A Flexible Reusable Space Transportation System

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***Abstract* — A crewless reusable vertical take–off horizontal landing space transportation system is investigated. The single stage launch vehicle goes into a very low orbit around the Earth. At burnout, the payload is deployed. At apogee, the upper stage fires to put the payload into its desired orbit. The launch vehicle continues in a single orbit of the Earth, re–entering the atmosphere and returning to the launch site. We call this near single stage to orbit (NSTO). For satellite payloads, the satellite and upper stage are carried in a reusable pod attached to the top of the vehicle. For crewed vehicles, the pod is replaced by a small winged vehicle which can be used for crew transfer and rescue from the International Space Station. This allows the launch vehicle to use a common bulkhead between the fuel and oxidiser tanks, further reducing launch vehicle mass. A number of propellant combinations are investigated. Computer simulations indicate that liquid oxygen with quadricyclene promises to give the largest payload mass for similar size vehicles. The launch vehicle can also be modified to be a fly–back booster for a heavy lift launch vehicle (HLLV). In this case the upper stage and payload are replaced with jet engines and kerosene fuel tanks. A pod is used to recover the second stage engines for reuse.**

***Index Terms* — reusable launch vehicles, satellite launch vehicles, crewed launch vehicles, heavy lift launch vehicles**

I. Introduction

REUSABLE space transportation systems have traditionally examined two types of systems; two stage to orbit (TSTO) and single stage to orbit (SSTO). TSTO was originally studied for the Space Shuttle using liquid oxygen/liquid hydrogen (O_2/H_2) [1]. Due to high development costs, this had to be scaled back to a partly reusable system. Recently, interest has concentrated on SSTO systems, most notably the O_2/H_2 VentureStar [2].

In this paper we present an alternative reusable transportation system. The majority of SSTO systems assume that the vehicle goes into the required low Earth orbit, deploys its payload, and then

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returns to Earth, requiring at least a day in orbit. The payload then manoeuvres to its required orbit if necessary. A more efficient way to perform this task is for the launch vehicle to make only one orbit of the Earth with an apogee of say 185 km and a perigee only high enough for the launch vehicle to return to its launch site after a single orbit of the Earth. We call this near single stage to orbit (NSTO). The payload and its upper stage is deployed soon after burnout. At apogee the upper stage fires its engine to go into a transfer orbit, for example on the way to geosynchronous orbit or the International Space Station (ISS). A similar technique was studied in [3] for horizontal take–off and horizontal landing vehicles.

To minimise structure mass we assume that a common bulkhead exists between the fuel and oxidiser tanks. Also, we assume that the payload is carried piggyback on the crewless NSTO vehicle (NV) which is launched vertically, similar to the latest version of VentureStar [4]. This allows almost complete freedom in the size of the payload, compared to other SSTO vehicle designs where one is restricted to the volume in the internal payload bay.

The payload can consist of a satellite with its upper stage or a small crewed vehicle (CV). In a flight emergency the CV can separate from the NV and return to Earth, unlike an SSTO vehicle in which the crew is inside the cargo bay from which escape is difficult. The NV can also serve as the first stage of a heavy lift launch vehicle (HLLV). In this case the upper stage and payload are replaced with jet engines and kerosene fuel tanks. The O_2/H_2 second stage is attached underneath the winged NV. At NV burnout, the NV separates and flies back to the launch site. The engines for the second stage can be designed to be recovered.

The traditional propellant for SV has been O_2/H_2 for its high effective exhaust speed. However, O_2/H_2 suffers from a very low density. Recently, there has been interest in high density propellants. To investigate this further we performed extensive computer simulations of a variety of propellant combinations. These included combinations of liquid oxygen or 98% hydrogen peroxide with liquid hydrogen, methane, ethane, propane, kerosene, methylacetylene, or quadricyclene. Six space shuttle main engine (SSME) size engines with constant propellant volume flow rate are assumed in our simulations. The nozzles are assumed to be non–extendable.

We first investigate the potential payload gains that can be achieved using an NSTO type orbit. We next investigate the performance of various types of propellants, especially in relation to the propellant's impulse density. This is followed by presentation of computer simulation results of an NV using the previously studied propellants.

II. NEAR SINGLE STAGE TO ORBIT

The NSTO concept can be applied to almost any SSTO vehicle. A requirement is that the payload and upper stage can be deployed after burnout and before apogee is reached. This usually takes about 35 minutes. Also, the payload bay or pod needs to be large enough to accommodate any increase in size of the payload and upper stage.

Since most payloads go into orbit above 800 km altitude, these payloads already have an upper stage either incorporated into the satellite or as a separate stage. This is especially true for payloads intended for geosynchronous orbit. These payloads need to have the upper stage or propellant tanks enlarged to take advantage of the payload increase of an NSTO mission.

Missions to ISS seem to be the exception to the above. In this case large payloads and crew are carried directly to ISS by the Space Shuttle. A similar technique is also proposed for VentureStar. For an NV, large payloads will need an upper stage and a crewed vehicle (CV) will need to be able to re–enter the Earth's atmosphere. Since a crew rescue vehicle (CRV) already needs to perform this function, the CV and CRV functions can be combined into a single vehicle.

A separate CV has a number of advantages compared to the crew being carried in a cargo bay. It gives the astronauts more autonomy in reaching and returning from ISS. The CV can stay for an almost indefinite time at the ISS, with reduced drag, and act as the CRV. The CV can also be used to deliver and return cargo to and from ISS. The reduced payload of a CV compared to the Space Shuttle could be made up by more frequent flights of the CV.

Example 1: The VentureStar can deliver a cargo mass m_c = 22.68 t (1 t = 1000 kg) into a 185.2 km, 28.5° orbit [5]. The VentureStar empty mass is m_s = 116.57 t, propellant mass m_p = 1049.16 t, and effective vacuum exhaust speed v_e = 4462 m/s [5] (divide by g = 9.80665 m/s^2 to obtain specific impulse in seconds).

Using the rocket equation

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

where Δv is the change in velocity and $m_f = m_v + m_c$ is the final mass, the total Δv of VentureStar is 9567 m/s. To go from a 20×185.2 km orbit to 185.2 km circular orbit requires a Δv of 50 m/s (see Appendix A for calculations). This implies that the payload mass into a 20×185.2 km orbit increases by 3.38 t to 26.06 t (see Appendix B for calculations).

The payload then needs to perform a circularisation burn. Assuming storable propellants with v_e = 3065 m/s, a propellant mass of 0.42 t is required. This reduces the payload mass to 25.64 t (a 2.96 t or 13% overall increase).

Example 2: To go from a 185.2 km circular orbit to a geosynchronous transfer orbit with an apogee of 35,786 km requires a Δv of 2459 m/s. For VentureStar and assuming an upper O_2/H_2 stage with v_e = 4402 m/s this implies that the final mass (including the empty mass of the upper stage) is 12.97 t. For an NSTO mission the final mass increases by 1.77 t to 14.74 t.

Example 3: The International Space Station (ISS) orbit is at 51.6° inclination and 354 km altitude. For VentureStar to go into this inclination from the Kennedy Space Center latitude of 28.45° increases the total Δv by 130 m/s. To go from a 185.2 km circular orbit to a 354 km circular orbit requires a Δv of 98.3 m/s. To go from a 354 km circular orbit to a 20×354 km re–entry orbit requires a Δv of 98.5 m/s.

For payloads deployed in a 20×185.2 km or 185.2 km circular orbit we assume that storable propellants with a v_e = 3065 m/s are used. For VentureStar reaching the ISS orbit we assume that the main engines with a v_e = 4462 m/s are used.

Table 1 shows the various payloads that can be achieved. VS orbit is the highest orbit that VentureStar reaches. PL is the payload mass including the upper stage mass that is delivered to the ISS orbit. m_p is the storable propellant mass. CV is the crewed vehicle and payload that is delivered to the ISS orbit and which then returns to Earth.

Table 1: Payloads to ISS orbit

VS orbit (km)	PL+m_p (t)	CV+m_p (t)
354	13.1+0	12.8+0
185.2	17.4+0.6	16.9+1.1
20×185.2	20.3+1.0	19.7+1.7

The Automated Transfer Vehicle (ATV) could be used as an upper stage [6]. This vehicle has a dry mass of 8.5 t and a propellant mass up to 4.5 t. Thus, the payload mass is 11.8 t. For VentureStar a Multipurpose Logistics Module (MPLM) with a mass of 4.1 t could be used [7]. This gives a payload mass of 9.0 t, a 2.8 t decrease. The CV could be derived from the X–38/CRV program [8].

