Investigation of Possible Heliocentric Orbiter Applications for Crewed Mars Missions

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Abstract
This study develops the use of heliocentric orbiters, or cyclers, to ferry crew in a safe and expedient manner to and from Mars while providing a sustainable and reusable exploration infrastructure. A Mars-scaled version of the Exploration Systems Architecture Study (ESAS) and the 1998 Mars Design Reference Mission Version 3.0 (DRM) provide architectural baselines. Figures of merit, which include performance, safety, reliability, cost and extensibility, are defined in order to make cross-architectural comparisons and determine the need for additional innovative technologies. Applied technologies and architectural variations are evaluated to determine the best possible implementation of the novel architecture, including costing, technology development and operational schedules for meeting the Space Exploration Initiative deadline of 2030.

Nomenclature
ΔV – Delta velocity, km/s
φ – flight path angle relative to the local horizontal
ρ∞ – density on free stream
ρs – density at sea level
ζ – heat shield density (kg/m²)
AB – Aerobraking
C – dimensional constant in heat transfer eqn
CaLV – Cargo Launch Vehicle
CER – Cost Estimating Relationship
CEV – Crew Exploration Vehicle
CM – Crew Module
ConOps – Concept of Operations
DRM – Design Reference Mission
ECCV – Earth Crew Capture Vehicle
ECLSS – Environmental Control and Life Support System
EAV – Emergency Abort Vehicle
EDL – Entry, Descent, and Landing
EDS – Earth Departure Stage
ERV – Earth Return Vehicle
ESAS – Exploration Systems Architecture Study
FOM – Figures of Merit
GCR – Galactic Cosmic Rays
ISRU – In-Situ Resource Utilization
LMO – Low Mars Orbit
LSAM – Lunar Surface Access Module
LV – Launch vehicle
MSAM – Mars Surface Access Module
NAFCOM – NASA Air Force Cost Model
NTP – Nuclear Thermal Propulsion
NERVA – Nuclear Engine for Rocket Vehicle Application
OEC – Overall Evaluation Criteria
Q – integrated heating at a specific location (J/m²)
Q’ – heat of ablation (J/kg)
QRAS – Qualitative Risk Assessment System
qs – thermal heating
R – effective nose radius of the vehicle in the flow
Rc – radius of the cross-section of the ring ballute
Rb – radius of curvature of a ballute
RASC-AL – Revolutionary AeroSpace Concepts – Academic Linkage
ROM – Rough Order of Magnitude
SEI – Space Exploration Initiative
SM – Service Module
SPE – Solar Particle Event
SVLCM – Spacecraft/ Vehicle Level Cost Model
TEI – Trans-Earth Injection
TPS – Thermal Protection System
TMI – Trans-Mars Injection
û – ratio of circumferential velocity component normal to the radius vector
V∞ – Hyperbolic excess speed
VASIMR – Variable Specific Impulse Magnetoplasma Rocket
VISIT – Versatile International Station for Interplanetary Transport

Introduction
In 1998, NASA released the results of a study performed to develop a baseline reference architecture for future Mars exploration planning efforts. This architecture, referred to as Mars Design Reference Mission Version 3.0 (DRM), described possible implementations of currently available and projected technologies to the development of a Mars transportation architecture. The focus of Mars DRM was to reduce the mass and cost and improve performance of the elements and mission, respectively, without introducing greater risk.

Conventional methods for space transportation architectures, such as those found in the Mars DRM and the Exploration Systems Architecture Study (ESAS), have been examined extensively for several years, but continue to pursue designs of minimal programmatic and economic risk. Innovative and original concepts have the

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Cyclers are stations on a heliocentric orbit that eventually return to the same position in space from where they started, at which time the trajectory cycle begins again. Cycler trajectories fall into two broad categories: ballistic, which acquires all necessary delta velocity (ΔV) from planetary fly-bys and powered, which requires periodic deep space ΔV maneuvers to maintain the trajectory. Ballistic trajectories are also known as free-return trajectories. Cyclers are further classified based on the number of synodic periods required to complete one cycle.

Free-return trajectories in general were first proposed in the 1960’s and were expanded upon to repeatable free-return trajectories by Rall and Hollister in 1969. The trajectories proposed by Rall and Hollister were ballistic and cycled every synodic period. Interest in these initial studies, however, was limited because of the high ΔVs required by the taxi to rendezvous and the short stay times provided at Mars. It wasn’t until the mid 1980’s, when two notable architectures were introduced, that interest in cyclers was rekindled. In 1985, Buzz Aldrin proposed an up and down escalator concept that repeats every synodic period. These cyclers allow for frequent and regular transfers between Earth and Mars. However, they require relatively large ΔV injections in deep space to maintain the trajectory and have large hyperbolic excess speeds (Vₚₚ) at the planets, which results in large ΔVs for the taxis. Shortly after the Aldrin cyclers were introduced, Niehoff presented his Versatile International Station for Interplanetary Transport cyclers, VISIT 1 and VISIT 2. The VISIT cyclers follow a seven synodic period repetition, passing between Earth and Mars on several occasions. The trajectory is purely ballistic and allows for smaller taxi ΔV requirements at the planets, but the encounters are infrequent and irregular. Over the past decade, increasing work has been conducted to find viable cycler solutions. Recent research focuses on developing generalized models to optimize trajectories based on a broad range of variables and actual ephemeris data.

