and oxygen liquefaction systems will be similar to those on the surface of Mars, but can perform nearly continually with solar energy, diminishing their size by more than a factor of two.

The water would be condensed, then electrolyzed, and the liquid hydrogen and oxygen stored in tanks on the Taxi vehicle preparing for its next rendezvous with the Astrotel. The storage system constitutes approximately half the system's mass. Assuming that the Taxi tanks can accommodate storage, the total mass of the system for producing propellant on Phobos should be quite small. The total system with storage can produce about 6 times its mass per year. If storage is provided separately, the system produces about 80 times its mass per year.

This option would be the preferred mode if the compositional assumption were correct; however, compositional evidence for the surface of Phobos does not appear to be water-bearing. Although hydrated material may occur just under the surface regolith, the assumption cannot be confirmed at present.

5.2.2.2 Oxygen from Dehydrated Carbonaceous Chondrite

We do not know the composition of the silicates and oxides on Phobos but assuming Phobos is of carbonaceous chondrite composition the following option may exist. In this option, Phobos is assumed to consist of dehydrated carbonaceous chondrite-like material. Although there may be iron oxide minerals on Phobos, which could be amenable to hydrogen reduction, it is probably safer to assume carbon reduction, particularly as the carbon can be derived locally. Carbon will reduce silicates to elemental silicon and iron with a residual oxide slag, when the mixture is heated above its melting point. Typically, approximately half of the mass of oxygen in a basaltic or similar planetary material can be released in this type of process, equivalent to about 20% of the mass of the material processed. A carbon reduction furnace was defined by Rosenberg et al (1996) for lunar oxygen production at a rate of 5mt/year. This reactor was designed to operate using solar energy on the Moon for 3500 hours per year. It would have a mass of about 150 kg and need a power system producing 22 kW if scaled for nearly continuous operation.

Heating the material to 1600° C in a furnace and allowing the indigenous carbon to reduce the iron and silicon in the regolith releases oxygen. Oxygen is assumed to constitute 40% of the regolith material and it is assumed that 85% of the oxygen can be removed in the process. The O_2 is liquefied and stored in the Taxi, which is left at the Mars Spaceport. The Taxi either arrives at Phobos with H_2 for its next trip to the Astrotel, or a separate tanker brings the H_2 from the Earth or L-1. This assumption is neither the most optimistic nor the most conservative; the performance will be somewhat poorer than the next option discussed. This model assumes that the Spaceport is located very near or on Phobos.

As was previously the case, the mass of the processing system is quite small. The mass of the storage system dominates the total system mass. If tanks on spacecraft can also be used for storage, a great deal of mass can be saved and perhaps the system can be simplified. However, more power may be required to maintain cryogens. If storage must be provided, this system will produce about 10 times its mass per year. If storage is not required, the system will produce about 80 times its mass of liquid oxygen per year.

This system has smaller mass than the system for producing propellant from Phobos water because it does not require hydrogen production, liquefaction and storage. This is replaced by the requirement to transport hydrogen from Earth or Moon to Phobos.

This option was selected for the baseline architecture studied during Phase I.

5.2.2.3 Carbon-poor and Dry Phobos

This is essentially the same case in 5.2.2.2, except that stores of carbon in some form (e.g. methane) have to be brought from Earth. A key question is how much carbon is lost in processing a batch of Phobos material to produce oxygen? Rosenberg, et. al. assumed that 3% of the carbon would be lost to an unrecoverable state with each batch. The carbon required for reduction of SiO_2 is approximately 15% of the mass of the SiO_2 reacted. Thus, to produce 20 mt of O_2 requires the reaction of 600 kg of carbon. If 3% is lost, this amounts to 20 kg of carbon that must be made up for one year's production. It is quite possible that human wastes can provide the carbon for this reactor from the Astrotel. On Earth, silicon reduction reactors commonly use coal, coke, wood chips or other carbon containing materials in the reaction.

5.2.2.4 Cost of Propellant at Phobos

Based on the above comparisons, the preferred mode is to find water at Phobos. However, both cases provide the capability of producing about 80-times the mass of hardware per year in propellant, excluding the storage requirement. Calculations of the cost must consider the development of the hardware, its transportation to Phobos and the operational costs of maintaining the facility.

5.2.3 Lunar Options

The discovery of enhanced concentrations of hydrogen near the lunar poles has been interpreted as signifying the presence of water ice in permanently shadowed craters (Feldman et al, 1998). More recent results seem to demonstrate that much of the hydrogen enhancement is correlated with large shadowed craters, lending credence to the ice interpretation (Binder, personal communication). If the amounts of excess hydrogen are correlated with the shadowed craters, the form of the hydrogen, whether trapped solar wind hydrogen or ice, is not crucial. Elsewhere on the Moon, solar wind hydrogen exists at the 50-100-ppm level, which is thought to be too low to mine economically. However, if energy costs are very low, or if mechanisms can be found to concentrate the finest grain sizes of the regolith that contain the majority of the hydrogen, or if hydrogen production can be integrated with other materials production (e.g. oxygen or ^3He), it may be economically feasible to obtain water anywhere on the Moon.

5.2.3.1 Lunar Polar Ice Deposits

The abundance of ice in the regolith is unknown, but estimated to be on the order of 1-10% (Binder, personal communication). We have assumed that the regolith contains 1% ice, so significant energy is required to heat the regolith to about 100°C for water extraction. An electrolysis plant would split the water and liquefaction systems would produce liquid hydrogen and oxygen. Such a plant could produce approximately 20 times its mass annually (Duke, 1998), including the mass of a nuclear power system. If solar energy is utilized, the plant can operate only about 35% of the time, but the power system mass is less, and the system still can produce many times its mass in water and propellants each year. The solar powered extraction system was baselined for this study. If hydrogen were present instead of ice, it might also be released at low heating temperatures (its excess concentration in the shadowed craters could only be explained if it were weakly bound). If it could be recovered directly as hydrogen, the electrolysis step might be bypassed, with considerable savings of energy. The exploration of the lunar polar regions to establish the nature of the hydrogen or water deposits should be a high priority task in preparation for the Astrotel scenario of Mars exploration.

Most of the power required for propellant production is for electrolysis and liquefaction. Nuclear power systems are favored traditionally, because they tend to have low specific mass (kg/kW) and can work continuously. Because access to near-continuous sunlight is available in certain locations near the poles, and because of recent technology advances in low-cost, high-efficiency flexible solar arrays, solar power generation may be preferred. If photovoltaic systems can be produced on the Moon from local materials (Ignatiev, NIAC study in progress), the cost of energy on the Moon could cease to be a significant issue relative to the cost of other architecture elements.

The scenario for lunar water production developed for the Astrotel program provides for the mining of lunar ice, followed by the electrolysis of enough water to provide liquid hydrogen and oxygen for a launch vehicle to transfer water as a payload to L-1. At L-1, the water would be electrolyzed to produce liquid hydrogen and oxygen for use by the Astrotel vehicles. Some of the propellant produced at L-1 is used to send the water tanker back to the Moon. Electrolyzing water at L-1 rather than producing all propellants on the Moon is chosen because of the relative ease with which water can be transported and the more effective access to solar energy at L-1, which will allow the facility to operate continuously rather than only 35% of the time that the lunar facility operates.

In this option, it is assumed that water ice exists on the Moon in the permanently shadowed craters near the lunar poles. Cold, water-bearing regolith is excavated and transported to a location where solar energy can heat it in a closed chamber to release the water. About half of the water is electrolyzed on the Moon to produce propellant for a reusable Lunar Water Tanker, which transports the other half of the water to L-1. At L-1, the remaining water is electrolyzed and the H_2 and O_2 liquefied and stored in the Taxi preparing for its next flight to the Astrotel and it is used in returning the Lunar Water Tanker to the Moon. This scenario requires the production of about three times as much propellant on the Moon as the useful propellant delivered to L-1.

This option was selected as the baseline for the Phase I study because NASA’s Lunar Prospector has shown significant concentrations of H_2 at both lunar poles and the current favored interpretation is that much of it is in the form of water ice.

5.2.3.2 Regolith Hydrogen and Oxygen

Hydrogen and oxygen are extracted by heating regolith to 900° C, which extracts the solar wind hydrogen (present at 50-100 ppm) and reacts it with ilmenite also in the regolith(Eagle Engineering, 1988). Water is produced in the reaction, most of which is delivered to L-1. Some of the water is electrolyzed to produce hydrogen and oxygen that is stored in propellant tanks for a Moon to L-1 transportation system. Excess oxygen produced in the process is also liquefied and stored for shipment to L-1. Only enough hydrogen is produced to transfer the oxygen to L-1.

Hydrogen is brought from Earth to L1. Some of the terrestrial hydrogen is used with lunar oxygen to transport the vehicle back to the Moon.

This option has not been selected for study during Phase I because of the excessive amount of regolith that must be excavated to produce the required hydrogen quantities. If the lunar polar hydrogen detected by Lunar Prospector is in the form of solar wind hydrogen, the hydrogen concentrations could reach several hundred parts per million. Then, a variant of this option could be exercised.

5.2.4 Summary of Baseline Resource Systems

The subsystem masses required for production of propellant from the three baseline processes are shown in Table 2. The subsystem masses are those required to produce the amount of propellant required annually at Mars, Phobos and L-1 to support the Astrotel architecture. It is assumed that systems operate for 3066 hours/year on Mars, Phobos and the Moon and for 8000 hours/yr at L-1. Footnotes give specific assumptions and/or sources of data.

Table 5-3 Baseline Resource System Masses

Element	Performance	Mars Surface Water	Phobos/ Mars Spaceport		Moon/L-1 Spaceport		Refurb factor (%/yr)
Amount of Propellant Required (kg/yr) (7:1 O_2/H_2)[1]		24,000		78,960	9,270		
Production of water, O_2		21,333		69,090	29,380 (Moon)		
Transport from Earth (kg/yr)				9,870			
Concentration of water/O_2		.01		.34	.01		
			Phobos	Spcpt	Moon	L1	
Excavation and transportation (kg)	Mars: 0.27 mt/mt excavated/hr; Moon: 0.2 mt/mt/hr Phobos: 1 mt/mt/hr	197[2]	70[2]		201[2]		20
Number of furnace tubes		30[3]			30[3]		
Mass of each furnace tube		19.6[4]			9.3[4]		
Total Furnaces + insulation (kg)		1686[5]			450[5]		1
Electrolysis(kg)	20 kg/kg/hr; 6.5 kWh/kg[6]	146.1			130	33.9	10
Carbothermal reactor			2274[7]				10
O_2 Liquefier(kg)	6.5kg/kg/hr; 0.5kWh/kg [6]	42.2	153.8		31.3	9.8	10
H_2 Liquefier(kg)	16.5 kg/kg/hr; 20 kWh/kg [6]	13.4			9.9	3.1	10
Water storage (kg)	Tank mass = 0.01 H_2O mass for L1, Phobos; 0.02 for Moon; 0.04 for Mars[8]	11.7			8.0	33.8	1
O_2 Storage (kg)	Tank mass = 0.07 O_2 mass[9]	327.3	604.5	604.5	0	210.8	1
H_2 Storage (kg)	Tank mass = 0.15 H_2 mass[9]	87.7		185.1	0	56.5	1
Fluids distribution		100	50	50	100	50	
Solar Power system and PMAD (kg)	8 kg/kW Mars, Phobos; 4 kg/kW Moon; L1[10]	1,594.6	3274		556	260.8	1
Total (kg)		**4,205**	**6,426**	**840**	**1,487**	**482**	

Notes to table:

[1]There will be excess oxygen on the Moon, L1 and on Mars, because the assumed oxidizer to fuel ratio is 7:1, rather than 8:1 as in water. This product should have economic value.

[2]Scaled from front-end loader and hauler combination for Moon by Eagle Engineering (1988); front-end loaders scale independently of gravity; hauler mass varies directly with gravity field; however, for Phobos, haulers must move more slowly because of very low gravity. Power for Moon, Mars from Eagle Engineering (1988)

[3] A simple batch-process reactor is assumed that consists of a set of parallel tubes, individually heated, with modest amounts of insulation for Mars, less for the Moon, within a larger cylindrical shell. Batches of material are introduced into the tubes. Material is heated for 4 hours, while water is being removed and condensed. The reactors are assumed to heat the material 600°C on Mars and 250°C on the Moon. No provision is made for thermal recuperation, as all heating is at relatively low temperatures (a trade study is needed for the Martian case to test this assumption). The number of tubes in the reactors is arbitrary. The tubes are assumed to be stainless steel for Mars and aluminum for the Moon.

[4] The diameter of the individual tubes is related to the volume of material that must be processed. It is assumed that the tube lengths are 2m and the volume of the tube is 2 times the volume of the material contained.

[5] The mass of the whole reactor includes insulation for each tube and an outer sheath of insulation that surrounds the assemblage of spherical tubes.

[6] From scaling equations given by Eagle Engineering (1988); Power for tanks estimated assuming reliquefaction of boiloff of 1%/day.

[7] From design of carbothermal reactor to produce 5mt O2/yr by Rosenberg et al (1996). The requirement in this model is for about 5 times the production rate assumed by Rosenberg et al.

[8] Water tankage specific masses estimated: .01 for L1, Phobos; .02 for Moon; .04 for Mars.

[9] M. Jacobs (SAIC, personal communication) estimates for hydrogen and oxygen tank specific masses. Tanks are provided for 90 days of product storage on Mars, at Phobos, and at L1. This is the estimated time that the spacecraft that will utilize the propellants is not at the production facility. Otherwise, propellants are stored in the vehicle that will utilize them. It may be more effective to turn off propellant production when the vehicles are not at the production site. This will increase the size of the production facility, but total system mass would be smaller. On the Moon, where the Moon-L1-Moon trips are of short duration and the transfer vehicle can be docked at the production facility, no additional storage of O2 and H2 is included.

[10] K. Nock (personal communication). Flexible solar array sheets on plastic film. Assumed capable of <4 kg/kW for full illumination at Moon or L1; 8 kg/kW at Mars and Phobos.

Note that the total propellant production rates shown above are not yet consistent with the actual Taxi propellant requirements shown earlier because these efforts were carried out in parallel; i.e. the resource system requirements were based on early estimates. Later, the resource requirements will be matched with the propellant requirements.

5.3 Resource System References

1. Duke, M. B. (1998) On the Discovery of Hydrogen (or Ice) on the Moon and The Implications for Solar Power Satellite Programs, IAF-98-R2.07, International Astronautical Federation, Paris.

2. Eagle Engineering (1988) Conceptual Design of a Lunar Oxygen Pilot Plant, Eagle Engineering, Inc. Contractor Report 88-182 to Johnson Space Center NAS9-17878.

3. Feldman, W. C., S. Maurice, A. B. Binder, B. L. Barraclough, R. C. Elphic, and D. J. Lawrence (1998) Fluxes of Fast and Epithermal Njeutrons from Lunar Prospector: Evidence for Water Ice at the Lunar Poles, Science 281, p. 1496.

4. Muff, T., R. H. King, and M. B. Duke (2000) Preliminary Analysis of a Small Robot for Martian Regolith Excavation, in Space Resources Roundtable II, LPI contribution 1019, Lunar and Planetary Institute, Houston, TX.

5. Rosenberg, S., Hermes, P. and Rice, E. E., (1996) Carbothermal Reduction of Lunar Materials for Oxygen Production on the Moon: Five Metric Tonne per Year Lunar Oxygen Plant Design Study, AIAA-96-07, American Institute of Aeronautics and Astronautics, Washington, D. C., 31pp.

6. Thomas, P., J. Veverka, J. Bell, J. Lunine, and D. Cruikshank (1992), Satellites of Mars: Geologic History, in Kieffer, H. et al (eds.) Mars, University of Arizona Press, Tucson, pp. 1257-1282.

6 Conceptual Architecture Design Description

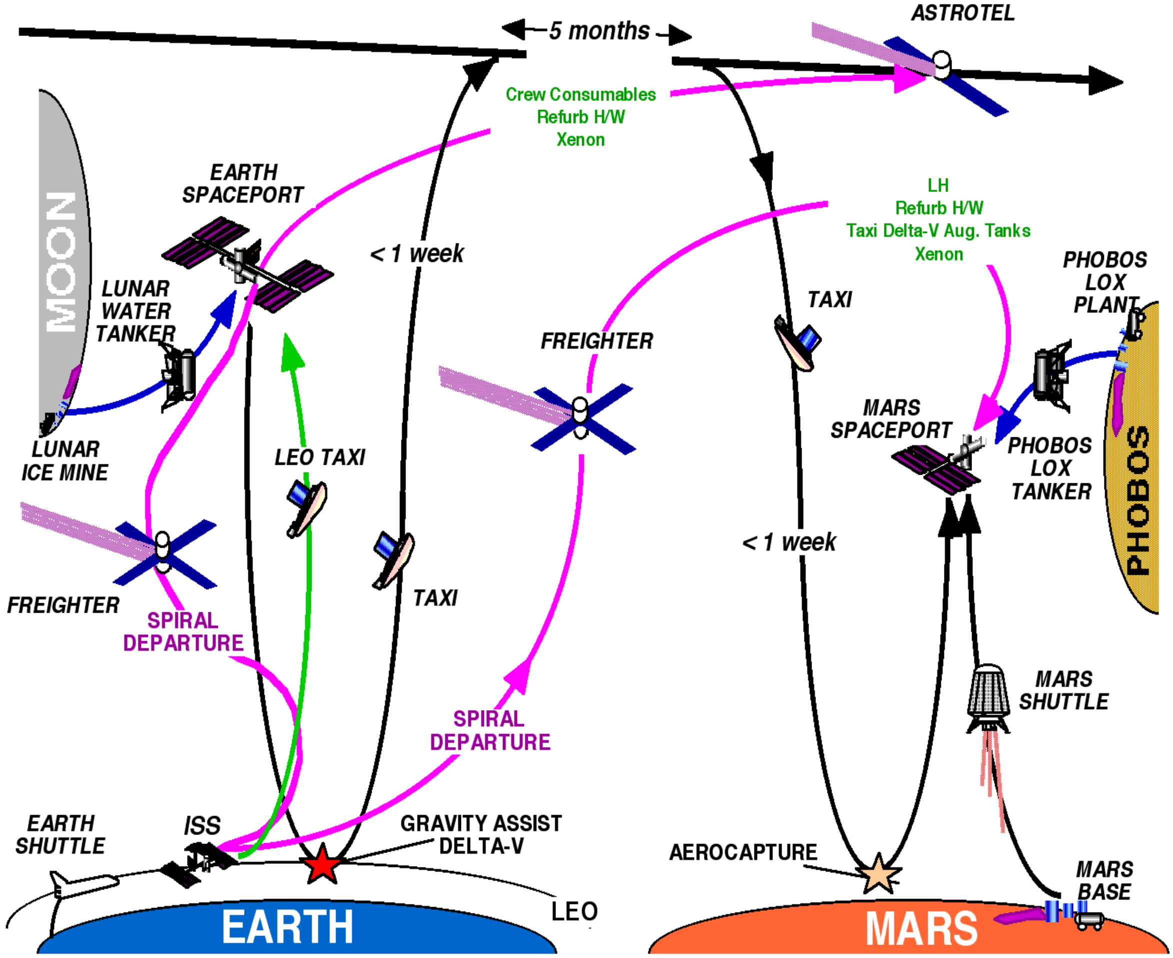
6.1 Introduction

This section describes the baseline architecture design for cyclic visits to Mars via astronaut hotels. This baseline design meets the requirements specified in Section 3 using technologies described in Section 4 and is based, in part, on MAMA analysis (Section 6) of various architecture and subsystem options discussed in Section 5.

