NASA Exploration Team (NEXT)

Design Reference Missions Summary

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National Aeronautics and
Space Administration

FOREWORD

<To be written>

The NASA Exploration Team (NEXT) has been chartered to develop a 21st Century integrated long-term space exploration strategy for NASA. The NEXT recommends the Agency adopt a paradigm for space exploration that is driven by science and discovery objectives, as modified by National interests, and enabled by technology advancements. The ultimate goal is to satisfy the newly defined Agency vision[1] and mission[2] statements.

This document summarizes in condensed form the vision and strategy for space exploration that has been developed and articulated by the NEXT, and applies that to design reference missions currently under consideration for implementing this vision. The design reference missions described within represent present efforts by the NASA Exploration Team to devise strategies for achieving science-driven exploration goals but should not be considered as mission proposals or preferred solutions. Vehicle and mission concepts are taken to levels of design fidelity appropriate for understanding technology benefits and overall feasibility of the studied approaches. It is our intent that this document serve as a reference from which we can continuously compare and contrast other new innovative approaches to achieve NASA's long-term goals for exploration.

[1] "To improve life here, to extend life to there, to find life beyond."

[2] "To understand and protect our home planet, to explore the universe and search for life, to inspire the next generation of explorers, as only NASA can."

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ACRONYMS AND ABBREVIATIONS

CTM	Chemical Transfer Module
CTV	Crew Transfer Vehicle
DART	Dual Anamorphic Reflector Telescope
DRM	Design Reference Mission
EVA	Extra-Vehicular Activity
FAIR	Filled-Aperture Infrared
FSS	Far Side Sentinel
HEDS	Human Exploration and Development of Space
IHS	Inner Heliospheric Sentinels
ISTP	Integrated Space Transportation Plan
ISS	International Space Station
LEO	Low Earth Orbit
LWS	Living With A Star
NEP	Nuclear Electric Propulsion
NEXT	NASA Exploration Team
OASIS	Orbital Aggregation and Space Infrastructure Systems
OSS	Office of Space Science
RLV	Reusable Launch Vehicle
SEC	Sun-Earth Connection
SEP	Solar Electric Propulsion

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1.0 INTRODUCTION

The NASA Exploration Team (NEXT) was chartered to develop a 21st Century integrated long-term space exploration vision for NASA. The NEXT is a cross-Enterprise, cross-Center team responsible for maintaining a multi-disciplinary approach toward future space exploration planning. The team is responsible for integrating a large part of NASA, encompassing multiple space programs within all five NASA Enterprises, into a single vision described below. The team is co-chaired by both the Office of Space Flight and the Office of Space Science. A Lead Scientist, also from the Office of Space Science, is assigned to the NEXT.

NASA chartered the NEXT to create, maintain, and implement an integrated long-term vision for human/robotic exploration of the Solar System and Universe. Under this chartered mission, the team is tasked to:

- Generate scientific, technical, and programmatic requirements to drive investments, which will enable each new phase of human/robotic exploration.
- Conduct advanced concepts analyses and develop new innovative approaches for space exploration.
- Integrate technology programs internal and external to NASA to align programs with the vision (where applicable).
- Identify and promote commercial and space development opportunities synergistic with the vision.

This document captures the details of mission concepts for achieving the science objectives. In accomplishing the science objectives, the design reference missions provide a common reference point to facilitate multidiscipline analysis to determine the critical areas for investments in technology.

2.0 EXPLORATION VISION

The NEXT Exploration Vision is addressed by a progressive, evolutionary approach for science discovery within our Solar System with humans and robots working together. The accomplishment of this expansion is envisioned in four discrete steps. The first step, currently underway, is to routinely permit humans to leave the Earth's influence and pursue productive scientific missions in space while expanding and enhancing the capabilities of robotic spacecraft to conduct scientific investigations beyond LEO and expand our understanding of the space environment. The second step is to gain knowledge through—and build on—these experiences. During this second phase and all subsequent stages, we will continue to expand our robotic capabilities through technology investments so that these spacecraft can conduct ever increasing scientific investigations safely and affordably. As these trip times become commonplace, humans can then move toward increasingly longer trip times inclusive of a stay at a particular destination as the robotic spacecraft pave the way. Finally, as we grow accustomed to such journeys, we can then establish a semi-permanent or permanent human presence at an extraterrestrial destination.

Top-level requirements for NASA space exploration missions evolve from the vision and mission statements. At the heart of this vision and mission are six fundamental questions.[3] The NEXT developed a set of Level 0 requirements to guide the Agency in executing the exploration aspect of the strategic plan, to answer the fundamental questions. These requirements, in expanded form[4], are intended as well to guide the further development of exploration strategy, in a manner that maintains strategic integrity across all Agency exploration activities.

The ability to discover is uniquely human. Because of the cost and risk associated with human presence in space, direct human participation in exploration beyond LEO has been limited. Presently, all space exploration beyond LEO is conducted with robotic spacecraft. Through investment in revolutionary technologies, we can begin to develop new robotic, human and integrated human/robotic capabilities, which will allow a systematic, affordable and safe expansion of humans and robots in the discovery process beyond LEO.

Exploration Grand Challenges

The imperative for space exploration by humans and robots has been articulated by NEXT in the form of three "grand challenges."[5] These challenges provide a link to exploration of life in the universe, and are formulated as:

How Did We Get Here?

- How did life arise on Earth?
- How did intelligence evolve on Earth?
- How did the Earth and Solar System form and evolve?

[3] See NASA 2000 Strategic Plan
[4] Space Exploration Top Level Requirements (Draft), 6/11/02
[5] NASA's Exploration Team: Products, Evaluations, and Priorities. February 2002.

Grand Challenge	Science Questions	Pursuits	Activities	Destina-
How Did We Get Here?	Solar System evolution 	History of major Solar System events	Planetary sample analysis: absolute age determination "calibrating the clocks"	Moon Mars Asteroids
Where Are We Going?	Humans adaptability to space 	Effects of deep space on cells	Measurement of genomic responses to radia-	Beyond Van-Allen belts
	Earth's sustainability and habitability 	Impact of human and natural events upon	Measurement of Earth's vital signs "taking the pulse"	Earth orbits Libration points
Are We Alone?	Life beyond the planet of origin 	Origin of life in the Solar Sys- Origin of life in the Universe	Detection of bio-markers and hospitable environments	Mars Europa Titan Cometary nuclei Libration points

Figure 2.0-1: Science Traceability to Exploration Grand Challenges

Where Are We Going?
- What is the fate of life on Earth?
- What is the interaction between life and the Earth's environment?
- How do we optimize the role of humans in space?

Are We Alone?
- Are there other abodes for life in the Solar System?
- Are there other abodes for life in the Universe?

The grand challenges also provide traceability to the basic science questions and pursuits currently under consideration by NEXT in the form of design reference missions, as mapped above in Figure 2.0-1.

"Stepping Stone" Capabilities

The basis of the new NASA exploration vision is sustained development of "stepping stone" capabilities that enable affordable, safe and reliable space exploration. The stepping stones – spheres of human/robotic presence – are differentiated as follows:

1) Earth and LEO ("Getting Ready")
2) Earth's Neighborhood ("Getting Set by Doing")
3) Accessible Planetary Surface ("Going for Visits")
4) Sustainable Planetary Surface ("Going Beyond and Staying")
5) Go Anywhere, Anytime

Each of these regimes can be associated with sites at increasing astrophysical distance from Earth, but their complete characterization over time requires as well specification of these parameters:

- Frequency of human presence
- Duration of human presence
- Degree of mastery of humans over operations (i.e., manual operation, teleoperation, or programmed operation)

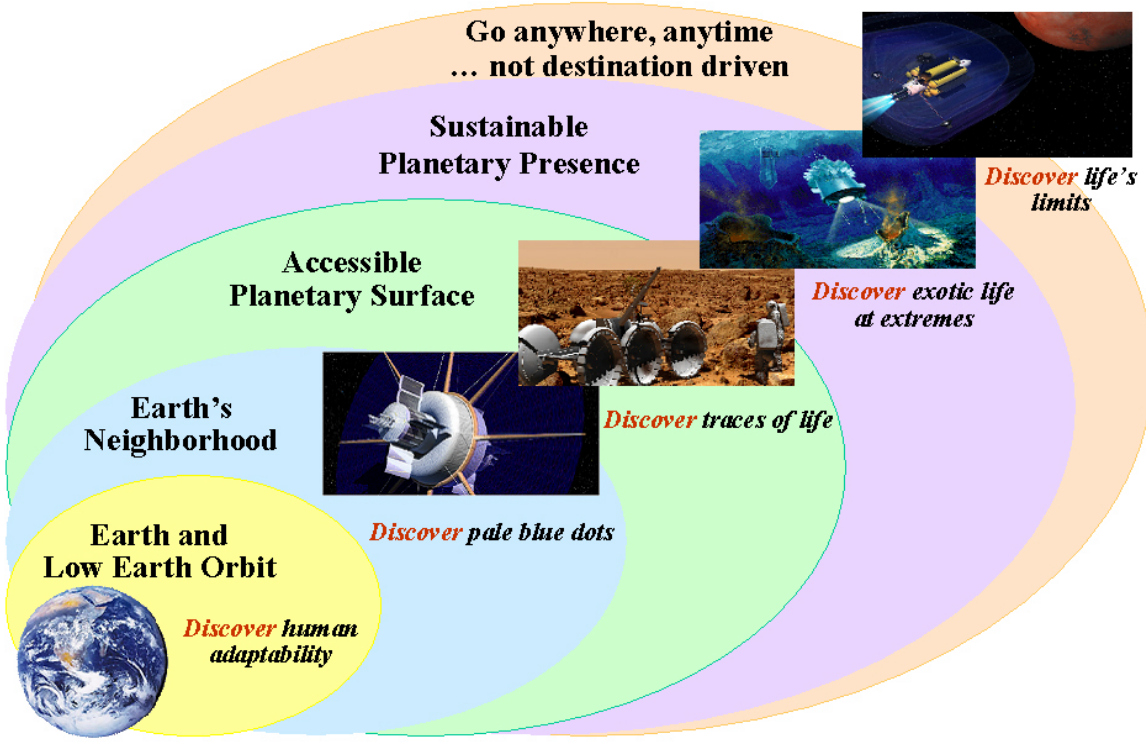


Figure 2.0-2: Exploration Stepping Stones

The NEXT envisions that infrastructure will be built and used in each regime as the basis for development of capabilities and technologies for subsequent regimes. That is, operations in each regime will serve to develop and test capabilities and technologies required to explore and establish the succeeding regime. While exploration concepts and technologies are yet in the early stages of development and roadmapping, the NEXT is proceeding on the basis of architectures that maximize reuse of systems, and minimize mass to Earth orbit and ΔV to destination.

Applying the foregoing, the logical structure for prioritizing transportation and technology investments over the long term is thus:

First priority is investment in infrastructure and technologies that support present science and discovery conducted in LEO and preparations for operation in Earth's neighborhood.

Second priority is investment in infrastructure and technologies that support science and discovery conducted in Earth's neighborhood, and that supports preparations for visits to accessible planetary surfaces.

Third priority is investment in infrastructure and technologies that support science and discovery conducted on accessible planetary surfaces, and that supports preparations for sustained presence on accessible planetary surfaces.

Fourth priority is investment in infrastructure and technologies that allow humans to go anywhere, anytime.

The NEXT stresses that this is a time sequence priority order, not an absolute importance priority order. Moreover, certain advance investments supporting later priorities must be made in earlier budget cycles, consistent with normal NASA program management practice. The peaks of lower priority investments appropriately may be deferred to future budget cycles, but it is essential that present investment decisions do not preclude execution of the sequential expansion inherent in the NEXT vision and strategy. In addition, the NEXT requirements are structured such that requirements and capabilities from one stepping stone phase carry forward to subsequent phases unless specifically modified.

Requirements for space exploration capability investments are mapped to the NEXT stepping stones below.

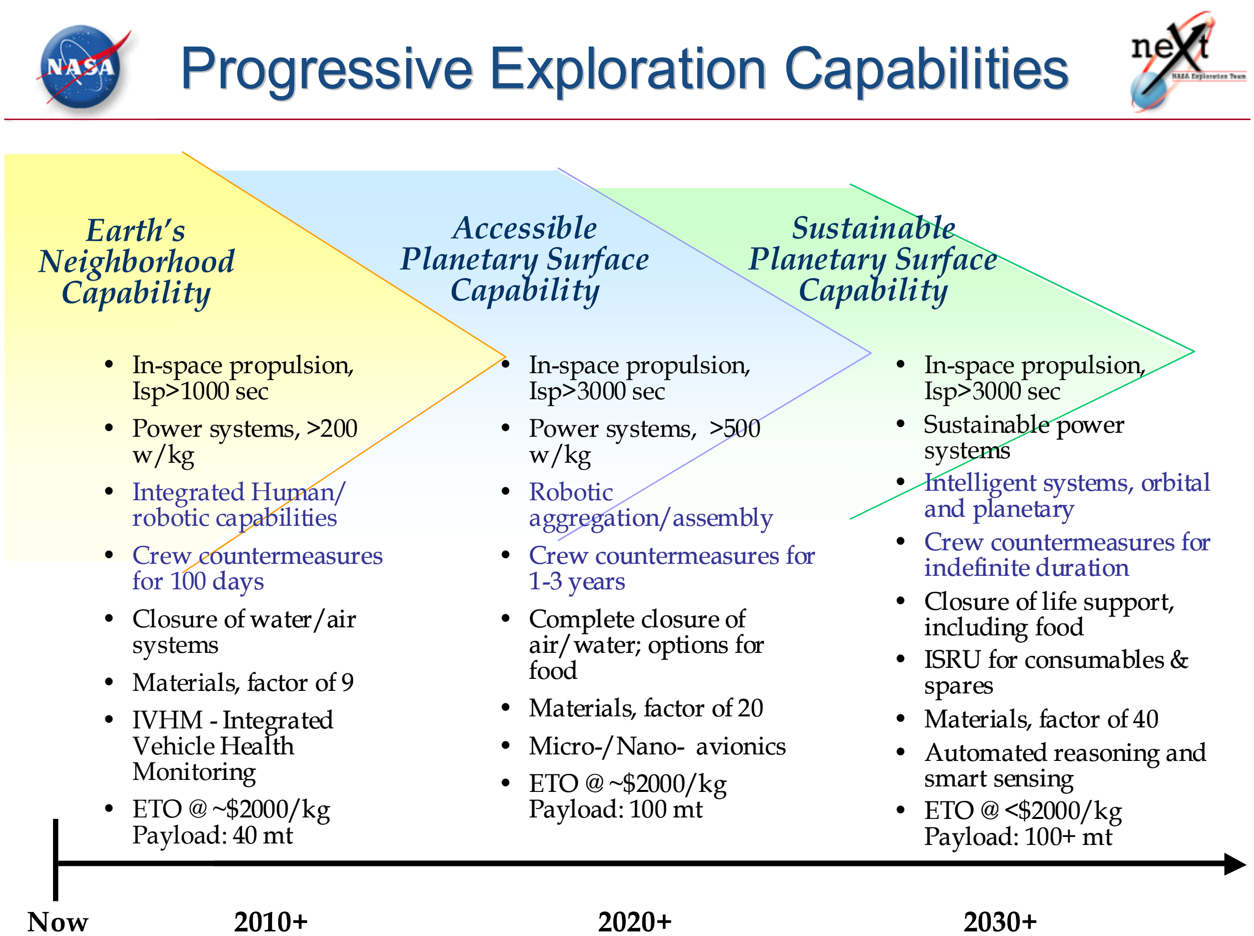


Figure 2.0-3: NEXT Philosophy of Progressive Capabilities for Exploration

3.0 DESIGN REFERENCE MISSIONS

This section of the document describes the design reference missions currently under consideration by the NASA Exploration Team. These missions describe the means by which humans and robots will leave Earth, travel to a destination, and achieve the scientific objectives identified in the NEXT integrated space exploration vision. The missions include, but are not limited to:

1. Earth's Neighborhood Transportation Infrastructure Development
2. Construct, Deploy, and Service Large Science Platforms
3. Lunar Exploration
4. Orbital Aggregation and Space Infrastructure Systems
5. Mars Exploration
6. Sun-Earth Connection Solar Sentinel Missions
7. Human Outer Planet Exploration

Each section devoted to a design reference mission (DRM) includes a brief overview of the mission and provides a linkage between the mission/program to the overall NEXT themes for integrated space exploration. Mission options for various architectural approaches and technology implementations are identified to scope the DRM trade space. DRM stakeholders have also furnished a description outlining the major mission elements that have been studied, key technology investments and mission requirements to enable implementation of the DRM, and sources of additional information for readers of this document.

3.1 Earth's Neighborhood Transportation Infrastructure Development

The design reference missions under consideration for exploration within the Earth's Neighborhood require reliable, low-cost transportation of mission crews and cargo elements between Earth and the various destinations of interest. Here, the region of space referred to as "Earth's Neighborhood" by the NASA Exploration Team includes the Lagrange points of the Earth-Moon system, the lunar surface, and the collinear Lagrange points of the Sun-Earth system. The geometry of the Earth's Neighborhood is illustrated in Figure 3.1-1. It is at these destinations that various high-value scientific missions may be conducted through the use of human explorers and robotic systems, namely the exploration of the lunar surface, the construction, deployment, and servicing of advanced science observatories such as solar sentinels and astronomical telescopes, and other potential investigations.

The cornerstone of this design reference mission is the emplacement of a construction, servicing, and mission-staging platform in the vicinity of the Moon, specifically at the Lunar L_1 Lagrange point. This facility will serve as a "gateway" to future human exploration of space, including other Lagrange points, the lunar surface, and various other destinations within Earth's Neighborhood and beyond. Also under consideration are human transportation systems for delivering crews between Earth, Lunar L_1, and the lunar surface, high-efficiency electric propulsion stages for the delivery of mission elements to the above destinations, and the utility of low-energy pathways between Lagrange points called "invariant manifold trajectories". An alternative approach to transportation infrastructure development for exploration in the Earth's Neighborhood is outlined in Section 3.4, Orbital Aggregation and Space Infrastructure Systems.

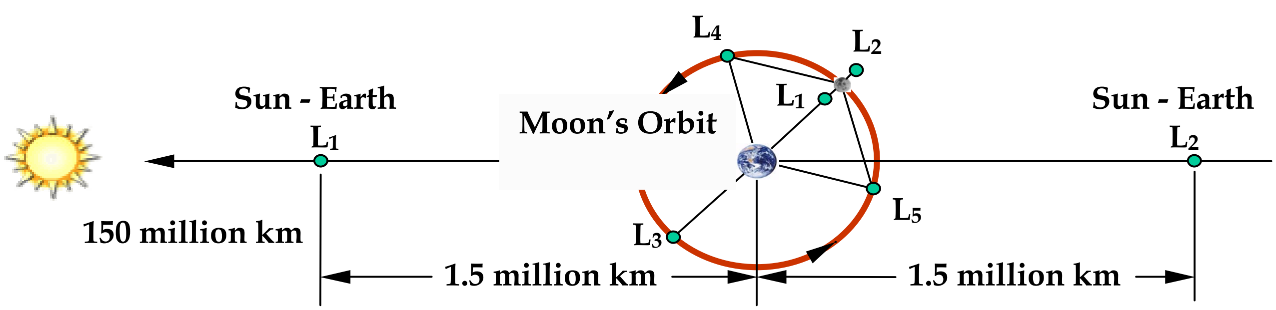


Figure 3.1-1: Geometry of the Earth's Neighborhood

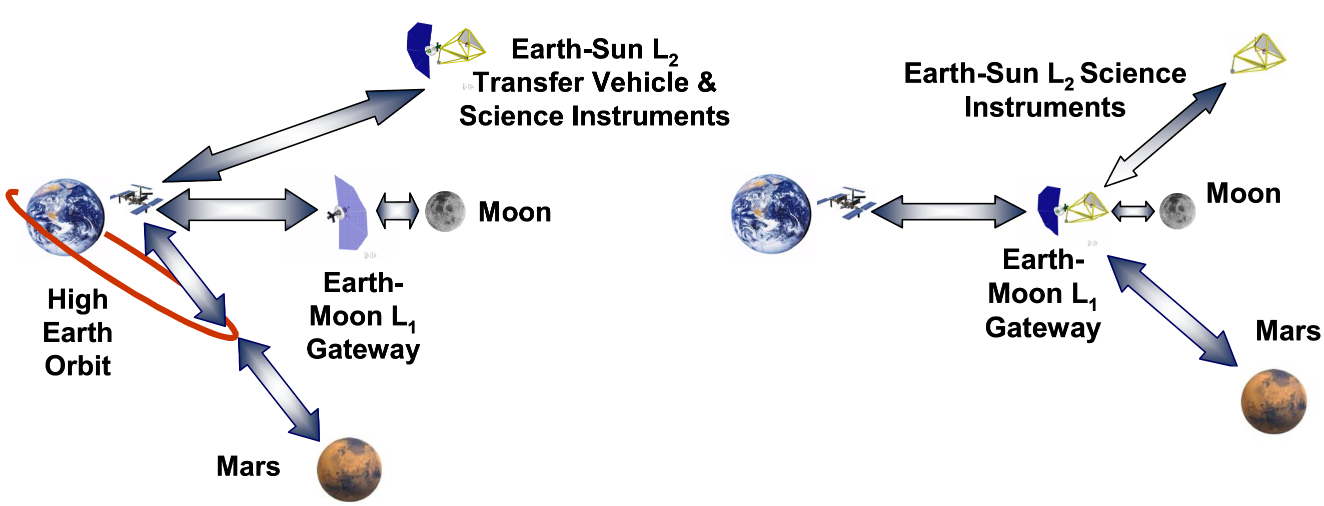
3.1.1 Connection to NEXT Themes and Goals

The NEXT vision for integrated space exploration places its emphasis on investment in transportation infrastructure to enable a progressive expansion of capabilities outward from Earth, and to utilize existing infrastructure when sensible to minimize overall investment. The transportation infrastructure for Earth's Neighborhood is entirely consistent with this philosophy. A strategy to utilize common crew transfer and cargo delivery elements, specifically a single Crew Transfer Vehicle and Solar Electric Propulsion Stage concept, has been adopted for the Earth's Neighborhood architecture. These elements can support the multiple design reference missions envisioned for this architecture by centralizing mission operations at the Lunar L_1 Lagrange point gateway. It is here that a single mission staging and spacecraft assembly servicing depot will be located, namely the L_1 Outpost. This strategy can potentially also support the next missions in the proposed “stepping stone” hierarchy, such as those to the immediately accessible planetary surfaces.

3.1.2 Mission Description

Recent scientific discoveries in the lunar polar regions have sparked renewed interest in human exploration of the Moon, and building the large astronomical facilities to necessary to explore the origins and evolution of the universe will require shifting the point of assembly from Earth-based facilities to on-orbit assembly with human and robotic partners. These new opportunities for scientific investigation in Earth’s Neighborhood have led architecture designers to take revolutionary new approaches for accommodating these various missions in a sensible, integrated fashion. In the past, such mission objectives and destinations were considered on their own basis, with each mission consisting of its own set of transportation infrastructure. This type of approach is typically referred to as an “expeditionary” architecture. Although expeditionary architectures can usually be characterized by having lower cost per mission because of a lower up-front investment in infrastructure and the development of infrastructure specialized for a specific mission, the aggregate program cost for all missions in the architecture is usually higher than an integrated approach. By contrast, an “evolutionary” architecture is one which places its emphasis on the development of infrastructure common to all missions within the architecture and which can be built upon to support future missions. Favoring the evolutionary approach has led to a particular architecture for exploration within Earth’s Neighborhood, central to which is the emplacement of a mission-staging platform near the Moon, specifically at the Lunar L_1 Lagrange point. This facility, the L_1 Outpost, will provide a staging location for human exploration expeditions to the lunar surface and an operational facility for constructing, deploying, and servicing large science platforms. The contrast between expeditionary and evolutionary architectures for

Figure 3.1-2: Expeditionary vs. Evolutionary Mission Architectures



Earth's Neighborhood exploration is illustrated in Figure 3.1-1.

As the figure above demonstrates, instead of developing crew and cargo transportation to support each of the various destinations of interest individually, a common infrastructure can potentially be put in place that supports in part each destination. The central node of this system of infrastructure is the Lunar L_1 Lagrange point.

3.1.2.1 Benefits of Lunar L_1 Staging

The primary goals of the Earth's Neighborhood exploration architecture are to enable both short-duration and extended-stay exploration of the entire lunar surface as well as on-orbit assembly, deployment, and servicing of large astronomical observatories and other science platforms. Utilizing the collinear Earth-Moon L_1 Lagrange point as a mission staging node allows access to all lunar latitudes for essentially the same transportation costs while providing a continuous launch window to and from the lunar surface. In addition, future large-aperture Gossamer telescopes will require on-orbit assembly, calibration, and servicing, and as a result, extensive infrastructure to support these tasks. For many of the science platforms under consideration by NEXT, the collinear Lagrange points of the Sun-Earth system are favored as the final mission operating points. As an alternative to sending humans, robots, and mission support equipment to these far-off destinations to support such tasks, the facility located at Lunar L_1 for staging lunar exploration missions may also be used for construction and servicing science platforms while avoiding other issues arising from assembly in Low Earth Orbit or at the science facility's final destination. An emerging field in orbital mechanics known as invariant manifold analysis has identified potential trajectories between Lagrange points for very little transportation cost.[6] Therefore, utilizing the L_1 gateway as a construction and servicing node for these platforms will enable such low cost transfers while consolidating in-space infrastructure.

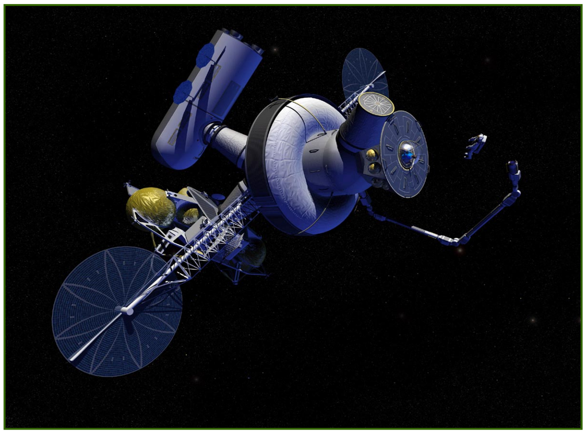
3.1.2.2 Earth's Neighborhood Transportation Infrastructure

The Earth's Neighborhood transportation infrastructure consists primarily of three elements:

1. L_1 Outpost
2. Crew Transfer Vehicle
3. Solar Electric Propulsion Stage

As described above, the L_1 Outpost is a mission staging and science platform construction/servicing facility emplaced at Lunar L_1. It will serve as a node for transferring exploration crews to the lunar surface, a center for the assembly, deployment, and servicing of large science platforms, and the point of departure of these facilities to other Lagrange points within the Earth's Neighborhood. The Crew Transfer Vehicle (CTV) will transfer exploration crews between Earth and the L1 Outpost to conduct these missions of interest in the Earth's Neighbor-

[6] Lo, M., Ross, S. "The Lunar L1 Gateway: Portal to the Stars and Beyond". AIAA Space 2001 Conference, Albuquerque, N.M, August 2001.



Mission: The Lunar L_1 Outpost is a mission staging and crew habitation platform stationed at the Lunar L_1 libration point for assembling and maintaining large astronomical observatories and conducting expeditions to the lunar surface.

Element Mass:
- Launch: 23,000 kg
- Outfitting: 1,000 kg
- Post-outfitting: 24,000 kg

Figure 3.1-3: L_1 Outpost

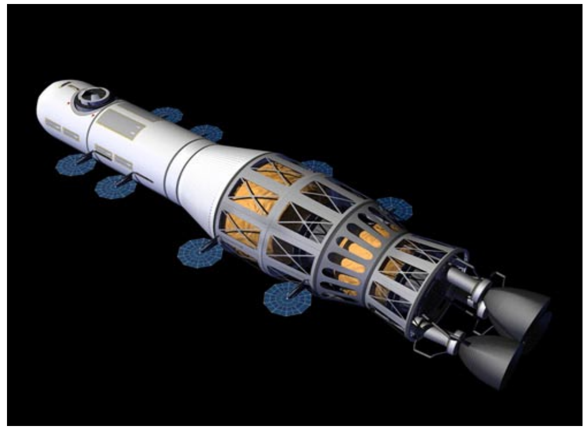
hood architecture. Finally, the Solar Electric Propulsion (SEP) Stage will provide a reusable means for transporting high-value mission cargo assets between Earth and their various destinations through the use of high-efficiency propulsion systems.

L1 Outpost

The L_1 Outpost is a unique crew habitation and mission staging platform for supporting the design reference missions in the Earth's Neighborhood exploration architecture, and for expanding and maintaining human presence beyond Low Earth Orbit. This mission element is envisioned as a single-launch deep space habitat, with the required volumetrics for radiation protection, consumables, spares, and crew accommodations provided. The L_1 Outpost will support space construction and servicing capabilities consistent with the requirements for large science platform assembly and checkout. These requirements are still under development. Docking accommodations are also provided for the CTV and other visiting vehicles. For initial delivery from LEO to Lunar L_1, the use of the Solar Electric Propulsion Stage has been baselined.

To serve as a technology testbed for future human exploration beyond Earth's neighborhood is a driving factor in the L_1 Outpost design. By demonstrating the operability of system technologies prior to use, mission planners can drastically reduce the cost and risk of such missions. Previous studies have identified key thrusts in the areas of advanced habitation, life support, in-space transportation, and power. For example, inflatable structures can provide large habitable volumes and integrate passive radiation protection methods while minimizing mass and packaged volume. Closed-loop life support is an enabling technology for human exploration beyond Low Earth Orbit by radically reducing total consumable mass requirements. A routine EVA capability will be needed for robust exploration of planetary surfaces and human in-space assembly tasks. It is in these areas and others that the focus of the Outpost design has been placed, and wherever possible, such systems have been selected.

For long-duration human space flight, a large habitable volume will be required for maintaining positive crew welfare, and inflatable habitation systems may be a promising solution to this need. As a primary design goal of the Outpost is to demonstrate such advanced technologies for future human exploration, an inflatable section was used to provide the primary habitable volume. However, such a structure presents major design challenges when massive external load-producing systems must be attached. For the Outpost, a number of systems, such as an EVA work platform, assembly and servicing infrastructure, visiting vehicle docking ports, a robotic arm, photovoltaic arrays, and others must be attached to the exterior structure. These needs, coupled with the desire to use inflatable technologies, led to a hybrid structure design for the Outpost as illustrated in Figure 3.1-3. A core pressure shell will provide rigidity for attaching external components and packaging systems during launch, while an inflatable section will provide a large habitable volume for the crew.



