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In-Orbit Reusable Kick-Stage Enabled by a Sustainable Space Mobility Solution Harvesting Solar Energy for Onboard Fuel Production

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Abstract

The EU-EIC Pathfinder project Green SWaP (Green Solar-to-Propellant Water Propulsion) develops a sustainable in-space mobility architecture that directly converts water into hydrogen (H_2) and hydrogen peroxide (H_2O_2) using solar energy. This approach enables a reusable propulsion system combining a 200 N chemical thruster for primary manoeuvres with 1 N solar-thermal thrusters (I_{sp} ~500 s) for attitude control. By harvesting energy in orbit and producing propellant onboard, the system enhances operational safety, supports water circularity in space, and reduces dependence on Earth-supplied resources. Such a capability extends spacecraft lifetime, enables in-orbit refueling and in-situ resource utilization (ISRU), and broadens the feasibility of reusable orbital stages. To evaluate this potential, a dedicated mission analysis was performed after a selection process, focusing on a reusable kick-stage concept as a case study. The results provide preliminary sizing of key enabling technologies, such as bi-modal propulsion, inflatable hydrogen storage, and solar-to-fuel conversion, and demonstrate the transformative impact of Green SWaP on sustainable space logistics and future mission architectures.

Acronyms/Abbreviations

ADR Active Debris Removal. 4–7

AHP Analytical Hierarchy Process. 2, 5

FoM Figures of Merit. 2, 3

GEO Geostationary Earth Orbit. 4

ISRU In-Situ Resources Utilization. 3, 4, 15

ISS International Space Station. 5, 6

LEO Low Earth Orbit. 3, 4, 6, 7

MR Mass Ratio. 5, 6, 8, 10, 12

RCS Reaction Control System. 12–14

STT Solar Thermal Thruster. 2, 13

1. Introduction

The increasing scale and diversity of space missions have amplified the demand for propulsion technologies that are not only high-performing but also sustainable, reusable, and adaptable to multiple mission scenarios. Conventional chemical propulsion systems, while mature, are inherently constrained by their reliance on finite onboard propellant reserves, limited reusability, high logistical costs and dangerous operation. These limitations become particularly restrictive in emerging mission domains, where long-term operability and cost-effectiveness are critical. Recent advances in alternative propellant concepts have highlighted water as a promising medium due to its safety, storability, and potential for in-situ availability. Leveraging water as a feedstock for propulsion aligns with broader goals of green space operations and circular resource utilization. However, despite significant research interest, the integration of water-based propellant systems into reusable orbital transportation architectures remains underexplored. This represents a key gap in current space mobility capabilities. Green SWaP addresses this challenge by developing an in-orbit system that converts water into hydrogen (H_2) and hydrogen peroxide (H_2O_2) using solar energy. These products are employed in a dual-propulsion configuration: a 200 N chemical thruster (H_2O_2 + H_2) with a target specific impulse exceeding 360 s, and a 1 N solar-thermal thruster using H_2 with a target specific impulse of 500 s. Together, these thrusters enable versatile, high-efficiency propulsion that can be replenished in orbit, thus laying the foundation for reusable mission architectures. More details regarding the Green SWaP solution and its features are reported in [1] and [2].

Before defining the mission in detail, it is important to clarify its meaning within the Green SWaP framework. In this context, the mission does not refer to a single spacecraft, but rather to a broader, flexible, and scalable in-space mobility architecture designed to integrate propulsion, energy conversion, and energy concentration subsystems. The propulsion system, as the core element of this architecture, must support a variety of platforms and applications with distinct requirements. Accordingly, the general mission statement is formulated at a high level—"to provide a cutting-edge sustainable in-space mobility solution"—and subsequently translated into concrete objectives. The primary objectives are to enable novel mission concepts through advanced propulsion capabilities and to enhance the overall sustainability of space operations, while a secondary objective is to extend the operational lifetime of space assets. This approach ensures that the mission definition remains aligned with the overarching goals of the Green SWaP initiative and provides a clear framework for decomposing requirements into logical and functional components that will guide the modular development of the enabling technologies.

2. Mission selection

Although Green SWaP aims to remain a comprehensive propulsion solution, scalable and applicable to multiple mission scenarios, a case study was needed for the preliminary study of the various subsystems and technology developments. This defined mission will serve as a guiding framework for the design, testing, and validation of the propulsion and conversion systems throughout the remainder of the project. Choosing an appropriate mission is a critical step, as it provides a concrete context in which the system's capabilities, constraints, and performance requirements can be assessed. Building up from the technological constraints imposed by the chosen solution, it was possible to define the main boundaries and characteristics that the candidate missions should have followed to take advantage of Green SWaP. These features would have a dual impact: on the mission concept and operations, and on the spacecraft's configuration.

The propellant combination and its production fall under the first category. As mentioned, Green SWaP will use solar energy to convert water into propellant. This means the choice of propellant is already determined for both the main and secondary propulsion systems. Since the system needs water, any mission that cannot be resupplied with H_2O, either from Earth or in situ, will be excluded. Furthermore, sunlight availability is equally critical. Both the conversion system and the solar thermal thruster require direct sunlight to function, so missions involving long or frequent periods in shadow, or operating in regions with low solar intensity, must also be ruled out. The configuration, instead, will be impacted by two different constraints. First, Solar Thermal Thruster (STT)s require a solar concentrator, a large mirror focusing sunlight to heat the hydrogen. Second, once hydrogen is produced, it must be stored. Because implementing a cryogenic refrigeration system is not feasible, the hydrogen will remain in gaseous form. Due to its extremely low density, storing gaseous hydrogen in any tank would demand a very large volume. To address this, one or multiple inflatable tanks will be designed. This tank can be launched compactly and deployed in space, significantly saving space inside the launch vehicle fairing. Addressing the technical challenges of these two components is outside the scope of this work, as they will be evaluated over the coming years. At this stage, what matters is their impact on the mission constraints and system capabilities.

With these constraints in mind, a Analytical Hierarchy Process (AHP) analysis considering six core Figures of Merit (FoM) was used to rank 10 candidate missions. The AHP is a structured multi-criteria decision-making method. Experts from diverse fields, such as spacecraft design, orbital mechanics, sustainability, and mission planning, were independently consulted to assign relative weights to each FoM and evaluate the mission concepts. Through pairwise comparisons, the importance of each criterion was determined in alignment with Green SWaP's sustainability goals. These weighted scores were then used to calculate a final ranking of the proposed missions. Below is a brief explanation of each FoM:

- **Water Circularisation**: assesses the benefit of using water as a propellant, independent of performance.
- **Sunlight Availability**: evaluates how accessible solar energy is throughout the mission, which impacts the efficiency of the solar thermal system.
- **Mission Duration Enhancement**: measures the po-

tential mission lifetime extension enabled by Green SWaP compared to conventional solutions.

- **Novelty**: reflects how dependent the mission is on Green SWaP for its feasibility.
- **Feasibility**: considers how close the mission is to being achievable with current technology.
- **Sustainability**: assesses the environmental and resource advantages over existing alternatives.

The weighting of each FoM was established by means of the Averaged Pairwise Comparison Matrix, where values ranged from 1 (equal importance) to 9 (extremely more important), ensuring the most mission-relevant criteria were prioritised in the selection process. The feasibility in the short term was identified as the most impactful parameter according to expert evaluations, with a score of 32.80%.