We begin with a summary of the concept and proceed to discuss orbit design, surface bases, spaceports, Astrotels, Taxis, Tankers, Mars Shuttles, Cargo Freighters and in situ resource utilization.

6.2 Summary

The following chart summarizes the current overall architecture for the Mars transportation system.



The systems architecture consists of a Mars Base; a Lunar Base; two Astrotels, one each in the Up and Down Escalator orbits (Down Escalator Astrotel not shown); two Spaceports one at Earth at L-1 and the other at Mars in a Phobos–like orbit; a Space Station in LEO (shown as the ISS) that is supplied by an Advanced Space Shuttle and a low-cost heavy lift launcher; a LEO Taxi to transport crew to and from the Space Station to the Earth Spaceport; propellant manufacturing facilities on the Moon (Ice Mine), at the Earth Spaceport (water electrolysis, LOX/LH liquefaction and storage), Phobos (LOX Plant), Mars Spaceport (LOX/LH storage) and Martian surface (Sand Dune Water Mine, LOX/LH production and storage); two Taxis transporting crew to and from Astrotels and Spaceports; a Mars Shuttle to carry crew and refurbishment cargo to the surface of Mars; a Lunar Water Tanker and Phobos LOX Tanker to carry resources to the Spaceports; and at least four SEP-powered Cargo Freighters (two shown) to transport consumables, refurbishment hardware and Xenon to the Astrotels and refurbishment hardware, Taxi propellant augmentation tanks and LH to the Mars Spaceport.

Figure 6-1 Mars Transportation Architecture Schematic

The following table summaries the key features of the various vehicles in terms of vehicle type, propulsion type, current design heritage, purpose, location or nodes serviced, delta-V capability, reusability and dry mass. Those systems that have been studied is some detail in Phase I are shown in **bold** type.

Table 6-1 Mars Transportation Systems Summary

Systems	Vehicle Type	Propulsion	Heritage	Primary Purpose	Location or Nodes
Vehicles and Surface Systems					
Astrotel	**Space**	**SEP**	**Transhab**	**Crew Transport**	**Mars/Earth**
Escape Pod	Aero-assist	Chem	X-38	Crew Escape	Astrotel to Planets
Earth Spaceport	Space	SEP	Astrotel	Crew & cargo transfer/storage & Propellant production & storage	L-1
Mars Spaceport	Space	SEP	Astrotel	Crew & cargo transfer/storage & Propellant storage	Phobos Orbit
Taxi	**Aero-assist**	**LOX/H**	**Apollo/AOTV**	**Crew Transport**	**Astrotel/Spaceports**
Mars Cargo Freighter	**Space**	**SEP**	**Astrotel propulsion**	**Fuel, refurb cargo to Mars Spaceport**	**LEO/Mars Spaceport**
Astrotel Cargo Freighter	**Space**	**SEP**	**Astrotel propulsion**	**Consum, fuel, refurb cargo to Astrotel**	**LEO/L-1/Astrotel**
Space Station	Space	Chem	ISS	Crew, cargo, propellant transfer	LEO
Space Shuttle	Aero-assist	SRM/LOX/H	Advanced STS	Crew to LEO Space Station	Earth Surface/LEO
HLLV or Magnum	Surface L/V	SRM/LOX/H?	??	Consum, H , Refurb cargo to LEO	Earth Surface/LEO
LEO Shuttle	Aero-assist	LOX/H	Taxi	Crew, Consum, H , Refurb cargo to LEO	LEO/L-1
Lunar Water Tanker	**Space**	**LOX/H**	**Repic, 1992**	**Lunar water to L1**	**Lunar Surface/Earth Spaceport**
Mars Shuttle	**Aero-assist**	**LOX/H**	**NCOS Mars Shuttle**	**Crew and refurb hardware to/from Mars Surface**	**Mars Surface/Mars Spaceport**
Phobos LOX Tanker	Space	LOX/H	Lunar Water Tanker	Propellant transport	Phobos/Mars Spaceport
Mars Base	**Surface**	**-**		**Crew accomodation and science**	**Mars Surface <30 Lat**
Lunar Base	Surface	-		Science and support water mine	Lunar S. Pole
Resources Systems					
Lunar Water Mine				**Mine Lunar ice and extract water**	**Lunar south pole**
L1 Water Elect/Cryo/Strg				**Electrolyze Lunar water, liquefy and store LOX/LH**	**L-1 near Earth Spaceport**
Phobos LOX Plant				**Mine roglith and extract, liquefy and store LOX**	**Phobos surface**
Mars Surface Water Plant				**Mine roglith and extract, liquefy and store LOX/LH**	**Near Mars Base**
Mars Spaceport LOX/LH Storage				**Store Phobos LOX and Earth LH**	**Near Phobos**

6.3 Orbit Design Parameters

6.3.1 Cyclic Orbits

SAIC generated 15 years (2011-2026 opportunities were used and are typical of later opportunities) of Aldrin Up and Down Cycler conic trajectory optimization runs describing planetary flyby dates, altitudes, V-Infinities, and transfer time between planets. The key parameter that was optimized was total mission delta-V over the 15-year sequence. The simplified analysis is conic in nature, which means that between planets only the gravity of the

Sun is considered and at the flybys only the mass of the planet is considered, hence the orbits are all *conic* sections (hyperbolas and ellipses). The mid-course (M/C) delta-Vs are shown taking place near the apoapsis of the orbits where impulsive delta-Vs would be most effective. As discussed earlier, these delta-Vs are required to rotate the interplanetary orbit to ensure repeated encounters. In practice, these delta-Vs will be carried out using low-thrust propulsion systems (Xenon ion propulsion) at optimum times during the orbit. The minimum swingby altitude was set at 200 km above planetary surfaces, still about 75 km above the sensible atmosphere for both planets. These data are shown in the following tables.

Table 6-2 Aldrin Outbound (Up Escalator) Cycler

Planet	Date	Flight-time, days	V-Infinity, km/s	Astrotel Flyby Altitude, km
Earth	13-Nov-11	start	5.808	5,740
Mars	24-Apr-12	163	10.142	17,444
Earth	18-Dec-13	603	5.854	3,699
Mars	18-May-14	151	11.449	55,595
Earth	26-Jan-16	618	5.902	1,180
Mars	16-Jun-16	142	11.488	7,586
M/C Δ V	***18-Apr-17***	***306***	***0.211***	***Δ V1***
Earth	17-Mar-18	333	5.791	200
Mars	9-Aug-18	145	8.914	683
M/C Δ V	***20-Jun-19***	***315***	***0.679***	***Δ V2***
Earth	3-Jun-20	349	5.765	200
Mars	10-Nov-20	160	5.682	3,387
M/C Δ V	***18-Aug-21***	***281***	***0.671***	***Δ V3***
Earth	22-Aug-22	369	5.831	4,803
Mars	24-Jan-23	155	7.142	25,372
Earth	26-Sep-24	611	5.961	4,516
Mars	14-Mar-25	169	8.156	13,279
Earth	1-Nov-26	597	5.946	5,740

Table 6-3 Aldrin Inbound (Down Escalator) Cycler

Planet	Date	Flight-time, days	V-Infinity, km/s	Astrotel Flyby Altitude, km
Earth	22-May-10	start	5.978	5,102
Mars	8-Jan-12	596	9.102	13,506
Earth	25-Jun-12	169	5.954	5,498
Mars	15-Feb-14	600	7.744	19,152
Earth	4-Aug-14	170	6.102	1,964
M/C Δ V	***16-Sep-15***	***408***	***0.046***	***Δ V1***
Mars	19-Apr-16	216	7.221	147,195
Earth	15-Sep-16	149	6.207	200
M/C Δ V	***7-Oct-17***	***387***	***0.626***	***Δ V2***
Mars	11-Jul-18	277	6.826	1,491
Earth	8-Dec-18	150	5.963	200
M/C Δ V	***10-Jan-20***	***398***	***0.933***	***Δ V3***
Mars	24-Sep-20	258	9.781	472
Earth	18-Feb-21	147	5.750	338
Mars	9-Nov-22	629	11.936	51,804
Earth	30-Mar-23	141	6.010	2,041
Mars	1-Dec-24	612	11.114	20,810
Earth	28-Apr-25	148	5.528	5,102

6.3.2 Low-thrust Analysis

6.3.2.1 Astrotel Orbit Corrections

SAIC generated the low-thrust trajectory analysis of SEP performance of 15 years of Up and Down Cyclers. The Astrotel SEP system carries out all propulsion maneuvers of the vehicle including the three large delta-Vs three orbits out of seven. The following tables describe the SEP performance requirements for two cases: a) near-optimum power for maximum payload and b) near-minimum power. Key assumptions for the analysis include 1) specific mass of the combined power and propulsion system is 8 kg/kW, 2) 15% tankage and reserve mass and 3) 75 mt initial vehicle mass. For 8 kg/kW specific mass, the power and propulsion system is only 1200 kg for a 150 kW_e system design. Propellant mass is about 3 mt for an ion engine operating at 5000 s specific impulse. The total propellant mass requirement results in a requirement for Xenon delivery to the Astrotel of only about 430-kg average each orbit. Propulsion on time is consistent with projections of ion propulsion technology.

Table 6-4 SEP Performance Requirements for Aldrin Up (Outbound) Cycler Over 15 years

SEP Performance Requirements for Correcting the Aldrin Outbound Cycler Over 15 years

Assume: Power/Propulsion System Specific Mass = 8 kg/kWe
Propellant Tankage & Reserve Fraction = 15%
Initial Mass = 75,000 kg

Po (kWe @ 1AU)	Isp (sec)	Mps† (kg)	ΣMp (kg)	0.15*ΣMp (kg)	Mpayload (kg)	Propulsion On-Time 3 Corrections (days)		
* Near-Optimal Power for Maximum Payload								
150	2000	1200	5813	872	67,115	74	219	222
150	3000	1200	4044	607	69,149	91	273	279
150	4000	1200	3184	477	70,139	113	345	351
150	**5000**	**1200**	**2767**	**415**	**70,618**	**135**	**430**	**438**
* Near-Minimum Power								
60	2000	480	8847	1327	64,346	190	628	649
75	3000	600	6149	922	67,329	183	624	649
98	4000	784	4589	688	68,939	192	589	640
120	5000	960	3506	526	70,008	190	576	605

Table 6-5 SEP Performance Requirements for Aldrin Down (Inbound) Cycler Over 15 years

SEP Performance Requirements for Correcting the Aldrin Inbound Cycler Over 15 years

Assume: Power/Propulsion System Specific Mass = 8 kg/kWe
Propellant Tankage & Reserve Fraction = 15%
Initial Mass = 75,000 kg

Po (kWe @ 1AU)	Isp (sec)	Mps (kg)	ΣMp (kg)	0.15*ΣMp (kg)	Mpayload (kg)	Propulsion On-Time 3 Corrections (days)		
* Near-Optimal Power for Maximum Payload								
150	2000	1200	5645	847	67,308	16	216	276
150	3000	1200	4001	600	69,199	20	269	352
150	4000	1200	3329	499	69,972	25	335	460
150	**5000**	**1200**	**3007**	**451**	**70,342**	**30**	**410**	**552**
* Near-Minimum Power								
70	2000	560	8153	1223	65,064	37	484	637
85	3000	680	5633	845	67,842	38	495	648
109	4000	872	4200	630	69,298	38	475	640
120	5000	1030	3338	501	70,131	36	461	635

As a result of these analyses it was decided to select the 150 kW, 5000 s I_{sp} case for the baseline Astrotel SEP system. The baseline case is denoted by boldface type in the table above. Figure 6-2 shows orbit geometry for the low thrust maneuvers for the second correction of the Up Cycler orbit. It can be seen that the low thrust orbit correction takes a substantial portion of the orbit, 430 days, as opposed to the impulsive correction mode.

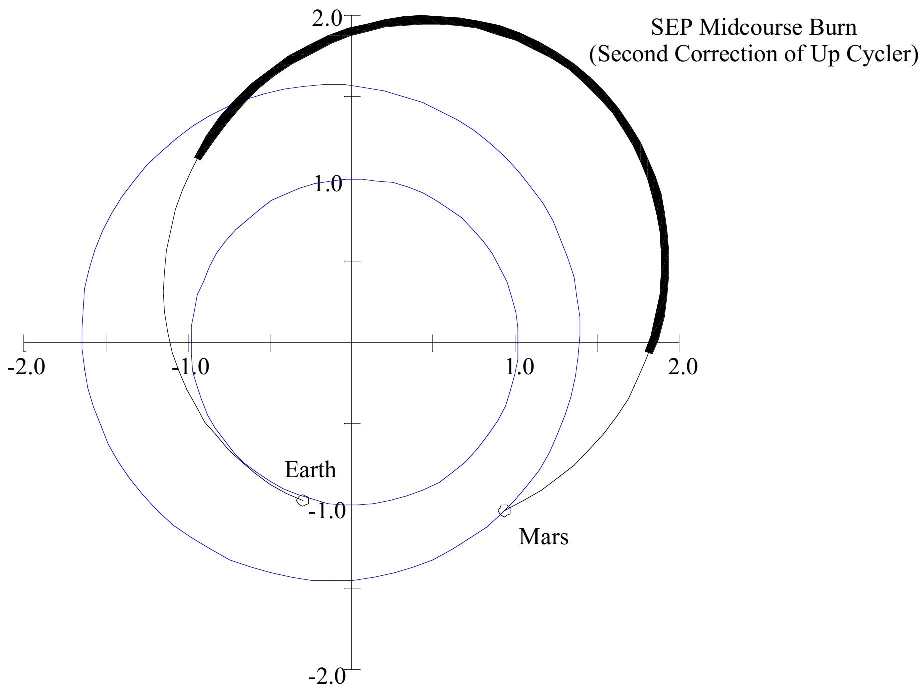


Figure 6-2 SEP Orbit Correction on Up Cycler Orbit

6.3.2.2 Reusable Cargo Freighter Orbit Analysis

SAIC has developed the analysis of SEP trajectories from LEO to both the Astrotel and the Mars Spaceport near Phobos. Cargo Freighters that deliver cargo to both cycling Astrotels and the Mars Spaceport will use these trajectories.

6.3.2.2.1 Expendable Cargo Freighter (One-way Astrotel resupply)

One-way SEP rendezvous trajectories of various flight times (500-800 days in heliocentric transfer) were generated for the end of the nominal cycle (i.e. the 7th sortie of the Outbound Astrotel). The results are representative of the Inbound Astrotel as well since the orbits are about the same. It was assumed that the transport vehicle would begin its mission in LEO (1000 km altitude) and spiral out to parabolic escape energy (C_3 = 0) before starting the heliocentric rendezvous phase, and not be reusable at end of mission. Performance is quite flat over heliocentric flight times of 650 to 800 days, but 700-day transfers appear to be best. Launch from LEO occurs about 12 months before the Astrotel’s Earth gravity-assist. Rendezvous occurs on the Astrotel's subsequent inbound leg (i.e. after the Mars gravity-assist and aphelion) at 1.59 AU and 5 months before the Astrotel's next Earth gravity-assist. The propellant-to-initial mass and payload-to-initial mass ratios are 0.2435 and 0.6399, respectively, at a power-to-initial mass ratio of 10 Watts/kg at 1 AU and operating Isp = 5000 sec. For comparison, the Astrotel's propulsion system propellant-to-initial mass and payload-to-initial mass ratios are about 0.037 and 0.914 at a power-to-mass ratio of 2.1 W/kg, with the same Isp. For reference, the analysis assumed transporting all the cargo required for 15 years of operation of the Astrotel. In actuality this approach is impractical (especially for the Mars Cargo Freighter) because of the large size of cargo on one trip, not to mention the cargo storage requirements at the destinations, and the large size of the SEP system required.

6.3.2.2.2 Reusable Cargo Freighters (Round-trip Astrotel resupply)

A more practical option is the transport of the cargo on reusable Freighters. SAIC analyzed this option for the Outbound Astrotel orbit. For the example opportunity studied, the Earth launch date is October 11, 2023 with return to Earth on May 14, 2027. This analysis resulted in a flight time (including planetocentric spirals of 966 days) to the Astrotel, only 335 days back to LEO for total round-trip flight time from LEO of 1311 days. Total propulsion on time is 608 days and there is an Earth turnaround time of 304 days before departure to the Astrotel again. For this analysis, total assumed payload mass delivered in 15 years was set at about 88 mt. Scalable mass fractions at the optimal power-to-initial mass ratio of 10 W/kg at 1 AU were 0.6016 for the payload-to-initial mass ratio and 0.2769 for the propellant-to-initial mass ratio. The following table describes the initial, payload, propellant and propulsion/power system masses and the solar array size as a function of number of sorties to the Astrotel in 15 years given the assumed cargo to be transported.

Table 6-6 Astrotel Cargo Freighter Parameters vs Number of Sorties in 15 years

# of Sorties	P_o (kW_e @ 1AU)	M_o (kg)	M_{ps} (kg)	ΣM_p (kg)	0.15* ΣM_p (kg)	M_{cargo} (kg)
7	211	21,104	1,688	5,843	877	12,696
6	246	24,621	1,968	6,818	1,023	14,812
5	295	29,545	2,360	8,183	1,228	17,774
4	369	36,932	2,952	10,228	1,534	22,218
3	492	49,240	3,936	13,636	2,045	29,623

For cargo transport to the Mars Spaceport analysis the earth launch date is January 5, 2011 with Earth return date of October 26, 2014. This analysis resulted in a flight time to Mars of 753 days (including a 263 day Earth escape spiral and 90 Mars capture spiral). A stay time of 178 days at the Mars Spaceport days was assumed followed by flight time of 459 days back to LEO (including a 22 day Mars escape spiral and 37 day Earth capture spiral). Total round-trip flight time from LEO is 1390 days. Total propulsion on time is 733 days and there is an Earth turnaround time of 140 days before departure to the Mars Spaceport again. For this analysis, total derived payload mass delivered in each opportunity was between 6-17 mt. Scalable mass fractions at the optimal power-to-initial mass ratio of 10 W/kg at 1 AU were 0.5753 for the payload-to-initial mass ratio and 0.2997 for the propellant-to-initial mass ratio. The following table describes the initial, propellant and propulsion/power system masses and the solar array size as a function of cargo mass to the Mars Spaceport delivered in each opportunity. Note this table only goes to a 30 mt payload level, but the results are directly scalable to higher payload mass values. In addition, these numbers assume all structure and spacecraft bus avionics are included in M_{ps}.

