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# ENABLING EXPLORATION MISSIONS NOW: APPLICATIONS OF ON-ORBIT STAGING

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**&**
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*Abstract*

Future NASA exploration objectives are difficult to meet using current propulsion architectures and fuel-optimal trajectories. We introduce the concept of On-Orbit Staging and combine it with the idea of pre-positioned fuel and supply depots to increase payload mass and reduce overall cost, schedule, and risk for missions proposed as a part of the NASA Vision for Space Exploration. The On-Orbit Staging concept extends the implementation of ideas originally put forth by Tsiolkovsky, Oberth and Von Braun to address the total mission design. Applying the basic staging concept to all major propulsive (orbit) events and utilizing technological advances in propulsion efficiency and architecture allows us to demonstrate that exploration and science goals can be met more effectively and efficiently. As part of this architecture, we assume the readiness of automated rendezvous, docking/berthing, and assembly technology, all of which will be required for any credible exploration architecture. Primary cost drivers are identified and strategies that utilize On-Orbit Staging to reduce these costs are discussed.

## INTRODUCTION

The objectives of the Vision for Space Exploration announced by President George W. Bush on January 14, 2004 include[1]:

- Implementing a sustained and affordable human and robotic program to explore the Solar System and beyond
- Extending human presence across the Solar System, starting with a human return to the Moon by the year 2020, in preparation for the human exploration of Mars and other destinations

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- Developing the innovative technologies, knowledge, and infrastructures both to explore and to support decisions about destinations for future human exploration

- Promoting international and commercial participation in exploration to further U.S. scientific, security, and economic interests

This paper addresses concepts that we believe will be critical to the achievement of each of the first three of these objectives, and essential to achieve the second objective in particular.

OOS begins with a set of launch vehicles that place an assembly spacecraft, propulsive elements and bulk supplies into LEO in advance of mission hardware or crew. Once in LEO, the assembly spacecraft (launched first) assembles these propulsive elements into several larger collective elements that permit optimal staging and a significant increase in the ratio of payload mass to initial wet mass. This staging can be thought of as similar to that used for any launch vehicle, although the elements may not be stacked like traditional Earth-to-orbit vehicles (i.e., one on top of the other). While no single design has been put forward, the general idea is to have multiple stages that have a cluster of propulsive elements, either liquid or solid.

Using OOS, a significant increase in payload mass is achieved utilizing existing propulsion technologies. The utility of this concept is evaluated with analytical methods and high-fidelity analysis to demonstrate feasibility and validate assumptions. We demonstrate the advantages of the concept by showing the increase in payload mass for missions to several destinations, including sample cases involving one-way and round-trips to the Moon and Mars. We analyze the benefits and efficiencies of using multiple launches to place individual propulsion and resource elements into LEO, where they can be assembled to form larger, staged vehicles for the remainder of the journey. We then compare the performance of such vehicles to direct single-stage transfers to the Moon, Mars, and other destinations in the Solar System.

Additional performance gains can be realized by enhancing the OOS concept with pre-positioned fuel in orbit about the destination body and at other strategic locations. We demonstrate with multiple cases of a fast (< 245-day) round-trip to Mars, that using OOS combined with pre-positioned propulsive elements and supplies sent via fuel-optimal trajectories can reduce the propulsive mass required for the journey by an order-of-magnitude. Substantial increases in payload mass to the Moon using optimal and fast transfers are also demonstrated.

OOS can be applied with any class of launch vehicle, with the only measurable difference being the number of launches required to deliver the necessary assets to LEO for a particular mission. The analysis presented addresses the use of existing high-performance ($LO_2/LH_2$) and bi-propellant propulsion systems as well as more advanced propulsion system concepts.

Finally, cost drivers for sample fast-transfer missions to Mars and the Moon are examined. It is demonstrated that the primary drivers are the cost of manufacturing space flight hardware and the launch costs for inexpensive cargo. We discuss how the relaxed mass constraints realized through the use of OOS and pre-positioned depots can act as a catalyst to reduce the cost of manufacturing space flight hardware and evaluate the relative effect on the total cost of these sample missions. We also evaluate the relative cost savings of developing an economical launch capability for inexpensive cargo.

**Analysis Tools**

The primary analysis tool developed to investigate the merit of OOS is a Microsoft Excel spreadsheet with custom functions based on the equations found in the discussion of optimal staging for similar stages by Prussing and Conway[2]. The spreadsheet is designed to accept as input: payload mass; the specific impulse; structural mass fraction; and number of stages of the on-orbit propulsion systems; the $\Delta V$ required for each propulsive event of a defined mission; and launch vehicle LEO payload capability. The number of stages in the current implementation is limited to 4, as the performance gain from additional staging is minimal. The output generated is the number of launch vehicles required to lift the required total mass into LEO to accomplish the defined mission. In addition, the optimal staging equations were implemented in MATLAB® functions and scripts to facilitate graphical analysis of data.

To compute $\Delta V$s, a key input to the Excel tool, we used several software programs to design the trajectories. The force models of these programs include planetary bodies (DE405-based) and solar radiation pressure as perturbations. A numerical integration was performed both for the coast phases and during the finite maneuver modeling using a Runge-Kutta-Verner 8/9 method to include maneuver accelerations. The simulations included mass updates as fuel mass was computed and depleted and hardware was ejected between maneuver events. The software programs used for this purpose were Analytical Graphics' Satellite Tool Kit® (STK)/Astrogator module and GSFC's Swingby, both operational trajectory design tools. For optimal planetary trajectory design, Space Flight Solutions' Mission Analysis Environment (MaNE™) software was used to compute departure and arrival $\Delta V$s, which were then used as a-priori estimates in Astrogator or Swingby. In addition to the Excel tool output, we also simulated several scenarios from end-to-end with the high-fidelity modeling.

## ON-ORBIT STAGING (OOS) CONCEPT

The OOS concept is derived directly from the concept of optimal staging for launch vehicles. Detailed discussions of optimal staging can be found in references by Prussing and Conway[2], Wiesel[3], and Curtis[4], among others. Nomenclature to be used in discussion of the concept is as follows…

$g$ = Earth gravitational acceleration
$I_{sp}$ = Specific impulse
$\Delta V$ = Change in velocity
N = Number of stages
MT = Metric ton

$M_0$ = Total mass
$M_P$ = Propellant mass
$M_S$ = Structural mass
$M_L$ = Payload mass
$M_V$ = Vehicle mass ($M_S + M_P$)
$\varepsilon$ = Structural mass fraction

Beginning with the formulation found in Prussing and Conway[2], the structural mass fraction is defined as…

$$\varepsilon \equiv \frac{M_S}{M_S + M_P} \qquad (1)$$

Using this definition, equations for the total mass ($M_0$), propellant mass ($M_P$), and structural mass ($M_S$) for a single-stage vehicle can be derived in terms of the payload mass ($M_L$), mass fraction ($\varepsilon$), specific impulse ($I_{sp}$), gravitational constant ($g$) and required $\Delta V$ as…

$$M_0 = \frac{M_L(1-\varepsilon)}{(e^{(-\Delta V/g\cdot Isp)} - \varepsilon)} \quad (2) \quad M_P = \frac{M_L(1-\varepsilon)(1-e^{(-\Delta V/g\cdot Isp)})}{(e^{(-\Delta V/g\cdot Isp)} - \varepsilon)} \quad (3) \quad M_S = \frac{M_L\varepsilon(1-e^{(-\Delta V/g\cdot Isp)})}{(e^{(-\Delta V/g\cdot Isp)} - \varepsilon)} \quad (4)$$

Note that these equations share a common denominator…

$$(e^{(-\Delta V/_{g\cdot Isp})} - \varepsilon) \quad (5)$$

…and that they break down as it approaches zero. As the ΔV requirement increases, ε must decrease and/or $I_{sp}$ must increase for the equations to close. For a single-stage vehicle, the ratio of payload mass to initial mass is…

$$\frac{M_L}{M_0} = \frac{(e^{(-\Delta V/_{g\cdot Isp})} - \varepsilon)}{(1-\varepsilon)} \quad (6)$$

For our analysis, we chose ε = 0.075 (Saturn V S-II stage) as a practical lower limit.[5] Assuming a constant $I_{sp}$ = 465 sec the maximum theoretical ΔV for a single-stage vehicle is approximately 11.8 km/sec. Practical considerations dictate that staging is required well before the ΔV requirement approaches this limit. Figure 1 depicts the payload mass ratio as a function of ΔV for a single-stage vehicle for $I_{sp}$ values of 465 sec and 320 sec with ε of 0.075 and zero (0.0).

