Fly Me to the Moon on an SLS Block II

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The first Lunar mission will be the beginning. Later missions will stay for longer periods on the Moon and continue its exploration. But getting to the Moon is like getting to first base. From there we'll go on to open up the solar system and start in the direction of exploring the planets. This is the long range goal. Its a learning process. As more knowledge is gained, more confidence is gained. More versatile hardware can be built. Simpler ways of doing things will be found. The flight crews will do more and more. "Fly Me to the Moon — And Back," National Aeronautics and Space Administration, Mission Planning and Analysis Division, 1966.

***Abstract* —** We examine how a 140 t to low Earth orbit (LEO) Block II configuration of the Space Launch System (SLS) can be used to perform a crewed Lunar landing in a single launch. We show that existing RSRMV solid rocket motors can be used to achieve Block II performance by using a core with six RS–25E engines and a large upper stage (LUS) with two J–2X engines. A cryogenic propulsion stage (CPS) with four RL–10C–2 engines is used to perform trans Lunar injection (TLI), Lunar orbit insertion (LOI) and 75% of powered descent to the Lunar surface. A Lunar module (LM) initially carrying two crew and 509 kg of cargo is used to perform the remaining 25% of Lunar descent. The LM is in two parts consisting of a crew and propulsion module (CPM) and non–propulsive landing and cargo module (LCM). The CPM returns the crew and 100 kg of samples to the waiting Orion in Lunar orbit for return to Earth.

I. Introduction

IT has been 48 years since humans first set foot upon the Moon on 20 July 1969 and 44.5 years since humans last left their footprints there. During that short 3.5 year period, six landings were performed by the Apollo program of the United States. Apollo demonstrated that crewed Lunar missions were possible, achieving the political goal of landing a man on the Moon and returning him safely to Earth by the end of the decade. In addition, a large amount of information was learnt about the Moon, but there is much more to be learnt. The poles, the far side and many other areas of the Moon remain largely unexplored.

Recently, the United States decided to develop the Space Launch System or SLS, initially in a 70 t to LEO configuration (Block I) and later in a 130 t to LEO configuration (Block II) [1]. Block I uses two five segment RSRMV solid rocket motor (SRM) boosters derived from the four segment

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RSRM boosters used on the Space Shuttle. A new 8.4 m diameter core using four liquid hydrogen/liquid oxygen (LH2/LOX) RS–25D engines (again from the Space Shuttle) and an upper stage from the Delta–IV Heavy with one LH2/LOX RL–10B–2 engine is used to complete the Block I configuration [2].

Current planning for Block II assumes that advanced boosters (AB) are needed to obtain the required performance [3]. One option is to use a new SRM with composite casings and hydroxyl terminated polybutadiene (HTPB) propellant and new five engine core [4]. The other option is to use new liquid boosters with LOX and rocket propellant kerosene (RP–1) engines [5, 6]. All these configurations require the use of a new LUS with two already developed LH2/LOX J–2X engines for 130 t to LEO. A possibly cheaper alternative is to use the existing RSRMV boosters with a new core that has six RS–25E engines. This only requires two major developments (the core and LUS) compared to three major developments (SRM, core and LUS or booster, engine and LUS) if using advanced boosters.

To send the crew to the Moon in their Orion multipurpose crew vehicle (MPCV) and LM a CPS with four LH2/LOX RL–10C–2 engines is used. The design of this stage is similar to the exploration upper stage (EUS) proposed in [7], but using a common bulkhead in order to meet vehicle height restrictions. We examined the case where the LUS performs partial TLI as in [8], but we found best performance is achieved when the CPS performs all of TLI due to the higher performance of the RL–10 engines and lower dry mass of the CPS.

To simplify mission design we assume the LUS places the CPS and spacecraft into a 37x200 km trajectory at apogee. This results in the LUS being safely targeted for reentry without requiring a deorbit burn. The CPS performs a small burn at apogee to circularise the orbit. While in LEO Orion separates from its spacecraft launch adaptor (SLA). At the same time the SLA is ejected. Orion then performs a transposition and docking manoeuvre and docks with the LM below. The CPS then performs TLI and LOI. This will require the CPS to have a low boil–off rate, as the LH2 and LOX are stored at cryogenic temperatures.

Due to the large mass of Orion at 26,520 kg [9], this puts significant limits on the LM. To overcome this limitation we propose using the high performance of the CPS to also perform 75% of Lunar descent. The LM then performs the remaining 25% of Lunar descent to touchdown. This requires a critical stage separation and ignition by the LM at the end of the CPS burn. To increase the reliability of this event the LM has a CPM and an LCM. The LCM is a non–propulsive stage which carries cargo, has landing legs and supports the CPM.

The CPM can carry up to four crew (two crew are carried in the initial flights), all the propellant and has two sets of engines, descent and ascent. The ascent engine is centrally located beneath the CPM and protrudes through the middle of the LCM. Two descent engines are at the sides of the ascent engine. The descent engines can throttle and rotate in two axis to enable precise landing control. The ascent engine nominally performs Lunar ascent, carrying the crew and 100 kg of Lunar samples to Orion waiting in low Lunar orbit (LLO). This engine is of fixed thrust and position for maximum reliability.

During Lunar descent, if the descent engines fails to ignite or experiences an anomaly, the CPM separates from the LCM with the ascent engine being used for abort. If the LM fails to separate from the CPS, the CPM separates from the LCM and performs an abort, using either the descent or ascent engines. If the ascent engine fails or experiences an anomaly during Lunar ascent, the descent engines can be used as a backup.

Unlike the Apollo LM descent stage, the LCM can have a large cargo volume as it is free from carrying propellant. Only the space where the ascent and descent engines passes through the LCM is used. The surrounding volume can be used for carrying a Lunar rover, tools, experiments, antenna, solar panels and supplies. For future more capable versions of the SLS Block II configuration presented in this paper, a small habitation module could also be carried. This would allow missions up to 14 Earth days. For a future Lunar base, the LCM can carry pressurised and unpressurised supplies for the base, in addition to the crew. Thus, even though using staged descent carries some risk (which we have tried to minimise) it has some great advantages, including increased payload and future mission flexibility.

A detailed analysis of the SLS Block II configuration we have selected is presented in the following sections.

II SPACE LAUNCH SYSTEM BLOCK II

The SLS Block II consists of three main stages. The first stage consists of twin boosters. The second stage is an 8.407 m diameter core using RS–25D or RS–25E (expendable more cost efficient versions of the RS–25D) engines. The 8.407 m diameter third stage or LUS uses one or more J–2X engines. We have analysed SLS in a number of different configurations, with RSRMV, advanced solid, advanced liquid (using either two F–1B engines or three dual nozzle AJ1E6 engines), four to six RS–25D or RS–25E engines on the core and one to three J–2X engines on the upper stage [10]. For SLS configurations with a Block I core and an LUS, the boost and post–boost phase of flight suffers from low acceleration, typically around 20 m/s^2 maximum. This results in large gravity losses and limits the size of the upper stage and payload that can be carried.

To overcome this, NASA has proposed using advanced boosters to increase the impulse during the boost phase. With advanced solid boosters, we obtain a payload mass of 124.8 t [10] into a 200 km circular orbit, below the 130 t value required by Congress. We use a 200 km reference orbit as that is close to the 185 km orbit typically used during Apollo. We increased this to 200 km to allow the orbit to be more stable during transposition and docking (an operation performed after TLI in Apollo). With F–1B powered boosters we obtain 133.2 t and with AJ1E6 powered boosters we obtain 136.2 t [10]. This is using a non–modified core with four RS–25E engines. All these configurations used an LUS with two J–2X engines.

However, there is another way of increasing acceleration (and thus reducing gravity losses) during boost and post–boost flight. Simply increase the number of engines on the core. With existing RSRMV boosters, four RS–25E engines and one J–2X engine, the payload is only 113.6 t. With five RS–25E engines and two J–2X engines payload increases to 130.6 t. With six RS–25E engines the payload increases to 137.0 t, beating all other configurations except advanced solids which also requires a new core stage.

Thus, we have chosen a six–engined SLS core as our baseline configuration as that is the most cost effective option (as we will show later). However, the Lunar mission can also be completed with any of the other Block II configurations, so we are not limited to using this option alone.

In the following, we present our assumptions used in the design of the SLS Block II vehicle.

II.A RSRMV Boosters

The usable propellant mass is m_{p1} = 628,407 kg and the ejected inert mass is m_{p2} = 4,082 kg [7]. We combine these masses into a total propellant mass of $m_p = m_{p1} + m_{p2}$ = 632,489 kg. The exhaust speed of the propellant (not including the inerts) is v_{e1} = 2622.3 m/s (267.4 s) [8] with the inerts having zero exhaust speed (v_{e2} = 0 m/s). The average exhaust speed is $v_e = (m_{p1}v_{e1} + m_{p2}v_{e2})/m_p$ = 2605.4 m/s (265.7 s). The burnout mass is 96,751 kg (95,844 kg dry and 907 kg slag) [7] and the action time is 128.4 s [8]. Using the graph of vacuum thrust verses time in [11], we manually plotted the graph and calculated the total impulse. This was then used to adjust the curve for the actual impulse of $m_p v_e$ = 1,647,887 kNs. Figure 1 plots the vacuum thrust against time.

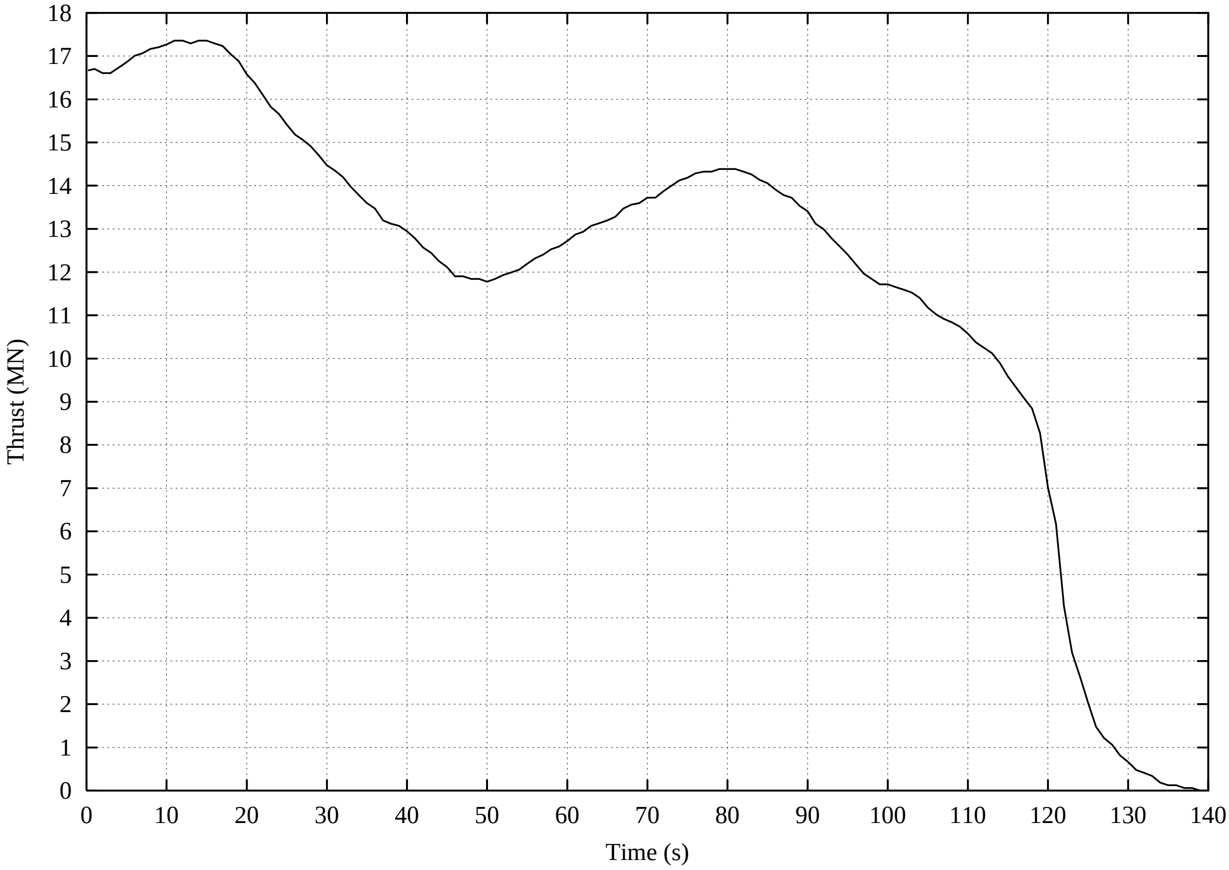

Figure 1: RSRMV vacuum thrust against time.

