

# MICROGRAVITY RESTART OF LIQUID ROCKET ENGINE WITH LOW PROPELLANT RESIDUALS

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The active deorbiting and passivation of launch vehicles has become key for the implementation of modern space debris mitigation guidelines. Appropriate engine restart conditions must be provided as part of this process. Ullage motors have been traditionally employed to induce active settling and ensure a gas-free propellant supply to the engines. Although robust and reliable, ullage rockets are also heavy, which motivates the study of alternative approaches to the problem. This paper explores for the first time several high-risk-high-return propellant settling strategies that may result in significant benefits for future space systems. In particular, three distinct Magnetic Positive Positioning concepts, an hydrogen-peroxide-based Propellant Gasification System, and an hybrid device that combines both approaches are introduced. The preliminary feasibility analysis indicates that the successful development of these technologies may lead to mass savings of hundreds of kilograms and benefits of several hundred thousand dollars per launch. However, the robustness of some of these methods may be compromised by complex fluid-structure interactions that require a careful numerical and/or experimental analysis.

## INTRODUCTION

The exploration and commercialization of space has led to the increasing contamination of the Low Earth Orbit (LEO) by non-functional man-made objects. Space debris represents a serious safety hazard for current and future satellites due to the risk of in-orbit collisions, and a concern for the general population during uncontrolled re-entry events. The minimization of debris release during normal operations has consequently become a major goal for the international space community.<sup>1</sup>

Launch vehicles (LVs) represent more than 42% of the abandoned intact objects in orbit and account for 57% of the abandoned mass.<sup>2</sup> Recent studies have identified the most dangerous elements in an attempt to guide future active debris removal efforts,<sup>2-4</sup> resulting in a list that includes 290 second stages of the Soviet/Russian “Cosmos-3M” LV, 7 of which have been considered among the 50 most concerning space debris objects,<sup>4</sup> 110 third stages of the Soviet/Russian “Cyclone-3” LV, 54 units of the American first and second stages of the “Delta” LV, as well as 38 third stages of the Chinese “CZ-4” and “CZ-2D” LVs. Further concerns arise due the presence of propellant residuals in the tanks, which account for up to ~3% of the initial fuel mass.<sup>5</sup> During the long stay of a stage in orbit, the remaining fuel evaporates under the influence of solar radiation, which leads to an increased risk of explosion in orbit and, therefore, to the generation of space debris.<sup>1</sup> The uncontrolled descent of first LV stages in sensitive drop areas can also lead to environmental

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pollution caused by the depressurization of toxic fuels, fires in the drop sites, and the contamination of water bodies. This problem is more relevant for Russian LVs like “Soyuz”, “Proton”, and “Angara”, where most of the drop zones are located on land.<sup>6,7</sup>

Modern launch vehicle operations are subjected to strict space debris mitigation policies.<sup>1</sup> When graveyard orbits are not attainable, the orbital lifetime is limited and systems are passivated by removing all energy sources. Active deorbiting represents a highly attractive alternative to those strategies, but it is not exempt from risks and technical challenges.<sup>8</sup> Among them, proper engine restarting conditions must be provided once the stage is separated from the rest of the vehicle in order to ensure a safe reorbiting or reentry. This decoupling induces strong disturbances on the propellant residuals and leads to highly non-linear sloshing dynamics, compromising the operation of the engine feed system.<sup>9</sup>

Ullage engines have been traditionally employed to settle the propellant residuals during insertion, orbital coast, or on-orbit operations in an approach known as active settling.<sup>10</sup> These independent rockets induce accelerations that can be as weak as  $10^{-4}$  to  $10^{-3}$  m/s<sup>2</sup> and involve solid, mono-propellant, bi-propellant, or cold gas technologies, sometimes fed by vaporized propellant vented from the main tanks.<sup>11</sup> Some examples include the Saturn IV-B’s hypergolic liquid bi-propellant Auxiliary Propulsion System (APS),<sup>12,13</sup> SpaceX’s Falcon 9 nitrogen cold gas thrusters for coast attitude control,<sup>14</sup> or the two *Sistema Obespecheniya Zapuska* (SOZ) ullage engines of the Blok DM-2 Proton upper stage. This last unit has raised concerns in the space debris community after being responsible for up to 50 on-orbit explosions between 1984 and 2019.<sup>15</sup>

The technical specifications of ullage engines are not usually accessible to the general public, which hampers any effort to perform an “external” evaluation of these systems. However, numerous reports from the Apollo era can still be consulted. The two Saturn IV-B APSs were usually fired in three consecutive ullaging burns for a total of  $\sim 245$  s, consuming  $\sim 13.5$  kg of propellant ( $\sim 23.5\%$  of the total propellant mass of each APS).<sup>16</sup> The dry mass of the APS is unknown to the authors but seems of the order of several hundred kg judging by the volume of the system. The dry mass of Saturn IV-B was about 13.5 t. On the other hand, each one of the two SOZ units of the Block DM-2 upper stage had a dry mass of  $\sim 106$  kg and a total propellant mass of up to 114 kg, while the stage itself weighted 2.1 t. Although determined by the characteristics of the vehicle and its mission profile, the total mass of the ullage rocket system should be around  $\sim 200$  kg.<sup>17</sup> With a Falcon 9’s launch cost to LEO of  $\sim 2700$  \$/kg,<sup>18</sup> an economic penalty per launch and stage of  $\sim 500.000$  USD may be estimated. This value is doubled for GEO orbits, and multiplied by an even larger factor in a Mars mission.

Ullage engines are a robust and well-established solution to deal with the restart of rocket engines in microgravity conditions. However, that should not preclude the study of alternative approaches to the problem with lower mass budgets and/or enhanced reliability. In this paper, the feasibility and performance of a Magnetic Positive Positioning (MP<sup>2</sup>),<sup>19</sup> an on-board Propellant Gasification System (PGS),<sup>20</sup> and an hybrid device are explored for the first time. The historical background of each system is presented together with a preliminary technical analysis. The ultimate goal is to initiate an open discussion on these technologies and inform the design of new-generation propellant settling systems.

## LAUNCHER CHARACTERISTICS

### Overview

Although applicable to multiple low-gravity propellant settling systems, the discussion that follows focuses on the first and second launch stages of a Falcon-9-like LV. The basic parameters of the vehicle are reported in Table 1, with the geometrical definitions being depicted in Fig. 1. Some of these values are found in SpaceX’s Falcon User’s Guide,<sup>14</sup> while others can only be estimated from unofficial sources\*.

### Propellant behavior during stage separation

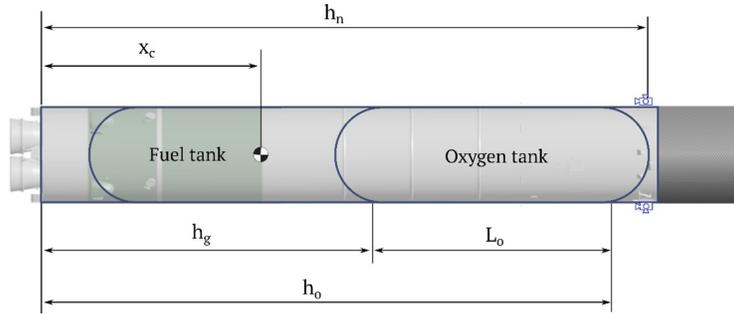
The acceleration profile experienced by each stage during separation is key to understand the dynamic behavior of the propellant. A simple mechanical model is introduced in the Appendix A and employed to

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\*See [www.spacelaunchreport.com/falcon9ft.html](http://www.spacelaunchreport.com/falcon9ft.html). Consulted on 13/01/2022.

**Table 1:** Geometrical and inertial parameters of the launch vehicle.

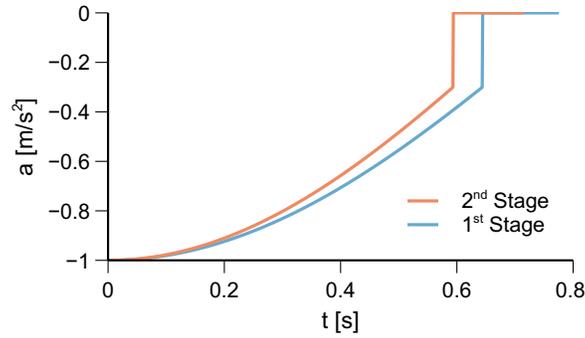
Parameter	First Stage	Second Stage
Propellant type	LOX + RP1	LOX + RP1
Empty mass [t]	22	4.5
Propellant mass [t]	411	111.5
Oxygen tank capacity [t]	287.4	78
Kerosene tank capacity [t]	123.5	33.5
Total mass [t]	433	116
Propellant mass after stages separation [t]	13	3.5
Residual propellant mass after landing [t]	1	0.3
Thrust (stage total) [kN]	7686	981
Throttle capability [kN]	4381 to 7686	626 to 981
Number of engines	9	1
Diameter [m]	3.66	3.66
LOX tank height $L_0$ [m]	22.5	8.7
Length of 1 stage $h_0$ [m]	29	11.2
Length of fuel tank $h_g$ [m]	19.2	7.4
First stage mass center $x_c$ [m]	14.9	5.7
Moment of inertia [ $\text{kg}\cdot\text{m}^2$ ]	$2.68 \cdot 10^6$	$3 \cdot 10^4$
Length to PGS nozzles $h_n$ [m]	39	15

**Figure 1:** Geometrical parameters of a launch vehicle stage.

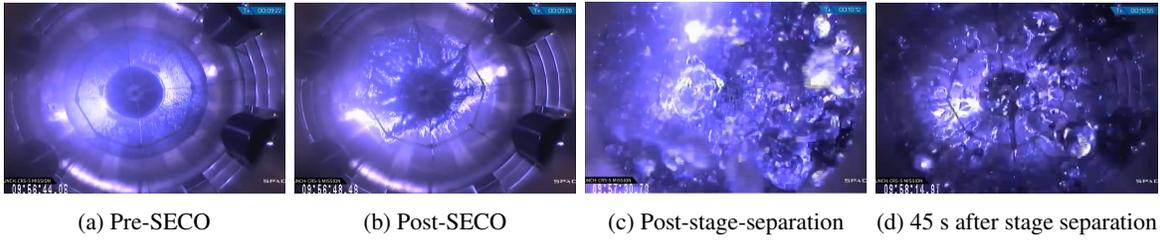
obtain the acceleration curves reported in Fig. 2 using representative values. Peak accelerations of  $\sim 1 \text{ m/s}^2$  are applied to the system and sustained for less than 1 s. Additional effects that may impact the propellant behavior include the release of strain energy from the walls of the tank, the flow movement induced by engine suction, or thermal convection.<sup>11</sup>

Modeling this problem is far from trivial, and experimental data is not easily available because of its consideration as Export-Controlled material. However, a partial recording of Falcon 9's first and second stage liquid oxygen (LOX) tanks during the CRS 5 mission is publicly available<sup>†</sup>, allowing for a qualitative analysis of the problem. Figure 3 shows the sequence of video frames for (a) the instant before second-stage engine cut off (SECO), (b) the lateral sloshing wave caused by the structural relaxation after SECO, (c) the cloud of LOX bubbles generated after separation, and (d) the state of the cloud 45 s after separation. It can be readily concluded that (i) the SECO induces a mild lateral sloshing wave, but does not significantly disturb the liquid, (ii) the stage separation atomizes the residual LOX in a myriad of droplets that reach the top of the tank in less than 40 s (i.e. the droplets move at least at  $\sim 20 \text{ cm/s}$ ), and (iii) the droplets keep moving for at least 6 minutes while coalescing with each other. This behavior is also (briefly) observed in the first stage, where the droplets seem to move at about 0.5 m/s. This value has been employed in the derivation of the

<sup>†</sup>The interested reader is referred to <https://youtu.be/p7x-SumbynI>. Consulted on 13/01/2022.