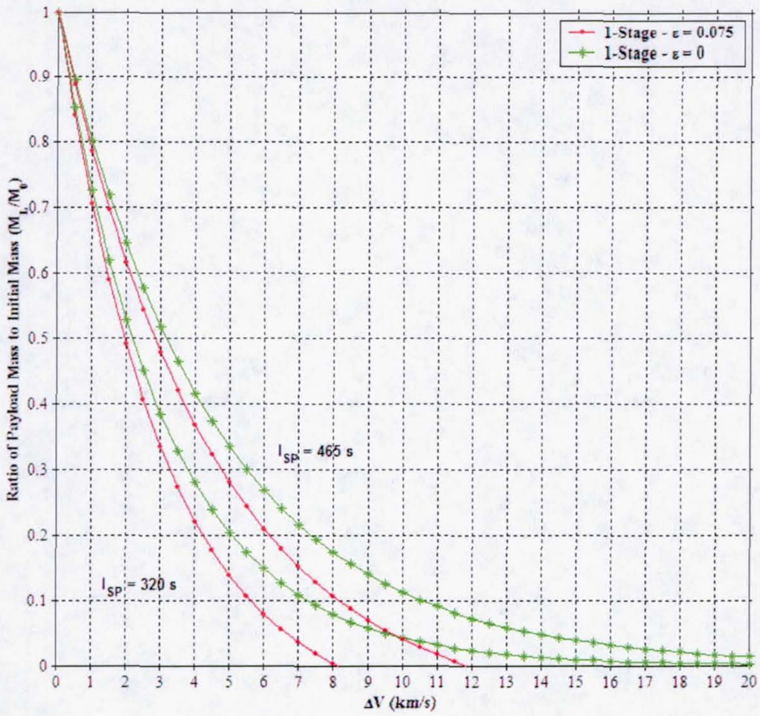


**Figure 1: Payload Mass Ratio as a Function of ΔV for a Single-Stage Vehicle for $I_{sp}$ Values of 465 sec and 320 sec**

For an N-stage vehicle with similar stages ($\varepsilon_1 = \varepsilon_2 \ldots = \varepsilon_N$; $I_{sp1} = I_{sp2} \ldots = I_{spN}$), it can be shown that the ratio of payload mass to initial mass for each stage is…

$$\frac{M_L}{M_0} = \frac{(e^{(-\Delta V/_{N\cdot g\cdot Isp})} - \varepsilon)}{(1-\varepsilon)} \quad (7)$$

…and that the overall payload mass ratio of the N-stage vehicle is…

$$\frac{M_L}{M_0} = \left[ \frac{\left(e^{\left(-\Delta V / N \cdot g \cdot Isp\right)} - \varepsilon\right)}{(1-\varepsilon)} \right]^N \quad (8)$$

Note that for ε = 0 and N=1, Equation 8 collapses into the more familiar form…

$$\frac{M_L}{M_0} = e^{\left(-\Delta V / g \cdot Isp\right)} \quad (9)$$

In this ideal case where the vehicle is assumed to have a massless structure it can be seen that for a given $M_0$ and ΔV, increasing $I_{sp}$ is the only option available to improve payload performance; however, in considering the realistic case represented by Equation 8, two other options – decreasing the structural mass of the vehicle (ε) and increasing the number of stages (N) – come into play so that now there are three avenues for improving payload performance that can be utilized in any combination. This has significant implications, especially in the case of long-duration missions where propellant storability is an important factor. In the case represented by Equation 9, improving payload performance requires using a high-energy propellant combination such as $LO_2/LH_2$. Storability considerations make this combination impractical for long-duration missions; however, Equation 8 indicates that the performance reduction in using an easily storable propellant can be mitigated by developing lighter vehicle structure and employing the use of staging.

**Advantages of On-Orbit Staging**

For ε = 0.075 and $I_{sp}$ = 465 sec, as the number of stages increases, $M_L/M_0$ approaches a limiting value of approximately 0.077 for a ΔV of 10.8 km/sec, which is representative of the ΔV required for a fast-transfer orbit insertion at Mars. Figure 2 depicts the payload mass ratio as a function of ΔV for single-stage and multi-stage (2-to 4-stage) vehicles using $I_{sp}$ values of 465 sec and 320 sec. The overall payload mass ratio increases for the multi-stage vehicles, and the advantage grows quickly with the required ΔV. At the lower $I_{sp}$ value, a single-stage vehicle cannot achieve the ΔV necessary for a fast transfer to Mars; therefore, to complete such a mission performance-enhancing features must be employed in the trajectory design, including gravity assist, aerobraking, etc. Figure 3 depicts the effects that staging has on the achievable payload mass ratio or ΔV for this representative case. Assuming an $I_{sp}$ = 465 sec and ε = 0.075, the plot indicates that for a single-stage vehicle $M_L/M_0 \approx 0.02$, and for a 4-stage vehicle, $M_L/M_0 \approx 0.071$. Assuming that $M_0$ is the same for both vehicles, this plot reveals that a 4-stage vehicle can impart a ΔV of 10.8 km/sec to a payload mass that is 355% greater than that of a single-stage vehicle. Conversely, the original single-stage payload mass can be accelerated to a higher ΔV of 15.66 km/sec or the total mass ($M_0$) required to accelerate the same mass to a ΔV of 10.8 km/sec can be reduced by ≈70% by using a 4-stage vehicle.

For lower ΔV requirements, OOS improvements are derived less from staging and more from higher $I_{sp}$ performance. This is a consequence of the exponential components in the equations above. Figure 4 shows the percentage increase in payload mass ratio due to the effects of staging only. The figure shows that staging is more valuable for propulsion systems with lower $I_{sp}$. For example, at a ΔV of 6 km/sec and an $I_{sp}$ of 320 sec, the improvement from staging ranges from approximately 40% to 55%, depending on the number of stages. For an $I_{sp}$ of 465 sec, the improvement is approximately 10%. Recall that, as shown in Figure 1, for a ΔV of 6 the improvement due solely to higher $I_{sp}$ is over 100%. OOS becomes the only solution for increasing mass as the ΔV increases and the achievable payload mass limit is reached for a single stage. Also shown is the approximate range of departure or arrival ΔVs for specific mission types.

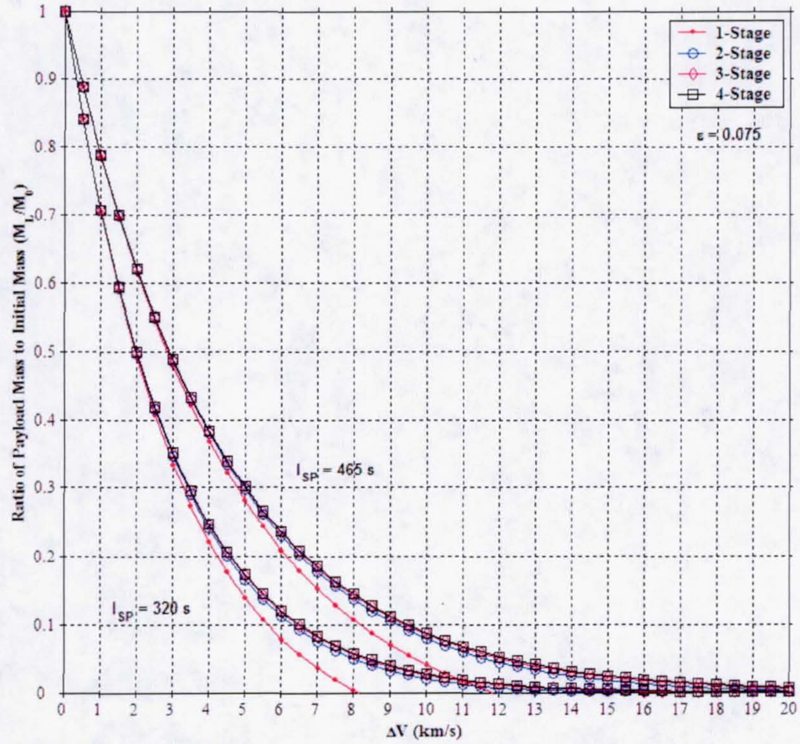


**Figure 2: Ratio of Payload Mass to Initial Mass as a Function of ΔV for 1- to 4-Stage Vehicles for $I_{sp}$ Values of 465 sec and 320 sec**