The nozzle exit diameter is 3.875 m [11]. The aft skirt diameter is d_s = 5.288 m [12]. The exposed area of the RSRMV hold down posts, separation motors and attachments was estimated to be A_{ha} = 0.763 m^2 from Figure 6–1 of [13]. There is an overlap between the aft skirt and core with diameter

d_e = 8.407 m [14] with a centreline distance of d = 6.363 m [14] (the Space Shuttle and SLS are assumed to have the same dimensions in this area). This area is given by [15]

$$A_{es} = A(d_e/2, x) + A(d_s/2, d - x) \tag{1}$$

where x is the horizontal distance between the core centre and the intersection with the aft skirt and $A(r,h)$ is the circular segment area with radius r and segment height h. We have that

$$x = \frac{d^2 + (d_e/2)^2 - (d_s/2)^2}{2d} = 4.021 \text{ m} \tag{2}$$

and

$$A(r, x) = r^2 \cos^{-1}(x/r) - x\sqrt{r^2 - x^2}. \tag{3}$$

This gives A_{es} = 0.301 + 0.500 = 0.801 m^2. The total additional area is then $A_{sa} = A_{ha} - A_{es}$ = –0.038 m^2. The above values are summarised in Table 1. The residual propellant is the propellant remaining after the action time.

Table 1: RSRMV Parameters

Aft Skirt Diameter (m)	5.288
Additional Area (m^2)	–0.038
Nozzle Exit Diameter (m)	3.875
Sea Level Thrust at 0.2 s (N)	15,471,544
Vacuum Isp (m/s)	2605.4
Total Mass (kg)	729,240
Usable Propellant (kg)	631,185
Residual Propellant (kg)	1,304
Burnout Mass (kg)	96,751
Action Time (s)	128.4

II.B Core Stage

The SLS Block I core with four RS–25D engines has a dry mass of m_{s1} = 100,062 kg [7]. Subtracting the mass of four RS–25D engines at m_{e1} = 3,545 kg each [16] gives $m_{se} = m_{s1} - 4m_{e1}$ = 85,882 kg. Other than for the engine mass, it is not known how much the dry mass will increase with the addition of two additional engines. For want of a better estimate, Boeing previously used a higher mass of m_{s2} = 115,575 kg for the core [8]. Thus, we will increase the core mass by $m_{sd} = m_{s2} - m_{s1}$ = 15,513 kg. This is an 18% increase in the tank and structure mass. The RS–25E engines are a little heavier at m_{e2} = 3,700 kg each [16]. The total dry mass is thus estimated to be $m_{se} + m_{sd} + 6m_{e2}$ = 123,595 kg.

The total propellant mass is m_p = 982,663 kg [7]. With four engines, the startup mass is $m_{ps,r}$ = 8,437 kg [7] and the nonusable propellant mass is $m_{pn,r}$ = 1,678 kg [8]. Thus, with six engines the startup mass is $m_{ps} = 1.5m_{ps,r}$ = 12,656 kg and the nonusable mass is $m_{pn} = 1.5m_{pn,r}$ = 2,517 kg. The total nonusable and reserve propellant mass in [7] for SLS with a LUS is $m_{pnr,r}$ = 9,662 kg.

This gives a reserve propellant mass of $m_{pr} = m_{pnr,r} - m_{pn,r}$ = 7,984 kg. The usable propellant mass is $m_u = m_p - m_{ps} - m_{pn} - m_{pr}$ = 959,506 kg.

Figure 2 illustrates two possible engine configurations. Note that the edge of the RSRMV aft skirt is about 1.7 m higher than the RS–25E engine nozzle outlet and thus does not interfere with operation of the engine. The first configuration has two engines that are only 0.936 m away from each RSRMV nozzle, compared to one engine that is 1.903 m for the second configuration. For this reason, we have chosen the second configuration. With both configurations, the core could also be used with five or four engines, although thrust is slightly asymmetric with five engines.

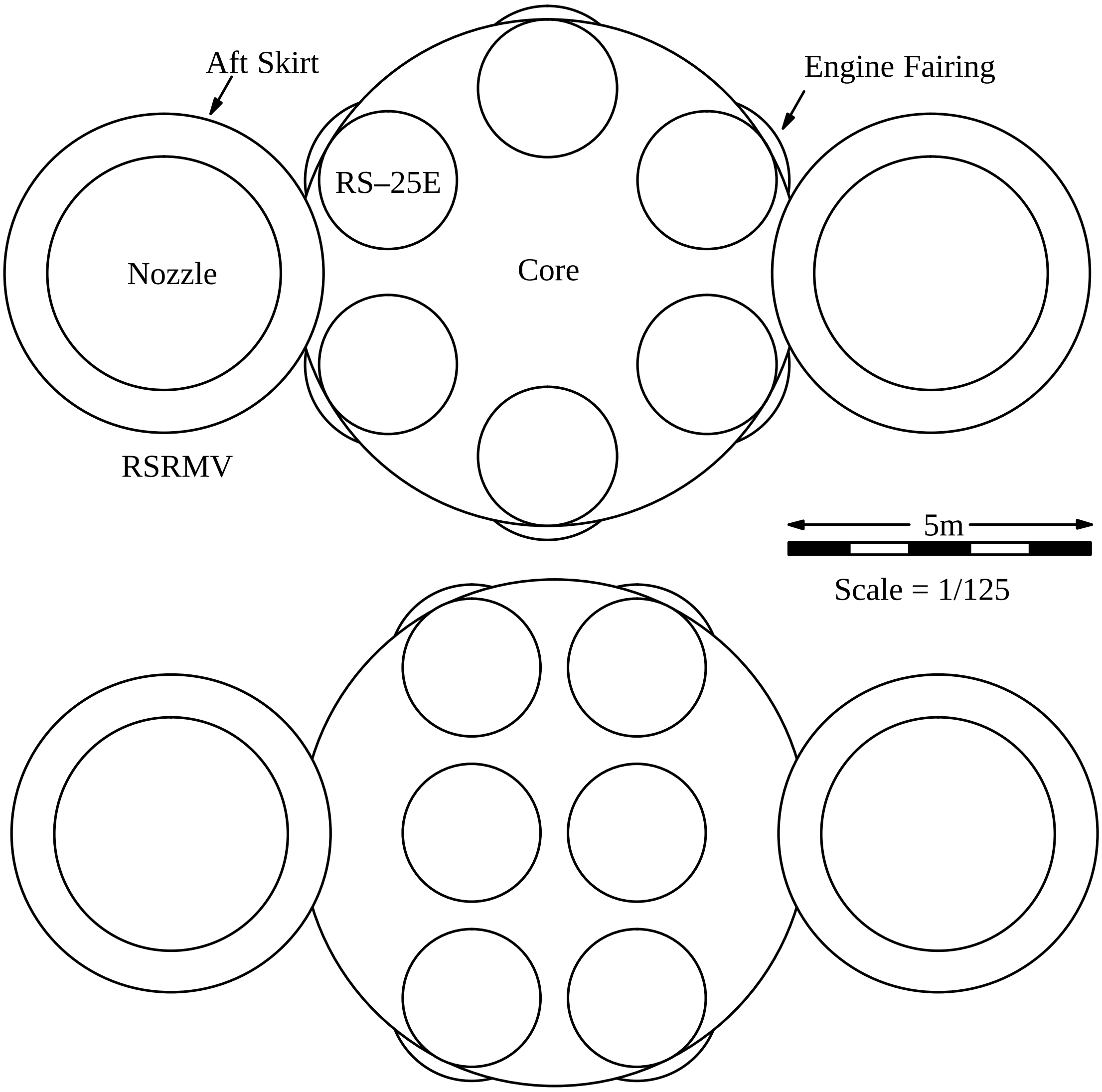

Figure 2: RSRMV and Core engine configurations.

For the RS–25E, the vacuum exhaust speed is 4420.8 m/s (450.8 s) [16]. A constant maximum vacuum thrust of 111% of rated power level (RPL) [16] or 2,320,637 N is used. The nozzle exit diameter is 2.304 m [17]. The core diameter is assumed to be the same as the Space Shuttle external tank of 8.407 m [14]. From Figure 6–1 of [13] we estimate the areas of each liquid oxygen feed line to be A_{cf} = 0.608 m^2, each engine fairing to be A_{ce} = 0.3045 m^2 and the tunnel to be A_{ct} = 0.045 m^2. The Block I core has two feed lines and four engine fairings. For the chosen six engine configuration we require three feed lines (this may be designed as two larger feedlines), four engine fairings and one tunnel. Thus, the total estimated additional area for the core is $A_{ca} = 3A_{cf} + 4A_{ce} + A_{ct} = 3.087$ m^2. The above values are summarised in Table 2.

Table 2: Core Parameters with RS–25E engines

Diameter (m)	8.407
Additional Area (m^2)	3.087
Nozzle Diameter (m)	2.304
Single Engine Vacuum Thrust (N) 111% RPL	2,320,637
Vacuum Isp (m/s)	4420.8
Number of Engines	6
Total Mass at Liftoff (kg)	1,093,602
Dry Mass (kg)	123,595
Usable Propellant (kg)	959,506
Reserve Propellant (kg)	7,984
Nonusable Propellant (kg)	2,517
Startup Propellant (kg)	12,656

II.C Large Upper Stage

The upper stage mass is determined in an iterative fashion. We start with a fixed total interstage, upperstage and payload mass (m_t). By adjusting the turn time of the first stage and maximum angle of attack of the core and LUS, the desired 37x200 km orbit is reached. This process is semi–automated as the program calculates a new angle based on the previous angle and the difference between the current and desired orbit. New parameters for the interstage, upperstage and payload are calculated and substituted back into the program. This process is repeated until the remaining usable propellant is zero. This gives the payload achievable for a given total m_t. The usable propellant mass is then increased or decreased in several further iterations until the payload mass is maximised. Typically, about 100 to 200 simulations are required to find the optimum mass.

As shown in Section II.H, in order for the vehicle to meet the height restriction of the Kennedy Space Center (KSC) Vehicle Assembly Building (VAB), the LUS and CPS must both use a common bulkhead design. A common bulkhead also has the advantage of lower mass and thus greater payload to LEO, at the expense of greater development and manufacturing cost.

The optimum m_t for this SLS configuration was found to be 383,500 kg. This gave a payload mass into LEO of 143,165 kg. This includes an additional 6,206 kg of payload due to using a common bulkhead design for the LUS. However, the vehicle was found to be over 2 m too high to fit the VAB. The solution we chose for this problem was to reduce m_t to 344,300 kg. This resulted in the LUS propellant mass being reduced by 34,434 kg, obtaining the necessary reduction in height. Payload decreased by only 2,498 kg to 140,667 kg.

