Analysis of Propellant Tank Masses

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***Abstract* — For the Direct 3.0 architecture the dry weight of the Earth departure stage (EDS) is critical for the architecture to work. We provide an independent analysis using actual data of previous liquid hydrogen and oxygen stages to show that the Direct 3.0 EDS dry stage mass is theoretically possible. However, we recommend a standard 15% margin be added to the EDS mass due to the combination of new technologies and techniques the stage will use.**

***Index Terms* — Jupiter, Direct, EDS**

I. INTRODUCTION

IN the Direct 3.0 architecture the Jupiter 246 launch vehicle is first used to launch an Earth departure stage (EDS) into low Earth orbit. A second Jupiter then launches the Altair Lunar lander and Orion crew exploration vehicle. The Altair/Orion stack then docks with the EDS which then fires to send Altair/Orion to the Moon.

A controversial aspect of the Direct architecture has been the dry weight of the EDS. The current design has a dry weight of 11,238 kg (similar to the Saturn V S–IVB) for a propellant mass of 175,519 kg (65% greater than the propellant carried in the S–IVB).

In order to understand how Direct achieves such a low mass we first analyse the stage masses of all previously constructed liquid hydrogen and oxygen stages. This allows us to obtain a very simple model of stage mass as a function of propellant mass. We then apply two key technologies used by Direct, a common bulkhead and the use of Aluminium Lithium alloy. The resultant model almost perfectly matches the stage masses that Direct have obtained.

However, the Direct EDS uses a number of other new technologies and techniques that have the potential to increase stage mass. For this reason we recommend a standard 15% margin be added to the Direct EDS mass. As the Direct architecture has quite large margins, this increase can be easily absorbed.

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II. STAGE MASS MODELS

In [1] the EDS dry mass is given as 11,238 kg with a gross propellant mass of m_p = 175,519 kg. The stage also contains 450 kg of RCS propellant and six RL–10B–2 engines of 301 kg each [2]. Therefore the empty (less propellant and engines) stage mass is m_s = 11,238 – 6 × 301 = 9,432 kg. This gives a stage (less engines) to propellant mass ratio m_s/m_p = 5.37%.

Table 1 gives the total mass m_t, propellant mass m_p, dry mass $m_f = m_t - m_p$, engine mass m_e, dry mass less engine mass $m_s = m_f - m_e$ and ratio m_s/m_p of all liquid hydrogen and oxygen stages. Most of the data for the table was obtained from [2]. The S–II and S–IVB data was obtained from [3]. In Table 2 we give proposed stage mass data for Direct 2.0 [4], Direct 3.0 [1], Ares–I and Ares–V [5].

In Figure 1 we plot empirical data of m_s/m_p in percent versus m_p on a log–log graph for the liquid hydrogen/oxygen stages in Tables 1 and 2. Using the data from Table 1 (excluding the Space Shuttle External Tank), we averaged all the data to obtain the formula

$$m_s = 0.19 m_p^{0.848}. \tag{1}$$

where m_s and m_p are in tonnes (1 t = 1000 kg). As the Direct 3.0 EDS uses a common bulkhead design we rescale our model so that it passes though the S–II point. Doing so, we obtain the formula

$$m_s = 0.1583 m_p^{0.848}. \tag{2}$$

We see that this line passes close to other stages with a common bulkhead.

III. EFFECT OF TANK MATERIAL

The yield stress for a structure is given by

$$Y_s = \frac{F_s}{A_s} \tag{3}$$

where F_s is the force (in Newtons) that causes the material to yield and A_s is the cross sectional area (in m^2) the force is applied over. As F_s is directly proportional to A_s then Y_s is a constant, only depending on the material being used as well as its temperature. As yield stress is force over area, the unit for Y_s is in Pascals (or more commonly kilo Pascals, kPa or mega Pascals, MPa). The mass of a structure is given by

$$m_s = L_s A_s d_s \tag{4}$$

where L_s is the length of the structure and d_s is the density in kg/m^3. Using (3), we can rewrite the above as

$$m_s = L_s F_s \frac{d_s}{Y_s}. \tag{5}$$

Assuming that L_s and F_s are constant for a structure, then a structure's mass is directly proportional to the density of the material and inversely proportional to its yield strength. Another way of defining a materials property is with its specific yield strength $\lambda_s = Y_s/d_s$ (which has units m^2/s^2). That is, structure mass is inversely proportional to specific yield strength.

