High Density Liquid Rocket Boosters for the Space Shuttle

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Abstract — **The use of high density hydrogen peroxide/kerosene liquid rocket boosters (LRB) for the Space Shuttle is investigated as a replacement for the existing solid rocket boosters (SRB). It is shown that hydrogen peroxide/kerosene outperforms both solids, LOX/Kero, and LOX/LH_2 as a general booster propellant due to its high density and moderate exhaust speed. With the same propellant mass and size as that of the current SRB's, computer simulations indicate that payload mass can be increased by a third from 24,950 kg to 33,140 kg for a 28.45°, 203.7 km circular orbit. Recovery of the boosters is performed at sea.**

Index Terms — **hydrogen peroxide, space shuttle, liquid rocket boosters.**

I. Introduction

THE USE of liquid rocket boosters (LRB) on the Space Shuttle has a number of advantages compared to solid rocket boosters (SRB). These include increased safety in integrating the booster (since the propellants are only loaded shortly before launch), more abort options if a failure were to occur during launch (since liquid boosters are inherently more controllable), and a cleaner ozone friendly exhaust (for most liquid propellants). There is also the potential of decreased costs in maintaining the boosters and in increased performance.

The choice of propellant for Space Shuttle LRB's has traditionally been liquid oxygen/kerosene (LOX/Kero). In this paper, we investigate the use of high density hydrogen peroxide/kerosene (H_2O_2/Kero) as a propellant for LRB's. The rocket equation can be expressed as follows

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

where Δv is the change of speed, v_e is the effective exhaust speed (divide by 9.80665 m/s^2 to obtain specific impulse in seconds), V_p is the propellant volume, d_p is the propellant density, and m_f is the

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final mass. For small $d_p V_p/m_f$ such as that obtained in the first stage of a rocket we can approximate (1) by

$$\Delta v \approx I_d V_p/m_f. \tag{2}$$

where $I_d = v_e d_p$ is the impulse density. This indicates that for a first stage where the propellant volume to final mass ratio is fixed that the stage's performance is mainly determined by I_d, and not just by v_e alone. We will in fact show that this approximation is valid for a number of propellant combinations and V_p/m_f ratios. That is, a propellant's density is just as important as its exhaust speed in the first stage of a rocket. For the second and higher stages, the exhaust speed is much more important since it significantly affects m_f.

We first analyse various propellant combinations which indicate that H_2O_2/Kero gives very good first stage performance. This is then followed by our design of the LRB, simulation results, a discussion of the recovery method, and properties of H_2O_2.

II. Propellant Performance

Table 1 gives the mixture ratios (MR), vacuum exhaust speed, density (in kilograms per litre), and I_d of selected propellants. The LOX/LH_2 and LOX/Kero values are for the Space Shuttle Main Engine (SSME) and RD–170, respectively [1,2]. Both of these engines represent the state of the art in staged combustion sea level liquid propellant engines. The ammonium perchlorate/aluminium/polybutadiene acrylic acid acrylonitrile terpolymer (AP/Al/PBAN) is the propellant used in the SRB. The bulk density of AP/Al/PBAN is about 1.75 kg/l but the large empty core of the propellant grain reduces the average density to about 1.3 kg/l. The H_2O_2/Kero values were estimated with [3] using the same engine parameters (combustion pressure and area ratios) as the RD–170. The H_2O_2 was assumed to be 98% pure with a 2% water impurity.

Table 1: Propellant Performance.

Propellant	MR (O:F)	v_e (m/s)	d_p (kg/l)	I_d (Ns/l)
98%H_2O_2/Kero	7.30	3017	1.306	3940
AP/Al/PBAN	4.37	2637	$\sim$1.3	3428
LOX/Kero	2.60	3305	1.025	3388
LOX/LH_2	6.00	4444	0.361	1604

As can be seen H_2O_2/Kero has about the same density as solid propellant, but with much higher v_e. It also has the best performance in terms of I_d. LOX/LH_2 has the worst performance and would be a very poor propellant choice for the first stage of a launch vehicle. To illustrate this further, we plot Δv versus V_p/m_f for the three liquid propellants in Table 1 in Figure 1. It can be clearly seen that up to Δv of about 7500 m/s, H_2O_2/Kero provides better performance than LOX/Kero and significantly better performance than LOX/LH_2. Note that for single stage to orbit (SSTO) vehicles requiring $\Delta v \approx 9000$ m/s the choice of propellant will affect the final mass due to a greater launch mass. Thus the apparent advantage that H_2O_2/Kero or LOX/Kero has over LOX/LH_2 is not so clear.

