A Survey of Modern and Future Space Propulsion Methods

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Earth is the cradle of humanity, but one cannot live in a cradle forever.

Tsiolkovsky, 1911

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Firstly, I must thank Professor Paulo Gil for all the guidance, motivation and especially patience throughout this process.

To all my close friends and family, that put up with all my complaining and whining during this phase, thank you.

Resumo

A presente tese é o resultado de uma pesquisa de métodos de propulsão espacial, no contexto de uma rápida missão tripulada a Marte. Os recursos de diferentes tecnologias (tecnologias químicas, não químicas e avançadas de propulsão) foram explorados de forma a conseguirmos identificar as principais tendências, e avaliar o nível de desenvolvimento de cada uma delas. A partir da literatura consultada, encontrámos ”valores típicos” da relação impulso / peso de cada uma das diferentes tecnologias de propulsão. Adicionalmente comparámos os dois resultados para estabelecer, por fim, novos ”valores típicos”.

Para isso, foi criada uma base de dados com informação obtida através da ”Web of Science” e do repositório da NASA. No total foram identificados 249 sistemas de propulsão espacial, dos quais 133 são químicos, 90 são não químicos, e 23 são sistemas avançados de propulsão. A partir das informações reunidas na base de dados, conseguimos desenvolver 17 gráficos. Em cada um deles delineámos o impulso específico e a relação impulso / peso, de acordo com os diferentes parâmetros dos sistemas de propulsão de cada tecnologia.

Os resultados demostram que a propulsão química é a tecnologia mais avançada, sendo porém limitada por desempenhos de impulso baixos. A propulsão não química oferece um desempenho mais desenvolvido em comparação às tecnologias químicas. Não obstante, demonstrou ter um baixo nível de desenvolvimento nos sistemas de propulsão térmica nuclear e baixos valores de impulso / peso da propulsão elétrica. Já as tecnologias avançadas de propulsão revelaram estar ainda longe das atuais capacidades tecnológicas. O desenvolvimento deste tipo de sistemas permitiria cenários de missões espaciais inacessíveis no contexto das tecnologias atuais.

Palavras-chave: Pesquisa, Propulsão Espacial, Impulso Específico, Rácio de Impulso por Peso.

Abstract

This thesis is the result of a survey of space propulsion methods, in the context of a fast manned mission to Mars. The capabilities of different technologies (chemical, non-chemical and advanced propulsion technologies) are explored to identify trends and evaluate their level of maturity. From the accessed literature, ”rules of thumb” were found regarding the specific impulse and thrust-to-weight ratio of different propulsion technologies. Additionally, we compared both results in order to establish new ”rules of thumb”.

To accomplish this, a database was made with data found in the ”Web of Science” and in the NASA repository. A total of 249 space propulsion systems were identified, of which, 133 were chemicals, 90 non-chemicals and 23 advanced propulsion systems. With the information gathered in the database we were able to produce 17 scatter plots. These plotted the specific impulse and the thrust-to-weight ratio as a function of different propulsion systems parameter of the different technologies.

The results show that chemical propulsion is the more mature technology, however it is limited to low specific impulse performances. Non-chemical propulsion offers improved performance over chemical systems, but low maturity level of nuclear thermal propulsion systems and low thrust-to-weight ratio of electrical propulsion. Advanced propulsion technologies revealed to be too far from present technological capabilities. The development of these systems would enable space missions scenarios unreachable with present technologies.

Keywords: Survey, Space Propulsion, Specific Impulse, Thrust-to-Weight Ratio.

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Nomenclature

Greek symbols

η	Thruster efficiency.
γ	Heat capacity ratio.
ρ	Density.
ξ	Specific power.

Roman symbols

A	Area.
I_{sp}	Specific impulse.
$\bar{M}$	Molecular weight.
$\dot{m}$	Mass flow rate.
m	Mass.
p	Pressure.
P	Power.
R	Universal gas constant.
T	Temperature.
T	Thrust.
c	Effective exhaust velocity.
v	Vehicle velocity.
w	Weight.

Subscripts

0	Initial.
a	Ambient.

c	Chamber.
e	Exit.
in	Input.
l	Loaded.
t	Throat.
ext	External.

Acronyms

BEP Beamed Energy Propulsion.

CH$_4$ Liquid Methane.

EP Electric Propulsion.

EPPP External-Pulsed Plasma Propulsion.

FFP Fission Fragment Propulsion.

HET Hall-Effect Thrusters.

ICBM Intercontinental Ballistic Missile.

ICF Inertial Confinement Fusion.

LEO Low Earth Orbits.

LH$_2$ Liquid Hydrogen.

LHC Liquid Hydrocarbon.

LOX Liquid Oxygen.

LPRE Liquid Propellant Rocket Engine.

MAA Matter-Antimatter Annihilation Propulsion.

MCF Magnetic Confinement Fusion.

MMH Monomethylhydrazine.

MPD Magnetoplasma Dynamic.

NRT New Rules of Thumb.

NTO Nitrogen Tetroxide.

NTP Nuclear Thermal Propulsion.

PPT Pulsed Plasma Thrusters.

RF Radio-Frequency.

RT Rule of Thumb.

RTG Radioisotope Thermoelectric Generators.

SL Sea Level.

SLV Space Launch Vehicle.

SPS Space Propulsion System.

SRB Space Shuttle Solid Rocket Booster.

STP Solar Thermal Propulsion.

TRL Technology Readiness Level.

UDMH Unsymmetrical Dimethylhydrazine.

V Vacuum.

VASIMR Variable Specific Impulse Magnetoplasma Rocket.

Chapter 1

Introduction

1.1 Objective

A manned mission to Mars is one of the next major global objectives in space exploration. There are several new space propulsion technologies being developed promising improvements over the modern systems in use. In this work we make a survey on modern and future space propulsion methods, to evaluate their maturity and development trends to help accomplish a manned mission to Mars.

1.2 Space Propulsion

The purpose of a propulsion system is to produce force which is used to change the motion of a body with respect to an inertial reference frame. Propulsion systems used on Earth use their surrounding conditions, solid or fluid. For example, sail boats use the air pressure for propulsion and, when walking, the push to accelerate a body is the result of the friction between the soles of the shoe against the ground. As a result of space being a vacuum and with no standing places, Space Propulsion System (SPS) need to have unique characteristics in order to give an impulse to a vehicle [1].

There are several types of SPS. Most of them produce thrust T by ejecting matter (called propellant), stored in the vehicle. As an example, chemical propulsion systems often use a fuel and an oxidizer as propellants. When combined and ignited they release enormous amounts of heat, the resulting thermodynamic energy from the expanded hot gas is then turned into kinetic energy using a convergent-divergent nozzle that accelerates the gas to supersonic velocities. By Newton's third law of motion, the hot exhaust leaving the nozzle with high speed applies an equal and opposite reaction to the vehicle, thus creating the accelerating force. Some Electric Propulsion (EP) systems, do not need to rely on the internal energy of chemical propellants, instead they use electric and/or magnetic fields to accelerate charged molecules or atoms, at very low densities, with the same effect [2].

1.2.1 History of Space Propulsion

Rockets have been in use since the year of 970, when the Chinese used gunpowder and bamboo tubes as fire works [2]. Although also widely used as a bombardment weapon and signaling tool for almost 1000 years, significant development of the rocket only took place in the early twentieth century, thus marking the beginning of modern rocketry [2]. With three pioneers standing out: Konstantin E. Tsiolkovsky and Hermann J. Oberth being the foremost pioneers of rocket flight theory, and, Robert H. Goddard who in 1926 launched the first successful liquid propellant rocket [3].

October 1942 marks an important historical moment in space propulsion with the first successful flight of the German missile V-2. The V-2 was the first large Liquid Propellant Rocket Engine (LPRE) built. Using Liquid Oxygen (LOX) and alcohol, offered a thrust of $249\ \mathrm{kN}$ at Sea Level (SL), surpassing the thrust of any other engine built at the time by a factor of ten. To date still remains the only large LPRE produced in a relatively large quantity, with over 5000 units built [3]. Even though about $25\ \%$ of the V-2 fired against England failed to reach or explode on their target, propulsion system failures did not occur very often. The lack of success was mainly due to failures in flight control, warheads that did not explode or for breaking apart during flight or reentry [4].

Both the first Intercontinental Ballistic Missile (ICBM), the R-7, and the first orbiting artificial satellite, Sputnik 1 in October 1957, were achievements from the Soviet Union. These accomplishments were made possible by the two highest thrust LPRE at the time, the RD-107 and RD-108. Developed in parallel they had very similar designs and throughout the years were built at least eight different version of each, nowadays they are still in use. Likely, making them the most utilized satellite booster engine in the world, with over 1630 rockets launched and the longest living rocket-engine project, with a success rate of $97.5\ \%$ [5]. Sputnik 1 Space Launch Vehicle (SLV), had four droppable RD-107 booster engines and one RD-108 sustainer in the center of the SLV. RD-107 and RD-108 engines had four fixed thrust chambers for thrust plus two and four verniers, respectively. Verniers are smaller hinge mounted thrust chambers that provide flight-path control and some of the axial principal thrust, usually $10\ \%$ to $20\ \%$ of the total thrust with analogous design, the main difference beside the number of verniers used was the slightly lower chamber pressure of the RD-108, granting longer working duration of the engine at the cost of a lower thrust. Parameter values for a version of each engine (different versions have different values) and the rest of the SPS depicted in this subsection can be seen in table 1.1. The propellants used were Rocket Propellant-1 (RP-1 — see Section 2.1) an highly refined form of kerosene, combined with LOX [6].

The year of 1969 was marked by the first manned lunar mission, the Apollo 11 lunar module reached the surface of the moon on the 20th July of that year. This mission was carried by the Saturn V, the largest SLV ever built, which performed an eight day journey carrying three men. To achieve sufficient thrust to lift the $3 \times 10^6\ \mathrm{kg}$ vehicle, the first stage clustered five F-1 engines, based on the aforementioned German V-2 engine [4]. Each produced $6770\ \mathrm{kN}$ of thrust at SL using LOX and RP-1, for more than a decade was the highest thrust any SPS could provide, nowadays is still one of the highest thrust engines ever built. The second and third stages were marked by yet another important breakthrough, the first operational use of Liquid Hydrogen (LH_2) as propellant, combined with LOX. Saturn V second stage

Table 1.1: Key parameters of historical SPS at Sea Level (SL) and Vacuum (V), taken from [4, 5, 7, 8]

SPS	Thrust, kN	Specific Impulse, s	Dry mass, kg	Propellants	Date, year
V-2	250(SL)/285(V)	203(SL)/242(V)	930	LOX/Alcohol	1942
RD-107	814(SL)/1000(V)	256(SL)/313(V)	1190	LOX/RP-1	1957
RD-108	745(SL)/941(V)	248(SL)/315(V)	1278	LOX/RP-1	1957
J-2	1023(V)	425(V)	1567	LOX/LH_2	1966
F-1	6770(SL)/7775(V)	265(SL)/305(V)	8444	LOX/RP-1	1967

clustered five J-2 engines while the third stage used one, with each J-2 providing more than 1000 kN thrust in vacuum [7].

1.2.2 Impact of the Propulsion System on Vehicle Performance

To evaluate and compare the impact of different SPS have on vehicle performance it is important to gain insight into the space propulsion fundamentals.

Any vehicle that produces thrust by ejecting matter follows the same laws of momentum conservation. Assuming there are no external forces acting on the vehicle, the velocity for matter ejecting vehicles can be given as a function of the effective exhaust velocity c, the vehicle mass at ignition m_0 and its current mass m, as [2]

$$v = c\ln(m_0/m). \tag{1.1}$$

This is the Rocket Equation, as the vehicle expels propellant in one direction its velocity increases in the opposite. The more propellant expended, the greater the mass ratio m_0/m becomes and so does the vehicle velocity v. For a fixed amount of propellant expended, the vehicle velocity is directly proportional to the effective exhaust velocity c, defined as [6]

$$c = V_e - \frac{p_e - p_a}{\dot{m}} A_e. \tag{1.2}$$

The effective exhaust velocity is the important parameter of any mass ejection system, a higher number often indicates better engine overall performance. This parameter depends mainly on V_e, which is the velocity of the exhaust gas in relation to the vehicle. The second term of the equation includes the mass flow rate of the propellant $\dot{m}$, the effect of the ambient pressure p_a and the exhaust gas pressure p_e at the end of the nozzle with area A_e [9].

Optimal nozzle expansion is when $p_e = p_a$, as it happens when the vehicle is in vacuum and the exhaust gas is expanded to zero pressure. Assuming there are no external forces acting on the vehicle, the second term on the right of equation (1.2) equals zero and the effective exhaust velocity equals the exhaust velocity ($c = V_e$). In this conditions, if we start from zero velocity the final vehicle velocity depends only on how much of the vehicle is propellant and on the exhaust velocity. The vehicle can match it's own exhaust velocity if the mass ratio $m_0/m = e$ and even surpass it if the ratio is higher [2].

The effective exhaust velocity is usually stated as its equivalent, the specific impulse I_{sp}. This parameter is equal to the effective exhaust velocity divided by the standard Earth's surface acceleration of gravity g_0 ($I_{sp} = c \div g_0$), its unit "seconds" is the same in the imperial and metric systems. It can be treated as a measurement of the propellant efficiency, as it is the amount of momentum gained by the vehicle per sea level weight unit of propellant expended. A higher I_{sp} means smaller amounts of propellant needed to perform the same maneuver [6].

The total thrust force acting at the vehicle's center of mass T is given as [6]

$$F_{ext} + \dot{m}V_e - (p_e - p_a)A_e = T, \tag{1.3}$$

where F_{ext} are the external forces acting on the vehicle. The second term is the *momentum thrust*, it's the product of the mass flow rate of propellant depleted with a velocity relative to the vehicle V_e, it is the biggest contributor to the vehicle's thrust. The third term is the *pressure thrust*, includes the effects of the atmosphere with pressure p_a and the pressure of the exhaust gas p_e. The thrust-to-weight ratio T/w gives the acceleration, in multiples of g_0, that the SPS is capable of giving to its own loaded propulsion system mass. For constant thrust, the acceleration of the vehicle increases as the propellant is burned and the vehicle mass decreases. Reaching its maximum right before the thrust termination. This parameter is useful in comparing different types of SPS and for identifying launch capability. A ratio above one is required to overcome Earth's gravity, otherwise the rocket will not lift [6].

1.2.3 SPS Types and Development

Each mission type can require very different types of propulsion systems to satisfy distinctive needs. Interplanetary manned missions need to be fast and safe to the crew, while robotic missions to the outer planets of the solar systems can endure continued low thrust for a long time in order to accelerate to the very high velocity required. In addition, along a given mission, the propulsion requirements can change dramatically, the lift-off requires a system which can provide a considerable amount of thrust in a very short time period, while station keeping requires small amounts of thrust over a long time period. As a result of this disparities there is no type of propulsion that will suit all mission classes or even all missions of the same type. It is therefore relevant to be able to differentiate and categorize the different types of SPS [10].

One important definition is of the "Technology Readiness Level" (TRL). It is a measurement system used to determine the technical maturity of instruments and spacecraft sub-systems. It is defined by ISO standard 16290 on a scale of 1 to 9, with TRL-1 being the lowest and TRL-9 the highest level of maturity. The meaning of all TRL levels are presented in figure 1.2 [11].

SPS Main Basic Functions

As the needs of each mission vary according to the phase they are in, we will separate SPS in their three main basic functions, this way we can understand better the capability of any given type of SPS to perform each function. The main basic functions of SPS can be divided as [12]:

- Lifting the launch rocket and its payload from the surface of the Earth and delivering the payload

Table 1.2: TRL definition [11]

TRL	Level Description
1	Basic principles observed and reported
2	Technology concept and/or application formulated
3	Analytical and experimental critical function and/or characteristic proof-of-concept
4	Component and/or breadboard functional verification in laboratory environment
5	Component and/or breadboard critical function verification in relevant environment
6	Model demonstrating the critical functions of the element in a relevant environment
7	Model demonstrating the element performance for the operational environment
8	Actual system completed and accepted for flight ("flight qualified")
9	Actual system "flight proven" through successful mission operations

into Low Earth Orbits (LEO). Generally, thrust is the important parameter. A T/w above 1 is required to launch from Earth's surface. These are generally called boosters or first stage engines.

- Transfer payloads from LEO into more energetic orbits, such as interplanetary or interstellar, and retro-action while approaching the moon or a planet. The need to accelerate to very high velocities using less propellant makes the specific impulse the important parameter. Depending on the mission the T/w ratio requirement changes drastically. In a robotic mission to the outer solar system, where transit time in an unimportant factor, can be performed by a very low T/w ratio SPS. On the other hand a manned mission, where transit time is a key factor, requires a high T/w ratio. Additionally, some missions require variable thrust ad engine restart capability.
- Small maneuvers and reaction control, this includes station keeping, docking maneuvers and spacecraft orientation. The necessary thrust depends on the vehicle mass and speed needed to perform the action, usually can be done with a low T/w ratio. An high I_{sp} is beneficial to reduce the required propellant mass. Some maneuvers often need thousands of thrust pulses, restarting and stopping is essential for systems performing this functions.

