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Review

Bridging the Technology Gap: Strategies for Hybrid Rocket Engines †

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Abstract: Hybrid rocket propulsion, first demonstrated by the Russian GIRD-09 rocket in 1933, combines liquid oxidizer and solid fuel for thrust generation. Despite numerous advantages, such as enhanced safety, controllability, and potential environmental benefits, hybrid propulsion has yet to achieve its full potential in space applications. In recent years, the research on hybrid propulsion has gained enormous momentum in both academia and industry. Recent accomplishments such as the altitude record for student rockets (64 km), the launch of the first electric pump-fed hybrid rocket, and a successful 25 s hovering test highlight the potential of hybrid rockets. However, although the hybrid community is growing constantly, industrial utilizations and in-space validations do not yet exist. In this work, we reassess the possibilities of hybrid rocket engines by presenting potential fields of applications from the literature. Most importantly, we identify the technical challenges that hinder the breakthrough of hybrid propulsion in the space sector and evaluate the technologies and approaches necessary to bridge the gaps in hybrid rocket development.

Keywords: gap analysis; case study; hybrid rocket engines; perspectives

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1. Introduction and Historical Context

Research into Hybrid Rocket Engines (HREs) emerged roughly at the same time as the early works on liquid and solid propulsion, mainly driven by the search for a propulsion system with less propulsive hazard than solids. In fact, Hermann Oberth stated that "powder believes it must explode all at once; from the old use in shells and guns, it is too well-trained always to destroy" [1]. While pioneers such as Robert Goddard started shifting from solid to liquid propulsion, the first launch that is considered to be propelled by hybrid propulsion was the GIRD-9 (Group for the Study of Reactive Motion) rocket of Korolev and Tikhonravov on 17. August 1933 [1]. Although hybrid rocket research never halted, until today hybrid propulsion has never been able to leave the shadow of solid and liquid propulsion, despite having several advantages (see Table 1). Indeed, the extensive work on solid and liquid propulsion has advanced these technology to the extent that the initial steep increase in specific impulse (I_{sp}) for solids and liquids has recently stagnated [2,3], whereas hybrids could have the potential for breakthrough advancements [4].

This raises the question of why, if they have distinct advantages over solids and liquids alike, no HRE has to date surpassed the sounding rocket stage, let alone reached orbit. Moreover, in the current NewSpace era , most notably starting after 2015, a considerable number of entities from at least fifteen countries are researching or planning hybrid rocket-powered space transportation vehicles [2]. Thus, it is relevant to ask what the hurdles and challenges are that have hindered hybrid propulsion reaching its full potential.

Table 1. Advantages and disadvantages of HREs (from [5], with permission).

Advantages	Disadvantages
Simplicity	Low regression rate
Fuel inertness	Low volumetric loading
Sustainability [1]	Low combustion efficiency
Robustness of grain	Slower transient
Throttleable	O/F shift during operation
Re-ignition possible	Regression rate estimation [2]
Propellant versatility	Residual fuel mass

[1] Depending on the propellant choice. [2] Similar to solid rocket motors.

In the present work, we address this question. First, a selection of case studies is presented to show the significant potential of HREs compared to liquid or solid propulsion depending on the use case. In a next step, we investigate the stages at which hybrid rocket engines have failed to transit from the theoretical state to the real-world application. Therefore, we collect relevant test data from the literature and distribute them according to the main challenges that have arisen during different campaigns and projects. Further, we present solutions that have been found for the various technical hurdles that have arisen. Indeed, we believe that with reignited research efforts on hybrids (notably, the number of hybrid propulsion papers and citations increased by a factor of seven from 2010 to 2021 [2]) a breakthrough for hybrid propulsion appears to be imminent.

2. Perspectives and Applications of Hybrid Rocket Engines

Davydenko et al. [6] stated that lower cost for series production is to be expected due to the simplicity of hybrids. They presented three main reasons for this statement [6]:

1. Hybrids are likely to have lower development periods (6–10 months) as compared to the 4–5 years required for Solid Rocket Motors (SRMs) and Liquid Rocket Engines (LREs).
2. Costs are reduced by a factor of 1.5 to 2 compared to LREs because advances made for SRMs can translate to HREs.
3. Reduced launching cost due to lower material cost, two to three times shorter prelaunch procedure, 40–50% decreased operating expenses, and lower cost of fire and explosion safety system.

Generally speaking, applications for hybrids (or any other propulsive system) can be separated into [2,7,8]: (a) upper stages; (b) suborbital flights and space tourism; (c) small(er) launch vehicles; (d) tactical/defence applications; (e) boosters; (f) planetary and lunar landers; and (g) in-space propulsion.

While the hybrid community is sometimes divided on the choice of oxidizer and fuel, the general agreement is that most probably the best initial applications of HREs are suborbital (space tourism) cases, kick stages (see Figure 1), or upper stages [2,4,7–10], the latter because of their lower thrust-to-weight ratio requirements. Moreover, these systems typically use pressure-fed systems, reducing the complexity of the first demonstration of an HRE system.

Mazzetti et al. [7] estimate a market potential of 400–70,000 t per year for upper stages and suborbital flights together. However, by far the highest volume is to be expected for tactical/defence applications, which (due to the lack of public information) can be estimated to yield up to 50,000,000 tons per year. They show that the 'merit parameter' (impulse per cost, kNs/€) for paraffin-based fuels is a factor of four higher than RP-1 (Rocket Propellant 1) and as much as eight times more than solid fuel formulations [7].

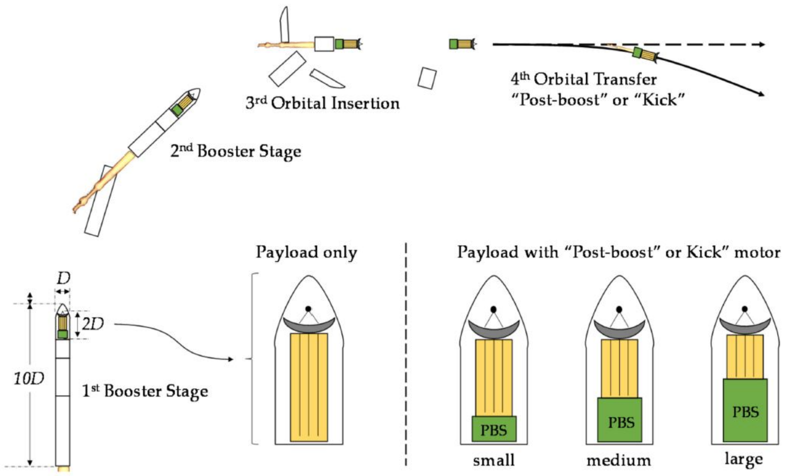


Figure 1. Definition of kick stage. Reprinted from [10] under Creative Commons 4.0 License.

Nonetheless, these predictions are rather generic; for this reason, in the following, we present case studies on the different fields of applications for hybrids and the predicted performance of the proposed hybrid propulsion system. For a better overview, the main findings are listed in Table A1 in the appendix.

2.1. Hybrid Upper Stages

A popular study to illustrate the potential of hybrid propulsion is the replacement of solid propulsion upper stages with a hybrid upper stage to show increased payload capacities with additional throttleability, reduced explosive hazard, and only a slight increase in cost.

Karabeyoglu et al. [4] investigated substituting the Orion 38 solid upper stage of the Taurus/Pegasus launch vehicle with a LOX (liquid oxygen)/paraffin HRE. They concluded that the payload increase could yield 40% and make the system 15–18% lighter when using advanced tank materials that are placed in parallel around the combustion chamber.

Casalino et al. [11] applied a Multidisciplinary Design Optimization (MDO) approach with regression rate uncertainty analyses to study the possibilities of substituting the Zefiro 9 (solid) and liquid AVUM stage (N_2O_4/UDMH (unsymmetrical dimethylhydrazine)) of Vega with a single HRE. The initial payload capacity of Vega is 1430 kg for a 700 km polar orbit. With a hydrogen peroxide and polyethylene HRE, the payload can be increased to 1971 kg (+38%). Using a LOX/wax HRE, the new payload would yield 2311 kg (+62%). When electric pump-feeding is considered, the payload even increases by another 250 kg.

The same Zefiro 9/AVUM replacement study was conducted by Božić et al. [12]. Using an H_2O_2/HTPB (hydrogen peroxide/hydroxyl-terminated polybutadiene) and paraffin blend with metal additives, the initial Vega design can be competed or even surpassed in terms of geometry, final mass, and performance. While the cost is estimated to increase by a factor of 1.7, the resulting HRE design would be non-toxic and considerably safer. Moreover, a comparable NTO/MMH (nitrogen tetroxide/mono-methyl hydrazine) liquid upper stage would be toxic and over five times more expensive than the initial Vega design.