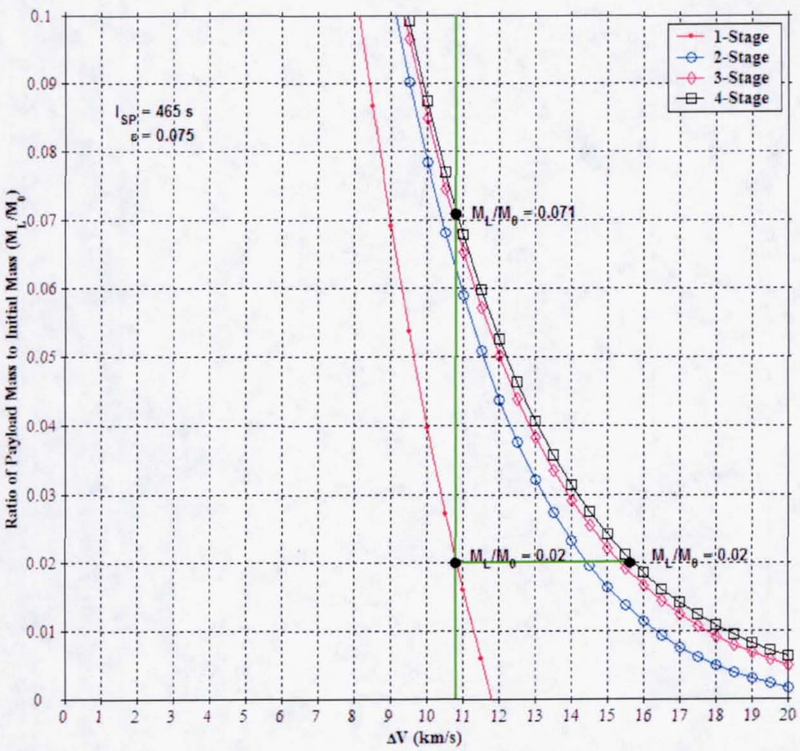


**Figure 3: Ratio of Payload Mass to Initial Mass as a Function of ΔV for 1- to 4-Stage Vehicle for $I_{sp}$ = 465 sec**

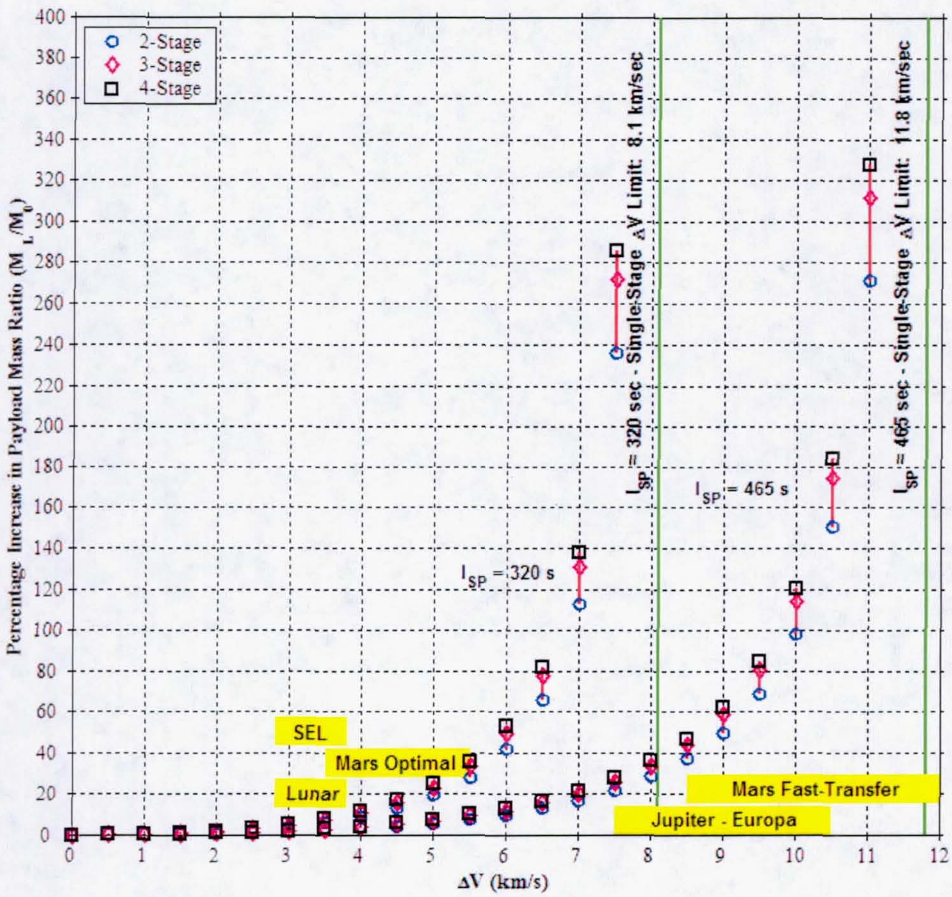


**Figure 4: Percentage Increase in Payload Mass Ratio for Staging**

## APPLICATIONS OF ON-ORBIT STAGING

In addition to enabling fast-transfer trajectories to the Moon, Mars, and other destinations in the Solar System, OOS also greatly enhances the payload capability for traditional fuel-optimal trajectories in that more payload mass can be delivered with an aggregation and staging of propulsive elements than by utilizing them separately without staging.

### Science and Robotic Missions

An increase in payload mass capability can be achieved for robotic missions by the use of OOS vs. the use of a direct transfer by the launch vehicle upper stage to the mission destination. For example, a mission to place an instrument into lunar orbit typically uses a single propulsive maneuver to place the spacecraft into a transfer orbit. Once in the vicinity of the Moon, an insertion maneuver is performed to achieve orbit capture. OOS will enhance both components of this sequence by applying multiple stages to each maneuver.

Tables 1 and 2 present a summary of the payload masses (in metric tons) that can be placed into several sample mission orbit types by the use of OOS and compares these to the payload mass capability achieved by the current launch and injection method, labeled "direct". These mission types include: a Sun-Earth libration orbit similar to that to be used for the James Webb Space Telescope (JWST); a lunar mission to place resources both into a lunar polar orbit similar to the Lunar Reconnaissance Orbiter (LRO) and onto the surface; Mars orbiters and landers, assuming fuel-efficient (vs. fast) transfers; a mission to place a spacecraft into a circular orbit about Jupiter at the radial distance of its moon, Europa; a 200-km circular orbit about Europa; and a polar orbit about the Sun with periapsis at 12 solar radii and apoapsis at 1 a.u.

The payload mass information is based on the mission parameters indicated in columns 2 through 4 of Table 1 and presumes the ability of the launch vehicle to place a given metric tonnage into LEO, assumed here to be 200 km circular. The $\Delta$Vs for these missions were computed using the previously mentioned programs. Another parameter, C3, also computed from these programs, can be used as the representative orbit energy parameter. For a mission such as JWST, the transfer insertion C3 values typically range between -0.69 $km^2/s^2$ and -0.40 $km^2/s^2$, yielding an injection $\Delta$V of between 3.12 km/sec and 3.20 km/sec. The C3 and injection $\Delta$Vs for the LRO mission range from -2.0 $km^2/s^2$ to -1.8 $km^2/s^2$ and 3.10 km/sec to 3.18 km/sec, respectively. The $\Delta$Vs for insertion into lunar orbit and to land on the surface are 0.9 km/sec and 1.7 km/sec, respectively. Mars transfer departure $\Delta$Vs range from 3.5 km/sec to 4.0 km/sec with Mars orbit insertion and landing $\Delta$Vs of 2.8 km/sec and 3.4 km/sec, respectively. For a transfer to Jupiter, the C3 values can be in excess of 84 $km^2/s^2$, with the amount of $\Delta$V required for braking and insertion into Jovian orbit being 10.9 km/sec, and the insertion $\Delta$V into Europa orbit being 8.9 km/sec. The payload mass capability of existing launch vehicles to LEO or direct transfer orbits is taken from the Kennedy Space Center (KSC) launch vehicle performance web site using either the C3 or LEO altitude as input.[6] The term "heavy lift" (a generic term here) assumes three different LEO payload delivery capabilities: 40 MT, 70 MT and 120 MT. The number of launches to LEO is listed in the table headers.

The OOS payload masses are calculated based on the $\Delta$V requirements, an $I_{sp}$ of 465 sec, a structural mass fraction $\varepsilon = 0.075$, and the use of a 4-stage vehicle for each maneuver. Equations 3, 4 and 8 are used to compute fuel and structural masses and the payload mass ratios. In all these cases, we use a structural mass fraction derived from the total mass estimates for the required tank hardware, propulsion system hardware, and guidance, navigation and control system hardware. The propulsion hardware mass was based on information for the RL10 and RL60 propulsion systems from the Pratt & Whitney web site or in-house information on bi-propellant systems, plus a percentage for other propulsion system hardware, e.g. valves, tanks, lines, etc.

By optimally combining vehicles, optimal payload mass for particular mission objectives can be achieved. In general, the larger the $\Delta$V required, the greater the improvement in the performance by employing OOS. For example, compared to a single Delta-II launching a payload directly into a transfer orbit, OOS with two Delta-IIs increases the payload mass into a Sun-Earth libration orbit, lunar orbit, or onto the lunar surface by a factor of three. A similar change for a Mars orbiter increases payload by a factor of four. Increasing the number of launch vehicles beyond two to place more fuel into LEO increases the payload mass to much larger values – well beyond the simple multiple. For example, for a Mars landing, using 4 Delta II launches yields an 800% improvement over the payload delivered by a single Delta II launch while OOS enables a Europa mission with over a metric ton of payload.

**Advantages of Pre-Positioned Fuel Depots**

A simple example of the relative effects of staging and the use of fuel depots is shown in Figure 5. The data shown is based on arrival and departure $\Delta$Vs at the destination that are equal in magnitude at 4.5 km/sec and the payload mass is set at 10 MT. $I_{sp}$ and $\varepsilon$ are assumed to be 465 sec and 0.075, respectively. The figure shows the propulsive mass, $M_V$, as a function of $\Delta$V, where $M_V$ is given by…

$$M_V = M_L\left[\left(\frac{1-\varepsilon}{e^{(-\Delta V/_{N\cdot g\cdot Isp})} - \varepsilon}\right)^N - 1\right] \quad (10)$$

Figure 5 shows that if the propulsive mass for the total required $\Delta$V of 9 km/sec is carried from the outset, the use of a 4-stage vehicle results in a savings of approximately 57 MT (79 MT vs. 134 MT) relative to a single-stage vehicle. If a fuel depot is employed in orbit at the destination, the exponential growth in propulsive mass is limited, since the propulsive mass for the departure maneuver is not carried through the insertion maneuver. The total propulsive mass for the insertion and departure maneuvers drops to 21 MT per maneuver for a single-stage vehicle and 19 MT per maneuver for a 4-stage vehicle. In the following example, we demonstrate that the reduction in propulsive mass required for maneuvers at the destination will result in a huge reduction in the propulsive mass required in LEO.