The potential to revolutionize the current space exploration efforts, but they are often disregarded due to their lack of heritage and programmatic aversion to less economical solutions. To answer the call for a revolutionary architecture (the ‘R’ in RASC-AL), cyclers are proposed as an architecture that could provide the most potential advantages over several other innovative options.

Advantages and disadvantages to the implementation of cyclers as a novel Mars transportation architecture. Cyclers are now becoming an appealing alternative to conventional planetary transfers for several reasons. The foremost reason is that they are permanent, reusable, and sustainable over long periods of time. For this study, the cycler travels between Earth and Mars, but the cycler architecture may readily be placed on a heliocentric trajectory to any target planet with minimal modification.

The proceeding study discusses the application of cyclers as a novel Mars transportation architecture. Advantages and disadvantages to the implementation of the cycler architecture were examined. Improvements in technologies that will make cyclers even more beneficial are presented. Comparisons were made to previously explored Mars architectures.

Study Requirements

Requirements for the study were defined from several sources including those put forth by RASC-AL, NASA, and ESAS, as well as those defined as being internal to the project. A driving requirement for the study was defined by the call for revolutionary architectures and technologies to be applied to human exploration of Mars. Other requirements specified by RASC-AL made it necessary for the whole scope of the architecture to fall within the NASA budget, provide a sustainable permanent infrastructure for exploration and provide a potential scientific payoff. Components of the architecture must utilize technology that will be sufficiently developed for implementation at the beginning of mission operations. NASA requirements make it necessary that the architecture provide a flexible abort opportunity so that the crew can return safely to Earth at any point of the mission, as well as have a time constraint on the beginning of mission operations; all elements of the architecture must be able to be implemented on or before 2030. The possibility of
making the developed architecture extensible to future exploration opportunities is put forth as a requirement by ESAS, as well as the need to provide realistic goals for both the overall mission and probability of mission success. Project specific requirements were also established for cost, safety, and reliability as well as setting the time constraint on the maximum crew transfer period to not exceed one year.

NASA’s latest approach for returning to the Moon and Mars was published in the ESAS report in the summer of 2005. An independently proposed Mars transportation architecture that reuses many of the components and technologies from that report would be advantageous. It would minimize the development cost of new technologies and make replacement parts easy to manufacture. The proposed architectures described in this report utilize combinations of ESAS’s heavy lift launch vehicle, crew exploration vehicle (CEV), solar panel power system, and LOX/LH2 propulsion system.

The ESAS report, while focusing on lunar missions, also suggests extensions to Mars exploration missions. Technology decisions for the lunar architecture must take into consideration the extensibility of those technologies to the Mars exploration architecture. The same is true for extending Mars architectures to other destinations. An optimal transportation architecture will be applicable to future missions.

In the post-Columbia era, NASA is largely focused on safety and reliability. Another requirement for the architecture is that the risk presented by the selected architecture must be acceptable. This is one area in which cyclers could provide an advantage, in that they provide a free return option. More of their advantages are discussed in a later section.

Performance issues that must be examined for any space transportation architecture include large system masses, radiation exposure, propellant boil-off, and insufficient power systems. The development of advanced propulsion systems, Mars surface production of propellant and resources, and inflatable transfer and surface habitats are desired technologies that have the ability to enhance performance. Cycler architectures with these technologies in various combinations were examined to determine the configuration for optimal cycler performance.

The final major driver that any Mars architecture will have to meet is cost. The government and public will never support an unaffordable Mars program. Cyclers can be a benefit in this area when taking multiple excursions to Mars, in that the main vehicle is reusable. Once the cycler has been put into its orbit, only small trajectory corrections will be required to maintain its orbit, decreasing the lifetime performance requirements.

**Baseline Architectures**

Two existing exploration architecture studies were evaluated and defined for cross-architectural comparisons based on past NASA studies. The baseline utilized for the study was based on Mars DRM. An additional reference architecture was derived from NASA’s lunar mission outlined in the ESAS report. The ESAS lunar mission was scaled for Mars in order to provide a useful comparison architecture. Both the baseline and reference missions are considered to be semi-direct in that they require orbit rendezvous around either Earth or Mars.

In order to compare the proposed architectures, the system and subsystem elements must first be defined. These elements can best be represented by a description of the architectural concept of operations. The Mars DRM concept of operations (ConOps) consists of thirteen individual elements (six unique elements), which are seen in Figure 1. During the first mission opportunity, an Earth Return Vehicle (ERV) and Trans-Mars Injection (TMI) stage will be launched on two separate launch vehicles. The ERV and TMI stage rendezvous in Earth orbit and fly to Mars using a Hohmann transfer, where the ERV aerobrakes into a parking orbit. Also during the first launch opportunity, the Cargo Lander (cargo, a propellant production plant, power systems, an inflatable habitat, and an ascent vehicle) and the Earth Crew Capture Vehicle (ECCV) launch on one launch vehicle, shortly followed by its TMI stage on a second launch vehicle. These vehicles rendezvous in Earth orbit, transit to Mars, aerobrake in the Martian atmosphere, and land on the surface. Between the first and second launch opportunities, the ISRU plant produces the propellant which will bring the crew back to Earth. During the second launch opportunity, the Crew Lander and TMI stage launch on two separate launch vehicles. These elements rendezvous in Earth orbit and transfer to Mars, where they aerobrake and land on the surface. After a surface stay of approximately 500 days, the ECCV and ascent vehicle launch into Mars orbit and rendezvous with the orbiting ERV. The ERV is used to perform the transit burn back to Earth and serve as a transfer habitat for the return trip. The ECCV then undocks with the ERV and descends to the surface of Earth.

