CZECH TECHNICAL UNIVERSITY IN PRAGUE

FACULTY OF TRANSPORTATION

Department of Air Transport

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HYBRID ROCKET PROPULSION

Diploma Thesis

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ČESKÉ VYSOKÉ UČENÍ TECHNICKÉ V PRAZE
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K621...**Ústav letecké dopravy**

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- Přehled současného stavu problematiky
- Výhody a nevýhody hybridního pohonu
- Popis konkrétního motoru
- Příklady možného využití
- Srovnání s konvenčním pohonem (ekonomie)

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Rozsah průvodní zprávy: minimálně 55 stran textu (včetně obrázků, grafů a tabulek, které jsou součástí průvodní zprávy)

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Rhee, Lee, Lee: Optimal Design For Hybrid Rocket Engine For Air Launch Vehicle

Bondarchuk, Vorozhtsov, Zhukov, Borisov: Regularities of Physical Processes in Hybrid Solid-Propellant Rocket

Vedoucí diplomové práce: **Ing. Bc. Jakub Hospodka, Ph.D.**

doc. Ing. Martin Bugaj, Ph.D.

Datum zadání diplomové práce: **30. června 2015**
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MASTER'S THESIS ASSIGNMENT

(PROJECT, WORK OF ART)

Student's name and surname (including degrees):

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Guides for elaboration

During the elaboration of the master's thesis follow the outline below:

- Current State Overview
- Pros and Cons of Hybrid Propulsion
- Description of a Specific Engine
- Examples of Possible Use
- Comparison with Conventional Propulsion (Economics)

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Graphical work range: according to the instructions of the mateŕs thesis supervisor

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Davydenko, Gollender, Gubertov, Mironov, Volkov: "Hybrid Rocket Engines: The Benefits And Prospects"

Rhee, Lee, Lee: Optimal Design For Hybrid Rocket Engine For Air Launch Vehicle

Bondarchuk, Vorozhtsov, Zhukov, Borisov: Regularities of Physical Processes in Hybrid Solid-Propellant Rocket

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b) in case of postponing the submission of the thesis, next submission date results from the recommended time schedule

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I confirm assumption of master's thesis assignment.

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Student's name and signature

Prague .... December 30, 2016

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Čestné prohlášení

Prohlašuji, že jsem předloženou práci vypracoval samostatně a že jsem uvedl veškeré použité informační zdroje v souladu s Metodickým pokynem o etické přípravě vysokoškolských závěrečných prací.

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V Praze dne 29. 5. 2017

……………………………………

Tomáš Cáp

Abstrakt

Autor: Bc. Tomáš Cáp

Název práce: Hybrid Rocket Propulsion

Škola: České vysoké učení technické v Praze, Fakulta dopravní

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Vedoucí práce: doc. Ing. Jakub Hospodka, PhD.

Klíčová slova: hybridní raketový motor, raketový pohon, raketové palivo, komerční lety do vesmíru

Cílem této diplomové práce je představení hybridního raketového pohonu coby perspektivní technologie, která do budoucna zřejmě výrazně ovlivní směřování kosmonautiky. Součástí práce je stručný popis konvenčních raketových motorů na tuhá a kapalná paliva a základní popis problematiky hybridních raketových motorů. V druhé části práce jsou zmíněny přibližné náklady na provoz těchto systémů a je provedeno srovnání těchto finančních nákladů v poměru k poskytovanému výkonu. V závěru jsou doporučeny oblasti možných využití vhodné pro hybridní raketové motory vzhledem k současnému stupni jejich technologické vyspělosti.

Abstract

Author: Bc. Tomáš Cáp

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The aim of this master's thesis is to introduce the hybrid rocket propulsion as a perspective technology that is likely to significantly influence the aerospace industry in the coming years. In the first part of the thesis, conventional, solid- and liquid-fuel rocket engines are briefly described, as well as basic processes and considerations for the hybrid rocket engines. In the second part of the paper, operating costs of specific systems from the different rocket engine families are determined and compared in relation to their performance. The thesis is concluded with recommandations for possible applications of hybrid propulsion with regard to their current level of technological maturity.

Contents

List of Acronyms

ADN - ammonium dinitramide

ADS-B - automatic dependent surveillance – broadcast

AP - ammonium perchlorate

APCP - ammonium perchlorate composite propellant

ATC - air traffic control

CEO - chief executive officer

CSP - Controllable Solid Propulsion

CTN - casing, throat and nozzle

GNSS - global navigation satellite systém

HNF - hydrazinium nitroformate

HRE - hybrid rocket engine

HTHP - high test hydrogen peroxide

HTPB - hydroxyl terminated polybutadiene

LH_2 - liquid hydrogen

LOX - liquid oxygen

LRE - Liquid Rocket Engine

MPS - moteur propergol solide, solid rocket booster

N_2O - nitrous oxide

NASA - National Aeronautics and Space Administration

OMS - Orbital Maneuvering System

RCS - Reactive Control System

RDX - cyclonite

RNAV/RNP - area navigation / required navigational performance

SABRE - synergistic air-breathing rocket engine

SESAR - Single European Sky ATM Research

SRB - solid rocket booster

SRM - solid rocket motor

SSTO - single stage to orbit

USD - United States Dollar

WWII - World War II

USG - US gallon

1 Introduction

There is hardly a more significant example of how far has the mankind come than all the achievements that have been made at the edge of the Earth's atmosphere and beyond. In as little as 15 years, man has progressed from successfully launching the very first artifical satellite to Earth's orbit to walking on the surface of the Moon. Spaceflight is a symbol of mankind's hopes and dreams, as well as a testament to its tenacity and its will and ability to overcome difficult technical challenges.

Apart from all this, it is also one of the most difficult scientific disciplines which only a select few get to comprehend and even fewer can actually personally experience. Rocket scientists and astronauts are an exclusive club – procedures that are ridiculously simple while dealt with on the surface of Earth are becoming unbelievably complicated under space conditions, and it does not really matter if it is one of numerous technical aspects of rocket propulsion, such as controlling the flow of liquid oxidizer and directing it to the combustion chamber, or something so mundane as sleeping and performing personal hygiene during a mission to the International Space Station.

The aim of this thesis is to introduce devices that have the potential to unlock even more of the universe to human exploration. Hybrid rocket engines are a concept that has been known for quite some time, in fact for as long as almost 90 years. Due to specifics of its internal ballistics, however, only now are we starting to truly understand it – and it will still take some time to fully comprehend all its perks and peculiarities.

The humble aim of this thesis is to make the process of basic familiarization with hybrid rocket propulsion somewhat easier. Main goals of this thesis are to map the current landscape in the field of rocket propulsion, summarize both positive and negative aspects associated with the utilization of hybrid rocket engines in spaceflight applications, offer practical examples of currently active HREs propose a field of their effective application along with associated financial aspects.

In chapter 2, conventional rocket engines will be first introduced to provide a benchmark for subsequent comparisons. Each of the systems is described in terms of its main parts, its propellants and present applications. Additionally, briefly explored is the potential for the systems' future development. Chapter 3 is dedicated to hybrid rocket engines, specifically presenting baseline differences from the legacy propulsion systems, specifics of their internal processes and considerations for their efficient and satisfactory operation. Chapter 4 is intended to be a practical part of this thesis, containing data with regard to efficiency and financial aspects of operation of the respective rocket engines. Finally, the results will be briefly assessed in a brief conclusion.

The results I expect to receive should confirm high measure of efficiency of hybrid rocket engines which should prove as the most cost-effective solution for powering launch vehicles for broad scale of applications, ranging from sub-orbital flights to light and possibly medium-class launchers. Moreover, I expect to discover ways to keep increasing the performance of hybrid rocket engines in the future, so that they would eventually become competitive even in the heavy launcher class.

I would also like to explore the ways in which hybrid rocket engines would benefit not only the spaceflight applications, but also global aviation industry.

2 Conventional Rocket Engines

In the first part of this thesis, I provide a general overview of conventional rocket propulsion technology that has dominated spaceflight since 1950s [1]. It is crucial to first understand the key advantages and issues associated with these legacy systems, which still play dominant roles in propelling the mankind to Earth's orbit and beyond, to be able to comprehend both the improvements and setbacks that are introduced by hybrid engines. For both solid rocket motors and liquid rocket engines, this chapter offers basic overview of key components and systems as well as pros and cons of each from the perspective of their operational use. Included are also perspectives on future use of these legacy systems which still remain highly competitive owing to ongoing research and development of new systems and solutions.

2.1 Solid Rocket Motors

Solid rocket motor (further referred to as SRM) is a type of rocket engine which utilizes solid-state fuels and oxidizers. It is an oldest form of rocket propulsion which has been used since as early as the 13th century and also the simplest one, with SRMs generally not having the capability of adjusting the thrust or stopping and subsequently restarting its operation. In modern spaceflight applications, SRMs are mostly being utilized as booster and auxiliary rockets. However, a few space vehicles have used SRMs as their main means of propulsion, such as US-developed Scout (displayed in Figure 1 below) and Athena rockets, Russian Start-1, Indian PSLV or European Vega). [2]

2.1.1 General Description

Main parts of an SRM are the casing, nozzle, fuel grain and fuel igniter. The whole casing acts as a combustion chamber and is entirely filled with fuel.

While the simplest rockets burn fuel from the bottom, high-performance SRMs utilize sophisticated fuel grain geometries and differently shaped ducts in order to optimize the burn area. Therefore, the fuel can burn simultaneously, throughout its volume, maximizing the thrust or even enabling a certain level of control over the burn.

Figure 1 - Scout launch vehicle [48]

The fuel mix is a composition of an oxidizer, fuel and other additives and under normal circumstances acts as a solid element. After the ignition, the burning fuel produces a great amount of exhaust. The gas exits the combustion chamber through a nozzle which is specially shaped to maintain the operating pressure inside the chamber.

Simple SRMs cannot be shut off or reignited, nor can the thrust be regulated in any way. Once ignited, they burn until all fuel is consumed. More advanced designs are able to regulate the thrust by auxiliary openings which vent the gas away from the nozzle or by specifically designed fuel geometry in order to achieve different levels of thrust during specific phases of flight. Some modern engines can even be stopped or restarted. To be able to do this, the modern systems employ either a combustion chamber extinguishing system, or they are assembled out of numerous consequently ignited segments (e.g. Space Shuttle SRBs which can be seen on Figure 2). [2] [3]

Figure 2 - Space Shuttle Columbia, assigned to mission STS-1, moments after lift-off from Pad 39A, Cape Canaveral, Florida on April 12th, 1981. The two SRBs (Solid Rocket Boosters) are attached to the large external fuel tank, one on each side. [48]

Modern solid propulsion systems can also utilize additional devices, e.g. thrust vectoring systems, guidance systems, auto-destruction systems, chutes, smaller auxiliary engines etc. Reasons and methods of their use, however, lay beyond the scope of this paper.

2.1.2 Key Elements of SRM Design

With a certain measure of simplification, a key prerequisite for the SRM design is the total impulse I_{sp} required for the vehicle to perform satisfactorily in the intended application. Based on the specific impulse and the propellant consumption (which is also known as a result of numerous calculations corroborated by experimental firings), the amount of fuel and oxidizer needed to fulfill the task the engine is designed for is implied. With this knowledge, first estimate of the SRM's dimensions can be safely made, as well as the selection of propellant and fuel grain set-up.

Main parameters of an SRM are:

- Type of propellant
- Fuel grain geometry

- Nozzle shape
- Casing resiliency

In following paragraphs, these components will be described in more detail. Generally, these elements need to be designed with a vision of the intended application in mind in order to enable them to function in a way that is acceptable.

Large part of that is to design an environment where the fuel combustion will be stable and predictable. Two main parameters that need to be figured out to achieve such a burn are the surface area of the fuel grain and the desired chamber pressure. The latter depends on the nozzle set-up (geometry, material, thermal protection) and on the fuel grain's burn rate and must not exceed the bounds given by the casing material. Additional important factor that is implied by the very application the SRM is being designed for is the required burn time which directly depends on layer thickness and overall amount of fuel [2].

Fuel

Fuel burns on its surface and burns through continually. Therefore, the fuel geometry which determines the burning surface area at any given time plays an important role in overall performance of the engine. The fuel surface's shape is changing over time and therefore the mass flow is a function of fuel density ρ, immediate burning gas surface A_s and linear speed of burn of specific fuel b_r as shown below:

$$Q_m = \rho \cdot A_s \cdot b_r$$

Generally, a number of geometric configurations are used, for instance [2]:

- Circular bore
- C-slot
- Moon burner
- Star

While different combinations of fuel and oxidizer may be used based on different requirements for the specific impulse, the most common is a substance based on polybutadiene binder, ammonium perchlorate (AP) oxidizer and aluminum additive [4].

