Contents

1 Abstract

This paper explores the current and future applications of a Reusable Post-Low Earth Orbit (LEO) Chemical Rocket Booster, along with calculations for its common use cases. This in-orbit booster is designed to propel spacecraft, space stations, and other vehicles from LEO to specific transfer orbits, then execute a controlled flip, perform a boost-back burn, and return to LEO for refueling and reuse. By significantly increasing payload capacity for interplanetary missions—whether in the form of cargo, fuel for return journeys, or larger spacecraft—this system offers substantial advantages for deep-space exploration.

The booster design is based on SpaceX's Starship, chosen for its industry-leading reusability, modularity, and high-thrust propulsion capabilities. This paper evaluates and compares existing and proposed post-LEO propulsion systems, such as NASA's Nuclear Thermal Propulsion (NTP), Nuclear Electric Propulsion (NEP), and ion thrusters. Key performance calculations for the proposed chemical booster—featuring nine Raptor 2 Vacuum engines and an initial mass of 3,650 tonnes—demonstrate sufficient Δv to insert payloads into geostationary, lunar, and Mars transfer orbits. Estimated maximum payload capacities to these transfer orbits are approximately 2,978 tonnes to GEO, 1,780 tonnes to the Moon, and 840 tonnes to Mars. After payload deployment, the booster performs a boost-back burn and returns to LEO for refueling, without reliance on aerobraking. The design achieves a balance between reusability and thrust efficiency, supporting future applications including interplanetary cargo transport, Mars Aldrin Cycler integrations, and the development of deep-space infrastructure.

2 Introduction

This section introduces the evolution of reusable rocket technology, the need for post-LEO reusable chemical boosters, and the challenges and motivations driving their development. It outlines the thesis's focus on designing a reusable transorbital booster, inspired by SpaceX's Starship, to enable cost-effective deep-space missions in the 2025–2035 timeframe.

2.1 Overview of Current Reusable Rocket Technology

Reusable rocket technology has revolutionized spaceflight by drastically reducing launch costs [9]. SpaceX's Falcon 9 and Falcon Heavy lead the industry, with Falcon 9 boosters reused up to 24 times and a fleet total of 369 reflights by March 2025 [31]. Starship aims for full reusability for Earth orbit, lunar, and Mars missions [29].

Other contributors include Blue Origin's New Shepard and forthcoming New Glenn, Rocket Lab's Electron and in-development Neutron, as well as international efforts [2, 27]. While these systems demonstrate impressive performance for Earth-to-orbit operations, post-LEO reusability remains limited [37]. See Section 3.1 for detailed reviews.

2.2 The Need for Reusable Chemical Boosters Beyond Low Earth Orbit (LEO)

Most current reusable rockets are optimized for LEO missions, often relying on expendable upper stages—such as Delta IV's DCSS—for transfer to higher orbits [35]. As exploration shifts toward the Moon, Mars, and deep space, delivering large payloads like space stations requires reusable transorbital chemical boosters [24].

SpaceX's Starship advances this capability through orbital refueling and second-stage reusability [29]. However, the increasing scale of missions and projected industry growth—including Aldrin Cycler networks and Lunar stations like Gateway—demand dedicated post-LEO solutions [10, 19]. These boosters could significantly reduce mission costs and increase the feasibility of ambitious projects [28].

2.3 Key Challenges and Motivations for Post-LEO Reusability

Post-LEO reusable boosters are crucial for transporting large structures—such as 500-tonne space stations—and saving propellant mass for deep-space missions [37]. While Falcon 9 proves the viability of LEO reusability, deep-space operations present greater challenges in propulsion efficiency, recovery, and refurbishment [31, 32].

Chemical propulsion, though reliable, lacks efficiency for long-duration missions. Nuclear thermal propulsion (NTP), with an I_{sp} of approximately 900–1,000 seconds, is unlikely to be operational before the 2030s–2040s. Ion propulsion systems offer extremely high I_{sp} (up to $\sim$ 10,000 seconds) but provide insufficient thrust for rapid transfers or heavy payloads [3, 6].

A key motivation is transitioning large LEO-assembled structures to transfer orbits—such as GEO, Lunar, Mars, and Aldrin Cycler paths [10]. LEO assembly allows for reduced launch costs, rapid repairs, human-assisted construction, and simplified refueling [28]. Post-LEO boosters then handle the transfer phase, supporting interplanetary habitats too massive for direct Earth launch while reducing orbital debris [24].

2.4 Objectives and Scope of the Thesis

This thesis explores the design and feasibility of a reusable chemical rocket booster, referred to as the "Deep Space Super Heavy Booster" or "Reusable Transorbital Chemical Rocket Booster," intended for post-LEO missions to GEO, the Moon, Mars, and Aldrin Cycler orbits. Inspired by SpaceX's Starship, it uses Raptor 2 engines with an I_{sp} of approximately 380 seconds and is designed for future upgrades to Raptor 3 engines [29].

It examines:

- **Design Principles:** Adapting Methalox propulsion for vacuum-only reusability.
- **Refueling Infrastructure:** Establishing LEO stations for rapid redeployment.

- **Recovery Challenges:** Evaluating boostback burns versus aerobraking alternatives.
- **Payload Calculations:** Estimating mass budgets and Δv requirements ($\sim$ 2.4–6.6 km/s) for GEO, Lunar, Mars, Ceres, and Jupiter missions, based on calculations presented in Section 6.

The scope evaluates chemical propulsion's near-term role relative to the development timelines of NTP and ion propulsion systems, proposing a scalable solution for the 2025–2035 window [3].

3 Background & Literature Review

This section reviews reusable rocket technologies and prior studies on post-LEO propulsion, framing the context for the Deep Space Super Heavy Booster. It compares chemical propulsion with alternatives like nuclear and electric systems, emphasizing their implications for reusable deep-space missions in the 2025–2035 timeframe.

3.1 Review of existing reusable rocket technology

Reusable rocket technology has transformed spaceflight economics over the past two decades [9]. SpaceX's Falcon 9 and Falcon Heavy pioneered reusable first stages, landing on Earth or sea platforms with precision [31]. Falcon 9's booster has flown up to 20 times (as of March 2025), slashing launch costs from approximately $60 million to approximately $30 million by reusing hardware instead of building a new booster from scratch each time [31]. The Starship system is composed of two parts: Super Heavy Booster, with 3,400-tonne fuel capacity and 33 Raptor engines, and the Starship upper stage. It targets full reusability, supporting missions from Earth's surface to LEO, the Moon, and Mars [29]. Starship's lunar role in NASA's Artemis program, under the Human Landing System contract, extends its ambition beyond LEO to the Moon, while Super Heavy focuses on Earth-to-orbit lift [16].

Blue Origin's New Shepard, a suborbital rocket, lands after each flight, providing rapid reuse for short missions like space tourism [2]. Its New Glenn, a heavy-lift 2024 orbital rocket, debuted on January 16, 2025, from Cape Canaveral, reaching orbit with a Blue Ring Pathfinder payload [2]. Powered by seven BE-4 engines using Methalox, generating approximately 3.85 million pounds of thrust, its reusable first stage targets 25 flights but failed to land on a barge during its maiden attempt [2]. With a 45-tonne LEO capacity and a 7-meter fairing, New Glenn rivals Falcon 9 and is well-suited for both commercial and military missions. Including the U.S. Space Force's National Security Space Launch (NSSL) program, though its full reusability has yet to be demonstrated [34].

Rocket Lab's Electron, a small orbital launcher, advances reusability via parachute and helicopter capture, with 12 launches in 2024 and a record 60 by March 2025 [27]. Its Neutron, a medium-lift rocket under development since 2021, targets a mid-2025 debut from Wallops Island, Virginia [27]. With nine Archimedes engines using Methalox, generating approximately 1.45 million pounds of total thrust, Neutron aims to lift 13,000 kg to LEO with a reusable first stage, landing on a 400-ft vessel, *Return On Investment*, starting in 2026. Two commercial launches are booked for mid-2026, focusing on mega constellations. Unlike Electron's small-payload niche, Neutron competes with Falcon 9, though its post-LEO recovery remains untested.

NASA's Space Launch System (SLS), designed for lunar and Mars missions, contrasts as a non-reusable platform, highlighting a gap in reusable post-LEO technology [17]. Each launch of SLS costs an eye-watering $2–4 billion [20]. Falcon, New Glenn, Electron, and Neutron prioritize LEO or Earth-to-orbit, while Starship signals steps toward reusable deep-space infrastructure [29].