There are several ways to differentiate SPS categories. For example, according to the size, type of propellant, by the energy source or by their basic functions. Following the *2015 NASA Technology Roadmap - TA2* [13], in this work we will categorize SPS as: chemical propulsion, non-chemical propulsion and advanced propulsion technologies, each of them having its own subcategories. Different types of SPS have completely different T/w and I_{sp} capabilities, with each technology having their strengths and weaknesses. Depending on the mission or flight application the requirements of what the propulsion system needs to provide change drastically [6]. Here we will introduce the basic characteristics of each family and in which they excel. In chapter 2 we will see their structure and how they work in more detail.

Chemical Propulsion

Chemical propulsion converts the internal energy of the propellants into kinetic energy. The combustion of the propellants increases the temperature and pressure of the product gases which are then expanded in a converging-diverging nozzle to supersonic velocities, generating thrust. Chemical propulsion is the more mature technology and can reach T/w ratios superior to 100, presently is the only viable option for first stage engines. On the other hand, chemical propulsion is the lowest performance technology with a I_{sp} limited to several hundreds of seconds. This is due to the energy in chemical propulsion coming solely from the chemical reaction of the propellants. The highest possible I_{sp} is achieved when using energetic chemical reactions with low mass exhaust products. Chemical propulsion can be further classified according to the physical state of the propellant: solid, liquid, hybrid (solid + liquid) and gel [14].

Non-Chemical Propulsion

To achieve high performance the energy cannot be exclusive out of chemical reactions. Non-chemical propulsion systems are those that use electrostatic, electromagnetic, field interactions, fission reactions, photon interactions, or externally supplied energy to accelerate a spacecraft. These propulsive technologies offer improved performances over chemical propulsion, however, the T/w ratio is usually very small [9]. Non-chemical propulsion can be further grouped into the following categories: electric propulsion, solar and drag sail propulsion, thermal propulsion and tether propulsion [13].

Advanced Propulsion Technologies

Advanced Propulsion Technologies are all concepts that are at TRL below 3. The development and breakthrough of this technologies could enable new types of space missions, today unreachable even with unlimited costs. This is achieved either by increasing the performance, or reducing propellant needs by reducing dry mass or mission velocity. This group of technologies include any basic principle or concepts formulated which are simply beyond present technical capabilities. The concepts detailed in chapter 2 are: beamed energy, advanced fission, fusion reactors and electromagnetic drive [13].

1.3 Manned Space Flight

A manned mission to Mars is the ideal benchmark for new types of SPS. As it is the next milestone in space exploration and minimizing the voyage time and the mission mass are the two key barriers for humans to explore Mars [15–18].

Manned space travel is more complicated than robotic missions. It is essential to ensure equipment and resources to maintain a healthy and safe crew during the travel to and from Mars, as well as during the stay on its surface. NASA's Human Research Program has been studying the adverse effects of a manned mission to Mars. The exposure to solar and galactic cosmic radiation have harmful physiological effects on humans as well as zero-g voyages, both become more demanding with longer exposure times. Space propulsion systems capable of performing fast transfers would mitigate the danger of a manned mission to Mars, as the trip time decreases so do the adverse effects to human health and performance [19]. Reducing transfer times implies an increase of propulsive energy, this extra energy comes from increasing the propellant mass, making the two key barriers reducing voyage time and reducing vehicle mass mutually exclusive. To reduce the trip duration and the mass required it is necessary to develop systems capable of delivering higher overall performances and efficiencies of those currently available. To achieve this in future manned missions to Mars and other space exploration or scientific discovery missions is fundamental to develop advanced propulsion concepts [16].

1.4 A Survey on Space Propulsion Systems

The performance of today propulsion systems makes manned planetary missions prohibitively long. In this work we analyze the different existing and proposed technologies in SPS with the objective to understand how to circumvent constraints to perform a fast transfer to Mars. In order to achieve this, the core of this work is divided into two main parts: the first is the construction of the database of SPS, and the second one is the analyses of the database, where we identify the trends in propulsion development and the performance capabilities of different technologies. The gathered data will help deduce if it is possible, and in what conditions, for each technology to carry out a transfer to Mars.

Chapter 2

Rocket Propulsion Fundamentals

2.1 Chemical Propulsion

2.1.1 Chemical Propulsion Fundamentals

In Section 1.2.2 we saw how, both, the velocity and the mass flow rate of the exhaust gases have an impact in the velocity and thrust of the vehicle. Equations (1.1 and 1.3) apply to all systems that use gas expansion as the mechanism for ejecting matter at high velocities. Here we present which characteristics of SPS have an impact on the mass flow rate and the exhaust velocity. The following equations are an ideal one-dimensional nozzle flows, that represent a simplification of the real aerothermochemical behavior [2].

The exhaust velocity can be expressed in terms of independent parameters such as the combustion chamber temperature T_c and pressure p_c, as [2]

$$V_e = \sqrt{\frac{2\gamma}{(\gamma-1)}\frac{RT_c}{\bar{M}}\left[1-\left(\frac{p_e}{p_c}\right)^{(\gamma-1)/\gamma}\right]}. \tag{2.1}$$

Where R is the universal gas constant, $\bar{M}$ is the molecular weight of the exhaust gases and the index γ is the ratio between the specific heat at constant pressure and the specific heat at constant volume, of the exhaust gases. The value of γ has a significant impact on the exhaust velocity. For a SPS exhaust the value is generally smaller than the value for air at normal temperature and pressure which is 1.3. Typical value for the exhaust gases of SPS is around 1.2 [2].

A particularity of chemical SPS, is that the size of the propulsion system does not have an impact on the exhaust velocity. In fact, a small SPS used for station keeping can have the same I_{sp} as a 1 MN used to lift an heavy vehicle. The parameter that is dependent on the size, and consequently on the mass, of the SPS is the mass flow rate. The mass flow rate can be expressed as [6]

$$\dot{m} = p_c A_t \sqrt{\left(\frac{2\gamma}{(\gamma-1)}\right)^{\frac{(\gamma+1)}{(\gamma-1)}}\frac{\bar{M}}{RT_c}} \tag{2.2}$$

where the cross-sectional area of the throat of the nozzle A_t is the only new parameter.

For (2.1) and (2.2) to give a good approximation of the real aerothermochemical behavior, the following criteria must be met [6]:

- The working fluid in the chamber needs to be homogeneous, a good propellant injection system can approach this condition.
- The propellant flow rate must be constant. Fluctuations in the propellant flow rate and pressure are usually under 5 % and can be neglected.
- The inner walls of the nozzle need to be smooth, so that wall friction losses are negligible.
- The heat losses to the walls in the chamber are usually bellow 1 % of the total energy, for that reason it can be neglected. The exception is for very small chambers, where the heat losses are higher.

When these requests are met, equations (2.1) and (2.2) give reasonable approximations with measured values of chemical SPS usually between 1 % to 6 % below the theoretical predictors, and has become recurring practice in designing new propulsion systems [6].

From (2.1), we can see that increasing the temperature and pressure of the combustion in the chamber increases the exhaust velocity. It also depends inversely on the square root of the molecular weight of the exhaust gases $\bar{M}$. The combustion temperature and molecular weight are both a consequence of the propellants of choice. The ratio between the thrust chamber pressure and the pressure at the exit of the nozzle p_e/p_c is determined by the design of the nozzle. The exhaust velocity also depends on the ambient and nozzle conditions, although the main determinant of exhaust velocity is the propellant combination [6].

In LPRE the highest combustion temperature for a fixed propellant combination happens when the ratio of fuel and propellant is at, or close to, the stoichiometric mixture ratio. This is when the quantities of the reactants are such that all of the reactants are consumed and none remain after the chemical combustion. In the case of SPS using a combination of LOX/LH_2 as propellants the stoichiometric ratio is 8:1, however in most cases this ratio is not used. Although the obtained combustion temperature would be greater it is usually preferable to increase the amount of LH_2. As LH_2 has a lower molecular weight than LOX, it decrease the mean molecular weight of the exhaust products. Even though the chemical reaction is not complete and fuel molecules remain unreacted, the decrease of the mean molecular weight of the products can be advantageous to a certain extend, where the drop in the combustion temperature remains small. For LPRE using the propellant combination LOX/LH_2 the mass ratio is, usually, between 4.5:1 and 6:1 [2].

The main factor influencing the mass flow rate is the product of the throat area and the pressure in the combustion chamber. The throat area is an important parameter of the SPS, with importance on the thrust and on the size of the propulsion system. Unlike the exhaust velocity, the mass flow rate decreases with the combustion temperature and becomes larger with the molecular weight. In propulsion systems in lower stages, where thrust is the main factor, may be beneficial to use high molecular weight propellants. This allows a larger thrust through a smaller throat area in a smaller propulsion system, at the cost of a lower exhaust velocity [6].

The nozzle is an important part of any SPS that is dependent on a gas expansion to produce thrust, as it influences the p_e. The exit pressure depends on the length of the nozzle, there is a local maximum thrust at the nozzle throat and drops along it. For a maximum exhaust velocity, and consequently thrust, the nozzle should be optimized so that p_e equals the ambient pressure p_a, this is called optimal nozzle expansion. As the atmospheric pressure decreases with altitude, optimal nozzle expansion is only possible at one point. For a launcher the nozzle is usually short with a high exit pressure to match the high ambient pressure. For a maximum thrust in altitude the nozzle should be longer, in fact for the vacuum of space the nozzle should be infinite so that the exhaust is expanded to zero pressure [2]. A measurement of the gas expansion is the nozzle area ratio A_e/A_t. The area ratio of a nozzle is the ratio of the exit area to the throat area of the nozzle. For low altitude operations the ratio is usually between 3 to 30. This nozzles have high exit pressure combined with low atmospheric drag. For nozzles designed to fly at high altitudes the ratio need to be higher in order to decrease p_e to the low atmospheric pressure found at high altitudes. These nozzles have a ratio typically in the range 40-200. Nozzles designed to flight only in vacuum would need to be infinite to achieve optimal nozzle expansion, in practice there is no benefit from additional length and ratio after a certain point as the extra mass outweighs the benefits of the extra thrust and exhaust velocity [6].

2.1.2 Types of Chemical Propulsion

Chemical propulsion can be further classified according to the physical state of the propellant: solid, liquid, hybrid (solid + liquid) and gel [14].

Solid propellant SPS, or solid rockets, is the most mature propulsion technology in use [20]. In solid rockets the oxidizer and fuel are stored directly in the combustion chamber in solid state. Generally solid rockets are simple in design with few or no moving parts making them easy to operate, resulting in low cost and high reliability, around 99 % [21]. However, there are two main disadvantages: the thrust cannot be controlled in flight and the specific impulse is limited, lower than 300 s [2], making the usage of this type of SPS limited. They are widely used in military applications as solid propellants can be stored safely and ready to be used for 10 to 30 years. Solid propellants have high density, which offers the possibility of creating a small and compact rocket, with less drag. This characteristic is particularly beneficial for 1st stage boosters, where the atmosphere drag is more accentuated. The thrust can vary widely, from small thrusters providing 2 N to large rockets with 1.2×10^7 N of thrust as is the case of the Space Shuttle Solid Rocket Booster (SRB) [6].

Liquid propellant SPS, or LPRE, even though have a higher complexity when compared to solid propellant SPS, can provide a higher I_{sp}. Most LPRE also have the capability to be randomly throttled, stopped and restarted during flight [22]. This type of SPS can use several types and combinations of propellants. Monopropellant SPS use a single chemical compound and are broadly used for low thrust orbit corrections due to its simple design. Usually the propellant used is hydrazine, with specific impulse generally inferior to 250 s but it can be stored for more than 15 years and it is easily decomposed, which means that it has multiple restart capability [1]. The most common type of all SPS are liquid bipropellant rocket engines, this type of systems use a combination of an oxidizer with a fuel, the propellants are stored separately and are mixed inside the combustion chamber. There are three main types of combinations of liquid bipropellants: Liquid Oxygen-Liquid Hydrogen (LOX/LH_2), a combination of storable propellants and Liquid Oxygen-Liquid Hydrocarbon (LOX/LHC) [2].

LOX/LH_2 are both cryogenic propellants, that is liquefied fuels at very low temperatures (16 K to 127 K). Cryogenic propellant systems are complex in order to prevent leakages of propellant and to maintain the very low temperatures needed. However, this type of bipropellant has very high specific impulse, for a chemical SPS, achieving values above 450 s in vacuum. LH_2 has very low density (about 0.07 kg/m^3 at 20.4 K), and it evaporates when kept for long periods of time in space [1]. The low density combined with the hard vacuum storability means there is a need for large and bulky propellant fuel tanks, which comes with additional structural mass. The high specific impulse is specially beneficial for upper stages performing high velocity transfers. However, LOX/LH_2 propellants are also used in lower stages [4].

Propellants are considered storable when are liquid at ambient temperatures and low pressures, with the capability to be kept for long periods of time in sealed tanks [3]. The most used combinations in today's systems use, as oxidizer, Nitrogen Tetroxide (NTO) combined with one of two hydrazine based compound fuels, Unsymmetrical Dimethylhydrazine (UDMH) or Monomethylhydrazine (MMH). Both NTO/UDMH and NTO/MMH combinations provide similar specific impulse performances, around 340 s with an area ratio of 40 [1]. Some authors call this combinations hypergolic for they ignite spontaneously when combined, making multiple restarts easy to accomplish [1]. The long-term storage and instant readiness to start characteristics makes this kind of bipropellant combination a appropriate option for attitude, orbit correction maneuvers and interplanetary transfers. With the capacity to achieve very high thrust this kind of SPS are also used in first and second stage boosters [2].

A large variety of Liquid Hydrocarbon (LHC) fuels and related chemicals were investigated. However, most SPS using LHC work with petroleum derivatives. Many early LPRE adopted gasoline and kerosene as fuel, this had the drawback of forming carbon deposits on the inner walls of flow passages under certain conditions. In order to reduce the formation of carbon deposits a refined form of kerosene, RP-1, was developed [4]. LPRE using a combination of RP-1 and LOX have a vacuum specific impulse of approximately 357 s, this is lower when compared to LOX/LH_2 propellants but offers a higher average density. Another noteworthy LOX/LHC propellant combination is LOX with Liquid Methane (CH_4), capable of reaching I_{sp} superior to 370 s. LOX/LCH_4 propellant combination offers a compromise between the low density and high performance of LOX/LH_2 and the high density and low performance of LOX/RP-1. Liquid hydrocarbons are easily storable, with no need for active refrigeration in space, this makes the design of the SPS simpler but combined with LOX does not form a hypergolic combination, thus restarting the system is more complicated. The aforementioned characteristics makes this kind of systems specially beneficial for high thrust low stage boosters and some second stages systems [1].

Hybrid propulsion systems is relatively simple, generally uses a combination of a solid fuel with a liquid oxidizer, with many suited propellant combinations available. Like solid systems, this method grants higher levels of safety and low cost production due to lower fabrication and operational costs, when compared to bi-propellant liquid systems, and, unlike solid systems, hybrids have the possibility of re-ignition and thrust throttle ability [23]. The mixture between solid and liquid propellants provides higher I_{sp} than solid SPS. Although, due to the lower density of the liquid propellant, the volume is higher. The safety, low cost and versatility of hybrid SPS makes them suited for educational purposes and most data and tests made are of low thrust applications [24].

When compared with the other chemical propellants (solid, liquid and hybrid), gel systems promise to offer performance improvements (increased I_{sp} and fuel density), by adding low-molecular-mass metal powder in the gel, and diminished problems when handling the reactive and toxic propellants [25]. Despite of these apparent advantages, gel systems are still not in use due to a low maturity level.

Both solid and liquid propellant propulsion systems have been researched and used since the dawn of space travel, with both technologies reaching a very high level of maturity, most of these systems are higher than TRL-6. Gel propellant propulsion systems, on the other hand, is a newer technology with only flight demonstration tests done, it has a TRL-6 as of 2010 [26]. With less investment and research, gel technology does not possess the maturity needed to be a viable choices when opting for a propulsion system. Once the present potential problems are overcome (boil-off and shifting in propellant loading), gelled propellants will become an option for space propulsion [27].

As one of the key objectives of this work is to determine the future possibilities of SPS for manned flight mission we will take the TRL into account when studying each technology, trying to make an educated guess of how they will evolve in the future.

2.2 Non-Chemical Propulsion

2.2.1 Non-Chemical Propulsion Fundamentals

Non-chemical propulsion does not use the internal energy of the propellants to produce thrust. Instead it uses a separate energy source for heating and/or directly ejecting propellant, this type of propulsion encompasses both thermal and non-thermal systems. Possible energy sources are solar sails, Radioisotope Thermoelectric Generators (RTG) and fuel cells. The relations (2.1 and 2.2) apply to non-chemical propulsion systems that use thermal processes. However, to understand and compare non-thermal SPS new concepts are needed. The mass of the power supply can be high when compared to chemical propulsion, and the exhaust velocity depends, not only on the nature of the propellant, but also on the amount of power delivered and how the SPS transfers momentum to the propellant [6].

The thruster efficiency η is defined as the ratio of the energy carried away per second by the exhaust to the total electrical power input P_{in} applied to the thruster. η is [6]

$$\eta = \frac{\frac{1}{2}\dot{m}V_e^2}{P_{in}}. \tag{2.3}$$

Thruster efficiency, takes into account all energy losses which do not result in propellant kinetic energy. It is a measurement of how efficiently the electrical power and propellant are used in the production of thrust [6].

