IAF–98–V.3.07

A FLEXIBLE REUSABLE SPACE TRANSPORTATION SYSTEM

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49th International Astronautical Congress
Sept 28–Oct 2, 1998/Melbourne, Australia

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Abstract

A 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 and upper stage are 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). The payload and upper stage are piggy–backed on the launch vehicle to allow unlimited payload volume. This also allows the launch vehicle to use a common bulkhead between the fuel and oxidiser tanks, further reducing the launch vehicle mass. A number of propellant combinations are investigated. Computer simulations indicate that liquid oxygen with either kerosene or subcooled propane promise to give the largest payload mass. The payload can be a small winged crewed vehicle for crew transfer and rescue from the International Space Station. 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.

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). 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.

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 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 200 km and a perigee only high enough for the launch vehicle to return to its launch site after its 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 [1].

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 NSTO vehicle (NV) which is launched vertically. This allows complete freedom in the size of the payload, compared to existing SSTO vehicle designs where one is restricted to the volume in the 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 from orbit.

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 (including O_2/H_2, O_2/subcooled methane, O_2/subcooled ethane, O_2/subcooled propane, O_2/kerosene, and 98%H_2O_2/kerosene). Six space shuttle main engine (SSME) size engines with constant propellant volume flow rate are assumed in our simulations.

We first investigate the potential payload gains that can be achieved using an NSTO type orbit. We next investigate the performances 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.

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 shroud 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 = 26.8 t (1 t = 1000 kg) into a 185.2 km, 28.5° orbit [2]. The VentureStar empty mass is m_s = 89.8 t, propellant mass m_p = 875.0 t, and effective vacuum exhaust speed v_e = 4462 m/s [2] (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 $m_f = m_s + m_c$ is the final mass, the total Δv of VentureStar is 9551 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 2.7 t to 29.5 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 0.5 t is required. This reduces the payload mass to 29.0 t (a 2.2 t or 8.2% 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 15.3 t. For an NSTO mission the final mass increases by 1.3 t to 16.6 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	18.8+0	18.4+0
185.2	22.2+0.7	21.5+1.4
20×185.2	24.4+1.2	23.6+2.0

The proposed Ariane Transfer Vehicle (ATV) could be used as an upper stage [3]. This vehicle has a dry mass of 2.7 t and a propellant mass up to 1.8 t. Thus, the payload mass could be increased by 2.9 t from 18.8 t to 21.7 t, a 15.4% increase. The CV could be derived from the X–38/CRV program [4].

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 de–orbit 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 well the launch vehicle dimensions 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. The VentureStar attempts to overcome this problem by splitting the Hydrogen tank into two and using a lifting body shape.

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. This has a number of advantages and disadvantages.

A large advantage is that their is practically no limit to the physical dimensions of the payload. This implies that large payloads with low density high performance propellants can be carried. 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 encapsulated in an expendable shroud. The cost and mass of the shroud should be more than offset by the decreased structure mass and simpler design of the NV. Multiple satellites can also be carried in a similar way to existing launch vehicles. The payload would then be attached shortly before launch using techniques similar to that of boosters to the sides of launch vehicles.

A disadvantage for externally mounted satellite payloads is that the payload may be lost if the NV underperforms and cannot reach orbit. Re–entry heat flux may be too great for the shroud, resulting in destruction of the payload. This may result in higher launch insurance fees. The externally mounted payload also increases drag on the vehicle.

Carrying crewed vehicles that are externally mounted instead of being internally carried offers significant safety and performance advantages. If a launch mishap were to occur, the CV can quickly separate and return to Earth. Also, no shroud 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. To take advantage of the knowledge learnt from the Space Shuttle and X–33, we shall assume vertical take–off and horizontal landing (VTHL).

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. 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.

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.

An important question is the choice of propellant for the launch vehicle. We discuss this in more detail in the next section.