Later, Božić et al. [13] conducted a similar study for the Brazilian Microsatellite Launch Vehicle (VLM-1) [14], which is a three-staged solid launcher in its reference design. To replace the upper stage, a pressure-fed H_2O_2 and HTPB/PE blend with metal additives are used as propellants. The HRE substitute is superior to the solid motor reference design in terms of thrust, mass, and energy efficiency. However, the cost is higher by a factor of 1.5 compared to the solid reference [13]. Nonetheless, an HRE upper stage would come with the benefits of better throttleability and reduced explosive hazard.

During the EU-funded ORPHEE (Operational Research Project on Hybrid Engine in Europe) project between 2009–2011 [8,9,15], hybrid propulsion applications were assessed using the same use case of the AVUM and Zefiro 9 replacement of Vega. Substituting the reference motors with a single LOX/HTPB HRE would gain over 800 kg of payload, an increase of roughly 60%. Kamps et al. [10] showed that a one cubic meter H_2O_2 (85%) hybrid rocket engine can achieve a change in velocity (Δv) of its 916 kg mass by over 4000 m/s.

To conclude, hybrid upper stages are an ideal candidate for replacing solid rocket upper stages in terms of payload capacity, throttleability, toxicity, and more.

2.2. Large Boosters and Main Stages

Hybrid rocket boosters for first stage applications have been investigated as well, partly driven by the Challenger Space Shuttle catastrophe, as a means to develop a throttleable system able to abort in case of failure [1]. In 1985, the company AMROC (American Rocket Company) started development on the Aquila launch vehicle. The Aquila launcher was supposed to be a four-staged hybrid launch vehicle with the capability to service 1450 kg into a 185 km circular polar orbit. The modular first (2–3 boosters) and second stage were designed to be powered by pressure-fed LOX/HTPB engines. The third stage was planned to be a solid motor, while the last and fourth stage was designed to be an N_2O/HTPB hybrid engine again. Through the throttleable and restartable fourth stage, even delivering multiple payloads to multiple orbits would have been theoretically possible. Thanks to the throttleable booster stages, excessive g-loading on the payload during launch would have been avoided [16,17]. However, AMROC went bankrupt in 1995; its intellectual property was acquired by SpaceDev, who ultimately developed the SpaceshipOne (hybrid suborbital vehicle) that won the X-prize for being the first privately funded company to carry three people above 100 km [18].

During the ORPHEE project, the replacement of the semi-cryogenic (LOX/RP1) first stage of Falcon-1 by a hybrid rocket engine was investigated. Theoretically, the HRE replacement can double the payload capacity [9].

Markopoulos et al. [19] have shown that adding two hybrid boosters to the ATLAS 2AR increases the payload capacity from 3900 kg (8600 lbs) in GTO to 4763 kg (10,500 lbs). Lengellé et al. [20] stated that the replacement of the solid booster for Ariane 5 with a hybrid LOX/HTPB system would reduce its weight by around 20 tons, leading to higher payload capacity [20].

2.3. Planetary and Lunar Lander or Ascent Vehicles

The ORPHEE project [8,9,15] evaluated the application of hybrids to lunar and Mars landers. For the Moon lander concept, LOX/HTPB was selected because short mission durations were assumed, meaning that the non-storability of LOX would not be a limiting factor. For the proposed hybrid lunar lander inserted using the Ariane 5ME, it would be possible to land 2000 kg dry mass (280 kg engine inert mass) on the Moon by having an initial weight of 3378 kg (1378 kg propellants) [9]. For the Mars lander concept, H_2O_2 was selected as the oxidizer for increased storability. The dry landing mass of the Mars lander would be 350 kg in a scenario similar to the Mars Pheonix lander [21].

Schmierer [22,23], who later co-founded the hybrid propulsion launcher start-up HyImpulse, investigated the possibility of a lunar sample return mission powered by an HRE lander (LOX/paraffin). The mission design was based on an Ariane 5 launching 8800 kg into a lunar transfer orbit. Schmierer et al. [22,23] calculated that the theoretically possible sample return weight could reach up to 75 kg (return rocket weight 1416 kg), comparable to the 87 kg (return rocket total weight 1566 kg) achieved by more complex and expensive LRE (LOX/methane) missions [22].

Hybrid uncertainty-based design optimization has been used to show that a hybrid propulsion system would have been well capable of substituting for the liquid ascent propulsion module of the Apollo missions [24]. Cho et al. [25] optimized the landing trajectory of a lunar lander with an initial mass of 300 kg. Starting from a lunar parking orbit (100 km circular orbit), the optimized lander would be able to land 160.6 kg on the lunar surface while consuming 139.4 kg of propellant.

For lunar applications, in situ utilization of regoliths has been researched as well. For example, Stoll et al. [26] investigated the use of oxygen as oxidizer, aluminium (as pure metal fuel) from regoliths, and alloys for tanks and fairings. The combustion of pure metal fuels is assumed to be possible, though with performance losses. Stoll et al. [26] showed that it is theoretically possible to insert 500 kg into a 300 km lunar orbit using fuel, oxidizer tanks, and fairings from lunar regoliths (using 349.1 kg material from earth and 1410 kg in situ material). This could be a use case for a future lunar gateway. Aluminium (20%) and magnesium (70%) (both metals that can be obtained from lunar soil) with sodium silicate (10%) as binder has been investigated by Yelken et al. [27] as hybrid rocket fuel.

Ascent rocket applications and in situ utilization have been assessed for Mars as well. The Mars Ascent Vehicle (MAV) has been under investigation at NASA following the success of the Mars Pathfinder mission in 1997 [28]. Many concepts have been investigated, including solid, liquid, gelled propellants, mono-prop, hybrids, and more. Of all options, the hybrid option is capable of single stage to orbit while clearly showing the lowest estimated mass and best packaging [28]. Chandler et al. [29,30] have shown in a system study that a hybrid concept Mars return rocket using Nytrox and paraffin decreases the system mass by up to 30% over the solid rocket reference for delivering a 36 kg payload into a 500 km orbit. This is due to the higher performance of the hybrids and because no thermal 'igloo' has to be used (which weighs 50 kg for the solid concept) [30].

Casalino et al. [11,31] proved the theoretical feasibility of a hybrid Mars sample return rocket and a manned return mission to a 170 km orbit. The payload can yield around 13 tons for the manned mission and around 86 kg in the case of the sample return. These payload values are similar to those achieved by comparable liquid engine designs while being less expensive and simpler [11,31].

For in situ utilization of propellants from Mars, CO_2/N_2O and paraffin hybrid engines (with a high percentage of aluminium additives for storability) have been investigated [32]. The CO_2 can be obtained from the Martian atmosphere.

2.4. In-Space Propulsion

Evidently, in-space applications, such as GTO/GEO and longer orbital transfer manoeuvres where a low regression rate could be beneficial [33–36], have been evaluated as well. Lengellé et al. [35] found that, the mass savings are between 3 to 4 kg when changing from a hydrazine mono-propellant system to an H_2O_2/PE hybrid for a small satellite (total mass 100 kg), which is significant considering that the payload mass is only 15 kg. Furthermore, for a Mars aero-capture manoeuvrer, the hybrid propulsion option would be an alternative to the hydrazine mono-propellant [37]. For propulsive Mars orbit insertion, CubeSats (25–100 kg) with hybrid engine solutions can outperform the liquid mono-propellant reference by 6% payload mass [38]. Similar results have been found by Ingenito et al. [39] for 24U CubeSats (less than 30 kg) that can be captured for Mars orbit using a 200 N class N_2O/paraffin motor. Even applications for satellite formation flight have been considered [40]. Another interesting HRE utilization could be active debris removal, where an HRE is transferred onto a piece of space debris for subsequent de-orbiting [41]. Exploration missions to Uranus or Jupiter powered by HREs can meet the same mission requirements as liquid rocket motors (Hydrazine/NTO) while being considerably simpler and having lower cost and power requirements [42].

2.5. Concluding Remarks for Hybrid Perspectives

The previous sections have shown the enormous potential of hybrid rocket engines for replacing both solid and liquid propellants. Concerning the replacement of solids, HREs typically show increased performance due to higher I_{sp}, are considerably less explosive and, most importantly, are throttleable, which increases flexibility and can limit losses due to gravity. Concerning cost, hybrids tend to be more costly than solids because they need to include tanks and valves, which a solid motor does not need. However, their decreased explosion hazard and easier handling could reduce the life cycle costs of hybrids considerably.

Concerning the replacement of liquid propulsion systems, HREs typically show slightly lower performance than liquid references, with the exception of liquid mono-propellants, which are usually below HREs in terms of performance. Nonetheless, hybrids are significantly less complex and less expensive than liquid systems. Moreover, typical toxic bi-propellant hypergolic systems (hydrazine and NTO) can be replaced by non-toxic hybrids.

