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Research Article

Optimizing Sounding Rocket Development Through Propulsion System Testing

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Abstract

Propulsion system testing plays a critical role in the development of experimental sounding rockets by providing experimental validation, reducing engineering uncertainty, and supporting informed design decisions. While test campaigns require additional infrastructure and operational resources, they enable improvements that are often unattainable through analytical methods alone. This paper presents both theoretical and practical aspects of propulsion system testing for sounding rockets, including measurement techniques for thrust, pressure, temperature, and propellant mass, together with selected solutions implemented in the propulsion system test stand developed by the student research group PWr in Space.

The study investigates the technical and economic impact of propulsion system testing throughout the sounding rocket development process. Particular emphasis is placed on how experimental validation supports the optimization of structural mass, propellant mass, subsystem reliability, and manufacturing methods. Case studies covering injector plate redesign, FEM-based structural optimization, and hybrid rocket fuel grain manufacturing demonstrate how experimental campaigns reveal design deficiencies, validate engineering assumptions, and guide design improvements beyond the capabilities of theoretical analysis alone.

The results indicate that, despite the additional cost of conducting test campaigns, propulsion system testing can substantially reduce the overall cost and risk of sounding rocket missions. Improvements such as reduced propellant consumption, lighter structural components, and enhanced subsystem reliability contribute to greater mission efficiency and lower development costs. These findings demonstrate that propulsion system testing should be regarded not only as a validation procedure but also as a strategic engineering and economic optimization tool in the development of experimental sounding rockets.

Keywords: sounding rocket, propulsion system, test campaign, reduce costs

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Introduction

Sounding rockets and their missions

Despite the space industry's ongoing efforts to reduce the cost of orbital flights, launching a payload into orbit aboard a Falcon 9 rocket in 2026 still costs around 74 million USD (SpaceX, 2026). As a result, there remains a strong demand within the industry for a low-cost, reliable, and versatile research and testing platform. In this context, one notable category comprises smaller, suborbital rockets known as sounding rockets (National Aeronautics and Space Administration, 2026). The term "sounding" originates from the nautical practice, which refers to measuring the depth of water. This nomenclature aptly captures the primary function of these rockets: to measure, investigate and validate various atmospheric and space-related phenomena. A significant advantage of this type of rocket is its relatively low-cost compared to orbital rockets (e.g., Falcon 9), which is approximately 1 million USD (Choucair, C., 2025). Sounding rockets have played and continue to play a significant role in aerospace industry, where evolving technology allows for more complex experiments (Heatwole, S. E., 2024). The use of this type of rocket continues to grow, so minimizing costs through appropriate technical optimization of the rocket becomes crucial. These parameters may include the propulsion system and the amount of fuel used to achieve a given altitude with minimal fuel consumption (Galletly, M. and Verstraete, D., 2025).

Indication of the need to test rocket propulsion systems

In rockets, one of the biggest challenges are propulsion systems, the actual operation of which is difficult to simulate using numerical methods, and some phenomena may be impossible to predict – e.g. problems with fuel combustion instability (National Aeronautics and Space Administration, 2025). For this reason, experimental testing should be used to validate this system. Although these tests can be expensive, they are ultimately cost-effective and reduce the overall cost of rocket operation (Abdullah, M., et al., 2024). Thanks to them, we can, among other things, increase fuel efficiency, reduce the weight of the propulsion system and optimize operating parameters (Suksila, T., 2024). However, failure to perform tests may result in an explosion or loss of the rocket, which is associated with high financial losses (Bennewitz, J. W. and Frederick, R. A., 2013).