Another important factor is the affect of thrust on a stages performance. Solid boosters have in general very high thrust which greatly reduces gravity losses during the boost phase. Together with their high I_d this is why they are so efficient as a booster stage. A propellant's I_d can also have an affect on its thrust. The propellant flow rate through an engine in litres per second (l/s) is equal to F/I_d where F is the vacuum thrust. If the same flow rate is maintained then an engines thrust is proportional to I_d. The increased thrust helps to compensate for increased propellant mass for high density, high I_d propellants such as H_2O_2/Kero.

III. Liquid Rocket Booster Design

We shall use the LOX/Kero booster rocket of the Energia launch vehicle as the basis of our design. These boosters were designed to be recovered from land. Recovery from sea will involve waterproofing the stage as well as protecting the engine from water contamination. A relatively simple technique for protecting the engine is described in Section V. Our first assumption is that the LRB has the same propellant mass as the SRB of 501.8 t (1 t = 1000 kg). Since H_2O_2/Kero is very similar in density to solid propellant then the size of the LRB will be almost identical to the SRB.

Our next assumption is that the RD–170 can be modified to use H_2O_2/Kero. We shall call this engine the RD–17X. In practice, the use of H_2O_2/Kero will require many design changes and in essence will result in a new engine design. We make the assumption that the RD–17X has the same chamber pressure (25 MPa) and expansion ratio (37) as the RD–170. We used [3] to calculate the vacuum v_e for both the RD–170 and RD–17X with 98%H_2O_2 (varying the mixture ratio to maximise v_e). Since v_e is known for the RD–170 the thrust efficiency could be determined and applied to the RD–17X v_e. Table 2 gives the performance of the two engines. The SSME parameters are also listed.

The Energia boosters have an empty mass of 35 t and a propellant mass of 320 t. With the larger propellant volume and the greater thrust of the LRB, the LRB empty mass was estimated at 53.5 t. We assume that in each LRB 10.0 t of propellant is used in the thrust buildup and 7.7 t of residual propellant remains after LRB cutoff. Table 3 gives the masses of the various elements of the Space Shuttle at lift–off. A lightweight external tank (ET) is assumed. The orbital maneuvering system (OMS) propellant is that required to reach and deorbit from a 203.7 km circular orbit (the maximum

OMS propellant load is 10.8 t). Figure 2 gives a cross section of the SRB and LRB designs for comparison. A diameter of 3.71 m (the same as the SRB) is assumed for the LRB.

Table 2: Engine Performance.

Engine	RD–170	RD–17X	SSME
$v_{e,vac}$ (m/s)	3305	3016	4444
F_{vac} (kN)	7911	9192	2091
F_{sl} (kN)	7259	8545	1668
d_p (kg/l)	1.025	1.306	0.361
MR (mass)	2.60	7.30	6.00
MR (volume)	1.81	4.08	1/2.69

Table 3: Space Shuttle masses at lift–off (all masses in tonnes).

Booster	SRB	LRB
Booster Empty	2×87.7	2×53.5
Booster Prop.	2×501.8	2×491.8
ET Empty	29.9	29.9
ET Prop.	726.9	726.9
Orbiter	89.3	89.3
OMS Prop.	4.7	4.7
Payload	24.9	33.1
Total	2054.7	1974.5

The next design assumption is that the LRB's should have sufficient thrust to lift the Space Shuttle safely away from the pad in case all three SSME's were to fail soon after launch (this is the same level of protection as provided by the current SRB's). With each LRB having 8545 kN of sea level thrust, the Space Shuttle could only survive one SSME engine failure soon after launch. To ensure three engine out survivability, the LRB thrust will need to be increased to 112% of rated thrust (10,295 kN vacuum and 9648 kN sea level).

Since the Space Shuttle will initially be accelerating slower than with SRB's, there is no need for the SSME's to throttle down from 100% to 65% and then up to 104% during the period of maximum dynamic pressure (the "thrust bucket"). Instead, the initial thrust setting for the SSME's can be 104%, only decreasing when the 29.4 m/s^2 (3g) acceleration limit is reached near the end of the main propulsion system (MPS) burn. With 112% RD–17X and 104% SSME, the liftoff thrust is 24,551 kN giving a liftoff acceleration of 12.4 m/s^2 (1.27g). This is slower than currently experienced, but faster than the Saturn V.