III. Example NSTO Vehicle Configuration

The previous section showed how the payload mass of an SSTO vehicle (SV) can be increased by adopting an NSTO strategy. By designing specifically for NSTO, further increases in payload mass may be possible. For example, an orbital manoeuvring system (OMS) may be deleted since no deorbit burn is required for the NV.

Nearly all proposed SVs have an internal cargo bay. The dimensions of the cargo bay are approximately 5 m in diameter and 20 m in length. This does not fit launch vehicle dimensions very well which usually have a diameter greater than 5 m and less than 20 m. The payload bay is also usually placed between the fuel and oxidiser tanks. This leads to significant wasted space and large structural mass between the tanks.

As demonstrated by the second stage of the Saturn V, significant reductions in structure mass can be achieved by using a common bulkhead between the fuel and oxidiser tanks. Modern launch vehicles such as the Ariane 5 also use this technique. This reduction in structure mass leads to a direct increase in payload mass or a reduction in the size of the launch vehicle.

Since achieving low structure mass for an SV or NV is going to be an already difficult problem, we assume a common bulkhead design. To further increase mass efficiency, we assume that the payload is externally mounted to the vehicle.

A large advantage is that payloads are not limited to the dimensions of a payload bay. This implies that low density or unusually shaped payloads can be carried, e.g., a crewed lunar lander. Also, the NV is only in space for only 1.5 hours, allowing a potentially higher utilisation rate.

Processing of satellite payloads should be similar to that of existing expendable launch vehicles. In this case, the satellite would be attached to its upper stage and then placed in a reusable pod. The cost and mass of the pod should be more than offset by the decreased structure mass and simpler design of the NV. The pod would then be attached shortly before launch using techniques similar to that of boosters to the sides of launch vehicles.

Carrying crewed vehicles that are externally mounted instead of being internally carried offers safety and performance advantages. If the NV were to fail, the CV can quickly separate and return to Earth. Also, no payload bay is required which further increases the advantages of a common bulkhead design NV.

There has been much argument over whether an SV should be horizontal or vertical in taking off and landing. Like the Space Shuttle and X–33, we choose vertical takeoff and horizontal landing (HTVL). This allows the NV to be used as a flyback booster in an HLLV.

We assume that six engines are used in our design. The initial acceleration is assumed to be 11.77 m/s^2 (1.2g) so as to allow single engine out survivability at lift–off. The maximum acceleration is assumed to be 29.42 m/s^2 (3g), the same as for the crewed Space Shuttle. The main diameter of the NV is assumed to be 8.4 m, the same as the external tank (ET) of the Space Shuttle. Figure 1 illustrates an approximation of what the NV might look like. The top view shows a satellite payload in a reusable pod (the satellite is ejected from the back like that in [3]) while the side view shows a CV as the payload. The side view also shows the forward reaction control system near the top of the vehicle and the forward and rear landing gear.

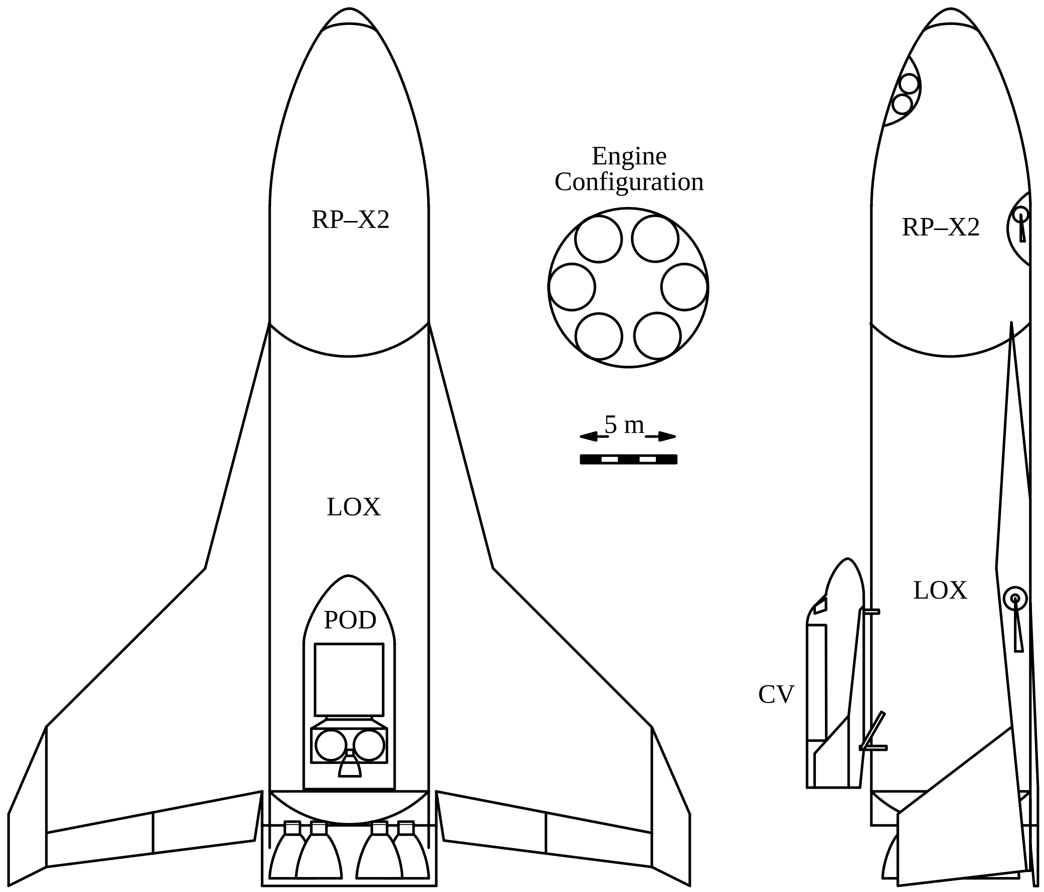

Figure 1: NSTO vehicle with satellite payload (overhead view)
and crewed vehicle (side view).

The wing span is 36 m and the vehicle length is 47 m. Wing area is 483 m^2. An important question is the choice of propellant for the launch vehicle. We discuss this in more detail in the next section.

IV. Choice of Propellant

Most SV's have assumed that O_2/H_2 is used due to its high exhaust speed. However, O_2/H_2 suffers from a low density. This implies that for a fixed propellant volume, not as much propellant can be carried as for a higher density propellant. To analyse this effect further, let us rewrite the rocket equation as

$$\Delta v = v_e \ln(1 + d_p V_p / m_f) \tag{2}$$

where d_p is the propellant density (kg/L, kilograms per litre) and V_p is the propellant volume. For low Δv's, we can approximate (2) with

$$\Delta v \approx I_d V_p / m_f \tag{3}$$

where $I_d = v_e d_p$ is the impulse density (Ns/L) of the propellant. One can think of the impulse density as the impulse (in Ns) per litre of propellant. Similarly, the effective exhaust speed v_e is the impulse

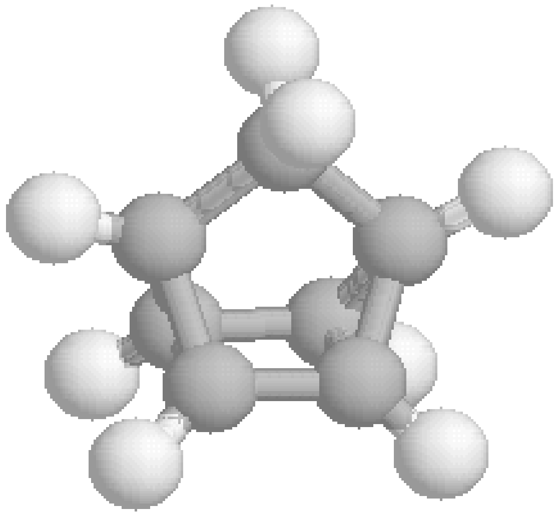

Figure 2: Quadricyclene (C_7H_8).

per kilogram of propellant (Ns/kg is the same as m/s).

From (3) we can immediately see that for a fixed propellant volume to final mass ratio, it is the impulse density that is most important. That is, we must take into account both exhaust speed and propellant density when considering which propellant is best. However, this is true only for low Δv's. For higher Δv's, the exhaust speed becomes more important, but the propellant density could still affect which propellant is best. That is, the best propellant is a function of the required Δv.