The ESAS Mars ConOps consists of fifteen individual elements (six unique elements), which are seen in Figure 2. During the first mission opportunity, the first Mars Surface Access Module (MSAM1) and the Earth Departure Stage (EDS) will be launched on separate launch vehicles. Due to the large Trans-Mars Injection propellant required to perform the burn, it becomes necessary to transport additional fuel to a
sufficiently sized, partially fueled EDS. It is assumed that orbital cryogenic refueling technologies will exist by the time of operations commence. In Earth orbit, the EDS is fueled and then docks with MSAM1, which will then transit to Mars, where both elements will aerobrake and enter into orbit. MSAM1 will undock and land on the surface. During the second launch opportunity, a second MSAM (MSAM2), a scaled Service Module (SM) and Crew Module (CM), EDS, and EDS fuel are launched on four separate launch vehicles. The EDS fuels in Earth orbit and docks with MSAM2, SM, and CM. This combination transfers to Mars and aerocaptures in Mars orbit. The crew will transfer into MSAM2 and descend to the surface. When the surface stay is completed, they will ascend to Mars orbit in MSAM1, which has produced propellant using ISRU, and rendezvous with the CM and SM, while MSAM2 remains on the surface for the next crew rotation. The CM and SM will transfer to Earth and aerocapture into orbit. The CM will descend to Earth’s surface.

Figure 1: Concept of operations for Mars DRM architecture

Figure 2: Concept of operations for ESAS architecture

Cycler Alternative Architecture
The cycler vehicle consists of a habitat and propulsion unit that passes at regular intervals (based on the Earth-Mars synodic period) between the Earth and Mars. Many of the components used for the cycler can be taken directly from Mars DRM or ESAS, such as the Cargo Launch Vehicle (CaLV), CEV and propulsion systems. This increases the extensibility of the ESAS lunar mission and reduces costs by reusing previously developed and proven components.

Unlike Mars DRM, which requires the launch of a new habitat for the crew transfer every mission, the cycler provides an adequate crew transfer habitat. Crews
need only take a CEV taxi from Earth or Mars to rendezvous with the cycler. Prepositioned surface habitats can be sent to Mars along low-energy transfers as needed, instead of every mission, regardless of whether or not they are required. Similarly, Mars DRM must send an additional full sized vehicle and habitat for the crew’s return; cycler missions return the crew to the cycler on the same CEV they arrived in. All of these factors significantly reduce the amount of mass that must be launched for every mission.

Another critical step to advance the use of cyclers has recently been realized. Cycler trajectory development has progressed far enough to demonstrate taxi to cycler ΔVs on par with conventional transfer trajectories. This allows cyclers to reasonably compete with other architectures.

A representative diagram of the cycler ConOps is shown in Figure 3, which consists of 21 elements (eight unique elements). During the first launch opportunity, six launch vehicles are used. The ERV and associated TMI/EDS will rendezvous and perform a Hohmann transfer to Mars, where they will aerobrake into a parking orbit. The ERV will remain in this orbit until an emergency situation arises and its use is required (i.e. the cycler is not available for the return trip). The Surface Habitat will rendezvous with its TMI/EDS, perform a Hohmann transfer to Mars, directly enter its atmosphere and land on the surface. The Surface Habitat, since it is the only predeployed surface asset, will be equipped with an ISRU plant to produce propellant to fuel the ascent stage. The Cargo Lander and associated TMI/EDS performs a Hohman transfer to Mars and aerobrakes into a parking orbit.

The cycler will inject into its heliocentric orbit, where only small ΔVs will be required for station keeping purposes. During the second launch opportunity the crew, SM, CM, and associated TMI/EDS are launched on one vehicle. A ΔV is performed to insert these three vehicles into the cycler orbit and to meet with the cycler. The cycler is used as a transfer habitat for the crew. When the cycler reaches Mars, the crew will transfer to the SM/CM and a ΔV is performed to enter Mars orbit. The SM undocks with the CM after the CM docks with the Cargo Lander currently in orbit. The CM and Cargo Lander then descend to the surface. After the surface stay is completed, the crew ascends in the ascent vehicle and CM configuration and docks with the SM. After the lander is discarded, the SM then performs a ΔV to get back onto the cycler for return to Earth. Upon arrival at Earth, it performs another ΔV to get off of the cycler. The CM and SM undock in Earth orbit. The CM aerobrakes and descends to Earth’s surface.

As can be seen in Figure 4, the cycler architecture is developed in such a way that most components are reused. One of the criticisms of Mars DRM is the number of elements, particularly propulsive stages, that are discarded every mission. Additionally, the permanent nature of the cycler eliminates the need to inject massive pieces of equipment to Mars that do not contribute to the surface mission. The crew lander is the only piece of precursor equipment required for every mission. Surface habitats are only sent as new bases are needed, which may not be necessary every mission.

Looking beyond the mass savings, there are some additional benefits to cyclers that cannot be achieved with conventional transfer missions: modular growth and extended deep space research. The cycler supports a modular design, which provides for potential expansion. Depending on the trajectory chosen, the cycler may have one or more Earth fly-bys that are not designated for crew transfer. During this time, additional modules may be launched to rendezvous with the cycler, ultimately constructing a large facility. Such large structures would allow for significantly larger crews to transfer as missions and goals evolve. It would be impracticable to use similarly sized structures using conventional transfers.

