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Conceptual study of technologies enabling novel green expendable upper stages with multi-payload/multi-orbit injection capability

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Abstract

The growing demand for cheaper space access calls for a more economically and environmentally sustainable approach for launchers. Concurrently, the shift to smaller satellites and the rise of constellations necessitate launchers capable of precise multi-payload/multi-orbit injection. ASCenSIon (Advancing Space Access Capabilities - Reusability and Multiple Satellite Injection), a Marie Skłodowska-Curie Innovative Training Network funded by Horizon 2020 (H2020), aims to respond to these demands. This paper describes the activities explored within ASCenSIon dedicated to developing novel green upper stages with multi-payload/multi-orbit injection capability. The aspects investigated here include the general system architecture, innovative solutions for the propulsion system (e.g., Hybrid Rocket Engines (HREs), green propellants and electric pump feeding), Guidance Navigation and Control (GNC) solutions for the multi-payload/multi-orbit injection capability, and reliability aspects of upper stages. First, relevant space market considerations are raised. Then, solutions for more environmentally friendly propulsion systems are proposed. Since identifying a good substitute for toxic hydrazine recently became a priority, the use of green propellant technologies will be assessed, tackling specific problems such as benchmarked propulsive performances, storability and material compatibility. Another promising solution for future propulsion systems with lower environmental impact are HREs. They bring benefits in terms of flexibility, safety and cost. However, high residual mass, oxidizer-to-fuel ratio (O/F) shift during operation, low regression rate and combustion inefficiency are some of the challenges that still need to be addressed in their application. In addition, electric pump fed systems, powered by green propellants, may be a game-changer technology for future upper stages. Compared to pressure-fed, it can provide improved performance and lower inert mass. With respect to turbopumps, it may also be advantageous in terms of simplicity and costs. On the other hand, battery mass and thermal control represent some of the drawbacks to overcome. Additionally, the implementation of novel GNC solutions is critical to ensure the multi-payload/multi-orbit injection capability. The challenges brought by the design of such a system are presented, including the correlation with the overall upper stage definition. Finally, the reliability of the launchers is a key aspect to protect both the space environment and the safety of the missions. Novel methods for reliability modelling of launchers are discussed and advantageous system architectures are proposed. These novel technologies being jointly assessed, this paper presents a preliminary analysis of the discussed topics and their interconnections within ASCenSIon, aiming at satisfying new requirements for novel green upper stages.

Keywords: **Future Upper Stages, Propulsion System, Hybrid Rocket Engines, Green Propellants, Electric Pump Feeding, GNC Solutions, Multi-payload delivery, Multi-orbit injection, Reliability and Safe Disposal, ASCenSIon Project**

Nomenclature

H_2O_2	Hydrogen Peroxide
I_{sp}	Specific Impulse
L/D	Length over Diameter
N_2H_4	Hydrazine
N_2O	Nitrous Oxide
O/F	Oxidizer-to-Fuel ratio

Acronyms/Abbreviations

ASCenSIon	Advancing Space Access Capabilities - Reusability and Multiple Satellite Injection
ADN	Ammonium Dinitramide
ADR	Active Debris Removal
AHRES	Advanced Hybrid Rocket Engine Simulation
AVUM	Attitude and Vernier Module (of VEGA)
DRAMA	Debris Risk Assessment and Mitigation Analysis
EIL	Energetic Ionic Liquids
EPF	Electric Pump Feeding
ETA	Event Tree Analysis
FMECA	Failure Modes, Effects and Criticality Analysis
FTA	Fault Tree Analysis
GEO	Geostationary Orbit
GHS	Global Harmonized System
GNC	Guidance, Navigation & Control
GP	Green Propellants
HAN	Hydroxyl Ammonium Nitrate
HRE	Hybrid Rocket Engine
HTP	High-test Peroxide
HTPB	Hydroxyl Terminated Poly Butadiene
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
LRE	Liquid Rocket Engine
MDO	Multi-Disciplinary Optimization
MMH	Monomethyl Hydrazine
OOS	On-Orbit Servicing
RBD	Reliability Block Diagram
SRM	Solid Rocket Motor
SSO	Sun Synchronous Orbit
TACS	Trajectory and Attitude Control System
TRL	Technology Readiness Level
TSP	Travelling Salesman Problem
TSTO	Two Stage to Orbit
UDMH	Unsymmetrical DiMethyl Hydrazine
VEGA	Vettore Europeo di Generazione Avanzata

I. INTRODUCTION

Space activities are in continuous evolution and in recent years the sector is experiencing a fast and abrupt expansion with novel public and private players entering the market every year. To face the growing space market challenges, topics such as environmental lifecycle assessment in early mission phase and sustainable space logistics are gaining everyday more importance aiming towards a more efficient use of resources. In this context, the European Union (EU) Space Programme is supporting Green Deal initiatives to reach net zero emissions [1], pushing forward the use of more eco-efficient and cost-efficient technologies to secure European space access in the long run.

Many companies and research institutions are trying to quantify the environmental impact of future space missions over their entire life cycle and minimize it with the introduction of novel sustainable solutions. While the upper stage by itself is aimed to be developed in a more sustainable way, especially with the implementation of greener end-to-end propulsive systems, it must also meet new requirements from both the GNC and the reliability sides. The latter involves specific requisites, especially for the propulsion system, when still in the early design phase. This aspect will be further explored in this paper. In addition to these demands, the upper stage propulsive system must already cover a wide range of requirements related to the type of mission. Indeed, those for the injection of CubeSats to multiple orbits are, for example, very different to the requests of a Geostationary Orbit (GEO) satellite injection.

