Lunar Orbit Propellant Transfer

Steven S. Pietrobon

Abstract — **An investigation of various crewed Lunar transportation schemes using liquid oxygen and liquid hydrogen is made. These include the traditional "direct ascent" approach as well as more advanced schemes that use Lunar oxygen and hydrogen. One such scheme involves a lunar orbit rendezvous between a spacecraft returning to Earth and another spacecraft heading for a Lunar base. The returning spacecraft delivers Lunar oxygen to the landing spacecraft in Lunar orbit. We call this Lunar orbit propellant transfer (LOPT). Since the oxidiser to fuel ratio is very high (greater than five) this reduces the required propellant mass that is delivered into trans Lunar injection (TLI). In fact, we show that the higher the oxygen to hydrogen mass ratio, the smaller the propellant mass that is delivered to TLI. Lunar surface propellant transfer (LSPT) using Lunar oxygen, oxygen/hydrogen, and oxygen/hydrogen with LOPT are also investigated. In general, LOPT has about 23% better performance than LSPT, at the expense of increased complexity and doubling the amount of required Lunar oxygen.**

Index Terms — **Lunar oxygen, Lunar hydrogen, Lunar transportation, propellant transfer**

I. Introduction

Future reusable launch vehicles are expected to only deliver between 20 to 30 t (1 t = 1000 kg) into low Earth orbit (LEO). For example, the proposed VentureStar vehicle can only deliver 26.8 t into a 185.2 km, 28.5° orbit [1]. This greatly reduces the available payload that can be delivered to the Lunar surface if a crewed Lunar base needs to be established. Schemes have been proposed where only Earth hydrogen fuel is delivered to the Lunar surface. Lunar oxygen oxidiser extracted from Lunar soil [2,3] is then used to return a crew back to Earth [4]. We call this Lunar surface propellant transfer (LSPT). A more advanced scheme involves a lunar orbit rendezvous between a spacecraft returning to Earth and another spacecraft heading for a Lunar base. The returning spacecraft delivers Lunar oxygen to the landing spacecraft in Lunar orbit. We call this Lunar orbit

A paper submitted to the *8th International Aerospace Congress* 20 April 1999. The author is with Small World Communications, 6 First Avenue, Payneham South SA 5070, Australia.

propellant transfer (LOPT). Since the oxidiser to fuel ratio is very high (greater than five) this reduces the required propellant mass that is delivered into trans Lunar injection (TLI).

We investigate five different techniques for transporting a crew to and from the Moon. The first is the "direct ascent" method where no Lunar resources are used. The second and third method are LSPT and LOPT where only Lunar oxygen is used. With the discovery of large amounts of water ice at the poles of the Moon [5], we also investigate LSPT and LOPT using the hydrogen and oxygen in the ice.

II. Delta V Calculations

We first determine the Δv's for trans Lunar injection (TLI), Lunar orbit insertion (LOI), trans Earth injection (TEI), Lunar descent (LD) and Lunar ascent (LA). These Δv's are then used to determine the spacecraft masses for the various methods given in later sections.

II.A *Trans Lunar Injection*

We assume the Lunar spacecraft is injected into a trans Lunar orbit. This requires a worst case delta V of 3141 m/s if the initial circular orbit around Earth has an altitude of 185 km above Earth's surface (see Appendix A). The rocket equation can be written as

$$\Delta v = v_e \ln(1 + m_p/m_f) \tag{1}$$

where Δv (m/s) is the change in speed, v_e (m/s) is the exhaust speed of the rocket engine (divide by $g = 9.80665$ m/s$_2$ to obtain specific impulse in seconds), m_p is the propellant mass, and m_f is the final mass. If we assume that the O_2/H_2 RL–10B–2 is used with v_e = 4531 m/s [9] then $m_p = m_f$.

Due to the limited payload of reusable launch vehicles, the TLI stage and Lunar spacecraft are launched separately. The two vehicles then perform an Earth orbit rendezvous (EOR) before continuing to the Moon. However, with $m_p = m_f$ this implies that the Lunar spacecraft cannot take full advantage of the available payload mass since m_f includes both the TLI stage and Lunar spacecraft mass. We solve this disadvantage by having the Lunar spacecraft complete the TLI burn immediately after the TLI stage has completed its burn. This puts the TLI stage into a highly elliptical Earth orbit instead of into a Solar orbit.

In our paper we have assumed a "clean space" policy. That is, no uncontrolled stages or spacecraft are left in Earth, Lunar, or Solar orbit where they can potentially impact other operating spacecraft or bases on the Moon. The above scheme fits this perfectly since the TLI stage can perform a small burn at apogee and re–enter the Earth's atmosphere where it will be harmlessly destroyed.