Mission: The CTV transports crews of four between ISS and Lunar L_1. Upon completion of a mission, the crew aerocaptures into Earth orbit and docks to ISS. The crew returns to Earth via an independent return vehicle such as the Shuttle.

Element Mass:

– CTV: 25,000 kg
– Injection Stage: 48,000 kg
– Total Stack: 73,000 kg

Figure 3.1-4: Crew Transfer Vehicle

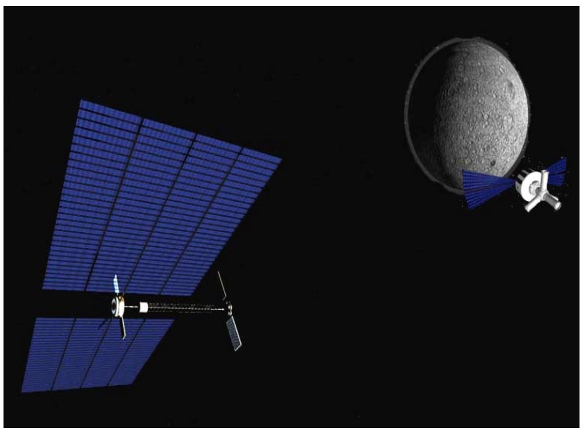
Crew Transfer Vehicle

The Crew Transfer Vehicle (CTV) is the crew transportation element in the Earth's Neighborhood architecture that ferries exploration crew from the International Space Station (ISS) to and from the L_1 Outpost. The CTV is capable of sustaining a crew of four for up to 22 days - nine of which are spent in transit between the ISS and Lunar L_1. The additional days allow for the CTV to perform proximity operations for the translunar injection (TLI) Kickstage, L_1 Outpost, and ISS. It also allows the crew to loiter at L_1 while waiting for a return opportunity to the ISS in the case where the CTV is unable to dock with the Gateway.

The CTV is composed of two distinct parts. The forward CTV contains the pressurized crew cabin with docking hatch, life support system, power system, avionics, crew accommodations (including food and medical supplies), forward reaction control system, forward aeroshell, suit stowage, and radiation protection system. The aft half of the CTV contains crew consumables, the thermal control system, the CTV main propulsion system return stage, contingency parachute landing system, and the aft aeroshell. Following launch, the CTV remains at the ISS when not in use in order that it may be reused on subsequent missions to avoid recurring launch costs, while the aft module is removed following the mission and returned aboard the Shuttle for refurbishment.

The CTV is designed to satisfy two mission scenarios. The first of these is the nominal mission in which the CTV returns to the ISS following a single-pass aerocapture maneuver. Upon its return to Low Earth Orbit, the CTV returns the crew to the Station where they transfer to the Shuttle for return to Earth. Between missions, the CTV receives resources from the ISS and EVA support to replace thermal protection system panels on the forward CTV. The second mission of the CTV is a one-time return of the crew to the surface of Earth in the event of a contingency such as a system failure or crew injury. Once it is determined that a contingency Earth return is necessary, the CTV performs a direct aeroentry maneuver to de-orbit through the upper atmosphere with the aft module attached to aid with vehicle braking and onboard reaction control system (RCS) for vehicle stabilization.

Three launches are required to bring the CTV and Kickstage to Low Earth Orbit. The forward section of the CTV is first launched aboard a Shuttle and is docked at the ISS. A second Shuttle is launched to ferry the exploration crew and CTV aft module to the ISS where the two modules are mated. The crew then transfers into the CTV, undocks and performs a rendezvous and docking maneuver with the TLI Kickstage which was launched near the ISS following successful mating of the CTV. The Kickstage, with its CTV payload, sends the crew to Lunar L_1. Following completion of their mission, the crew transfers back to the CTV for the return trip to Earth.



Mission: High-efficiency solar electric propulsion (SEP) is used in the Earth's Neighborhood architecture to deliver uncrewed elements from low-Earth orbit to a final destination. The SEP Stage subsequently returns to Earth for reuse.

Element Mass:

–	SEP Stage:	35,000 kg
–	Payload:	30,000 kg
–	Total Stack:	65,000 kg

Figure 3.1-5: SEP Stage

Solar Electric Propulsion Stage

The efficient and cost-effective delivery of both cargo and humans to and from various exploration destinations is critical for human exploration missions. Given the total mass involved in many exploration architectures, this area is of prime importance and thus has been the focus of many studies and technology development efforts. Historically, propulsion technologies include high and low thrust propulsion systems involving chemical, nuclear, solar, and aeroassist forms of energy exchange. Electric propulsion is one class of in-space transportation that has benefits for human exploration. Electric propulsion concepts utilize solar or nuclear power to accelerate propellant to higher exit velocities than those from a chemical reaction. Such systems have the advantage of a high specific impulse system that maximizes engine efficiency usually at the expense of thrust.[7] One common requirement for all electric propulsion systems is a substantial supply of power generated either from solar or nuclear energy. For solar power, large photovoltaic arrays can be built and deployed to power the systems for a trip to Earth's Neighborhood locations, while none of the shielding requirements or concerns with launching and operating in Low Earth Orbit that all nuclear options carry

[7] George, Jeffrey A., "Piloted Mars mission planning - NEP technology and power levels," Space nuclear power and propulsion; Proceedings of the 10th Symposium, Albuquerque, NM, Jan. 10-14, 1993. Pt. 1; New York, American Institute of Physics, 1993, p. 493-500.

would be an issue. Therefore, solar electric propulsion (SEP) was baselined for use in the Earth's Neighborhood transportation infrastructure.

In the current architecture, it is envisioned that the Solar Electric Propulsion Stage will be used to transport high-value assets such as the L_1 Outpost, Lunar Landers, the Lunar Habitat, and re-supply items between Earth and the various destinations. These mission elements have the advantage of being uncrewed during flight portions to the point of deployment, therefore transit time is not a major concern as with crewed spacecraft. SEP can utilize high-efficiency, continuous thrust propulsion systems to save on propellant cost at the expense of transit time. A concept for the SEP Stage is provided in Figure 3.1-5.

3.1.3 Mission Requirements

Description	Requirement	Rationale
Total mass to Low Earth Orbit (LEO) for Earth's Neighborhood Exploration Missions	The system shall be capable of delivering a total of 110 metric tons in 6 months to low-Earth orbit.	Current Exploration Office analysis indicates approximately 60-110 metric tons initial mass in LEO for a range of Earth's Neighborhood missions. The total mission mass is dependent on mission type and destination.
Delivery of Earth's Neighborhood exploration mission crew	The system shall be capable of delivering a minimum of 4 exploration mission crew to low-Earth orbit (407 km circular, 51.6 deg.).	Current mission planning requires four exploration crewmembers to perform the missions included in the Earth's Neighborhood architecture.
Earth's Neighborhood Exploration mission rate	The system shall support a minimum of 2 exploration missions per year.	Scientific objectives defined by the NASA Exploration Team require a minimum of two missions per year to accomplish.

Payload upmass to LEO for Earth's Neighborhood Exploration Missions	The system shall be capable of delivering a minimum of 40 metric tons per launch to low-Earth orbit (407 km circular, 51.6 deg.).	Wide ranges of launch package masses for Earth's Neighborhood missions have been studied. Payloads of 40 metric tons represent a good balance between required size of the payload and the number of launches required. Package sizes in the range of current launch capabilities (20 metric tons) show significant disadvantages from both a mass and volume perspective including: 1) Significant mass efficiency losses due to non-optimal packaging (ISS experience indicates a 70% utilization efficiency), 2) Design inefficiencies increase with the number of launches due to increased number of interfaces and additional functional requirements (bulkheads, docking mechanisms, plumbing, etc.), 3) Probability of mission success (launch) is decreased with increasing number of launches, and 4) Significant increase in the level of on-orbit assembly required for vehicle and systems.
Payload downmass to LEO for Earth's Neighborhood Exploration Missions	The system shall be capable of returning a minimum of 18 metric tons from low-Earth orbit (407 km circular, 51.6 deg.).	Crew transportation elements may be returned to Earth upon completion of a mission for refurbishment and reuse. Analysis has shown these elements to weigh up to 18 metric tons.
Payload volume to LEO for Earth's Neighborhood Exploration Missions	The system shall be capable of delivering payloads with minimum volumetric dimensions of 6 m diameter x 18 m length per launch.	Launch package sizes in the range of current launch capabilities show significant disadvantages including: 1) Significant mass efficiency losses due to non-optimal packaging (ISS experience indicates a 70% utilization efficiency), 2) Design inefficiencies increase with the number of launches due to increased interfaces and additional functional requirements (bulkheads, docking mechanisms, plumbing, etc.), 3) Probability of mission success (launch) is decreased with increased number of launches, 4) Significant increase in on-orbit assembly required for vehicle and systems.

Reliability goal for Earth's Neighborhood Exploration Mission Launches	The system shall provide an overall payload delivery reliability of at least 99.7%.	The probability of total mission success is directly related to the launch vehicle reliability. Given the current worldwide launch vehicle reliability history, the probability of launch success for current launch capabilities would be less than 70%. A launch vehicle system reliability approaching that of the Shuttle, in excess of 99%, is required to maintain a total launch success probability of 90% or greater.
Automated Rendezvous and Capture	The system shall provide the capability to perform automated rendezvous and capture with previously delivered payloads in low-Earth orbit.	Mission planning requires that crew transportation elements be launched individually due to launch mass limitations. These elements should be docked without the aid of human piloting.
Launch of cryogenic propellants	The system shall be capable of launching payloads containing significant quantities (50-80%) cryogenic propellants.	The crew transportation elements will contain significant amounts of cryogenic propellants to perform injection maneuvers.

Table 3.1-1: Earth's Neighborhood Transportation Infrastructure Mission Requirements

3.1.4 Key Technology Investments

Technology	Summary Description	Current TRL	Additional Applications
Reusable and Ablative Thermal Protection Systems	High temperature reusable and ablative thermal protection systems for aerocapture and Earth entry		Space Launch Initiative, other HEDS applications
Electric Propulsion Thrusters	50 kWe gridded ion engines, operating on Xenon at >2500 s Isp	2-3	Long-duration spacecraft, human Mars missions, outer planet exploration
Inflatable Structures	Inflatable materials for high deployed volume to mass habitation modules		ISS modules, other HEDS applications
Composite Structures	Use composite structures to reduce the weight of vehicle primary structure and fluid storage tanks	6	Space Launch Initiative
Advanced Aluminum Alloys	High strength-to-weight aluminum alloys (Al-Li) for primary structure	5-6	All spacecraft structures
Radiation Protection	Passive and active radiation shielding strategies and materials		Other HEDS applications
Docking Adapters	Advanced mechanisms and materials for light-weight, reliable docking adapters		Other HEDS applications
Integrated Cryogenic OMS/RCS Systems	High cycle life LH_2/LO_2 and LCH_4/LO_2 main propulsion engines and RCS thrusters	5-6	Upper stages, other HEDS applications

Zero-Boiloff Cryogenic Fluid Storage	Long-lifetime cryocoolers to remove thermal energy for long-term fluid storage	4	Sensor cooling
Photovoltaics	High-efficiency photovoltaic cells (41% AM0) for in-space power generation	5	All spacecraft power applications
Batteries	Lithium-based batteries (>200 Wh/kg, 70% DoD)	2-3	All spacecraft energy storage applications
Power Processing	Light-weight power conversion and switching electronics	3-4	All high-power spacecraft applications
Integrated Energy Storage and Attitude Control	Composite flywheels to provide spacecraft momentum management and energy storage	2-3	Long-duration/large spacecraft applications
Thermal Control	Light-weight flexible radiator materials operating at high temperatures	2-6	All TCS applications
Life Support	Closure (>95%) of air and water loops to reduce consumables and rate of resupply		Other HEDS applications
Micro-Electro-Mechanical Systems (MEMS)	Integration of mechanical elements, sensors, actuators, and electronics on a single chip		Unlimited applications
Autonomous Navigation System	Precision autonomous navigation	5	All spacecraft applications

Table 3.1-2: Earth's Neighborhood Key Technology Investments

3.1.5 References

1. NASA Exploration Team, Annual Report, November 2001.
2. "NASA Decadal Planning Team, Earth's Neighborhood Exploration Architecture Study," NASA-JSC, November 2000.
3. L_1 Gateway Habitat Study Update, JSC, August 2001
4. "Crew Transfer Vehicle Design Report," NASA-JSC, September 2001.
5. "Lunar L_1 Gateway Design Report," NASA-JSC, October 2001.

DRM POC: NASA/JSC Advanced Development Office

3.2 Construct, Deploy, and Service Large Science Platforms

Key mission objectives achievable within the Earth's Neighborhood, as identified by the NASA Exploration Team, include the assembly and maintenance of large-scale advanced scientific platforms in space. Ambitious science facilities will be extremely difficult to deploy, construct, rescue, service, and repair in space without sophisticated capabilities for manipulation and mobility. Such capabilities may be provided through the collaborative partnering of advanced robots, autonomous or remotely operated systems, and/or humans on-site.

Critical to this design reference mission is the emplacement of a construction, servicing, and mission-staging platform in space to support human and robotic explorers. Studies by NEXT have focused on one potential facility located in the vicinity of the Moon, specifically at the Lunar L_1 Lagrange point. This Lagrange point gateway is particularly advantageous as an assembly and servicing node as it enables very low-energy transfers to the Earth-Sun Lagrange points – locations considered ideal for advanced astronomical instruments and solar weather monitors – while remaining within relative proximity of Earth for accessibility by humans and robots.

Several representative advanced science platform concepts are presently under development by the NEXT Earth's Neighborhood Science Team, including the Filled Aperture Infrared (FAIR) Telescope concept. The FAIR telescope is a post NGST, large-aperture far-infrared and submillimeter telescope to meet anticipated high-priority science objectives that may potentially be assembled and serviced by humans and robotic systems at the L_1 Outpost.

3.2.1 Connection to NEXT Themes and Goals

This design reference mission will enable NASA to achieve some of the ambitious science missions currently under study in the Office of Space Science (OSS). Unlike today's space-based telescopes, the facilities envisioned to answer the fundamental questions of astronomy are beyond the pre-assembled payload volume capability of Earth-to-orbit launch vehicles. Therefore, future large-aperture observatories will require in-space assembly, calibration, and as a result, significant support infrastructure to enable these tasks. This mission will also allow science platforms once in operation to be serviced for routine system maintenance or equipment replacement, extending the science-gathering lifetime of the facility. Or, specific science instruments may be upgraded or replaced, enhancing overall science capability. This model for on-orbit servicing and upgrades of science platforms has been applied with spectacular results to the Hubble Space Telescope over the course of four Space Shuttle servicing missions.

The observatories to be supported by this design reference mission are derived from the science objectives identified in the NEXT vision for exploration and the NASA Stategic Plan. A FAIR telescope would determine how planetary system-forming disks evolve. With its keen infrared vision, it would probe deeper into protostellar disks and jets to investigate the physical processes that govern their formation, evolution, and dissipation, as well as those that determine their temperature, density, and compositional structure. With this knowledge, coupled with an understanding of how our own Solar System formed and evolved, it may be possible to extrapolate where life beyond the planet of origin may exist in the Universe and how that life may have originated. These objectives directly address the third grand challenge of exploration as identified by NEXT – the question "Are we alone?" This science pursuit seeks to answer whether other abodes for life may exist in the Universe. Through the construction, deployment, and operation of advanced observatories such as the FAIR Telescope, NASA may begin to solve such questions.

3.2.2 Mission Description

Several destinations were initially considered for the location of the assembly, deployment, and servicing facility. As the collinear Sun-Earth Lagrange points (Sun-Earth L_1 and L_2) are considered advantageous for operating astronomical observatories and solar sentinels, it was first thought that human and robot assembly agents could be sent from Earth to those locations to construct and deploy the facilities on-site. This would alleviate the need for transporting the facilities to a final destination after assembly. However, analysis quickly revealed that long transit times would be required for the human transportation infrastructure to deliver assembly teams to the worksite, meaning that vehicle would be dissimilar to other crew transportation systems in the Earth's Neighborhood architecture due to additional consumables and habitable volume required. This was contrary to the NEXT philosophy of consolidating and building off existing infrastructure when sensible.

Cursory analysis of an assembly and servicing platform in LEO was also performed, and is still on going. This approach was initially discounted, though, because of impacts to the science platform once assembled and high transportation costs for servicing. Assuming assembly in LEO, a completed telescope could be delivered to its final destination by one of two means: (1) a high-thrust, impulsive transfer, or (2) a continuous, low-thrust transfer. In the former, extremely high loads would be imparted to the reflector truss structure, requiring stronger truss members thus adding mass to the overall facility. In the latter, sensitive telescope instruments would undergo multiple passes through the Van Allen radiation belts. Also, servicing the telescope in LEO once operational would require a significant propulsive capability on the facility to enter and then re-escape from Earth's gravity well.

Locating the construction facility at the Lunar L_1 Lagrange point appears to be an optimal comprise between the two former options. As outlined in the 'Earth's Neighborhood Transportation Infrastructure DRM' section, this point has been baselined for lunar exploration because it enables global access to the surface with continuously available launch windows. Therefore, a crew transportation system between LEO and L_1 would already be available to support those missions. In addition, very-low-energy transfers (as low as 14 m/s) have recently been identified between the Lagrange points of the Earth-Moon and Sun-Earth systems through the use of "invariant-manifold" trajectories.[8] These trajectories will allow deployed science platforms to be put into operation and returned for servicing for very little transportation cost. Therefore, infrastructure to support assembly, deployment, and servicing would be placed on the same facility that supports lunar exploration mission staging, the L_1 Outpost. Additional information on the L_1 Outpost can be found in Section 3.1 of this document.

[8] Lo, M., Ross, S. "The Lunar L1 Gateway: Portal to the Stars and Beyond". AIAA Space 2001 Conference, Albuquerque, N.M, August 2001.

Telescope assembly at Lunar L_1 can also solve some of the environmental and operational concerns of assembly in Low Earth Orbit. Contamination of hypersensitive telescope detection instruments and reflector surfaces is a major concern, and it is questionable whether assembly from the Shuttle or ISS could meet these stringent requirements. Adherence of atomic oxygen found in LEO to telescope components may also be an issue. In addition to contaminants, construction in LEO implies a high risk of micrometeoroid and orbital debris impact, an issue that is greatly reduced at the L_1 Outpost. The orbital debris flux is essential zero at Lunar L_1, and the micrometeoroid flux is equivalent to that of interplanetary space. Finally, this destination offers a more attractive thermal environment for telescope assembly. An outgassing and bake-out phase may be desired to eliminate any lingering contaminants from the telescope structure prior to deployment of the reflector surfaces, thus requiring a sustained high-temperature environment. Inversely, telescope instruments must be passively cooled to cryogenic temperatures for operation. As Lunar L_1 is located in a deep space environment, it offers constant sunlight viewing for telescope bake-out while use of the assumed sunshield can achieve the low temperatures required for instrument testing. The temperature environment of Low Earth Orbit involves constant orbital day/night cycling and thermal albedo from Earth, therefore is less likely to satisfy telescope assembly thermal requirements.

3.2.2.1 FAIR-DART Telescope

Many concepts have been identified for large aperture reflectors. These concepts range from segmented, solid-surface reflectors that must be deployed or erected on-orbit to membrane monoliths that must be inflated or stretched on-orbit. The first space-based telescope to use a deployable mirror will be the Next Generation Space Telescope (NGST) that is currently planned for launch in 2010. Although very technically aggressive, even the NGST's 6.5m-diameter deployable reflector is small enough that it can be segmented into a single circumferential ring of panels and folded for launch using a relatively simple arrangement of hinges and latches. So-called "one-ring" segmented reflectors are desirable due to their mechanical simplicity, but they are limited to deployed diameters no more than a factor of 2.5 to 3 larger than the launch vehicle shroud. For large aperture diameters of 5 to 10 times the launch vehicle shroud diameter, it is necessary to consider either multi-ring segmented reflectors or unfurled membrane reflectors. In the future, it is hoped that advances in active-control and wavefront correction technology will make membrane reflectors practical. Application of membrane reflectors is the basis of the Dual Anamorphic Reflector Telescope (DART), a proposed concept for the FAIR Telescope mission.[9]

[9] Lake, Mark S, et al., "Evaluation of Hardware and Procedures for Astronaut Assembly and Repair of Large Precision Reflectors", NASA/TP-2000-210317, August 2000.

Science Overview[10]

FAIR-DART (Figure 3.2-1) is intended to cover the far infrared and sub-millimeter spectral range, 40 - 500 micrometers. A telescope in this wavelength regime is well suited to studying the behavior of interstellar gas and dust over a wide range of redshifts. It is this interstellar matter that feeds super massive black holes in the nuclei of galaxies, causing Active Galactic Nuclei (AGN) to brighten; fuels starbursts in galaxies; establishes and renews stellar populations by the formation of new stars in molecular clouds; and collects and transports the heavy elements that shape stellar evolution and make life possible.

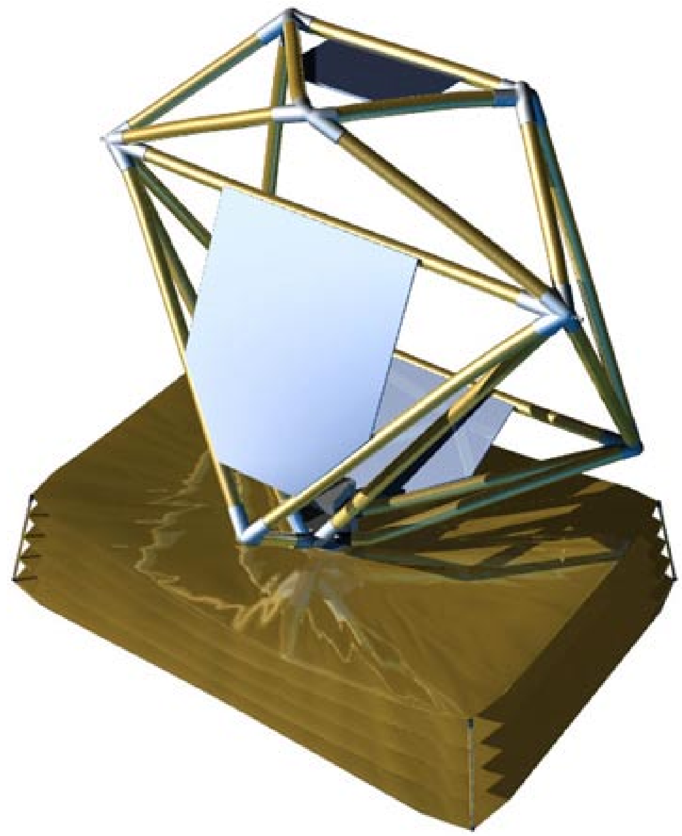


Figure 3.2-1: FAIR-DART Telescope Concept

The far infrared and sub-millimeter spectral ranges are critical for probing the interstellar medium. Regardless of the original emission process, cosmic energy sources glow in the far infrared due to the effectiveness of interstellar dust in absorbing visible and ultraviolet photons and reemitting their energy. For example, the Milky Way and other galaxies show two broad spectral peaks, one produced directly by stars and extremely thoroughly studied in the visible and near infrared and the second comparatively unexplored in the far infrared. Warm, dense interstellar gas cools predominantly through low energy fine structure lines and also emits profusely in rotational transitions of the most abundant molecules; both systems of lines emerge predominantly in the far infrared and sub-millimeter ranges. These lines are key participants in the process of collapse that regulates formation of stars and AGN's. They also provide detailed insights to the temperature, chemical composition, density, and ionization state of the collapsing clouds.

The FAIR-DART telescope will enable a substantial body of high priority science tasks to be carried out. Four high priority topics that help to drive the top-level technical requirements are:

1. Deep Extragalactic Surveys: These will be crucial in understanding the evolution of galaxies. These surveys will also be important in beginning to resolve the far-infrared/sub-millimeter background, discovered by COBE, into individual galaxies.

[10] "JPL Team-X Study Report, FAIR-DART 2002-01". Bob Oberto, Leader, NASA JPL, February 2002.

2. Determining power source for the infrared (IR) bright galaxies: Here the issue is to determine whether the main power source in a given IR bright galaxy is due to a massive burst of star formation (starburst phenomenon) or to intense activity in the nucleus of the galaxy (AGN or active galactic nucleus phenomenon). Fine structure emission lines from the interstellar gas in the galaxy can provide the answer, because the excitation of the lines is different in the two cases. This will attempt to probe these fine structure lines in high redshift galaxies.

3. Studying the warm cores of star and planet forming regions: The purpose here is to spatially resolve the 1000 AU sized cores to get at the structure and physics they contain. Since the dark clouds in which these cores form are so opaque, it is necessary to go to the far IR in order to have visibility into the core.

4. Studying properties of Kuiper Belt objects in our solar system and elsewhere: These "debris disks" are likely to be the remainder of the planetesimals out of which the planets formed.

3.2.2.2 Human and Robot Assembly Servicing Agents

Because of the variety of task complexity, risk, and frequency associated with the FAIR-DART Telescope, a number of human and robot agents of differing capabilities have been identified to aid in assembly and servicing operations. It is assumed that depending on the nature of the task at hand, certain agents can be paired to a team, or squad, optimal for the job. Though the FAIR-DART Telescope has been assumed for study purposes, the capabilities of most agents are generic enough to be applied to any science platform. One potential team may include, but is not limited to:

<u>Human Agents</u>

1. EVA Astronauts
2. Remote Manipulator System (RMS) Operator
3. Dexterous Robot Teleoperator
4. Mission Control Team

<u>Robotic Agents</u>

1. Dexterous Robot (Robonaut, Ranger, etc.)
2. Remote Manipulator System
3. Assembly/Servicing Flight Support Equipment
4. Inspection Robot Free-Flyer (Mini-AERCam)

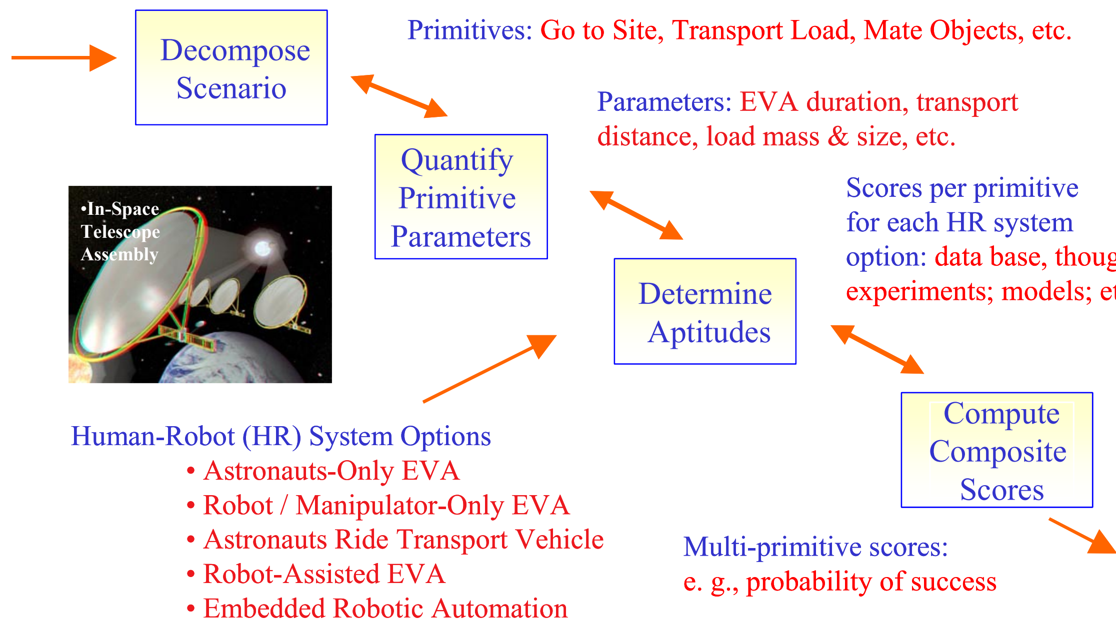


Figure 3.2-2: Human-Robot Performance Case Study Process

3.2.2.3 Human/Robot Assembly Optimization

Construction, deployment, and servicing operations of large science platforms, such as post-NGST astronomical telescopes, are expected to fall within a broad spectrum of task complexity and frequency. Such tasks will likely be accomplished through an optimal combination of human EVA and robotic system capabilities, however this combination is not well-understood, and will depend on the nature of the task or series of tasks. The relative strengths of humans and robots have been well established conceptually through anecdotal evidence. Whereas robotic systems are generally suited to more frequent, high-risk access, less complex tasks, humans excel in non-repetitive, non-linear situations requiring cognitive abilities and innovative thinking.[11,12] This has been witnessed in the rescue of the Hubble Space Telescope and Gamma Ray Observatory, multiple satellite servicing missions (Westar/Palapa, Intelsat, Spartan), and in numerous other examples throughout the history of human spaceflight. As for robots, evidence can be provided through the application of terrestrial robots for assembly line manufacturing and the fact that robotic spacecraft have visited on numerous occasions harsh planetary environments (Venus, Jupiter) currently inaccessible to humans. However, opinions and hunches about the value of humans and robots for in-space assembly and servicing tasks significantly exceed in-depth study and formal assessment. What is needed are standardized metrics to quantify performance and rigorously defined criteria to evaluate relative performance. The NEXT Human-Robot Working Group (HRWG) is currently performing studies to examine optimized assembly of the FAIR telescope. One such method for achieving these objectives has been outlined in Figure 3.2-2 for various human/robot "squad" options.

[11] Chmielewski, A., Cadogan, D. "Robotics/EVA Optimization for Construction of Gossamer Telescopes", September 2000.
[12] Thronson, H., et al. "The Lunar L1 Gateway Concept: Supporting Future Major Space Science Facilities". NASA HQ, March 2002.

3.2.2.4 Assembly Platform Infrastructure

Using the model of the DART concept assembled in space by humans and robots, a preliminary set of infrastructure needs has been developed for the construction platform. As telescope components will likely be launched and delivered separately from the assembly team in the shroud of a launch vehicle, the platform must provide a docking cradle for the telescope components. During assembly, there must be a means to stabilize the telescope during assembly and react loads back to the platform primary structure. A system for transportation of components and assembly agents from the platform to the worksite will be required, possibly in the form of scaffolding or other climbing structure, mobile, extensible cranes (e.g. ISS RMS), free-flyer robots and EVA pods, or a "light-rail" system. Power and data transmission equipment for providing power to the telescope systems and robotic agents, and communicating with the EVA and robotic agents, is required. Depending on the environment for assembly, an unpressurized, partially closed shelter for EVA astronauts may be necessary to limit total radiation dose.