The European Space Agency (ESA) allocated 18.1% of its 2021 budget to Space Transportation [2], highlighting the high motivation of developing more efficient, reliable, cost-effective and eco-effective launchers. The upper stage design is especially complex as it must cover the last miles, while relying on the lower stage(s) performances, to deliver the precious payloads onto the right orbits. The innovative technologies presented in this paper hold promises for future upper stages covering different areas such as the propulsive system, GNC solutions and reliability assessment.

II. AN INNOVATIVE PROPULSIVE SYSTEM FOR THE UPPER STAGE

The technologies featured here aim at bringing a real revolution in the way upper stages are conceived, making them more sustainable, efficient, cost-effective and more versatile to respond to the wide range of future missions by extending their portfolio. Regarding the propulsion system, innovative technologies such as Hybrid Rocket Engines (HREs), Green Propellants (GP) and Electric Pump Feeding systems are outlined.

II.I Hybrid Rocket Engines

One of the most promising technologies for the endeavor of sustainable upper stages that will be researched and assessed within the ASCenSIon project are Hybrid Rocket Engines.

HREs use a liquid (typically the oxidizer) and a solid propellant (in general the fuel). To allow the integration of HREs in upper stages, the Technology Readiness Level (TRL) of the system has to be increased by developing design methodologies and creating a robust and validated experimental database. HREs low maturity and technical challenges related mainly to their low regression rate, combustion inefficiency, and high residual inert mass, hindered their application to launchers, where conventional Liquid Rocket Engines (LREs) and Solid Rocket Motors (SRMs) are widely used. However, the recent shift in the market towards smaller, cheaper and sustainable launch systems led to an increased research interest in hybrid propulsion. The advantages of HREs (listed below) make this technology promising for small launcher applications and for advanced upper stages [3, 4].

- *Environmental Friendliness:*

Green propellant combinations can be chosen thanks to the versatility of the propellants.

- *Safety:*

Fuel and oxidizer are physically separated resulting in a negligible self-ignition risk. The fuel is inert and can be handled without special precautions. The combustion is diffusion controlled, reducing susceptibility to failures induced by instabilities.

- *Simplicity:*

Compared to LREs, HREs are simpler to design and require less components.

- *Low Cost:*

The choice of an inert solid fuel greatly reduces the cost of manufacturing, handling, storage and transportation.

- *Throttle-ability:*

HREs can be throttled by regulating only the liquid propellant mass flow rate, since the solid propellant consumption is related only to it.

- *Re-ignitability:*

Compared to SRMs, HREs can be shut down and re-ignited.

- *Propellants Versatility:*

The propellant choice is highly versatile, the desired performance can be achieved using different oxidizers and fuel compositions. The selection of 3D printable fuels can also allow grain designs that enhance the combustion efficiency of the engine [5].

- *Grain Stability:*

The inert fuel grain is insensitive to cracks and imperfections, that can be catastrophic in SRMs.

- *Performance:*

In general, the ideal specific impulse of a HRE is higher than the one of a SRM.

Nonetheless, considerable research efforts to tackle the aforementioned drawbacks of HREs have to be carried out. This includes:

- *Low Regression Rate:*

The regression rate of the solid fuel is more than one order of magnitude lower than that of SRMs, due to the diffusion flame behavior and the blocking effect on heat exchange and mixing. With non-conventional fuels and several other regression rate enhancement techniques under investigation the issue can be partially tackled [6], but it still poses a problem especially in large engines.

- *Low Volumetric Loading:*

The low regression rate leads to a long combustion chamber or use of multi-port grain fuel to increase the mass-flow ratio of the fuel and thus the thrust. A long combustion chamber means a small web thickness of the fuel grains leading to a poor volume loading. The use of a multi-port grain, on the other hand, leaves a moderate amount of unburnt fuel slivers. Moreover, the need for a post-combustion chamber to enhance the mixing of the propellants and the efficiency of combustion increases the mass fraction losses.

- *Combustion Efficiency:*

Due to the diffusion flame mechanism, a good mixing of oxidizer and fuel is difficult to achieve, inducing a combustion efficiency lower than that of LREs and SRMs.

- *Oxidizer-to-Fuel Ratio (O/F) Shift:*

The oxidizer to fuel ratio evolves over the time of combustion as the area of solid burning fuel varies. Consequently, the fuel mass flow rate changes while the oxidizer mass flow usually remains constant. Moreover, the regression rate is typically related to the oxidizer mass flux that changes as the port area of the grain changes. It is possible to keep the oxidizer to fuel ratio fixed at the optimal value but only with dedicated design choices, leading in general to a shift of performance and specific impulse.

- *Slower Transients:*

Having the oxidizer and the fuel in different phases, HREs present a longer transient ignition than SRMs and a slower response to throttling than LREs.

Research Efforts for Upper Stages

Launch systems where all stages are powered by hybrid propulsion are limited in their payload capabilities, given some thrust limitations of today's HREs. However, combining the high thrust of traditional chemical propulsion systems for the lower stage(s) with the low-cost, overall flexibility and re-ignitability of an HRE in the upper stage can potentially disrupt the launcher market. The expensive, sophisticated main stages can be reused and the cheaper, yet efficient HRE can be disposed after multiple payload injection.

In 1992, Estey *et al*. [7] exemplary described the design choices of upper stage HREs they considered for the AMROC Aquila launcher. They pointed out that it would be beneficial to utilize the main oxidizer of the upper stage HRE also for attitude control during non-powered flight phases. For this, Nitrous Oxide (N_2O) was selected as an oxidizer, due to its high vapor pressure. However, Hydrogen Peroxide (H_2O_2) and its capability to ignite using a catalyst bed also poses a suitable alternative. Furthermore, the authors pointed out several trade-offs to be considered. Choosing a higher I_{sp} propellant like Liquid Oxygen (LOX) would increase the payload capability but at the expense of system complexity and cost.