2.1 Mission description

A total of ten missions were outlined following a study of future applications and potential opportunities for Green SWaP technologies. Central to all of them is the availability of water in situ (in orbit or at celestial bodies) and the potential to advance current propulsion and resource-utilization methods. Each mission demonstrates how Green SWaP can enable sustainable operations in different orbital environments and exploration contexts.

A total of ten missions were outlined following a detailed study of future applications and potential opportunities for Green SWaP technologies. Central to all of them is the availability of water in situ (either in orbit or at celestial bodies) and the potential to advance current propulsion and resource-utilization methods. Together, these missions span a broad range of operational domains, from low Earth orbit logistics to lunar infrastructure, asteroid mining, and Martian exploration, demonstrating the versatility of Green SWaP as an enabling technology. The proposed scenarios were deliberately selected to cover both near-term applications, such as orbital debris removal and station resupply, and long-term visionary concepts, including asteroid resource exploitation and interplanetary communication infrastructures. By addressing this wide spectrum, the missions illustrate how Green SWaP can progressively expand human presence in space, reduce dependence on Earth-based logistics, and lay the foundations for a self-sustaining space economy. In addition to their technical diversity, the missions also serve as use cases to evaluate system-level performance, operational scalability, and sustainability. Each one highlights specific challenges in propulsion, resource management, and autonomy, while showing how Green SWaP provides a coherent and flexible solution across different environments. Collectively, they present a roadmap for advancing space operations through sustainable propulsion, resource efficiency, and innovative mission design.

2.1.1 Lunar Resupply Module (Ascent Phase)

Designed to extract and process water from lunar ice deposits, this module resupplies an orbiting lunar station with propellant derived from in-situ resources. Its main engine executes ascent and transfer maneuvers, while solar-powered In-Situ Resources Utilization (ISRU) systems and lightweight, reusable hardware minimize logistics from Earth. By enabling autonomous refueling and docking, the system reduces dependence on terrestrial supply chains and advances a sustainable lunar economy.

2.1.2 Lunar Resupply Module (Phasing Phase)

In this variation of the previously described mission, the ascent phase is managed by a dedicated lunar launch system, allowing the resupply module to focus on transport, storage, and orbital phasing. Freed from the burden of ascent propulsion, the module increases its payload capacity for water delivery and simplifies onboard systems. Green SWaP technologies support precise phasing, attitude control, and docking, strengthening the scalability and efficiency of cislunar logistics.

2.1.3 LEO Small Space Station

Envisioned as a crewed platform for science, technology testing, and logistics in Low Earth Orbit (LEO), this station employs Green SWaP thrusters for orbit maintenance, debris avoidance, and controlled reboosts. A modular design, solar power, and autonomous control ensure resource-efficient operations. Serving both as a testbed for deep-space technologies and a hub for near-term human activity, the station supports long-duration spaceflight and strengthens orbital infrastructure.

2.1.4 Lunar Space Station

Acting as a strategic gateway for Moon and deep-space missions, a human-occupied station in lunar orbit will rely on Green SWaP thrusters for station-keeping, altitude corrections, and collision avoidance in the complex Earth–Moon environment. Solar energy powers both systems and propulsion, while advanced navigation ensures precise, low-intervention maneuvering. This outpost enhances operational flexibility, reduces resupply demands, and lays the groundwork for sustainable human presence in cislunar space.

2.1.5 GEO Space Factory

A manufacturing hub in Geostationary Earth Orbit (GEO) would exploit the microgravity and vacuum environment to produce advanced materials and large structures directly in space. Station-keeping and collision avoidance are managed by Green SWaP-powered thrusters, countering perturbations unique to GEO. High-efficiency solar arrays provide energy for both fabrication and propulsion, while autonomous controls optimize resource use. By advancing sustainable in-orbit manufacturing, this mission paves the way for large-scale infrastructure and an expanded space economy.

2.1.6 LEO Active Debris Removal

Integrated with a small orbital station, this system tackles the growing challenge of orbital debris. The Active Debris Removal (ADR) unit, fueled by in-situ water converted to propellant, detaches to capture and deorbit defunct satellites before returning for refueling and reuse. Combining advanced guidance and capture mechanisms with sustainable propulsion, the mission offers a scalable and cost-effective approach to improving orbital safety and ensuring long-term access to LEO.

2.1.7 LEO Kick Stage

Functioning as a reusable orbital tug, the Kick Stage performs precise orbit insertion for payloads deployed from a station in LEO or lunar orbit. Refueled using station water converted to propellant, it docks with payloads, executes transfer burns, and returns for reuse. This capability reduces reliance on large onboard propulsion systems for satellites, strengthens orbital logistics, and establishes the backbone of a sustainable in-space transportation network.

2.1.8 Asteroid Resource Utilization

By harvesting resources directly from asteroids, this mission delivers water and raw materials to a lunar orbital hub. Extracted water doubles as both a resource and fuel, eliminating Earth-supplied propellant for the round trip. Designed for reusability and in-situ refueling, the system supports lunar infrastructure, manufacturing, and sustained exploration, marking a step toward a self-sufficient space economy.

2.1.9 Icy Moons Sample Return

Targeting water-rich moons, this mission collects surface ice and mineral samples for return to Earth. By generating propellant on-site from extracted water, the spacecraft extends its operational endurance and reduces dependence on Earth-launched fuel. Returning pristine samples enables transformative science while also demonstrating the viability of sustainable exploration architectures for distant solar system bodies.

2.1.10 Mars Constellation

A satellite network around Mars will provide GPS and communication coverage for future surface operations and exploration. Supported by a refueling station that extracts Martian water and converts it into propellant, the constellation maintains formation and ensures long-term operability without Earth resupply. This infrastructure forms the foundation for a self-sufficient Martian communications system, directly enabling both robotic and crewed missions on Mars.

2.2 Trade-off results

The trade-off results are presented in Table 1. From this analysis, mission scenarios with the lowest scores were identified and excluded from further consideration. An exception was made for the Mars Constellation mission, which received one of the lowest overall rankings due to its limited feasibility score (3.39%). Although feasibility is a critical driver in mission selection, the team recognised that the underlying concept remains valuable in a different context. In particular, a lunar constellation dedicated to communication or navigation could offer a more practical and near-term application. For this reason, the Mars Constellation concept was retained for further examination in the second stage of the selection process, with the aim of exploring potential adaptations and alternative implementations.

Subsequent analyses focused on a subset of missions: the Kick Stage / ADR, the Asteroid ISRU, the LEO Space Station, and the Lunar Constellation. These were evaluated in greater detail with respect to their ΔV budgets and mission phases, in order to identify the most feasible baseline concept for both the propulsion system and the other subsystems. Following this high-level assessment, the Kick Stage mission was selected as the primary study case to drive the project. The LEO Station concept was integrated with this selection in the form of a refueling hub, complementing the Kick Stage architecture. For clarity, the following section presents the detailed mission analysis only for this chosen study case.

Table 1: Summary of the AHP analysis results.