Naturally, new advancements in material science are also making their way to the SRM design processes and one of the promising venues in the mid-term scope is the use of chlorine-free oxidizers such as ADN (ammonium dinitramide, $NH_4N(NO_2)_2$), a material very similarly sensitive to RDX (cyclonite, $(O_2NNCH_2)_3$). The replacement of AP with ADN results in increase of specific impulse but at the same time propellant density decreases for only a small density-impulse gain. It is quite promising when used with high strength composite casings, however.

Another approach utilizes HNF (hydrazinium nitroformate, $CH_5O_6N_5$). It is friction sensitive with moderate impact sensitivity. The product is needle-shaped, therefore crystallization is necessary before manufacturing process of the fuel grain can be started. [5]

Casing

The casing can be manufactured from a wide range of different materials. For spaceflight applications, the materials of choice usually include aluminum alloys or steel. Modern approaches utilize carbon-fibre composite materials and epoxy resin. The casing must by sufficiently resilient to withstand significant pressure (1-10 MPa dependent on material and application) and high temperatures. Therefore, from a construction point of view, casings are designed as pressure vessels. To protect the casing from the effects of high temperature and corrosive effect of the exhaust, ablative coating is often applied to the inner side of the casing. Important point is that all elements within the casing must be properly sealed (e.g. O-rings) [2].

With recent technological advances, new engines have almost no bounds in terms of diameters or lengths. While manufacturing a steel case measuring 6.60 m in diameter used to be a challenge in 1960s, today's experience with production of large aircraft, e.g. Boeing 787's full barrel fuselage, prove that production of very large composite rocket engine casings is not only possible but in fact quite feasible [5].

As the capabilities of building high-strength composite motor casings steadily grow, so does the possibility of designing SRMs capable to work with heightened operating pressures. If we allow that optimum pressure for a metallic case is approximately 6-7 MPa, similar engines built from filament wound carbon fiber composite may operate in the range of 8-9 MPa. In case that these cases can be built as monolithic structures, they are able to withstand 9-10 MPa. This proves especially crucial while the SRM is running at ground level. With the diameter of exit cone being an inherent limitation, the only way to increase the specific impulse is the increase in combustion pressure.

Another promising area is the use of nano-technologies. They hold a great potential of enhancements of mechanical properties in both metals and composites. Their main advantages lie in improvements of fracture toughness as well as thermal conductivity of composite materials leading to more responsive thermal control [5].

Nozzle

Nozzle of an SRM is similar to the one used in liquid fuel rocket engines. It utilizes a shape of a de Lavall nozzle. Unlike the liquid-fueled engine, an SRM nozzle cannot be cooled regeneratively (by flow of fuel or other coolant through ducts in the nozzle walls) and therefore the nozzle must withstand high temperatures. In order to achieve the required thermal resistance, the nozzle is usually made of graphite or carbon laminate. Some SRMs may have a thrust vectoring system (e.g. Space Shuttle SRBs). Usually, these systems employ flex-seals and external actuators. With such set-up, the nozzle can rotate to an extent of 5-6 degrees in all directions. Some special designs may reach even more, 15-20 degrees. Alternatively, Liquid Injection Thrust Vectoring system can be used. This system injects liquid to the rocket engine exhaust. As it gets vaporized, it locally increases the thrust and offers a control momentum.

Convective heating typically reaches the maximum at the nozzle throat (converging – throat – diverging geometry) and therefore this is the area where heat resistant materials are employed. Typically used are carbon-carbon composite thermostructures, graphite or graphite phenolic composites. In the divergent section, ablative materials are used, such as carbon fiber or silica fiber reinforced plastic [5].

Since carbon/carbon composites and carbon fiber reinforced plastics possess different recession rates, discountinuities („steps") in the nozzle surface may occur. This may render the engine prone to failures as the flow may be disturbed and eventually grooves may form down the stream. This makes careful contour design and material assessments a must. Modern computerized fluid dynamic calculations combined with advanced material modelling are capable of such optimization that these problems are no longer a factor [5].

2.1.3 Performance and Use

Modern SRM with a high-energetic fuel mix may achieve a vacuum specific impulse of up to 286 seconds as compared to liquid fuel engines running on oxygen and RP-1 with vacuum I_{sp} reaching 339 s and liquid oxygen with liquid hydrogen with vacuum I_{sp} as high as 452 s. Apparently, SRMs are not as efficient and their performance is lacking in comparison to liquid fuel rocket engines. However, their significantly lower costs may render them useful in certain applications. In spaceflight applications, they are mostly being used as auxiliary lift-off engines (e.g. the Space Shuttle Solid Rocket Boosters), providing power in the Mega Newton range. Under certain circumstances, they have also been successfully used in the final phases of guidance to the orbit for some types of payload (satellites) [2][6].

Thanks to the possibility of long-term storage and immediate deployment, they have also a broad range of military applications. However, these applications lay beyond the scope of this thesis.

2.1.4 Perspective of Future Use

In the last 60 years, solid propulsion has been used in a number of launch vehicles for its properties that provided their designers with plenty of performance at a very competitive price and with relatively low risks. The ability to easily store SRMs also meant that the engines were capable of almost instant use, making them originally the first choice for military applications from which some of the first space launchers were initially converted (for instance Juno I, the first US launcher, being developed from the Redstone ballistic missile and the first Soviet Sputnik rocket being derived from the R-7 intercontinental ballistic missile [7]). With ever-changing technical capabilities, challenges and priorities in space exploration, however, comes a need for constant modernization. Thus, there is a need for mid- to long-term visions and roadmaps that pave the way to the future of space exploration.

These documents allow us to understand how SRMs might evolve well into 2030s [5]. Primarily, the use of modern, high-strength composite materials used in small launch vehicles (e.g. European Vega launcher [8]) and strap-on boosters has enabled them to achieve considerably higher levels of thrust and, by definition, specific impulses. Case in point, composite filament casings for SRMs proved to withstand 5 times higher operating pressures when compared to the metallic ones. This allows for significant mass reductions and consequently also results in higher specific impulses.

[9] reveals the performance boost in Ariane 5 MPSs (solid rocket boosters) permitted by new composite casings as shown in Table 1.

Table 1 - Comparison between legacy Ariane 5 MPS and composite case upgraded version

	Legacy MPS	Upgraded MPS
M_p	237.7	247.8
M_i	30.8	27.5
I_{sp}	275.3	280
A_e	6.99	6.99
T_b	128	125
P_{max}	6.1	8.5

Figure 3 below [5] represents a mass breakdown for a typical SRM enclosed in a filament wound case based on medium-performance carbon. Interestingly, there is potential for further decrease in the inert mass of the booster, leading to additional performance increase – especially in case that high-strength fibers (e.g. treated with nano-technologies) become available. Additionally, implementation of lower density insulation or improved fuel grain production proces might also contribute to further optimizations.

All these improvements could result in as much as 20 – 30 percent of increase in payload massjust based on the inert mass optimization.

From the financial and procedural viewpoints, while the use of metallic hardware is well established, the composite solution is still a fairly new one. Even though today's prices already are competitive, predictions show that the performance-price ratio is yet to decrease to offer even more favorable economical conditions [5].

As an example, the Atlas V SRBs manufactured by Aerojet offers a vacuum thrust of over 1160 kN and weighs slightly over 45 tons. 87 SRBs have successfully flown on the Atlas V as of May 2017 with 100% success rate since the first flight in August 2002. The average market cost of a single Atlas V SRB is 6.8 million USD, while the launch costs of the entire Atlas V system range between 109 and 153 million USD [10]. Future advancements in material science and high-performance composites in particular may press these prices further down and make the launch costs more affordable.

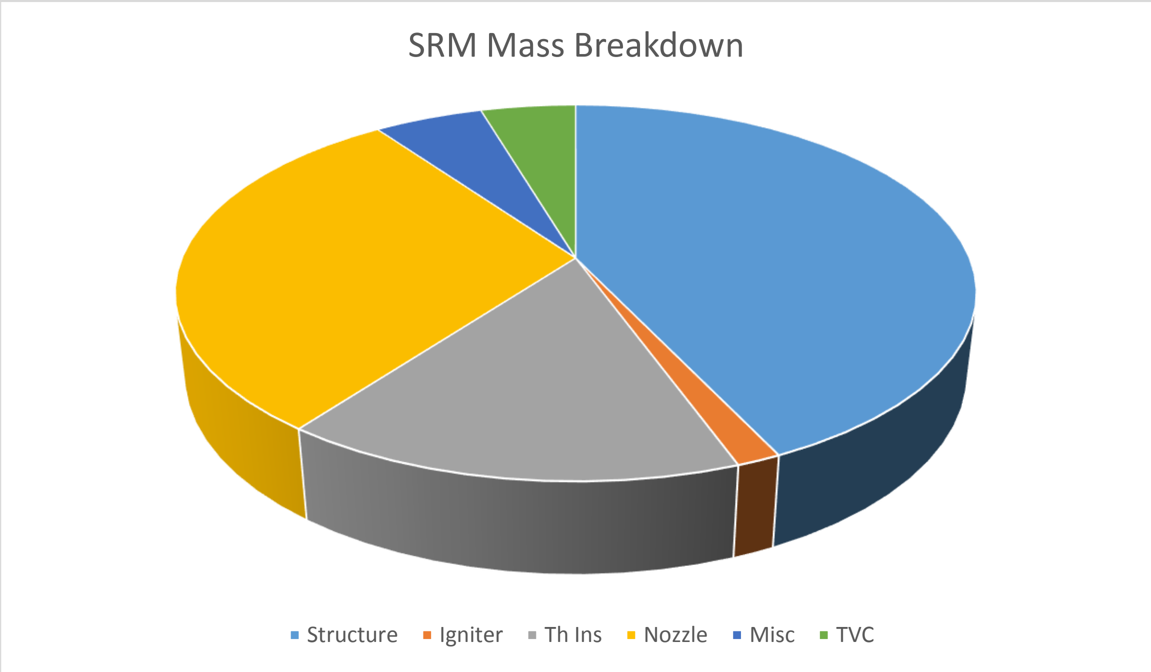

Figure 3 - Mass breakdown for an advanced 90-ton class SRM. [5]

In case that the casing can be processed as a monolithic structure, filament wound carbon presents large advantage over steel from the inert mass point of view. However, if this is not possible and the casing has to be manufactured in segments, the joints that are necessary to connect the individual parts constitute a penalty in terms of mass and by association, costs. To make matters worse, specific stiffness is required in certain parts of the construction (launcher attachment points, propellant bonding etc.), which may be difficult to achieve. However, even if these construction details remain difficult parts of every SRM design, present technology already offers reliable solutions to overcome these issues [5].

2.1.5 Controllable Solid Propulsion (CSP)

CSP is a concept that is set to negotiate one of the key disadvantages of solid propulsion and possibly redefine its field of application. With Aerojet Rocketdyne being a key player developing and promoting this technology, their team is now looking at a possibility of employing the concept for stabilizing the crew/payload capsule in both manned and unmanned vehicles before touching down, among other uses. Especially for crewed missions, there always is a measure of unpredictability in atmospheric conditions during reentry and landing. CSP technology might be used to neutralize issues associated with high winds and strong gusts. Therefore, employing CSP stabilization systems might contribute to mature, low-risk approach to increase safety during critical phase of every mission and to ensure the safe return of crew and/or payload back to Earth [5].

Currently, the whole concept is in the stage of testing with multiple ground test firings already successfully completed. The system is being prepared for implementation with NASA's new Orion vehicle that should carry out its first crewed mission in August 2021 [11] [12].

2.2 Liquid Rocket Engines

A liquid rocket engine is a workhorse of spaceflight missions of the last 60 years. While successfully negotiating the main disadvantages of SRMs, it has been plagued by issues of its own, mostly associated with volatile, unstable nature of liquid propellants and mechanical complexity that add greatly to the launcher's mass as well as costs and length of preparations before launch. For decades, space missions were exclusively carried out by governmental organizations only, for both financial as well as political (and military) reasons. In last 20 years, however, private companies have emerged that declared a goal to reach space on their own and make it accessible for everyone. Today, we already know that we are fortunate enough to live in an era when these pledges and dreams are slowly coming true. There are companies that are successfully carrying out tests of new propulsion systems as well as entire launch vehicles and day by day, they are getting closer to making spaceflight available to public. Even NASA is now using private contractors to deliver supplies and cargo to the International Space Station and crewed flights on commercial spacecraft will follow soon.

Thus, we can say that spaceflight is entering a truly commercial age. This has a wide scale of implications, with one of them being the ever-present aim of increasing reliability and driving down costs. This tendency manifests in a host of creative solutions, such as the recent high-profile introduction of reusable first stages of launch vehicles which is becoming more feasible every day.