To investigate propellant performance we need to find the performance of various propellants. Table 2 gives the chemical formula, density, and heat of formation of various fuels and oxidisers [9]. Figure 2 shows the interesting shape of quadricyclene which was recently tested with in an Atlas vernier engine [10].

Table 2: Fuel and oxidiser parameters

Name	Formula	kg/L	kJ/mol
Liquid Oxygen	O_2	1.149	–12.98
Hydrogen Peroxide	H_2O_2	1.4424	–187.78
Liquid Hydrogen	H_2	0.0709	–9.01
Methane	CH_4	0.4239	–89.50
Ethane	C_2H_6	0.57	–99.37
Propane	C_3H_8	0.5853	–123.85
Kerosene (RP–1)	$CH_{1.9532}$	0.8	–24.10
Methylacetylene	C_3H_4	0.7	162.34
Quadricyclene (RP–X2)	C_7H_8	0.985	302.08

To determine the performance of various propellant combinations we assumed that the engine uses the same parameters as the space shuttle main engine (SSME). That is, a chamber pressure of 20.7 MPa and an expansion ratio of 77.5:1. The SSME was chosen since it is a high performance staged combustion engine that can operate from sea–level to vacuum. A bell–nozzle design was chosen instead of an aerospike design since a bell–nozzle can be used in any launch vehicle, while an aerospike needs to be specifically designed for the launch vehicle. For non O_2/H_2 propellant combinations a new engine is probably desirable, instead of modifying the SSME. This will allow the latest technology to be used so as to decrease engine maintenance costs and increase lifetime.

A program based on [9] was used to determine propellant density and exhaust speed. All exhaust speeds were normalised to the same efficiency of the SSME (97.4%). Except for O_2/H_2, the mixture ratio (MR) was chosen so as to maximise the exhaust speed. Table 3 gives the parameters for various propellant combinations, from lowest to highest impulse density. HTP is 98% H_2O_2 with 2% H_2O. The MR is by mass and oxidiser to fuel.

Except for O_2 with C_3H_4 and C_7H_8 and HTP with C_7H_8, as impulse density increases, propellant density increases and exhaust speed decreases. The MR for the SSME is 6.0. To get a higher impulse density we have increased the MR to 7.5 (below the stoichiometric ratio of 7.936).

Table 3: Propellant performance

Propellants	MR	d_p (kg/L)	v_e (m/s)	I_d (Ns/L)
O_2/H_2	5.0	0.3251	4455	1448
O_2/H_2	6.0	0.3622	4444	1610
O_2/H_2	7.5	0.4120	4365	1798
O_2/CH_4	3.6	0.8376	3656	3062
O_2/C_2H_6	3.2	0.9252	3634	3362
O_2/C_3H_8	3.1	0.9304	3613	3362
O_2/C_3H_4	2.4	0.9666	3696	3573
O_2/RP–1	2.8	1.0307	3554	3663
O_2/C_7H_8	2.4	1.0954	3628	3974
HTP/C_3H_4	6.5	1.2553	3319	4166
HTP/RP–1	7.3	1.3059	3223	4209
HTP/C_7H_8	6.6	1.3496	3288	4437

To understand the effect of impulse density further we plot Δv versus V_p/m_f (L/kg, litres per kilogram) using the exact rocket equation from (2) in Figure 3. We can see that up to about 2–3 km/s, the curves are nearly linear with a slope equal to the impulse density. This clearly indicates that for

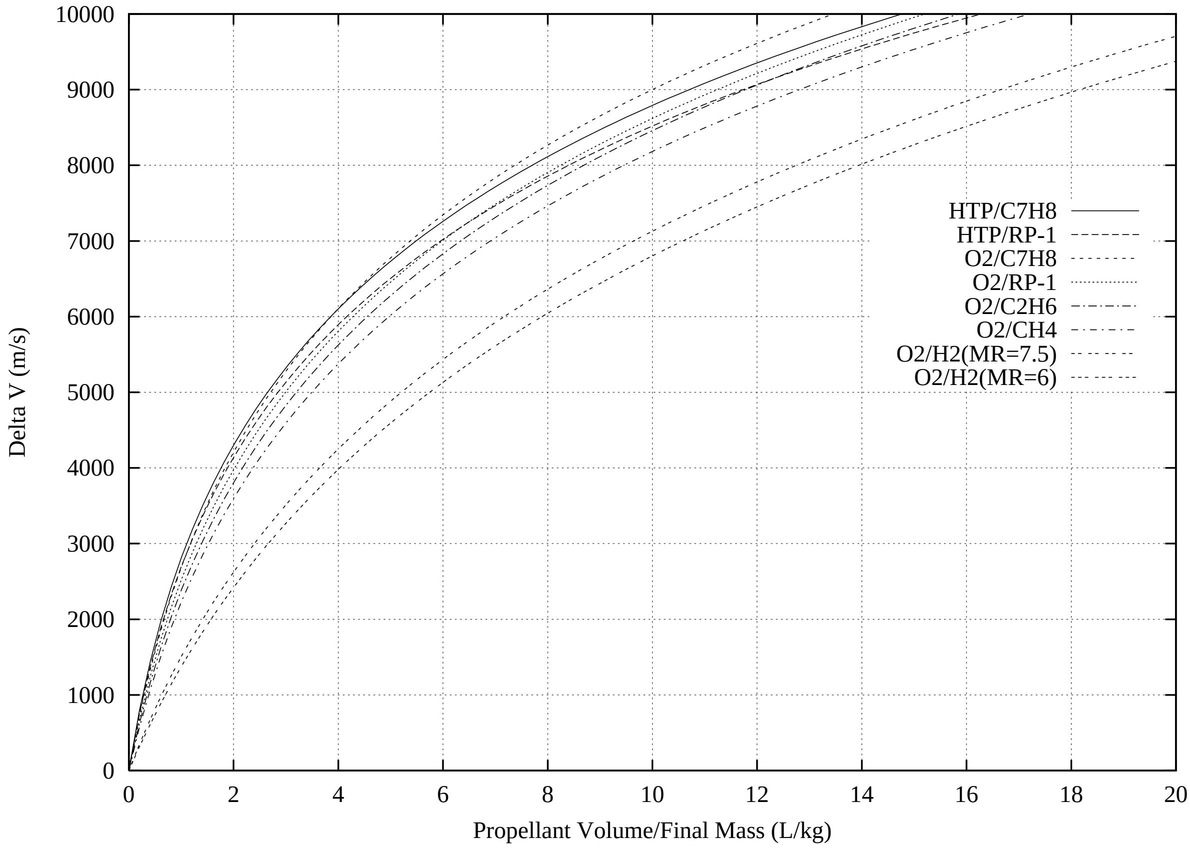

Figure 3: Delta V versus propellant volume to final mass ratio.

the first stage of a multistage launch vehicle one should choose a propellant that has the highest impulse density. In this case, the best propellant is HTP/C_7H_8 with the worst propellant being O_2/H_2.

Since the second stage of a multistage launch vehicle is very sensitive to mass we should choose the propellant with the highest exhaust speed, in this case O_2/H_2. Most launch vehicles reflect this, although the first stage propellant is usually a solid. For example, the solid rocket boosters on the Space Shuttle have an overall density of approximately 1.3 kg/L, a vacuum exhaust speed of 2637 m/s, and an impulse density of 3428 Ns/L [11].

For the NV we are interested in orbital speeds from 9 to 9.5 km/s. In this case, Figure 3 indicates that the best propellant is O_2/C_7H_8. However, the higher launch mass due to the greater propellant mass will result in increased structural loads and thus structure mass. We investigate this in the next section by performing computer simulations of an NV using various propellants.

V. Computer Simulations

To determine the performance of various propellants, a computer simulation of an NV into an 80×185 km orbit inclined at 51.6° was performed. The launch latitude was also assumed to be 51.6°. Since we assume that the propellant volume flow rate is constant (R_v = 1299 L/s for each engine at 100% throttle) the engine vacuum thrust (F_v) is proportional to the impulse density of the propellant. That is, $F_v = I_d R_v$.