3.2 Chemical Propulsion

Chemical propulsion dominates deep-space missions with its high thrust-to-weight ratio, ideal for rapid maneuvers and heavy payloads [32]. NASA's SLS, though expendable, delivers approximately 8.8 km/s Δv for lunar missions using LOX/LH_2 engines with an I_{sp} of 450 seconds [17]. The proposed Deep Space Super Heavy Booster, a reusable adaptation of SpaceX's Super Heavy, employs 9–10 Methalox Raptor Vacuum engines, with an I_{sp} of approximately 380 seconds and approximately 258 tonnes of force per engine, producing a total thrust of approximately 2,322–

2,580 tonnes [29]. Methalox supports in-situ resource utilization (ISRU) on Mars via the Sabatier reaction:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

producing fuel from local resources [38]. Unlike Starship's ceramic tiles for atmospheric reentry at destination, the proposed LEO stainless steel, vacuum-only design avoids the need for thermal shielding [29].

3.3 Nuclear Thermal Propulsion (NTP)

NTP boosts efficiency ($I_{sp} \sim 900$–1,000 s) by heating propellant via a nuclear reactor, halving propellant needs for Mars missions compared to chemical systems [3]. NASA's relaunched NTR program (2021) aims for a prototype by 2029–2032, targeting crewed Mars flights [15]. However, major challenges remain, including reactor miniaturization, safety, and propellant handling, likely delaying operational use beyond this project's 2025–2035 focus [3].

3.4 Electric Propulsion

Electric propulsion, like ion and Hall-effect thrusters, achieves an I_{sp} of up to 10,000 s, with NASA's Dawn xenon ion propulsion system reaching approximately 3,100 s [26]. Its efficiency suits extremely long duration, uncrewed deep-space probes, but the low thrust of approximately 0.1 N limits rapid total change in velocity (Δv) or crewed missions [6]. Scaling for heavy payloads is currently impractical, positioning it as a niche use case and far off from real-world utilization [8].

3.5 Comparison of Propulsion Options

Chemical propulsion excels in thrust—Raptor engines provide approximately 258 tonnes—but lags in I_{sp} (~ 380 s) compared to NTP (900–1,000 s) and electric systems (3,000+ s) [29, 3, 6].

NTP's efficiency could cut Mars propellant needs by approximately 50%, yet its 2030s timeline delays adoption [3]. Nuclear Electric Propulsion (NEP), merging nuclear power with electric thrusters, promises higher I_{sp} but faces a 2040s horizon [24]. Hybrid systems like VASIMR (Variable Specific Impulse Magnetoplasma Rocket), with $I_{sp} \sim 5{,}000$ s, bridge chemical and electric, but development lags into the 2030s [5]. Experimental options, including photon propulsion (Breakthrough Starshot) and solar sails (IKAROS, LightSail 2), offer theoretical speed but lack practicality for crewed or heavy missions [11, 33]. Chemical propulsion remains the most viable solution for 2025–2035 missions, particularly with in-development reusability and orbital refueling infrastructure [32].

4 Theoretical Foundations

This section establishes the theoretical underpinnings of post-LEO propulsion, focusing on the Tsiolkovsky Rocket Equation, vis-viva equation, orbital mechanics, and propulsion efficiency. These principles guide the design and operation of the Deep Space Super Heavy Booster, a reusable system for delivering payloads to transfer orbits and returning to LEO.

4.1 The Rocket Equation and Its Implications for Post-LEO Propulsion

Understanding the Tsiolkovsky rocket equation is fundamental to evaluating propulsion requirements beyond Low Earth Orbit (LEO). This section establishes the mathematical framework used to calculate mass ratios, propellant needs, and velocity changes for orbital transfers and recovery maneuvers. By outlining the derivation and implications of the rocket equation, it provides the basis for analyzing the feasibility of reusable post-LEO boosters, highlighting how specific impulse and exhaust velocity critically affect mission design and fuel efficiency.

The Tsiolkovsky rocket equation describes the relationship between a rocket's change in velocity and the ratio of its initial and final mass, accounting for conservation of momentum in a system with constant exhaust velocity.

Definitions

Δv Total change in velocity (mission Δv).

I_{sp} Specific impulse of the rocket engine (in seconds).

g_0 Standard gravitational acceleration at sea level ($9.80665\,\mathrm{m/s^2}$).

v_e Effective exhaust velocity ($v_e = I_{sp} \cdot g_0$).

m_0 Initial mass (before propellant is burned).

m_f Final mass (after propellant is expended).

Tsiolkovsky Rocket Equation

$$\Delta v = I_{sp} \cdot g_0 \cdot \ln\left(\frac{m_0}{m_f}\right)$$

Exhaust Velocity Relationship

The exhaust velocity can be expressed in terms of specific impulse:

$$v_e = I_{sp} \cdot g_0$$

Substituting this into the rocket equation yields:

$$\Delta v = v_e \cdot \ln\left(\frac{m_0}{m_f}\right)$$

Mass Ratio Rearrangement

Solving for final mass:

$$
\begin{aligned}
\Delta v &= v_e \cdot \ln\left(\frac{m_0}{m_f}\right) \\
\frac{\Delta v}{v_e} &= \ln\left(\frac{m_0}{m_f}\right) \\
e^{\frac{\Delta v}{v_e}} &= \frac{m_0}{m_f} \\
m_f \cdot e^{\frac{\Delta v}{v_e}} &= m_0 \\
m_f &= \frac{m_0}{e^{\frac{\Delta v}{v_e}}}.
\end{aligned}
$$

Derived from conservation of momentum, this equation shows that thrust results from expelling mass at high velocity, accelerating the rocket [1]. For post-LEO reusable boosters, it quantifies the propellant needed for transfer orbit insertion 2.397–3.576 km/s and boostback of equal magnitude, per Section 6, highlighting fuel efficiency as a design driver [28, 37].

4.2 Mass Ratio and Exponential Fuel Requirements

The mass ratio (MR), defined as $MR = \frac{m_0}{m_f}$, reflects the proportion of mass that must be carried as propellant to achieve a given Δv. The rocket equation's logarithmic term demonstrates the exponential nature of fuel requirements: as Δv increases, the required propellant grows disproportionately [32]. Table 1 illustrates this relationship:

Table 1: Mass Ratio vs. Velocity Ratio

Mass Ratio	$\Delta v/v_e$
2	0.69
3	1.10
4	1.39
10	2.30

For a Raptor Vacuum engine ($v_e = 3.726527\,\text{km/s}$), a Δv of $6.553\,\text{km/s}$ (Jupiter Insertion Burn) requires an MR of $e^{\frac{6.553}{3.7}} \approx 5.9$. This implies that approximately 83% (as shown in Table 2) of the initial mass (m_0) is fuel [29]. This exponential scaling presents a significant limitation for chemical rockets, highlighting the importance of mass optimization and in-orbit refueling for reusable designs [3, 37].

The fuel mass fraction, which quantifies this relationship, is given by:

$$\text{Fuel Mass Percent} = 1 - e^{-\frac{\Delta v}{v_e}}$$

For the Deep Space Super Heavy Booster, the fuel mass percentages for various destinations are shown in Table 2:

Table 2: Fuel Mass Percentage for Deep Space Super Heavy Booster

Destination	Fuel Mass Percent
GEO	47.45%
Lunar	56.27%
Mars	64.82%
Ceres	75.23%
Jupiter	82.77%

These fuel mass fractions, tied to the booster's 3,400-tonne tank, illustrate the trade-offs inherent in high-Δv missions: while these missions require large amounts of propellant, they leave minimal mass for payload and structure. This underscores the need for mass optimization and in-orbit refueling to reset m_0, as well as the development of lightweight materials and efficient propulsion systems [8, 28, 36].

4.3 Vis-viva Equation

The Vis-viva equation is a fundamental orbital mechanics relation that describes the velocity of an orbiting body at a given distance from the central mass. It applies to elliptical, parabolic, and hyperbolic trajectories and is derived from the conservation of mechanical energy [1, 12].

Definitions

$v_{perigee}$ Velocity required at perigee to achieve desired orbit.

μ Standard gravitational parameter of the central body ($\mu = GM$, where G is the gravitational constant and M is the mass of the central body).

r Instantaneous distance from the orbiting body to the center of mass.

a Semi-major axis of the orbit.