Preliminary cost assessment of Sounding rockets missions and test

The development and operation of sounding rockets involve multiple cost sources related to both the propulsion system and the launch infrastructure. The most significant expenses are associated with propulsion system manufacturing, propellant procurement, structural manufacturing, avionics, payload integration, and launch operations. In the case of propulsion system testing, additional costs include the construction of the test stand, instrumentation, data acquisition systems, and consumable resources used during static hot-fire tests. Although propulsion system testing increases the initial development cost, it enables validation of engine performance and identification of design flaws before flight. As a result, testing may reduce the overall mission cost by reducing engineering uncertainty and enabling identification of potential failure modes before flight, decreasing unnecessary propellant mass, and enabling optimization of structural components and operational procedures. Potential areas for cost reduction include optimization of propellant mass, reduction of structural mass through validated load analysis, improvement of subsystem reliability, and simplification of manufacturing and operational procedures. These aspects are further discussed in the following sections.

Procedure for Testing Rocket Propulsion Systems

Rocket engine testing in theory

Engine parametres

The rocket engine is a complex module which has many parameters, some of which cannot be calculated analytically. In addition, some of the parameters that can be calculated have to be verified experimentally in order to validate the calculations. Below are listed some examples of engine parameters that can be measured during the test.

The rocket engine, as a critical component of a rocket, has several key parameters that must be thoroughly tested:

- **Thrust** – the force produced by the engine: T
- **Engine burn time** – the duration of engine operation: t_b
- **Pressure** – the pressure in the combustion chamber and in the injector feed system
- **Temperature** – the temperature in critical areas of the engine structure
- **Propellant mass flow rate** – the mass of propellant flowing through the engine per unit time

Once these values are measured, several derived engine parameters can be calculated, including:

- **Total impulse** – the integral of thrust over time: $I_t = \int_{t_0}^{t_b} T(t)dt$
- **Specific impulse** – the total impulse divided by the weight of the propellant: $I_s = \frac{I_t}{g \times m_p}$

where: m_p – propellant mass g - gravitational acceleration

Measuring methods

Measurement of Thrust. Thrust is measured on a test bench. The engine is mounted on the bench and, after ignition, exerts force on a load cell. The load cell receives the input signal in the form of stress induced by the thrust force. These stresses are then converted into voltage differences, which are read by an ADC (analog-to-digital converter) and interpreted by an MCU (micro-controller unit) as a thrust force.

Thrust can be measured in more than one configuration:

- vertically with thrust directed downwards
- vertically with thrust directed upwards
- horizontally

All configurations are presented on fig 1.

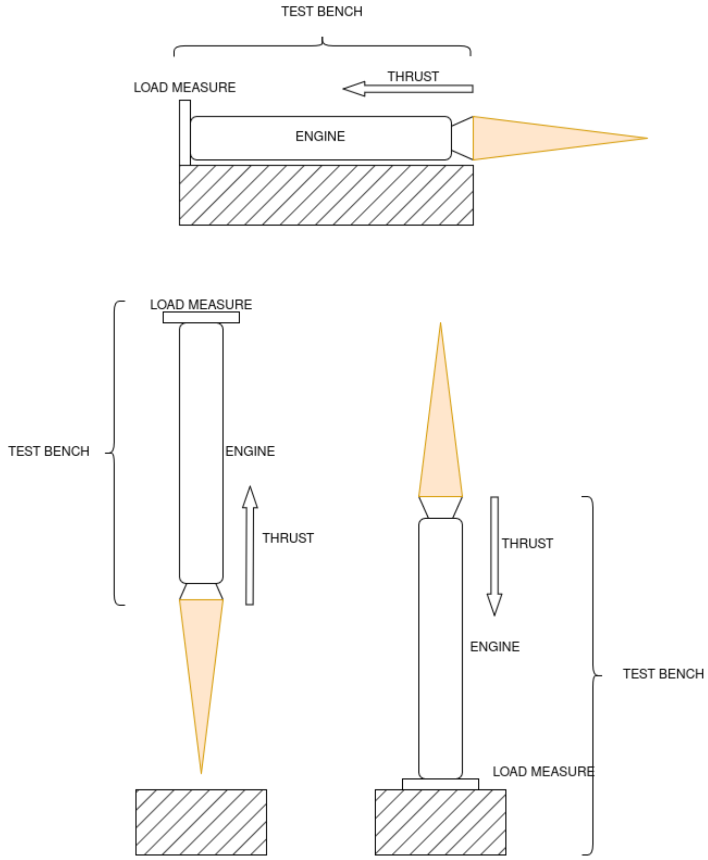

Fig 1. Test bench directions

Measurement of pressure. Pressure is usually measured:

- inside combustion chamber
- on injector
- inside pressure tanks filled
- in the end of propellant channels

Pressure is most commonly measured using pressure transducers. These devices convert the force exerted on a sensing membrane within the measurement channel into an electrical signal, typically voltage or current. Most transducers exhibit an approximately linear relationship between input pressure and output signal, which allows for straightforward and accurate estimation of the actual pressure values.

Combustion chamber pressure is a particularly important parameter, as accurate measurements can be used to estimate thrust. Therefore, it can serve as an indirect substitute for direct thrust measurement. To calculate the thrust force using the chamber pressure, the following formula is used (Sutton, G. P. and Biblarz, O., 2017):

$$F = C_F \times A_t \times p_c, \tag{1}$$

where:

- F is the thrust force,
- A_T is the nozzle throat area,
- p_c is the combustion chamber pressure,
- C_F is the thrust coefficient.

The ideal thrust coefficient (excluding friction or flow issues) is calculated with the following equation (Sutton, G. P. and Biblarz, O., 2017):

$$C_F = \sqrt{\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}\left[1-\left(\frac{p_e}{p_c}\right)^{\frac{k-1}{k}}\right]} + \frac{A_e}{A_t}\left(\frac{p_e-p_a}{p_c}\right) \tag{2}$$

where additionally:

- A_e is the area of the nozzle exit,
- k is a specific heat ratio,
- p_e is the pressure at the nozzle exit,
- p_a is the ambient pressure. In case of the propulsion testing, it will be the ground level pressure.

With the appropriate corrections (Sutton, G. P. and Biblarz, O., 2017), this value can provide a valid thrust expectation.

Measurement of temperature. Temperature within such ranges is usually measured by two types of sensors: a thermocouple and an RTD sensor. Both of them have the different working principle. In case of thermocouple, there are two metals connected in one point making a probe. The temperature difference creates the potential, allowing for the current flow. An RTD sensor is a resistor that changes its resistance with temperature. Thermocouples are better for extreme temperatures, up to thousands of degrees; however, they lack in linearity and accuracy, where RTD sensors like Pt100 are superior, which are better for lower temperature ranges. Both of the sensor types are suitable to measure the temperatures of combustion chamber.

Measurement of propellant mass. Specific impulse is an important rocket propulsion parameter. It gives an estimate of engine "efficiency", showing how much thrust the engine is able to produce per unit mass of propellant. It allows to calculate the effective exhaust velocity and velocity increment in potential spaceflight.

The optimal scenario is to measure the mass flow continuously during the engine burn; however, this requires a special flow meter, which is often difficult to include in the hydraulic system. An easier way is to measure mass of the engine before and after the burn, adapting the load cell for thrust measurement. It might result in less accurate measurements, due to opposite direction of force, however the simplicity of setup makes it viable. To achieve more accurate measurement, an additional mass measurement of the whole propulsion system might be introduced.

All measurement methods described above are affected by uncertainties resulting from sensor accuracy, calibration procedures, data acquisition systems and environmental conditions during testing. Consequently, measured values of thrust, pressure, temperature and propellant mass should be interpreted within the accuracy limits of the applied instrumentation. These uncertainties may influence the calculated values of total impulse and specific impulse and constitute one of the factors contributing to differences between theoretical predictions and experimental results.

Rocket engine testing in practice

PWr in Space Scientific Association

Sounding rocket development is not confined to the private sector or to major space agencies such as NASA or ESA. A significant share of contributions comes from university-based scientific associations across the world, which play an increasingly important role in advancing experimental rocketry and space technologies. One such association is PWr in Space, operating at the Wroclaw University of Science and Technology. The team is currently developing its first liquid-propellant engine intended for self-designed sounding rockets, which creates an increasing demand for comprehensive engine testing and validation.