Another important parameter is the power-to-mass ratio or specific power ξ

$$\xi = \frac{P_{in}}{m_l}, \tag{2.4}$$

where m_l is the mass of the loaded propulsion system, including the power source. In some cases, as it is with electric propulsion, a massive and relatively inefficient energy source is needed, granting specific powers much lower than that of chemical propulsion, where the energy comes from the propellant combustion. The ξ is a measurement of utility of the mass of the SPS, including power source [6].

With the thruster efficiency and the power-to-mass ratio defined, we have for the exhaust velocity V_e

$$V_e = \sqrt{\frac{2\eta P_{in}}{\dot{m}}} = \sqrt{\frac{2\eta\xi m_l}{\dot{m}}}. \tag{2.5}$$

The exhaust velocity, and consequently the specific impulse, are no longer free parameters, they are fixed by the electrical power input and the mass flow rate. Increasing the exhaust velocity requires an increase in the power supplied to the propulsion system or a decrease in $\dot{m}$, which in turn decreases the thrust. This means that for a constant P_{in} the thrust is inversely proportional to the specific impulse [28].

2.2.2 Types of Non-Chemical Propulsion

Non-chemical propulsion provides thrust without chemical reactions and combustion, it is divided into: Electric Propulsion (EP), solar and drag sail propulsion, thermal propulsion and tether propulsion [13].

Electric propulsion is a technology based on accelerating matter using electric forces. The energy used by this systems are not provided by propellants, as it happens in chemical propulsion, but from energy sources (batteries, solar radiation or nuclear). Usually the thrust is low, normally less than 1 N, but the performance is comparatively high [13]. This means a considerable reduction in propellant mass needed and the possibility to perform longer duration missions. There are three different types of EP technologies: electrothermal, electrostatic and electromagnetic [6].

Electrothermal thrusters use electric energy to heat propellant, this can be accomplished either by resistojets, which use solid resistors to dissipate power and heat the propellant by convection, or by arcjets, using electric arcs to ionize the propellant for same effect. In both system types, the heated propellant then goes through a nozzle, expanding and accelerating to supersonic speeds, thus producing thrust. Resistojets offer two main advantages, they are able to use a large array of propellants and provide with the highest thruster efficiency of all EP technologies, up to 85 %. This means resistojets grant the highest thrust-to-power ratio, with a wide range of working power levels that go from a few watts to tens of kilowatts, however they generate a limited maximum specific impulse of around 300 s. Arcjets can achieve more than double the specific impulse of resistojets, nevertheless more than half of the electrical power is lost, most for residual internal energy and ionization, making the usual thruster efficiency bellow 50 %. Both resistojets and arcjets in use today prefer hydrazine as the propellant of choice. Electrothermal technology is the most simple and common type of EP, usually, it is used to perform small maneuvers and reaction control, with both arcjets and resistojets having achieved a TRL-9

[29].

The propulsion systems that produce thrust using electrostatic (or Coulomb) forces to generate ions and accelerate them to very high exhaust speeds are named electrostatic thrusters [30]. This kind of technology, often called ion thrusters, have specific impulse values usually between 2000 s to 10 000 s, limited thrust below 0.5 N and a very long demonstrated operational life time of 30 000 h [31]. The preferred propellant is xenon, for being the highest molecular mass stable inert gas, it is easily stored and does not present any problems of condensation or toxicity. This characteristics makes ion thrusters attractive for high velocity missions and station keeping applications. Like electrothermal technology, ion propulsion enjoys a TRL-9 [32].

Electromagnetic thrusters accelerates the propellant gases that have been heated to a plasma state using electromagnetic forces generated in a very high current plasma discharge [32]. Plasma gases are with temperatures normally above 5000 K, that are affected by electric and magnetic fields. The current plasma discharge ionizes the propellant and it produces an intense magnetic field that in turn pushes the ions in the plasma out of the vehicle at high velocities. There are several types of electromagnetic thrusters [6].

Magnetoplasma Dynamic (MPD) thrusters draw thrust from both electromagnetic acceleration and electrothermal expansion of the propellant, although the latter is generally insignificant at high power levels. MPD systems have typically worked with low efficiencies of 25 % to 35 % with moderate specific impulse around 2000 s. To achieve higher efficiencies high power is required (above 200 kW), which is currently an order of magnitude larger than the power sources used on current spacecraft. The difficulty to achieve steady high power makes MPD uncompetitive with other propulsion systems and in recent years not much investment was done in this technology. With some tests and prototypes it is currently at TRL-6. The low efficiency at low power limits the use of MPD. Admitting spacecraft power level at values above 5 MW, with an hydrogen fed MPD would be possible to achieve an I_{sp} higher than 10 000 s with a efficiency up to 75 % making it an suitable option for piloted missions to Mars [33].

Hall-Effect Thrusters (HET) consist of a cylindrical channel with an interior anode, a magnetic circuit that generates a primarily radial magnetic field across the channel and a cathode external to the channel [4]. The anode and cathode create a electrical field while the magnetic field creates a crossed magnetic field. The electrons present in this crossed field generate a plasma discharge that ionizes the propellant injected into the axial channel. The transverse magnetic field reduces axial electron mobility which results in an axial electrical field that accelerates the ions to form the thrust beam [30]. As the thrust of HETs is a result of ions accelerated by an electrical field some authors consider them electrostatic thrusters [30]. Other authors consider them electromagnetic thrusters [4] as the magnetic field is critical to the process. HET have efficiency about 50 % to 60 %, whit thrust below 1 N and I_{sp} below 3000 s. Hall thrusters are being developed and used for orbit raising and station keeping on geostationary satellites due to the high performance and relatively simple construction and operation. To be capable of providing different mission types higher specific impulse is required [30].

The Variable Specific Impulse Magnetoplasma Rocket (VASIMR) is a two stage system. Firstly, radio waves ionize and heat a propellant. Secondly, magnetic fields accelerates the resulting plasma to generate thrust. This engine requires an ambient vacuum to work and has a low thrust-to-weight ratio, making it improper for first stage usages [34]. With a promised I_{sp} ranging from 2000 s to 5000 s, 5 N of thrust and thruster efficiency up to 72 % at 100 kW power level, the VX-200SS prototype performed a 100 hour uninterrupted test in 2018, upgrading to TRL-5. Like MPD, VASIMR needs scaling to megawatt class power levels to support human flight scenarios [35].

Pulsed Plasma Thrusters (PPT) provide thrust via short pulses, the firing rate is usually below 20 Hz although some can operate with rates up to 50 kHz, most systems use solid Teflon as propellant. When compared with other technologies the efficiencies and specific impulse are low. However it can function at power levels below 10 W with a simple, robust and functional system, making satellite attitude and control applications the most used for PPT systems [36].

Electric propulsion systems can provide serious propellant savings over chemical thrusters due to their high exhaust velocities, this is accentuated for high ΔV missions, such as a transfer to Mars. In recent years there was a renewed interest in nuclear space power options. This could enable high power EP systems to conduct such mission with shorter trip times and a lower propellant mass than chemical propulsion is capable of [37].

Solar sails reflect sunlight from a large, mirror like, sail made of a lightweight, very thin reflective material. The pressure from the reflected sunlight grants a continuous propulsion, without using any propellant. This allows for very high ΔV maneuvers and long duration missions, as long as its not too far away from the sun [38]. A limitation of solar sails is the drop in sunlight intensity moving away from the sun, by a factor of $1/R^2$, whit R being the distance of the vehicle to the sun. The T/w ratio on the other end is very low, up to 10^{-4} at Earth's distance from the sun. The first generation of solar sail propulsion systems developed by NASA have sails diameters ranging from 40m to over 100 m and a areal density of less than 13 $\mathrm{g/m^2}$. With some systems flown, solar sails have TRL-7 [39].

Thermal propulsion works similar to chemical SPS. This technology uses solar or nuclear fission as power sources, to heat the propellant before thermal expansion through a nozzle. Solar Thermal Propulsion (STP) uses an absorber cavity to concentrate sunlight. Nuclear Thermal Propulsion (NTP), uses a solid core fission reactor to heat the propellant [13]. While both systems can provide high I_{sp} ($<$ 800 s), STP has a low thrust. Nuclear thermal propulsion, on the other hand, also provides high thrust, which means that it is able to carry a bigger payload and perform short trip duration when compared with chemical propulsion systems [40]. Several proposed manned mission concepts to Mars suggest NTP systems as one of the most viable options. However, most research done in NTP was with the Rover/NERVA nuclear rocket programs (1955-1972) with 20 reactors designed and ground-tested. The reactors operated at thrust levels from 10 kN to 100 kN [41]. Current modifications to fuel forms and materials reduced the TRL to 4 [13].

Tether propulsion generates thrust using long cables, up to 100 km long, attached from the spacecraft to a ground or space station. Without the use of any propellant, the acceleration is given by the force of Earth's magnetic field applied to the cable (the tether) carrying an electrical current [32]. This technology can be used for small spacecrafts in LEO or any planet with a magnetosphere to make altitude and inclination changes or station keeping [13].

2.3 Advanced Propulsion

2.3.1 Types of Advanced Propulsion

In Beamed Energy Propulsion (BEP) a beam of electromagnetic radiation is transmitted from a laser or microwave energy source station based on the ground or space. Thrust can be generated in two ways: the energy is collected by the vehicle and used to heat a propellant or the energy is reflected to create a momentum [13]. The big advantage of this technology is the potential for significant weight reduction from a lack of a heavy power supply thus increasing the performance of the rocket. This system provides very low T/w, around 10^{-2}, but a I_{sp} that could reach 1000 s [32].

Similar to the solar sail concept, electric sails do not need propellant to provide thrust. Instead, it reflects charged particles from the solar wind, mostly protons, utilizing a magnetic field [13]. Although the velocity of solar wind maintains constant throughout the solar system, around 300 km s^{-1} to 800 km s^{-1}, just like the sunlight intensity the solar wind intensity drops as the square of the distance from the Sun. Unlike solar sails, electric sail technology requires a power system to create the magnetic field and, due to the radial and highly symmetrical nature of solar wind, tangential thrust is limited [32].

Advanced Fission has three alternatives to solid core NTP: gas and liquid core NTP, fission-fragment and external-pulsed plasma [13]. Gas and liquid core NTP, instead of a solid core, possess a gaseous or liquid reactor and work in a similar way. The fission from the liquid or gaseous core heats a propellant (generally hydrogen), which is then expanded through a nozzle to produce thrust. Solid, liquid or gas core NTP systems, are limited by engine materials structural temperatures, solid cores restrict the most being capped at 1000 s of I_{sp}, core temperatures can be enhanced using both liquid and gaseous cores instead of solid, thus increasing the restriction in I_{sp} [40]. For liquid core NTP has an expected maximum I_{sp} of approximately 1700 s while a gas core could achieve 7000 s [32].

Fission Fragment Propulsion (FFP) and External-Pulsed Plasma Propulsion (EPPP) are concepts that are not limited by engine materials structural temperatures. Fission Fragment Propulsion does not need to heat a propellant, uses naturally occurring fission fragments to directly or indirectly produce thrust [32]. The propellant is the fission fragments traveling with high exit velocity (several percent of the speed of light), which could provide an I_{sp} of more than 10^6 s. External-Pulsed Plasma Propulsion produces thrust with small fission pulse units being dropped from the rocket separated from 1 s to 10 s apart. The units would detonate at distances ranging from 30 m to 300 m to the rocket, the blast from the explosion interacts with a pusher plate, which gives impulse through a shock attenuation system. This would solve the problem from the resulting high temperatures, since the time of interaction with the rocket structure would be shorter. This type of technology should perform a 250 days round trip to Mars, with a I_{sp} of 2000 s and a T/w of about 4 [40].

There are many different fusion reactor concepts based on two different principles to produce energy: Inertial Confinement Fusion (ICF) and Magnetic Confinement Fusion (MCF). The ICF approach, uses high-intensity lasers or particle beams to strike the fusion target from multiple directions to compress and heat the fuel, generally deuterium-tritium, to fusion conditions. Magnetic confinement fusion principle uses magnetic fields to contain the fusion plasma, composed mainly by electrons and ions. Theoretically both ICF and MCF are expected to provide a I_{sp} that could reach 10^5 s and a $T/w > 1$. If confirmed the potential of this technology, it could accomplish in less than 100 days a piloted round-trip to Mars, carrying 100 t of payload [32].

Matter-Antimatter Annihilation Propulsion (MAA) has the possibility to offer the highest possible energy density of any known reaction substances. It works by annihilating atomic particles (protons) with their antiparticles (antiprotons), converting up to 75 % of fuel mass into propulsive energy, in the form of photons [13]. Since photons travel at the speed of light, according to Einstein's theory $(E/M = c^2)$ the ideal energy density from MAA would be of 9×10^{16} J/kg orders of magnitude higher than chemical (1×10^7 J/kg), fission (8×10^{13} J/kg) and fusion (3×10^{14} J/kg) reactions. With the exhaust speed maxed at the speed of light, this technology could enable outer solar system and interstellar missions. However the amount of thrust is limited by the mass of matter and antimatter reacting, a simple orbit transfer maneuver would require milligram masses of antimatter, this comes with hardships. Generating and storing antimatter is complicated, it does not appear in nature and worldwide the production of antiprotons is limited to some nanogram quantities per year [32].

Roger Shawyer proposed in 2001 a new principle of electric propulsion allowing direct conversion from electrical energy at microwave frequencies to thrust without the need for propellant [42]. This is called the Radio-Frequency (RF) resonant cavity thruster, or Electromagnetic Drive. A NASA team conducted tests on the Electromagnetic drive, consistently performing a thrust-to-power ratio of 1.2 mN/kW, seemingly violating Newton's third law of physics. The tests showed thrust on a low-thrust torsion pendulum without the use of propellant or a medium with which to exchange momentum. Although there is no consensus on how the thrust is being produced, the achieved thrust-to-power ratio is two orders of magnitude higher when compared to other technologies that function without the use of propellant [43].

2.4 Manned Mission to Mars

A useful concept for comparing different mission scenarios is of Mission Velocity Δ_V. It is a figure of the energy consumed by the vehicle to achieve the desired velocity, it is equal to the sum of all the velocity increments needed during the duration of the mission. In an hypothetical gravity-free and drag-free environment, it would be the obtained velocity if all the propulsive energy was applied in the same direction [2].

In reality Δ_V takes larger values than those of the actual velocity, due to the energy necessary to overcome gravity losses and potential energy. For example, the actual velocity of a rocket V to maintain a circular orbit at 500 km altitude is 7600 m s^{-1}, tangential to the gravitational field, the needed velocity increment Δ_V is higher, approximately 8700 m s^{-1}. The amount of energy used fighting gravity losses can be diminished by increasing the mass flow rate $\frac{\mathrm{d}m}{\mathrm{d}t}$ of the propellant, this implies higher thrust, thus higher acceleration. However, higher acceleration usually comes at the cost of more structural and propulsion system mass, which in turn cause the mass ratio to augment [44].

For interplanetary transfers, the minimum energy transfer can be achieved using an ellipse transfer. In the case of transfer to Mars this happens when the Earth at launch and Mars at arrival are on opposite sides of the Sun, the minimum energy path is the ellipse tangent to both planetary orbits. To achieve this two big velocity increment are needed, one that inserts the vehicle into the interplanetary orbit from the circular orbit around the Earth, other that inserts the vehicle into a Mars orbit from the interplanetary orbit. To reduce gravity losses thrust levels need to be high enough to minimize the operating time of the propulsion system, acceleration of at least $0.01\ g$ is necessary. Theoretically, to have no gravity losses the total impulse would have to be given instantaneously as the vehicle reaches the apogee of the trajectory. In practice the total impulse is given by the SPS in a length of time. Consequently, for Earth to Mars transfer 12.5 % to 15 % of the total mission velocity can be lost to overcome gravity losses [6].

Earth and Mars Orbits are not exactly circular nor coplanar, so the trajectories of each opportunity are not all equal, a minimum energy transfer from Earth to Mars takes about 259 days. To achieve faster transfer times, higher energy transfers are required [9].

2.4.1 Architectures

There are two different scenarios for manned missions to Mars, which can be divided as short-stay and long-stay missions. Short-stay missions are characterized by the short-stays on the surface of Mars (typically 30 to 60 days) and high propulsive requirements as the departure of Mars is on a non-optimal energy return. This means higher total energy requirements and large variation in Δ_V from opportunity to opportunity. Long-stay missions can have as much as 550 days at Mars surface and represent the global minimum energy solution for a given launch opportunity, unlike short-stay missions time would be spent waiting for a lower energy return. Although long-stay missions have a longer round-trip duration of approximately 900-1100 days, due to the lower energy transfer each trip time is shorter (less than 200 days). Human Exploration of Mars Design Reference Architecture 5.0 Addendum 2 suggests this two alternatives [15].

To accurately understand the needed propulsive characteristics of the mission, beside the propulsive characteristics of the SPS, it is necessary the mass of the vehicle. Proposed architectures for manned missions to Mars are usually divided in two stages [16, 37]. The first stage carries the cargo to Mars. A fast transit time is unnecessary, sending the cargo before on a minimum energy transfer reduces the mission mass as well as enabling pre-deployment of necessary facilities for human survival on Mars surface. The second stage would carry the crew and supplies in a higher energy transfer. This would reduce the risk to the astronauts by shortening the trip time.