The lift–off thrust is equal to $F_i = 6(R_i F_v - F_d)$ where R_i is the throttle setting (initially 1.04) and F_d is the sea level back–pressure force (F_d = 422.6 kN for the SSME). Since the lift–off acceleration is a_i = 11.77 m/s^2 (1.2g) the lift–off mass is equal to $m_i = F_i / a_i$. The vehicle then follows a vertical trajectory to an altitude of 56 m where it pitches over. A pitch over time is input to the program to specify the time the vehicle deviates from the inertial trajectory at an angle of –0.03°. The vehicle then follows a gravity turn such that the thrust vector is equal to the velocity vector of the vehicle relative to a rotating Earth. This maintains a zero angle of attack to the surrounding air. When the maximum acceleration of 29.42 m/s^2 (3g) is reached for the first time the angle of attack is made to gradually increase to a maximum positive value that is input to the program. As centrifugal forces increase on the vehicle, this causes the angle of attack to gradually decrease. More details of this algorithm can be found in [11]. The pascal source code and a 32–bit DOS executable for our 2–D simulation program are freely available from [12].

When an acceleration of 29.42 m/s^2 is reached the vacuum thrust of all the engines is reduced by a 1% increment. This repeats until the engine thrust reaches 65% (the current minimum of the SSME). In this case, a single engine is then shut down to reduce the acceleration. This process then repeats until the vehicle has reached an inertial speed of 7890.9 m/s. The one or two engines that are still firing are then shut down. This technique maximises the time that all engines are firing, thus allowing more abort options if an engine were to fail. A lower minimum thrust than 65% is probably more desirable since it would reduce gravity losses and have a larger number of engines still firing at engine cut–off.

To determine the final mass of the vehicle the Δv of the vehicle is determined using (1). This Δv value is then increased by a 1% safety margin and the final mass determined using this new Δv. By adjusting the pitch–over time and maximum angle of attack values, the vehicle can be usually placed into the desired orbit.

To obtain an 80×185 km orbit, the pitch–over time varied from 2.8 to 4.9 s. Maximum acceleration usually occurred at an altitude of around 40 km and a speed of 1750 to 2000 m/s. Maximum angle of attacks varied from 4.3° to 5.3°. Engine cut–off occurred at altitudes from 85 to 87 km. At this altitude, there is still significant drag and so the orbit at engine cutoff was higher than desired (usually about 82×218 km). As the 5 m diameter, 25 t payload ascended to apogee the orbit is reduced to the desired 80×185 km. At apogee, the upper stage fires its storable propellant engine to put it in a 185 km circular orbit. Total firing time is quite short at less than 6.5 minutes.

Figures 4 and 5 plot speed and altitude versus time for O_2/C_7H_8 propellant. The uneven plot of Figure 4 after 200 s is caused by the shutdown of five engines one after the other.

Table 4 gives the simulation results for the various propellant combinations given in Table 3. The Δv does not include the 1% overhead. The initial or lift–off mass $m_i = m_p + m_f$. The propellant volume is given in kL (kilolitres) which is equivalent to cubic metres (m^3). Note that to determine the payload mass, the structure, tank, and engine mass must be subtracted from the final mass. Thus, provided that any increases in structure, tank, or engine mass are not too great, the greater the final mass, the better the performance.

Table 4: Simulation Results

Propellants	Δv (m/s)	m_i (t)	V_p (kL)	m_f (t)
O_2/H_2 (6)	9339	893.0	2170	106.9
O_2/H_2 (7.5)	9303	1022.9	2194	118.8
O_2/H_2 (7.5–5)	9290	1022.9	2258	122.4
O_2/CH_4	9124	1893.1	2079	152.3
O_2/C_2H_6	9107	2099.9	2089	167.1
O_2/C_3H_8	9100	2099.4	2079	164.9
O_2/C_3H_4	9104	2244.8	2129	186.5
O_2/RP–1	9088	2306.9	2069	174.3
O_2/C_7H_8	9086	2521.1	2118	201.0
HTP/C_3H_4	9049	2653.5	1979	169.0
HTP/RP–1	9038	2682.8	1933	158.0
HTP/C_7H_8	9038	2840.2	1973	176.9

Figure 6 shows the vehicle (final) and propellant masses versus propellant. The propellant mass is broken down into fuel and oxidiser mass. Figure 7 plots propellant volume against propellant, showing the volumes of the fuel and oxidiser.

For O_2/H_2 three mixture ratios were investigated (these are shown in brackets in Table 4). The first MR is the same as the SSME and shows a final mass of only 106.9 t. By using a 7.5:1 MR, the final mass increases by 11.9 t. The third O_2/H_2 result is where the lift–off MR is 7.5. When the maximum acceleration is reached for the first time, one engine changes its MR from 7.5 to 5. This repeats until all engines are at 5:1. The engines are then throttled and shut down as before. A separate program was written to simulate this (the pascal source code and 32–bit DOS executable can be found in [12]). As can be seen, the final mass increases by only an additional 3.6 t.

Much larger increases in final mass can be achieved by using using higher density propellants. The highest final mass is with O_2/C_7H_8 which is 78.6 t greater than the best result achieved with