To increase payload mass we assume that a near single stage to orbit (NSTO) orbit is used [6]. That is, the reusable vehicle deploys the payload soon after burnout and lands back at the launch site

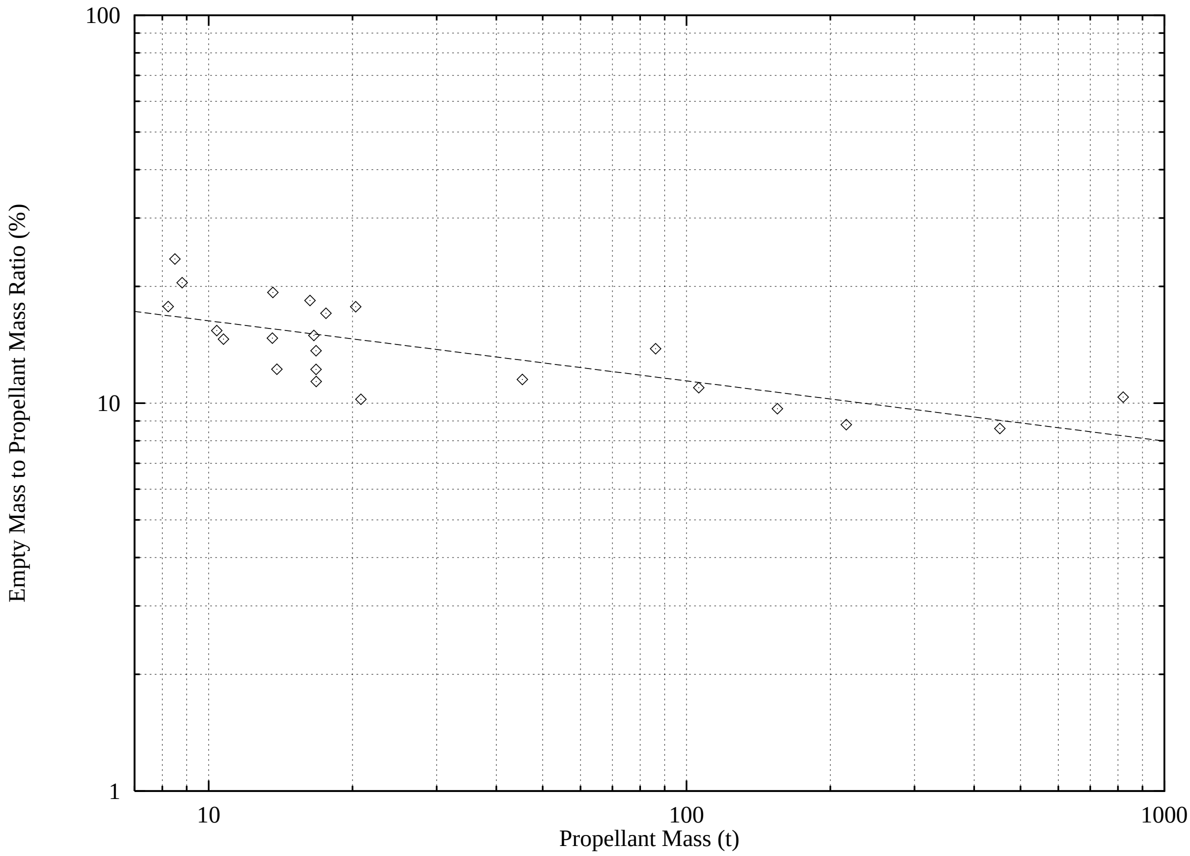

Figure 1: Mass ratio versus propellant mass for O_2/H_2 stages.

after a single 20×185 km orbit around the Earth. The payload performs a 50 m/s burn at apogee to circularise its orbit. For VentureStar the payload mass into the 20×185 km orbit is 29.5 t [6].

We now wish to estimate the propellant and empty mass of the TLI stage. Figure 1 plots the mass ratio m_e/m_p (%) against m_p (t) for various O_2/H_2 stages that were found in [9] (m_e is the empty mass). Also plotted is a line of best fit which gives the relation

$$m_e = 0.233 m_p^{0.875} \tag{2}$$

where m_e and m_p are in tonnes. Given $m_e + m_p = 29.5$ t and (2) we obtain $m_e = 3.97$ t and $m_p = 25.33$ t. For the 50 m/s circularisation burn 0.33 t of propellant is required giving a total mass into orbit of 29.17 t, an 8.8% increase over the normal 26.8 t. The Lunar spacecraft performs a similar burn.