**Figure 2:** Estimated acceleration profile of the 1<sup>st</sup> and 2<sup>nd</sup> stages after separation.



**Figure 3:** Top view of the second-stage LOX tank of SpaceX's Falcon 9 during the CRS-5 mission. Source: [www.youtube.com/watch?v=mVAGoWJuDKk](http://www.youtube.com/watch?v=mVAGoWJuDKk)

acceleration profiles shown in Fig. 2.

### Engine restart conditions

The propellant must be settled over the fuel outlet to ensure a safe engine restart. Although this usually implies bringing *all* the liquid back to the bottom of the tank before ignition, the requirement may be instead reformulated by noting that the engines will also accelerate the stage. The goal is thus to have enough propellant at the outlet so that, when the thrust-induced settling brings all the liquid to the bottom of the tank, no gas bubbles have made their way into the engines.

The first factor to consider is mass flow rate: higher thrust will settle the propellant droplets faster, but will also require a larger initial mass. The relation between thrust  $T$  and mass flow rate  $\dot{m}_p$  is given by

$$T = I_{sp}g_0\dot{m}_p, \quad (1)$$

where  $I_{sp}$  is the specific impulse and  $g_0 = 9.81 \text{ m/s}^2$  is the standard gravity acceleration. For a LOX-RP1 chemical engine with a mass ratio of  $\sim 2.3$  the specific impulse should be around 285 s at sea level.<sup>21</sup> In fact, the old SpaceX website attributed to Falcon 9's Merlin engines the values of 282 s at sea level and 311 s in vacuum<sup>‡</sup>. The second factor is the propellant settling time, that can be divided into two phases. On the first, the propellant residuals return to the intake device, while on the second, gas bubbles are removed from the liquid. The total settling time can thus be expressed as

$$t_s = t_I + t_{II}. \quad (2)$$

For a tank of length  $L$ , the duration of the first phase is bounded by the kinematic result

$$t_I = \sqrt{\frac{2Lm_0}{T}}, \quad (3)$$

<sup>‡</sup>See [web.archive.org/web/20130501002858/http://www.spacex.com/falcon9.php](http://web.archive.org/web/20130501002858/http://www.spacex.com/falcon9.php). Consulted on: 13/01/2022.

with  $m_0$  being the dry mass of the stage. The duration of the second phase, assuming a constant terminal velocity of the bubbles in the liquid, is<sup>22</sup>

$$t_{II} = l \left[ \frac{8}{3} \frac{r_b T}{C_d m_t} \left( 1 - \frac{\rho_g}{\rho_l} \right) \right]^{-1/2} \quad (4)$$

where  $r_b$  is the bubble radius,  $C_d$  is the drag coefficient,  $\rho_g$  is the gas density,  $\rho_l$  is the liquid density,  $m_t$  is the total mass of the stage after settling, and  $l$  is the height of the longest liquid column. Consequently, the initial mass of propellant required to complete the maneuver for a giving thrust level is

$$m_p = \dot{m}_p t_s = \frac{T}{I_{sp} g_0} \left\{ \sqrt{\frac{2Lm_0}{T}} + l \left[ \frac{8}{3} \frac{r_b T}{C_d m_t} \left( 1 - \frac{\rho_g}{\rho_l} \right) \right]^{-1/2} \right\}, \quad (5)$$

which increases with  $\sqrt{T}$ , showing that small thrust values are convenient to minimize the mass of propellant required to restart the engines. Table 2 reports the stage acceleration, settling time, and initial oxidizer and fuel masses for different thrust configurations of Falcon 9's first and second stages. The values  $C_d = 0.47$  (sphere),  $r_b = 5$  mm,  $\rho_g = 0.1785$  kg/m<sup>3</sup> (He),  $\rho_l = 1141$  kg/m<sup>3</sup> (LOX), and  $l = m_r / (\pi R^2 \rho_l)$ , with  $R$  being the tank radius and  $m_r$  the residual LOX mass, are employed in combination with those presented in Table 1 using the larger LOX tank as a reference. The masses reported in Table 2 are the minimum absolute values required near the fuel outlet to initiate the restart maneuver. Unofficial sources claim that Falcon 9's first stage restart acceleration is less than 50 m/s<sup>2</sup> by employing reverse engineered telemetry data<sup>8</sup>, which indicates that either the central engines (maximum thrust) or three outer engines (minimum thrust) are actually employed in this process. However, the authors were not able to verify this information.

	First Stage				Second Stage			
	$g$ [m/s <sup>2</sup> ]	$t_s$ [s]	LOX [kg]	RP1 [kg]	$g$ [m/s <sup>2</sup> ]	$t_s$ [s]	LOX [kg]	RP1 [kg]
<b>Maximum Thrust</b>	350	0.66	1287	401	218	0.39	88	38
<b>Minimum Thrust</b>	200	0.88	971	303	140	0.49	70	30
<b>Single Engine</b>	22	2.63	324	101	140	0.49	70	30

**Table 2:** Stage acceleration, settling time, and minimum initial oxidizer and fuel masses for different restart configurations of Falcon 9's first and second stages.

## MAGNETIC POSITIVE POSITIONING

### Concept and overview

The ability of controlling the position of susceptible liquids by means of magnetic fields in microgravity leads to several potential space applications. Those include, but are not limited to, mass transfer,<sup>23-25</sup> thermo-magnetic convection,<sup>26,27</sup> or micropropulsion.<sup>28,29</sup> The volume force density that enables these technologies is induced by inhomogeneous magnetic fields on susceptible liquids, and adopts the form

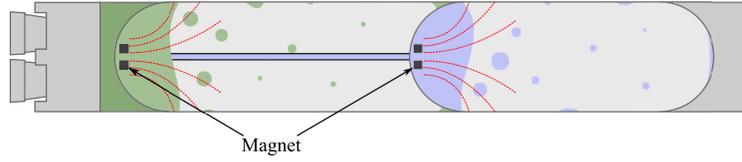
$$\mathbf{f}_m = \mu_0 M \nabla H, \quad (6)$$

with  $\mu_0$  being the permeability of free space, and  $M$  and  $H$  denoting the magnetization and magnetic fields, respectively. In addition, the *magnetic normal traction*

$$p_m = \mu_0 \frac{M_n^2}{2}, \quad (7)$$

should be considered at the liquid interface, where  $M_n$  is the normal magnetization component.<sup>30</sup> This pressure-like term is usually neglected for *natural* liquids, such as LOX, but becomes relevant for highly

<sup>8</sup>See <https://github.com/shahar603/SpaceXtract> for a remarkable example of reverse engineering. Consulted on 17/01/2022



**Figure 4:** Magnetic Positive Positioning

susceptible materials like high-density ferrofluids.<sup>31</sup> Since both  $H$  and its gradient decay with the distance to the source, the magnetic force vanishes relatively quickly. Therefore, powerful magnets or coils are needed for most applications.

The Magnetic Positive Positioning approach, sketched in Fig. 4 for the system under study, seeks to induce a magnetic acceleration that holds, collects, and/or traps the liquid near the fuel outlets. The concept was first proposed in 1963 by Steve Papell in the same patent where he invented ferrofluids.<sup>32</sup> The idea was abandoned until 2001 when, motivated by the advent of stronger permanent magnets and high-temperature superconductors, the NASA Magnetically Actuated Propellant Orientation (MAPO) experiment explored the positioning of ferrofluid solutions in a series of parabolic flights.<sup>33</sup> Such ferrofluids were selected to approximate the magnetization curve of LOX for different magnetic field intensities. It should be noted that LOX is the most susceptible natural paramagnetic liquid,<sup>34</sup> making it particularly appropriate for this application. Subsequent publications by Marchetta and coworkers presented refined numerical models and results of technical relevance for the development of liquid oxygen magnetic positioning devices.<sup>35–43</sup> Recent works have also explored the free surface oscillations of ferrofluids in microgravity, which may be relevant for slosh control and the development of novel propellant management devices (PMDs).<sup>44–49</sup> A comprehensive review of the field can be found in Ref. 19.

Significant advances have been made in the modeling and fundamental understanding of magnetic positive positioning devices during the last two decades. However, none of the aforementioned works explored the feasibility of this approach as part of the operation of LVs. Although limited by the lack of reliable technical information, this study aims at covering this knowledge gap by exploring the application of magnetic positive positioning to the restart of Falcon 9’s first and second stages.

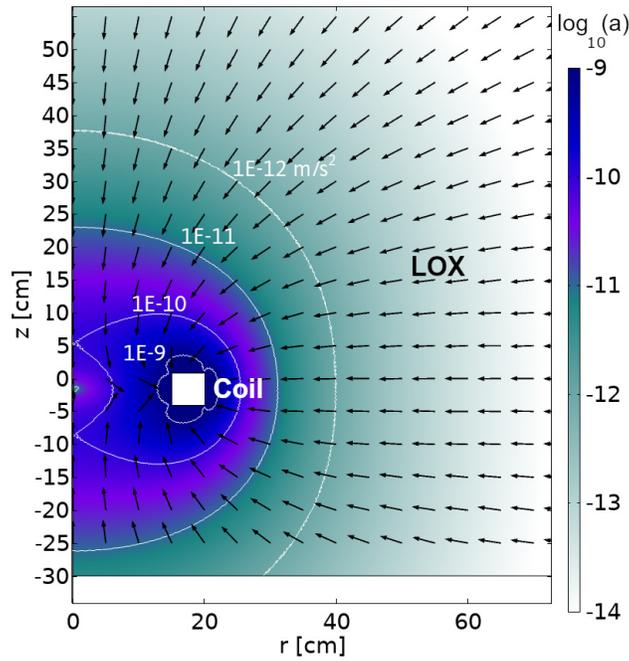
### Passive retention strategy

The first and most intuitive approach to magnetic positive positioning is the liquid retention strategy, where a magnet or coil is used to hold the paramagnetic liquid in the presence of adverse accelerations that tend to destabilize the free liquid surface. In the classical literature, the critical Bond number

$$Bo^* = \frac{\rho g^* R^2}{\sigma} \quad (8)$$

is employed to compute the critical acceleration load  $g^*$  for which surface tension cannot longer stabilize the meniscus. Myshkis and coworkers provide a best-case  $Bo^* = -3.32$  for cylindrical tanks at a contact angle of  $90^\circ$ ,<sup>50</sup> which results in  $g^* = 2.9 \cdot 10^{-6} \text{ m/s}^2$  for the LOX tank considered in this work. In other words, surface tension does little or nothing to prevent the atomization of the residual LOX volume observed in Fig. 3 under the action of the acceleration loads estimated in Fig. 2. An obvious question is whether the magnetic force can hold the liquid against adverse accelerations of  $\sim 1 \text{ m/s}^2$ .