A mission to Mars would require a LEO payload estimated to be on the order of 800-1200 t, this is two to four times larger than any previously attempted mission. The Human Exploration of Mars, DRA 5.0 [16] presents a nuclear thermal rocket reference with a total weight of 848.7 t, divided by two cargo vehicles weighting 246.2 t each plus the crew vehicle with 356.4 t . Nine launches, with capability to place both large mass volume into LEO, would be required [16].

Chapter 3

A SPS Database

In this work we built a database to analyse the evolution trends and evaluate performance capabilities of different technologies. The database will also be used to verify typical values of SPS types identified in the literature.

3.1 Construction of the Database

The main focus of the database was gathering and organizing data about different types of SPS, with the goal to comprehend their level of maturity and analyse the development trends. The focal point for the database construction was on quantitative data (such as how much thrust or I_{sp} a SPS can provide), nevertheless we also gathered qualitative information, when pertinent, for the analyses. A combination of quantitative and qualitative data allows for a more in-depth insight into each SPS.

The selected software for the database construction was Microsoft Office Excel, as it has a user friendly interface and it is widely available.

3.2 Data collection

To better understand the development trends, we gathered information not only from the most modern designs flying today, and the proposed concepts that might fly in a near or distant future, but also of SPS designs from the dawn of space exploration. The information was obtained primarily from the Web of Science [45] and, also, through the NASA Technical Report Server [46]. However it is difficult to detect relevant and accessible literature. In order to keep the content valid and reliable, the preferred sources were websites of SPS producers, and space oriented websites such as Spaceflight Now [47] and Encyclopedia Astronautica [48].

We performed a thorough survey of the literature and collected all data which included information of the attributes that can be found in table 3.1.

Out of all the literature consulted in the making of this work, 148 references were used for the construction of the database. Of which 39 were articles and books and 109 were web-pages, resulting in the

Table 3.1: List of attributes present in the database

Attribute	Unit
Type of SPS (in accordance to Chapter 2)	
Sub-type of SPS (in accordance to Chapter 2)	
Specific Impulse (sea level and vacuum values)	s
Thrust (sea level and vacuum values)	N
SPS Mass	kg
Diameter	m
Length	m
TRL	
Date of first flight	year
Propellants	
Mass Ratio of the Propellants	
Chamber Pressure	kPa
Chamber Temperature	K
Nozzle Aspect Ratio	
Thrust-to-Weight Ratio	
Thruster Efficiency	
Engine Cycle	
SPS Application (in accordance to Section 1.2.3)	
Country of Developer	
Manufacture	
Problems	
Additional Comments	

accumulation of data of 250 different propulsion systems. Of the 250 SPS found, 141 are chemical SPS, 88 are non-chemical SPS and 21 are advanced propulsion.

In reference [4] George P. Sutton estimates an approximate total of 1300 different liquid propellant rocket engines designed, built and tested. Considering all other technologies the total number of SPS designed should be considerably higher. The omitted engines in the database are a result of limitations on the amount of data known or accessible. Since each type of SPS can have multiple versions different from each other, when searching for information, there are many sources that refer to the SPS without clarifying which version is being mentioned. That can be a handicap to understand the development trends. In those cases, it was made an information match by crossing all the gathered data to detect possible patterns and, that way, to understand which type of SPS is being mentioned. Also, in the database there are included a few number of signalized contradictory information (with the following reference in the ”Comment” section). In those cases, the information that was selected to be displayed in the database followed two criteria: which information is more recent or more quoted by the main bibliography.

3.3 Database Division

The database consists of four worksheets: ”User Guide”, ”Raw Data”, ”Processed Data” and ”Bibliography”. The ”User Guide” worksheet is an introduction of the database and explains how it works, it includes all the information needed to understand and navigate in the database. The worksheets ”Raw Data” displays all the gathered data, organized by SPS and each set of attributes, with references. The

”Processed Data” is similar to the ”Raw Data” but contains data that was not found but could be deduced. For example, T/w ratio can be deduced if the thrust and mass of the SPS is known, or an estimation of the dimensions can be extrapolated if pictures are available. In order to easily differentiate the deduced and the collected data a color code was used. The meaning of each color can be found in the ”User Guide” worksheet. Also, the ”Bibliography” worksheet has the list of all the sources used for the construction of the database with date of retrieval, in the case of websites.

The worksheets ”Raw Data” and ”Processed Data” were built with the same structure, each SPS entry occupies one line, while the different attributes referenced in Section 3.2 occupy three columns each: ”Ref.”, ”Comment” and ”Value”. The column ”Ref.” gives the reference of the source of the data respective to the attribute in question (between box brackets) and, when suitable, the page number (after ”pp”), the ”Comment” column is allocated for complementary information about the attribute and the ”Value” column gives the value of the attribute in conformity with its unit.

Figure 3.1 is a portion of the ”Raw Data” worksheet, it shows the I_{sp} and thrust in vacuum of three different types of SPS. As we can see from 3.1 the BBM-2 MESC Thruster has a vacuum thrust of 0.15N, displayed in the ”Value” column, the ”Comment” column gives additional information about the thrust value of this engine and the ”Ref:” column indicates that this data was collected in page 68 from reference number 15 of the database. Note that the bibliography numbers in the database do not correspond to the ones in this work.

	Isp (Vacuum) [s]			Thrust (Vacuum) [N]		
SPS	Ref:	Comment	Value	Ref:	Comment	Value
BBM-2 MESC Thruster	[15] pp68		3518	[15] pp68	Nominal thrust 0.15N, Maximum thrust 0.18N	0.15
NEXT-C	[124]	Isp ranges from 1200-4200s	4200	[124]	range 0.025-0.235N (0.323N mode in development)	0.235
ORION	[2] pp106	3000-5000s	5000	[2] pp106	Value for 5000s Isp (300000N at 3000s Isp)	1600000

Figure 3.1: Excerpt from ”Raw Data”

Some noteworthy functions of the database are the ”filter” and ”sort” functions. By extending the arrows next to the ”Value” column, the user can filter what values appear. Figure 3.2 shows a portion of the ”Processed Data” worksheet, and it was selected in the column ”Types of SPS” to filter out non-chemical and advanced propulsion systems. Another function used in the figure, is the ”sort” function, this allows to order the values by increasing or decreasing value, in this case we order by increasing value of the ”Date”. Another functionality displayed in figure 3.2 is the color code, both the LE-5B and the CHASE-10 SPS have a different color in the ”Date” attribute. This helps to differentiate them from the SPS that have a first flight ”Date”. For example, in the LE-5B the ”Date” is correspondent to the end of development, in opposition to the RL-10A-4-2 that has the date of first flight. Finally to make the database more user friendly we used the ”Group” function. This allows to minimize the ”Ref.” and ”Comment” columns, leaving on display only the ”Value” column. This can be done by pressing the ”minus squares” found on the top of 3.2, and to display back all three columns, the ”plus square” just needs to be pressed. As can be seen in 3.2, ”Types of SPS” and ”Thrust” attributes are grouped, while the attributes of ”I_{sp}” and ”Date” are expanded.

B	E	F	G	H	K	L	M	N
	Type of SPS	Isp (Vacuum) [s]			Thrust (Vacuum) [N]	Date [year]		
SPS	Value	Ref:	Comme	Valu	Value	Ref:	Comment	Value
LE-5B	Chemical	[12] pp821		450	140000	[12] pp821	End of development	2002
RL10A-4-2	Chemical	[116]		451	99195	[116]	First Flight	2002
Merlin 1A	Chemical	[48]		300	378949	[48]	First Flight	2006
AJ-60A	Chemical	[141]		275	1270000	[141]	First Flight	2006
CHASE-10	Chemical	[6] pp479		321	107000	[114] pp1	Prototype firing tests	2006
Kestrel	Chemical	[51]		320	30700	[51]	First Flight	2006
RD-861K	Chemical	[39]		330	77472.535	[125] pp455	First Flight	2006
Merlin 1C	Chemical				614700	[49]	First Flight	2008
Merlin 1C Vacuum	Chemical	[50]		342	411400	[50]	First Flight	2009

Figure 3.2: Excerpt form ”Processed Data”

Chapter 4

Database Analyses

4.1 Rules of Thumb

We can analyse the gathered data to compare the "Rules of Thumb" (RT) found in the literature and that are supposedly typical values for SPS. In [6, p. 39] there is a figure containing the heuristics of I_{sp} and T/w for chemical and non-chemical SPS, and also a scatter plot of the specific impulse in function of the thrust-to-weight ratio of those types of SPS can be found in figure 4.1.

4.1.1 Rules of Thumb for Specific Impulse

- **RT1 - Chemical Propulsion Systems** values of I_{sp} around 170-468 s, typical values for system at SL with P_1 =6895 kPa and exit pressure = ambient pressure [6].
- **RT1a** - Typical values for I_{sp} of solid propulsion 170-220 s [9] [1].
- **RT1b** - Chemical bipropellant systems typical operating I_{sp} is about 300-468 s [30], depending on the propellant combination. Reference [32] gives representative values of contemporary and advanced chemical systems with a chamber pressure p_c = 6895 kPa and a nozzle area ratio $A_e/A_t = 81$.
- **RT1b.1** - Systems using NTO/MMH as propellants have a I_{sp} = 317 s [32]. This appears as a representative for storable bipropellant SPS as the different combinations have very similar I_{sp} capabilities [1].
- **RT1b.2** - Systems using LOX/LH_2 have I_{sp} = 423 s [32].
- **RT1b.3** - Systems using LOX/RP-1 have I_{sp} = 358 s [32].
- **RT1b.4** - Systems using a LOX/CH_4 have I_{sp} = 330 s [32] [2].
- **RT2 - Electric Propulsion Systems** I_{sp} can go from 300 s to 12 000 s, depending on the type of technology used [32].
- **RT2a** - Electrothermal system typically have specific impulses ranging from 300 s up to 1200 s, depending on the technology [30].

[1] In [49] says typical maximum I_{sp} for solid propulsion systems is 260 s.

[2] Reference [9] says current and conceptual propulsion systems using LOX/LH_2 have an $I_{sp} = 455$ s, while systems using LOX/LHC have an I_{sp} between 200-350 s.

- **RT2a.1** - Resistojets have a I_{sp} level typically between 300-400 s with an electrical power input of 0.5-1 kW and a thruster efficiency of 65-90 % [30, 32] [3].
- **RT2a.2** - Arcjets using hydrogen as propellant usually have I_{sp} about 900-1200 s, with typical power range in the 100 s of W [32], while systems using hydrazine have I_{sp} 500-600 s with electrical power input of 0.9-2.2 kW and 25-45 % thruster efficiency [30].
- **RT2b** - Electrostatic systems, or ion thrusters I_{sp} is frequently around 2500-3600 s, with electrical power input of 0.4-4.3 kW and a thruster efficiency from 40 % to 80 % [30] [4].
- **RT2c** - Electromagnetic systems can provide I_{sp} from 850 s to 12 000 s, depending on the technology employed [50].
- **RT2c.1** - Pulsed plasma thrusters have a I_{sp} between 850-1200 s, with the lowest electrical power input $<$200 kW, and lowest thruster efficiency 7-12 % [30] [5].
- **RT2c.2** - Hall effect thruster have a I_{sp} around 1500-2000 s, with a electrical power input level between 1.5-4.5 kW and thruster efficiency of 35-60 % [50].
- **RT2c.3** - Magnetoplasmadynamic thrusters are capable of delivering I_{sp} between 3000-12 000 s, depending on the electrical power input that can go from 100 kW to 1 MW [32] [6].
- **RT3 - Nuclear Thermal Propulsion System** using a nuclear solid core usually provide I_{sp} around 800-1000 s [32].
- **RT4 - Advanced Propulsion Systems** have the least amount of information, due to their theoretical nature and far reached technological developments needed.
- **RT4a** - Advanced fission SPS have I_{sp} around 2000-15 000 s [32].
- **RT4a.1** - Systems using a gas core have I_{sp} around 2000-7000 s [32].
- **RT4a.2** - Fission fragment propulsion are capable of providing a I_{sp} of 2000-5000 s [32].
- **RT4a.3** - External-pulsed plasma systems enjoy the highest I_{sp} of the advanced fission systems, are capable of 2500-15 000 s [32].
- **RT4b** - Fusion systems have I_{sp} that can go from 20 000 s to 10^6 s [32].

4.1.2 Rules of Thumb for Thrust-to-Weight Ratio

- **RT1 - Chemical Propulsion Systems** ratio of thrust force to the full propulsion system sea level weight up to 200 [6].
- **RT1a** - System using solid propellants have T/w ratio up to 200 [6].
- **RT1b** - Systems using liquid propellants have T/w capabilities limited to 200 [6].
- **RT2 - Electric Propulsion Systems** have T/w ratios usually between 10^{-6}-10^{-2} [6].
- **RT2a** - In the case of electrothermal systems the T/w ratio is usually comprehended between 10^{-4}-10^{-2} [6].
- **RT2a.1** - Resistojets T/w ratio is equal, usually about 10^{-4}-10^{-2} but can produce higher thrust

[3] In [6] the typical I_{sp} is 200-350 s, this may be due to a smaller electrical power input.
[4] In [6] the I_{sp} typical values are 1500-8000 s, while in [32] the given I_{sp} for this type of systems is 2000-10 000 s with electrical power inputs from W to 100 kW.
[5] In [6] the I_{sp} for PPT is around 600-2000 s, this may be due to lower electrical power input levels.
[6] Reference [6] gives more conservative typical values of I_{sp} around 2000-5000 s.

Table 4.1: A summarize of the RT found in the bibliography

RT	Type of SPS	I_{sp}, s	T/w	Ref.
RT1	Chemical	170-468	$<$ 200	[6]
RT1a	Solid Propellant	170-200	$<$ 200	[6, 9]
RT1b	Bipropellant	300-468	$<$ 200	[6, 30]
RT1b.1	Storable	317	-	[32]
RT1b.2	LOX/LH_2	423	-	[32]
RT1b.3	LOX/RP-1	358	-	[32]
RT1b.4	LOX/CH_4	330	-	[32]
RT2	Electric	1000-12000	10^{-6}-10^{-2}	[6, 32]
RT2a	Electrothermal	300-1200	10^{-4}-10^{-2}	[6, 30]
RT2a.1	Resistojets	300-400	10^{-4}-10^{-2}	[6, 30]
RT2a.2	Arcjets	900-1200	10^{-4}-10^{-2}	[6, 30]
RT2b	Electrostatic	2500-3600	10^{-6}-10^{-4}	[6, 30]
RT2c	Electromagnetic	850-12000	10^{-6}-10^{-3}	[6, 50]
RT2c.1	PPT	850-1200	10^{-6}-10^{-3}	[6, 30]
RT2c.2	HET	1500-2000	10^{-6}-10^{-3}	[50, 51]
RT2c.3	MPD	3000-12000	10^{-6}-10^{-3}	[32, 51]
RT3	NTP	400-1000	$<$ 30	[6, 32]
RT4	Advanced Propulsion Systems	2000-10^6	-	[32]
RT4a	Advanced Fission	2000-15000	-	[32]
RT4a.1	Gas Core	2000-7000	-	[32]
RT4a.2	FFP	2000-5000	-	[32]
RT4a.3	EPPP	2500-15000	-	[32]
RT4b	Fusion	20000-10^6	-	[32]

levels of 200-1000 mN [6].

- **RT2a.2** - Arcjet have the lower thrust range between 200-300 mN and a T/w around 10^{-4}-10^{-2} [6].
- **RT2b** - Ion propulsion systems have low T/w ratio typically from 10^{-6} to 10^{-4} and a thrust range typically between 0.01 up to 500 mN [6] [7].
- **RT2c** - Electromagnetic systems have a T/w ratio that can go from 10^{-6} to 10^{-3} across all technologies, depending on the electrical power input [6, 50].
- **RT2c.1** - PPT systems usually have a small thrust range comprehended between 0.05-10 mN and a T/w ratio around 10^{-6} - 10^{-3} [6].
- **RT2c.2** - Current state-of-the-art HET have a thrust level usually of 10-50 mN and a T/w ratio capability of 10^{-6} - 10^{-3} [51].
- **RT2c.3** - MPD typical value of thrust level is 0.001-2000 mN with a T/w ratio usually from 10^{-6} up to 10^{-3} [51].
- **RT3 - Nuclear Thermal Propulsion** systems can reach T/w ratios up to 30 [50].
- **RT4 - Advanced Propulsion systems** - No information was found regarding the T/w of this type of systems.

All RT relative to the I_{sp} and T/w values found in the bibliography are summarized in table 4.1.

[7] In [50] the typical T/w ratio is around 10^{-5}-10^{-4}.

4.2 Database Overview

In Sections 4.3-4.6 are displayed and discussed 15 scatter plots, from the data gathered in the database. The displayed data will help to better understand each technology capabilities and identify trends. From the displayed data New Rules of Thumb (NRT) are identified, to be compared with the results from the literature.