IV. Simulation Results

To estimate the performance of the Space Shuttle with LRB's trajectory simulation programs were written for the Space Shuttle with SRB's and LRB's. Each program uses a set of Pascal procedures that can accurately simulate a rocket in flight in two dimensions (range and height). These procedures were originally written for a Saturn V trajectory simulation program [4] but can be applied to any rocket on any planet. The program uses the Runga–Kutta fourth order method and a standard atmosphere model. The program is able to model thrust which changes proportionally with time. This is very useful in accurately simulating the thrust curve of solid motors, as well as thrust buildup and dropoff of liquid propellant engines. A table of nominal vacuum thrust versus time in one second increments at 15.6 C of the SRB's [5] was used in the simulation. A variety of sources were used to determine the masses and other parameters of the Space Shuttle [1,2,6,7].

Only two parameters are required to shape the trajectory into the required orbit. This is the pitch over time soon after launch and the maximum angle of attack after SRB or LRB separation. After pitch over the shuttle follows a gravity turn such that the air angle of attack is zero. After SRB or LRB separation the angle of attack is automatically increased to its maximum value and then automatically decreased. This is achieved via an algorithm that forces h_2 to be proportional to $-\operatorname{sign}(h_1)|h_1|^2$ where h_0 is height above the planet's surface, $h_1 = dh_0/dt$, $h_2 = dh_1/dt$, and sign(x) is the sign of x. Thus, if h_1 is positive (meaning that h_0 is increasing) then h_2 is made to decrease, slowing the rate of altitude increase. If h_1 is negative (the vehicle is now heading back towards the planet), then we make h_2 positive so as to push the vehicle back up. Although this is a crude algorithm, we have found it to be very effective and provides good performance (coming to within a few percent of payload mass of trajectories that use optimal algorithms).

After SRB or LRB separation there is not enough thrust to maintain a positive rate of altitude increase and so the angle of attack quickly increases to its maximum value. Once centrifugal forces build up to a sufficient degree the angle of attack gradually decreases. The trajectory then attains a maximum altitude and then gradually decreases. When the 3g maximum acceleration limit is reached the angle of attack (positive in normal trajectories) is fixed at its current value. This ensures that h_1 is positive at main engine cutoff (MECO) implying that the orbiter will ascend to apogee. If h_1 is negative at MECO, the orbiter will descend to perigee and re–enter the Earth's atmosphere.

Figures 3 and 4 plot speed and altitude versus time, respectively, for the Space Shuttle SRB's and LRB's. A DOS executable and pascal source code for these programs is available from [8,9]. As can be seen from Figure 3, the LRB's take off much slower than the SRB's. However, the higher v_e of H_2O_2/Kero eventually allows the LRB's to eventually pass and then exceed the SRB's. This reduces the required Δv for the MPS burn allowing for a payload increase. The reduced MPS burn time is due to the SSME's operating at 104% from liftoff, instead of 100% and then 65% during the thrust bucket. The elimination of the thrust bucket increases safety since two potential failure modes (throttle down and throttle up) are no longer necessary.

At T+127.01 s, the 29.4 m/s^2 maximum acceleration value is reached. The RD–17X's thrust is then decremented in 1% steps from 112% to 92% with LRB cutoff occurring at T+143.07 s. Maximum dynamic pressure (maxQ) is 27.1 kPa at T+74 s compared to 31.2 kPa at T+41.5 s for the SRB's (a 13% decrease). In [1] maxQ is stated as being less than 31.4 kPa which closely agrees with our SRB simulation results.

At MECO the orbiter is in an elliptical orbit ascending to apogee. The desired orbit is 203.7 km, 28.45° circular where the Space Shuttle is specified as being able to deliver 24,950 kg of payload. We found that to get to a 203.7 km orbit MECO occurred at between 80 to 90 km altitude. At this altitude there is still significant drag and so the apogee had to be somewhat higher than 203.7 km. For the SRB's MECO occured at 84.5 km with a 76.2×216.4 km orbit. After ascending to apogee the orbit was reduced to 74.8×203.7 km due to drag on the orbiter. The OMS then fired for 85.6 s to put the orbiter into circular orbit. The LRB's initially went into an 87.4×219.4 km orbit at 87.7 km, ending up in an 86.8×203.7 km orbit after ascent. The OMS then fired for 82.7 s to put the orbiter into a circular orbit.