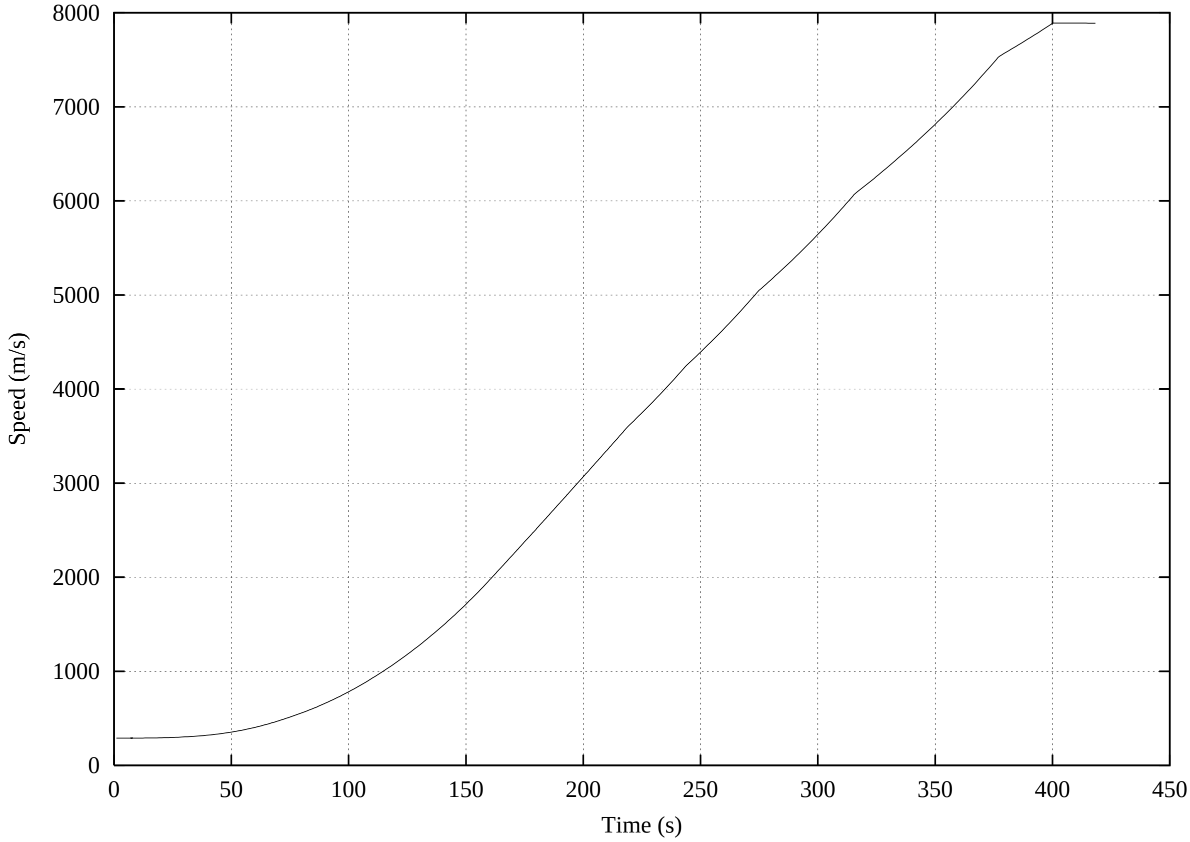

Figure 4: Speed versus time for O_2/C_7H_8 NSTO vehicle

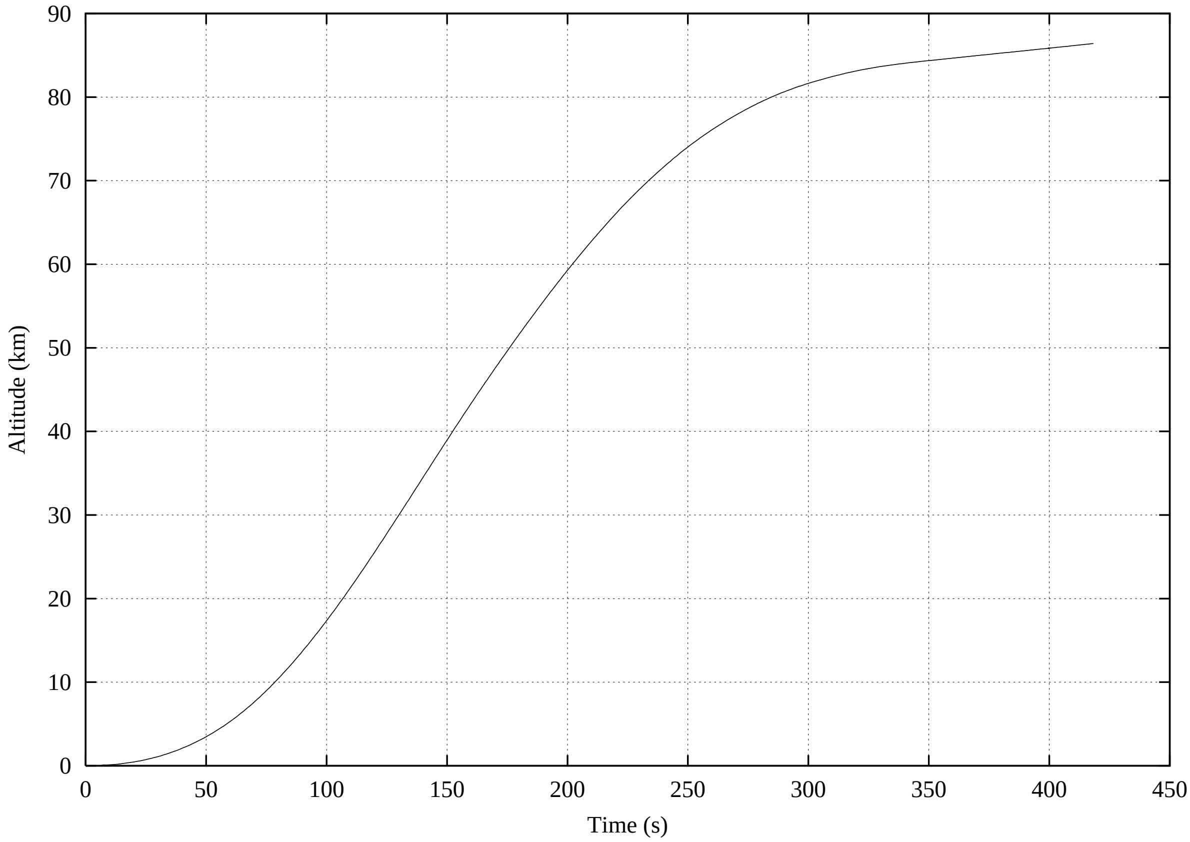

Figure 5: Altitude versus time for O_2/C_7H_8 NSTO vehicle

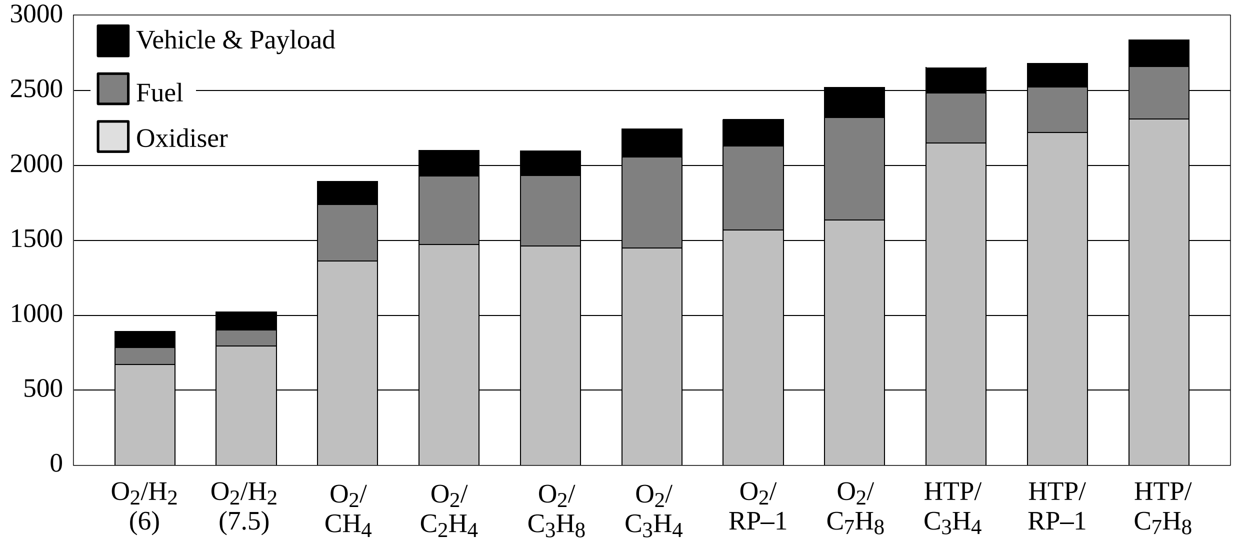

Figure 6: Vehicle and propellant mass (t) versus propellant

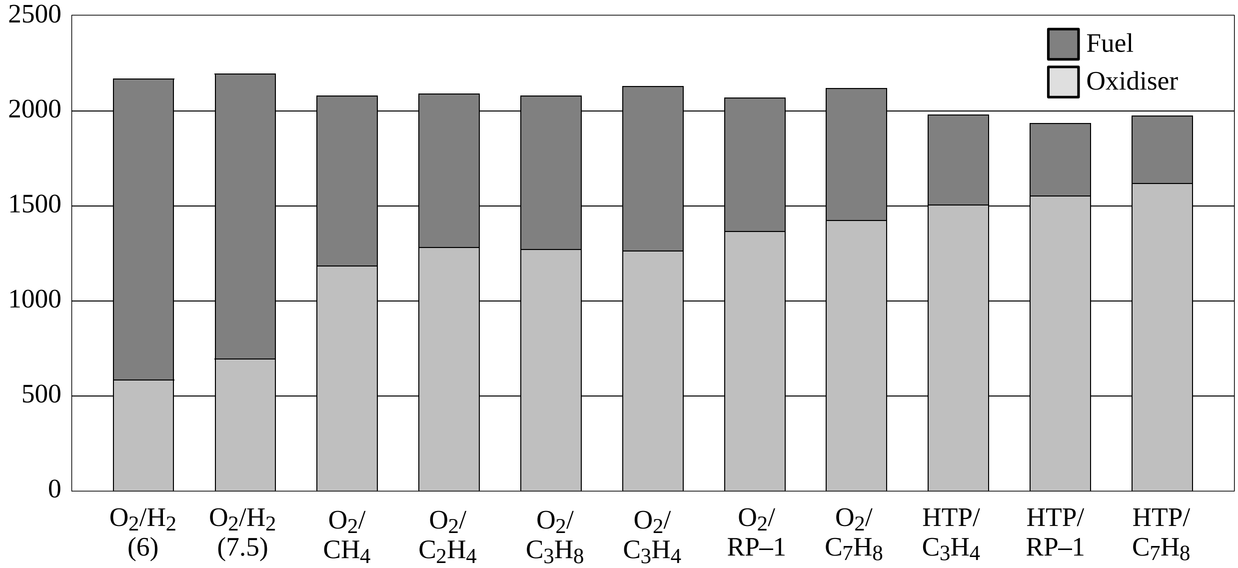

Figure 7: Propellant volume (kL) versus propellant

O_2/H_2. This potentially could lead to a 78.6 t increase in payload mass! However, since the initial mass is 146% greater than for O_2/H_2, the increased structural mass due to higher loads will reduce this increase by some degree.