Test stand

The team has built its own test stand, shown on figure 2, suitable for vertical testing of engines with up to around 10 kN of thrust.

Fig 2. PWr in Space test stand right after the engine static hot-fire test

This test stand consists of:

- **Engine mounting structure**,
- **Reaction structure integrated with a load cell**,
- **Propellant loading hydraulic system**,
- **Loading line disconnection system**,
- **Oxidizer supply bottle weighing system**,
- **Data acquisition system**,
- **Electronic control system**,
- **Imaging system**.

Since the liquid-propellant engine developed by PWr in Space utilizes ethanol as fuel, only the oxidizer requires remote loading, which significantly simplifies the system architecture. Furthermore, as nitrous oxide is used as the oxidizer and nitrogen serves as the pressurant, the implementation of cryogenic systems is not required.
A potential test stand for sounding rocket engine may be extended using:

- **Oxidizer vessel cooling system**,
- **Oxidizer loading and cooling system**,
- **Fuel loading system** (for remotely handled fuels),
- **Cryogenic oxidizer and/or pressurant system**,
- **Additional subsystems as required**.

Specific solutions

The test stand developed by the team incorporates several distinctive design features tailored to the requirements of the engine, enabling time and cost optimization across various processes while improving reusability and enhancing the overall versatility of the system.

Loading lines disconnection system. The first of these solutions focuses on the rapid and reliable disconnection of the loading lines. The system utilizes compressed air to actuate a pneumatic cylinder mounted on an adjustable stand, enabling remote and controlled retraction of the connections. This approach increases operational safety and reduces turnaround time between successive tests.

Engine mounting structure. The engine mounting structure was designed to accommodate engines of varying sizes and configurations. The load cell can be adjusted vertically, allowing for flexible positioning of the propulsion system and ensuring proper alignment with the thrust measurement axis. This adaptability facilitates testing of different engine geometries without requiring significant modifications to the stand.

Concrete test bed. The test bed, constructed from concrete blocks and reinforced with additional concrete layers, provides effective shielding of the surroundings while directing the exhaust plume in a controlled direction. Its robust design enhances structural durability and ensures safe dissipation of thermal and mechanical loads generated during engine operation.

Oxidizer supply bottle weighing system. The oxidizer supply bottle weighing system is mounted on a tripod structure, which ensures stable and reliable positioning of the tank regardless of ground conditions. This solution allows for operation on uneven or unprepared terrain, improving the portability and field-readiness of the test stand. Additionally, the tripod configuration minimizes the influence of external disturbances on the measurement, contributing to increased accuracy in oxidizer mass estimation during the test.

Cost assessment

The cost of propulsion system testing can be divided into two main categories: installation costs and operational costs. The first category includes any permanent and long-lasting equipment, while the second consists of any resources that are used on limited test events (usually on a single test).

Permanent costs. Estimating the cost of the test stand is difficult since many modules are constructed with the support of external sponsors, however the majority of costs may be divided into the following subcategories:

- **Concrete** — The PWr in Space test stand required approximately 110 bags of concrete, each weighing 25 kg. Assuming the use of standard C20/25 concrete, with an estimated unit price of approximately 15 PLN per bag (Leroy Merlin Polska, 2026), the total material cost amounted to approximately 1,650 PLN (approximately 390 EUR).

- **Hydraulic and pneumatic systems** — These systems comprise cryogenic

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solenoid valves as well as numerous passive components. Due to the complexity and custom nature of the installation, a precise valuation is difficult to determine. However, internal estimates conducted within the association indicate a total cost in the range of 80,000–100,000 PLN (approximately 19,000–23,500 EUR).

- **Electronic systems** — Similarly to the hydraulic and pneumatic subsystems, the valuation of the electronic infrastructure remains approximate because of the diversity of components involved. Based on internal cost assessments, the estimated value of the electronic systems is approximately 70,000 PLN (approximately 16,500 EUR).