For the LRB's the payload performance was determined iteratively. The initial payload was set at 24,950 kg and the excess ET propellant determined (varying the pitch over time and maximum angle of attack until the desired orbit is found). This excess propellant (whether positive or negative) was added to the payload mass and the orbit finding process repeated. After three or four iterations a payload mass of 33,140 kg was determined. That is, a 33% payload increase of 8,190 kg was achieved. Note that in practice this payload mass would not be available for a 203.7 km orbit due to limits on the orbiter landing mass in abort and nominal flights (the maximum payload mass due to landing weight limits is 22,910 kg for all orbiters except Columbia which has a limit of 19,100 kg [6]). However, for higher orbits and higher inclinations (such as the 354 km, 51.6° inclination orbit of the International Space Station), where the payload mass is normally under 22,910 kg, the extra performance of the LRB's can be used to increase payload mass.

V. LRB Recovery

To minimise the empty mass of the LRB's recovery at sea is assumed. Since each of the two LRB's are identical this also allows reduced development costs compared to a single large booster. Since the LRB is almost identical in size to the SRB, only minimal changes will be required to the launch platform. It may also be possible to use both SRB's and LRB's on the same launch platform. A similar recovery system as that used on the Energia boosters can be used. This system can be seen in the large pods forward and aft of the Energia boosters. It is assumed that a system of parachutes and either airbags or retro rockets are used to successfully land the boosters in a horizontal orientation. The softer water landing compensates for the increased mass of the LRB compared to the Energia boosters which were designed for recovery from land.

After separation from the ET, the LRB's will need to vent any residual H_2O_2 overboard to reduce the landing mass and to increase safety in the recovery area. The large empty volume of the propellant tanks will easily keep the LRB's afloat. As for the SRB, waterproofing of areas containing mechanical and electrical items will need to be made. The engine should also be protected from seawater contamination. This could be achieved by a set of half–rings with high strength water proof cloth between the half–rings. The half–rings and cloth are stored on one side of the bottom edge of the aft skirt. Similar to a foldable roof for a sports car, the half–rings are automatically deployed to cover the engine before water impact. Figure 5 illustrates the deployment process.

For increased safety the RD–17X could be replaced by two H_2O_2/Kero derivatives of the LOX/Kero RD–180 (as used on the Atlas IIAR). This would allow single engine out survivability. Other engines that could be used are H_2O_2/Kero derivatives of the LOX/LH_2 SSME and Russian RD–0120 (two would be required for each LRB). As described in the next section, the large number of modifications required to use H_2O_2 will in essence result in a new engine design.

VI. Properties of H_2O_2

Hydrogen peroxide is a dense, colourless, water–like liquid with a pungent, acid odour. It is miscible with water in any proportion. It is commercially available in large quantities up to concentrations of 70% by weight where it is used in chemical synthesis, paper pulp bleaching, metallurgy, textile bleaching, water and effluent treatment, and other applications [11]. Higher concentrations can also be produced. Table 4 lists basic properties of pure H_2O_2 at 20 C and one atmosphere pressure [12]. H_2O_2 decomposes in the presence of impurities or catalysts with the formula

$$H_2O_2 \rightarrow H_2O + 0.5O_2 + 54.22 \text{ kJ}. \tag{3}$$

Commercially produced H_2O_2 is very pure with normally very low decomposition rates [11] (less than 1% per year). However, fast homogeneous decomposition will occur (often with extremely low levels of contaminants such as a few parts per million) if contact occurs with salts of metals such as iron, copper, chromium, vanadium, tungsten, molybdenum, silver and metals from the platinum group [11]. Fast heterogeneous decomposition can also occur if H_2O_2 contacts insoluble solids such as ruthenium, manganese, iron, cobalt, nickel, lead and mercuric oxides, platinum, osmium, iridium, palladium, rhodium, silver, and gold [11].

Table 4: Properties of H_2O_2.

Molecular Weight	34.016
Freezing Point	–0.4 C
Boiling Point	150.2 C
Density	1.4425 kg/l
Heat of Formation	–187.8 kJ/mol
Heat Capacity	2.63 kJ/kgK
Thermal Conduct.	569.4 W/Km
Viscosity	1.249 mPa s

The choice of material for storing and handling H_2O_2 must be made with care. Materials such as aluminium (at least 99.5% pure such as 1050 and 1060), Al–Mg alloys (5254 and 5652), fully austenitic stainless steel, polyethylene, glass, and teflon can be used [11]. Most metallic materials will need cleaning and degreasing with detergent, pickling to remove metal impurities, passivating and conditioning [11].