General Vis-viva Equation

$$v_{perigee} = \sqrt{\mu \cdot \left(\frac{2}{r} - \frac{1}{a}\right)}$$

Special Cases

Escape Velocity. Escape velocity occurs when the semi-major axis approaches infinity ($a \to \infty$), implying a parabolic trajectory. The Vis-viva equation simplifies to:

$$v_{esc} = \sqrt{\frac{2\mu}{r}}$$

This represents the minimum velocity required to escape the gravitational influence of the central body [7, 36].

Circular Orbital Velocity. For a circular orbit, where $r = a$, the Vis-viva equation simplifies to:

$$v_{circ} = \sqrt{\frac{\mu}{r}}$$

This is the velocity at which an object must travel to maintain a stable circular orbit at a given radius.

For a LEO-to-GEO transfer (Section 6.1), where $r_{LEO} = 6778\,\text{km}$ and $a_{transfer} = 24,471\,\text{km}$, we calculate the velocity at perigee (transfer orbit radius equal to radius at LEO):

$$v_{perigee} = 10.066\,\text{km/s}$$

and the initial LEO circular orbital velocity is:

$$v_{LEO} = 7.669\,\text{km/s}$$

subtracting v_{LEO} from $v_{perigee}$ results in a required Δv of:

$$\Delta v = 2.397\,\text{km/s}$$

This equation is crucial for calculating the required orbital changes, particularly for mission planning involving boostback maneuvers, as the booster relies solely on engine burns without aerobraking [37].

4.4 Specific Impulse and Propulsion Efficiency

Specific impulse (I_{sp}) measures the efficiency of a rocket engine, and is directly related to exhaust velocity (v_e) by the following equation:

$$v_e = I_{sp} \cdot g_0$$

where $g_0 = 9.81\,\text{m/s}^2$ is the standard gravitational acceleration. A higher I_{sp} results in reduced propellant needs for a given mission Δv, which is key to improving the performance and efficiency of space propulsion systems [32].

Table 3 compares the propulsion efficiency of various engine types based on their exhaust velocity and specific impulse:

The Raptor Vacuum's specific impulse of approximately 380 seconds provides a good balance of high thrust—essential for rapid burns—and efficiency. This makes **Methalox** (liquid oxygen and methane) an ideal choice for reusable post-LEO boosters, particularly when combined with

Table 3: Propulsion Efficiency

Propulsion Type	Exhaust Velocity (m/s)	I_{sp} (s)
Solid Rocket	2,500	250 [37]
Liquid Oxygen / RP-1	3,000	300 [37]
Liquid Oxygen / Methane	3,300–3,800	330–380 [30]
Liquid Oxygen / Hydrogen	4,500	450 [37]
Nuclear Thermal	9,000	900 [3]
Ion Thrusters	30,000+	3,000+ [6]

the potential for in-orbit refueling [30] to reduce the mass fraction of propellant for long-duration missions.

4.5 Impact on Post-LEO Booster Design

The rocket equation plays a crucial role in defining the design constraints of a reusable chemical rocket booster intended for payload deployment beyond LEO [32]. Several key considerations directly influenced by this equation are outlined below:

- Δv **Requirements** – The total velocity change (Δv) required for inserting payloads into transfer orbits, such as GEO, lunar, or Mars, and returning the booster to LEO must be optimized. Balancing these requirements with minimal fuel consumption is critical, as inefficient use of propellant directly impacts mission feasibility [37, 12].

- **Mass Optimization** – The dry mass of the booster, including engines, structural components, and propellant tanks, must be minimized to improve the mass ratio. A lower dry mass increases the effectiveness of the propellant used, improving the overall mission performance [28, 38].

- **Refueling Considerations** – The ability to refuel in orbit is a game changer for post-LEO missions. This capability effectively resets the initial mass (m_0), reducing the required launch fuel and increasing the payload capacity. Orbital refueling also enables the booster to perform multiple mission cycles without needing to return to Earth for a full refuel, significantly enhancing reusability [10, 29].

- **Engine Selection** – The selection of engines with high specific impulse (Isp) is critical for balancing performance and reusability. A methane-oxygen engine, such as SpaceX's Raptor engine, offers an optimal balance between efficiency and the ability to perform multiple burns over the course of a mission. The Raptor engine's reusability factor is particularly important for reducing operational costs in post-LEO missions [14, 29].

Since aerobraking is not considered in this design, all Δv requirements for the return leg to LEO must be met through engine burns alone (Wertz et al., 2011). This necessitates a well-optimized fuel budget and highlights the need for efficient propulsion systems [32, 38].

In summary, the Tsiolkovsky Rocket Equation serves as the foundation for evaluating post-LEO propulsion strategies. Its exponential nature emphasizes the challenges related to fuel efficiency, mass constraints, and reusability—factors that must be carefully considered when designing a reusable chemical booster for deep-space missions [28, 32].

4.6 Orbits

Spacecraft orbits vary by mission, and the corresponding Δv values are critical for mission planning and booster design. Each orbit has different requirements for velocity change, and understanding these is key to mission success.

- **LEO (Low Earth Orbit) (160–2,000 km):** LEO is commonly used for satellite deployment and space station operations. It requires a low Δv for circular orbit insertion ($\sim 7.67\,\text{km/s}$). This is often the first step in interplanetary missions, serving as a staging point for further trajectory burns [37].

- **GEO (Geostationary Earth Orbit) (35,786 km):** GEO is used for communications and weather satellites. Reaching GEO requires a Δv of $\sim 2.397\,\text{km/s}$ from LEO. This orbit is unique because satellites in GEO appear stationary relative to Earth [37].

- **Lunar Orbit:** The transfer to lunar orbit from LEO requires a Δv of $\sim 3.083\,\text{km/s}$, with an additional $\sim 0.9\,\text{km/s}$ for lunar orbit insertion. These values are based on the Hohmann transfer trajectory, which is an energy-efficient path to the Moon [37].

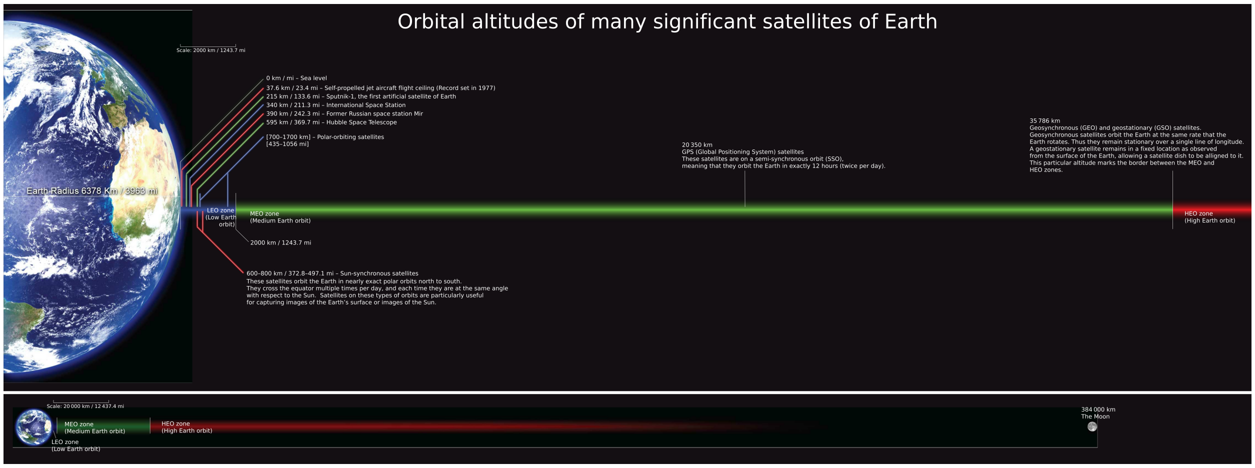

Figure 1: Comparison of different spacecraft orbits including LEO, GEO, and lunar orbit. Source: `https://commons.wikimedia.org/w/index.php?curid=16302497`.

- **Interplanetary:** Hohmann transfers are used for deep-space missions, providing the most energy-efficient route between planets. These transfers minimize the fuel required for interplanetary travel [37].