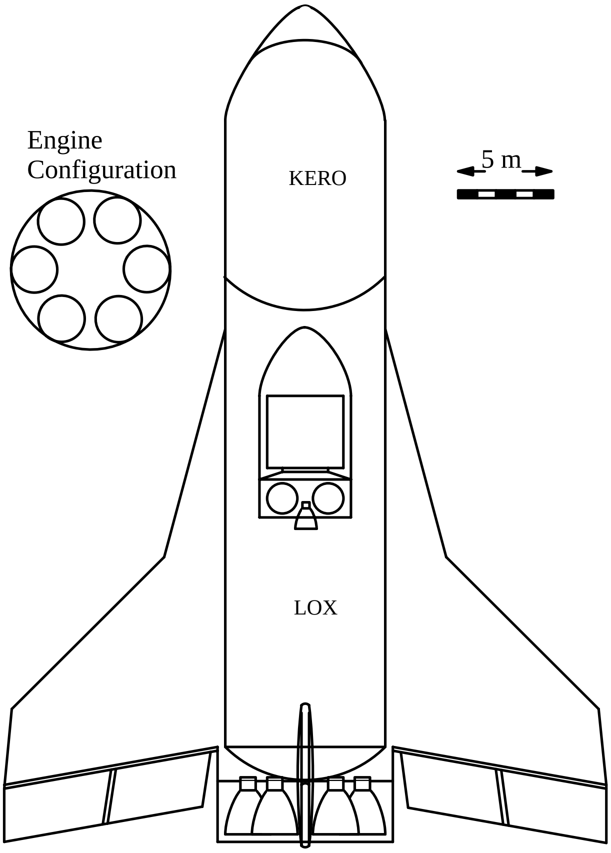

Figure 1: Possible NSTO vehicle configuration

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

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

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

$$\Delta v \approx I_d V_p / m_f \quad (3)$$

where $I_d = v_e d_p$ is the impulse density (Ns/l). 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 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 [5].

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
Nitrogen Tetroxide	N_2O_4	1.431	–19.58
Liquid Hydrogen	H_2	0.0709	–9.01
Methane at 90K	CH_4	0.451	–90.71
Ethane at 90K	C_2H_6	0.652	–111.29
Propane at 90K	C_3H_8	0.728	–136.48
Kerosene (RP–1)	$CH_{1.9532}$	0.8	–24.10
Hydrazine	N_2H_4	1.004	50.42
MMH	N_2CH_6	0.874	54.18
UDMH	$N_2C_2H_8$	0.7861	49.79
Methylacetylene	C_3H_4	0.7	162.34

To increase their density, methane, ethane, and propane were subcooled to the boiling point of oxygen at 90 K. MMH is monomethyl hydrazine and UDMH is unsymmetrical dimethyl hydrazine.

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 program based on [5] 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 N_2O_4 with MMH and UDMH, as impulse density increases, propellant density increases and exhaust speed decreases. With N_2O_4 the best fuel is obviously N_2H_4 since it has the highest propellant density and exhaust speed of the three candidates. 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	0.325	4455	1448
O_2/H_2	6.0	0.362	4444	1609
O_2/H_2	7.5	0.412	4365	1798
O_2/CH_4	3.5	0.855	3652	3122
O_2/C_2H_6	3.1	0.969	3614	3502
O_2/C_3H_8	3.0	1.004	3597	3611
O_2/RP–1	2.8	1.031	3554	3664
N_2O_4/UDMH	2.9	1.182	3350	3960
N_2O_4/MMH	2.4	1.205	3366	4056
N_2O_4/N_2H_4	1.4	1.216	3371	4099
HTP/C_3H_4	6.5	1.255	3319	4165
HTP/RP–1	7.3	1.306	3223	4209

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 2. 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 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/RP–1 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, an exhaust speed of 2637 m/s, and an impulse density of 3428 Ns/l [6].

For the NV we are interested in orbital speeds from 9 to 9.5 km/s. In this case, Figure 2 indicates that the best propellant is O_2/C_3H_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.

Computer Simulations

To determine the performance of various propellants, a computer simulation of an NV into a 80×200 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 = 1300 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 $F_{lo} = 6(R_t F_v - F_d)$ where R_t 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_{lo} = 11.77 m/s^2 (1.2g)