The Δv for the TLI stage with $m_p = 25$ t and $m_f = 33.14$ t is 2547 m/s. We decrease this to 2496 m/s so as to have a 2% propellant margin. This puts the TLI stage into a $185 \times 38{,}289$ km orbit and implies that the Lunar spacecraft needs to perform a 633 m/s burn for Lunar perigee or a 645 m/s burn for Lunar apogee. Together with the 50 m/s circularisation burn and a 2% margin, this implies the total Δv to TLI for the Lunar spacecraft is 697 m/s or 709 m/s for Lunar perigee or apogee, respectively.

II.B *Lunar Orbit Insertion and Trans Earth Injection*

For LOI the task is to insert the spacecraft into a circular orbit around the Moon. An important consideration is whether the LOI burn is performed on the near or far side of the Moon. A near side burn puts the Lunar spacecraft on a posigrade orbit of the Moon. This allows the LD and LA to reduce their burns by up to 4 m/s each due to the rotation of the Moon. Using a far side burn, a retrograde orbit is achieved. This allows a morning landing to be achieved with the Sun behind the spacecraft. Also, if LOI is not achieved, the gravity assist from the Moon changes the orbit so that the spacecraft heads back to Earth. For a missed near side LOI burn, the orbit is changed so that the spacecraft heads out to space. Thus, like the Apollo missions, a far side LOI is selected due to its landing and safety advantages at the penalty of a very small increase in Δv.

The Moon's orbit has an inclination that varies from 4.95° to 5.35° to the ecliptic. However, the nodes of the Moon's orbit has a period of 18.6 years which causes the inclination of the Moon's orbit relative to the Earth to vary from 23.45–5.35 = 18.1° to 23.45+5.35 = 28.8° [16]. A launch at 28.45° inclination thus results in a worst case inclination difference of θ = 28.45–18.1 = 10.35°.

The relative velocity v_{MS} between the Moon with velocity v_M and the Lunar spacecraft at velocity v_S with angle θ is given by

$$v_{MS} = \sqrt{v_M^2 + v_S^2 - 2v_M v_S \cos\theta}\,. \tag{3}$$

For Lunar perigee we have v_M = 1099 m/s and v_S = 200 m/s giving v_{MS} = 903 m/s. For Lunar apogee we have v_M = 963 m/s and v_S = 176 m/s giving v_{MS} = 790 m/s. To go into a circular orbit around the Moon the required Δv is

$$\Delta v_{\mathrm{LOI}} = \sqrt{2v_o^2 + v_{MS}^2} - v_o \tag{4}$$

where v_o is the circular velocity around the Moon [7]. For an altitude of 100 km v_o = 1633 m/s. Thus Δv_{LOI} is 847 m/s and 808 m/s for Lunar perigee and apogee, respectively. We include a 4% propellant margin to give the total TLI and LOI Δv of 1578 and 1549 m/s for Lunar perigee and apogee, respectively. Since a Lunar perigee TLI and LOI has the higher Δv, we let the combined TLI and LOI burns be Δv_1 = 1578 m/s.

For TEI the spacecraft is in a Lunar orbit of 100 km and performs a burn to put the re–entry capsule on a direct path to the surface of the Earth. An alternative technique has the capsule going partially into the atmosphere and performing a small burn to put the capsule into LEO. However, this will require another launch and rendezvous to bring the astronauts back to Earth. Also, the capsule will need to be serviced in LEO or on the Moon. Thus, it is more practical and efficient to have a direct descent since we avoid an additional launch and servicing can be performed on the Earth.

For TEI, there is no need to change the inclination (unless another return inclination is desired), so $\theta = 0$. For the worst case Lunar perigee TEI we want the re–entry altitude to be $h_p = 122$ km [8] which gives $v_S = 199$ m/s. Thus $v_{MS} = v_M - v_S = 902$ m/s and from (4) $\Delta v_{\mathrm{TEI}} = 846$ m/s. Increasing this with a 4% margin gives $\Delta v_4 = 880$ m/s.

II.C *Lunar Descent and Ascent*

For LD we wish to deorbit the Lunar spacecraft in orbit around the Moon and land it on the Moon. The initial burn puts the spacecraft into a 0×100 km orbit with $v_p = 1703$ m/s and $v_a = 1610$ m/s. For a retrograde orbit, the minimum Δv to land the spacecraft is $\Delta v_{\mathrm{LD}} = v_o - v_a + v_p + v_r$ where v_r is the rotation speed at the equator of the Moon given by

$$v_r = \frac{2\pi R}{T}. \tag{5}$$

T is the sidereal period of the Moon (2,551,440 seconds) and R is the radius of the Moon. With $v_r = 4$ m/s we have $\Delta v_{\mathrm{LD}} = 1730$ m/s. For LD we add 20 seconds of hover time (for a Δv of 32 m/s under Lunar surface gravity of 1.6 m/s^2) and 8% margin for gravity losses to give $\Delta v_2 = 1903$ m/s.