Mission	Value (%)
LRM (ascent)	9.76
LRM (phasing)	9.74
LEO Small Space Station	10.31
Lunar Space Station	10.12
GEO Space Factory	9.50
LEO ADR	10.14
Kick Stage	11.47
Asteroid Resource Utilization	10.62
Icy moon mission	9.11
Mars Constellation	9.09

2.3 High-level mission analysis

After the trade-off phase, the four selected mission concepts were analysed in more depth. In order to determine the most suitable reference mission, it was necessary to move from a qualitative evaluation to a quantitative one by estimating the theoretical ΔV requirements involved in executing the proposed manoeuvres. The reference [3] shows the equations forming the analytical foundation for the high-level evaluation of mission profiles. They allow for the estimation of total ΔV budgets required for each concept and help identify which use cases are most compatible with the Green SWaP system's performance. To evaluate the feasibility and efficiency of the proposed missions, an initial analysis was conducted using the propellant Mass Ratio (MR) as a key performance metric. The propellant mass ratio will be defined as:

$$MR = \frac{m_p}{m_w} \tag{1}$$

Where m_p is the propellant mass and m_w is the wet mass (total mass of the vehicle, including propellant, structure, and payload). This metric serves as a direct indicator of the propulsion system's efficiency in the context of the specific mission profile. A lower mass ratio implies a more efficient system, capable of achieving the required maneuvers while reserving more mass for payload or structural components. In other references, the propellant mass ratio is referred as the ratio between the propellant mass and the dry mass. For our means, the alternative definition just presented was adapted for usability reasons. In the following sections, this metric, together with the ΔV budget, was used to compare mission architectures and guide the selection of a baseline configuration for the subsequent phases of system design and prototyping.

2.3.1 Kick Stage / ADR

Building from the results of the AHP evaluation, it was decided to merge the Kick Stage concept and the ADR system into a single spacecraft architecture. This decision was driven by the realization that both missions share similar operational requirements and can be effectively carried out by the same spacecraft, with only minor modifications in interface definitions and mission phases. From a manoeuvrability standpoint, the two mission profiles are highly compatible. The Kick Stage function involves raising the orbit of a payload from a parking orbit to its designated operational orbit, whereas the ADR function follows a similar trajectory, with the main difference being that it relocates decommissioned satellites to a disposal orbit rather than boosting active payloads. Hence, this study will focus primarily on orbital relocation manoeuvres. In the case of ADR operations, this operational orbit will be a graveyard orbit, ensuring the long-term sustainability of the space environment without compromising the Kick Stage's ability to conduct future missions.

The preliminary analysis of the Kick Stage/ADR mission was conducted under the assumption that the system would initially be attached to the International Space Station (ISS). Within this scenario, a modular approach was adopted, separating the propulsion vehicle from the fuel conversion system. This led to the definition of a dedicated refueling module, which would be docked to the ISS and serve as a fuel depot. This setup presents a major logistical advantage, as water resupply operations could leverage existing ISS cargo missions. In particular, resupply vehicles such as Northrop Grumman's Cygnus are already transporting approximately 900 liters of water annually to the ISS. This means that Green SWaP could integrate into an already established logistics chain, reducing the need for additional dedicated launches. An in-orbit Kick Stage able to perform refueling was never designed before. However, the upper stages of launcher can be used as references, since they performs similar operations, at least during the target commission phase. The upper stage of a launcher is the topmost part of a multi-stage rocket. Its main job is to carry and precisely place the payload (like a satellite or spacecraft) into its intended orbit or trajectory after the lower stages have provided the initial boost. It usually contains its own propulsion system, allowing it to operate in space, sometimes with multiple burns to adjust the orbit or escape Earth's gravity entirely. According to the state of the art shown in Table 2, upper stages typically exhibit a higher MR compared to scientific satellites or telecommunications platforms. MR values for upper stages can range from 0.80 to more than 0.90, depending on the specific mission and propulsion technology.

Name	Company	Payload [kg]	Thrust [N]	MR
Star 48BV	NG	1200	2077	0.95
Star 63F	NG	2200	4750	0.97
Curie	Rocket Lab	150	160	0.80
Photon	Rocket Lab	170	260	0.84

Table 2: Representative data for common Kick Stages including payload capacity [4–6].

Upper stages are usually decommissioned after having placed the target into its operational orbit, while this system will need a complex communication and autonomous systems of rendezvous and docking, as well as a structured feeding system for the refueling phase. This means that any operation requiring an MR above 0.85 would be considered unfeasible due to excessive structural or propulsion constraints. For this reason an operational MR range of [0.75 – 0.85] was adopted during this study, which is slightly lower than typical values of similar systems. The system design would aim to maximize MR within the defined range to ensure efficient fuel utilization while maintaining structural integrity and maneuverability.

The primary maneuvers required to reach the target, deliver it to its operational orbit, and return to the refueling station consist of at least three main phases, each involving a minimum of two burns of the main engine. At this stage, all maneuvers are assumed to be impulsive, meaning they are executed as instantaneous velocity changes. However, the feasibility of this assumption depends on the spacecraft's thrust capabilities. The highest the thrust, the nearest is the the manouvre to an impulsive one. Usually, a manouvre is considered to be impulsive when $\frac{t_{burn}}{T_{orbit}} \ll 1$, where $t_{burn} \propto \frac{1}{F}$, and F is the thrust. If the required burn time is too long, the impulsive approximation may no longer be valid, in which case multiple main engines might be necessary to achieve the required acceleration. Alternatively, a more in-depth analysis could explore the possibility of a multi-stage transfer utilising several small burns. This approach would allow for greater manoeuvring flexibility while avoiding the added complexity of integrating multiple large main engines. Table 3 shows the ΔV budget estimated under the assumption of two-stage impulsive Homann transfers. The ΔV required for attitude control and docking operations was neglected at this stage.

To provide a clearer overview of the consumptions, Figure 2 illustrates the total ΔV budget required for a single mission departing from the ISS and returning after delivering the payload. As observed, the manoeuvre cost increases as the operational orbit altitude rises above the parking orbit of the refueling module. This trend indicates that missions to lower-altitude orbits will be more efficient, as they require less propellant, thereby allowing for higher spacecraft mass and greater payload capacity. Based on the performance parameters of the main engine, including thrust and specific impulse, as well as a comparative assessment with reference missions, the Kick Stage mission concept has been deemed a feasible and promising application of the GreenS WaP architecture. The proposed system demonstrates adequate margins in terms of mass ratio, ΔV capabilities, and operational flexibility, making it well-suited for a variety of tasks such as orbit insertion, phasing, and ADR operations. Furthermore, the integration of in-space refueling from water-based resources adds a unique and sustainable advantage to the mission concept, aligning with the long-term goals of space sustainability and cost reduction.

Maneuver	ΔV [m/s]
Target Capture (Min)	117
Target Delivery (Min)	58
Return Transfer (Min)	0.1 (phasing)
Target Capture (Max)	117
Target Delivery (Max)	884
Return Transfer (Max)	768
Min Total	**235**
Max Total	**1770**

Table 3: ΔV requirements for each mission phase of the Kick Stage/ADR system.

2.4 Mission selection conclusions

Following the mission selection and high-level feasibility assessment, the Kick Stage/Active Debris Removal scenario emerged as the most promising and suitable reference mission for the development and validation of the Green SWaP propulsion and refueling system, as anticipated.