Table 1: Liquid Hydrogen/Oxygen Stage Masses

Stage	m_t (kg)	m_p (kg)	m_f (kg)	m_e (kg)	m_s (kg)	m_s/m_p (%)
S–IV	50,580	45,360	5,220	6×131	4,430	9.77
S–IVB	117,774	106,189	11,585	$1\times1{,}438$	10,147	9.56
S–II	487,789	452,308	35,481	$5\times1{,}438$	28,291	6.25
Centaur C	15,558	13,604	1,954	2×131	1,692	12.44
Centaur D/E	16,258	13,627	2,631	2×131	2,369	17.38
Centaur I	15,558	13,880	1,678	2×141	1,396	10.06
Centaur II	18,833	16,778	2,055	2×141	1,773	10.57
Centaur IIA	19,073	16,778	2,295	2×168	1,959	11.67
Centaur G	23,877	21,105	2,772	2×141	2,490	11.80
Centaur 3A	18,710	16,805	1,905	1×167	1,738	10.34
Centaur 3B	22,956	20,829	2,127	2×167	1,793	8.61
Centaur V1	22,825	20,799	2,026	1×167	1,859	8.94
Centaur V2	23,050	20,800	2,250	2×167	1,916	9.21
Ariane H–8	9,678	8,221	1,457	1×149	1,308	15.91
Ariane H–10	12,000	10,400	1,600	1×155	1,445	13.89
Ariane H–10+	12,310	10,740	1,570	1×155	1,415	13.18
Ariane H–155	170,800	158,100	12,700	1×625	12,075	7.64
Ariane H–173	186,000	173,300	12,700	1×811	11,889	6.86
Ariane ESC–A	16,500	14,400	2,100	1×155	1,945	13.51
Ariane ESC–B	27,500	24,100	3,400	1×280	3,120	12.95
CZ H–8	10,500	8,500	2,000	4×236	1,056	12.42
CZ H–18	21,000	18,200	2,800	2×550	1,700	9.34
Japan H–1–3	10,600	8,800	1,800	1×245	1,555	17.67
Japan H–2–1	98,100	86,200	11,900	$1\times1{,}714$	10,186	11.82
Japan H–2–2	16,700	14,000	2,700	1×242	2,458	17.56
Japan H–2A–1	113,600	100,000	13,600	$1\times1{,}800$	11,800	11.80
Japan H–2A–2	19,600	16,600	3,000	1×269	2,731	16.45
GSLV–3	14,600	12,400	2,200	1×282	1,918	15.47
Delta 3–2	19,300	16,824	2,476	1×301	2,175	12.93
Delta IV–1	226,400	199,640	26,760	$1\times6{,}597$	20,163	10.12
Delta IV–2	24,170	21,320	2,850	1×301	2,549	11.96
Delta IVH–2	30,708	27,220	3,488	1×301	3,187	11.71
Energia	905,000	820,000	85,000	$4\times3{,}450$	71,200	8.68
Shuttle ET	747,974	721,212	26,762	0	26,762	3.71

Table 2: Proposed Liquid Hydrogen/Oxygen Stage Masses

Stage	m_t (kg)	m_p (kg)	m_f (kg)	m_e (kg)	m_s (kg)	m_s/m_p (%)
Ares–I US	151,400	138,000	13,400	$1\times 2{,}472$	10,928	7.92
Ares–V Core	1,761,253	1,603,630	157,623	$6\times 6{,}747$	117,141	7.30
Ares–V EDS	278,460	254,193	24,267	$1\times 2{,}472$	21,795	8.57
Direct 2.0 Core	808,687	735,360	73,327	$3\times 6{,}747$	53,086	7.22
Direct 3.0 Core	802,255	735,360	66,895	$4\times 3{,}177$	54,187	7.37
Direct 2.0 EDS	381,500	359,065	22,435	$2\times 2{,}472$	17,491	4.87
Direct 3.0 EDS	186,757	175,519	11,238	6×301	9,432	5.37

