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# LUNA AND MARS FOR LESS: A MEDIUM-LIFT LAUNCH SYSTEM ARCHITECTURE FOR THE VISION FOR SPACE EXPLORATION

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## ABSTRACT

The current Vision for Space Exploration (VSE) requires near-term, developmentally simple manned expeditions to the moon and Mars to restore confidence in American programs of human spaceflight. Pursuant to these requirements, it may be necessary that robust, cost-sustainable mission architectures be initially de-prioritized in favor of those which – while more operationally complex – are programmatically easier to initiate with current technology.

This study proposes a method of undertaking manned expeditions to the moon and Mars with existing medium-lift launch systems. In this architecture, 20-tonne high energy propulsion stages are placed individually in low-Earth orbit, where they are mated to moon or Mars-bound spacecraft and ignited at successive perigees to execute trans-planetary injections. Contrary to NASA mission plans which utilize evolved expendable launch vehicles (EELVs), this architecture does not require complex spacecraft systems integration on orbit, nor does it necessitate orbital construction beyond initial spacecraft rendezvous and docking. Most significantly, the mission architecture in this paper does not require the development of heavy-lift launch vehicles (HLLVs): highly effective yet costly prerequisites which could fatally hinder the current Vision for Space Exploration. It is concluded that the mission design herein represents a less robust, though easier to develop method of conducting manned expeditions to the moon and Mars than current EELV or HLLV-based mission designs – an alternative method by which space may be opened to humanity.

## BACKGROUND

Decades of analysis have been dominated by Mars mission architectures that require heavy-lift launch vehicles (HLLVs) to initiate[1,2]. Whether involving orbital assembly of large, 1000-tonne spacecraft or – more practically – the direct throw of smaller payloads onto Mars-bound trajectories, the vast majority of mission designs require the development of launch systems with at least 100-tonne to low-Earth orbit (LEO) capacities, lacking since the days of Apollo.

Many have proposed developing such heavy-lift launch technology from NASA's Space Transportation System (STS), which alone can place approximately 80 tonnes in LEO without modification. Typical designs for STS-derived boosters involve either replacing the shuttle orbiter with an expendable cargo pod, or extending the external tank fairing vertically to accommodate a payload mounted to a high-thrust upper stage.

However, confidence in the reliability of the Space Transportation System wears increasingly thin. The recent failure of the shuttle Columbia has only created more doubt in the aerospace community about a launch system that, by all accounts, should have been retired some time ago[3]. Accordingly, NASA has slated the STS to be decommissioned in the next decade – and to base the future of manned spaceflight on its longevity would seem both problematic and inadvisable.

Moreover, the modification of the STS to carry a payload other than the orbiter is a potentially expensive project; and while certainly not impossibly expensive with respect to available resources, it nevertheless requires significant change – something that NASA has strongly opposed regarding its launch systems, and another significant obstacle that any STS-based mission design must negotiate. While there do exist other options for heavy lift launchers, such as the Russian Energia, arguments levied against the STS can be similarly applied, with the added fact that Energia hasn't flown in years. Reworking it now, in a space exploration climate even more unstable and uncertain than in the United States, would seem difficult at best.

Developing heavy-lift launch technology is neither easy nor inexpensive; but while many would argue that such costs and efforts would be justified, the decision makers have thus far disagreed. Eventually, engineers may be forced to abandon the HLLV prerequisite in their mission designs – the need for such technology may permanently anchor manned space exploration to the ground.

## MARS FOR LESS

The *Mars for Less* architecture circumvents the need for heavy-lift vehicles entirely. Rather than create a mission that necessitates as-yet undeveloped technology, this study designs around what exists and is being flown *today*: specifically, medium-lift launch systems with 20-tonne to low orbit payload capacities. There are several boosters in this category presently in use, the majority of which cost between $100 and $200 million dollars – or for perspective, between 1/6 and 1/3 the cost of a single shuttle launch. By dividing a Mars-bound spacecraft and its propulsion into smaller components, and launching them on existing rockets rather than proposed heavy-lift boosters, the need for new launch systems can be eliminated; consequently, the only new vehicles to develop are those actually being sent to Mars, resulting in a program of exploration that is fundamentally easier to initiate.

### Mission Overview

*Mars for Less* is essentially Robert Zubrin's *Mars Direct* mission architecture, divided into smaller components and launched by systems in use today. Comprehensively discussed in reference [2], each complete mission in the *Mars Direct* plan requires two spacecraft: an Earth Return Vehicle (ERV) which flies to Mars unmanned, and fuels itself automatically with propellants manufactured from indigenous resources; and a crewed Mars Transfer and Surface Vehicle (MTSV), which houses astronauts during the outbound leg of each mission and for the majority of each surface stay.