3.2.3 Mission Requirements

Description	Requirement	Rationale
Science platform assembly components injected mass	The system shall inject a 5,000 kg payload to a C_3 of –0.50 km^2/s^2	The DART telescope components may weigh up to 5,000 kg, and must be injected to Lunar L_1, which is the currently baselined point-of-assembly for the telescope.
Earth's Neighborhood Exploration mission rate	The system shall support a minimum of 2 exploration missions per year.	Scientific objectives defined by the NASA Exploration Team require a minimum of two missions per year to accomplish.

Table 3.2-1: Science Platform Assembly and Servicing Mission Requirements

3.2.4 Key Technology Investments

Technology	Summary Description	Current TRL	Additional Applications
EVA Suits	Suit development is required for: 1) minimizing consumables and environment contamination, 2) improving dexterity and mobility, 3) mechanical augmentation, and 4) supplemental instrumentation and information technologies		Lunar/Mars exploration, other zero-g EVA applications, terrestrial users

Robotic Systems	Robotic systems with the improved capabilities over the state-of-the-art in the following areas are required: 1) increased autonomy, 2) improved dexterity and mobility, 3) intelligent interaction with humans, 4) low resource requirements, 5) human-compatible interfaces	3-4	System components applicable to variety of terrestrial and space robotic systems
Airlock	Advanced inflatable airlock to minimize packaging volume and mass, and to reduce atmosphere loss during depressurization		Space Launch Initiative, other HEDS applications

Table 3.2-2: Science Platform Assembly and Servicing Key Technology Investments

3.2.5 References

1. "FAIR Telescope Assembly Study", NEXT Earth's Neighborhood Science Team, December 2001

2. JPL Team-X Study Report, FAIR-DART Opt. 1, 2001-08, August 2001.

3. JPL Team-X Study Report, FAIR-DART 2002-01, August 2001.

4. Lake, Mark S, et al. "Evaluation of Hardware and Procedures for Astronaut Assembly and Repair of Large Precision Reflectors", NASA/TP-2000-210317, August 2000.

5. NASA Exploration Team, Annual Report, November 2001.

6. "NASA Decadal Planning Team, Earth's Neighborhood Exploration Architecture Study," NASA-JSC, November 2000.

7. Fullerton, Richard. "What's Next for EVA". NASA JSC/HQ, April 2002.

8. Rodriguez, G., et al. "NEXT-HRWG Case Study 2001/02: In-Space Assembly of Large Gossamer Telescopes". NASA JPL, September 2001.

9. "Lunar L_1 Gateway Design Report," NASA-JSC, October 2001.

DRM POC: NEXT Earth's Neighborhood Science Team
NEXT Telescope Team
NEXT Human-Robot Working Group
NASA JSC/Advanced Development Office

3.3 Lunar Exploration

Recent scientific discoveries of potential water ice deposits in the lunar polar regions by the Clementine and Lunar Prospector spacecraft have sparked renewed interest in robotic and human exploration of the Moon. A significant return from scientific activities may result from investigations by humans and robots on the lunar surface. These include a clearer understanding of the impact history of comets in near-earth space, better knowledge about the composition of the lunar mantle, past and present solar activity, lunar ice at the poles, and the history of volatiles in the solar system. In particular, the lunar north and south polar regions represent excellent initial footholds for human planetary exploration beyond Earth. This is because of the Moon's relative proximity to the home planet, high scientific return potential, possible access to a variety of natural lunar resources, and potential Mars surface and deep space exploration analogs.

Indeed, exploration of the Moon may be a necessary precursor to other accessible planetary surface missions. Previous lunar experience has been lost due to the time-span (approximately 40-50 years) between Apollo and reasonable assumed lunar and Mars mission dates. NASA must re-establish the knowledge and critical skills required for human exploration beyond earth orbit and in the context of another orbital body. Given the expense and remoteness of Mars missions, it may be imperative that lunar missions be used to mitigate the significant risks associated with a Mars mission. From an operations perspective, to safely execute planetary surface exploration missions, there are a number of critical knowledge and skill elements that must be developed. Because of its close proximity to Earth, the Moon is the ideal proving ground to develop critical tools, knowledge, and skill, and should be used to assess and reduce the risks associated with future missions.

Finally, the exploration and development of the Moon may present a number of commercialization opportunities. Here, commercial potential includes the extraction of oxygen and water for propellant/consumables resource depots on the surface and in space, extraction of metals from the lunar soil, and materials processing.

3.3.1 Connection to NEXT Themes and Goals

Exploration of the lunar surface by humans and robots is consistent with the NEXT philosophy of a "stepping stone" approach to solar system exploration outward from Earth and the "grand challenges" for exploration of life in the universe. Before humans are ready to embark on exploration missions to other accessible planetary surfaces, the Moon may serve as an operational testbed for systems technologies and surface operations while remaining within the realm of the Earth's Neighborhood. The commitment to explore one of Earth's neighboring terrestrial planets will likely require a multi-year mission away from Earth with limited opportunities for early return. The Moon, on the other hand, is relatively near Earth and is an ideal intermediate step between first leaving Low Earth Orbit with humans and traveling to other planetary surfaces. Further, the infrastructure put in place to support other missions within the Earth's Neighborhood may be utilized to enable lunar surface expeditions for minimum additional investment. As previously articulated, the NEXT vision for exploration favors a progressive expansion in capability and investment in infrastructure that supports science and discovery conducted in Earth's neighborhood. The goals of lunar exploration are fully consistent with this philosophy.

In addition, lunar exploration may be a possible means for investigations related to the NEXT "grand challenges" of space exploration. The first grand challenge, "How did we get here", examines the history and evolution of the solar system, and the Moon is an excellent site for understanding that question. Lacking a sensible atmosphere and the geologic processes that recycle planetary surfaces, the Moon is a 4+ billion year-old history book of solar activity, solar wind composition, and planetary impacts. This data may possibly be extracted from samples of the lunar surface. Exploration of the Moon may also address the second grand challenge articulated by the NASA Exploration Team, "Where are we going". This question, in part, attempts to understand the adaptability of humans to deep space and other planetary surfaces. As was previously stated, the Moon may serve as such a testbed prior to committing to the long-duration exploration of nearby accessible planetary surfaces.

3.3.2 Mission Description

When human beings once again walk on the surface of the Moon, their missions will be designed around objectives of science, exploration, and technology validation. Over the past fifteen years a number of advisory committees and working groups have issued reports and recommendations for future lunar missions. These recommendations can be set within the context of five themes, three of which are primarily scientific in nature.[13]

> "**The Moon as a Planet** is a theme that builds on the scientific legacy of the Apollo program. The discipline of Planetary Science was born out the Apollo Lunar Science Conferences. Investigators realized that the Moon has undergone a geologic and geochemical evolution since its formation. A strong similarity existed between the development of the Moon and that of the Earth. Scientists realized that studies of the Moon were key to understanding the nature of the other terrestrial planets as well as many of the minor planets in the solar system. Astronauts on future missions will conduct experiments, make observations, and collect samples to address remaining questions about the structure, the composition, and the history of the Moon as a planet.
>
> **The History of the Earth-Moon System** acknowledges the fact that the surface of the Moon harbors a record of exogenous processes that have affected the Earth as well as the Moon. That record certainly includes a history of solar activity and solar wind composition. Now that we understand the importance of impacts on the evolution of life on the Earth, we are eager to examine the impact record on the Moon for evidence of periodicity in impact flux or signatures of singular groupings of impacts. Signatures may exist for gravitational interactions early in lunar history. For example, we now know that the existence of a large satellite has moderated variations in the obliquity of the Earth, enhancing the long-term favorable environment for life. This scientific theme was not fully appreciated at the time of the Apollo missions, and the technology for carrying out some of the relevant investigations did not exist.
>
> **The Moon as a Laboratory** recognizes the potential of the lunar surface environment as a setting for unique experiments. The absence of a sensible atmosphere allows observation of the Universe across the electromagnetic spectrum. Charged particles in the form of cosmic rays or solar particle events can be observed directly. The far side of the Moon is the only location in the solar system permanently shielded from the manmade radio interference generated on Earth. Conversely, the Earth hangs in the sky permanently on the near side, allowing continuous synoptic observation from the Moon. The seismicity of the lunar surface is many orders of magnitude lower than that of the Earth, providing an extremely stable platform for certain classes of physics investigations.

[13] "Themes for Future Human Missions to the Moon", Mendell, Wendell. NASA Johnson Space Center, May 2001.

> At depths greater than a meter, the thermal environment is extremely stable. The remoteness and isolation of the Moon make it a good location for carrying out hazardous investigations, particularly those of a biological nature. Once access to the lunar surface is reliable, many ideas for experiments will emerge.
>
> **Learning to Live in the Solar System** refers to activities directed toward collection of information and testing of technology and processes for future human habitation off the Earth. A major subtheme is In-situ Resource Utilization (ISRU), referring to utilization of local materials for various habitation needs such as consumables, construction materials, or radiation shielding. A number of schemes have been discussed in the literature, including extraction of oxygen from regolith or the mining of He-3 or the manufacture of solar cells. No processes associated with these ideas have been actually tested in the lunar environment on lunar materials. Some proposed processes require concentrations of certain types of minerals (ores). In addition to ISRU investigations, technologies for life support, surface exploration, mobility, power generation, and other basic needs require validation in the lunar environment. Finally, the human beings themselves must learn to adjust to long exposures to extraterrestrial conditions if habitation is a goal.
>
> **Preparing for the Exploration of Mars** addresses the immediate requirement to create and validate technologies (in the broadest sense of the word) critical to the successful human exploration of the planet Mars. The 3-year NASA Mars Reference Design Mission requires performance by the crews and by the technology that far exceeds the current NASA experience base. Life support and other critical technologies must perform under challenging conditions for the duration. Data does not exist for human health and crew performance either for time duration of the mission or for conditions of low gravity. Mission operations must be designed around different philosophies than current models, given the long time delay. Missions to the Moon offer unique opportunities to explore and to test the parameter space, beginning with short, Apollo-like missions and evolving to long duration, martian exploration missions. The lunar mission set will be structured to meet this programmatic need."

To achieve the objectives outlined within the five themes above, an exploration strategy of visiting many scientifically compelling surface sites for various durations has been devised. It is envisioned that several expeditionary "Apollo-style" missions will be conducted to feature-rich high-latitude sites of the lunar near side and all latitudes of the lunar far side for durations on the order of three days. These sites were not visited during the Apollo program due to architecture limitations and program constraints, yet they offer the potential for high-value science return. Coupled with these expeditionary missions are longer-duration surface stays to the lunar polar regions. These areas have been identified as possible analogs for eventual Mars surface exploration. Such analog objectives may include, but are not limited to:

- Testing of Mars surface equipment in lunar polar environment
- Autonomous operations may be required when Earth out of line-of-site
- Lunar ice utilization technologies may be similar to those relating to Martian permafrost

3.3.2.1 Mission Architecture

The lunar exploration design reference mission takes advantage of transportation infrastructure put in place to support other missions within the Earth's Neighborhood, namely the crew transportation system between Earth and Lunar L_1, a cargo transportation system for high-value assets between Earth and various destinations of interest, and a mission-staging platform located at Lunar L_1 (the L_1 Outpost). For the lunar surface mission scenario currently under consideration, a lunar lander will be delivered to Lunar L_1 by a solar exploration stage and docked to the L_1 Outpost prior to mission commencement. The Lunar Lander is the crew transportation element in the Earth's Neighborhood architecture that ferries exploration crews from Lunar L_1 to the lunar surface and back again. As was previously described, exploration of the entire lunar surface is desired to achieve the scientific and operational objectives of this design reference mission. Using Lunar L_1 as the staging point for such missions enables global surface access for no additional transportation cost, a consideration that weighed heavily in its ultimate selection. Though a lunar orbit rendezvous approach as used in the Apollo program requires less total ΔV than Lagrange point rendezvous, launch phasing from the lunar surface constraints are a significant concern. For rendezvous in lunar orbit, the ascent window from the surface opens when the rendezvous orbital plane rotates over the launch site. In the case of high-latitude sites on the Moon that are of particular scientific interest, launch opportunities may be separated by as much as fourteen days. However, as the Lagrange point maintains a fixed position relative to the lunar surface, launch opportunities to and from the L_1 Outpost are continuously available.

Upon arrival of the exploration crew at the Outpost, the crew will prepare for lunar surface departure. The Lunar Lander is designed to fulfill two types of missions. The first of these missions is the expeditionary-type mission where the lander is capable of sustaining a crew of four for three days at any location on the lunar surface. In this mode, the crew uses the lander as its primary base and habitation basecamp for short duration missions. The second mission for the Lunar Lander is to ferry the crew to and from the Lunar Habitat pre-deployed at either the lunar North or South Pole. In this mission the crew will live in a 30-day Habitat for extended lunar missions while the Lunar Lander awaits crewed ascent in survival power mode.

3.3.2.2 Architecture Elements

Lunar Lander

The Lunar Lander (Figure 3.3-1) is capable of supporting a crew of four for a total of nine days—three of which are spent on the lunar surface. The lander is comprised of two stages—an ascent and a descent stage. The descent stage is composed of landing gear, main propulsion system descent tanks, descent reaction control system (RCS), and support structure while the ascent stage hosts the crew module, avionics, ECLSS, ascent propulsion tanks, ascent RCS, and main propulsion system. In order to minimize the payload mass to the L_1 Outpost, the descent stage is left behind on the lunar surface. In addition to the crew, the ascent stage is capable of delivering 50 kg of Lunar samples to the Gateway for transfer back to Earth for

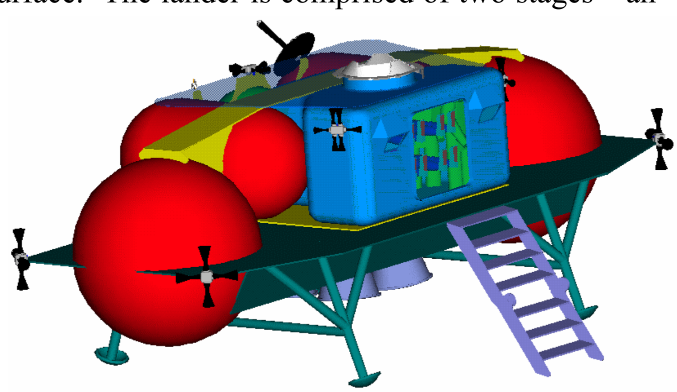


Figure 3.3-1: Lunar Lander

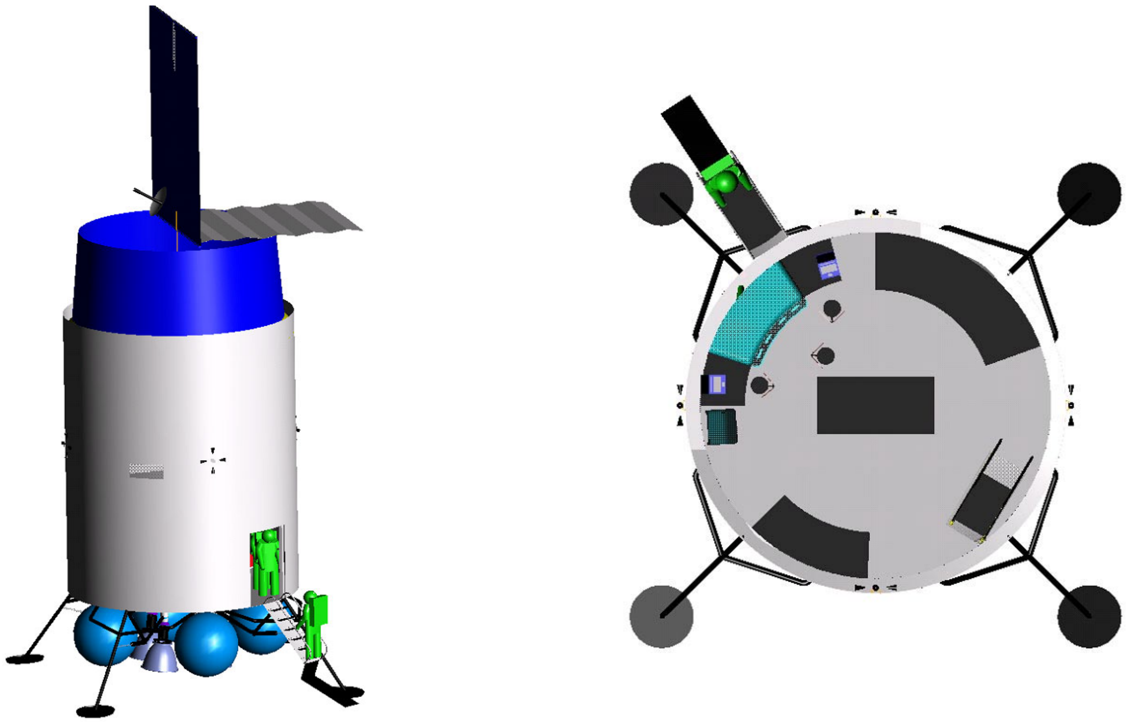


Figure 3.3-2: 30-Day Lunar Habitat

scientific analysis.

Housed on the descent stage is an unpressurized rover capable of transferring the crew to and from the Lunar Habitat and which is also used as a mobility aid for crew traverses of the lunar surface during extra-vehicular activities (EVAs). In addition to the rover, the descent stage also houses a pallet containing science payloads for use during expeditionary surface missions. Alternately, this payload pallet could be used to resupply the Lunar Habitat.

Lunar Habitat

The Lunar Habitat is a surface habitation/science laboratory that can support a crew of four at the lunar North or South Pole for 30 days. The spacecraft is cylindrically shaped, lands vertically, and is supported by a four-legged main landing gear. The lander is propelled by four lox-methane pressure-fed engines, and is powered by deployable solar arrays that generate electrical power for all onboard electrical systems. Terrains at the lunar polar regions may provide nearly continuous sunlight throughout the lunar day; therefore photovoltaic power generation is possible. The configuration of the Lunar Habitat is a vertical cylinder that is divided into three decks. The outer surface of the vehicle serves as the integrated payload launch shroud with a coned portion at the top (which protects the solar arrays and radiators during launch) which is jettisoned during launch to take full advantage of the payload mass and packaging capabilities of the launch vehicle. The Lunar Habitat is illustrated in Figure 3.3-2.

The lower deck houses two two-person airlocks for EVA crew egress/ingress and which can also be used as a radiation shelter to protect the crew during solar events. It also provides an unpressurized porch where the crew may dust off prior to vehicle ingress. Deployable stairs provide access to and from the lunar surface. The second deck houses mechanical and avionics systems as well as the science laboratory facilities for processing of lunar samples, while the third deck houses the crew quarters, galley, and wardroom.

3.3.2.3 Lunar Surface Timeline

This section aims to provide a brief overview of the possible surface timeline for the lunar exploration DRM and factors that have been taken into account when studying such a timeline. Time allocations were designed in accordance with previously developed manned space flight crew scheduling constraint guidelines, and are similar to the generic crew workweek defined for ISS operations. They also reflect Phase 1/Mir flight experience and therefore are applicable to in-transit and on-orbit operations. The timeline begins from initial touch down of the ascent vehicle on the lunar surface and finishes at ascent engine ignition.

<u>Available Crew EVA/IVA Time</u>

The time available for each mission in terms of intra-vehicular-activity (IVA) and extra-vehicular activities (EVA) are shown below. It has been assumed in this study that the time requirements for arrival day checks and ascent vehicle preparation are the same for both the 3-day and 30-day mission scenarios.

ACTIVITY	3 DAY	30 DAY
Surface Stay (hrs)	72	720
Number of Crew	4	4
Surface Stay (person-hours)	288	2880
On-Duty (hr/day)	8.5	8.5
Total On-Duty (hrs)	102	1020
Arrival Day Checks (hrs)	12*	12*
Ascent Vehicle Prep. (hrs)	12*	12*
Departure Day Checks (hrs)	12*	12*
Available Time for Surface Ops (hrs)	66	984
IVA support & Service Needed per EVA (hr/hr)	1	1
Number of Crew/EVA	2	2
EVA Duration	6.5	6.5
Maximum Number of EVAs	3	23
Maximum Possible EVA (hrs)	39	299
IVA Support EVA (hrs)	39	299
Available Time Margin (hrs)	14	386

Table 3.3-1: Crew IVA and EVA Time Available for Lunar Surface Missions

Daily Schedule

The crew daily activities shall be divided between IVA and EVA in proportions dictated by the mission duration and objectives. The overall daily schedule shall be of a 24-hour cycle divided between on-duty activities, post-sleep, pre-sleep, sleep, exercise and meals. For a 30-day mission, the standard crew workday will remain constant throughout the 28-day lunar cycle thus requiring systems and protocols to maintain the crew circadian rhythm over the 28-day lunar cycle. These activities will be scheduled for all four crewmembers each day in the durations shown below.

ACTIVITY	DURATION (HRS.)	DESCRIPTION
On Duty	6.5	IVA and EVA vehicle maintenance, payload deployment, maintenance and operations, construction operations
Post Sleep	1	The 1-hour period following each sleep period. This allows time for personal hygiene for each crewmember and 'wake-up' reconfiguration of Habitat systems.
Pre-Sleep	2	The 2-hour period before each sleep period. This allows time for personal hygiene for each crewmember and 'sleep' reconfiguration of Habitat systems.
Sleep	8	6 hrs. Minimum if shifting sleep cycles
Exercise	2	Required per day unless EVA or strenuous payload activity of equivalent exercise rigor and duration
Meals	3	Three 1-hour periods for the morning, mid-day, and evening meals for each crewmember. This includes meal preparation and cleanup of the dining area.
Uplink Review	0.5	The 30-minute period to review the 'morning mail' from the MCC, immediately following receipt of the daily Earth communication uplink.
Report Prep. & Planning	1	The 1-hour period following the evening meal for each crewmember to prepare a daily report and for the crew to do the detailed planning and scheduling for the following day. (Should be completed prior to daily Earth LOS to allow downlink to MCC).

Table 3.3-2: Generic Daily Activity Time Allocation

These durations shall be generic for both the 3-day and 30-day missions except for the time allocated for exercise. It is a commonly held opinion that the exercise requirement can be ignored for a 3-day mission where it is felt that the EVAs will provide sufficient amounts of exertion.

Each of these activity time blocks shall occur at the same time every day and it is suggested that when possible, they will be synchronized with the MCC shift schedule. A typical one-shift workday cycle is shown in Figure 3.3-3 below.

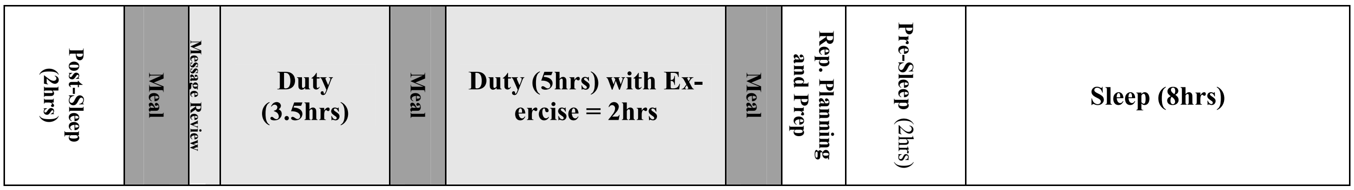


Figure 3.3-3: Typical Workday Schedule

Figure 3.3-4 below shows a typical a workday cycle for a 3-day, or a 30-day, mission split into two teams, the top showing the timeline for an EVA team while the lower shows the timeline for the EVA support team.

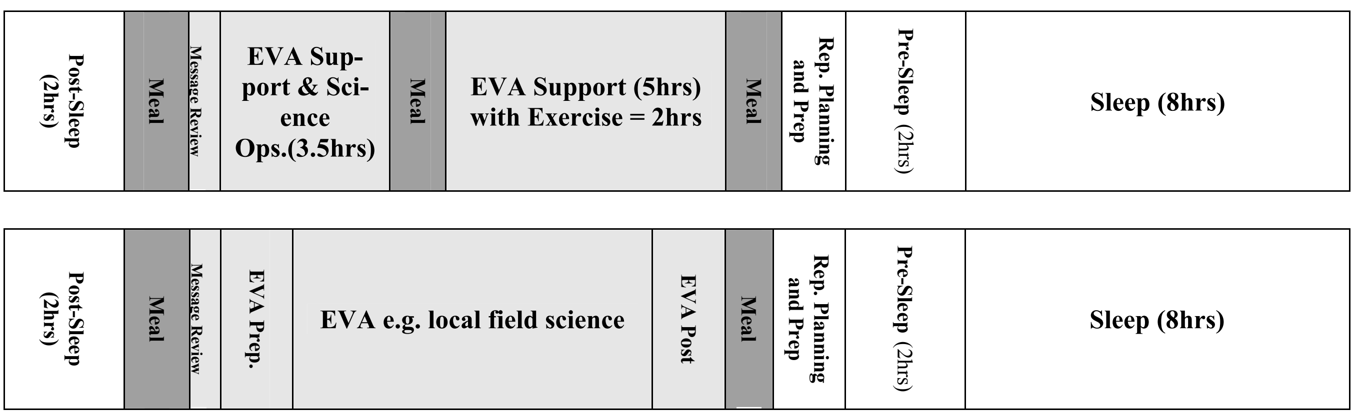


Figure 3.3-4: 3-Day or 30-Day Workday Cycle

Daily EVA Activities

EVAs are envisioned as the focal point for all on-duty activities. Each will require two crewmembers to reflect the "buddy system" for assistance in case of trouble. As the mission concept calls for four crewmembers on the surface, these four individuals would form two EVA teams of two persons per team. While one pair is outside, the second pair would remain inside, with one individual performing EVA support, and the other monitoring vehicle health and status.

With emphasis being placed on the 30-day mission, sufficient time off is required to prevent exhaustion and burnout. Therefore the two pairs of EVA teams will perform surface activities on alternating days, or as the schedule demands. The maximum nominal EVA duration for productive external work is 6.5 hours with safety-dedicated reserves available to last at least 8 hours (from egress through ingress).

EVA operations will not be conducted beyond 5 km walking distance from the habitat without a cache of suit recharge consumables. The rationale for this is that in the event of a rover breakdown, the EVA crew can walk safely back to the surface lander or habitat. When EVA work being conducted extends beyond 5 km from the lander/habitat, the farthest point on the traverse will be the initial destination. For standard EVA operations, each team will be monitored by at least one EVA support crewmember inside the vehicle. This could include monitoring the activities of the EVA team or operating tele-robotic equipment in direct support of the EVA team.

A possible EVA scenario for the 30-day mission that has previously been studied is as follows[14]. During 28 days of surface stay, 23 EVAs of 6.5-hour duration are supported, for a total EVA duration of 149.5 hours. The tasks are distributed:

- Crew transfer EVAs between vehicles at both ends of the surface mission. This will involve all crew (4) simultaneously performing the transfer between the vehicles
- 5 local area (acclimation, area science, rover assembly) EVAs
- 5 rover-assisted geological surface surveys EVAs
- 11 rover-assisted core drills at three different sites (Core A is assumed to take only 3 EVAs, the others require 4 each). Core drilling is assumed to normally require a day (EVA) for setup, two days for drilling, and a day for teardown/relocation

This timeline is considered optimistic and may be decreased by at least one EVA in the future to accommodate vehicle assembly, maintenance or other support functions.

3.3.2.4 Lunar Surface Science

<u>3-Day Surface Missions</u>

While considering the five lunar science themes, a generic set of science instruments and tools was created for potential three-day surface missions. Since the missions can be conducted at any location on the lunar surface, scientists will have the ability to compare measurements from a similar compliment of instruments at many locations on the moon as a method of determining the variances of the geologic structures of the moon. Though most of the science package will be duplicated for subsequent three-day missions, a portion of the payload has been set aside for location specific science. Of the total science payload capacity of the Lunar Lander, a portion

[14] Operations Concept Definition for Human Exploration of Mars, Human Operations Team, Second Ed. DV-00-014, NASA, Lyndon B. Johnson Space Center, May 17, 2000

will be devoted to science at that particular location. The science package for a three-day mission is shown below to show the generic science instruments as well as the mission specific payload chosen for such a mission.

The science package includes an Apollo-style rover to give the crew mobility on the lunar surface to deploy science experiments to be left on the surface, collect measurements at a distance of up to 10 km from the lander, and collect samples using the geologic tools they will be taking. The geology tools, sample containers, and Lander Geoscience Laboratory are to be used to gather, store, down select, and prepare lunar samples for return to Earth. Most of the science instruments will require minimal set up time by the crew and will then be controlled from the earth for use long after the crew departs. These packages, such as the Lunar Surface Experiment Pack, are designed to gather data about the environment on the surface of the moon, including the atmosphere, solar wind, and seismic activities. Other science instruments, such as the Traverse Geophysical Package and the Lunar Portable Magnetometer, are to be carried on the rover to take measurements during the traverses. The location specific instrument for this first mission is a small, steerable telescope that can be controlled from the earth. The purpose of this instrument is to test the feasibility of placing larger telescopes on the moon.

All of these scientific instruments and tools are to be used and set up within the time constraints set forth in the Lunar Surface Timeline section above.

30-Day Surface Missions

For the initial 30-day lunar surface mission, the crew time on the surface will be increased tenfold from the 3-day mission, but the science payload will only be approximately doubled. The time on the surface will be an intense time for the crew and more time must be allowed for the crew to rest to avoid mistakes. However, the science activities must be carefully considered in order to maximize the crew time on the surface.

During this mission, the science activities will be more time-consuming, with more crew interaction than on the 3-day missions. The primary science package that will accomplish this, as well as provide good science data, is deep core drilling. Also, the return payload capability will remain at 50 kg of sample, so more time should be dedicated to analyzing the samples at the habitat to increase the amount of data from the samples, as well as to select the most interesting specimens for return to Earth. Continued work with the science community will enable the creation of specific science packages for this mission using the suggested general concepts.