Marchetta and coworkers explore the problem of magnetic LOX retention in a 12 cm diameter 24 cm height cylindrical tank under the influence of a point dipole of 1.4 cm diameter and an inertial acceleration along its major axis.<sup>42</sup> The dipole strength required to hold  $\sim 30 \text{ ml}$  of liquid is shown to be about  $10 \text{ Am}^2$  for  $g^* = 1 \text{ m/s}^2$ . Similarly, in Ref. 19 an analytical model is developed to study the stability of magnetic interfaces and applied to a 10 cm diameter cylindrical tank, showing that a 60 g magnet can increase the critical load by 31.5%. None of these low-gravity studies can be easily extended to Falcon 9’s 3.66 m diameter LOX tank, where the liquid outlet has a diameter of about 90 cm and the maximum estimated



**Figure 5:** Magnetic acceleration contours induced on LOX by a 35 cm diameter coil operating at 1 A.

acceleration is about  $1 \text{ m/s}^2$ . The problem, however, can be easily addressed by plotting the axisymmetric magnetic acceleration contours induced by a 1 A cylindrical coil on the LOX tank volume as done in Fig. 5. The figure depicts the magnetic acceleration levels in a logarithmic scale and its direction using black arrows. A coil mean diameter of 35 cm is chosen to ensure that the liquid gets attracted toward the PMD located at the the tank outlet. The minimum mass to be retained is 70 kg (second stage), which translates to a LOX sphere of 25 cm radius. The magnetic acceleration at this distance is about  $10^{-11} \text{ m/s}^2$ . Since the acceleration scales with the square of the current intensity,<sup>19</sup> values of  $\sim 10^6 \text{ A}\cdot\text{turn}$  (i.e. coils current intensity times number of coil turns) would be required to retain the oxidizer against accelerations of 1 to  $10 \text{ m/s}^2$ . Further computations are not required to conclude that the mass and/or power requirements of this approach are well beyond reason with existing technologies, particularly for the first stage.

### Recovery strategy

The passive magnetic retention strategy sets an upper limit for the magnetic field strength. Since this limit is hard to reach with existing technologies, alternative strategies must be explored. The first of them is here introduced and seeks to collect the LOX droplets after they are atomized rather than holding part of the liquid at the bottom of the tank<sup>¶</sup>. In order to evaluate this idea, the time required to settle a LOX droplet is first derived with a simplified framework of analysis.

*Magnetic settling time:* If the settling of the propellant is induced by the magnetic interaction and not by a uniform acceleration  $g$ , the derivation of  $t_s$  is complicated by the presence of an inhomogeneous acceleration field. A strict approach to the problem would require solving the Navier-Stokes equations with a magnetic force source term. Although less computationally expensive than the fully coupled fluid-magnetic simulations that are necessary for highly susceptible ferrofluids,<sup>19</sup> this approach is still prohibitive for a preliminary study. Instead, the movement of a perfectly spherical droplet along the symmetry axis of an axisymmetric coil or magnet is analyzed. The droplet is small in comparison with the variation of the magnetic field and exhibits linear magnetization with susceptibility  $\chi \ll 1$ . It is further assumed that external, internal, and magnetization fields are collinear, that residual liquid volumes do not contribute to the magnetic field,

<sup>¶</sup>Following the popular saying, “if you can’t beat them, join them”.

and that magnetic surface force terms are negligible. In this simplified framework, the total magnetic force induced on the liquid droplet by a circular coil with  $n$  turns, radius  $R$ , and current intensity  $I$  at a distance  $z$  along the symmetry axis  $e_z$  is<sup>||</sup>

$$\mathbf{F}_m \approx -\frac{3\mu_0\chi(nI)^2R^4}{4} \frac{z}{(R^2+z^2)^4} \mathbf{e}_z. \quad (9)$$

This expression can also be applied to axially magnetized cylindrical magnets with magnetization  $M_m$ , radius  $R$  and height  $l_m$  by considering an equivalent circular coil with the same radius and current intensity  $nI = M_m l_m$ . Its main advantage is that it allows the derivation of a quasi-analytical expressions for  $t_I$ . After considering Newton's second law and solving the resulting second-order differential equation with initial position  $z(0) = L$  and initial velocity  $\dot{z}(0) = 0$ , the duration of the first phase becomes

$$t_I(L) = \sqrt{\frac{4\pi\rho_l}{\mu_0\chi(nI)^2R^4}} \int_0^L \left[ \frac{1}{(z^2+R^2)^3} - \frac{1}{(L^2+R^2)^3} \right]^{-1/2} dz, \quad (10)$$

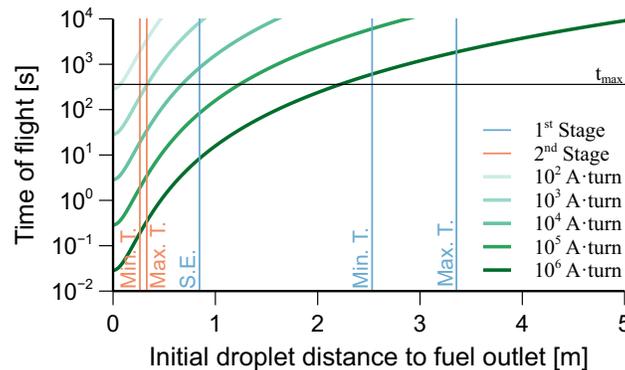
where it should be noted that  $t_I$  is inversely proportional to  $nI$  (or, if a magnet is employed, to  $M_m l_m$ ) and  $R^2$ . Of these, only the current intensity can be considered a design parameter, because  $R$  is bounded by the fuel intake radius (see previous subsection).

The time required to debubble the multiphase mixture near the fuel outlet using the paramagnetic force can be derived in a similar way, as done in Ref. 52 after adopting the Stokes' law for viscous drag (which is valid for  $Re \ll 1$ ) instead of the drag coefficient. However, the magnetic debubbling process is much faster than the first phase because the liquid is closer to the magnetic source, and hence it is further assumed that  $t_s \approx t_I$ .

Even though the assumptions employed in the derivation of Eq. 9 are not appropriate for highly susceptible ferrofluids, the volume magnetic susceptibility of the liquids employed in this work is bounded by that of LOX ( $\chi_{\text{LOX}} = 0.0034$  at 90 K and 1 atm<sup>34</sup>). It would not make sense to use high density ferrofluids in the fuel tank when the most demanding requirements are associated with the LOX tank.

*Performance analysis:* The time of flight of the droplets for a coil diameter of 35 cm is represented in Fig. 6 as a function of the initial droplet distance to the coil and the applied current intensity. A LOX settling time of  $t_{\text{max}} \approx 6$  minutes, estimated in the analysis of the second stage in Fig. 3, is superposed and treated as

<sup>||</sup>Equation 9 can also be found in the literature divided by the term  $(1+\chi)^2$  or  $(1+D\chi)^2$ , with  $D = 1/3$  being the demagnetization factor of a sphere.<sup>19,31,51</sup> Each choice denotes a different modeling of the internal magnetic field inside the droplet. Since in the problem here considered  $\chi \ll 1$ , the simplest approach, that assumes  $H_0 \approx H$ , is adopted.



**Figure 6:** Time required by a LOX droplet to reach the bottom of the tank as a function of its distance to the coil and the applied current intensity. The minimum tank settling length required by the single engine (S.E.), minimum thrust (Min. T.) and maximum thrust (Max. T.) configurations is superposed.

a deadline for the collection process. However, unofficial telemetry data shows that the first stage restarts 2 to 3 minutes after stage separation (see footnote at the end of page 5). As it will be seen, this does not change the qualitative results of the analysis. It is arbitrarily assumed that, after atomization, the LOX droplets are uniformly distributed in the tank volume. In this framework, the vertical lines represent the tank length that needs to be settled for each one of the configurations detailed in Table 2 before the LOX droplets stop moving (i.e. get attached to the walls of the tank).

A qualitative difference is first observed between first and second stages. The LOX mass required to restart the engines, listed in Table 2, drops by an order of magnitude in the second stage, and thus the length of the tank that needs to be settled is much smaller under the uniform droplet distribution assumption. The second factor that should be considered is the density of residual LOX per unit tank length. Although the lower LOX tank has a length of 22.5 m and the upper of just 8.7 m, the density of residual LOX per unit length is relatively similar, decreasing from 383.5 kg/m to 267 kg/m, respectively. Figure 6 shows that a coil with a configuration of  $10^3$  A·turn can satisfy the requirements of the second stage. However,  $10^4$  to  $10^5$  A·turn are needed to settle the first stage using the single engine restart configuration, with  $10^6$  to  $10^7$  A·turn being required for the rest. In other words, the liquid recovery strategy can potentially reduce the coil strength requirements by one order of magnitude in the first stage and three orders of magnitude in the second. These conclusions do not change if  $t_{\max}$  drops to 120-180 s for the first stage, as indicated by unofficial telemetry data.