Employing the same mission philosophy, the *Mars for Less* plan would require 6 medium-lift boosters per complete spacecraft. For each vehicle, the first launch would deliver a two-level common habitat element, to which the second would mate mission-specific modules, consisting of propulsion stages, hydrogen feedstock and a chemical reactor for the ERV and garage/storage modules for the MTSV. Subsequent boosters would deliver 4 high-energy propulsion stages to each payload, which would be mated aft like train cars and ignited at successive perigees, widening spacecraft orbits until final stages impart sufficient energy to initiate trans-Mars injection (TMI). The Earth Return Vehicle would proceed to Mars on a near-minimum energy transfer of approximately 260 days, while the crewed launch would be assembled 2 years later and follow on a slightly faster trajectory of 120 to 190 days depending on the specific opportunity. Piloted spacecraft have the option of generating tether-based artificial gravity using burnt-out TMI stages as counterweights. Each spacecraft would aerobrake into Mars orbital capture (MOC) and land with the aid of parachutes and rockets at a common site. As in *Mars Direct*, each crew would remain on the surface for 500 days, during which broad exploration could be performed. Astronauts would be afforded the natural protection of the Martian atmosphere from the hazards of interplanetary space for the duration of their stay, at the conclusion of which they would board their fully-fuelled ERV and launch directly from the surface of Mars back to low-Earth orbit. Upon return, it may be possible to refit the habitat module of the ERV for reuse on subsequent expeditions.

While mating spacecraft and propulsion stages in orbit is less robust an approach than launching directly from the Earth's surface, it does represent an easier architecture to *develop*, having been designed around systems already flying today and which require no modification. The 20-tonne components of *Mars for Less* can be placed in LEO using any combination of existing medium-lift boosters, such as the Russian Proton, French Ariane V, Lockheed Titan IVB or Boeing Delta IV Heavy. Only the absolute minimum of technology need be developed for missions conducted this way, with the only new vehicles being those sent to Mars: a fact that may bring piloted expeditions within the grasp of private ventures.

While any Mars mission design involving orbital assembly may have been frowned upon a decade ago, today's space agencies have since acquired considerable experience mating spacecraft in Earth orbit through operations involving Mir and Space Station Alpha; and that notwithstanding, the connection of propulsion stages in LEO is not nearly as difficult as assembling space stations or candidate spacecraft from past studies, in which systems integration requirements are considerably more complex. No life support or redundant power systems need be extended through a Mars Transfer Vehicle's modular propulsion, nor would any support infrastructure be required for spacecraft assembly (i.e. – there is no need for space stations, lunar colonies, or any other dedicated construction outposts). Using existing launch vehicles to assemble *Mars Direct*-style missions would allow for a dramatic reduction in mission development costs, which is essential for initiating – if perhaps not perpetuating – the current Vision for Space Exploration.

## MISSION CONSIDERATIONS

### Interplanetary Trajectories

The lowest energy transfer between two orbits is known as a Hohmann transfer; named for the German engineer and architect who derived its properties, the Hohmann transfer is an elliptical trajectory whose periapsis and apoapsis are tangent to the departure and arrival orbits respectively. The design of a Hohmann transfer is based on its semi-major axis, or _ the long axis of the transfer ellipse; for a minimum-energy trajectory, this can be approximated as an average of the departure and arrival orbits' mean radii, or:

$$a = \frac{r_{earth} + r_{mars}}{2} \qquad (1)$$

With $r_{earth} = 1.50\text{x}10^8$ km, and $r_{mars} = 2.28\text{x}10^8$ km. The departure, or *hyperbolic* velocity needed for a spacecraft to enter a heliocentric transfer orbit is its relative velocity about the sun with respect to Earth at the beginning of its trajectory, or:

$$v_h = v_{transfer} - v_{earth} \qquad (2)$$

Where $v_{earth}$ is the Earth's orbital velocity around the sun and $v_{transfer}$ is the spacecraft's velocity at the beginning of the transfer. Expressing $v_{earth}$ and $v_{transfer}$ in terms of the Earth's radius and the transfer semi-major axis, equation (2) becomes:

$$v_h = \left[\mu_{sun}\left(\frac{2}{r_{earth}} - \frac{1}{a}\right)\right]^{\frac{1}{2}} - \left(\frac{\mu_{sun}}{r_{earth}}\right)^{\frac{1}{2}} \qquad (3)$$

where $\_{sun}$ is the sun's gravitational constant ($\approx 1.33\text{x}10^{11}$ $km^3 s^{-2}$). A minimum-energy transfer between Earth and Mars has a semi-major axis of approximately $1.89\text{x}10^8$ km, and substituting this value into equation (3) gives a hyperbolic velocity of approximately 3.0 km/s, which is the velocity needed by each spacecraft at a point where the Earth's gravitational influence becomes negligible. The velocity change needed to achieve this hyperbolic velocity is given by:

$$(\Delta v + v_0)^2 = v_h^2 + v_{esc}^2 \qquad (4)$$

In a typical low-Earth orbit of 185 km, $v_0$ = 7.8 km/s, $v_{esc}$ = 11.0 km/s, and the total _$v$ required for the Hohmann transfer, excluding gravity losses, is 3.61 km/s.

While Hohmann transfers are ideal for launching Earth Return Vehicles and other cargo to Mars, such trajectories have flight times of almost 260 days – and while that may suffice for unmanned missions, faster transfers are desired for piloted spacecraft.

By increasing the semi-major axis of the transfer ellipse, an orbit can be created that intersects Mars much sooner; however, the extent to which travel time can be reduced is restricted by the _$v$ budget of the rockets being used and by the speeds with which Mars-bound spacecraft encounter the red planet's atmosphere. Choosing a trajectory leaving Earth with $a = 2.35\text{x}10^8$ km and $v_h$ = 5.0 km/s allows for transit times between 120 and 190 days depending on the specific opportunity, and which is also suitable for a 2-year free return trajectory if the need arises. Substituting into equation (8), we find the velocity change required for this transfer to be 4.30 km/s excluding performance losses; much more suitable for manned flights.