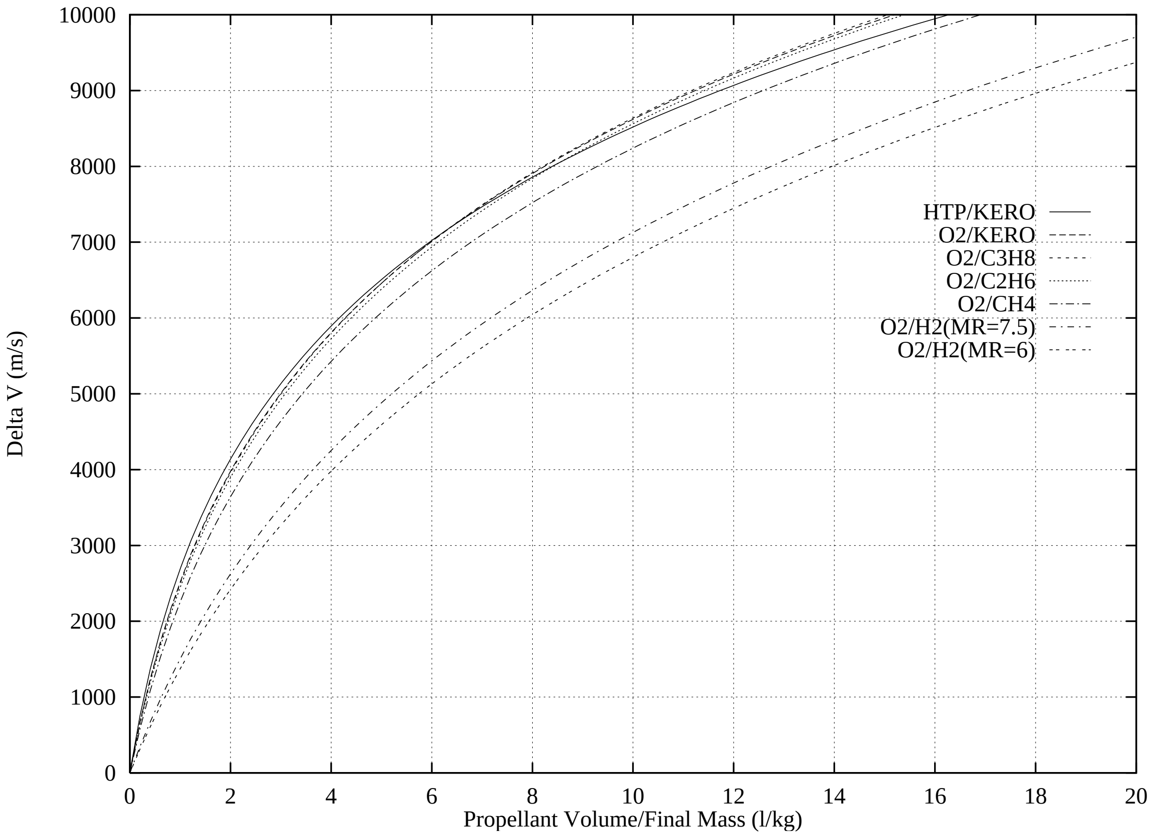

Figure 2: Delta V versus propellant volume to final mass ratio.

the lift–off mass is equal to F_{lo}/a_{lo}. 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 [6]. The pascal source code and a 32–bit DOS executable for our 2–D simulation program are freely available from [7].

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 7891 m/s. The two to three 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 80×200 km orbit, the pitch–over time varied from 2.8 to 4.8 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 4.5° to 5°. 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 82×216 km). As the 10 m diameter, 25 t payload ascended to apogee the orbit is reduced to the desired 80×200 km. At apogee, the upper stage fires its storable propellant engine to put it in a 200 km circular orbit. Total firing time is quite short at less than 6.5 minutes.

Figures 3 and 4 plot speed and altitude versus time for O_2/RP–1 propellant. The uneven plot of Figure 3 after 200 s is caused by the shutdown of four engines one after the other.

Table 4 gives the simulation results for the various propellant combinations given in Table 3. The Δv does