An easily overlooked but potentially valuable asset of a cycler is its long-term operation in deep space. The cycler can operate as a science platform throughout its lifetime. Many characteristics of the region beyond Earth are poorly understood, especially the solar wind and radiation environment. Placing scientific instruments on the cycler will return a wealth of information over the life of each vehicle that would otherwise require a dedicated spacecraft to obtain.
Selection and Definition of Investigated Architectures

A systems engineering methodology was implemented to reduce the original design space (initially containing 8.7x10^14 alternatives) to a few architectures to be compared to the baselines. This general systems methodology is depicted in Figure 5 below.

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the results section of the report. These FOMs were then mapped to the large number of possible technological and architectural variations from the original design space to determine the architecture components that will have the greatest impact on the value function. Reducing the drivers to the 5 shown in the figure downsized the number of possible alternatives to 192. At this point, multiple alternative transportation architectures are developed. A Matrix of Alternatives contains the top driving trade spaces, both revolutionary and conventional, for the concept of operations, propulsion, method of Earth entry, radiation protection, and power systems. The revolutionary technologies include ballute aerocapture, Nuclear Thermal Propulsion (NTP), Variable Specific Impulse Magnetoplasma Rocket (VASIMR), and LOX/CH₄.

Ballute aerocapture is an atmospheric assisted ΔV maneuver using a large, inflated, trailing drag surface. There is a potential to reduce propellant mass as well as reduce the acceleration loads on the crew and heating loads on the vehicle. However, this Earth entry option reduces the reliability of the architecture when compared to conventional propulsive maneuvers.

The following three technologies introduced in the reduced morphological matrix are all theoretically advantageous, revolutionary propulsion options. NTP is based on a wealth of testing done in the Nuclear Engine for Rocket Vehicle Application (NERVA) program in the 1960’s and 1970’s. The system consists of a nuclear fission reactor in which propellant, usually liquid hydrogen, is heated by the reactor and expanded through a nozzle. This technology provides both high thrust and high specific impulse, reducing the required propellant mass. However, this desirable performance is also linked to added radiation shielding mass. Using a combination of methane and liquid oxygen, LOX/CH₄, in the architecture opens the door to In-Situ Resource Utilization (ISRU). The ability to use propellant generated on Mars reduces the mass needed to launch from Earth. However, this combination of chemical fuels results in a lower specific impulse, as compared to a combination of liquid oxygen and liquid hydrogen. VASIMR is a plasma rocket with a range of Isp extending to 3000 seconds and beyond, which allows optimization to the trajectory. However, this technology currently resides at a low technology readiness level (TRL), is quite massive, and requires approximately 1 MW of power.

Fifteen theoretically advantageous alternative architectures were selected from the revolutionary and conventional architecture components in the matrix of alternatives. These were further reduced to five architectures using a Pugh matrix selection which compared the alternatives to a baseline qualitatively across all of the FOMs.

The baseline used for this study is Mars DRM v3.0. In addition to the five selected alternative architectures, the lunar ESAS architecture extended to a Martian application was also analyzed and scaled. The remaining alternate architectures are all based on a revolutionary cycler orbit ConOps. The following table displays the driving components of the chosen alternative architectures.

### Alternatives

<table>
<thead>
<tr>
<th>Alternatives</th>
<th>Conops</th>
<th>Propulsion</th>
<th>Radiation</th>
<th>Power System</th>
<th>EDL method</th>
</tr>
</thead>
<tbody>
<tr>
<td>DRM v3.0</td>
<td>Semi-direct</td>
<td>NTR</td>
<td>Shielded room</td>
<td>Solar panels</td>
<td>Direct</td>
</tr>
<tr>
<td>ESAS Mars</td>
<td>Semi-direct</td>
<td>LOX/LH2</td>
<td>Shielded room</td>
<td>Solar panels</td>
<td>Direct</td>
</tr>
<tr>
<td>Alternative 7</td>
<td>Semi-direct</td>
<td>VASIMR</td>
<td>Shielded room</td>
<td>Fission</td>
<td>Ballute aerocapture</td>
</tr>
<tr>
<td>Alternative 13</td>
<td>Cycler habitat</td>
<td>LOX/LH2</td>
<td>Shielded hull</td>
<td>Solar panels</td>
<td>Ballute aerocapture</td>
</tr>
<tr>
<td>Alternative 14</td>
<td>Cycler habitat</td>
<td>NTR</td>
<td>Shielded hull</td>
<td>Fission</td>
<td>Direct</td>
</tr>
<tr>
<td>Alternative 16</td>
<td>Cycler habitat</td>
<td>LOX/CH4</td>
<td>Fuel/Water shield</td>
<td>Fission</td>
<td>Ballute aerocapture</td>
</tr>
<tr>
<td>Alternative 18</td>
<td>Cycler habitat</td>
<td>NTR</td>
<td>Polymer/Water shield</td>
<td>Bimodal NTR</td>
<td>Ballute aerocapture</td>
</tr>
</tbody>
</table>

### Trajectory Analysis

Trajectory options for this study were selected from previously published optimized solutions because of the sheer number of potential trajectories and variables involved in their optimization. We selected five potential trajectories that provided the best fit with our mission requirements. Additional analysis of these trajectories allowed us to select the optimized S1L1 trajectory described in McConaghy, et al. The trajectory, optimized for minimum combined cycler and taxi ΔV, was used in subsequent analyses for all cycler alternatives.
Figure 6: S1L1 cycler trajectory schematic

Four vehicles are needed to ensure one mission can occur during each synodic period. The cyclers themselves will need to be launched sufficiently in advance to complete their first gravity assist maneuver prior to its first crew transfer. Based on flight times and mission durations, initial cycler launches may be spread out annually over four years.