The combustion of kerosene with H_2O_2 is given by the formula

$$CH_2 + 3H_2O_2 \rightarrow CO_2 + 4H_2O \tag{4}$$

where CH_2 is the approximate formula of kerosene. This compares with the combustion of kerosene with LOX

$$CH_2 + 1.5O_2 \rightarrow CO_2 + H_2O. \tag{5}$$

We can see that the exhaust of H_2O_2/Kero is predominantly water. This results in a very clean exhaust (second only to LO_2/LH_2) and a distinctive clear flame. The low molecular mass of water also helps to increase the performance of H_2O_2/Kero.

Designing a H_2O_2/Kero engine needs to take into account the properties of H_2O_2 in its design. Previous H_2O_2/Kero engines such as that successfully used in the Black Knight sounding rocket and Black Arrow launch vehicle used silver gauze to first decompose the 85% H_2O_2 in the combustion chamber [13]. For higher concentrations of H_2O_2 another catalyst is required, such as platinum. Modern H_2O_2 contains inhibitors which can poison a catalyst. These inhibitors may need to be removed for rocket–grade H_2O_2. No ignition source is required since the very hot decomposed H_2O_2 will spontaneously combust with kerosene. Due to the high mass ratio of H_2O_2 to kerosene and the superior heat characteristics of H_2O_2 compared to kerosene, the H_2O_2 should be used to cool the engine nozzle before combustion. In a staged combustion engine, the pre–combustion chamber needs only to decompose H_2O_2 to provide the energy for the turbines, simplifying the engine design. Since H_2O_2 is non–cryogenic this helps to increase reliability (of the 22 Black Knight and 4 Black Arrow launchers, involving 128 Gamma engines, there were zero engine or propulsion unit failures). The NF–104D research aircraft also successfully used the AR–2–3 90% H_2O_2/kerosene engine to set records that still stand to this day [14].

VII. Conclusions

The high density and moderate exhaust speed of hydrogen peroxide gives it very good performance as a first stage propellant. Since H_2O_2/Kero has the same density as solid fuel, but with a superior exhaust speed, this implies that a liquid rocket booster for the Space Shuttle could be used that is identical in size to the existing solid rocket boosters. The increased performance of H_2O_2/Kero is able to overcome increased gravity losses due to a lower initial acceleration to provide a nearly one third increase in payload mass (from 24,950 kg to 33,140 kg). The RD–170 liquid oxygen/kerosene engine could be used as the starting point for designing a new engine that uses H_2O_2/Kero.

Acknowledgment

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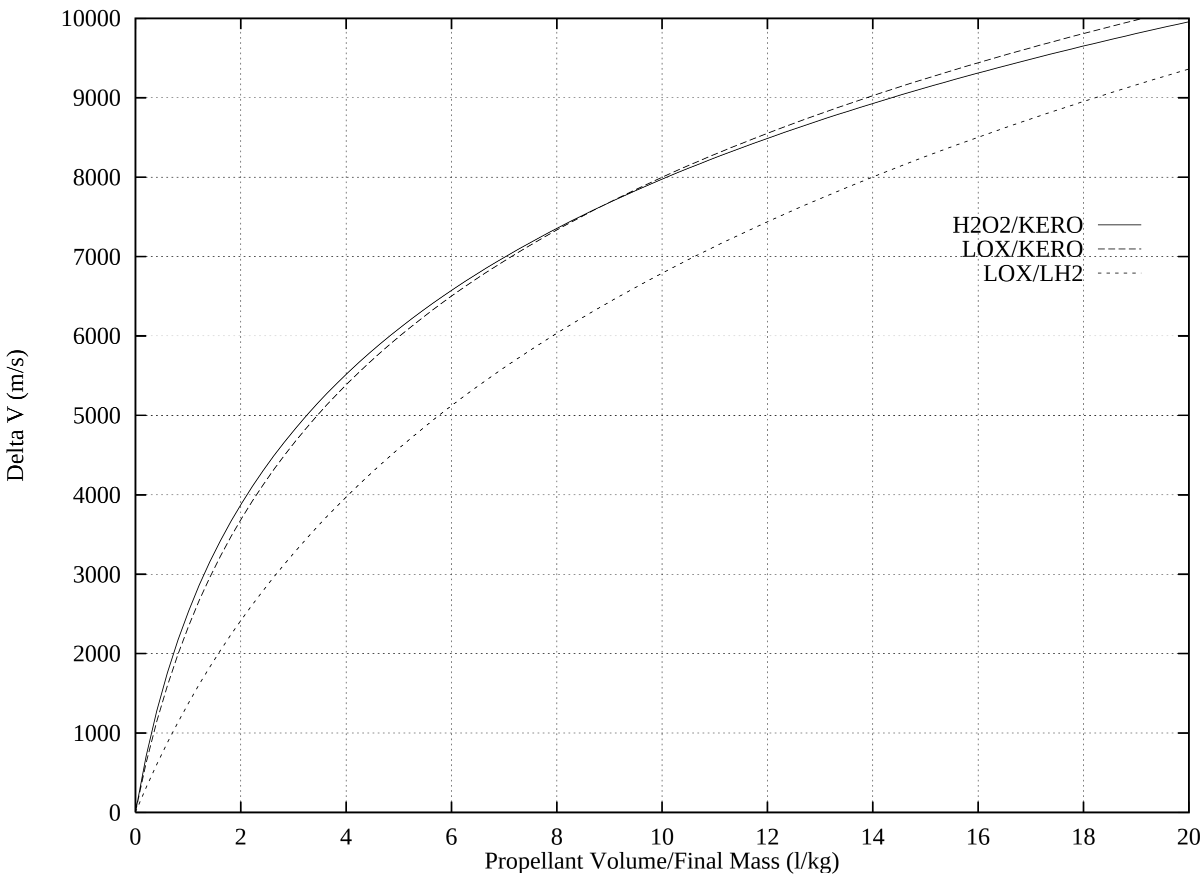