The interstage mass was determined from a trajectory simulation of the vehicle in [8]. This vehicle has an interstage mass of $m_{i,r}$ = 7,394 kg and height of $h_{i,r}$ = 15.0 m (estimated from Fig. 9 of [8]). From Section II.H, the interstage height for a common bulkhead design is h_i = 7.5 m. It was found that the maximum weight of m_t due to acceleration and dynamic pressure acting on the reference vehicle was $F_{i,r}$ = 7,989,605 N. From our simulation, m_t experienced a maximum weight of F_i = 9,992,646 N at 304.05 s into flight. Thus, the interstage mass is $m_i = m_{i,r}(F_i/F_{i,r})(h_i/h_{i,r})$ = 4,624 kg. For comparison, the S–IC/S–II interstage of the Apollo 14 Saturn V launch vehicle has a smaller dry mass of only 3,957 kg [18], even though the interstage has a larger 10 m diameter, a larger m_t of 488,027 kg, a higher maximum acceleration of 37.5 m/s^2 and a higher dynamic pressure of 32 kPa.

With two J–2X engines, the startup propellant mass is m_{su} = 771 kg [8]. To determine the unusable propellant mass, we use as reference data from the S–II second stage of the Saturn V [18], where gaseous oxygen and hydrogen were used to pressurise the tanks. Table 3 summaries the respective data.

Table 3: Apollo 14 S–II Predicted Propellant Data

	Mass (kg)	Symbol
LOX In Tank at Separation	679	$m_{ito,r}$
LOX Below Tank at Separation	787	$m_{bto,r}$
LOX Ullage Gas at Separation	2,254	$m_{ugo,r}$
Total LOX at Liftoff	379,876	$m_{po,r}$
Fuel In Tank at Separation	1505	$m_{itf,r}$
Fuel Below Tank at Separation	123	$m_{btf,r}$
Fuel Ullage Gas at Separation	599	$m_{ugf,r}$
Total Fuel at Liftoff	72,476	$m_{pf,r}$

Five J–2 engines have oxidiser and fuel rates of $R_{o,r}$ = 1053.9 kg/s and $R_{f,r}$ = 190.4 kg/s, respectively [18]. For an oxidiser to fuel mixture ratio of r_m = 5.5, two J–2X engines have oxidiser and fuel rates of R_o = 503.7 kg/s and R_f = 91.6 kg/s, respectively. Normalising the below tank propellant mass by these propellant rates, we obtain a below tank oxidiser mass of $m_{bto} = m_{bto,r}R_o/R_{o,r}$ = 376 kg, below tank fuel mass of $m_{btf} = m_{btf,r}R_f/R_{f,r}$ = 59 kg and below tank propellant mass of $m_{bt} = m_{bto} + m_{btf}$ = 435 kg.

We assume the reserve oxidiser mass $m_{ro,r}$ is the in tank oxidiser mass $m_{ito,r}$ = 679 kg, the reserve fuel mass is $m_{rf,r} = m_{ro,r}/r_{m,r}$ = 142 kg (the mixture ratio at engine cutoff is $r_{m,r}$ = 4.8 [18]) and the fuel bias mass is $m_{fb,r} = m_{itf,r} - m_{rf,r}$ = 1363 kg. The fuel bias is to ensure that engine cutoff is fuel rich, to prevent the oxidiser from burning any metallic engine components. Normalising by the fuel rate we obtain a fuel bias of $m_{fb} = m_{fb,r}R_f/R_{f,r}$ = 656 kg.

The oxidiser and fuel ullage gas masses are given by

$$m_{ugo} = f_{ugo}\left(\frac{m_{ms} + m_r}{1 + 1/r_m}\right) \tag{4}$$

$$m_{ugf} = f_{ugf}\left(\frac{m_{ms} + m_r}{1 + r_m} + m_{fb}\right) \tag{5}$$

where m_{ms} is the mainstage propellant mass (including startup propellant), m_r is the reserve propellant mass, $f_{ugo} = m_{ugo,r}/(m_{po,r}–m_{bto,r}–m_{ugo,r})$ = 0.5981% and $f_{ugf} = m_{ugf,r}/(m_{pf,r}–m_{btf,r}–m_{ugf,r})$ = 0.8348%. From our simulation, we obtained m_{ms} = 166,819 kg and m_r = 449 kg for a 0.5% increase in delta–V. This gives m_{ugo} = 847 kg, m_{ugf} = 220 kg and $m_{ug} = m_{ugo} + m_{ugo}$ = 1,067 kg. The total propellant mass $m_p = m_{ms} + m_r + m_{ug} + m_{bt} + m_{fb}$ = 169,426 kg.

To estimate the dry mass of the upperstage, we use a nonlinear model. Using historical data, we showed in [19] that the dry stage mass for cryogenic upper stages without the engines can be modelled by

$$m_s = \alpha m_p^{0.848} \tag{6}$$

where α is a constant depending on the materials and technology used in the stage. This model is more realistic than a linear model since it reflects a higher dry mass fraction for low values of m_p and low values for high m_p. To determine α, we use the total S–II dry mass of $m_{st,r}$ = 35,402 kg [18] which includes five J–2 engines. The J–2 dry mass is $m_{e,r}$ = 1,584 kg [20] and the J–2X dry mass is m_e = 2,472 kg [3]. We have the reference dry mass as $m_{s,r} = m_{st,r} – 5m_{e,r}$ = 27,482 kg. This gives $\alpha = m_{s,r}/m_{p,r}^{0.848}$ = 0.43975. Thus, the total dry mass is estimated to be $m_{st} = \alpha m_p^{0.848} + 2m_e$ = 16,894 kg.

To ensure the propellants are settled prior to engine start, solid motors are used like that in the S–II stage of the Saturn V. To model the required thrust we use as reference the ullage motors of the second and third stages of the Saturn V [18]. The total mass of the vehicle after first and second stage separation are m_{ut2} = 666,299 kg and m_{ut3} = 166,258 kg, respectively. The total vacuum thrust is F_{u2} = 409,236 N and F_{u3} = 30,159 N. We use a nonlinear model where

$$F_u = \alpha_u m_{ut}^{\beta_u}. \tag{7}$$

Using the reference values we have $\beta_u = \ln(F_{u3}/F_{u2})/\ln(m_{ut3}/m_{ut2})$ = 1.8786 and $\alpha_u = F_{u3}/m_{t3}^{\beta_u}$ = 4.6976x10$^{–6}$. Thus for, $m_{ut} = m_t – m_i$ = 339,676 kg we have F_u = 115,425 N. The ullage motors are offset θ = 30° from the centreline, so the inline thrust is reduced to $F_u\cos(30°)$ = 99,961 N.

We use a linear model of the ullage motor propellant mass as a function of thrust. For the S–IVB, we have m_{up3} = 53.5 kg and m_{us3} = 61.2 kg. Thus $m_{up} = m_{up3}F_u/F_{u3}$ = 205 kg. For the case mass, we use a nonlinear model where $\alpha_{us} = m_{us3}/m_{up3}^{0.848}$ = 2.0946. Thus $m_{us} = \alpha_{us}m_{up}^{0.848}$ = 191 kg. We use the same event times as for the S–IVB [18]. The ullage motors are started 0.18 s before core separation and have an action time of 3.87 s. Separation of the ullage motor casings occurs 11.72 s after core separation.

The above values are summarised in Table 4. The J–2X parameters are from [16].

Table 4: Large Upper Stage Parameters with J–2X engines

Diameter (m)	8.407
Nozzle Diameter (m)	3.048
Single Engine Vacuum Thrust (N)	1,307,777
Vacuum Isp (m/s)	4393.4
Number of Engines	2
Total Mass at Liftoff (kg)	186,716
Dry Mass (kg)	16,894
Total Propellant (kg)	169,426
Startup Propellant (kg)	771
Main Stage Propellant (kg)	166,048
Reserve Propellant (kg)	449
Ullage Gas Propellant (kg)	1,067
Below Tank Propellant (kg)	435
Fuel Bias Propellant (kg)	656
Ullage Motors Propellant (kg)	205
Ullage Motors Dry Mass (kg)	191
Ullage Motors Thrust (N)	141,615
Ullage Motors Action Time (s)	3.87
Ullage Motors Offset Angle (°)	30
Interstage Mass (kg)	4,624

II.D Cryogenic Propulsion Stage

The CPS first burn is to circularise the orbit to 200 km circular. Four RL–10C–2 engines are used, the same as the EUS in [7]. To avoid a trajectory that rises and then falls to Earth, the upper stage releases the CPS near 200 km altitude. After 1.8 s, the CPS fires to circularise the orbit. The upperstage returns to Earth to burn up in the atmosphere. Before engine start the mass of the interstage, CPS and payload is m_i = 143,933 kg. For a separate tank design, this mass is reduced by 5,864 kg to 138,069 kg, indicating the significant performance advantage of a common bulkhead for the LUS. From Section II.H, the CPS interstage height is h_i = 6.3 m. The maximum weight for the total is F_i = 4,471,756 N at 81 s. This gives an interstage mass of $m_i = m_{i,r}(F_i/F_{i,r})(h_i/h_{i,r})$ = 1,738 kg.

To perform Earth orbit insertion (EOI) and trans–Lunar injection, these were simulated to show that Δv_{eoi} = 49.0 m/s and Δv_{tli} = 3184.9 m/s are required. If an engine fails to start at the beginning of the burn, then $\Delta v_{tli,3}$ = 3220.2 m/s which is a 1.1% increase. Thus, we include a 1.1% delta–V margin for TLI. All other delta–V's are increased by a 1% margin.

The initial mass is $m_t - m_i$ = 142,195 kg before LEO insertion. From [21], the highest Lunar orbit insertion delta–V was Δv_{loi} = 960.4 m/s for Apollo 14. Here we assume LLO insertion is into an approximate 110 km circular orbit, instead of with a perilune of 15 km (921.2 m/s to 107.6x313.0 km plus 62.7 m/s to 16.9x108.9 km minus 23.5 m/s to 103.7x118.3 km). A total powered descent of Δv_{tpd} = 2041.6 m/s from Apollo 17 is used. The CPS performs 75% of powered descent, giving $\Delta v_{pd} = 0.75\Delta v_{tpd}$ = 1531.2 m/s.

We assume a boil–off rate of r_{bo} = 0.17% per day, which is 70% greater than [22] claims can be achieved for the Centaur stage with modifications. In [23] a low boil–off version of the Delta–IV Heavy upper stage is examined. Figure 3–2 of [23] indicates that an independent cooling system can have a boil–off rate of only 9.3 kg/day using 500 kg of additional thermal protection. That corresponds to a rate of only 0.034% per day for an initial propellant mass of 27,200 kg [24], five times less than our assumed value. The calculated boiloff mass in each flight segment i is $m_{boi} = T_i r_{bo} m_p$ where T_i is the number of days for slight segment i and m_p is the initial total propellant mass.

To allow sufficient time to perform transposition and docking in case there are problems, 0.25 days or four orbits are spent in LEO. This value is taken from Apollo 14 where the CSM/LM separated from the S–IVB at 5 hours and 47 minutes into the mission [21]. Lunar transit can take up to 3.5 days (Apollo 17 was 3.46 days). We assume a stay time in Lunar orbit before descent of 1.25 days, the same time as Apollo 16, where additional time was needed to resolve a problem with the SM engine. Once more experience is gained though, the number of orbits can be reduced.