Over a 30-year period, this trajectory provides 14 possible mission opportunities. Each opportunity provides an average mission timeline of 168 days in transit from Earth to Mars, 522 days surface stay, and 213 days in transit from Mars to Earth, which is comparable to DRM mission times (Figure 7). The taxi will require an average $\Delta V = 4.49$ km/s for the Earth to Mars leg and $\Delta V = 2.83$ km/s for the return leg (Figure 8). In order to maintain this trajectory, the cycler will have to make periodic deep space trajectory adjustments. When averaged over the 14 missions, these maneuvers amount to $\Delta V = 0.11$ km/s per synodic period.

ECLS Analysis

The goal of the Environmental Controls and Life Support analysis was to compute the consumable mass for the various architecture alternatives. This was computed using a system breakdown from Larson and Pranke. Consumable mass includes food, cooking utensils, clothes, and sanitation supplies. The main drivers for this analysis are the time of flight and surface stay. An assumption for this analysis is that the number of crew is six and constant for all of the architectures. The trajectory analysis gave the time of flight numbers and surface stay based on launch window opportunities for minimum $\Delta V$'s.

The difference in the masses between the architectures is 300 kg. This is a very small fraction of the total mass of the vehicle and thus this system is not a major driver in the mass determination. Figure 4 shows the consumables mass for the various architectures.

Radiation Analysis

Providing a crew with sufficient radiation shielding without adding excessive mass is a significant challenge in spacecraft design. Space radiation poses a significant threat to humans during long duration missions, and the exposure to radiation must be mitigated by adding...
enough shielding to keep short term and accumulated doses within safe limits. Because of the mass of shielding required, it is impractical to fully shield the entire vehicle to meet all radiation hazards. Designs, therefore, often incorporate a small shelter inside the vehicle to optimize protection. The hull protects from nominal levels of solar particle radiation (SPR) and some galactic cosmic rays (GCR) while a sheltered room, such as a sleeping area, provides additional GCR shielding and protection against potentially lethal radiation from solar flares.

Typical spacecraft hulls are constructed of aluminum alloys, which have proven to give poor protection. Hydrogen provides the best shielding, and several lightweight, high hydrogen content polymer materials have been developed to supplement basic hull shielding. These materials serve no other purpose, however, and are carried as deadweight. Water stores and fuel can also be used effectively as supplemental shielding. Since the water and fuel mass must be carried during the mission, using these stores for shielding can significantly reduce the amount of nonfunctional mass required.17,18

Models are required to calculate the estimated radiation doses, but current knowledge of the radiation environment beyond the moon is incomplete. During previous missions, some measurements were taken of GCR radiation during solar maximum and solar minimum periods. It has been observed that periods of greater solar activity have a diminishing effect on GCR, and vice-versa. Vehicles must also be designed to protect against SPE levels four times greater than the 1972 flare, a particularly violent storm. These GCR and SPE measurements were combined to develop a model that calculates the estimated dose based on radiation type and mission time with respect to the solar cycle.19,20 The model was applied to each alternative based on the type of shielding dictated for each. In this study, alternatives 13 and 14 called for only hull shielding, Mars DRM, ESAS Mars, and alternative 17 place water stores around sleeping quarters, and alternative 18 combines the water with a supplemental polymer material. The analysis revealed that combining water stores and high hydrogen content materials (Alternative 18) provides the greatest reduction in overall dose for the least amount of additional mass.

![Figure 10: Comparison of shield configurations reveals that water/polymer shielded room (Alt 18) provides lowest dose](image)

![Figure 11: Combination water/polymer shielded room (Alt 18) requires least additional mass to implement](image)

Earth EDL Analysis

Analysis of entry, descent, and landing (EDL) upon Earth return is important since it imposes restrictions on the trajectory (flight path angle), resulting from human23,24 and material limits and targeting requirements. The results of this analysis propagate back and impose requirements throughout the mission timeline, especially in the launch mass. The results also determine mission success and crew survivability. The goals of this analysis are to provide an EDL trajectory that ensures crew and vehicle survivability, provides a proper landing site, and has a minimum impact on all other parts and phases of this mission.

Ballute aerocapture29,31 at Earth return from Mars is investigated because of its perceived benefits in the reduction of mass. Figure 12 shows a representative aerocapture maneuver; the intent is to use one pass through the atmosphere to dissipate enough energy to capture into orbit. Although propulsive maneuvers are required to adjust the orbit after the maneuver, the propellant mass will be less than for an all-propulsive orbit capture. Even with the mass required for thermal protection, the combined mass of the thermal and propellant system is less than it would be without aerocapture.
Once a target orbit is selected, the aerocapture corridor can be found. Trajectories are evaluated using full lift up (bank angle equal to zero degrees) and full lift down (bank angle equal to 180 degrees) to define the corridor. If the vehicle enters the atmosphere at a flight path angle steeper than the flight path angle for the full lift up trajectory, the vehicle lands. If the vehicle enters the atmosphere at a flight path angle shallower than the flight path angle for the full lift down trajectory, the vehicle is not captured.