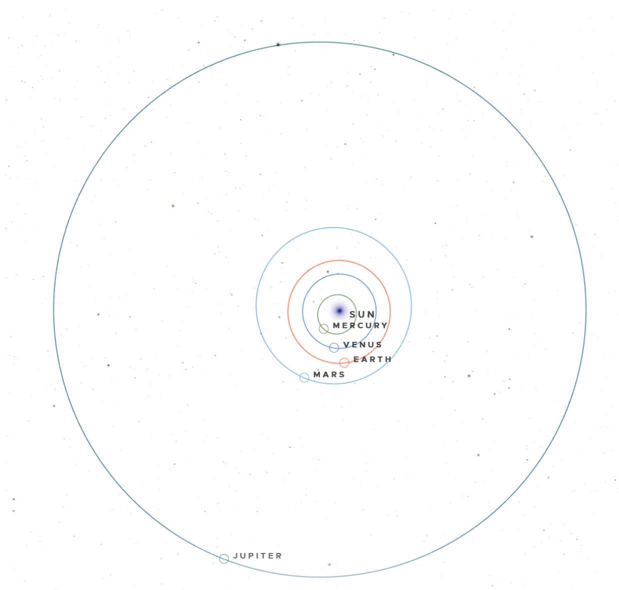

Figure 2: The Orbits of the planets around the sun up to Jupiter. Source: `https://eyes.nasa.gov/apps/solar-system/#/sun`.

4.7 Delta-V Requirements

Key transitions between orbits and planetary destinations have specific Δv requirements, as summarized below:

- **LEO to GEO:** $\sim 2.397\,\mathrm{km/s}$ (insertion only). This Δv value is for moving from LEO into a geosynchronous orbit, where a satellite can remain fixed over one point on Earth [37].
- **LEO to Moon:** $\sim 3.083\,\mathrm{km/s}$ (transfer). This is the velocity required to transfer from Earth to the Moon using a Hohmann transfer trajectory [37].
- **LEO to Mars:** $\sim 3.892\,\mathrm{km/s}$ (Hohmann injection). A Hohmann transfer is typically used for Mars missions, requiring a precise burn to inject the spacecraft on an elliptical path toward Mars [37].
- **LEO to Ceres:** $\sim 5.199\,\mathrm{km/s}$ (Hohmann injection). This transfer to the asteroid belt follows a similar trajectory as the Mars transfer, but with a higher Δv due to the greater distance [37].
- **LEO to Jupiter:** $\sim 6.553\,\mathrm{km/s}$ (Hohmann injection). Jupiter missions also use Hohmann transfers, with the Δv increasing as the distance to the outer solar system increases [37].
- **Deep Space:** Escape velocity ($\sim 11.2\,\mathrm{km/s}$) is required for a spacecraft to break free from Earth's gravitational influence. Additional Δv may be needed for interplanetary transfers, such as $4-6\,\mathrm{km/s}$ for a mission to Ceres [37].

For reusable boosters, the Δv values for boostback (return to LEO) are typically doubled compared to those for orbital insertion or interplanetary transfers. For example, the round-trip Δv for Mars missions is $\sim 7.784\,\mathrm{km/s}$, accounting for both the departure and return [37].

4.8 Retrograde Burn

A retrograde burn involves firing the spacecraft's engines in the opposite direction of its motion, resulting in a decrease in velocity. This maneuver is commonly used for orbit lowering or returning the spacecraft to a lower orbit or the surface. For reusable chemical boosters, such as those used in Mars missions, a retrograde burn at the perigee of the Low Earth Orbit (LEO) trajectory can be used to reduce the spacecraft's velocity, ensuring a safe return to orbit and avoiding the stresses associated with atmospheric reentry.

To restore a circular orbit and prevent reentry, a burn equal to the insertion Δv (for example, $3.892\,\text{km/s}$ for Mars) is typically performed at the perigee of the LEO trajectory. This is crucial because the booster must not enter the atmosphere at steep angles or excessive speeds, which would result in high thermal and mechanical stresses that could damage the vehicle. By conducting the retrograde burn at the appropriate location and velocity, the booster can return to a safe circular orbit without incurring these dangers.

Furthermore, the precise timing of retrograde burns is essential to optimizing fuel usage and turnaround time. Rapid reuse of the booster is one of the most critical aspects of cost reduction for space missions, and the efficiency of each burn significantly impacts the overall mission success. Therefore, the booster must execute these burns with high precision to minimize fuel consumption while ensuring that it meets the necessary trajectory corrections. Such precision timing is particularly important for booster turnaround as rapid and efficient reuse is fundamental to the economic viability of space missions, especially for interplanetary travel [28].

Key Considerations for Retrograde Burns:

- **Engine Thrust and Burn Duration:** The thrust level of the engines and the duration of the burn must be precisely calculated to achieve the required Δv for orbit restoration. An insufficient burn may result in insufficient velocity reduction, while an excessive burn could result in overshooting the desired orbit.

- **Orbital Mechanics:** Retrograde burns must be executed at specific orbital locations, such as at perigee for circular orbits, to ensure that the spacecraft maintains the correct trajectory and avoids reentering the atmosphere.

- **Optimization of Fuel Usage:** Since fuel is limited on any space mission, the retrograde burn must be optimized to balance fuel efficiency with mission requirements. Accurate prediction of fuel consumption is essential for planning subsequent mission stages, especially in long-duration interplanetary missions [4].

4.9 Propulsion Considerations

Chemical propulsion systems are favored for high-thrust applications, essential for launching large payloads and providing the required velocity changes (Δv) in a short period of time. Among the most prominent chemical engines are:

- **LOX/Methane (Raptor):** SpaceX's Raptor engine, which utilizes liquid oxygen (LOX) and methane, delivers a specific impulse (I_{sp}) of approximately 380 seconds at sea level, with a vacuum exhaust velocity (v_e) of around 3.7 km/s. It also provides a high thrust-to-weight ratio (285 tonnes-force per engine) which is crucial for the rapid acceleration required for booster launches and maneuvers. Its design optimizes fuel efficiency while ensuring the performance needed for space travel [29].

- **LOX/Hydrogen:** The combination of liquid oxygen (LOX) and liquid hydrogen (LH2) is another common choice for high-performance engines, offering an impressive specific impulse of around 450 seconds. While the LH2/LOX engine provides greater efficiency than LOX/Methane engines, it sacrifices thrust due to the lower density of liquid hydrogen. This, combined with the extreme difficulty in storing LH2 long-term, presents significant operational challenges. Moreover, the engine's complexity and lower thrust make it less suited for rapid turnaround missions compared to more modern alternatives like the Raptor engine [37].

- **Nuclear/Ion Propulsion:** While nuclear and ion engines are more efficient for deep-space missions due to their ability to provide higher specific impulses over extended periods, they lack the immediacy of chemical engines. Nuclear engines, for instance, provide better fuel efficiency but face significant hurdles, including radiation protection and thermal management, while ion engines are primarily used for low-thrust, long-duration missions where high efficiency is more critical than rapid acceleration [3, 6].

The Raptor Vacuum engine, in particular, aligns with the growing trend of Methalox refueling and Starship compatibility, making it a viable option for reusable missions that require frequent launches and rapid turnaround times. This further enhances the feasibility of reusing boosters for both low Earth orbit (LEO) and deep-space missions [29].

5 System Design and Engineering Considerations

This section outlines the design and engineering principles of the Deep Space Super Heavy Booster, a reusable propulsion system tailored for post-LEO missions. Key considerations include booster configuration, reusability challenges, guidance systems, and multi-booster architectures, balancing efficiency, flexibility, and scalability for deep-space operations.

Table 4: Starship and Deep Space Booster Specifications

Parameter	Starship [29]	Deep Space Booster
Fuel Mass	1,500 tonnes	3,400 tonnes [29]
Dry Weight	150 tonnes	250 tonnes
Payload	150 tonnes	-
Engine Count	6 Raptor 2	9 Raptor 2 Vacuum
Total Initial Mass	1,800 tonnes	3,650 tonnes
Thrust [13]	230 tf 258 tf	258 tf
Specific Impulse (I_{sp})	355 s	380 s
Exhaust Velocity (v_e)	3,481.361 m/s	3,726.527 m/s

5.1 Booster Design

The Deep Space Super Heavy Booster mirrors SpaceX's Super Heavy in scale, 9 meters in diameter and 70 meters tall, offering a robust, modular platform [29]. Its size supports adjustable fuel tank configurations during assembly, enabling optimization for specific mission profiles or future improvements [28]. Operating solely in space, the booster eliminates grid fins, reducing mass and complexity compared to atmospheric reentry designs [29].