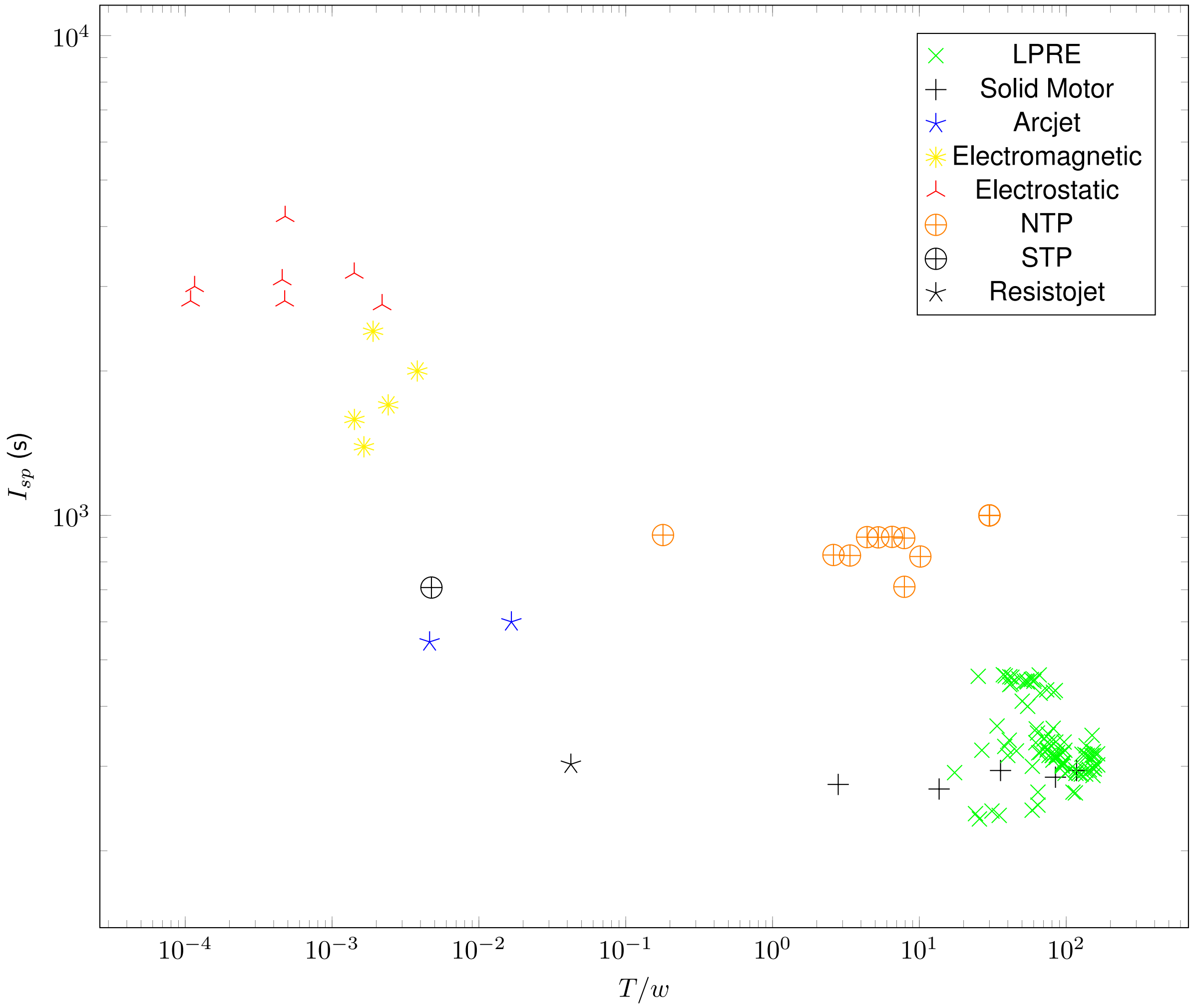

Figure 4.1: Vacuum I_{sp} plotted as a function of T/w, of chemical and non-chemical SPS

Figure 4.1 depicts a scatter plot similar to the image presented by Sutton [6, p. 39]. Figure 4.1 contains all of chemical and non-chemical propulsion systems, with the I_{sp} plotted in function of the SPS T/w, while the figure by Sutton [6, p. 39] gives approximate performance of exhaust velocity as a function of the vehicle acceleration. As seen in Section 1.2.2, the specific impulse is the exhaust velocity divided by the standard Earth's surface acceleration, and the thrust-to-weight ratio gives the acceleration in multiples of g_0, that the SPS is capable of giving to its own loaded propulsion system mass. Figure 4.1 is also useful for comparing the RT1-RT3 with the data gathered in the database.

Figure 4.2 shows the vacuum I_{sp} of the various types of propulsion in function of the year of the first flight [8]. The plot includes all systems in the database that have a value of vacuum I_{sp}, with the exception

[8] Systems that have not yet flown or with no data about the year of the first flight, the date used was of the ground tests. If this

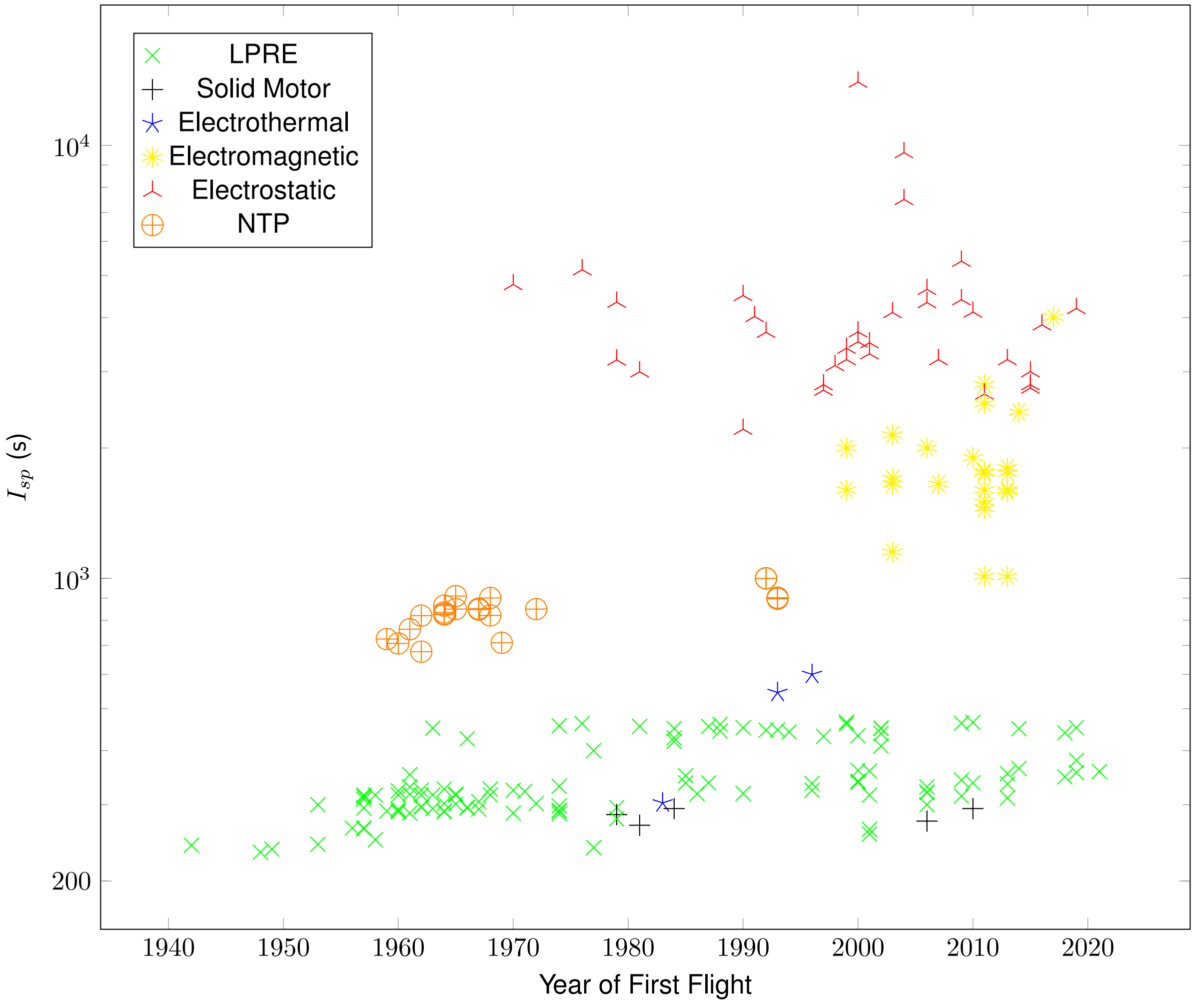

Figure 4.2: Vacuum I_{sp} plotted as a function of year of first flight, of chemical and non-chemical SPS

of advanced propulsion technologies. These were left out of figure 4.2 due to the very high I_{sp} when compared to chemical and non-chemical propulsion. Data relative to advanced propulsion technologies can be found in figure 4.16.

In chapter 2 we saw what the differences of chemical and non-chemical propulsion are, and in 4.1 exactly how different the expected performance is. Figures 4.1 and 4.2 evinces that disparity in terms of I_{sp} and T/w.

Chemical Propulsion has the lowest specific impulse of the represented technologies in figures 4.1 and 4.2, limited to low specific impulse levels ($I_{sp} < 500$ s), due to the energy used coming entirely from the chemical reaction of the propellants [14]. On the other hand, chemical SPS have the highest T/w of the displayed technologies, standing alone as the only technology that has reached a ratio above 100. In Section 4.3 is made a more detailed analyses of the chemical SPS present in the database.

In order to achieve higher I_{sp} a separate energy source is required, as is the case of non-chemical propulsion. NTP systems use nuclear fission, while STP uses solar radiation, to achieve a better performance compared to chemical propulsion, with a I_{sp} level comprehended between 600-1000 s. While

information was also unavailable a relevant date relative to the SPS in question was used, such as end of development date (in the database can be found additional info about the dates used).

granting similar specific impulse, by inspecting figure 4.1 is noticeable the difference in T/w. STP has a T/w below 1×10^{-2}, on the other hand, NTP can reach T/w above 10. In Section 4.5 is made a more detailed analyses of the NTP systems present in the database.

EP systems using electrostatic or electromagnetic forces to accelerate, can deliver the highest performance in terms of specific impulse in the technologies analysed in figures 4.1 and 4.2, with most systems capable of a $I_{sp} > 1000$ s. Of all EP types, electrothermal thrusters is the technology with the lowest I_{sp} levels. Resistojets have similar performance to chemical propulsion with a I_{sp} around 300 s, while arcjets grant increased I_{sp} between 500 up to 700 s. While EP is the technology capable of reaching the highest I_{sp} of all chemical and non-chemical systems, the T/w capabilities are several orders of magnitude below the possibilities of both chemical and nuclear thermal systems. In 4.5 a more detailed analyses of the EP systems present in the database is made.

An identifiable trend in figure 4.2 is the amount of new SPS flying. In the 1960s and early 1970s, during the space race, there was a big investment and development of aerospace industry, with USA and Russia (then USSR) being the biggest contributors to this development. After the moon landing in 1969 the interest and investment slowly decreased. In the end of the twentieth century a renewed interest appears, as space propulsion had become a essential technology with most communication traveling through space, relying on satellites [2].

4.3 Chemical SPS analyses

4.3.1 Specific Impulse

Figure 4.3 shows the vacuum I_{sp} for all chemical SPS in function of the year of the first flight. As expected by RT1b the highest I_{sp} of all chemical propulsion systems are achieved by SPS using LOX/LH_2 as propellants. This propellant combination is the only to have surpassed the 400 s barrier with the exceptions of the RD-701, the RD-704 (a derived version of the RD-701) and the RD-301. Both the RD-701 and the RD-704 were tripropellant engines, for low altitude operations used LOX/RP-1, then switched to a lower density combination of LOX/LH_2. Initial tests showed to be an effective SPS, but cuts in funding stopped the development of these systems. The displayed value for both tripropellant SPS are when the systems are using LOX/LH_2. RD-301 has $I_{sp} = 400$ s, using a combination of liquid fluorine and ammonia (LF_2/NH_3). Although very energetic and capable of better specific impulse than a LOX/LH_2 SPS, fluorine is highly corrosive and toxic at ambient temperatures, at high temperatures it changes the morphology of the engine. RD-119 used LOX/UDMH, this propellant combination proved to be highly energetic and capable of improvements over SPS using storable propellants and LOX/RP-1. Due to the highly toxic propellant combination the project was abandoned. These three SPS have a low TRL and have never operationally flown [6]. Contemporary SPS using LOX/LH_2 have a specific impulse between 430-465 s.

Storable propellants SPS have values around 280 s to 340 s, when analysing only contemporary SPS the range of specific impulse decreases, with all systems providing I_{sp} between 310 s to 340 s. SPS using

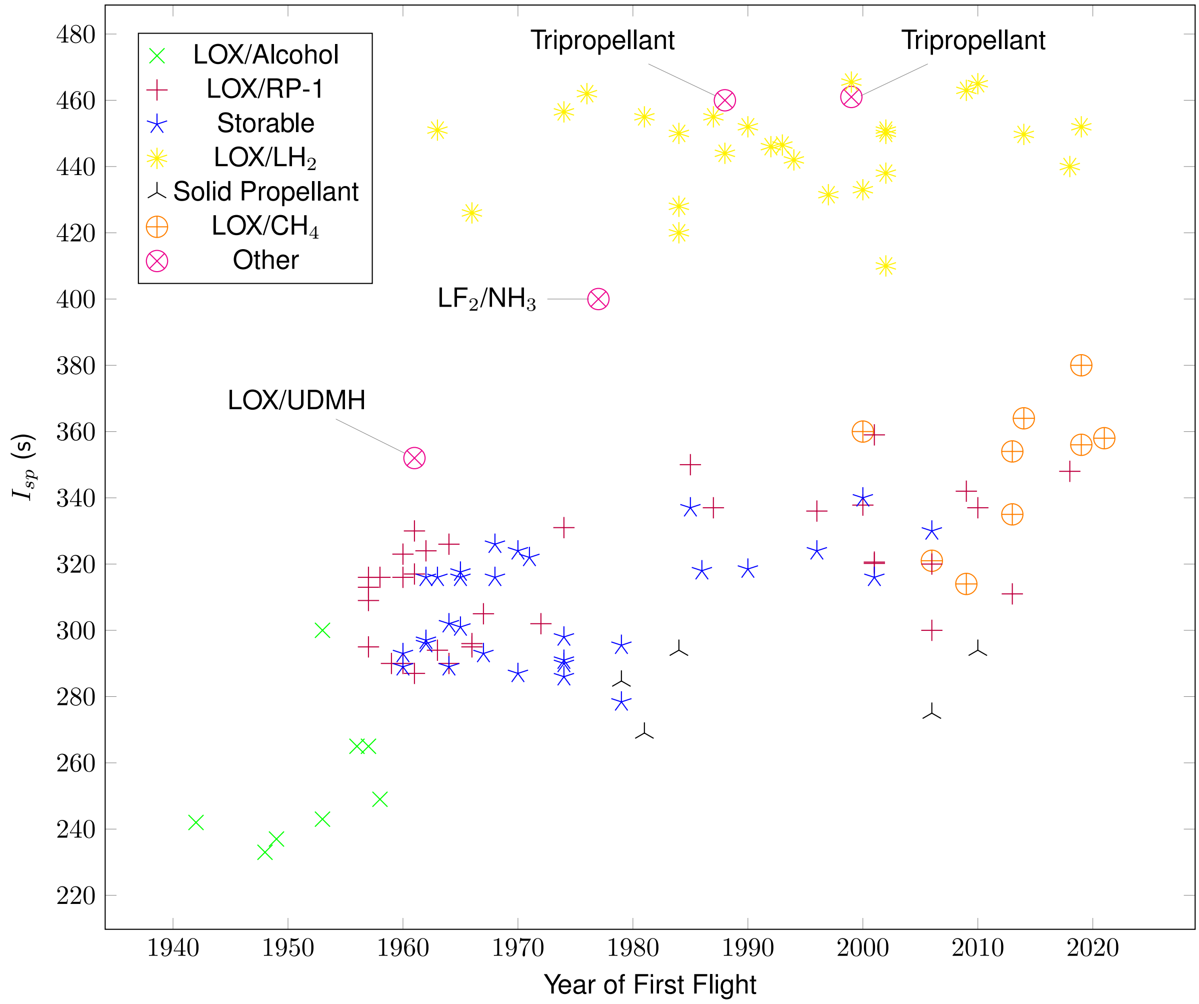

Figure 4.3: Vacuum I_{sp} plotted as a function of year of first flight, of chemical SPS

LOX/RP-1 have slightly higher specific impulse capabilities than those using storable propellants, with the I_{sp} of contemporary SPS around 300 s to 360 s.

Solid propellant systems have specific impulse capabilities different than those given by RT1a. There was a limited amount of data found concerning these systems, nonetheless all five solid motors displayed have a specific impulse considerably higher than that of RT1a. With the specific impulse around 260 s to 300 s. As we saw in Section 2.1.2 solid rockets are often used for military applications. As the focus of this research is in space propulsion, and not military, the focus of the research was not on military applications. As we can observe, the ISP values of this research are higher then the ones from the typical values from the main bibliography studies. A possible explanation for this difference of values could be determined because the military applications I_{sp} is not as important as it is for space applications.

In the last twenty years, methane (CH_4) saw an increased interest as a possible solution for both manned and unmanned missions to Mars. With the I_{sp} values about 310 s to 380 s, can achieve higher performance than systems using storable or LOX/RP-1 as propellants. Even though it grants lower performance than LOX/LH_2, liquid methane is approximately six times more dense than liquid hydrogen, meaning the fuel tanks can be smaller.