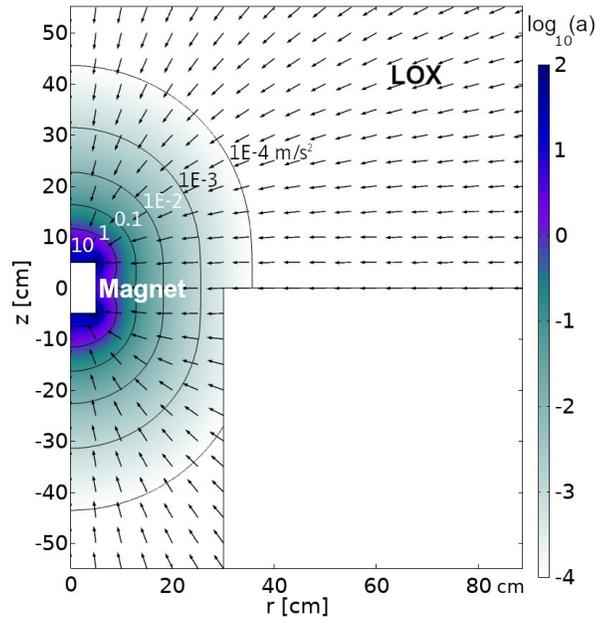
These results should be taken with care due to the number of assumptions employed in the derivation of Eq. 10. In particular, fluid-structure interactions have been completely neglected, but Fig. 3 shows that, after a few minutes, the liquid droplets tend to get stuck to the walls of the tank. This is a natural consequence of the presence of corner geometries in the interface between PMDs and the walls. The robustness of the liquid recovery strategy may thus be compromised by this effect, which should be evaluated with flight data that is not available to the authors. Possible mitigation strategies include the elimination of gaps and corner geometries or the application of a LOX-phobic treatment to the internal surfaces.

### Active retention strategy

The third and final approach seeks to soften the requirements imposed on the magnetic system by storing the restart liquid in a smaller auxiliary tank during the stage separation process. The propellant is later released near the fuel outlet, where it is held against disturbing accelerations by means of a permanent magnet or coil. The restart liquid remains thus unaffected by large stage separation accelerations. The main drawbacks of this method are the addition of ancillary components and the limitation of the number of restart events.

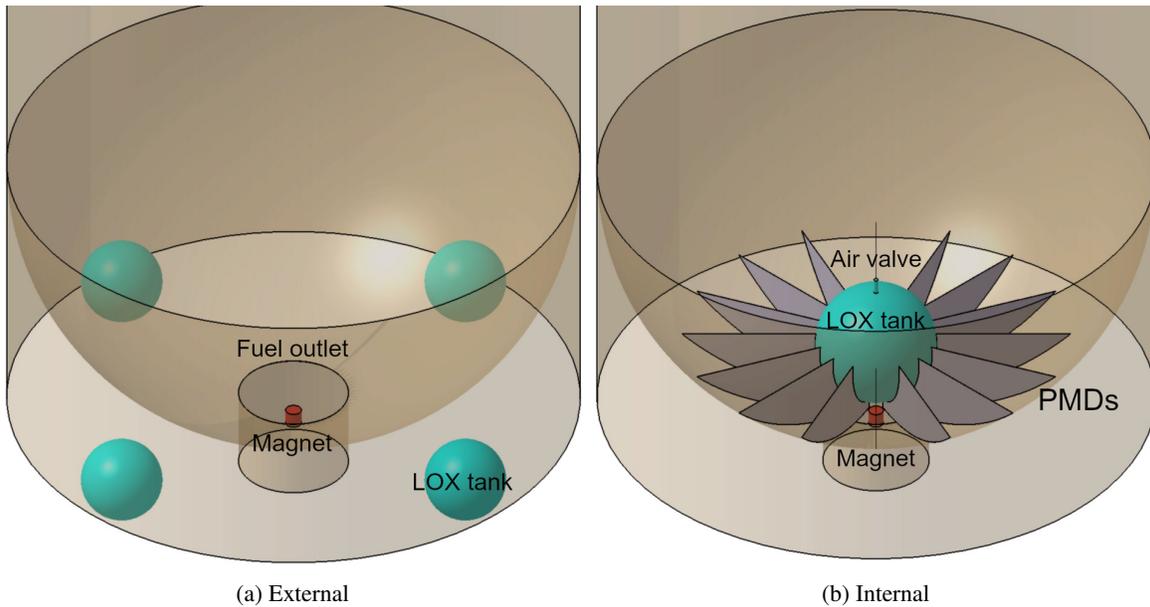
A representative magnetic configuration is displayed in Fig. 7, which depicts the magnetic acceleration field on LOX of a 5 cm radius 10 cm height cylindrical magnet magnetized at 1300 kA/m. The magnet geometry differs from that of Fig. 5 in the adoption of a smaller magnet radius, which increases the magnetic force close to the source.<sup>19</sup> The magnet volume is chosen to impose an acceleration of  $10^{-4}$  m/s<sup>2</sup> (one order of magnitude larger than microgravity disturbances<sup>11</sup>) at  $\sim 40$  cm from the magnet. This leads to the approximate LOX volume that needs to be retained in the single-engine first stage scenario reported in Table 2. Although the mass of the magnet is just  $\sim 5.5$  kg, ancillary components like the tanks or PMDs must also be taken into account.

Four 26 cm radius spherical containers are first considered to store the LOX volume in the first stage. The four tanks are located around the fuel outlet and outside the main tank space, as shown in Fig. 8a. Although this choice leaves the internal configuration virtually unaffected, an alternative internal positioning would minimize the pressure difference supported by the walls, and thus the mass requirements. The total mass of this system is hard to estimate without access to further technical details, but it would not be surprising to discover that ancillary components account for most of it. For instance, emptying the tanks in the microgravity environment would require a bladder<sup>12</sup> or outlet magnet<sup>45</sup> to ensure a gas-free expulsion and a small pressure difference applied, for instance, by means of the same helium tank employed to pressurize the LOX vessel. An alternative system that minimizes the number of ancillary components at the expenses of providing a single



**Figure 7:** Magnetic acceleration field induced by a 5 cm radius 10 cm height praseodymium magnet magnetized at 1300 kA/m.

engine restart opportunity is shown in Fig. 8b. This configuration employs one  $\sim 40$  cm radius internal LOX tank with a top gas valve and an open-ended bottom. After the stage separation, the gas valve is opened and the LOX flows under the influence of the magnet to the minimum-energy configuration given by the constant acceleration lines in Fig. 7. The total mass of this system would be close to 20 kg after considering a 1 mm thick Aluminum LOX tank. The major risk of this approach is the potential onset of Rayleigh-Taylor



**Figure 8:** Conceptual active liquid retention configurations for Falcon 9's first stage LOX tank.

instabilities (see Ref. 53) at the open bottom end if the LOX interface is accidentally exposed to the gas. A careful fluid dynamic analysis of the problem is thus needed to ensure a robust operation.<sup>19,40,41</sup>

The same system can be applied to the second stage and would require four 16 cm radius tanks or a single 25 cm radius vessel with a magnet of just 0.5 kg. The mass of the single tank aluminum walls would account for  $\sim 2.2$  kg. That is, the active liquid retention strategy may potentially achieve mass savings of one to two orders of magnitude with respect to existing ullage engines.

### Magnetic field generation

With the  $nI$  parameter already sized for the different engine restart configurations, the next logical step is to determine how to produce the required magnetic fields. Three technologies are subsequently studied: copper/aluminum coils, rare earth permanent magnets, and superconducting coils.

The magnetic field generated by a coil is linearly dependent on the  $nI$  parameter, that can be increased by adding more wire turns or employing higher currents. Total coil mass and power dissipation are the driving factors of the design. The mass of the coil can be estimated as

$$m = 2\pi R n S \rho_w, \quad (11)$$

where  $S$  is the cross-section of the wire and  $\rho_w$  its density. The heat dissipated by the coil can be derived from Ohm's law, resulting in

$$P = 2\pi R I n \rho_e \frac{I}{S}, \quad (12)$$

with  $\rho_e$  being the resistivity of the material. In a worst-case scenario this heat is stored in the coil instead of being dissipated, causing a temperature increase of

$$\Delta T = \frac{P t_{\max}}{m C_p} = \frac{\rho_e t_{\max}}{\rho_s C_p} \left( \frac{I}{S} \right)^2, \quad (13)$$

where  $C_p$  is the heat capacity of the wire. In order to constrain the design, the heat dissipated by the coil is limited by considering two worst-case scenarios: in the first, the heat is fully transferred to the residual LOX. The maximum power is arbitrarily set to the one that vaporizes 1 kg of LOX during the 6 minutes operation of the coils. The latent heat of vaporization of LOX is 6.82 kJ/mol (or 213.13 kJ/kg) at atmospheric pressure,<sup>34</sup> which results in a maximum coil power of 592 W. Because  $nI$  is fixed, the ratio  $I/S$  is determined by Eq. 12. In the second scenario, the heat is stored in the coil, causing a temperature increase that is limited by choice to 10 K. Therefore, from Eq. 13 the  $I/S$  parameter is extracted. The most restrictive constraint is chosen for each design so that the thermal runaway of the material and the vaporization of the residual LOX volume are avoided. Then, the mass is computed for the  $I/S$  value from Eq. 11. The second requirement concerns the voltage of the coil, set to 24 V to ease integration with Falcon 9's power subsystem. After inserting the  $I/S$  value in Eq. 12 and employing Ohm's law, the current intensity of the coil and its resistance are computed. At the boiling temperature of LOX (90 K) the resistivities of copper and aluminum are  $3.5 \cdot 10^{-9} \Omega \text{ m}$  and  $4.5 \cdot 10^{-9} \Omega \text{ m}$ , respectively.<sup>54</sup> Although copper is slightly more conductive than aluminum, its density and heat capacity are  $8960 \text{ kg/m}^3$  and  $0.385 \text{ kJ/kgK}$ , while aluminum has a density of  $2700 \text{ kg/m}^3$  and a heat capacity of  $0.89 \text{ kJ/kgK}$ . Therefore, aluminum is chosen to minimize the mass of the design.

The second approach focuses on employing rare earth permanent magnets to generate a constant, unpowered magnetic field. Neodymium ( $\text{Nd}_2\text{Fe}_{14}\text{B}$ ) is the most popular rare earth material, has a density of  $7008 \text{ kg/m}^3$ ,<sup>34</sup> and exhibits a remanent magnetization of  $M_m \approx 1200 \text{ kA/m}$ . It is classified as a "hard material", implying that it can be used to manufacture magnets of any shape.<sup>55</sup> As previously noted, the sizing parameter  $nI$  of a cylindrical coil can be translated to the length  $l_m$  of an equivalent cylindrical magnet with the same diameter by means of the expression  $In = M_m l_m$ . Magnet tessellation strategies such as Halbach arrays can be employed to boost the paramagnetic force on one side, while partially canceling it on the other.<sup>56</sup> Halbach arrays have already been proposed for space applications<sup>52</sup> and would be particularly well suited to the LOX settling problem for two reasons: the reach of the magnet is increased, and the interaction between the LOX magnet and the droplets generated in the fuel tank is reduced.

Neodymium magnets experience a slight increase of their magnetic flux as temperature decreases. At around 135 K, a transition point is reached and the magnet undergoes spin reorientation (i.e. a change in the preferred direction of the magnetization vector) that decreases the flux by no more than a 14%. This process is reverted as soon as the temperature increases.<sup>57</sup> If needed, the transition point could be avoided by isolating the magnet in the LOX tank and actively controlling its temperature. A more elegant solution is, however, to employ praseodymium magnets to avoid the spin reorientation. Praseodymium magnets do not suffer from spin reorientation and have been shown to reach a remanent magnetization of  $\sim 1300$  kA/m at 85 K,<sup>58,59</sup> which makes them ideal for LOX control applications.