The main findings of the aforementioned studies on hybrid rocket applications are collected in Table A1. Unfortunately, these studies are all of a theoretical nature, along with a few experimental proofs of concepts; for this reason, in the next section we discuss why, given the benefits discussed in the preceding sections, hybrids have not yet become the dominant chemical propulsion system.

3. Problems Encountered in Hybrid Rocket Development

The different use cases and promising performance assessments laid out in the last section give rise to the question of why, until now, HREs have not been more prominent in the space sector. It appears that there are obstacles that have not yet been fully solved between the theoretical investigations and the experimental applications. To offer more insight, we present typical downsides and technical hurdles that have been described and encountered in the literature backed by experiments. Finally, proposed solutions to these problems are discussed.

3.1. Scalability

The large majority of HRE experiments and tests are conducted on lab-scale or smaller motors. Therefore, it is important to understand how results on small-scale motors (e.g., regression rate) can be translated to medium- or full-scale applications. One major reason for the difficulty of scaling HREs can be found in their unique boundary layer diffusion combustion [43]. When the diameter of the fuel port is increased (as is the case for larger applications) the regression rate typically decreases even at the same mass flux of comparable lab-scale motors. At larger fuel port diameters, the core flow of the oxidizer and the boundary layer at the grain surface interact less with each other, leading to lower regression rates [44].

Therefore, it is necessary to develop similarity rules for the following phenomena [45]:

1. Geometry (constant length-to-port-diameter ratio).
2. Transport phenomena (most importantly the Reynolds number).
3. Heating regime (constant ratio between the heat transfer to the wall and the overall heat addition to the flow from the fuel)
4. Chemistry aspects (most notably constant O/F).
5. Compressibility (constant Mach number).
6. Liquid phase and injector characteristics (droplet lifetime, spray penetration, and momentum).

According to Gany [45], it is crucial to respect geometric similarity, to use the same propellant combination for both smaller and larger scales, and finally to scale the port diameter proportionally to the oxidizer mass flow.

The importance of proper scaling and similarity considerations can be seen in the example of the 1100 kN (multi-port) hybrid boosters of AMROC. AMROC performed upscaling based on bi-port large size motors and predicted the performance of the full-scale tests efficiently. After the bankruptcy of AMROC, testing on the boosters continued in HPDP (Hybrid Propulsion Demonstration Program). Within HPDP [46–48], upscaling was carried out using small-scale multiport sizes, which had a hydraulic diameter only a quarter that of the full-scale booster ports. Consequently, the predicted regression rate was missed by 30%. Therefore, it can be concluded that scale-up should be done based on the largest available port sizes [44]. This, however, indicates that scaling from small- to full-scale is best done by gradually increasing the motor size, which can be a costly endeavour. Nonetheless, AMROC's development of the 1100 kN booster showed that a full-scale hybrid booster (the most powerful tested to date) can be designed, manufactured, and tested within only 10–13 months [44].

Another example of the negative impact of scaling can be found in the efforts on the single-stage-to-orbit (SSTO) Mars ascent vehicle (MAV) [49]. During scale-up, the regression rate was underpredicted, and the team had to artificially lower the regression rate by changing the formulation of the fuel.

However, the development of scalable regression rate formulations [50], better understanding and theoretical investigation of the scale effect [51], and the increasing overall availability of data on HREs is leading to more successful scaling efforts, such as upscaling from low-pressure and low-mass flow to a larger scale for paraffin [52,53], upscaling from 2.5 kN to 40 kN on the CAMUI–Cascaded Multistage Impinging-Jet [54] (see Figure 2), and multi-port upscaling from 250–2500 kN with only 2.6–8.4% error [55].

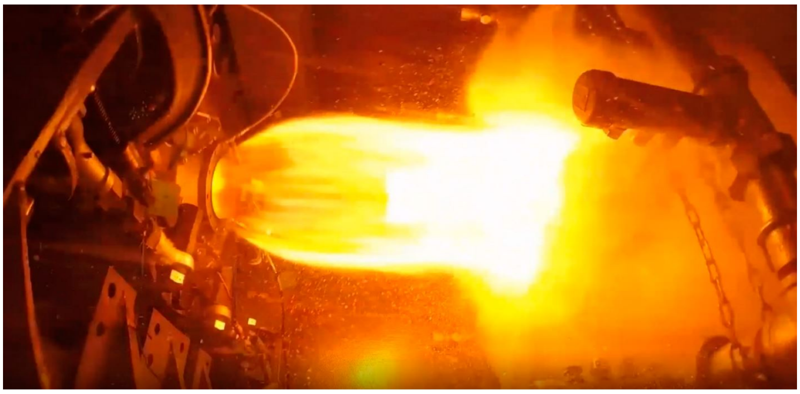


Figure 2. 40 kN test of CAMUI. Reprinted from [54] under Creative Commons 4.0 License.

Two other considerations for large- or full-scale applications need to be briefly discussed as well. The first is the structural integrity of large fuel grains. For wax-based fuel grains, this is due to the properties of the grain material, e.g., high brittleness, melting, and deformation, which can be problematic in large-scale applications. For example, during the investigations of the MAV, cracking of the fuel grain during the casting and burn was constantly observed [49]. There are, however, additives that can increase the mechanical properties while impacting the liquid fuel layer; an extensive review can be found in [56]. Moreover, embedded structures such as armoured grain [57,58] or nested helix [59] can simultaneously increase both the structural integrity and regression rate. Furthermore, recent achievements such as large-scale tests and launches of paraffin-fuelled hybrid rockets (see Section 3.12) show that these problems can be effectively tackled.

For classical fuels such as HDPE or HTPB, multi-port solutions (which are often necessary to increase the thrust) can lead to large-scale fuel break-off during the end of the burn due to the merging of the ports [44,52]. For HTPB, the formation of bubbles during the HTPB manufacturing process needs to be avoided [60].

The second consideration of large-scale applications is the use of additives, which for the most part are metallic. Here, the mass of additives is increased, leading to a more expensive grain and a more difficult manufacturing process. Moreover, with an increasing number of additives, problems such as combustion instabilities, agglomeration, and slag formation [61–63] increase. However, Farbar et al. [62] have shown that a turbulator at the grain entrance increases recirculation while considerably reducing agglomerations, thereby avoiding chuffing instabilities.

3.2. Throttling

Theoretically, throttling of an HRE should be easier than for a liquid engine because only the oxidizer needs to be throttled, which reduces the complexity and needs fewer parts. However, throttling an HRE changes the O/F ratio, which can be a significant problem. Kamps et al. [10] estimated that the combination of nozzle erosion (Section 3.4) and O/F shift (Section 3.5) for their kick stage HRE amounted to 19% of Δv losses. The impact of the O/F shift due to throttling is discussed in Section 3.5; in this section, we present the different difficulties, hurdles, and solutions around throttling of hybrids. Because (deep) throttling is one of the major selling points of hybrids compared to solids, the research into throttleability is important.

In the 1960s, ONERA showed a 5:1 throttling (from 10 kN) in the LEX sounding rocket [64]. The US Air Force Sandpiper [65,66] (8:1, 2.3 kN) and HAST (High Altitude Supersonic Target [67]) with 10:1 throttling range from 5.3 kN maximum thrust fall within the same era. The Firebolt drone, a successor of HAST, achieved a 10:1 throttling range in the 1980s [1]. In 2014, Whitmore et al. [68] demonstrated 67:1 throttling (from 800 N to 12 N) of an N_2O/HTPB engine with a commercial off-the-shelf throttle valve, and stated that to their knowledge this is the lowest turndown achieved by a hybrid or liquid engine (the space shuttle throttled at 1.67:1, while other LREs have reached 20:1). Moreover, the combustion stability of deep throttled HREs with nitrous oxide is superior to LREs. During deep throttling of an HRE, the injector pressure drop increases, whereas for LREs the pressure drop typically decreases. Whitmore et al. [68] stated this effect as the reason for increased deep throttling combustion stability in HREs that use nitrous oxide.

Other successful throttleable hybrids have been reported in ranges of 3:1 [69,70] with soft landing experiments (see Figure 3), roughly 4:1 (from around 700 N) [71], 8.88:1 (from 950 N) [72], 1.66:1 [73], and 10:1 [74,75]. Ruffin et al. [76] achieved throttling ratios of 12.6:1 and Bhadran et al. [77] throttled down to 5.5:1. For a better overview, a selection of throttleable hybrid motors is shown in Table 2.

