Aerospace Design Project 2
Initial Concept Report
Team 17

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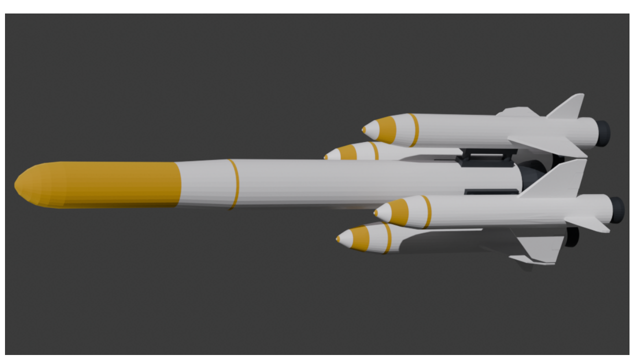
Abstract

This report introduces a novel launcher concept designed to deliver payloads to Geostationary Transfer Orbit (GTO) and Low Earth Orbit (LEO) more cost-effectively than existing designs by implementing a multi-objective launcher. The concept allows for various configurations to be constructed within a family of launchers, enhancing modularity and adaptability. A detailed explanation of the methodology used to calculate efficiency values is provided. The report conducts a comprehensive analysis of existing launcher designs and presents an extensive literature review, examining launcher configurations, various engine types, and their efficiencies. Additionally, the report explores potential solutions for reusable booster stages, considering their feasibility and impact on cost reduction. The findings indicate that, despite its complexity, the proposed launcher concept demonstrates greater cost-effectiveness and structural efficiency than contemporary designs.



Contents

1 Introduction

Designing a sustainable and impactful launcher family and a sufficient primary launcher is an immense challenge. It requires an in-depth analysis of modern designs. Due to their complexity, one of the major issues in modern-day launchers is their lack of ability to do multi-objective missions. This report aims to provide a primary concept rocket launcher to have the capability of bringing two payloads, $21,000$kg and $10,500$kg to Low Earth Orbit (LEO) and Geostationary Transfer Orbit (GTO) respectively, within a single mission [1] The concept launcher is designed to be as sustainable as possible, using reusable stages. This report also provides a possible secondary launcher configuration which can deliver a payload into the low-Earth orbit, for faster missions. Lastly, the report provides design concepts for fly-back booster stages for partial reusability.

Literature Review: In recent years, the development of Rideshare by NASA SMD has provided research and applications on providing secondary and tertiary (auxiliary) payloads into orbits with similar altitudes using tugs such as SHERPA. As provided in the SMD Rideshare 101 document [2], the SHERPA (tugs) can only reduce the altitude of the payload, meaning the current Rideshare practices cannot transport a payload from LEO to GTO. Another issue highlighted by this report is the SHERPA's inability to manoeuvre high-mass payloads, which is crucial in many missions such as the one analysed in this report. The purpose of the concept launcher is to tackle the issue that the SMD Rideshare is not suitable. Additionally, it provides important information regarding current-day technology, allowing further research to be built on top of it.

The dual payload launcher poses challenges such as increasing the complexity of the mission and the propellant mass, limiting the capacity of the payload and increasing the launcher size. However, it will achieve significant cost savings by addressing two orbital requirements in one mission and therefore increasing efficiency and maximising the launcher capabilities, with a minimal ΔV value.

Report Struture: This report is structured as follows: Section 2 provides theory and equations regarding the efficiency calculations that are used in the following sections to analyse different launcher configurations. Section 3 analyses the efficiencies and capabilities of 3 existing launchers: Ariane 5, Pegasus XL launch system, and Stratolaunch-based system using the equations introduced in the previous section. Next, in Section 4, the primary concept launcher is introduced, giving information regarding its design considerations, possible secondary launchers, engine choices and providing an insight into the reusable liquid propellent boosters that return to the Earth's surface after staging off. In Section 5, the efficiency calculations are applied to the primary concept design, and the values obtained are compared to the previous 3 launchers. Lastly, Section 6 summarises all the previous sections and concludes the research.

Notation: This report uses the following notation. Low-earth-orbit is represented as LEO and geostationary transfer orbit is represented as GTO. Each individual section of the launchers are referred to in their stage, such as boosters. Sections of the mission are referred to as phase, such as booster stage. The evaluated property B at phase $[n]$, of stage i is shown as $B_i^{[n]}$. Time derivatives of properties are given with a dot, such as $\dot{B}$.

2 Analysis and Methodology

2.1 General Rocket Equations

To evaluate the efficiency of the concept launcher and compare it with modern-day launchers, a mass ratio analysis at each phase of the mission should be performed. For any stage i, we define the mass ratio μ_i as the ratio of the mass at the end of the phase to the mass at the start of the phase as

$$\mu_i = \frac{M_{s,i} + M_{p,i} + M_{F,i}}{M_{s,i} + M_{p,i}} \tag{1}$$

where,

$M_{s,i}$ [kg], is the *structural mass* of stage i,

M_p [kg], is the *payload mass*,

$M_{F,i}$ [kg], is the *propellant mass* of stage i.