On subsequent 30-day missions, the surface habitat life support consumables must be replenished. This gives an excellent opportunity for the addition of large amounts of scientific payload on the resupply vehicle. This will give scientists the opportunity to take large items to the lunar surface and conduct science that was not possible before. The main science themes that can be targeted during these missions may be in-situ resource utilization and the testing of Mars mission hardware and operational practices. Devices that can extract oxygen from the lunar regolith will enable future lunar residents to stay for extended durations, and machines that can form building materials from the lunar soil will be key in the construction of larger and larger habitats on the

Task/Activity	Time (hrs)	Personnel	Equipment Mass (kg)	Power (W)	Volume (m3)	Reference
Unpressurized Rover	0.45	2	200	self	6.5	Apollo Exp. Ops.
Generic Seismic Package	NA	1	45	self	0.05	Wendell Mendell
Lunar Portable Magnetometer	NA	2	4.6	self	0.01	Apollo Exp. Ops.
Soil Mechanics	NA	NA	2.3	NA	NA	Apollo Exp. Ops.
Traverse Geophysical Package	Several	2	39	38	0.08	FLO Ch. 4
	QTY	Mass (kg)	Total Mass (kg)	Power (W)	Volume (m3)	Total Vol. (m3)
Electromagnetic Sounder	1	10	10	10	0.02	0.02
Electrical Properties Exp.	1	16	16	10	0.04	0.04
Profiling Magnetometer	1	5	5	10	NA	NA
Lunar Geology Tools	NA	1	28.7	self	0.18	Lunar Tools
	QTY	Mass (kg)	Total Mass (kg)	Power (W)	Volume (m3)	Total Vol. (m3)
Contact Soil Sampling Device	2	0.5	1.0	-	0.00	0.00
Contingency Soil Sampler	2	1.2	2.4	-	0.03	0.06
2cm Core Tubes	2	0.3	0.7	-	0.01	0.03
4cm Drive Tubes	2	0.5	1.0	-	0.06	0.13
Rake	2	1.5	3.0	-	0.02	0.03
3m Drill	1	13.4	13.4	430, self	0.02	0.02
Extension Handle	2	0.8	1.6	-	0.03	0.06
Hammer	2	1.3	2.6	-	0.00	0.00
Small Scoop	2	0.2	0.3	-	0.01	0.02
Large Adjustable-Angle Scoop	2	0.6	1.2	-	0.00	0.00
32-Inch Tongs	2	0.2	0.5	-	0.00	0.00
Gnomon	2	0.3	0.5	-	0.00	0.00
Sample Scale	2	0.2	0.5	-	0.00	0.00
Lunar Sample Containers	NA	1	35.5	-	0.06	Lunar Tools
	QTY	Mass (kg)	Total Mass (kg)	Power (W)	Volume (m3)	Total Vol. (m3)
Apollo Lunar Sample Return Container (ALSRC)	4	6.7	26.8	-	0.03	0.12
Sample Collection Bag	2	0.8	1.5	-	0.01	0.03
Extra Sample Collection Bag	2	0.6	1.1	-	0.01	0.03
Core Sample Vacuum Container	4	0.5	2.0	-	0.00	0.00
48 Cup-shaped Sample Bags	1	0.4	0.4	-	0.00	0.00
20 Rectangular Sample Bags	2	0.4	0.9	-	0.00	0.00
Gas Analysis Sample Container	4	0.2	0.7	-	0.00	0.00
Special Environment Sample Container	2	0.4	0.7	-	0.00	0.00
Organic Sample Monitor Bag	1	0.8	0.8	-	0.00	0.00
Protective Padded Sample Bag	1	0.2	0.2	-	0.00	0.00
Magnetic Shield Sample Container	1	0.4	0.4	-	0.00	0.00
Laser Ranging Retroreflector	0.10	1	27.0	0	0.14	Apollo Exp. Ops.
Steerable Automatic lunar Ultraviolet Telescope	AUTO	0	200	80	TBD	CLMSP
Cosmic Ray Detector	NA	1	0.2	0	0.01	Apollo Exp. Ops.
Lunar Neutron Probe Experiment	NA	1	2.3	-	0.002	Apollo Exp. Ops.
Lunar Surface Experiemnts Pack. (ALSEP)	2.30	1	94.1	NA	3.68	Apollo Exp. Ops.
	Time (hrs)	Personnel	Equipment Mass (kg)	Power (W)	Volume (m3)	Reference
Central Station	0.50	1	25.0	NA	3.48	Apollo Exp. Ops.
Passive Seismic Experiment	0.17	1	11.5	NA	0.05	Apollo Exp. Ops.
Suprathermal Ion Detector	0.17	1	8.8	NA	0.02	Apollo Exp. Ops.
Solar Wind Spectrometer	0.07	1	5.3	NA	0.01	Apollo Exp. Ops.
Lunar Surface Magnetometer	0.25	1	8.6	NA	0.04	Apollo Exp. Ops.
Cold Cathode Gauge	0.03	1	5.7	NA	0.01	Apollo Exp. Ops.
Charged Particle Lunar Env. Exp.	0.08	1	2.5	NA	0.01	Apollo Exp. Ops.
Lunar Dust Detector	0.00	0	0.3	NA	-	Apollo Exp. Ops.
Heat Flow Experiment	1.00	1	9.9	NA	0.02	Apollo Exp. Ops.
Lunar Ejecta and Meteorites	0.03	1	7.4	NA	0.02	Apollo Exp. Ops.
Lunar Atmospheric Composition Exp.	NA	1	9.1	NA	0.02	Apollo Exp. Ops.
Kodak DCS760 Digital Camera and Lenses (2 Cameras)	NA	1	7	self	0.003	kodak.com
Large Tool Carrier	NA	1	5.9	-	0.074	Lunar Tools
U.S. Flag	0.1	1	1.2	-	NA	Apollo Exp. Ops.
Lander Geoscience Laboratory	TBD	TBD	46	144	0.083	CLMSP
	QTY	Mass (kg)	Total Mass (kg)	Power (W)	Volume (m3)	Total Vol. (m3)
Binocular Microscope	1	5	5	20	0.01	0.01
Mossbauer Spectrometer	1	2	2	4	0.00	0.003
Paleomagnetics	1	10	10	20	0.02	0.02
Sample Prep. and Preservation Equipment	1	20	20	100	0.05	0.05
	Time (hrs)	Personnel	Equipment Mass (kg)	Power (W)	Volume (m3)	
Totals			732	262	11	

Figure 3.3-5: Science Package for Typical 3-Day Mission

lunar surface. The moon will provide an excellent test bed for Mars mission technology, as it is relatively close in the event of the need for an emergency return. Mars mission planners can experiment with crew formulations for psychological and group interaction information, test regenerative and self-sustaining life support systems, and develop protocols for mission command and communications by adding 20 minute delays to simulate communications to Mars. Planning with the science community and human exploration missions to Mars will ensure the proper usage of the resources available on these subsequent missions to the surface habitat.

3.3.3 Mission Requirements

Description	Requirement	Rationale
Payload upmass to LEO for Lunar Exploration Missions	The system shall be capable of delivering a minimum of 40 metric tons per launch to low-Earth orbit (407 km circular, 51.6 deg.).	Wide ranges of launch package masses for Near-Earth missions have been studied. Payloads of 40 metric tons represent a good balance between required size of the payload and the number of launches required. Package sizes in the range of current launch capabilities (20 metric tons) show significant disadvantages from both a mass and volume perspective including: 1) Significant mass efficiency losses due to non-optimal packaging (ISS experience indicates a 70% utilization efficiency), 2) Design inefficiencies increase with the number of launches due to increased number of interfaces and additional functional requirements (bulkheads, docking mechanisms, plumbing, etc.), 3) Probability of mission success (launch) is decreased with increasing number of launches, and 4) Significant increase in the level of on-orbit assembly required for vehicle and systems.
Lunar Exploration mission rate	The system shall support a minimum of 2 exploration missions per year.	Scientific objectives defined by the NASA Exploration Team require a minimum of two missions per year to accomplish.

Payload volume to LEO for Lunar Exploration Missions	The system shall be capable of delivering payloads with minimum volumetric dimensions of 6 m diameter x 18 m length per launch.	Launch package sizes in the range of current launch capabilities show significant disadvantages including: 1) Significant mass efficiency losses due to non-optimal packaging (ISS experience indicates a 70% utilization efficiency), 2) Design inefficiencies increase with the number of launches due to increased interfaces and additional functional requirements (bulkheads, docking mechanisms, plumbing, etc.), 3) Probability of mission success (launch) is decreased with increased number of launches, 4) Significant increase in on-orbit assembly required for vehicle and systems.
Reliability goal for Lunar Exploration Mission Launches	The system shall provide an overall payload delivery reliability of at least 99.7%.	The probability of total mission success is directly related to the launch vehicle reliability. Given the current worldwide launch vehicle reliability history, the probability of launch success for current launch capabilities would be less than 70%. A launch vehicle system reliability approaching that of the Shuttle, in excess of 99%, is required to maintain a total launch success probability of 90% or greater.
Launch of cryogenic propellants	The system shall be capable of launching payloads containing significant quantities (50-80%) cryogenic propellants.	The crew transportation elements will contain significant amounts of cryogenic propellants to perform injection maneuvers.
Automated Rendezvous and Capture	The system shall provide the capability to perform automated rendezvous and capture with previously delivered payloads in low-Earth orbit.	Mission planning requires that crew transportation elements be launched individually due to launch mass limitations. These elements should be docked without the aid of human piloting.

Table 3.3-3: Lunar Exploration Mission Requirements

3.3.4 Key Technology Investments

Technology	Summary Description	Current TRL	Additional Applications
Composite Structures	Use composite structures to reduce the weight of vehicle primary structure and fluid storage tanks	3	Space Launch Initiative
Advanced Aluminum Alloys	High strength-to-weight aluminum alloys (Al-Li) for primary structure	5-6	All spacecraft structures

Radiation Protection	Passive and active radiation shielding strategies and materials		Other HEDS applications
Docking Adapters	Advanced mechanisms and materials for light-weight, reliable docking adapters		Other HEDS applications
Integrated Cryogenic OMS/RCS Systems	High cycle life LH_2/LO_2 and LCH_4/LO_2 main propulsion engines and RCS thrusters	5-6	Upper stages, other HEDS applications
Zero-Boiloff Cryogenic Fluid Storage	Long-lifetime cryocoolers to remove thermal energy for long-term fluid storage	4	Sensor cooling
Photovoltaics	High-efficiency photovoltaic cells (41% AM0) for in-space power generation	5	All spacecraft power applications
Batteries	Lithium-based batteries (>200 Wh/kg, 70% DoD)	2-3	All spacecraft energy storage applications
Power Processing	Light-weight power conversion and switching electronics	3-4	All high-power spacecraft applications
Life Support	Closure (>95%) of air and water loops to reduce consumables and rate of resupply		Other HEDS applications
Thermal Control	Light-weight flexible radiator materials operating at high temperatures	2-6	All TCS applications
Micro-Electro-Mechanical Systems (MEMS)	Integration of mechanical elements, sensors, actuators, and electronics on a single chip		Unlimited applications
EVA Suits	Suit development is required for: 1) minimizing consumables and environment contamination, 2) improving dexterity and mobility, 3) mechanical augmentation, and 4) supplemental instrumentation and information technologies		Mars exploration, other zero-g EVA applications, terrestrial users
Science Instruments	Low-mass/power/volume science instruments with simple human and robot compatible interfaces		Other spacecraft science applications
Surface Rovers	High reliability unpressurized rovers for repeated-use, longer-distance surface exploration traverses		Mars exploration, robotic rover systems

Table 3.3-4: Lunar Exploration Key Technology Investments

3.3.5 References

1. NASA Exploration Team, Annual Report, November 2001.
2. "NASA Decadal Planning Team, Earth's Neighborhood Exploration Architecture Study," NASA-JSC, November 2000.
3. "1999 Human Exploration Architecture Study, Representative Human Missions to the Moon", December 1999
4. "Lunar Lander Design Report," NASA JSC, November 2000.
5. "Lunar Habitat Design Report," NASA JSC, December 2000.
6. Fullerton, Richard. "What's Next for EVA". NASA JSC/HQ, April 2002.

7. Kennedy, G., Hair, J. "Lunar L1 Architecture: Surface Timeline, Science Complement, and Resupply Options". NASA JSC, May 2001.
8. Lunar Surface Operations Study, NASA Contract Number NAS 9-17878, EEI Report #87-172, December 1, 1987

DRM POC: NASA/JSC Advanced Development Office

3.4 Orbital Aggregation and Space Infrastructure Systems

An architecture composed of common in-space transportation elements was derived to support both human exploration and commercial applications in the Earth-moon neighborhood. Mission concepts utilizing this architecture are predicated on the availability of a low-cost launch vehicle for delivery of propellant and re-supply logistics. Industry, NASA and other users would share infrastructure costs.

The Orbital Aggregation and Space Infrastructure Systems (OASIS) architecture minimizes point designs of elements in support of specific space mission objectives and maximizes modularity, reusability and commonality of elements across many missions, enterprises and organizations. A reusable Hybrid Propellant Module (HPM) that combines both chemical and electrical propellant in conjunction with modular orbital transfer/engine stages was targeted as the core OASIS element. The HPM provides chemical propellant for time critical transfers and provides electrical propellant for pre-positioning or return of the HPM for refueling and reuse. The HPM incorporates zero-boil off technology to maintain its cryogenic propellant load for long periods of time. The Chemical Transfer Module (CTM) is an OASIS element that serves as a high-energy injection stage when attached to an HPM. The CTM also functions independently of the HPM as an autonomous orbital maneuvering vehicle for proximity operations such as payload ferrying, refueling and servicing. The Solar Electric Propulsion (SEP) Stage serves as a low thrust transfer stage when attached to an HPM for pre-positioning large/massive elements or for the slow return of elements for refurbishing and refueling. The Crew Transfer Vehicle (CTV) is used to transfer crew in a shirt sleeve environment from LEO to the L_1 Earth–Moon Lagrange point and back as well as to the International Space Station (ISS) and any other crewed infrastructure elements.

3.4.1 Connection to NEXT Themes and Goals

Successful development of Low Earth Orbit (LEO) and beyond will require a coalescence of events and technologies anticipated to span decades. Event occurrence and technology development are a function of budgetary, scientific and political variables. The timeframe and order in which these events develop will be gradual and evolutionary in nature unless paradigm shifting technology breakthroughs are introduced. Two developments that are major drivers in the future scenario are cost effective Earth-to-orbit transportation and discovery of commercially viable LEO business opportunities. As an example, there is a school of thought that space tourism will drive the initial development of inexpensive launch capability and space infrastructure. The future scenario that leads to the OASIS architecture is driven by the concurrent needs of the NASA, military and commercial (including space tourism) sectors.

Through all but the last phases of this scenario, crew transportation to LEO is assumed to be provided by the current or upgraded U.S. Space Shuttle along with Russian Soyuz vehicles and, possibly, Chinese derivatives. "Affordable" human transportation to LEO is essential for space tourism and requires significant improvements in efficiency over current human-rated launch vehicles. However, nearly all mass sent into space is in the form of hardware and propellant that does not require a human-rated launch vehicle. Expendable launch vehicles (ELVs) such as the Delta IV-Heavy can be used in the near future to launch valuable hardware while a new generation of mass-produced, inexpensive ELVs may be developed to launch propellant and raw materials that are aggregated in LEO. The reliability of this new generation of ELVs would not have to be as high as conventional launchers since a lost payload would typically be just a tank of liquid hydrogen or oxygen. If technology permits, a non-human rated reusable launch system for aggregation of propellant in LEO could replace the mass-produced ELVs later in the scenario. Systems for facilitating the aggregation of resources in LEO are already under development through the Department of Defense (DoD) Orbital Express program. Orbital Express is a system for maintaining and refueling satellites in support of military objectives. The technologies (e.g., automated rendezvous and docking, on-orbit refueling) and standards developed for the military are assumed to migrate to the commercial sector. Once automated on-orbit servicing of both military and commercial satellites is the norm, the next natural extension is the ability to deliver and transport satellites utilizing a space based infrastructure. This is a leap in scale beyond Orbital Express requiring a large, reusable Orbital Transfer Vehicle (OTV) with cryogenic propellants. The OASIS HPM and CTM are the next step in the evolution of capabilities beyond a military/commercial OTV.

The International Space Station (ISS) offers the potential for reinvigorating the development of space. The key factor is the discovery of processes or products unique to the LEO environment that can form the basis of commercially viable enterprises. Whether these are new wonder drugs or valuable materials difficult to produce on Earth, a commercial demand for ISS resources will quickly follow. It is assumed that when ISS resources can no longer be expanded to accommodate the demand, unpressurized, crew-tended commercial platforms or pressurized, crewed platforms will be deployed in LEO. A reusable on-orbit infrastructure will be required to economically maintain a large number of LEO processing platforms. Economical transportation of materials to and from LEO will also be required if large-scale production occurs. Crewed processing platforms could have much in common with NASA's L_1 Outpost and could yield a core design that may eventually be utilized as a commercial space hotel in support of space tourism.

Satellite systems for telecommunications and remote sensing certainly will be more capable than today's systems. Communications over more frequencies with higher bandwidth along with increased military and civilian remote sensing applications will either require larger satellites with more power and on-orbit upgrade capability or increased constellations of smaller, more disposable systems. Reality will likely be a combination of the two. Both system concepts will benefit from an on-orbit infrastructure and reduced launch costs.

3.4.2 Mission Description

The initial focus areas for this OASIS study were the transportation elements in support of a given set of exploration Design Reference Missions (DRMs) and future low-Earth orbit (LEO) commercialization scenarios (Figure 3.4-1).

A reusable Hybrid Propellant Module (HPM) that combines both chemical and electrical propellant in conjunction with modular orbital transfer/engine stages was targeted as the core OASIS element. The fundamental concept for an HPM-based in-space transportation architecture requires two HPMs and two propulsive transfer stages; one chemical-based and one electric-based. The basic philosophy is to utilize the chemical propellant stored onboard the HPM in conjunction with a chemical transfer/engine stage to provide high thrust during the time critical segments of a

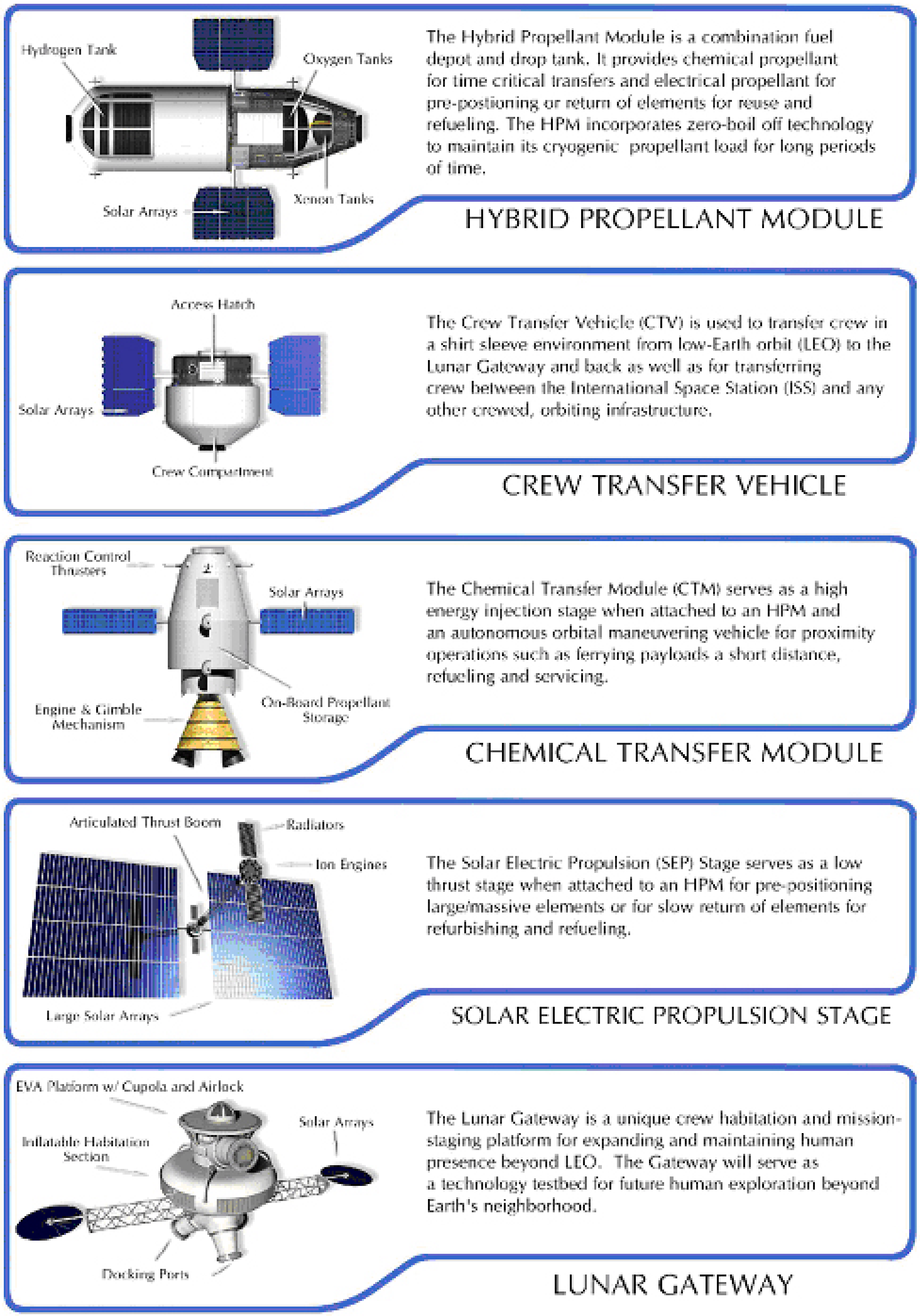


Figure 3.4-1: OASIS Elements (not to scale)

mission (e.g., crew transfers), and utilize the electric propellant with a solar electric transfer/engine stage during non-time critical segments of the mission (e.g., pre-positioning an HPM for the crew return segment of the mission, and return of an HPM to its parking orbit). This architecture can save a significant amount of propellant when compared to an all-chemical mission assuming that the efficiency of the electric propulsion system is sufficiently greater than the chemical propulsion system. For the currently baselined propellants, liquid oxygen (LOX) and liquid hydrogen (LH_2) are assumed to have a specific impulse (Isp) of 466 seconds, and the electric propellant, xenon, is assumed to have an Isp of 3,000 seconds or greater. Although chemical propellant is still required for each crew transfer segment of the mission, the mass penalty for carrying the return trip chemical propellant is substantially reduced due to the substantially higher specific impulse of the electric propulsion system. In other words, the larger the difference between the chemical and electric Isp values, the greater the benefit of employing an HPM-based architecture.

OASIS Exploration Architecture

The Earth-Moon L_1 mission scenario for the OASIS architecture is based on the assumptions that humans will return to the lunar surface for scientific operations and that the L_1 Outpost with Lunar Lander have been deployed to their operational L_1 Lagrange point location. The L_1 Outpost will also provide a facility for in-space science missions and missions beyond the moon.

After the L_1 Outpost/Lunar Lander stack has completed its journey to the L_1 Lagrange point, an HPM is sent to the Outpost to be pre-positioned for the crew return-to-Earth flight. This first HPM is launched on a Shuttle-class launch vehicle. The HPM will be partially fueled based on launch vehicle cargo-to-orbit capability and center of gravity constraints. The sequence of events for this initial HPM deployment is as follows:

- The HPM solar arrays are deployed and tested in LEO.
- While the HPM is in LEO, it is fueled or topped off with liquid oxygen, liquid hydrogen, and xenon delivered by a next generation, low-cost ELV.
- After HPM on-orbit fueling, a SEP Stage is launched to LEO on a Shuttle-class launch vehicle. The SEP Stage deploys its solar arrays, activates its systems, and uses its internal xenon propellant and engines to phase with the orbiting HPM.
- The SEP Stage gaseous hydrogen/oxygen reaction control system (RCS) is used to autonomously rendezvous and dock with the HPM.
- The SEP Stage/HPM stack then begins a 270-day trip to the L_1 Outpost. During the journey the HPM supplies xenon to the SEP Stage while using zero-boil off systems to maintain and store the liquid hydrogen and liquid oxygen that will later be used to transfer the crew from the L_1 Outpost back to LEO.
- The SEP Stage/HPM stack arrives at the L_1 Outpost with almost all of the xenon propellant expended. The SEP Stage utilizes its RCS system for final approach and docking.
- Once the HPM and SEP Stage arrive at the L_1 Outpost, the HPM is checked out to ensure that it is ready for the crew return-to-Earth flight.

Once the crew-return HPM and L_1 Outpost have been verified ready for the crew, the lunar expedition crew is transported to the L_1 Outpost utilizing a second HPM, CTM, and CTV in the following sequence:

- A Shuttle-class launch vehicle delivers the CTM and CTV to LEO.
- The CTM is deployed and loiters until the HPM is delivered and fueled.
- After CTM deployment, the Shuttle-class launch vehicle performs a rendezvous with the ISS and berths the CTV to the station via an International Berthing & Docking Mechanism (IBDM) located on the nadir face of the ISS. The CTV is then configured and outfitted for the journey to the L_1 Outpost.
- The HPM for crew transport to the L_1 Outpost is launched to LEO and fueled/topped off with liquid oxygen, liquid hydrogen and xenon delivered to orbit by a next generation, low-cost ELV. This HPM contains enough liquid oxygen and hydrogen to deliver the crew from LEO to the L_1 Outpost in less than four days. The HPM also carries enough xenon propellant so that the HPM can be returned from L_1 using a SEP Stage.
- The CTM performs a rendezvous and docks with the HPM.
- The CTM performs a rendezvous and docks the CTM/HPM stack to the CTV on the ISS. The crew enters the CTV from the ISS and is now ready to begin the journey to the L_1 Outpost.
- The CTM/HPM/CTV stack departs from the ISS. The CTM utilizes its RCS to separate the stack a sufficient distance to fire its main engines. Then the CTM/HPM/CTV stack begins a series of engine burns that will transport the crew from LEO to the L_1 Outpost.
- The CTM/HPM/CTV stack arrives and docks to the L_1 Outpost.

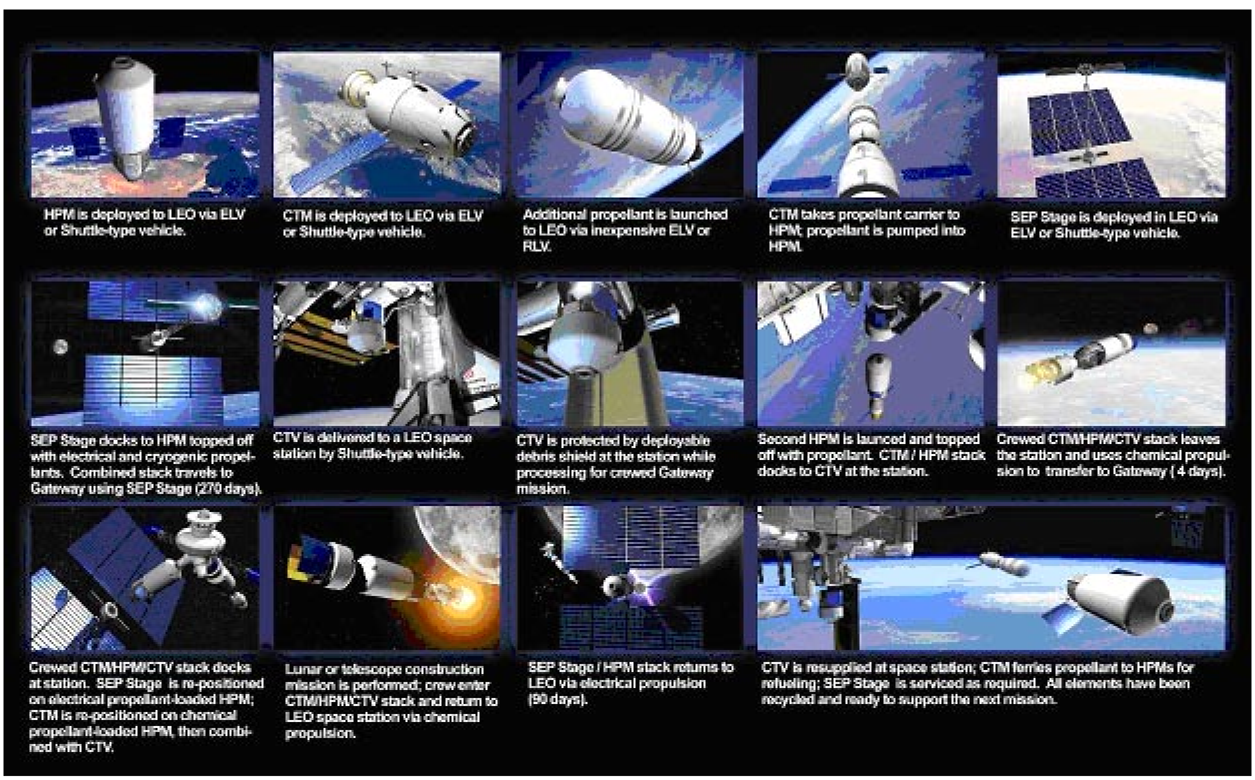
Crew and all elements required to perform a lunar excursion are now at the Outpost. Before the lunar excursion is performed, the CTM, SEP Stage and HPMs must be repositioned such that (1) the HPM with the full load of liquid hydrogen and liquid oxygen is connected to the CTV and CTM, and (2) the HPM with the full load of xenon propellant is attached to the SEP Stage. The repositioning begins with the CTM pulling the HPM loaded with xenon off the CTV and holding it a safe distance from the L_1 Outpost. Next, the SEP Stage utilizes its RCS to transfer the HPM loaded with liquid hydrogen and liquid oxygen to the L_1 Outpost port where the CTV is docked. The HPM stacks approach the desired ports on the Outpost in sequential order. Once this phase is complete, the HPM loaded with hydrogen and oxygen is attached to the CTV. Now, the CTM and SEP Stage separate from the HPMs. They exchange places so that the CTM is attached to the HPM loaded hydrogen and oxygen and the SEP Stage is attached to the xenon-loaded HPM. Once they have been checked out in this configuration, both stacks are ready for the return voyage to LEO. The lunar excursion can now be performed.

After the lunar excursion is complete and the crew has returned to the L_1 Outpost, the return-to-Earth mission sequence begins:

- The crew enters the CTV from the L_1 Outpost.
- The CTM separates the CTM/HPM/CTV stack from the L_1 Outpost.
- The CTM then propels the HPM and crewed CTV back to LEO. The stack docks to the ISS where the crew will depart for Earth on a Shuttle flight.
- The CTV is refurbished on the ISS.
- The HPM and CTM perform a rendezvous with ELV-delivered propellant carriers, refuel and are ready for the next L_1 Outpost mission sortie.