2.2.1 General Description

A liquid rocket engine (further referred to merely as LRE) is a device that converts the chemical energy of liquid-state fuel and oxidizer into kinetic energy, or thrust. These engines have been used extensively for all kinds of spaceflight applications as they provide a significantly higher specific impulse compared to an SRM. There are multiple criteria of discriminating among different kinds of LREs, the most common being either monopropellant LREs utilizing a single propellant (typically hydrazine) or bipropellant engines running on a combination of fuel and oxidizer. As the use of hydrazine as a propellant is associated with significant environmental hazards and risk for the support personnel due to its severe toxicity and as most of present-day launchers use either the combination of liquid oxygen for oxidizer and either liquid hydrogen or RP-1 (highly refined form of kerosene), I will adhere to the description of bipropellant LREs for the sake of practicality. [13]

2.2.2 Transport of Fuel and Oxidizer

As opposed to SRMs where the propellant is stored directly in the combustion chamber, this is not possible in LREs as the propellants and oxidizers are often too unstable and volatile substances that must be stored separately. Therefore, the need for transfer of both the fuel and the oxidizer to the combustion chamber arises, with a key requirement being that this transfer is speedy and is taking place under a significant pressure. Since as early as WWII, this task has been carried out by turbopumps. Their parameters are defined based on operating requirements for the engine and key properties of the selected propellants, with the main ones being density, viscosity and boiling point. Generally, the lower the propellant density (liquid hydrogen is a typical example), the sturdier and more robust have the turbopumps to be in order to provide sufficient mass flux. This, of course, carries a significant mass penalty to the launcher. However, simpler engines may not need turbopumps and utilize different methods of propellant transfer instead. According to the principle employed in the solution of propellant transfer, we talk about different LRE operation cycles [13]. The four major types of cycles are as follows:

Pressure-fed Cycle

Fuel and oxidizer are being pushed to the combustion chamber by a pressure of inert gas (usually helium) from a separate tank. The gas is being warmed by the combustion chamber. This is a fairly simple method which often does not require turbopumps. Usually, engines utilizing pressure cycle are capable of a repeated reignition. Pressure-fed cycled LREs may be controlled simply by opening or closing the pressurizing gas valve. These are controlled by either electromagentic or pneumatic means. The ingition is commonly carried out separately (hypergolic propellants) or by an igniting agent.

Pressure cycle has been notably used in maneuvering engines of the OMS (Orbital Maneuvering System) of the Space Shuttle orbiters or in the RCS (Reactive Control System) of the Apollo Command/Service module [14].

Expander Cycle

Expander cycle is such where the propellant is first used as a coolant running through cooling shafts in the combustion chamber and nozzle walls. The fuel expands due to the absorbed heat and before entering the combustion chamber, it is used to spin a turbine which in turn powers the turbopumps. The expander cycle requires use of cryogenic fuel which easily achieves the boiling point, e.g. hydrogen, methane or propane. While using a conventional bell-type nozzle, the thrust is limited to approximately 300 kN as the quantity of heat available to expand and vaporize the fuel is limited. [13][14]

Gas Generator Cycle

Gas generator cycle (also called the open cycle) is one of the most commonly used methods of powering the turbopump system. A small portion of fuel along with a certain amount of oxidizer is combusted in a pre-burner turbine which powers the fuel transfer turbopumps and then expelled through either the main nozzle, or a separate, smaller one. The main disadvantage of this system is the reduced propulsion efficiency due to having to use up a certain portion of propellants which generate little to no thrust. However, the energy generated by the pre-burner is sufficient for large, powerful turbopumps which allows for high power engines.

Staged Combustion Cycle

Staged combustion (or closed) cycle is the most complicated and advanced concept of an LRE. Its principle is similar to that of the open cycle, but the difference lies in the fact that the entire propellant stock transits through the gas generator in either a fuel-rich or an oxidizer-rich mixture. From the gas generator which powers the turbopumps, the high-pressure exhaust is routed to the combustion chamber where the unspent propellants are combusted. This allows for very high pressure and excess of thrust, making it a very efficient solution. Major disadvantage is the fast amortization of turbine and the necessity for an excessively complex fuel injection system. [13][14]

2.2.3 Injection System

The injection system serves as means of transfer of fuel and oxidizer to the combustion chamber. Fundamentally, it is a set of small-diameter holes (nozzles) shaped in such a way that the fuel mix is dispersed in a specific manner permiting efficient combustion to take place. The first injectors used in German A-4/V-2 rockets (presented in Figure 4 below) during WWII were merely parallel nozzles which were not able to sufficiently mix the fuel with the oxidizer. As a result, the efficiency of the first rocket engines was poor. However, modern injection systems consist of several nozzles targeted into specific points in the combustion chamber for achieving optimal mixture of the propellants and atomization to small droplets which perform significantly better as opposed to larger propellant particles. [14]

Main Types of Injectors

- Shower head
- Self-impinging Doublet
- Cross-impinging Triplet
- Centripetal / Swirling
- Pintle

Figure 4 - A-4/V-2 rocket replica. Author: AEIfwine, fr.wikipedia.org [48]

Stability of Combustion

Combustion chamber instability may manifest in a number of different ways ranging from light vibration through engine choking to strong shocks which can lead to engine failure. In order to prevent them, it is necessary to secure a sufficient pressurization of the injector, normally about 20% higher than the combustion chamber pressure. However, in large engines with fast combustion, there is a tendency for strong vibrations that have not been plausibly explained. These high-frequency vibrations tend to spoil the airflow around the engine and the flow of coolant in the cooling shafts of the nozzle which can lead to failure of the cooling system and eventually to the vehicle's destruction. [14]

2.2.4 Cooling System

In order to achieve high propulsion efficiency, an extremely high exhaust temperature is required (up to 3500 K). In other words, a temperature far above melting point of most common materials except graphite and wolfram. Therefore, to ensure safe and efficient operation of the nozzle, a robust cooling system must be in place. In case adequate cooling system is employed, normal materials such as steel, aluminium, copper and nickel alloys may be used.

Regenerative Cooling

The principle lies in the flow of fuel through shafts in the combustion chamber or nozzle walls. Required cooling output is significant, up to 20 MW per cubic metre. The amount of absorbed heat depends on multiple circumstances.

- Chamber vs coolant temperature difference
- Thermal conductivity
- Heat transfer coefficient
- Fuel flow velocity
- Exhaust flow velocity in chamber or nozzle [14]

Ablative Cooling

Usually used as a reentry protection of landing modules. Some engines utilize it to cool the nozzle or the combustion chamber. Ablative cooling works on the principle of subsequent vaporization of layers material from the walls of the combustion chamber or the nozzle. [14]

Other Methods

- Radiative cooling
- Curtain cooling
- Film cooling [15]

2.2.5 Propellant Types

In present applications, quite a significant number of different propellants are used. The most frequent ones are bipropellants as a fuel for main engines because of their high effectivity. Monopropellants are usually used for simple maneuvering engines. Monopropellants require a catalyst to operate (e.g. iridium).

As a basic means of categorizing the propellants, we recognize whether they have to be supercooled in order to function (cryogenic or semi-cryogenic propellants) or whether they are capable of storage and operation at normal temperatures and pressures (storable propellants).

CRYOGENIC PROPELLANTS

LOX and LH_2

LREs utilizing the combination of LOX (liquid oxygen) and LH_2 (liquid hydrogen) for propellants have been researched since as early as late 1900s, only a few years after the very first droplets of liquid hydrogen were obtained in laboratory conditions. It took more than 50 years, however, until the concept became not only feasible, but also practical enough for the first LOX/LH_2 upper stage to be introduced (CENTAUR). Subsequent research which brought a better understanding of the use of liquid hydrogen as well as its thermodynamic properties then led to a start of wider development of use of cryogenic engines, with first example being RL10. Today, most launchers use the combination of LOX and LH2 in some of their upper stages. Notable example of a launcher powered exclusively with this combination of propellants is the US-built Delta IV (shown in Figure 5).

In early 1960s, development of cryogenic engines has been kickstarted in Russia and also in France, leading to introduction of RD56 and RD57 engines and HM4 engines, respectively. [16]

Liquid hydrogen provides the highest specific impulse of all currently used propellants and its energetic density is also the highest of all rocket fuels – 143 MJ/kg, which is as much as a 40% increment compared to other propellants. Additionally, it generates zero emissions as the only product of its combustion is water vapour. In case of a leak, it is not toxic and does not contitute an environmental threat (unlike hydrazine). [14]

Figure 5 - Delta IV [48]

However, there are also disadvantages that limit the use of liquid hydrogen. One of the most important ones is the costly production and problematic storing. To produce liquid hydrogen, it is necessary to achieve high pressure and to supercool it to a temperature of 20 K (-252,87°C). This bears a problem of high energetic demand. The storage tanks must be very well isolated and even then, it is extremely complicated to maintain the required temperature. Liquid hydrogen's density is only 70.99 kilograms per cubic meter, which means that large tanks are necessary as well as larger and stronger turbopumps to succeed at transporting large enough quantities to the combustion chamber. The combined mass of larger tanks, their isolation and turbopumps is a significant limiting factor for an engine's effectivity and therefore makes it impossible to fully utilize its advantages. Other risk lies in leaks and vaporization which normally reach up to 1% of volume per day. To store and manipulate liquid hydrogen, all the regulations and limitations as for regular hydrogen are still in place. All these drawbacks significantly increase the overall launch costs of LOX/LH_2 powered vehicles. [14]

SEMI-CRYOGENIC PROPELLANTS

LOX and RP-1 (kerosene)

RP-1 is a highly refined form of classic petrol. It does not reach such efficiency as liquid hydrogen, but thanks to undemanding manipulation, low price and the possibility of easy storage, it is widely used as a first stage fuel. Other advantages include much higher density and boiling point which eliminate problems with vaporization and enormous size of turbopumps. From ecological point of view, RP-1 combustion is somewhat “dirtier“ because burn products include carbon monoxide and other toxic products. Also, the very refining process presents an environmental burden. However, RP-1 is not nearly as toxical as hydrazine.

The main disadvantage of RP-1 lies in its proportionally lower efficiency. Other problems are tied with the above-mentioned presence of carbon monoxide in the exhausts as its molecules are relatively heavier and cause a decrease in an effective exhaust speed and, by extension, in specific impulse. High boiling point and low pressure of its fumes requires an additional system for volume and tank pressure recharge. Unburned RP-1 may polymerize and carbonize when in contact with hot engine parts and therefore create deposits, which may in turn cause a malfunction. This problem is present mainly in ground tests and also with engine restarts. [14]

STORABLE PROPELLANTS

Dinitrogen Tetroxide (N_2O_4) and hydrazine (N_2H_4)

Both fluids can be stored in reasonable temperatures and pressures. Their use is mainly military and in long-mission satellites. The fluids are hypergolic (they combust when in mutual contact). Hydrazine constitutes a serious environmental and health hazard. [14]

2.2.6 Advantages

The most obvious advantage and a reason of such broad use of LREs is that these engines provide high thrust-mass ratios (e.g. thrust-mass ratio of Soviet/Russian NK-33, shown in Figure 6 below, is as much as 133:1). Density of liquid propellants is similar to that of water, around 700-1400 kilograms per cubic meter (except for liquid hydrogen with dramatically lower density). This makes for relatively lower mass of propellants when compared to SRMs and as pressurization required to preclude autonomous vaporization is relatively low, the tanks in which the propellants are stored can be rather thin and lightweight. Even for dense elements such as kerosene, the fuel tank mass contitutes mere 1 % of overall mass. In case liquid hydrogen is used for fuel, the fuel tank accounts for 10% of total mass of the vehicle as a proper insulation is required due to its low density.

To successfully inject the fuel and oxidizer to the combustion chamber, a proper pressurization is needed. Therefore, a turbopump is installed which has high outputs and low mass. Separate turbopumps for fuel and oxidizer enable relatively accurate control over the propellant mixture in the combustion chamber. The system also allows for fluent increases and decreases of thrust and if an appropriate ignition system is utilized, also for a total shutdown and restart of the engine. [14]

Figure 6 - Russian NK-33/43 LRE at MAKS 2011 exhibition, Zhukovsky, Russia. [48]

2.2.7 Disadvantages

As LREs are significantly more mechanically complicated machines as opposed to SRMs, this unfortunately brings about certain disadvantages that designers must be well aware of in order to compensate for them adequately.

The first and the most obvious disadvantage when compared to SRMs is the mechanical complexity of LREs, which is tied to a host of issues that need to be dealt with in order to succesfully operate LRE-powered vehicles. Besides the general tendency of mechanically complicated systems to fail or fault, this also manifests in long and complicated preparations needed to be carried out before each launch.

The list, however, does not end there. There are quite a few complications associated with the liquid fuel itself. It is a well-known fact that liquid fuels are plagued with instability and volatility that may make them a hazard for the safety of the mission under certain circumstances, as evidenced by the 1986 disaster of the Space Shuttle Challenger.