Propulsion relies on 9–10 Raptor Vacuum (RVac) engines, replacing the 33 sea-level Raptors of the Super Heavy [29]. RVac engines, optimized for vacuum efficiency with a specific impulse of roughly 380 s for Raptor 2, with potential Raptor 3 upgrades to 400 s, enhance performance for in-space burns [13, 14, 29]. Fuel selection aligns with the Starship system: liquid methane (CH_4) and liquid oxygen (LOX), or Methalox [29]. This choice ensures high thrust-to-weight, compatibility with already planned in-orbit refueling, and future in-situ resource utilization potential on Mars, where CH_4 can be synthesized from CO_2 and H_2O [38].

The structural design leverages stainless steel for durability, cost-effectiveness, and existing production methods [29]. Modular payload attachments, or ”toppers,” enable flexibility, supporting deep-space transport, cargo deployment, or station transfers, adaptable via standardized interfaces [28]. This modularity positions the booster as a versatile workhorse for diverse missions.

5.2 Reusability Challenges

Unlike Earth-returning boosters, the Deep Space Super Heavy avoids atmospheric reentry, negating the need for extensive thermal protection seen on Starship [29]. Stainless steel's inherent heat resistance suffices for space conditions, simplifying maintenance. Future variants might incorporate aerobraking for specific missions, lunar slingshot returns, or Mars orbit insertion, but this study focuses on vacuum-only operations [37].

Fuel constraints dominate reusability, driven by transfer orbit insertion and boostback burns. Total fuel capacity is 3,400 tonnes (CH_4 and LOX combined), sufficient for high-v missions, 7.786 km/s for Mars, per Section 6 [29]. For long-duration missions, boil-off insulation could mitigate fuel loss, though short-term operations render this optional [37]. Micrometeoroid shielding is discretionary, as stainless steel withstands minor impacts, with repairs at an orbital station. In the future, large stations could shield stored boosters externally [28].

A LEO refueling and docking station enables rapid reuse. Initially, Starship tankers can deliver Methalox via multiple flights, as planned for NASA's Artemis program [16]. A dedicated station, pre-stocked with at least 3,400 tonnes for one booster and 1,500 tonnes of fuel for a Starship, streamlines operations. Thermal insulation on the station prevents boil-off, ensuring availability [37]. This infrastructure reduces turnaround time, minimizes waste, and scales to support multiple boosters [28].

5.3 Guidance and Control

Operating in microgravity, the booster minimizes engine gimbaling. Only 3–4 central RVac engines retain limited gimbaling for fine adjustments, reducing weight [29]. Primary attitude control relies on cold-gas nitrogen retro thrusters at the booster's top and bottom, pending a proven alternative, Methalox RCS [37]. These proven thrusters handle payload deployment, flip maneuvers for boostback, and station-keeping during docking, ensuring precision with minimal refurbishment [28].

For initial launches from Earth, an aero cover protects the booster and payload, detaching in space and reentering the atmosphere for recovery [29]. Future iterations will adopt modular toppers in orbit for mission-specific payloads, including multi-booster setups (see section 5.4) or tug-assisted docking [28]. Boosters remain on standby at the refueling station, enabling rapid redeployment once fueled and paired with a payload. On-site repairs extend lifespan, while storing multiple boosters supports concurrent launches, minimizing downtime and maximizing flexibility [28].

5.4 Multi-Booster Configuration

For missions exceeding a single booster's capacity, like 2,500-tonne space stations, large interplanetary payloads, or missions with a destination booster, multi-booster configurations distribute v and mass requirements [37]. Two or more Deep Space Super Heavy Boosters operate in tandem, each with 3,400 tonnes of fuel and 9–10 RVac engines, synchronized during the LEO perigee burn [29]. For example, a 2,500-tonne payload to lunar orbit could split insertion mass of payload across the two boosters:

- Single Booster Limit: With a dry mass of 250 tonnes, 3,400 tonnes fuel, and $v_e = 3.726527$ km/s, one booster delivers 1,798 tonnes payload for 3.082 km/s insertion with payload and 3.082 km/s booster-only return burn, per Tsiolkovsky's equation [32]; [29].
- Two Boosters: Each handles 1,798 tonnes payload, doubling payload capacity to 3,596 tonnes to the same 3.082 km/s, leaving fuel for individual boostback while doubling the payload. Total fuel: 6,800 tonnes across two boosters [37]; [29].

Synchronization requires precise burn timing, achieved via retro thrusters and limited gimbaling [37]. Post-insertion, boosters separate simultaneously, releasing the payload to coast while each performs its own boostback burn [28]. Challenges include structural coupling, via a reinforced topper linking boosters, and trajectory alignment. These are mitigated by station-based, LEO assembly and pre-launch simulations [28]. Benefits include scalability; three boosters could lift 5,394 tonnes to lunar orbit, enabling pre-built stations or habitats for lunar, Martian, or deep-space destinations like Ceres or Jupiter [37]. Tug-assisted docking could refine payload insertion, reducing booster v needs.

6 Calculations

This section quantifies the Δv budgets for post-Low Earth Orbit (LEO) missions using transorbital boosters, focusing on the insertion of payloads into transfer orbits to Geostationary Orbit (GEO), the Moon, and Mars, and the subsequent boostback of reusable boosters to LEO. Both expendable and reusable configurations are evaluated, analyzing fuel requirements, payload capacities, and operational trade-offs. Calculations are grounded in orbital mechanics, utilizing the vis-viva equation for velocity changes and the Tsiolkovsky Rocket Equation for propellant estimation [37]. The gravitational parameter for Earth ($\mu = 3.986 \times 10^{14}\ \mathrm{m}^3/\mathrm{s}^2$) underpins all computations [37]. Specific Δv values and equations are presented using LaTeX.

6.1 Maximum Payload for Expendable and Reusable Missions

The Δv required to insert payloads into Hohmann Transfer orbits from LEO (400 km altitude, circular velocity at LEO ≈ 7.67 km/s) to GEO, the Moon, and Mars is calculated using the vis-viva equation [37]. The booster performs a single prograde burn at LEO perigee to achieve the transfer orbit's perigee velocity, after which the payload separates and coasts to its destination. The Earth's gravitational parameter ($\mu = 3.986 \times 10^{14}$ m^3/s^2) drives these computations, assuming minimum-energy elliptical trajectories [37]. The overall calculation chain proceeds by inputting known orbital parameters, applying the vis-viva equation to determine the transfer orbit velocity, deriving the fuel mass fraction, determine initial and final Masses using the Tsiolkovsky Rocket Equation, and finally calculating the available payload mass based on fuel constraints.

Several of the velocities presented in Tables 5 and 6 are calculated values, derived using standard orbital mechanics formulas such as vis-viva and Hohmann transfer approximations, demonstrated earlier in this section. Where not explicitly shown, calculations follow identical methods to the examples provided.

Table 5: Earth System Values

Earth System	Values
Gravitational Parameter Earth	398,600 km^3/s^2 [25][21]
Radius	6,378 km [25][21]
LEO Orbit	6,778 km
Velocity LEO	7.6686 km/s
GEO Orbit	42,164 km
LEO to GEO Perigee Velocity	10.0661 km/s
Moon Orbit	384,400 km [22]
LEO to Moon Perigee Velocity	10.7507 km/s
Escape Velocity	11.1799 km/s [21]

Table 6: Solar System Values

System	Values
Gravitational Parameter Sun	132,700,000,000 km^3/s^2 [25]
Earth Orbit	149,600,000 km [25][23]
Velocity Earth	29.7831 km/s [25][23]
Mars Orbit	227,990,400 km [25][23]
Earth to Mars Perigee Velocity	32.72898 km/s
Earth to Mars Perigee Velocity Earth Centered	11.5616 km/s
Ceres Orbit	418,880,000 km [25][23]
Earth to Ceres Perigee Velocity	36.15529 km/s
Earth to Ceres Perigee Velocity Earth Centered	12.86846 km/s
Jupiter Orbit	777,920,000 km [25][23]
Earth to Jupiter Perigee Velocity	38.57363 km/s
Earth to Jupiter Perigee Velocity Earth Centered	14.22202 km/s

- **Lunar Transfer:** For a lunar transfer (Other Earth Centric Destinations use this method).

This section calculates the Δv and fuel requirements for a translunar injection from LEO, using Earth as the central body.

Key Parameters

$v_{perigee}$ Orbital velocity at perigee required to reach destination orbit.

$\mu = 398{,}600\ \mathrm{km}^3/\mathrm{s}^2$ Earth's gravitational parameter.

$r = 6{,}778$ km Radius of LEO (starting orbit).

$a = 195{,}589$ km Semi-major axis of the elliptical transfer orbit.