LA is the reverse operation, but without the necessity of hovering. Thus $\Delta v_3 = 1868$ m/s.

III. Getting To The Moon And Back

In this section we investigate five different techniques for getting to the Moon and back. In all cases we assume that the payload mass into NSTO orbit is 29.5 t. Also, we assume that the maximum tank diameter for the Lunar vehicle is 2 m and the payload mass to the Moon and back are the same.

III.A *Direct Ascent*

In direct ascent, a two stage vehicle is used. The first stage is used for LEO circularisation, TLI completion, LOI, and LD. This requires a total $\Delta v = \Delta v_1 + \Delta v_2 = 3481$ m/s. The second stage is used for LA and TEI requiring $\Delta v = \Delta v_3 + \Delta v_4 = 2748$ m/s. From the rocket equation we have for these two burns

$$\Delta v_1 + \Delta v_2 = v_{e1} \ln\left(1 + \frac{(1 + R_1)m_{F1}}{m_e + (\gamma_{F1} + \gamma_{O1}R_1)m_{F1} + ((1 + \gamma_{F2})(1 + \gamma_{FB}) + R_2(1 + \gamma_{O2})(1 + \gamma_{OB}))m_{F2}}\right) \tag{6}$$

$$\Delta v_3 + \Delta v_4 = v_{e2} \ln\left(1 + \frac{(1 + R_2)m_{F2}}{m_e + (\gamma_{F2}(1 + \gamma_{FB}) + R_2\gamma_{O2}(1 + \gamma_{OB}))m_{F2}}\right) \tag{7}$$

where m_e is the vehicle mass (excluding tanks), R_i is the oxidiser to fuel mass ratio, v_{ei} is the vacuum exhaust speed, m_{Fi} is the fuel mass, γ_{Fi} is the fuel to fuel tank mass ratio, γ_{Oi} is the oxidiser to oxidiser tank mass ratio, γ_{FB} is the fuel boiloff to fuel mass ratio, and γ_{OB} is the oxidiser boiloff to oxidiser mass ratio, for stages $i = 1$ and 2.

Table 1: Estimated exhaust speeds for RL–10B–2.

R (O:F)	v_e (m/s)
5.0	4530.7
5.5	4528.6
6.0	4516.6
6.5	4495.5
7.0	4465.0
7.5	4423.2

Since we are given the total mass m =29.5 t, we wish to find m_e. We thus have a third equation

$$m = m_e + (1 + R_1 + \gamma_{F1} + \gamma_{O1}R_1)m_{F1} + ((1 + \gamma_{F2})(1 + \gamma_{FB}) + R_2(1 + \gamma_{O2})(1 + \gamma_{OB}))m_{F2}. \tag{8}$$

We can rearrange (6–8) into three linear equations that are a function of m_e, m_{F1}, and m_{F2}. These three equations are then solved by the Gaussian elimination with backward substitution algorithm [10].

Since we are assuming that O_2/H_2 propellants are used, there will be some boiloff of propellants while the Lunar vehicle is on the Moon. From [11], the optimum thickness of multi–layer insulation (MLI) for one Lunar day (30 Earth days) is 5 cm for H_2 and 7 cm for O_2. This corresponds to boiloff rates of γ_{FB} = 5.17% and γ_{OB} = 0.63% with insulation densities of 1.77 kg/m^2 for H_2 and 2.48 kg/m^2 for O_2. These values were used for the second stage propellant tanks.

We initially assume that γ_{Fi} = 0.5 and γ_{Oi} = 0.03 based on rough estimates of existing stages. However, these ratios will change depending on the volume of the propellant tanks. Given the fuel and oxidiser masses, we determine the tank volumes using 70.9 kg/m^3 for H_2 and 1149 kg/m^3 for O_2. In order to maintain a low centre of gravity, we have four H_2 tanks and two O_2 tanks for the first stage and two H_2 and one O_2 tanks for the second stage. Using the volume for each tank V in m^3, we assume the tank mass in kilograms is given by $32.3V^{0.795}$ for H_2 and $27.0V^{0.843}$ for O_2 [12]. In addition to this we add the MLI insulation mass. We also add the MLI mass to the first stage tanks as an estimate of the landing legs and other structure. These tanks masses are then used to calculate new values of γ_{Fi} and γ_{Oi} which are used to calculate new propellant masses. After only a few iterations, the propellant masses quickly converge to a constant value, in effect solving the non–linear equations.