On January 28th, 1986, Space Shuttle Challenger assigned on mission STS-51-L exploded and disintegrated 73 seconds after lifting off, killing all seven crewmembers. The primary cause of the disaster was a failed O-ring seal in the vehicle's right solid rocket booster. As the O-ring in question was not designed to withstand the unusually cold temperature during that particular launch, it gave way and caused a breach in the SRB joint it was supposed seal. This caused the high-pressure burning gas to gradually compromise the structural integrity of the external liquid fuel tank. Eventually, a leak of liquid hydrogen started to form that eventually provided an additional propulsive force that caused the liquid hydrogen tank inside the external tank to ram into the liquid oxygen tank in front of it. At this point, both propellants mixed and ignited, creating a fireball that engulfed the vehicle. Subsequently, aerodynamic forces broke the orbiter apart following deviation from its path caused by the semi-separated SRB providing thrust in a conflicting vector and a significant rise of load factor of up to 20 G (see Figure 7).

In case that cryogenic fuel is used, there is a risk of accreation of ice on areas adjacent to the fuel tank which is extremely dangerous to sealing elements and valves. This in turn creates a possibility of a leakage of the liquid propellants which may, as shown above, easily ignite or explode.

However, even if no such events occur, the very consumption of liquid propellants is an issue in itself as it leads to the center of gravity shift towards the rear of the vehicle. Moreover, vibrations and shocks encountered while transiting through Earth's atmosphere may cause dynamic movements of the propellants in their tanks. Both issues may result in loss of control if not countered appropriately.

Another critical point is the turbopump. Generally, the turbopumps are quite complex from the mechanical point of view and they may malfunction if they run on idle (in case the flow of propellants is distorted), in case they are contaminated with metal debris or if they suck up gas due to presence of vortices in propellants caused by inertia.

The vehicle's movement in a no gravity environment also comes with its own set of challenges. Due to general tendency of all liquids to form droplets due to their surface tensions, the liquid fuel can detach from the throat of the fuel pump. Usually, this is negotiated by utilization of acceleration engines which provide a small inertial force sufficient for the flow initiation.

2.2.8 Current Status of LRE Development

Based on analysis of literature regarding LREs [16][17], we can divide the development of LREs into four stages or generations. While the first generation's distinct feature is only a single ignition of the main engines in flight, later booster designs possess the capability of transfers between orbits with prolonged periods of inaction and multiple ignitions.

The second generation is significant because of the introduction of intermittent (relay-based) information on the state of pneumohydraulic supply system permitting its controllability. For the first time, the launchers of this generation were able to generate an indication when the fuel tanks were nearly depleted.

The tell-tale sign typical for third generation is that the intermittent information from the previous generation of vehicles was replaced by continuous monitoring of the system. Also, first diagnostic and emergency egress systems were introduced.

Fourth generation is co-defined by the fact that the engine control system system monitors and gives feedback on both normal and emergency engine cutoff procedures.

Naturally, there has been a trend to use fewer flight tests in order to negotiate the issue of astronomocial costs of the development of new space vehicles. This leads to an increased demand for effective statistical analysis of test and operational data. For these purposes, mathematical simulation along with computerized modelling is used.

For future designs, further improvements to the level of monitoring and control of individual systems is to be expected, namely through improved real-time telemetrics utilizing heightened bandwidths and downlink/uplink speeds. Together with increasing safety margins permitted by fine-tuning new LRE performance, this will lead to heightened safety of spaceflight which is demanded by customers due to financial constraints of each payload delivery (and extremely adverse effects in cause of a launch failure).

The level of safety, however, will always be somewhat relative when designing a new vehicle as the designer is forced to compromise between objectives that are potentially mutually exclusive – reliability, performance, low development and recurring costs.

The reliability of modern space transport systems is crucial. Main reasons for this include the above mentioned high costs of payloads as well as the significance and media coverage of a potential launch failure resulting in a decline of confidence between the payload customers, insurers and other players in the spaceflight industry.

Performance issue is critical for upper stages as every additional second of the specific impulse implies the ability to transport and successfully deliver heavier payloads to higher orbits.

It may be difficult to balance these requirements, especially under significant pressure from investors and customers, but it is imperative nevertheless. Hence, clear priorities have to be set in each program in order to reach its goals and make it a success.

Fortunately for researchers and producers, there is a growing set of technologies and tools that enable them to better simulate and predict different conditions and issues associated with the development even before the vehicle is ever built and test-launched; these are the same technologies that revolutionized the automotive and aviation industries during past three decades. Now, they are equally significant in the spaceflight industry.

The most prominent among these, enabled by the frantic pace of improvements in processing performance of modern computers, are apparently advanced numerical simulations along with a wide range of computerized design tools. Together, these power tools help significantly shorten the developmental process while assisting with finding the most viable and most beneficial solutions for the specific applications.

New instrumentation technologies together with all-present miniaturization have, on the other hand, completely transformed the testing process. The ground-breaking new systems, such as flush pressure gauges, laser, optical and magnetic instruments and thermal and infra-red imaging equipment have largely contributed to the ability to immediately visualize, analyze and comprehend the behaviour of processes that are key to the resulting vehicle's performance. [16]

2.2.9 Commercial Spaceflight Considerations

The biggest challenge and simultaneously the largest driving force behind the development of LREs is presently the commercial spaceflight market. There are several companies that compete in the new “space race“, that is in the quest to secure commercial contracts for delivering payload to different orbits around the Earth. The specifics of this race include regular increase in the satellite mass as well as a strong push for reliability as the manufacturing costs of the space hardware are astronomical.

Every spaceflight provider strives to possess a variety of launch vehicles in order to be able to offer a tailored solution for each launch. This capability is attainable only by operating a fleet of heavy, medium and small vehicles. Simultaneously, the operators are forced to continuously increase the flexibility of existing launchers, mainly by development of new upper stages featuring necessary upgrades such as different types of fairings able to accommodate payloads of different dimensions, additional propellant capacity or advanced engine control capabilites.

Choice of next-generation propellants for upper stages is still up for debate: cryogenic (especially LOX/liquid methane which is considered for missions to Mars), LOX/RP-1, storable propellants. However, a significant reduction in use of toxic storables, mainly of hydrazine, is visible.

Very promising are the hydrocarbon propellants, especially methane as suggested above. Especially their density and boiling point are a significant asset, with additional advantage of easy storage and manipulation.

One of the new trends is the emergence of commercially developer, very cheap launchers, especially for small payloads. However, the simplification of the engine design process appears not to be the right way as these attempts, made with the aim to reduce costs, have so far resulted in discovering unexpected issues along the way.

The last few years have consistently shown, however, that despite some initial shortfalls and failures, the establishment of private companies such as SpaceX and Blue Origin caused nothing short of a revolution in the aerospace industry. These companies have so far proved very effective in lowering costs-per-launch and coming up with innovative designs, which are currently paving the way to complete reusability of the first stages of their respective vehicles. When they succeed, this will bring about an unprecedented ability to significantly increase the number of launches per year while deliver further savings in the range of as much as 30% of today’s launch costs. [18]

2.2.10 Future Perspective of LRE Development

Mid and long term-wise, the probable way forward is to pursue improvements in presently established areas rather than aiming for breakthroughs by introducing entirely new concepts of rocket propulsion. These key areas include further improving reliability while pressing down costs, expanding availability of a wide array of launchers for different payloads and applications, shortening the development times and increasing performance metrics through use of advanced materials in both structure and propellants as well as through state-of-the-art software modelling tools.

Out of all these targets, the priority criterion will still be the reliability, but with an increased accent on balancing the various requirements in new designs.

New technologies are expected to significantly increase the operating temperatures or enable complex hydraulic shapes – metal deposition, new welding technologies and especially additive manufacturing, just to name a few.

With ongoing efforts to achieve complete reusability of the first stages (and possibly upper stages as well in some time), the engine life will become a serious issue. Through utilization of new materials and engine designs, the manufacturers will have to come up with ways to minimize the amount of overhauls required between the launches. [16]

3 Hybrid Rocket Engines

In the third part of this thesis, I introduce the concept of a hybrid rocket engine (further referred to as HRE). Even though this concept is not new – first theoretical works on HREs were published as early as 1930s – it is revolutionary as it has the capability to overcome most of disadvantages of both SRMs and LREs. The reason that it still has not seen a widespread use lies in the fact that successful design of HRE requires a detailed knowledge of its internal processes; a knowledge that was not available until relatively recently, as it requires complex modelling and computational tools to properly predict the behaviour and performance of the engine and achieve agreement with the results of experimental firings.

The research has advanced significantly in only a handful of years, however, and with the shift of space exploration towards commercial operations discussed earlier, it is safe to say it will not be long before new HRE-powered launchers will be introduced, taking advantage of their inherent safety and efficiency while staying cost-efficient.

In the end of the section, I also present another take on the meaning of the word “hybrid“. That is, an approach that does not combine solid and liquid-fuel rocket engines into one, but rather is a combination of an air-breathing engine and a rocket. While very different from HREs that are the primary subject of this thesis, it is an interesting concept nevertheless, one that may contribute to introduction of a fully re-usable space vehicle and one that together with HREs can contribute to a concept of eventually “aviationizing the spaceflight“ which I discuss in the last part of the thesis.

3.1 General Description

Hybrid engines are such devices which are running on propellants which are stored separately and in different phases. While a “direct“ hybrid engine runs on a combination of solid fuel and a liquid oxidizer, a so-called “indirect“ engine uses, vice versa, liquid fuel coupled with solid oxidizer [19]. Considering HRE’s specific output and mass characteristics, they stand in the middle between liquid rocket engines utilizing cryogenic propellants and solid rocket motors. While being fairly simple, they generally permit adjustments of thrust, thrust vectoring, multiple firings and are capable of long-term operation under space conditions. The schematic drawing of a common HRE set-up is shown in Figure 7.

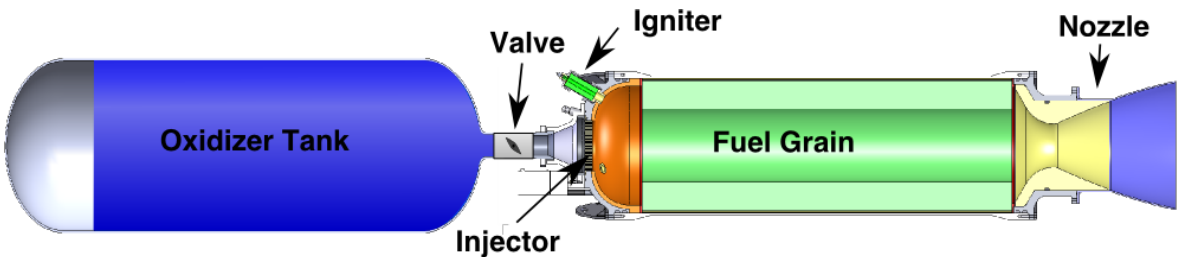

Figure 7 - Schematic drawing of HRE [48]

HREs are advantegeous because of the stability in their internal processes as well as practical absence of irregular combustion. Moreover, they possess an intrinsic capability of chamber and nozzle protection through the design of the propellant which is something that SRMs and LREs lack and have to be specifically designed just in order to secure these important properties.

From construction viewpoint, HREs are capable of a very high level of main engine unit unification which can, and will, further drive down the costs while improving efficiency and also safety of operation of these systems. [20]

When accounted for a lower cost of propellants and the above-mentioned facts, the bottom line is that wide application of hybrid engines to spaceflight domain would lead to a significant reduction of costs of spacecraft production. It appears that this heightened financial efficiency together with handling characteristics of HREs could contribute to a process of "aviationizing" spaceflight, i.e. making spaceflight a much more routine and safe matter than it is today.

The most significant advantages of higher utilization of HREs in spaceflight industry seem to be heightened reliability and consequent reduction in costs of payload delivery to the orbit which specifically come from the following:

- Reduction of development time of new propulsion systems (when compared to current LREs and SRMs from 4-5 years to as short as 6-10 months thanks to the utilization of unified assemblies and pre-constructed units.
- Reduction of as much as 50% in manufacturing costs when compared to LREs of the same class thanks to application of simple ablative cooling on the basis of carbon or carbon-ceramics composite materials instead of regenerative cooling systems, when designing the combustion chamber and nozzle sub-assemblies.

- Decrease in launching costs due to less expensive construction material, up to 60% reduction in time required for pre-launch preparations, significantly lower operating costs due to mechanical simplicity of the whole propulsion system, and so on [20].

3.2 Basic processes in HREs

Generally, hybrid propellants are challenged by the same requirements as other chemical propellants. Therefore, they shall containe substances that are preferably dense, interact chemically with the highest possible heat output and their combustion generates exhaust with the lowest possible molecular weight. [20]

Most widely used are polymer hydrocarbon fuels (synthetic rubbers) together with liquid oxygen or hydrogen peroxide as an oxidizer, for they are simple and of adequate efficiency. To increase power and to stabilize the burning process, metals, such as aluminium or lithium, may be added.

A problem arises in an approach to determining an optimum flow of oxidizer to ensure a stable burn. The solution is based on obtaining and analyzing a sufficiently accurate thermodynamic model of the combustion process. This includes solving equations of turbulent boundary layer for binary chemically reacting mixture. There are still areas of this field that are not properly explored and as a result, a certain measure of uncertainty while designing HREs is still present.