Table 6-7 Mars Cargo Freighter Parameters as a Function of Cargo Mass

P_o (kW_e @ 1AU)	M_o (kg)	M_{ps} (kg)	ΣM_p (kg)	0.15*ΣM_p (kg)	M_{cargo} (kg)
100	10,000	800	2,997	450	5,753
150	15,000	1,200	4,496	674	8,630
200	20,000	1,600	5,994	899	11,507
250	25,000	2,000	7,492	1,124	14,384
300	30,000	2,400	8,990	1,349	17,261

The nominal plan is to have two Astrotel Cargo Freighters for the Astrotels (one for each Astrotel), each delivering cargo every other opportunity and two to four Cargo Freighters for the Mars Spaceport (one or two leaving every opportunity). This plan ensures a cargo delivery every Mars opportunity and every other opportunity for each Astrotel thus potentially building in some reliability and margin.

6.3.3 Taxi Delta-V Requirements

The delta-V requirements have been determined for the Taxi vehicles. The assumptions are that the Spaceport at Earth is at L_1 and in Phobos orbit at Mars. In addition, aerocapture is assumed for Taxi orbit capture from an Astrotel at both Mars and Earth.

Leaving an Astrotel, there is a relatively small deflection delta-V required of the Inbound Taxi to target it to the planetary atmosphere at 125–km altitude, which is entry interface. The second delta-V occurs after the aerocapture maneuver to rendezvous with the Spaceport. There are small delta-V factors included for expected orbit plane changes.

Outbound Taxis required a delta-V in order to depart the Spaceport and lower periapsis to about 200-km close approach altitude. At closest approach, where it is most efficient, a delta-V is performed that injects the Taxi onto a hyperbolic intercept trajectory toward the Astrotel. At Astrotel rendezvous, a small delta-V is required to match the Taxi velocity with that of the Astrotel. Phasing between the Taxis, Astrotel and Spaceport may require multiple orbits of the primary planet in order to sync up the trajectories. The delta-Vs shown in the next two tables include all the appropriate delta-Vs that have been discussed here.

Table 6-8 Taxi Delta-V for Up Cycler Rendezvous

Encounter	Date	ΔVin (m/sec)	ΔVout (m/sec)	Flyby Altitude (km)
Earth	13-Nov-11	9	2,473	5,740
Mars	24-Apr-12	749	9	17,444
Earth	18-Dec-13	0	2,490	3,699
Mars	18-May-14	812	0	55,595
Earth	26-Jan-16	0	2,508	1,180
Mars	16-Jun-16	733	0	7,586
Earth	17-Mar-18	0	2,454	200
Mars	9-Aug-18	721	0	683
Earth	3-Jun-20	0	2,442	200
Mars	10-Nov-20	726	0	3,387
Earth	22-Aug-22	0	2,482	4,803
Mars	24-Jan-23	762	0	25,372
Earth	26-Sep-24	0	2,542	4,516
Mars	14-Mar-25	742	0	13,279
Earth	1-Nov-26	0	2,538	5,740

Table 6-9 Taxi Delta-V for Down Cycler Rendezvous

Encounter	Date	Taxi ΔVin (m/sec)	Taxi ΔVout (m/sec)	Flyby Altitude (km)
Earth	22-May-10	928	0	5,102
Mars	8-Jan-12	0	6,897	13,506
Earth	25-Jun-12	929	0	5,498
Mars	15-Feb-14	0	5,732	19,152
Earth	4-Aug-14	922	0	1,964
Mars	19-Apr-16	0	5,506	147,195
Earth	15-Sep-16	918	0	200
Mars	11-Jul-18	0	4,940	1,491
Earth	8-Dec-18	918	0	200
Mars	24-Sep-20	0	7,478	472
Earth	18-Feb-21	918	0	338
Mars	9-Nov-22	0	9,528	51,804
Earth	30-Mar-23	922	0	2,041
Mars	1-Dec-24	0	8,719	20,810
Earth	28-Apr-25	928	0	5,102

As seen in the preceding tables, the Taxi delta-Vs departing Earth toward the Up Escalator are modest at about 2.5 km/s. However, the Taxi delta-Vs departing Mars are quite substantial at 5-10 km/s. These large delta-Vs will require staging of Taxi propulsion systems (expendable propellant augmentation tanks [PATs]) for delta-Vs over 3.4 km/s. In addition to the PATs, separate larger engines may be required for the first stage of the Taxi vehicle for delta-Vs larger than about 6.6 km/s in order to reduce burn time and subsequent gravity losses to an acceptable level.

6.3.4 Spaceport Locations

A primary criterion for selecting a Spaceport location is total transportation architecture propulsion requirements. The existence and proximity of *in situ* resources needed in the transportation system will drive these requirements. Very clearly the overall propulsion requirements for a Spaceport in an Earth-Moon cycler orbit are smallest. However, there are the phasing penalties between the Earth Spaceport and Astrotel in terms of flighttime and delta–V between interplanetary and Earth-Moon cycler orbits, which do not exist for the L_1 location. The optimization of Spaceport location at Earth and Mars is an area of future proposed study in

Phase II. Options for Earth Spaceport locations, and their propulsive delta-V requirements for staging to interplanetary cyclers, are shown below in Table 6-10 [Section 1 Ref. 2].

Table 6-10 Delta-V Requirements for Staging in Earth Moon Space, m/s

Location	Trans Mars* Injection	From LEO	From Lunar Surface
LEO	4470	–	2670
GEO	3540	3820	3520
Earth-Moon L_1	2050	3670	2510
Earth Moon Cycler	1408	3058	2550
Lunar Orbit	2230	3880	1730
Lunar Surface	3960	5610	–

* - Escape energy of 30 $(km/s)^2$ typical of Aldrin Up Cycler rendezvous.

6.4 Lunar Base

A human outpost near a lunar pole might be established for several reasons, including the scientific exploration of the Polar Regions, the development of lunar ice deposits, or preparation for the exploration of Mars. However, the scale of propellant production that is required for the Astrotel scenario is relatively small and could possibly be carried out entirely by robotic means. Access to humans at a lunar base would be helpful for maintenance purposes, but should be a small increment to a lunar base operated for other purposes. The water, oxygen and hydrogen production facilities that are required for the Astrotel scenario would be complementary to the needs of the lunar outpost for life support and propellant.

6.5 Mars Base

Studies of human exploration of Mars that were carried out by the National Commission on Space (NCOS) in 1985 established several features of a permanent, sustained Mars base of roughly 2035. We shall use the key elements of that base concept as the starting point for our study of an innovative transportation system that could support its logistical requirements [Section 1 Ref. 1].

The level of capability envisioned supports significant surface activities in the areas of science exploration, resource surveys, life cycle maintenance, propellant production, and materials processing and fabrication. These activities would take place at one or two fixed-site facilities on Mars and on distant traverses from base. Such operations would require a high degree of mobility, appropriate levels of automation with efficient man-machine interfaces, and they require crews that combine the need for individual specialization with job sharing abilities. An artist illustration of the NCOS Mars base is shown in Figure 6-3. A crew complement averaging 20 persons on the Martian surface will carry out these activities; the resident population at any time could fluctuate substantially from the average depending on the phase of the crew rotation cycle dictated by the interplanetary transportation options. It is assumed that half the crew is exchanged at any one time.

Figure 6-3 Artist's conception of a Mars base c. 2035

In addition to the Mars surface facilities, the overall Mars base infrastructure could include (1) *in situ* resource production plants on the Martian surface and at Phobos, (2) a spaceport near Phobos serving as a work station and a transportation hub, and (3) several shuttle transports operating between the Martian surface and the spaceport. The Mars surface is assumed to be permanently inhabited in this scenario thereby providing staggered crew rotations and, thus, overlap between "experienced" and "fresh" personnel.

Self-sufficiency and *in situ* resources utilization would be the major focus of the Mars base and dominate much of the crew activity and attention. The environmental control and life support systems would be regenerative to a large degree but not entirely closed. Supplies of oxygen, water and carbon dioxide can be extracted from the soil and atmosphere. Plants of different varieties will be grown in greenhouse enclosures while other types of food will be produced using such methods as aquaculture. The operational objective would be a bioregenerative system with minimal replenishment from Earth. This same objective is true for propellant resources used for long traverse mobility on the surface and in the atmosphere, for rocket vehicle transportation between a Mars base and a Mars Spaceport, and for refueling the transportation vehicle systems used in departure to Earth return. To support such a base at Mars, a means of transporting crews and equipment between the planets is needed. It is the crew and logistical support to this base that is the subject of this Phase I study. The following table summarizes the content and system masses of the Mars Base.

Table 6-11 Mars Base Equipment and System Masses

Systems	#	Unit Mass, mt	Total Mass, mt	15-Year Refurbish-ment Mass, %	15-Year Refurbish-ment Mass, mt	Source of Estimate
Life Critical Systems						
Habitat	4	38.5	154.0	12%	18.5	Transhab-based system, JSC Ref v1.0
Washdown facility	2	0.9	1.8	10%	0.2	NCOS
			155.8			
Mission Support Systems						
120 kW Power Source (solar array @100W/kg)	2	1.2	2.4	100%	2.4	WAG
Power Management, Distribution and Maintenance	2	0.3	0.6	30%	0.2	WAG
Energy Storage (RFC packages)	2	1.1	2.2	100%	2.2	WAG
Suitup/Maintenance Facility	2	1.8	3.6	20%	0.7	NCOS
Pressurized Transporter	3	9.1	27.3	15%	4.1	NCOS
Open Rovers	3	1.0	3.0	10%	0.3	NCOS
Inflatable Shelter w/Airlock	10	0.5	5.0	50%	2.5	NCOS
Communication Satellites	3	0.8	2.4	100%	2.4	WAG
Crane	2	5.0	10.0	10%	1.0	NCOS
Trailer	2	2.0	4.0	10%	0.4	NCOS
			60.5			
Science and Exploration Systems						
Base Laboratory	2	13.6	27.2	20%	5.4	NCOS
Mobile Laboratory	3	9.1	27.3	20%	5.5	NCOS
200 m Drill	1	2.3	2.3	10%	0.2	NCOS
10 m Drill	3	0.1	0.3	100%	0.3	NCOS
UAV	3	0.3	0.9	100%	0.9	NCOS
Robotic Rovers	10	0.2	2.0	100%	2.0	Athena 2003
Weather Station	5	0.2	1.0	30%	0.3	NCOS
Survey Orbiters	2	0.8	1.6	100%	1.6	WAG
			62.6			
Total			**278.9**		**51.0**	
Total Brought to Mars Surface					**48.6**	
Total Brought to Mars Spaceport					**2.4**	
Refurb Mass per Opportunity					**6.9**	

6.6 Spaceports

Spaceports are collection points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels. In past architectures such Spaceports were large, rotating, permanently crewed platforms. In this new concept, a Spaceport is based on the Astrotel design philosophy. Crew stay times would be limited in order to minimize effects of zero-g. Crew maintenance is minimized by maximum application of autonomy in order to shorten stay times. Stationkeeping, orbit corrections, orbit-phasing delta-Vs could easily be performed by the same or even smaller SEP system envisioned for the Astrotels. While the detailed design of Spaceports is deferred to Phase II we have assumed Astrotel refurbishment mass requirements, since there is considerable commonality with the Astrotel.

6.7 Astrotels

This new Mars transportation system architecture concept uses small, highly autonomous space ships, we dub *Astrotels*, for transporting humans to and from Earth and Mars on cyclic or near-resonant orbits between these planets. Human flight time each way is reasonably short at between 5 and 6 months.

6.7.1 Astrotel Design Description

Key elements of these ships are that they are highly autonomous and transport only human and other high value cargo, use highly efficient solar electric propulsion, and not require artificial gravity. These features reduce the size of these vehicles from over the 460 mt found in the NCOS studies [Section 1 Ref.1] to about 70 mt including SEP systems. Reducing its mass significantly reduces the total propulsive energy budget required for course corrections to the 2767-kg propellant required for all major corrections over 15 years. The 70 mt mass includes a habitability module for a crew of ten. The size and volume of this system would provide a crew volume of about 6-times that available to today's Space Shuttle crew (7). The astronaut living space is a three-story structure patterned after the *TransHab* modules currently under study by NASA. Figure 6-4 is a schematic of one concept for an Astrotel that is approaching Mars. The two smaller modules between the TransHab and the solar array are cargo bays. The Astrotel Cargo Freighter autonomously delivers all cargo to the Astrotel contained within a standard cargo bay. These are pressurized modules to facilitate crew unloading of consumables and RRU hardware. Once emptied the cargo bay could be discarded or used to provide added crew volume.

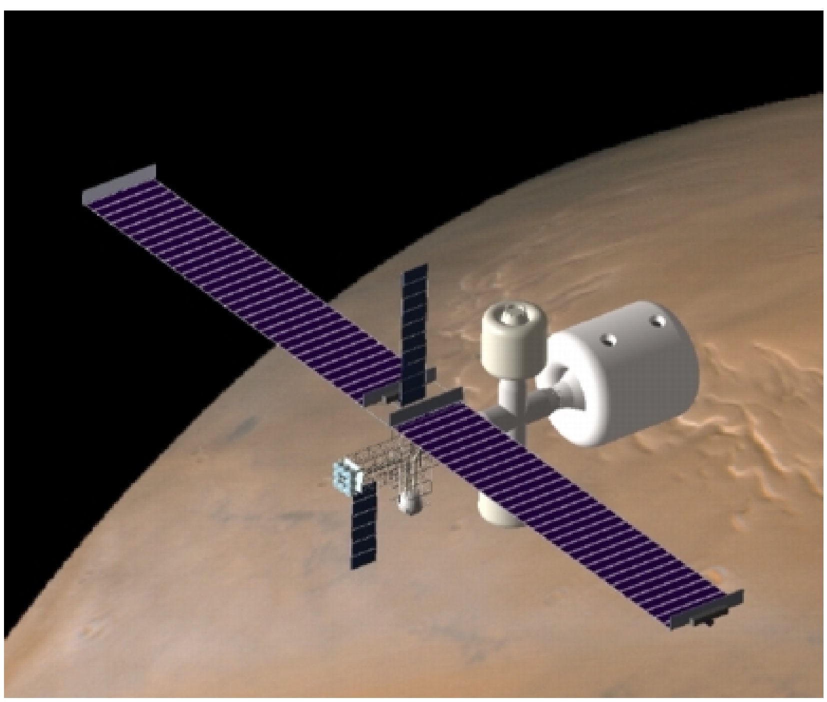


Figure 6-4 Computer Design of one Concept for an Astrotel

6.7.2 Astrotel IPS Module

During three orbits out of seven in 15 years, the Astrotel orbit will need to undergo modification by use of its ion propulsion system. The low thrust analysis presented earlier optimized the maneuvers by timing them for minimum propellant usage, not necessarily near the optimum impulsive delta-V at the orbit aphelion. In addition, since propellant mass has a cost associated with it, the focus is on the maximum payload cases, in particular the 150 kW, 5000 s I_{sp} case. This case was best achieved by an ion propulsion system. The required propellant supply averages only about 400 kg per cycle to meet the 2767-kg propellant required for 15 years of corrections. Xenon has the advantages of being inexpensive, easy to store, and having considerable ion propulsion experience. A 644 kg, Xenon propellant, ion propulsion system (IPS) thruster system is included in the Astrotel design. Xenon ion thrusters are situated at one end of the Astrotel in order to facilitate pointing the thrust vector toward the center of gravity of the system. The ion propulsion system is based on the scaling presented in Section 4.3.2.2. The following figure shows the details of the propulsion system including the eight 50-cm engines, radiators, and the xenon propellant tank. Not shown, but located behind the thrusters is the power processing electronics for the ion propulsion system. The thermal radiator assembly for the IPS is shown below the thruster assembly and oriented 90° to the direction to the Sun. The 3 m^3 volume xenon tank is sized to contain all the propellant required for the 15 years cycle.

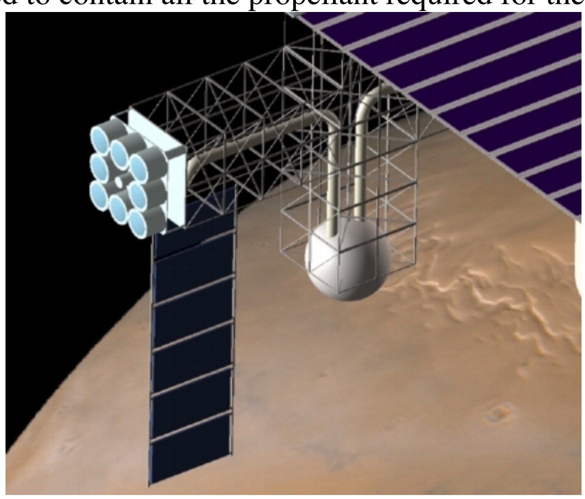


Figure 6-5 Astrotel Propulsion Module

6.7.3 Astrotel Solar Array

As discussed earlier, various options exist for solar array cell selection. At this point we have specified the use of multi-component, mechanically stack solar cells in a concentrator solar array configuration at a specific power of about 250 W/kg (or ~4 kg/kW). We assumed a 10 kW requirement in addition to the propulsion power requirement for a total array size of 160 kW. During crew habitation the ion propulsion system will usually be off allowing significant reserve power for crew activities. It is estimated that such a solar power generation system will mass about 785 kg and require about 384 m^2 of area, which could be accommodated in two separate deployable array panels each 6-m wide by 32-m long. The solar arrays can be articulated ±90° along their long axis in order to track the sun while orienting the propulsion system thrust vector optimally.

6.7.4 Astrotel Mass Summary

The total estimated Astrotel mass is about 70 mt, which includes about 28 mt of hab module, propulsion/power systems, and cargo storage plus about 32 mt for reserve, radiation shielding and an escape pod. The mass breakdown, in Table 6-12, will be further developed in the proposed Phase II effort as the radiation shielding requirements and the need and requirements for an escape pod are evaluated.

The Astrotel sizing was evaluated based on the revised JSC Mars reference mission systems designs. It is based on the JSC Reference Mission Version 3.0 (JSC Adv. Dev. Off. Report #EX13-98-036, June 1998) Earth Return Vehicle sizing. The JSC mass numbers were combined with the 160 kW power and IPS SEP subsystem. The JSC Mars reference mission consumables numbers for the Earth return vehicle were used though scaled up for a 12-member crew (the baseline crew size is 10, so there is some reserve included). Consumables are 2 kg/person/day for Physical Chemical Life Support and 2.2 kg/person/day for crew accommodation (food, etc.). This is not a totally closed life support system (LSS). The next table is the mass summary for the current Astrotel vehicle.