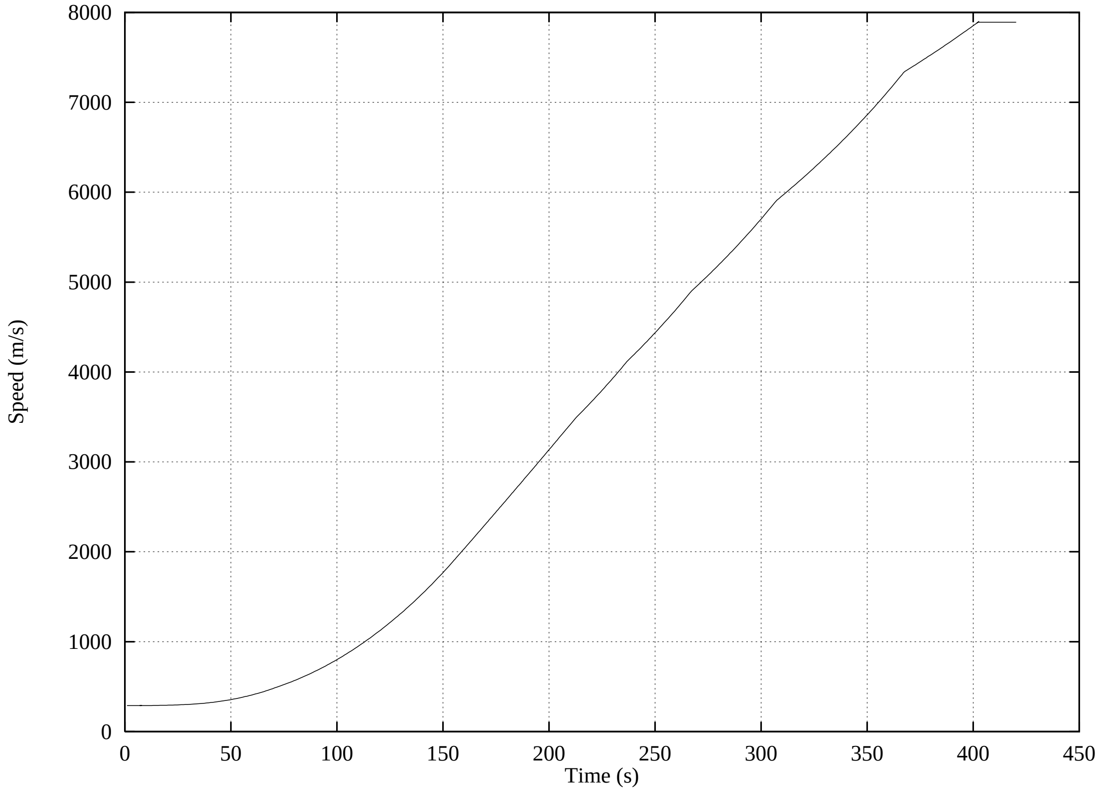

Figure 3: Speed versus time for O_2/RP–1 NSTO vehicle

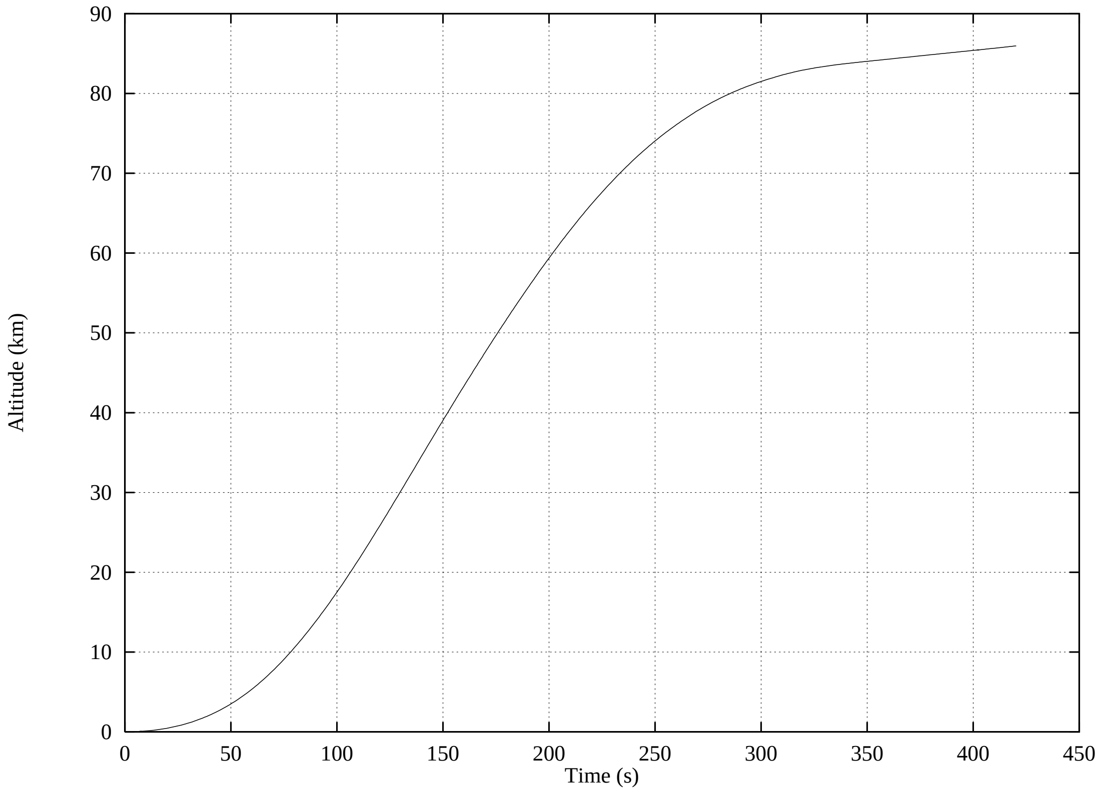

Figure 4: Altitude versus time for O_2/RP–1 NSTO vehicle

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)	9323	893.0	2169	107.3
O_2/H_2 (7.5)	9279	1023.6	2194	119.6
O_2/H_2 (7.5–5)	9277	1023.6	2259	122.9
O_2/CH_4	9114	1935.9	2082	155.7
O_2/C_2H_6	9096	2197.4	2089	172.9
O_2/C_3H_8	9087	2272.8	2088	177.2
O_2/RP–1	9076	2309.1	2071	175.1
N_2O_4/UDMH	9051	2512.8	1987	164.1
N_2O_4/MMH	9051	2579.1	1999	170.6
N_2O_4/N_2H_4	9051	2608.8	2004	173.3
HTP/C_3H_4	9042	2654.4	1980	169.4
HTP/RP–1	9028	2684.7	1934	158.6

Figure 5 shows the vehicle (final) and propellant masses versus propellant. The propellant mass is broken down into fuel and oxidiser mass. Figure 6 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 107.3 t. By using a 7.5:1 MR, the final mass increases by 12 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 [7]). As can be seen, the final mass increases by only an additional 3.3 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_3H_8 which is 54.3 t greater than the best result achieved with O_2/H_2. O_2/RP–1 also provides excellent performance with a 52.2 t increase in final mass. This potentially could lead to a 50 t increase in payload mass! However, since the initial mass is 130% 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 7 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 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 [8]. Ignoring sea level performance losses we have that