Around 2000, NASA funded investigation on upper stage HREs with the Hydrogen Peroxide Hybrid Upper Stage Program. The program opted for a gas-generator aft-injection design using 90% H_2O_2. The motor design was capable of throttling and shutdown. However, some after-burning was observed, which is postulated to not occur under vacuum conditions [8].

Within the European ORPHEE (Original Research Project on Hybrid Engine in Europe) project, among other applications, an upper HRE was inspected [9]. The Use-case was a substitute of the Vega upper stage motors (*AVUM* and *Z9*) with a single HRE. A first turbopump-fed LOX concept (both paraffin and HTPB with metal additives as fuels) promised payload increases of up to 60% [9]. However, in the final report, the project partners prioritized HTPB formulations and acknowledged H_2O_2 (99%) as an interesting alternative.

At Space Propulsion Group, Karabeyoglu *et al*. [10] applied their newly researched liquefiable fuel approach to investigate the replacement of the *Orion 38* solid upper stage motor (I_{sp} = 290s) with a LOX/Paraffin HRE. The authors state that rather than implementing the HRE technology directly to a whole launcher system, it could be beneficial to apply the HRE to relatively small systems like upper stages. This would help accelerating the maturing of HREs. The authors also argued that the upper stages' technology can be adapted to suborbital touristic flights, where a huge growth is to be expected. In an extensive systems design study, the authors calculated that the launchers that use *Orion 38* could benefit from the shift to an HRE upper stage with a payload capability increase of up to 40%.

Starting from 2010 until today, Casalino *et al*. [11, 12, 13, 14] extensively research a potential HRE application of the third and fourth stage of a *Vega*-like mission profile. The HRE would combine the initial 3rd and 4th stage in one single stage. The ingenuity of their approach is to optimize the design (grain geometry, initial thrust, initial mixture ratio, nozzle expansion ratio, initial tank pressure, initial chamber pressure, port to throat area ratio) and the trajectory simultaneously in a multidisciplinary optimization (MDO) approach to maximize the launcher payload for a 700 km circular polar orbit. In several iterations over the years, Casalino *et al*. [14] concluded in 2019 that a LOX/Paraffin design with electric pump feeding (considering most advanced battery technology found in literature at that time [15], the total payload mass achievable would be 2,467.7 kg as compared to the 1,500 kg of the initial *Vega* design [14].

At the German Aerospace Center (DLR), the program AHRES (Advanced Hybrid Rocket Engine Simulation) [16, 17] was started in 2011. AHRES combines CFD tools and software engineering tools for design and optimization of HREs. Two studies were conducted: In 2012 the substitution of the 3rd and

4th stages of *Vega* with a single HRE [16] and later in 2014 to evaluate an HRE substitute for the third stage of the Brazilian Space Agency's Microsatellite Launch Vehicle (*VLM-1*) [14, 17]. In both cases, the chosen oxidizer is a highly concentrated H_2O_2 and the fuel a HTPB blend (either with polyethylene or paraffin) with metal additives. According to the authors, the proposed designs can compete or even surpass the initial solid stage designs when it comes to architecture, final mass and performance [16, 17]. However, the cost is estimated to increase by a factor of 1.5-1.7. For a liquid engine substitute, on the other hand, the cost factor in the *Vega* case would be at least 5 times higher.

In 2020, Barato *et al.* [18] developed (semi-) analytical expressions for cylindrical single-port HREs. Using these functions, Barato [19] reviewed the specific case of upper stage HRE applications in 2021. Based on his investigations, he drew three major conclusions. First, he pointed out that contrary to boosters where a maximization of the regression rate is important, for upper stages, the regression rate sometimes has to be capped in order to avoid unreasonable ratios of initial and final diameter and overall grain design. Additionally, it was shown that HREs can replace LREs predominantly for small Δv or cases of minimum acceleration in a fixed time frame (e.g. capture maneuvers). In order to compete with solids, the mission requirements have to justify the hybrids advantage of throttling and restarting. As for the dimensions of the upper stage, Barato highlights the importance of the length-over-diameter (L/D) ratio. Hybrids cannot compete with the compact designs possible with LREs or SRMs in upper stages (the Russian space tugs having L/D smaller than 1). However, novel designs of the oxidizer tanks (like annular around the fuel grain) as also proposed by Karabeyoglu *et al.* [10] and Bozic *et al.* [17] can counter the higher L/D ratios of HREs. Lastly, Barato stresses the problematic of the heat soak-back effect (after shut-down heat from the nozzle can potentially damage the remaining fuel inside the combustion chamber) for ablatively cooled HREs. This effect needs to be thoroughly considered if an upper stage HRE needs to restart reliably.

Summarizing the collected works, it is evident that although HREs can pose a real alternative for upper stages, many challenges are yet to be overcome. Among others, in-space re-ignitability, tailored regression rates, excessive L/D ratios and the overall low maturity of HREs pose serious obstacles. Moreover, the use-case of the upper stage has to really demand for an HRE application (sustainability, safety, true benefit of throttling and restart capabilities). Nonetheless, with the rise of the "New Space" era, several start-ups are incrementally increasing the TRL of HREs. Several suborbital rockets have flown in the recent past powered by an HRE, like Virgin Galactic SpaceShipTwo, and several launchers are under development, with a list given in Table *1*.