This approach has been widely tested and justified by confrontation of calculated results to experimental data which corroborated each other. The experiments allowed a research into regularities of polymer material burning processes in both oxygen and hydrogen peroxide. Particularly important is the fact that owing to the variations in heat exchange properties, the temperature and concentration of propellants, the regression rate and consequently the distortion of cylindrical channel of the fuel grain is a non-monotone function of the x coordinate, or the length of the combustion chamber (fuel grain).

Some results indicate that the regression rate of fuel increases proportionally with increasing oxidizer density.

A key issue (and one of major disadvantages of HREs) is that to be able to successfully compete with other types of rocket engines (mainly the LREs), more complete propellant combustion must be ensured. Indeed, due to specifics of hybrid propellant burning process, the HRE is by default incapable of achieving complete propellant combustion as the oxidizer and the propellant simply do not have sufficient length of time to mix with each other properly in the fuel grain channel.

Geometric properties of the fuel grain duct can greatly influence the completeness of combustion η. Among others, length and diameter of the fuel grain channel influence the combustion as much as the type of fuel itself, along with the operating pressure in the combustion chamber, oxidizer-to-fuel ratio α and other factors.

Available experimental results confirm that combustion completeness of more than 0.9 or 0.95 can't be obtained when using hydrogen peroxide or oxygen as oxidizer, respectively.

By using special devices (such as optimized injector designs), the combustion completeness can be improved to up to 0.93 for hydrogen peroxide and 0.98 for oxygen.

3.3 Oxidizers

As fuels used in HREs, unlike in SRMs, are usually inherently stable, safe substances which do not require any special attention while being handled, the most treatment sensitive element in hybrids is generally the oxidizer. As with fuels, there is too many substances that are used in this role to keep track of and for the purposes of this thesis, I will only introduce the three most common. Special emphasis is placed on the nitrous oxide which has been used in the only two large-scale applications of HREs so far, the Scaled Composites / Virgin Galactic SpaceShipOne (pictured in Figure 8) and SpaceShipTwo.

Figure 8 - Scaled Composites SpaceShipOne. Author: lkluft [48]

3.3.1 Nitrous oxide (N_2O)

In most hybrid applications, the most popular oxidizer has so far been the nitrous oxide. One of its key advantages is that it is self-pressurized in atmospheric conditions. Thus, there is no requirement for pumps or any external means of generating pressurized environment. The simplicity of this approach is, on the other hand, somewhat limiting in a certain way and not without issues. The boil-off pressure varies with outside air temperature and often, the ullage (the part of oxidizer tank filled with gas) needs to be heated in order to increase the pressure and to secure the desired flow-rate necessary for the engine to function properly and according to its design. N_2O is also prone to deflagrations or detonations from a wide array of ignition sources. Various events (hard starts, backshocks) can contribute to a mid-flight explosion. The danger persists even as the system is being filled, when running at idle or even under the cold-flow conditions.

Another issue is the tendency of N_2O to boil when the self-pressurized system is activated. The activation is accompanied by a drop in the pressure and this makes the oxidizer to produce gas pockets throughout its volume. This in turn makes it compressible, with resulting rapid changes in operating pressure manifesting as “pulsing“. The injection is thus less stable and gradually becomes unpredictable and uncontrollable.

To prevent this, the ullage may be filled with an inert gas for pressurization. An added advantage is the provision of non-compressible fluid to the injector. However, with the oxidizer tank emptying throughout the flight, the inert gas can exit through the injector and reduce the combustion effectiveness or even cause the motor to choke. This can be negotiated by implementing a dip-tube.

The most ideal inert gases for the task are helium or argon. To maintain the pressure as the tank empties, the pressurant gas can be fed to the tank from an additional, separate tank. With high-flow regulator, the pressure provided by the inert gas remains constant throughout the operation, resulting in smooth and stable performance. [21]

Another method is more complex from the engineering point of view but removes the risks considerably. It uses a piston or a bladder which separates the oxidizer from the pressurant gas. This makes for a very satisfactory engine performance.

A large drawback of the N_2O is that in a temperature of 36 degrees Celsius, it forms a supercritical fluid which is compressible and highly shock-sensitive. Moreover, it will dissolve hydrocarbons quite aggresively, with the resulting substance consequently becoming even more sensitive to shock.

3.3.2 High-Test Hydrogen Peroxide (HTHP)

Being a highly corrosive and reactive substance, the HTHP has to be treated with utmost care. It does not self-pressurize and therefore means to pump or press it into the injector have to be present.

So far, there have been some troubling issues with starting the combustion with the use of HTHP. Fuel grain modifications and inclusion of brought limited success.

3.3.3 Liquid Oxygen

The use of liquid oxygen as the oxidizer brought about significant advances for HREs. Some of the early experients with LOX proved that with liquid oxygen, HREs can be quite comfortably scaled to sizes which might be practical for spaceflight applications. There have been some problems with fuel grains has enabled to scale HREs to considerable sizes and allows for reliable startups.

Especially initial tryouts of the Space Propulsion Group utilizing novel fuel grains on basis of paraffin wax yielded satisfactory results. When combined with LOX, they were able to produce surprisingly high specific impulses that might be competitive even when compared with high-performance LREs. [21][22]

3.4 Different Injector Configurations

Generally, two primary parameters considered to most affect the fuel regression rate in HREs are the oxidizer mass flux and the chamber operating pressure. The amount to which these properties correlate, however, depends largely on the propellant used. In other words, there is no universal and reliable way to calculate propellant consumption in an HRE. Values gained from practical experiments are not interapplicable to other engines, notably because of scaling issues. However, there may be another reason for this that is not accounted for – that is, the influence of oxidiser injector configuration. Some recent research works have shown that by changing the injector set-up, the whole engine’s behaviour will change.

Remarkably, by using different injector types, we may recieve significantly different regression rate distributions along the length of the combustion chamber. With radial injector, instead of the expected uniform flow field, a zone recirculation may form. This may in turn lead to an uneven regression rate along the rest of the fuel grain and thus, under certain configurations, excessive amounts of unburned fuel may remain in the combustion chamber.

Therefore, the injection of oxidizer through an axial nozzle is preferable as it permits avoiding the threat of excessive turbulent flow in the combustion chamber, effectively countering the negative influence described above.

3.5 Tribrid

A tribrid is a special case of HRE which features an additional injection system for liquid fuel. This carries a penalty to the overall engine mass caused by additional plumbing and hardware. However, such engine can operate even when the solid propellant grain is consumed.

Experimental results show that tribrid engines are able to produce higher thrust than simple hybrids. However, they also tend to behave slightly more regressively, in other words, the thrust decreases slightly quicker over time as opposed to classical hybrids.

Overall, the tribrid, despite its additional complexity, significantly enhances the scope of application of HREs as in this configuration, it can be employed both as a booster and a sustainer at the same time.

However, tribrids still remain at a purely conceptual and experimental phase with no specific application in development as of May 2017. [19]

3.6 Thermal Protective Materials in HREs

To improve power-and-mass characteristics of HREs, it has been suggested to implement composite materials in the combustion chamber and nozzle unit design. With this solution, the design process is open to procedures and technologies used in the development of much simpler and vastly proven SRMs, making the design stage of HRE development significantly simpler. With equal heat fluxes and operating loads, the HRE operating conditions are distinctive because of significant content of oxygen in their exhaust. This is quite a difference when compared to the application of thermal protective materials in SRMs. Therefore, to secure the nozzle faillless operation and to prevent thermal protection ablations, enhanced oxidizing resistance must be ensured. Along with conventional ablative protection, this fact also calls for implementation of carbon-ceramics composites. Apart from these passive methods, active thermal protection may also be employed.

In laboratory setting trials, it has been observed that carbon-carbon composites are suitable for ablative protection of throat region and carbon- and glass-phenolics can be employed in large area expansion ratio nozzles for burn lengths of up to 100 seconds. For longer burn times or for repeated ignitions, carbon-ceramics materials are preferable.

In case that the operating conditions of HREs surpass those mentioned above, it is strongly advisable to use active thermal protection in order to maintain section stability as well as nozzle contours. These may include either protective application of low-temperature films of neutral or weakly acid combustion products. By employing this method at a rate of consumption of these substances of about 0,5 to 1,5 % of total engine consumption, the oxidizing potential of the exhaust products drops by a factor of 3 to 5. This may, in turn, lead to precluding the heat accumulation in the throat section of the nozzle by as much as 300 to 500 K. To put this into perspective, these factors allow this active thermal protection method to reduce the ablation rate of carbon-graphite composite materials from 0,1-0,2 mm/s to 0,01-0,015 mm/s. Moreover, a potential decrease of specific impulse is fully countered by the positive effect of absence of erosion of the nozzle throat and subsequent decrease of effective area expansion ratio. For large area expansion ratio nozzles, the losses caused by the protective film application are smaller by a factor of 3-5 when compared to losses brought about by the nozzle contour violation. [5]

3.7 Fields of Application

Thanks to their simplicity and relatively higher safety of operation, HREs may become a method of choice while designing future launch vehicles of various classes, provided that present issues will be solved satisfactorily. However, even now, they are versatile enough to be eligible to help solving other significant problems in today's rocket space technology:

- Orbital transfers
- Orbit eccentricity modifications;
- Docking maneuver control;
- Efficient maneuvering engines in vacuum conditions.

Today's technology makes the design of high-powered, large HREs a realistic task. Present efforts in this field show that these applications of HREs may be promising in mid-term future:

- Light-weight launch vehicles for payloads of up to 1.5 tons into low earth orbit;
- Technical maintenance of space objects using reusable interorbital towing vehicles;

- Space containers designed for cleaning of near-Earth space from “space junk“
- Hybrid rocket boosters.

As mentioned before, the introduction of HREs constitutes a dawn of a new era in spaceflight as a routine, widely accessible activity. On the forefront of this effort are private companies which develop space vehicles capable of repeated flights. The most notable example of such HRE-powered vehicle that already goes throught extensive flight testing is the SpaceShipTwo (displayed in Figure 9) operated by Virgin Galactic. [20]

Figure 9 – Virgin Galactic’s SpaceShipTwo. Author: Ron Rosano [48]

3.8 Disadvantages

HREs possess two main disadvantages compared to SRMs and LREs, which are the combustion inefficiency and low regression rate. [19]

In a generic HRE, the efficiency of the burn heavily depends on the mixture richness – the ratio between the oxidizer and the fuel before the oxidizer’s egress from the combustion chamber. Because of this, a typical HRE is longer than an average SRM. Apart from adjusting dimensions, the engineers also aim to increase the burning surface area through arranging the fuel grain in different geometries.

One of the major criteria determining the length of burn and the specific impulse is the shape of the fuel grain. Due to the fact that the burn occurs on the fuel surface and gradually progresses to the inside of the grain, it generally is an intention of the engineers to design the grains in such a way as to maximize the burning surface at any given moment. However, there are a few complications that have to be addressed during the design process.

First and foremost, there is a phenomenon known as the blocking effect. As fuel burns, the fuel grain particles evaporate. Paradoxically, these vaporized particles blow from the solid fuel grain surface, modifying the boundary layer and further fueling the diffusion flame, effectively blocking the transfer of heat back into the surface of the fuel grain, thus reducing the speed of burn and the regression rate (for polymers, the regression rate is only about 1 mm / s). Because of this, it is critical to maximize the initial fuel grain surface in order to ensure sufficient thrust for the first moments of flight (which tend to be by far the most energetically demanding). [23][24]

To combat the low regression rate, the method of weak oxidizer doping of the solid fuel may be used. This means incorporating a small amount (usually ranging from 3 to 6% of the total fuel grain mass) of solid, powdered oxidizer into the solid fuel grain. It has been experimentally proven that by utilizing this method, the regression rate might be increased as much as five- to ten-fold. Thus, the largest drawback of HREs might be successfully suppressed. [19]

Naturally, the low oxidizer doping comes with a set of its own challenges, namely the production of homogenous fuel grain with evenly distributed particles of the solid oxidizer might prove difficult. However, these issues are certainly negotiable with further material research.

3.9 Advantages

As mentioned above, there is a wide range of benefits when compared to both SRMs and LREs. The common denominator of all of them is mainly higher reliability and safety of operation at lesser costs of development and construction. These can be broken down to more fundamental properties, such as the propellant's non-explosive character and the system's insensitivity to the initial propellant temperature (as opposed to LREs). Moreover, they are also quite insensitive to cracks. [22]

Regarding the engine controllability, HREs offer reliable shut-off and throttlability capability. These properties may aid in increasing the level of accuracy in the flight control as well as achieving increasing levels of "aviationizing" the spaceflight in the more distant future.