Next, we obtain the effective exhaust velocity of a rocket engine (V_e [ms^{-1}]) from its specific impulse I_{sp}[s] via

$$V_e = I_{sp}\, g \tag{2}$$

where g is the gravitational acceleration ($g \approx 9.81\text{ms}^{-2}$). To compute the change in velocity ΔV between any two stages, we apply

$$\Delta V = V_e\, \ln(\mu). \tag{3}$$

For an n-stage *serial* rocket, the total velocity increment for the i^{th} stage is given by

$$V = \sum_{i=1}^{n} \Big(V_{e,i}\, \ln(\mu_i) \Big). \tag{4}$$

However, this formula does *not* apply to *parallel-stage* rockets, where the core and booster stages operate simultaneously but are jettisoned at different times. In such a configuration, the "start mass" of stage two is not equal to the launch mass minus the booster mass—because the core has also consumed some of its propellents while the boosters were firing.

To handle parallel staging, the mass flow rate $\dot{m}$[kgs^{-1}] can be taken into account using the formula

$$\Delta m = \dot{m}\, \Delta t_{\text{burn\_time}} \tag{5}$$

and updating the core's remaining propellant accordingly as

$$M_{p,\text{core}} = M_{p,\text{srb}} \;-\; \dot{m}_{\text{core}}\, \Delta t_{\text{burn\_time\_srb}}. \tag{6}$$

Hence, for the *first* (booster) phase of a parallel-stage launcher, the change in velocity can be written as:

$$\Delta V_{\text{srb}} = V_{e,\text{srb}}\, \ln\!\left(\frac{M^{[1]}_{\text{total}}}{M^{[1]}_{\text{total,launch}} \;-\; M^{[1]}_{\text{total,core}} \;+\; \dot{m}_{core}\, \Delta t^{[1]}_{\text{burn\_time}}} \right) \tag{7}$$

where the term $\dot{m}\, \Delta t_{\text{burn\_time\_booster}}$ subtracts the *core's* partially consumed propellant at the time the boosters are jettisoned. $\dot{m}$ is given by,

$$\dot{m} = \frac{T}{I_{sp} g} \tag{8}$$

where $T[N]$ is thrust. It is also important to know the structural efficiency σ of the launcher stage at every phase. This value provides insight into how much of the mass is allocated to the structure indicating how much mass is effectively wasted. This makes the structural efficiency desirable to keep as low as possible. Note that σ can be calculated as

$$\sigma = \frac{M_s}{M_s + M_F} \tag{9}$$

For simplicity, defining a new parameter, $\alpha \triangleq M_S/M_F$ is beneficial. Therefore,

$$\alpha = \frac{\sigma}{1 - \sigma} \tag{10}$$

and,

$$M_S = \frac{\sigma M_F}{1 - \sigma} \tag{11}$$

Lastly, the payload ratio, λ is the payload mass ratio to the rest of the rocket mass. The payload mass should be as large as possible relative to the rocket mass to ensure the payload ratio is high.

$$\lambda = \frac{M_p}{M_S + M_F} \tag{12}$$

2.2 Maximum Available Payload

There may be instances where structural efficiency σ, required change in velocity ΔV, exit velocity V_e, and total mass M_{total} is known for an n stage rocket. It is therefore important to know the maximum payload mass M_P the launcher can hold. This can be obtained by rearranging (3). For Phase 1, if the total, initial mass at the beginning of the booster phase is given as $M_{total}^{[1]}$, then the used fuel during this phase can be found as

$$M_F^{[1]} = M_{total}^{[1]}(1 - e^{-\frac{\Delta V^{[1]}}{V_e^{[1]}}}) \tag{13}$$

Therefore the structural mass can be obtained, using the Equation 11 (where $K = e^{\frac{\Delta V}{V_e}}$) as

$$M_S^{[1]} = \frac{\sigma M_{totoal}^{[1]}(1 - K^{-1})}{1 - \sigma} \tag{14}$$

At the beginning of the next phase, the initial mass is given as

$$M_{total}^{[2]} = M_{totoal}^{[1]} - M_S^{[1]} - M_F^{[1]} \tag{15}$$

Equations (14) and (15) can be used within a loop, until the remaining payload mass, $M_{totoal}^{[n+1]}$, is obtained.

Finally, it is possible to bring all of the previously defined equations to obtain the *next phase total mass* after staging using

$$M_{totoal}^{[2]} = M_{totoal}^{[1]}\left(1 - \frac{1 - K^{-1}}{1 - \sigma}\right) \tag{16}$$

By using (16), for each phase, it is possible to obtain the remaining *payload mass*. It should be noted that within the code, the data for the liquid propellent boosters is given as a combination of the 4 boosters when necessary, such as thrust, T[N]. This means that if there exists a target payload, performing the above calculations over a wide range of structural efficiencies σ, can help find the maximum structural efficiency, σ_{max}. This is important because as σ_{max} decreases, the launcher would need less structural mass which would be difficult to achieve.

The required ΔV values could be obtained as they should be the change in velocities needed to stay in orbit at a certain altitude. Newton's law of gravity could be used as

$$\begin{aligned} F &= mr\omega^2 \\ &= \frac{GM_Em}{r^2} \\ \frac{mV^2}{r} &= \frac{GM_Em}{r^2} \\ V &= \sqrt{\frac{GM_E}{r}} \end{aligned} \tag{17}$$

where m [kg] is the mass of payload needed to stay in orbit at an altitude r [m] above the centre of Earth, $G \approx 1 \times 10^{-11}[\mathrm{m^3kg^{-1}s^{-2}}]$ is the gravitational constant, F[N], is the force due to gravity, ω $rads^{-1}$ is the angular velocity of the mass about the orbit and M_E[kg] is the mass of Earth. The obtained values of the needed velocities are 7 [$\mathrm{kmh^{-1}}$] for LEO and 11 [$\mathrm{kmh^{-1}}$].