Assuming an oxidiser to fuel mixture ratio of r_m = 5.88 [25], four RL–10C–2 engines have oxidiser and fuel rates of R_o = 83.0 kg/s and R_f = 14.1 kg/s, respectively. Using the S–II model, we obtain m_{bto} = 62 kg, m_{btf} = 9 kg, m_{bt} = 71 kg and m_{fb} = 101 kg. From our program, we obtain m_{ms} = 94,100 kg (including boiloff) and m_r = 460 kg. This gives ullage gas masses of m_{ugo} = 483 kg, m_{ugf} = 116 kg and m_{ug} = 599 kg. The total propellant mass is $m_p = m_{ms} + m_r + m_{bt} + m_{fb} + m_{ug}$ = 95,330 kg.

The RL–10C–2 dry mass is assumed to be the same as the RL–10B–2 dry mass of m_e = 301 kg [25]. As for the LUS, a common bulkhead design for the CPS is required in order to meet vehicle height requirements. In [26], a common bulkhead design with four RL–10 engines called ACES 41 is presented. The reference inert mass is $m_{st,r}$ = 5,000 kg with propellant mass $m_{p,r}$ = 40,800 kg. We obtain $\alpha = (m_{st,r} - 4m_e)/m_{p,r}^{0.848}$ = 0.46718. The exhaust speed of the RL–10C–2 is v_e = 4535.6 m/s (462.5 s) [7].

The total trans Lunar (TL) trajectory correction manoeuvre (TCM) CPS reaction control system (RCS) delta–V is Δv_{tcm1} = 3.8 m/s (Apollo 16). This is the largest value of the three Apollo J missions. For powered descent initiation (PDI), we have CPS RCS Δv_{pdi} = 24.9 m/s (Apollo 16) and assume powered descent (PD) CPS RCS burns of Δv_{pdr} = 5.5 m/s, half of the total given in [27]. The other half is performed by the LM during descent. For the CPS RCS, we assume gaseous hydrogen and oxygen is used (GH_2/GO_2). In [28] an actual GH_2/GO_2 RCS thruster was tested which has an exhaust speed of $v_{e,crs}$ = 3432.3 m/s (350 s).

Due to the complex non–linear model used, we used an iterative algorithm to determine the total propellant mass of the CPS. Table 5 gives the parameters for the CPS. Note that due to rounding errors, the sums of the subtotals may be slightly different from the total values.

Table 5: CPS Parameters with RL–10C–2 engines

Diameter (m)		8.407
Nozzle Diameter (m)		2.146
Single Engine Vacuum Thrust (N)		110,093
Vacuum Isp (m/s)		4535.6
Number of Engines		4
Total Mass at Liftoff (kg)		104,330
Dry Mass (kg)		9,000
Total Propellant (kg)		95,330
EOI Propellant (kg)	49.0 m/s	1,528
LEO Boiloff (kg)	0.25 days	41
TLI Propellant (kg)	3184.9 m/s	70,038
TCM RCS Propellant (kg)	3.8 m/s	76
TL Boiloff (kg)	3.5 days	567
LOI Propellant (kg)	960.4 m/s	13,004
LLO Boiloff (kg)	1.25 days	203
PDI RCS Propellant (kg)	24.9 m/s	213
PD Propellant (kg)	1531.2 m/s	8,383
PD RCS Propellant (kg)	5.5 m/s	47
Reserve Propellant (kg)	60.8 m/s	460
Ullage Gas Propellant (kg)		599
Below Tank Propellant (kg)		71
Fuel Bias Propellant (kg)		101
Interstage Mass (kg)		1,738

II.E Orion Multipurpose Crew Vehicle

The total Orion command module (CM) mass including four crew members is m_{cm4} = 10,387 kg [9]. Assuming m_{cm} = 125 kg for each crew member [8], this gives a CM mass of $m_{cm} = m_{cm4} - 4m_{cm}$ = 9,887 kg. The European service module (ESM) inert mass is m_{sm} = 6,858 kg with up to 8,602 kg of storable propellant [9]. The Orion adaptor mass is m_{oa} = 510 kg [29]. The reference SLA mass is $m_{sla,r}$ = 2,300 kg [8]. From Figure 4 in [8], we estimate the height of this SLA to be

$h_{sla,r}$ = 9.535 m. As determined from Section II.H, the SLA height is h_{sla} = 5.326 m. This the SLA mass is $m_{sla} = m_{sla,r} h_{sla} / h_{sla,r}$ = 1,285 kg.

The Service Module Fairing (SMF) and Launch Abort System (LAS) masses are m_{smf} = 1,384 kg and m_{las} = 7,643 kg, respectively [29]. These are jettisoned at t_{smf} = 375 s and t_{las} = 380 s after launch [30]. The orbital manoeuvring system (OMS) engine from the Space Shuttle is used with an exhaust speed of $v_{e,o}$ = 3069.5 m/s (313 s) [31]. The exhaust speed of the Orion 220 N RCS thrusters is $v_{e,or}$ = 2650 m/s [32].

We use the unusable propellant mass fraction of the total propellant from the Apollo 11 LM descent stage of f_u = 0.5279% [21]. We assume Orion RCS burns of Δv_{tad} = 0.6 m/s for transposition and docking (TAD) in LEO. Before the LM ascent stage returns to LLO, Orion performs a plane change (PC) of up to Δv_{pc} = 46.2 m/s. Higher values are not possible due to the limited amount of available propellant. This allows latitudes to be reached on the Lunar surface that are about half that of Apollo, or approximately 12°. For Orion RCS burns in LLO, we use Δv_{llo} = 5.5 m/s. The trans Earth injection (TEI) burn is Δv_{tei} = 1168.7 m/s (Apollo 14) with TCM burns of Δv_{tcm2} = 1.7 m/s (Apollo 15).

At TLI, the maximum acceleration is 6.401 m/s^2 and Orion mass is 25,716 kg giving a maximum load on the LM of 164.6 kN. This is well within the maximum compressive axial load of 300 kN of the International Docking System Standard [33]. For LOI, two of the four RL–10 engines can be fired to reduce axial loads. Table 6 gives the parameters for Orion.

Table 6: Orion Parameters

Diameter (m)		5.029
Vacuum Isp (m/s)		3069.5
Total Mass at Liftoff (kg)		35,259
Launch Abort System Mass (kg)		7,643
Crew Mass (kg)		375
Crew Module Mass (kg)		9,887
Service Module Inert Mass (kg)		6,858
Service Module Fairing Mass (kg)		1,384
Service Module Adaptor Mass (kg)		510
Total Propellant (kg)		8,602
TAD Propellant (kg)	0.6 m/s	6
PC Propellant (kg)	46.2 m/s	380
LLO RCS Propellant (kg)	5.5 m/s	53
TEI Propellant (kg)	1168.7 m/s	8,037
TCM RCS Propellant (kg)	1.7 m/s	11
Reserve Propellant (kg)	12.2 m/s	69
Unusable Propellant (kg)		45
Spacecraft Launch Adaptor Mass (kg)		1,285

II.F Lunar Module

Table 7 gives the parameters for the LM. The Lunar Module carrying two crew members at 125 kg each performs the remaining of powered descent of Δv_{ds} = 0.25*2041.6 = 510.4 m/s. It is assumed that Lunar ascent is performed with the abort engine. The descent and ascent RCS delta–V are Δv_{dsr} = 5.5 m/s and Δv_{asr} = 5.5 m/s, respectively. For the descent engine, we use the exhaust speed of the VTR–10 Lunar Module descent engine of 2991.0 m/s (305 s) [34]. For the ascent engine, we use the exhaust speed of the RS–1801 Lunar Module ascent engine of 3040.1 m/s (310 s) [34]. We assume R–4D 44:1 expansion ratio engines are used for the LM RCS thrusters with an exhaust speed of $v_{e,lmr}$ = 2942.0 m/s (300 s) [35]. The ascent delta–V is Δv_{as} = 1890.0 m/s (Apollo 11).

Table 7: LM Parameters

Landing Engines Isp (m/s)		2991.0
Ascent Engine Isp (m/s)		3040.1
Total Mass at Liftoff (kg)		10,348
CPM Dry Mass (kg)		3,558
LCM Mass (kg)		588
LM Adaptor Mass (kg)		602
Cargo Mass (kg)		509
Total Propellant (kg)		5,092
Descent RCS Propellant (kg)	5.5 m/s	19
Descent Propellant (kg)	510.4 m/s	1,568
Ascent RCS Propellant (kg)	5.5 m/s	14
Ascent Propellant (kg)	1890.0 m/s	3,432
Reserve Propellant (kg)	24.1 m/s	33
Unusable Propellant (kg)		27
Crew Mass (kg)		250
Return Sample Mass (kg)		100

In [8], an LM adaptor mass of $m_{lma,r}$ = 1,000 kg is used for an LM mass of $m_{lm,r}$ = 16,200 kg. Thus, we use the scale factor of $m_{lma,r}/(m_{lm,r}+m_{lma,r})$ = 5.814% of the total LM and adaptor mass to determine the adaptor mass. We assume the LCM mass is 7% of the total landed mass. The CPM includes 2,207 kg for a multi–mission space exploration vehicle (MMSEV) cabin [36]. For the ascent stage propulsion system, for want of a better model, we use as reference the Apollo 11 Lunar

Module descent stage [21] with $m_{st,r}$ = 2,033 kg and $m_{p,r}$ = 8,248 kg which gives $\alpha = m_{st,r}/m_{p,r}^{0.848}$ = 0.9707.

For comparison, the Apollo 11 descent stage dry mass was 27.7% of the landed mass (which included the descent stage engine and propellant tanks, which are not included in the LCM) and ascent stage dry mass of 2,179 kg. For return to Earth, the CPM carries 100 kg of Lunar samples. For the above configuration, the LCM is able to carry 509 kg of cargo, which can be used for a Lunar roving vehicle, tools and experiments.

II.G Trajectory Simulations

To estimate the performance of the Block II SLS a trajectory simulation program called *sls*2 was written. A 32–bit DOS executable and Pascal source code for this program is available from [37] for configuration SLS1C6J2C4. Software for also determining the CPS, Orion and LM masses called *lunar* is also given in [37]. The 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 [38] but can be applied to any rocket on any planet. The program uses the Runga–Kutta fourth order method to solve the differential equations and a standard atmosphere model. The program is able to model thrust which changes proportionally with time. This is useful in accurately simulating the thrust curve of solid motors, as well as thrust buildup and dropoff of liquid propellant engines.

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 booster separation. After pitch over the vehicle follows a gravity turn such that the air angle of attack is zero. After booster 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 $-\,\mathrm{sign}(h_1)|h_1|^p$ 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. Values of p = 2 are used after booster separation and p = 1 after core separation. 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 booster separation there is not enough thrust to maintain a positive rate of altitude increase and so the angle of attack increases to its maximum value. Once centrifugal forces build up to a sufficient degree the angle of attack gradually decreases.

The launch latitude is θ_l = 28.45°, but the required orbital inclination for Lunar missions is θ_o = 32.55° [21]. As we are using a 2–D program, we approximate this by reducing the inertial speed at liftoff. Using the spherical law of cosines [39], the orbital plane azimuth (where East is 0° and North is 90°) is given by $\delta = \arccos(\cos(\theta_o)/\cos(\theta_l))$ = 16.52° (note that this is not the same as the launch azimuth). The launch site inertial speed is $v_l = 2\pi R_e\cos(\theta_l)/T$ = 408.9 m/s where the

Earth radius is R_e = 6,378,165 m and the sidereal rotational period is T = 86,164.09 s. The orbital speed at altitude h_o = 200,000 m is $v_o = \sqrt{\mu/(R_e + h_o)}$ = 7783.2 m/s where μ = 3.986005×10^{14} m^3/s^2 is Earth's gravitational constant. Using the planer law of cosines, this gives the required delta–V of $\Delta v_r = \sqrt{v_s^2 + v_o^2 - 2v_s v_o \cos(\delta)}$ = 7393.1 m/s. We thus use an adjusted surface speed of $v_o - \Delta v_r$ = 391.1 m/s. Note that this is less than launching from a latitude equal to θ_o where the inertial speed is 392.0 m/s.

