PHARO: Propellant Harvesting of Atmospheric Resources in Orbit

Technology Enabled Human Mars Mission

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Conventional humans-to-Mars architectures require high masses in Low Earth Orbit boosted by numerous launch vehicles. Depending on the architecture, up to 80 percent of the required mass is the propellant needed for transit to and from Earth. Replacing this Earth-to-orbit supplied propellant mass with an orbiting system that collects and supplies this transit propellant reduces the number of heavy-lift launches required by 80%, thus leading to a more cost efficient architecture. This architecture takes advantage of significant advanced propulsion and power technologies to provide for continuous operations propellant collection vehicles and for Mars transfer vehicles that can take advantage of the collected gases.

NOMENCLATURE

ΔV  Δelta-V (change in velocity)
DDT&E  Design, Development, Testing, and Evaluation
DRA 5  Design Reference Architecture 5.0
IMLEO  Initial Mass in Low Earth Orbit
IOC  Initial Operational Capability
Isp  Specific Impulse, s
ISRU  In-Situ Resource Utilization
LCC  Life Cycle Cost
MHD  Magnetohydrodynamic
MTV  Mars Transfer Vehicle
NAFCOM NASA—Air Force Cost Model
NTR  Nuclear Thermal Rocket
PHARO  Propellant Harvesting of Atmospheric Resources in Orbit
TFU  Theoretical First Unit
1.0 Introduction

1.1 Overview

Human missions to Mars require significant amounts of mass to be deployed in low-Earth orbit (DRA 5). Under current architectures, this mass is launched via nine heavy lift vehicles (or twelve launches with no nuclear thermal propulsion). Mars transfer vehicles (MTV) are assembled, with payload and propellant ferried on separate launches. Thus, many launches are required to facilitate even a short duration stay at Mars. Any technology that presents an opportunity to reduce that launch mass in a safe and economic manner deserves consideration; on-orbit propellant harvesting and related technologies appear to provide that capability.

1.2 Problem Definition

To evaluate the advantages of implementing on-orbit propellant harvesting within a humans-to-Mars architecture, a baseline architecture was required. Such a baseline was derived from a synthesis of the RASC-AL 2010 guidelines and the Mars Design Reference Architecture (DRA 5). Once this baseline architecture (hereafter referred to as DRA 5 Derived) was established, analysis of propellant harvesting and related advanced technologies was used to determine the benefits relative to the baseline.

Both the DRA 5 Derived mission and the propellant harvesting enhanced mission (hereafter referred to as the PHARO mission) aim to deliver a crew of four on a fast transfer, enabling a surface stay of thirty days and a total (crew) mission time of less than two years. Prior to the crewed leg of the mission, one or more cargo vehicles would be dispatched to predeploy assets: surface habitat, ascent vehicle, and in-situ resource utilization (ISRU) elements. The timing of each leg of the mission is such that ΔV requirements are minimized.

1.3 Concept of Operations

Figure 1 shows the DRA 5 Derived architecture. Ten Ares-V class launches are required to deploy the cargo vehicle and crew vehicle, as well as their respective propellants and payloads, into orbit. The cargo vehicles depart for their transfer to Mars, arriving and deploying their payloads in orbit and on the surface. At the next launch opportunity, the crew vehicle departs on a fast trajectory, arriving at Mars orbit approximately 180 days later. The crew descends to the Martian surface, performs their mission, and uses the ascent vehicle fueled by ISRU elements to return to the crew MTV. The MTV quickly returns to Earth, completing the mission in just under two years.

Figure 2 shows the PHARO architecture, which utilizes propellant collection and advanced power and propulsion technologies to reduce the initial mass in low Earth orbit (IMLEO). As before, the cargo vehicle departs first, carrying the surface habitat, ascent vehicle, and ISRU system. The cargo vehicle then returns to Earth to begin refueling for the next opportunity.
At that next opportunity (five years later), the crew MTV departs for Mars, alongside the cargo MTV for the next mission cycle. As in the DRA 5 Derived architecture, the crew descends, performs their mission, and returns to the MTV over the course of thirty days. Both MTVs then return to Earth, completing the mission and beginning the cycle anew.