3 Part 1: Analysis of Current Designs

An efficiency analysis of successful, current-day launchers is necessary to be able to compare the concept design. The launchers Ariane 5, and Pegasus XL, along with a general Stratolaunch system, are good standards to aim for. All of the following questions have been calculated using the scripts available in the GitHub repository accessible via this link.

3.1 Ariane 5

The *Ariane 5* is a heavy-lift launch vehicle developed by the European Space Agency (ESA) and operated by Arianespace. It is designed primarily to deliver payloads to LEO. The launch vehicle consists of two solid rocket boosters (SRBs), a cryogenic core stage (parallel staged with the SRBs), and an upper stage. The payload is enclosed in a *payload fairing* and mounted on a *payload adapter*.

Table 1, presents key data for each stage of the Ariane 5 launch vehicle:

Table 1: Ariane 5 Stage Data

Stage	**Launch Mass (kg)**	**Propellant Mass (kg)**	**Thrust (kN)**
Core Stage	184,700	170,000	1,390 (vac), 960 (SL)
Solid Rocket Booster (each)	268,000	237,800	6,470 (SL)
Upper Stage	19,440	14,900	62.7 (vac)
Payload Fairing	2,000	-	-
Payload Adapter	500	-	-

A set of questions are answered using this data for the Ariane 5. The LEO and GTO payloads are $21,000$kg and $10,500$kg respectively.

Q1(a) What is the Thrust-to-Weight ratio of the system at lift-off for the maximum LEO payload?
Answer: ($\frac{T}{W} = 1.855$)

Q1(b) What is the Thrust-to-Weight ratio of the system at lift-off for the maximum GTO payload?
Answer: ($\frac{T}{W} = 1.811$)

Q1(c) Calculate the structural efficiency of the core stage
Answer: (Using Equation 9, $\sigma = 0.0796$)

Q1(d) Calculate the structural efficiency of the upper stage
Answer: (Using Equation 9, $\sigma = 0.2335$)

Q1(e) Calculate the structural efficiency of the solid-propellant boosters (SRBs)
Answer:(Using Equation 9, $\sigma = 0.1127$)

Q1(f) For the maximum LEO payload, determine the speed increases ('Delta V's) for each of the three phases of the launch

Answer: (Using Equation 7 and Equation 1, $\Delta V_{srb} = 3,947\text{ms}^{-1}$, $\Delta V_{core} = 4,981\text{ms}^{-1}$, $\Delta V_{upper} = 1,980\text{ms}^{-1}$, $\Delta V_{total} = 10908\text{ms}^{-1}$)

Q1(g) For the maximum GTO payload, determine the speed increases ('Delta V's) for each of the three phases of the launch

Answer: (Using Equation 7 and Equation 1, $\Delta V_{srb} = 4050\text{ms}^{-1}$, $\Delta V_{core} = 5588\text{ms}^{-1}$, $\Delta V_{upper} = 2941\text{ms}^{-1}$, $\Delta V_{total} = 12580\text{ms}^{-1}$)

3.2 Pegasus XL Launch System

The *Pegasus XL* is an air-launched rocket developed by Orbital Sciences Corporation. It was designed for launching small satellites into Low Earth Orbit (LEO) during the 1980s and 1990s. The Pegasus XL uses a three-stage solid-propellant design and is deployed mid-air from a modified Lockheed L-1011 aircraft. Table 2, presents key data for each stage of the Pegasus XL launch vehicle:

Parameters	**Stage 1**	**Stage 2**	**Stage 3**
Launch mass $M_{totoal}^{[1]}$ [kg]	16383	4306	872.3
Structural mass M_S [kg]	20	15.2	102.1
Propellant mass M_F $[kg]$	28	8.91	770.2
M_F/M_S	10.97	10	7.5
Isp [s]	295 (Vac)	289	287
T [kN]	726 (Vac)	158	32.7
Burn time [s]	68.6	71	67
Mass of delta wing [kg]	100	-	-
Maximum LEO payload $M_{P,max}$ [kg]	443		

Table 2: Pegasus XL Stage Data

A set of questions are answered using the data provided in the above table.

Q2(a) For the three stages, calculate the structural efficiencies, as well as the payload fractions (λ).
Answer: Using Equation 9 and Equation 12, σ and λ for each stage is shown in Table 3.

Stage	**Structural Efficiency** (σ)	**Payload Fraction** (λ)
First Stage	0.1363	0.3234
Second Stage	0.0908	0.3055
Third Stage	0.117	0.5079

Table 3: Pegasus XL Q2a

Q2(b) Compute the average propellant mass-flow rates and from that estimate the exhaust gas velocity for each stage.

Answer: Using Equation 8 and Equation 1, $\dot{m}$ and V_e for each stage are shown in Table 4.

Stage	**Average propellant mass-flow rate,** $\dot{m}$ [kgs^{-1}]	**Exhaust gas velocity,** V_e [ms^{-1}]
First Stage	218.863	3317.144
Second Stage	55.1408	2865.3895
Third Stage	11.5015	2843.1093

Table 4: Pegasus XL Q2b

Q2(c) For the maximum LEO payload, compute the speed increases as well as the total ΔV. Use ISP data for V_e. State any assumptions made.

Answer: Horizontal vacuum conditions are assumed. Using Equation 4, the speed increases are calculated and presented in table 5.