$v_{LEO} = 7.6686$km/s Velocity at LEO Table 5

$v_e = 3,726.53$m/s Exhaust velocity of engines Table 4

m_0 Total Initial Mass

m_f Total Final Mass

m_p Payload Mass

Vis-viva Equation

$$
\begin{aligned}
v_{perigee} &= \sqrt{\mu \cdot \left(\frac{2}{r} - \frac{1}{a}\right)} \\
&= \sqrt{398{,}600 \cdot \left(\frac{2}{6{,}778} - \frac{1}{195{,}589}\right)} \\
&= 10.75071 \text{ km/s}
\end{aligned}
$$

Required Δv from LEO

$$
\begin{aligned}
\Delta v &= v_{perigee} - v_{\text{LEO}} \\
&= 10.75071 - 7.6686 \\
&= 3.08208 \text{ km/s} = 3{,}082.08 \text{ m/s}
\end{aligned}
$$

Fuel Mass Percent, using $v_e = 3,726.53$ is computed as follows:

$$
\begin{aligned}
\text{FMP} &= 1 - e^{-\frac{\Delta v}{v_e}} \\
&= 1 - e^{-\frac{3{,}082.08}{3{,}726.53}} \\
&= 0.56267 = 56.267\%
\end{aligned}
$$

Fuel Mass for Boostback, using $m_f = 260$ is calculated as:

$$\begin{aligned} m_0 &= m_f \cdot e^{\frac{\Delta v}{v_e}} \\ &= 260 \cdot e^{\frac{3{,}082.08}{3{,}726.53}} \\ &= 594.52 \text{ tonnes} \\ \text{Boostback Fuel Mass} &= m_0 - m_f = 334.52 \text{ tonnes} \end{aligned}$$

Mass Budget Without Return

$$\begin{aligned} m_0 &= \frac{\text{Initial Fuel Mass}}{\text{FMP}} = \frac{3{,}400}{0.56267} = 6{,}042.62 \text{ tonnes} \\ m_p &= m_0 - 3{,}400 - 250 = 2{,}382.62 \text{ tonnes} \end{aligned}$$

Mass Budget With Return

$$\begin{aligned} m_{fuel\ with\ return} &= \text{Initial Fuel Mass} - \text{Boostback Fuel Mass} - \text{Circularization Fuel Mass} \\ m_{fuel\ with\ return} &= 3{,}400 - 334.52 - 10 = 3{,}055.48 \text{ tonnes} \\ m_0 &= \frac{3{,}055.48}{0.56267} = 5{,}430.34 \text{ tonnes} \\ m_p &= 5{,}430.34 - 3{,}400 - 250 = 1{,}780.34 \text{ tonnes} \end{aligned}$$

- **Mars Transfer:** For a trans-Mars injection (Other Sun Centric Destinations use this method).

  This section calculates the required Δv and fuel budgets for a heliocentric transfer from Earth to Mars.

  Key Parameters

$v_{perigee}$ Orbital velocity at perigee in heliocentric transfer orbit.

$\mu = 132.7 \times 10^9 \text{ km}^3/\text{s}^2$ Gravitational parameter of the Sun.

$r = 149{,}600{,}000$ km Earth's orbital radius.

$a = 188{,}795{,}200$ km Semi-major axis of transfer orbit to Mars.

$v_{\text{escape}} = 11{,}561.59$ m/s Earth escape velocity Table 5.

$v_e = 3,726.53$m/s Exhaust velocity of engines Table 4.

$m_{bf} = 260$ tonnes Booster Dry Mass and circularization fuel mass.

$v_{Earth} = 29.78308$km/s Earth's orbital velocity around the sun Table 6.

v_∞ Difference of planet orbital velocities.

Vis-viva Equation

$$\begin{aligned} v_{perigee} &= \sqrt{\mu \cdot \left(\frac{2}{r} - \frac{1}{a}\right)} \\ &= \sqrt{132.7 \times 10^9 \cdot \left(\frac{2}{149.6 \times 10^6} - \frac{1}{227.99 \times 10^6}\right)} \\ &= 32.72898 \text{ km/s} \end{aligned}$$

Earth Departure Velocity, using v_{Earth}=29.78308 is computed as:

$$v_\infty = v_{perigee} - v_{Earth}$$

$$v_\infty = 32.72898 - 29.78308 = 2.94589 \text{ km/s}$$

$$\begin{aligned} v_{\text{perigee (earth-centered)}} &= \sqrt{v_\infty^2 + v_{\text{escape}}^2} \\ &= \sqrt{(2{,}945.89)^2 + (11{,}179.98)^2} = 11{,}561.59 \text{ m/s} \end{aligned}$$

Required Δv from LEO, using current velocity of 7,668.6 is:

$$\begin{aligned} \Delta v &= \text{Desired Velocity} - \text{Current Velocity} \\ &= 11{,}561.59 - 7{,}668.6 \\ &= 3{,}892.96 \text{ m/s} \end{aligned}$$

Fuel Mass Percent

$$\begin{aligned} \text{FMP} &= 1 - e^{-\frac{\Delta v}{v_e}} \\ &= 1 - e^{-\frac{3{,}892.96}{3{,}726.53}} \\ &= 0.64819 = 64.819\% \end{aligned}$$

Fuel Mass for Boostback

$$
\begin{aligned}
m_0 &= 260 \cdot e^{\frac{3{,}892.96}{3{,}726.53}} = 739.03 \text{ tonnes} \\
\text{Fuel Mass} &= m_0 - m_{bf} \\
&= 739.03 - 260 = 479.03 \text{ tonnes}
\end{aligned}
$$

Mass Budget Without Return

$$
\begin{aligned}
m_0 &= \frac{\text{Initial Fuel Mass}}{\text{FMP}} = \frac{3{,}400}{0.64819} = 5{,}245.38 \text{ tonnes} \\
m_p &= 5{,}245.38 - 3{,}400 - 250 = 1{,}595.38 \text{ tonnes}
\end{aligned}
$$

Mass Budget With Return

$$
\begin{aligned}
m_{fuel\ with\ return} &= \text{Initial Fuel Mass} - \text{Boostback Fuel Mass} - \text{Circularization Fuel Mass} \\
m_{fuel\ with\ return} &= 3{,}400 - 479.03 - 10 = 2{,}910.97 \text{ tonnes} \\
m_0 &= \frac{2{,}910.97}{0.64819} = 4{,}490.92 \text{ tonnes} \\
m_p &= 4{,}490.92 - 3{,}400 - 250 = 840.92 \text{ tonnes}
\end{aligned}
$$

These Δv values represent both the expendable case, where the booster's role ends after insertion. For reusable configurations, additional boostback Δv is required, where the booster must boostback the same Δv for rapid reuse.

In reusable configurations, the booster separates from the payload at LEO perigee immediately after the insertion burn and performs a retrograde burn to return to its original circular

Table 7: Δv to Various Destinations

Destination	Δv (km/s)	Δv (m/s)
GEO	2.39751	2,397.51
Moon	3.08208	3,082.08
Mars	3.89296	3,892.96
Ceres	5.19983	5,199.83
Jupiter	6.55339	6,553.39

Table 8: Mass Booster and Payload Calculations

Parameter	GEO	Lunar	Mars	Ceres	Jupiter
Retrograde Burn m_0 (tonnes)	494.75	594.52	739.03	1,049.47	1,509.09
m_0 Fuel Mass Retrograde Burn (tonnes)	244.75	344.52	489.03	789.47	1,249.09
Initial Fuel Mass Percent	0.4745	0.5627	0.6482	0.7523	0.8277
Total Initial Mass (No Return) (tonnes)	7,165.78	6,042.62	5,245.38	4,519.75	4,107.72
Payload (No Return) (tonnes)	3,515.78	2,392.62	1,595.38	859.75	457.72
Burnable Fuel Mass (Return) (tonnes)	3,145.25	3,055.48	2,910.97	Not Possible	Not Possible
Total Initial Mass (Return) (tonnes)	6,628.88	5,430.34	4,490.92	Not Possible	Not Possible
Payload (Return) (tonnes)	2,978.88	1,780.34	840.91	Not Possible	Not Possible

LEO orbit [37]. This boostback maneuver cancels the excess velocity imparted during transfer insertion, requiring a Δv equal to the insertion burn, as both occur at the same or very similar altitude ($r_{\text{LEO}} = 6{,}778$ km) [37].