Table 6-12 Astrotel Equipment and Mass Summary, kg

Subsystem or Item		Dry Mass	Consumables	Subtotal Mass	15-year Consumables Mass	15-year Refurbish-ment Mass
Physical/Chemical Life Support		2,778	3,840	6,618	26,880	1,389
Crew Accomodation		5,000	4,224	9,224	29,568	2,500
Structure		5,500		5,500		
EVA Equipment and Consumables		1,183	446	1,629	3,122	1,183
Communications and Information		320		320		320
Thermal Control		550		550		275
Power		785		785		785
Solar Array	640					
Internal Electrical Power Distribution	100					
Energy Storage	45					
Propulsion		644		644		644
Thrusters	147					
Power Processing Units	291					
Radiators	24					
Propellant Management	60					
Gimbals	44					
Cabling, structure, thermal, DCIU	78					
Attitude Control		500		500		250
Radiation Shielding, Escape Pod and Reserve		32,000		32,000		
Crew		1,200		1,200		
Utility Module Base		5,000				
Permanent Cargo Bay		3,000				
Spares		2,100		2,100		
Total Mass		**60,560**	**8,510**	**69,070**	**59,570**	**7,346**

6.8 Taxis

Taxis provide transportation between Spaceports and Astrotels. In order to minimize propulsive energy use, Taxis use advanced aeroassist technologies for planetary orbit capture. Aerocapture takes maximum advantage of planetary atmospheric drag to slow the vehicle on its approach from planetary space.

6.8.1 Key Sizing Assumptions

The initial sizing of the Taxi vehicle has been carried out. The key assumptions were:

e) Minimal radiation protection (equivalent to ~10 cm polyethylene surrounding crew module) for the crew is provided since transfer times to/from the Astrotels could be 7-10 days

f) No cargo is transported to the Astrotel by the Taxi except crew

g) 15% of the entry mass is aeroshell

h) LOX/LH propulsion system at Isp of 460 s and thrust of 60,000 lbs./engine

i) Fuel cell energy storage, no solar array power source

j) Propellant tank augmentation (expendable drop tanks and in some cases additional engines) are required at Mars

6.8.2 Taxi Design Description

The nominal Taxi system aeroshell design is an elliptical raked cone (see Section 4.6.3). Taxis utilize LOX/LH propulsion to escape planets and place them and their crew onto hyperbolic rendezvous trajectories with the interplanetary orbiting Astrotels. Figure 6-6 depicts a Taxi departing the Earth Spaceport at with the Moon in the background. This figure illustrates the crew module, propellant tanks, rocket engines (in their deployed position), and the aeroshell. Propellant capacity of the basic Taxi vehicle is 20.6 mt. Rendezvous time to Astrotels would be measured in days in order to reduce the duration of crew time in the expected cramped quarters. Crew volume is comparable to what the Apollo astronauts had on their flights to the Moon and back.

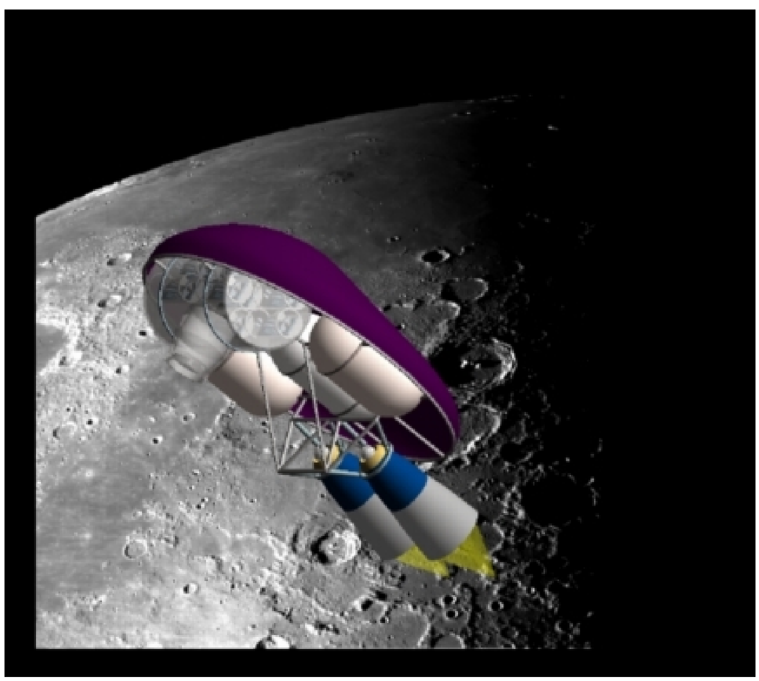


Figure 6-6 Taxi Departing L-1

The following figure is a scale drawing of the Taxi as it undergoes aerocapture at Mars. The view is as seen from 50 km above Valles Marineris during aero-cruise. Note the rocket engines are in their stowed position. During this time the tanks are almost empty, containing only the propellant necessary to rendezvous with the Mars Spaceport after aerocapture. The crew module is shown in *see-through* mode so one can observe the crew g-seats, which rotate in order to accommodate the varying g-load direction and the quite different thrust direction during propulsive maneuvers.

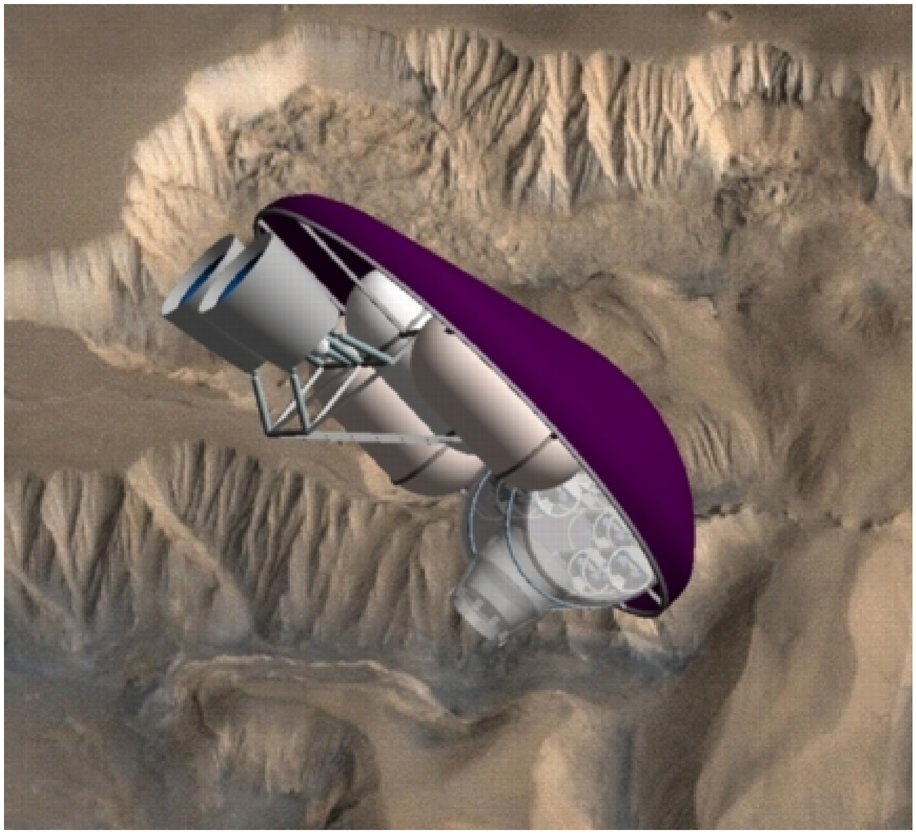


Figure 6-7 Taxi During Mars Aerocapture

The details of the crew module are shown in the following figure. The usable volume of the crew module is roughly comparable to of the Apollo Command Module, or about 2-3 m^3 per crewmember. The crew module has 10 crew pods, which includes a rotating g-seat, communications and computer panels, and hygiene systems. At least two crewmembers will have additional Taxi control and monitoring equipment adapted to their g-seat. Crew pods rotate in order to accommodate the varying acceleration vector during aerocapture maneuvers and very different acceleration vectors during staged propulsive rocket burns. Maximum g-load is nominally 5. The crew module is cylindrical in shape and surrounded by about 3-cm thick polyethylene radiation shield to protect the crew during major solar particle events. In future work we will look at increasing the shielding thickness by making the crew module smaller and/or increasing the mass of shielding allowed. In the figure, the structure located on the side is an airlock and docking port for providing emergency EVA access or docking access to Astrotels or Spaceports. ECLSS equipment is located external to the crew module and is not shown in this figure.

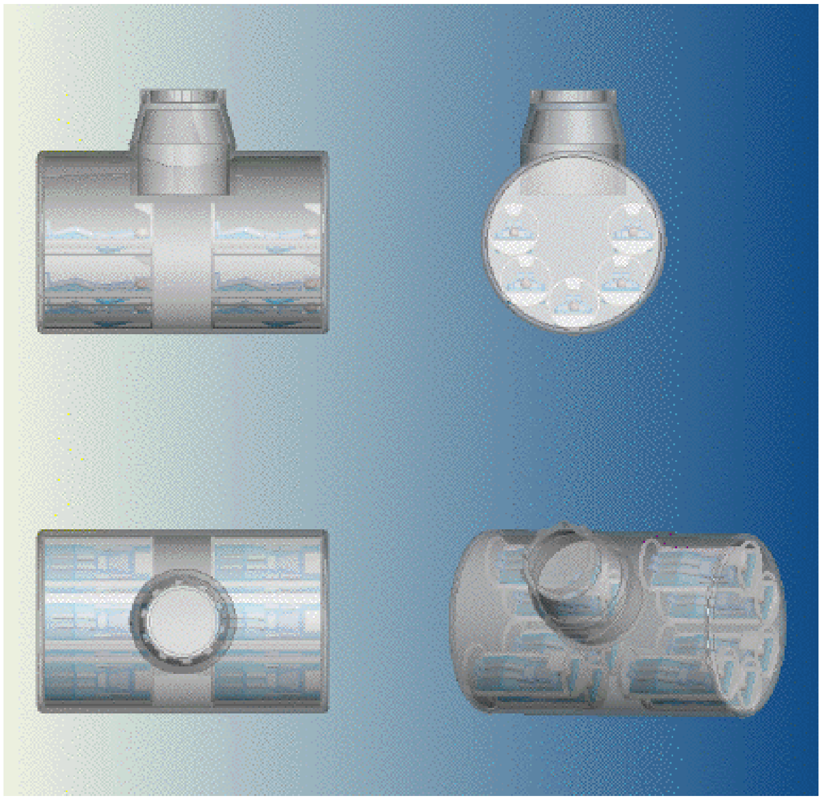


Figure 6-8 Crew Module Configuration

Figure 6-9 illustrates the Taxi vehicle docked at an Astrotel.

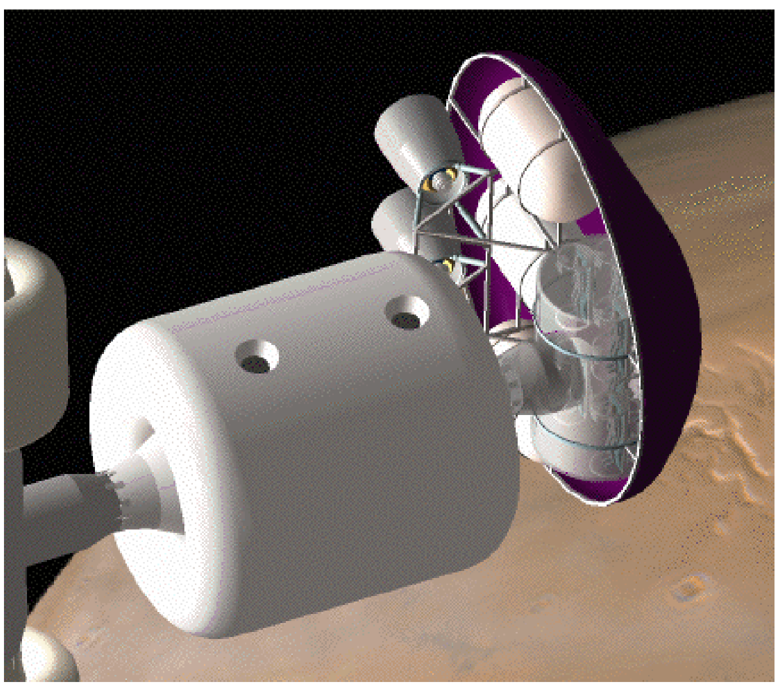


Figure 6-9 Taxi Docked at Astrotel

6.8.3 Taxi Propellant Requirements

At Mars the departure delta-Vs are significantly larger than at Earth due to the higher V-infinity of the Astrotel Down Escalator orbit as it passes Mars (see Section 6.3.3). These larger delta-Vs mean that the Taxi must be staged at Mars. Two stages are required for 3 of 7 opportunities, where the delta-V is less than 6.7 km/s. Staging is accomplished with the addition of up to 7.9 mt of expendable propellant augmentation tanks (PATs). For these opportunities up to 52.5 mt of additional propellant must be used. Three stages are required for the other 4 opportunities, where the delta–V can reach up to 10.5 km/s. Staging is accomplished by adding three more engines in addition to 24.6 mt of tanks. These three engines burn the additional 164 mt of propellant in the first stage and keep the gravity losses low.

Propellant requirements for the Taxi vehicles have been estimated from the Taxi system design and the delta–Vs generated in Phase I and shown above. The following tables describe the 15-year propellant requirements (kg) and in the case of Mars departures, the additional tanks required for staging due to these large delta-Vs. The propellant requirements will drive the propellant production system requirements at Mars and Earth. For the purposes of this analysis the augmentation tankage was calculated as exact percentages of additional propellant (rubber tanks) as opposed to fixed increments of tank sizes. In actuality, there will likely be fixed tank sizes and each set of tankage will be optimized depending on actual delta-V requirements of each opportunity.

Table 6-13 Up Escalator Taxi Propellant Requirements, kg

		ΔVin (m/sec)	ΔVout (m/sec)	Total ΔV (m/s)	Mf	Mi	Total Propellant
Earth	13-Nov-11		2,473	3,222	19,105	39,021	19,916
Mars	24-Apr-12	749					
Earth	18-Dec-13		2,490	3,302	19,105	39,725	20,620
Mars	18-May-14	812					
Earth	26-Jan-16		2,508	3,240	19,105	39,184	20,079
Mars	16-Jun-16	733					
Earth	17-Mar-18		2,454	3,175	19,105	38,619	19,514
Mars	9-Aug-18	721					
Earth	3-Jun-20		2,442	3,167	19,105	38,555	19,450
Mars	10-Nov-20	726					
Earth	22-Aug-22		2,482	3,244	19,105	39,213	20,108
Mars	24-Jan-23	762					
Earth	26-Sep-24		2,542	3,284	19,105	39,569	20,463
Mars	14-Mar-25	742					
Earth	1-Nov-26		2,538	2,538			
			Total 15-year Propellant at Earth				**140,150**

Table 6-14 Down Escalator Taxi Propellant Requirements, kg

	Date	ΔVin (m/sec)	ΔVout (m/sec)	Total ΔV (m/s)	ΔV 1	ΔV 2	ΔV 3	Total Propellant	Aug Tanks, Mt1	Aug Tanks, Mt2	Expend-able Engines	Total Aug Tanks and Expendable Engines
Earth	22-May-10	928										
Mars	8-Jan-12		6,897	7,825	1,168	3,352	3306	104,118	4,645	7,875		12,520
Earth	25-Jun-12	929				-			-			-
Mars	15-Feb-14		5,732	6,654		3,348	3306	73,050		7,860		7,860
Earth	4-Aug-14	922				-			-	-		-
Mars	19-Apr-16		5,506	6,424		3,118	3306	67,208		6,984		6,984
Earth	15-Sep-16	918				-			-	-		-
Mars	11-Jul-18		4,940	5,858		2,553	3306	54,803		5,123		5,123
Earth	8-Dec-18	918				-			-			-
Mars	24-Sep-20		7,478	8,397	1,739	3,352	3306	123,837	7,603	7,875		15,478
Earth	18-Feb-21	918				-			-			-
Mars	9-Nov-22		9,528	10,449	3,792	3,352	3306	237,610	24,669	7,875	1,500	34,044
Earth	30-Mar-23	922				-			-			-
Mars	1-Dec-24		8,719	9,648	2,990	3,352	3306	182,771	16,443	7,875	1,500	25,818
Earth	28-Apr-25	928										

Stages	Code
2	(green)
3	(yellow)

Total 15-year Propellant at Mars **843,397** **107,827**

Total 15-year Propellant at Mars, mt **843.4**

Total 15-year Tanks at Mars, mt **107.8**

6.8.4 Taxi Mission Profile

Two Taxis operate in the Mars transportation architecture. In a typical sequence a Taxi departs Earth and rendezvous 7-10 days later with the Up Astrotel to Mars for its 5-month trip to Mars. Several days before Mars arrival, the Taxi departs the Astrotel and deflects its trajectory to a Mars aerocapture. After aerocapture the Taxi, now in orbit, rendezvous with the Mars Spaceport, where it docks. This Taxi remains docked to the Mars Spaceport as the crew departs the Spaceport on the Mars Shuttle toward the Mars Base. After an average of 2.3 years, the next crew boards the Taxi and departs the Mars Spaceport to rendezvous with the Down Astrotel for its 5-month trip back to Earth. Total Taxi mission duration is an average of 2.8 years from Earth departure to return. Once back at the Earth Spaceport, major refurbishment and upgrades are planned including possible replacement of aeroshell components. Figure 6-10 illustrates the Taxi profile departing the Mars Spaceport on its way back to Earth.

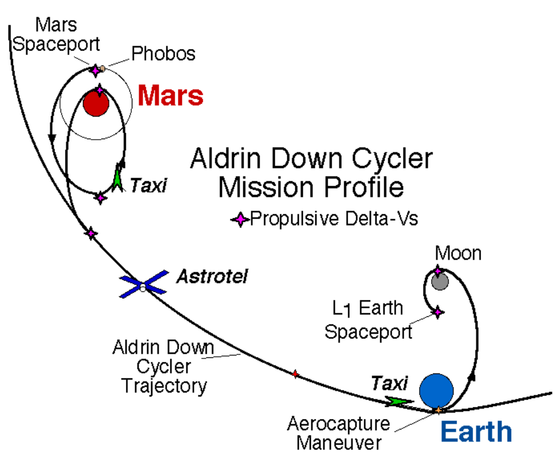


Figure 6-10 Taxi Mission Profile on Return to Earth

6.8.5 Taxi Mass Summary

The following table summarizes the mass for the basic Taxi vehicle that can depart Earth, rendezvous with the Up Astrotel, carry out the Mars aerocapture maneuver and perform propulsive maneuvers to rendezvous with the Mars Spaceport.