**Table 1**
**MISSION MASS (MT) USING DIRECT INSERTION**

| **Mission Type** | **Mission $\Delta$Vs Depart / Arrival** (km/sec) | **C3** ($km^2/s^2$) | **Flight Time** (days) | **D-II 1 Direct C3 Based** | **D-IV 1 Direct C3 Based** |
|---|---|---|---|---|---|
| Sun-Earth $L_2$ | 3.14 / 0.01 | -0.4 | 120 | 1.4 | 9.3 |
| Lunar Orbiter | 3.12 / 0.90 | -2.0 | 5 | 1.2 | 7.8 |
| Lunar Lander | 3.12 / 2.60 | -2.0 | 5 | 0.8 | 5.1 |
| Mars Orbiter | 3.66 / 2.28 | 7.8 | 204 | 0.7 | 4.7 |
| Mars Lander | 3.66 / 5.68 | 7.8 | 204 | 0.3 | 2.4 |
| Jupiter Orbiter | 6.56 / 10.87 | 84 | 788 | 0.0 | 0.0 |
| Europa Orbiter | 6.65 / 8.90 | 84 | 788 | 0.0 | 0.1 |
| Solar Polar | 26.5 / 0.00 | 1050 | 55 | 0.0 | 0.0 |

**Table 2**
**MISSION MASS (MT) USING OOS**

| **Mission Type** | **D-II** 2 @ 5760 MT to LEO | **D-II** 4 @ 5760 MT to LEO | **D-IV** 2 @ 23060 MT to LEO | **D-IV** 4 @ 23060 MT to LEO | **Heavy Lift** 3 @ 40000 MT to LEO | **Heavy Lift** 3 @ 70000 MT to LEO | **Heavy Lift** 3 @ 120000 MT to LEO |
|---|---|---|---|---|---|---|---|
| Sun-Earth $L_2$ | 5.5 | 10.9 | 21.9 | 43.7 | 56.8 | 70.7 | 121.2 |
| Lunar Orbiter | 4.4 | 8.8 | 17.7 | 35.4 | 46.9 | 80.5 | 138.0 |
| Lunar Lander | 2.9 | 5.9 | 11.7 | 23.5 | 30.5 | 53.4 | 91.6 |
| Mars Orbiter | 2.8 | 5.6 | 11.2 | 22.3 | 29.1 | 50.8 | 87.2 |
| Mars Lander | 1.2 | 2.4 | 4.8 | 9.6 | 12.6 | 22.0 | 37.7 |
| Jupiter Orbiter | 0.2 | 0.3 | 0.7 | 1.3 | 1.7 | 3.0 | 5.2 |
| Europa Orbiter | 0.3 | 0.5 | 1.1 | 2.2 | 2.9 | 5.0 | 8.6 |
| Solar Polar | 0.0 | 0.0 | 0.0 | 0.1 | 0.1 | 0.2 | 0.3 |

Currently, a common perception is that a human mission to Mars is "too far, too hard". A primary obstacle is that the mass required for crew habitation and life support infrastructure is prohibitive utilizing current propulsion architectures and fuel-optimal trajectories. One way around the obstacle is to use high-energy, fast-transfer (180 - 245 day round-trip) trajectories for crew transfer, eliminating the need for an elaborate life support infrastructure. Current propulsion architectures do not support the high $\Delta$V required for fast-transfer trajectories for even the most optimistic payload mass estimates for such a mission. OOS, in conjunction with pre-positioned fuel depots, has the potential to bring the fast-transfer option into the realm of possibility - and a human mission to Mars within reach in the near-term - with current propulsion systems and technology.

For round-trip fast transfers between Earth and Mars, carrying the fuel from the outset to complete the entire trip is prohibitive. Placing a pre-positioned fuel depot for the return trip in orbit about Mars enables such a mission. A sample case assuming a 10 MT payload for each leg and a $\Delta$V for insertion of 10.8 km/sec demonstrates the advantage of a pre-positioned fuel depot in Martian orbit. As in the previous example, we assume that the $\Delta$V for insertion and departure are equal, for a total required $\Delta$V of $\approx$ 21.6 km/sec (using two 4-stage propulsion units). To minimize the fuel required, we assume use of OOS, breaking the insertion maneuver into 4 stages to yield a payload mass ratio of approximately 0.071 and provide an increase in payload mass of 355%. Note that our earlier results show that a $\Delta V$ of this magnitude can be achieved only through the use of OOS. Carrying all of the fuel for the round trip on the outbound fast-transfer leg requires an initial mass of 15,666 MT in LEO. In contrast, carrying only the fuel to insert into Mars orbit on the outbound leg requires an initial mass of just 1,045 MT in LEO. Pre-positioning the fuel required for the return leg from Mars orbit by way of a $\Delta V$-optimal Hohmann transfer utilizing OOS requires just another 573 MT in LEO. Adding the 1,045 MT to get to Mars to the 573 MT to get back yields a total round-trip requirement of only 1,618 MT - an order-of-magnitude savings over OOS.

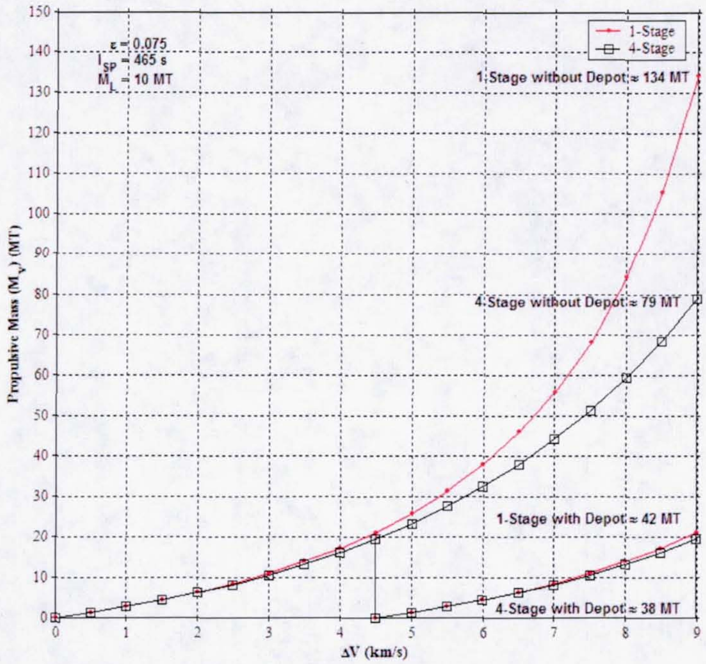


**Figure 5: Relative Effects of Staging and Fuel Depots**

OOS also has benefits for lunar missions. For human lunar missions, high-energy solar events are a serious threat to crew health and safety. OOS enables the use of fast-transfer trajectories (24-36 hours vs. 96 hours) to enhance crew safety by reducing crew exposure to radiation in cislunar space from high-energy solar events. OOS also increases the amount of payload mass that can be placed onto the lunar surface so that lunar base architectures requiring substantial resources can be realized more quickly.

## MARS AND LUNAR FAST-TRANSFER OPTIONS

Several case studies are presented below to show the benefits of OOS and the use of pre-positioned fuel depots. These studies cover human missions to Mars and the Moon.

### Mars Fast Transfer

A fast transfer to Mars can be achieved by launching into a heliocentric orbit that arrives at Mars at Earth-Mars conjunction. Figure 6 shows a fast transfer of 120 days from Earth to Mars, a 14-day stay and a 75-day return. The trajectory shown in the figure requires much higher $\Delta V$s than a slow conjunction or opposition trajectory. While slow (Hohmann) transfers require $\Delta V$s in the range of 3-4 km/sec, the fast-transfer $\Delta V$s can be several times as large and are driven by the relative geometry between Earth and Mars at the beginning and end of the transfer period. Target parameters are the incoming trajectory asymptote with regard to the required B-plane (the B-plane of Mars is targeted for the optimal incoming trajectory to minimize the insertion $\Delta V$), the final orbit semi-major axis and eccentricity (we assume a circular orbit of 200 km), and the surface stay time.

As seen in Table 3, the 4 major maneuvers required for a total 180-day transfer time, 90 days for each leg, require ΔVs that range from approximately 9 km/sec to 15 km/sec for each Mars insertion or departure maneuver. The total ΔV for all legs remains fairly constant, implying that a trade must be performed between the insertion and departure legs at Mars in order to optimize the amount of usable payload mass in the OOS equations. Figure 7 presents a contour plot of the Mars insertion and departure ΔVs for various surface stay times and launch dates.