Our equations also include two different oxidiser to fuel ratios. This is because the large H_2 tank volume is reduced with larger R's, which can compensate for the reduced exhaust speed. Table 1 gives the estimated exhaust speeds for various R's using [13] for the RL–10B–2. The largest R is 7.5 so as to be smaller than the stichometric ratio of 7.936 (an oxygen rich exhaust may not be feasible since it will burn a metal combustion chamber). The optimum ratios were then found by finding the maximum m_e out of all 36 combinations of R_1 and R_2.

III.B *Lunar Surface Propellant Transfer* (O_2)

In LSPT, the Lunar vehicle lands on the Moon and is supplied with Lunar O_2 for the return trip. Alternatively, the O_2 could be supplied on another mission from the Earth. The first stage consists of four H_2 tanks and the second stage of two H_2 and two O_2 tanks. The O_2 tanks are initially filled with enough O_2 to get to the Lunar surface. They are then partially filled for the return trip. The four empty H_2 tanks in the first stage are left on the Lunar surface.

The two main equations are

$$\Delta v_1 + \Delta v_2 = v_{e1} \ln\left(1 + \frac{(1 + R_1)m_{F1}}{m_e + (\gamma_{F1} + \gamma_{O2}R_1)m_{F1} + (1 + \gamma_{F2})(1 + \gamma_{FB})m_{F2}}\right) \tag{9}$$

$$\Delta v_3 + \Delta v_4 = v_{e2} \ln\left(1 + \frac{(1 + R_2)m_{F2}}{m_e + \gamma_{O2}R_1 m_{F1} + \gamma_{F2}(1 + \gamma_{FB})m_{F2}}\right). \tag{10}$$

The third equation for the total mass can be found from the numerator and denominator in (9).

III.C *Lunar Surface Propellant Transfer* (O_2/H_2)

In this version of LSPT, the Lunar vehicle is only a single stage. On the Lunar surface the four H_2 and two O_2 tanks are filled with Lunar O_2 and H_2 (either or both may also come from the Earth on another mission). If both propellants come from the Earth, one method may be to transport the propellant as water. The water would then be electrolysed and liquefied on the Lunar surface. This eliminates the need for large H_2 storage tanks on the refueling flight. This could be an intermediate step before full scale Lunar O_2 and H_2 production is achieved.

The two main equations are

$$\Delta v_1 + \Delta v_2 = v_{e1} \ln\left(1 + \frac{(1 + R_1)m_{F1}}{m_e + (\gamma_{F1} + \gamma_{O2}R_1)m_{F1}}\right) \tag{11}$$

$$\Delta v_3 + \Delta v_4 = v_{e2} \ln\left(1 + \frac{(1 + R_2)m_{F2}}{m_e + (\gamma_{F1} + \gamma_{O2}R_1)m_{F1}}\right). \tag{12}$$

III.D *Lunar Orbit Propellant Transfer* (O_2)

In LOPT, the Lunar vehicle enters Lunar orbit without enough propellant to land on the Lunar surface. Instead, it must rendezvous with another Lunar vehicle, returning from the Lunar surface to Earth. The returning vehicle supplies enough Lunar O_2 to the landing vehicle for it to land on the Moon. In case the landing and returning vehicles do not rendezvous, the landing vehicle carries sufficient O_2 for it to abort the mission and return to Earth. Similarly, the returning vehicle carries its own H_2 so it too can return to Earth if rendezvous is not achieved. In an abort the landing vehicle will need to dump excess H_2 and the returning vehicle dump excess O_2.

The first stage has four H_2 tanks and the second stage has two H_2 tanks and two O_2 tanks. The first stage tanks are left on the Lunar surface, while the O_2 tanks are filled with Lunar O_2. Mixture ratio R_1 is used for LOI and abort TEI and R_2 for LD, LA, and TEI. A high mixture ratio is expected to be used for R_2 in order to minimise the H_2 and H_2 tank mass from the Earth.

The five main equations for LOI, LD, LA, TEI, and abort TEI are

$$\Delta v_1 = v_{e1} \ln\left(1+\frac{(1+R_1)m_{F1}}{m_e+\gamma_{F1}m_{F1}+(1+\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+[(1+\gamma_{F2})(1+\gamma_{HB})+\gamma_{O2}R_2](m_{F3}+m_{F4})+(1-\gamma_{O2})R_1m_{F5}}\right) \quad (13)$$

$$\Delta v_2 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F2}}{m_e+\gamma_{F1}m_{F1}+(\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+[(1+\gamma_{F2})(1+\gamma_{HB})+\gamma_{O2}R_2](m_{F3}+m_{F4})-\gamma_{O2}R_1m_{F5}}\right) \quad (14)$$

$$\Delta v_3 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F3}}{m_e+(1+\gamma_{O2})R_2m_{F2}+[\gamma_{F2}(1+\gamma_{HB})+\gamma_{O2}R_2](m_{F3}+m_{F4})+(1+R_2)m_{F4}-(1+\gamma_{O2})R_1m_{F5}}\right) \quad (15)$$