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

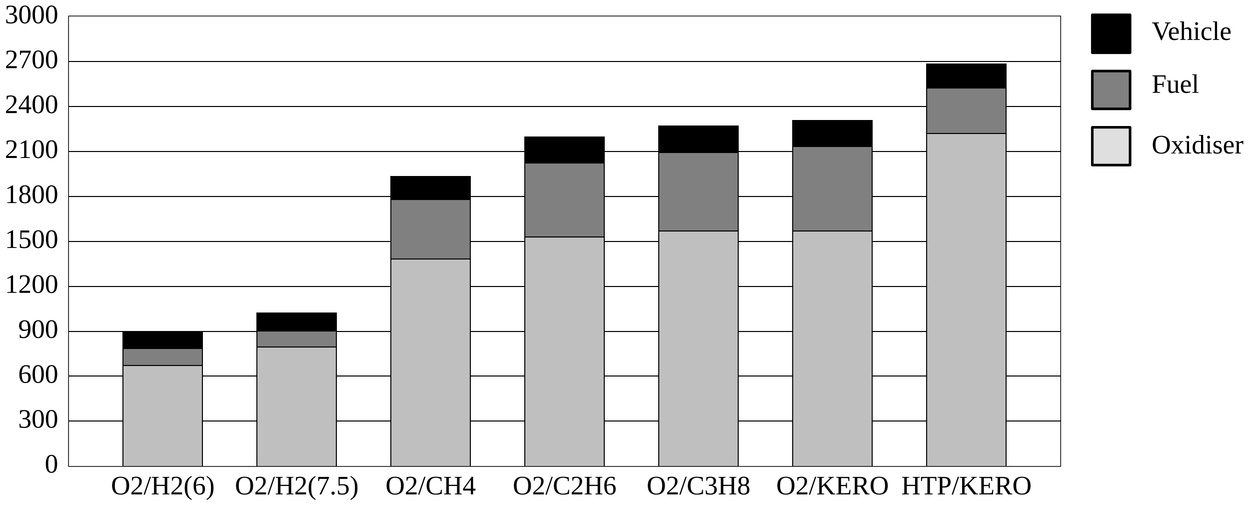

Figure 5: Vehicle and propellant mass (t) versus propellant

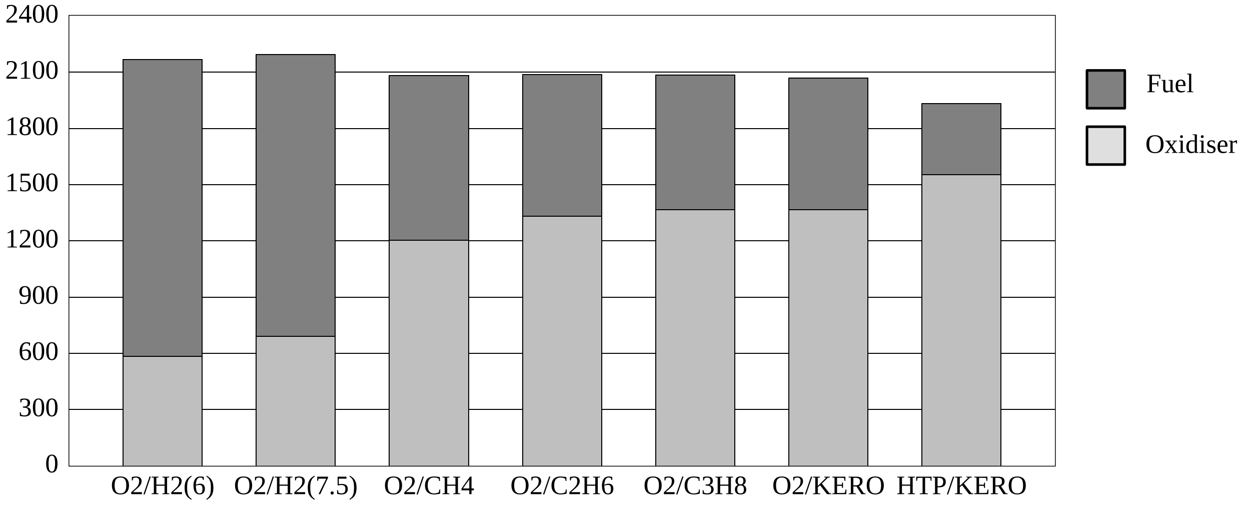

Figure 6: Propellant volume (kl) versus propellant