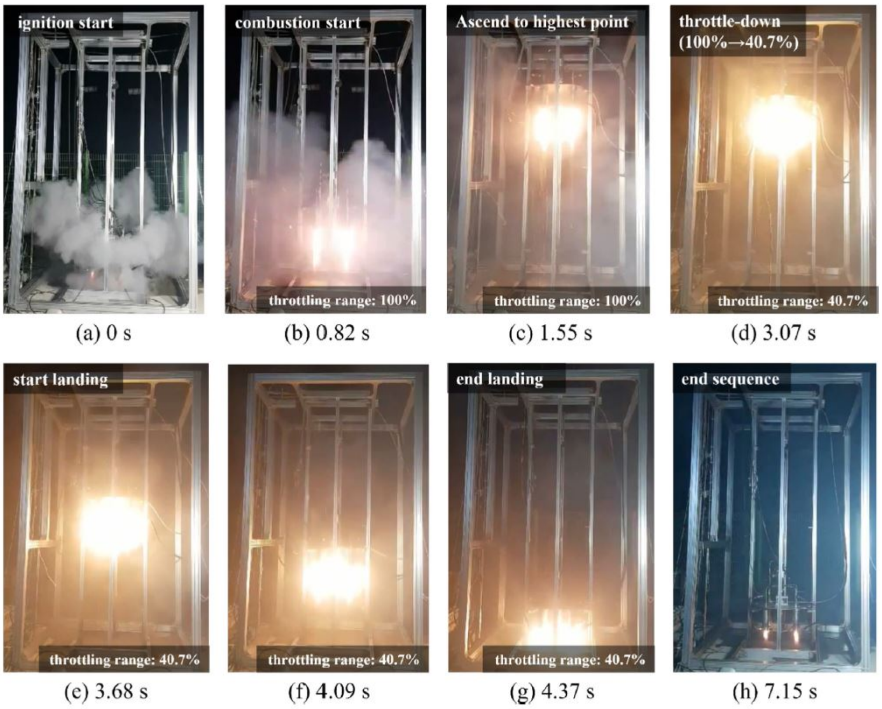


Figure 3. Soft landing of four clustered 50 N HREs. Reprinted from [70] under Creative Commons 4.0 License.

From 2011–2014, the European Union funded an international consortium called SPARTAN (SPAce exploration Research for Throttleable Advanced eNgine) [78–80] to investigate throttleable hybrid rocket engines capable of planetary and lunar soft precision landing, as qualified throttleable engines did not exist in Europe during that era [78]. To avoid complex liquid propellant systems, a hybrid rocket using HTPB as fuel and H_2O_2 (87.5%) as oxidizer was chosen, showing throttling capabilities of 10:1, from 1800 N to 180 N. Tower and helicopter drop tests were carried out as well [78,80,81].

Table 2. Overview of achieved throttling ranges (selection).

Thrust	Throttling Range	Ref.
800–12 N	67:1	[68]
841.4–66.8 N	12.6:1	[76]
max. thrust 5300 N	10:1	[67]
1800–180 N	10:1	[78,81]
500–50 N	10:1	[74]
motor class 444 N	10:1	[75]
950–107 N	8.88:1	[72]
max. thrust 2300 N	8:1	[65,66]
∼186–34 N	5.5:1	[77]
10,000-2000 kN	5:1	[64]
700–175 N	4:1	[71]
∼56–18.4 N	3:1	[69,70]
1200–722 N	1.66:1	[73]

Typical technical hurdles (similar to those around throttling with liquid engines) involve the choice of throttling method. These include flow control valves, variation of the injection area, and secondary gas injection [71,82]. Another difficulty is the nonlinear flow dynamics of the ball valves [83].

A different method of throttling is Pulse Width Modulation (PWM). Instead of changing the oxidizer mass flow, the engine is operated in pulsed mode to achieve the desired thrust profile. Anthoine et al. [84] have demonstrated multi-pulsed operation of an HRE that accumulates up to one minute with constant performance. The downsides of PWM thrust modulation are inability to perform thrust vector control during the coast phases and I_{sp} penalties due to the repeated ramp-up phase during ignition and shutdown [82]. In fact, because of the diffusion flame and the finite conductivity of the solid fuel, hybrid engines have a thermal lag; Karabeyoglu et al. [85] listed the lag in the order of magnitude of 10^{-1} s. For normal throttling operations, this lag is typically considered negligible, as repeatability is deemed more important than response speed [86]; however, for PWM this lag plays a major role.

3.3. Repeatability and Active Control

One major downside of hybrid rocket engines is the high amount of randomness in the thrust and regression rates, resulting in decreased repeatability. For example, Whitmore et al. [60] stated that their run-to-run variability was close to 10%, although the exact same motor design, automated fire sequence, and same construction technique was used for each test. Whitmore et al. [60] admitted that the variability in the industry appears to be lower; however, the deviation of each run is an inherent downside of hybrids. They postulate that a combination of poorly understood boundary layer combustion processes in HREs, two-phase flows of certain HRE oxidizers (e.g., N_2O), and the low maturity of the fuel manufacturing process of hybrids compared to, for example, solid rocket motors lead to these inconsistencies [60].

If the performance (thrust, total impulse) of an HRE cannot be confidently predicted and reproduced for each motor, hybrid propulsion will never succeed in real applications. Therefore, either the fuel production and general development of HRE needs to reach a more mature stage (with the growing number of successful tests of large engines of numerous hybrid startups hinting at considerable progress), or the engine needs to be actively controlled to match the desired thrust levels. The signal to close the loop can be either the oxidizer mass flow, the chamber pressure, the thrust, or the instantaneous regression rate [60,87,88].

To couple the oxidizer mass flow to the thrust as feedback, it is often necessary to accurately develop (or test) the relationship between the oxidizer mass flow and the thrust, as well as to ensure that the commanded and actual oxidizer mass flow rate are identical [88]. For this, the internal ballistics are of upmost importance, including information on the regression rate parameters, combustion efficiency, and nozzle erosion [88]. Using thrust as feedback is the most straightforward of these, however, it is difficult to measure in-flight and needs to be estimated using flight mechanics, velocity, acceleration, and angular momentum [60]. Moreover, the load cell measurements during ground tests can be noisy. For the chamber pressure, the main problem is that the thrust can only be estimated. In the best case, there exists a strongly linear relationship between thrust and chamber pressure. However, nozzle erosion (see Section 3.4) can eradicate this linearity. Moreover, the pressure probes typically read the pre-chamber pressure. This could vary from the true chamber pressure due to the diffusion combustion process [88] and motor acoustic resonant frequencies [60]. In all three cases (thrust, chamber pressure, and oxidizer mass flow as feedback), extensive and expensive tuning and debugging of the controller gains through simulations, cold flow tests, and open loop tests are necessary. In addition, the general problems with throttling (see Section 3.2) come into play.

Nonetheless, the results of closed-loop control are promising. Whitmore et al. [60] were able to reduce their run-to-run variability from 10% to 3.9% or even 1.5% when accounting for nozzle erosion in their N_2O/HTPB motor. They used low-cost off-the-shelf commercial components. The feedback parameter was either the chamber pressure or the thrust.

Guang et al. [88] achieved 0.5% thrust control accuracy in a range of 400–600 N with an H_2O_2/PE (polyethylene) engine. The same group used closed-loop feedback of an H_2O_2 HRE using 58% aluminized HTPB [89] with a control error of less than 0.5% and a steady state error of less than 5%, using thrust as the feedback parameter. These findings are similar to the results obtained by the same group using polyethylene [88], which is interesting because HREs with high percentages of aluminium show higher combustion instabilities, resulting in noisier feedback [89].

Bhadran et al. [77] used the chamber pressure as feedback in their compressed air/paraffin engine. The controlled thrust yielded a 2–6 N mean error for 78–127 N (4.25%).

Velthuysen et al. [83] showed closed-loop throttling using pressure or thrust signals on an N_2O/paraffin engine. Although suffering from increased thermal lag and significant O/F shift because of the fast-burning paraffin and the oxidizer tank pressure decay, a steady thrust profile could be maintained within 2.4% of the maximum thrust of the motor.

One of the biggest hurdles of closed-loop control for hybrids is that these methods typically rely heavily on the observation that the oxidizer mass flow or chamber pressure has a linear relationship to the thrust in smaller-scale motors [60,88,89]. For fast-regressing fuels such as paraffin, larger motors, longer burn times, deep throttling, or significant nozzle erosion, the linear relationship is most probably not valid any more. Nonetheless, closed-loop throttling can solve the problem of run-to-run variations.

Moreover, the thrust response delay due to the finite conductivity of the solid fuel (see Section 3.2) and the different behaviour of the engine when throttling up or down can lead to hurdles around controlling the thrust of an HRE [90]. However, the integrated absolute error can be reduced significantly with proper gain scheduling of the controller [90].

On a related note, closed-loop control has been investigated for A-SOFT (Altering-intensity Swirling Oxidizer Flow Type) engines [87,91]; however, the goal in these studies was to minimize the O/F shift. Therefore, the studies concerning this issue are discussed in Section 3.5.

3.4. Nozzle Erosion

Nozzle erosion of the ablatively cooled graphite nozzles in HREs poses a significant problem. When the material of the nozzle throat erodes, the pressure and the I_{sp} of the engine [5,92,93] changes as well. Generally speaking, the nozzle erosion limits the burn duration. Although this problem is present in SRMs as well, the erosion for hybrids is typically around 1.5 times higher than for aluminized solid rocket motors because of higher mass fractions of the oxidizing agents in HRE combustion [92].