Stage	Speed increase [ms^{-1}]
First Stage	3060.215
Second Stage	3380.0954
Third Stage	2480.009
Total ΔV	8920.3201

Table 5: Pegasus XL Q2c

3.3 Stratolaunch-Based System

The *Stratolaunch-based* launch system is an air-launch approach in which a massive carrier aircraft (designed by Stratolaunch) carries a rocket to high altitude before releasing it for ascent into space. This method can accommodate both satellite launchers and high-speed experimental vehicles, leveraging mid-air deployment to improve performance and flexibility compared to traditional ground-launched rockets.

Table 6 presents key data for each stage of the Stratolaunch System:

Parameter	Value / Assumption
Maximum "drop" mass from Stratolaunch	$200,000$ kg
Total ΔV required (LEO mission)	10.0 kms^{-1}
Baseline effective exhaust velocity (V_e)	3.4 kms^{-1}
Two-Stage Cases	
Q3(a) structural efficiency (both stages)	0.07
Q3(b) same as Q3(a), but V_e varied	(start from 3.4 km/s, then increase as found)
Q3(c) same as Q3(a), but σ varied	(start from 0.07, then decrease as found)
Three-Stage Cases	
Q3(d) structural efficiency (all 3 stages)	0.07
Q3(e) structural efficiency (all 3 stages)	0.08

Table 6: Stratolaunch Stage Data.

A set of questions are answered using the data provided in the above table.

Q3(a) What is the maximum achievable payload mass to keep the launch mass below 200,000 kg? For this maximum payload mass, what are the propellant and structural masses for both stages?

Answer:

Table 7: Stratolaunch Q3(a)

Maximum Payload	5904.3[kg]	
Stage	**Propellant Mass [kg]**	**Structure Mass [kg]**
First Stage	154042	11595
Second Stage	24647	1992

Q3(b) Compute the required increased value for V_e to achieve a payload of $6,500$kg

Answer: V_e should be raised from 3400ms^{-1} to 3480ms^{-1}

Q3(c) Compute the required decreased value for structural efficiency to achieve a payload of 6,500 kg.

Answer: Structural efficiency should be reduced from 0.07 to about 0.06.

Q3(d) What is the maximum achievable payload mass to keep the launch mass below 200,000 kg? For this maximum payload mass, what are the propellant and structural masses for both stages?

Answer: The results are tabulated in Table 8

Maximum Payload	7861.1[kg]	
Stage	**Propellant Mass [kg]**	**Structure Mass [kg]**
First Stage	124967	9466
Second Stage	41806	3806
Third Stage	13455	1218

Table 8: Stratolaunch Q3(d)

Q3(e) What is the maximum achievable payload mass to keep the launch mass below 200,000 kg? For this maximum payload mass, what are the propellant and structural masses for both stages?

Answer: The table of results in found in Table 9

Maximum Payload	6625[kg]	
Stage	**Propellant Mass (kg)**	**Structure Mass [kg]**
First Stage	124905.7	10864.5
Second Stage	40106.4	3487.5
Third Stage	12890.7	1120.1

Table 9: Stratolaunch Q3(e)

4 Part 2: Concept

4.1 General Concept

This report presents a contemporary launcher family design for multi-orbit missions. The primary launcher concept is comprised of 3 phases. A booster phase with 4 liquid propellent boosters using the engine *RS-25*, which have data as presented in Table 11. The core phase uses a *Merlin 1D* engine with data presented in Table 12. Lastly, the upper stage uses the engine *RL-10* as shown in Table 13. Primary and secondary launcher models are shown from different angles in Figure 1 and Figure 2.

The concept launcher also allows many configurations to be made, creating a *family of launchers*. While this report mainly focuses on the primary launcher, it is important to note that the stages used on the primary launcher can be used in any amount to create a diverse set of launchers to fit any mission, such as the secondary launcher which is explored in Section 4.2.

The primary launcher is designed to reach LEO within the first two phases. For partial reusability, the 4 liquid propellent boosters have control surfaces and two-stage parachutes, allowing them to manoeuvre to the Earth surface. Also, at the LEO stage, the first payload is deployed using a large SHERPA-like tug system, allowing for auxiliary payloads to launch along with the main LEO payload. Next, the upper stage engine turns on, and by using an advanced optimal automatic control system such as an linear quadratic regulator (LQR) [3], the launcher is guided into a new *Bi-elliptic transfer orbit*.

Stage	**Launch Mass [kg]**	**ISP [s]**	**Thrust Vacuum [kN]**	**Thrust Sea Level [kN]**	$\Delta V [ms^{-1}]$
Booster	100000×4	363	N/A	2279×4	3000
Core	200000	300	N/A	900	4000
Upper	30000	410	66700	N/A	4000