A key technology used by the Direct EDS is Aluminium Lithium 2195 alloy, as used by the Space Shuttle External Tank. This alloy has a composition of 1.0% Lithium, 4.0% Copper, 0.4% Magnesium, 0.4% Silver, and 0.12% Zirconium by weight [6]. Al, Li, Cu, Mg, Ag and Zr have densities of 2698, 534, 8933, 1738, 10501, and 6506 kg/m^3, respectively at 0 C and 100 kPa [7]. This gives a density of 2685 kg/m^3 (the actual density at 25 C will be slightly less).

Al 2219 has a density of d_s = 2840 kg/m^3 [8]. Thus, the decreased density of AlLi 2195 results in 5.5% decreased weight. Also, AlLi 2195 has a greater yield strength Y_s than Al 2219. Table 3 gives yield strength Y_s and specific yield strength $\lambda_s = Y_s/d_s$ for these alloys at varying temperatures [9].

Table 3: Aluminium properties.

Alloy	Al 2219	AlLi 2195
Density (kg/m^3)	2840	2685
Specific Yield Strength at 20 K (m^2/s^2)	170240	226680
Specific Yield Strength at 80 K (m^2/s^2)	163000	219690
Specific Yield Strength at 25 C (m^2/s^2)	136700	194270
Yield Strength at 20 K (MPa)	483.5	608.6
Yield Strength at 80 K (MPa)	462.9	589.9
Yield Strength at 25 C (MPa)	388.2	521.6

At room temperature (25 C) AlLi 2195 is 24.6% stronger than Al 2219. This implies that for the same size upper stage, stage mass can be reduced by 29.6%. At cryogenic temperatures this total reduces to 24.9% at 20 K (for LH2) and 25.8% at 80 K (for LOX).

Assuming an average reduction of 26%, we apply this to the common core model to obtain the formula