**Propulsion Systems**

Dividing spacecraft propulsion into stages allows a vehicle to discard otherwise parasitic mass incrementally. A staged spacecraft abandons each unit after its propellant has been exhausted – thus, no further energy is required to accelerate empty stages after separation, and the overall system does not incur performance penalties associated with extra mass.

Consider a rocket of mass $m + dm$, which discards propellant mass $dm$ during an instant $dt$ at a constant exhaust velocity $c$. Letting $dv$ represent the velocity gained by the rocket, and applying conservation of linear momentum, the expression:

$$mdv = -cdm$$

is obtained. Differentiating with respect to time produces the force equation:

$$m\frac{dv}{dt} = -c\frac{dm}{dt}$$

Accounting for external forces (drag and gravity losses) and dividing through by $m$:

$$\frac{dv}{dt} = -\frac{c}{m}\frac{dm}{dt} - g\sin\theta - \frac{D}{m}$$

Where $D$=__$AC_dv^2$. Separating variables and integrating:

$$\int_{v_1}^{v_2} dv = \Delta v = c\ln\left(\frac{m_0}{m_f}\right) - \int_{t_1}^{t_2} g\sin\theta dt - \int_{t_1}^{t_2}\frac{D}{m}dt \qquad (4)$$

This is a form of the rocket equation; in the absence of external forces, the total velocity change that a rocket stage can impart to a spacecraft can be idealized according to:

$$\Delta v_i = c\ln\left(\frac{m_{0i}}{m_{fi}}\right) = gIsp_i\ln\left(\frac{m_{0i}}{m_{fi}}\right) \qquad (5)$$

Where *Isp* is the specific impulse of the stage in question; $m_{oi}$ is the mass of the spacecraft when stage $i$ is ignited; and $m_{fi}$ is the final mass of the spacecraft including the burnt-out stage when all fuel is exhausted. The total _$v$ that a system with $n$ stages can generate is the sum of the individual stage _$v$ values, or:

$$\Delta v_{total} = \Delta v_1 + \Delta v_2 + \ldots + \Delta v_n = \sum_{i=1}^{n} \Delta v_i \qquad (6)$$

An upper limit on $n$ must be imposed on the focus mission – while any number of 20-tonne stages can be used to propel spacecraft to Mars, for every additional stage another launch is needed to lift it, and each additional launch increases both cost and complexity. At some point, the savings achieved in mission development would be negated by the sheer expense of putting hardware in orbit.

**Hydrogen/Oxygen ($H_2/O_2$) Stages:**

The most efficient bipropellant in use today, rockets burning $H_2/O_2$ can yield specific impulses between 440 and 465 seconds. If hydrogen/oxygen propellants are used, then both cargo and piloted flights to Mars can be accomplished with only 4 propulsion stages, assuming individual dry masses of 2.5 tonnes, propellant masses of 17.5 tonnes, 500 kN thrust and *Isp* = 450 seconds.

Hydrogen/oxygen bipropellants are very cryogenic, and will tend to boil off when stored in space for appreciable amounts of time. Fortunately, this effect can be largely mitigated using multi-layer insulation – 50 sheets of MLI (specifically, insulation consisting of aluminized capton with dacron net separations) can slow the rate of $H_2/O_2$ propellant loss to roughly 1% of the total mass per month[4]. While options may exist for chilling oxygen tanks with hydrogen gas as it boils away, and thus potentially restricting propellant losses to ≈0.55% per month, the feasibility of such a system is beyond the scope of this paper, and its use is not assumed.

The performance degradation of a 4-stage $H_2/O_2$ system resulting from propellant losses between 1 and 6 months on orbit is shown in Graph 1. For all intervals considered, the total _*v* capability after losses remains sufficient to launch at least cargo missions, though times ranging between 3 to 5 months offer the best, most realistic schedules for both cargo and manned spacecraft assembly.

**Methane/Oxygen ($CH_4/O_2$) Stages:**

Methane/Oxygen is the highest-energy space storable propellant available, with a specific impulse ranging from 370 to 380 seconds. In addition to being less cryogenic than hydrogen, liquid methane has the added advantage of higher density (162 $kg/m^3$ as opposed to 71$kg/m^3$); this not only allows for more compact fuel tanks, but also serves to further restrict boil-off, the rate of which being thermodynamically governed and proportional to the exposed surface area of a fluid.

With an individual dry mass of 2 tonnes, propellant mass of 18 tonnes and a specific impulse of 375 seconds, 5 $CH_4/O_2$ stages would be needed to launch both cargo and piloted spacecraft in the focus mission; if the specific impulse was increased to 390 seconds, then cargo could be launched with only 4 propulsion stages, though this is not assumed.

Since methane/oxygen is the ideal fuel for Earth Return Vehicles, as it can be readily produced on Mars, it may be possible to design common rocket engines for both TMI and ERV propulsion.

Even partial commonality between Mars-bound spacecraft and their stages would be extremely beneficial in terms of development costs; the specifics of this, however, are beyond the scope of initial mission design.

Propellants with lower energy than methane/oxygen – such as kerosene/oxygen or mono-methyl hydrazine/nitrogen tetroxide – can all successfully launch cargo to Mars; for crewed flights however, greater than minimum energy trajectories are desirable, and propellants with lower performance cannot expedite piloted flight times in less than 6 stages. Therefore, the focus mission restricts TMI propulsion consideration to hydrogen/oxygen and methane/oxygen systems, in the interest of minimizing the number of launches required for vehicle assembly.