The design points of the aluminum coil and praseodymium magnet are shown in Table 3 as a function of the  $nI$  parameter. In all cases but  $10^6$  At, the design of the coil is driven by the thermal requirement (maximum temperature increase of 10 K). Magnets are orders of magnitude lighter for all  $nI$  values, incurring in a – still reasonable– mass penalty of 52 kg at  $10^5$  At.  $nI$  values beyond  $10^5$  At seem unreachable without incurring in large mass penalties, and it is in this context where high-temperature superconductors (HTC) can become a game-changing alternative. An HTC wire exhibits zero resistance in a certain operational range, resulting in no heat loss and a potential reduction in mass and power requirements. This happens when (i) it is operated below its critical temperature  $T_c$  –greater, by definition, than the boiling point of nitrogen (77 K)–, (ii) it is subjected to a magnetic field below the critical field  $B_c$ , and (iii) the critical current  $I_c$  is not exceeded. The simultaneous satisfaction of these three requirements is far from trivial; in fact,  $I_c$  decreases continuously with increasing temperature and magnetic field.<sup>60</sup> For example, Bi2223 ( $\text{Bi}_2\text{Sr}_2\text{Ca}_2\text{Cu}_3\text{O}_{10+\delta}$ ) has a critical temperature of 110 K, but its critical current drops to zero when the material is exposed to a field of less than 1 T. RE-123 ((RE) $\text{Ba}_2\text{Cu}_3\text{O}_7$ , where RE stands for Rare Earth element) superconductors (also known as REBCO), on the contrary, can resist up to 10 T, but only well below a critical temperature of around 90 K.<sup>61</sup> It is nowadays feasible to generate very strong magnetic fields at the boiling point of Helium (4.22 K), the best example being the 32 T superconducting magnet<sup>62</sup> of the National High Magnetic Field Laboratory\*\*. Reaching similar values at higher temperatures seems, unfortunately, still beyond our technical capabilities. In the application here discussed the superconductor would be immersed in LOX, which would act as a cooling agent only if  $T_c \gg 90$  K, and the maximum magnetic field imposed near the coils would be  $\sim 10$  T at  $In = 10^6$  At. The results presented in Ref. 60 for different commercial REBCO conductors seem to indicate that such operation point cannot be reached with current technologies. However, the  $I_c$  value of 4 mm wide superconductors is shown to be 450–1000 A/mm at 12 T and 77 K, a value that jumps up to 60 kA/mm<sup>2</sup> at 18 T and 4.2 K. This indicates that cooling mechanisms need to be put in place to reach the  $10^6$  At configuration with HTCs, which may open an opportunity for multiple-use of the helium tanks employed for tank pressurization. This possibility, although attractive, would require a deeper technical analysis that is beyond the scope of this paper.

\*\*See <https://nationalmaglab.org/magnet-development/magnet-science-technology/magnet-projects/32-tesla-scm>. Consulted on: 26/12/2021

**Table 3:** Mass and power budget for different magnetic configurations.

Current · Turns [At]	Aluminum coil <sup>1</sup>					Praseodymium magnet <sup>2</sup>	
	I [A]	P [W]	n [# turns]	d [mm]	m [kg]	h [mm]	m [kg]
$10^2$	$2.51 \cdot 10^{-3}$	0.06	4	$1.62 \cdot 10^{-1}$	2.44	0.08	0.052
$10^3$	$2.51 \cdot 10^{-2}$	0.60	40	$5.12 \cdot 10^{-1}$	24.4	0.77	0.52
$10^4$	$2.51 \cdot 10^{-1}$	6.03	405	1.62	244	7.7	5.2
$10^5$	2.51	60.26	4054	5.12	2438	77	52
$10^6$	24.67	592	40541	16.2	24814	769	519

<sup>1</sup> Coil of 35 cm diameter operating at 24 V and 90 K.

<sup>2</sup> Cylindrical magnet of 35 cm diameter magnetized at 1300 kA/m at 90 K.

## Fuel tank

From the magnetic actuation perspective, LOX determines the design envelope of the system. On one hand, the LOX tank is more than two times larger than the fuel tank, and therefore a given magnetic source will reach a larger portion of the latter. On the other, LOX is a paramagnetic substance with volume magnetic susceptibility  $\chi_{\text{LOX}} = 0.0034$ , while kerosene is a diamagnetic with  $\chi_{\text{Ke}} \approx -8 \cdot 10^{-6}$ .<sup>63</sup> In order to apply the magnetic retention and settling approaches to the fuel tank, it must be transformed into a para/ferromagnetic by adding magnetic nanoparticles and creating a kerosene-based ferrofluid. With this approach, the susceptibility of the solution is bounded by the concentration of magnetic nanoparticles.

Kerosene has been employed as a carrier liquid since the invention of ferrofluids in 1963<sup>32</sup> and kerosene-based ferrofluids are synthesized and used in numerous fields.<sup>64–67</sup> Commercial solutions like Ferrotec's EMG-905<sup>††</sup> are now widely available at a relatively low cost. The initial susceptibility of a monodisperse, colloidal ferrofluid can be estimated as<sup>30</sup>

$$\chi_{\text{ini}} = 8\phi\lambda, \quad (14)$$

where  $\phi$  is the volume fraction of magnetic solids and  $\lambda$  is the coupling coefficient, given by

$$\lambda = \frac{\mu_0 M_d^2 V}{24kT}, \quad (15)$$

with  $M_d$  being the saturation moment of the bulk magnetic solid,  $V$  the nanoparticle volume,  $k$  the Boltzmann constant, and  $T$  the absolute temperature. Assuming an iron oxide nanoparticle radius of 5 nm, an absolute temperature of 293 K, and a saturation moment of 446 kA/m,<sup>68</sup> the approximate volume fraction required to match the magnetic susceptibility of LOX starting from the value of kerosene would be just  $\phi \approx 3.2 \cdot 10^{-4}$ . Not surprisingly, this value is within the range tested by Martin and Holt in the NASA MAPO experiment.<sup>33</sup>

If the whole kerosene tank volume is magnetized, the magnetic nanoparticles add  $\sim 40$  kg and  $\sim 11$  kg to the first and second stages, respectively, in addition to negligible variations in density and specific impulse.<sup>19</sup> The very low ferrofluid concentration should prevent damage to the engines. Although simple, this approach is expensive and inefficient in comparison with the active liquid retention strategy, where only the kerosene employed to restart the engine would need to be enhanced with ferromagnetic nanoparticles. In this case, the mass penalty associated with such nanoparticles would be of just 40 to 140 g based on the total RP-1 masses presented in Table 2. The same approach could be applied to the passive liquid retention and liquid recovery strategies if a concentrated ferrofluid volume is mixed with the RP-1 residuals shortly before MECO/SECO.

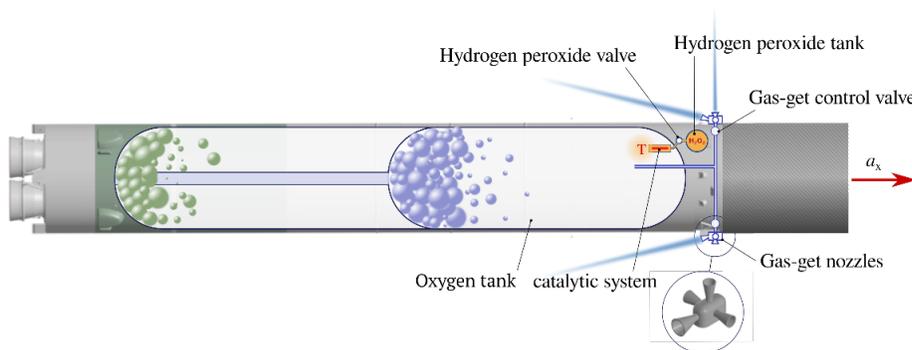
## Operation

The discussion provided in this section focuses on moving the propellant toward the bottom of the tank. A logical follow-up question is what to do once that happens. Unless some sort of valve or screen mesh is put in place, it must be assumed that the propellant stored in the tubes leading to the engines is partially released to the tank. Thus, once the liquid is attracted back to the tank outlet, a gas bubble is generated in the tubes leading to the engines. This would represent an unacceptable safety risk.

There are several ways to face this problem, and all of them require a deep knowledge of Falcon 9's tank outlet design. A possibility would be to employ a combination of PMDs to hold the liquid at the entrance of the engine against accelerations of  $1 \text{ m/s}^2$ . This should effectively prevent the development of a Rayleigh Taylor instability and the entrance of gas bubbles. Active strategies may involve purge valves and membranes located in strategic positions along the liquid path. Unfortunately, a more detailed analysis requires technical information that is not available to the authors.

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<sup>††</sup>See <https://ferrofluid.ferrotec.com/products/ferrofluid-emg/oil/emg-905/>. Consulted on: 28/12/2021



**Figure 9:** Propellant Gasification System

## PROPELLANT GASIFICATION SYSTEM

### Concept and overview

The injection of hot gases into the tanks for chemical pressurization through propellant evaporation and combustion has long been known and used in the “Proton”, “Rokot” and “Dnepr” launch vehicles.<sup>69</sup> These systems do not require heat exchangers, which are used to heat cold helium gas from 90 K to 300 K in current pressurization systems, and are instead based on feeding a nitric acid-based oxidizer and fuel into the fuel and oxidizer tanks, respectively. The components ignite and heat is released, causing the fuel to vaporize, which increases the pressure in the tanks. Referring to Sutton, while this type of pressurization system is small and lightweight, it has generally not yielded reproducible tank pressures due to the difficulties to stabilize the combustion reaction. For instance, fuel sloshing caused by vehicle maneuvers results in sudden cooling of the hot pressurizing gas and leads to erratic tank pressure changes. This problem can be avoided by physically separating the hot gas from the liquid propellant. If the hot gas is generated from a solid propellant reaction or from mono-propellant decomposition instead of a high-pressure gas supply, a significant reduction in the mass of the pressurizing system can be achieved.<sup>21</sup>