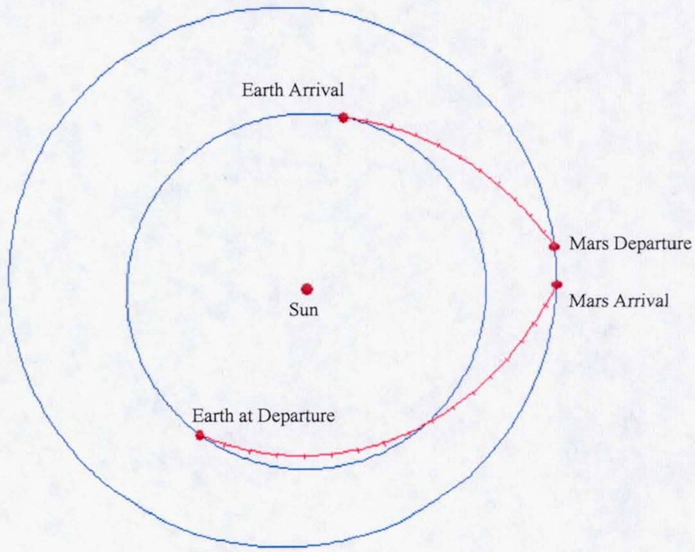


**Figure 6: Sample Fast Transfer to Mars**

**Table 3**
**ΔVs REQUIRED FOR 180 DAY ROUND TRIP TO MARS WITH VARIOUS STAY TIMES ON THE SURFACE**

| Launch Date (2020) | Stay Time (days) | Earth Departure (km/sec) | Mars Insertion (km/sec) | Mars Departure (km/sec) | Earth Return (km/sec) | Total (km/sec) |
|---|---|---|---|---|---|---|
| 1-Jun | 7 | 9.64 | 14.50 | 7.87 | 4.85 | 36.86 |
| 1-Jun | 14 | 9.64 | 14.50 | 8.84 | 4.79 | 37.77 |
| 1-Jun | 21 | 9.64 | 14.50 | 9.90 | 4.89 | 38.93 |
| 1-Jun | 28 | 9.64 | 14.50 | 11.03 | 5.14 | 40.31 |
| 15-Jun | 7 | 8.32 | 12.89 | 9.90 | 4.89 | 36.00 |
| 15-Jun | 14 | 8.32 | 12.89 | 11.03 | 5.14 | 37.38 |
| 15-Jun | 21 | 8.32 | 12.89 | 12.22 | 5.56 | 38.99 |
| 15-Jun | 28 | 8.32 | 12.89 | 13.44 | 6.15 | 40.80 |
| 1-Jul | 7 | 7.01 | 11.17 | 12.48 | 5.68 | 36.34 |
| 1-Jul | 14 | 7.01 | 11.17 | 13.71 | 6.31 | 38.20 |
| 1-Jul | 21 | 7.01 | 11.17 | 14.97 | 7.13 | 40.28 |
| 1-Jul | 28 | 7.01 | 11.17 | 16.24 | 8.16 | 42.58 |
| 15-Jul | 7 | 6.05 | 9.86 | 14.97 | 7.13 | 38.01 |
| 15-Jul | 14 | 6.05 | 9.68 | 16.24 | 8.16 | 40.13 |
| 15-Jul | 21 | 6.05 | 9.68 | 17.52 | 9.42 | 42.67 |
| 15-Jul | 28 | 6.05 | 9.68 | 18.79 | 10.93 | 45.45 |

Table 4 shows the ΔVs for all legs for a constant surface stay time of 7 days but with various trip times. The ΔVs here range from approximately 4 km/sec to 30 km/sec and are primarily dependent upon the relative geometry between Earth and Mars. Figure 8 presents this data in a plot to show that, like the stay time, a transfer time trade must be performed to minimize the impact of the high arrival and departure ΔVs at Mars, which again are the dominant drivers.

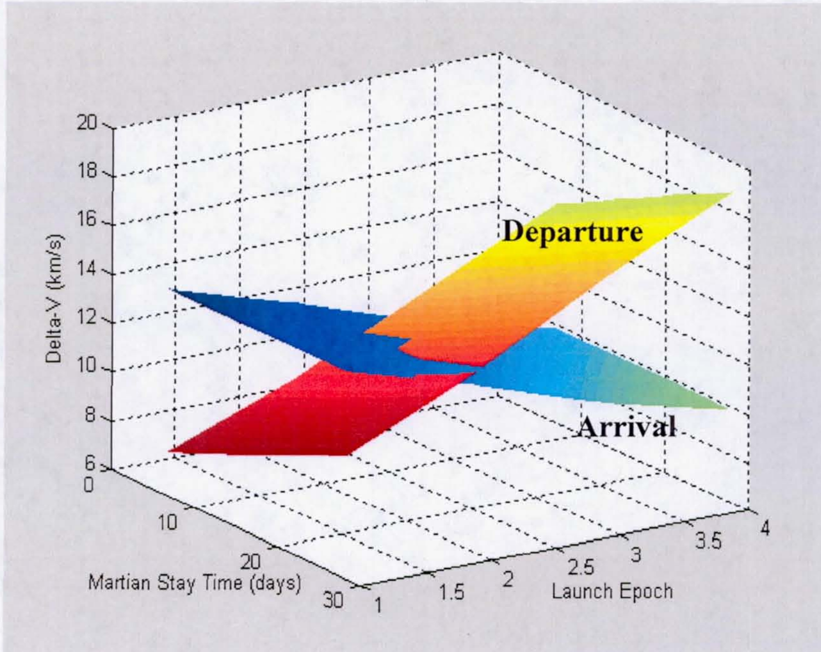


**Figure 7: Mars Arrival and Departure ΔV for Various Surface Stay Times and Launch Opportunities with 90-Day Transfer Legs**

**Table 4**
**ΔVs REQUIRED FOR VARIOUS TRANSFER TIMES WITH 7-DAY SURFACE STAY AT MARS**

| Launch Date (2020) | Outbound Duration (days) | Return Time (days) | Earth Departure (km/sec) | Mars Arrival (km/sec) | Mars Departure (km/sec) | Earth Return (km/sec) | Total (km/sec) |
|---|---|---|---|---|---|---|---|
| 1-Jun | 90 | 90 | 9.64 | 14.5 | 7.87 | 4.85 | 36.86 |
| 1-Jun | 100 | 80 | 8.23 | 12.07 | 9.82 | 5.36 | 35.48 |
| 1-Jun | 110 | 70 | 7.19 | 10.06 | 12.35 | 6.30 | 35.90 |
| 1-Jun | 120 | 60 | 6.43 | 8.40 | 15.65 | 8.13 | 38.61 |
| 1-Jun | 130 | 50 | 5.86 | 7.03 | 20.10 | 12.03 | 45.02 |
| 15-Jun | 90 | 90 | 8.32 | 12.89 | 9.90 | 4.89 | 36.00 |
| 15-Jun | 100 | 80 | 7.15 | 10.69 | 12.25 | 5.67 | 35.76 |
| 15-Jun | 110 | 70 | 6.31 | 8.89 | 15.21 | 7.10 | 37.51 |
| 15-Jun | 120 | 60 | 5.70 | 7.41 | 19.03 | 9.86 | 42.00 |
| 15-Jun | 130 | 50 | 5.26 | 6.20 | 24.16 | 15.66 | 51.28 |
| 1-Jul | 90 | 90 | 7.01 | 11.17 | 12.48 | 5.68 | 36.34 |
| 1-Jul | 100 | 80 | 6.08 | 9.18 | 15.34 | 7.07 | 37.67 |
| 1-Jul | 110 | 70 | 5.45 | 7.61 | 18.80 | 9.49 | 41.35 |
| 1-Jul | 120 | 60 | 5.01 | 6.34 | 23.23 | 14.10 | 48.68 |
| 1-Jul | 130 | 50 | 4.69 | 5.31 | 29.19 | 22.19 | 61.38 |
| 15-Jul | 90 | 90 | 6.05 | 9.86 | 14.97 | 7.13 | 38.01 |
| 15-Jul | 100 | 80 | 5.35 | 7.92 | 18.19 | 9.36 | 40.82 |
| 15-Jul | 110 | 70 | 4.89 | 6.55 | 22.08 | 13.19 | 46.71 |
| 15-Jul | 120 | 60 | 4.57 | 5.46 | 27.07 | 19.75 | 56.85 |
| 15-Jul | 130 | 50 | 4.35 | 4.59 | 33.80 | 29.14 | 71.88 |