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

$$\frac{F}{F_v} = 1 - \frac{F_d}{I_d R_v} \qquad (5)$$

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

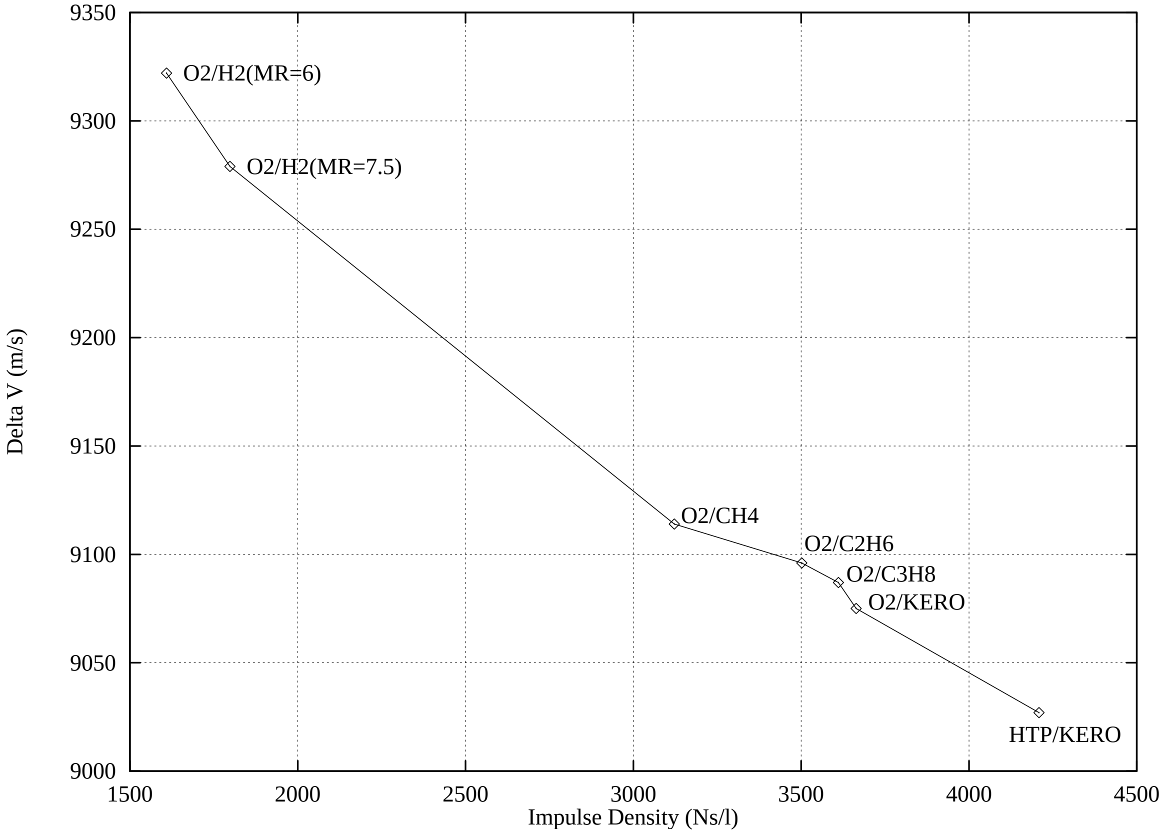

Figure 7: Delta v versus impulse density for NV.