Table 10: Launcher stage data

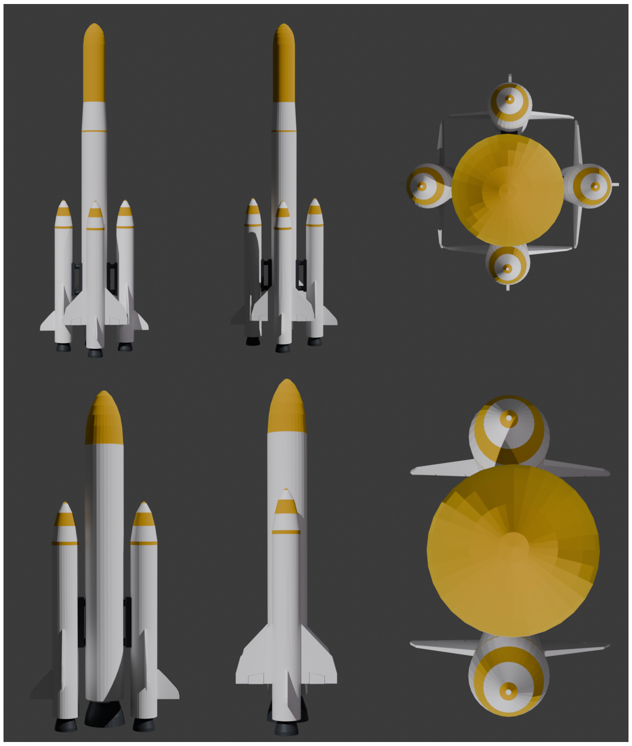


Figure 1: 3D drawings of the primary (top row) and secondary (bottom row) launchers showing front, side and top views respectively. Models designed using Blender

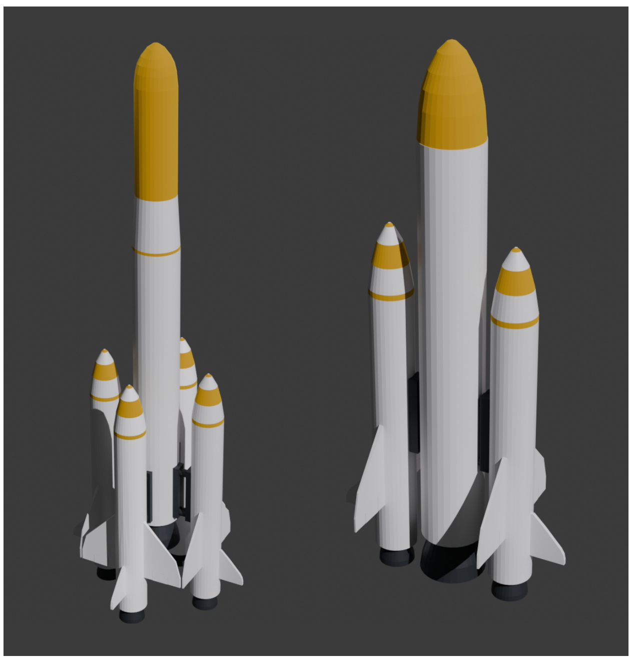


Figure 2: Isometric views of primary (right) and secondary (left launchers). Designed using Blender.

4.2 Secondary Launcher and Family of Launchers Concept

As mentioned in Section 4.1, it is possible to make a smaller secondary launcher for faster missions. While the primary launcher can deliver payloads to both LEO and GTO, due to the high demand of LEO satellites, a smaller secondary launcher can be launched only for LEO.

The secondary launcher is made up of only 2 boosters and a core. As proved in Section 5, the primary launcher already gets to LEO using 4 boosters and a core stage, with additional mass.

With no change made to the structural efficiency of the launcher stages, the thrust can be adjusted, by adapting the Merlin 1D engine to use a smaller thrust force as it is scalable between 40% - 100%, this can be paired with using less propellant in the fuel tank to get a much lighter and more efficient launch. This idea could also come with decreasing the number of boosters to reduce the overall weight for launch. The secondary launcher can therefore be made up of the same stages as the primary launcher. Note that many other configurations could be made for other missions.

Due to the *family of launchers* idea, the partial reusability applies to all configurations that include a booster phase.

4.3 Engines

Table 11, Table 12 and Table 13, contain the relevant data for engines within the three stages booster, core and upper respectively. Each chosen engine also has a mission success rate greater than 90%, with the exception of the Raptor engines, which remain in testing.

Engine	**Propellant**	**ISP Vacuum [s]**	**ISP Sea Level [s]**	**Thrust [kN]**	**Dry Mass [kg]**	**Success Rate**
SpaceX Raptor	LCH4	380	327	2500	1630	Unknown
RS-25	LH2	452.3	366	2279	3177	100%

Table 11: Booster Stage Engines

Engine	**Propellant**	**ISP Vacuum [s]**	**ISP Sea Level [s]**	**Thrust [kN]**	**Dry Mass [kg]**	**Success Rate**
RD-171	RP-1	338	309	7887	9750	97%
Merlin 1D	RP-1	348	283	900	630	99%

Table 12: Core Stage Engines

Engine	**Propellant**	**ISP Vacuum [s]**	**ISP Sea Level [s]**	**Thrust [kN]**	**Dry Mass [kg]**	**Success Rate**
RL-10	LH2/LOX	410	N/A	66.7	131	99%
Raptor Vacuum	LCH4/LOX	380	N/A	2000	630	Unknown

Table 13: Upper Core Engines

Note that engines designed to operate in a vacuum do not have values for ISP at sea level as this is not their intended operation level.

The engine we are using for the first stage is the Merlin 1D rocket engine. This a liquid-based rocket engine designed and developed by SpaceX, it was designed for the Falcon 9. The engine is fuelled by a kerosene (RP-1) and liquid oxidiser mixture (LOX). It is most used due to its reusability, high thrust-to-weight ratio, and overall high efficiency. [4]

Property	Value
Propellant	PR-1 & LOX
Thrust	900kN
ISP (Vaccum)	348s
ISP (Sea Level)	283s
Chamber Pressure	10MPa
Thrust:Weight	180
Dry Mass	630kg
Average Burn Time	140s
Cooling Method	Regenerative Cooling

Table 14: Engine Specifications

The Merlin engine is powered by RP-1 and LOX, these are stored at an initial pressure of 344kPa[5]. The propellant mixture is then pumped through the system and the pressure increases to around 10MPa. The propellant is then circulated through the nozzle and cooled before igniting in the combustion chamber. A pre-burner ignites a small fraction of the propellant mixture to drive the turbine. This then powers propellant and oxidiser pumps.[**?**] The combustion chamber operates at around 7MPa and reaches high temperatures of 4145K which generates the thrust. Depending on the amount of propellant needed at any one time many valves are used to regulate the amount combusting.