Developer	Launcher	Stages	Payload	Propellants
Equatorial Space Systems (Singapore)	Volans	2	220 kg @ LEO 150 kg @ SSO	N_2O / Proprietary composition
Gilmour Space (Australia)	ERIS	3	215 kg @ SSO	LOX / 3D Printed Fuel
HyImpulse (Germany)	SL1	3	500 kg @ LEO	LOX / Paraffin
Nammo Raufoss (Norway)	North Star	3	50 kg @ SSO	H_2O_2 / HTPB
tiSpace (Taiwan)	Hapith V	3	390 kg @ LEO 350 kg @ SSO	N_2O / Butadiene
VAYA Space (USA)	Dauntless	2	1,000 kg @ LEO 610 kg @ SSO	N_2O / 3D Printed Fuel

Table 1 – HRE launchers under development.

II.II Green Propellants

When discussing the environmental impact of upper stages, it is key to mention the growing importance of the propellant choice. The currently most used technologies for upper stages are liquid thrusters. Depending on the level of thrust and on the generic requirements of the mission, there are two main categories of upper stage systems: mono-propellants and bi-propellants systems. The former exploits compounds that spontaneously ignites in contact with a catalyst bed while in the latter a mixture of fuel and oxidizer initiates the combustion.

In both cases, the most widely used propellant is Hydrazine (N_2H_4) and its derivatives, namely Unsymmetrical DiMethyl Hydrazine (UDMH) and Monomethylhydrazine (MMH). Hydrazine is and has been the most common rocket propellant for decades due to its ideal thermo-physical properties and high propulsive performances. Its use in space propulsion was introduced in the 1960s and there is extensive literature describing its properties and operational aspects, showing great maturity [20].

Unfortunately, the compound is also a well-known toxic agent, harmful for both the humans and the environment, and in the EU's list of Substances of Very High Concern (SVHC) since 2011 [21]. It is recognized as a very dangerous agent in most countries worldwide besides being the primary suspected cause of abnormally high rates of hormonal and blood disorders around the launching site in Kazakhstan [22]. This high health hazard makes Hydrazine's handling and storage procedures extremely dangerous and expensive as strict safety measures are required.

In this context, the need to identify a suitable alternative to Hydrazine is crystal clear. This is an opportunity to implement a more global approach to the environmental impact of the different propellants through the overall chain from production to their use in space. This task is however not simple and, while a lot of research is currently ongoing and some prototypes have already flown in space, requires extensive work. Some alternative compounds, usually referred to as "green propellants", are already well-known and studied for decades but their technologies are often still immature and need further developments.

Green propellants are generally defined as non-toxic storable propellants with low environmental impact and high performances [23]. The definition of what is considered "green" is broad and depends as well on its foreseen application. Indeed, while the ideal final goal remains to identify a low-environmental impact propellant over its entire lifecycle, from production to use, the factors to consider vary drastically with respect to the given stage. For instance, when studying a propellant's impact on the environment, analyzing the combustion exhaust products holds a crucial role when looking at main stages while this aspect is not relevant for upper stages as the emissions happen above the "space limit" and do not endanger our atmosphere anymore.

The other way around, considerations irrelevant for main stages, become primordial for upper stages, like the system re-ignitability to comply with the space debris mitigation guidelines, which needs to be enforced in order to keep space clean. Indeed, to not become an orbital debris, future upper stages should ensure successful disposal within 25 years with a certainty of at least 90% [24]. Within this time, passivation measures must be applied and therefore integrated upstream in the design of the system. One of the most important is reducing the risk for the propulsion system to explode by venting the tanks and emptying the batteries [24]. Venting and depressurization measures bring direct requirements for the propulsive system, should it be a monopropellant system in blow-down mode or a pressure-fed bipropellant. The implementation of those, while re-designing the propulsive system to be greener and more flexible to different missions, is a real challenge and each one of the technologies presented here, could be able to tackle this challenge in the future.

The overly large number of space debris is a well-known threat and a widespread international concern to mitigate. While the methods to do so are still under discussion, the current mitigation guidelines requiring the presence of de-orbiting strategies represent the best options. Incorporating de-orbiting strategies stands as a valuable initiative but imposes at the same time further requirements which must be addressed during the design phase, including the cost calculation for the additional propellant needed on board. ESA invested time on this topic and its tool DRAMA (Debris Risk Assessment and Mitigation Analysis) is, for example, a useful asset to assess this question.

Promising fuel candidates, especially studied for monopropellant systems, able to replace hydrazine can be divided into three main families [25]: Energetic Ionic Liquids (EILs), liquid Nitrous Oxides compounds (NOx) and Hydrogen Peroxide aqueous solutions. In order to fall under the "green" umbrella, a propellant is usually evaluated with respect to different categories such as toxicity, storability, performance, cost and availability.

- *Toxicity*

Toxicity is one of the most important factors as it influences most of the others. Compound toxicity is generally evaluated using the Acute Toxicity Classification (ATC) level, a 1:5 scale delivered by the Global Harmonized System of classification and labelling of chemicals (GHS) [26]. A score of 1 is attributed to the most toxic species and a score of 5 to the less toxic ones. While there is no international agreement on the GHS labelling, the hazard posed by many substances currently in use is widely recognised. Hydrazine, for example, is highly toxic and carcinogenic and displays a GHS level of 2. Very often a propellant can be referred to as "green" by displaying a GHS level only one unit higher than hydrazine, i.e., from 3 and higher [23]. This wide definition includes a substantial number of substances under analysis for years such like the EIL ADN-based compounds which are only moderately non-toxic and display a GHS of 3. Most of other green monopropellants show, on the contrary, a GHS closer to 4 or even 5, such as N_2O which is considered non-toxic. Beyond the health hazard itself, the most interesting aspect of moving towards less-toxic propellants is the drastic reduction of handling and storage costs by limiting the necessary but expensive safety procedures.