The Propellant Gasification System concept was first proposed in the early 2010s as a method to vaporize the propellant residual of 2<sup>nd</sup> launch vehicle stages and provide attitude and orbit control capabilities by means of dedicated vapor-fed thrusters.<sup>70–72</sup> The original idea was to inject the combustion products of two-component propellants (AA and NDMH) in the tank to move the stage from its initial circular orbit to an elliptical orbit. Such orbit would ensure a successful deorbiting in the time frame of 25 years. In 2015, the use of solid fuel instead of a two-component propellant was investigated to simplify the design and improve the energy performance of the PGS. Further analyses on the Soyuz 2.1v launch vehicle showed that the PGS could also lead to launch vehicle characteristic speed enhancements of up to 5%.<sup>73</sup> The PGS baseline design has currently evolved to reduce its mass and environmental impact using a green mono-propellant (hydrogen peroxide) that adds the possibilities of i) controlling the movement of the stage to reach a given drop area, ii) providing conditions for LRE restart by executing the flip around and propellant settling maneuvers, and iii) passivating the propellant after a normal or emergency cutoff of the LRE.<sup>74</sup> Highly concentrated hydrogen peroxide (85%) has already been employed as a green mono-propellant in substitution of hydrazine on the “Soyuz” launch vehicle for the operation of turbo-pump units.<sup>75</sup>

The PGS considered in this work vaporizes the propellant residuals in the oxidizer tank of the launch vehicle using the catalytic decomposition of hydrogen peroxide, which is placed in an auxiliary tank and used as a heat source. The mono-propellant is passed through a catalyst chamber that leads to the formation of up to 823K hot oxygen and steam. The vapor-gas mixture is then transferred into the oxidizer tank, which leads to vaporization of the liquid phase and a pressure increase. The gas, consisting of vaporized propellant and pressurizing agent (helium), is used to feed a set of gas thrusters that are employed for attitude control and tank settling, as sketched in Fig. 9. This approach can be regarded as the active equivalent of the hydrogen venting strategy employed in the Apollo era.<sup>11</sup>

It is important to remark that the risk of combustion or explosion is virtually non-existent in the proposed PGS approach. Since only hot hydrogen peroxide decomposition byproducts (water vapor and oxygen) are injected into the oxidizer tank and a reducing agent at auto-ignition temperatures is not present, the combustion reaction cannot be produced. Such a hazard could only occur if the PGS was used in the propellant tank. However, this approach has already been applied to the aforementioned chemical pressurization systems, and the reaction was safely controlled by tuning the pressurizer gas flow rate.

The following main subsystems compose the PGS: (i) a hot gas generator that includes a bladder-controlled hydrogen peroxide tank and a catalyst chamber where the exothermic decomposition of hydrogen peroxide happens, producing a high-temperature vapor-gas mixture (VGM) with a 34%  $H_2O$  and 66%  $O_2$  composition, (ii) a system of nozzles installed after the catalyst chamber that injects the VGM into the LOX tank minimizing tank wall heating, and (iii) a system of gas nozzles used to discharge the VGM from the tank and produce the required thrust. The PGS provides control over the tank discharge valves, hydrogen peroxide feeding, and gas nozzles. Cold helium gas, which is stored in balloons at the bottom of the oxidizer tank, can also be used to reduce the temperature of the VGM (see next section).

### Concept of Operations:

Figure 10 shows the ConOps for the PGS approach. Three seconds after MECO, the first stage separates from the second stage using a pneumatic pusher (or equivalent). The control system of the 1<sup>st</sup> stage starts the PGS to increase the pressure in the oxygen tank. Nine seconds after MECO, the PGS opens the attitude control nozzles with a total thrust of 1 kN to start the flip around maneuver, which involves an acceleration phase, a constant angular velocity phase, and a deceleration phase. About 55 seconds after MECO, a set of nozzles provide 5 kN of axial acceleration before engine re-start.

The PGS can be operated in three distinct modes in combination with the helium pressurization system:

1. Standalone PGS operation: In this case, the main task of the PGS is to vaporize the liquid residuals. The highest gas temperature is employed (823 K).
2. Alternate operation: The residual cold helium pressurizer gas (90 K) is fed into the tank in order to reduce the VGM temperature. This reduces or stops the evaporation of liquid propellant and ensures that the proper quantity of liquid is vaporized to operate the gas thrusters.
3. Combined operation: In this case, the VGM does not vaporize the propellant residuals, acting instead as a heat exchanger that warms up the pressurizer gas up to an accepted operational temperature (300 K), replacing the engine heat exchanger.<sup>76</sup>

### Flip around maneuver:

The flip around maneuver to the required  $\Delta\phi$  angle consists of three phases that are evaluated in Fig. 11. The acceleration phase is produced with constant angular acceleration  $\epsilon$  until the speed  $\omega$  is reached, and lasts

$$t_\epsilon = \frac{\omega}{\epsilon}, \quad (16)$$

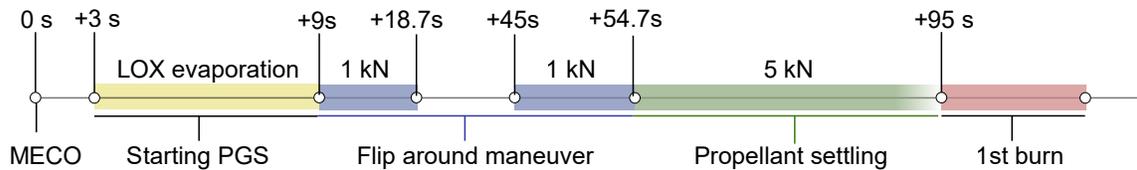
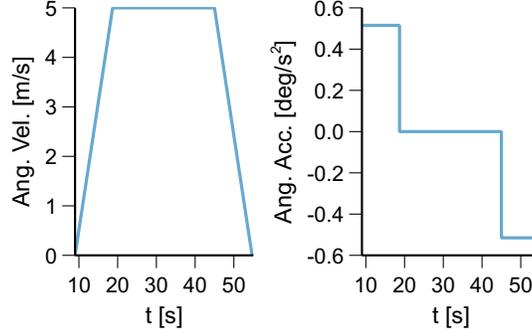


Figure 10: PGS Operation cycle



**Figure 11:** Angular velocity and angular acceleration profiles of the 1<sup>st</sup> stage during flip around maneuver using the attitude control nozzles with a total thrust of 1 kN

where zero initial speed and angle are assumed. The second phase is produced with constant angular velocity  $\omega$  and duration  $t_\omega$ . Finally, the deceleration phase happens with the same angular deceleration. The total turnaround time  $t_r$  is given by

$$t_r = \frac{\omega}{\epsilon} + \frac{\Delta\psi}{\omega}. \quad (17)$$

The magnitude of angular acceleration  $\epsilon$  is determined by the torque produced by the attitude control nozzles relative to the center of mass and by the lateral moment of inertia of the stage. For instance, around the z-axis the acceleration becomes

$$\epsilon = \frac{F(h_n - x_c)}{J_z}, \quad (18)$$

with  $F$  being the total thrust of the nozzles,  $h_n - x_c$  the thrust arm, and  $x_c$  the distance from the bottom section of the stage to the center of mass. The parameters of the flip around maneuver are given in Table 4.

### Oxygen vaporization model

Vaporization in a tank can occur according to three mechanisms: evaporation without boiling, film boiling, and nucleate boiling. The criterion that determines whether vaporization belongs to one of three types is the temperature of the liquid. The mass rate of vaporization during bubble and film boiling is calculated following a thermodynamic approach that assumes that all the heat supplied is employed to vaporize the liquid.

The mathematical model of the oxygen vaporization process is based on the first law of thermodynamics, and leads to<sup>74</sup>

$$\frac{dp}{dt} = \frac{k-1}{V} \left( H_u \dot{m}_{hc} + i_{hc} \dot{m}_{hc} + i_{ev} \dot{m}_{ev} - i_{out} \dot{m}_{out} - \frac{k}{k-1} p \frac{dV}{dt} \right), \quad (19a)$$

$$\frac{d\rho}{dt} = \frac{1}{V} \left( \dot{m}_{hc} + \dot{m}_{ev} - \dot{m}_{out} - \rho \frac{dV}{dt} \right), \quad (19b)$$

**Table 4:** Parameters of the flip around maneuver

Parameter	Value
Angular velocity, $\omega$ [deg/s]	5
Total attitude nozzle thrust force, $F$ [N]	1000
Acceleration and deceleration phases, $t_\epsilon$ [s]	9.7+9.7
Constant angular velocity phase, $t_\omega$ [s]	26.3
Total flip around time, $t_r$ [s]	54.7

$$\frac{dT_w}{dt} = \frac{q_{\text{rad}}^{\text{mix-w}} + q_{\text{con}}^{\text{mix-w}} + q_{\text{rad}}^{\text{w-lox}} + q_{\text{con}}^{\text{w-lox}} - q_{\text{rad}}^{\text{w}} + q_{\text{ext}}}{c_w m_w} \quad (19c)$$

$$\frac{dT_{\text{mix}}}{dt} = \frac{-q_{\text{con}}^{\text{mix-w}} - q_{\text{rad}}^{\text{mix-w}} + q_{\text{rad}}^{\text{w-lox}} + q_{\text{con}}^{\text{mix-lox}} - q_{\text{ev}} + H_u \dot{m}_{\text{hc}}}{c_{\text{mix}} m_{\text{mix}}}, \quad (19d)$$

$$\frac{dT_{\text{lox}}}{dt} = \frac{q_{\text{rad}}^{\text{mix-lox}} - q_{\text{con}}^{\text{mix-lox}} + q_{\text{rad}}^{\text{w-lox}} + q_{\text{con}}^{\text{w-lox}} - q_{\text{ev}}}{c_{\text{lox}} m_{\text{lox}}}. \quad (19e)$$

The system given by Eqs. 19a-e describes the change of the pressure in the tank  $p$ , density of the vapor-gas mixture  $\rho$ , and the temperatures of the vapor-gas mixture  $T_{\text{mix}}$ , LOX  $T_{\text{lox}}$ , and tank walls  $T_w$ . The temperature of the vapor-gas mixture depends on the radiative and convective heat flux to the wall of the tank ( $q_{\text{con}}^{\text{mix-w}}$ ,  $q_{\text{rad}}^{\text{mix-w}}$ ), radiative and convective heat flux to LOX ( $q_{\text{rad}}^{\text{w-lox}}$ ,  $q_{\text{con}}^{\text{mix-lox}}$ ), heat of vaporization of LOX  $q_{\text{ev}}$ , and the heat generated during decomposition reaction of  $H_2O_2$  with mass of  $m_{\text{hc}}$ ,  $H_u \dot{m}_{\text{hc}}$ . The temperature of the walls of the tank depends on the external heat flux from the atmospheric heating  $q_{\text{ext}}$ , heat flux from the vapor-gas mixture in the tank  $q_{\text{rad}}^{\text{mix-w}}$  and  $q_{\text{con}}^{\text{mix-w}}$ , heat flux to LOX ( $q_{\text{rad}}^{\text{w-lox}}$ ,  $q_{\text{con}}^{\text{w-lox}}$ ), and radiative heat flux  $q_{\text{rad}}^{\text{w}}$ . The temperature of LOX is determined by the heat flux from the wall, vapor-gas mixture in the tank and heat of vaporization of LOX.