Figure 4.4 shows the influence of the chamber pressure p_c on the vacuum specific impulse of chem-

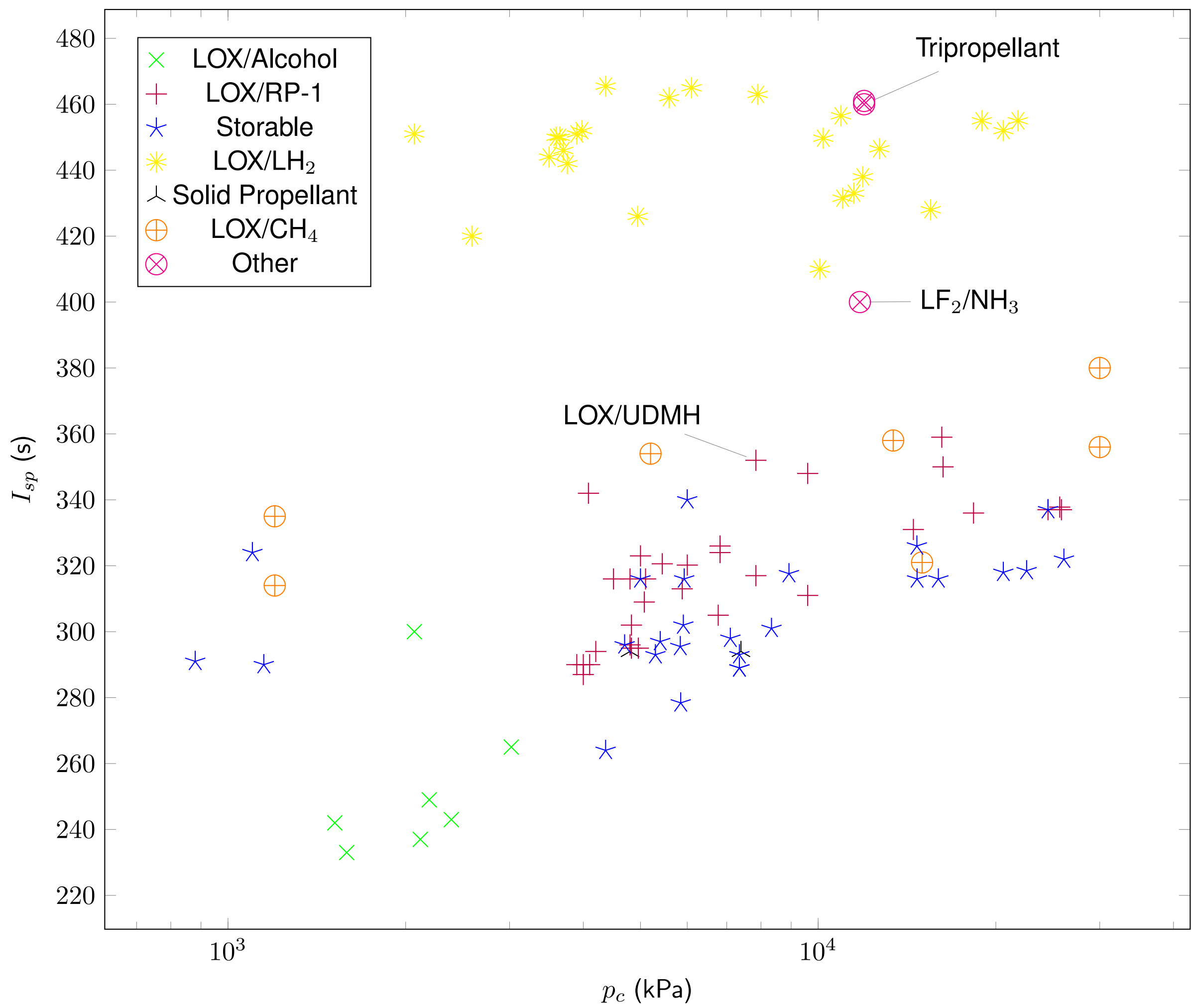

Figure 4.4: Vacuum I_{sp} plotted as a function of chamber pressure, of chemical SPS

ical SPS. Across all sub-types of chemical propulsion there is a positive weak relation between the increase of p_c and the I_{sp} provided by the SPS. This is in accordance with equation 2.1, increasing the chamber pressure as a positive influence on the exhaust velocity and consequently on the specific impulse.

Figure 4.5 shows chemical SPS vacuum specific impulse plotted against the nozzle area ratio. The plot contains all Chemical SPS with vacuum thrust and A_e/A_t. As seen in Section 2.1, the nozzle area ratio has an impact on the exit pressure of the nozzle. The bigger the ratio, the more expanded the gas is in the nozzle, and the closer the exit nozzle pressure gets to 0. As expected across all types of chemical SPS, the I_{sp} increases with A_e/A_t.

From figures 4.3-4.5 the NRT relative to specific impulse of chemical propulsion systems can be extrapolated:

- **NRT1 - Chemical Propulsion Systems** values for vacuum I_{sp} of conceptual SPS are typically between 260-470 s.
- **NRT1a** - Solid propellant SPS have a vacuum I_{sp} around 260-300 s.
- **NRT1b** - Chemical bipropellant conceptual systems typical operating vacuum I_{sp} is about 300-470

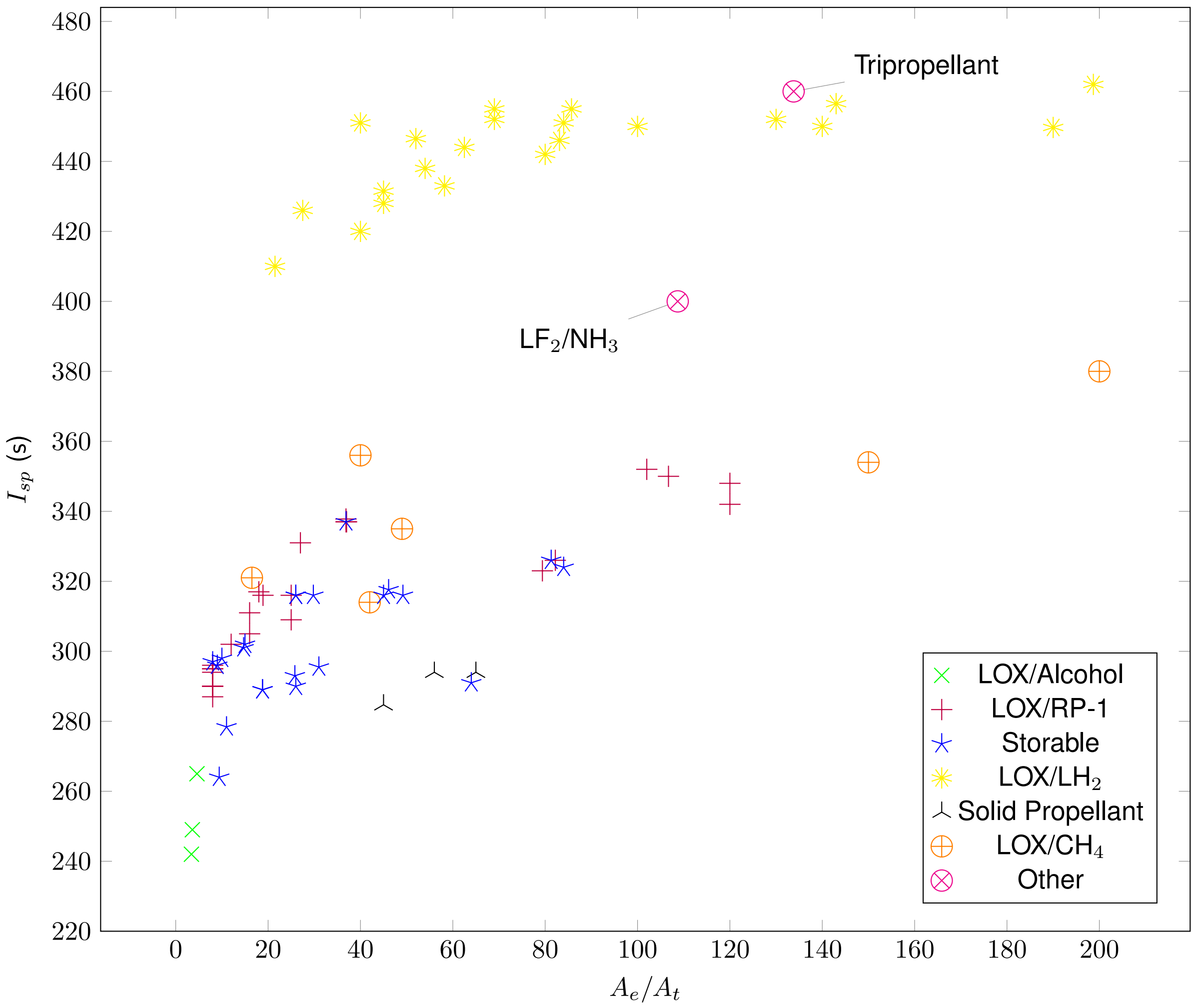

Figure 4.5: Vacuum I_{sp} plotted as a function of nozzle area ratio, of chemical SPS

s.

- **NRT1b.1** - Systems using a combination of storable propellants have a I_{sp} between 310-350 s.
- **NRT1b.2** - Systems using LOX/LH_2 have I_{sp} around 410-470 s.
- **NRT1b.3** - Systems using LOX/RP-1 have I_{sp} about 300-360 s.
- **NRT1b.4** - Systems using a LOX/CH_4 have I_{sp} among 310-380 s.

4.3.2 Thrust-to-Weight Ratio

Figure 4.6 shows the vacuum I_{sp} plotted as a function of T/w. In all sub types of chemical propulsion, with the exception of systems using LOX/Alcohol as propellants, increasing the I_{sp} has a negative impact in the T/w. This was expected, in Section 2.1, when optimizing a SPS trade between specific impulse and thrust-to-weight ratio must be done. This is demonstrated by plots 4.5 and 4.8, where we can see the opposite effect of increasing A_e/A_t. The development of I_{sp} results in a decrease of T/w.

Special attention to NK-33 and the Merlin 1D Vacuum, both engines excelling in both specific impulse as well as thrust-over-weight ratio. The NK-33 was developed in the 1960s for the N-1 Russian moon vehicle. It had a simple and light weight design, high chamber pressure and an advanced combustion

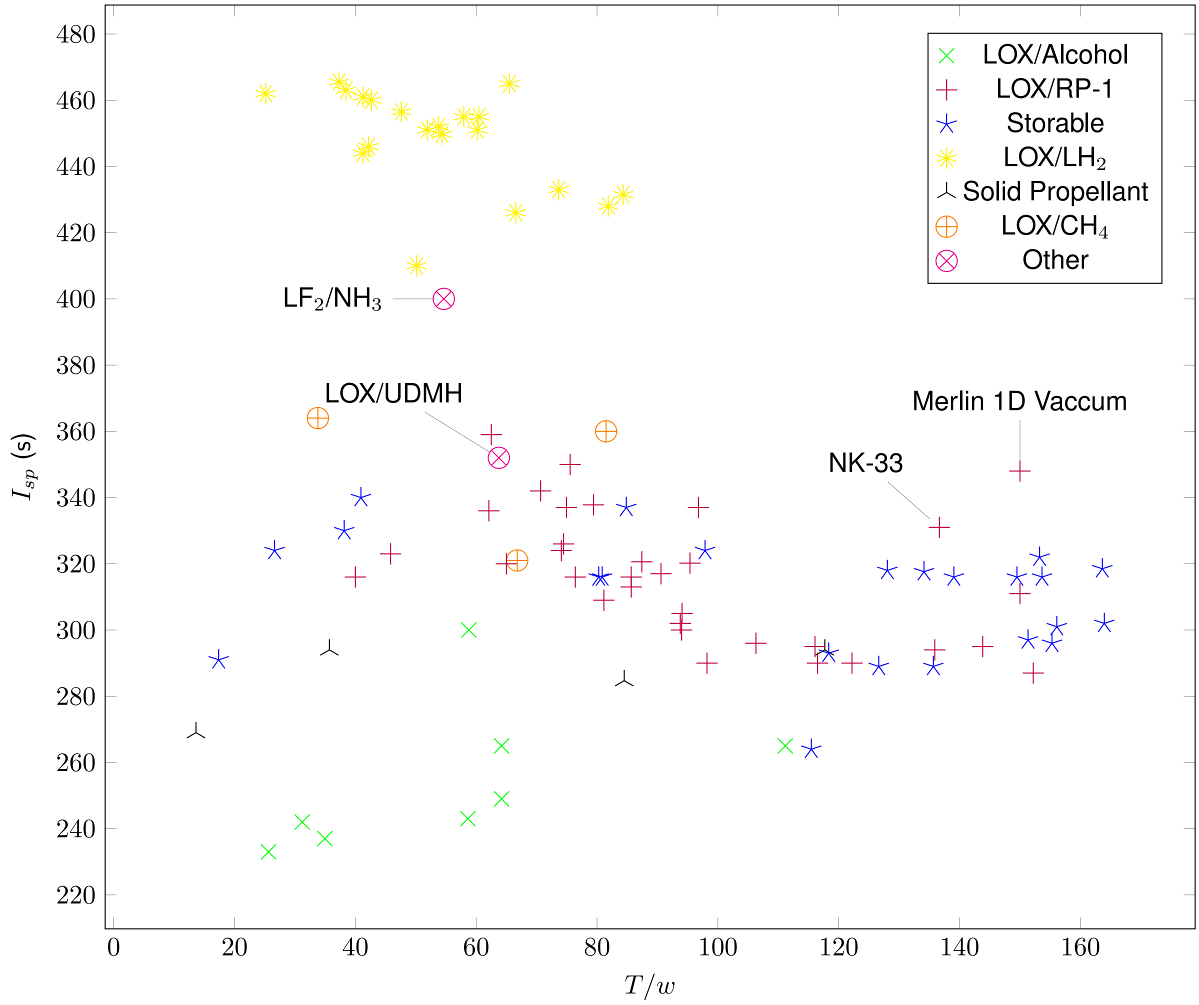

Figure 4.6: Vacuum I_{sp} plotted as a function of T/w, of chemical SPS

cycle which offered both high thrust and specific impulse. Although the program was canceled in 1974 after four consecutive failures, in 1993 it was sold to the American Aerojet, as the company believed the SPS had improved combustion cycle technology when compared to the existing American SPS using the same propellants [52]. Since 2018 Merlin 1D Vacuum is powering the second stage of SpaceX's Falcon 9 SLV. The very high performance in T/w is accomplished by a combination of very high chamber pressure (increasing T) and the development of light weight materials (decreasing w) [53].

As we saw in Section 2.1.1, the thrust chamber pressure p_c has a great impact on the thrust of the SPS. This is confirmed by figure 4.7, where we can see tendency between increase of T/w and p_c. Again we see some outliners, these can be explained by the purpose of the SPS, as we saw in Section 2.1.1 engines doing first stage usually are built with focus on development of thrust in determinant of specific impulse.

Figure 4.8 shows the thrust-to-weight ratio of the SPS plotted as a function of the nozzle area ratio. Contrary to figure 4.5, where an increase in A_e/A_t helped the development of the specific impulse. Here as we saw in Section 2.1.1 the increase of the nozzle adds extra weight, this has a toll on the thrust-to-weight ratio. Excluding the propellant combination the area ratio is the parameter with biggest

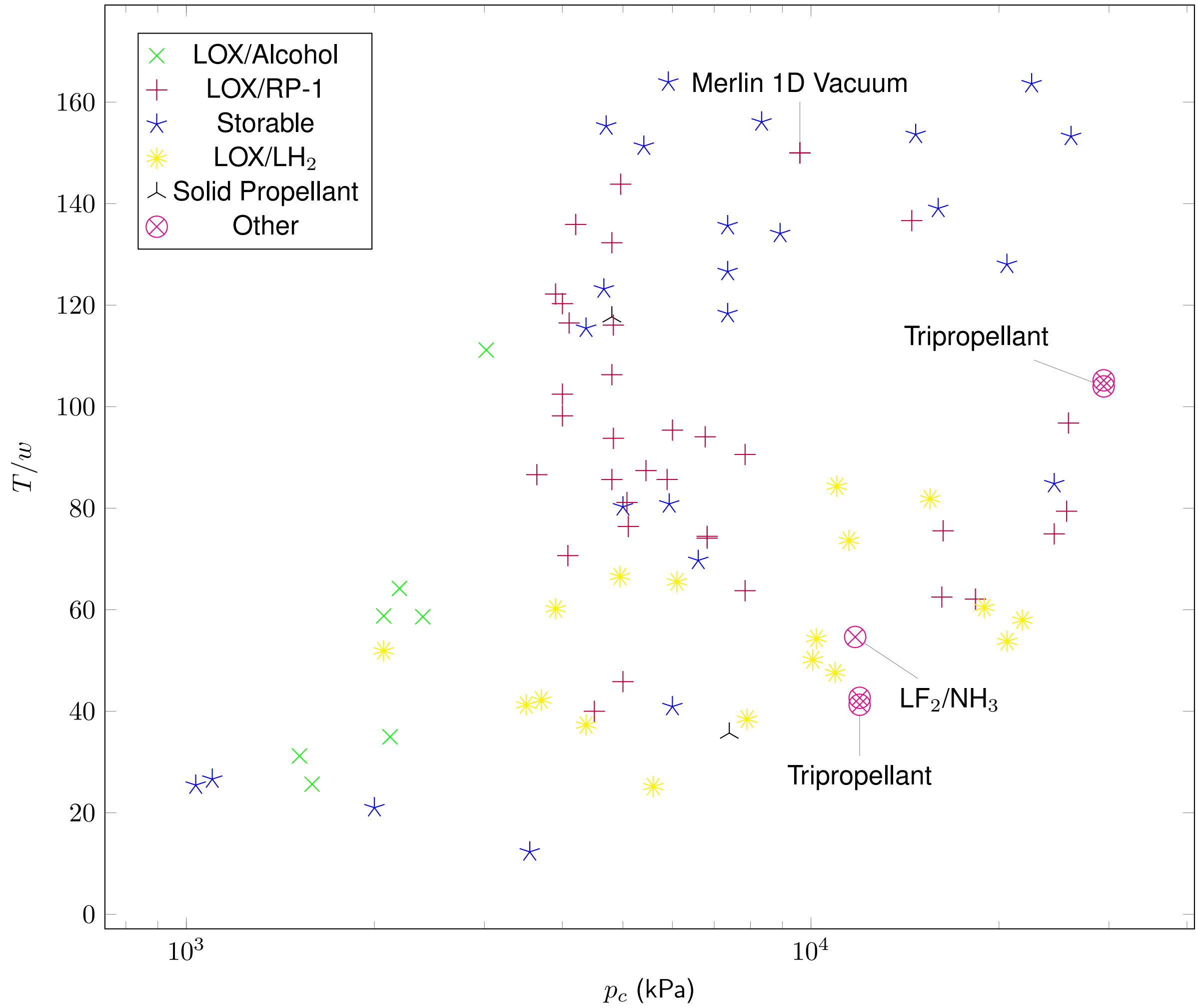

Figure 4.7: Vacuum T/w plotted as a function of chamber pressure, of chemical SPS

influence in the T/w. A big difference in values happens around $A_e/A_t = 40$. The majority of the SPS with $A_e/A_t < 40$ have a T/w between 70 and 160, while systems with $A_e/A_t > 40$ have a T/w between 20 and 70. This is in conformity with what we saw in Section 2.1.1, systems designed for low altitude operations have nozzle area ratio below 30, while systems for high altitudes have the ratio generally above 40.