- *Storability*

Storability considerations for Earth and in-space operations must examine not only the compound toxicity, but also other crucial aspects such as being in liquid form within the range of operation, having a low freezing point, being stable (no decomposition, no spontaneous ignition) and being compatible with common tank materials. These factors are major drawbacks to overcome in the development of new propulsive systems based on new substances. Material compatibility is, for example, one of the most important disadvantages of Hydrogen Peroxide which is not compatible with Titanium.

- *Performance*

Regarding performance, in order to be competitive on the market, green propulsion systems must display an equivalent or higher specific impulse (I_{sp}) compared to state-of-the-art hydrazine-based components. An additional and ideal property of new systems is the possibility to work in multi-mode (i.e., in both mono- & bi-propellant systems). This feature has been initially studied for Hydrogen Peroxide, Nitromethane, and some HAN-based EILs, showing promising results [25].

- *Cost & Availability*

While crucial for the choice of propellant, cost and availability are also difficult to evaluate due to the volatile nature of both the supply chain and the market. These aspects are therefore not investigated in detail in this article. An example worth mentioning is the high cost of ADN-based monopropellant compared to Hydrogen Peroxide which is relatively cheap [27]. However, an increasing market interest in exploiting ADN-based fuels would optimize its production and lower its cost, like it has been the case to produce high-quality High-test peroxide (HTP).

Considerations on some of the most promising compounds are summarized in Table *2*.

In short, the perspective of a ban of Hydrazine by the REACH (Registration, Evaluation, Authorization and Restriction of Chemicals) regulation is enhancing the research on green propellants. This could also respond to the new space market needs by increasing the flexibility of space missions and increasing the performance of the propulsion system while lowering the total lifecycle cost of operation. However, introducing a new propellant for space activities also brings new challenges such as re-designing the whole propulsive system, testing and developing it. While there are many good candidates with their respective advantages and disadvantages, the matching between a propellant and its use-case requires further maturation in view of its extensive introduction and utilization [28, 29].

Propellants	Pros	Cons
Hydrazine (N_2H_4)	- High I_{sp} - Well-known technology (High TRL) - Simple system design	- Toxic (GHS 2) - Extra cost for handling (strict safety measures)
Hydrogen Peroxide	- High maturity - High performance in bipropellant mode - Can be used as an oxidizer in bipropellant and pure as monopropellant (multi-mode) - Hypergolic ignition - Cheap, commercially available	- Not compatible with Titanium - Low performance in monopropellant mode - Careful handling required - Significant self-decomposition rate
EIL HAN-based	- Good performances - Low vapor pressure - Good stability in pulsed or continuous mode	- Low TRL, except for AF-M315E - High Combustion Temperature
EIL ADN-based	- High maturity - Flexible ignition (catalytic, electric or thermic)	- Moderate toxicity - High vapor pressure - High cost
NOx compounds	- Self-pressurization properties - Good storability & stability - Low toxicity	- Extremely high chamber temperature - Low density

Table 2: Promising Green Propellants compounds

II.III Electric Pump Feeding Systems

On the feeding system's side, electric pump feeding may represent a promising alternative to both conventional pressure-fed systems and turbopumps. While the first method relies on highly pressurized propellant tanks, the second uses pumps to provide the required pressure inside the combustion chamber. Systems where turbines are used to power these pumps are known as turbopump-fed systems. Expander, staged, and gas generator cycles belong to this category. As opposed to classical turbopump systems, electric pump-fed systems use electric motors powered by batteries rather than turbines to drive the pumps [15].

When implemented in upper stages, electric pump feeding can become a game-changing technology optimizing the performance while limiting mass and complexity. To assess this possibility, this section reports the main advantages and disadvantages of this innovative pressurization system with respect to the traditional pressure and turbopump-fed.

Electric Pump Feeding vs Pressure Fed

Pressure fed systems are classically used for small to medium thrust applications. Their principal assets are simplicity and reliability. Its substitution by electric pump feeding can bring the following advantages:

- *Dry Mass Reduction*

The implementation of electric pump feeding allows a decrease in the propellant tank pressures. Lower tank pressures entitle the use of thinner walls and, ultimately, lighter tanks. The second consequence of this reduction in pressure is a decrease of the Helium volume required to pressurize the tanks, and hence, a further reduction in mass [30]. This dry mass reduction leads to an increase of the system Δv capabilities and payload maximization.

- *Improved System Performance*

The utilization of pumps allows achieving higher chamber pressures [30]. This translates into an increase in specific impulse, and thus, enhanced propellant mass efficiency and system Δv capabilities.

- *Transient Reduction & Weaker Coupling*

By tailoring the pump speed, and hence its frequency, start-up, shut down, and combustion instabilities can be damped [31]. Moreover, a softer transient and a weaker coupling between the feeding system and the combustion chamber are also beneficial for both the payload and the electronics.

- *Update-ability*

Improvements can be made with new software updates or by the implementation of new technologies, i.e. new battery technologies.

Electric Pump Feeding vs Turbopump Fed

When confronted with turbopump fed systems, the combination of short burning times and high thrust leads to a penalty for electrically driven pumps [32]. Thus, when high amounts of power and energy must be delivered within a very short time, as it is often the case for first and boost launcher stages, the turbopump fed option remains unbeatable in terms of performance.

Instead, for low to medium thrust and not extremely short burning times, electric pump feeding can be a feasible alternative not only to pressure fed but also to turbopumps. Turbopumps have indeed been proven highly inefficient for these thrust ranges due to, among other things, their high complexity. The main advantages of electric pump systems with respect to turbopumps are therefore:

- *Reduction of System Complexity and Components*

Electric pump-fed systems are easier to handle due to their decreased mechanical and thermal complexity. For instance, the huge temperature gradients, characteristic of turbopumps, can be avoided. Moreover, plumbing and other components such as gas generators, pyro-starters or other valves can be left behind. This component reduction leads to a higher simplicity, which at the same time turns into lower development and operational costs and higher reliability [15].