To obtain a 200.0 km circular orbit inclined at 32.55° a turn time of 5.051 s and a maximum angle of attack of 10.9612° was used. Figures 3, 4, 5 and 6 plot speed, altitude, acceleration and dynamic pressure versus time, respectively. Maximum dynamic pressure (maxQ) is 28.9 kPa at T+61 s compared to 31.4 kPa for the Space Shuttle [40]. Maximum acceleration with no throttle changes is 29.02 m/s^2 at the end of core burnout at T+304.05 s. This is less then the maximum value of 29.42 m/s^2 (3g). Table 8 summaries the vehicle performance into LEO.

Table 8: SLS Block II Summary

Orbit (km)	200.0 ± 0.0
Inclination (°)	32.55
Liftoff Thrust at 0.2 s (N)	42,332,715
Liftoff Mass (kg)	2,895,882
Liftoff Acceleration (m/s^2)	14.63
MaxQ (Pa)	28,878
Maximum Acceleration (m/s^2)	29.02
LAS Jettison Time (s)	375
SMF Jettison Time (s)	380
Total Payload (kg)	140,667
Total Delta–V (m/s)	9,155

II.H Vehicle Height

With three stages using low density liquid hydrogen, there is a potential problem that the vehicle may be too high for the KSC VAB. The maximum vehicle length is limited to be no greater than 118.872 m [41]. The core length is 64.86 m [42].

To estimate the vehicle heights, we assume that the dome height is one third of the tank diameter. The ullage volume was estimated to be f_{ul} = 7% of the propellant volume using propellant mass data from [18] and volumes estimated from Saturn V drawings. The LOX and LH2 nominal boiling point (NBP) densities are d_o = 1,149 kg/m^3 and d_f = 70.9 kg/m^3, respectively [43]. The volume of a domed cylindrical tank is given by

$$V = \pi D^2(L/4 + D/9) \tag{8}$$

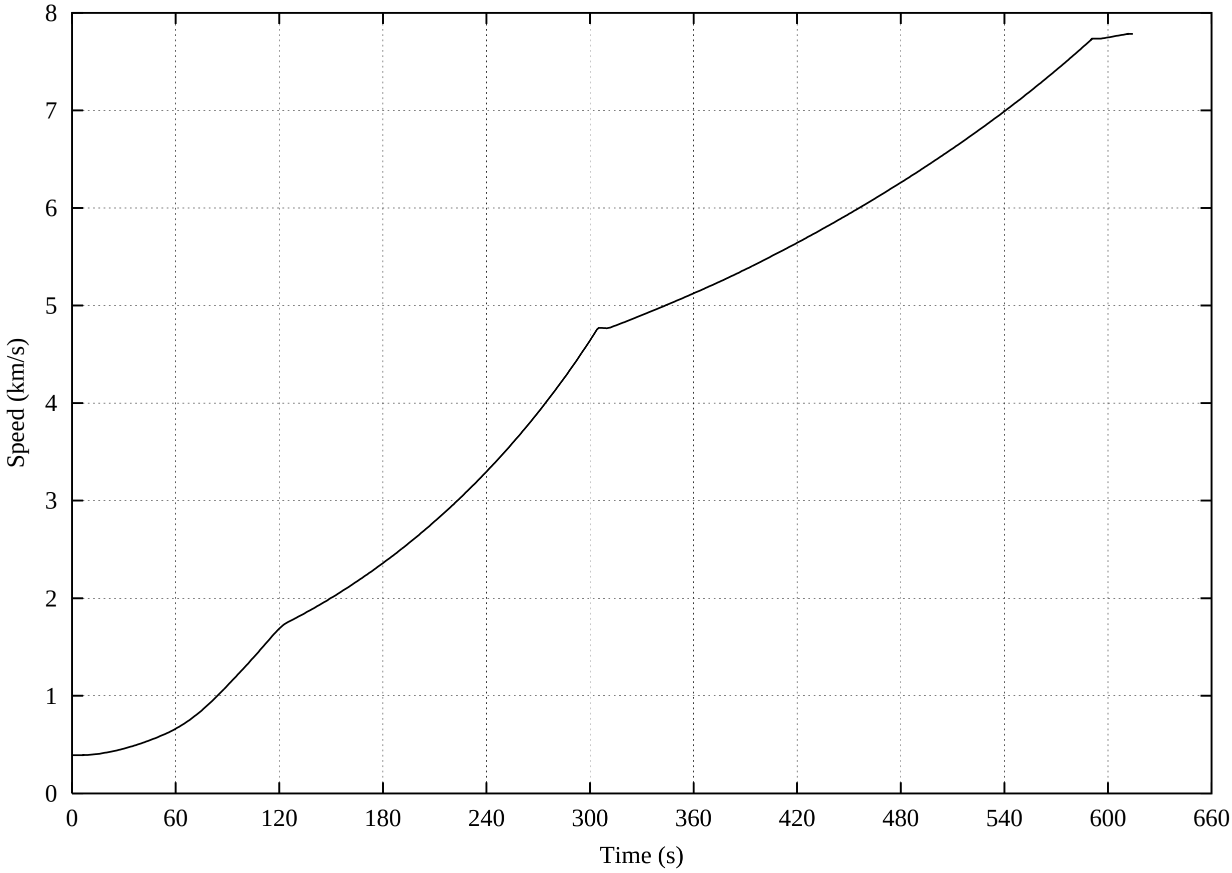

Figure 3: Speed versus time.

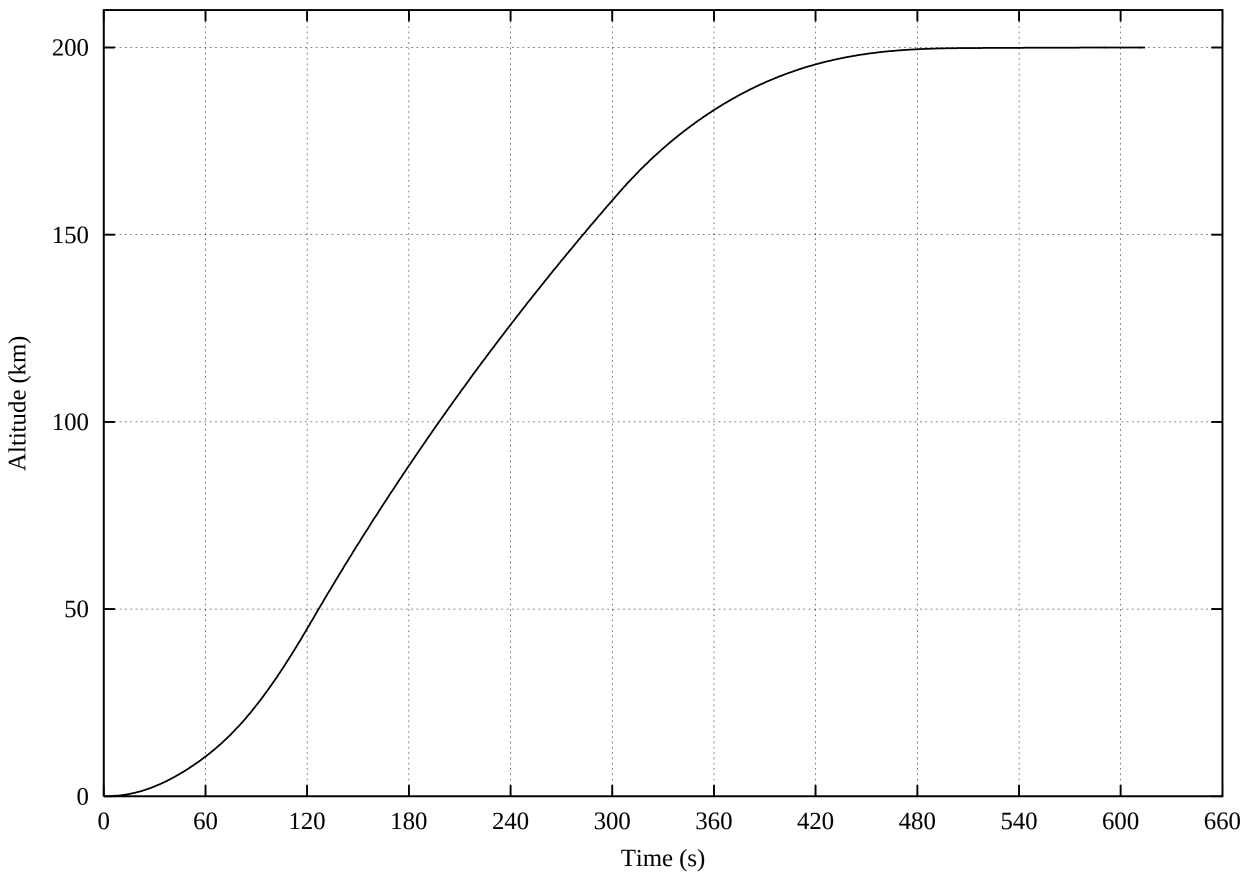

Figure 4: Altitude versus time.

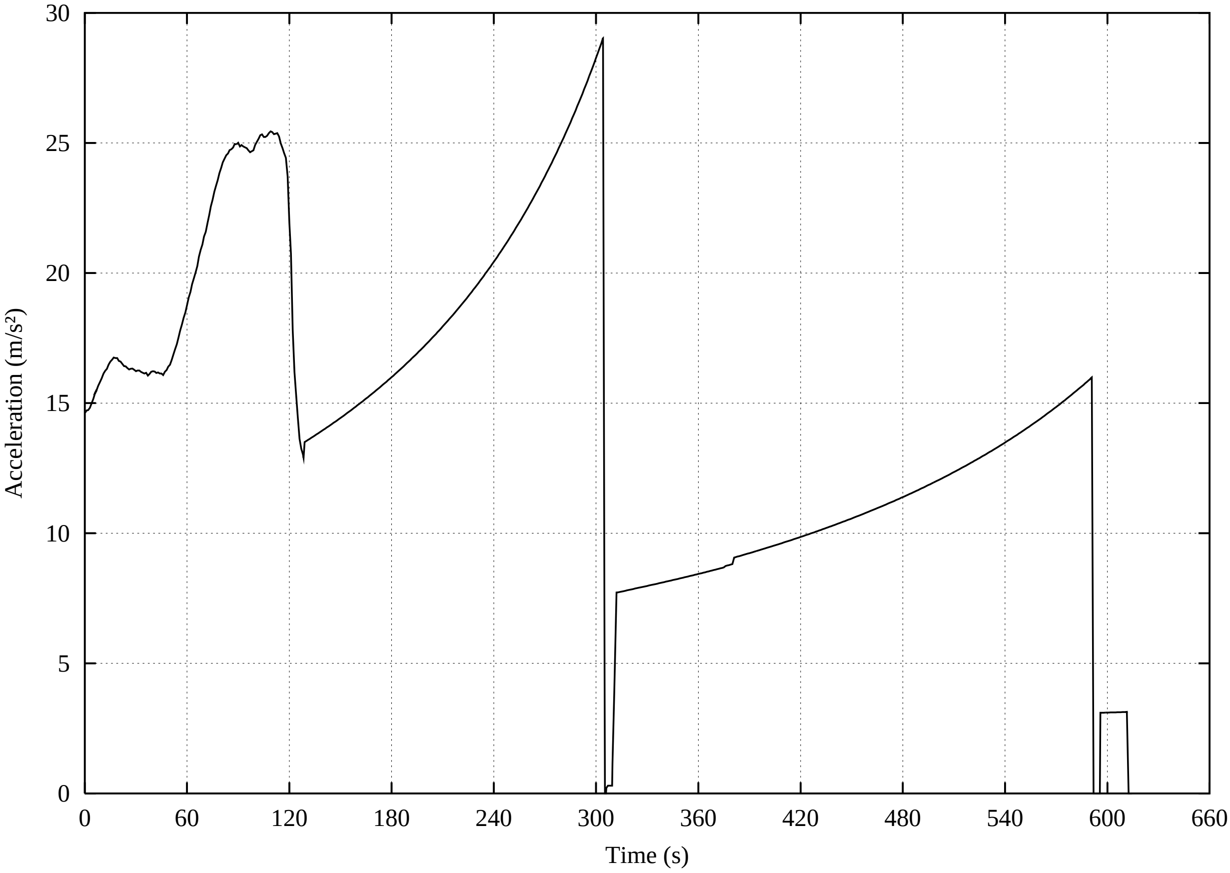

Figure 5: Acceleration versus time.