Advantages Against Liquid Rockets

- One of key advantages of HREs is that they are mechanically simpler, thus avoiding the need for countless valves, pumps, complicated plumbing system and resulting in simpler operation and inherent reliability. Implications of this simple fact are complex, ranging from superior safety of launch to reduced costs per launch.
- As solid fuel possesses higher density, HRE propellants have lower overall volume when compared to the LRE propellants. This is partly negated by the general tendency of HREs to be longer due to peculiarities of hybrid propellant burning process but it is still safe to assume that by employing HRE for a given application, the vehicle can be smaller and thus more lightweight as opposed to the one employing an LRE, which in turn leads to lower costs or the ability to store and transport larger payload.
- HREs typically do not have issues with combustion instabilities. Solid fuel grain does not suffer from acoustic waves which cause problems in an open LRE combustion chamber.
- HRE completely negate the need for propellant pressurization. In LREs, one of the most complicated elements to design are turbopumps which have to fulfill quite a few, at times conflicting, requirements in order for the whole system to operate successfully. In HREs, however, there is a significantly lower demand for transfering fluids and the pressurization can be secured by a blow-down system. Alternatively, the oxidizer may be self-pressurizing, such as N_2O.
- There is signicantly lower demand for cooling in HREs as opposed to LREs where complex cooling systems employing the flow of fuel through chamber and nozzle walls has to be present. In HRE, the combustion chamber walls are protected by the fuel grain itself and the nozzle is usually coated in graphite or ablative materials. The result is inherent safety of operation while keeping development and construction times to a minimum as opposed to LREs.

Advantages Against the SRMs

- HREs possess higher values of specific impulse. Generally, the limits of known solid oxidizers are far below the currently used liquid oxidizers.
- As HRE propellant grain is much more tolerant of processing errors (e.g. cracks) because the burn rate depends on oxidizer mass flux rate. The HRE fuel grain also cannot be ignited by a stray electrical charge and is very unlikely to self-ignite due to heat. Also, as oxidizer and propellant are stored separately by default, safety is improved when compared to SRMs.

- As opposed to the SRM, the propellant grains in HREs are very stable both chemically and thermally. Unlike in SRMs, additives are not required to keep the grain from deteriorating.
- While there are efforts to introduce some extent of control to SRMs (the CSP concept), they generally do not have this capability. HREs, on the other hand, are capable of repeated restarts as well as adjusting the thrust.

3.10 SpaceDev N_2O/HTPB Engine – SpaceShipOne

Even though HREs are quite common in amateur rocketry as well as in sounding rockets, there have actually been only two successful operational large scale HRE types so far. The first one was developed by SpaceDev for Scaled Composites SpaceShipOne, while the second is the RocketMotorTwo used in Virgin Galactic's SpaceShipTwo. As information about the latter are still not publicly available, we are left with the SpaceShipOne for detailed description. While large amount of data on SpaceShipOne's engine is also proprietary, there has been significantly more leads and details that were publicized due to high-profile, historical flights of SpaceShipOne, among others the Ansari X-Prize winning flights.

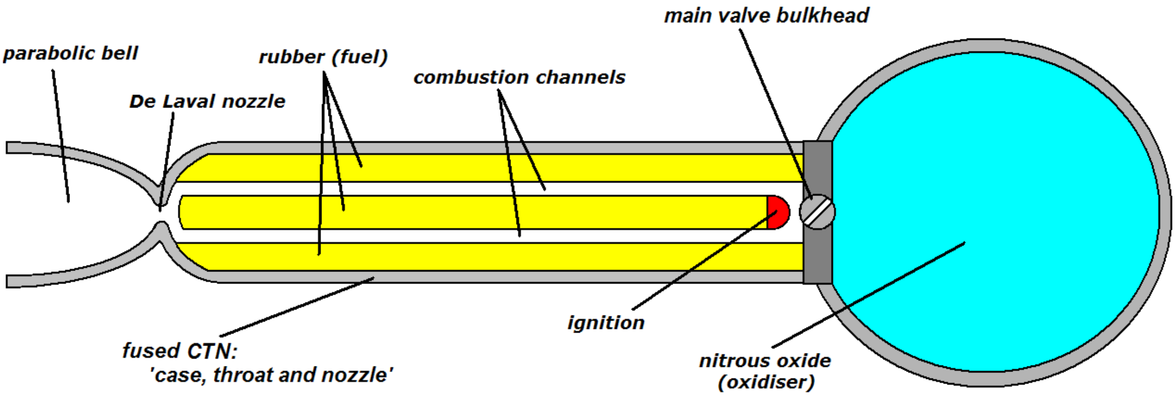

Figure 10 - SpaceDev CTN engine used in SpaceShipOne [48]

SpaceShipOne used an HRE (schematic is shown in Figure 10) powered by the mix of nitrous oxide oxidizer and hydroxyl-terminated polybutadiene. These engines are commonly designated as N_2O/HTPB engines for brevity.

From construction point of view, the engine is exceptionally simple. Running from front to back, there is the oxidizer tank which is designed as a part of the space ship's fuselage structure.

Attached to the tank is the casing which houses the rubber fuel grain. The nozzle then protrudes from the casing and out of the vehicle’s rear part. This set-up has been actually patented under the designation CTN (casing, throat and nozzle). The casing together with the nozzle are constructed as a single structure.

The HTPB fuel grain is manufactured by casting and has 4 combustion channels. For the simple flight test program of SpaceShipOne, the engine did not have any throttling or reignition capability (however, the burn could have been extinguished).

For each flight, the engine was fitted with 2400 kg of solid propellant. The oxidizer tank was pressurized to 4.8 MPa at room temperature. Together, the propellants generated 88 kN of thrust with a burn time of 87 s (some sources indicate up to 90 s). The engine provided a specific impulse of as much as 250 s.

Throughout the engine, similarly to the entire vehicle, composite materials were used in large quantities, allowing for a very lightweight construction. [24][42][43]

3.11 SABRE Engine

SABRE (stands for Synergistic Air-Breathing Rocket Engine, shown in Figure 11) is a concept of a hybrid jet/rocket engine with performance supposedly high enough to achieve a single-stage-to-orbit (SSTO) capability. It is under development by United Kingdom-based Reaction Engines Ltd. Although the engine is a hybrid in a very different sense from HREs that are covered in this thesis, it certainly merits a mention due to the significant change it promises to bring to the aerospace industry.

Figure 11 - SABRE Engine. Author: Science Museum London [48]

Conceptually, SABRE is a hypersonic precooled hybrid air-breathing rocket engine. It promises to provide single-stage to orbit capability. Currently, it is scheduled to be used in the work-in-progress reusable space launch vehicle Skylon set to fly in early 2020s.

Fundamentally, the design incorporates a classical axial compressor known from air-breathing jet engines together with a rocket engine in a single casing. The aircraft can take off conventionally, as a fixed-wing airplane, and acts as a jet in the initial stages of its flight. As the aircraft moves forward, the outside air is entering the engine through the compressor and both its pressure and temperature increases. However, after exiting the compressor, the air mass proceeds to a pre-cooler positioned behind the compressor where the temperature of the mass of the air drops significantly, while its density increases. This leads to a considerable pressure ratio gain. Subsequently, the air enters the aft-positioned combustion chamber where it serves as an oxidizer. The air is mixed with liquid hydrogen fed to the chamber from a separate tank before being burned.

After surpassing a certain altitude and speed (in the hypersonic range), the jet portion of the engine is shut down, the intake is closed and the combustion continues using internal stores of liquid hydrogen and oxygen for propellants. Using this configuration, the spacecraft can be controlled by aerodynamic control surfaces, very much like a conventional airplane during the atmospheric phase of flight. Another significant advantage of this concept lies in the fact that due it doesn't have to withstand high temperatures, therefore allowing for a light-weight construction. In case it would prove completely reusable, requiring only a refill of propellants, it would truly mean a revolution from the financial point of view as it should be ready in early 2020s if it stays on the schedule – 100% reusability of conventional launchers is sure to be much further down the road.

While the design sounds promising, the response is mostly sceptical [39]. Projected costs of the program are also far above those of modern LRE-powered medium and heavy launchers. Especially with ongoing attempts at partial reusability of these launchers, which are becoming more and more successful, the viability of this solution is questionable.

However, British Aerospace has recently invested significant amount of financial assets in the program [40]. At the very least, the additional funding will sustain this interesting program for an additional period of time, giving it a chance to achieve the promised breakthrough advance. [41]

4 Assessment of Feasibility and Cost-Efficiency of HRE Applications

In the final chapter of this thesis, I offer a summary based on most of information presented above. However, the perspective is somewhat different – instead of that of purely academical, informative reports, I intend to present a practical approach with specific results that can provide a clear idea of potential of the different propulsions systems in relation to each other. Each of the previous chapters included an assessment of both virtues and shortfalls of the main kinds of rocket propulsion systems. Now, let us finally carry on with a comparison of these engines in real, easily imaginable terms. In this chapter, we are dealing with fuel costs and total launch expenditures as well as other peculiarities of the respective engine types. Based on this data, I also come to conclusions regarding the areas of possible implementation of hybrid rocket propulsion in which it would be most beneficial from today's point of view and that of mid-term future – areas that would make maximum use of HRE's advantages while countering the influence of their drawbacks.

Additionally, the final chapter introduces a trend that is sure to bring about a significant breakthrough in all fields of space exploration, the additive manufacturing. With this technology which has already made a significant impact in other fields including the automotive and commercial aircraft industries, designers and manufacturers obtain a capability to build complex structures as well as miniscule devices with unprecedented accuracy and in a way that enables superior material properties while also minimizing the weight of such components. It is yet to be seen which areas of launch vehicle construction will this technology be the most beneficial but there has been so far more than enough evidence that additional manufacturing is one of the most promising new tools out there.

Finally, the practical part of this thesis concludes with my own proposal that takes into account the synthesis of all knowledge gained during my work on this thesis and puts them into perspective with my primary field of study as well as the branch I have spent my whole professional life in – that is aviation, especially air traffic control, aircraft management and associated specializations.

In this proposal, I suggest a way to "aviationize the spaceflight". In theory, let us assume that hybrid propulsion will be able to advance and evolve in such a way that it will enable development and construction of new types of aircraft powered by HREs and equipped with systems giving them unprecedented level of control over the rocket engine's operation. In the next 20-30 years, these new designs could then serve in the role of super- or hypersonic transport aircraft (only in altitudes much higher than those that are suitable for today's airliners fitted with air-breathing engines), or even as sub-orbital point-to-point transport spacecraft.

First proposals and concepts of this new category of air transport are beginning to surface which shows that with a lot of additional research and design work, this approach might be realistic. The biggest advantage lies in the fact that it might solve the ever-present issues that today's experts in the field of air transportation are painfully aware of – issues associated with airspace capacity that are becoming increasingly difficult to compensate for. This is especially visible in Europe and North America, but as aviation is a sharply rising industry in Asia, it is generally expected that in course of the next 20 years, similar problems will arise there as well. [46]

While present approach to these problems lies in optimizing the capacities of individual airports, ATC routes and airspace sectors and in introducing sophisticated air traffic flow management systems, it is apparent that if the volume of flights worldwide will continue to rise, we will eventually reach a point where all these measures simply will not suffice.

My assumption, however, is based on a different prerogative – instead of optimizing processes and airspace capacity that is used these days, I calculate with introduction a new class of aircraft that will perform long-range flight in the upper layers of the atmosphere. By presenting this purely personal view, I intend to demonstrate one of the possible ways of long-term advancement of the aerospace industry.

4.1 Propellant Cost Comparison

To properly evaluate feasibility and competitiveness of HREs as opposed to other types of rocket propulsion, we would logically start with the most distinctive feature that differentiates HREs from other types of rocket engines – the propellants.

Specifically, we are looking for a comparison of market prices of different types of propellants. That is not the only criteria, however, upon which to base the analysis. Naturally, we will also need a host of other data – for instance, how much fuel does any particular engine need to operate in the specific scenario and to succeed in the task it has been selected to accomplish? How many percent of total launch costs the cost of propellants accounts for?

To carry out a proper comparison, let us first narrow down the group of propellants to compare. Obviously, there are far too many combinations of fuels and oxidizers to choose from for propulsion of liquid rockets and there is even wider spectrum of available solid propellants mixtures. For practical reasons, we will only select a few propellants (or combinations of propellants) that are most commonly used. As we merely need to determine the baseline differences between the respective propellant groups, this approach will be sufficient while still staying conveniently simple.

For liquid rockets, this would certainly be the combinations of liquid oxygen and kerosene (RP-1) and liquid oxygen and liquid hydrogen. For solid rockets, let us select the most common option, which is the Ammonium Perchlorate-based propellant family with a content of aluminium (acting as high-performance propellant) and HTPB (acting as low-performance fuel and binder). As a typical representative of hybrid fuel, HTPB (hydroxyl-terminated polybutadiene) will be considered.

The main issue that makes these calculations complicated is the fact that information regarding exact fuel formulae and pricing are often proprietary, especially in case of HRE propellants. Therefore, the below calculations are based on a combination of hard data (in case it is available directly from manufacturers), information gathered by synthesis of data from multiple sources (e.g. press releases) or, in the worst case, qualified estimates.