Figure 1: Rocket propellant performance.

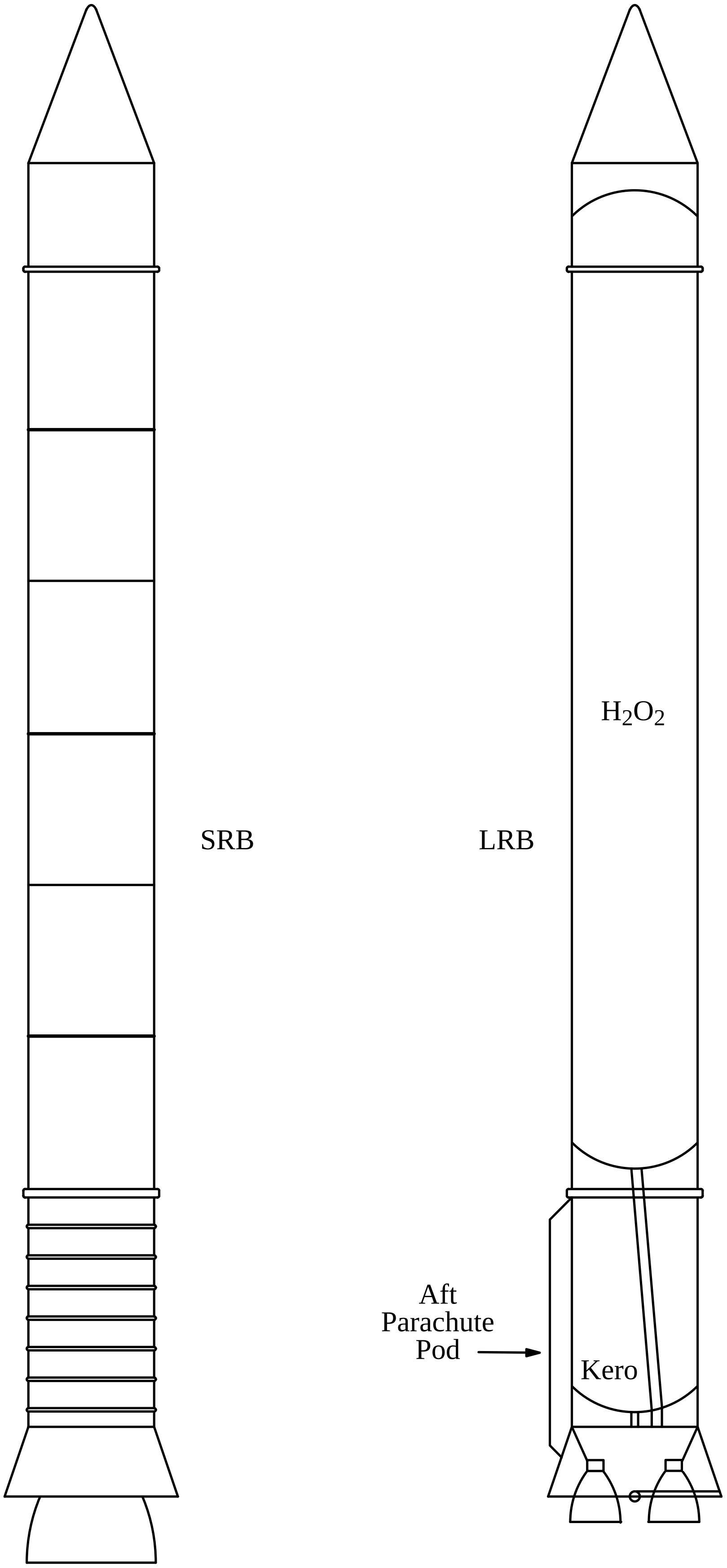

Figure 2: Comparison of SRB and LRB design.