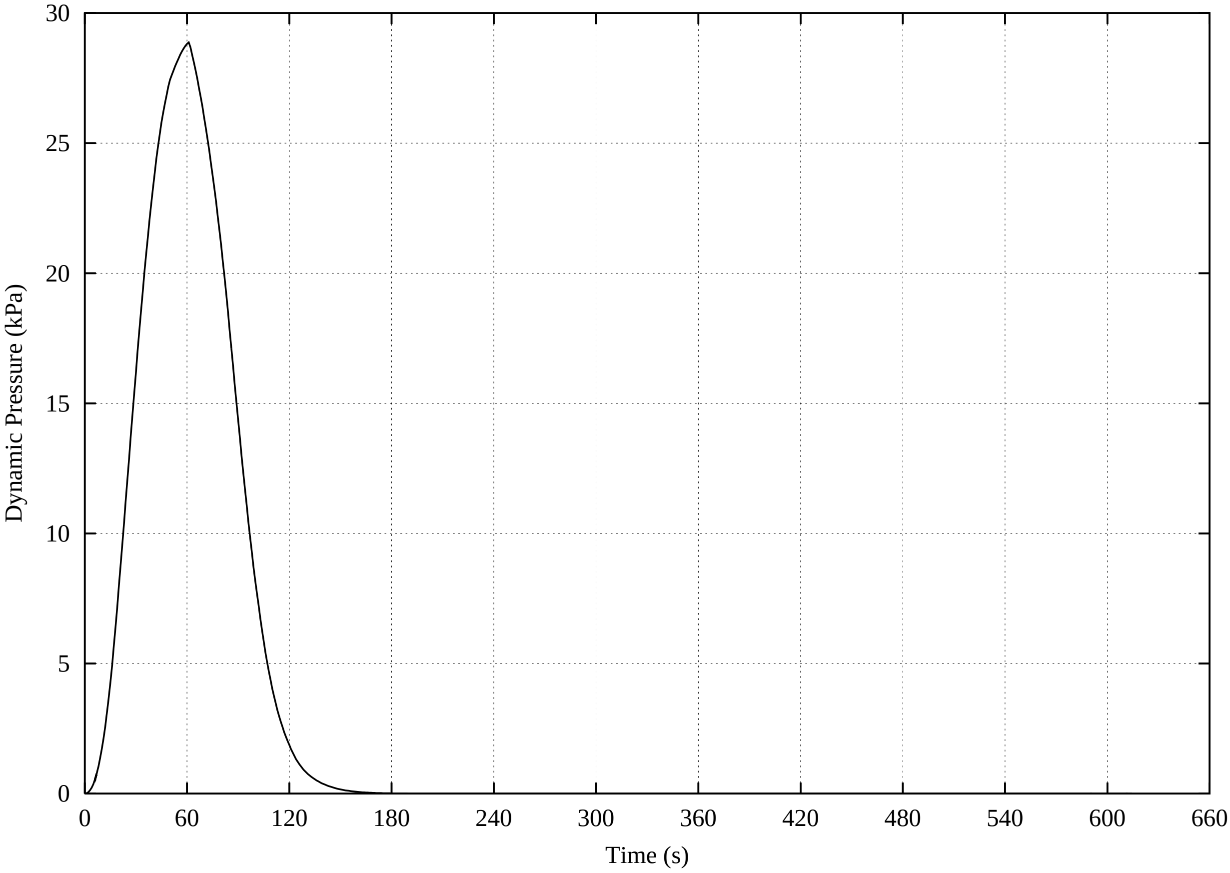

Figure 6: Dynamic pressure versus time.

where D is the tank diameter and L is the length of the tank side walls. The oxidiser and fuel tank volumes are

$$V_o = \frac{(1 + f_{ul})}{d_o}\left(\frac{m_{ms} + m_r}{1 + 1/r_m} + m_{ugo}\right) \tag{9}$$

$$V_f = \frac{(1 + f_{ul})}{d_f}\left(\frac{m_{ms} + m_r}{1 + r_m} + m_{ugf} + m_{fb}\right). \tag{10}$$

For the LUS we have m_{ms} = 166,819 kg, m_r = 449 kg, m_{ugo} = 847 kg, m_{ugf} = 220 kg, m_{fb} = 656 kg and r_m = 5.5 which gives V_o = 132.592 m^3 and V_f = 401.582 m^3. For a common bulkhead design, we let $V = V_o + V_f$ = 534.174 m^3 and D = 8.407 m to give L = 5.887 m.

For the CPS we have m_{ms} = 94,100 kg, m_r = 460 kg, m_{ugo} = 483 kg, m_{ugf} = 116 kg, m_{fb} = 101 kg and r_m = 5.88 which gives V_o = 75.709 m^3 and V_f = 210.696 m^3. For a common bulkhead design, we let $V = V_o + V_f$ = 286.405 m^3 and D = 8.407 m to give L = 1.422 m.

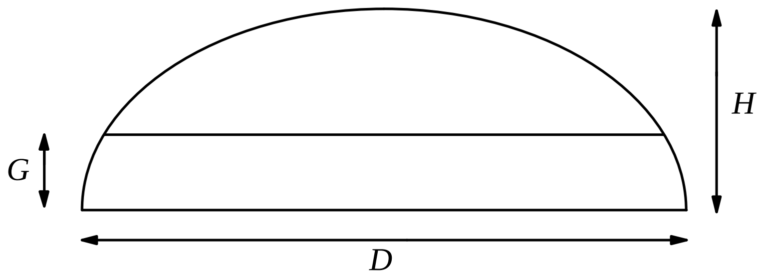

Figure 7: Clamshell Dome

For the LOX tank, we use a bishell design where a normal dome has a height G cut from a dome of height $H = D/3$ as shown in Figure 7. This reduces the common bulkhead area and requires less structural mass compared to having an upward facing bulkhead. The total volume of the LOX bishell tank in terms of D, G and H is

$$V_o = \pi D^2(2H + G^3H^2 - 3G)/6. \tag{11}$$

We solve this using Newton's method to give G = 0.688 m and 1.274 m for the LUS and CPS, respectively.

For the LM, we use four spherical tanks to hold the storable nitrogen tetroxide (N_2O_4) and Aerozine–50 (50% unsymmetrical dimethyl hydrazine (UDMH) and hydrazine (N_2H_4)). The propellant densities are d_o = 1431 kg/m^3 and d_f = 881.8 kg/m^3. For m_p = 5,092 kg and r_m = 1.6 [34], we obtain $V_o = (1+f_{ul})m_p/(d_o(1+1/r_m))$ = 2.343 m^3 and $V_f = (1+f_{ul})m_p/(d_f(1+r_m))$ = 2.376 m^3. We will use the larger volume so that all four tanks are of equal diameter $D = \sqrt[3]{3V_f/\pi}$ = 1.314 m. The cabin diameter is 2.4 m, slightly larger than the Apollo LM at 2.337 m [44]. The LCM height, not including the landing legs, is 1.265 m, compared to 1.65 m for the Apollo 11 descent stage [44].

Figure 8 shows our design assuming 0.25 m spacing between a stage engine and the bulkhead below. Dimensions of the Orion spacecraft were obtained from [29]. The vehicle height is 118.872 m, equal to the maximum allowable.

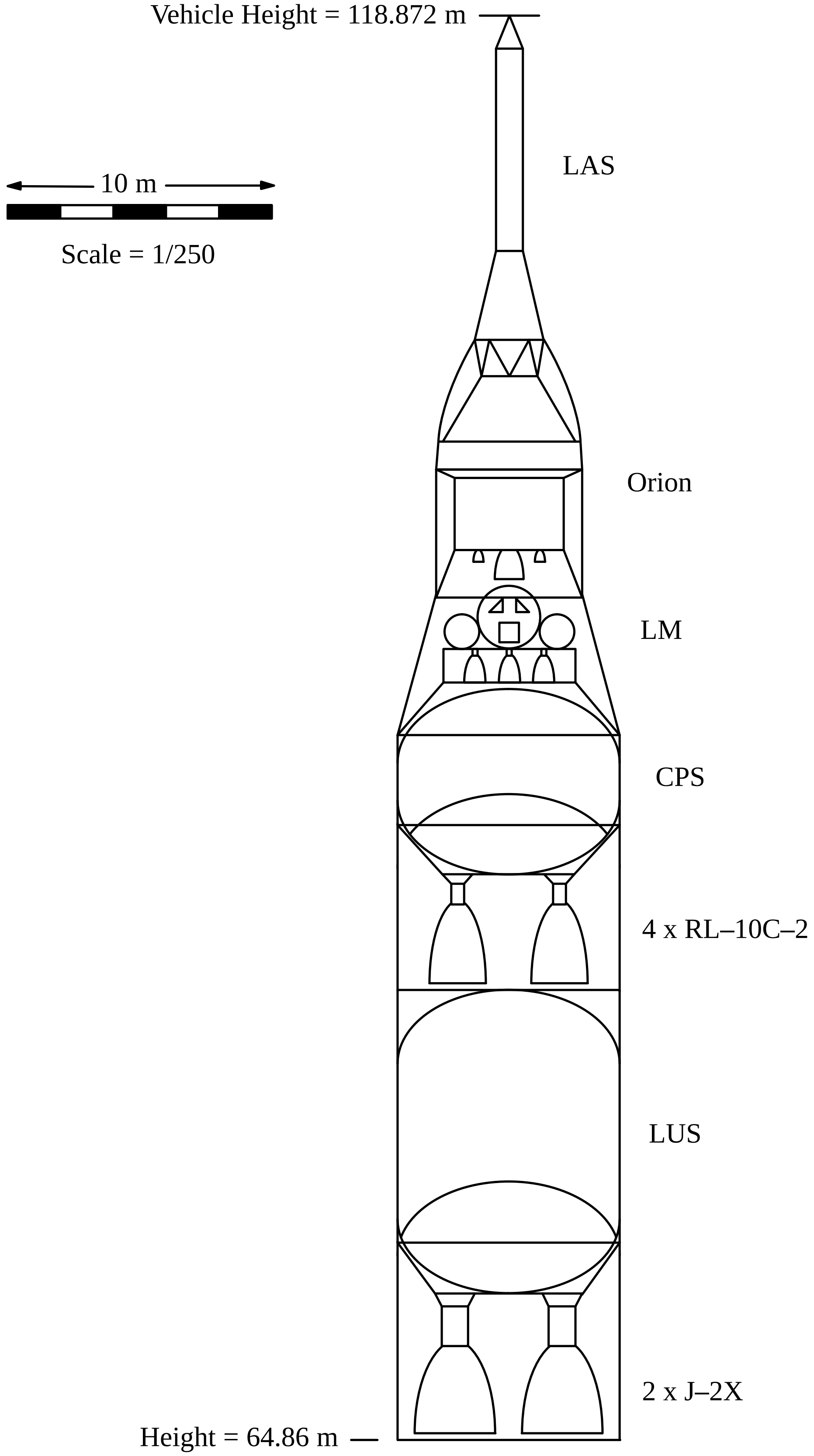

Figure 8: Large Upper Stage, Cryogenic Propulsion Stage, Lunar Lander, Orion and LAS.