During the three-year latency time between flights, the two MTVs are refueled by the PHARO propellant collection system. This system consists of seven propellant collection vehicles, powered by fourteen solar power beaming satellites, which collect and store the atmosphere, then transfer the collected gases to the MTVs. Each collector maintains its orbit by using a fraction (30 percent) of the collected gases in a solar power energized, magnetohydrodynamic (MHD) electric propulsion system. This provides the thrust to counteract atmospheric drag, while also allowing the collectors to transfer to and from the parking orbit of the MTVs.

2.0 Systems Engineering

A systems engineering approach was applied to the design and comparison of the two architectures. Several figures of merit were identified as being relevant: initial and recurring IMLEO, program life cycle cost (LCC), number of launches required per mission, and probability of loss of mission. These figures of merit were then used to determine the degree to which the PHARO architecture represents an improvement over the DRA 5 Derived architecture.
2.1 Design Process—MTV
The design process began by developing the DRA 5 Derived mission. The first step in this process was determining whether the orbit would be based on impulse-like maneuvers or low-thrust maneuvers. The key to determining this is in the working fluid used in the propulsion system. The PHARO system collects air, composed mainly of nitrogen and oxygen. There are two viable means of using this fluid. The first is separation into molecular oxygen and nitrogen, using the oxygen as an oxidizer for a chemical MTV. The separated nitrogen is used in the PHARO propulsion system. The second method is to use air as the working fluid in the propulsion system of both vehicles.

The former method was rejected early in the design process as it provided marginal cost benefit and negative reliability impact over a similar architecture that simply brought oxidizer with it on each mission. The latter concept was considered in both a high-thrust, impulse mode and a low-thrust, continuous mode. A model relating mass flow rate through a propulsion system to power input and thrust output was developed based off methods presented in the literature (Litchford et al., 2000). The power input was then used to determine system masses for power and propulsion using relationships derived from the literature (Brown et al., 2010) (“Power to…”). This information was combined to produce a closed system, that would meet the given mission requirements (Figure 3).
Air has a high molecular weight (29 g/mole) compared to hydrogen (2 g/mol), and therefore has an $I_{sp}$ of 250s compared to 950s for hydrogen. This low performance leads to the need for MHD augmented propulsion systems. High thrust nuclear thermal rockets have a thrust-to-mass ratio of approximately one for impulsive maneuvers. This leads to an enormous 14 GW power requirement, on the scale of terrestrial nuclear plants. Alternately, the low-thrust concepts require several orders of magnitude less power, even though the $I_{sp}$ is dramatically higher. This leads to the selection of low-thrust spiral orbits for the mission. The orbit is based on one presented in (Chang Diaz et al. 2001) that had a mission timeline that met the given requirements.

The DRA 5 Derived architecture uses hydrogen as a working fluid, and therefore follows a more conventional NTR approach which allows for a high thrust orbit profile. Aside from this, the key difference in the two architectures is in reusability. The PHARO architecture uses reusable MTVs that return to Earth at the end of the mission, requiring a $\Delta V$ of 5,100 m/s. For the DRA 5 Derived mission, launch mass is minimized by using a ballistic Earth return for the crew and a one way mission of the cargo vehicle, discarding both MTVs at the end of the mission. This gives substantial fuel savings for the DRA 5 Derived mission as it eliminates several burns. Since all fuel must be brought up from Earth for the DRA 5 Derived architecture, this can translate into significant savings in propellant mass, even on the order of the mass of the discarded MTV. Of course, there is no need to launch propellant in the PHARO architecture, so there is great incentive to maximize reusability, as this is how launch mass is minimized for this architecture. Thus, the mass of propellant required for a mission is calculated, and is used to design the collector system (Table 2).

2.2 Design Process-Collector

The concept of operations and collection requirement of the collector system are derived from the needs of the MTV. There are many possible design options available in this design space, as shown in Figure 4. Mass and volume on the collector is constrained by the launch capability of the Falcon 9 launch vehicle (SpaceX). As in the MTV case, the feasibility of the collector is dependent on the power and propulsion requirements that are driven by the amount of propellant needed and the collection time available. Because the power requirements of the collector are approximately 175 kW, onboard power production or storage is eliminated based on both mass and safety (nuclear power) limitations. Another important consideration is in transfer of the propellant to the MTV. The altitude of the collector (123 km) is much lower than that of the MTV parking orbit (407 km), so a maneuver must be performed to rendezvous with it. This transfer maneuver uses 23% of the total atmosphere collected or 30% more than the propellant required by the MTV. MHD propulsion is also chosen as the propulsion option due to the high available $I_{sp}$ (400 s) which is required by the vehicle.
The actual technologies and design options used on the collector are highlighted in red on the morphological matrix (Figure 4). The final design of the collector, along with its major elements, is shown in Figure 5.