4.3.1 Structural Design

The Merlin 1D engine has been optimised for multiple reuses; this makes it a key component in our engine choice as we want to maximise reusability. The engine includes various key design features to maximise its performance, reusability, and efficiency.
These include:

- **Lightweight Materials:** To improve the thrust-to-weight ratio, various components have been optimised to be as light as possible. Components like the fuel tank are made of an Aluminium-Lithium alloy (Al-Li), which is much lighter than the traditional Aluminium. Lithium is the least dense element therefore, its inclusion within the compound is vital. While components are made as light as possible, they also still need to be strong enough to hold under the harshest of conditions.
- **Regenerative Cooling:** The fuel (RP-1) circulates around the nozzle to minimise the chance of overheating. This increases the engine durability by minimising the overall fatigue of materials and thermal expansion. This means that the engine can endure multiple missions without experiencing excess wear, increasing the reusability. The RP-1 also becomes less viscous which increases its combustibility before it is injected. This results in a more efficient combustion process and increases the specific impulse of the engine [6]

4.3.2 Performance and Reliability

Since its first flight in 2013 the Merlin 1D has powered many missions for SpaceX and very rarely fails. Throughout this period there have been 3 in-flight failures. The Merlin 1D engine has a success rate of 99.7%. This demonstrates its extreme reliability and great performance. In 2020 during a Falcon 9 launch, one of the 9 Merlin engines shut down, but the rocket compensated by burning the remaining 8 for longer.
Other key advantages of the Merlin 1D are:

- Efficient Specific Impulse.

- Deep Throttling Capabilities: Merlin 1D can throttle between 100% down to 40%, this allows extremely precise control of manoeuvres and adjustments.

4.3.3 Alternative Engine Considerations

During the engine selection process, we looked at and compared various alternate engines shown in Table 11 [7], Table 12 [8] and Table 13. While engines like the Raptor provide a extremely high thrust and efficiency, we selected the Merlin 1D for its proven reliability, cost-effectiveness, high efficiency and reusability.[9]

4.3.4 RL-10 as High Stage Option

For the Upper stage propulsion, the Rl-10 was chosen due to its high ISP and multiple restart capability. The RL-10 uses a LH2 and MOX propellant mixture. It has been widely used in the upper stages of many NASA missions[10], demonstrating its high reliability and efficiency. It has a vacuum ISP of 410s, having a high ISP is important during the upper stages of a mission as it reduces the amount of fuel needed at any one time which improves the capacity of payloads and reduces costs[11]. Another reason we selected this engine is because of its multiple restart capability, this allows more complex manoeuvres such as precise payload insertions into an orbit to be carried out[12].

4.3.5 Advantages and Disadvantages

The first stage rocket engine Merlin 1D has:

Advantages

- High thrust to weight ratio
- Low cost of production
- High reusability
- Reliable and efficient

Disadvantages

- Lower ISP compared to other engines.
- Optimised for first stage only making it unsuitable for deep space missions.

The Merlin 1D's reusability, low cost and high efficiency vastly outweigh its disadvantages and make it a great engine to use in the first stage of missions, reducing the overall launch costs and enables a quick turnaround between different missions.

The second stage rocket engine RL-10 advantages and disadvantages.

Advantages

- High ISP, giving a high efficiency in space
- Multiple restart capability, allowing for precise and complex manoeuvres
- Lightweight, maximising payload capacity
- Highly reliable

Disadvantages

- Lower thrust, requiring additional engines or staging before upper stage
- Quite costly, due to complex manufacturing.

The RL-10's high ISP and restarting capability make it an essential engine to be used during deep space exploration or high-orbit insertions. However, its high costs limits its use primarily to the most important payloads.

4.3.6 Conclusion

The Merlin 1D is one of the most efficient and cost-effective first-stage engines which plays a key role in modern reusable launches. Its high thrust, reliability and reusability are the main factors we considered when selecting the engine to launch multiple payloads to different orbits. In contrast, the RL-10 is ideal for upper-stage propulsion due to its high ISP and restart capabilities. Allowing for the most precise and complex manoeuvres to be carried out when delivering payloads to GTO.

4.4 Return-Home System

To ensure partial reusability and save costs, the four liquid propellants are designed to return to the Earth's surface, safely and with minimal damage. This means that from a structural perspective, the boosters should be designed to withstand high pressures and extreme temperatures as the LPBs re-enter. As mentioned before this will not only apply to the primary launcher but also to all other configurations from the family of launchers. To ensure full controllability, the boosters are fitted with two wings that have flaps, along with a rudder for horizontal stabilisation. Due to the lack of landing gear, the booster is required to land on water and is also fitted with a dual-stage parachute on the nose for emergency landing on land [13]. However, this is not recommended as the booster would ideally have to go into a high angle of attack which might cause a potentially unrecoverable unstable stall.