It is interesting to see that the propellant volume is approximately 2000 kL for all the propellants investigated. This volume is about the same as in the S–IC first stage of the Saturn V. Interestingly, the highest volumes are for O_2/H_2. The lowest volume is for HTP/RP–1. As expected, the initial mass is roughly proportional to the impulse density of the propellant.

Figure 8 plots Δv versus impulse density for various propellants. As can be seen, there is an almost linear relation between between these two parameters. That is, the higher the impulse density, the

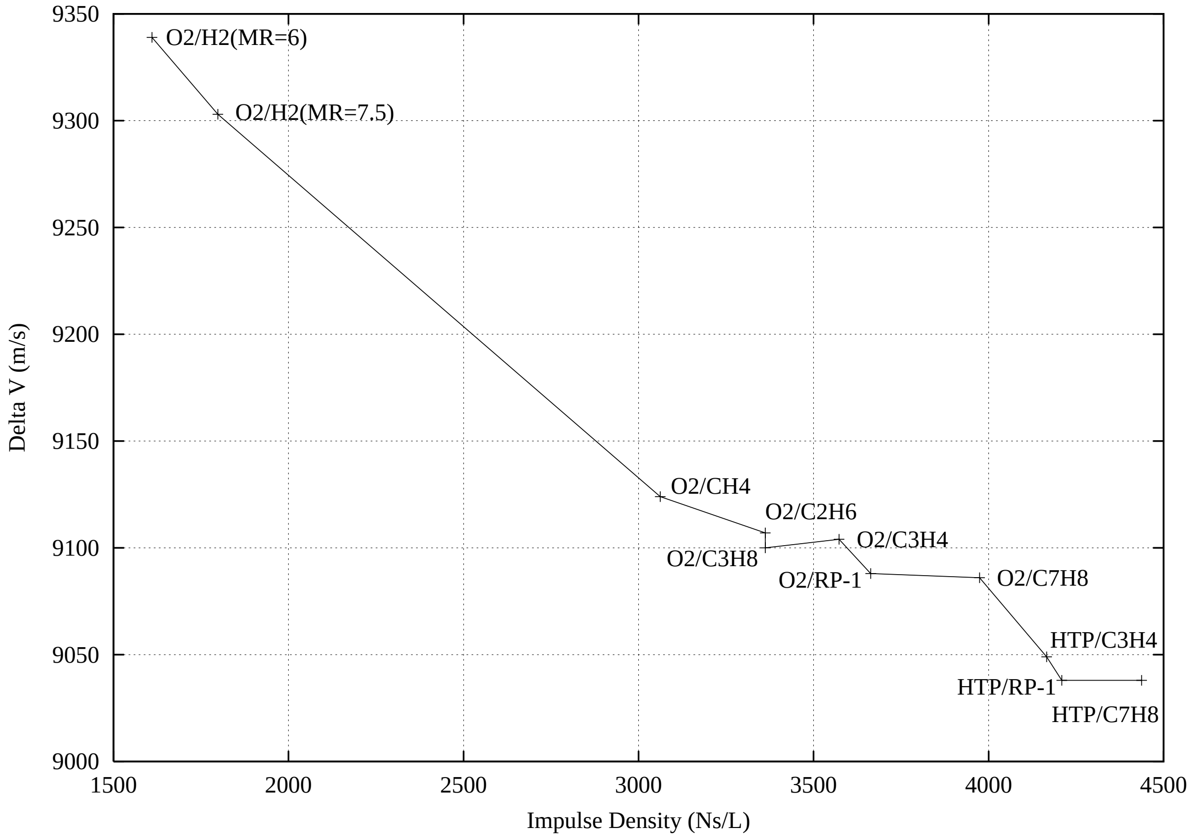

Figure 8: Delta v versus impulse density for NV.

lower the required Δv. There is some 300 m/s difference between the lowest and highest I_d propellants.

There are a number of reasons of why Δv is dependent on which propellant is used. The first reason is due to differences in exhaust speed [13]. Ignoring sea level performance losses we have that

$$a = \frac{F_v}{m_i - R_m t} = \left(\frac{1}{a_i} - \frac{t}{v_e}\right)^{-1} \tag{4}$$

where R_m is the mass flow rate and $a_i = F_v/m_i$ is the initial acceleration which is constant. The smaller v_e is, the faster that a increases with time. Higher accelerations therefore result in decreased losses due to gravity.

Secondly, for a fixed size engine with a constant propellant volume flow rate (R_v) and engine size we have

$$\frac{F}{F_v} = 1 - \frac{F_d}{I_d R_v} \tag{5}$$

where F is the engine thrust and F_d is the atmosphere back pressure force (equal to the nozzle exit area times air pressure). Thus, the higher the impulse density, the smaller the losses due to atmospheric back pressure.

Thirdly, the deceleration due to drag decreases with the higher launch mass (since the cross sectional area of the vehicle is assumed to be the same). As can be seen, all these three affects are dependent on each other, thus requiring computer simulations to determine the required Δv for each propellant.

We now attempt to estimate the tank, engine, and structure masses of the NV. From [14], carbon fibre composite tank mass for a horizontal tank–off vehicle is given as

$$m_t = aV^b \tag{6}$$

where m_t is the tank mass in kg, V is the tank volume in kL, and a, b are constants dependent on the propellant. We have a = 27.0, 32.3, 30.5 and b = 0.843, 0.794, 0.824 for liquid oxygen, liquid hydrogen, and JP–4 (a kerosene), respectively. Note that the tank volumes from Table 4 were increased by 2% to allow for ullage and engine start–up. Using the JP–4 values for all fuels except hydrogen and liquid oxygen values for all oxidisers, Table 5 gives the tank masses that were found. As can be seen, there is very little variation in tank mass. A protective liner for the oxidiser tank is assumed, similar to that proposed for the LOX/Kero SSTO Roton [15].

Table 5: Spacecraft masses

Propellants	m_f (t)	m_t (t)	m_e (t)	m_v(t)	m_c (t)
O_2/H_2 (6)	106.9	17.4	21.2	121.5	–14.6
O_2/H_2 (7.5)	118.8	17.8	21.6	124.0	–5.2
O_2/H_2 (7.5–5)	122.4	18.2	21.6	125.2	–2.7
O_2/CH_4	152.3	19.1	24.1	136.1	16.1
O_2/C_2H_6	167.1	19.1	24.7	138.2	28.9
O_2/C_3H_8	164.9	19.1	24.7	138.0	26.9
O_2/C_3H_4	186.5	19.5	25.2	140.6	45.9
O_2/RP–1	174.3	18.9	25.3	139.5	34.8
O_2/C_7H_8	201.0	19.3	26.0	142.6	58.3
HTP/C_3H_4	169.0	18.1	26.4	140.0	29.0
HTP/RP–1	158.0	17.6	26.4	138.7	19.2
HTP/C_7H_8	176.9	17.8	26.9	141.0	35.9

Table 6 gives the vacuum thrust, impulse density, and engine mass of various staged combustion engines [16]. A practical assumption is that the engine mass is proportional to propellant volume flow rate R_v. To test this assumption we plotted m_e/R_v (also given in Table 6 with units kgs/L) against thrust in Figure 9.

Table 6: Staged combustion engine performance

Engine	F_v (kN)	I_d (Ns/L)	m_e (kg)	m_e/R_v (kgs/L)
LE–7	1078	1583	1714	2.518
NK–15	1677	3387	1247	2.466
RD–253	1745	3663	1280	2.687
RD–0120	1961	1615	3500	2.883
SSME	2091	1609	3175	2.443
RD–180	4149	3387	5294	3.506
RD–170	7904	3387	8755	3.752