4.1.1 Liquid Propellants

LOX/LH$_2$

Liquid hydrogen and liquid oxygen are quite common substances not just in aerospace industry, but in many industrial applications. Paradoxically, this makes it quite difficult to obtain reliable information regarding their pricing because of sheer quantity of data. Eventually, I was able to retrieve information about prices of liquid hydrogen and liquid oxygen from a 2001 factsheet published by NASA [26] regarding a total use of propellants and other liquids by the Space Shuttle system. According to this publication, the NASA costs of liquid hydrogen and liquid oxygen in 2001 were 0.98 USD/USG and 0.67 USD/USG, respectively. As a matter of interest, NASA, as well as most other space agencies, does not produce these substances in-house but uses contractors instead [27]. This makes for attractively low unit prices based on high-quantity bulk acquisitions at contracted rates. For purpose of this approximate comparison, we will not take into account additional costs associated with complicated conditions of storage of cryogenic propellants, their handling, safety measures that have to be in place etc. These secondary costs are certainly a factor when accounting for total launch costs but we will factor them in in later phases of our estimating process.

By simple conversion of units and comparison with known values of density of both liquid hydrogen and oxygen, we calculate prices of both substances per kilogram. The results are shown in Table 2.

Table 2 - Costs of liquid H_2 and liquid O_2 per kilogram

Costs of liquid H_2 and liquid O_2 per kilogram			
Substance	**Cost in USD per US gallon**	**Density**	**Cost in USD per kilogram**
Liquid H_2	0.98 USD/USG	71 kg/m3	3.45 USD/kg
Liquid O_2	0.67 USD/USG	1140 kg/m3	0.15 USD/kg

LOX/RP-1

Market costs of RP-1 proved to be quite difficult to acquire, so a creative solution had to be adopted in order to provide an approximation of the figure in order to be able to carry out any comparison.

As an example, let us use one of the most perspective launchers of today, the Falcon 9 (shown in Figure 12) built by SpaceX. While being a fairly new design, this vehicle has already attained quite a few firsts, such as the first commercial launcher to reach low earth orbit and the first commercial launcher ever to dock with the International Space Station and to deliver government-agency payload. At approximately 62 million USD per launch as an off-the-shelf price [28], it is the most cost-efficient and versatile launcher currently in use – particularly due to visionary approach adopted by the company's leadership as well as revolutionary processes (for an aerospace company), which allow them to come up second-to-none pricing and innovations that make the launcher set for success in coming years.

Figure 12 - SpaceX Falcon 9 v 1.1 [48]

While hard data regarding the propellant prices are not available, the SpaceX CEO, Elon Musk, has stated in his press briefing on April 25^{th}, 2014 [29], that propellants account only for 0.3 % of total costs of the vehicle, which in this case is roughly 200 000 USD.

Information about the quantity of propellants in Falcon 9 are publicly available [30]. The first stage of Falcon 9 (v1.0) used 150 000 l of liquid oxygen and 95 000 l of RP-1, while the second stage used 28 000 l of liquid oxygen and 17 000 l of RP-1. That means a total of 178 000 l (203 t) of liquid oxygen and 112 000 l (91 t) of RP-1 (density of RP-1 equals to 810 kg/m3 [14]. As we know the total cost of propellants and we already know the approximate price of liquid oxygen, we can easily calculate the estimated cost of RP-1 per kg, which is then roughly 1.92 USD/kg.

4.1.2 Solid propellants

Unfortunately, the information regarding the cost breakdowns of solid propellants are scarce. According to [31], the approximate price per kg of a ammonium perchlorate based solid propellant with HTPB and aluminium is 5.56 USD. This is, however, a very approximate value as there can be a virtually unlimited number of solid propellant formulae differing in content of various additives as well as their concentrations. Therefore, let us assume that the price in today's values would be generally somewhere between 5 and 6 USD per kilogram, which should be a reasonably conservative estimate.

4.1.3 Hybrid propellants

The situation with hybrid propellant pricing availability is even more difficult than that with solid propellants as there is virtually no history of use of hybrid engines in spaceflight applications besides a few designs that are currently in an experimental and flight test phase (with Virgin Galactic's RocketMotorTwo being a notable example). That way, we have no choice but to move to an uncertain field of qualified estimates.

Compared to solid propellants, hybrids are simpler as they do not usually include molecules of oxidizer. Furthermore, the substances used in hybrid propulsion tend to be more stable and require less additives in order to keep the fuel grain from deteriorating. In fact, one of the most promising hybrid propellants (although dominantly in amateur rocketry and sounding rockets applications) is a substance as simple as paraffin wax. While it is yet to be seen if such propellants can be successfully scaled up for spaceflight applications – some research works hint that this should indeed be possible [31], the point remains: hybrid propellants are often inert, homogeneous substances and the actual price will be reflecting this fact. Let us estimate the price somewhere in the range of 3-4 USD per kg.

Additionally, HREs require a liquid oxidizer to operate, much like LREs. For HTPB-based hybrid fuel, let us select nitrous oxide as an oxidizer. For our purposes, I will estimate the market price of nitrous oxide equal to the price of liquid oxygen.

The values we have acquired above for all groups of propellants are presented in Table 3.

Table 3 - Propellant costs per kilogram

Propellant costs per kilogram	
Propellant	**Price per kg**
Liquid H_2	3.45 USD
Liquid O_2	0.15 USD
RP-1	1.92 USD
Solid	5-6 USD
Hybrid	3-4 USD

Visibly, mere costs per kg of propellant do not tell us much. The highest cost is that of solid propellants which are consistently rated as the most affordable, cheapest means of achieving spaceflight. There must apparently be other factors in play and therefore, we have to look another way of determining the cost-efficiency of rocket propulsion.

To embark on the path to achieve a consistent comparison, we need to look at specific impulses of the propellants.

4.2 Performance Metrics Comparison

Specific impulse is one of the key parameters of a rocket engine as it is proportionate to a measure of efficiency of given engine. According to the definition, it is the total impulse (force over time) provided by a unit of propellant. It can be expressed based on mass or weight units, in which case we receive the specific impulse either in units of velocity or time, respectively. Specific impulses for the selected fuels are presented in Table 4.

Table 4 - Specific impulses of select propellants

Specific impulses of select propellants		
Propellant	**Cost per kilogram**	**Specific impulse**
Liquid O2 + liquid H_2	0.15 + 3.45 USD/kg	381
Liquid O_2 + RP-1	0.15 + 1.92 USD/kg	289
APCP (AP+HTPB+Al, solid)	5-6 USD/kg	277
N_2O + HTPB (hybrid)	0.15 + 3-4 USD/kg	248

Simply said, the higher the specific impulse, the less fuel is needed to achieve a certain value of thrust in given time. If we compare two engines with different specific impulses, the one with higher specific impulse is providing higher trust with the same quantity of propellant.

This is directly governed by the Tsiolkovsky equation, also called ideal rocket equation, which is

$$\Delta v = v_e ln\frac{m_0}{m_f}$$

where Δv is the maximum value of change of the launch vehicle's speed, v_e is the effective exhaust velocity, m_0 is the initial total mass of vehicle and m_f is the initial dry mass (total vehicle mass excluding the propellants).

Using the specific impulse, it can also be noted in the following form

$$\Delta v = \ g_0 I_{sp} ln\frac{m_0}{m_f}$$

where g_0 is the gravitational acceleration at the surface of Earth and I_{sp} is the specific impulse. This leads to the I_{sp} being possible to denote as

$$I_{sp} = \frac{v_e}{g_0}$$

To put this into perspective, the value of specific impulse in N.s/kg is equal to the effective exhaust velocity. To receive the value of specific impulse in seconds, we need to divide it by g_0.

To determine thrust, we simply need to multiply the exhaust velocity and the exhaust mass flow:

$$F_{thrust} = v_e \cdot \dot{m}$$

where $\dot{m}$ is the propellant mass flow. When the specific impulse (commonly given in seconds) is used, the equation can be denoted as

$$F_{thrust} = I_{sp} \cdot g_0 \cdot \dot{m}$$

As specific impulses are known for all commonly used propellants, it would be fairly easy to divide these values by the respective propellant's cost per kilogram and receive a comparison of a “specific impulse per dolar expended“. However, we would not be getting a full picture. That is because the performance of a single kilogram of propellant is not the only important value to look out for. There is a significant impact of the density of these propellants as well, which becomes quite apparent once we consider the differences between the liquid hydrogen and RP-1 – or the solid propellants for that matter.

Therefore, let us introduce the density impulse which is nothing more than the specific impulse corrected for the propellant density. It is calculated simply by multiplying the I_{sp} with the respective density. The result implies the performance of the propellant not based on the weight, but rather the energetic output of a given volume of the substance. The results are shown in Table 5 which makes the perspective shift introduced by density impulse self-evident:

Table 5 - Comparison of specific and density impulses for selected propellants

Comparison of Specific and Density Impulses			
Propellant	**Cost per kilogram**	**Specific impulse**	**Density impulse**
Liquid O2 + liquid H_2	0.15 + 3.45 USD/kg	381	124
Liquid O_2 + RP-1	0.15 + 1.92 USD/kg	289	294
APCP (AP+HTPB+Al, solid)	5-6 USD/kg	277	474
N_2O + HTPB (hybrid)	0.15 + 3-4 USD/kg	248	290

This makes for some rather confusing implications. Apparently, it is not too informative to compare propellant performance on the basis either of specific or density impulses. We are forced to come up with an alternative way to compare performance of the propellants in question. The only remaining set of information that is readily available is the total impulse which is recieved by multiplying the thrust and the burn time. The resulting value can then be divided by total cost of propellants to receive the total amount of energy in N.s per USD expended. This seems to be the most objective figure we can come up with and one that we can make some valid observations and implications from. However, we must take notice that values of thrust and burn times (and total impulses by extension) are different for each type of vehicle. Therefore, we must accept these type-specific values with “a pinch of salt” to draw generalizations suitable for the entire propellant families.

The results are shown in Table 6 below.

Table 6 - Impulse per USD comparison for selected launchers

Impulse per USD comparison for selected launchers					
Vehicle	**Thrust**	**Burn time**	**Total impuls**	**Total launch costs**	**Impulse per USD**
Delta IV (1st Stage)	3140 kN	245 s	769.3 MN.s	375 mil USD	2 N.s/USD
Falcon 9 FT (1st Stage)	7607 kN	162 s	1232.3 MN.s	62 mil USD	19.9 N.s/USD
Space Shuttle SRBs	11520 kN	124 s	1428 MN.s	23.3 mil USD	61 N.s/USD
SpaceShipOne	88 kN	87 s	7.6 MN.s	300 000 USD [32]	25.3 N.s/USD

These are interesting results from a number of perspectives. First, it shows a shocking difference between costs of launches carried out by governmental organization as opposed to those of a private company. It also shows that the utilization of technically simple SRMs in booster applications is simply impossible to beat (for the time being) as they provide significantly higher impulse per USD expended than any other modern rocket systems. At present technological level and well beyond, no other engine is even close.

However, these results also clearly show the previously proclaimed superior efficiency of HREs. First of all, even though the SpaceShipOne was a vehicle with no real value from the payload perspective, the disproportionately cheap cost per launch is completely unprecedented. These figures prove that as soon as the issues with HRE scalability and combustion incertainties are solved, we may have access to an exclusive line of HRE-powered launchers that will be either significantly cheaper (by a margin of as much as 20%) than liquid powered launchers of Falcon 9 category, or will outperform the Falcon 9 Full Thrust at comparable costs.

Of course, to some extent, we compare the uncomparable. Note that numbers for both Delta IV and Falcon 9 FT only include their booster stages while the SpaceShipOne is technically a second stage. However, it is quite common that such comparisons end up in a sort of hyperbole. Moreover, there is currently no operational hybrid-powered booster stage so in order to extend parallels and draw conclusions, we have to make do with what numbers are available. All in all considered, while sober and cautious approach to the above numbers is required, they are promising and of interest nevertheless.

4.3 Possible Applications

Based on the above information and calculations, we can draw logical conclusions as per the area of applications that would be the most suitable for wide, systematic employment of HREs.

First and foremost, we need to properly recognize the traits of HREs that are the most beneficial. It is apparent that at the level they are presently at, HREs do not offer any improvement in performance figures over the conventional rocket propulsion – they achieve specific impulses that generally fall somewhere in the middle between SRMs and LREs.

That being said, HREs do offer significant advantages in terms of simplicity, controllability, versatility and safety of operation. This combination of properties is not possessed by neither SRMs nor LREs and makes them unique. It also makes them predetermined for certain applications – mainly those where instead of peak performance, safety and comfort are the primary concern.

We are on the brink of a commercial spaceflight age. Private companies are gearing up for this new era, already spending large sums of money on research and development of a new generation of spacecraft that will eventually allow private citizens to experience the feeling of spaceflight for relatively affordable cost. These spacecraft will be very different from all those we have known so far – they will not be built just for performance but for the experience. This makes for a very different philosophy which will resonate in all aspects of the vehicles‘ design.