Either prior to or shortly after the crew departs from the Gateway, the SEP Stage and xenon-loaded HPM leave the Gateway for the return to LEO. Once the SEP Stage/HPM stack is back in LEO, the HPM is refueled via the ELV-delivered propellant logistics carriers. The SEP Stage internal tank is also topped off with xenon. The SEP Stage arrays may need replacement at the ISS. At this point, all of the elements that were utilized for crew and supply transfer with the exception of the Lunar Lander have returned to LEO and are ready to support another mission.

Figure 3.4-2: OASIS Exploration Architecture Mission Profile



OASIS Commercial Applications

An on-orbit reusable propellant depot could perform a number of missions to support commercial and military orbital assets in the future. Some of these missions are not practical with today's aerospace infrastructure. This section discusses potential usage of the OASIS elements and illustrates specific scenarios for each mission.

The HPM when combined with a propulsion module such as a CTM is envisioned to be used as an upper stage to augment the launch capability of a low cost RLV or ELV that would only provide access to LEO (altitude < 400 km). One potential mission is the deployment of a satellite to its final orbital position. Figure 3.4-3 illustrates the deployment scenario. With HPMs paired with CTMs and pre-positioned in storage orbits, mission planners would select the HPM/CTM closest to the final orbit position of a payload for use on this mission. Prior to launching the satellite, one or more ELVs would launch LH2 and LOX propellants into LEO. The HPM/CTM (or perhaps CTM only) would rendezvous and dock with the propellant delivery stage and transfer the propellants into the HPM. The satellite would then be launched on another ELV or RLV to LEO. The HPM/CTM would rendezvous and dock with the satellite and use CTM propulsion to move the combined stack to the final deployment orbit position and release the satellite. It may be possible to deliver more than one satellite per mission with the HPM/CTM maneuvering to release each satellite at the correct true anomaly. Following deployment, the HPM/CTM would perform the necessary engine burns to return to the parking orbit to await the next mission.

The figure illustrates a satellite delivery to a final orbit requiring no additional propellant usage for maneuvering by the satellite to complete the delivery. For satellites destined for orbits requiring velocity increments greater than the velocity capability of the HPM/CTM, the system could be used to transfer the satellite(s) from LEO into a transfer orbit (e.g., geosynchronous transfer orbit (GTO)) as is frequently done by the present day launch industry. The scenario would be the same; however, the satellite would be required to carry a propulsion system such as an apogee kick motor with enough propellant to complete delivery to the final orbit position.

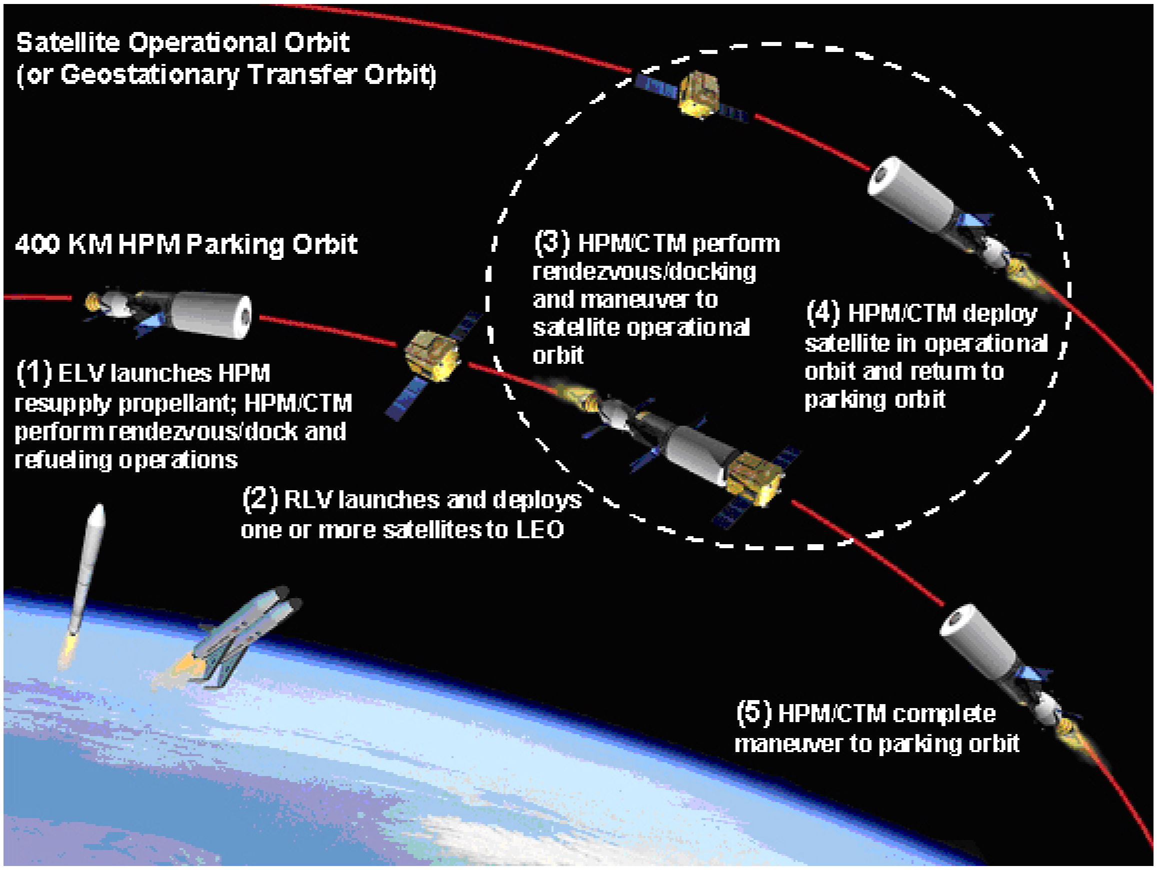


Figure 3.4-3: HPM Commercial Satellite Deploy Scenario

One advantage of a reusable propellant depot with autonomous operations capability is the opportunity to directly service satellites already in orbit. Servicing could extend their life beyond original design and delay the need to replace these expensive assets. Satellite lifetime is primarily governed by the depletion of station keeping propellant and, secondarily, by degradation of power-generating solar panel cells. The ability to refuel and refurbish satellites could significantly extend their useful lives. The capability of changing out components of healthy satellites with newer technology components could improve satellite performance without the cost of designing, manufacturing and launching entirely new spacecraft. While there are minor differences in the details of the refueling and refurbishing missions, they can generally be combined into a category of on-orbit servicing. Figure 3.4-4 illustrates a servicing mission scenario. Most of the steps in the mission sequence are the same as for the deployment scenario.

One form of on-orbit servicing for which the OASIS architecture is uniquely suited is refueling those satellites designed to use xenon propulsion systems for station keeping and maneuvering. Rather than using its supply of xenon to fuel a SEP Stage, an HPM/CTM stack could use the xenon supply to refuel one or more satellites nearing the end of their useful life due to propellant depletion. This mission would require that the HPM have the plumbing lines and valves to control the transfer of xenon to the satellite. For this to be a viable market, a good share of the satellite industry would need to adopt xenon propulsions systems and provide a common refueling port to accommodate the transfer of fuel.

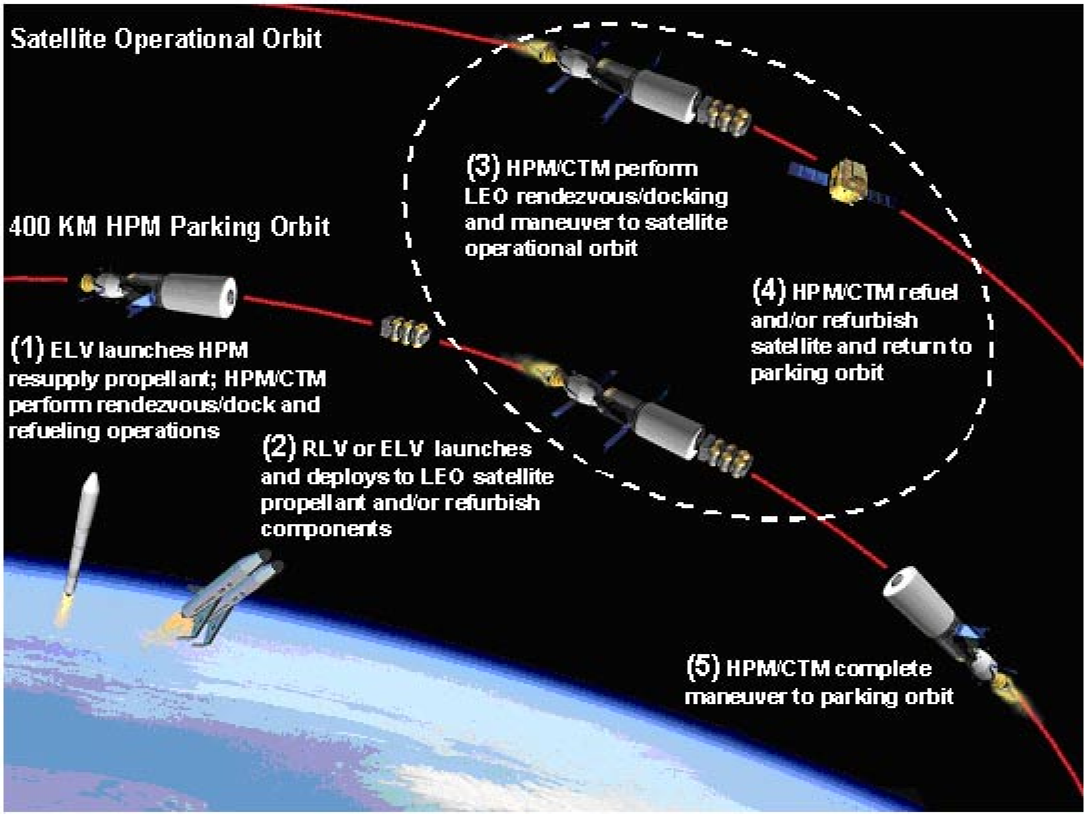


Figure 3.4-4: HPM Commecial Satellite Servicing Scenario

The refurbishment mission would be conducted in the same manner; however, the HPM/CTM stack would require the capability of removing old components and installing the replacements. This may be accomplished by formation flying in close proximity to the satellite or by docking with the satellite. In either case, a robotic arm controlled either remotely by ground controller or autonomously would be required to accomplish the mission. Hence, HPM subsystems in addition to those required for the exploration missions may need to be designed and developed to support the variety of potential commercial missions.

Additional commercial missions for which the HPM would be suited include rescue and subsequent retrieval or deployment in correct final or transfer orbits. Removal of older satellites into disposal orbits or possibly even self-destructive reentry orbits may be a possible commercial application for OASIS elements. Details of each of these scenarios would differ slightly from those discussed above but the major scenario steps would be similar in all of these missions.

3.4.3 Mission Requirements

The OASIS architecture is predicated on the assumption that in the 2015+ period, a low-cost launch system will have been developed to provide transportation from Earth launch sites into LEO. These LEO parking orbits are assumed to be circular and between 200 and 400 km in altitude. Orbits below 200 km sustain orbit decay rates that are too rapid to support timelines needed for orbital operations.

Launch systems are assumed to be one of two types depending on payload. Highly reliable, reusable launch vehicles (RLVs) or expendable launch vehicles (ELVs) may be developed to provide launch services for high value payloads such as human crews, cargo and satellites. Potentially lower reliability ELVs may be developed to provide very low cost launch services for lesser-valued payloads such as re-supply propellant.

Description	Requirement	Rationale
Crewed Space Access	7 Crew to and from LEO each month in support of gateway missions, ISS and commercial ventures	Requirement based on concurrent gateway and ISS utilization coupled with TBD commercial activities
Heavy Lift for Sensitive Cargo	Shuttle/35 Mt, 5 meter shroud Delta IV derivative type performance	All OASIS elements sized for Shuttle performance. More margin is available with the delta IV derivative type performance. There is no need for sending crew with sensitive cargo.
Propellant Delivery	5 to 20 Mt of cryogenic propellant (H2, O2, Xenon) a week to LEO at around $1000/kg	The size (5 MT or 20Mt) or the type (ELV vs RLV) is not as important as the economics of getting the propellant to orbit.

Table 3.4-1: OASIS Mission Requirements

3.4.4 Key Technology Investments

The advanced technologies necessary to make the OASIS architecture a reality, including technologies specifically applicable to the HPM, CTM, CTV, and SEP Stage, are listed in Table 3.4-2 and described below:

- Zero boil-off cryogenic propellant storage system for the HPM providing up to 10 years of storage without boil-off.
- Extremely lightweight, integrated primary structure and micrometeoroid and orbital debris shield incorporating non-metallic hybrids to maximize radiation protection. This is required for all OASIS elements.
- High efficiency power systems such as advanced triple junction crystalline solar cells providing at least 250 W/kg (array-level specific power) and 40% efficiency, along with improved radiation tolerance. Required for the HPM, CTM, and CTV.
- Long-term autonomous spacecraft operations including rendezvous and docking, propellant transfer, deep-space navigation and communications, and vehicle health monitoring (miniaturized monitoring systems). Applicable for all OASIS elements.

- Reliable on-orbit cryogenic fluid transfer with minimal leakage using fluid transfer interfaces capable of multiple autonomous connections and disconnects

- Lightweight composite cryogenic propellant storage tanks highly resistant to propellant leakage

- Advanced materials such as graphitic foams and syntactic metal foams. Required for all OASIS elements.

- Long-life chemical and electric propulsion systems with high restart (>50) capability, or systems with on-orbit replaceable and/or serviceable components.

- High thrust electric propulsion systems (greater than 10 N).

- Integrated flywheel energy storage system combining energy storage and attitude control functions.

These technologies needed to enable the OASIS elements require targeted research and development. With the proper funding levels, many of the technologies could be available within the next 15 years. Accelerated funding levels could make this timeline significantly shorter.

Technology	Summary Description	Current TRL	Additional Applications
Integrated Energy Storage and Attitude Control	Composite flywheels to provide spacecraft momentum management and energy storage	2-3	Long-duration/large spacecraft applications
Photovoltaics	High-efficiency photovoltaic cells (41% AM0) for in-space power generation	5	All spacecraft power applications
Photovoltaics	Large deployable thin film arrays	2-3	All spacecraft high power applications
Zero-Boiloff Cryogenic Fluid Storage	Long-lifetime cryocoolers to remove thermal energy for long-term fluid storage	4	Sensor cooling
Multi-Function Structure	Integrated primary structure, radiation shielding, and micrometeoroid/orbital debris shielding	5	All spacecraft applications
Autonomous Navigation System	Precision autonomous navigation	5	All spacecraft applications
Cryogenic Fluid Transfer	Efficient transfer of large quantities of cryogenic liquids in low gravity	4	All deep space HEDS missions
Composite Structures	Use composite structures to reduce the weight of vehicle primary structure and fluid storage tanks	6	Space Launch Initiative
Graphitic Foam	Light-weight filler for debris shielding that also adds thermal protection	5	Various applications
Carbon-Carbon Composite Radiators	Light-weight radiator materials operating at high temperatures	2-6	All spacecraft TCS applications
Integrated Cryogenic OMS/RCS Systems	High cycle life LH_2/LO_2 main propulsion engines and RCS thrusters	5-6	Upper stages, other HEDS applications

Life Support	Closure (>95%) of air and water loops to reduce consumables and rate of resupply		Other HEDS applications
Electrostatic Ion Thrusters	High power 60-70 cm dia. gridded ion engine operating at 50 kW producing 3300s Isp on xenon	2-3	Long-duration spacecraft, human Mars missions, outer planet exploration

Table 3.4-2: OASIS Key Technology Investments

3.4.5 References

1. "Orbital Aggregation & Space Infrastructure Systems (OASIS), Preliminary Architecture and Operations Analysis – FY2001 Final Report". June 2002.

2. "OASIS In-Space Architecture – A Commercialization Analysis: PowerPoint Presentation". May 2002.

POC: Pat Troutman, NASA Langley Research Center, Hampton Virginia 23861. p.a.troutman@larc.nasa.gov

3.5 Mars Exploration

The human exploration of Mars will be a complex undertaking. It is an enterprise that will confirm the potential for humans to leave our home planet and make our way outward into the cosmos. Though just a small step on a cosmic scale, it will be a significant one for humans, because it will require leaving Earth on a long mission with very limited return capability. The commitment to launch is a commitment to several years away from Earth, and there is a very narrow window within which return is possible. This is the most radical difference between Mars exploration and previous lunar explorations. During the past decade, NASA has studied various mission approaches for expanding human presence beyond low-Earth orbit.[i, ii, iii, iv, v] Each of these mission studies, referred to as an "architecture" provides descriptive information of the overall exploration theme and its derivation from, and links to, driving national needs. Each of these architectures identifies governing objectives, ground rules and constraints, the mission strategy to be used in developing scientific implementation approaches, implementation and technology options, and important programmatic decision points. Architectures for human exploration consist of the integrated set of functional building blocks that describe the method and style by which humans leave Earth, travel to destinations beyond low-Earth orbit, carry out a set of activities to accomplish specified goals, and subsequently return to Earth.

Human exploration missions beyond low-Earth orbit have been an integral part of NASA's strategic vision.[vi, vii, viii] During the past several years, personnel representing several NASA field centers have formulated a "Reference Mission" addressing human exploration of Mars. This reference approach has undergone numerous revisions and improvements. This report summarizes the current exploration architecture work and describes alternative approaches for conducting the first human exploration of Mars including several technological alternatives.

3.5.1 Connection to NEXT Themes and Goals

Scientific Discovery: A robust human exploration program will enable key discoveries that expand our understanding of the Universe, Earth, and ourselves. Key questions that drive our discoveries include[ix]: How did the universe begin and evolve? We seek to explain the earliest moments of the universe, how stars and galaxies formed, and how matter and energy are entwined on the grandest scales. How did we get here? Investigating how the chemical elements necessary for life have been built up and dispersed throughout the cosmos. Where are we going? Comparing the climatic histories of Earth and Mars to understand our planets past and future. Are we alone? This is perhaps the grandest central human question seeking to understand if life exists elsewhere in the universe. As we continue to explore with both humans and robots we will gather new information, new knowledge, and new questions to answer. Each step we take will help us answer these fundamental questions.

Technology Advancement: Space exploration initiatives have shown to stimulate a wide range of technological innovations that make their way into the marketplace. The human exploration of Mars currently lies at the ragged edge of achievability. Some of the technology required to achieve this mission is either available or on the horizon. Other technologies will be developed based on the needs of the mission. Proper investment in the development of high-leverage technologies will enable safe, effective, and affordable human and robotic exploration. The new technologies or the new uses of existing technologies will not only benefit humans exploring Mars but will also enhance the lives of people on Earth.

Leadership: A fundamental objective guiding United States space activities has been, and continues to be, space leadership. Human exploration endeavors provide the nation an opportunity to establish and maintain the United States as a leader in the human exploration and development of space. The exploration programs of the past have been symbols of power. The U.S. can be the leader of the exploration programs of tomorrow that represent common pursuits by nations of the world to improve the quality of life for all.

Inspiration: Bold new endeavors can serve as a catalyst for the nation. Long-range commitments to space will stimulate our national educational system and inspire students to learn. Motivated students are essential to excellence in education. Human exploration missions will motivate and inspire new generations on which our future as a nation depends.

Evolutionary Approach: The human exploration of Mars cannot be an end to itself. The implementation approach must be progressive, sustainable, and one that maximizes the use of existing and previously developed capabilities. Integrating all aspects of the mission including fundamental research, technology development, data acquisition, and flight systems is vital to reducing the cost and risk of the exploration endeavor. Whereas the space program of the past was the responsibility of the government, the program of tomorrow will combine the efforts of the government and private sector to fulfill the exploration objectives while satisfying and capturing new economic commercial markets.

3.5.2 Mission Description

The principal use of the Reference Mission is to lay the basis for comparing different approaches and criteria in order to select better ones. That is, it is used to form a template by which subsequent exploration strategies may be evaluated for consideration as alternate or complementary approaches to human exploration of Mars. When comparing architectures specific measures of merit are considered including human health and safety, cost, performance, mission return, and schedule. With this in mind, the Reference Mission may be used to:

- Understand requirements for human exploration of Mars in the context of other space missions and research and development programs.
- Establish an end-to-end mission baseline against which other mission and technology concepts can be compared.
- Derive technology research and development plans.
- Define and prioritize requirements for precursor robotic missions.
- Define and prioritize flight experiments for precursor human missions, such as those involving the Space Shuttle and International Space Station.
- Open a discussion with international partners in a manner that allows identification of potential interests of the participants in specialized aspects of the missions.

- Provide educational materials at all levels that can be used to explain various aspects of human interplanetary exploration.
- Describe to the public, media, and political system the feasible, long-term visions for space exploration.

The choice of the overall mission strategy has a profound influence on the safety, mission return, and subsequent cost of the exploration endeavor. Special consideration of specific mission related design choices including the mission class, human health hazards, aborts, and mission sequencing must be made early in the mission design process as discussed in this section. Finding the proper balance between these inter-related parameters requires stringent systems engineering processes and careful evaluation of the resulting designs and strategies.

3.5.2.1 Mars Mission Classes

Over the past several years, piloted Mars mission analyses have focused primarily on understanding the differences between available interplanetary trajectory classes and their associated energy (and therefore propulsion system) requirements. Traditionally, these mission classes have been treated as distinct and separate options, with first order parameters such as round-trip mission time and required propellant mass used as figures of merit. Such analyses have failed to include considerations that may prove to be critical in formulating operationally sound mission strategies. Such considerations include crew health effects relative to times spent by the flight crew on the Martian surface and in transit, mission return, and overall mission operational approaches. Round-trip human missions to Mars can be characterized by the length of time spent on the surface, short-stay and long-stay, as discussed below.

<u>Short-Stay Mars Missions</u>

The first Mars mission class consists of short stay-times (typically 40 days) and round-trip mission times ranging from 365-660 days. This is often referred to as an opposition-class mission, although the exploration community has adopted the more descriptive terminology “short-stay” mission. Trajectory profiles for typical short-stay missions are shown in Figure 3.5-1. This class of mission has high propulsive requirements even when employing a gravity-assisted swing-by of Venus or performing a deep space maneuver to reduce the total mission energy. Short-stay missions always have one short transit leg, either outbound or inbound, and one long transit leg, the latter requiring close passage by the sun (0.7 AU or less). After arrival at Mars, rather than waiting for a near-optimum return alignment, the spacecraft initiates the return after a brief stay and the return leg cuts well inside the orbit of the Earth to make up for the “negative” alignment of the planets that existed at Mars departure. Distinguishing characteristics of the short-stay mission are: 1) short-stay at Mars, 2) short to medium total mission duration, 3) perihelion passage inside the orbit of Venus on either the outbound or inbound legs, and 4) large total energy (propulsion) requirements.

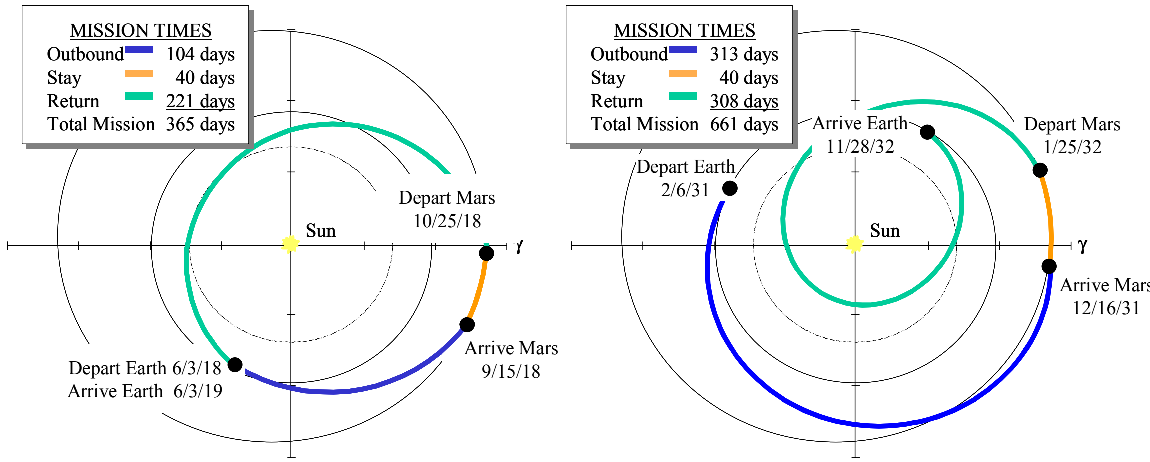


Figure 3.5-1: Example Short Stay Mission Profiles

Examination of Figure 3.5-2 and Figure 3.5-3 shows that the short-stay mission approach has some distinct disadvantages. First, the total energy requirement, as measured in velocity change (delta-v or ΔV), varies greatly for each mission opportunity to Mars, repeating across the synodic cycle. (The synodic cycle is the period of time required for the relative phasing between Earth and Mars to repeat itself). The variation in total energy is also highly dependent on the total round-trip mission time and, in fact, can vary by as much as 88% across the synodic cycle. For all Mars mission classes, as the trip time decreases, the required injection velocity and Mars arrival velocity both increase. This is important not only because higher total energies require exponentially greater propellant quantities, but also higher approach velocities can eliminate some leading technologies from consideration, such as aero-capture at Mars. In addition, Figure 3.5-3 shows that the total round-trip mission times can be quite large up to 660 days, with only 40 days at Mars. These long periods in the deep-space environment raise many human health and performance issues that must be considered during the mission design process. Constructing a proper balance between the propulsion system requirements (velocity change and resulting vehicle size) and desired total round-trip times is discussed later.

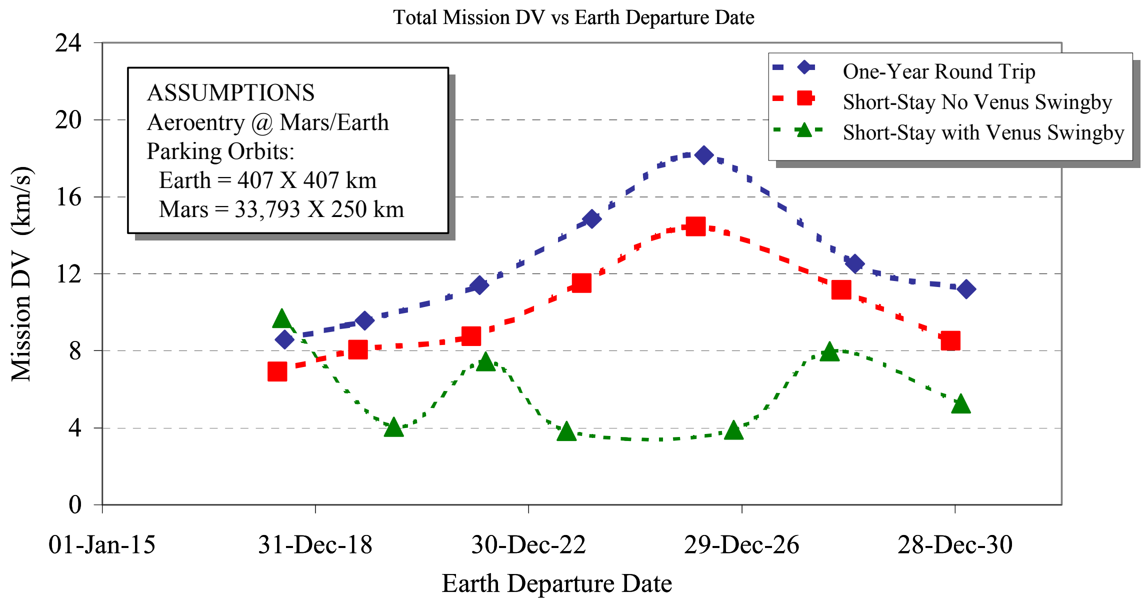


Figure 3.5-2: Mars Short-Stay Trajectory Energies

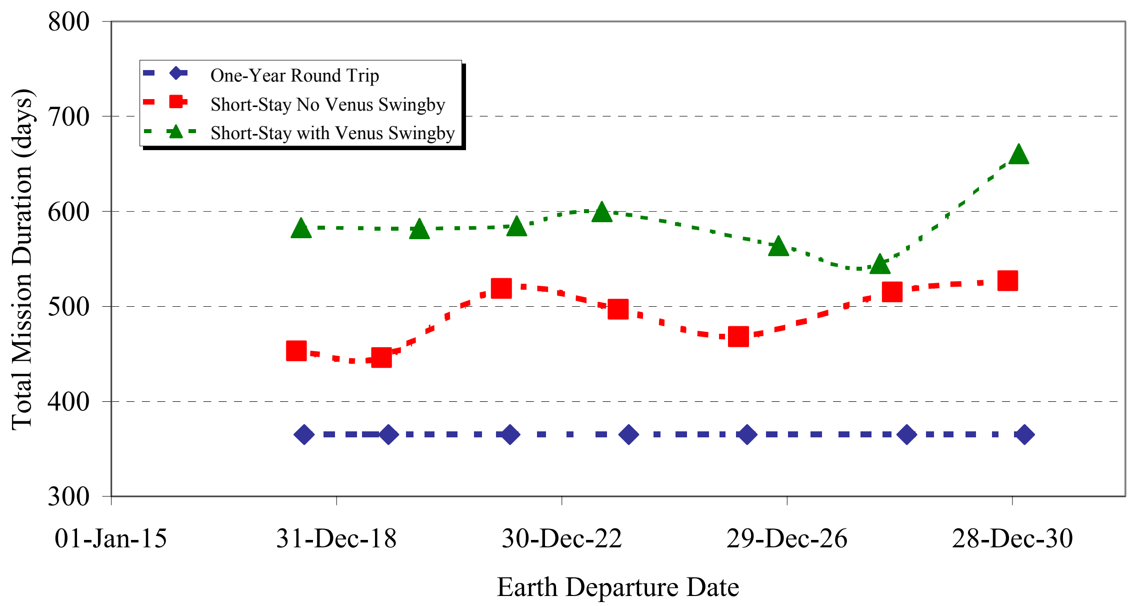


Figure 3.5-3: Mars Short-Stay Trajectory Round-Trip Times

Long-Stay Mars Missions

The second Mars mission class is typified by long-duration stay-times (as much as 600 days) and long total round-trip times (approximately 900 days). This mission type is often referred to as conjunction-class, although the exploration community has adopted the more descriptive terminology "long-stay" mission. These missions represent the global minimum-energy solutions for a given launch opportunity. Unlike the short-stay mission approach, instead of departing Mars on a non-optimal return trajectory, time is spent at Mars waiting for more optimal alignment for lower energy return. A variation of this long-stay mission type has recently gained attention.[x] This mission has a total round-trip time comparable to those of the minimum-energy, long-stay missions, but the one-way transits are substantially reduced. Distinguishing characteristics of the long-stay mission include: 1) long total mission durations, 2) long-stays at Mars, 3) relatively little energy change between opportunities, 4) bounding of both transfer arcs by the orbits of Earth and Mars (closest perihelion passage of 1 AU), and 5) relatively short transits to and from Mars (less than 200 days). The mission flight profile for a typical fast-transit mission is shown in Figure 3.5-4 and a comparison of the short-stay and long-stay mission energies is shown in Figure 3.5-5.