One of the most important consequences of this design process was the resultant need for a system of orbiting power beaming satellites. This adds another major element to the PHARO architecture that itself requires significant development. This network is dedicated to, and enables the operation of, the collectors. Power beaming is required because nuclear systems have safety issues and other power sources are extremely heavy. The design of this network and its interaction with the collectors is discussed in section 3.1.4.

2.3 Figures of Merit

Comparison of the two architectures is based on several figures of merit. There are many more components in the PHARO architecture than in the DRA 5 Derived architecture, so reliability is an obvious point of comparison between the two. This is characterized by the probability of loss of mission, as derived from the reliability analysis of the MTV and (in the PHARO case) the collector system.

Cost is another important comparison point. While many components of the PHARO architecture are reusable, their lifespan certainly is not infinite. The savings for a given mission in terms of launch costs for the PHARO approach must be weighed against the replacement costs (manufacture and launch) of new components as old ones reach the end of their life. Also, the
development and deployment costs for the PHARO components add significant overhead to this architecture, before the first mission begins. So the question in terms of cost is not whether developing the PHARO architecture is cheaper than developing the DRA 5 Derived architecture, for it will not be, but how long, or how many missions, it will take for the PHARO architecture to produce savings over the DRA 5 Derived architecture. If the breakeven point is only one or a few missions, and the expected campaign is many missions long, then this gives weight to investing in the development of PHARO. However, if the breakeven point only occurs after many missions, then it is hard to justify the development, unless there are other applications to share the development cost.

3.0 Technical Analysis

3.1 Modeling

3.1.1 DRA 5 Derived MTV Modeling

The Mars transfer vehicle models calculate system level masses for the main structure, power system, propellant and tanks, propulsion systems, and other system level masses using both physics and empirical based methods verified by Mars DRA 5. The rocket equation was used to calculate propellant required for a given mission, propulsion performance, and total system dry mass. These models were linked together and allowed to iterate so that the system would attain closure of thrust and ΔV required and all masses would be calculated for the mission.

3.1.2 PHARO MTV Modeling

Similar methods were used for the Mars Transfer Vehicle and Mars architecture, as well as a PHARO enabled Mars Transfer Vehicle and Mars architecture. These were used later in the process to compare architectures on the basis of reliability and cost.

The MHD propulsion system was modeled after the method in (Litchford et al., 2000). The analysis was adapted to the current problem, eliminating fuel input and replacing the heat of combustion with an input power. Chamber pressure and temperature from the nuclear thermal rocket in DRA 5 were used as the initial design conditions for the MHD augmentation; thrust and Isp were determined as a function of input power and mass flow rate.

3.1.3 PHARO Collector Modeling

Design of the collector followed a method previously developed by the authors (Jones et al. 2010). In summary, this method begins with the collection and storage requirements of the mission (collected mass, collection time, and the fraction of mass stored), and, using estimates of major vehicle parameters (collection area, drag coefficient), determines the required orbit and propulsion system requirements to permit collection.

These requirements were then fed into the MHD propulsion model previously discussed, to estimate the mass and power requirements of the propulsion system. For the collection vehicle, it was found that 99% of the power required came from the propulsion system.

A conceptual thermal analysis was performed to estimate the mass and power requirements of the thermal system used to cool, liquefy, and store the collected gases. Supporting data were used to estimate heat pump, plumbing, and radiator sizes (Larson and Pranke 1999). Tank sizing was performed using thin membrane theory with a non-optimum factor of 50%. Likewise, first order estimates for the inlet and nozzle masses were made using common techniques in spacecraft design (Humble et al. 1995); although work remains to
characterize the design for this high altitude application.

Integrating these analyses permits the parametric sizing of the collectors based on the requirements of the MTV. Trade variables include the number of collectors used, the number of trips each collector makes in a mission cycle, and the storage fraction.