The other scenarios such as a lunar communication constellation, an interplanetary asteroid retrieval mission, or servicing/refueling of a LEO or lunar station either rely on speculative infrastructure (e.g., a fully operational lunar refueling depot), suffer from extreme ΔV demands (e.g., asteroid retrieval), or pose operational limitations due to power and mass constraints (e.g., deep space constellations). While each presents exciting and forward-looking opportunities for Green SWaP, they either lack near-term feasibility or involve high degrees of uncertainty in terms of mission architecture, launch cadence, or infrastructure availability. The Kick Stage/ADR mission, on the other hand, stands out since it is technically demanding but bounded, operating in Earth orbit with manageable and tailorable ΔV budgets and predictable orbital dynamics. Additionally, it is near/mid-term feasible, building upon existing mission concepts with realistic launch and integration timelines, and it is scalable and modular, as multiple Kick Stages can be refueled and reused from a single LEO depot, enhancing both economic and operational efficiency. Last but not least, the Kick Stage mission is demonstrative of Green SWaP's core value, proving the capability to perform multiple orbital maneuvers using water as propellant with a refuelable system.

3. Mission analysis

Building upon Section 2.3.1, a well-defined scenario was established to set more precise mission requirements in terms of ΔV and operational constraints. As already discussed, the ΔV directly influences the mass budget and propellant requirements, and the selection of orbital manoeuvres is therefore an iterative process. At this stage, a full mission optimization is not within scope. Instead, all orbital manoeuvres are assumed to be impulsive, providing a simplified first-order estimation of the required ΔV. This approximation serves as a foundation for defining early system-level requirements, particularly for the main propulsion subsystem, including parameters such as specific impulse, total throughput, and burn duration. The requirements for the RCS were determined in a subsequent step, as they depend on the spacecraft configuration and operations to be performed. Consequently, all conclusions from this mission analysis refer primarily to the main engine. Figure 1 summarises the mission profile defined for this preliminary analysis.

3.1 Refueling Station

A baseline mission scenario has been defined as a reference point for the performance envelope of the Kick Stage. In this scenario, the integration of a refueling module within a crewed orbiting space station was relaxed. Instead of relying on the ISS, a dedicated refueling station is envisioned. The ISS, whose operational lifetime has already been extended several times, is approaching the end of service. An ageing infrastructure not aligned with the long-term objectives of Green SWaP would be counterproductive. One of the project's primary goals is mission duration enhancement, with a refueling station expected to operate for roughly 20 years and a Kick Stage with a

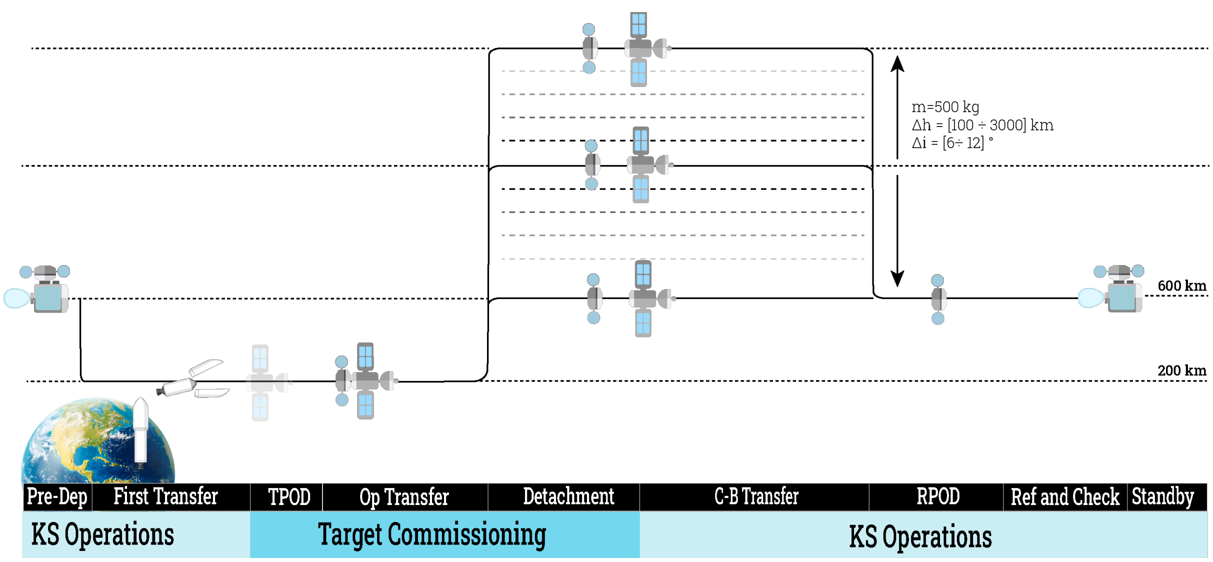

Fig. 1: Graphical representation of the concept of operation. The mission profile was divided in two main phases, the Kick Stage (KS) operations and the Target Commissioning. The overall duration of each operation strongly depends on the target operational orbit.

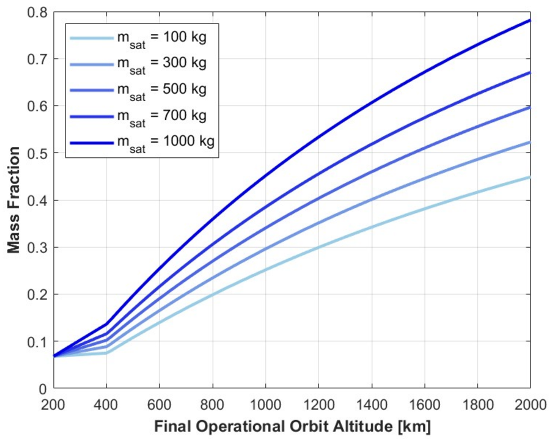

Fig. 2: Kick Stage MR evolution as a function of the operational orbit altitude. The Kick Stage wet mass is assumed to be 500 kg, while m_{sat} is the mass of the target satellite to be operated.

projected lifespan of about 10 years. Developing a new, purpose-built refueling station better meets these requirements of sustainability and operational robustness. Orbital inclination plays a critical role in fuel efficiency, particularly for plane change manoeuvres, which are among the most ΔV-demanding operations in space. Placing the refueling station in the ISS orbit would impose an inclination of 51.6°, a value chosen for reasons unrelated to Green SWaP's objectives, such as accessibility from Baikonur, compatibility with multiple launch providers, Earth observation optimisation, and historical factors. By removing the constraint of aligning with the ISS orbit, the refueling station can instead be placed in an orbit tailored to Green SWaP's operational needs, selecting inclination and altitude that minimise propellant consumption for a variety of missions. The main drawback of decoupling from the ISS is the loss of logistical resupply advantages, such as supplementary water deliveries from cargo missions. To mitigate this, an alternative resupply strategy is proposed. The Kick Stage can be repurposed to collect water from a nearby parking orbit and deliver it to the refueling station. Since the relative distance between the two orbits can be kept small, fuel consumption is minimized. The detailed placement of both the refueling station and the parking orbits will be further analysed to optimise ΔV requirements. While feasible, this strategy introduces additional design, thermal, mechanical, and configurational constraints that must be addressed in future feasibility studies. At this stage, the key driver remains the mass ratio. To assess the feasibility of resupply, the Kick Stage mass and fuel mass ratio were defined. Figure 3 illustrates the variation of mass ratio with cargo mass and highlights the corresponding fuel consumption for water resupply operations. As shown, the water resupply mission appears largely feasible and could therefore serve as a baseline for defining resupply operations in the system analysis.