Table 6-15 Basic Taxi Vehicle Mass Summary, kg

Subsystem or Item		Dry Mass	Consum-ables	Single Stage Mass
Physical/Chemical Life Support		700	140	840
Crew Accomodation		1300	154	1454
Structure		2500		2500
EVA Equipment &Consumables		465	225	690
Radiation Shielding		1843		1843
Communications and Information		100		100
Thermal Control		550		550
Power		1900		1900
Fuel Cell	1800			
Internal Electrical Power Dist	100			
Attitude Control		500		500
Crew		800		800
Subtotal		10658	519	11177
Propulsion		4407		4407
Engines	1000			
Tankage	3,098			
Residual and Reserve	310			
Total		15065	519	15584
Aeroshell		3521		3521
Grand Total Dry Mass		**18586**	**519**	**19,105**
Final Stage Mass				**19,105**
Propellant for each Stage				**20,650**
Grand Total Wet Mass				**39,755**
Delta-Vs, m/s				**3306**

6.9 Lunar Water Tanker

The Lunar Water Tanker (LWT) is a reusable vehicle that can be fueled either on the Moon or at the Earth Spaceport. It transports water from the lunar surface to the Earth Spaceport where it refuels and then returns empty to the lunar surface. The LWT is not crewed thus minimizing additional infrastructure. The key power and LOX/LH propulsion elements are assumed to be common with other elements of the Mars transportation architecture in order to minimize development and recurring cost.

The maximum payload of the LWT is expected to be about 15.3 mt of Lunar water. Leaving the Moon for L-1, the required delta-V is 2160 m/s, which requires a propellant load of about 13.5 mt of LOX/LH. The departure delta-V propellant is assumed to be produced totally on the Moon. Once delivered to L-1, some of the water is processed into LOX/LH for the LWT to return to the

Moon. Since the LWT is empty leaving L-1, only about 4.8 mt of LOX/LH propellants (processed Lunar water) are required to return to the Moon. The net materials left at L-1 is about 9.3 mt of LOX/LH for use as propellants and some excess oxygen. Total water produced at the Moon is 30.5 mt every 2 1/y year cycle.

Design details will be developed further during Phase II. The following figure is a sketch of one possible configuration.

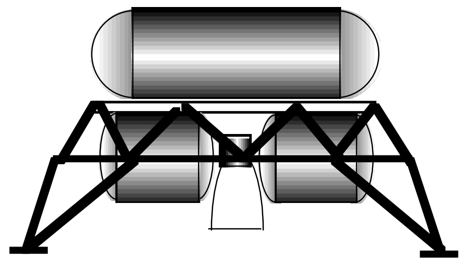


Figure 6-11 Sketch of Lunar Water Tanker

6.10 Mars Shuttle

The Mars Shuttle is patterned after the design developed during the NCOS effort in 1985. The Mars Shuttle transports a crew to and from the Martian surface base and the Mars Spaceport near Phobos. The Mars Shuttle supports crew needs during the very short transit (<2 days) between the Mars Base and the Mars Spaceport. In addition, the Mars Shuttle carries out delta-V maneuvers, performs aero-entry and landing maneuvers within the Martian atmosphere, navigates autonomously during all maneuvers, provides electrical power to its subsystems and carries RRU cargo from the Mars Spaceport to the Mars Base. The Mars Shuttle is designed to travel only between the Mars surface and the Mars Spaceport at Phobos. The vehicle developed by NCOS only carried a crew of four so this Mars Shuttle was scaled in size to carry a crew of 10.

The basic vehicle is a low lift/drag (L/D) ratio design with a deployable 15.2-m diameter aerobrake used during entry and landing. At take-off, the aerobrake is stowed to reduce atmospheric cross-section and minimize drag. The low lift/drag ratio design offers reduced mass, ease of fabrication, reduced cost and growth accommodation over higher L/D designs.

The following figure is the NCOS Mars Shuttle system configuration [see Section 1 Ref. 2]. In the figure, drawing #1 illustrates the entry configuration. Drawing #2 is the Mars Shuttle at landing showing some cargo on the backside of the deployable aerobrake. Drawing #3 illustrates the system prior to take-off with the aerobrake restowed. The following table summarizes the evolution of the design concept from the NCOS study to this Phase I effort.

Table 6-16 is a summary of the mass of the Mars Shuttle along with a comparison of the NIAC design with the NCOS design.

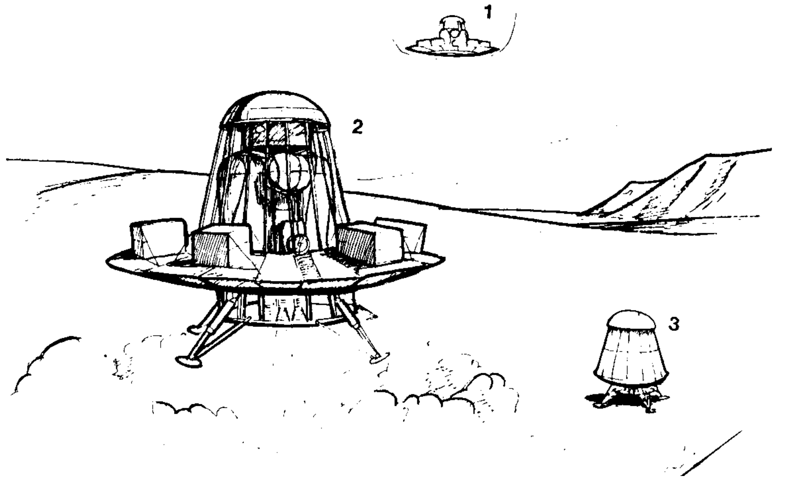


Figure 6-12 Mars Shuttle Design Concept

Table 6-16 Mars Shuttle System Mass Summary, kg

Subsystem	NCOS Reference 1985	Astrotel Study	Refurb Mass in 15 years, %	Refurb Mass, kg
Crew Module				
Primary Structure	255	893	5%	45
Couches, restraints	36	90	50%	45
Hatches, windows	55	60	25%	15
Docking	77	80	50%	40
Panels, supports	23	30	20%	6
Power System	436	1,090	75%	818
PMAD	105	105	30%	32
Comm	95	50	100%	50
Guidance and Nav	102	50	100%	50
Controls & Displays	91	50	100%	50
Instrumentation	86	50	100%	50
Life Support System	583	1,458	50%	729
Crew	318	818	0%	-
Total Crew Module	2,262	4,823		1,928
Propulsion Module				
Tanks, Insulation & Plumbing	2,923	4,610	5%	230
Engines	2,000	2,000	100%	2,000
Landing Gear	336	423	10%	42
Aerobrake	5,780	6,451	30%	1,935
Attitude Control (dry)	228	50	100%	50
Attitude Control (prop)	481	704	0%	-
Primary Structure	2,475	3,522	5%	176
Total Propulsion Module	14,222	17,760		4,434
Total Mars Shuttle	16,484	22,584		6,363
Tankage Factor	8.4%	10.0%	of propellant mass	
Proportional Mass Factors				
Structure	4.71%	5.0%	of initial mass	
Landing Gear	0.6%	0.6%	of initial mass	
Attitude Control	0.9%	1.0%	of initial mass	
Aerocapture	16.6%	15.0%	of Entry mass	
Mars Cargo Sized	10,000	10,000	kg	

Though the vehicle is sized for carrying up to 10 mt of cargo to the Mars Base, the requirement is only about 7.7 mt, thus indicating considerable reserve exists in the design. The delta-V requirements for the Mars Shuttle and appropriate mass fractions are presented in the next table.

Table 6-17 Mars Shuttle Delta-Vs and Mass Fractions

Delta-Vs		
Mars Surface to Phobos	5,100	m/s
Exhaust Velocity	4,511	m/s
Mass Fraction #1	3.10	
Phobos to Mars Entry	562	m/s
Mass Fraction #2	1.13	
Landing		
	0	m/s
Landing Deceleration	920	m/s
Wind	80	m/s
Hover	230	m/s
Total Landing	1,230	m/s
Mass Fraction #3	1.31	
Total Phobos to Landing	1,792	
Mass Fraction #4	1.49	

The performance for the Mars Shuttle is shown in the following table, which includes the per cycle propellant requirements at Mars and at the Mars Spaceport.

Table 6-18 Mars Shuttle Performance Analysis

Overall Masss Fraction = MF	Mass Fraction #3	A = Tankage Factor	B = Proportional Mass Factors	C = Aeroshell Mass Factor	Fix Masses = Mfx	Initial Mass = Mi	Entry Mass = Me	Landed Mass	Final (empty) Mass = mf	Propellant Mass leaving Mars = mp1	Propellant Mass leaving Phobos = mp2
3.10	1.31	10%	6.6%	15.0%	6,873	70,448	43,008	32,744	22,744	**46,096**	**48,714**
Mass of Propellant required at Mars every cycle (2 1/7 years)										322,673	
Mass of Propellant required at Phobos every cycle (2 1/7 years)											341,001

6.11 Integrated Propellant and Cargo Use Profiles

In this section we select and summarize the overall propellant and cargo masses that move through nodes of the transportation architecture over a 15-year period. Vehicle propellant, crew consumable and RRU mass requirements over 15 years have been summarized and collected in MAMA at their eventual transportation node in the following tables. All masses are kilograms.

Table 6-19 Overall 15-year Cargo Use Profile vs Transportation Node and System, kg

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Users						
Taxis (2)						
Propellant						
LOX		122,631	737,972			
LH		17,519	105,425			
Tot		140,150	843,397			
Water required at Spaceport		157,669				
Augmentation Tanks	107,827		107,827			
Taxi Refurb Mass	22,475	22,475				
LEO Shuttle (1)						
Refurb Mass	22,475					
Mars Shuttle (1)						
Propellant						
LOX			298,376	282,339		
LH			42,625	40,334		
Total			341,001	322,673		
Water required on Mars				363,007		
Refurb Mass	6,363		6,363			
Lunar Polar Base						
Polar Ice Mine Refurb Mass	567	567			567	
Mars Base						
Base Refurb Mass	51,045		51,045	51,045		
Propellant Plant Refurb Mass	3,016		3,016	3,016		
Phobos LOX Plant						
Propellant Plant Refurb Mass	4,763		4,763			
Astrotels (2)						
Crew Consumables	119,140					119,140
Xenon Propellant	5,751					5,751
Refurb Mass	14,692					14,692
Mars Spaceport (1)						
Refurb Mass	14,692		14,692			
Phobos LOX Storage Refurb Mass	379		379			
Station-keeping Xenon propellant Mass						
Earth Spaceport (1)						
Refurb Mass		14,692				
LOX/LH Production Plant Refurb mass	66	66				
Station-keeping Xenon propellant Mass						
Lunar Water Tanker (1)						
Refurb Mass						
Astrotel Cargo Freighter						
Xenon Propellant	64,246					
Refurb Mass	2,652					
Mars Cargo Freighter						
Xenon Propellant	175,108					
Refurb Mass	6,677					

These use profiles have been summarized as to where they are needed throughout the transportation architecture for the 15 years of operation and displayed in the following table (all masses are in kg).

Table 6-20 Cargo Use Profile Summary

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Total Cargo Requirements						
LOX		122,631	1,036,348	282,339		
LH		17,519	148,050	40,334		
Water		157,669		363,007		
Xenon	245,106					5,751
Refurb Mass	149,863	37,800	80,259	54,061	567	14,692
Augmentation Tanks	107,827		107,827			
Crew Consumables	119,140					119,140
Communications Satellites						
Total Mass Required	621,935	335,619	1,372,483	739,741	567	139,583

Finally, this cargo has been divided into the various delivery systems, which is displayed in the next chart.

Table 6-21 Cargo Delivery Systems Requirements Summary

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Delivery Systems						
SEP Freighter Delivered Cargo		37,800	336,135			139,583
Cargo Freighter Each Trip		5,400	48,019			19,940
Lunar Water Tanker Total		157,669			567	
Lunar Water Tanker Each Trip		22,524			81	
Mars Shuttle				54,061		
Mars Shuttle Each Trip				7,723		

These propellant and cargo use profiles provide the derived requirements for the transport of cargo and propellant throughout the architecture and the requirements on the ISRU production rates to be discussed later.

6.12 Cargo Freighters

Two types of cargo transporters are planned, an Astrotel Cargo Freighter and a Mars Cargo Freighter. These vehicles deliver cargo from LEO to Astrotels and Spaceports. See Section 6.11 for the integrated cargo requirements. Cargo Freighters use xenon ion propulsion systems to spiral out of Earth orbit, shape the interplanetary trajectory to rendezvous with Astrotels or spiral into Mars orbit to Phobos. The Astrotel Cargo Freighter delivers a standard pressurized cargo bay module to the Astrotel. The cargo bay approach facilitates crew unloading.

The Freighter solar arrays consist of multiple sets of identical Astrotel solar arrays (80 kW panels). The propulsion system shares high degree of technology heritage with the Astrotel IPS. The following charts describe the design parameters for these important vehicle systems, which will be a subject of more design during Phase II.

Astrotel Cargo Freighter Parameters

Mass of Cargo each trip to Astrotel	**19,940**	kg
Initial mass of Cargo Freighter in LEO	33,146	kg
Propellant Mass	9,178	kg
Final Mass of Freighter	4,027	kg
P_o	331.5	kW
Power/Propulsion Mass (M_{ps})	2652	kg

Figure 6-13 Astrotel Cargo Freighter Sizing

Mass of Cargo each trip to Mars Spaceport	**48,019**	kg
Initial mass of Cargo Freighter in LEO	83,468	kg
Propellant Mass	25,015	kg
Final Mass of Freighter	10,434	kg
P_o	834.7	kW
Power/Propulsion Mass (M_{ps})	6,677	kg

Figure 6-14 Mars Cargo Freighter Sizing

6.13 In Situ Resource Utilization Systems

The following sections summarize the *in situ* resource systems for the Moon, the Earth Spaceport, Mars surface, Phobos and the Mars Spaceport.

6.13.1 Lunar Ice Mine

The automated Lunar Ice Mine is assumed to operate only during periods of sufficient solar illumination on its surface solar array, thus 34% of the time. During the night batteries or non-regenerative fuel cells provide keep-alive power. Ice is mined and melted into water and then transported to the Earth Spaceport for production of LOX/LH propellants.

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required		30,473	kg					
Mining and Excavation (0.16-g)	Mine regolith (kg/hr)	1016	kg/hr	34%	0.20	201		1.80
Soil Hauler (0.16-g)	Haul soil to reactor and slag away							
Reactor/Condensor	Heats soil from -200oC to 50°C (kg H2O/hr)	10.16	kg/hr	34%	45.00	450	7.5	76.18
Water Storage	Stores extracted water (.33 x annual water production)	10.16	kg/hr	100%	0.02	203		
Distribution	Distributes liquids and gases to where they are processed					100		
Electrolyzer	Produces propellant for launching water to L1 (kg/hr)	4.50	kg/hr	34%	20.00	90	6.5	29.28
LH liquefaction	Provides LH for transfer vehicle (kg H2/hr)	0.56	kg/hr	34%	16.50	9	20	11.26
Lox liquefaction	Provides Lox for transfer vehicle (kg O2/hr)	3.94	kg/hr	34%	6.50	26	0.5	1.97
Lox, LH storage	Assumes immediate transfer to L1						Total Power-->	120.50
Solar Array - XX kW	Produces electrical power during Lunar day (kW)			34%	4.00	482		
Power Mgmt & Dist	Power voltage/freq control and distribution					48		
					total mass	1,609		

6.13.2 Earth Spaceport LOX/LH Production and Storage

The Earth Spaceport is an ideal location for the electrolysis of lunar water into oxygen and hydrogen, liquefaction into LOX/LH and finally storage of these propellants. Nearly continuous power is available using high efficiency solar arrays. The Earth Spaceport ISRU facility is expected to operate at a 90% duty cycle.

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Water Delivered to L1		15,270	kg					
Water Storage	Stores Lunar water after delivery to L1 (3 mo. supply)	3,818	kg	100%	0.01	38		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	1.59	kg/hr	91%	20.00	29	6.5	10.3
Distribution	Distributes liquids and gases to where they are processed	50	kg	91%	1.00	46		
LH liquefaction	Liquefies H2	0.18	kg/hr	91%	16.50	3	20	3.5
LoX liquefaction	Liquefies Ox	1.42	kg/hr	91%	6.50	8	0.5	0.7
LH Storage	H2 gas storage at X°C (kg) - 3 mo. supply, rest in Taxi	387	kg	100%	0.15	58	0.0016	0.6
LOx Storage	Cryogenic gas storage at -183oC - 3 mo. supply, rest in Taxi	3099	kg	100%	0.07	217	0.001	3.1
						Total Power-->		18.3
Solar Array - XX kW	Produces electrical power 100% of time (kW)			100%	4.00	73		
Power Mgmt & Dist	Power voltage/freq control and distribution					7		
					Total Mass--	480		

6.13.3 Mars Dune Water Mine

Water is recovered from the regolith of Mars. Dune fields could be an excellent location for an excavation site. Once water is recovered from the soil it is electrolyzed into oxygen and hydrogen, liquefied into LOX/LH and finally stored within the Mars Shuttle tanks. The operation on Mars is fully automatic and operates at a 34% duty cycle along with the normal day/night periods.

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required		24,200	kg/yr					
Dune Collection (0.38-g)	Excavates likely water bearing soil (kg/hr)	807	kg/hr	34%	0.27	218	0.00743	6.0
Soil Hauler (0.38-g)	Transports soil to reactor and slag away							
Reactor/Condensor	Heats soil to 500°C to remove water (kg/hr)	8.07	kg/hr	34%	205	1,654	14.8	119.4
Water Storage	Stores water before electrolysis (kg)	1147	kg	100%	0.04	46		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	8.07	kg/hr	34%	20.00	161	6.5	52.4
Distribution	Distributes liquids and gases to where they are processed	8.07	kg/hr		3.70	30		
LH Liquifaction	Liquifies hydrogen (kg/hr)	1.01	kg/hr	34%	16.50	17	20	20.2
LOX Liquifaction	Liquifies oxygen (kg/hr)	7.06	kg/hr	34%	6.65	47	0.5	3.5
LOX Storage	Crygenic gas storage at -183°C (25% of annual production, rest on Mars Sh	6776	kg	100%	0.07	474	0.5	1.41
LH Storage	Crygenic gas storage at Y°C (25% of annual production, rest on Mars Shutt	968	kg	100%	0.15	145	20.0	8.07
						Total Power-->		211.0
Solar Array	Produces electrical power 33% of time			34%	9.24	1,950		
Power Mgmt & Dist	Power voltage/freq control and distribution					97		
					Total Mass--	4,839		

6.13.4 Phobos LOX Plant

Phobos regolith is excavated and processed to recover oxygen. The oxygen is then liquefied and transported to the Mars Spaceport where it can be stored in Taxi propellant tanks or propellant augmentation tanks (PATs). The operation on Phobos is fully automatic and operates at a 34% duty cycles along with the day/night periods.