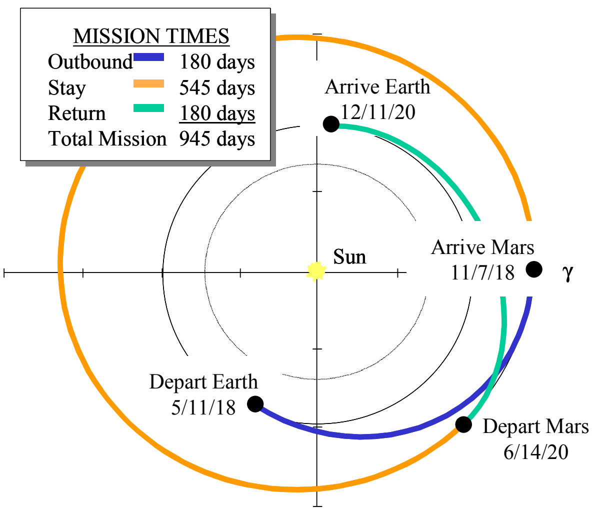


Figure 3.5-4: Example Long-Stay Profile

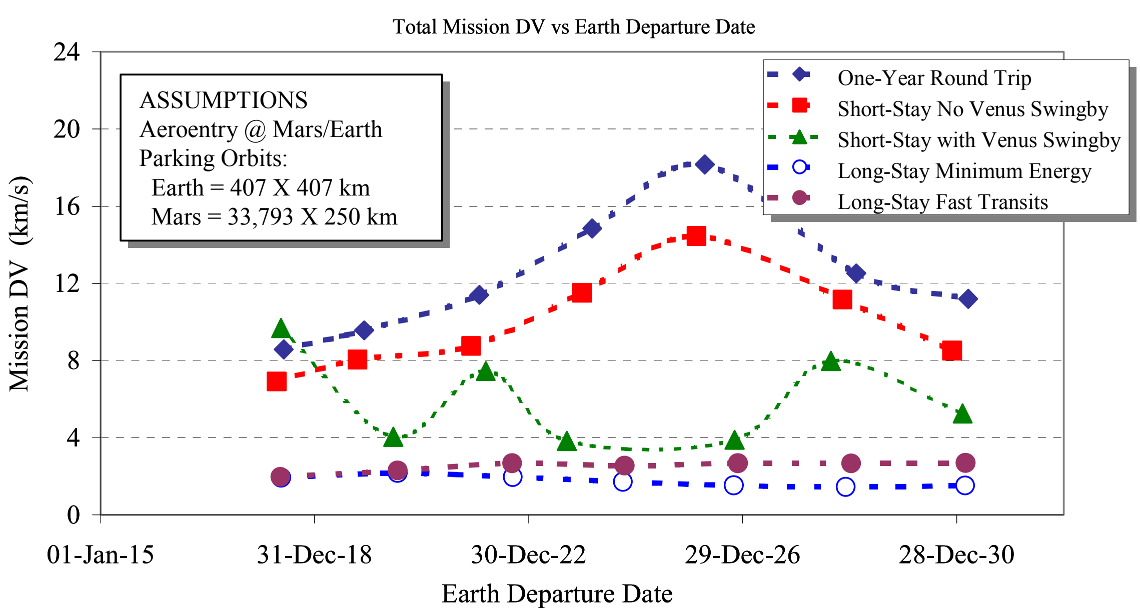


Figure 3.5-5: Mars Long-Stay/Short-Stay Trajectory Energy Comparison

3.5.2.2 Comparison of Mars Mission Classes

The applicability of each of the previously discussed mission types to the preliminary human expeditions to Mars has been the subject of much debate. The opinion has generally been held that the initial flights should be short-stay missions performed “as fast as possible” (so-called “sprint” missions), ostensibly to minimize crew exposure to the zero-gravity and space radiation environment, to ease requirements on system reliability, and to enhance the probability of mission success. When considering “fast” Mars missions, it is key to distinguish whether one is referring to fast *round-trip* or fast *transit* missions. In fact, past analyses have shown that decreasing round-trip mission times for the short-stay missions does not equate to fast transit times (i.e., less exposure to the zero-gravity and space radiation environment) as compared to the long-stay missions. *Indeed, fast transit times are available only for the long-stay missions.* This point becomes clear when looking at Figure 3.5-6, which graphically displays the transit times as a function of the total round-trip mission duration. Although the short-stay mission has shorter total duration of the long-stay missions, over 90% of the time is spent in deep-space transit, compared to 30% for the fast-transit mission. In order to appreciate the significance of this distinction, the hazards of the interplanetary environment to the crew must be discussed in more detail.

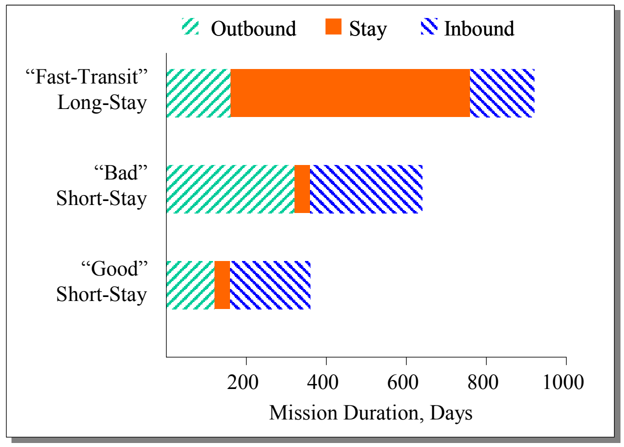


Figure 3.5-6: Mission Comparisons

Radiation Hazards

The interplanetary ionizing radiation environment of concern to mission planners consists of two components: galactic cosmic radiation (GCR) and solar particle events (SPEs). GCR is an isotropic bombardment of light and high-energy heavy ions originating from outside the solar system. Due to interaction with the solar magnetic field, GCR levels increase significantly in years near solar minimum and increase slightly with increasing distance from the sun. Other than these effects, the GCR effluence and composition are constant. In contrast, SPEs are sporadic, short-duration events (increasing in probability during solar maximum), consisting primarily of medium energy protons and alpha particles ejected from the sun. While SPE radiation levels can be lethal in an unshielded environment, the brevity of the event (several hours to a few days) allows a “storm shelter” type of crew protection to be feasible. Neither of these sources is a major concern in low Earth orbit due to the shielding effect of the Earth’s magnetosphere.

Estimating the biological damage or *dose-equivalent* of the interplanetary radiation environment is a difficult process at the present time. While the composition and effluence of GCR is relatively well understood, there are large uncertainties involving the radiation interaction with biological systems and the effectiveness of various shielding materials. These uncertainties arise from the fact that the heavy ions (primarily iron ions) that constitute the majority of the GCR dose-equivalent are not common in terrestrial radiation sources. Due to this lack of GCR ground simulation, space radiation transport computer models have been developed at NASA's Glenn Research Center and other locations.[xi, xii] Currently, there are no laboratory validated GCR transport codes; however, estimates of interplanetary radiation exposure levels and the effectiveness of various shielding materials have been made.[xiii, xiv, xv] These estimates, combined with the trajectory options previously discussed, form the basis of a relative evaluation of mission classes.

Legally, NASA space flight crews are judged to be federal agency radiation workers and have been covered by Occupational Safety and Health Administration (OSHA) radiation protection regulations since 1980.[xvi] These regulations, however, have not been promulgated for space flight activities and the law allows the agency to adopt supplemental standards in the case where no appropriate OSHA standards exist. In 1982, NASA received approval of such supplementary standards based on National Academy of Sciences recommendations. In 1990, NASA legally adopted the more restrictive recommendations of the National Council on Radiation Protection and Measurements (NCRP) as the agency's supplementary standard for space flight crew dose limits to be applied to all but "exceptional exploration missions."[xvii] *There are currently no radiation exposure standards for human exploration missions.* These regulations bind NASA to provide a preflight appraisal of radiation hazards to be encountered and advise NASA to adhere to the ALARA (As Low As Reasonably Achievable) principle. ALARA takes the position that no level of exposure to ionizing radiation is considered safe and that all reasonable protective measures should be taken even if the legal standards have been met. Thus, from a radiation perspective, important considerations when constructing a mission approach include the total mission time spent in the various environments, deep-space and Mars surface, as well as the perihelion passage as shown in Figure 3.5-7.

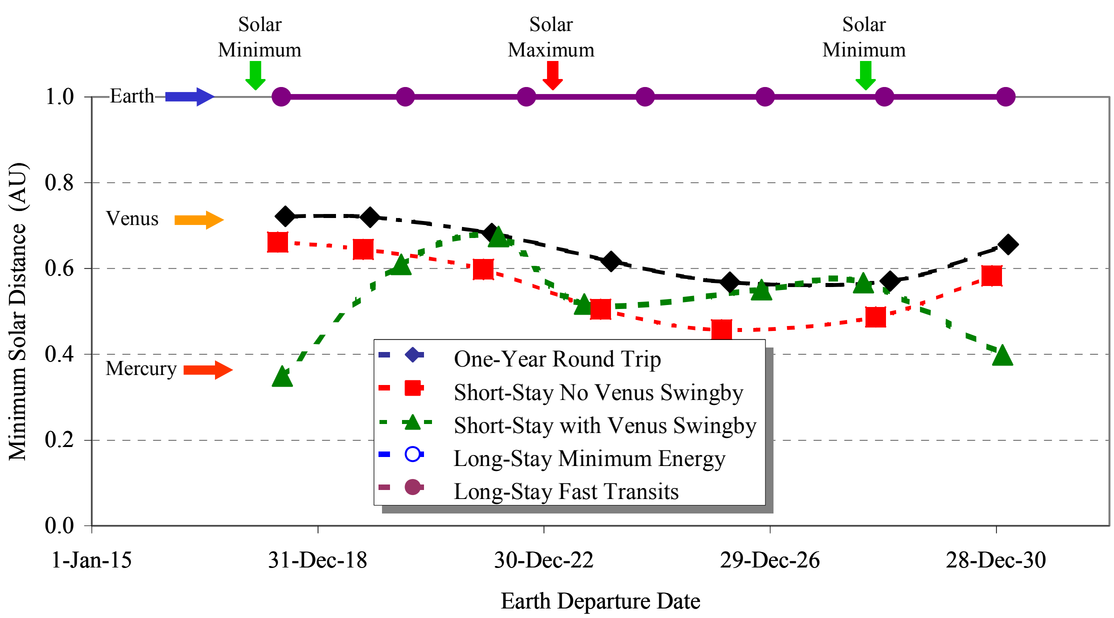


Figure 3.5-7: Mars Long-Stay/Short-Stay Minimum Solar Distance Comparison

Zero-gravity Hazards

An additional threat to the Mars mission crew is the extended period of time required in the zero-gravity environment. This represents another discipline in which our understanding of long-term exposure is rudimentary at best. It is known, however, that significant physiological changes occur when zero-gravity time begins to be measured in months. Bone decalcification, immune and cardiovascular system degradation, and muscular atrophy are a few of the more unpleasant effects. Research on the effects of long-term zero-gravity on the human body is in an elementary stage. The longest U.S. missions to date include 188 days on the Russian Space Station Mir by astronaut Shannon Lucid, 196 days by astronauts Daniel Bursch and Carl Walz, and the longest Soviet mission was 438 days. In none of these cases were crews exposed to zero-gravity/partial-gravity/zero-gravity sequences similar to that projected for Mars missions.

Upon arrival on the Martian surface, the crew must spend some time readapting to a partial-gravity field. Current data indicates that recovery in a 1-g environment can be fairly rapid (on the order of a few days), but development of full productivity could require significantly more time. This may be of concern for the short-stay missions where a substantial portion of the surface stay time could be consumed by crew adaptation to 0.38 g's. Conversely, ample time will be available for the crew to regain stamina and productivity during the long surface stays missions.

Several potential solutions to the physiological problems associated with zero-gravity transits to and from Mars include: countermeasures (exercise, body fluid management, lower body negative pressure), artificial-gravity spacecraft, and reduced transit times. The usefulness of countermeasures to reduce some of the zero-gravity effects is still unknown. Soviet long-duration crews have experienced physiological degradation even when rigorous exercise regimens have been followed. However, most of these effects seem to be quickly ameliorated upon return to a 1-g environment, at least when immediate medical aid is available.

Rotating the Mars Transfer Vehicle (MTV) is a method of providing an artificial gravity environment for the crew and is most often associated with low-performance propulsion systems, or the short-stay class of trajectories (since both require long transit times). Studies have indicated that the Mars transfer vehicle design mass penalties are on the order of 5-20% if artificial-gravity is incorporated.[xviii] Depending upon the specific configuration, there may also be operational complications associated with artificial-gravity spacecraft including EVA, maintenance, and the spin-up/spin-down required for mid-course maneuvering and rendezvous/docking.

The inbound and outbound transits for short-stay missions are typically separated by a short duration, typically 30-40 days. It is fair to question whether such a short time spent in a 0.38-g field will counteract five to eight months of outbound zero-gravity exposure. In contrast, the one-way trip times of the fast-transit missions are *within the current U.S. zero gravity databases*, which will certainly be augmented by normal International Space Station operations prior to executing human interplanetary missions. Also, note that the fast-transit mission's zero-gravity transfer legs are separated by a substantial period of time in the Martian gravitational field. This long period on the surface of Mars may well prove sufficient to ameliorate the effects of the relatively short outbound transit.

3.5.2.3 Other Mission Design Considerations

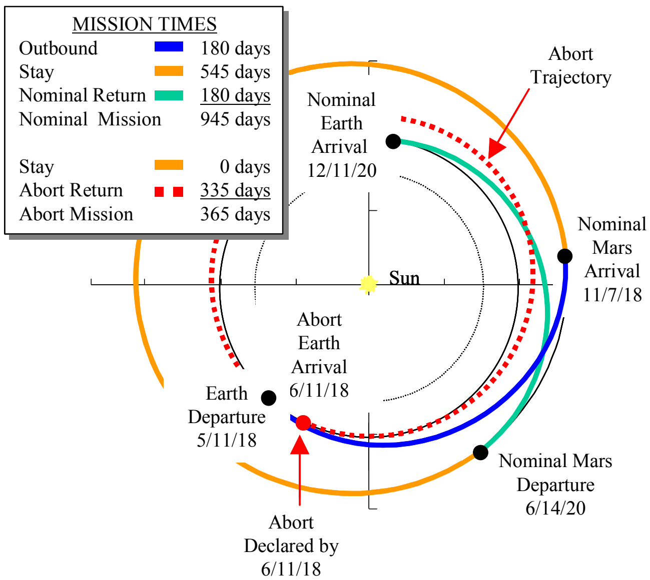
<u>Abort Considerations</u>

An end-to-end mission abort strategy generally requires the vehicle and mission to be defined to a level sufficient for a failure-mode analysis to be performed. While such a comprehensive effort for piloted Mars flights would be premature at this point, it is an accepted risk-management practice to identify possible failure scenarios, assess their impacts, and suggest design approaches which eliminate highly undesirable consequences, (i.e., loss of crew) even if the associated failure statistics are unknown.[xix]

During studies of Mars missions conducted previously, the primary emphasis was on the fast-outbound legs in order to have the flight crew reach Mars as quickly as possible, reducing the effects of zero-gravity and thus enhancing mission success (even though the need for a long return leg was recognized in some cases).

Although an abort option protecting against a Mars transfer vehicle propulsion failure during all mission phases is desirable, it is important to understand the cost of protecting against certain failure modes and, perhaps more importantly, what condition the crew is place during certain abort modes. Traditionally, this concept has paralleled that of the "free-return" abort, which became familiar to mission planners during the Apollo program; that is, the spacecraft is injected onto a trajectory that eventually returns it to the vicinity of Earth with no subsequent major propulsive maneuvers. (Actually, mid-course corrections are required for the re-encounter to occur.) While such trajectories exist in the case of Mars, they can impose high costs on the nominal mission. The free-return abort profile is forced into a multi-revolution, Mars gravity-assist with a round-trip time on the order of two to three years. This multi-year abort exposes the crew to the interplanetary radiation and zero-gravity environment for periods far outside current or near-future databases. It should also be noted that free-return aborts have a limited period of usefulness during the mission. A flawless injection onto the outbound trajectory is required and the abort capability is lost subsequent to final targeting for Mars orbit insertion. Free-returns are effective *during the ballistic trans-Mars coast only.* A complete failure of the main propulsion system at any other time during the mission (other than prior to attaining Earth escape velocity) implies loss of vehicle and crew.

In general, it is felt that for the challenging Mars mission, free-return aborts do not provide acceptable crew return options in the event of a main propulsion system failure, nor is the extent of abort coverage sufficient. Also, having a low-probability event like an abort substantially drive vehicle designs (e.g., artificial-gravity, large-scale radiation protection, and extended system lifetimes) with no gain in mission productivity increases the cost and complexity of the nominal mission. Because of these limitations, it is recommend that the transfer vehicle contain redundant main propulsion capability, in the form of multiple engines, tanks and propellant lines, along with a *powered* abort strategy which allows a faster return of the vehicle and crew using a degraded (or less redundant) propulsion system during as much of the mission phase as possible. An example of this powered abort strategy is shown in Figure 3.5-8. Earth return capabilities are present early in the mission phase, depending on the specific vehicle design, to return the crew post-TMI certainly within hours, if not days or perhaps months, of the injection event.



Post-TMI Abort Option Examples

1. Early Return Option 1
   - Abort declared within 8 hours after TMI
   - Crew lives in Mars transfer vehicle after abort declaration
   - Crew returned 30 days later
2. Early Return Option 2
   - Abort declared within 30 hours after TMI
   - Crew returned in the Earth entry capsule
   - Return transit time 1-2 days
3. Heliocentric Aborts
   - Abort declared within 1-2 months after TMI
   - Return transit times range from 360-570 days
   - Crew lives in the Mars transfer vehicle
   - Can perform this abort only for some (3 of 7) opportunities

Figure 3.5-8: Example Early Mission Abort Strategy

Mission Mass and Launch Strategy

In addition to evaluating trip time, human health and performance, and abort strategies during the mission design phase, it is equally important to consider the overall launch and vehicle assembly strategy for the mission. Proper balance between mission strategy, technology push, and operational strategies must be developed in order to determine the best architectural approach. As was discussed earlier, the total mission energy, measured in velocity change (delta-v) has an exponential impact on the overall vehicle size. In fact, the size of the vehicle is directly related to mission strategy and propulsion technology employed. Parametric modeling techniques are generally used early in the mission design phase to determine the sensitivity and relative magnitude of the mission approaches under consideration. Utilizing parametric techniques provides the mission designer an approach to quickly assess various vehicle, technology and mission strategies to determine those combinations that make the best “integrated” strategy. An example of this parametric analysis is shown in Figure 3.5-9 that shows an estimate of the initial mass in low-Earth orbit (IMLEO) for a conceptual nuclear thermal propulsion system. Results for both the short-stay and long-stay mission approaches are provided. It must be noted that estimates for the payloads taken to Mars and back to Earth are different depending on the mission strategy employed. For instance, additional surface systems, such as a surface habitat, must be deployed to Mars for the long-stay mission, systems which are not required for the short-stay approach. Several important results can be seen from Figure 3.5-9 including:

- Very short, one-year, round trip missions can only be conducted in the 2018 opportunity, that is for reasonable initial mass in Low-Earth Orbit (IMLEO).
- There is very little mass change for all of the long-stay opportunities.

- A vehicle designed for the 2018 one-year mission can also be used for the hardest 2028 Venus swing-by opportunity.

Results from this quick parametric analysis allowed the exploration team to focus design efforts on a set of mission cases that encompass the range of missions that balance the various mission needs.

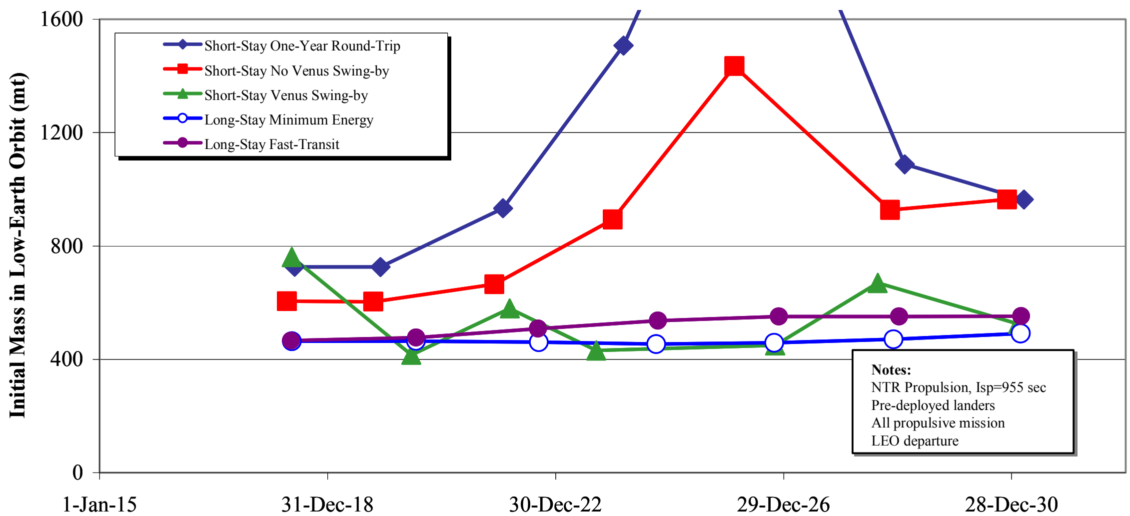


Figure 3.5-9: Example Parametric Mass Estimates for Various Mission Classes

Split Mission Strategy

Another useful mission approach that should be considered during the architecture design phase is the overall mission sequencing approach. The use of minimum energy trajectories to pre-deploy mission hardware is an excellent way of reducing overall mission mass. Sending cargo elements ahead of the crew allows those elements to utilize the more energy efficient trajectories, rather than forcing the cargo elements on the faster, energy intensive trajectories with the crew. With this approach, the cargo elements are injected toward Mars one opportunity, approximately 26 months, prior to the crew. These elements can then be in place, checked out and verified to be operational *before* the crew departs Earth. When making this design decision, other factors must be considered including: 1) the additional time that these pre-deployed systems must operate in the overall mission architecture, 2) the criticality of the mission assets in terms of crew safety, and 3) the overall affect that the pre-deployed elements have on other portions of the mission. For instance, pre-deploying the lander has no affect on the crew's ability to return from Mars orbit and thus pre-deploying it ahead of time is viewed be an acceptable strategy. On the other hand, pre-deployment of the crew's return propellant introduces an additional mission critical event, namely rendezvous in Mars orbit, that is mandatory in order to return the crew. In addition, with this strategy it must be noted that fewer powered abort options exist since the return propellant is pre-deployed separately from the crew. An example of the mass advantage of various pre-deployment strategies is shown in Figure 3.5-10 that shows a comparison of three different degrees of pre-deployment.

Pre-deployment strategies can also provide some operational advantages, depending on the overall mission strategy. Figure 3.5-11 provides an overview of the mission dependence between the pre-deployed cargo and piloted missions for both short and long-stay missions. There is a direct one-to-one correspondence between the cargo mission and piloted mission for the short stay approach. That is, each cargo mission supports only the next piloted mission because there is no overlapping of subsequent cargo missions and the piloted missions. As can be seen from the figure, the crew must depart Mars before the next cargo vehicle has arrived.

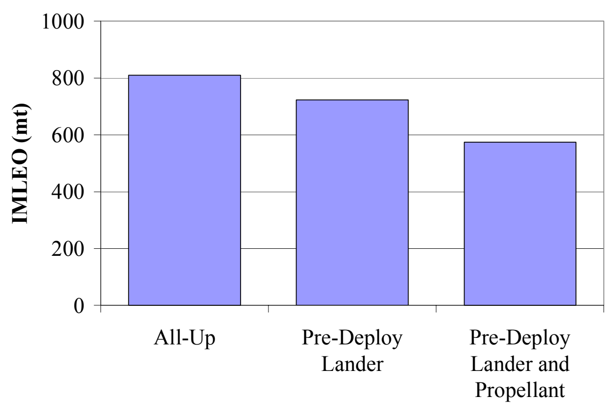


Figure 3.5-10: Pre-Deployment Advantages

On the other hand, for the long-stay mission approach, there is redundancy between the cargo missions and the piloted missions. As can be seen from Figure 3.5-11, since the crew is at Mars for a longer period, they have two distinct operational paths: 1) Primary – utilize the cargo elements sent the opportunity before the crew, or 2) Contingency – utilize the cargo elements sent during the same opportunity as the crew. This overlapping of mission resources for the long-stay mission approach can provide significant risk reduction techniques, thus enhancing the crew safety aspects of the mission.

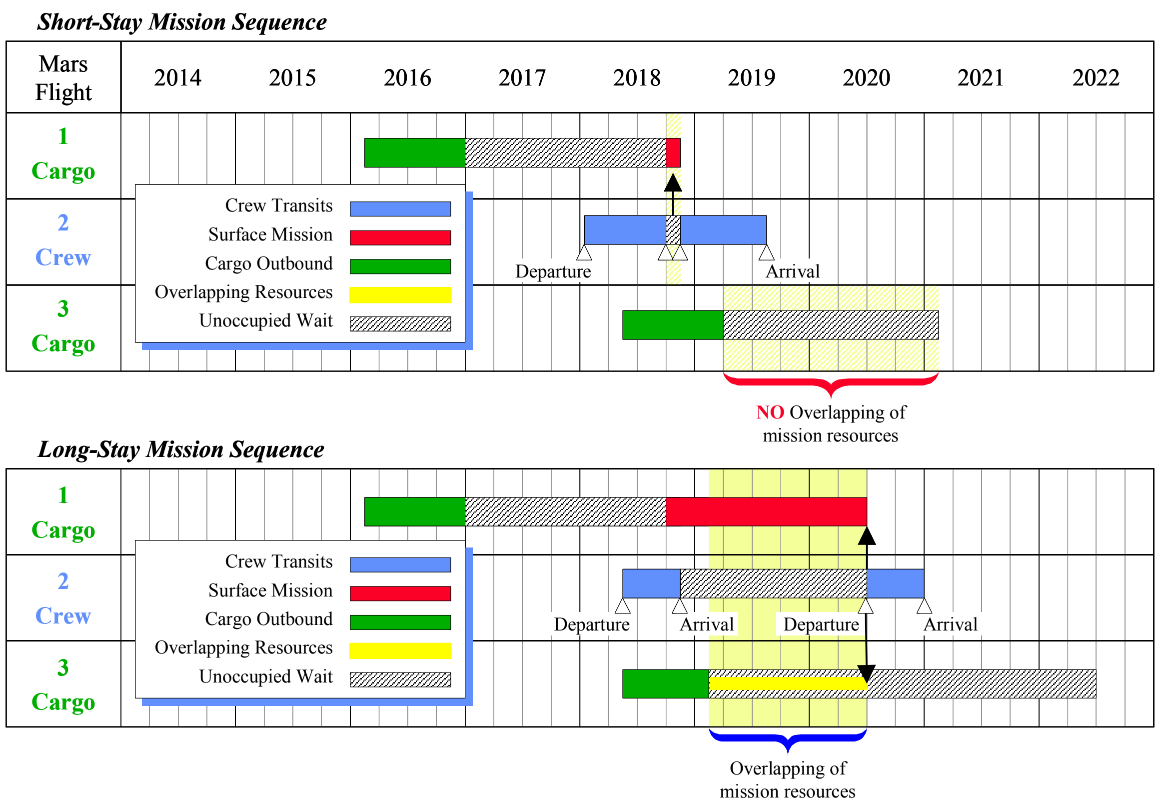


Figure 3.5-11: Example Pre-Deployment Mission Sequence

Incorporation of Advanced Technologies

Incorporation of advanced technologies can have a significant impact on the resulting total mission mass and overall mission approach. Techniques used for measuring the benefits of various technologies are not straightforward due to the inherent complexity of human exploration missions. Traditionally, total mission mass, specifically the mass savings from the application of a specific technology, has been used as the first order measure of benefit. Total mission mass does indeed provide the mission designer some degree of understanding of the resulting size of the mission architecture, but it also provides insight into other related figures of merit including mission complexity, cost, and to a certain level, mission risk. Although each of these other criteria is related to mass, specific comparisons of the merit of each technology toward these other criteria must be made.

Perhaps an equally important aspect of measuring the value of a technology is the mission toward which it is applied, specifically the combined technologies that are included. The measured value of a specific technology is highly dependent on the interrelationship of it in concert with other applied technologies. For instance, the value of an advanced technology as applied to a mission comprised of "today's" technologies will provide more mass reduction than applying that same technology to a mission comprised of "tomorrow's" technologies. In addition, the order in which the technologies are applied can change the measured savings. This technology bundling effect can be seen in Figure 3.5-12. As can be seen from this figure, the total savings from start to end is the same, but the relative savings between incremental steps in the process is different.

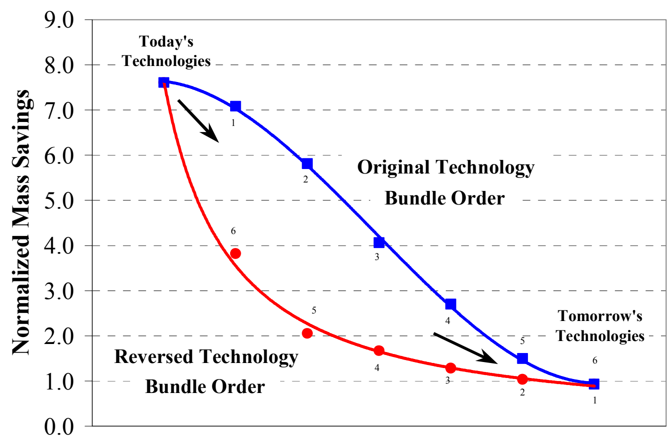


Figure 3.5-12: Technology Bundling Effect

Another important criterion to consider when evaluating specific technologies is the ancillary benefit that a technology can provide to the total system or mission architecture. This measure is often overlooked, but it can have a profound impact on the overall mission. One example of this is the incorporation of microminiaturized electronic technologies. Many electronic components are currently small and represent a small fraction of the overall mission mass, and thus further miniaturization will not have a significant impact on the resulting total mission mass. However, one must consider other important mission criteria, such as risk, when evaluating the value of the technology. Since the size of these components continues to decrease, the mission designer can begin to evaluate the risk reduction benefits of incorporating multiple redundancies in the systems designs. One can even begin to investigate the safety benefits of not just dual or triple redundancy, but redundancy strategies of 10's or even 100's of similar components.