The $V_c$ is assumed to be 13 km/s for a direct Mars return and 7.84 km/s for a Cycler return. During the aerocapture pass, acceleration limits for the crew and heat shield thermal limits must not be exceeded.

The $q_s$ is given by:

$$ q_s = \frac{C}{\sqrt{R}} \left( \frac{\rho_s}{\rho_s} \right)^n \left( \frac{u}{\cos \varphi} \right)^m $$

(1)

Equation (1) shows Chapman’s heating rate in hypersonic flow at a stagnation point, which is used for the heating analysis. C, n, and m are constants that depend on the type of boundary-layer flow. For laminar flow, $n = \frac{1}{2}$, and C and m can be determined using testing or theory and found in references. R is the effective nose radius of the vehicle in the flow, $u$ is the ratio of the circumferential velocity component normal to the radius vector to the circular orbital velocity, and $\varphi$ is the flight path angle relative to the local horizontal. The effective nose radius for the crewed vehicle is the radius of the heat shield.

$$ R_s = \left[ \frac{1}{2} \left( \frac{1}{R_n} + \frac{1}{R_b} \right) \right]^{-1} $$

(2)

The effective ballute nose radius is calculated with equation (2), where $R_n$ is the radius of the cross section of the ring ballute and $R_b$ is the radius of curvature the ballute, assumed to be infinity here.

$$ t = \frac{Q_{total}}{(Q^*)^n (\zeta)} $$

(3)

For the heat shield thickness (equation 3), t is the heat shield thickness in meters, $Q_{total}$ is the integrated heating at a specific location (J/m²), $Q^*$ is the heat of ablation (J/kg), and $\zeta$ is the heat shield density (kg/m³).

For the present analysis, the following assumptions are made regarding the heating:

1. The bow shock of the crewed vehicle is inboard of the ring ballute – there is no shock interaction with the ballute shock.
2. Chapman’s heating rate equation is applied to the ballute by changing only the effective nose radius – the other variables are no different compared to the crewed vehicle.
3. To account for radiative heating, the convective heating rate from Chapman’s equation is multiplied by a factor of two.
4. The trailing distance is not taken into account.

For the simulation, the duration of ballute deployment can be varied to keep within acceleration loading constraints and achieving the target apopasis.

The Earth EDL analysis provides comparisons of masses, heat shields and ballutes, measured acceleration and maximum heating rates during the entry. Figure 13 compares the masses for each alternative with the baseline architecture. The largest masses correspond to the ballute aerocapture option. Alternative 14 results a low EDL mass since it has a low entry velocity after separating from the cycler. Figure 14 compares the maximum measured acceleration for the alternatives. Figure 15 compares the maximum heating rate upon entry. The largest benefit of the ballute aerocapture is the minimization of the heating rate.
assumption will result in a large mass decrease since the solar arrays are used on multiple architectural elements. Reusing ESAS elements, such as the CEV (Crew Exploration Vehicle) CM (Crew Module) also results in a larger mass than the Mars DRM v3.0 study. However, the credibility lies with this assumption since it is a much more recent and thorough crew return vehicle than the capsule assumed in Mars DRM v3.0. A power budget identical to Mars DRM v3.0 is assumed since a reevaluation of the power needs is deemed outside the scope of this project. The nuclear fission reactor mass sizing is based on a combination of historical masses, projected performance for spacecraft reactor systems and paper studies.

Figure 16 compares the power system masses against the baselines. Alternative 7 utilizes VASIMR propulsion, which requires 1 MW of power driving up the mass. ESAS Mars has the lowest mass, but it also has the lowest power requirement.

Reliability Analysis

The reliability analysis was performed on each of the seven architecture alternatives using the Qualitative Risk Assessment System (QRAS). The method used is built up by components, so the reliability of each component in the architectures was computed individually. The final reliability number was computed as the product of the reliability of the individual components. The reliability for each component in the architecture was computed in one of two ways. The number of redundant components also factors into this calculation.

Since the main purpose is to compare the reliability of each architecture relative to the other architectures, only the varying technologies between the alternatives are used to calculate the relative reliability. These technologies include the propulsion and power systems.
and the aerocapture method. The technologies that drove the reliability numbers were the VASMIR propulsion system and the ballute aerocapture. Since ballute aerocapture has not been implemented on a manned spacecraft, this is modeled using the parachutes on the Space Shuttle and by changing the duration that the ballutes/parachutes are in use. From the entry, descent, and landing analysis, the ballute operation duration was calculated to be fifty minutes. The reason that VASMIR decreases the reliability is that it doesn’t include a redundant engine. VASMIR is massive and requires a large amount of power (on the order of megawatts), so a second engine would increase the mass to an undesirable amount for the marginal reliability gain we would receive.

Table 2 shows the final reliability numbers for the different architectures. The most reliable architecture, after ESAS Mars, is alternative 14, which consists of the cycler reactor, direct orbit entry, a shielded hull, and fission power.