It is also important to properly recognize all the limitations that will also shape the field of application of HREs. To properly account for them, we must first make sure we properly understand the simplicity and reliability that are being celebrated as HRE's most distinctive virtues.

When we talk about HREs being conveniently simple, for the most part we are talking about the engine's mechanical simplicity. It is true that this feature is without doubt inherently present in HREs, but it is also true that the internal processes of HREs are based on complex set of criteria which we have only started to understand in the last few years. HREs are arguably more difficult to design when compared to both SRMs and LREs based on this very set of complicated phenomena we are slowly uncovering, due to no small part thanks to advanced computational software, finite element modelling method and other modern software tools.

One of key issues that makes it complicated to perform any tests is that HREs in particular are very difficult to scale up, thus making it largely useless to perform any small-scale testing and then simply applying the results in the desired scale. The internal combustion processes just do not behave the same way. There is a host of issues that make it difficult to design a safe and efficient hybrid-powered large-scale launcher that might replace today's medium and heavy launchers used for manned spaceflight. First and foremost, fuel mass flow cannot be controlled directly as the regression rate is not simply based on a combustion chamber pressure. Both chemically and physically, hybrid fuel is highly dependent on a number of parameters which we still do not entirely understand, just as we are still struggling to fully comprehend the internal fluid dynamics in the HRE combustion chamber.

Unlike both SRMs and LREs, HREs do not suffer from pressure-related instabilities which may be catastrophic to conventional propulsion systems. However, several kinds of instabilities may occur in hybrids. For instance, they can be related to the boundary layer combustion which is hybrid-specific which is still not properly mapped. The complexity of hybrid combustion process and the very reason its modelling is quite difficult lies in the fact that while with LREs, the energy release occurs in a specific zone of the combustion chamber and in SRMs, the energy is produced at or near the fuel grain's surface, in HREs, this process occurs in a diffusive flame with significant changes in temperature and velocity along its surface that arises from the fuel grain surface. The energy release is therefore distributed along the length of the fuel grain port with different axial as well as radial values [33].

The overall list of complications associated with the development of HREs could go on and on and is well beyond the scope and complexity of this thesis. It is clear, however, that the incertainties are still numerous and while future research will most likely successfully overcome them, they are the main reasons why hybrids today are still a technology that is promising but its potential is still largely neglected.

For applications most suitable for HREs with present extent of knowledge regarding their internal processes, however, a few assumptions may be safely made.

With regard to their efficiency which is still not on par with advanced LREs used in today's launchers (most notably the Merlin 1D engines powering the SpaceX's Falcon 9 and perspectively the Falcon Heavy), it is clear that beyond small-scale applications such as amateur rocketry and sounding rockets where HREs are a popular choice, HREs can be successfully used in sub-orbital flights. There is sure to be a dynamically growing market in this field with development of commercial space trips being imminent.

The first contestant in this “commercial space race“, Virgin Galactic with its hybrid-powered SpaceShipTwo, was originally set to commence commercial passenger flights to the edge of space as early as 2016. However, the company was forced to reconsider this ambitious plan after a disaster during a test flight that occured on October 21st, 2014, resulting in a total loss of the spacecraft and death of one crewmember.

Nevertheless, the company recovered and eventually started flight testing of a second SpaceShipTwo named VSS Unity in September 2016. As of April 2017, first commercial flights have been expected to commence sometime in 2018.

As of May 2017, SpaceShipTwo (along with its predecessor, SpaceShipOne) is the only large-scale spacecraft powered by an HRE that has achieved flight.

A very promising area that could eventually mean a significant expansion of the application field of HREs is the paraffin wax propellant. Using this basically very common substance, the involved research teams were able to come up with designs that according to some of these researchers could, in some time and with a sufficient investment, provide specific impulses comparable to those of high-performance LREs (even engines as powerful as Russian-produced RD-180 used in Atlas V) at a fraction of their operating costs.

With such powerful hybrid motors in existence, it is quite possible we might witness a significant breakthrough in terms of cost-per-launch that would resonate and significantly influence the entire aerospace industry.

4.4 Role of Additive Manufacturing in Future Spaceflight Applications

Additive manufacturing, more commonly known as 3D printing, is a phenomenon which is slowly starting to bring breakthrough advancements in a multitude of scientific disciplines, to include medicine, automotive applications, architecture and others. It is only natural that researchers are coming up with ways to introduce 3D printing to aerospace industry as well.

It is not hard to see the areas belonging to the very scope of this thesis where additive manufacturing may come handy. Especially when considering often very complicated patterns and geometries of fuel grains that are delicately designed to bring the desired characteristics to the specific engine, there really seems to be a wide avenue of approaches where 3D printing might come as crucial in developing new patterns which might be previously very difficult to manufacture.

Indeed, such projects already exist and are enjoying a fair measure of success. One of them is spearheaded by a start-up company called Rocket Crafters Inc. based in Florida, USA. By specially shaped fuel grains, the company promises to enable a construction of engines that will enjoy the advantage of consumption of high energy fuel at accelerated pace while also solving the issue of generating vibrations inherently associated with hybrid rocket engines for a very long time [34]

However, start-up companies are not alone in recognizing potential of this new technology. Experiments with 3D printed parts have been going on for years. NASA has successfully developed and tested a 3D printed injector which successfully supplied oxidizer into a combustion chamber of an experimental engine during a test firing as early as 2013 [35]. Since then, experiments in design and implementation of 3D printed parts steadily grow and are producing incrementally encouraging results. NASA is now well on the path to building a completely 3D printed cryogenic rocket engine [36].

A major success has been recorded in 2014 when the first 3D printed combustion chamber liner made from copper was successfully tested by SpaceX [37]. It is the construction of similar mechanically highly sophisticated and complicated engine parts where the 3D printing is the most promising, reducing the unit costs greatly – compared to highly complicated process of production of molded or cast components, 3D printed counterparts are significantly cheaper and quicker to build.

Directly tied to manufacturing and operating costs is the question of savings in terms of vehicle mass. With additive manufacturing, even parts that are extremely complicated in shape and structure may be manufactured as monolithic structures, which, despite lacking any high strength connecting agents, are capable to withstand multiple-fold loads when compared to similar parts built conventionally. Or vice versa – additive manufactured parts built to exhibit the same strength and toughness values may be significantly lighter. In terms of launcher mass, these savings could be worth literally tens of millions of US dollars.

Owing to rapid technology development, even dimensional constraints are no longer a significant factor – recent experiments with extremely large machines being capable of 3D printing entire buildings act as proof of concept for construction of extremely large objects [38].

4.5 Aviationizing the Spaceflight

While I acknowledge the fact that by writing this chapter, I may be accused of moving into the realm of science fiction, I firmly believe that not only my line of thinking is quite plausible mid- to long-term, but it is possibly the only way forward for aviation in terms of increasing airspace capacity and thus, volume of air traffic in some of the most congested airspaces of the planet. My intention is to offer a vision of solution of some of the most painful problems of modern civil aviation through careful research and a little bit of out of the box thinking.

There are regions on this planet where airspace is so saturated that further growth in the volume of air traffic seems to be very difficult. While modern technology such as precision surveillance radars, ADS-B, GNSS and RNAV/RNP has enabled the air traffic to grow thanks to making smaller and smaller separation minima possible, this process cannot continue forever. Costly programs are being implemented around the world – the most significant being the SESAR in the EU and NextGen in the USA – to incorporate both procedural and technological solutions in order to increase aviation safety, minimize delays and improve cooperation between different entities taking part in the air transportation industry. Advanced concepts stemming from these programs, such as the free flight concept (autonomous navigation in specially designated portions of airspace with sophisticated technology onboard the aircraft guarding the separation minima), will help somewhat, but mainly in terms of optimizing routes and travel times and minimizing delays. It will increase the airspace capacity to some extent but it does have limitations. [44][45]

It is inevitable that at some point, researchers in the field of aviation will start focusing on utilizing parts of airspace where controlled flight is still possible, yet very rarely achievable thanks to inherent limitations of today's modern air-breathing engines. If there was a way to conduct a safe and economically viable flight above FL500, this would help significantly with all the presently seemingly unnegotiable issues – namely the capacity saturation and the need to push the separation minima even further.

One of the answers may apparently be to replace conventional air-breathing engines with other means of propulsion. Aviation already knows some alternatives – ramjets are hardly a new concept, being in operation since as early as 1960s.

However, ramjets have significant disadvantages. In order for them to function properly, the aircraft must first accelerate to sufficient speed. To attain it, other means of propulsion must be present, therefore making the ramjet impractical.

If we turn to rocket engines, however, we find that we have a propulsion system that is capable of functioning independently – indeed, it matters very little in what kind of outside environment we choose to operate it. Robustness of rocket engines allows for wide range of temperatures, pressures and speeds in which they may operate, as evidenced by their spaceflight applications.

It is my intent to show that while both solid and liquid fuel motors may be chosen and successfully used for this hypothetic operation, it is the hybrid motor that would be ideal for such application, all because of the reasons analyzed in the previous parts of this paper. It is apparent that HREs combine all the positive merits of both SRMs and LREs while negotiating most of both propulsor's flaws: HREs are very easily controllable, stable, they can deliver more than adequate specific impulses while achieving high level of safety of operation. That is far more that could be said of the first generation of jet engines, yet they turned out to be dominant means of aircraft propulsion in the second half of 20^{th} century and beyond.

It is my belief that in a few decades, air transport will change significantly - while present-day short- and medium-haul aircraft's successors will operate in "conventional" flight levels suitable for air breathing engines, long-haul air transportation may be almost exclusively a domain of new generation of aircraft, equipped predominantly with hybrid rocket engines. These aircraft won't be bound by present-day limitations and will be able to cruise well above FL660 which is currently considered to be the "roof" of the controlled airspace. Additionally, they will be able to travel at speeds that will enable them to significantly reduce times of transcontinental journeys. Another plausible alternative is point-to-point suborbital flight which already has been proven possible with current technology [47].

It is always difficult to predict long-term developments in any given field. Sometimes it may feel more like dreaming or guessing. Only time will tell if the vision outlined above is feasible or if the whole industry will develop in a completely different way. Even if that turns out to be the case, it is very likely that we will still witness the start of commercial sub-orbital flights in a foreseeable future. It is apparent, therefore, that base technology needed to make the above vision a reality is well within reach and as the capabilities of HREs will grow over time, so will the chance of fielding whole new transportation systems that will significantly influence the face of future's aviation.

5 Conclusion

To reiterate, the objectives set prior to starting works on this thesis were as follows:

- Overview of current state of affairs in rocket propulsion
- Pros and cons of hybrid propulsion
- Description of a specific engine
- Examples of possible use
- Comparison with conventional propulsion in terms of economics

Generally, all of the original objectives have been met, although I have experienced significant difficulties along the way. These were mainly associated with lack of publicly available information regarding speific formulae of solid and hybrid propellants as well as pricing of some of the propellant substances covered in this thesis. In some cases, I have had to resort to estimating some of the needed figures. These estimates have surely introduced additional systematic errors to the results presented in Chapter 4.

Moreover, the data I was able to retrieve were not consistent and fully comparable as some of the sources I was using were published over a period of as much as 10 to 12 years. As an example, prices of propellants are sure to have changed significantly over this period but unfortunately, I do not have sufficient training nor experience to properly estimate and account for the influence of inflation and other economical mechanisms. As a consequence, some of my numerical results are apparently inaccurate. However, for a purely informative comparison of different families of rocket engines, I believe that my research has still been sufficiently consistent with available data and has not been in opposition to any of the key principles that are valid for the world of rocket propulsion.

Regarding my personal experiential gain from my work on this thesis, I have progressed from the viewpoint of a person only vaguely interested in the world of spaceflight to the point that if presented with the opportunity, I would like to get more involved with the problematics of rocket propulsion, possibly with hybrid engines specifically.

Looking back at the volume of my research, I do realize that I have only barely scratched the surface as there is an overwhelming amount of high-quality research available, but due to a significant time pressure, I was only able to read through a fraction of the relevant works.

Moreover, lacking the background of rocket physics or associated sciences, my understanding of the more complex research papers is still somewhat limited.

If given a chance, I would like to continue my research in some way. It is apparent that hybrid engines are still a topic that is largely unexplored and bears a great potential for future research. It would be most beneficial if theoretical research could be directly coupled with experimental approach but it is clear that opportunities for such effort would be extremely limited in our conditions.

Possible way forward might be to establish cooperation with researchers from other faculties or universities in Czech Republic and Slovakia and to pursue joint projects together. In case that such cooperation will be established, I sincerely hope that this thesis may help to form a kind of overview needed to assess the current situation in the field or rocket propulsion and thus to contribute to the success of any future projects that may be started in effect.

In any case, I am looking forward to the introduction of the first HRE-powered vehicles, notably the SpaceShipTwo. I am sure I will continue to watch this field with great interest.

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