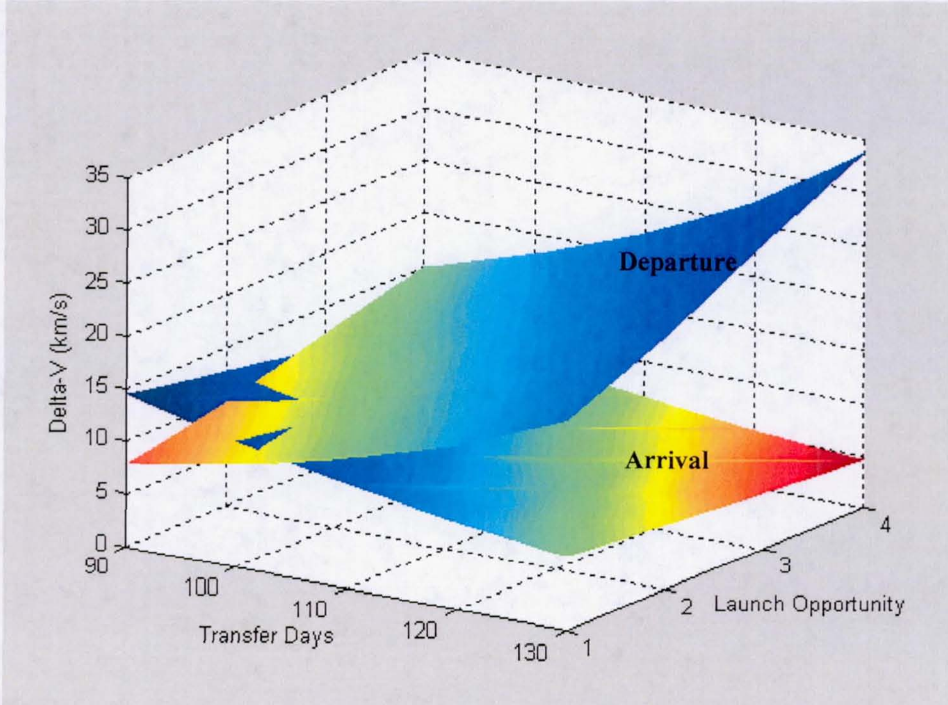


**Figure 8: Mars Arrival and Departure ΔVs for Various Transfer Durations and Launch Opportunities with a 7-Day Surface Stay**

Prior to the first human mission, robotic missions can be used to retire risk associated with pre-positioned fuel depots, fast-transfer issues, precision landings, Mars ascent, and rendezvous and docking. Additionally, the insertion and departure ΔVs must be compared to determine the most optimal design and minimize the total ΔV cost. Figure 8 provides a 3-D view of the ΔV cost for insertion and departure from a 200 km circular Mars orbit based on a fast transfer of 90 days. This figure indicates that a near-optimal solution can be chosen but is dependent upon the launch epoch.

**Human Mission to Mars**

We have developed three strawman options for human missions to Mars by varying trip times and delivered masses for 14-day surface stay durations. Each of these scenarios allows round trips within a single Mars conjunction opportunity. The short trip time reduces mission support requirements significantly and greatly decreases the risk to the crew.

As shown above, the use of OOS allows large cargo payloads to be moved to/from Mars on fuel-optimized transfers of 6-7 months duration. In our architecture, we would use this capability to pre-position all required hardware and stores for use at Mars, as well as the fuel and provisions for the return trip. Ideally, we would minimize the size and mass of the crew transfer craft to reduce the amount of propellant required for the fast transfer. Therefore, everything not absolutely required by the crew to make a safe transfer would be pre-positioned using a fuel-optimal Hohmann trajectory.

Table 5 portrays the basic mission parameters for each option, including payload masses, trip times, duration on the surface, and the number of launches required using launch vehicle capabilities to LEO of 40 MT or 120 MT. Mission mass details include the crew fast-transfer vehicle mass to and from Mars, the amount of dry mass landed and returned to Mars orbit, and the amount of depot fuel mass sent on a ΔV-optimal Hohmann trajectory. Using OOS and depots as described above, a significant amount of mass can be sent to Mars and returned. The 20 MT fast-transfer vehicle is consistent with the assumptions in Reference Mission Version 3 of the NASA Mars Exploration Study Team (June 1998)[7]. The data in the tables also provides the option of either an $I_{sp}$ of 465 sec ($LO_2/LH_2$ equivalent) for both legs, or an $I_{sp}$ of 320 sec (bi-propellant equivalent) on the return leg if long-term fuel storage is not permissible and easily storable propellant is required. A separate system is assumed for a time-insensitive return of a 3 MT payload of surface material (soil and rock samples, etc).

Table 6 lists the number of launches required to support the missions outlined in Table 5. The number of launches varies from several to hundreds depending on the option chosen, the $I_{sp}$ assumed, and the launch vehicle payload mass capability to LEO. The $\Delta$Vs assumed for these trips are taken from the transfer shown in Figure 6 and are listed in Table 7. In seeking to optimize the missions, we found that the best allocation of mission $\Delta V$ shifts the distribution of the total $\Delta V$ budget toward the propulsion system with the best performance, e.g., a higher Mars insertion $\Delta V$ for the higher performance ($LO_2/LH_2$) insertion system in exchange for a lower Mars departure $\Delta V$ for the lower performance (bi-propellant) departure system. This was accomplished by adjusting the launch date to shift the $\Delta V$ allocation toward the higher performing system. Also, the effect of stay time on the departure $\Delta V$, assuming a fixed time of arrival and Earth to Mars transfer duration, was to increase the departure $\Delta V$ by approximately 1 km/sec for each week of additional stay time. The landing and ascent $\Delta$Vs are each assumed to be 3.7 km/sec for this case based on non-atmospheric landing simulations. Note that orbit entry and landing at Mars is assumed to be fully propulsive to ensure a conservative estimate of fuel required to complete the mission. Other approaches, including aerobraking/aerocapture and non-propulsive entry, descent and landing systems must obviously be investigated to ensure the safest, most cost-effective mission.

One of the more striking outcomes of this analysis is the small number of dry (sensitive or human-rated) launches required versus the number of launches required for the fuel. This fact would strongly suggest that finding a way to deliver fuel and other acceleration-insensitive cargo to LEO cheaply would drastically reduce the cost of a Mars mission. There is no denying that this concept requires a large number of launches and assembly steps, which in other architectural concepts presented significant reliability issues for exploration missions. We believe that the risks in this concept can be mitigated effectively by decoupling the launch of propellants and other g-insensitive payloads from the launch of crew and high-value payloads, and by building a robust assembly capability in LEO.

In our approach, g-insensitive payloads would be launched on very different vehicles than crew and high-value payloads (which should also have separate launch vehicle types). G-insensitive payloads could also be launched on multiple launch systems, since differences in launch loads will have no effect on them. Given the low cost and easy replacement of these g-insensitive payloads, the loss of some number of them is not a threat to the mission – we can afford to procure and launch a large number of spares if we reduce the launch costs by relaxing g-limits and reliability requirements. Much of the propellant launched could also be storable, eliminating the schedule pressure on the rest of the program, as the propellant modules could be launched years ahead of the crew. In fact, the critical, high-cost mission hardware and crew would be launched only after all required propellant had been launched and configured as required for the mission.

Docking/berthing/assembly risks can be addressed by limiting docking to only the mission elements that carry the crew. All other proposed Mars mission architectures have this requirement as well; therefore, there is no additional risk in our approach relative to any others. For the remaining mission elements, we would use berthing, a far more reliable approach than docking.

G-insensitive payloads would be berthed to, and assembled into their final mission configuration by, a specialized LEO assembly spacecraft. The assembly spacecraft would further reduce the risk of berthing and assembly by taking advantage of the fact that, in the proposed architecture; dry mass is not at as high a premium as in other approaches. Additional mass can be used to provide more robust berthing and assembly systems, and multiple, independent means of capturing payloads can be used including tethers, robotic arms, etc. Multiple berthing stations can be provided on the assembly spacecraft to ensure that each propellant module has redundant berthing locations. If more redundant capability is desired, additional robotic arms can be placed on the assembly spacecraft. One could be placed on the crew vehicle itself, or a separate assembly spacecraft could be pre-positioned at Mars on a slow transfer to reduce risk in the Mars orbital portions of the mission.

Based on these arguments, we believe that the large number of propellant launches and assembly actions do not significantly increase the risk to mission success as a constant replenishing strategy for the depots can be assumed. The mission reliability calculation should then include only the high-value cargo and man-rated systems, the same as for any other Mars mission architecture.

**Table 5**
**INPUT FOR MARS FAST TRANSFER WITH 14-DAY STAY USING OOS WITH 40 MT OR 120 MT LAUNCH CAPABILITY**

| Mission Event<br>( Mass in MT)<br>( Time in Days ) | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return |
|---|---|---|---|---|---|---|
| $M_V$ to Mars Optimal | 108 | 108 | 48 | 48 | 108 | 108 |
| $M_L$ to Mars Fast | 20 | 20 | 10 | 10 | 15 | 15 |
| $M_L$ to Surface | 30 | 30 | 15 | 15 | 30 | 30 |
| $M_L$ from Surface | 15 | 15 | 5 | 5 | 15 | 15 |
| $M_L$ to Earth Optimal | 3 | 3 | 3 | 3 | 3 | 3 |
| $M_L$ to Earth Fast | 20 | 20 | 10 | 10 | 10 | 10 |
| Trip time to Mars | 120 | 120 | 135 | 135 | 120 | 120 |
| Trip Time to Earth | 75 | 75 | 93 | 93 | 75 | 75 |
| Total Trip Time | 209 | 209 | 242 | 242 | 209 | 209 |