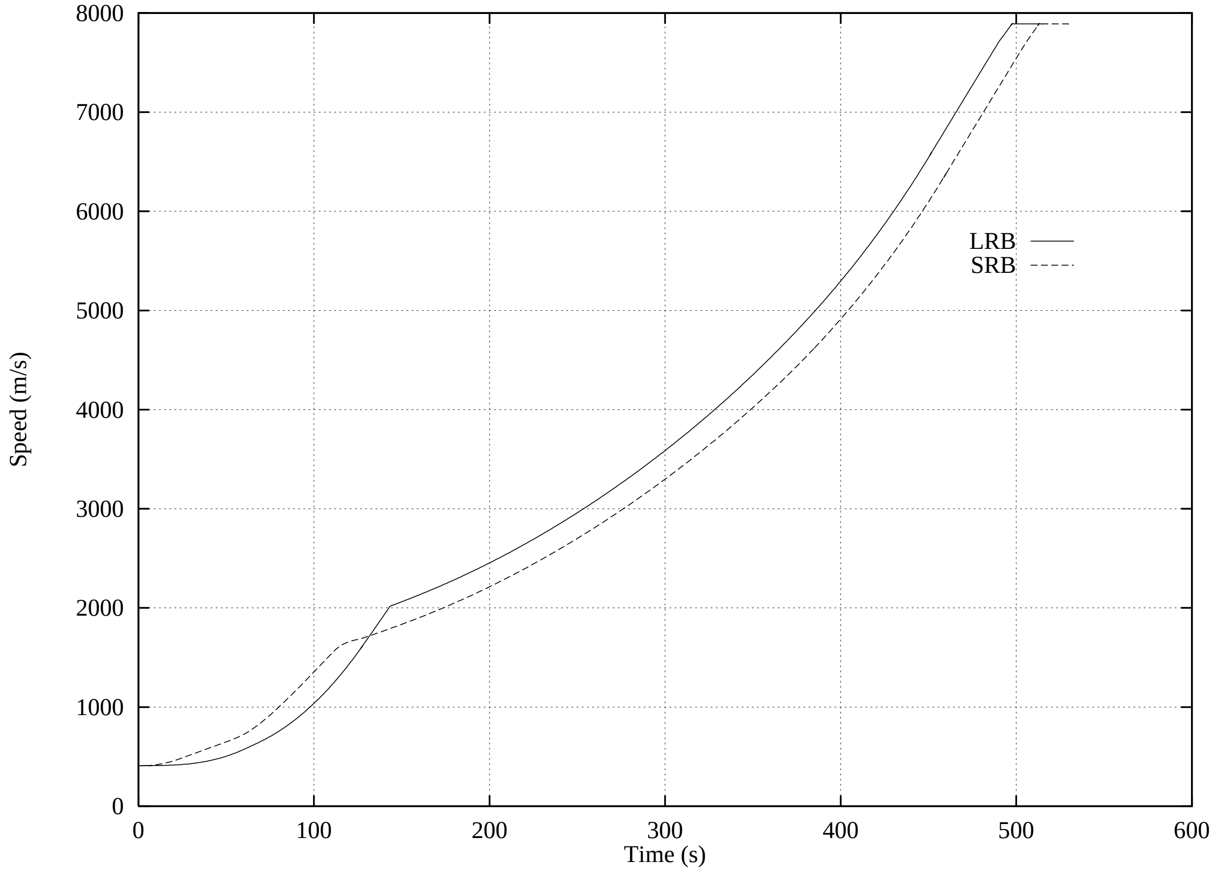

Figure 3: Space Shuttle trajectory (speed versus time).