3.5.3 Mission Options

The choice of the overall exploration mission sequence and corresponding trajectory strategy has perhaps the greatest single influence on the resulting architecture. The ideal mission would be one that: 1) provides the shortest overall mission in order to reduce the associated human health and reliability risks, 2) provides the most time on the surface, thus increasing the science return, and 3) provides low mission mass which in turn reduces the overall cost and mission complexity. Unfortunately the "ideal" mission does not exist, and tough choices must be made between design options. Table 3.5-1 provides a top-level comparison of how the two mission approaches, short and long stay, meet the driving mission design goals developed for this study. Further comparative information is provided in Table 3.5-2. With advanced propulsion concepts and acceptance of larger mission masses, the short-stay mission can provide short round-trip missions on the order of one-year, but only for very selective mission dates, namely 2018. Trip times for the more difficult opportunities can be as long as 660 days (22 months). In addition, the short-stay missions are characterized by large mission masses and limited time on the surface to conduct the required exploration activities. On the other hand, the long-stay mission eliminates many of the mission mass concerns by providing a non-varying, low mass approach, with ample time to conduct the surface exploration, but it is characterized by long total mission durations, up to 32 months long. The exploration team settled on two primary options for the first human exploration mission of Mars: Option 1) Short-stay approach for the first mission transitioning to the long-stay approach for subsequent missions, and Option 2) Conduct long-stay missions from the beginning.

Mission Goals	**Short Stay**	**Long Stay**
Balance risks	◐	●
Operationally simple mission	◐	◐
Flexible implementation strategy	○	●
Maximize human health & safety	◐	◐
Robust surface exploration	○	●
Short mission duration	●	○
Low mission mass	◐	●

○ Doesn't Meet Goal
◐ Moderately Meets Goal
● Meets Goal

Table 3.5-1: Design Goal Comparison

Parameter	Short-Stay Mission	Fast-Transit Long-Stay Mission
Mission Duration (days)	365-661	892-945
Surface Stay	30	501-596
One-Way Transits	104-357	134-210
Total Transit Time	335-631	296-413
Trajectory Characteristics	Venus Swing-by	No Venus Swing-by
Closest Approach to Sun	0.35 – 0.72 AU	1.0 AU
Human Health	Certification process of long zero-g space missions unknown. Crew exposure to surface environment minimized.	Mission transits within US zero-g spaceflight experience on MIR. Extended exposure of crew to surface environment.
Transportation	Advanced propulsion required for reasonable mass.	Advanced propulsion enhances missions (lower mass or transits).
Earth-to-Orbit	Large mission mass necessitates high launch rate and/or larger launch vehicle	Lower mission mass relieves launch requirement and launch rate.
System Reliability	12-22 months	30-32 months
Mission Focus	Transportation and propulsion.	Surface and mission return.

Table 3.5-2: Mars Mission Characteristics Comparison

3.5.3.1 Mission Sequence for the Short-to-Long-Stay Option

The focus of the short-to-long-stay mission option is to take advantage of the favorable mission opportunities of the synodic cycle, which occurs in 2018, in order to reduce the overall mission duration. As was mentioned earlier, a proper balance must be struck between the length of the overall mission and the total mass that must be launched. As the mission duration is shortened, the total mission mass grows exponentially. The philosophy behind the short-to-long stay transition approach is that as experience is gained by conducting the short-stay missions, a transition to the long-stay mission is made. An example of this transition is shown in Table 3.5-3 that shows the total mission durations for the various opportunities. As can be seen from this table, the total mission duration increases after the 2018 injection date.

	Short-Stay Mission Mission Times (Days)				Long-Stay Mission Mission Times (Days)			
Opportunity	Out	At Mars	Back	Total	Out	At Mars	Back	Total
2018	104	40	221	365	180	585	180	945
2020	242	40	164	446	180	556	200	936
2022	226	40	253	519	207	510	200	917
2024	316	40	244	600	203	501	210	914
2026	216	40	308	564	194	522	193	909
2028	274	40	231	545	184	548	170	902
2030	263	40	224	527	162	596	134	892

Table 3.5-3: Short-to-Long-Stay Mission Option Timeline

A split mission approach is used whereby mission cargo is delivered to Mars one opportunity before the crew. This provides a significant advantage in reducing the total mission mass. In fact, for the harder mission opportunities, pre-deployment of mission assets is required to obtain reasonable initial masses. An overview of this mission architecture is shown in Figure 3.5-13. As can be seen from this figure, the exploration team settled on the approach of pre-deploying both the descent/ascent vehicle and the crew Earth Return Vehicle. Each human mission to Mars is comprised of three vehicle sets, two cargo vehicles and one piloted vehicle. An overview of the mission manifest for the three vehicle sets is provided in Table 3.5-4.

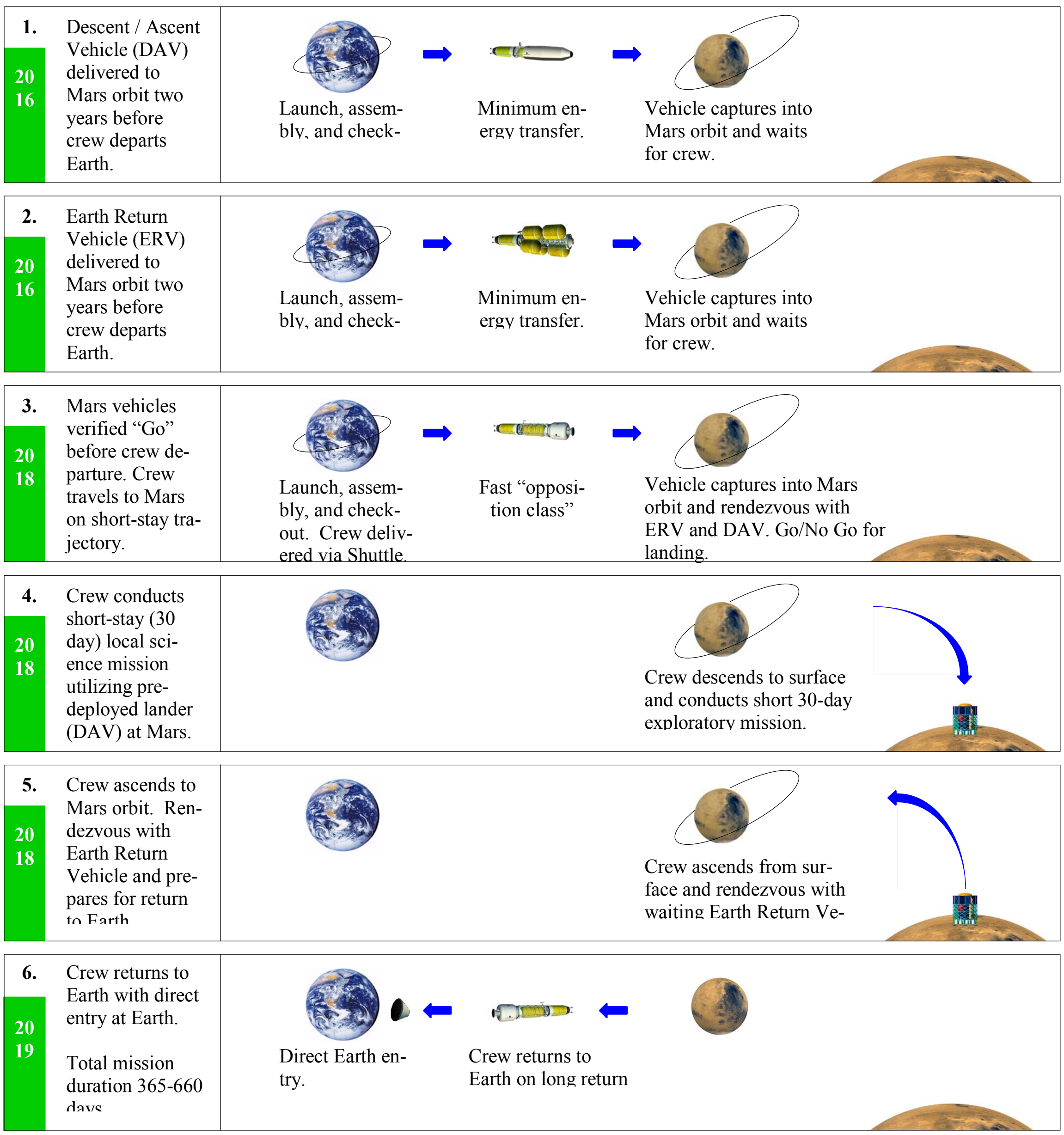
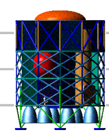
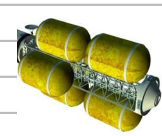


Figure 3.5-13: Mission Sequence for the Short-to-Long Stay Architecture.

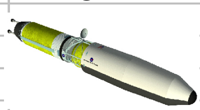
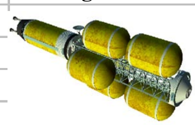
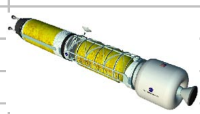
The first phase of the short-to-long-stay mission architecture begins with the pre-deployment of two cargo elements to Mars orbit which includes the Descent / Ascent Vehicle (DAV) and the Earth Return Vehicle (ERV). These two vehicle sets are launched, assembled, and checked out in low-Earth orbit. After all systems have been verified and are operational, the vehicles are injected into minimum energy transfers from Earth orbit to Mars. Upon arrival at Mars the vehicles are captured into a high-Mars orbit and remain in a semi-dormant mode, waiting for the arrival of the crew approximately 24 months later. Periodic vehicle checks and orbital maintenance are performed in order to place the vehicles in the proper orientation for crew arrival.

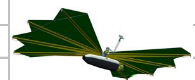
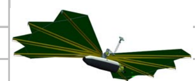
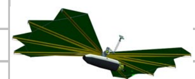
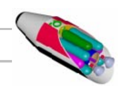
Flight	Payload	Current Best Mass Estimate (kg)	
1: Cargo	**Descent / Ascent Vehicle**	**72,100**	
	Ascent Vehicle	31,400	
	Mobility / Science	2,700	
	EVA Systems	10,600	
	Entry / Descent System	27,400	
2: Cargo	**Return Propellant**	**70,900**	*
	MOI/TEI Propulsion Stage	54,800	
	EOI Propulsion Stage	2,100	
	Aerobrake	10,000	
	Attitude Control System	4,000	
3: Crew	**Mars Transfer Vehicle**	**27,100**	*
	Transit Habitat	21,100	*
	EVA Systems	900	
	Cruise Science	600	
	Earth Crew Capsule	4,500	
	Contingency Consumables	0	

* System mass dependent on specific opportunity under consideration. Data shown for 2018.

Table 3.5-4: Short-to-Long Stay Mission Manifest

The specifics of the Earth departure and Mars arrival scenarios are dependent on the transportation technologies chosen. Both Nuclear Thermal Propulsion Rocket (NTR) propulsion and Solar Electric Propulsion (SEP) were considered by the exploration team to be leading technologies for the in-space transportation function. These technologies were believed to best balance technical risk, technology advancement, mission complexity, mission mass, and overall development and operational cost. The transportation mission manifest for both propulsion options is shown in Table 3.5-5. It must be noted that this table provides the mass estimates for the 2018 injection opportunity, which represents the minimum mass across the synodic cycle.

Flight		Mass Estimate
1: Cargo	**Descent / Ascent Vehicl**	**161,100** *
	Payload	72,100
	Stage	30,000
	Propellant	59,000
2: Cargo	**Earth Return Vehicle**	**212,000** *
	Payload	0
	Stage	54,000
	Propellant	158,000
3: Crew	**Crew Transit Vehicle**	**199,100** *
	Transit Habitat	27,100
	Stage	58,000
	Propellant	114,000
4: Crew Delivery	**Crew**	**558**
	Crew	558

Flight		Mass Estimate
1: Cargo	**Descent / Ascent Vehicl**	**148,100** *
	Payload	72,100
	SEP Vehicle	26,000
	Chemical Stages	4,000
	Propellant	46,000
2: Cargo	**Earth Return Vehicle**	**187,000** *
	Payload	0
	SEP Vehicle	26,000
	Chemical Stages	31,000
	Propellant	130,000
3: Crew	**Crew Transit Vehicle**	**120,100** *
	Transit Habitat	27,100
	SEP Vehicle	26,000
	Chemical Stages	10,000
	Propellant	57,000
4: Crew Delivery	**Crew Taxi Vehicle**	**25,000**
	Crew Taxi	11,000
	Stage	2,000
	Propellant	12,000

* System mass dependent on specific opportunity under consideration. Data shown for 2018.

Table 3.5-5: Short-to-Long Stay Transportation System Manifest

The second phase of this architecture begins with the launch, assembly, and checkout of the Mars Transfer Vehicle during the next injection opportunity. The Mars Transfer Vehicle serves as the interplanetary support vehicle for the crew as well as the outbound transportation system. A vehicle checkout crew is delivered to the Mars Transfer Vehicle in Earth orbit to perform vital systems verification and any necessary repairs prior to departure of the flight crew. After all vehicles and systems, including the Mars Descent / Ascent Vehicle, the Earth Return Vehicle, and the Mars Transfer Vehicle are verified operational, the flight crew is injected on the appropriate short-stay trajectory. The length of the outbound transfer to Mars is dependent on the injection opportunity, and ranges from 104-316 days. Upon arrival at Mars, the crew must rendezvous with both the Earth Return Vehicle and Descent / Ascent Vehicle. A proper rendezvous with the Earth Return Vehicle is mission critical since this vehicle contains all of the return propellant for the crew. After arriving at Mars, the crew has up to forty days to make all of the necessary orbital adjustments for the return trajectory and conduct the surface mission. During this period the transit habitat is transferred from the Mars Transfer Vehicle to the waiting Earth Return Vehicle.

The Descent / Ascent Vehicle serves as the primary transportation and crew support element for the planetary exploration phase of the mission. The vehicle is designed to transport the mission crew from a high Mars orbit to the surface of Mars, support the crew for up to 30 days while on the surface, and return the crew from the surface to the high Mars orbit whereby it performs a rendezvous with the Mars Transfer Vehicle. The functional capabilities of the Descent / Ascent Vehicle must accommodate the ability to operate in a fully automated mode since it is anticipated that the crew will not be capable of performing complicated tasks due to the long exposure to micro-gravity while in transit. Vehicle terminal phase targeting/control, post-landing safing, initial flight-to-surface transition, and appendage deployments must occur without crew exertion. Thus, the vehicle must provide adequate time for the crew to re-adapt to 0.38 G on Mars. During this period, no strenuous activities (e.g., EVA) will be scheduled for any crewmembers and the focus of the operations will be on developing adequate crew mobility and maintaining systems operability.

The focus of the surface exploration phase is to conduct scientific investigations of the local landing vicinity. Of the 30 days on the surface of Mars, up 21 potential Extra-Vehicular Activity (EVA) sorties can be conducted. This strategy provides time for the crew to acclimate to the Martian environment as well as perform the closeout and vehicle checks necessary at the end of the surface mission prior to ascending back to orbit. During the science investigations, a ten-kilometer radius has been established as a reasonable traverse radius about the landing zone. This radius is derived from the maximum unassisted walk-back distance of a suited crewmember due to rover failure. This radius also considers the rate life support consumables within the EVA system are depleted while returning from that distance.

After completion of the surface mission, the crew performs the necessary closeout and shutdown operations of the vehicles. Surface elements, including science instruments are placed in an automated operations mode for earth-based control. The crew then ascends in the Descent / Ascent Vehicle and performs a rendezvous with the waiting Earth Return Vehicle. This vehicle is used to return the crew from Mars, ending with a direct entry at Earth.

Short-to-Long-Stay Architecture Concerns

- Human health and safety: 365-660 days in deep space radiation environment
- Operational risks: 40 days for surface ops, contingency planning, dust storms
- Limited return: only 4-8% of mission time on surface
- High-degree of sensitivity and variability including transition issues to long-stay (different vehicles)

3.5.3.2 Mission Sequence for the Long-Stay Option

The philosophy of the long-stay mission architecture approach is to minimize the exposure of the crew to the deep space radiation and zero gravity environment while at the same time maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of the planets for both the outbound and return trajectories by varying the stay time on Mars, rather than forcing the mission through non-optimal trajectories as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order to six-months, while allowing them to stay on the surface of Mars for a majority of the mission, on the order of eighteen months.

The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface or Mars orbit and checked out in advance of committing the crews to their mission. The split mission approach also allows the crew to be transported on faster, more energetic trajectories, minimizing their exposure to the deep-space environment, while the vast majority of the material sent to Mars is sent on minimum energy trajectories. The trajectory analysis discussed earlier was used to insure that the design of the space transportation systems could be flown in any opportunity. This is vital in order to minimize the programmatic risks associated with funding profiles, technology development, and system design and verification programs. An overview of this mission architecture is shown in Figure 3.5-14. As can be seen from this figure, each human mission to Mars is comprised of three vehicle sets, two cargo vehicles and one round-trip piloted vehicle. An overview of the manifest for the three vehicles is provided in Table 3.5-6. The transportation manifest for both the Nuclear Thermal Rocket and Solar Electric Propulsion transportation system concepts is provided in Table 3.5-7. It must be noted that the objective of the long-stay mission architecture is to design the systems to operate in all mission opportunities to Mars, thus the mission masses provided in this table represent the worst case mass throughout the synodic cycle.

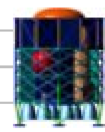
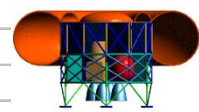
Flight	Payload	Current Best Mass Estimate (kg)
1: Cargo	**Descent / Ascent Vehicle**	**72,100**
	Ascent Vehicle	31,400
	Mobility / Science	2,700
	Crew Support Systems	10,600
	Entry / Descent System	27,400
2: Cargo	**Surface Habitat**	**58,200**
	Surface Habitat	15,000
	Power System	6,000
	Consumables	6,800
	EVA & Mobility Systems	2,500
	Regional Science Lab	900
	Entry / Descent System	27,000
3: Crew	**Mars Transfer Vehicle**	**38,000**
	Transit Habitat	24,700
	EVA Systems	900
	Cruise Science	600
	Earth Crew Capsule	4,500
	Contingency Consumables	7,300

Table 3.5-6: Long-Stay Mission Manifest

The first phase of the long-stay mission architecture begins with the pre-deployment of the first two cargo elements, the Descent / Ascent Vehicle and the Surface Habitat. These two vehicle sets are launched, assembled, and checked out in low-Earth orbit. After all systems have been verified and are operational, the vehicles are injected into minimum energy transfers from Earth orbit to Mars. Upon arrival at Mars the vehicles are captured into a high-Mars orbit. The specifics of the Earth departure and Mars arrival scenarios are dependent on the transportation technologies chosen. The Descent / Ascent Vehicle (DAV) remains in Mars orbit in a semi-dormant mode, waiting for arrival of the crew two years later. The Surface Habitat (SHAB) is captured into a temporary Mars orbit, and then performs the entry, descent, and landing on the surface of Mars at the desired landing site. After landing the vehicle is remotely deployed, checked out, and all systems verified to be operational. Periodic vehicle checks and remote maintenance are performed in order to place the vehicles in proper orientation prior to crew arrival.

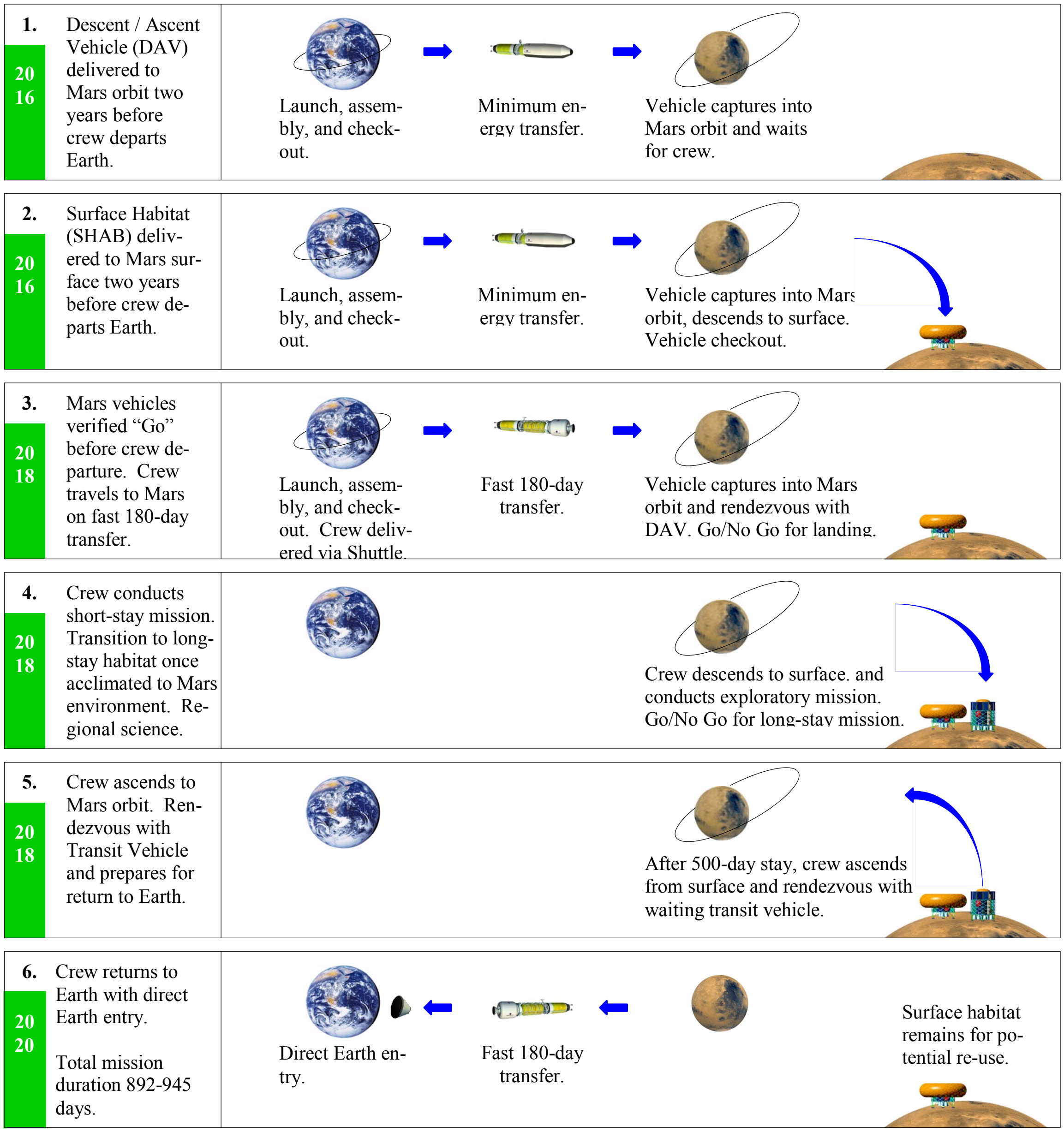
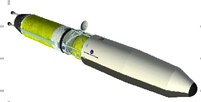
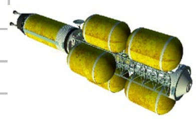
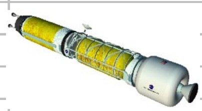
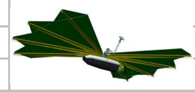
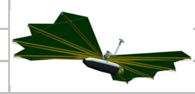
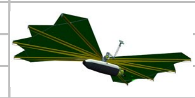
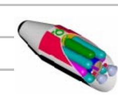


Figure 3.5-14: Mission Sequence for the Long-Stay Architecture

A key feature of the long-stay mission architectures is the deployment of significant portions of the surface infrastructure before the human crew arrives. This strategy includes the capability for these infrastructure elements to be unloaded, moved significant distances, connected to each other, and operated for significant periods of time without humans present. In fact, the successful completion of these various activities will be part of the decision criteria for launch of the first crew from Earth.[xx] Pre-deployed and operated surface elements include the surface habitat, power system, thermal control system, communications system, robotic vehicles, and navigation infrastructure.

Flight		Mass Estimate
1: Cargo	Descent / Ascent Vehicl	163,100 *
	Payload	72,100
	Stage	30,000
	Propellant	61,000
2: Cargo	Surface Habitat	140,200 *
	Payload	58,200
	Stage	30,000
	Propellant	52,000
3: Crew	Crew Transit Vehicle	163,000 *
	Transit Habitat	38,000
	Stage	39,000
	Propellant	86,000
4: Crew Delivery	Crew	558
	Crew	558

Flight		Mass Estimate
1: Cargo	Descent / Ascent Vehicl	149,100 *
	Payload	72,100
	SEP Vehicle	26,000
	Chemical Stages	4,000
	Propellant	47,000
2: Cargo	Surface Habitat	130,200 *
	Payload	58,200
	SEP Vehicle	26,000
	Chemical Stages	4,000
	Propellant	42,000
3: Crew	Crew Transit Vehicle	132,000 *
	Transit Habitat	38,000
	SEP Vehicle	26,000
	Chemical Stages	10,000
	Propellant	58,000
4: Crew Delivery	Crew Taxi Vehicle	25,000
	Crew Taxi	11,000
	Stage	2,000
	Propellant	12,000

* System mass dependent on specific opportunity under consideration. Data shown for 2018.

Table 3.5-7: Long-Stay Transportation System Manifest

The second phase of this architecture begins during the next injection opportunity with the launch, assembly, and checkout of the Mars Transfer Vehicle. The Mars Transfer Vehicle serves as the interplanetary support vehicle for the crew for a round-trip mission to Mars orbit and back to Earth. Prior to departure of the flight crew, a separate checkout crew is delivered to the Mars Transfer Vehicle to perform vital systems verification and any necessary repairs prior to departure of the flight crew. After all vehicles and systems, including the Mars Descent / Ascent Vehicle, Surface Habitat, and the Mars Transfer Vehicle are verified operational, the flight crew is injected on the appropriate fast-transit trajectory towards Mars. The length of this outbound transfer to Mars is trajectory dependent, and ranges from 180-205 days. Since the crew is delivered to Mars on their round-trip vehicle including the return propellant, the crew does not have to perform any rendezvous or other complicated orbital maneuvers in order to return from Mars back to Earth. Upon arrival at Mars, the crew performs a rendezvous with the Mars Descent / Ascent Vehicle, which serves as their transportation leg to and from the Mars surface. After arriving at Mars the crew as ample time, up to eighteen months, to make all of the necessary orbital adjustments for the return trajectory and conduct the surface mission.

The Descent / Ascent Vehicle serves as the primary transportation element for the crew in the vicinity of Mars. The vehicle is designed to transport the mission crew from a high Mars orbit to the surface of Mars, support the crew for the initial post-landing acclimation period, up to 30 days, and return the crew from the surface to the high Mars orbit whereby it performs a rendezvous with the Mars Transfer Vehicle. The functional capabilities of the Descent / Ascent Vehicle must accommodate the ability to operate in a fully automated mode since it is anticipated that the crew will not be capable of performing complicated tasks due to the long exposure to microgravity while in transit. Vehicle terminal phase targeting/control, post-landing safing, initial flight-to-surface transition, and appendage deployments must occur without crew exertion. Thus, the vehicle must provide adequate time for the crew to re-adapt to 0.38 G on Mars. During this period, no strenuous activities (e.g., EVA) will be scheduled for any crewmembers and the focus of the operations will be on developing adequate crew mobility and maintaining systems operability.

Current human health and support data indicates that it may take the crew up to one week to acclimate to the partial gravity of Mars. After the crew has acclimated, the focus of the initial surface activities is on transitioning from the lander to the surface habitat. This includes performing all remaining setup, checkout, and maintenance that could not be performed remotely from Earth. The crew has up to 30 days after landing to perform all necessary startup activities of the surface habitat. During this period local science is also conducted to insure that the initial science objectives can be met if early ascent from the surface is required. Lastly, the lander is connected to the surface habitat power system and placed in a semi-dormant mode since it will not be needed again until ascent from the surface is required. Although the lander is in a semi-dormant mode, emergency abort-to-orbit is available throughout the surface exploration phase of the mission.

The long-stay mission architecture lends itself to a very robust surface exploration strategy. The crew has approximately eighteen months to perform the necessary surface exploration activities and thus the strategy follows a less rigorous, less scheduled approach. Ample time is provided to plan and re-plan the surface activities, respond to problems, and readdress the scientific questions posed early in the mission. The focus during this phase of the mission will be on the primary science and exploration activities that will change over time to accommodate early discoveries. A general outline of crew activities for this time period will be provided before launch and updated during the interplanetary cruise phase. This outline will contain detailed activities to ensure initial crew safety, make basic assumptions as to initial science activities, schedule periodic vehicle and system checkouts, and plan for a certain number of sorties. Much of the detailed activity planning while on the surface will be based on initial findings and therefore cannot be accomplished before landing on Mars. The crew will play a vital role in planning specific activities as derived from more general objectives defined by colleagues on Earth.

Before committing the crew to Mars ascent, full systems checkout of the ascent vehicle and the Mars Transit Vehicle is required. Because both vehicles are critical to crew survival, sufficient time must be provided prior to launch to verify systems and troubleshoot any anomalous indications prior to crew use. In addition, the surface habitat will be placed in a dormant mode for potential re-use by future crews. This includes stowing any nonessential hardware, safing critical systems and their backups, and performing general housekeeping duties. Lastly, surface elements, including science instruments are placed in an automated operations mode for earth-based control. The crew then ascends in the Descent / Ascent Vehicle and performs a rendezvous with the waiting Mars Transfer Vehicle. This vehicle is used to return the crew from Mars, ending with a direct entry at Earth.

<u>Long-Stay Mission Architecture Concerns</u>

- Long mission
- Long exposure to 3/8 g

3.5.3.3 Artificial-gravity Mission Option

To be supplied. This section will describe current mission studies pertaining to the artificial-gravity Nuclear-Electric Propulsion concept. Since this is work in progress, it will deal with the mission concept and results to date.