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Oxygen Required kg/yr		69,090	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/yr)	203,205	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/hr)	} 67.7	kg/hr	34%	1.00	68	0.05	3.1
Soil Hauler (zero-g)	Transports soil to reactor and slag away							
Carbothermal Reactor	Reactions at 1600 °C to generate oxygen (kgO2/hr)	23.20	kg/hr	34%	96.00	2,227	16	371.2
Distribution	Distributes oxygen to where it is liquified	50	kg		1.00	50		
Liquifaction	Liquifies oxygen (kg/hr)	23.20	kg/hr	34%	6.50	151	0.5	11.6
LOX Storage	Assumes immediate transfer to Marsport							
							Total Power-->	385.8
Solar Array - XX kW	Produces electrical power 33% of time (kW)			34%	9.24	3,566		
Power Mgmt & Dist	Power voltage/freq control and distribution					178		
					Total Mass--	6,239		

6.13.5 Mars Spaceport Propellant Storage

Phobos LOX and Earth LH are stored at the Mars Spaceport, in Taxi PATs and Mars Shuttle tanks. The operation at the Mars Spaceport is nearly continuous, except for occasional eclipses of the Sun by Mars.

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Phobos LOX Storage*	Crygenic gas storage at -183oC - (25% ann. prod., rest in Taxi tanks)	17,272	kg	100%	0.07	1,209	0.00018	3.1
Earth LH Storage*	Crygenic Earth LH storage at Y°C - (25% ann. Req't, rest in Taxi tanks)	2,467	kg	100%	0.15	370	0.0084	20.7
							Total Power-->	23.8
Solar Array - XX kW at ~1.5 AU	Produces electrical power 100% of time				9.24	220		
Power Mgmt & Dist	Power voltage/freq control and distribution					11		
					Total Mass--	1,810		

7 Mars Astrotel Model Architecture (MAMA)

A Mars Astrotel Model Architecture (MAMA) model was developed to facilitate integration of various system elements and to support life cycle cost analysis. The approach allows independent development of individual elements and supporting analyses by focusing on the relationships among the system elements and establishing element-to-element links for selected inputs/outputs. Each element generates a semi-standardized output that is used to support life cycle requirements analyses.

The following figure is a schematic of the Mars Astrotel Model Architecture (MAMA) Information Flow and System Integration System design that was started during Phase I. In Phase I, this system is a highly integrated and interrelated model of the baseline Mars transportation vehicles; ground systems; subsystem technology assumptions; *in situ* resource assumptions and systems; and celestial mechanics analysis. This format was constructed in order to facilitate overall architecture trade studies and costing. The Phase I MAMA design, though limited in the degree to which we can perform trade studies, has built in most of the features required to expand the model in Phase II.

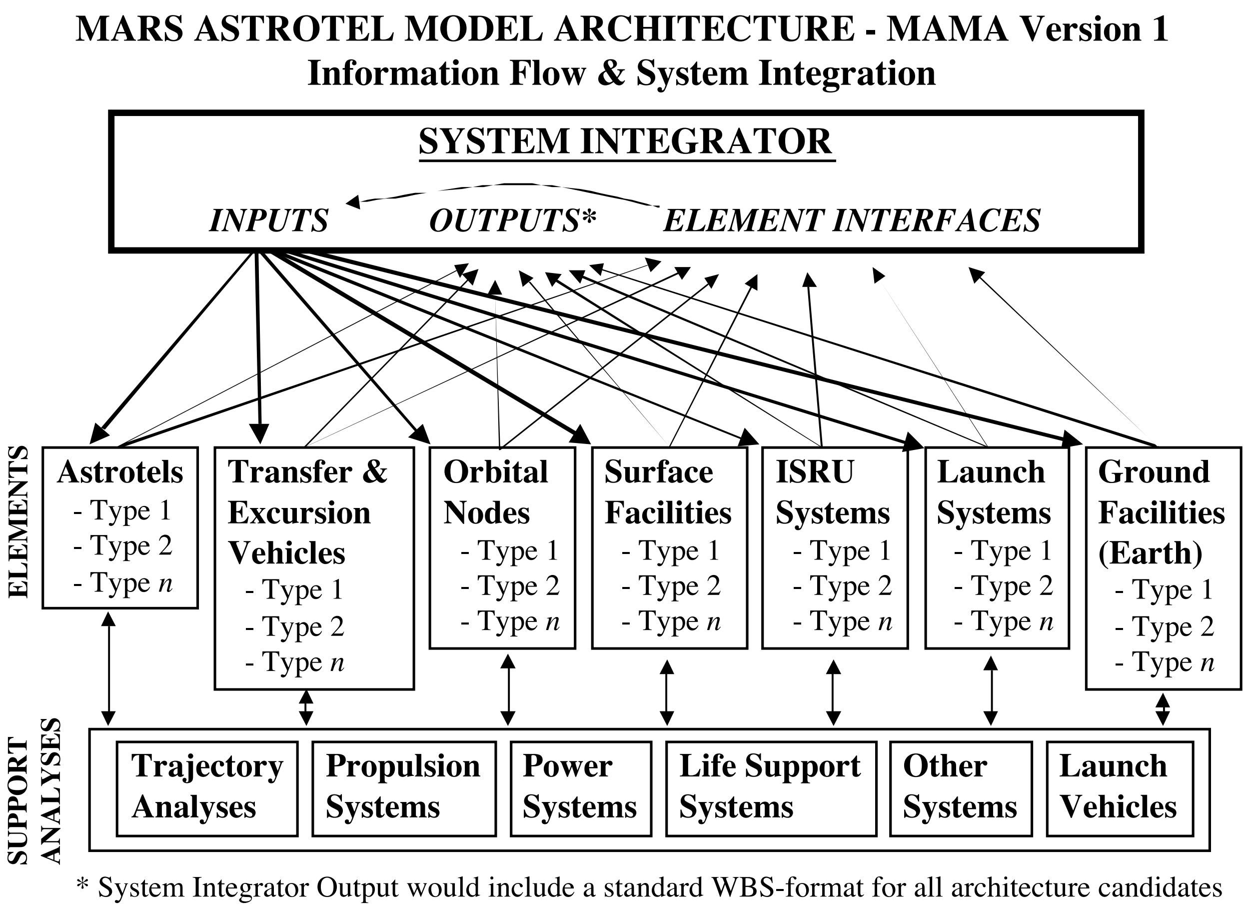


Figure 7-1 Mars Astrotel Model Architecture (MAMA) Information Flow and System Integration System Design

The following figure is an example of one spreadsheet of MAMA that focuses on the Mars Shuttle design. The resultant design, which includes propellant requirements, is factored into the resource system requirements at Phobos and Mars Cargo Freighter requirements.

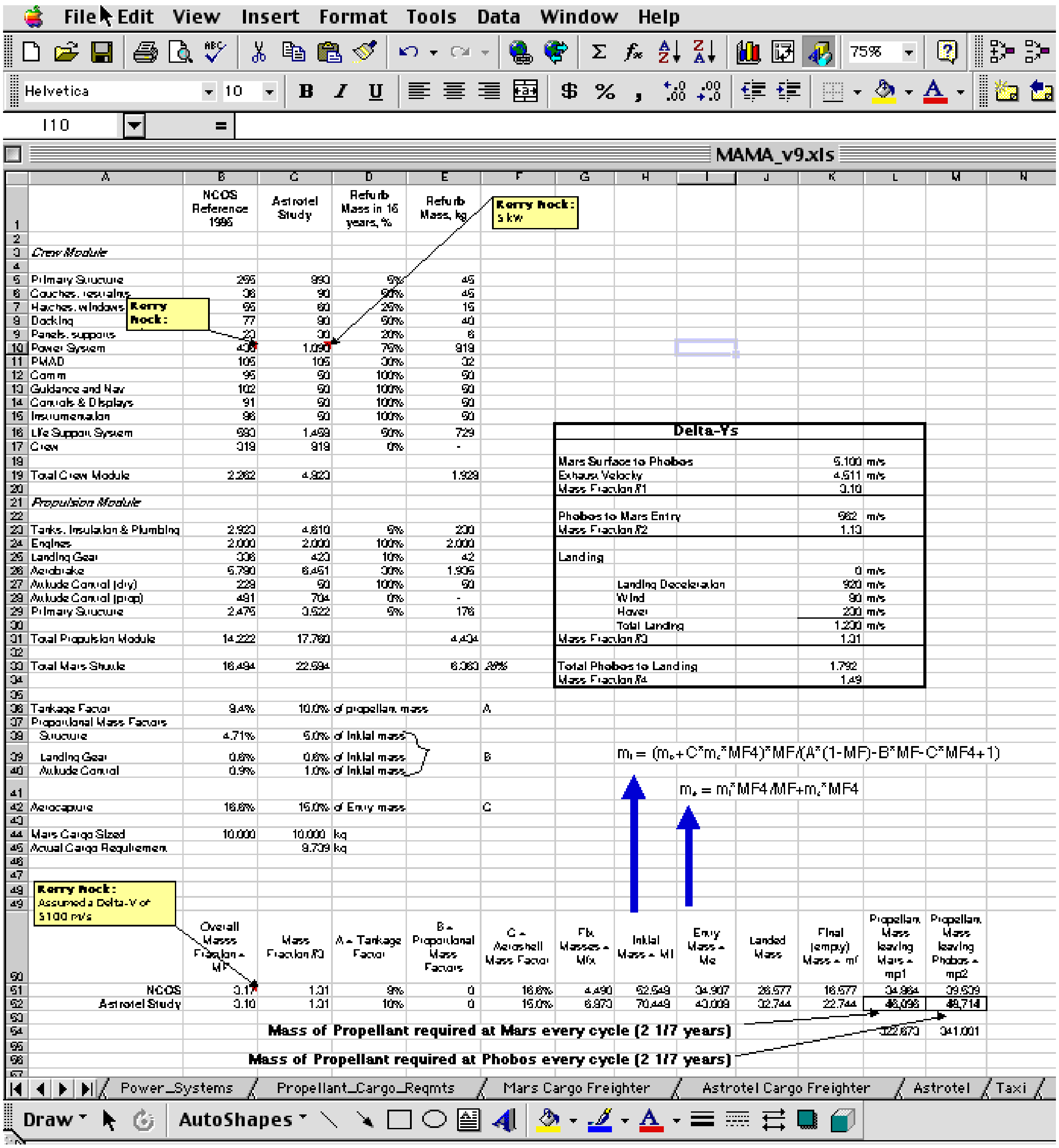


Figure 7-2 Example MAMA Spreadsheet for the Mars Shuttle Vehicle

System requirements and life cycle cost estimates are uniquely generated for each system sub element shown in Figure 7-1. The MAMA System Integrator (MAMA SI) collects the information from each MAMA system element required to support costing. Hardware design and development costs for each system sub element are estimated based on the number of units, unit mass, refurbishment mass, and type (specific to each sub element). All other development costs are estimated as a function of the system and sub element hardware costs. Also, Advanced Technology Development (ATD), and Operations costs are currently functions of development costs. Launch costs are based on required mass to LEO and an assumed launch vehicle cost per kg. MAMA SI is used to summarize all design and cost results for all Mars Astrotel system elements. Sample system mass outputs are shown in Figure 7-3 and Figure 7-4.

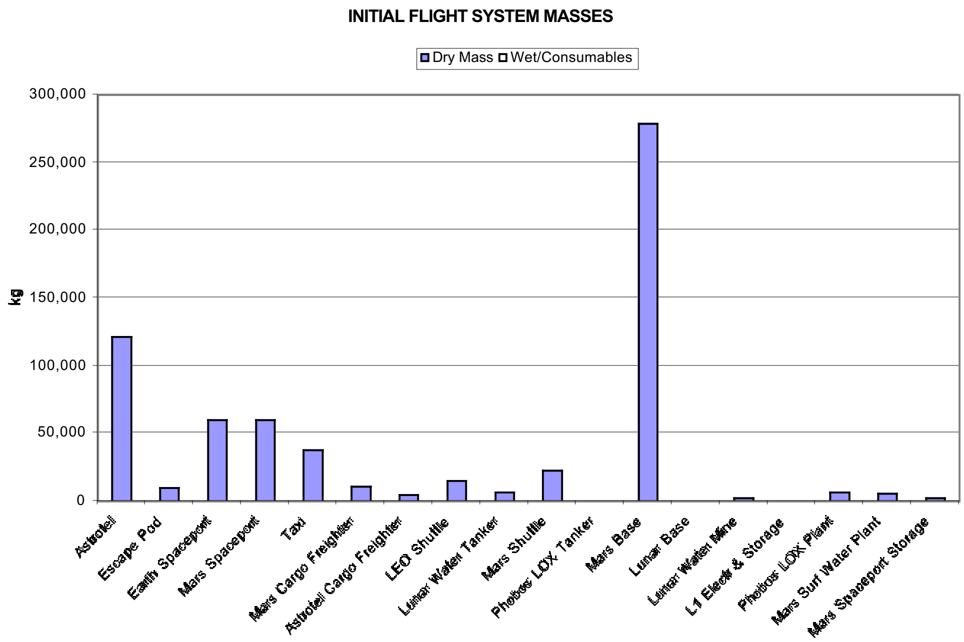


Figure 7-3 Example MAMA SI Output: Initial Hardware Mass

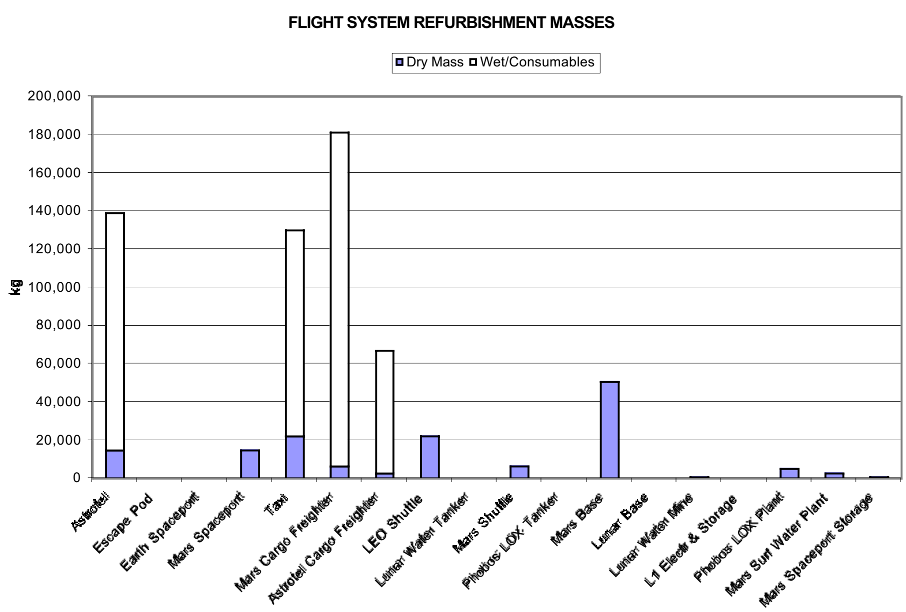


Figure 7-4 Example MAMA SI Output: Refurbishment Hardware Mass

8 Cost Analysis

8.1 Introduction

Cost analysis of Mars Astrotel scenario options cover life cycle costs for all required system elements. All costs are in FY 2000 dollars. To facilitate cost tracking, a detailed WBS was developed showing costs for each WBS element across each life cycle phase, summarized in Table 8-1.

Table 8-1 Life Cycle Cost Analysis Summary

	Life Cycle Phases			
Life Cycle Cost Elements	Adv Tech Devel	Flight Sys Devel	Launch	Operations
1.0 Advanced Technology Development	Facilities, demos, and prep for Flight Sys Dev & Ops			
2.0 Flight System Development	Validation of flight system & components w/ adv technology	Design, fabrication, assembly, integ. & test		Fabrication of Refurb HW
3.0 Launch		Launch approval & EIS	Launch vehicle and services	Launch vehicle and services
4.0 Operations				Emplacement, steady-state, refurbishment

8.2 Life Cycle Cost Elements

A Mars Astrotel WBS was established to capture life cycle costs for all project elements. For most cost elements, estimates are made at the level of detail shown in Table 8-2, the exception being WBS 2.1, which has another more detailed level to track additional cost categories associated with each flight system element.

Table 8-2 Life Cycle Cost Elements

1.0 Advanced Technology Dev	**2.0 Flight System Development**	**3.0 Launch**	**4.0 Operations**
1.1 General R&D 1.2 Facilities 1.3 Demos/Tests 1.4 System-Unique 1.5 Other	2.1 Flight Elements 2.2 Proj Support 2.3 Other 2.4 Reserves	3.1 Approval/EIS 3.2 Processing 3.3 Launch Vehicle 3.4 Other	4.1 Proj Mgmt 4.2 Integ Logistics 4.3 Flight Ops 4.4 Training Ops 4.5 Launch Ops 4.6 Crew Support 4.7 Comm/Data 4.8 Proj Support 4.9 Other 4.10 Reserves

8.2.1 Advanced Technology Development (ATD)

ATD cost elements capture requirements for validation of all new technologies prior to entering Flight System Development. "WBS 1.1 General R&D" covers research for items with low maturity. "WBS 1.2 Facilities" covers all new facilities and support equipment required to support validation testing. "WBS 1.3 Demos/Tests" covers the actual testing costs, which is mostly labor and materials. "WBS 1.4 System-Unique" captures any costs unique to a specific technology type and "WBS 1.5 Other" is a placeholder for costs that may not fit cleanly into another WBS element.

8.2.2 Flight System Development

Flight system development cost elements include all hardware systems used to support a Mars Astrotel scenario option. "WBS 2.1 Flight Elements" includes several sub elements: 2.1.x.1 Subsystem Components, 2.1.x.2 Subsystem Integration & Test, 2.1.x.3 System Integration & Test, and 2.1.x.4 System Support & Management ("x" refers to each individual flight element). Estimates are made at the Subsystem Component level and all other WBS elements are calculated as a percent of the Subsystem Component costs. A sample output showing WBS 2.1 results for each flight element is shown in Figure 8-1 (note that NRC – non-recurring, RC – recurring).