- *Quick and Safe Updating Capabilities*

In the same way as for pressure fed systems, improvements can be easier to achieve by software updating and implementing the latest version of batteries, motors, and inverters. These components can be more easily replaced without compromising some of the already performed test campaigns. Most of the problems faced by turbines and gas generators are turned into software problems in electric feeding systems which are way easier to manage.

- *Improved Re-ignitability*

The ignition sequence is dramatically simplified thanks to the use of electric motors instead of turbines, and consequently, the re-ignitability capacity is highly simplified by the electric pump-fed approach [15]. Satellite multi-injection capabilities, as well as final shot to de-orbit, could then easily become a reality.

- *No Loss in Specific Impulse*

Contrary to what happens in gas generator cycle all the propellant contributes to the gain in specific impulse [15].

Electric pump-driven systems clearly show promising features but are not immune to possible drawbacks. Among these challenges, the most relevant ones are summarized below:

- *Decreased Pump Efficiency for Low Mass-Flow*

For low and medium thrust applications, pump efficiency can be a concern. Its improvement can notoriously reduce the overall pressurization system mass of electric pump-fed systems [31].

- *Battery and Electric Motors Mass and Efficiency*

Current developments in electric motors and batteries allow getting high power and energy densities [31]. However, the battery mass is still the main design concern [15]. Moreover, battery efficiency is extremely temperature-dependent and may require heavy thermal insulation.

- *Thermal Control*

Thermal control of batteries and electric motors is critical to secure the systems proper behavior [15]. Power conversion releases heat that needs to be adequately managed by the employment of a cooling system (radiative, regenerative, etc.). The selection of the appropriate technology is a critical design parameter as it contributes significantly to the inert mass.

- *Single or Double Shaft Selection*

Implementing a single shaft configuration for driving both propellant pumps reduces the overall mass but at the expense of the system efficiency.

- *Number of Electric Motors*

Whereas using more than one electric motor can improve reliability and refrigeration (as the surface to area ratio increases), it also entails more complex plumbing [31].

With still some challenges ahead to overcome, namely the battery mass, electric pump fed system offer significant attractive perspectives. Indeed, more performing than pressure-fed systems it is also offers a net reduction of the complexity brought by turbopump-fed systems. The gain in simplicity, cost and reliability of the overall propulsive system makes it a ground-breaking technology in need for more study and maturation.

III. GNC SOLUTIONS FOR DELIVERING MULTIPLE PAYLOADS TO DIFFERENT ORBITS

To complete the design of the upper stage, the GNC system that will drive its motion during the operational phase must be defined as it will conditionate the success of the overall mission. Typically, such a design is relatively standard regarding both the guidance algorithms and hardware, as usually they must deploy a single payload. However, the increasing number of new satellites envisaged in the near future, due to the higher amount of constellation missions and the escalating interest in small satellites [33], demands of newer injection strategies. Indeed, the current delivery strategy of several satellites relies on piggybacking, in which secondary payloads are inserted in orbits alongside the primary load.

Such a strategy poses a burden in the flexibility of these satellites' missions, which are conditioned both in terms of operational orbit and launch scheduling by the primary load. As a solution to this problem, ASCenSIon proposes the upper stage to have the capability of delivering multiple payloads into differentiated orbits, increasing the number of satellites that can be launched at once and reducing the time-dependence, thus increasing the cost-effectiveness of the launcher.

To achieve this capability, the upper stage must be provided with a GNC system that ensures the effective delivery of the satellites according to customer requirements. For this purpose, a correct design of the guidance algorithm, the upper stage shape, and the mounted hardware are crucial.

III.I The Motion of the Upper Stage for Multi-Orbit Multi-Payload Injection

One of the most complex parts in the development of the GNC system for the multi-orbit multi-payload injection is the design of a guidance and control strategy such that the different orbits are visited while minimising the consumption of fuel and the overall mission time. The definition of this guiding trajectory involves a multi-target rendezvous motion between different orbits. Reaching an optimal trajectory for this problem requires the analysis of both the visiting sequence and the transfer between two consecutive orbits. This problem has already been studied before with the applications for Active Debris Removal (ADR) and On-Orbit Servicing (OOS) activities.

The selection of the ideal visiting sequence is a combinatorial problem in the optimisation domain, similar to the Travelling Salesman Problem (TSP) in which the solution is the shortest path allowing the salesman to visit once every city and return to the origin. However, when dealing with orbits as the "cities", several differences to this classical problem arise. Firstly, due to the nature of orbital dynamics, the problem is in fact time-dependant and the cost of travelling between two nodes will vary with time. Secondly, the cost of traveling from one orbit to the next does not necessarily mean that the opposite motion would have an equal cost, making the problem asymmetric. Finally, while the typical TSP is a closed route, the orbit visiting one is not as it starts at a certain parking orbit (or at a ground launch point) and finishes at a disposal orbit to comply with the mitigation guidelines.

Such a problem has been already investigated in literature, for which both exhaustive and heuristic methods have been proposed. The former, while providing with more accurate results, are computationally unfeasible for large numbers of orbits due to the factorial growth of the search space in combinatorial problems [34]. The latter, which trade optimality in the solution for time efficiency, have been proved to reach sufficiently accurate results in this type of problem and are therefore more interesting in terms of flexibility.