## VEHICLE DESIGN AND ASSEMBLY

The principal design consideration for both spacecraft in the *Mars for Less* architecture is that their components be compatible with existing launch systems – vehicles must be modularly designed to fit within the payload dimensions and mass limits of today's medium-lift boosters, and must be placed in orbit with as few launches as possible.

### Structure

The recommended spacecraft for the focus architecture would have exterior diameters not exceeding 5 meters, and either folding or rigid-conical aeroshields that can be adapted to the payload fairing of candidate launch systems. A common crew cabin for both Mars transfer and Earth return spacecraft is assumed in all reference designs, with additional sections dedicated to storage or propulsion systems respectively. Both vehicles would be conical in design, optimized primarily for vertical takeoff, flight, and landing. With only one load path, structural reinforcement can be restricted to one principle axis, thus sparing otherwise significant mass penalties.

The common habitat for both spacecraft would be two levels, with additional vehicle-specific units consisting of a storage compartment for the MTSV and 2 methane/oxygen stages for the ERV. Habitable sections would be connected by a half-cylinder 1.4 meters in diameter running through the center of the spacecraft; with deck 1 containing the galley, washroom and two sleeping quarters, and deck 2 consisting of laboratory and exercise sections, command systems, and two additional sleeping quarters. There would be three central access points per deck: swinging hatches would open into common areas, and metal sliding doors on the flat section of the hub would permit access to individual crew quarters. The curved portion of the hub would be filled with water, and this combined with absorbent metal doors would serve as a protective environment from radiation in the event of a solar flare. The sleeping quarters for the crew would be small but functional, permitting each astronaut approximately 40 square feet of floor space and 8 feet of overhead. Beds can be elevated by about 4.5 feet, beneath which storage or work areas might be placed to conserve space.

Aluminium alloys are assumed as primary materials for spacecraft structures; such alloys are low cost, possess high strength to weight ratios, and are generally easy to machine and form. While aluminium-lithium alloys in the 8xxx series are increasingly being utilized as aerospace materials on account of their lower density and superior mechanical properties, it is specifically recommended that such materials not be considered for major structural components here. 8xxx alloys exhibit high mechanical anisotropy, and are particularly weak under compressive loading conditions that reduce or eliminate crack closure, such as those typical to spacecraft takeoff and landing. Moreover, high crack growth rates for microstructurally short cracks have been observed, which can cause early cracking in regions of high stress concentration; this represents an especially problematic phenomenon during prolonged spaceflight[5]. Consequently, aluminium material considerations are restricted to alloys in the 2xxx, 6xxx and 7xxx series.

Spacecraft aeroshields may be either composite or ablative. Fasteners and attachment fittings for advanced composites would be titanium alloys, and nickel or cobalt-based alloys offer the best properties for areas of the vehicle exposed to higher-heating, especially leading edges and secondary thermal protection systems.

**Thermal**

Thermal control subsystems would be kept as simple as possible, with passive radiators positioned exterior to the spacecraft above the aeroshield, linked by conduction bars and heat pipes. If deemed necessary, active cooling could also be employed, which would cycle fluids to hotter areas of the hull – such a system working through the aeroshield may be used to reduce its mass, though the specifics of this are beyond the scope of initial reference designs.

Mars orbital capture (MOC) and atmospheric entry profiles would be ballistic in nature to minimize the amount of thermal protection required; while spacecraft such as the shuttle tend to fly on lifting trajectories rather than ballistic, such profiles actually maximize the surface area subjected to high heating as well as flight time. The recent failure of the shuttle Columbia unfortunately serves to illustrate this point – the less ballistic the trajectory, the longer the spacecraft is exposed to thermal stressors, which significantly increase failure potential for heat shields and protection systems. Spacecraft fuselage may be curved slightly, angled inwards to protect against aerodynamic flow-field effects – the specific design of aeroshields and their duration of use would dictate whether this would be necessary.

All vehicles in the focus architecture would be exposed to the Martian surface for lengthy periods of time; consequently, the environmental impact on long duration stays must be considered, especially for each mission's return transportation. The ERV is the primary mission-critical concern after the crew is safely landed, and once on the surface it is recommended that ERV propulsion be kept completely sealed from the external environment. Temperatures on Mars can vary by as much as $100^{o}$C daily, which can induce cracks in spacecraft materials. The primarily aluminium structures of reference spacecraft will not experience ductile-to-brittle behaviour under temperature changes (aluminium has a face-centred cubic structure, which maintains a large number of slip systems even at low temperatures[6]); however, such materials are susceptible to phenomena such as pitting, crevice and galvanic corrosion. Martian fines – which are microns in size – are often of a highly reactive species, and can be expected to compromise spacecraft materials during prolonged exposure. Care must be taken to treat or anodize exposed areas of vehicles to combat these forms of deterioration.

**Power**

While in space, power for both vehicles would be provided by a 5 kWe photovoltaic array, and propulsion stages would be battery-powered. Once on Mars, these systems would be supplemented with the 100 kWe nuclear reactor imported for propellant production by the Earth Return Vehicle; in addition to supporting surface energy requirements, the radioisotope generator would allow astronauts to transmit considerably more data back to Earth than if communications were exclusively solar-powered.