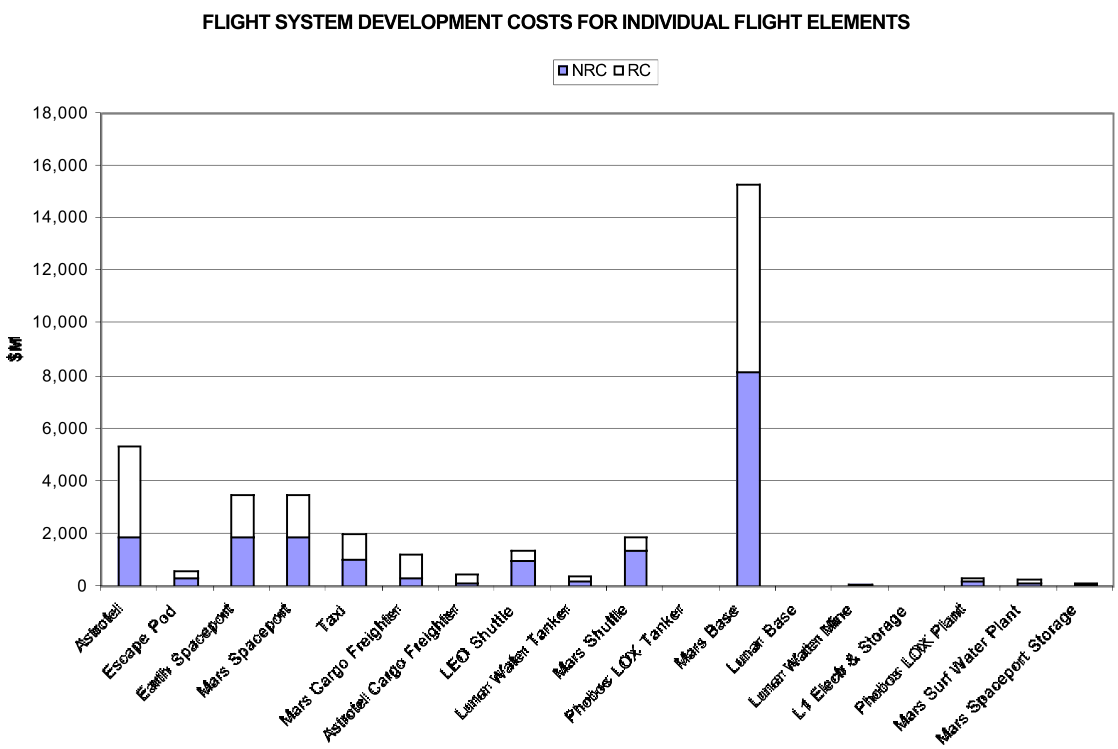


Figure 8-1 Example Flight System Development Cost Results

Costs shown in Figure 8-1 are estimated by building up costs from each flight system sub element (subsystem/component-level), and then adding subsystem integration & test, system integration & test, and system support/management as factors of the sub element costs. For each lower level item, a cost reference is defined that can be weight or performance based. Cost references typically use actual lower level costs from past projects and include cost reduction factors based on how the estimated item compares to the reference item. In the current version of MAMA, there are over 100 individual sub elements and 17 different system elements. Cost references are a mix of actual data from past missions and component-level performance parametrics developed by technology specialists in NASA, industry and academia. Table 8-3 provides a summary of cost (inflated) references used for each flight element.

Table 8-3 Basis of Estimate for Costing by Flight

Item	Basis of Estimate for Costing
TRANSPORTATION SYSTEMS & ORBITAL ELEMENTS	
Astrotel	$500/W for Solar Array; NSTAR-based estimates for SEP PPUs and Thrusters; Spacelab for Other Elements
Taxi	STS Orbiter
Mars Cargo Freighter	Taxi for Tanks; Astrotel for SEP and Power
Astrotel Cargo Freighter	Taxi for Tanks; Astrotel for SEP and Power
Earth & Mars Spaceports	Spacelab
Mars Shuttle	STS Orbiter
LEO Shuttle	STS Orbiter
Escape Pod	Spacelab
LUNAR/MARS BASES AND ISRU TANKERS	
Lunar Base	Tbd
Mars Base	Spacelab
Lunar Water Tanker	Taxi
Phobos LOX Tanker	TBD
IN SITU RESOURCE UTILIZATION (ISRU) SYSTEMS	
Lunar Water Mine	Spacelab
L1 Electrolysis & Cryo Storage	Spacelab
Phobos LOX Plant	Spacelab
Mars Surface Water Plant	Spacelab
Mars Orbit Propellant Storage	Spacelab
Assumed Improvements	50% reduction for Design; 30% reduction for Fabrication (applied to all flight elements)

Further investigation may identify significant additional cost savings by using common components/subsystem types for as many flight elements as reasonable.

"WBS 2.2 Project Support" captures costs for mission operations and ground system development, system engineering, product assurance, management and other non-Flight System costs. "WBS 2.3 Other" is a placeholder for cost items that do not cleanly fall into another WBS category. "WBS 2.4 Reserves" currently includes a 20% reserve on all Flight System Development costs.

8.2.3 Launch Costs

Space launch costs and specific launch costs ($/kg) are shown in the next figure for operational and soon-to-be-operational launch vehicles. These specific costs were calculated for placing payloads into low-inclination (28 degrees), 200-km altitude orbits. Note that most specific launch costs are between $10,000 and $20,000 per kilogram, except for the more expensive Pegasus, Taurus and Athena launch vehicles. Although these launchers have low total cost, the specific launch costs are high. In addition, these small launchers only have payload capabilities between 470-720 kg to these low inclination orbits. For the purposes of the Phase I Mars transportation architecture preliminary cost estimation, we will assume a future specific launch the cost of $2,000 per kilogram by 2035.

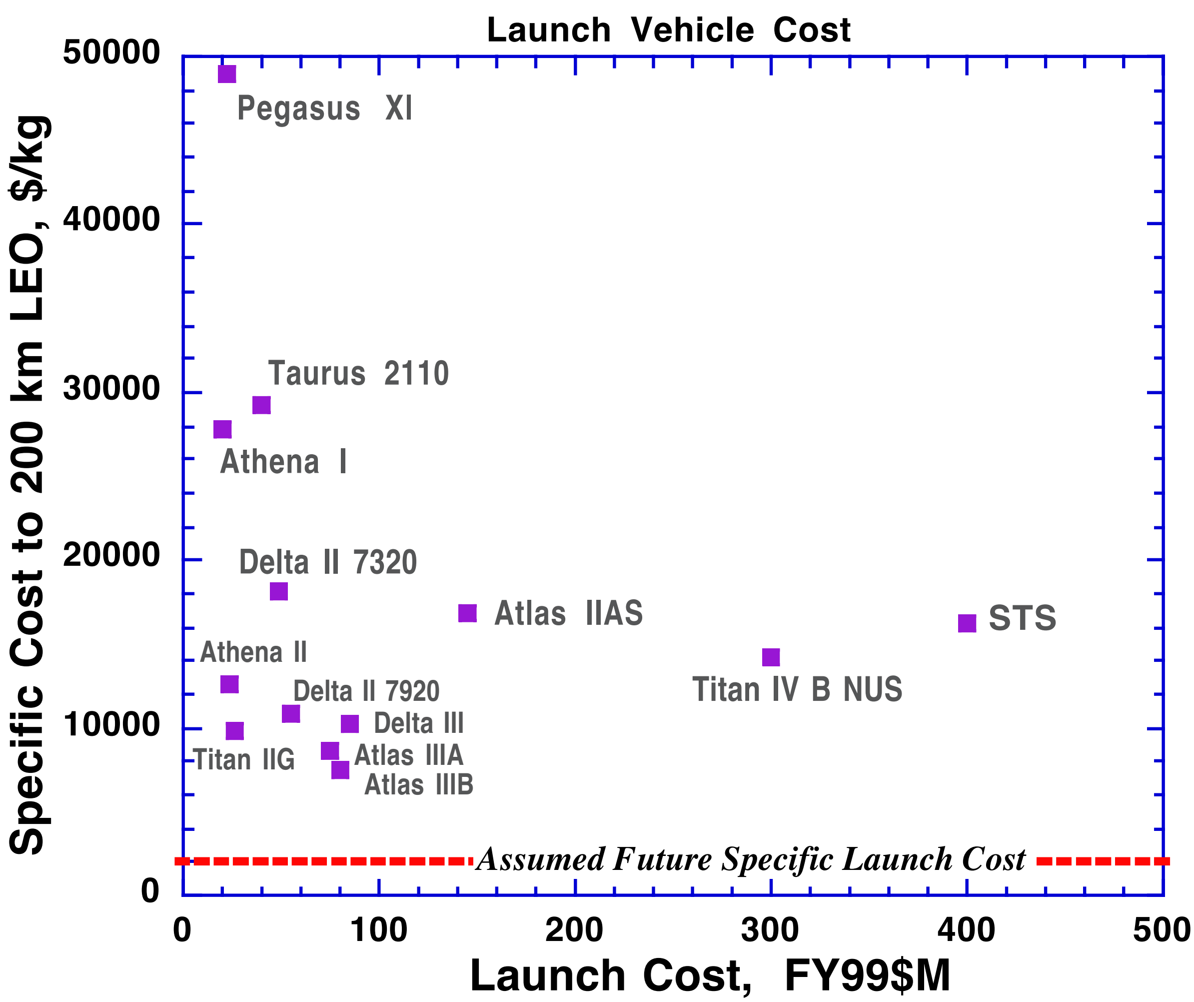


Figure 8-2 Launch Vehicle and Specific Launch Costs

Although the launch vehicle community has made several projections of $1,000/kg, all concepts to date rely on large up-front investments required to emplace ground facilities capable of supporting hundreds of flights per year. No current launch vehicle concept has validated an order of magnitude reduction of launch costs and a source for the required large up-front investment has not been identified. For this reason, a more conservative estimate of $2,000/kg is currently being used. Provided future costs come down to this level, launch costs do not seem to be a major cost driver for the Mars transportation architecture because the total over 15 years of operation total less than $3B of a $90B life cycle

8.3 Cost Modeling Considerations

8.3.1 Operations

Operations costs are calculated as a percentage of the recurring development costs based on the JSC online Mission Operations Cost Model (MOCM) using analogies to "Manned" Mission Types (http://www.jsc.nasa.gov/bu2/MOCM.html). MOCM estimate results show approximately 6.6% of any development effort is spent annually on operations. This percentage is currently applied to all Mars Astrotel systems and then the result is spread across the Operations WBS elements. Costs for On-Orbit Assembly, Startup Operations, and Disposal are currently not included. This approach is over-simplified and not sensitive to many operational requirements, but does generate a reasonable ROM estimate. A more refined approach for estimating operations will be developed in Phase II.

8.4 Life Cycle Project Phases

Each cost element described in Section 8.2 has a unique funding profile requirement based on the specific sub elements and components comprising each system. This impact is captured by separately estimating various project phases for each cost element. Specific project life cycle phases included are summarized in Table 8-3.

Table 8-4 Life Cycle Project Phases

1.0 Advanced Technology Dev	2.0 Flight System Development	3.0 Launch	4.0 Operations
- Adv Tech Dev - Adv Tech Testing	- Design - Subsys Fab - Sys Assy/Int/Test - Launch Processing - Orbit Checkout	- Launch Processing - Launch - Checkout	- Orbital Assy - Startup - Steady State - Refurbishment - Disposal

8.4.1 ATD Phases

ATD phases bring specific items requiring technology investment to a flight-ready status. Early in ATD, focus is on specific system sub elements and components. As ATD progresses, more demonstrations may be necessary for validation. Also, for certain items, new facilities and/or facility modifications may be required to support these demonstrations.

8.4.2 Flight System Development Phases

Flight system development begins after all new technologies required have been developed to a flight-ready status. The Design phase ends with CDR, after which Subsystem Fabrication efforts ramp up quickly. Subsystems are developed and integrated into each flight element prior to delivery to System Assembly/Integration/Test (AIT). After System AIT, the flight elements are shipped to the launch site. At the launch site, flight system development engineering support is included to capture ground-processing costs that are in addition to the support from the Launch Vehicle provider. The Development phase ends with on-orbit checkout of the flight systems (typically 30 days after launch).

8.4.3 Launch Phases

Launch begins with generation of Launch Approval and Environmental Impact Statement documentation (completed during Development). The next phase captures the launch vehicle provider's support at the launch site for ground processing and integration of flight elements with the launcher. Specific launch vehicle costs are currently estimated as a fixed $/kg to LEO. The Launch phase ends with checkout of the actual launch vehicle injection performance.

8.4.4 Operations Phases

Operations phases begin immediately after on-orbit checkout. Costs for Orbital Assembly, Emplacement, Startup Operations, and Disposal have been identified, but not yet included. Refurbishment costs cover fabrication and launch of replacement hardware required to support the Mars Astrotel systems over a 15-year lifetime.

8.5 Mars Transportation System Architecture Costing Assumptions

There are a number of assumptions made to estimate life cycle costs. It is important to note that many of these assumptions are placeholders for a more rigorous estimating methodology currently under development. Costing assumptions include:

- All costs are estimated in Fixed Year 2000 dollars
- Costs reflect full cost accounting pricing and cover all life cycle elements

8.5.1 Advanced Technology Development

- ATD costs for R&D and Facilities are estimated based on system sub element advanced technology costs

8.5.2 Flight System Development

- Sub element advanced technology costs are estimated at 1% of their Design cost (Design is the first phase of Development)
- Development Non-Recurring Costs (NRC) and Recurring Costs (RC) use a different design basis for each sub element – the design basis starting point is the most similar system from a past mission (with available cost actuals).
- Subsystem Integration & Test (I&T) costs are estimated at 15% of the subsystem component totals
- System I&T costs are estimated at 15% of the subsystem component and I&T totals
- System Support and Project Management Costs are estimated at 15% of the subsystem component, subsystem I&T, and system I&T costs
- Flight System Development Support costs are estimated at 10% of the Flight System totals (WBS 2.1)

- Development Reserves are estimated at 20% of all other Development costs

8.5.3 Launch Costs

- Launch Approval (& EIS) is estimated at 1% of the Design cost
- Launch site ground processing costs are estimated at 3% of the Launch Vehicle cost
- Specific Launch Vehicle cost is assumed to be $2,000/kg
- Launch Vehicle costs do not account for volume restrictions and assume 100% utilization (each launch vehicle is filled to its maximum mass capability)

8.5.4 Operations

- Assumes 15-year operating lifetime
- Includes maintenance and refurbishment of select system sub elements and consumables/propellants
- Steady-state annual costs are estimated at 6.6% of Development RC total
- Operations costs do not include reserves

8.6 Cost Summary

Given the assumptions discussed above, a very preliminary total estimated life cycle cost for the Mars transportation architecture studied in Phase I was estimated at ~$90B over 15 years of operation. An example of a life cycle cost summary output is shown in Figure 8-3. Cost estimates for Development totaled ~$50B ($5B/year for 10 years) and Operations for 15 years totaled ~$40B (averages to <$3B/year). These are initial estimates and the ability to evaluate potential cost saving approaches is currently under development. It needs to be emphasized that more important than the actual cost estimate is our ability, with the models developed in Phase I, to begin to study the effects of different mission, cost, vehicle, and resource options.

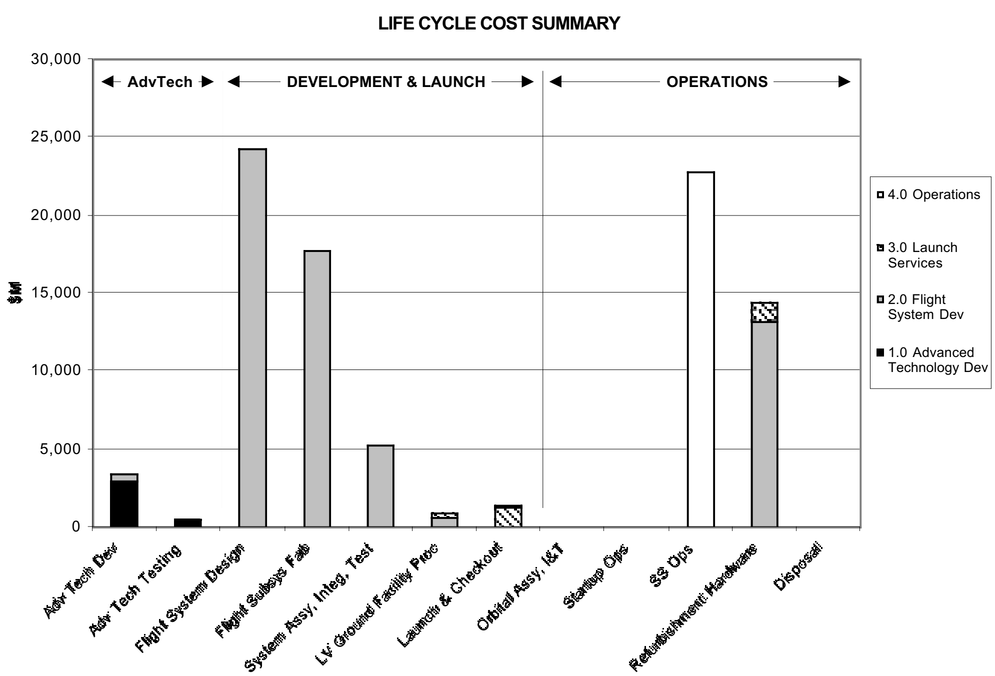


Figure 8-3 Example Life Cycle Cost Results

9 Potential Implications to Mars Exploration Planning

9.1 Introduction

In this section we provide an initial evaluation of the potential implications of this new concept for a Mars transportation architecture on the planning and technology development for future human and robotic exploration of Mars. For the purposes of this discussion, we shall make the assumption, which has not yet been tested fully in Phase I, that the Mars transportation architecture presented in this report is ***the*** desired approach to supporting a future permanent Mars Base. We then ask, what are the implications to future NASA Mars exploration planning of having the goal of establishing a Mars transportation architecture by 2035.

If one can envision an optimized interplanetary transportation systems architecture, then one can take steps today that will enable it. These steps could include the setting of key technology goals to insure technology advance meets the future need. Other steps include embarking on robotic pathfinder missions to explore Mars, Phobos and the Moon and to search for *in situ* resources that are useful in any transportation systems architecture.

9.2 Robotic Exploration

The implications of this concept to robotic Mars exploration planning potentially include 1) a different approach for the use of interplanetary orbits, 2) the selection of alternative mission targets, 3) the development of technology demonstration experiments, and 4) the detailed search for *in situ* materials and the surface context in which they are found.

9.2.1 New Interplanetary Orbit Technology and a Mother/Daughter Spacecraft Concept

Current robotic missions to Mars, including possible surface sample return missions use traditional point-to-point trajectories between Earth and Mars. If cyclic orbits are to become an important aspect of future human Mars exploration, the robotic exploration program might consider the application of cyclic orbits in robotic exploration. The use of cyclic orbits might have some advantages over traditional point-to-point trajectories because of the ability to reduce mission-specific spacecraft requirements. For example, a very reliable, capable interplanetary ***Mother*** spacecraft could be placed on an Up Escalator orbit. Smaller, mission specific ***Daughter*** satellites could be launched toward the Mother spacecraft on hyperbolic trajectories. Once reaching the vicinity of Mother spacecraft the more capable Mother spacecraft would rendezvous and dock with the Daughter satellite. The Mother spacecraft could provide all the interplanetary communications, navigation and propulsive capability for the Daughter satellite. Because these capabilities can be relatively expensive elements for a spacecraft to have, eliminating these elements may make the Daughter satellite less expensive compared to traditional interplanetary spacecraft. Once at Mars the Daughter satellite could take advantage of planned Mars orbital communications relay satellites for communications back to Earth. The Daughter satellite could bring additional propellant to the Mother if necessary. In addition to potential long-term cost advantages, there would be good experience gained on the operations of spacecraft in cyclic orbits (hyperbolic rendezvous, deep space autonomous docking, shared resource allocation) that would be useful in the future. Thus, an implication of developing a Mars transportation architecture is near-term focus on the development of cyclic orbit technologies with possible benefits of robotic exploration vehicle cost savings.