**Table 6**
**NUMBER OF LAUNCHES REQUIRED FOR MARS FAST TRANSFER USING OOS WITH 40 MT OR 120 MT LAUNCH CAPABILITY**

| Launch Resources Required | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return | Isp<br>465 sec | Isp<br>465 sec<br>Outbound<br>320 sec<br>Return |
|---|---|---|---|---|---|---|
| *Launch Capability of 40 MT to LEO* | | | | | | |
| Transfer and Arrival Propellant | 93 | 190 | 32 | 60 | 56 | 107 |
| $M_L$ | 4 | 4 | 3 | 3 | 4 | 4 |
| Lander | 12 | 12 | 7 | 7 | 10 | 10 |
| Return Propellant | 60 | 145 | 15 | 30 | 31 | 73 |
| ***Total*** | 97 | 194 | 35 | 63 | 60 | 111 |
| *Launch Capability of 120 MT to LEO* | | | | | | |
| Transfer and Arrival Propellant | 32 | 64 | 12 | 21 | 20 | 37 |
| $M_L$ | 2 | 2 | 2 | 2 | 2 | 2 |
| Lander | 4 | 4 | 2 | 2 | 4 | 4 |
| Return Propellant | 21 | 49 | 6 | 11 | 11 | 25 |
| ***Total*** | 34 | 66 | 14 | 23 | 22 | 39 |

Note: Additional launches would be required for the crew and assembly spacecraft

Table 7
ΔV FOR MISSION SCENARIOS

| | May 14 2020<br>ΔV (km/sec) | May 1 2020<br>ΔV (km/sec) | April 17 2020<br>ΔV (km/sec) | April 01 2020<br>ΔV (km/sec) |
|---|---|---|---|---|
| Earth Departure | 6.97 | 7.71 | 7.26 | 8.61 |
| Mars Arrival | 9.12 | 9.95 | 8.85 | 10.39 |
| Mars Departure | 11.04 | 9.06 | 8.14 | 6.61 |

**Human Mission to the Moon**

Similar to a Mars mission, a lunar mission can be designed using OOS to enable an increase in payload mass to the lunar surface or an overall increase in mass for a fast transfer. This fast transfer would take 24 to 36 hours and could be used to mitigate the risk of longer transfer times in cislunar space. The application of OOS and fuel depots are similar to a Mars mission with the obvious exception that the ΔV magnitudes are lower and the minimum ΔV transfer times to the Moon range from 4.2 to 5.1 days. The ΔVs for a lunar fast return are shown in Figure 9 for a 36-hour return and in Figure 10 for a 24-hour return over an entire lunar month. The variation in ΔV is due to the geometry of the orbit plane with respect to the return trajectory asymptote. Note the ΔV variation with true anomaly, indicating that each orbit has a minimum ΔV maneuver location. A maneuver performed in an orbit that is perpendicular to the Earth-Moon line-of-sight has a nearly constant ΔV over the orbit (day 3 or 17 for example), while a geometry in which the orbit plane is aligned with the Earth-Moon vector may result in a maneuver on the "outbound" portion of the orbit instead of the "inbound" (day 10 and 25 for example). The maximum ΔVs for insertion upon Earth return are 3.91 km/sec for the 24-hour case and 3.42 km/sec for the 36-hour case. The information in Table 8 uses these ΔVs and the same masses as those in the first Mars scenario for comparison. Again these masses are 20 MT to and from the Moon, 30 MT to the lunar surface, and 15 MT back into lunar orbit. Table 8 also presents the number of launches required assuming payload capabilities of 40 MT or 120 MT to LEO. Since rounding up to the next integer resulted in a considerable increase in the total launches after summing, it was decided to carry one significant digit to show a realistic estimate. The table shows the required number of launches for 24-hour, 36-hour, and 96-hour (optimal ΔV) outbound and return trajectories.

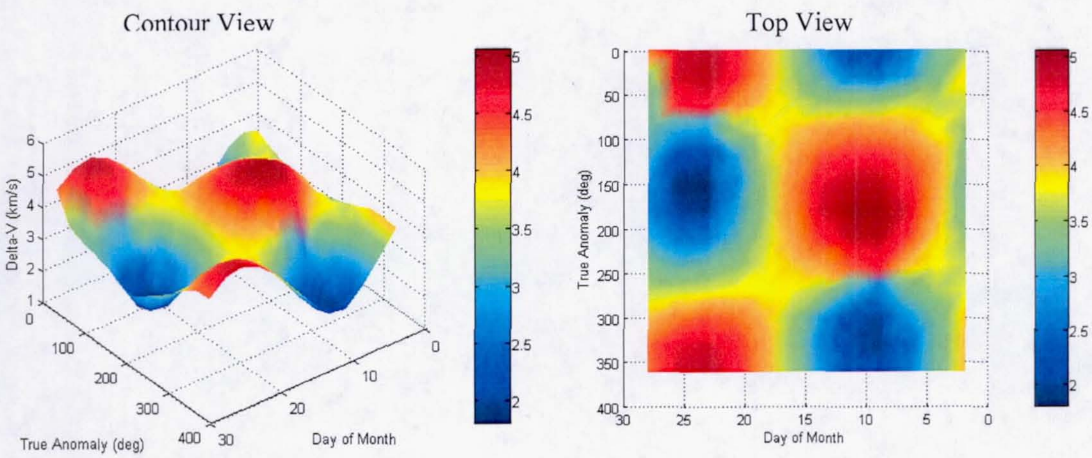


**Figure 9: ΔV Contour Plot For 36 Hour Fast-Transfer Return from Moon**

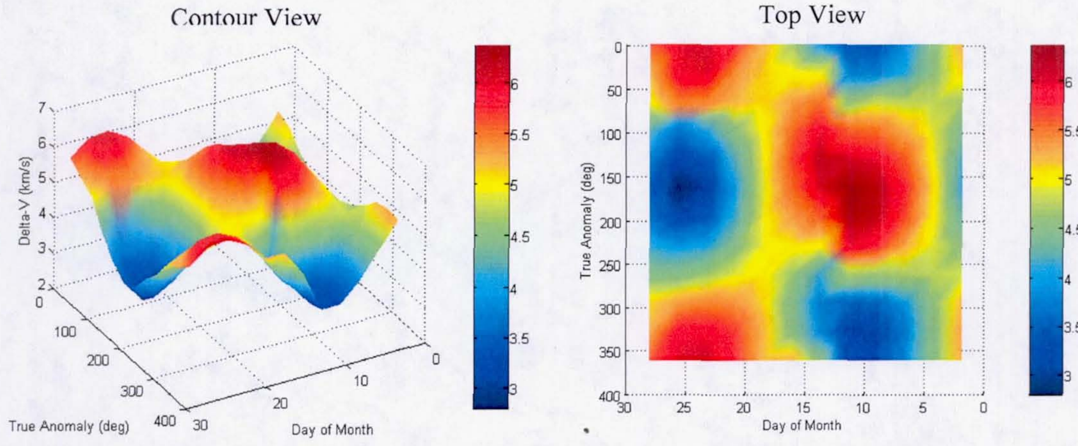


**Figure 10: ΔV Contour Plot For 24 Hour Fast-Transfer Return from Moon**

**Table 8**
**NUMBER OF LAUNCHES REQUIRED FOR LUNAR TRANSFER USING OOS WITH 40 MT OR 120 MT LAUNCH CAPABILITY**

| Launch Resources Required | Isp 465 sec | Isp 465 sec Outbound 320 sec Return | Isp 465 sec | Isp 465 sec Outbound 320 sec Return | Isp 465 sec | Isp 465 sec Outbound 320 sec Return |
|---|---|---|---|---|---|---|
| *Launch Capability of 40 MT to LEO* | | | | | | |
| One-Way Trip Time | 24 | 24 | 36 | 36 | 96 | 96 |
| $M_P$ Launches (total) | 13.3 | 31.4 | 8.6 | 16.5 | 4.9 | 4.9 |
| $M_L$ Launches | 2.5 | 2.5 | 2.5 | 2.5 | 2.5 | 2.5 |
| Lander $M_L$ | 3.7 | 3.7 | 3.7 | 3.7 | 3.7 | 3.7 |
| Return Propellant Only | 9.7 | 27.8 | 5.7 | 13.6 | 2.3 | 2.3 |
| ***Total*** | *15.8* | *33.9* | *11.1* | *19.0* | *7.4* | *9.4* |
| *Launch Capability of 120 MT to LEO* | | | | | | |
| One-Way Trip Time | 24 | 24 | 36 | 36 | 96 | 96 |
| $M_P$ Launches (total) | 4.4 | 10.5 | 2.9 | 5.5 | 1.6 | 2.3 |
| $M_L$ Launches | 0.8 | 0.8 | 0.8 | 0.8 | 0.8 | 0.8 |
| Lander $M_L$ | 1.3 | 1.3 | 1.2 | 1.2 | 1.1 | 1.1 |
| Return Propellant Only | 3.2 | 9.3 | 1.9 | 4.5 | 0.8 | 1.4 |
| ***Total*** | *5.3* | *11.3* | *3.7* | *6.3* | *2.5* | *3.1* |

**Increasing Lunar Surface Payload Mass**

Figure 11 shows the results of analysis to determine the increase in the amount of payload mass placed on the lunar surface using OOS. The analysis is based on a 10 MT vehicle for transfer of the crew to and from the Moon along with a 10 MT lander, and a launch vehicle payload mass capability to LEO of 120 MT; the number of launch vehicles is restricted to 3. The use of OOS and pre-positioned fuel depots is compared to that of sending the aggregate mission mass on the initial transfer leg. As indicated, the increase in payload mass to the surface can be significant as the trip time is reduced from 96 to 24 hours and the associated ΔV increases. In each case, the payload mass to the lunar surface increases by over 10 MT, the 36-hour case showing the greatest improvement at approximately 25 MT. The advantage of using OOS for the 96-hour trip time is somewhat lower due to the low ΔV required.