From an avionics and control perspective, the challenge is to control a highly unstable and reactive system with the least fuel usage, using the control surfaces all the while avoiding the other boosters and performing an online path planning algorithm. Due to the need for optimal fuel usage (and therefore optimal control signal), an optimal control technique such as a Linear Quadratic Regulator is used. This would be achieved using the linearised model for a 6-DoF booster rocket, presented [14]. Laser-based guidance should be used as presented in [15].

Due to there being 4 rockets needed to be safely guided to the target location without crashing into one another, an online obstacle avoidance and a multi-agent mission distribution algorithm is required.

4.5 Payload Launch System (HOPES)

This sub-section looks into the payload launch system. To ensure the payload can launch at staging and stay in the required orbit, a payload launch system (Heavy Optimal Payload Extraction System (HOPES)) is proposed in this sub-section. HOPES design considerations include; structure and avionics.

The HOPES system is designed to hold a major payload such as the LEO payload and multiple minor (auxiliary) payloads such as CubeSats and SmallSats, which are fitted onto an ESPA ring under the major payload. The design is similar to the SHERPA [2], however larger to accommodate bigger payloads. Just as the SHERPA, HOPES is fitted with controllable and retractable (to ensure it can fit) solar panels. The HOPES system is also fitted with a fuel tank at the centre of the ESPA ring to allow small, engines to be used. Lastly, HOPES uses ion cannons to generate angular momentum, to change direction without consuming fuel therefore lowering costs.

The HOPES computer calculates the required altitude and current states of HOPE. This data is then used to generate an optimal trajectory and velocity profile using "non-linear programming representation of an optimal control problem" as presented in [16]. The reference points generated by this algorithm is then sent to a proportional, integral, derivative controller (PID), which navigates HOPES to the desired altitude.

5 Results and Discussion

These results were obtained from the Python script (accessible via this link) which uses the theory established in Section 3 of this document. It calculates an optimal value for the structural efficiency using a naive search algorithm along with launch thrust-to-weight ratio. This is done utilising the data presented in Table 15. It is assumed that the structural efficiencies are the same for all stages of the concept launcher. The initial parameters and characteristics for the script, such as engine metrics and desired payload masses, are taken directly from the methodology and analysis part of the report. A comparison of these results to other missions have been outlined below. It should be noted that these missions were largely single-objective and only launched to one orbit. Masses for the LBP are the sum of all four boosters.

Stage	**Structural Mass [kg]**	**Propellent Mass [kg]**
LPB	24025	376401
Core	12365	193719
Upper	1347	21107
Other Data		
Thrust-to-Weight (Booster Phase)	1.406	
Structural Efficiency σ	0.06	

Table 15: Result Data

The results could then be compared against the existing launchers analysed in section 3. The comparison will be made using the GTO values for all launchers, as the concept launcher is designed to get to both LEO and GTO.

The thrust-to-weight ratio of the concept launcher is lower than the thrust-to-weight ratio of *Ariane 5* launcher ($1.406 < 1.811$). The decrease in unbalanced force upwards implied in this result is expected as the concept launcher is intended for much larger payloads than were used in the Ariane 5

The structural efficiency of the concept launcher (0.06) is much smaller than the Ariane 5's ($\sigma_{srb} = 0.0796$, $\sigma_{core} = 0.2335$, $\sigma_{upper} = 0.1127$). It is also smaller than Pegasus XL Launch System ($\sigma_{srb} = 0.0796$, $\sigma_{srb} = 0.0796$, $\sigma_{srb} = 0.0796$). When compared to the *Stratolaunch based system*, the two structural efficiencies are the same. This outlines the superiority of the heavier launcher, such as the concept, as it shows that structural efficiencies of lighter launchers can be replicated in much heavier launchers.

It is also important to note the cost implications of the concept launcher compared to that of other launchers. While launchers such as the *Stratolaunch based system* requires less fuel consumption, the concept launcher is designed to transport two payloads. This dramatically increases cost efficiency as it will require less launches.

6 Conclusion

This report has presented a novel launcher concept that can successfully deliver payloads to LEO and GTO in a single mission, using reusable stages - hence improving efficiency and decreasing cost. The analysis of similar existing launchers and engines has proven that the proposed design achieves the mission objectives through a careful stage separation, controlled reusability and an advanced guidance system. The launcher family design consists of 3 stage rockets with RS-25 boosters, a Merlin 1D core, and an RL-10 upper stage, to provide thrust efficiently and sustainably. Partial reusability is achieved through a controlled booster recovery approach which further enhances cost-effectiveness. The use of a SHERPA-like system allows multi-payload deployment, expanding the mission's versatility. Similarly, the integration of an advanced control system such as an LQR-based guidance mechanism, ensures precise orbits which is crucial for such a complex mission. Ultimately, the results clearly show that this launcher is a viable and competitive alternative to current designs, offering improved efficiency and flexibility for multi-orbit missions. Further research will be necessary to refine the design and control techniques as well as ensure long-term feasibility.