$$\Delta v_4 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F4}}{m_e+\gamma_{O2}R_2m_{F2}+[\gamma_{F2}(1+\gamma_{HB})+\gamma_{O2}R_2](m_{F3}+m_{F4})-\gamma_{O2}R_1m_{F5}}\right) \quad (16)$$

$$\Delta v_4 = v_{e1} \ln\left(1+\frac{(1+R_1)m_{F5}}{m_e+\gamma_{O2}R_2m_{F2}+[\gamma_{F2}(1+\gamma_{HB})+\gamma_{O2}R_2](m_{F3}+m_{F4})-\gamma_{O2}R_1m_{F5}}\right) \quad (17)$$

III.E *Lunar Orbit Propellant Transfer* (O_2/H_2)

This scheme is very similar to the previous method, except that Lunar H_2 is also used for the return trip. To simplify operations and since the amount of H_2 mass involved is small, no Lunar H_2 is transferred from the returning vehicle to the landing vehicle. This results in a single stage vehicle with four H_2 tanks and two O_2 tanks. When the vehicle lands, the tanks will be empty. They are then refilled with the appropriate amounts of Lunar H_2 and O_2.

For the Δv's involved, the Lunar O_2 determines the size of the O_2 tanks and the LOI and LD determines the size of the H_2 tanks. The five main equations are

$$\Delta v_1 = v_{e1} \ln\left(1+\frac{(1+R_1)m_{F1}}{m_e+\gamma_{F1}m_{F1}+(1+\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+\gamma_{O2}R_2(m_{F3}+m_{F4})+(1-\gamma_{O2})R_1m_{F5}}\right) \quad (18)$$

$$\Delta v_2 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F2}}{m_e+\gamma_{F1}m_{F1}+(\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+\gamma_{O2}R_2(m_{F3}+m_{F4})-\gamma_{O2}R_1m_{F5}}\right) \quad (19)$$

$$\Delta v_3 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F3}}{m_e+\gamma_{F1}m_{F1}+[\gamma_{F1}+(1+\gamma_{O2})R_2]m_{F2}+\gamma_{O2}R_2m_{F3}+(1+\gamma_{O2})R_2m_{F4}-(1+\gamma_{O2})R_1m_{F5}}\right) \quad (20)$$

$$\Delta v_4 = v_{e2} \ln\left(1+\frac{(1+R_2)m_{F4}}{m_e+\gamma_{F1}m_{F1}+(\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+\gamma_{O2}R_2(m_{F3}+m_{F4}) - \gamma_{O2}R_1m_{F5}}\right) \quad (21)$$

$$\Delta v_4 = v_{e1} \ln\left(1+\frac{(1+R_1)m_{F5}}{m_e+\gamma_{F1}m_{F1}+(\gamma_{F1}+\gamma_{O2}R_2)m_{F2}+\gamma_{O2}R_2(m_{F3}+m_{F4}) - \gamma_{O2}R_1m_{F5}}\right) \quad (22)$$

IV. Results

A pascal program was written that solves the above five cases given m or m_e [14]. Table 2 gives the propellant and tank mass breakdowns (m_{Fi} and m_{Oi}), tank numbers (n_{Fi} and n_{Oi}) and lengths (L_{Fi}

and L_{Oi} assuming a maximum diameter of 2 m) and the optimum mass ratios with $m = 29.5$ t. Table 3 summarises Earth (m_{FE} and m_{OE}) and Lunar (m_{FM} and m_{OM}) propellant masses, total tank mass (m_T) and payload mass (m_e). Note that some masses may not add exactly due to roundoff errors.

Table 2: Mass breakdown of Lunar vehicle propellant and tank masses.

Method	m_{F1} m_{O1} (kg)	m_{F2} m_{O2} (kg)	m_{F3} m_{O3} (kg)	m_{F4} m_{O4} (kg)	m_{F5} m_{O5} (kg)	m_{FB} m_{OB} (kg)	m_{FT1} m_{OT1} (kg)	L_{F1} L_{O1} (m)	n_{F1} n_{O1}	m_{FT2} m_{OT2} (kg)	L_{F2} L_{O2} (m)	n_{F2} n_{O2}	R_1 R_2
Direct Ascent	2264 13586	811 4866				42 31	817 305	3.21 2.55	4 2	326 123	2.58 2.02	2 1	6.0 6.0
LSPT O_2	2264 13586	1175 8813				61	817	3.21	4	438 321	3.44 2.55	2 2	6.0 7.5
LSPT O_2/H_2	2264 13586	1383 10373								817 321	3.21 2.55	4 2	6.0 7.5
LOPT O_2	1446 7230	1086 8144	1430 10721	372 2793	514 2569	93	893	3.51	4	617 425	4.92 3.31	2 2	5.0 7.5
LOPT O_2/H_2	1446 7230	1061 7954	1545 11584	434 3257	599 2996					885 438	3.48 3.41	4 2	5.0 7.5

Table 3: Summary of Lunar vehicle masses.