The key heat fluxes that determine the vaporization process of LOX are now considered. The radiant heat flux from the vapor-gas mixture to the LOX surface is

$$q_{\text{rad}}^{\text{mix-lox}} = \sigma \epsilon_{\text{mix}} F_{\text{lox}} (T_{\text{mix}}^4 - T_{\text{lox}}^4), \quad (20)$$

where the area of the surface of vaporization of LOX,  $F_{\text{lox}}$ , depends on the mechanical condition in the tank. If LOX is accumulated at the bottom of the tank, then  $F_{\text{lox}}$  equals the cross section area of the tank. If the liquid oxygen is in the drop state, then  $F_{\text{lox}}$  is the total surface of all the drops. The convective heat flux from the vapor-gas mixture to the oxygen surface is

$$q_{\text{con}}^{\text{mix-lox}} = \alpha_{\text{mix}} F_{\text{lox}} (T_{\text{mix}} - T_{\text{lox}}), \quad (21)$$

with

$$\alpha_{\text{mix}} = \frac{\lambda_{\text{mix}} N u_{\text{mix}}}{l} \quad (22)$$

being the heat-exchange coefficient of the vapor-gas mixture to the surface of evaporation of the oxygen in the tank, and where  $\lambda_{\text{mix}}$  is the thermal conductivity of the vapor-gas mixture in the tank, and  $l$  is the characteristic dimension that is equal to the diameter of the oxygen tank.  $\alpha_{\text{mix}}$  depends also on the Nusselt number  $N u_{\text{mix}}$ , which is determined by the vaporization type of LOX: evaporation from the free surface, nucleate boiling or film boiling.  $q_{\text{con}}$  and  $q_{\text{rad}}$ , among other contributors to the heat exchange process, are defined similarly to Eqs. 21 and 22.

If free surface vaporization is the dominant vaporization mechanism, the Nusselt number can be expressed as<sup>77</sup>

$$N u_{\text{ev}} = C_1 R e_{\text{ev}}^{0.8} P r_{\text{ev}}^{0.43} \quad (23)$$

where  $R e_{\text{ev}}$  is the Reynolds number,  $P r_{\text{ev}}$  is the Prandtl number for the vapor-gas mixture, and  $C_1$  is equal to 0.037.

In the case of nucleate boiling, the Nusselt number becomes<sup>77</sup>

$$N u_{\text{nuc}} = \frac{\alpha d_0}{\lambda} = 75 (P e \cdot F o)^{0.7} P r^{-0.2}. \quad (24)$$

For the nucleate boiling case, the characteristic dimension is the average diameter of the bubble departure diameter  $d_0$ , that can be considered independent on the heat flux  $q$ . Consequently, the Peclet number for heat exchange during nucleate boiling is

$$P e = \frac{q d_0}{r_* \rho_v a}, \quad (25)$$

with an average diameter of the bubble departure diameter defined by Fritz's equation<sup>78</sup>

$$d_0 = 1.2\theta \sqrt{\frac{\sigma}{g(\rho_l - \rho_v)}}, \quad (26)$$

where  $\theta$  is the contact angle in rad,  $\sigma$  is the surface tension coefficient,  $g$  is the inertial acceleration,  $\rho_l$  and  $\rho_v$  are the densities of the liquid and gas oxygen,  $q$  is the heat flux density,  $r_*$  is the specific heat of evaporation, and  $a$  is the temperature conductivity coefficient. The Fourier number for the nucleate boiling of oxygen is

$$Fo = \frac{a}{d_0^2 f}, \quad (27)$$

with

$$f \sqrt{d_0} = 0.56 \left( \frac{\rho_l - \rho_v}{\rho_v} g \right)^{1/2} \quad (28)$$

being the frequency of formation of the vapor bubbles.

Finally, in a film boiling scenario with a large liquid volume on vertical surfaces as well as on horizontal cylinders and spheres, it is recommended to use the similarity equation

$$Nu_f = 0.18 Ra^{1/3}, \quad (29)$$

where

$$Ra = \frac{g l_0^3}{\nu_v a_v} \frac{\rho_l - \rho_v}{\rho_v} \quad (30)$$

is the Rayleigh number that characterizes the behavior of a fluid with a temperature gradient, and where

$$l_0 = \frac{\sigma}{g(\rho_l - \rho_v)} \quad (31)$$

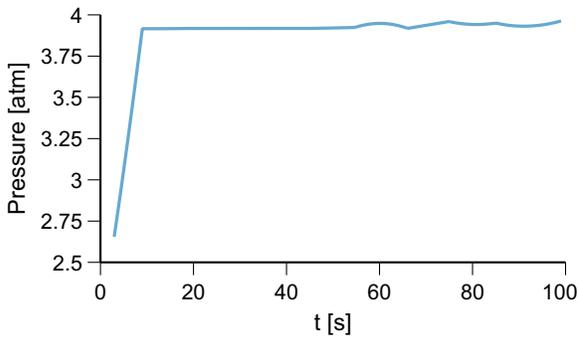
is the capillarity constant, selected as a characteristic length.

### Performance analysis

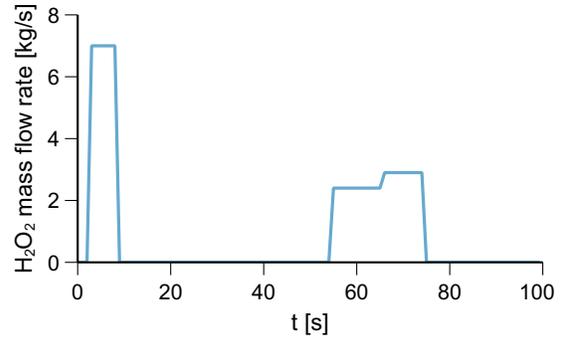
The alternate PGS mode, that reduces the average tank temperature in comparison with a standalone PGS operation, is employed in the analysis. A combined operation with the tank pressurization system reduces oxygen evaporation and preserves it for the operation of the LRE. The ratio of hydrogen peroxide and helium mass flow is not optimized in the results presented in this work, where one of the possible combinations was chosen. However, even this sub-optimal set point reflects the efficiency of the alternate operation of the PGS and helium pressurization systems.

Figure 12 shows the time evolution of different variables of the PGS for the nozzle thrust values reported in Table 4. After stage separation, the supply of hydrogen peroxide to the LOX tank leads to an operating pressure of  $\sim 4$  atm in the oxygen tank. Figure 12a shows the time evolution of this variable. During the first six seconds after separation of the stage and supply of heat to the tank, the pressure increases from 2.7 to 4 atm. As noted above, the given pressure level in the tank is maintained by the joint operation of the PGS and the supply of helium to the LOX tank, that are set manually and may be optimized in future works. From 9 to 95 s, the stage performs the flip-around and propellant settling maneuvers as described in Fig. 10. Figures 12b and 12c show the mass flow rates of hydrogen peroxide and helium during this process. Figures 12f and 12e show thrust plots and mass flow rates of the PGS nozzles. During the entire operation, the gas temperature in the tank increases up to 275 K, which does not violate the standard thermal stability limits of 300-350 K.

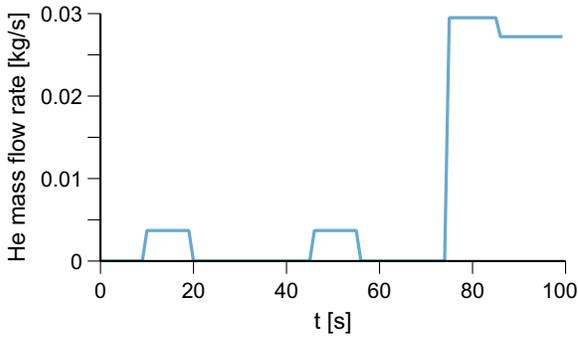
Table 5 shows the results of the flip around and propellant settling maneuvers separately. A total of 327.9 kg of gas are employed by the nozzles during the operation of the PGS, with 354 kg of oxygen, 95 kg of hydrogen peroxide, and 0.7 kg of helium being consumed. After the PGS operation, the gas residuals (composed of vaporized oxygen, water vapor, and helium) are needed to maintain the tank pressure of 4 atm



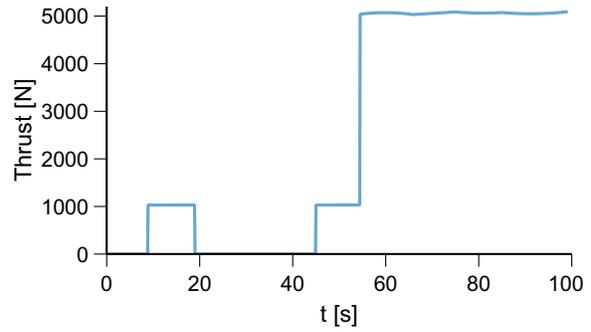
(a) Pressure in the oxygen tank



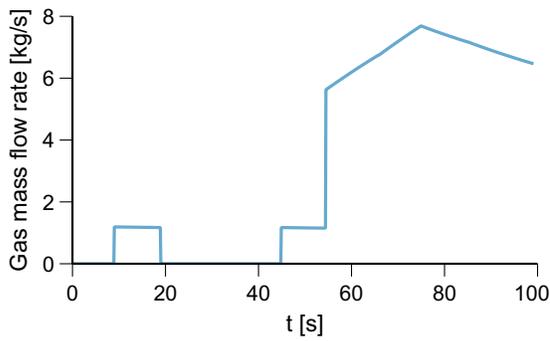
(b) Hydrogen peroxide mass flow rate



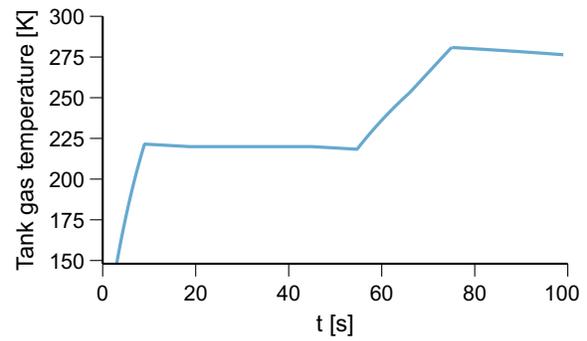
(c) Helium mass flow rate



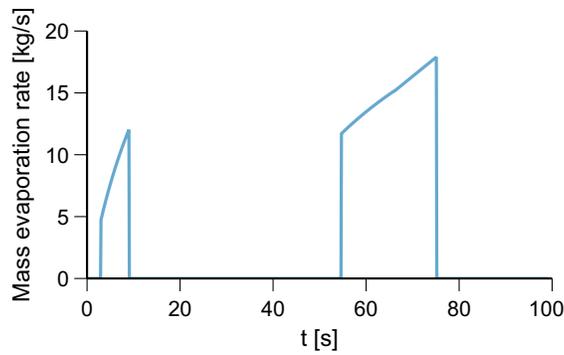
(d) Nozzle thrust in turning and fuel shrinking modes