**Propulsion**

$N_2O_4/N_2H_4$ reaction control systems with *Isp* = 330 seconds are assumed for both spacecraft. RCS rockets would control spacecraft attitude during orbital assembly, and replacement fuel modules that fit within the excess LEO capacity of early launches would be delivered to supplant propellant expended on orbit. For both spacecraft, these thrusters would perform the approximately 30 m/s mid-course correction required in transit, and would aid with orbit and descent manoeuvres at Mars. For the MTSV, this system would be more than capable of generating the maximum velocity change needed to return a stranded spacecraft from between Earth-escape and minimum-energy Mars transfer, or approximately 350 m/s. If the total _*v* capability of both spacecraft was increased to 500 m/s, which would allow them to land on Mars without the aid of parachutes, the additional propellant required would be approximately 1 tonne for the MTSV and 2 tonnes for the ERV, enhancements which are manageable with candidate propulsion systems.

**Assembly**

Each spacecraft would require two launches to place in Earth orbit. For the Earth Return Vehicle, launch one would deliver the empty propulsion stages, nuclear reactor, propellant plant, hydrogen feedstock and aeroshield, a total mass of 19.2 tonnes. The second launch would contain the ERV cabin, massing 15.8 tonnes for a combined mass of 35 tonnes. The first MTSV launch would carry a 17 tonne habitat module and its supplies, to which the second would mate the storage compartment and shield for a total of 30.1 tonnes. After each vehicle is delivered to orbit, TMI stages would be launched and mated one by one; the base-mounted aeroshields of both spacecraft would initially be oriented to protect propulsion stages from solar heating for the duration of orbital operations.

Integration of spacecraft and propulsion subsystems in this architecture would be far less complex than in previous studies; structurally, rocket stages would be connected to each other with a simple truss, ideal for supporting axial loads (no shear or moment). Truss designs for this application would essentially be statically determinate, having no more components than are necessary to introduce and react propulsion loads.

**Cabin Recovery**

At the conclusion of surface exploration, the Earth Return Vehicle would launch directly from Mars on a low-energy return to Earth; the habitat module would possess its own aeroshield, which would be used to capture back into Earth orbit. The commonality between MTSV and ERV habitats make it possible to reuse this vehicle, either by keeping it in Earth orbit, or returning it to the surface for refit.

The mass allocations for reference spacecraft are presented in Table 1, with science equipment masses elaborated upon in Table 2. The number of crew members per mission is limited to 4; the rationale for this choice is discussed at length in reference [2].

## ORBITAL RENDEZVOUS

Typical Mars mission studies involving orbital assembly have called for huge, 1000-tonne plus spacecraft to be built in orbiting drydocks and fuelled with cryogenic propellants in LEO. Such designs have caused many to immediately disregard even the idea of orbital assembly, and with just cause. While much simpler than previous studies, the focus approach is still less robust than missions involving simpler, direct throws of payload with heavy-lift vehicles.

However, while orbital construction architectures of any variety may have been non-starters a decade ago, today's space agencies have acquired much more experience with such operations; and as mentioned, mating candidate spacecraft in this architecture would be far less complex than assembling space stations or spacecraft designs from previous studies.

Docking one vehicle to another involves a phasing orbit which allows the interceptor to achieve a specific geometry with respect to the target, and then initiating a Hohmann transfer to a corresponding rendezvous point. For coplanar interceptions, the spacecraft remains in its initial orbit until the desired geometry is achieved, at which point it injects onto a minimum-energy transfer. The wait time in the initial orbit prior to this burn is the relative angular distance between the spacecraft in question divided by their relative angular velocity, or:

$$t = \frac{\phi_i - \phi_f + 2(k-1)\pi}{\omega_{\text{int}} - \omega_{target}} \qquad (6)$$

Where $\phi_f$ is the phase angle (angular separation of target and interceptor) needed for rendezvous, $\phi_i$ is the initial phase angle, k is the number of rendezvous opportunities, and $_{int}$, $_{target}$ the angular velocities of the interceptor and target spacecraft respectively. The total time required for the rendezvous is the transfer flight time (the period of the transfer orbit) plus the wait time[7], and the $\Delta v$ which the interceptor must generate to accomplish this linkup is the magnitude of its relative velocity with respect to the target at the rendezvous point. The initial and required velocities of each spacecraft in a conservative system can be obtained using the *specific mechanical energy,* or *energy per unit mass* equation:

$$\varepsilon = \frac{v^2}{2} - \frac{\mu_{earth}}{r} = -\frac{\mu_{earth}}{2a} \qquad (7)$$

The target orbit for focus mission spacecraft and propulsion stage mating is a typical low-Earth orbit of approximately 185 kilometres, and NASA has been docking spacecraft at comparable altitudes since the 1960s. Assembling large interplanetary spacecraft with dedicated construction and fuelling outposts in LEO would be a decided detriment to any Mars mission, both in terms of cost and complexity. But docking propulsion stages to Mars-bound payloads without committed infrastructure or propellant transfer is not nearly as problematic, and represents an acceptable trade-off for reductions in mission development costs.