Several algorithms have been studied and proven such as Ant Colony Optimisation (ACO) [35], the Particle Swarm Optimisation (PSO) [36], Simulated Annealing [37], or Genetic Algorithms (GA) [38]. Most of these solutions, however, greatly simplify the cost calculation of the transfers themselves, focusing on the TSP part of the problem. A proper transfer cost computation is crucial to correctly optimize the problem, as it will determine a more realistic cost in terms of needed transfer time and fuel consumption. Therefore, to obtain the optimal trajectory guidance for the translational part, a hybrid optimal control strategy is to be followed in the current GNC design process, such that both time and fuel consumption are minimised, by optimising the visiting sequence and the individual transfers at the same time.

While the translational control of the upper stage is of great importance, the attitude control is also necessary, as it will ensure that the transfer manoeuvres are correctly performed and that the payloads are accurately injected. The attitude control law, therefore, must be included in the optimisation problem stated before as a constraint in the motion of the upper stage, as it will affect the accuracy of insertion. Such a consideration has not been done in literature before, to the current knowledge of the authors, ignoring a characteristic of the spacecraft's motion that will significantly affect its performance. The design of such control algorithm, however, is not simple due to the varying inertia of the body, both due to fuel consumption and to the deployment of the payloads, which makes the inertia tensor value change discontinuously. To control the body under this condition, several approaches could be followed, including switching control strategies or a single adaptive controller. These options shall be studied in the design of the GNC of the upper stage for multi-payload multi-orbit injection.

Finally, an additional design feature that must be considered for this attitude control is the shape of the upper stage itself, and the distribution of the different payloads stored within it, that will define the initial inertia tensor and its evolution. An analysis of the upper stage optimal shape should be done when defining controller design, framing and foreseeing effective collaborations between the GNC and system designs.

III.II The Effect of GNC Hardware on the Design

The design and the performance of the GNC highly rely on the set of hardware that is mounted on the upper stage. Regarding the translational motion, the strategy is highly dependent on the propulsion system performance, explained in previous sections. The upper stage thrusters' operation modes will drive the transfer strategy, either by means of finite thrust or continuous low thrust manoeuvres. Fuel consumption and total mission time will be differently affected depending on the chosen architecture and GNC must adapt accordingly.

The other important limitation is the upper limit in number of re-ignitions that the propulsion system can provide. This will determine the maximum number of payloads that can be delivered, constraining the complete TSP problem in terms of the possible visiting orbits. Due to the variability of these factors, both continuous and discrete thrust strategies need to be considered when defining the transfer optimization in the design of the trajectory guidance. On the other hand, as stated before, attitude control is crucial for the correct injection of the payloads. Therefore, the decision on the hardware implemented in the Attitude Control System is of great importance.

As a matter of fact, it is necessary to select a set of control actuators that accurately follow the commands of the attitude controller in a timely and precise manner. The decision on the actuators will not only affect the insertion accuracy, but if a set of thrusters are selected for this purpose (such as a Reaction Control System) the additional fuel consumption generated by their usage needs to be considered in the optimization problem. The analysis of the implications in terms of control thrusters needs and requires a close coordination between GNC, propulsion and system design activities.

Finally, while not directly affecting the performance of the optimal trajectory, the knowledge of the upper stage with respect to its own dynamic state will affect the performance of the overall mission. As such, coordination with the decision on the navigation sensors to be mounted must be done, and an analysis of the impact that these have on the state knowledge must be performed.

This includes not only the type and number of sensors to be used, but also their performance parameters, in order to design a proper navigation filter. In such a way, the design of the GNC algorithm needs to be developed with constant feedback from the hardware design of the upper stage.

III.III Future Applications

As stated earlier, two of the main applications for which the multi-target rendezvous problem is studied are the ADR and the OOS. From the upper stage design point of view, it is more interesting to analyze the cases in which the vehicle can continue its mission after the delivery of the payloads.

The design of the spacecraft would not differ much from that of the injection purposes, except for the necessary specific equipment for the other mission scenarios (for instance deorbiting kits for ADR or bigger fuel tanks for OOS). In these cases, the satellite must optimize both the visiting sequence and the transfers, and the mass would also change discretely after performing an activity at a certain orbit. Therefore, the GNC design principles would be the same. It would therefore be of interest to test such design for these mission purposes and analyse whether the new applications would affect the overall upper stage definition (whether in the GNC system, the shape, or the hardware). This study needs to be done with constant feedback from the upper stage design point of view, like the previous ones, to ensure correct definition and performance.

IV. RELIABILITY AND SAFE DISPOSAL

Reliability can be defined as the probability that a system will perform its required function without failure under stated conditions for a stated period of time [39]. Ensuring the reliability of space launch vehicles is essential to protect the sustainability of both Earth and space environment, as well as the missions being launched. Traditional approaches to reliability modelling of launch vehicles include the use of methods such as Fault Tree Analysis (FTA); Failure Modes, Effects and Criticality Analysis (FMECA); Event Tree Analysis (ETA); Reliability Block Diagram (RBD) and so on. However, these methods require extensive knowledge about the system and are very time consuming.

The main focus within ASCenSIon will therefore be on the development of new reliability methods that can already be applied during early design stages, allowing to make choices at the system architecture level that will benefit the reliability of the launchers. Additionally, a special focus will be put on the reliability of the upper stages. These stages are the most worrisome for the sustainability of the space environment, being usually the only ones that reach orbital velocity and therefore the ones that could stay in orbit for long periods of time, turning into orbital debris.

IV.I Current Reliability Methods

Literature [40] provides a good description of the most widely used methods to assess reliability, with the main ones being briefly described below.

- **FMECA**: consists of analyzing the potential failure modes and their consequences on the system. The failures are then classified according to their criticality and likelihood of occurrence.
- **FTA**: deductive analysis inspecting the sequence of events leading to potential failures. The combined probability of these events can lead to a quantitative estimation of the reliability.
- **ETA**: studies all the possible system responses for a determined initiating event.
- **RBD**: logical representation of a complex system. It can be used to get a quantitative estimate of the reliability from the combinatorial probability of failure of its components.