Thus, nozzle erosion renders it difficult to predict the performance of an HRE because the chamber pressure and I_{sp} change over time. Another delicate effect of nozzle erosion is the link between the oxidizer mass flow, chamber pressure, and thrust. As discussed above (see Section 3.3), closed-loop throttling systems often rely on the proportionality between the oxidizer mass flow and the thrust [88]. With nozzle erosion, this direct link changes over time.

In essence, the problem of nozzle erosion can be tackled in two ways: (a) avoiding nozzle erosion with new methods and materials and (b) better understanding of nozzle erosion and development of advanced modelling techniques to predict, calculate, and incorporate nozzle erosion into the design process, as was done, for example, in [92,94–97].

Several methods have been proposed and tested to tackle nozzle erosion. The most straightforward is to change the material of the nozzle. Chandler et al. [98] proposed materials such as the refractory metals tungsten (3695 K melting point), rhenium (3459 K), tantalum (3290 K), molybdenum (2896 K), and niobium (2750 K). Kamps et al. [97] coated the nozzle throat with silicon carbide (SiC). D'Elia et al. [99] tested SiC-based refractory microconcrete nozzles, which could potentially be a cheaper alternative to refractory metals. Whitmore et al. [100] tested pyrolytic graphite throat inserts surrounded by high heat capacity-absorbing layers.

Another rather simple method is to trigger cooling films (unburnt fuel) that decrease the throat temperature. Narsai [101] proposed placing a small fuel sliver before the nozzle. Likewise, Karakas et al. [93] used a ring of aluminium-doped paraffin to create a cooling film, decreasing erosion by 40%.

More complex methods involve active cooling of the nozzles. Successful suppression of erosion with water by keeping the temperature below 1500 K has been reported [102], and experiments involving regenerative cooling with LOX [103,104] have shown no erosion on the motors, where usually the throat diameter increases by 10–25% [104]. However, regenerative cooling with the oxidizer remains in its early stages and is not very mature.

3.5. Oxidizer-to-Fuel Ratio Shift

A unique (though undesired) characteristic of HREs is O/F shifting over time due to expansion of the fuel port diameter. At a constant oxidizer mass flow, the burning surface increases; additionally, the regression rate depends on the mass flux (and as such the port diameter), meaning that it changes over time. O/F shifting reduces the theoretical performance, as it deviates the mixture ratio from the optimal stoichiometric value. The absolute impact of the natural O/F shift is sometimes debated [105]; on the whole, however, the hybrid community agrees that the O/F shift is more pronounced during throttling because only the oxidizer can be throttled (see Section 3.2). A special case involving O/F shifting involves the propellant couples and motors when the Marxman law ($\dot{r} = a\, G^n$ [106]) exponent n equals 0.5 [82,105]. In this case, the burning surface due to the enlarging fuel port increases by the same order of magnitude as the regression rate decreases due to the lower mass flux (i.e., a higher port diameter equals a lower mass flux at a constant oxidizer mass flow). Consequently, the O/F remains constant for $n = 0.5$.

Two major concepts have been investigated to counter the O/F shift: A-SOFT [87,91], and HREs in which a portion of the oxidizer is injected at the aft end of the chamber [105,107,108]. These aft chamber designs can change the O/F directly by varying the amount of oxidizer injected in the aft chamber without impacting the regression rate. However, twice the number of valves and feed lines are needed to achieve this [82]. For the A-SOFT types, the O/F is controlled by changing the swirl intensity, which directly pilots the regression rate and consequently the fuel mass flow from the surface. In both engine designs, instantaneously determining the regression rate, and consequently the current O/F, is difficult and poses the biggest problem for closing the feedback loop [87]. In flight simulations, O/F-controlled sounding rockets show up to 2.42% increased I_{sp} and 8.1% higher altitude than uncontrolled rockets [91].

3.6. Estimation of the Regression Rate

The regression rate of HREs (or solid propulsion, for that matter) is rather difficult to estimate during tests. However, to properly understand and design an HRE it is necessary to measure the regression rate. Most techniques rely on investigations of the fuel grain after the burn, and more precisely, the final fuel port diameter. For a given initial fuel diameter (D_0), final fuel port diameter (D_f), and burn time (t_b), the regression rate ($\dot{r}$) can be calculated as follows:

$$\dot{r} = \frac{D_f - D_0}{2t_b}. \tag{1}$$

To obtain the final diameter, the most common method is the mass loss method, where the fuel is measured before and after the burn to estimate the mass of fuel loss. Assuming even and axisymmetrical fuel consumption, the final diameter, and consequently the regression rate, can be estimated. Methods that allow a more detailed resolution of the final diameter and the local regression rate include the slicing method, where the fuel grain is cut along its longitudinal axis and the final diameter is digitized using an office scanner [109], structured light 3D scanners [110] (Figure 4), and even X-Rays [111,112] and CT scans [113] (see Figure 5).

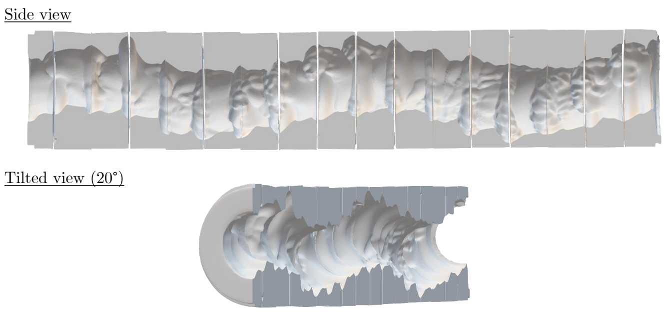


Figure 4. 3D scan of a stepped helix grain after testing. Based on [110], used with permission.

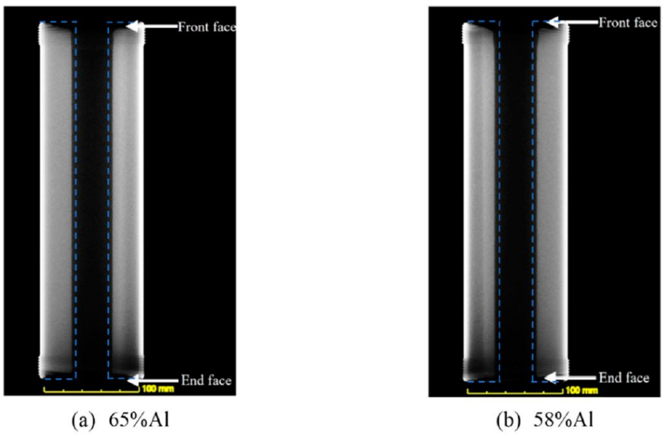


Figure 5. CT scans of paraffin fuel grains with different densities of aluminium. Reprinted from [63] under Creative Commons 4.0 License.

However, all of these methods except for X-Rays only work a posteriori. To obtain the temporal evolution of the regression rate, other methods have been developed. These include, for example, the use of X-ray tomography during the burn [111,112], the application of ultrasonic transducers to measure the web thickness during the burn [114], placing resistors in the fuel grain [87,115], and the so-called ballistic reconstruction technique, which dates back to 1966 [116].

Ballistic reconstruction is based on the mass balance inside the combustion chamber; the total propellant mass (oxidizer and fuel) equals the mass expelled through the nozzle, which is calculated using the characteristic velocity c^*, nozzle throat area A_t, and chamber pressure P_c) [115]:

$$\dot{m}_{\mathrm{ox}} + \dot{m}_{\mathrm{fuel}} = \frac{P_c\, A_t}{c^*}\ . \tag{2}$$

Through ballistic reconstruction, the temporal variation of the average regression can be obtained during testing.

3.7. Low Regression Rate

One of the most prominent downsides of hybrid rocket engines is their low regression rate, which can be smaller by a factor ten compared to, for example, solid rocket engines. A low regression rate and fuel mass flow directly translates to lower thrust. Because of the low regression rate, multiport solutions are often needed to increase the burning fuel surface and fuel mass flow. However, multi-port solutions suffer from low structural integrity, fuel break-off at the end of the burn, high residuals (10–20% [117]), and resonances between the fuel ports [44]. Moreover, HRE fuel grains need to be relatively long in order to increase the fuel mass flow, as this signifies a larger burning surface. This limits design freedom and could lead to unacceptably long designs.

Consequently, many researchers have focused on increasing the regression rate. Because a complete list of methods and designs to increase the regression rate is out of scope for this article, the reader is referred to dedicated reviews of the matter, such as [5,118]. Generally speaking, the regression rate can be increased by [5]:

1. Adjustments to the solid fuel chemical properties such as liquefiable fuels and additives.
2. Advanced injection methods and concepts such as swirl injection and vortex engines.
3. Improving the combustion chamber design by using diaphragms or steps to increase mixing and heat transfer.