III. LUNAR MISSION COST

We use the Spacecraft/Vehicle Level Cost Model [45] derived from the NASA/Air Force Cost Model (NAFCOM) database to estimate the total development and production costs for one development flight and five or ten operational flights. We multiply the FY99 amounts by 1.469 in order to obtain 2017 dollar amounts [46]. We also compare this cost to a Lunar mission which uses two 93.1 t Block IB SLS vehicles for each Lunar mission [47].

III.A SLS Block II Lunar Mission Cost

As the LUS and CPS use a common bulkhead, we increase their development and production costs by 15% to take into account the extra difficulty of this technology. As the cost model does not include solid stages, we use the Launch Vehicle Stage model, but with the calculated cost reduced by 65%. This allows the cost values to be matched to the Advanced Missions Cost Model for Rocket Missiles [48] where only the total development and production cost is given. For the LAS, we reduce its cost by 30% to take into account that it is a complex solid stage. Table 9 gives the estimated development and production costs for each element.

Table 9: SLS Block II Lunar Mission Costs

Element	Dry Mass (kg)	Quantity per mission	Development Cost $M	Production Cost 11 Missions $M	Production Cost 29 Missions $M
RSRMV	96,751	2	2,023.9	1,854.2	3,894.5
Core	101,395	1	5,933.6	3,214.5	6,751.7
LUS	11,950	1	2,105.1	897.6	1,885.2
CPS	7,796	1	1,664.3	676.4	1,420.8
LM	4,145	1	2,592.3	1,300.1	2,730.7
Orion	16,745	1	5,587.0	3,276.3	6,881.5
LAS	5,044	1	797.3	308.6	648.3
RS–25E	3,700	6	3,880.0	1,324.4	2,781.7
J–2X	2,472	2	3,108.1	437.3	918.5
RL–10C–2	301	4	976.2	184.4	387.4
Total	250,299	20	28,667.8	13,473.8	28,300.3

As the RSRMV, Orion, LAS, RS–25E, J–2X and RL–10C–2 have already or will be developed, excluding their development costs gives a total development cost of $12,497.7M. This includes 10% of the development cost or $202.1M to restart RSRMV steel segment production. The total development and production costs are $25,971.5M for 11 missions and $40,798.0M for 29 missions. Per mission costs are $1,224.9M and $975.9 for 11 and 29 missions, respectively.

III.B SLS Block IB Lunar Mission Cost

The Block IB SLS uses a standard Block I SLS, where the Delta–IV upper stage is replaced with an EUS with four RL–10C–2 engines. The first SLS launches a two stage LM into LLO with the second SLS launching Orion into LLO. Orion docks with the LM, which then performs a standard Apollo type mission. To estimate the dry mass of the LM we assume the total mass is the same as Orion in LLO of m_t = 25,848 kg. Using the Apollo 17 LM [21] we have the reference dry mass $m_{s,r}$ = 4,937 kg and reference total mass of $m_{t,r}$ = 16,448 kg. Using a simple linear model, the LM dry mass is $m_s = m_{s,r} m_t / m_{t,r}$ = 7,758 kg. The Block IB masses are obtained from [7].

Table 10: SLS Block IB Lunar Mission Costs

Element	Dry Mass (kg)	Quantity per mission	Development Cost \$M	Production Cost 11 Missions \$M	Production Cost 29 Missions \$M
RSRMV	96,751	4	2,023.9	3,152.1	6,620.7
Core	85,898	2	5416.3	4,896.5	10,284.3
EUS	10,650	2	1,718.1	1,229.4	2,582.2
LM	7,758	1	3,659.4	1,968.7	4,135.1
Orion	16,745	1	5,587.0	3,276.3	6,881.5
LAS	5,044	1	797.3	308.6	648.3
RS–25E	3,700	8	3,880.0	1,650.6	3,467.0
RL–10C–2	301	8	976.2	313.6	658.6
Total	226,847	27	24,058.2	16,795.8	35,277.7

As the RSRMV, Core, Orion, LAS, RS–25E and RL–10C–2 have already or will be developed, excluding their development costs and including RSRMV steel segment restart gives a development cost of \$5,579.9M. The total development and production costs are \$22,375.7M for 11 missions and \$40,857.6M for 29 missions. Per mission costs are \$1,526.9M and \$1,216.5 for 11 and 29 missions, respectively.

The high development costs of a new core and LUS implies that the total cost for this version of the SLS Block II is \$3,595.8M greater for 11 missions. However, as the per mission costs are about 20% less for Block II, for 29 or greater missions Block II becomes cheaper.

Note that we have not specified a launch frequency, which may effect total operations costs. A nominal two Lunar missions per year would be desirable, similar to what was achieved during the last Apollo missions. This allows sufficient time to analyse results before the next mission. This is certainly achievable with single Block II missions. Dual Block IB missions may have additional overhead costs due to requiring four launches per year.

III.C Comparison With Other SLS Block II Configurations

We investigate the development and production costs for other SLS Block II configurations that achieve 130 t or more into LEO. The dry mass and payload results were for an earlier lighter version of the LAS and SMF (8,314 kg total instead of 9,027 kg) which were ejected together at an earlier time of 330 s. The dry mass model of the LUS used the separate tank design of [8] where α = 0.65554. The LUS puts the payload directly into a 200 km orbit inclined at 28.45° instead of 32.55°. Details of the trajectory simulations and the data used can be found in [37].

Configuration SLS1C6J2.1 uses RSRMV boosters with a six engine core, SLS2C4J2.2 uses LOX/RP–1 boosters with two F–1B engines each and a four engine core, SLS3C4J2.2 uses LOX/RP–1 boosters with three staged combustion AJ1E6 engines each and a four engine core and SLS4C5J2.2 uses advanced HTPB composite case solid boosters with a five engine core. For the F–1B dry mass, we assume that it is the same as the F–1A [49]. For the AJ1E6 dry mass, we assume that it is the same as the RD–180 [50]. Tables 11 to 14 gives the development and production costs of the four different versions.

Table 15 gives the total development and production costs excluding the development costs of elements that have already or will be developed (RSRMV boosters, four engine core, RS–25E and J–2X). The RSRMV steel segment restart cost is included for SLS1C6J2.1. Per flight costs are also given.

Table 11: SLS1C6J2.1 (137.0 t to LEO)

Element	Dry Mass (kg)	Quantity per flight	Development Cost $M	Production Cost 11 Flights $M
RSRMV	96,751	2	2,023.9	1,854.2
Core	101,395	1	5,933.6	3,214.5
LUS	20,642	1	2,472.4	1,120.7
RS–25E	3,700	6	3,880.0	1,324.4
J–2X	2,472	2	3,108.1	437.3
Total	224,960	12	17,418.0	7,951.1

Table 12: SLS2C4J2.2 (133.2 t to LEO)

Element	Dry Mass (kg)	Quantity per flight	Development Cost $M	Production Cost 11 Flights $M
Pyrios AB	106,754	2	6,104.1	5,654.3
Core	100,775	1	5,913.6	3,201.5
LUS	16,158	1	2,160.8	953.0
F–1B	8,618	4	6,177.3	1,699.4
RS–25E	3,700	4	3,880.0	971.0
J–2X	2,472	2	3,108.1	437.3
Total	238,477	14	27,343.9	12,916.5

Table 13: SLS3C4J2.2 (136.2 t to LEO)

Element	Dry Mass (kg)	Quantity per flight	Development Cost $M	Production Cost 11 Flights $M
Liquid AB	101,500	2	5,937.0	5,468.5
Core	100,775	1	5,913.6	3,201.5
LUS	16,097	1	2,156.4	950.6
AJ1E6	5,393	6	4,773.4	1,699.5
RS–25E	3,700	4	3,880.0	971.0
J–2X	2,472	2	3,108.1	437.3
Total	229,937	16	25,768.5	12,728.4

Table 14: SLS4C5J2.2 (144.1 t to LEO)

Element	Dry Mass (kg)	Quantity per flight	Development Cost $M	Production Cost 11 Flights $M
Solid AB	96,615	2	2,022.3	1,852.5
Core	101,395	1	5,933.6	3,214.5
LUS	18,912	1	2,356.2	1,057.6
RS–25E	3,700	5	3,880.0	1,151.8
J–2X	2,472	2	3,108.1	437.3
Total	223,094	11	17,300.2	7,713.7

Table 15: SLS Block II Costs in $M (11 Flights)

Configuration	Total Flights	Per Flight
SLS1C6J2.1	16,559.4	722.8
SLS2C4J2.2	27,358.7	1,174.2
SLS3C4J2.2	25,595.2	1,157.1
SLS4C5J2.2	18,025.8	701.2

The cheapest option for the SLS Block II vehicle is the configuration we have chosen in this paper, which uses a new six engine core, existing RSRMV boosters and a two J–2X engine LUS. The next cheapest is using advanced solid boosters, which costs $1.5B (9%) more for 11 flights, respectively. Per flight rates are only 3% cheaper. Using liquid boosters costs 53% to 66% more due to the high development and production costs of the booster stages and engines.

IV. Future Improvements

There are a number of options for increasing the performance of the Block II vehicle as well as the performance of the overall Lunar mission. The first restriction that must be overcome is the vehicle height, as this currently limits overall vehicle performance for single launch Lunar missions. The current SLS launch mount uses vehicle support posts (VSP) [51] to mount the RSRMV boosters. These were not used for the Space Shuttle. Eliminating these posts would provide 1.727 m of additional vehicle height, at the expense of having to modify the launch mount as well as the location of the core umbilicals on the launch tower.

The RL–10B–2 engine has a stowed length of 2.197 m [25], compared to a length of 3.767 m that we have used in our design. This would allow an increase of 1.57 m in tank length as well as increased performance due to a higher Isp and shorter interstage. There is additional risk though from nozzle deployment failures. However, the RL–10B–2 has flown 35 times in the Delta IV launch vehicle without any deployment failures. Also, the increase in delta–V due to a single nozzle deployment failure is only 1.1%, which we have included in the mission design.

Replacing the LAS with the max launch abort system (MLAS) [52] would provide much larger increases in tank length, of up to 12.2 m, which far exceeds what is required of at least 2 m. MLAS was partially developed and performed one successful flight test. Another alternative is to replace the Orion spacecraft with a Block II configuration with a 3.18 m diameter headlight shaped capsule that can carry four astronauts, a separate orbital module that would provide a much larger internal volume then available in Orion and an MLAS like abort system. This could reduce the 9,887 kg mass of Orion to 5,870 kg (similar to the Apollo command module), which would allow significant performance improvements. Not including any reduction in the SM mass or increase in mass to LEO, this would increase the LM cargo mass from 509 kg to 2,296 kg and allow Orion plane changes up to 12.1° (Apollo had a maximum plane change of 3.9°). This should allow much higher lattitudes to be reached with perhaps stay times of up to 14 days. The LM could be landed with wheels and act as its own pressurised rover, allowing a total traverse distance of over 200 km. To obtain the equivalent increase in performance using Orion, we would need to increase the total mass after LUS separation from 142,195 kg to 162,960 kg.