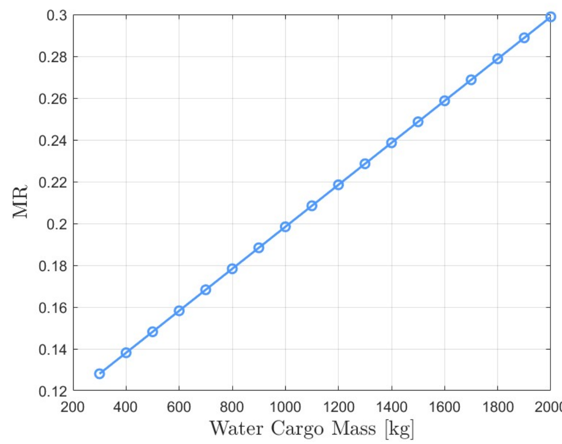

Fig. 3: Analysis result of the water resupply feasibility by using the Kick Stage as a cargo retrieval system. The MR is clearly under the threshold to declare the cargo resupply mission feasible.

The choice of the refueling station orbit must balance two sets of constraints: those related to continuous and efficient solar-powered propellant production, and those related to operability and mission support. From the production perspective, stable and prolonged sunlight exposure is essential to power the onboard conversion systems, while thermal differentials across the spacecraft surfaces ensure optimal performance of concentration and conversion processes. Water availability is assumed to be guaranteed as previously discussed. From the operational perspective, station keeping must remain feasible using the Green SWaP bi-modal propulsion system, orbital inclination should be compatible with common launch vehicle deployment inclinations, and the selected orbit should align with operational orbits frequently used by Earth observation satellites. Balancing these factors requires a trade-off. Inclination strongly affects ΔV for plane changes, while altitude mainly impacts solar exposure and atmospheric drag. After evaluating different configurations, a dusk-descending Sun-Synchronous Orbit (SSO) was identified as the most suitable compromise. This near-polar orbit maintains a constant local solar time, ensuring consistent illumination and thermal conditions. A dusk-descending SSO crosses the equator at local dusk (18:00), placing the spacecraft near the terminator and providing near-continuous sunlight with only minor eclipses around equinoxes. The typical inclination of about 97.6°, retrograde, is also widely used in Earth observation, making it compatible with many target missions.

Given these considerations, the inclination of the refueling station should match that of the target satellite, with minor adjustments possible to accommodate launcher misalignments. The remaining free parameter is orbital altitude, which directly impacts propellant consumption. A detailed analysis shows that propellant requirements are minimized when the refueling station is close to the parking orbit. Transfers to higher final orbits are less sensitive to the station's altitude, while transfers to lower orbits remain fuel-intensive. Placing the refueling station near the parking orbit reduces mission ΔV, but station keeping favours higher altitudes. To balance these needs, an altitude of 600 km was selected as the best compromise. Below 500 km, atmospheric drag imposes prohibitive hydrogen consumption for station keeping, while altitudes much higher than 600 km complicate tracking and communications. The chosen orbit also benefits Earth observation missions by providing favourable lighting conditions, strong surface temperature contrasts, stable solar power generation, and enhanced imaging performance. Finally, since SSO is among the most commonly used inclinations for modern launchers, particularly for small and medium payloads, the Kick Stage is expected to require minimal inclination changes, thereby reducing total ΔV. As a contingency, a secondary orbital configuration has also been identified, based on the second most common inclination used for commercial launches. This ensures flexibility and compatibility with a broader set of mission profiles.

3.2 Kick Stage

After defining the orbit of the refueling station, a detailed mission analysis was carried out to establish the operational constraints of the Kick Stage and to size its subsystems. The reference scenario assumes a Kick Stage wet mass of 500 kg, with a maximum propellant mass ratio of 0.85. Of this, 90% is allocated to orbital manoeuvres, while the remaining 10% is reserved for margins, trajectory corrections, docking, and attitude control. The nominal target satellite mass is 500 kg, with an assumed maximum inclination correction of $\Delta i = 3°$ to account for potential misalignments between the parking and operational orbits. The operational altitude range for the target satellites is set between 300 km and 3000 km. Based on these inputs, the total deliverable ΔV in the most demanding scenario reaches approximately 2500 m/s. The ΔV contributions from the different phases of the mission were estimated under both minimal and maximal effort conditions, as shown in Table 4.

Maneuver	**ΔV min**	**ΔV max**
Target Capture	220 m/s	220 m/s
Target Delivery	70 m/s	1260 m/s
Return Transfer	60 m/s	1070 m/s
Total	**450 m/s**	**2500 m/s**

Table 4: ΔV breakdown for the Kick Stage manoeuvres.

It should be noted that the minimum and maximum values reported in Table 4 are not additive across scenarios. The totals correspond to the actual best- and worst-case missions, while the manoeuvre-specific rows report the individual range of values for each phase. For instance, the minimum total ΔV is not obtained by summing the minimum values of each manoeuvre, as these conditions do not occur simultaneously in a single trajectory. Similarly, the maximum total value reflects the worst-case combination, rather than a sequence of individually extreme manoeuvres.

From a design perspective, the maximum requirement of 2500 m/s is adopted as the driver case to ensure the system is sized for the most demanding missions within the operational envelope. This conservative approach also results in a generous margin for lower-energy cases, where operations may require less than 20% of the available ΔV. In such situations, the Kick Stage could perform multiple missions between refueling cycles, particularly when servicing smaller payloads or executing low-altitude orbit changes. While the propulsion subsystem and refueling infrastructure must be designed to withstand the most challenging cases, scalability and flexibility remain intrinsic to the architecture, making the system adaptable to a wide range of applications. Figure 4 shows how the total ΔV requirement scales with increasing inclination correction. Each line represents a specific target altitude and illustrates the sensitivity of the budget to Δi. Higher orbital altitudes combined with large inclination changes quickly approach or exceed the system's performance limit, thereby setting clear operational boundaries.

Figure 5 further highlights the link between ΔV and fuel consumption. Each curve corresponds to a different altitude, illustrating how increasing Δi drives higher propellant demand. The analysis confirms that the Kick Stage's maximum ΔV capability of 2500 m/s is sufficient to cover the

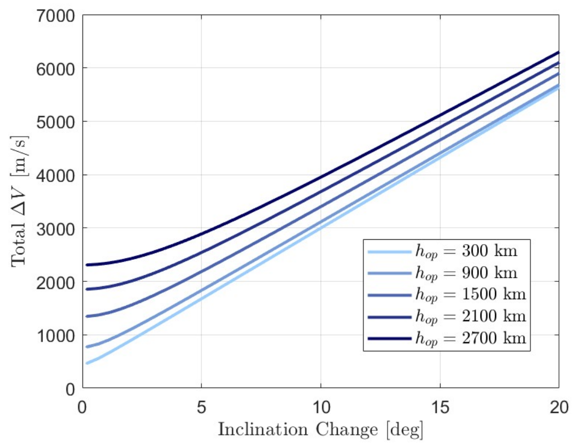

Fig. 4: Total ΔV vs. inclination correction for different altitudes.

full range of intended missions, up to altitudes of nearly 3000 km, even with moderate inclination changes. This validates the design-driving requirement and demonstrates that the system can operate reliably within the Green SWaP concept.