## PERFORMANCE LOSSES

### Gravity Losses

Returning to the expression derived earlier for $\Delta v$ capability,

$$\Delta v = c \ln\left(\frac{m_0}{m_f}\right) - \int_{t_1}^{t_2} g \sin\theta dt - \int_{t_1}^{t_2} \frac{D}{m} dt$$

The integrals on the right hand side of the equation respectively denote the gravity and drag losses which a rocket incurs during a flight interval $[t_1,t_2]$. Since the capabilities of focus mission launch systems have all been previously characterized, analysis here is restricted to performance losses during trans-Mars injection; and appropriately, for manoeuvres above 150km altitude, $D \approx 0$.

Gravity loss is a function of both time and flight path angle, the latter of which is variable on $[t_1,t_2]$, and increases as spacecraft trajectories go from elliptical to parabolic. From the Taylor series expansion of sin_, for small angles we can make the first order approximation sin_ ≈ _ and rewrite the gravity loss expression as:

$$\Delta v_{loss} = -g\int_{t_1}^{t_2} \theta(t) dt \qquad (8)$$

where the flight path angle is treated as a function of time. For both spacecraft in the focus mission, TMI propulsion stages would be ignited individually at sequential perigees, where they would be travelling at maximum speed. Since kinetic energy varies as the square of velocity, executing burns deep in the Earth's gravity well allows each rocket stage to achieve maximum performance, permits total burn time to be divided evenly between individual stages, and allows flight path angles to be minimized.

### Propulsion Requirements

In the focus mission, the total _*v* requirements for spacecraft departing low-Earth orbit, including 7 % gravity losses, are 4.0 km/s for Earth Return Vehicles and 4.6 km/s for crewed Mars Transfer Vehicles. Plane changes, mid-course corrections, post-aerocapture manoeuvres and landing are achieved using spacecraft reaction control systems.

## LAUNCH SYSTEMS

Table 3 shows characteristics for candidate medium-lift vehicles, with low-Earth orbit given as 185 km circular, inclined 28.5 degrees. Optimal vehicles are the Atlas V, Ariane V and Proton variants, which offer the lowest costs-per-unit at present.

## LUNA FOR LESS

While Mars represents the probable hub through which manned exploration of the solar system will progress in coming centuries, many still champion the moon as humanity's primary near-term goal in spaceflight. In addition to scientific prospects, the moon also offers an ideal training ground for astronauts bound for more distant locations, as well as an estimated million tonnes of Helium-3 ($^3He$), which does not occur naturally on Earth and is the ideal fuel for deuterium-helium fusion reactors, important for mankind's increasing energy requirements. While some have viewed the extraction of $^3He$ from lunar regolith as an unrealistic proposition, the prospects of obtaining it elsewhere – such as the outer gas giant planets – are considerably more problematic, and will remain so for at least several decades. In short, the moon presently has much to offer humanity – and proposals for manned Mars expeditions are only improved if launch systems and spacecraft can also be used for lunar exploration.

The ability to conduct lunar missions is an integral component of Zubrin's *Mars Direct* architecture: in the *Mars Direct* plan, Earth Return Vehicles are optimized such that their upper stages, when fully-fuelled, are capable of propelling a return capsule through the approximately 2.8 km/s needed for Earth return from the lunar surface, assuming that vehicles aerobrake into Earth-orbital capture. In keeping with this design philosophy, the focus mission is similarly structured: for lunar mission implementation, a set of four 20-tonne propulsion stages would be launched into orbit, to which a fully fuelled ERV upper stage and Crew Return Capsule (CRC) would be mated, massing 21.61 tonnes. The propulsive $\Delta v$ which this assembly can generate, including 7% gravity losses, is approximately 5.9 km/s; enough to land the completed TEI-stage/CRC assembly on the surface of the moon when RCS propulsion is accounted for. After the CRC is safely landed, another 4-stage assembly would be placed in orbit, and would dispatch a crewed MTSV (or LTSV, in this case) to the same common site, excluding the aeroshield and several tonnes of consumables. The remainder of this scenario would play out as discussed in reference [2]; with considerably less restriction on return opportunities, the crew could fly back to Earth essentially whenever they please – a significant advantage which potential Mars explorers cannot be afforded. Mass allocations for this mission's spacecraft are outlined in Table 4. Lunar missions conducted according to this plan would require 5 total launches per spacecraft, or 10 medium-lift boosters per mission.

## COST ESTIMATES

Cost estimates for Mars missions using hydrogen/oxygen and methane/oxygen propulsion stages are presented in Tables 5 and 6. All figures assume spacecraft development costs of $15 million per tonne, with recurring per vehicle expenditures equal to 75% of initial investments. Individual mission estimates are based on single launch costs averaging $100 million, assuming each complete spacecraft is placed in orbit by a combination of Delta IV-Hs, IVs, Protons, and Ariane Vs. Each spacecraft dispatched to Mars is estimated to cost $1.25 billion using hydrogen/oxygen stages, for a total mission cost (with 2 complete spacecraft) of $2.5 billion.

## DISCUSSION

The true advantage of the focus architecture is not in its recurring costs, which are on-par with past studies; rather, the principle benefit exists in achieving radically low *development* costs: with no need for new launch technology, cost projections for initiating this architecture are at least an order of magnitude lower than any conventional Mars mission plan, even with a contingency factor of 100% taken into account.