V. Conclusions

We have presented a solution for achieving a Lunar landing mission using only one SLS Block II launch vehicle. To achieve this we use the existing RSRMV solid rocket boosters, the four engine core of the Block I vehicle modified to use six RS–25E engines, a dual J–2X LUS and a quad RL–10C–2 CPS. Due to vehicle height limitations, the LUS and CPS must use a common bulkhead design, which has the additional benefit of increased payload performance. There are also many options available to increase performance.

Compared to other Block II configurations, we have shown that this configuration is the cheapest in terms of total development and production costs. A dual Block IB Lunar mission is $3.6B cheaper for 11 Lunar missions. However, per flight costs of using a single Block II mission are 20% less, which for 29 or more missions would lead to lower overall cost. For future Mars missions, the 140 t capability of this SLS Block II version gives a significant advantage over the 93 t capability of SLS Block IB, requiring fewer flights for each mission and thus simplifying overall mission complexity.

By going to the Moon, which is an extremely difficult exercise as demonstrated by Apollo, the experience gained in actual beyond Earth exploration can be regained from that lost when the

Apollo program was prematurely curtailed. Lunar exploration also allows regular missions to be performed, compared to having to wait over two years between each Mars mission. With the experience gained in regular Lunar missions, the much greater effort and complexity required to go to Mars can then be tackled with much greater confidence.

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REFERENCES

[1] Senate and House of Representatives of the United States of America in Congress, "An act to authorize the programs of the National Aeronautics and Space Administration for fiscal years 2011 through 2013, and for other purposes," 111th Congress, Public Law 111–267, Washington, DC, USA, Oct. 2010.

[2] M. Ryschkewitsch, "Human space flight crew and cargo status and challenges," *AIAA Complex Aerospace Systems Exchange Conf.*, Pasadena, CA, USA, Sep. 2012.

[3] T. May, "NASA's Space Launch System overview," *SLS Industry Day*, Michoud Assembly Facility, New Orleans, LA, USA, Oct. 2012.

[4] D. Sauvageau and A. Corliss, "Advanced booster for NASA Space Launch System (SLS)," *IAF Int. Astronautical Congress*, Naples, Italy, IAC–12–D2.8.6, Oct. 2012.

[5] S. Cook, K. Doering, A. Crocker and R. Bachtel, "Enabling an affordable, advanced liquid booster for NASA's Space Launch System," *IAF Int. Astronautical Congress*, Naples, Italy, IAC–12–D2.8.10, Oct. 2012.

[6] C. Crumbly, "NASA's Space Launch System: Partnering for tomorrow," *ASEE Engineering Research Council Conf.*, Washington, DC, USA, Mar. 2013.

[7] B. Donahue and S. Sigmon, "The Space Launch System capabilities with a new large upper stage," *AIAA Space Conf. and Exhib.*, San Diego, CA, USA, Sep. 2013.

[8] B. Donahue and J. Bridges, "The Space Launch System capabilities for enabling crewed Lunar and Mars Exploration," *IAF Int. Astronautical Congress*, Naples, Italy, IAC–12–D2.8.7, Oct. 2012.

[9] Boeing, "The Space Launch System and the pathway to Mars," Annual Meeting of the Lunar Exploration Analysis Group, Laurel, MD, USA, Oct. 2014.

[10] S. S. Pietrobon, "Space Launch System trajectory simulations," Dec. 2013. http://www.sworld.com.au/steven/space/sls/index.html

[11] ATK, "Space propulsion products catalog," OSR No. 12–S–1902, Aug. 2012.

[12] T. Williams and S. Cannon, "Ares I first stage design, development, test and evaluation," *IAF Int. Astronautical Congress*, Valencia, Spain, pp. 7856–7870, Oct. 2006.

[13] NASA, "Space Launch System Program Imagery System Requirements Document (ISRD)," SLS–SPEC–167, Jan. 2013.

[14] NASA, "National Space Transportation System reference," June 1988.

[15] E. W. Weisstein, "Circle–Circle Intersection," MathWorld: A Wolfram Web Resource. http://mathworld.wolfram.com/Circle–CircleIntersection.html

[16] M. Kynard, "NASA's Space Launch System engines overview," *SLS Industry Day*, Michoud Assembly Facility, New Orleans, LA, USA, Oct. 2012.

[17] R. A. O'Leary and J. E. Beck, "Nozzle design," *Threshold*, Pratt & Whitney Rocketdyne, Spring 1992.

[18] Saturn Flight Evaluation Working Group, "Saturn V launch vehicle flight evaluation report AS–509 Apollo 14 mission," NASA George C. Marshall Space Flight Center, MPR–SAT–FE–71–1, April 1971.

[19] S. S. Pietrobon, "Analysis of propellant tank masses," Submitted to Review of U.S. Human Space Flight Plans Committee, 6 July 2009.

[20] MSFC, "J–2 engine," IND B1411E, I–E–J 005C, 26 May 1968.

[21] R. W. Orloff and D. M. Harland, "Apollo: The definitive sourcebook," Springer–Praxix Publishing, Chichester, UK, 2006.

[22] M. Schaffer, "A study of CPS stages for missions beyond LEO," *Future In–Space Operations Working Group*, Atlanta, GA, USA, May 2012.

[23] J. F. LeBar and E. C. Cady, "The advanced cryogenic evolved stage (ACES) – A low–cost, low–risk approach to space exploration launch," *Space 2006*, San Jose, CA, USA, AIAA 2006–7454, Sep. 2006.

[24] United Launch Alliance, "Delta IV launch services user's guide," June 2013.

[25] Pratt & Whitney Rocketdyne, "RL–10B–2," 2009.

[26] F. Zegler, B. F. Kutter and J. Barr, "A commercially based Lunar architecture," *AIAA Conf. & Exposition*, Pasadena, CA, USA, AIAA 2009–6567, Sep. 2009.

[27] R. B. Tierney, "NASA's ground rules and assumptions," NASASpaceflight.com forum, 10 Oct 2009. http://forum.nasaspaceflight.com/index.php?topic=16163.msg372945#msg372945

[28] J. Calvignac, L. Dang, T. L. Tramel and L. Paseur, "Design and testing of non–toxic RCS thrusters for second generation reusable launch vehicle," *AIAA/ASME/SAE/ASEE Joint Propulsion Conf. & Exhibit*, AIAA 2003–4922, Huntsville, AL, USA, July 2003.

[29] NASA, "Orion quick facts," FS–2014–08–004–JSC, Aug. 2014.

[30] NASA, "Orion flight test: Exploration flight test 1," NP–2014–11–020–JSC, Dec. 2014.

[31] NASA, "National space transportation system reference: Systems and facilities," Vol. 1, June 1988.

[32] Airbus Defence and Space, "Space propulsion: Chemical bi–propellant thruster family," 2013. http://cs.astrium.eads.net/sp/brochures/bipropellant–thrusters/bipropellant–thrusters.pdf

[33] "International docking system standard (IDSS) interface definition document (IDD)," 21 Sep. 2010.

[34] NASA, "Space vehicle design criteria (chemical propulsion): Liquid rocket engine nozzles," SP–8210, July 1976.

[35] Aerojet Rocketdyne, "R–4D 490 N (110–lbf) bipropellant rocket engine," 24 May 2005.

[36] A. F. J. Abercromby, S. P. Chappell and M. L. Gernhardt, "Multi–mission space exploration vehicle (MMSEV): An EVA and robotics worksystem for multiple destinations," *Workshop on Golden Spike Human Lunar Expeditions*, 6008, Houston, TX, USA, Oct. 2013.

[37] S. S. Pietrobon, "Space Launch System Trajectory Simulations," http://www.sworld.com.au/steven/space/sls/

[38] S. S. Pietrobon, "Saturn V simulation program," June 1997. http://www.sworld.com.au/steven/space/apollo/sim/

[39] W. Gellert, S. Gottwald, M. Hellwich, H. Kästner and H. Küstner, “The VNR Concise Encyclopedia of Mathematics,” 2nd ed., Van Nostrand Reinhold: New York, 1989.

[40] NASA JSC, “STS–89 flight requirements document – Level A groundrules and constraints,” NSTS–17462–89 Appendix A, Apr. 1997.

[41] NASA, “Space Launch System Program (SLSP) logistics support analysis (LSA) report,” SLS–RPT–108 Version 1, Apr. 2013.

[42] NASA, “SLS core stage flight article transportation plan,” SLS–PLAN–179, Draft for PDR, May 2013.

[43] C. Selph, “Input utility for specific impulse code.”

[44] Grumman, “Apollo news reference,” 1968.

[45] GlobalSecurity.org, “Spacecraft/Vehicle Level Cost Model,” http://www.globalsecurity.org/military/intro/reference/calc/SVLCM.htm

[46] Coin News Media Group, “US inflation calculator,” http://www.usinflationcalculator.com/

[47] J. Connolly, “Human Lunar exploration architectures,” *Annual Meeting of the Lunar Exploration Analysis Group*, Greenbelt, MD, USA, Oct. 2012.

[48] Federation of American Scientists, “Advanced missions cost model,” http://fas.org/news/reference/calc/AMCM.htm

[49] NASA, “Liquid engine comparison,” Slide PD24, 12 Jan. 1992.

[50] Pratt & Whitney, “The RD–180 engine,” Fact Sheet, Mar. 2004.

[51] C. Gebhardt, “NASA outlines SLS Mobile Launcher umbilical plans,” Nov. 2012. http://www.nasaspaceflight.com/2012/11/nasa–sls–mobile–launcher–umbilical–plans/

[52] A. P. Taylor, C. Kelley, E. Magner, D. Peterson, J. Hahn and D. Yucjnovicz, “The NASA MLAS flight demonstration – A review of a highly successful test,” *AIAA Conf. & Exposit.*, Anaheim, CA, USA, AIAA 2010–8706, Aug.–Sep. 2010.

Acronym List

AB	Advanced Boosters
CM	Command Module
CPM	Crew and Propulsion Module
CPS	Cryogenic Propulsion Stage
ESM	European Service Module
EOI	Earth Orbit Insertion
EUS	Exploration Upper Stage
GH_2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
HTPB	Hydroxyl Terminated Polybutadiene
KSC	Kennedy Space Center
LAS	Launch Abort System
LCM	Landing and Cargo Module
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LM	Lunar Module
LOI	Lunar Orbit Insertion

LOX	Liquid Oxygen
LUS	Large Upper Stage
maxQ	Maximum Dynamic Pressure
MLAS	Max Launch Abort System
MMSEV	Multi–Mission Space Exploration Vehicle
MPCV	Multi Purpose Crew Vehicle
NAFCOM	NASA/Air Force Cost Model
NASA	National Aeronautics and Space Administration
N_2H_4	Hydrazine
N_2O_4	Nitrogen Tetroxide
NBP	Nominal Boiling Point
OMS	Orbital Manoeuvring System
PC	Plane Change
PD	Powered Descent
PDI	Powered Descent Initiation
RP–1	Rocket Propellant Kerosene
RPL	Rated Power Level
RSRM	Reusable Solid Rocket Motor
RSRMV	Reusable Solid Rocket Motor Five Segment
SLA	Spacecraft Launch Adaptor
SLS	Space Launch System
SMF	Service Module Fairing
SRM	Solid Rocket Motor
TAD	Transposition and Docking
TCM	Trajectory Correction Manoeuvre
TEI	Trans Earth Injection
TL	Trans Lunar
TLI	Trans Lunar Injection
UDMH	Unsymmetrical Dimethyl Hydrazine
VAB	Vehicle Assembly Building
VSP	Vehicle Support Posts