Using different designs, the regression rate can be increased by anywhere from 20–800% [5]; see Figure 6. Therefore, it can be said that the regression rate problem is a downside of HREs that can be solved, sometimes by very creative means, though there is not yet one single 'correct' solution.

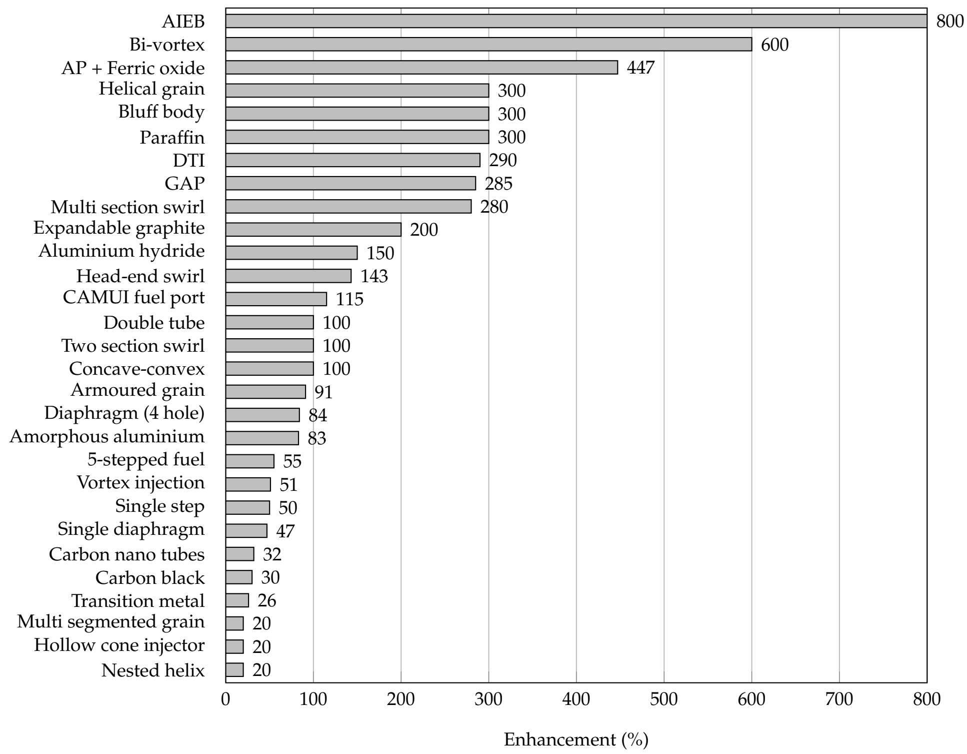


Figure 6. Experimental regression rate increase in selected engine designs. Taken from [5] with permission. References (from top to bottom): AIEB [119], Bi-vortex [120], AP + Ferric oxide [121], Helical grain [122], Bluff body [123], Paraffin [124], DTI [125], GAP [126], Multi section swirl [127], Expandable graphite [128], Aluminium hydride [129], Head-end swirl [130], CAMUI fuel port [131], Double tube [132], Two section swirl [133], Concave-convex [134], Armoured grain [58], Diaphragm (4 hole) [135], Amorphous aluminium [136], 5-stepped fuel [137], Vortex injection [138], Single step [139], Single diaphragm [140], Carbon nano tubes [141], Carbon black [142], Transition metal [143], Multi segmented grain [144], Hollow cone injector [145], Nested helix [59].

3.8. Instabilities

Combustion instabilities are often stated in the literature to be the dominant downside of HREs [2,7,44]. Due to the weak or non-existing link between regression rate and pressure at nominal mass flux levels, hybrids typically do not show catastrophic instabilities [146]. Oscillations are limited to 2–20% [85], or more conservatively to 50–60% of the mean chamber pressure [147]. Nonetheless, these oscillations can lead to unacceptably high thermal or structural loads [147].

The instabilities in HREs can be sorted according to [147]:

1. Low frequency (<200 Hz). Most common. Origin: feed system coupled or chuffing. Moreover, HREs have unique Intrinsic Low Frequency Instabilities (ILFI) caused by coupling of thermal transients in solid fuel, wall heat transfer blocking, transients of the boundary layer over the fuel surface, and vortex shedding in the aft chamber [148–150].
2. Medium frequency (200–2000 Hz). Coupled with the low frequency oscillations. Typically caused by the longitudinal acoustic modes of the chamber. Amplitudes are low. Additionally, hydrodynamic oscillations driven by vortex shedding in the aft chamber can be counted in this group or with the low frequency oscillations [148–150].
3. High frequency (>2000 Hz). Coupled to higher longitudinal or transverse acoustic modes. Usually negligible.

The low-frequency instabilities found by Chae and Lee [151] only occurred when using an aft chamber due to vortex shedding. However, the aft chamber of HREs is essential to increasing the mixing and continuing the combustion of propellants to increase the I_{sp} compared to solid rocket motors. To counter low-frequency pressure oscillations, fuel inserts of lower-regressing fuel (2.5% of total fuel mass) have been shown to delay oscillations and even completely counter instabilities by altering the phase difference between the combustion pressure and heat release, which leads to negative coupling of their amplitudes [151].

A prominent example of hybrid engine pressure oscillation can be found in the large-scale AMROC tests. Oscillations were first encountered when moving from 44.5 kN ($\sim$0.5 m diameter) 'small-scale' motors to the 146.8 kN 'half-scale' motor ($\sim$1.3 m diameter) [44]. These oscillations originated from inadequate LOX vaporization. The instabilities were countered by (a) increasing the Venturi pressure drop, (b) decreasing the injector manifold, (c) increasing the injector pressure drop to decrease LOX droplet size, and (d) fuel 'splash-blocks' to increase the residence time of the oxidizers [44]. During follow-up work on the large-scale hybrid booster in the HPDP, Triethylaluminum–Triethylborane (TEA-TEB) was constantly injected as a hypergolic heat addition to vaporize the LOX. Due to the toxicity of hypergolic fuels and the fact that this effectively eradicates the advantage of 'inert' hybrids, heat addition in the pre-chamber can be achieved using hybrid 'heater motors' [44]. During the development of the hybrid MAV concept [49], a similar necessity to add heat to help with oxidizer vaporization (in this case, MON (Mixed Oxides of Nitrogen)) was described on a larger scale.

To conclude, HREs have inherent susceptibility to pressure oscillations, even if they are typically bound in terms of amplitude. On the other hand, experience with the large-scale AMROC tests has shown that, with enough time and resources, solutions can be found to solve these problems. Karabeyoglu et al. [4], for example, have stated that they found a proprietary method to decrease oscillations without the need for heat addition.

3.9. Numerical Simulations

Numerical simulations of HREs are of the utmost importance, as they can be used to understand the ballistic behaviour (regression rates, injection, combustion instabilities) of an existing engine after a test or predict and optimise future designs. Moreover, simplified and considerably faster codes can be used for large parametric studies and systems analyses, or even for tuning the controllers for closed-loop systems.

However, due to the comparably low amount of research and test data into hybrids, the internal phenomena in the complex diffusion flame combustion of HREs are not yet fully understood. Another problem is posed by the change in fuel port geometry over time due to the fuel regressing. Finally, it can be argued whether Reynolds-averaged Navier–Stokes approaches (as in the majority of HRE simulations) are ideally suited for the complex wall blowing and pyrolysing phenomena at the fuel surface. Nevertheless, numerical simulations of HREs are improving significantly, be it in the modelling of hydrodynamic instabilities [149,150], swirl injection [138,152–154], shape changing simulations [153–157] (see Figures 7 and 8), complex geometries such as helices [158–160], rotated grains [144,161], throttling [72,162], or nozzle erosion [94,163].

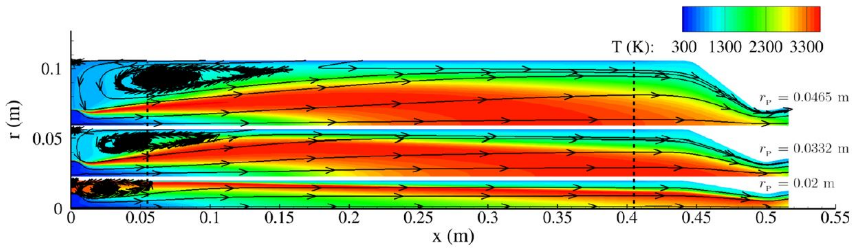
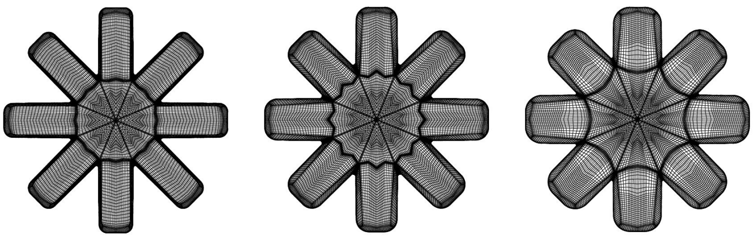


Figure 7. Shape change simulation at different instances. Reprinted from [154] under Creative Commons 4.0 License.