No additional or only a very minor circularization burn is needed, as the retrograde burn occurs at or very near LEO altitude, directly restoring the circular orbit [37]. Propellant margins of 0.29 (10 tonnes) of total fuel ensure precision and control [29]. Total Δv scales with mission energy: GEO (4.794 km/s), Moon (6.164 km/s), and Mars (7.786 km/s) [37].

6.2 Deriving an Equation for Maximum Δv with Return Capability

In this section, we derive the maximum available change in velocity (Δv_m) for a reusable booster system that must not only perform an insertion burn but also retain sufficient propellant for a return maneuver. The analysis accounts for both the outbound and return phases of flight, with constraints imposed by the final mass required for controlled recovery. This is a modified version of the Tsiolkovsky Rocket Equation discussed in Section 4.1.

Definitions

The following variables are used in the derivation:

Δv_m Maximum change in velocity.

v_e Effective exhaust velocity of the propulsion system.

m_0 Total initial mass of the integrated system (booster and payload).

m_{B_0} Initial mass of the booster.

m_p Mass of the payload.

m_{bb} Booster mass remaining after the insertion burn, required for return.

m_{ib} Total system mass after insertion burn, defined as $m_p + m_{bb}$.

m_f Final mass of the booster after the return maneuver is complete.

Derivation

The total mission Δv is constrained by both the insertion and return phases. The change in velocity for the insertion burn can be expressed using the Tsiolkovsky rocket equation as follows:

$$\Delta v_m = v_e \cdot \ln\left(\frac{m_0}{m_{ib}}\right) = v_e \cdot \ln\left(\frac{m_p + m_{B_0}}{m_p + m_{bb}}\right)$$

The return maneuver likewise consumes propellant, and its corresponding Δv is:

$$\Delta v_m = v_e \cdot \ln\left(\frac{m_{bb}}{m_f}\right)$$

Equating the two expressions for Δv_m and solving for m_{bb} yields:

$$v_e \cdot \ln\left(\frac{m_{bb}}{m_f}\right) = v_e \cdot \ln\left(\frac{m_p + m_{B_0}}{m_p + m_{bb}}\right)$$

$$\frac{m_{bb}}{m_f} = \frac{m_p + m_{B_0}}{m_p + m_{bb}}$$

$$m_{bb}(m_p + m_{bb}) = (m_p + m_{B_0})m_f$$

$$m_{bb}^2 + m_p \cdot m_{bb} - (m_p + m_{B_0})m_f = 0$$

This quadratic equation in m_{bb} can be solved analytically. The positive root yields the required booster mass at the end of the insertion burn:

$$m_{bb} = \frac{-m_p + \sqrt{m_p^2 + 4(m_p + m_{B_0})m_f}}{2}$$

Substituting this value back into the original insertion equation provides the maximum achievable Δv for a mission requiring return capability:

$$\Delta v_m = v_e \cdot \ln\left(\frac{m_p + m_{B_0}}{m_p + m_{bb}}\right)$$

Substituting the closed-form solution for m_{bb} directly into the insertion equation yields the complete expression for maximum Δv with return capability:

$$\Delta v_m = v_e \cdot \ln\left(\frac{m_p + m_{B_0}}{m_p + \left(\frac{-m_p + \sqrt{m_p^2 + 4(m_p + m_{B_0})m_f}}{2}\right)}\right)$$

This formulation ensures that the vehicle retains adequate propellant for both outbound delivery and recovery, enabling a reusable flight profile given only the initial mass values. So values for maximum Δv of boosting a fully loaded Starship or any other vehicle can be calculated. Pluging in an initial booster mass of 3650 tonnes and a payload of a fully loaded Starship with a mass of 1,800 tonnes. The max Δv with return is 3068.25 m/s (Nearly a complete lunar transfer insertion). Fuel mass saved depends on destination.

Table 9: Starship Fuel Mass Savings to Various Destinations

Destination	Fuel Mass (tonnes)
Moon	1,006.14
Mars	809.39
Ceres	569.97
Jupiter	396.38

6.3 Cost and Efficiency Trade-Offs

The trade-offs between expendable and reusable boosters are assessed through Δv budgets, propellant use, and operational impacts [32]. Expendable boosters, requiring only the insertion Δv (GEO: 2.397 km/s, Moon: 3.082 km/s, Mars: 3.893 km/s), maximize payload capacity by expending all propellant on the transfer burn [37]. For example, an expendable trans-orbital booster could deliver 1,595 tonnes to Mars transfer, compared to a reusable design at 840 tonnes, where 479 tonnes of propellant mass is reserved for boostback [29, 32]. The Tsiolkovsky Rocket Equation illustrates this:

$$\Delta v = I_{sp} \cdot g_0 \cdot \ln\left(\frac{m_0}{m_f}\right) \tag{1}$$

Here, reusable boosters increase m_f (dry mass plus return propellant), reducing payload fraction.

Reusability, however, yields long-term savings. A reusable booster with total Δv of 4.794–7.786 km/s, refueled in LEO via orbital depots, could amortize its $100M cost over 20 or more flights, versus expendable boosters requiring new builds per launch [9]. Operational complexity increases—two burns at LEO demand precise timing and refueling infrastructure—but current emerging technology, such as SpaceX's Starship, supports this approach [29]. In-orbit refueling would offset payload penalties, enhancing reusability's viability for sustained operations.

7 Implementation & Future Applications

The development of transorbital boosters—reusable chemical propulsion systems designed to accelerate payloads from Low Earth Orbit (LEO) to transfer velocities—offers significant potential for both near-term and long-term spaceflight applications. This section explores the feasibility of integrating such systems into existing programs, their synergy with emerging space infrastructure, and their role in advancing interplanetary exploration.

7.1 Feasibility of Near-Term Implementation

Transorbital boosters could revolutionize near-term programs like NASA's Lunar Gateway Station and SpaceX's Starship missions to Mars. The Lunar Gateway, a planned modular station in lunar orbit, is currently slated for piecewise delivery and assembly in a Near-Rectilinear Halo Orbit (NRHO), relying on multiple launches—via the Space Launch System (SLS) or, more likely, commercial rockets like Starship or Falcon Heavy—to transport components [19]. This approach introduces logistical challenges: repairs, replacements, or upgrades require additional launches from Earth, with transit times of days to weeks due to the $\sim$384,400 km distance [37]. A transorbital booster system mitigates these issues by enabling full assembly of the Gateway in LEO at 400 km altitude, where it is accessible to crewed missions via Crew Dragon, Starship, or Starliner, and robotic servicers [28]. Once assembled, tested, and repaired, the station could be accelerated as a single unit to lunar orbit using a transorbital booster [37]. This approach reduces mission risk, cuts assembly time, and leverages LEO's proximity to Earth's industrial base.

For SpaceX's Mars missions, transorbital boosters enhance payload capacity and safety. Starship's baseline design relies heavily on aerobraking to decelerate from a hyperbolic entry trajectory into Mars orbit or for direct landing, imposing thermal and structural stresses that limit reusability and safety margins. A transorbital booster would allow a higher fuel mass on Starship to a Mars transfer orbit, decreasing the Δv burn requirement from LEO [32]. This reserves fuel for a retrograde burn prior to entry. For example, a 1 km/s retrograde burn at Mars could lower perigee velocity, reducing aerobraking demands and enabling a safer, controlled descent [37]. This fuel efficiency stems from the Tsiolkovsky Rocket Equation: by offloading initial acceleration to the booster, Starship's mass ratio $\left(\frac{m_0}{m_f}\right)$ improves, saving hundreds of metric tons of propellant per mission [32]. Such savings are critical for crewed missions and large payloads, where redundancy and safety are paramount [28].

7.2 Potential Integration with Space Infrastructure

Transorbital boosters align seamlessly with emerging space infrastructure, such as orbital refueling depots and space tugs. Orbital depots, like those proposed by SpaceX for Starship refueling, could supply liquid oxygen and methane to boosters stationed in LEO, enabling rapid turnaround for multiple missions [29]. A booster could refuel in LEO, accelerate a payload to transfer velocity, and return to LEO for reuse, leveraging a retrograde burn to circularize its orbit.

Another promising element in this infrastructure is the use of Lagrange points—regions of gravitational stability between celestial bodies—for staging missions or hosting depots. For example, Earth–Moon L1 and L2 offer stable vantage points for lunar operations or waystations en route to deep space. Their predictable orbital mechanics reduce stationkeeping demands and enable efficient transfer trajectories to lunar or Martian destinations [18]. These points could serve as intermediate hubs where transorbital boosters deliver payloads, which are then relayed further by space tugs or interplanetary stages.