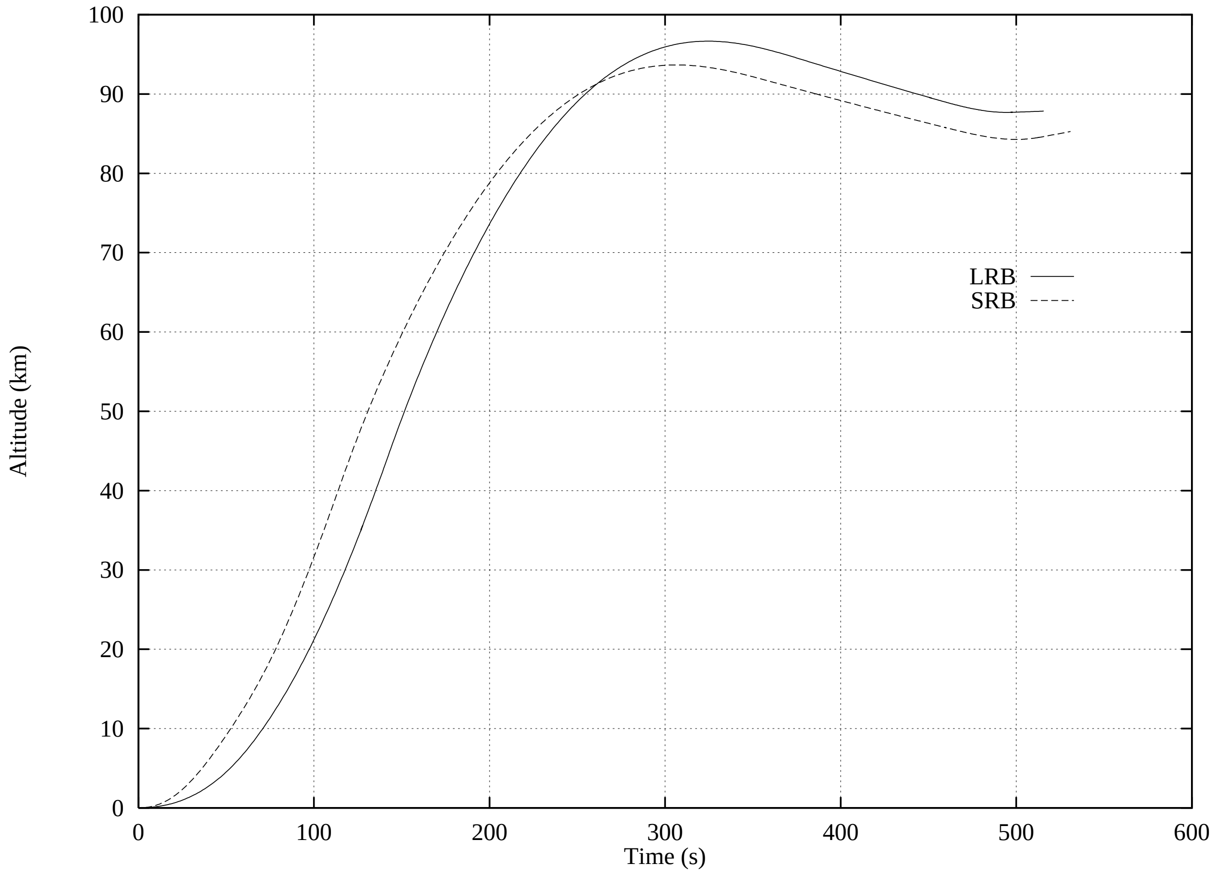

Figure 4: Space Shuttle trajectory (altitude versus time).

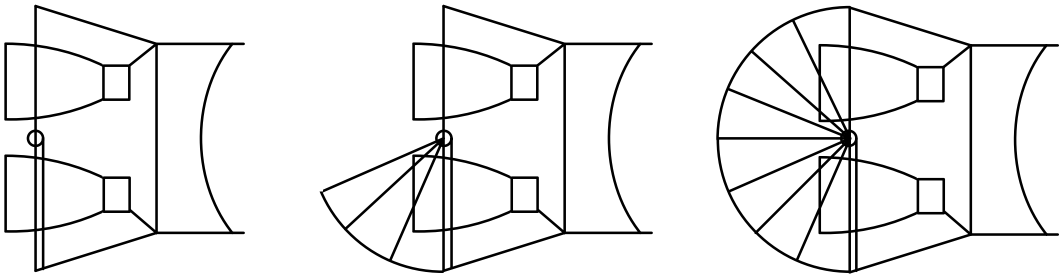

Figure 5: Deployment of LRB engine cover.