Temporary costs. Temporary costs are associated with each engine ignition and test operation. These costs are related to the following aspects:

- **Cost of preparing the ignition system for a single test** – this includes the combined cost of electric matches and pyrotechnic materials. The cost of a single electric match is relatively low, at approximately 3 EUR. The cost of a single self-manufactured pyrotechnic igniter is approximately 5 EUR.

- **Technical gases** – in the case of propellants, the cost of technical gases depends on the total impulse of the engine (higher impulse corresponds to a greater amount of required propellant). Technical gases may also be used for pressurization or as working fluids in hydraulic subsystems. The costs of selected gases commonly used in experimental sounding rockets are presented in Table 1 as the cost per *kg*.

- **Logistics costs** – these costs are associated with transportation of the propulsion system infrastructure and all necessary test equipment. They include fuel costs (depending on the distance to the test site and the number of required vehicles), as well as alimentation costs for the test personnel.

- **Costs related to component wear** – during testing, hardware components are subjected to wear and degradation. This includes hydraulic components, engine elements exposed directly to combustion products, and electrical hardware affected by extreme environmental conditions such as high and low temperatures, rain, or snow. Additional costs may also arise from damage to the ground support equipment. Although these costs are difficult to estimate precisely, their occurrence should be taken into consideration.

Design Aspects Optimized in the Course of the Test Campaign

Having test data, there are many possibilities to optimize the following engine design parameters:

- **mass of rocket structure elements,**
- **mass of propellants,**
- **reliability of individual modules.**

Optimization of rocket elements mass

Thrust of engine is one of the most significant mechanical loads, affecting the structural elements of experimental sounding rockets. Based on the estimated value of that load, mass optimization is being done. Below is presented an example of process of mass optimization done by FEM method, with assumed values of thrust, and with value of thrust known after test. Below is an example of mass yield during thrust ring optimization using the FEM method.

Optimization of design

During the test campaign process, many design details used in the construction of the engine, ground support equipment, and engine structure are evaluated. The design of these elements can be verified during the campaign, as they are used in practice.

Some potential design flaws can only be identified when the components operate together in real conditions, not merely in theoretical CAD models. Since validation is carried out during the test campaign, design shortcomings can be revealed and eliminated.

Optimization of mass of propellants

During engine tests conducted on a test bench, analytically calculated values of thrust, pressure, and temperature are verified experimentally. Knowing the thrust and the engine burn time, the total impulse of the engine can be calculated.

Having precise knowledge of the total impulse provides detailed information on the maximum altitude the rocket can reach. This allows us to reduce the amount of propellant used eliminating the need to carry unnecessary excess propellant. As a result:

- the mass of the rocket during flight is reduced,
- the amount of consumed propellant is minimized, which lowers the cost of a single launch.

Some examples of such optimizations, which affect costs, rocket mass, and the functionality of the entire rocket, are presented in the following chapter.

Analysis of Cost Savings Resulting from the Test Campaign

The analysis presented below combines experimental data with engineering estimates used for the economic evaluation. Assumptions introduced for the cost assessment are identified where applicable.

Reduction of costs is related with:

1. **Reduction of propellant mass** – performing a test makes it possible to better estimate the amount of propellants used in the mission.
2. **Reduction of structural mass** – knowing the forces, pressures, and temperatures occurring during testing enables structural optimization from a mass perspective.
3. **Identification of potential reliability improvements** – tests can reveal weak points of the design and indicate areas requiring further development.

All types of reduction are described in detail below.