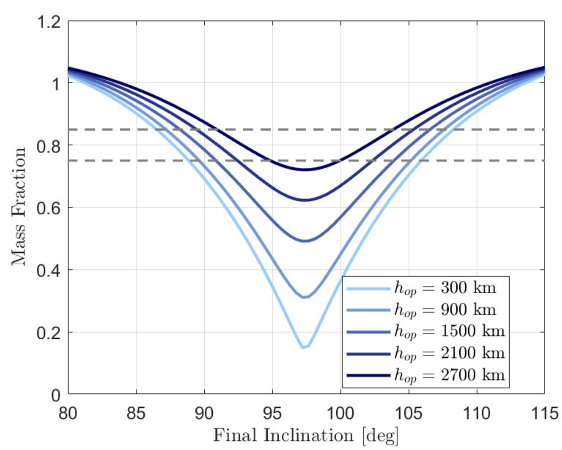

Fig. 5: MR required as a function of the final inclination of the target's operational orbit for different altitudes. In this figure, the Kick Stage mission is assumed to depart from and return to the refueling station's orbit.

A summary of achievable operational scenarios is given in Table 5. Once the Kick Stage design and the two main reference orbits (parking and refueling) are fixed, the target satellite mass becomes the primary factor influencing mission feasibility. The lower bound of the inclination range corresponds to the maximum Δi achievable for transfers to the highest operational orbit, while the upper bound applies to transfers toward a 300 km orbit. As the table shows, the achievable inclination correction is modest, underscoring that the Kick Stage is not intended for large plane changes. Instead, it relies on high-precision injection by the launch vehicle to minimize initial inclination offsets. For lighter payloads or lower operational orbits, Δi capability improves slightly, but significant inclination mismatches would still remain impractical to correct.

Target Mass	**Max Altitude**	**Deliverable Δi**
100 kg	5700 km	[10 ÷ 20] deg
500 kg	3000 km	[6 ÷ 12] deg
1000 kg	1900 km	[3 ÷ 6] deg
2000 kg	1300 km	[2 ÷ 4] deg
3000 kg	900 km	[1.5 ÷ 2] deg

Table 5: Summary of achievable Kick Stage manoeuvres.

The operational flexibility of the system also depends on the rate at which propellant can be produced in orbit. Table 6 presents the required production rates as a function of operational frequency, assuming continuous water-to-propellant conversion. Shorter resupply intervals demand higher production rates, while longer intervals reduce the demand but increase mission downtime between successive operations. This table highlights the trade-off between system productivity and mission cadence: a faster production cycle enables more frequent operations but requires higher conversion efficiency, whereas slower cycles relax technical requirements at the expense of responsiveness. The appropriate balance will depend on mission demand, available infrastructure, and long-term planning within the Green SWaP framework.

Operational Frequency	**Production Rate (g/h)**
15 days	1120
30 days	560
60 days	280
90 days	185
180 days	90
365 days	46

Table 6: Required production rate as a function of operational frequency.

4. Configuration

A preliminary configuration of both the Kick Stage and the refueling station was developed to begin exploring the feasible geometries and constraints associated with inflatable propellant tanks, as well as to start working on the requirements for the subsystems and RCS operations. This included an early estimation of internal volumes, integration challenges, and the impact of using multiple 1 N thrusters for attitude control maneuvers. This conceptual configuration enables a better understanding of the overall propellant consumption, integration feasibility with the launch vehicle, and the volume and mass requirements dictated by the previously estimated propellant needs. Additionally, this conceptual design served as a starting point for evaluating the available area for solar panels and conversion panels, which is directly tied to the system's power budget and the design of the power conversion and distribution subsystem.

The first step in defining the system was to take the total propellant mass derived from the mission analysis as input. In order to size the refueling station appropriately, it was crucial to determine the number of Kick Stage missions the system should be capable of supporting before requiring a water resupply from Earth. As an initial assumption, the refueling station includes one dedicated water tank per mission. Each tank contains the amount of water necessary to generate the full propellant load required by one Kick Stage mission. The system is also assumed to be launched with one fully filled water tank to support the station's own preliminary maneuvers (assuming that the station itself is propelled using Green SWaP technology). Under the assumption of three onboard water tanks, the total operational capacity of the refueling station would be equivalent to four full Kick Stage missions before requiring a resupply. This includes one internal use and three external refuels. When referring to "missions," this assumes the most demanding case. In scenarios involving less intensive operations, such as low-altitude commissioning, no orbital plane change, or targeting small spacecraft, the Kick Stage may be capable of performing multiple missions before requiring a return to the refueling station. A total dry mass limit of 2000 kg was assumed for the entire refueling station, including a maximum of 1600 kg allocated for propellant-related mass. This includes both unprocessed water and the already processed hydrogen and hydrogen peroxide. It is important to note that only one full resupply of ready-to-use propellant is available at a time. After the Kick Stage has been refueled, the station must wait for the next batch of propellant to be synthesized from stored water before being able to provide another refueling service, introducing a delay between successive refueling operations. The confuguration resulting from these preliminary consideration is shown in Figure 6.

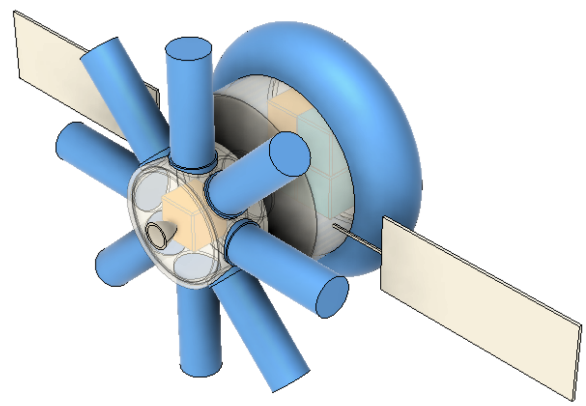

Fig. 6: Configuration of the refueling station attached to the Kick Stage. The station's external blue tanks store gaseous hydrogen, and the yellow tanks store hydrogen peroxide. Internal water tanks (in light blue) are located only within the station, as the Kick Stage has no conversion capabilities.

For the Kick Stage configuration (Figure 7), the results of the mission analysis were again used to size the propellant tanks. Starting from the required propellant mass, the dimensions of the cylindrical tank body were derived by computing the required height and diameter such that both the Kick Stage and the refueling station could be integrated within a single launch vehicle fairing. The remaining volume was assessed to ensure sufficient space for non-propulsion subsystems, such as avionics, power, and thermal control. Assuming a propellant mass ratio of 80%, approximately 100 kg of dry mass was available for these subsystems. A standardized docking ring interface was assumed, consistent with the ECSS standards adopted for the refueling station, to ensure mechanical and operational compatibility.

The design approach for the hydrogen tanks differed between the refueling station and the Kick Stage. While either approach could be used for both, it was preliminarily considered more effective to segment the hydrogen into multiple smaller tanks on the Kick Stage. This configuration simplifies pressure regulation by allowing the system to draw from individual tanks sequentially. For the Kick Stage, which requires stable and predictable thrust during its main maneuvers, maintaining high and controlled pres-

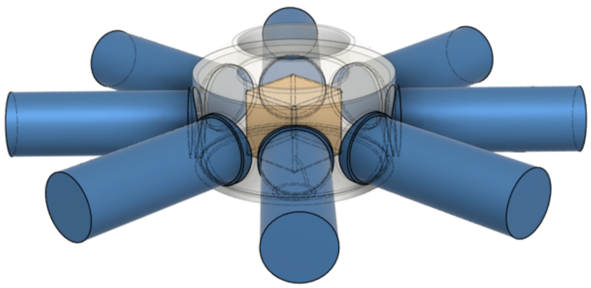

Fig. 7: Kick Stage preliminary configuration with 8 inflated cylindrical hydrogen tanks.

sure is essential. Using separate tanks helps mitigate the performance degradation associated with the pressure drop that occurs as inflatable tanks deflate during use. For the refueling station instead, the controllability of the system was preferred over high and fast operation performances, resulting in a more symmetric configuration. This would help control the system's dynamics during the conversion process, where the pointing accuracy must meet demanding requirements.