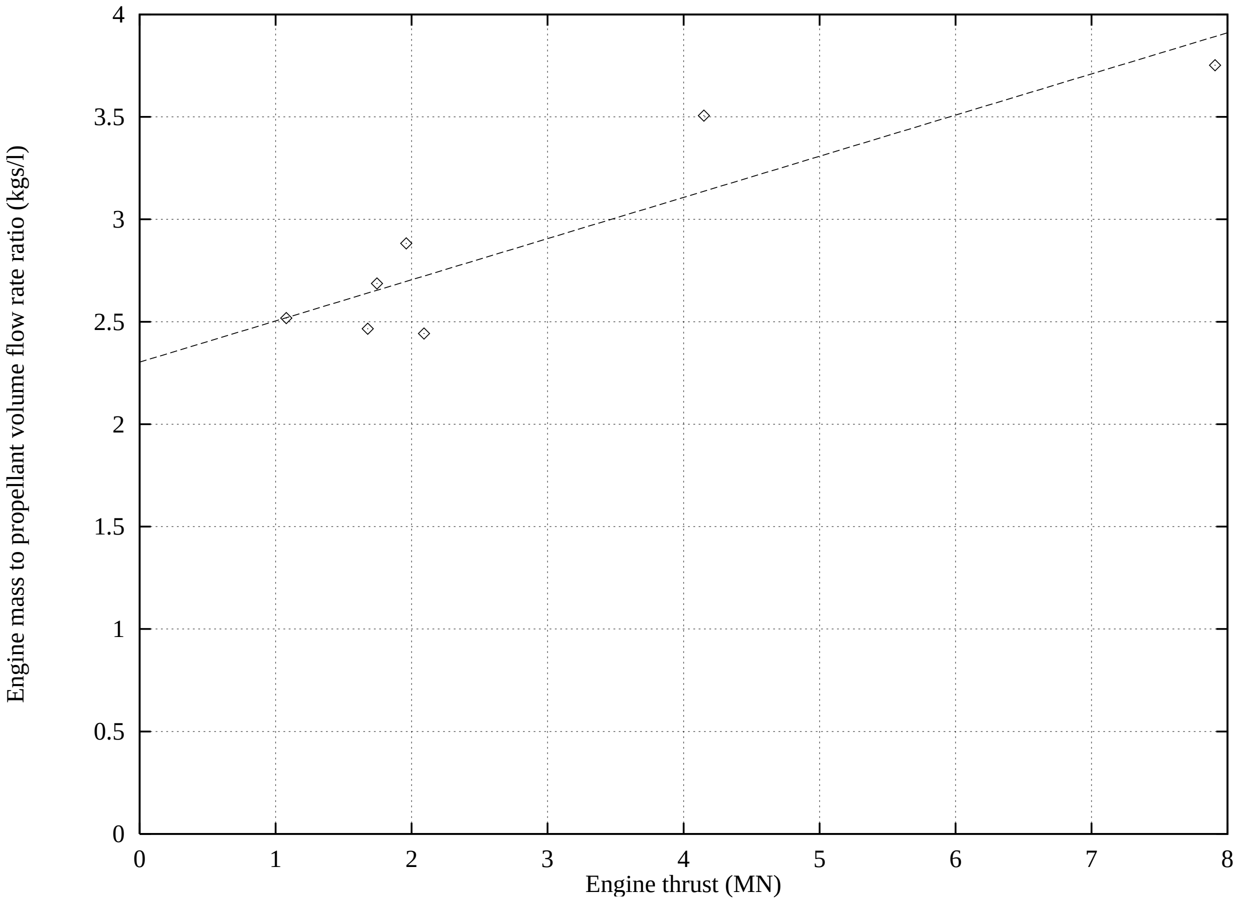

Figure 8: Engine mass to propellant volume flow rate ratio versus vacuum thrust.

requiring computer simulations to determine the required Δv for each propellant.

We now attempt to estimate the tank and engine masses of the NV. From [9] 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. 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.

Table 6 gives the vacuum thrust, impulse density, and engine mass of various staged combustion engines [10]. 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 8.

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 8 and has the formula

$$m_e/R_v = 2.303 + 0.201F_v. \tag{7}$$

Table 5: Tank and engine masses

Propellants	m_f (t)	m_t (t)	m_e (t)	m_{s+c} (t)
O_2/H_2 (6)	107.3	17.1	21.2	69.0
O_2/H_2 (7.5)	119.6	17.5	21.6	80.5
O_2/H_2 (7.5–5)	122.9	17.9	21.6	83.4
O_2/CH_4	155.7	18.8	24.3	112.6
O_2/C_2H_6	172.9	18.8	25.1	129.0
O_2/C_3H_8	177.2	18.8	25.3	133.1
O_2/RP–1	175.1	18.6	25.4	131.1
N_2O_4/UDMH	164.1	18.1	26.0	120.0
N_2O_4/MMH	170.6	18.2	26.2	126.2
N_2O_4/N_2H_4	173.3	18.2	26.3	128.8
HTP/C_3H_4	169.4	17.8	26.4	125.2
HTP/RP–1	158.6	17.3	26.5	114.8

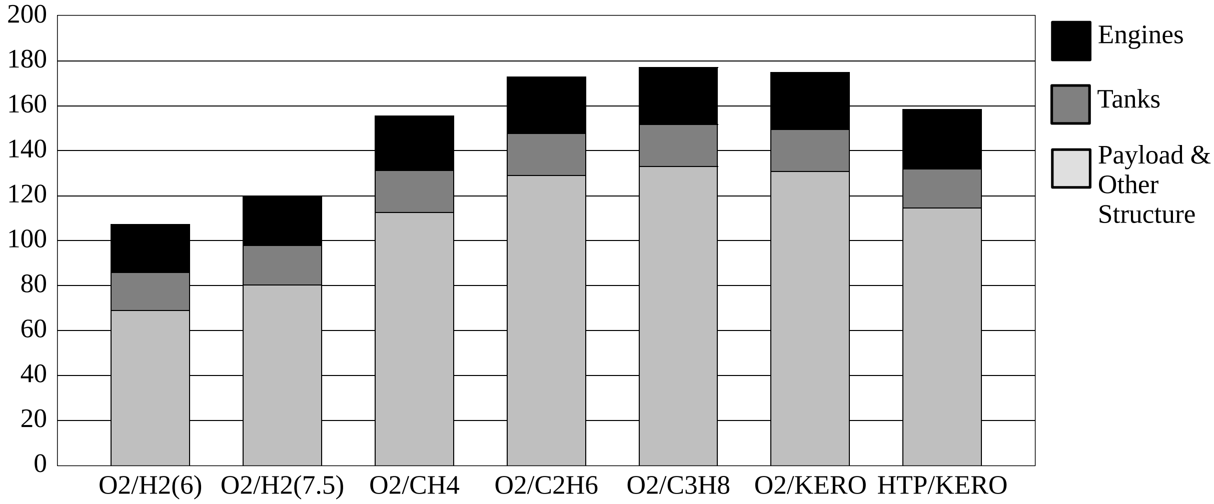

Figure 9: Vehicle mass (t) versus propellant.