Reduction of propellant mass

Propellant mass is estimated during the design process based on flight simulations and the assumed total impulse value. However, the value measured during real tests I_{tm} may differ from theoretical impulse I_{tt} due to multiple phenomena not accounted for in the design process:

- Deviations in the mixture ratio from the nominal design value, caused by instabilities in the feed system or injector performance, which directly affect combustion temperature and efficiency (Sutton, G. P. and Biblarz, O., 2017).
- Incomplete atomization and mixing of propellants, particularly in liquid and hybrid engines, leading to reduced combustion efficiency and the presence of unburned species (Huzel, D. K. and Huang, D. H., 1992).
- Variations in chamber pressure relative to the design point, influencing thrust generation and overall performance (Sutton, G. P. and Biblarz, O., 2017).
- Nozzle-related losses, including non-ideal expansion (overexpanded or underexpanded flow) and possible flow separation, reducing effective exhaust velocity.
- Heat losses through the chamber and nozzle walls, lowering the available thermal energy for conversion into exhaust kinetic energy (Huzel, D. K. and Huang, D. H., 1992).
- Simplifications in thermodynamic and chemical modeling (e.g., ideal gas assumption, equilibrium chemistry), contributing to discrepancies between predicted and measured performance.
- Measurement uncertainties, including load cell calibration errors, pressure sensor drift, and signal processing limitations, affecting calculated impulse values (National Aeronautics and Space Administration, 1967). The magnitude of these uncertainties depends on the specific instrumentation used during a given

test campaign and should be considered when interpreting propulsion system performance results.

- Inaccurate estimation of consumed propellant mass, for example due to residuals in tanks or incomplete combustion (Sutton, G. P. and Biblarz, O., 2017).

- Transient effects during ignition and shutdown phases, altering the time-averaged thrust profile and integrated impulse.

- Combustion instabilities, influencing thrust fluctuations and effective performance (Huzel, D. K. and Huang, D. H., 1992). Differences between test and design ambient conditions, particularly ambient pressure and temperature, affecting nozzle expansion and measured thrust (Sutton, G. P. and Biblarz, O., 2017).

All the aforementioned factors can lead to a discrepancy in the measured total impulse I_{tm}. Two cases can be distinguished:

1. $I_{tm} > I_{tt}$ – this means that the rocket will reach a higher altitude than expected unless the engine is shut down at the correct time corresponding to the specified flight altitude. In order to estimate cost losses, it is assumed that the excessive impulse is not known prior to flight; therefore, the propellant tanks are fully filled. The loss is then associated with the excess mass of propellants Δm carried by the rocket:

$$\Delta m = m_{filled} - m_{req} \tag{3}$$

where:

m_{filled} – mass of propellants loaded into the tanks,

m_{req} – required mass of propellants to reach the target apogee.

In this case, Δm represents lost mass.

2. $I_{tm} < I_{tt}$ – this means that the rocket will reach a lower altitude than expected, and the estimated apogee will not be achieved. The cost of this issue is difficult to estimate and depends on the mission scope, payload cost, mission concept, and mission requirements. If reaching the specified apogee is mandatory, the entire flight may be considered a failure, and the associated cost corresponds to the total launch cost. Although no quantitative model is provided, it should be noted that these costs can be significantly higher than in the first case.

In Table 1, the costs of propellants are presented as the cost of technical-grade substances per $1kg$. These values represent averaged prices from Linde PLC (2026) and Air Liquide (2026) and from local suppliers who provide technical gases for PWr in Space.

Table 1: Cost of technical gases in euro

type of propellant	N_2O	H_2O_2	LO_2	LH_2	LCH_4	Ethanol
cost per kg in euro	12	15	8	20	10	7

To evaluate the economic benefits associated with propellant mass reduction, a simplified cost assessment was performed. Since the actual propulsion system performance may differ from theoretical predictions due to injector characteristics, combustion efficiency, hydraulic losses, and modelling simplifications, a conservative uncertainty of ±25% of the total impulse was assumed. For the analysed system, the maximum total impulse was estimated as $I_{tmax} = 10^6\ Ns$, corresponding to a potential propellant mass deviation of approximately $100kg$. Average market prices of technical-grade propellants were adopted, and only direct propellant costs were considered. Consequently, the obtained values should be treated as order-of-magnitude estimates intended to illustrate the economic impact of propellant mass savings. Based on these assumptions, the cost associated with the additional propellant required to compensate for the performance uncertainty was determined and is presented in Fig. 3.