Figure 4.9 displays the T/w plotted as a function of the thrust it can produce, it contains all chemical SPS. This can help understand how well the thrust-to-weight ratio of SPS scale with size. There is a weak link between the increase of thrust and the increase of T/w.

With the exception of solid motors, there is a clear step on what acceleration the SPS has. For thrust bellow 10^5 N, the T/w is limited to values below 70, while systems with an order of magnitude higher, 10^6 N, can reach a T/w of 160.

A new set of values of T/w can be deduced from figures 4.6-4.9 relative to the capabilities of different propellant combinations. The NRT relative to thrust-to-weight ratio of chemical propulsion systems are as follows:

- **NRT1 - Chemical Propulsion Systems** values for T/w vary from 20 up to 170.

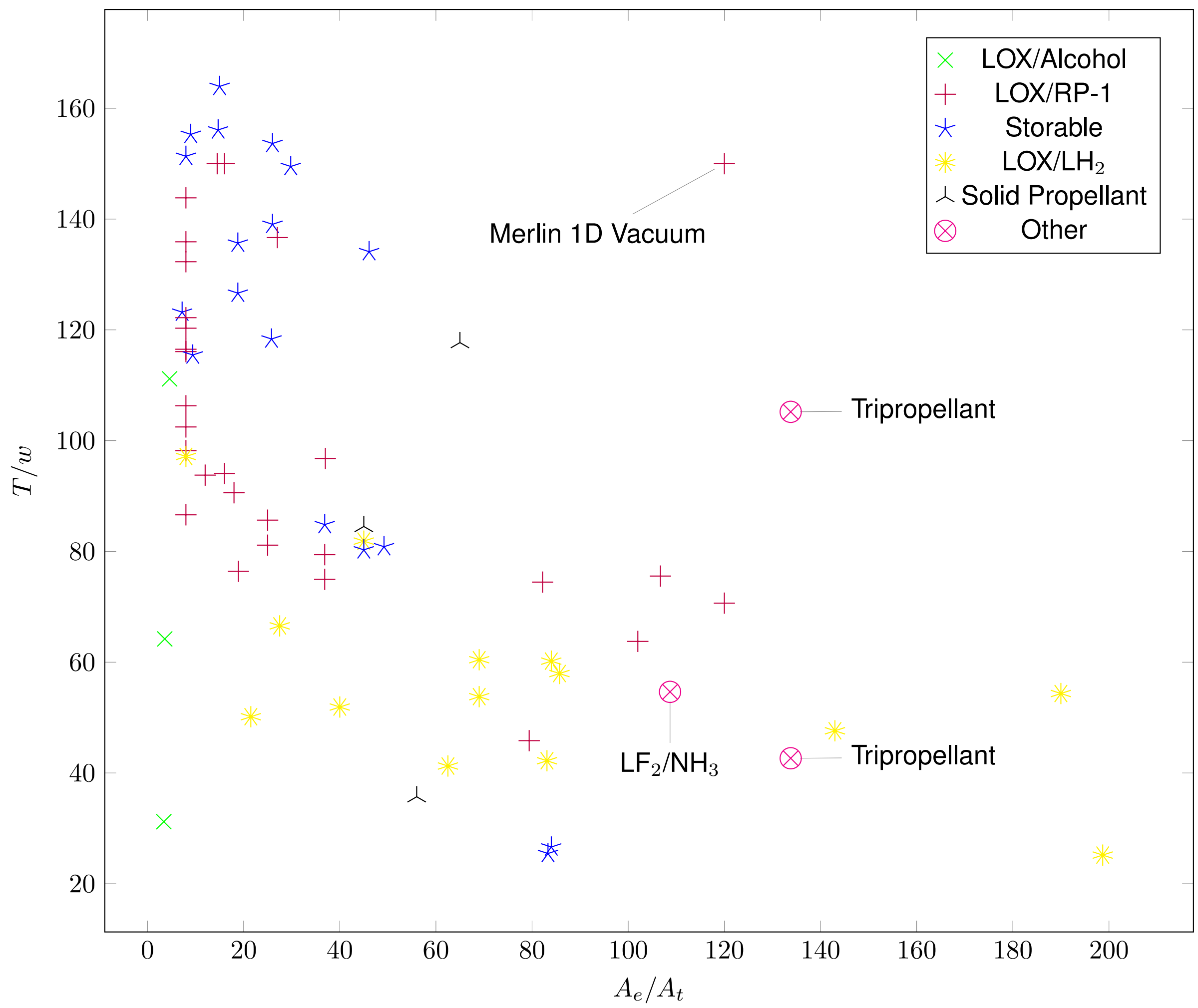

Figure 4.8: Vacuum T/w plotted as a function of nozzle area ratio, of chemical SPS

- **NRT1a** - Solid propellant SPS have a T/w below 120.
- **NRT1b** - Chemical bipropellant systems have values for T/w up to 170.
- **NRT1b.1** - Systems using a combination of storable propellants have a T/w usually between 80-170.
- **NRT1b.2** - Systems using LOX/LH$_2$ have a T/w around 30-100.
- **NRT1b.3** - Systems using LOX/RP-1 have T/w about 60-150.
- **NRT1b.4** - Systems using a LOX/CH$_4$ have T/w among 30-80.

4.4 EP Analyses

4.4.1 Specific Impulse

Figures 4.10 and 4.12 display the vacuum I_{sp} as a function of the thruster efficiency, and electrical power input of the EP gathered in the database, respectively. The SPS presented in table 4.2 are missing due to the very high I_{sp} or P_{in} when compared with the remaining EP systems, which would hamper the interpretation of the figures. Table 4.2 contains the data concerning the variables plotted for these SPS.

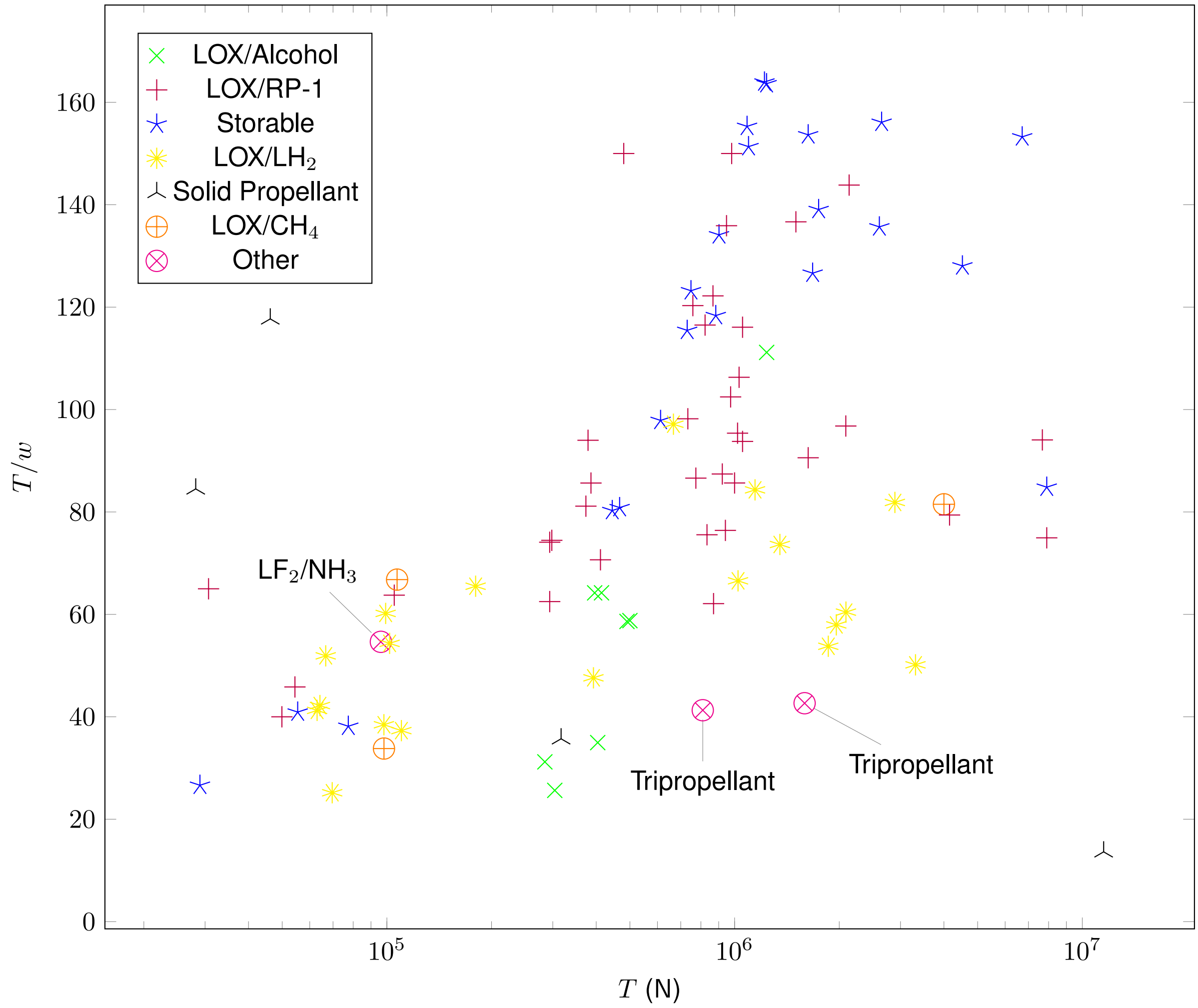

Figure 4.9: Vacuum T/w plotted as a function of thrust, of chemical SPS

The VX-200SS is a VASIMR, the remaining three SPS presented are ion thrusters.

Figure 4.10 displays the vacuum specific impulse as a function of the P_{in}. This figure contains all EP with data about the vacuum I_{sp} and P_{in}, except the SPS presented in table 4.2. As we saw in equation 2.5, the exhaust velocity, and by consequence the specific impulse, is a function of the square root of the electrical power input. The plot verifies the equation, across all subtypes of EP there is a positive non-linear tendency between the variables. Some outliners were expected, as the thruster efficiency and the mass flow rate of the propellant also impact the development of I_{sp}.

Figure 4.11 gives the vacuum I_{sp} plotted in function of the thruster efficiency η. From equation 2.5, we know that η as the same effect on I_{sp} development as P_{in}. That is, the specific impulse is a function of the square root of the thruster efficiency. The data displayed for ion thrusters seem to show more of a linear tendency between the variables in question, as for the rest of EP technologies not enough information was available. As we saw in Section 2.2.1, for a constant P_{in} the thrust and the specific impulse are inversely proportional. A possible explanation for the linear increase is that, increasing the I_{sp} passed 3000 s is less impactful than increasing the thrust. The displayed ion thrusters are mainly used for small maneuvers and reaction control. Of the four types presented, arcjets have the lower

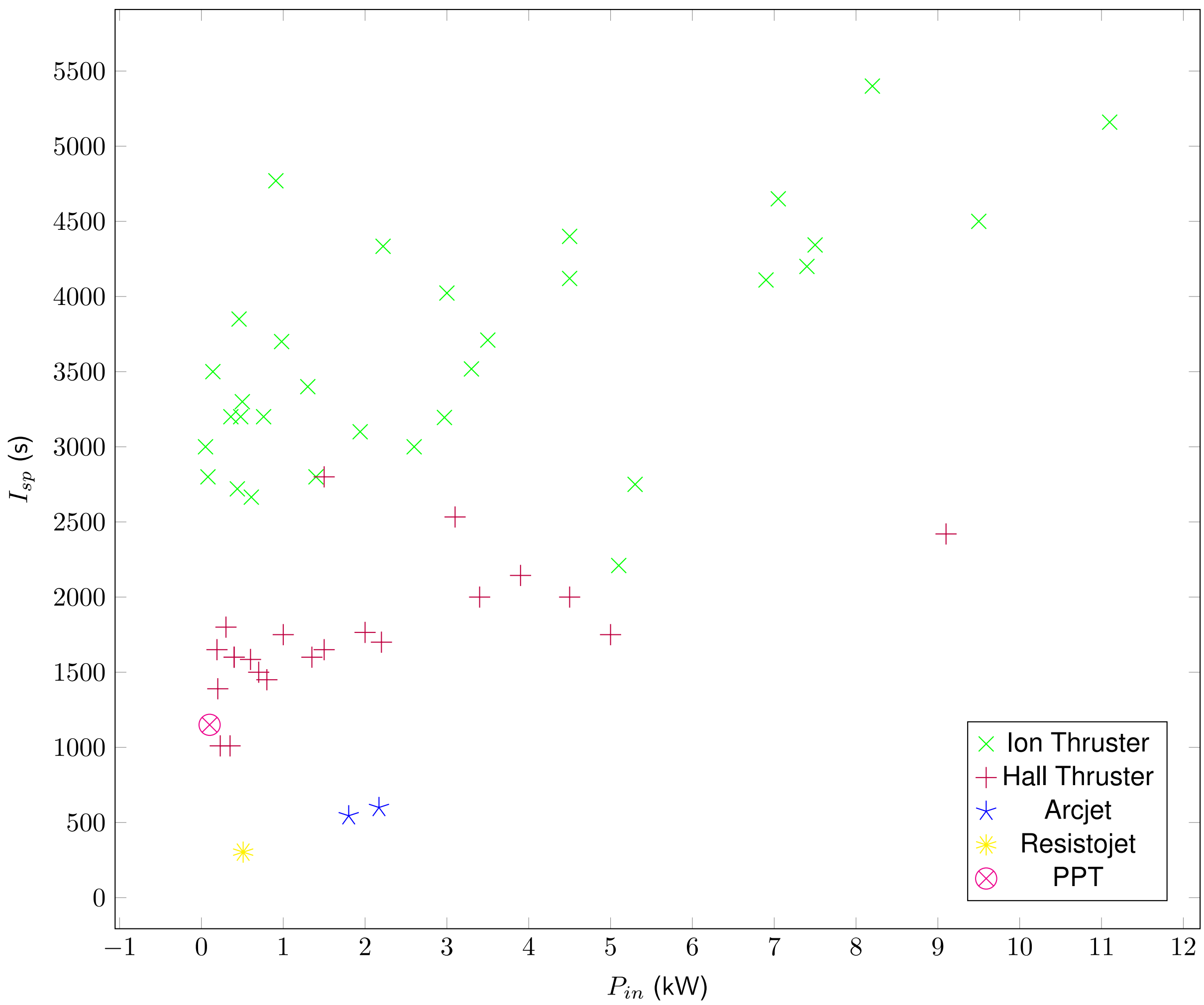

Figure 4.10: Vacuum I_{sp} plotted as a function of input power, of EP

efficiency, of about 30 %, while ion thrusters have efficiencies ranging from below 30 % up to 90 %. Hall effect thrusters have thruster efficiencies ranging between 45-60 %. The only PPT with info about both vacuum I_{sp} and efficiency is the PRS-101 with a η of 82 %.

Figure 4.12 displays vacuum I_{sp} in function of the T/w ratio. Unlike what we saw in figure 4.6, T/w does not seem to have an impact on the specific impulse inside one sub-type of EP, as different P_{in} and η have a major impact in both the I_{sp} a T/w. Electrothermal systems are the systems with lower I_{sp} but capable of achieving the highest T/w of EP, in the opposite side, ion thrusters have the highest I_{sp} capabilities of EP but the lower T/w.