Simplified geometries and simulations carried out on the average fuel port diameter are considerably less computationally costly compared to multiple instances of the fuel port progression, with both providing satisfactory representations of the HRE flow field and local regression rates [157,164]. HRE models (e.g., regression rate models [165] and nozzle erosion [166]) are currently being implemented and improved in system tools such as Ecosim/ESPSS to facilitate and improve systems analysis. MDO approaches with uncertainty analyses are a powerful tool for HRE trajectory and design optimization [11]. Moreover, 0/1D [167–171] and 1.5D [114,172–174] codes for HREs are being developed by researchers to improve calculation speeds, potentially leading to models that can provide real-time estimation of HRE behaviour during flights and testing. However, this will require considerable future investment in development and testing, especially for long-term projects.



(a) 10 s (b) 20 s (c) 30 s

Figure 8. Star-shaped mesh prepared for steady-state simulations at different time instances. Reprinted from [161] under Creative Commons 4.0 License.

3.10. Limited Experimental Database

Due to the low maturity of HREs (see Section 3.11), the available experimental data on hybrids is limited. This poses a considerable problem, as many solutions to the challenges of hybrids (such as throttling, repeatability, low residuals and scalability) rely on empirical data. For example, Barato et al. [117] have stated that if the regression rate is not predicted as precisely as possible through the Marxman parameters, then considerable amounts of residuals arise that limit the performance. The importance of the Marxman parameters ($\dot{r} = a\ G^n$ [106]) for hybrid rocket engines cannot be stressed enough. For instance, Jens et al. [42] used them to evaluate the optimal design needed to reach the outer planets, Guang et al. [88] used them for closed-loop throttling, and Kamps et al. [10] used them to evaluate different kick stage designs. These are only a few of the many examples to date, all of which have the common requirement that the Marxman parameters need to be known as precisely as possible. To ensure that this is the case, extensive testing of each propellant pairing and motor size needs to be conducted in order to create a sufficiently extensive database.

3.11. Low Maturity

Finally, we discuss a final downside that encompasses all the aforementioned disadvantages and challenges of hybrid propulsion, namely, the low overall maturity of hybrid technology. The first very obvious difference between solid, liquid, and hybrid propulsion is the amount of research that has been conducted on each propulsion type. Although hybrid, solid, and liquid propulsion systems emerged at roughly the same time, hybrids received relatively little attention during the space race era, and even afterwards were mostly confined to smaller scales, research, and academia, with the major exception of the AMROC Aquila launcher. Following such achievements as liquefiable fuels [124,175] and the first manned suborbital flight using hybrid propulsion SpaceshipOne [18], the hybrid community has begun to gain momentum, with a notable increase in papers published (as shown in [2]).

Liquid propulsion is widely used, with famous examples including the Saturn V F-1 engines, the Soyuz RD-family, Ariane 5's Vulcain engines, the space shuttle's main engines, and the Raptor engines used by SpaceX [176]. While to list all of the liquid engines that have flight heritage would be out of scope for this review, all of these engines have their own development history that together have progressively increased the maturity of LREs.

The same applies for solid propulsion, such as the boosters used for the space shuttle and Ariane. Moreover, solid rockets are typically dominant in the tactical/defence domain due to their simplicity and flight readiness. In the U.S. alone, several hundreds of thousands of units are produced yearly [7]; for example, unguided solid rockets are being produced at a rate of at least 250,000 units a year [176]. Compared to these numbers, the development and knowledge base of hybrid rocket technology is marginal. Certainly, components from solid and liquid propellant systems (e.g., ablatively cooled nozzles, HTPB manufacturing processes, igniters, and materials) can be transferred to a certain extent; however, the discrepancy in hybrid research and development remains obvious.

The problem that this low level of maturity poses to hybrids can be illustrated through the technology selection that was carried out for the MAV for NASA, as described by Story et al. [49]. Hybrids were considered a promising candidate with the capacity for a single-stage-to-orbit rocket. Development of the design started from a 'blank sheet' [49], and significant progress on the hybrid concept was achieved between 2015–2019. The motor was tested at full scale with vacuum ignition and restarts, achieving rapid ignition (<507 ms), stable combustion ($<5\%$), and 0.4% residuals. Despite all of this progress, however, NASA selected the solid propellant option. One of the main reasons was the high uncertainty margins of the hybrid system when comparing the expected cost of the hybrid and solid options. Due to this high uncertainty, the hybrid system had comparable costs to the solid system rather than a lower cost. The main reason the hybrid option was discarded, however, was the relative maturity of the technologies. Following NASA's recommendation of a TRL (Technology Readiness Level) of at least 6 for the preliminary design review, the estimated TRL 5+ of the hybrid system was deemed to be too risky, and the hybrid option was unanimously discarded [49]. Simply put, hybrid systems are not currently selected for missions due to their insufficient TRL. Nonetheless, mission integration is essential for advancing their TRL and achieving required maturity, resulting in a vicious cycle. For this reason, Kuo and Chiaverini [43] emphasised the necessity of long-term hybrid programs, e.g., HUP [177–179], ORPHEE [8,9,15], HyTOP [180], HPDP[46–48], Peregrine [181–183], SPARTAN [78,80,81], HYPROGEO [34,184,185], SMILE [186], ALTAIR [187], etc.

The effect that long-term development can have on the success of HREs may be shown by the following example. In 1981, the company Starstruck was founded; it developed a ∼156 kN hybrid sounding rocket (LOX/PB) that was launched from sea in 1984. Three seconds into the flight, a TVC (Thrust Vector Control) LOX valve froze and the termination command was issued [1]. The company dissolved in 1984. With new funding and under a new name, the same team continued as AMROC, which between 1985–1995 carried out extensive testing on LOX/HTPB hybrid motors, including the record-breaking 1100 kN motor. Due to funding problems, AMROC teamed up with NASA and other companies to demonstrate flight-worthiness under the HyTOP (Hybrid Technology Option Project 1993–1994). In 1995, AMORC lost its sponsor. The remaining project partners continued to show great interest in the engine. In 1999, the 1100 kN motor was tested again [18]. Later, SpaceDev acquired the patents and rights to AMROC technology and their test data [18]. Ultimately, SpaceDev's SpaceshipOne won the X-prize as the first privately funded company to carry three people above 100 km [18]. The successor, SpaceshipTwo of Virgin Galactic, launched their sixteenth 'astronaut' in September 2023 [188]. Looking at the heritage of Starstruck and AMROC, the success of SpaceshipOne/Two can be traced back to 1981. It is not entirely clear how much of the data and designs from the earlier developments can be found in the design of SpaceshipOne/Two (e.g., the oxidizer for Dolphin and AMROC was LOX, while SpaceshipOne/Two used N_2O, though the fuel remains HTPB-based). However, the experimental and other data that SpaceDev acquired can be assumed to have played a major role. This illustrates the importance of long-term funding and development, an area in which which hybrids are significantly lacking compared to liquids and solids.

Fortunately, with the new space age, commercial companies appear to have noticed the potential of hybrid rocket engines for simpler and cheaper propulsive solutions, with many recent startups choosing hybrid propulsion for their engines [2].

3.12. Discussion on Recent Milestones

While it has been shown that hybrids encounter many problems, it appears that each of these can be countered. At the core, the main problem is low maturity. Long-term projects and industry applications could help to break the vicious cycle, as exemplified by the following recent HRE milestones.

Nammo launched their Nucleus sounding rocket in 2018, reaching an altitude of 107.4 km using a 30 kN H_2O_2/HTPB hybrid engine [189]. In 2023, the student world record for a hybrid rocket engine (LOX/paraffin) was achieved, with the student team HyEnd from the University of Stuttgart reaching an altitude of 64 km [190]. InnoSpace successfully launched their 150 kN sounding rocket in 2023, though they did not disclose the altitude it reached [191]. In 2022, the Advanced Rocket Research Center (ARRC) launched their first sounding rocket, reaching 3 km instead of the planned 10 km altitude because of a lack of fuel. ARRC stated that they tested the first active GNC (Guidance, Navigation, and Control) controlled hybrid rocket in the world [192]. HyImpulse reported the final flight readiness tests of their 75 kN LOX/paraffin engine, and the maiden flight has been announced for the end of 2023 [193]. Likewise, Gilmour space expects their maiden flight by the end of 2023, including flight readiness tests of their 110 kN motor [194]. Vaya space tested a sounding rocket in 2022 [195]. In 2023, SpaceForest conducted their first launch of the Perun rocket, which reached an altitude of 22 km with their 30 kN LOX/paraffin motor [196].