Space tugs—small reusable spacecraft for orbital maneuvering—could complement this system by fine-tuning payload trajectories post-booster separation, reducing the booster's Δv burden [37]. For instance, a tug could perform the final circularization burn at a destination orbit, allowing the booster to focus on the initial high-energy transfer [28]. This integration enhances operational efficiency and scalability, leveraging infrastructure already under development for NASA's Artemis program and commercial ventures [16].

7.3 Long-Term Prospects for Reusable Chemical Boosters in Interplanetary Missions

In the long term, reusable chemical boosters could enable ambitious interplanetary architectures, particularly for multi-booster missions. Consider a large space station—a 1,500-tonne habitat for Mars or deep space—requiring multiple boosters to provide the initial Δv from LEO to transfer velocity, while a dedicated deceleration booster, launched with the station, performs a retrograde burn at the destination (1–2 km/s) to enter Mars orbit [37]. This modular approach scales payload capacity beyond single-launch limits and leverages chemical propulsion's maturity, high thrust, and reliability over emerging technologies like electric propulsion [32]. For missions to the Jupiter system or beyond, boosters could stage payloads to Earth escape, with subsequent transfers handled by nuclear-electric systems, optimizing the trade-off between thrust and specific impulse [24]. The reusability of these boosters—proven by systems like Falcon 9—promises cost reductions, making large-scale interplanetary infrastructure feasible [9].

8 Conclusion

8.1 Summary of Key Findings

This study demonstrates that transorbital boosters can significantly enhance spaceflight capabilities by drastically increasing the available propellant mass for key orbital transfers. By applying the Tsiolkovsky Rocket Equation to Hohmann transfers and retrograde maneuvers, it is shown that reusable chemical propulsion can bridge the gap between current launch capabilities and future interplanetary ambitions. These boosters offer a practical and scalable solution to enhance payload delivery in both cislunar and deep-space missions.

8.2 Max Payload Capacities to Key Destinations

The payload capacity of the Deep Space Super Heavy Booster varies significantly depending on the destination and mission profile. Based on the calculations outlined in Section 6, two cases are considered: maximum payload delivery without return capability (expendable) and payload delivery with booster recovery (reusable).

For geostationary transfer orbit (GEO), the booster can deliver approximately 3,515.78 tonnes in an expendable configuration, or approximately 2,978.88 tonnes with return capability. For lunar transfer orbits, the expendable payload capacity reaches 2,392.62 tonnes, while a reusable profile supports approximately 1,780.34 tonnes. For Mars transfer orbits, the expendable payload is approximately 1,595.38 tonnes, decreasing to 840.91 tonnes for reusable operations due to the additional propellant required for the return boostback burn.

Transfers to more distant destinations such as Ceres and Jupiter pose greater challenges. While expendable payloads to Ceres and Jupiter are calculated at 869.75 tonnes and 457.72 tonnes respectively, a full reusable profile to these destinations, with Δv coming from the booster alone, is not feasible with current booster performance, due to the extreme Δv requirements exceeding the booster's mass budget. If part of the required Δv comes from the payload (Such as Starship) it is feasible, because the booster's required Δv is reduced.

These results highlight that transorbital boosters dramatically enhance payload mass delivery beyond LEO, particularly for cislunar and near-planetary missions. They also underscore the need for multi-booster missions or in-orbit refueling for heavy missions targeting deeper solar system destinations like Ceres and Jupiter.

8.3 Specific Fuel Savings on Starship to Mars, the Moon, Ceres, and Jupiter

The calculations presented in this study show that a transorbital booster could save approximately 809.39 tonnes of propellant for a Starship-class vehicle en route to Mars. By reducing the Δv requirement by 3.068 km/s, the fuel needed for Mars transfer drops from 1166.74 tonnes to 357.35 tonnes, dramatically increasing Starship's delivered mass and enabling a Mars orbit insertion burn without relying solely on aerobraking. Similarly, for lunar missions, the analysis indicates a fuel savings of approximately 1006.14 tonnes, reducing the required fuel load from 1012.80 tonnes to just 6.67 tonnes.

Expanding the analysis to deeper space destinations, the transorbital booster could save approximately 569.97 tonnes of propellant for a Starship transfer to Ceres and approximately 396.38 tonnes for a transfer to Jupiter. Although the Δv requirements for these more distant missions are significantly higher, the booster still provides meaningful reductions in the propellant mass needed to depart Earth orbit. All fuel savings calculations are based on the Starship carrying its full 150-tonne payload configuration, maximizing the utility of each mission.

These results highlight the transformative impact of boosters on mission architecture, shorting timelines, reducing the cost for deepspace missions, and enabling larger payload deliveries to the Moon, Mars, and beyond.

8.4 The Role of Reusable Chemical Boosters in the Future of Spaceflight

The findings of this study suggest that reusable chemical boosters could play a pivotal role in the future of space exploration and infrastructure deployment. By significantly lowering launch mass requirements, boosters can enable the assembly and transport of large structures—such as Gateway stations or Mars transfer vehicles—with fewer launches and reduced cost. Furthermore, boosters provide the high-thrust capability necessary for fast initial transfers, complementing slower, high-efficiency systems for interplanetary trajectories. As humanity targets more ambitious destinations like Ceres's water resources, Jupiter's moons, and Venus's atmosphere, reusable boosters offer a critical component to achieving these goals efficiently and reliably.

8.5 Challenges and Areas for Further Research

While this study confirms the potential of transorbital boosters, it also highlights challenges that must be addressed to fully realize their benefits. Reducing reliance on aerobraking by performing large retrograde burns requires precise Δv budgeting and improved propulsion performance.

Engine upgrades, such as the anticipated Raptor 3 with higher specific impulse, could further enhance booster effectiveness; for example, a 10% increase in I_{sp} could yield an 8% propellant mass reduction for a Mars mission. Additionally, mission design strategies like shorter retrograde burns or lunar slingshot returns present promising avenues for further fuel savings, albeit with trade-offs in mission duration and complexity. Expanding the use of multiple boosters per transfer, as discussed in Section 5.4, could unlock even greater mass efficiencies for deep-space missions.

Appendix: Glossary of Terms, Abbreviations, and Acronyms

Term	Definition
Aerobraking	A maneuver that uses atmospheric drag to slow a spacecraft and reduce propellant use during orbital insertion or descent.
Delta-V (Δv)	The total change in velocity required for a spacecraft to perform orbital maneuvers.
Effective Exhaust Velocity (v_e)	The speed at which propellant is expelled; calculated as $v_e = I_{sp} \cdot g_0$.
GEO (Geostationary Earth Orbit)	A circular orbit 35,786 km above the equator where satellites remain fixed relative to Earth's surface.
Initial Mass (m_0**)**	The total mass of a spacecraft or booster before fuel is burned.
ISRU (In-Situ Resource Utilization)	The practice of extracting and using local resources (e.g., on Mars) to generate fuel, oxygen, or water.
Isp (Specific Impulse)	A measure of propulsion efficiency; the thrust per unit of propellant flow, expressed in seconds.
Lagrange Points	Locations in space where gravitational forces and orbital motion balance, allowing stable positioning of spacecraft.
LEO (Low Earth Orbit)	An Earth orbit with altitudes ranging from 160 km to 2,000 km, commonly used for satellites and stations.
Methalox	A propellant combination of liquid methane and liquid oxygen, used in modern reusable rockets.
NRHO (Near-Rectilinear Halo Orbit)	A highly elliptical orbit around the Moon suited for staging lunar missions.

Appendix (continued)

Term	Definition
Orbital Depot	A space station or platform used to refuel spacecraft in orbit.
RVac (Raptor Vacuum Engine)	A vacuum-optimized version of SpaceX's Raptor engine with high specific impulse.
Retrograde Burn	A burn in the opposite direction of travel to reduce a spacecraft's velocity or alter its orbit.
SSO (Sun-Synchronous Orbit)	A near-polar orbit that maintains consistent local solar time for each ground pass.
Standard Gravitational Parameter (μ)	The product of the gravitational constant (G) and the mass (M) of the central body: $\mu = GM$.
Tsiolkovsky Rocket Equation	An equation that relates Δv to the mass ratio and exhaust velocity in rocketry.
VASIMR	Variable Specific Impulse Magnetoplasma Rocket — a high-efficiency plasma engine under development.
Vis-viva Equation	An orbital mechanics equation used to compute orbital velocity at any point along an orbit.

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