3.1.4 Power Satellite Orbit Design

The solar power satellite network must transmit solar radiation at fifty times concentration to the collector in its orbit. This is chosen as the high end of the operating point of available solar cells on the collector, and the available surface area of the collector for mounting. Several factors govern the design of these satellites. The first is that beaming towards the Earth’s surface is not allowed, due to safety concerns of exposure to the intense light. The second is that continuous coverage of the entire collector network is needed. In addition, the solar satellites in beaming position will also need constant exposure to the Sun.

The density of satellites in the beamer orbit is governed by the required density of the collectors in the beamer’s sight at any given time. This is a function of the number of collectors needed. In general, one might expect that a sun synchronous orbit would be beneficial; unfortunately, the required inclination combined with the Earth’s tilt on its axis means that at some times the satellites in the beamer’s zone of sight will indeed be blocked by the Earth. The beamer satellites will themselves require active orbit maintenance to maintain their relative orientation to the sun over the course of a year, as well as to counter the effects of solar radiation pressure. The final orbit contains 14 satellites to power the seven collectors. The orbit has an altitude of 3950 km and an inclination of 58º. This orbit is shown in Figure 6.

![Figure 6: Collectors and power satellites in orbit.](image)

3.1.5 Cost Modeling

NAFCOM (NASA—Air Force Cost Model), a cost estimating program developed by NASA and the Air Force, was used to estimate the design, development, test and evaluation (DDT&E) costs and the theoretical first unit (TFU) costs for both the Collector and the MTV (PHARO and DRA 5 Derived architecture). The appropriate space vehicle type was selected for each vehicle and the mass breakdown for each vehicle was used to calculate the DDT&E and TFU costs.

NASA wraps were applied to account for yearly overhead costs such as management and contracting support. The NASA Wraps number was calculated as 15% of the peak year DDT&E cost (Wilhite 2009). Additionally, a learning curve of 90% was applied to successive units of the collector, assuming that as more units are built, the workforce “learns” to be more productive, and therefore the cost of building another unit is less expensive than the previous unit.
The power beaming satellite network will have to be put up specifically for the collectors. An estimated DDT&E cost of $1 billion was used based on similar solar power satellite studies. Each satellite, to be built, launched, and operated, is estimated to cost $300 million (Satellite Signals). Applying the same costing techniques (cost spreading, NASA wraps, and learning curve), the DDT&E and operations costs were calculated.

3.1.6 Reliability Modeling
Space Shuttle reliability numbers were used for the major systems of the collector and the MTV to calculate the total reliability for both. These were used as a baseline, with adjustments made to account for present and future improvements in technical reliability, as well as the low given reliabilities for items such as displays and screens.

For the major components, the failure mode was considered to be either cycle or operational time, depending on how the component is utilized. The shuttle-derived reliability for the relevant mode was used for each component.

The reliability of the PHARO mission was calculated given the design for success despite failure of a single collector. Two collectors would need to fail for not enough propellant to be collected for the mission; therefore the chance of failure of a collector over 5 years was squared to determine the chance of collector failure causing a mission failure (0.01). The probability of a MTV failure over the five year mission cycle is 0.056. Given that mission failure would be defined by a failure of two collectors or the failure of an MTV in the five year span, the total probability of loss of mission was calculated. A maintenance mission is planned after five years of service of the MTVs and therefore the five year reliability is used during subsequent missions. To further enhance reliability, there is a spare collector in orbit collecting propellant, and a replacement collector could be launched to replace a failed collector as needed.

The DRA 5 Derived mission has fewer components to take into account for mission reliability. One of the MTVs only needs to reach Mars, and therefore the reliability after half a year (0.971) can be used. The second MTV needs to return to Earth before being discarded, so the reliability after two years is used (0.962). These reliabilities are then used to determine the probability of loss of mission.

3.2 Architecture Results

3.2.1 DRA 5 Derived Architecture
The DRA 5 Derived Mars architecture was designed as a traditional, all-launch, direct transfer architecture. It utilizes two MTVs. One MTV is strictly for cargo and the other is for cargo and crew transfer. Both MTVs have nuclear thermal rocket (NTR) propulsion systems with hydrogen as the propellant. Both MTVs are non-reusable. The cargo MTV makes only a one-way transfer, is separated from the payload and heat shield, and is discarded. The crew MTV is responsible for returning the crew to Earth at the completion of the mission, so it must make a round-trip transfer. It is separated from the crew re-entry module at Earth and discarded. All mass for this architecture must be launched at the initiation of each mission. The launch mass includes two MTVs, crew payload, cargo payload and propellant mass.