Table 6: Staged combustion engine performance

Engine	F_v (kN)	I_d (Ns/l)	m_e (kg)	m_e/R_v
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

If we assume that R_v = 1300 l/s (the same as the SSME) then we can determine the engine mass as a function of impulse density

$$m_e = 2.303R_v + 0.201I_dR_v^2. \quad (8)$$

The m_e in Table 5 gives the mass for six engines using (8). We can now determine the remaining structure and payload mass (m_{s+c}). As can be seen, the high density propellants still outperform O_2/H_2 by a significant margin (up to 49.7 t). The best propellant is still O_2/C_3H_8. Figure 9 shows the performance difference graphically.

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 the vehicle utilisation.

To achieve high mass efficiency a common propellant bulkhead design with an externally attached payload is proposed. This allows practically no size limit on the payload and more abort options for a crewed vehicle. Disadvantages are greater drag on the vehicle, a satellite payload may be lost if the vehicle underperforms, and an expendable shroud is required for satellites.

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/sub–cooled propane with O_2/kerosene closely behind. O_2/kerosene may be a better propellant to choose since kerosene is much easier to handle than cryogenic propane.

The higher impulse density of O_2/RP–1 over O_2/C_3H_8 would imply a slightly better performance when the NV is used as the first stage of a heavy lift launch vehicle (HLLV). Due to its very low impulse density, choosing O_2/H_2 would halve the mass of the second stage, greatly reducing the payload mass. 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.

Future work will involve more accurate calculations of engine performance, trajectory simulations, and vehicle weights. However, the large improvement of high density propellants over O_2/H_2 that we have found should still give the same conclusion for propellant choice.

Simulations will also need to be made of the HLLV to show what payloads can be achieved (expected to be over 100 t if O_2/H_2 is not used for the NV).

Acknowledgment

The author would like to thank Jim Davis, Paul Dietz, Marcus Lindroos, and Henry Spencer for their help in providing information and references for this paper.

Appendix A

To determine the required Δv's for changing from elliptical to circular orbits we use the following equations [11]. 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 ($\mu = 3.986005 \times 10^{14}$ 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 50 m/s burn at apogee to circularise the orbit, deploys the payload, and then performs another 50 m/s burn to re–enter the Earth's atmosphere. We also assume that the total Δv includes a 1% overhead. That is, 95.5 m/s of the total Δv of 9551 m/s is overhead. This implies that 2.5 t of propellant remains in the tanks. This effectively increases the empty mass from 89.8 t to 92.3 t and decreases the propellant mass from 875 t to 872.5 t.

Assuming the O_2/H_2 engines are used, the deorbit burn requires $m_{p,3}$ = 1.05 t of propellant from

$$50 = 4462 \ln\left(1 + \frac{m_{p,3}}{92.3}\right) \tag{12}$$

The circularisation burn requires $m_{p,2}$ = 1.35 t of propellant from

$$50 = 4462 \ln\left(1 + \frac{m_{p,2}}{1.05 + 92.3 + 26.8}\right) \tag{13}$$

Thus, the Δv required to go into a 20×185.2 km orbit is Δv_1 = 9368 m/s from

$$\Delta v_1 = 4462 \ln\left(1 + \frac{872.5 - 2.4}{2.4 + 92.3 + 26.8}\right) \tag{14}$$

We can now determine that the new payload mass is m_c' = 29.5 t from

$$9368 = 4462 \ln\left(1 + \frac{872.5}{92.3 + m_c'}\right). \tag{15}$$

References

1. ESA, "Future European space transportation investigation programme (FESTIP)," progress meeting, June 1998.
2. M. A. Dornhein, "Follow–on plan key to X–33 win," *Aviation Week and Space Technology*, vol. 145, pp. 20–22, 8 July 1996.
3. A. Wilson, "Interavia space directory," 1989–90.
4. C. Covault, "Second X–38 set for flight," *Aviation Week and Space Technology*, vol. 149, pp. 58–62, 31 Aug. 1998.
5. B. McBride, "A computer program for estimating the performance of rocket propellants," NASA Lewis Research Center, Cleveland, Ohio, 1972.
6. S. S. Pietrobon, "High density liquid rocket boosters for the space shuttle," *J. British Interplanetary Soc.*, to appear. www.sworld.com.au/steven/pub/lrb.pdf
7. S. S. Pietrobon, "Near single stage to orbit simulation program," www.sworld.com.au/steven/space/nsto/
8. M. Burnside Clapp, *Space Access 97*, Phoenix, AZ, Apr. 1997.
9. J. A. Martin, "An evaluation of composite propulsion for single–stage–to–orbit vehicles designed for horizontal take–off," *NASA TM X–3554*, Nov. 1977.
10. M. Wade, "Encyclopedia astronautica," solar.rtd.utk.edu/~mwade/spaceflt.htm
11. A. C. Clarke, "Ascent to orbit: A scientific autobiography," 1984.