9.2.2 *In situ* Materials, Mission Targets and Technology Demonstrations

The existence of water, or even just hydrogen, on the Moon or Phobos could have a dramatic impact on future plans and technology development for Mars exploration. The use of *in situ* materials can significantly reduce the LEO launch mass requirements for a Mars transportation architecture. Water broken down into its component molecular states of oxygen and hydrogen is rocket propellant. Hydrogen could be combined directly with oxygen for propulsion as with the current Space Shuttle. Or hydrogen could be combined with carbon to make methane, a more easily stored form of chemical energy. If LEO launch costs remain high (they haven't been dropping quickly), ISRU could have a significant effect on the overall cost of a Mars transportation system.

At this point however a number of questions still exist about the availability of certain materials and the context in which they are found. We do not know whether or not water in the form of ice exists at the lunar poles or what is its abundance. We do not know whether or not useful quantities of water or oxygen can be found at Phobos. We do not know how much water will be found in Martian regolith in places that we might wish to explore. These are all questions that can be answered by relatively simple, low-cost robotic missions. Past robotic missions have not unambiguously resolved the issue of water at any of these bodies listed above. Unfortunately, there are also no planned missions to resolve the uncertainties at this time, yet the existence of water and knowledge of its physical context and abundance could have profound implications on future human Mars exploration and transportation systems. A concept for an Earth-to-Mars transportation system could generate the context necessary to get such missions off the ground.

Once we have established the location of various resources and the physical form in which they exist, we can then develop ISRU systems and perform technology demonstrations in relevant environments. An added benefit of ISRU is the use of *in situ* resources for robotic exploration. Thus, another implication of pursuing a Mars transportation architecture would be a near-term emphasis on understanding resource distribution at Mars, Phobos and the Moon.

9.3 Human Exploration and Development of Space (HEDS)

The implications of this concept to the NASA HEDS program potentially include 1) a focus on permanent Mars exploration instead of brief expeditions, 2) an initial emphasis on human exploration of the Moon and Phobos both as intermediate targets of exploration and as testbeds for space demonstration of self-sufficiency applicable to Mars exploration, 3) the identification of the vehicle and system steps to Mars transportation infrastructure and its development, 4) the development of relevant transportation, power and inhabitation technologies, and 5) a context for articulating the rationale for the incremental development of components of the Mars transportation architecture.

9.3.1 Flags and Footprints or Permanent Human Presence on Mars

The choices in future Mars exploration planning are to 1) follow the Apollo expedition paradigm characterized as *flags and footprints*, which has resulted in an absence of human exploration of the Moon for over a quarter century or 2) begin Mars exploration by establishing and using elements of the infrastructure needed to provide a permanent presence of humans on Mars. If we take the first choice, we will get to Mars but in the process we may have expended so much political capital and public will that, like Apollo, Mars exploration is eventually abandoned.

9.3.2 Phobos and the Moon

The Moon and Phobos clearly have an important role to play in any future transportation architecture of a sustained human base on Mars. However, at this time we do not have enough information to know exactly what resources exist on those natural satellites. At the current time there are no HEDS Lunar or Phobos exploration mission plans. This is despite the fact that each body could represent an intermediate and easier target of human exploration compared to Mars. Energetically it is easier to get to and from the Moon and Phobos than the surface of Mars. If the Moon were found to have enough water at the lunar poles or Phobos were found to have water in its regolith, they could be places to begin testing the philosophy of self-sufficiency in space before tackling the enormous problem of permanently living and working on the surface of Mars. In this fashion the Mars exploration program could grow in an evolutionary manner with more modest resource requirements and with a higher probability of success. Thus, an implication of pursuing a Mars transportation architecture is near-term robotic mission priority for exploring the lunar poles and the surface of Phobos.

9.3.3 Steps to Infrastructure and Relevant Technology Development

If we know what systems and vehicles will be required in the future to support a Mars transportation architecture, the steps to the development of this infrastructure can be constructed including the long-range planning and costing, advanced technology development, advanced system development and flight-testing. Intermediate vehicle or surface systems can begin to be used, perhaps for lunar or Phobos exploration, as they are developed with the knowledge that their efficiency is driven not necessarily by their immediate use but by their eventual application in the overall infrastructure. A key example of the to provide a context for future technology development is the need for nuclear power generation systems in a future architecture. If such systems are not absolutely needed, or in fact are more expensive than solar photovoltaic systems, then we can save considerable precious resources by pursing the solar option.

The concepts envisioned by this systems architecture have a potential role to play in the expedition phase of Mars exploration. The application of these orbit and systems concepts in the expedition phase of Mars exploration may serve to reduce overall mission development costs and improve overall mission reliability and safety. Once launched into cycling orbits Astrotels can orbit indefinitely as long as they are periodically maintained, improved and supplied with orbit correction propellants. In addition, the result of embracing such a mission concept early in an expedition phase means that a permanent inhabitation phase of Mars is all the more closer. An implication of pursuing this path toward a Mars transportation architecture is near-term development of intermediate systems of immediate benefit to human space exploration, which have a role to play in the expedition phase of human Mars exploration.

9.3.4 Context for Human Space Enterprise Development

Many people question the value of the International Space Station. One contributing reason for this skepticism may be the lack of future context of space stations beyond gaining zero-g experience, carrying out limited scientific investigations and exploiting the potential commercial uses (materials processing, entertainment and tourism). It is clear that in the Mars transportation architecture a space station is a key transportation node where crews will be transferred, vehicles will be refurbished and upgraded and cargoes will be collected for transport to deep space. A context for a space station sets its requirements on a firm foundation from which future plans can be formed and advocated. Thus the implication of pursuing a path toward a Mars transportation architecture is that the rationale for the International Space Station, and follow-on human enterprises, will be clear.

9.4 Summary of Implications

In short by establishing the goal of developing a Mars transportation architecture NASA will effectively provide context for and direction to planning and mission activities within its robotic exploration and HEDS enterprises. Without a framework to place in context the expensive future work, we are apt to expend two of our most valuable resources, time and money, on extraneous technology and system development paths.

10 Future Work

10.1 Introduction

Although we have achieved considerable progress in developing a Mars transportation system in support of a future Mars base, there is much additional work remaining to define the optimum Mars transportation architecture. Additional work is suggested in:

10.2 Research and performance comparison of various cyclic orbits, low-thrust trajectories, and innovative new orbit dynamics concepts;

10.3 Research, analysis and design of innovative aero-assist concepts;

10.4 Definition and development of ISRU options and their overall performance comparison;

10.5 Design and definition of various vehicles throughout the transportation architecture;

10.6 Further development of the Mars Astrotel Model Architecture (MAMA) engineering and life cycle cost model; and

10.7 Identification and pursuit of pathways to a Mars transportation architecture development.

10.2 Cyclic Orbit and Celestial Mechanics Concepts

Redesign of the Aldrin Cycler orbit would be desirable to eliminate large propulsive maneuvers. A search should be made for new cyclers and semi-cyclers and to explore the feasibility and benefit of Earth-Moon cycler concepts. Low-thrust trajectories (using Solar Electric Propulsion) will continue to be investigated for use in the transport of bulk cargo. The feasibility of employing aero-gravity assist lifting bodies to eliminate propulsive maneuvers should be examined.

10.3 Advanced Aeroassist Technology

An aero-assist trajectory model needs be developed to simulate controlled aerocapture of taxi vehicles at Mars, with special emphasis on the highest approach speed cases where it is difficult to create the necessary centripetal force by aero-assist means alone. This model should simulate the following aspects of the aerocapture process: entry and descent; level-off and/or pull-up maneuvers; engine burns in any direction; use of a ballute at low density under maximum g-load limits; altitude control by rolling the vehicle about the velocity vector; and navigation to the desired exit speed, path angle and location appropriate to the desired apogee. Such a model should be used to perform detailed analysis of new aero-assist concepts identified during Phase I, especially the innovative new concept for propulsive thrusting during aero entry to reduce g–load.

10.4 In Situ Resource Systems Concepts

In Phase I a preliminary selection of baseline systems was made from attractive ISRU options. In Phase II these options should be examined in greater detail and end-to-end performance in the Mars transportation architecture should be compared. Support to the Mars transportation system architecture development by experts in the field should be continued, thereby lending technical expertise to preliminary systems design trade studies of competing resource utilization options. Development of *in situ* resource utilization system models need to be pursued for propellant production on the Moon, Phobos, Mars' surface and associated space propellant depots. Development of system designs for resource operations and plants also need to be pursued. There needs to be an emphasis on the problems of excavating and processing materials in the near-zero-g, vacuum environment of Phobos.

10.5 Design Concepts and Options for Mars Transportation Systems

The Astrotel and the Taxi life support, radiation shielding and structural designs need further development and definition. During Phase I there was limited design definition of Earth and Mars Spaceports, the SEP Cargo Freighters, the Lunar Water and Phobos LOX Tankers, and the LEO and Mars Shuttles. Design and interface requirements for these vehicles needs to be identified; computer-aided designs developed; and refurbishment, repair and upgrade plans need refinement. The impact of subsystem commonality between vehicle systems needs to be assessed to reduce overall life cycle costs.

10.6 Mars Astrotel Model Analysis (MAMA) and Life Cycle Costing Models

During Phase I, there were two separate MAMA modules. One sub-module included the element requirements and supporting analyses and the other sub-module integrated those analyses into overall mass requirements and life cycle costs. In Phase II, the element requirements, supporting analyses sub-module and the system integrator (MAMA SI) sub-module need to be fully integrated into one module. If feasible, we should incorporate the capability to simultaneously evaluate multiple design options to greatly facilitate conducting trade studies.

In Phase I very rough cost references were used for each element of the architecture. In Phase II, cost references used for each transportation sub-element should be reviewed/revised to ensure that the most appropriate cost reference is being used for each subsystem/component. Enhanced estimating methodology should be developed including cost estimating relationships for lower-level items. Potential cost saving approaches need to be identified and methods developed to assess their impacts. If feasible, a set of costing inputs to capture cost savings approaches and transportation architecture design options within MAMA should be developed.

10.7 Pathways to Architecture Development

Someday scientists and explorers will regularly travel to Mars for research and exploration as they now travel to Antarctica. Knowing this eventuality enables us to plan effectively for the future. The pathways to the future implementation of a Mars transportation architecture and its potential impact on near-term space development should be explored with NASA robotic and human Mars program planners and managers. For example, robotic Mars exploration can potentially advance the date for an eventual establishment of a Mars transportation architecture by 1) gaining experience in use of a cyclic interplanetary orbits, 2) selecting missions to answer key ISRU questions, and 3) demonstrating technologies that are relevant to a Mars transportation architecture. In addition, the NASA HEDS program can potentially steadily advance toward the goal of permanent human habitation of Mars by a) focusing on permanent Mars habitation instead of brief expeditions, b) emphasizing human exploration of the Moon and Phobos both as intermediate targets of exploration and as testbeds for space demonstration of self-sufficiency applicable to Mars habitation, c) the identifying incremental vehicle and system steps to Mars transportation infrastructure and its development, d) developing relevant transportation, power and habitation technologies, and e) embracing the goal of permanent human habitation of Mars as a context for articulating the rationale for incremental development of components of a Mars transportation architecture.

11 Summary

This report describes the work accomplished and results obtained during Phase I of the development of an innovative and new concept for Cyclic Visits to Mars via Astronaut Hotels in support of the NASA Institute for Advanced Concepts. During Phase I, we developed a new Mars transportation architecture that is designed to provide permanent human habitation of Mars. During Phase I, we achieved our objectives by:

- Developing a conceptual Earth-to-Mars transportation system architecture for a sustained Mars base circa 2035 that focused on cyclic orbit and aeroassist technology, small, human transport vehicles, and maximum use of available *in situ* resources, and
- Developing and optimizing advanced celestial mechanics, aero-assist, ISRU, and vehicle systems concepts for low-cost, low-energy, frequent and short-duration trips to and from Mars.

Features of this new Mars transportation architecture include:

- Five month human flights between Earth and Mars on cyclic orbits,
- Small, highly autonomous human transport vehicles or *Astrotels*,
  - In cyclic orbits between Earth and Mars
  - Using Solar Electric Propulsion for orbit corrections
  - That are Untended for more than 20 out of 26 months
  - Without artificial gravity
- Fast-transfer, aeroassist vehicles, or *Taxis*, between Spaceports and the cycling Astrotels,
- Low energy, long flight-time orbits and unmanned vehicles for the transport of cargo,
- Use of *in situ* resources at Mars, Phobos and the Moon for propulsion and life support
- Environmentally safe, propulsion/power technology

This new architecture provides low-cost, frequent-access to Mars by scientists and explorers; has systems concepts that can be utilized in robotic and human expedition phases of Mars mission exploration; and sets a framework and context for future technology advance, HEDS development and robotic mission exploration.

There were five tasks during Phase I:

1) Task 1 Define Conceptual Design Requirements and Assumptions
2) Task 2 Analyze Celestial Mechanics
3) Task 3 Develop Conceptual Transportation System Architecture Design
4) Task 4 Estimate Transportation System Costs
5) Task 5 Planning and Reporting

During Phase I, we met 100% of the objectives by completing all the tasks. This document includes reports on the first four tasks. Its submission completes the fifth task.

12 Acronyms

Ah	Ampere hour
AIT	Assembly Integration Test
AM0	Air mass zero (a space-like environment)
AOA	Angle of attack
AOTV	Aerobraking Orbit Transfer Vehicle
aphelion	The point in the orbit of a planet or comet that lies farthest from the sun.
apoapsis	The point in an orbit where the moving body lies farthest from the celestial body around which it orbits.
Astrotel	*Astro*naut Ho*tel*
ATD	Advanced Technology Development
AU	Astronomical unit or A unit of distance equal to the average distance between the earth and the sun, about 93 million miles.
Ballute	A balloon-like parachute used as a braking system for a sounding rocket or other spacecraft.
BOL	Beginning of life
C_d	Drag coefficient
CIGS	Copper indium gallium diselenide
CIS	Copper indium diselenide
C_l	Lift coefficient
CDR	Critical Design Review
DA	Downward acceleration
Delta v	ΔV, a mathematical expression for a change in velocity, especially referring to spacecraft; designates the velocity change required to transfer a spacecraft from one orbit to another.
DOD	Depth of discharge
DS1	Deep Space One Mission or spacecraft
ECLSS	Environmental Control and Life Support Systems
EVA	Extra vehicular activity
g-load	The ratio of an applied force on an object to the force due to gravity acting on the body at sea level.
g	Average acceleration of gravity at Earth surface = 9.80665 m/s^2
g load	The ratio of an applied force on an object to the force due to gravity acting on the body at sea level.
GaAs	Gallium arsenide
GaSb	Gallium antimonide
GCR	Galactic cosmic radiation
g_{drag}	Drag component of g-load
GEO	Geosynchronous earth orbit

g_{lift}	Lift component of g-load
GSFC	Goddard Space Flight Center
HEDS	Human Exploration and Development of Space
I&T	Integration and Test
InGaP	Indium gallium phosphorus
IPS	Ion propulsion system
IR	Infrared
Isp or I_{sp}	Specific impulse
ISRU	*in situ* resource utilization
ISS	International Space Station
ITFT	Iowa Thin Film Technologies
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
km/s	Kilometers per second
kW	Kilowatt
kWe	Kilowatt electric
L/D	lift/drag ratio
L1	Lagrangian points: five points in the orbital plane of two large bodies at which any small (essentially massless) object can remain in equilibrium relative to the two large bodies.
LEO	Low earth orbit
LH	Liquid hydrogen
Libration point	Any of five positions in the plane of a celestial system consisting of one massive body orbiting another at which the gravitational influences of the two bodies are approximately equal.
Line of apsides	The major axis of an elliptical orbit.
LOX	Liquid oxygen
LSS	Life support system
LWT	Lunar Water Tanker
M/C	Mid course
m/s	meters per second
MAMA	Mars Astrotel Model Architecture
m_f	Final mass
m_i	Initial mass
MLI	Multi-layer insulation
mN	$1x10^{-3}$ Newtons
MOCM	Mission Operations Cost Model
MOD	Meteoroid and orbital debris
M_{ps}	Propulsion/power mass

mt	metric tonnes
MW	Megawatt
NASA	National Aeronautics and Space Administration
NCOS	National Commission on Space
NCR	Non-recurring costs
NIAC	National Institute for Advanced Concepts
NRC	National Research Council
NRFC	Non-regenerative fuel cells
NSTAR	NASA Solar Electric Propulsion Technology Application Readiness
NTR	Nuclear thermal rockets
PAT	Propellant augmentation tank
PBO	A structural adhesive polymer that yields useful bonds above 260°C and has good high-temperature aging characteristics.
periapsis	The point at which an orbiting celestial body makes the closest approach to its primary body during its orbital revolution around that body.
perigee	The point at which an object that is in orbit around the earth, such as the moon or an artificial satellite, is nearest to the earth.
perihelion	The point at which an object in orbit around the sun is nearest to it.
PMAD	Power management and distribution
PPU	Power Processing Unit
PV	Photovoltaic
R	Radius
RC	Recurring costs
Regolith	Soil
ROM	Rough order of magnitude
RRU	Refurbishment, repair and upgrade
SAIC	Science Applications International Corporation
SCARLET	Solar Concentrator Array with Refractive Linear Element Technology
SEP	Solar Electric Propulsion
SI	System Integrator
SIT	Segmented Ion Thruster
SLA	Stretched Lens Array
SPE	Solar flare particle events
SPE	Solar Proton (or Particle) Events
SSE	Space Science Enterprise
STS	Space Transportation System (Shuttle)
TBD	To be determined: estimated requirements shown in brackets.
TPS	Thermal Protection System
TRL	Technology Readiness Level

V^2	Velocity
V_e	Entry speed
V_e	Exhaust velocity
$V_{horizontal}$	Horizontal velocity
V_{inf}	Velocity at infinity
VISIT	Versatile International Station for Interplanetary Transport
W	Watt
WBS	Work Breakdown Structure
Wh	Watt hour
W/m^2	Watts per meter squared
W/cm^2	Watts per centimeter squared