To close this discussion on recent HRE achievements, let us consider the first example of (tethered) hovering of an HRE rocket [73]. In 2020, the team of the ARRC successfully demonstrated stable 25 s hovering of the H_2O_2/Polypropylene rocket 3 m above the ground using both attitude and position controls. The four engines generated 1200 N of thrust each [73]. This remarkable achievement proves that HREs are capable of vertical take-off and landing. This manoeuvre is only possible if the challenges discussed in the present article (regression rate prediction, throttling, feedback control, scalability, etc.) can be solved to a certain extent. However, the group has stated that the technology continues to need considerable future effort, especially because of high O/F shifting and difficulty in thrust vector control due to the length of the combustion chambers [73].

To conclude, hybrid propulsion technology appears to be close to a breakthrough; though this was already stated by Kuo and Chiaverini [43] in 2007, now, in 2023, this statement seems more relevant than ever.

4. Conclusions

> "There is probably no other good idea in chemical rocket propulsion that has had as long a development as the idea of the hybrid rocket." [52]

This quote from Cantwell et al. [52] accurately describes the history of hybrid propulsion systems. We have shown in this article that numerous case studies indicate the strong potential of hybrid rocket engines; however, until now, the technology has not been readily available. We have discussed the major challenges and technical hurdles encountered in HRE development over the years. However, while the hybrid community has found solutions to tackle the problems related to each challenge, the TRL of the system and the solutions developed to date remains low. At the most basic level, the major gatekeeper for HREs is their low level of technological maturity. Because of their low TRL, HREs are rarely selected for real applications, which in turn hinders the advancement of the technology in a vicious cycle. This cycle needs to be broken, either by long-term agency-funded research projects or private investments from the launcher industry and startups. Fuelled by the recent market demand for safe, reliable, and low-cost space access, hybrid rocket engines have regained significant attention and momentum. In fact, the ESA's European Space Research and Technology Center has identified "lowered production, operational, and transport expenses due to reduced propellant toxicity and reduced explosion hazards" [60,197,198] as necessary steps towards low-cost space access. Hybrid rocket propulsion could be on the brink of becoming capable of fulfilling this role, as shown by the numerous examples from the open literature discussed in this article.

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Conflicts of Interest: The authors declare no conflicts of interest.

Abbreviations

The following abbreviations are used in this manuscript:

A-SOFT	Altering-intensity Swirling Oxidizer Flow Type
AF	Advanced Fuel
AIEB	Axial-Injection End-Burning
ALTAIR	Air Launch Space Transportation Using an Automated Aircraft and an Innovative Rocket
AMROC	American Rocket Company
AP	Ammonium Perchlorate
ARRC	Advanced Rocket Research Center
CAMUI	Cascaded Multistage Impinging-Jet
DTI	Distributed Tube Injector
GAP	Glycidyl Azide Polymer
GEO	Geostationary Orbit
GIRD	Group for the Study of Reactive Motion

GTO	Geostationary Transfer Orbit
GNC	Guidance, Navigation, and Control
HDPE	High-Density Polyethylene
HAST	High-Altitude Supersonic Target
HPDP	Hybrid Propulsion Demonstration Program
HRE	Hybrid Rocket Engine
HTPB	Hydroxyl-Terminated Polybutadiene
HUP	Hydrogen Peroxide Hybrid Upper-Stage Program
HYPROGEO	Hybrid Propulsion Module for Transfer to GEO Orbit
HyTOP	Hybrid Technology Option Project
I_{sp}	Specific Impulse
ILFI	Intrinsic Low-Frequency Instabilities
LEX	Lithergol Experimental
LRE	Liquid Rocket Engine
LOX	Liquid Oxygen
MAV	Mars Ascent Vehicle
MDO	Multidisciplinary Design Optimization
MMH	Mono-Methyl Hydrazine
MON	Mixed Oxides of Nitrogen
NTO	Nitrogen Tetroxide
O/F	Oxidizer-to-Fuel Ratio
ORPHEE	Operational Research Project on Hybrid Engine in Europe
PB	Polybutadiene
PE	Polyethylene
PWM	Pulse Width Modulation
RP-1	Rocket Propellant 1
SMILE	Small Innovative Launcher for Europe
SSTO	Single-Stage-to-Orbit
SPARTAN	Space Exploration Research for Throttleable Advanced Engine
SRM	Solid Rocket Motor
TEA-TEB	Triethylaluminum–Triethylborane
TVC	Thrust Vector Control
TRL	Technology Readiness Level
UDMH	Unsymmetrical Dimethylhydrazine

Appendix A

Table A1. Summary of use case studies on HREs.

Use-Case	Propellants [a]	Description	Remarks	Ref.
Upper stage	LOX/paraffin	Replacing Orion 38 solid upper stage of Taurus/Pegasus launch vehicle	40% higher payload and 15–18% lighter than Orion 38 solid upper stage	[4]
Upper stage	LOX/wax	Replacing solid Zefiro 9 and liquid AVUM of Vega with single HRE	Increase in payload by 62% (turbo pump) and +79% (electric pump)	[11]
Upper stage	H_2O_2/PE	Replacing solid Zefiro 9 and liquid AVUM of Vega with single HRE	Increase in payload by 38% from 1430 to 1971 kg	[11]
Upper stage	H_2O_2/HTPB	Replacing solid Zefiro 9 and liquid AVUM of Vega with single HRE	Initial design competed or surpassed, 1.7 times pricier than a solid, 5 times cheaper than LRE	[12]
Upper stage	LOX/AF [b]	Replacing solid Zefiro 9 and liquid AVUM of Vega with single HRE	+800 kg (+60%) increase in payload capacity	[9]
Upper stage	H_2O_2/HTPB & PE	Replacing solid upper stage of VLM-1 microsatellite launch vehicle with HRE	HRE superior in thrust, mass, and energy efficiency. Cost increased by 1.5, but throttleable	[13]
Kick stage	H_2O_2/HDPE	Post boost kick stage	Over 4000 m/s velocity change for a one-cubic meter, 916 kg HRE	[10]
Main stage	LOX/AF [b]	Replacing LOX/RP-1 first stage of Falcon 1	+100% payload capacity	[9]
Booster/main stage	LOX/HTPB	Adding two hybrid boosters to ATLAS 2AR launch vehicle	Payload capacity to GTO increases from 3900 kg to 4763 kg	[19]

Table A1. *Cont.*

Use-Case	Propellants [a]	Description	Remarks	Ref.
Booster	LOX/HTPB	Replacement of Ariane 5 solid booster	20 tons lighter system weight of HRE	[20]
Lunar lander	LOX/AF [b]	Lander inserted with Ariane 5 ME, lander mission starts from 100 km lunar orbit	Landing 2000 kg dry mass on the lunar surface, of which 280 kg engine inert mass	[9]
Lunar sample return	LOX/paraffin	Ariane 5 inserts lander and return rocket (8800 kg) in lunar transfer orbit	With 95% combustion efficiency, the soil sample of HRE is 75 kg, comparable with 87 kg of LOX/methane liquid	[22]
In-situ lunar	O_2/aluminium	Transport of 500 kg payload from the Moon to a 300 km lunar orbit.	349 kg of the ascent rocket from earth and 1410 kg of lunar material (oxidizer, fuel, fairings, tanks) fulfil mission requirements	[26]
Mars lander	H_2O_2/AF [b]	Based on NASA Phoenix lander mission	Able to land 350 kg dry mass on the Mars surface (75 kg engine inert mass)	[9]
Mars ascent	MON-25/wax	16 kg payload (sample return)	Single stage to orbit possible with hybrid	[49]
Mars ascent	Nytrox/paraffin	36 kg payload in 500 km Mars orbit	HRE system mass 30% lower than solid propellant alternative	[29,30]
Mars sample return	LOX/wax	Return to 170 km Mars orbit	HRE capable of delivering around 86 kg, depending on configuration	[11,31]
Mars manned return	LOX/wax	Return to 170 km Mars orbit	HRE capable of delivering around 13 tons, depending on configuration	[11,31]
CubeSat	H_2O_2/PE	Replacement of mono propellant to HRE for 100 kg CubeSat, 15 kg payload	HRE can increase the payload by 3 to 4 kg	[35]
CubeSat	GOX/HDPE	Mars orbit insertion of 25–100 kg weight	Payload mass compared to liquid mono propellant increases by 6%	[38]
Outer planets	MON-3/PE wax	Δv required for Europa flyby 1.52 km/s Δv required for Uranus 1.96 km/s	HREs can fulfil mission requirements similar to hydrazine/NTO while being less toxic, simpler, and low cost	[42]

[a] Additives not listed. [b] 'Advanced Fuel', a hypothetical HTPB or paraffin fuel with additives [9].

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