The key disadvantages lie in the need for orbital assembly and multiple launches; but as discussed, the orbital assembly requirements in this proposal are nowhere near as complex as assembling large, 1000-tonne spacecraft or space stations – the latter with which today's space programs actually have considerable experience. With no dedicated outposts needed to construct or fuel candidate spacecraft, Mars-bound payloads can simply be mated to independent propulsion systems in LEO and dispatched directly to the red planet.

While the need for multiple launches may seem problematic in terms of increased mission complexity, this requirement may actually prove to be this plan's greatest strength. Indeed, multiple launches allow each mission to have a high level of redundancy: with the exception of the booster carrying the mission's crew, failure in any one launch can be offset by delivering a replacement component to orbit; conversely, any program of Mars exploration predicated entirely on the use of a single, dedicated heavy-lift system may find itself indefinitely delayed or permanently grounded as a result of even one failed launch.

Additionally, the sheer number of different launch systems capable of orbiting the requisite mission mass significantly increases the potential for international cooperation. For example, if the United States designed, built, and launched the 2 Mars-bound spacecraft required per mission, with Canada, Russia, France and Japan contributing the TMI propulsion stages for each, then the cost per nation could average $300 million dollars for each complete expedition, or approximately $150 million per year. Put simply, this is *dirt-cheap* for major space programs and well worth the investment if it means opening another planet to our species.

These estimates assume the use of existing medium-lift launch technology, and future systems will only further reduce per mission costs. The potential of single-stage to orbit (SSTO) spacecraft or supersonic-combustion ramjets ("scramjets") remains untapped to date; but if launch vehicles can be made to operate as conventional aircraft do, with full reusability and short turnaround times, the cost of orbiting Mars mission hardware – indeed, any payload – could drop by at least another order of magnitude.

## CONCLUSIONS

The architecture developed in this paper represents an alternative method by which manned exploration of both the moon and Mars can be initiated in the near-future; by eliminating the need for heavy-lift launch technology and dividing mission mass into components that can be placed in orbit with boosters flying *today*, the cost of both developing and launching piloted expeditions can be reduced by an order of magnitude over prevailing design studies. In the postshuttle world, a mission design such as this may simply be the only cost-effective method by which near-term lunar or Mars exploration can be initiated – and the current Vision for Space Exploration realized.