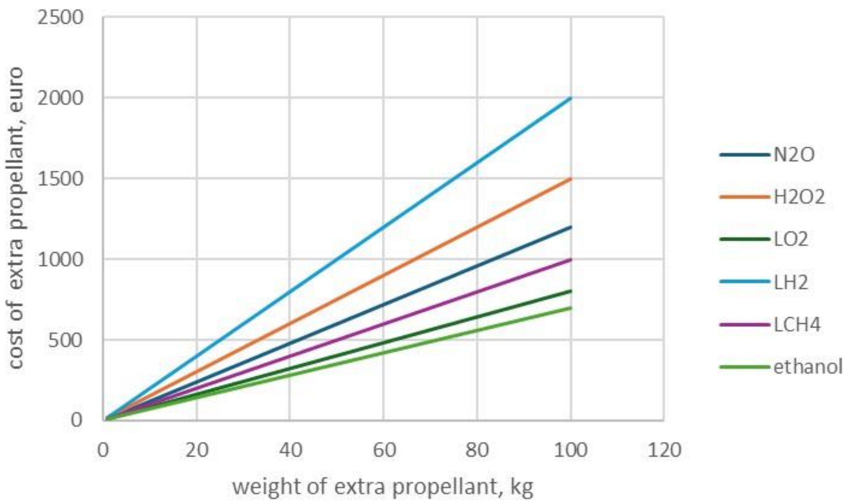

Fig 3. Cost of additional propellants

As shown in Fig. 3, for a single launch, these costs are not significant, even in the most pessimistic scenario. However, when the business model involves multiple flights, this cost increases, as it occurs in every launch.

Reduction of structural mass

Performing tests provides the exact value of thrust generated by the engine and may lead to decisions regarding adjustments of operating parameters in order to achieve optimal performance. Potential deviations in thrust compared to the theoretical value can influence the structural elements of the engine.

During the design phase, thrust is one of the most significant loads considered in static structural analysis. If this value differs after the testing phase, it may result in a situation where these analyses are not fully valid due to the use of non-representative loading conditions. Figure 4 presents selected structural elements from Engine C, developed by PWr in Space: the thrust ring (Fig. 4a) and the airframe cage (Fig. 4b).

(a) Thrust ring design

(b) Airframe cage design

Fig 4. Design of selected structural elements in Engine C

The structural optimization of the thrust ring and airframe cage was performed in Ansys Mechanical using the Finite Element Method (FEM). Linear static analyses were conducted assuming an Aluminium 6082 T6 material model and a tetra type mesh with an element size of 0.5mm. Engine thrust was applied as an axial load at the mounting interfaces, while the boundary conditions reflected the actual attachment locations within the propulsion system. The optimization objective was to minimize structural mass while maintaining a minimum safety factor of 2.0 according to the von Mises criterion. The resulting masses of the optimized components are summarized in Table 2.

Table 2: Structural element mass as a function of engine thrust load

value of thrust, N	thrust ring mass, g	airframe cages, g
2,800	142.1	375.2
2,500	124.6	340.5
2,300	114.3	305.4

For a single element, the difference is not significant; however, in the liquid-propelled rocket engine Nova, there are six airframe cages, which results in a total mass difference of over 400*g*, representing a significant value. Moreover, the mentioned components constitute only a subset of the structural elements in the overall rocket design affected by changes in thrust.

Design Improvements Identified During Testing

This type of cost optimization refers to identifying potential disadvantages of the design and provides areas for improvement. These improvements are demonstrated using examples from propulsion system tests conducted by the student research group PWr in Space. However, the types of potential improvements may vary and can include, for example:

- Improving the strength properties of engine structural elements,
- Detection of malfunctions or improper assembly,
- Improvement of operational procedures,
- Identification of potential failure modes that may be addressed before flight.