(e) Gas mass flow rate through gas jet nozzles



(f) Gas temperature in oxidizer tank



(g) Mass oxygen evaporation rate

**Figure 12:** Performance of the PGS.

**Table 5:** Mass from different components required for the flip around and settling maneuvers.

System	Flip around maneuver	Propellant settling
Gas through nozzles [kg]	22.9	305
Vaporized oxygen [kg]	53.9	304
PGS [kg]	40	-
Catalyst [kg]	0.5	-
$H_2O_2$ balloon [kg]	4.6	5.8
$H_2O_2$ [kg]	42	53
Helium [kg]	0.069	0.668

as a working body for the gas thrusters, that are in charge of stage orientation and stabilization during the flight. The PGS mass budget of 40 kg includes the gas generator (5 kg), nozzles (10 kg), valves, pipes and other hardware components. The total mass of the nozzles depends on the selected configuration: 8 nozzles are arbitrarily selected in this work, with 2 of them having a higher thrust. The mass values are strongly dependent on the technical decisions of the designers, who will have their own constraints and engineering vision of the PGS, including the length of the pipes, number of valves, etc. Active catalysts (e.g. potassium permanganate grains, silver mesh, copper, or other metals<sup>79</sup>) are used for hydrogen peroxide decomposition. The catalyst does not lose its properties when exposed to the low temperature of the cryogenic tank or the high temperatures of the combustion reaction. The temperature of the hydrogen peroxide decomposition products can be controlled by tuning the hydrogen peroxide concentration,<sup>80</sup> which allows the mass evaporation rate to be modified when the PGS is used in combination with the pressurizer gas helium.

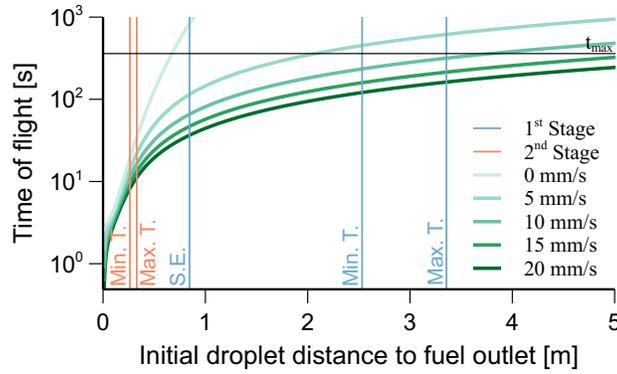
In order to minimize propellant residuals in the gas phase at the moment of PGS operation termination, it is necessary to optimize the helium and hydrogen peroxide input cyclogram (e.g. using the Pontryagin maximum principle or dynamic programming<sup>81</sup>). An optimum flow rate combination reduces the residual gas both in the balloons and in the tank after vaporization, and also the final PGS mass.

Computations for the second stage would follow the same process as for the first stage and will be addressed in future works.

## HYBRID MAGNETIC GASIFICATION

A combination of the MP<sup>2</sup> and PGS technologies can potentially enhance the robustness and performance of the overall propellant settling system. This hybrid approach involves a permanent magnet located at the fuel outlet and a smaller PGS aimed at carrying out the flip-around phase and a shorter propellant settling maneuver. The initial acceleration induced on the liquid residuals during the settling phase starts a slow movement toward the bottom of the tank, where the magnetic force is stronger and thus able to efficiently collect the liquid droplets.

Figure 13 depicts the time required by an LOX droplet to reach the bottom of the tank when subject to the magnetic field generated by a  $10^4$  At magnet as a function of its initial velocity and distance to the tank outlet. Based on Table 3, such magnet would have a mass of  $\sim 5.2$  kg. The curves are computed from an extended version of Eq. 10, and prove that an initial inertial kick can significantly extend the reach of the MP<sup>2</sup> system. Initial droplet velocities of just 5 to 10 mm/s allow the magnet to collect the necessary residual propellant mass under all engine restart configurations listed in Table 2 for both stages assuming that the droplets are uniformly distributed in the tank volume. The 5 kN PGS nozzles, that induce accelerations of  $\sim 0.23$  m/s<sup>2</sup> in the first stage, would theoretically need to operate for less than 0.05 s to induce these droplet velocities, reducing the propellant settling window in about 40 s. The operation window of the nozzles would need to be extended to account for transient effects and fluid-structure interactions, but this would only increase the performance of the system. Based on Table 5, the associated PGS mass savings would be close to 60 kg, resulting in a total hybrid system mass of  $\sim 92$  kg.



**Figure 13:** Time required by a LOX droplet to reach the bottom of the tank as a function of its initial position and velocity when subject to a  $10^4$  At coil located at the tank outlet. The minimum tank settling length required by the single engine (S.E.), minimum thrust (Min. T.) and maximum thrust (Max. T.) configurations is superposed.

## SUMMARY AND CONCLUSIONS

Five novel high-risk-high-return propellant settling approaches have been explored in this work: passive magnetic retention, magnetic recovery, active magnetic retention, propellant gasification, and hybrid magnetic gasification. The general advantages and disadvantages of each of them have been discussed, and preliminary mass budgets have been derived. Table 6 summarizes and extends the main results from the analysis.

While the passive magnetic retention strategy exceeds any reasonable mass budget, the active magnetic

**Table 6:** Comparison of different propellant settling approaches and mass budgets for first and second stages.

	Advantages	Disadvantages	Mass, 1 <sup>st</sup> [kg]	Mass, 2 <sup>nd</sup> [kg]
<b>Passive Magnetic Retention</b>	<ul style="list-style-type: none"> <li>- Simple</li> <li>- Robust</li> <li>- Thoroughly studied</li> </ul>	<ul style="list-style-type: none"> <li>- Beyond current technological capabilities</li> <li>- Massive unless stage operation is adapted</li> <li>- Limited control</li> </ul>	$\rightarrow \infty$	>1000
<b>Magnetic Recovery</b>	<ul style="list-style-type: none"> <li>- Lightweight</li> <li>- Simple</li> </ul>	<ul style="list-style-type: none"> <li>- Sensitive to fluid-structure interactions</li> <li>- Slow</li> <li>- Requires tank outlet redesign</li> <li>- Very low TRL</li> </ul>	104	1-10
<b>Active Magnetic Retention</b>	<ul style="list-style-type: none"> <li>- Lightweight</li> <li>- Simple</li> </ul>	<ul style="list-style-type: none"> <li>- Limited reusability</li> <li>- Requires careful tank design</li> <li>- Requires tank outlet redesign</li> <li>- Very low TRL</li> </ul>	40	6
<b>Propellant Gasification System</b>	<ul style="list-style-type: none"> <li>- Robust</li> <li>- Provides settling and attitude control</li> <li>- Fast to operate</li> <li>- More traditional design</li> </ul>	<ul style="list-style-type: none"> <li>- Complex</li> <li>- Sensitive to liquid movement</li> <li>- Very low TRL</li> </ul>	147	-
<b>Hybrid Magnetic Gasification</b>	<ul style="list-style-type: none"> <li>- Lightweight</li> <li>- More robust than MP<sup>2</sup> systems</li> <li>- Provides settling and attitude control</li> <li>- Fast to operate</li> <li>- Boosts magnet performance</li> </ul>	<ul style="list-style-type: none"> <li>- Complex</li> <li>- Sensitive to liquid movement</li> <li>- Requires tank outlet redesign</li> <li>- Very low TRL</li> </ul>	92	-

retention approach can potentially reduce the mass of existing propellant settling systems by one to two orders of magnitude, leading to more than half a million dollar savings per launch and stage. This comes at the cost of higher complexity and potential robustness and reliability issues. Although less efficient, the magnetic recovery system is also very competitive with respect to current technologies, particularly for upper stages. Because this approach depends on the availability of uniformly distributed free-floating propellant droplets, fluid-structure interactions can easily undermine its performance. The PGS, which represents a relatively more conventional approximation to the problem, can also lead to moderate mass savings that are significantly increased when operated in combination with a magnetic retention system. As with any other low-TRL technology, numerous technical challenges remain that can only be addressed with a more detailed numerical and experimental analysis.

Ullage engines have been employed since the beginning of the space era and are nowadays regarded as a robust active settling solution. However, publicly available data indicates that they also involve significant mass and economic penalties that may be reduced with novel approaches. Such approaches must demonstrate the same level of reliability and robustness in order to become competitive. Although the magnetic positive positioning and propellant gasification systems introduced in this work are still in a very early stage of development, the analysis here presented offers reasons to persevere in their development.

## APPENDIX A: STAGE SEPARATION MODEL

The accelerations experienced by each stage during separation are modeled after assuming that the process is carried out by spring pushers with total initial force  $F_b$ , final force  $F_e$ , and stroke  $h$ . Thus, the total force acting on the stage is

$$F(x_{12}) = F_b - cx_{12} \text{ if } x_{12} \leq h, 0 \text{ otherwise,} \quad (32)$$

with  $c = (F_b - F_e)/h$  being the total stiffness of the springs, and where the geometrical parameters of the problem are defined in Fig. 14. After applying Newton's second law and solving the resulting differential equation, the duration of the maneuver becomes

$$t_h = \sqrt{\frac{m_{12}}{c}} \arccos \frac{F_e}{F_b}, \quad (33)$$

where  $m_{12} = m_1 m_2 / (m_1 + m_2)$ . The acceleration of the stage with respect to the orbital frame is

$$a_1 = -\frac{F_b}{m_1} \cos \sqrt{\frac{c}{m_{12}}} t \text{ if } t \leq t_h, 0 \text{ otherwise.} \quad (34)$$

The values  $F_e = 0.3F_b$ , and  $h = 0.2$  m are adopted in this work, with  $F_b = 12.1$  kN and  $F_e = 4.5$  kN in the first and second stages, respectively. Empty and total masses are considered for the deployed and remaining stages with a payload mass of 15 t. The resulting propellant dispersion velocities range between 0.45 and 0.5 m/s, a range that seems to agree with observations from SpaceX's CRS-5 mission.

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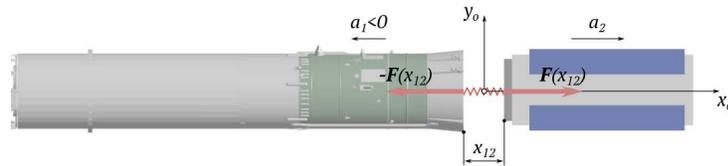


Figure 14: Stage separation model

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