3.5.4 Mission Requirements

Description	Requirement	Rationale
Mars exploration mission support	The system shall be capable of supporting human and robotic exploration of planetary surfaces such as Mars.	Ambitious human deep-space missions, including missions to Mars, are called out in the NASA Human Exploration and Development of Space Strategic Plan and as a part of the NASA Exploration Team's (NEXT) goals.
Total mass to Low Earth Orbit (LEO) for Mars Exploration Missions	The system shall be capable of delivering a total of 450 metric tons to low-Earth orbit.	NASA has conducted numerous human Mars mission studies incorporating a variety of mission goals, mission durations, and differing assumptions in terms of advanced technologies. Results from these studies indicate that the total mission mass of a single human mission to Mars can range from 400 metric tons (using a suite of advanced technologies) up to 1400 metric tons (utilizing more conservative technologies). Current Exploration Office analysis in support of the NASA Exploration Team indicates approximately 450 metric tons of initial mass in LEO.
Delivery of Mars exploration mission crew	The system shall be capable of delivering a minimum of 7 exploration mission crew to low-Earth orbit (407 km circular, 28.5 deg.).	Current mission planning calls for a minimum of seven exploration crewmembers to perform the missions included in the Mars exploration design reference mission.

Mars Exploration mission rate	The system shall support a minimum of 1 exploration mission every 20 months.	Injection opportunities to Mars occur every 26 months. In order to meet reasonable operations timelines, all exploration payloads must be launched with a reasonable time for vehicle and system checkout prior to the opening of the injection window.
Payload mass to LEO for Mars Exploration Missions	The system shall be capable of delivering a minimum of 100 metric tons per launch to low-Earth orbit (407 km circular, 28.5 deg.).	A wide range of launch package masses for human exploration missions of Mars has been studied in the past. Payloads of 100 metric tons represent a good balance between required size of the payload and the number of launches required. Package sizes in the range of current launch capabilities (20 metric tons) show significant disadvantages including: 1) Significant mass efficiency losses are incurred due to non-optimal packaging. ISS experience indicates a 70% utilization efficiency, 2) Design inefficiencies increase with the number of launches due to increased number of interfaces and additional functional requirements (bulkheads, docking mechanisms, plumbing, etc.), 3) Probability of mission success (launch) is significantly decreased with increasing number of launches, and 4) Significant increase in the level of on-orbit assembly required for vehicle and systems including aerobrakes and aeroentry shields, 5) The number of launches required for a single human Mars mission using only 20 metric ton launch packages would require a number of launches roughly equivalent to that required to assemble the Phase III International Space Station. This does not represent a feasible approach.

Payload volume to LEO for Mars Exploration Missions	The system shall be capable of delivering payloads with minimum volumetric dimensions of 8 m x 30 m length per launch.	Numerous vehicle configurations for human Mars missions have been studied including incorporating advanced technologies such as inflatable structures. A major driver for these missions continues to be those associated with the Mars entry phase. Due to the atmosphere of Mars, all surface elements and landing vehicles must be designed to fit within the envelope of an entry shield. Assembly of the entry shield in Earth orbit is an operationally difficult task and represents a significant risk. The launch vehicle shroud dimension of 8x30 meters allows the entry shield to be assembled on the Earth and launched intact.
Launch shroud serves as aeroshield	The system shall not preclude the utilization of the launch vehicle payload shroud as an aeroentry shield - Option	Due to the atmosphere of Mars, all surface elements and landing vehicles must be designed to fit within the envelope of an entry shield. In addition, mission studies have shown that aerocapturing into Mars orbit, rather than propulsively capturing, can reduce the total mission mass by up to 45%. Aero entry is an enabling technology for human missions to Mars. One option is to utilize the launch vehicle payload shroud as the entry shield, thus eliminating the complicated tasks associated with on-orbit assembly and checkout of small individual spacecraft and aero shield components.
Reliability goal for Mars Exploration Mission Launches	The system shall provide an overall payload delivery reliability of at least 99.7%.	The probability of total mission success is directly related to the launch vehicle reliability. Given the current worldwide launch vehicle reliability history, the probability of launch success for current launch capabilities would range from 20-40%. A launch vehicle system reliability approaching that of the Shuttle (in excess of 99%) is required to maintain a total launch success probability of 90% or greater.

Launch of spacecraft containing nuclear materials	The system shall be capable of safely launching spacecraft containing nuclear power and propulsion systems.	Exploration of planetary surfaces, deep-space, and near-solar destinations has been established as specific goals within both the Space Science and Human Exploration and Development of Space Enterprises. Advanced space power, for both power generation and advanced propulsion, has been identified by the Agency as an important technology gap that must be closed for these mission concepts. Nuclear space power has been identified as the most promising concept for filling that gap in the near future. The specific nuclear requirements for the various mission concepts currently under investigation include both fission and radioisotope concepts. The Agency has established a Nuclear Systems Initiative (NSI) in order to take positive steps towards filling this technology gap. Many Space Science missions have already identified the need for nuclear space power for mission concepts and many more are expected as a result of the establishment of the NSI.

Table 3.5-8: Mars Exploration Mission Requirements

3.5.5 Key Technology Investments

Technology	Summary Description	Current TRL	Additional Applications

Table 3.5-9: Mars Exploration Key Technology Investments

3.5.6 References

POC: Bret Drake/NASA JSC

[i] NASA, "Report of the 90-Day Study on Human Exploration of the Moon and Mars," NASA-JSC, 1989.

ii Hoffman, Stephen J, and Kaplan, David I., “Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team,” NASA Special Publication SP-6107, Government Printing Office, Washington, D.C., 1987

iii Drake, Bret G., “Reference Mission 3.0, Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team,” Addendum to NASA Special Publication 6107, NASA/SP-6107-ADD, June, 1998.

iv Weaver, David B., and Duke, Michael B., “Mars Exploration Strategies: A Reference Program and Comparison of Alternative Architectures,” AIAA 93-4212. AIAA Space Programs and Technology Conference, September 21-23, 1993.

, v Stafford, Thomas P. “America at the Threshold, Report of the Synthesis Group on America’s Space Exploration Initiative,” Government Printing Office, Washington, D.C., 1996.

vi NASA, “The NASA Strategic Plan,” 1992, 1994, 1996, 1998.

vii NASA, “NASA’s Enterprise for the Human Exploration and Development of Space, The Strategic Plan,” January, 1996.

viii NASA, “Human Exploration and Development of Space Strategic Plan,” 2000.

ix NASA, “Space Science Strategic Plan,” November, 2000.

x Soldner, J.K., “Round-Trip Miars trajectories – New Variations on Classic Mission Profiles,” AIAA Paper No. 90-2932, *AIAA/AAS Astrodynamics Conference*, Portland, OR, August 20-22, 1990.

xi Wilson, J.W., Townsend, L.W., Nealy, J.E., Chun, S.Y., Hong, B.S., Buck, W.W., Lamin, S.L., Ganapol, B.D., Kahy, F., and Cucinotta, F.A., “BYNTRN: A Baryon Transport Model,” NASA TP-2887, 1989.

xii Wilson, J.W., Miller, J., Konradi, A., Cucinotta, F.A., “Shielding Strategies for Human Space Exploration,” ANSA Conference Publication 3360, December 1997.

xiii Townsend, L.W. Nealy, J.E., Wilson, J.W., Simonsen, L.C., “Estimates of Galactic Cosmic Ray Shielding Requirements During Solar Minimum,” NASA TM-4167, 1990.

xiv Simonsen, L.C., Nealy, J.E., Townsend, L.W., Wilson, J.W., “Radiation Exposure for Manned Mars Surface Missions,” NASA TP-2979, 1990.

xv Cucinotta, Francis A., et al, “Space Radiation Cancer Risks and Uncertainties for Mars Missions,” Radiation Research 156, 2001, pgs 682-688.

xvi “Radiation Protection Guidance to Federal Agencies for Occupational Exposure: Recommendations Approved by the President,” Federal Register Vol, 52, No. 17, January 1987.

xvii “Guidance on Radiation Received in Space Activities,” NCRP Report No. 89, July 31, 1989.

xviii “Space Transfer Concepts and Analysis for Exploration Missions,” NASA Contract NAS8-37857, Boeing Defense and Space Group Advanced Civil Space Systems, March 1991.

xix Railsback, J.W., Simion, G.P., “A Probabilistic Approach of Incorporating Safety and Reliability in System Designs for a Manned Mission to Mars,” Proceedings of the International Topical Meeting on Probabilistic Safety Assessment PSA '99, Risk-Informed, and Performance-Based Regulation in the New Millennium, published by the American Nuclear Society, Inc. La Grange Park, Illinois, 1999.

[xx] Hoffman, Stephen J., "The Mars Surface Reference Mission: A Description of Human and Robotic Surface Activities - Draft," NASA/TP 199-209371, August, 1999.

3.6 Sun-Earth Connection Solar Sentinel Mission

We live in the extended atmosphere of an active star. While sunlight enables and sustains life, the Sun's variability produces streams of high-energy particles and radiation that can affect life.

Under the protective shield of a magnetic field and atmosphere, the Earth is an island in the Universe where life has developed and flourished. The origins and fate of life on Earth are intimately connected to the way the Earth responds to the Sun's variations. Understanding the changing Sun and its effects on the Solar System, life, and society is the goal of the Sun-Earth Connection Theme.

The Solar Sentinel mission will carry out scientific observations and research that will enable and improve space weather predictions. Living with a Star (LWS) scientists initially considered a complement of spacecraft for the Sentinels mission that included a spacecraft observing the far side of the Sun, a cluster of satellites at L1, and two next-generation STEREO spacecraft. Both remote images of the Sun and in-situ measurements would be taken. After much deliberation, the LWS science pre-definition team endorsed a four satellite inner heliospheric constellation with a suite of in-situ instruments and a solar far side observer with both remote and in-situ type instruments.

3.6.1 Connection to NEXT Themes and Goals

One of the three grand challenges identified by the NASA Exploration Team for future exploration of the solar system and beyond is to study how life developed on Earth and how the Earth and the solar system evolved together – the "How did we get here?" science pursuit. We seek to understand how and why the Sun varies, and how Earth responds to these changes. Knowledge of the dynamics and evolution of our own star, and the impact to present-day life on Earth, may provides clues to the origin of life on Earth and possibility for life on other planets and in extra-Solar systems.

Our Sun is an active star. This activity impacts Earth and human society in numerous ways. Terrestrial climate, ozone concentrations in the stratosphere, and atmospheric drag on satellites all respond to variations in the Sun's radiative output. Astronauts, airline passengers, and satellite electronics are all imperiled by the energetic particles produced in solar flares and coronal mass ejections (CMEs). Electrical power to our homes and businesses, communications, and navigation systems can all be interrupted by geomagnetic storms driven by blasts in the solar wind.[15] Solar Sentinels will study the structure and long-term (solar cycle and much longer) climatic variations of the ambient solar wind in the inner heliosphere. This mission seeks to determine how large-scale solar wind structures propagate and evolve in the inner heliosphere. Sentinels will also examine what dynamic processes are responsible for the release of geoeffective events in order to quantify the Sun's influence on global change and improve our characterizations and forecasts of space weather.

3.6.2 Mission Description

[15] "Solar Dynamics Observatory: Report of the Science Definition Team". NASA GSFC, October 2001.

The heliosphere is very inhomogeneous. Therefore, the propagation of transients is not uniform. To improve accuracy of space weather predictions, we need to know the environment through which the disturbances propagate.

LWS scientists have developed several primary scientific objectives for the Solar Sentinel design reference mission, as outlined below.

- Determine the structure and long-term (solar cycle and much longer) climatic variations of the ambient solar wind in the inner heliosphere
- Determine how large-scale solar wind structures propagate and evolve in the inner helio-sphere
- Determine what dynamic processes are responsible for the release of geoeffective events
- Determine how and where are energetic particles released and accelerated

Table 3.6-1: Solar Sentinel Mission Primary Science Objectives

Science Objectives	Space Weather Application	Measurement Requirement	Location
What is the ambient 3D structure of the heliosphere near the ecliptic?	Increase accuracy of forecasts Input to climatology models Radial profile	Solar wind plasma and composition	From various radial and longitudinal vantage points in the inner heliosphere
		Vector magnetic field	
		Energetic particles	
		Remote sensing of heliosphere	
How do large structures evolve during transit to Earth? (CMEs, Shocks, Fast Streams)	Increase accuracy of forecasts Input to models Radial profile	Solar wind plasma and composition	From various radial and longitudinal vantage points in the inner heliosphere
		Vector magnetic field	
		Radio burst tracker	
		Remote sensing of heliosphere	
		Remote sensing of photosphere/corona	Both sides of the Sun
What are the dynamic processes in the corona as can be determined from heliospheric observations?	Identify source regions and mechanisms hence provide forecasting capability.	Solar wind plasma and composition	From various radial and longitudinal vantage points in the inner heliosphere
		Vector magnetic field	
		Energetic particles	
		Remote sensing of photosphere/corona	Both sides of the Sun
How and where are energetic particles released and accelerated?	Develop SEP forecasting capability.	Energetic particles	From various radial and longitudinal vantage points in the inner heliosphere
		Radio burst tracker	
		Vector magnetic field	
		Remote sensing of photosphere/corona	Both sides of the Sun

These science objectives provide a link to the basic mission objective of improving the accuracy of space weather predictions, and can be used to derive in-situ and remote sensing measurements required to achieve such objectives. This information was then applied to determine locations in the solar system to operate solar sentinel spacecraft (see Table 3.6-1). LWS scientists concluded that a constellation of four inner heliospheric satellites and one solar far side observer would be required. In order to observe the development of regions of activity we must also observe the solar far side.

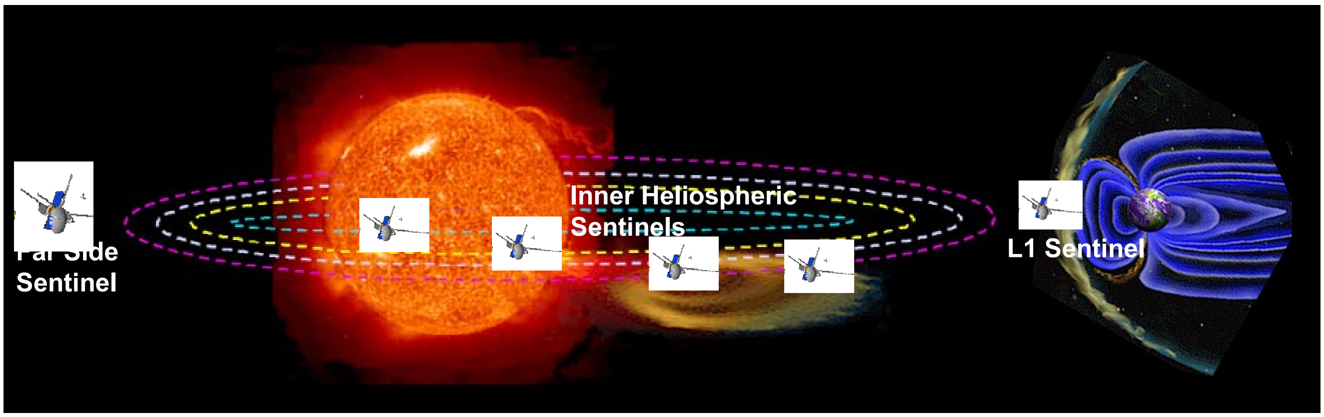


Figure 3.6-1: Sentinel Elements

3.6.2.1 Inner Heliospheric Sentinels

The Inner Heliospheric Sentinels (IHS) illustrated in Figure 3.6-1 will provide continuous observations of the heliosphere in order to characterize the environment through which solar transients propagate as well as track the full evolution of space weather disturbances to improve forecasting. The IHS mission relies on a small constellation of spinning spacecraft in concentric elliptical orbits at various distances from the Sun to achieve mission objectives. The constellation features four identical spacecraft spinning at 20 RPM normal to the orbit plane, each with a propulsion system for final orbit spacing. Each spacecraft carries a complement of in-situ instruments that will assist in the global characterization of the ambient structure of the inner heliosphere and will follow the evolution of large-scale features. Scientific knowledge gained from this mission will improve lead-time and accuracy of space weather forecasts.

<u>Science Instruments</u>

The baseline IHS instrument complement for each spacecraft consists of the four instruments listed below and their associated electronics:

- A three-axis magnetometer mounted on a 3-meter boom
- A solar wind analyzer on a 1-meter boom

- An energetic particle detector
- A radio waves experiment utilizing two 12-meter wire antennas

Instrument system parameters, shown in the table, are based on direct heritage from the WIND, ACE, STEREO, and MESSENGER missions.

Type/Classification	Size	Mass	Power	Data Rate
	LWH or DH		Avg/Peak	Avg/Peak
	(cm)	(kg)	(W)	(kbps)
Magnetometer*	5x5x10	5	2	0.1
Solar Wind Analyzer*	10x10x30	4.5	5	1.5
Energetic Particle Detector	10x10x15	4	3.5	0.2
Radio Waves Instrument	20x20x20	5.4	5	1.0
Digital Processing Unit	25x25x20	10	10	
Total		**28.9**	**25.5**	**2.8**
* Includes mass for boom				

Table 3.6-2: IHS Instrument Resource Accommodations

<u>Mission Profile</u>

The following serial time spans are assumed for mission planning:

- Extended pre-formulation and formulation phase for instrument, spacecraft, and ground system accommodation studies to match the available funding profile
- 12 months for conclusion of project formulation and definitization prior to approval years from approval to launch readiness
- Launch in December, 2008
- 4.5-month transit to final orbits
- 3 years for baseline mission operations including transit period

- 2-year mission extension (option for evaluation)

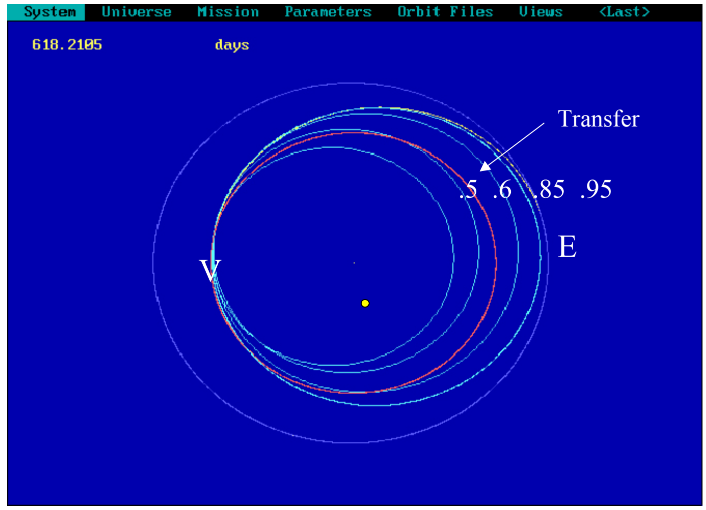


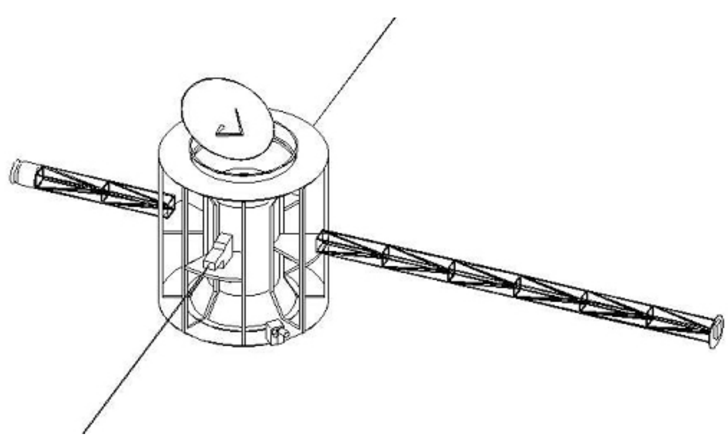
Figure 3.6-2: IHS Orbit Pictorial

The spacecraft are delivered to orbit by a single launch vehicle with the required performance for a C_3 of 10 km^2/sec^2. There is a 40-day launch window available for this C_3 value. The solar sentinel spacecraft will be launched into four elliptical heliocentric orbits (see Figure 3.6-2) at various distances from the Sun, all approximately in the plane of the ecliptic. A Venusian Gravity Assist (VGA) is employed to disperse the spacecraft orbits and an on-board propulsion system is used to provide the final orbit spacing. Launch vehicle options include the Pegasus XL and some Delta II configurations. The former requires four launches to accomplish the mission, while the latter (Delta II 7925H) with a stretched fairing may allow a single launch.

IHS Spacecraft

IHS spacecraft features include the following. As previously mentioned, four nearly identical spacecraft comprise the Inner Heliospheric Sentinels constellation, each spin-stabilized at 20 RPM with external solar array and radiator surface areas sized in accordance with its orbit proximity to the Sun. A mission-unique spacecraft design that has some commonality with the RBM and IM flight elements has been baselined. Also assumed in the design are deployable instrument booms and wire antennas, body-mounted solar arrays, and a Ka-band high gain antenna system on a despun platform for downlink of science data. A preliminary concept for the IHS spacecraft may be seen in Figure 3.6-3.

Figure 3.6-3: IHS Spacecraft Concept



3.6.2.2 Far Side Sentinel

The Far Side Sentinel (FSS) shown above in Figure 3.6-1 provides observation of activities on the far side of the Sun in order to track the full evolution of space weather disturbances and enable improved space weather forecasting. The FSS mission employs a three-axis stabilized spacecraft and its complement of remote sensing and in-situ instruments to make continuous solar observations of the far side of the Sun. These instruments may include a Doppler magnetograph, a EUV imager, a magnetometer, solar wind plasma and energetic particle detectors, and possibly others. Specific mission objectives for the Far Side Sentinel include the following:

- Search for disturbances forming in the Sun's convection zone and rising to the photosphere
- Track the evolution of these disturbances to enable daily/weekly space weather predictions
- Characterize coronal mass ejections (CMEs)
- Measure coronal magnetic fields
- Obtain global magnetic boundary conditions to model the three- dimensional structure of the heliosphere

Table 3.6-3: FSS Instrument Resource Accommodations

Type/Classification	**Mass**	**Power**	**Data**
		Avg/Peak	**Storage**
	(kg)	(W)	(Mb/day)
Doppler Magnetograph	14	30	290
EUV Imager	14	27.5	150
Magnetometer	1.5	2	1.6
Faraday Cup	1.5	2	0.2
Energetic Particle Detector	0.3	0.2	1.6
Total	**31.3**	**61.7**	**443.4**
Radio Waves Experiment Option*	17.8	115	
* Part Of Spacecraft Communications Subsystem			

Science Instruments

The FSS instrument complement consists of the following types of measurement devices:

- Doppler magnetograph for remote sensing of photospheric magnetic and velocity fields
- EUV imager for remote sensing of the solar corona
- Magnetometer for in-situ magnetic field measurements
- Faraday cup for in-situ solar wind plasma measurements
- Energetic charged particle detector for in-situ measurements
- Optional radio science experiment using the telecommunications system

All the instruments above use proven technology and have strong flight heritage. Instrument system parameters are shown in the table.

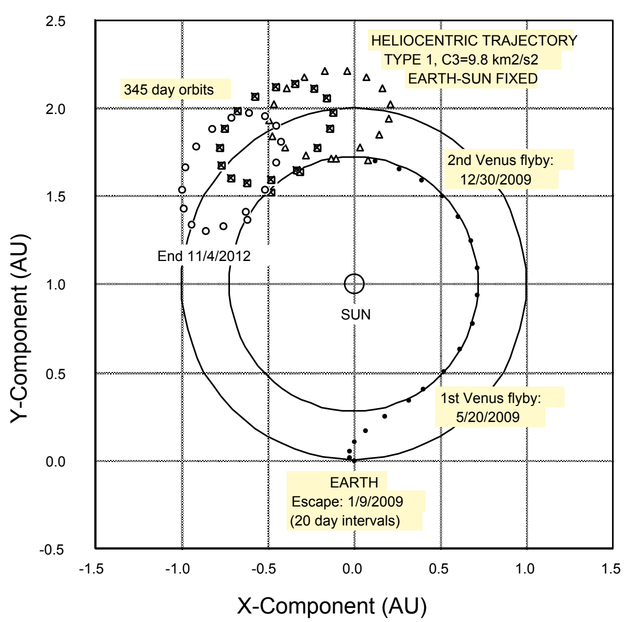
Mission Profile

The following serial time spans are assumed for mission planning:

- Extended pre-formulation and formulation phase instrument, spacecraft, and ground system accommodation studies to match available funding
- 12 months for conclusion of project formulation and definitization prior to approval
- 2.5 to 3.3 years from approval to launch readiness dependent upon LWS program funding profile
- January or November 2009 launch
- 12 month cruise to operational drift orbit
- 2 years for baseline mission operations
- 3-year mission extension (option for evaluation)

The FSS spacecraft is delivered to orbit by a single launch vehicle with the required performance for a C_3 of 14 km^2/sec^2. There is a 20-day launch window available for this C_3 value. The spacecraft will be launched into a solar far side

Figure 3.6-4: FSS Orbit Trajectory



drift orbit (see Figure 3.6-4), stationed behind the Sun within 30^o of the Earth-Sun line. Two Venusian Gravity Assists are employed to minimize C_3 and ΔV requirements. Launch vehicle options include the Delta II 7326 and Athena II with a Star-48 kickstage.

FSS Spacecraft

The FSS spacecraft includes the following features. The spacecraft is a three-axis stabilized, solar-pointed spacecraft with a propulsion system for orbit maneuvers with external solar array and radiator surface areas sized in accordance with its orbit proximity to the Sun. Also assumed in the custom bus design is a deployable non-articulated Ka-band high gain antenna system with telecommunications systems mounted on a side pallet with radiators to dissipate heat. A preliminary concept for the FSS spacecraft may be seen in Figure 3.6-5.

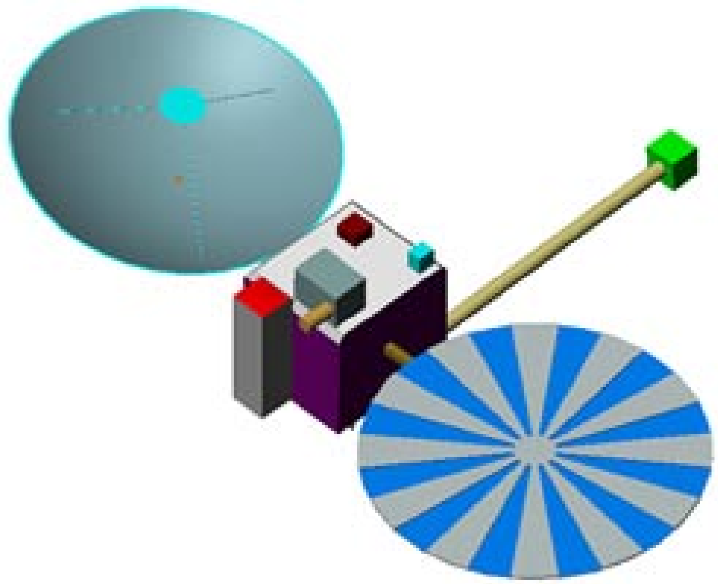


Figure 3.6-5: FSS Spacecraft Concept

3.6.3 Mission Requirements

Description	Requirement	Rationale
Launch of Inner Heliospheric Sentinels (IHS)	The launch system shall inject a 1,400 kg payload to a C_3 of 10 km^2/s^2	Preliminary analysis has determined the mass of four IHS spacecraft to be 1146 kg. Sufficient margin must be included to protect against future mass growth. The IHS spacecraft use a Venusian Gravity Assist to achieve elliptical heliocentric orbits at various distances from the Sun.
Launch of Far Side Sentinel (FSS)	The launch system shall inject a 210 kg payload to a C_3 of 14 km^2/s^2	Preliminary analysis has determined the mass of the FSS spacecraft to be 210 kg. The FSS spacecraft uses a Venusian Gravity Assist to achieve a solar far side drift orbit.
Payload Volume for IHS Launch	The launch system shall accommodate a payload with minimum volumetric dimensions of 2.5 m diameter x 3 m length	Launch packaging of the IHS spacecraft (stacked 2x2) for a single launch has determined that a 2.5 diameter x 3 m length payload volume will be required.
Payload Diameter for FSS Launch	The launch system shall accommodate a payload with a minimum diameter of 9.5 ft	Launch packaging of the FSS spacecraft has determined a 9.5 ft payload diameter will be required to accommodate the high-gain antenna.

Table 3.6-4: Solar Sentinel Mission Requirements

3.6.4 Key Technology Investments

Instruction: Add key technologies for the mission to the table below. Be sure to complete all columns for each technology.

Technology	Summary Description	Current TRL	Additional Applications
High-efficiency photovoltaic cells	Advanced triple-junction photovoltaic cells (41% AM0) for in-space power generation	5	All spacecraft power applications
Li-ion battery modules	High energy density Li-ion battery modules for energy storage		All spacecraft power applications
Low mass instruments	Smaller in-situ and remote sensing instruments for IHS and FSS spacecraft		Mission specific
Communications	X-band omni antennas for command uplink		All spacecraft communications applications
Communications	Ka-band high gain antenna for telemetry downlink		All spacecraft communications applications
Simple-slit star camera	Simple-slit star camera for spin axis orientation		All spacecraft attitude control applications
Composite propellant tanks	Composite pressure vessel (COPV) propellant tank for mass savings		All spacecraft propulsion applications

Minimum impulse thrusters	Minimum impulse thrusters for finer control and elimination of reaction wheels		All spacecraft attitude control applications

Table 3.6-5: Solar Sentinel Mission Key Technology Investments

3.6.5 References

1. http://lws.gsfc.nasa.gov/sentinels.htm

2. Szabo, A. "Inner Heliospheric Sentinels". LWS Science Workshop, NASA GSFC, May 2000.

3. "Sentinels". NASA GSFC. http://lws.gsfc.nasa.gov/docs/LWS_Pre-Formulation_Study/LWSIHS.pdf

POC: NASA GSFC / Living with a Star Program

3.7 Human Outer Planet Exploration

Instruction: This section should provide a brief introduction to the design reference mission. A few paragraphs should suffice.

3.7.1 Mission Description

Instruction: Describe the mission in this section, including important items such as architecture profile(s), major mission elements, IOC, etc.

3.7.2 NEXT Themes and Goals

Instruction: This section should link the design reference mission to NEXT themes and near-/long-term goals. A one-page summary of the science and commercial goals supported by the mission is requested.

3.7.3 Integrated Space Transportation Plan Requirements

Instruction: Compile DRM requirements for ISTP in this section. Include a brief rationale for each requirement.

Description	Requirement	Rationale

3.7.4 Key Technology Investments

Instruction: Add key technologies for the mission to the table below. Be sure to complete all columns for each technology.

Technology	Summary Description	Current TRL	Additional Applications

3.7.5 References

Instruction: Compile all applicable references here. Include the primary points of contact at the bottom of the section.

POC:

4.0 SUMMARY