As can be seen, the m_e/R_v ratio seems to be dependent on F_v. A practical explanation for this is that the higher the engine thrust, the greater the stress on the engine and thus the more engine mass that is required. A line of best fit is also shown in Figure 9 and has the formula

$$m_e/R_v = 2.3 + 2\text{x}10^{-7}F_v. \tag{7}$$

If we assume that R_v = 1299 L/s (the same as the SSME) then we can determine the engine mass as a function of impulse density

$$m_e = 2.3R_v + 2\text{x}10^{-7}I_dR_v^2. \tag{8}$$

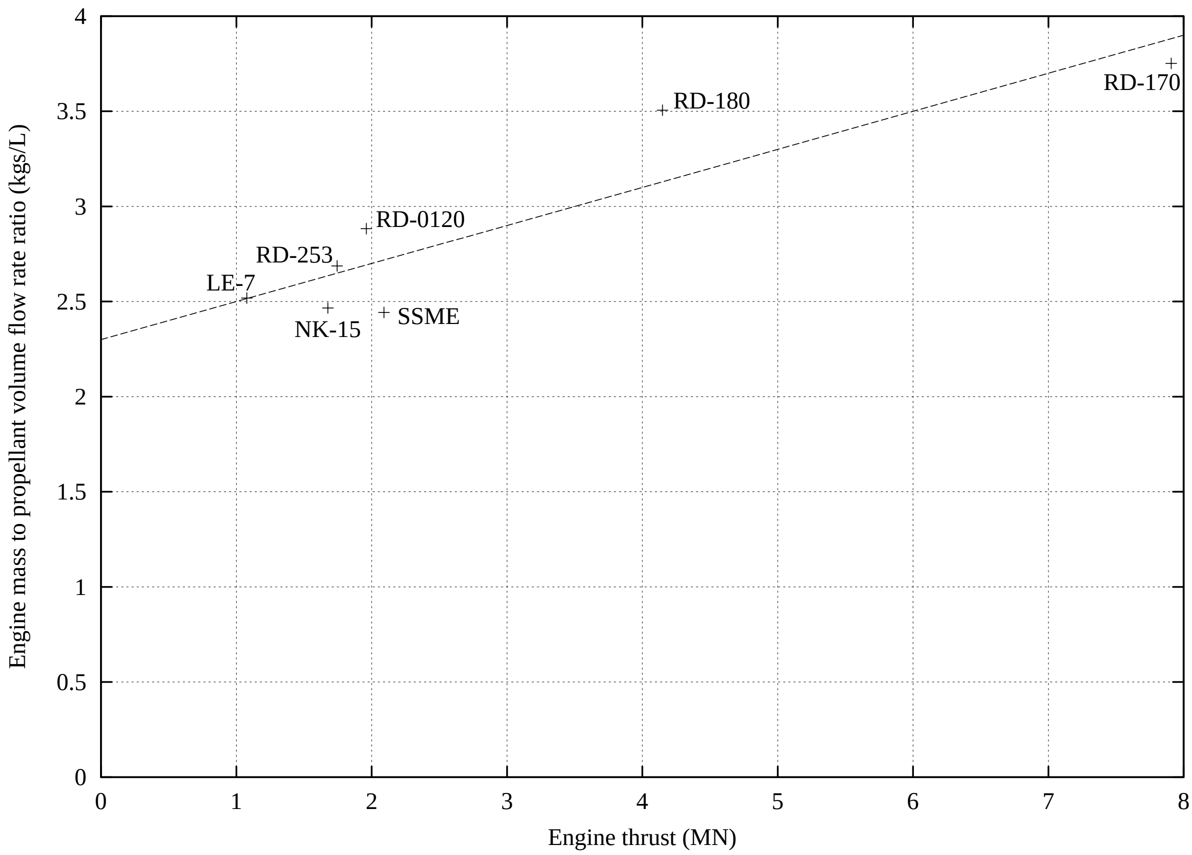

Figure 9: Engine mass to propellant volume flow rate ratio versus vacuum thrust.

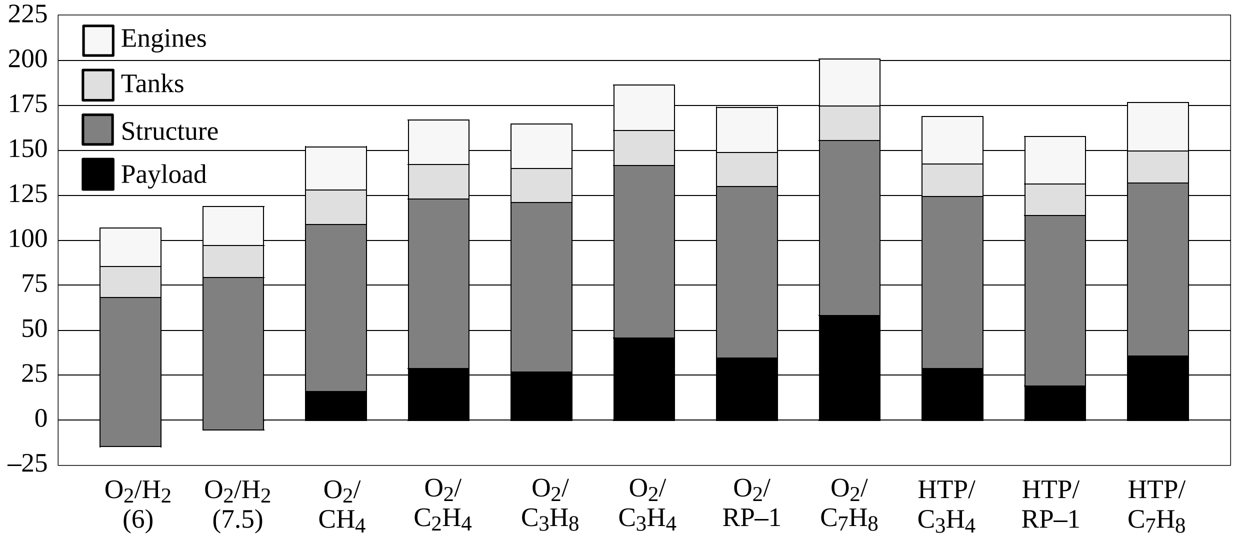

Figure 10: Vehicle mass (t) versus propellant.

The m_e in Table 5 gives the mass for six engines using (8). To determine the structure mass (including cold and hot structures, accessories, reaction control system, thermal protection, landing gear, subsystems and margins) we use the once–around Earth (OAE) vehicle in [3] as a baseline. Primary structures are in CFRP with ceramic, MMW and FEI thermal protection [3]. For the suborbital hopper and OAE vehicles in [3] $m_s = 2.15(m_t + m_e)$. Using this criteria, we determine the vehicle mass $m_v = m_s + m_t + m_e = 3.15(m_t + m_e)$ followed by the payload mass $m_c = m_f - m_v$ (values given in Table 5). Figure 10 shows the performance difference graphically. Surprisingly, all O_2/H_2 vehicles had negative payload mass. The larger engine mass of a vertical lift–off vehicle compared to the horizontal lift–off vehicle in [3] is the most likely reason for the negative payload mass. For all other propellants, a positive payload is achievable. O_2/C_7H_8 has the largest payload mass of 58.3 t as well as the largest payload fraction of m_c/m_f = 29%.

With a maximum landing mass of 206.1 t (used only in aborts) the wing loading is 4.2 kPa, slightly above that of the Space Shuttle with 4.1 kPa. For a nominal mission where the payload is deployed, the landing mass is 147.7 t and the wing loading is 3.0 kPa.

Some comment needs to be made as to why O_2/H_2 has been traditionally chosen for SSTO vehicles, whereas we have come to the opposite conclusion, that O_2/H_2 is the worst combination (at least for VTHL). We believe this is due to historical reasons where it was initially recognised that the high exhaust speed of O_2/H_2 gave significant payload increases when used in the second stage. It could have easily been thought that this advantage could flow into the first stage as has been demonstrated by the many single and two stage vehicles designed using O_2/H_2.

As we have shown, this is not true. For the first stage of a multistage vehicle, the propellant impulse density is the most important factor. O_2/H_2 has a poor impulse density and thus makes a poor first stage propellant. For NSTO or SSTO, high impulse density propellants are still desirable, although the highest impulse density propellants may not give the best performance.

VI. Heavy Lift Launch Vehicle

If the NV is used as a first stage of a HLLV, a VTHL configuration allows the externally mounted payload to be replaced with jet engines and kerosene tanks (flyback pod). This allows the NV to be flown back to the launch site after separation from the second stage of the HLLV. This further increases the flexibility of an NV.

For the HLLV an additional engine is mounted in the central position of the NV to give a total of seven engines. The second stage has three O_2/H_2 SSME engines in a recoverable pod attached to the bottom of the second stage O_2/H_2 tanks. Like the Space Shuttle, these engines are also burning at lift–off to increase the launch thrust and eliminate the need to ignite these engines at altitude. Assuming an initial acceleration of 11.77 m/s^2, m_i = 3387.7 t, 16% greater than the Saturn V. The flyback pod was estimated to have an empty mass of 13.9 t and a fuel mass of 37.8 t.