It is easily observable that these methods require a very detailed knowledge of the system, as well as of the mechanisms or events that can lead to a failure. Additionally, they can be extremely complex and time-consuming. As an example, seven different teams were required to perform the FTA for the investigation of the accident of the Columbia Space Shuttle, resulting in a fault tree of 1,560 elements [41]. This type of analysis is therefore not suitable for the early stages of the design. Consequently, reliability analyses are mostly performed in late stages when the most critical elements are already defined. It is crystal clear that a great benefit could come from assessing the systems reliability at earlier stages.

IV.II Reliability of Current Launchers

The first steps to implement models that can help to improve the reliability of launchers is to analyze the launch failures over the past years, in order to understand the subsystems that were involved in these failures and that should therefore be the focus of new reliability models. Additionally, it is interesting to divide these failures by mission phase, since the failures during the ascent will typically imply the loss of the mission but the failures during the payload injection phase can additionally imply the creation of orbital debris. This analysis was performed over the failures in the last 15 years in [42], and the main results relevant for this work are introduced here.

First, the analysis was performed without any distinction regarding different mission phases. In this case, a 52% of the failures was caused by the propulsion subsystem. The remaining failures were mainly caused by the Trajectory and Attitude Control System (TACS) and the Separation Systems, with a 20% each. The last 8% was shared between telemetry, structures, and the power storage and distribution subsystems.

The failures were then divided by mission phase. The phases considered were the ascent, occurring from the beginning of the launch until the separation of the upper stage, and the payload injection phase, from the separation of the upper stage until the injection of the last payload. The failures during the decommissioning phase were not considered due to the difficulties in retrieving this data but are expected to be analysed in future work.

The most relevant failures in the scope of this work are therefore those which occurred during the payload injection phase although the results do not differ a lot among the phases. With the propulsion still being responsible for about half of the failures, the main difference was found in more failures due to the separation subsystem in the ascent phase and more failures in the TACS during the payload injection, which is coherent with the characteristics of each mission phase. In any case, the results clearly show that the reliability efforts should be focused on the propulsion system in the first place.

A deeper analysis of the propulsion failures, also presented in [42], showed that 25 out of the 26 propulsion failures found occurred in Liquid Rocket Engines (LREs) while only one occurred in a Solid Rocket Engine (SRE). Moreover, more than half of these failures were provoked by the feeding system. In this respect, the foreseen synergy between more sustainable propulsive technologies studied in ASCenSIon, such as EP in combination with HRE or GP can bring the extremely fruitful advantage of increasing the reliability of the system.

IV.III Preliminary Methods to Model Engine and System Reliability

As stated before, the goal within ASCenSIon will be to develop a methodology that can allow to model the reliability of the launchers from early design stages. The resulting reliability model will aim to be included in a MDO, resulting in a more reliable vehicle design.

D. A. Young [43] shows how system reliability calculations can be included in a MDO process, using dynamically changing Fault Trees for these calculations. Z. Huang *et. al*. [44] identify key reliability drivers for liquid rocket engines identified and derives a parametric model to account for the main drivers. Finally, K. O. Kim [45] makes use of a baseline engine to estimate the reliability of a newly developed one, by assessing changes on propellant and engine cycle through a similarity analysis and utilizing mathematical relationships to account for different thrust design and testing time.

These studies show that it is possible to assess the system reliability, and more specifically the reliability of LREs, in early design stages where the detailed list of components is still unknown. Even though the reliability estimation will be less precise and will not account for some specific interactions or failure modes, it can provide very valuable information for the design of the vehicle.

V. SUMMARY AND WAYS FORWARD

The rapid expansion of space activities in the recent years has brought an enormous market pressure on the European space sector, namely with the emergence of new challenges for future launchers. Keeping up with these demands calls for technology innovation which ESA has planned around three main axes: competitiveness, versatility and diversification. While cost-driven competitiveness aims at promoting a more end-to-end approach of launchers development, a versatile and diversified space logistics will increase the launchers capability in performing different types of mission and to reach a wider range of orbits with each launch.

Assessing and mitigating the global environmental impact of space activities is another trending topic in which ASCenSIon members are involved at every step. Indeed, advancing the maturation, and more importantly the synergy, of the novel technologies under study in the project not only fully align with the EU space sector's trend but will also offer new insights. Among the technologies reviewed in this paper, it is clear that GNC delivery and safe disposal measures will give direct benchmarks for the propulsive performances, namely in terms of re-ignitability possibilities for multi-orbit injection and of passivation/safe disposal measures. Meeting these new requirements while, in parallel, investigating greener propulsive systems for future upper stage is a real challenge. Promising options such as HREs, green propellants and e-pump have been highlighted in this paper. While the TRL of HRE and of GP is not to be proven anymore, e-pumps, on the contrary, are quite new and could be combined with both.

Indeed, a preliminary study of GP & e-pump versus pressure-fed toxic propellant option for a kick stage [46] showed an increase in specific impulse while keeping the system mass constant. For what concerns the use of HREs in the private space transportation sector, several start-ups are developing their own small launcher using this propulsion system, as was shown in Table *2*. Among the ones listed, four are using a green oxidizer (N_2O or H_2O_2), and at least one of them is investigating into the exploitation of electric pump feeding to increase the performance of the launcher [47].

Advancing the maturation and the synergy of these technologies within ASCenSIon could lead to real game-changing perspectives in the European market providing cheaper, safer and more diversified space access to Europe. Indeed, cost remaining the most important design driver for future launchers, the possibility of decrease in complexity of these systems is very attractive. These technologies are initially developed independently with the final goal of implementing together the best possible solution.

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