5. RCS preliminary design

5.1 Moment of Inertia

The Kick Stage system was selected as the design driver for defining the system and subsystem requirements of the propulsion system. As discussed for the main engine, its attitude control operations are expected to be more demanding than those of the refueling station. Therefore, estimating the Kick Stage's principal moments of inertia about its x, y, and z axes is essential for evaluating its attitude dynamics and control requirements [7]. Such estimation requires a high-level definition of the spacecraft configuration. For a total spacecraft mass of $m_{\text{total}} = 500$ kg and assuming a propellant mass ratio (MR) of 0.8 (the average of the range considered during mission analysis), the total onboard propellant mass is 400 kg. Using the stoichiometric mixture ratio for the main engine and neglecting the hydrogen required for the Reaction Control System (RCS) maneuvers, this results in 378 kg of hydrogen peroxide and 22 kg of hydrogen. The assumption of neglecting the RCS hydrogen will be validated in the next section. In general, at this stage, it is considered negligible; a margin can be added to the inflatable tanks in a subsequent design phase. The current focus is on the internal mass distribution. The mass of the tanks themselves is neglected at this stage, as the storage technology, particularly for the inflatable hydrogen tanks, requires in-depth investigation and is beyond the scope of this preliminary work. The mass distribution is assumed to be concentrated in three main components: a central hydrogen peroxide (H_2O_2) tank with a mass of $m_{\text{HP}} = 378$ kg; an array of eight peripheral hydrogen (H_2) tanks with a combined mass of $m_{H_2} = 22$ kg (i.e., 2.75 kg each); the remaining subsystems with a combined mass of $m_{\text{other}} = 100$ kg. The reference frame is defined with its origin at the geometric center of the satellite. The z-axis is vertical, while the x and y axes lie in the horizontal plane containing the hydrogen tanks. The central H_2O_2 tank is approximated as a cube, the hydrogen tanks as vertical slender cylinders, and the remaining subsystems as a solid vertical cylinder. Their individual contributions to the moments of inertia are summarized in Table 7.

Table 7: Estimation of the moment of inertia of the main Kick Stage's subsystems.

Subsystem	**Inertia** $[kg \cdot m^2]$
H_2O_2 tank (x,y,z-axis)	30.87
H_2 tanks (z-axis)	22.0
H_2 tanks (x,y-axis)	23.41
Other (x,y-axis)	27.53
Other (z-axis)	50.0

By summing the respective contributions, the estimated principal moments of inertia for the satellite are being estimated to be $I_x = I_y = 81.81 kg \cdot m^2$ and $I_z = 102.87 kg \cdot m^2$. This simplified estimation assumes rigid and symmetrical mass distributions for analytical convenience. While higher-fidelity methods such as CAD-based modeling or finite element analysis could provide more accurate results, this analytical approach offers a reliable baseline for preliminary attitude dynamics and control studies.

5.2 RCS Operations

As an assumption, the Kick Stage will use a combination of solar thermal thrusters to control the attitude. In this section, the aim is to understand which manouvres are achievable with the target performance and what are the requirement of the reaction control system. Attitude control refers to the process of managing the orientation of a spacecraft with respect to an inertial frame or another object (such as Earth or a star). The RCS is one of the primary methods used to achieve attitude control [8].

Generally speaking, active control of spacecraft attitude can be achieved by combinations of Reaction wheels, Momentum wheels, Control moment gyros (CMG), Thrusters, and Magnetic torquers. The spacecraft must be controllable along all six degrees of freedom, encompassing three translational and three rotational axes. Additionally, to generate torque for rotational control, each axis requires at least a pair of thrusters operating in opposition. In a conventional configuration, this would imply a minimum of 12 thrusters. However, as discussed in [9], alternative configurations and advanced control laws can reduce this number. For the purposes of this simplified case study, we assume a total of 16 thrusters to include redundancy, organized into four groups of four thrusters strategically distributed around the spacecraft.

5.2.1 External disturbances

To assess the external disturbances the spacecraft may encounter,a worst-case gravity gradient torque of $4.5 \cdot 10^{-5}$Nm was considered. The nominal torque that the proposed RCS using the combined effect of two opposing thrusters can deliver is 4Nm, which is orders of magnitude greater than the expected disturbance torque. Consequently, the thrusters would need to operate well below their nominal thrust during fine maneuvers. For this reason, it would be advisable to rely on momentum or reaction wheels for high-precision operations, such as docking, while using the RCS for momentum dumping, large slew maneuvers (e.g., target acquisition, release operations, or pre-maneuver reorientation), and as a backup in the event of reaction wheel failure. Given the low magnitude of disturbance torques, the sizing of the RCS thrusters is likely to be driven more by the required slew rates than by disturbance rejection alone.

5.2.2 Slew

A thruster can modify the spacecraft's angular momentum by generating a control torque. To design an effective slew manoeuvre, two key mission parameters must be defined: the desired slew angle and the allowed manoeuvre time. These values significantly impact the sizing and control strategy of the attitude control system. The fundamental relation governing the control torque F_{slew} during a slew manoeuvre can be estimated by using the moment of inertia of the spacecraft for the most demanding axis and the expected angular acceleration to be provided. In Figure 8, the total time required to accelerate the spacecraft to the target angular velocity and then decelerate it is assumed to be 20% of the total manoeuvre time. A symmetrical deceleration phase is assumed, with the remaining 80% allocated to constant velocity rotation. Based on the previous inertia analysis, the most demanding axis (i.e., the one with the largest moment of inertia) was considered in the torque estimation. Using basic rotational dynamics, it is possible to estimate the required thruster force per engine as a function of the total manoeuvre time.

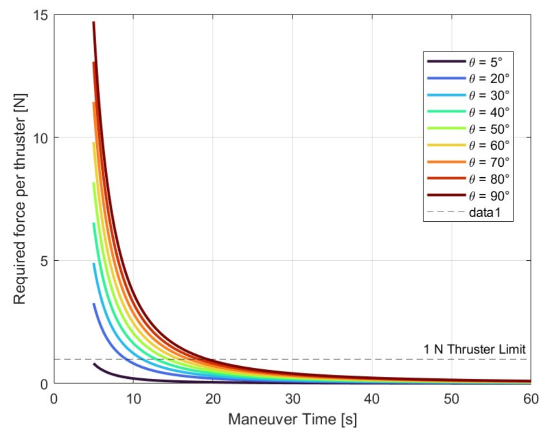

Fig. 8: Required thruster force as a function of slew angle and manoeuvre time.

This analysis reveals that increasing the allowed manoeuvre time significantly reduces the required thruster force. Hence, a trade-off exists between manoeuvre agility and propulsion system constraints (longer manoeuvres allow for smaller, more efficient thruster operation). From Figure 8, it can be observed that the required thruster force decreases non-linearly as the manoeuvre time increases, for all tested slew angles. This behavior underscores the importance of mission-level decisions in balancing attitude control performance with system efficiency and fuel consumption. These figures and considerations will be used to set reasonable requirements for the slew manoeuvres and, thus, the STTs design.