3.2.2 PHARO Architecture
The PHARO enabled Mars architecture was designed and optimized to require the lowest launch mass per mission and still have the same performance capabilities (i.e. complete the same surface mission) as the traditional DRA 5 Derived design. This
architecture utilizes two fully reusable MTVs on a low thrust spiral trajectory. They have MHD-augmented NTR propulsion systems with air as the propellant. Due to the desire to limit the MHD power requirement to approximately 100 MW (due to the limits of radiator performance on the MTV), the Isp was calculated at 5000 s. Since both MTVs are reusable, they must both make round-trip transfers. The cargo MTV will spiral out to Mars where it will deploy its payload and return to Earth. At the next launch window the crew MTV will spiral out to Mars and insert into Mars orbit, awaiting the crew to rendezvous and return to Earth. For this architecture only the payload mass needs to be launched at the beginning of every mission, as the MTVs are already in LEO and the propellant is being harvested by the PHARO system. However, there is significant mass that has to be launched initially.

3.2.3 Collector
Due to the size limitations of the collector (less than 5m diameter, to be deployed on a Falcon 9) and the degree of power concentration available without requiring active cooling, the upper limit of the power requirement for the collector was 250 kW. This led to the choice of six collectors making two trips each, with a power requirement of 175 kW (more would have required additional power). This power was used to energize a 15 N of thrust, 400 s of Isp propulsion system, with the collector operating at an altitude of 123 km. The collector’s total mass was 17.5 T, with a storage capacity of 98 T, of which 30% is reserved for transfer to and from the MTVs. The mass summary for the collector is shown in Table 1.

### Table 1: Collector mass summary.

<table>
<thead>
<tr>
<th>Collector Component</th>
<th>Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tank</td>
<td>6056</td>
</tr>
<tr>
<td>Inlet</td>
<td>1017</td>
</tr>
<tr>
<td>Liquefation System</td>
<td>116</td>
</tr>
<tr>
<td>Propulsion/Power</td>
<td>1288</td>
</tr>
<tr>
<td>Structure</td>
<td>3818</td>
</tr>
<tr>
<td>Other</td>
<td>1112</td>
</tr>
<tr>
<td>Growth</td>
<td>4022</td>
</tr>
<tr>
<td>Total Mass</td>
<td>17430</td>
</tr>
</tbody>
</table>

3.3 Comparison
The results for the two architectural MTVs are shown in Table 2. Of note is the significant difference in mass launched per mission. While the PHARO architecture requires initial deployment of the dry MTVs prior to the first mission (to permit fueling), subsequent launches require only the payloads for each mission, coming to 101 T mass. By comparison, the traditional architecture must launch the full mission mass (1032 T) every mission.

### Table 2: MTV Comparison

<table>
<thead>
<tr>
<th></th>
<th>DRA 5 Derived</th>
<th>PHARO</th>
</tr>
</thead>
<tbody>
<tr>
<td>Transfer Type</td>
<td>Direct</td>
<td>Spiral</td>
</tr>
<tr>
<td>Propellant</td>
<td>Hydrogen</td>
<td>Air</td>
</tr>
<tr>
<td>Propulsion</td>
<td>NTR</td>
<td>MHD Augmented NTR</td>
</tr>
<tr>
<td>Isp</td>
<td>950 s</td>
<td>5000 s</td>
</tr>
<tr>
<td>Payload Mass (Crew)</td>
<td>62 T</td>
<td>56 T</td>
</tr>
<tr>
<td>Dry Mass (Crew)</td>
<td>272 T</td>
<td>140 T</td>
</tr>
<tr>
<td>Prop Mass (Crew)</td>
<td>481 T (Round trip)</td>
<td>450 T (Round trip)</td>
</tr>
<tr>
<td>Payload Mass (Cargo)</td>
<td>35 T</td>
<td>45 T</td>
</tr>
<tr>
<td>Dry Mass (Cargo)</td>
<td>150 T</td>
<td>140 T</td>
</tr>
<tr>
<td>Prop Mass (Cargo)</td>
<td>120 T (One way)</td>
<td>450 T (Round Trip)</td>
</tr>
<tr>
<td>Launch Mass (Initial)</td>
<td>0 T</td>
<td>279 T</td>
</tr>
<tr>
<td>Launch Mass (Per Mission)</td>
<td>1032 T</td>
<td>101 T</td>
</tr>
</tbody>
</table>