Method	m_{FE} m_{OE} (kg)	m_{FM} m_{OM} (kg)	m_T m_e (kg)
Direct Ascent	3117 18483		1571 6329
LSPT O_2	3500 13586	 8813	1575 10838
LSPT O_2/H_2	2264 13586	1383 10373	1138 12512
LOPT O_2	4427 9799	 19089	1934 13340
LOPT O_2/H_2	2507 10226	1979 19799	1323 15445

Direct Ascent is only able to achieve a payload mass of 6.3 t. Since this includes engine, attitude control system, power system, and other masses, achieving this mass may be very difficult. A two person vehicle may just be possible.

With LSPT the payload mass increases by 71% to 10.8 t which gives much more freedom in the design. A three or four person vehicle should be able to be designed. With LOPT, we increase the payload mass a further 23.1% to 13.3 t, however Lunar O_2 production has to be increased by 117%.

Surprisingly, LSPT with O_2/H_2 has a smaller payload mass than LOPT. LOPT with O_2/H_2 has a 23.4% increase in payload mass over LSPT with O_2/H_2 and requires an 85% increase in Lunar O_2/H_2 production.

The tank lengths vary from 2 to 4.9 m. Figure 2 illustrates possible configurations of these vehicles using a single RL–10B–2 engine (diameter of 2.1 m and fully extended length of 4.1 m [9]). With a thrust of 110 kN, the engine can be used in all configurations. The RL–10 series of engines is capable of multiple restarts and can be designed to be throttleable from 30% to 100% as demonstrated by the RL–10A–5 on the DC–X [9]. Each vehicle has a circular diameter of around 7 m and heights from 9 to 12 m. Thus, it may not be possible to fit a Lunar vehicle within the payload bay of some reusable vehicles. In this case, an externally mounted payload such as that proposed in [6] may be a desirable solution to the problem. The externally mounted payload also allows the crew capsule to be ejected with a launch escape tower in case of a launch accident.