$$m_s = 0.1171 m_p^{0.848}. \tag{6}$$

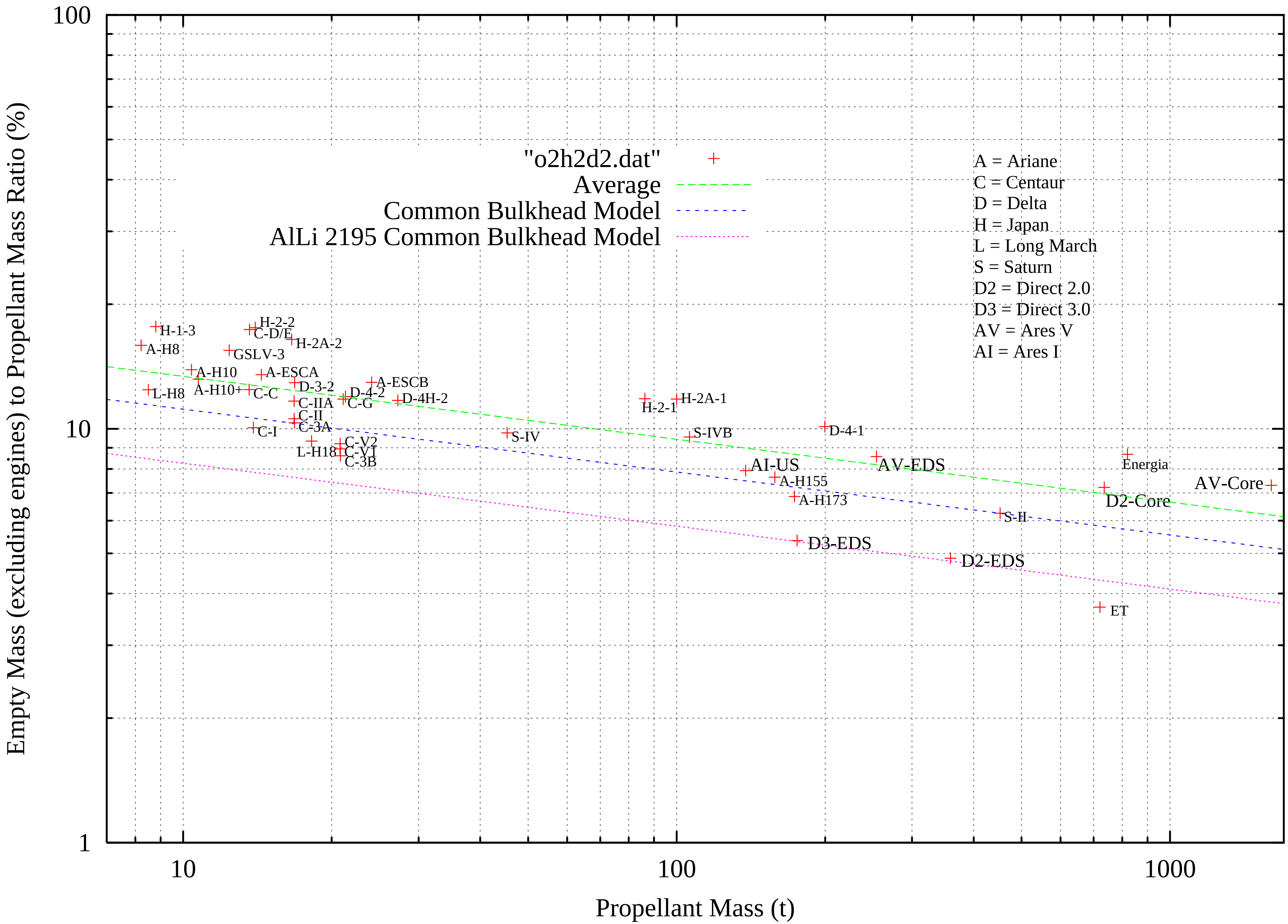

Figure 1: m_s/m_p ratio versus m_p.

This line is also plotted in Figure 1. We see that the line pass very close to the Direct 2.0 and 3.0 EDS values. This indicates that it is theoretically possible for the Jupiter EDS to meet its dry mass target. However, we have a number of concerns that could add to this dry mass.

(1) It has been over 40 years since the United States has constructed a wide body common bulkhead. This was the 10 m diameter Saturn V S–II. Current experience is limited to the 3 m diameter Centaur upper stage. As reported in [10], there were a large number of problems encountered in constructing the common bulkhead. This is due to the need for the bulkhead to be constructed in three layers without any air gaps between the layers. There is a top and bottom metal layer and a middle insulating layer to prevent the hotter liquid oxygen at around 80 K from boiling the liquid hydrogen at around 20 K. The Jupiter EDS is 8.4 m in diameter and similar or new problems may also be encountered.

(2) The Jupiter EDS is based on the integrated common evolved stage (ICES) [11]. One of the main characteristics of the ICES stage is that the common bulkhead is in an unusual configuration. Normal cryogenic stages with the liquid oxygen tank at the base have the common bulkhead facing down. This minimises the area of the liquid oxygen tank and thus the overall stage mass as the liquid oxygen is far heavier than the liquid hydrogen (by a ratio of 5 or 6 to 1). The ICES stage has the common bulkhead facing up, minimising the area of the liquid hydrogen tank. This minimises the amount of propellant loss due to heating, as the EDS is required to spend up to four days in LEO waiting for Orion/Altair to dock. Thus EDS mass is slightly increased compared to traditional designs, but for long duration stays, the mass is reduced overall.