From this, it is evident that the mass required per mission for the PHARO architecture is 90% less than that of the DRA 5 Derived mission. Spread over Ares V-like launch vehicles, with a throw of approximately 109 T (Axdahl et al.), the number of launches required per mission is reduced by 80%.
3.4 Reliability

Collector reliability is plotted in Figure 7. Because the collector is an unmanned vehicle, reliability could fall to 0.6 before the collector would be required to be replaced. The reliability of the collector does not fall below 0.6 until it gets to 30 years of active flight time. Replacement of individuals collectors over this period would maintain a higher average reliability that is used as margin rather than a replacement constraint.

MTV reliability is plotted in Figure 8. As the MTV is a manned vehicle, it will be replaced when the reliability falls below 0.9. The reliability of the MTV falls below 0.9 after 10 years, and therefore will be replaced after 10 years of active flight time.

Based on the analysis described previously, the probability for loss of mission for the DRA 5 Derived architecture is 6.6%. The probability for loss of mission for the PHARO architecture is 11.8%. This higher probability is driven primarily by the need for the MTVs to function successfully for the full five year mission cycle (due to being in orbit during fueling), rather than the shorter terms for the DRA 5 Derived mission. However, some systems (propulsion, primary on-board power) would not be required during the fueling time; hence, the reliability of the MTV for the PHARO mission may be conservative.

3.5 Cost Comparison

The baseline program assumes a five year DDT&E phase, and a five year operations phase for each mission. For the fully reusable PHARO architecture, there is an initial two-year build of the collector, solar satellites, and MTVs, and a two-year deployment of the reusable architecture preceding the operational phase.

The DRA 5 Derived architecture consists of the two expendable MTVs, which require ten Ares V launches per mission to launch the dry mass, hydrogen propellant, and payload.

Table 3 shows the DDT&E, TFU, and operations and associated launch vehicle costs. A learning factor of 0.9 is assumed for all architecture element production. For the PHARO architecture, Table 3 also shows the foundation costs for the fully reusable architectural elements, as well as the initial build and deployment costs.

Figure 9 compares the yearly cumulative costs for the DRA 5 Derived and PHARO
architectures. The PHARO architecture takes five additional years to reach initial operational capability (IOC) because of the initial builds and delivery. The initial costs for to reach IOC total approximately $10B. The breakeven point is shown to be slightly over 2 missions (ten years of operations) with the lower operation costs of saving eight Ares V launches and MTV replacements; this manifests as a $1B per year savings relative to the DRA 5 Derived architecture. The breakeven point could take longer if higher reliability is mandated and a net present value analysis is applied.

4.0 Conclusion
Implementation of propellant harvesting has the potential to dramatically reduce the launch mass for a multi-mission Mars architecture. Additionally, introducing such a technology can reduce the life cycle cost of the campaign. Although such an architecture does lead to a reliability penalty relative to a DRA 5 Derived, non-harvesting concept, this penalty could be mitigated by further improvements in technology. This study shows the potential of atmospheric propellant harvesting as an enabling
technology for future humans-to-Mars missions.

5.0 Education and Public Outreach

In conjunction with the technical design work, outreach activities were performed to promote interest in mathematics and the sciences. A presentation was given to a sixth grade math class, as well as several visiting eighth graders, at Crayton Middle School in Columbia, SC. The students participated in several activities related to distance measurements, unit conversions, and orders of magnitude, as applied to understanding the distances involved in space travel. A question-and-answer forum was also held, during which students’ questions regarding NASA’s plans were answered.

Additionally, the team visited a local children’s hospital (Children’s Hospital of the King’s Daughters, Norfolk, VA). There, they gave a similar presentation to the children on the planets of the solar system, including their size, composition, and distance from the sun.
References


Wilhite, A. Notes from Spacecraft and Launch Vehicle Design I. 2009.