## TABLES AND GRAPHS:

**Graph 1: Propellant Boil-off Impact on Delta-V Capability**

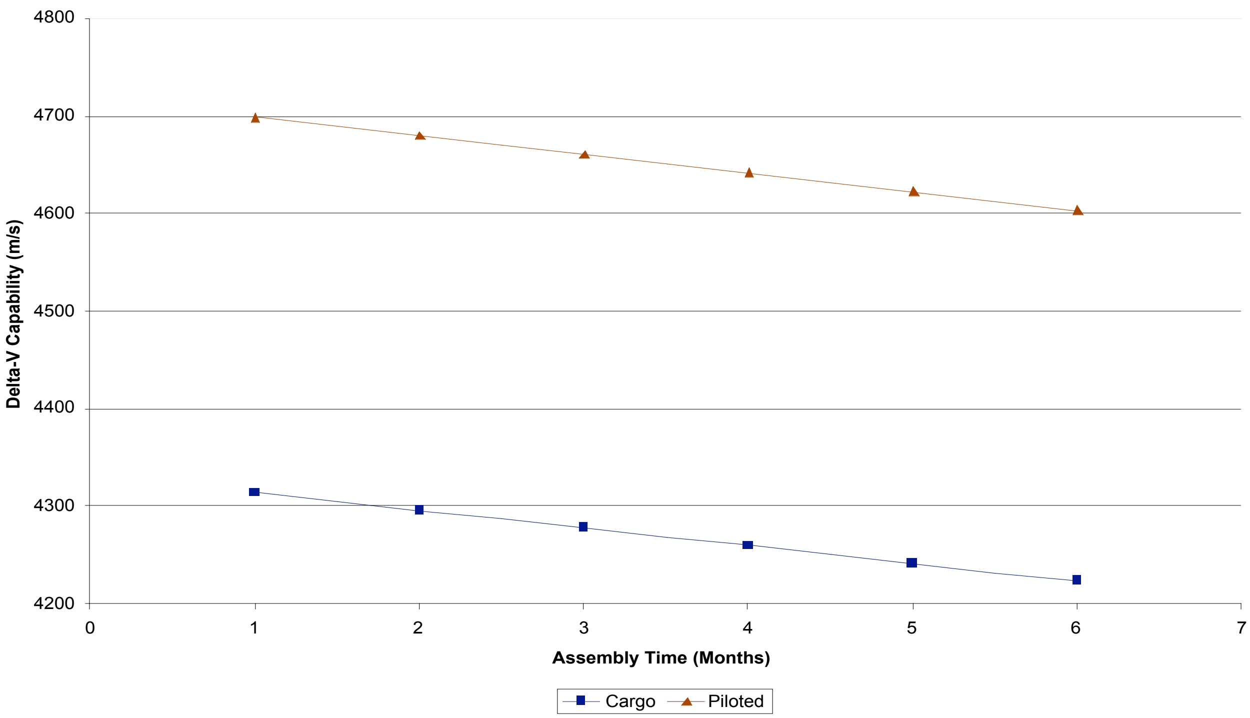


**Table 1: Mass Allocations for Earth Return and Mars Transfer Vehicles**

| Allocation – ERV | Mass ($10^3$kg) | Allocation – MTSV | Mass ($10^3$kg) |
|---|---|---|---|
| Habitat | 3.0 | Habitat | 3.0 |
| Life Support | 1.0 | Garage | 2.0 |
| Consumables | 3.4 | Life Support | 3.0 |
| Power (5 kWe PVA) | 1.0 | Consumables | 7.0 |
| Reaction Control System | 0.5 | Power (5 kWe PVA) | 1.0 |
| RCS Propellant | 3.3 | Reaction Control System | 0.5 |
| Comm. and Info. Management | 0.1 | RCS Propellant | 3.1 |
| Interior | 0.5 | Comm. and Info. Management | 0.2 |
| EVA Suits (4) | 0.4 | Science Equipment | 2.4 |
| Heat Shield | 1.6 | Crew (4) | 0.4 |
| Crew Allowances & Margin | 1.0 | EVA Suits (4) | 0.4 |
| RTG Truck | 0.5 | Interior | 1.0 |
| Hydrogen Feedstock | 6.3 | Open Rover | 0.4 |
| Propulsion Stages | 4.5 | Pressurized Rover | 1.4 |
| Propellant Production Plant | 0.5 | Crew Allowances | 0.4 |
| RTG (100 kWe) | 3.5 | Aeroshield | 3.93 |
| Aeroshield | 3.93 | **Total** | **30.13** |
| **Total** | **35.03** | | |

**Table 2: Science Equipment Masses**

| *Surface Science Equipment* | (kg) |
|---|---|
| Field Geology Package | 300 |
| Geosciences Laboratory Equipment | 110 |
| Exobiology Laboratory | 50 |
| Traverse Geophysical Instruments | 275 |
| Geophysical/Meteorology Instruments | 75 |
| 10-Meter Drill | 260 |
| Meteorology Balloons | 200 |
| Biomedical/Bioscience Laboratory | 300 |
| Discretionary Science | 200 |
| **Total** | **1770** |
| ***Cruise Science Equipment*** | (kg) |
| Particle and Field Science | 100 |
| Astronomy Instruments | 200 |
| Small Solar Telescope | 100 |
| Biomedical Instruments | 200 |
| **Total** | **600** |

**Table 3: Medium-Lift Vehicle Characteristics**

| Launch System | Upper Stage (if any) | LEO Capacity (kg) | Diameter (m) | Length (m) | Cost |
|---|---|---|---|---|---|
| Shuttle | – | 24 400 | 4.5 | 18.0 | $600 Million |
| Titan IVB | Centaur | 21 681 | 4.5 | 12.8 | $200 Million |
| Ariane V | L-9 | 18 000 | 4.5 | 12.0 | $150 Million |
| Proton | D1 | 20 900 | 4.1 | 15.6 | $70 Million |
| Proton K | DM | 20 100 | 4.5 | 15.0 | $70 Million |
| Proton M | Breeze M | 22 000 | 4.5 | 15.0 | $100 Million |
| Delta IVB | Delta 4H-2 | 20 500 | 5.0 | 18.0 | $170 Million |
| Atlas V | Cryo. RL-10a | 18 000 | 5.0 | 23.0 | $90 Million |

**Table 4: Mass Allocations for Lunar Mission Implementation**

| CRC – Allocation | Mass ($10^3$ kg) | LTSV - Allocation | Mass ($10^3$ kg) |
|---|---|---|---|
| Capsule | 1.0 | Habitat | 3.0 |
| Life Support | 0.5 | Garage | 2.0 |
| Consumables | 0.4 | Life Support | 3.0 |
| Power (2x2kWe Battery) | 0.3 | Consumables | 3.4 |
| Comm. and Info. Management | 0.1 | Power (5 kWe PVA) | 1.0 |
| Interior | 0.25 | Reaction Control System | 0.5 |
| Heat Shield | 1.3 | RCS Propellant | 3.0 |
| Reaction Control System | 0.5 | Comm. and Info. Management | 0.2 |
| RCS Propellant | 2.5 | Science Equipment | 2.4 |
| Margin | 0.75 | Crew (4) | 0.4 |
| **Total** | **7.6** | EVA Suits (4) | 0.4 |
| | | Interior | 1.0 |
| | | Open Rover | 0.4 |
| | | Pressurized Rover | 1.4 |

| | | | |
|---|---|---|---|
| | | Crew Allowances | 0.4 |
| | | **Total** | **22.5** |

**Table 5: Development Cost Estimates**

| | $H_2/O_2$ | $CH_4/O_2$ |
|---|---|---|
| Mars Transfer & Surface Vehicle | $300 million | $300 million |
| Earth Return Vehicle | $315 million | $315 million |
| Propulsion Stages | (use Centaur) | $30 million |
| Contingency (100%) | $615 million | $645 million |
| **Totals** | **$1.23 billion** | **$1.29 billion** |

(Actual masses for MTSV and ERV, excluding fuel and miscellaneous equipment, are 20 and 21 tonnes respectively)

**Table 6: Recurring Mission Cost Estimates**

| | $H_2/O_2$ | $CH_4/O_2$ |
|---|---|---|
| Trans-Mars Spacecraft | $225 million | $225 million |
| Propulsion Stages | 4 x $30 million | 5 x $23 million |
| Required Launches | 6 x $100 million | 7 x $100 million |
| Operations & Contingency | $300 million | $300 million |
| **Total** | **$1.25 billion** | **$1.34 billion** |

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