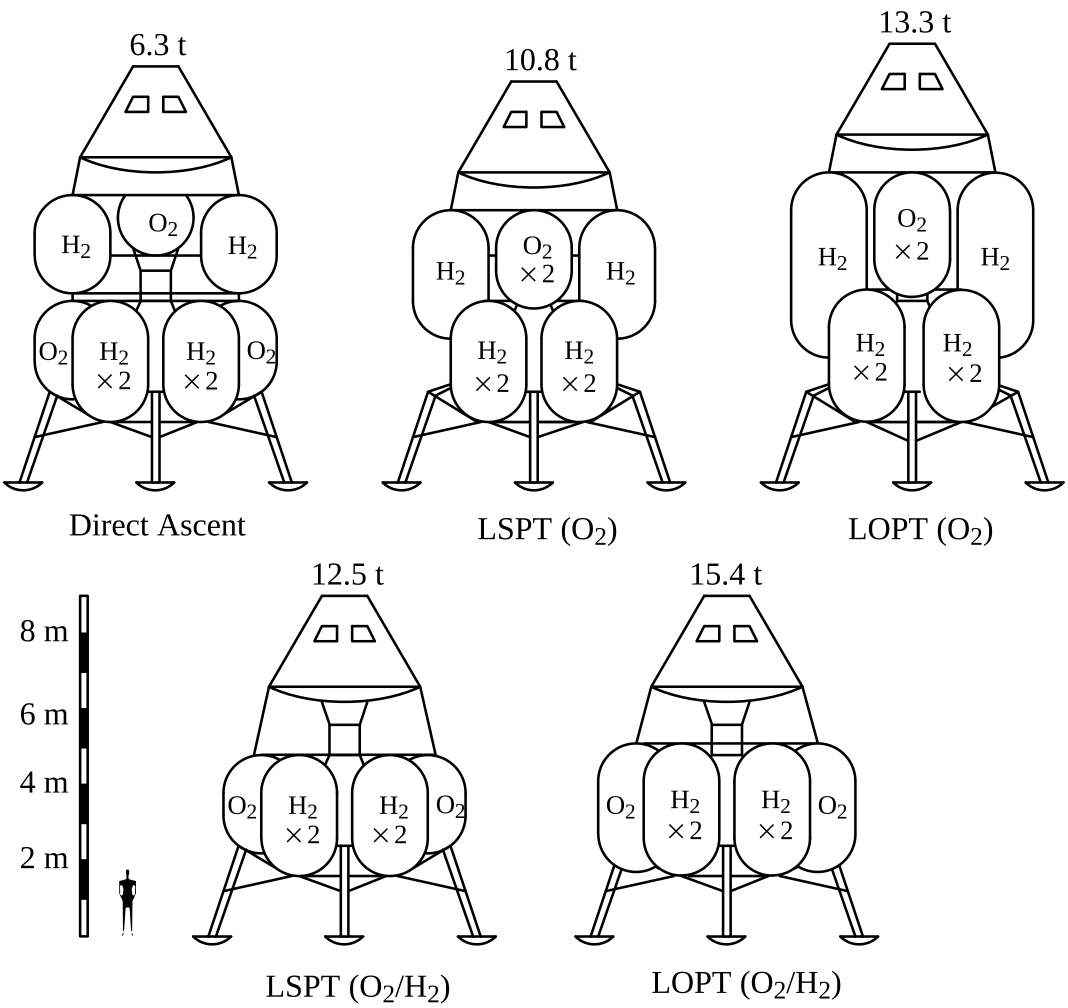

Figure 2: Lunar vehicle configurations for m = 29.5 t
(m_e given above each vehicle).

V. CONCLUSIONS

We investigated five different methods of getting a crewed vehicle to the Moon and back using O_2/H_2 propellants. It is shown that substantial payload gains can be achieved over Direct Ascent by using propellants from the Moon. Using Lunar O_2 only, payload gains of 71% and 108% were achieved by using Lunar surface and orbit transfer, respectively. With these schemes a combination of low and high mixture ratios will increase the payload mass. Using Lunar O_2/H_2, payload gains of 98% and 144% were achieved using Lunar surface and orbit transfer, respectively. LOPT can increase payload mass over LSPT by about 23%, however, the Lunar propellant production rate needs to be doubled.

The five schemes were investigated for a reusable launch vehicle (RLV) that is able to insert 29.5 t into an orbit of 20×185 km. Two launches were assumed, one for the trans Lunar injection (TLI) stage, and the other for the Lunar vehicle (LV). An Earth orbit rendezvous is then performed with the TLI stage which puts the LV in an elliptical orbit. After TLI stage burnout, the LV fires to complete the TLI burn so as make maximum use of RLV payload capability. Payload masses of 6.3 t for Direct Ascent were achieved, which may just be feasible for a minimum two person vehicle. LSPT and LOPT with Lunar O_2 achieved 10.8 t and 13.3 t, respectively. LSPT and LOPT with Lunar O_2/H_2 achieved 12.5 t and 15.4 t, respectively. A significant advantage of Lunar O_2/H_2 availability is that the Lunar vehicle is only a single stage. In fact, the whole vehicle could possibly be made reusable. Since the LV returns directly to Earth, orbital infrastructure is eliminated, greatly reducing costs and simplifying servicing of the LV.

Appendix A

To determine the required Δv's for changing from elliptical to circular (or vice–versa) orbits we use the following equations [15]. For a circular orbit we have

$$v_o = \sqrt{\frac{\mu}{R + h}} \tag{23}$$

where v_o is the speed for a circular orbit, μ is the combined gravitational parameter of the planet and satellite (μ = 398.6005 × 10^{12} and 4.90279 × 10^{12} m^3/s^2 for the Earth and Moon, respectively), R is the radius of the planet (R = 6,378,165 and 1,737,950 m [16] for the Earth and Moon, respectively), and h is the height above the planet's surface. For elliptical orbits we have

$$v_a = \sqrt{\frac{2\mu}{r_a(r_a/r_p + 1)}} \tag{24}$$

$$v_p = \sqrt{\frac{2\mu}{r_p(r_p/r_a + 1)}} \tag{25}$$

where v_a is the apogee speed, $r_a = R + h_a$ is the apogee radius, h_a is the apogee height, v_p is the perigee speed, $r_p = R + h_p$ is the perigee radius, and h_p is the perigee height. For the Moon, $r_{p,M}$ = 356,334 km and $r_{a,M}$ = 406,610 km [16].

Using (23) the circular orbit speed at h = 185 km is 7793 m/s. From (24), the perigee speed for a worst case trans Lunar injection (TLI) orbit with a perigee altitude of 185 km and an apogee radius of $r_{a,M} + R_M + h_M$ = 408,448 km is 10,934 m/s (R_M is the radius of the Moon and h_M = 100 km is the altitude of the Lunar orbit). Thus, the total Δv to change from a 185 km orbit to a TLI orbit is $v_p - v_o$ = 3141 m/s. To reach Lunar perigee the Δv is reduced by 12 m/s to 3129 m/s.

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