Case study - reduction of thrust on injector plate. The injector plate is a component responsible for injecting propellants into the combustion chamber in the appropriate quantity and direction. It should ensure proper atomization of the propellants. During the design of engine C, the geometry of the injector plate was modified as part of the development of a new engine iteration.
A comparison between two injector plate designs is shown in Fig. 5.

(a) Old injector plate design (b) New injector plate design

Fig 5. Comparison of injector designs in Engine C.

The design of the injector holes was modified: the number of holes was increased, while their diameter was reduced. In theory, this should improve the atomization of nitrous oxide during injection into the combustion chamber. However, static tests showed a decrease in engine thrust compared to tests conducted with the previous design. The observed thrust reduction was approximately 200 N, corresponding to about 8% of the engine thrust. Although the modification of the injector geometry was considered a possible contributor to this effect, the available data do not allow its influence to be quantified or confirmed. During this design phase, no CFD study of the injector plate was performed; therefore, experimental validation through testing was the only way this issue could be detected.

Case study - fuel grain manufacturing method. A hybrid rocket engine is a propulsion system in which one propellant is stored in solid form (typically the fuel), while the other is stored in liquid or gaseous form – typically the oxidizer (Sutton, G. P. and Biblarz, O., 2017). The fuel grain must be manufactured and then assembled before being placed in the combustion chamber. The initial design approach involved dividing the grain into three segments, casting each part, machining each segment, and subsequently bonding them into a single grain. However, this method required a more complex and expensive manufacturing process. The segments were still sufficiently long that it was necessary to use a steady rest on a lathe to reduce vibrations during machining.
Testing showed that the grain could be divided into a greater number of smaller segments. In the improved design, the grain consists of six segments, which are machined individually and then bonded together. This optimization reduced manufacturing costs by approximately 35%, resulting from a technological change in the design identified during the testing phase.

Summary

The performed analysis demonstrated that propulsion system testing is necessary due to limitations of analytical and numerical methods in predicting the real operating conditions of rocket engines. Experimental validation enables verification of propulsion system performance under realistic conditions and allows detection of phenomena that may be difficult to predict theoretically, such as combustion instabilities, hydraulic losses, injector-related issues, or deviations in thrust characteristics. The presented examples confirmed that test campaigns may significantly improve understanding of engine behaviour and provide valuable data for further optimization of propulsion systems. The paper presents several aspects of testing rocket engine including methods of rocket engine testing and measured parameters.

The paper also showed that propulsion system testing may contribute to optimization of multiple engineering aspects of sounding rockets. Experimental measurements of engine thrust and operating parameters allow reduction of excessive structural safety margins and optimization of structural element mass. Based on FEM analysis performed for selected structural components of Engine C, a reduction of several hundred grams in overall structural mass was demonstrated. Similarly, validation of propulsion system performance enables more accurate estimation of required propellant mass, reducing unnecessary propellant reserves carried during flight. This directly decreases launch mass and improves mission efficiency.

An important aspect presented in this study was the identification of potential subsystem reliability improvements through testing. The case study concerning injector plate redesign demonstrated that experimental testing may reveal design flaws that are not identified during the design phase or through simplified theoretical analysis. Additionally, the presented hybrid rocket fuel grain manufacturing case showed that testing may also influence technological and manufacturing processes. Dividing the fuel grain into a greater number of smaller segments reduced manufacturing complexity and lowered manufacturing costs by approximately 35%.

The economic analysis presented in this paper demonstrated that propulsion system testing generates both permanent and temporary costs associated with infrastructure, instrumentation, logistics, propellants, and hardware wear. However, despite these additional expenses, test campaigns may significantly reduce the overall cost of sounding rocket missions. Reduction of engineering uncertainty allows optimization of propulsion system design, reduction of unnecessary propellant usage, minimization of excessive structural mass, and reduction of mission-related technical risks. Consequently, propulsion system testing should be treated not only as a validation procedure, but also as an important engineering and economic optimization tool in the development of experimental sounding rockets.

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