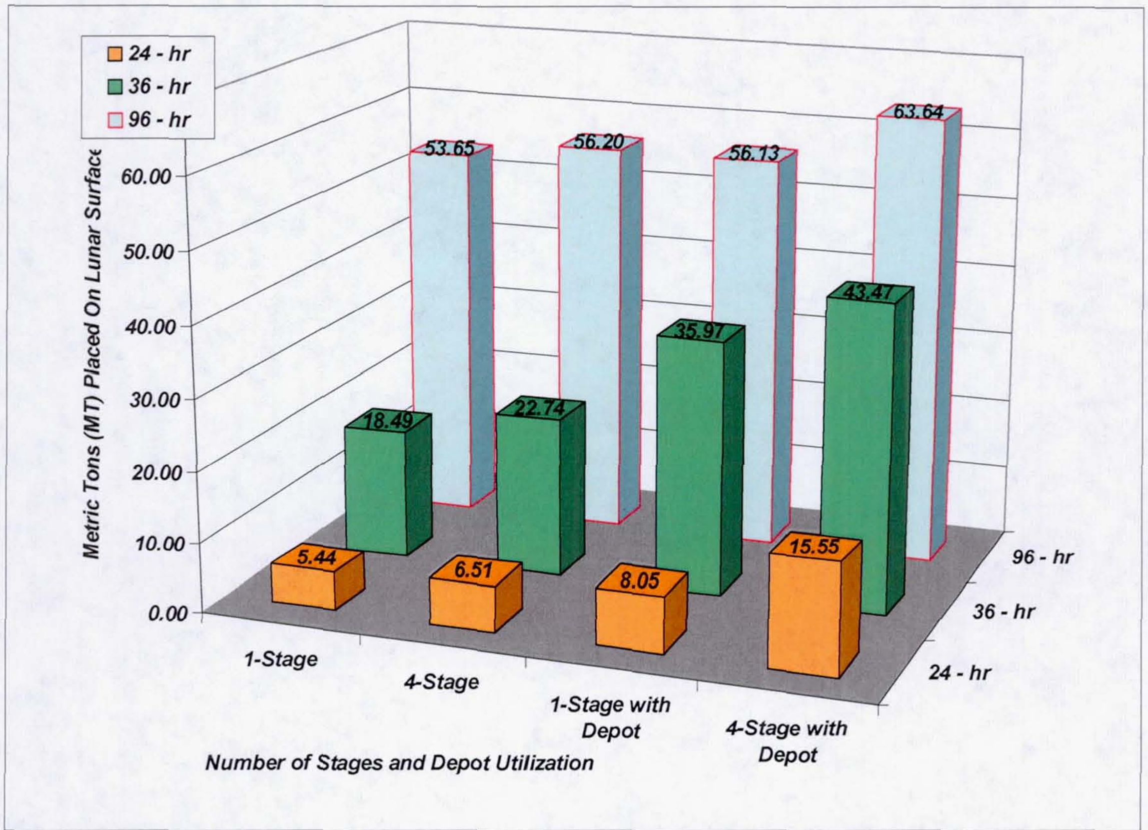


**Figure 11: Surface Payload Mass For 24, 36 and 96-Hour Lunar Transfers Using OOS With 120 MT Launch Capability**

## ADDRESSING EXPLORATION COST DRIVERS WITH OOS AND DEPOTS

One large cost driver in use of OOS and depots for Mars missions is the cost of launching the propellant - particularly for fast round trips using bi-propellants on the return leg. We believe that the cost of launching propellants can be reduced substantially, given the nature of the payload: inexpensive, and insensitive to acceleration. Optimizing launch trajectories, increasing g-loads, and eliminating redundant systems that are simply not worth buying to launch cheap payloads might reduce the cost of existing launchers by several tens of percentage points, at little or no development cost. Moreover, the potential savings from reducing the cost of launching such payloads from \$5000 per kilogram to \$1000 per kilogram would be tens of billions of dollars for our scenarios. Therefore, NASA would gain a substantial return on investment by investing several billion dollars to develop new launchers designed specifically to deliver propellant and similar cargos for that lower price.

The second major driver in the cost of the proposed Mars and lunar exploration missions is the cost of the dry spacecraft hardware. Current costs for hardware for human missions anywhere or for robotic Mars missions are on the order of \$400,000 per kilogram. Costs per unit mass of this magnitude make the total cost of any proposed exploration mission requiring several tens of metric tons of hardware prohibitive, regardless what mission architecture is used. OOS and depots can help alleviate this problem by relaxing the constraints on mass that drive up the cost per kilogram of space hardware. For decades, we have been willing to spend the extra millions to shave a few grams off of each component. If we are to achieve NASA's exploration goals, we need to save billions of dollars by accepting mass increases, as long as the net result is a reduction in total mission costs. To achieve revolutionary savings, we will need to begin using truly off-the-shelf commercial subsystems, meaning those used both inside and outside the aerospace industry, and we will need to integrate those commercial subsystems into a reliable space system by adding redundancy and shielding if needed. OOS and depots allow us to accept such mass penalties and still close the mission designs.

Another cost driver with most current Mars exploration architectures involving humans is the amount of infrastructure hardware and consumables mass that is required for a multi-year mission. The fast, 6 to 7-month total duration round trips enabled by OOS and depots eliminate the need for expensive, massive, difficult to develop systems such as closed environmental control and life support systems, extensive medical facilities, food production systems, long-life power systems, and in-situ propellant production facilities.

Finally with OOS and depots there is no need to invest in exotic, ultra-high-$I_{sp}$ propulsion capabilities to meet mission requirements, another major cost driver in many proposed Mars exploration architectures.

**NEW PROPULSION TECHNOLOGY**

Although OOS and depots do not require revolutionary advances in propulsion technology, there are several potential improvements that could enhance the performance of our proposed architecture. The first is improved cryogenic storage capabilities that would reduce mission risk by enabling longer storage, or lower propellant losses in LEO. We do not believe that cryogenic propellants will be viable at Mars for many decades, because the infrastructure required to store the propellants would be prohibitively expensive in terms of both mass/power and financial resources. As a result, all of our example cases include bi-prop alternatives.

The second potentially high-payoff improvement is development of alternative propellant formulations that would increase the performance of existing chemical systems. Nano-particles added to solids, gels, liquids and hybrids increased their performance, compared to current formulations, offering storable propellants with performance closer to that of cryogens. They appear to hold particular promise for applications where high thrust is acceptable, such as the launching of acceleration-insensitive payloads. Considerable work has already been done on such additives, and development costs would be relatively low.

**MAJOR FINDINGS FROM THIS ANALYSIS**

1. Equation 8 highlights the fact that additional variables, other than $I_{sp,}$ can be manipulated to increase payload performance of rocket propulsion systems. Lightweight vehicle structures and the utilization of OOS offer significant opportunities for enhanced performance.
2. OOS from LEO and at the destination using depots opens up the possibility of human exploration of Mars within the next two decades using conventional chemical propulsion.
3. OOS and depots provide significant improvements in mission capabilities for human lunar exploration, such as landed mass, transfer times, and return mass.
4. OOS and depots mitigate the performance reduction imposed by the use of lower $I_{sp}$ chemical propellants that have other favorable characteristics such as low-cost and storability.
5. OOS and depots point to a third class of payload aside from crew and high-value cargo: acceleration insensitive cargo such as food, water and propellant. Given that this type of payload makes up well over 90% of all Earth-to-orbit mission mass for the analyzed exploration missions, there is potential for a huge payoff from developing low-cost methods to launch these mission elements.
6. OOS and depots reopen the evaluation of mission architectures with respect to absolute mass, cost, risk and reliability.

**CONCLUSIONS**

On-Orbit Staging alone results in a substantial increase in payload mass over current methods as verified by simulations using operational software. OOS enables missions that are not feasible by current launch methods, increases the ratio of payload mass to launch mass, and can be applied to any class of launch vehicle or mission design. Adding pre-positioned fuel depots permits an order-of-magnitude reduction in required resources in the high $\Delta V$ fast-transfer cases we have analyzed. The combination of OOS with depots enables fast transfers of humans to Mars with robust amounts of hardware. The relaxing of mass constraints pays off in multiple ways. It reduces the design costs to build systems under tight mass budgets, enables the use of multiple-fault-tolerant systems, allows for flexible program schedules, and enables the use of off-the-shelf parts. OOS and depots enable robotic exploration of the entire Solar System, fast-transfer trajectories, payload masses measured in metric tons rather than tens or hundreds of kilograms, and robotic Mars sample returns on the order of hundreds of kilograms rather than grams. Although the assumption of optimal staging used in our analysis cannot be applied in the purest sense in real-world situations, the performance gains that it offers can be closely approached by judiciously selecting the sizes and combinations of standardized propulsion elements. Trades need to be performed on $\Delta V$ allocations, staging design, trip durations, and propulsion system parameters, in order to yield the most efficient fast-transfer scenarios.

## ACKNOWLEDGEMENTS

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