A simulation program was written to estimate the payload mass delivered into a 185 km circular orbit inclined at 28.45°. The payload is initially put into an 85×185 km NSTO orbit, with the payload firing its own engines to put it into circular orbit. The recoverable pod reenters the atmosphere so that the engines can be reused for another flight. The second stage pod, engine, and tank mass was estimated at 54.2 t. This gave a mass of 179.0 t into a 185 km orbit and a mainstage propellant mass of 631.4 t (including 1% overhead). The second stage and fairing (mass of 4.5 t) are separated at second stage burnout. For an O_2/H_2 engine with v_e = 4531 m/s, Δv = 4465 m/s, and a third stage empty mass of 12.6 t, a payload of 53.7 t could be sent on a trajectory to Mars. Figure 11 shows a possible configuration of this vehicle. With a 17 m high fairing, the vehicle height is 68 m.

VII. Development and Testing

Using the cost estimator at [17], it is estimated that the development cost for four O_2/C_7H_8 NV's in year 2000 US$ is $8512M or $2078M per vehicle ($1637M for 24 engines and $6596M for four vehicles). For a vehicle with a payload smaller than 58.3 t, the development costs can be substantially reduced compared to the other vehicles.

The flyback pod would be very useful to develop for the NV since it can be used to test landing procedures and used to transport the vehicle. Two flyback pods are estimated to cost $384M.

Development costs for the HLLV are estimated to cost $2134M for the second stage and $1354M for the third stage for a total of $3488M. With the first stage being fully reusable and the second stage engines being recoverable, this should substantially reduce the cost of a Lunar or Mars program.

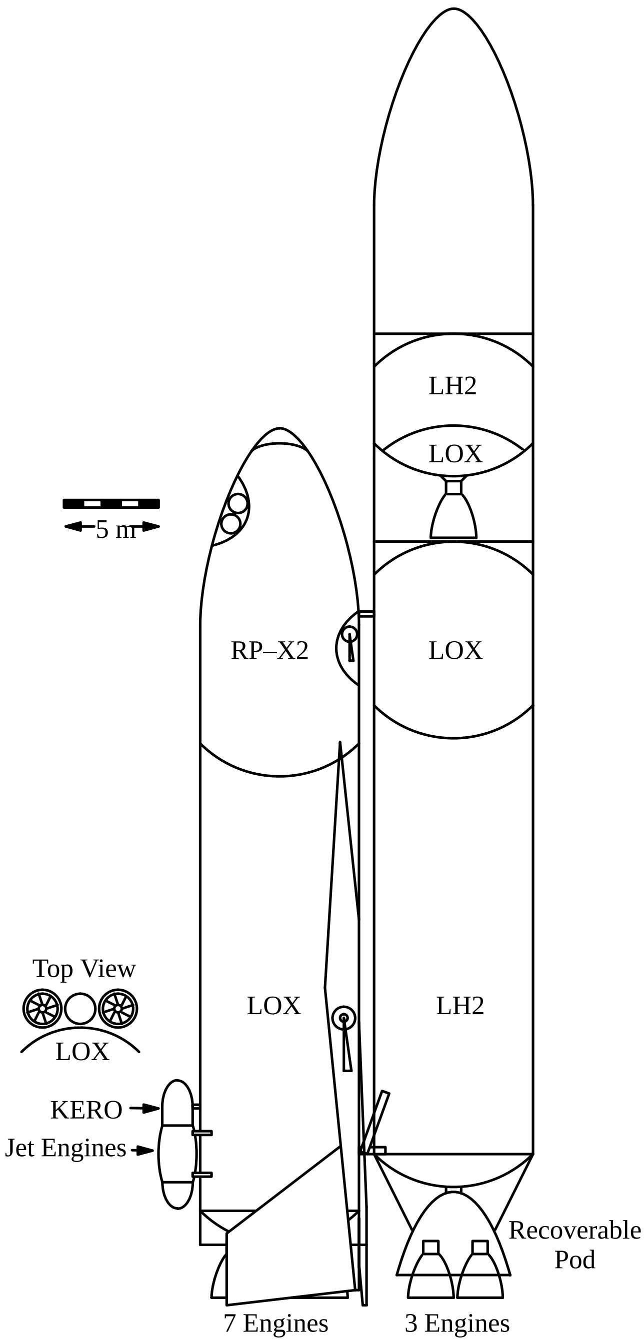

Figure 11: Heavy lift launch vehicle with upper stage.

VIII. Conclusions

A near single stage to orbit can provide increased payload mass compared to using a single stage to orbit. To achieve this, the propellant volume in the upper stage needs to be increased or an upper stage added to the payload. An NSTO also reduces the time spent by the launch vehicle in space, increasing vehicle utilisation.

To achieve high mass efficiency a common propellant bulkhead design with an externally attached payload is proposed. This allows almost no size limit on the payload and more abort options for a crewed vehicle.

Finally we performed an extensive analysis, both theoretical and analytical, on various propellant options for an NSTO vehicle (NV). Simulations show that total Δv is dependent on which propellant is used, with O_2/H_2 requiring up to 300 m/s more Δv than higher density propellants. O_2/H_2 was also shown to have the worst performance compared to higher density propellants. The best performance was achieved by O_2/C_7H_8 which could deliver 58.3 t into an NSTO 80×200 km orbit inclined at 51.6°.

The high impulse density of O_2/C_7H_8 also offers excellent performance when the NV is used as the first stage of a heavy lift launch vehicle (HLLV). In this case, the external payload would be replaced with jet engines and kerosene fuel tanks. The second stage should use O_2/H_2 due to its high mass efficiency. Computer simulations indicate a mass of about 179.0 t could be delivered into a circular orbit inclined at 28.45° or with an O_2/H_2 upper stage, 53.7 t on a trajectory to Mars.

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APPENDIX A

To determine the required Δv's for changing from elliptical to circular orbits we use the following equations [18]. For a circular orbit we have

$$v_o = \sqrt{\frac{\mu}{R + h}} \tag{9}$$

where v_o is the speed for a circular orbit, μ is the gravitational parameter of the planet (μ = 3.986005×10^{14} m^3/s^2 for Earth), R is the radius of the planet (R = 6,378,165 m for Earth), and h is the height above the planet's surface.

For elliptical orbits we have

$$v_a = \sqrt{\frac{2\mu}{r_a(r_a/r_p + 1)}} \tag{10}$$

$$v_p = \sqrt{\frac{2\mu}{r_p(r_p/r_a + 1)}} \tag{11}$$

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.

Using (9) the circular orbit speed at h = 185.2 km is 7793 m/s. From (10), the apogee speed for a 20×185.2 km orbit is 7743 m/s. Thus, the total Δv to change from a 20×185.2 km orbit to a 185.2 km orbit is $v_o - v_a$ = 50 m/s.

APPENDIX B

For VentureStar we assume that it initially goes into a 20×185.2 km orbit, performs a Δv_2 = 50 m/s burn at apogee to circularise the orbit, deploys the payload, and then performs another Δv_3 = 50 m/s burn to re–enter the Earth's atmosphere. We also assume that the total Δv includes a 1% overhead. That is, 95.7 m/s of the total Δv of 9567 m/s is overhead. This implies that 3.02 t of propellant remains in the tanks. This effectively increases the empty mass from 116.57 t to m_v = 119.59 t and decreases the propellant mass from 1049.16 t to m_p = 1046.14 t. The cargo mass is m_c = 22.68 t and exhaust speed v_e = 4462 m/s.

Assuming the O_2/H_2 engines are used, the deorbit burn requires $m_{p,3}$ = 1.35 t of propellant from

$$\Delta v_3 = v_e \ln\left(1 + \frac{m_{p,3}}{m_v}\right) \tag{12}$$

The circularisation burn requires $m_{p,2}$ = 1.62 t of propellant from

$$\Delta v_2 = v_e \ln\left(1 + \frac{m_{p,2}}{m_{p,3} + m_v + m_c}\right) \tag{13}$$

Thus, the Δv required to go into a 20×185.2 km orbit is Δv_1 = 9379 m/s from

$$\Delta v_1 = v_e \ln\left(1 + \frac{m_p - m_{p,2} - m_{p,3}}{m_{p,2} + m_{p,3} + m_v + m_c}\right) \tag{14}$$

We can now determine that the new payload mass is m_c' = 26.06 t from

$$\Delta v_1 = v_e \ln\left(1 + \frac{m_p}{m_v + m_c'}\right). \tag{15}$$

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