5.2.3 Momentum dumping

As introduced at the beginning of this section, it is both reasonable and recommended to couple the RCS with a set of reaction wheels. While reaction wheels provide fine attitude control with high precision and no propellant usage, they are subject to secular disturbances, such as gravity gradient torques, solar radiation pressure, or magnetic influences, that cause continuous accumulation of angular momentum. Over time, this pushes the wheels toward their maximum angular velocity limits, a condition known as saturation. To maintain control authority, reaction wheels must be regularly desaturated, that is, their stored momentum must be offloaded via external torques. The RCS is typically used to provide these torques by firing thrusters in a coordinated way that transfers angular momentum from the wheels to the spacecraft body and, ultimately, to space.

The force required to desaturate a single wheel depends on the angular momentum storage capacity of the reaction wheel h and the duration of the desaturation manoeuvre. The value of h is determined during the wheel selection and system design phase. Once the orbit is defined, the main driver of the required momentum storage becomes the level of attitude accuracy, particularly in yaw, that the mission aims to achieve. Since this yaw accuracy requirement has not yet been finalized, it was treated as a parametric variable for this preliminary study. Figure 9 illustrates the trade-off between desaturation time and yaw pointing accuracy, based on assumed values of wheel momentum storage and thruster configuration. It highlights that with 1 N thrusters, relatively high pointing accuracy can be achieved even with short desaturation times. The desaturation time is a critical parameter in designing both the RCS and the reaction wheel subsystem. Shorter desaturation times require higher thruster forces, which may demand larger thrusters or higher propellant consumption. This can be costly in terms of both mass and power. On the other hand, longer desaturation periods allow for gentler, lower-force manoeuvres, reducing thruster wear and enabling smoother attitude transitions. However, extended desaturation durations may interfere with mission operations, such as Earth-pointing or docking modes, if not carefully scheduled.

Therefore, selecting an optimal desaturation time involves a trade-off between thruster sizing, fuel efficiency, attitude accuracy, and mission scheduling flexibility. This parameter must be revisited and refined as mission-level constraints and pointing requirements become clearer in the subsequent design phases. The insights obtained here, alongside the slew manoeuvre analysis, will contribute directly to defining the final performance and operational requirements for the RCS subsystem.

Figure 9 can be extended into a three-dimensional representation to better visualize the performance of the RCS system. Specifically, Figure 10 and 11 present two different perspectives on the system design. In Figure 10, the external torque is treated as a variable parameter, as its value is difficult to define precisely at this stage. This analysis supports the final definition of the RCS by illustrating the range of external disturbances the system may encounter during its operational life. Conversely, Figure 11 generalizes the results shown in Figure 9, providing insight into the performance limits of the proposed RCS in terms of yaw control accuracy. This has an indirect but important influence on the selection and sizing of the reaction wheels.

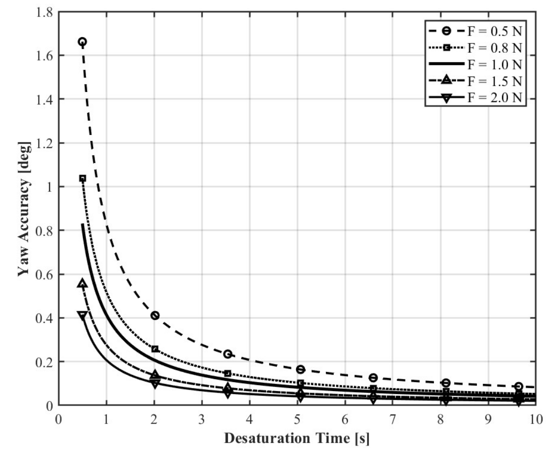

Fig. 9: Estimated yaw accuracy as a function of desaturation time using 1 N thrusters based on preliminary design assumptions.

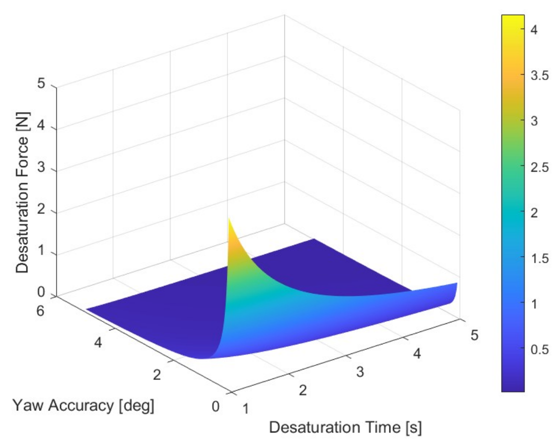

Fig. 11: 3D plot of the desaturation force needed as a function of the desaturation time of the reaction wheels and the yaw accuracy required by the RCS.

6. Conclusions

The comprehensive analysis of multiple mission scenarios has clearly identified the Kick Stage mission as the most suitable and promising use case for the Green SWaP system. This conclusion is based on both technical feasi-

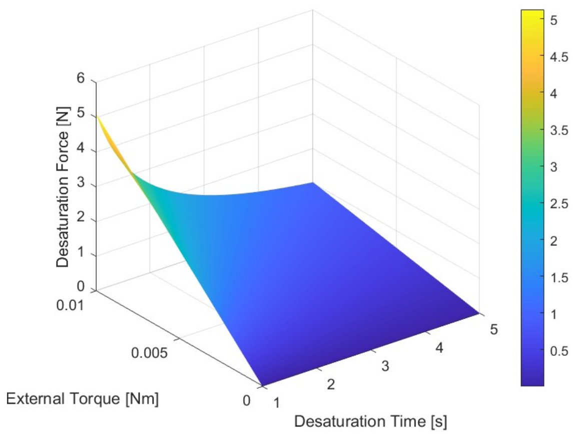

Fig. 10: 3D plot of the desaturation force needed as a function of the desaturation time of the reaction wheels and external torque level.

bility and alignment with the core objectives of sustainability, reusability, and in-orbit resource utilization. While the concept of a Kick Stage is not entirely novel, the innovation introduced by Green SWaP lies in its unique approach to sustainability and long-term usability. Specifically, it leverages a closed-loop, in-space refueling architecture based on water as propellant, with conversion systems already placed in orbit. This enables a mission that is not only reusable in principle but also operationally and environmentally more sustainable than current alternatives.

Using water as a propellant introduces several key advantages. It is inherently non-toxic and safe to handle, reducing risks during launch integration and on-orbit operations. It is also cheaper and more abundant than traditional chemical propellants. Importantly, the use of water opens up the potential for circular resource cycles in space (particularly through the recycling of wastewater from human-rated space stations or ISRU activities on the Moon). This aligns well with the long-term vision of in-situ resource utilization and regenerative mission architectures. From a system engineering perspective, the Kick Stage mission offers a rigorous and challenging use case. Because of its role in performing precise orbital injections, plane changes, or debris removal at various altitudes and inclinations, it imposes tight constraints on the propulsion system's thrust, responsiveness, ΔV delivery, and operational reliability. Selecting such a demanding mission as a baseline therefore ensures that the Green SWaP system is tested under one of the most challenging environments of the proposed scenarios.

Acknowledgements

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