(3) Another unusual aspect of the ICES design is the use of a sump in the common bulkhead. As the common bulkhead is facing down, traditional designs have feed lines external to the tank. With the bulkhead facing up in ICES and to minimise stage mass and input heating a sump is incorporated into the common bulkhead. This introduces complex curves into the common bulkhead which may result in further difficulty in constructing the bulkhead.

The above technologies are all worth pursuing as they can passively reduce boiloff to only 0.1% per day [11]. However, there are a number of risks involved so we recommend that the EDS mass (less engines) is increased by the standard amount of 15% from 9,432 kg to 10,847 kg. This consequently reduces propellant mass by 1415 kg to 174,104 kg. This increases the stage mass ratio from 5.37% to 6.23%. We study the effect of this mass increase on TLI payload in the next section.

IV. Rocket Equation

We can relate the cargo mass m_c to the propellant mass m_p with the rocket equation

$$v_d = v_e \ln\left(1 + \frac{m_p - c_{bo}(m_p + m_{pu})}{m_d + m_{pu} + m_{prcs} + m_c}\right) \tag{7}$$

where v_d is the change in speed, v_e is the engine exhaust speed (equal to ISP in seconds multiplied by g = 9.80665 m/s^2), m_p is the propellant mass (including reserve propellant), m_d is the dry mass of the stage, m_{prcs} is the RCS propellant, m_{pu} is the unusable residual propellant mass and c_{bo} is the boiloff fraction. That is, we have

$$\begin{aligned} m_c &= m_p(1 - c_{bo})\beta - m_{pu}(1 - c_{bo}\beta) - m_d - m_{prcs} \\ &= c_p m_p - c_{pu} m_{pu} - m_d - m_{prcs}. \end{aligned} \tag{8}$$

where $\beta = 1/(\exp(v_d/v_e) - 1)$.

As an example of a typical value for c_p and c_{pu} we have $v_d = 1.01 \times 3215 = 3247.2$ m/s for TLI injection [1] (this TLI value takes into account the gravity losses from using only four of the six RL–10B–2 engines with a 1% performance reserve), v_e = 4565 m/s for the RL–10B2 engine [12], and c_{bo} = 2.82% [1]. This gives c_p = 0.9374 and c_{pu} = 0.9728.

For the EDS TLI stage, we have that $m_{pd} = m_p + m_d$, m_{pu} and m_{prcs} are fixed in value. Thus, from (8) we have

$$m_c = c_p m_{pd} - c_{pu} m_{pu} - m_{prcs} - m_d(1 + c_p). \tag{9}$$

That is, for every kg that m_d increases, payload mass decreases by $1+c_p$ = 1.9374 kg. Thus, for a 1415 kg dry mass increase in the EDS, TLI mass decreases by 2741 kg. However, the Direct 3.0 architecture has a 79,053 kg TLI capability [1]. This would be reduced to 76,312 kg, 7.3% above NASA's requirement of 71,100 kg [5]. In fact EDS dry mass (less engines) could be increased by 4105 kg to 13,537kg (propellant mass decreases to 171,414 kg) to give a mass ratio of 7.9%. This would be above the line for the common bulkhead model using Aluminium 2219.

V. Conclusions

The Jupiter 3.0 EDS upper stage has the smallest dry mass (less engines) to propellant ratio of any existing stage of only 5.37%. By using empirical data we obtain models that show that this ratio is theoretically possible. Key to achieving that ratio are the use of a common bulkhead and Aluminium Lithium alloy. Due to the use of new techniques and methods, we recommend that the EDS mass be increased by 15%. Despite this increase, the Jupiter launch vehicle has sufficient margin so that NASA's requirement of 71.1 t TLI mass is exceeded by 7.3%. In fact, Direct's EDS dry mass (less engines) could increase by 43.5% and still meet NASA's requirement of 71.1 t TLI mass.

The empirical models we have obtained can also be used for other cryogenic stages. By plotting a proposed stage in Figure 1, it will be easily seen if a stage is too pessimistic or optimistic.

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