### Table 2: Final reliability numbers for varying architectures

<table>
<thead>
<tr>
<th>Alternative</th>
<th>Reliability</th>
</tr>
</thead>
<tbody>
<tr>
<td>DRM</td>
<td>0.8651</td>
</tr>
<tr>
<td>ESAS</td>
<td>0.9422</td>
</tr>
<tr>
<td>Alt. 7</td>
<td>0.7692</td>
</tr>
<tr>
<td>Alt. 13</td>
<td>0.7967</td>
</tr>
<tr>
<td>Alt. 14</td>
<td>0.9242</td>
</tr>
<tr>
<td>Alt. 16</td>
<td>0.8444</td>
</tr>
<tr>
<td>Alt. 18</td>
<td>0.7314</td>
</tr>
</tbody>
</table>

Figure 17: Reliability of each architecture

**Mass Sizing Analysis**

A mass sizing analysis is performed for each of the baselines and investigated architectures to quantify the performance of each and to obtain information necessary to calculate the costs of the vehicle. The analysis is performed using a simple, structural mass fraction and mass ratio sizing approach which scaled currently proposed propulsion stage structural masses based upon: 1) the propellant needed to perform the necessary ΔVs obtained from the trajectory analysis and 2) the payload mass that the propulsion system needs to accelerate. The structural mass fractions for the propulsive stages were obtained from an investigation of the mass breakdowns of similar systems on previous architectures (for example all NTP stages were based upon the NTP stages reported in Mars Drm v.3.0). The total system masses are determined starting at Earth arrival and progressing backwards through the concept of operations by building up the mass of each successive vehicle using the elements to be transferred as payloads according to the ConOps. Total system masses of each of these architectures are calculated based upon the previously stated set of assumptions about the architectural element complement of each of the alternatives. The use of ISRU for Mars surface propellant production, which is assumed for all architectures, is captured in the mass sizing by accounting for the transport to the surface of Mars on the first Mars surface payload of the hydrogen feedstock necessary to generate the necessary ascent propellant and the ISRU system which is referenced from Mars DRM². Aerocapture at Mars is assumed for the determination of the arrival ΔVs at Mars. Different propulsion systems are modeled by varying I_{sp} and the propulsion system masses. Radiation protection masses, power system masses, consumable masses, the propulsion system masses and varying concepts of operations were all accounted for in the mass statement. The analysis reported three masses. First is the mass of one crewed mission to Mars from start to finish. This is used to gage the initial difficulty of setting up any exploration infrastructure and to cost all of the necessary elements. The second reported mass is the total mass necessary for five crewed transfers assuming that each transfer lands at the same outpost location at the surface. This is used to compare with the third mass, which is the total mass necessary to perform five crewed missions assuming that the mission lands at a different surface location each time in a sortie class mission. These two types of missions are modeled to capture: 1) the potential benefits of certain architectures due to the available reusability of certain elements of that architecture and 2) the extensibility of certain architectures to permanent habitation efforts.

Figure 18 shows the results of the mass sizing analysis. It was seen that the VASIMR alternative (Alternative 7) provided the most mass reduction over DRM. This follows from the high I{sp} capability with the VASIMR propulsion system. All the advantageous solutions involve the use of either VASIMR or NTP, emphasizing the impact of high performance propulsion options. Conversely, the more conventional chemical options showed the worst performance. It was also observed that the anticipated advantage of cycler orbits in non-sortie, permanent outpost options was verified by
the large relative gap between the one time masses and the five time masses on the cycler architectures.

![Figure 18: One and five trip system mass comparisons normalized to Mars DRM v.3.0 masses](image)

**Costing Analysis**

The costing analysis is used to compare the relative lifecycle development and production costs of each of the investigated architectures by building up the total costs for each of the architectural elements. Only development and production were considered due to the large variability of these across architectures. The masses obtained from the mass sizing analyses are used as inputs to a Rough Order of Magnitude (ROM) Spacecraft/ Vehicle Level Cost Model (SVLCM) for all the airframe development and production costs. The results of using this cost model to predict the total development and production costs were checked against the costs for similar elements on other vehicles and found to correspond reasonably. The propulsion system costs were predicted using Cost-Estimating Relationship (CER) based upon the NASA Air Force Cost Model (NAFCOM).

Using this model to separately cost the propulsion system allowed for the application of complexity factors to capture the higher costs of developing revolutionary propulsion systems. The element developments costs were spread over 8 years per element using a cost spreading worksheet. These element costs were then placed in the order of necessary development and spread over a 12 year development timeframe. This development timeframe was followed by the flight testing timeframe. Test flight hardware costs for flights preceding the manned missions were accounted for in the production cost. Production costs were modeled by spreading the necessary one mission hardware cost over the 2.5 year (~1 Earth-Mars synodic period) time between available transfer opportunities. A 90% Crawford learning curve and a 3% rate of inflation were included in this cost. The combination of development, test hardware, and production costs (all without fees or margin) necessary for the five crewed transfer non-sortie option was used to compare the different architectures in the overall evaluation criteria (OEC). The costs associated with each of the masses shown in the Figure 18 are shown in the Figure 19. These costs are normalized to the similar Mars DRM v.3.0 costs for ease of cost benefit recognition.

![Figure 19: One and five trip system cost comparisons normalized to Mars DRM v.3.0 costs](image)
It can be seen that ESAS Mars is the most affordable architecture. This follows from the small number of elements (especially newly developed elements). This cost for ESAS Mars is also slightly over priced due to the neglecting of the heritage in the landers and service module. Cycler alternatives with advanced space propulsion are seen to compare to those without due to the increased mass of conventional propulsion means.