A Concept Launcher Evaluator Script

```
###===-------------------------------------------------===###
# Script:       main.py
# Authors:      Demir Kucukdemiral 2883935K, Charikleia Nikou 2881802N,
#               Cameron Norrington 2873038N, Adam Burns 2914690B,
#               Ben Maconnachie 2911209M, Jeremi Rozanski 2881882R
# Created on:   2025-02-28
# Last Modified: 2025-02-28
# Description:  optimal structural efficiency solver for a concpet launcher
# Version:      1.0
###===-------------------------------------------------===###


import numpy as np
from dataclasses import dataclass
"""
This script uses a dataclass to store the information about the stages
of a launcher. The Launcher class uses this data to calculate the
thrust-to-weight ratio and the final payload mass given some initial
totoal mass, engine data, phase information and required final velocity.
"""


@dataclass
class Stage:
    name: str
    launch_mass: float
    Isp: float
    thrust_vac: float
    thrust_sl: float
    burn_time: float
    delta_v: float
    number: int

class Launcher_Data:
    def __init__(self):
        """
        Initi function to declare or parameters, phases and stage informations
        also declares all engine informations.
        """
        self.gravity = 9.81

        self.mass_payload_1 = 21100
        self.mass_payload_2 = 10010

        self.lpb = Stage(
            name="Liquid Propellant Booster",
            launch_mass=100000 * 4,
            Isp=363,
            thrust_vac=2279000 * 4,
```

```
    thrust_sl=2279000 * 4,
    burn_time=130,
    delta_v=3000,
    number=1
)

self.core_stage = Stage(
    name="Core Stage",
    launch_mass=200000,
    Isp=300,
    thrust_vac=900000,
    thrust_sl=900000,
    burn_time=540,
    delta_v=4000,
    number=1
)

self.upper_stage = Stage(
    name="Upper Stage",
    launch_mass=30000,
    Isp=410,
    thrust_vac=66700,
    thrust_sl=0,
    burn_time=500,
    delta_v=4000,
    number=1
)


self.stages = {
    "lpb":   self.lpb,
    "core":  self.core_stage,
    "upper": self.upper_stage
}


self.total_mass = (
    self.mass_payload_1
    + self.mass_payload_2
    + self.lpb.launch_mass * self.lpb.number
    + self.core_stage.launch_mass * self.core_stage.number
    + self.upper_stage.launch_mass * self.upper_stage.number
)


self.phases = [
    {
        "name": "lpb",
        "active_stages": ["lpb", "core", "upper"],
        "drop_stages":   ["lpb"],
        "jettison_fairing": False,
```

```
                "active_engine": ["lpb"]
            },
            {
                "name": "core",
                "active_stages": ["core", "upper"],
                "drop_stages":   ["core"],
                "jettison_fairing": True,
                "active_engine": ["core"]
            },
            {
                "name": "upper",
                "active_stages": ["upper"],
                "drop_stages":   ["upper"],
                "jettison_fairing": False,
                "active_engine": ["upper"]
            }
        ]
class Launcher:

    def __init__(self):
        self.data = Launcher_Data()

        self.time = 0
        self.total_mass = self.data.total_mass

        self.structural_efficiency = 0.06

        self.number_of_stages = 3

        self.phases = self.data.phases
        self.stages = self.data.stages

    def Thrust_to_weight(self, phase: str) -> float:
        #thrust to weight calculation (of the booster phase) using given paramaters
        phase_info = None
        for p in self.phases:
            if p["name"] == phase:
                phase_info = p
                break
        if phase_info is None:
            raise ValueError(f"Invalid phase '{phase}' in Thrust_to_weight().")

        total_thrust_sl = 0.0
        for engine_name in phase_info["active_engine"]:
            stage_obj = self.stages[engine_name]
            total_thrust_sl += stage_obj.thrust_sl * stage_obj.number

        tw_ratio = total_thrust_sl / (self.total_mass * self.data.gravity)
        return tw_ratio

    #Final mass calculation using the given structural efficiency
```

```python
    def final_payload_mass(self, structural_efficiency, verbose=True):
        Ve = 0.0
        alpha = structural_efficiency / (1 - structural_efficiency)
        current_m0 = self.total_mass

        for stage_key in ["lpb", "core", "upper"]:
            stg = self.stages[stage_key]
            Ve = stg.Isp * self.data.gravity

            K = np.exp(stg.delta_v / Ve)
            m_after_burn = current_m0 / K
            m_propellant = current_m0 - m_after_burn
            m_structure = alpha * m_propellant

            if verbose:
                print(f"Stage {stage_key} => dv={stg.delta_v:.1f} m/s | "
                    f"m_prop={m_propellant:.1f} kg | m_struct={m_structure:.1f} kg")

            current_m0 = m_after_burn - m_structure

            if stage_key == "core":
                current_m0 -= self.data.mass_payload_1

            if current_m0 <= 0:
                return 0.0

        return current_m0

    def optimal_efficiency(self, target_payload: float):
        #searching for the optimal structural efficiency until desired final mass is reached
        current_eff = self.structural_efficiency

        while current_eff > 0:
            payload = self.final_payload_mass(structural_efficiency=current_eff, verbose=False)
            if payload >= target_payload:
                print(f"Found structural efficiency ~ {current_eff:.2f} => payload ~ {payload:.1f} kg")
                return
            current_eff -= 0.001

        print("Could not achieve the desired payload with the given model.")


if __name__ == "__main__":
    launcher = Launcher()

    tw_lpb = launcher.Thrust_to_weight("lpb")
    print(f"T/W ratio during LPB phase: {tw_lpb:.3f}")

    leftover = launcher.final_payload_mass(launcher.structural_efficiency, verbose=True)
    print(f"Final payload mass (default efficiency={launcher.structural_efficiency:.2f}) ~ {leftover:.1
```

```
print("\nSearching for an optimal structural efficiency to get 1000 kg payload...")
launcher.optimal_efficiency(10500)
```

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