The element development cost scheduling is shown for ESAS and Alternative 13 in Figure 20 and Figure 21. Development cost in this analysis is investigated because this cost is the most likely to exceed budget constraints. The costs here include a 20% development cost margin, 7% SEI, 10% fee, and a 25% full cost wrap in agreement with the ESAS Report. This development timeframe will likely be pushed back slightly to accommodate for the Lunar ESAS flights, but for the purposes of this study, the available funds for the development timeframe start in accordance with the NASA FY2007 proposed budget date for the starting of Mars research funding. Both ESAS Mars and the cycler alternative were found to exceed projected NASA budgetary constraints (yearly costs) for a portion of their development cycles. A more condensed development timeline pushed to later dates may have alleviated this problem. Mars mission development will occur during continuing Lunar ESAS operations. Scheduling of ESAS Mars (already overpriced from heritage issue) could easily be adjusted to fit the program within budgetary constraints. The cycler architecture could be modified, with the elimination of some elements and a possible delay to initiate the development timeframe, to limit the number of years the program would exceed the NASA budget.

**Conclusions**

When the final seven architectures were compared without the revolutionary metric, the ESAS Mars architecture was found to be the most advantageous choice according to the predefined value function. This is the most conservative architecture, and therefore has high safety and reliability factors. As shown in the following radar plot (Figure 22), which compares the architectures using their performances in the value function metrics, this architecture scores the best in cost, safety and reliability, and, predictably, ESAS reuse.
Table 3 displays the relative rankings of all alternatives by their value function score. Cyler alternative 13 scored highest among cycler architectures. This alternative provided moderate reliability and performance but scored well on cost and ESAS reuse because the architecture uses many components very similar to ESAS. The heavy weighting of our value function to cost and reliability metrics influenced the final result. A value function with a higher emphasis on performance or extensibility would have produced a different result.

![Radar plot of the value function for final architecture selection](image)

Figure 22. Radar plot of the value function for final architecture selection

Table 3: Value function scores

<table>
<thead>
<tr>
<th>Alternative</th>
<th>Value function weighting</th>
</tr>
</thead>
<tbody>
<tr>
<td>ESAS</td>
<td>1.3492</td>
</tr>
<tr>
<td>Alt 13</td>
<td>1.0408</td>
</tr>
<tr>
<td>Alt 14</td>
<td>1.0383</td>
</tr>
<tr>
<td>Alt 7</td>
<td>1.0244</td>
</tr>
<tr>
<td>DRM</td>
<td>1.0000</td>
</tr>
<tr>
<td>Alt 18</td>
<td>0.9707</td>
</tr>
<tr>
<td>Alt 16</td>
<td>0.9449</td>
</tr>
</tbody>
</table>

This study reveals that cycler architectures can provide many advantages over conventional approaches, provided that certain technologies and other considerations are implemented. As a permanent infrastructure for Mars missions, cyclers capitalize on reusability to significantly reduce lifecycle costs despite higher initial mission costs. Reuse of certain architectural elements, particularly the cycler transfer habitat, eliminates the need to launch unnecessary elements and greatly reduces the total mass required to complete each mission. The cycler architecture is also readily extensible to other missions with minimal modifications.

Cyclers are best used for a non-sortie mission, where a habitat is not delivered at each launch. This allows the original surface habitat to be reused, reducing the number of architectural elements on each mission and thus lessening the complexity of future missions. Cyclers are, however, still complex systems with a large number of elements and docking maneuvers, which will reduce reliability.

The use of revolutionary technologies can improve the performance of cyclers, but their low heritage reduces reliability and safety and increases cost. For example, cyclers with advanced propulsion systems perform as well as, or better than, the Mars DRM architecture, but until these systems are proven they will experience cost and reliability penalties. LOX/CH₄ systems are beneficial because they use ISRU for ascent propellant. LOX/CH₄ is not practicable for interplanetary transit, however, because of its low Iₚₑₚ.

The results of this study demonstrate that even a first cut analysis of cycler architectures is able to identify several potential advantages. Future work to improve the viability of cyclers should focus on several areas, such as technology development and optimizing the architectural design. Enabling technologies, such as NTR and ISRU, can provide significant performance advantages but must be developed more fully to improve reliability and cost. More research into deceleration technologies for aerocapture can drive down the masses and costs required while improving safety. Studies to further optimize cycler trajectories and incorporate low thrust options will improve performance and capitalize on the benefits of newer technologies.

Yearly development costs for the cycler do overrun the NASA budget in some years. However, adjustments to the cycler development schedule combined with scaled back Lunar ESAS operations can help the project remain within budgetary constraints.

Investigating the impact of exchanging certain elements of the architecture can also increase safety and reduce cost and complexity. For instance, use of an EAV for an abort scenario was found to penalize the cycler architectures due to additional complexity and elements in the system. Replacing the EAV with an orbiting habitat, however trades the massive, service intensive, and underutilized element for a simple, lightweight refuge. An orbiting habitat will provide a safe haven for the crew if they need to abort surface operations before the cycler returns. Since the cycler will return, the crew does not need a separate vehicle to bring them home and they can remain in the habitat until rendezvous. The habitat, therefore, has no need for a trans-Earth injection stage, which greatly reduces the mass that must be sent to Mars from Earth.
References