From figures 4.10-4.12, the different I_{sp} capabilities and trends for EP technologies are presented. Both electrothermal systems, arcjet and resistojet, have the lower I_{sp}. Electrostatic or ion thrusters achieve the highest I_{sp} of all types of EP considered. Although this type of SPS has the capability of reaching values superior to 10 000 s (as is the case of the NASA Interstellar), most systems using this technology deliver between 2500 s and 5500 s of specific impulse. Electromagnetic thrusters have different capabilities depending on the technology used. Hall effect thrusters are the most discussed in the bibliography out of all types of electromagnetic thrusters, most HET have a level of I_{sp} comprehended

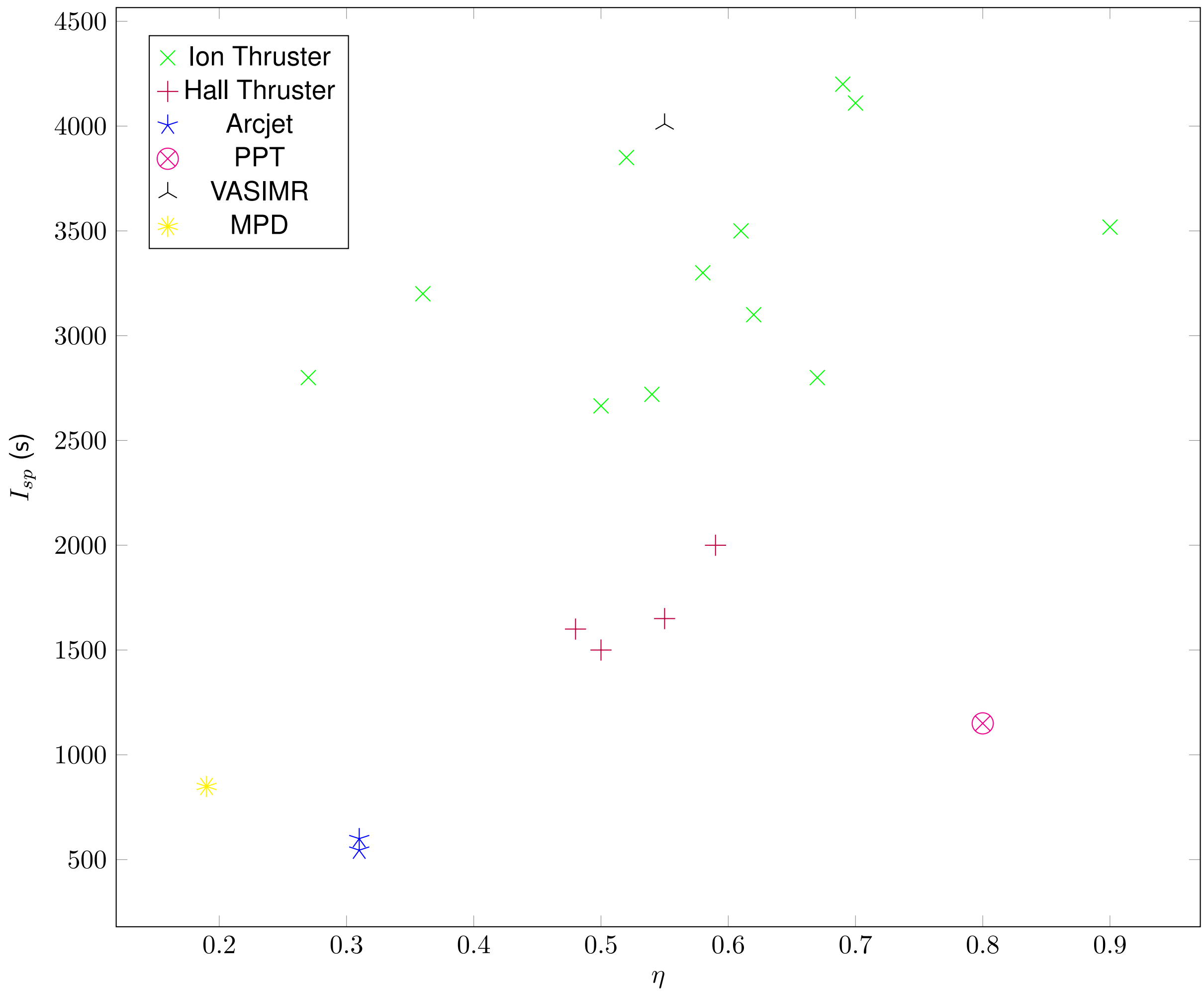

Figure 4.11: Vacuum I_{sp} plotted as a function of thruster efficiency, of EP

between 1000-3000 s.

Table 4.2: Key parameters of original and upgraded versions of SPS at Sea Level (SL) and Vacuum (V), taken from [8, 54–56]

SPS	Type of EP	I_{sp} (s)	P_{in} (kW)	η (%)
NASA Interstellar	Ion thruster	14000	30	-
HiPEP	Ion thruster	9620	39.3	80
NEXIS	Ion thruster	7500	20	-
VX-200SS	VASIMR	4010	108	55
MPD	MPD	849	150	19

The big limitation of EP is the electrical power available in space and its weight. Therefore, if MW class powers are achieved, it would result in a significant improvement of I_{sp}. Also, if the specific power (or power-to-mass ratio) is increased the T/w capabilities of EP would also improve.

From figures 4.10-4.12 the NRT relative to specific impulse of EP can be extrapolated:

- **NRT2 - Electric Propulsion Systems** I_{sp} can go from 300 s to 14 000 s, depending on the type of technology used. [32].
- **NRT2a** - Electrothermal system typically have specific impulses ranging from 300 s up to 600 s,

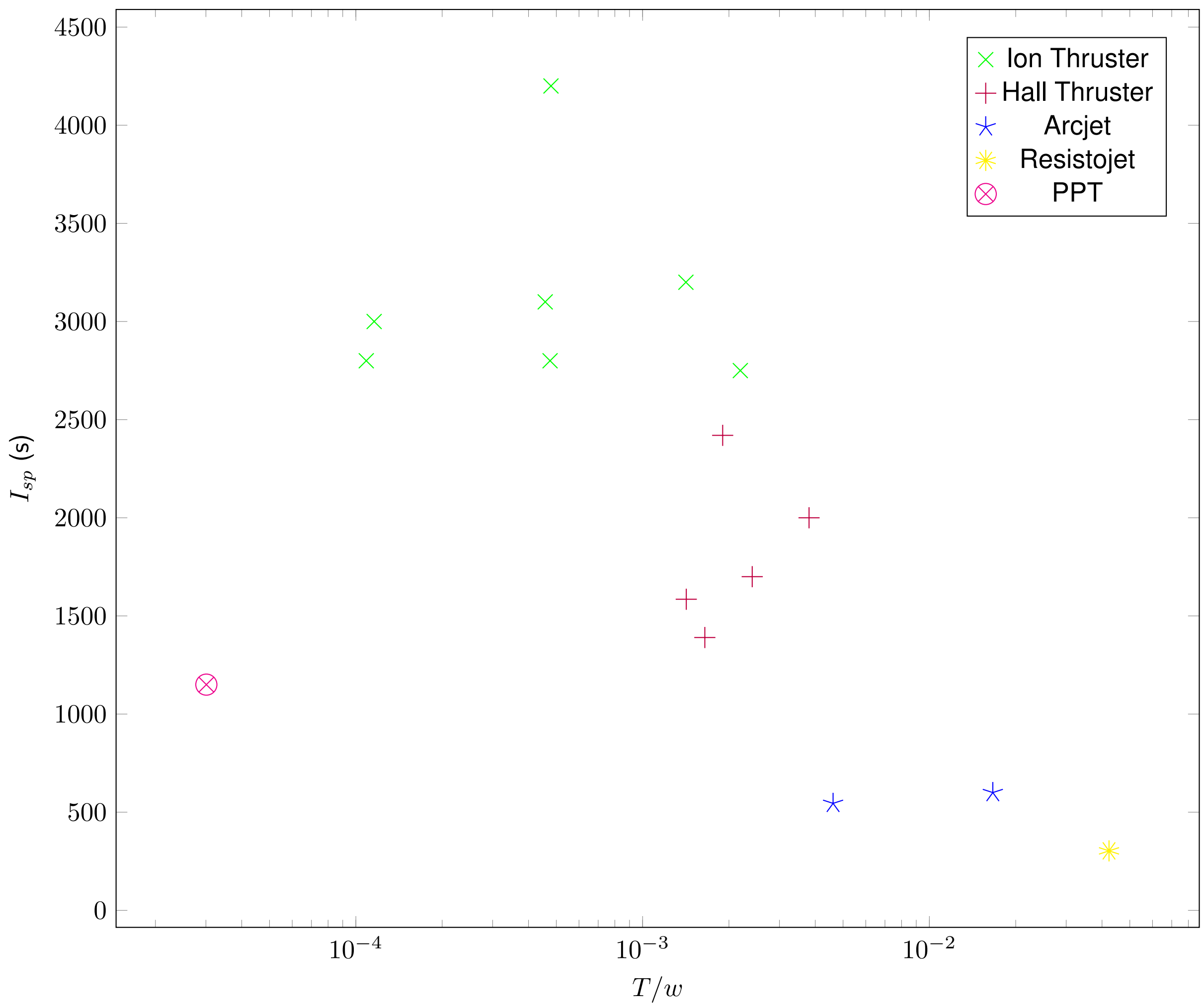

Figure 4.12: Vacuum I_{sp} plotted as a function of T/w, of EP

depending on the type of electrothermal system used.

- **NRT2a.1** - Resistojets have a I_{sp} level around 300 s.
- **NRT2a.2** - Arcjets have I_{sp} about 500-600 s.
- **NRT2b** - Electrostatic systems, or ion thrusters I_{sp} is often comprehended around 2500-4500 s, with electrical power input up to 10 kW and a thruster efficiency from 40 % to 70 %. When the P_{in} is scaled up to 30 kW these systems can reach values up to 14 000 s of specific impulse.
- **NRT2c** - Electromagnetic systems can provide I_{sp} from 1000 s to 4000 s, depending on the technology employed and the electrical power input.
- **NRT2c.1** - PPT have a I_{sp} around 1150 s.
- **NRT2c.2** - Hall effect thruster have a I_{sp} between 1000-3000 s, most systems with an electrical power input level up to 5 kW and thruster efficiency of 50-60 %.
- **NRT2c.3** - MPD thruster have a I_{sp} around 850 s at 150 kN power level.
- **NRT2c.4** - VASIMR have a I_{sp} around 4000 s.

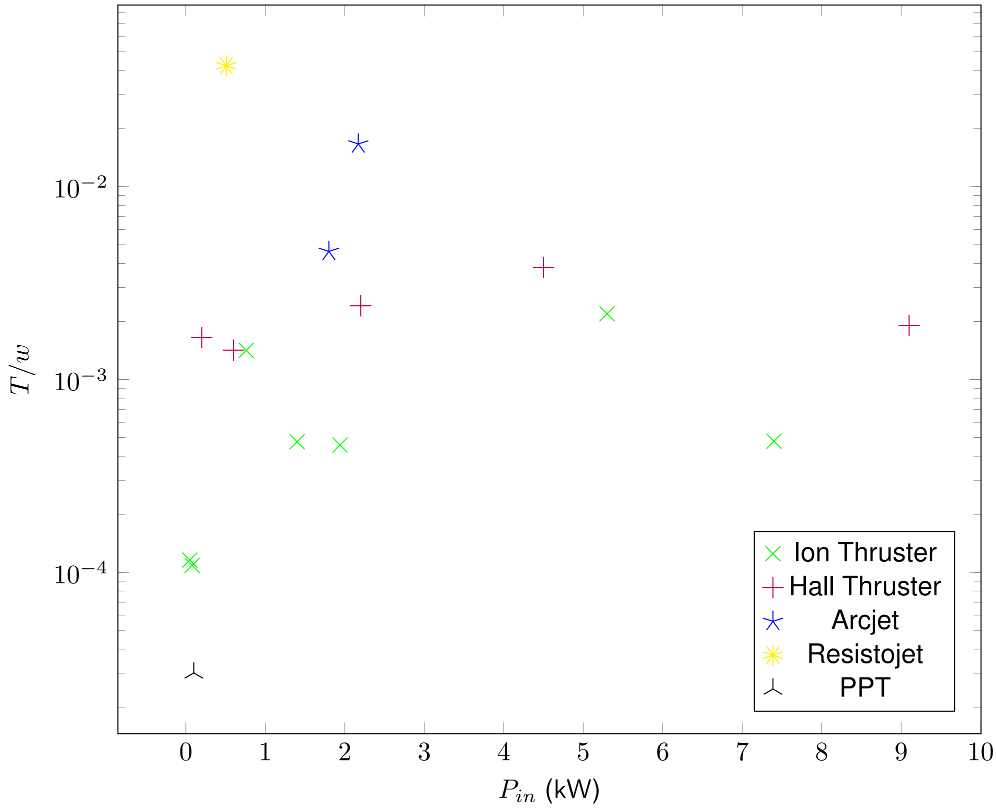

Figure 4.13: Vacuum T/w plotted as a function of input power, of EP

4.4.2 Thrust-to-Weight Ratio

Figure 4.13 gives the T/w of EP as a function of the electrical power input. There is a positive tendency between the plotted variables, increasing the P_{in} has a positive consequence to the T/w ratio. Some out-liners were expected, as for a constant P_{in} the specific impulse is inversely proportional to the T/w, as seen in 2.2.1.

Figure 4.14 displays the T/w ratio as a function of η. The thruster efficiency is a measurement of how efficiently the electrical power and propellant are used in the production of thrust [6]. Although not enough data was found for the majority of the systems to confirm this, ion thrusters have a positive tendency, with the increase of the thruster efficiency there is an increase in T/w.

From figures 4.13 and 4.14 the NRT relative to thrust-to-weight ratio of EP can be extrapolated:

- **NRT2 - Electric Propulsion Systems** T/w can go from 10^{-5} up to 10^{-1}, depending on the type of technology used.
- **NRT2a** - Electrothermal system typically have thrust-to-weight ratio ranging from 10^{-3} up to 10^{-1}, depending on the type of electrothermal system used.
- **NRT2a.1** - Resistojet have a T/w level around 5×10^{-1}.
- **NRT2a.2** - Arcjets have T/w comprehended between 10^{-2}-10^{-3}.
- **NRT2b** - Electrostatic systems, or ion thrusters T/w is comprehended around 10^{-4}-10^{-3}.
- **NRT2c** - Electromagnetic systems can provide T/w from 10^{-5} to 10^{-3}, depending on the technology employed.

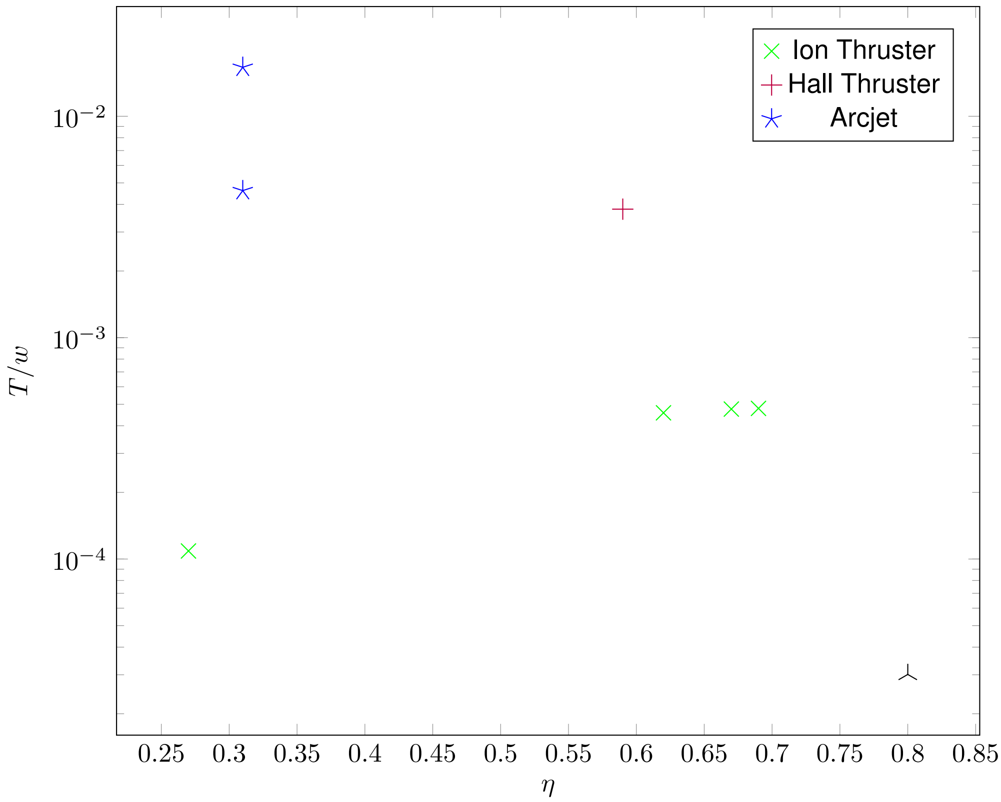

Figure 4.14: Vacuum T/w plotted as a function of thruster efficiency, of EP

- **NRT2c.1** - PPT have a T/w around 3×10^{-5}.
- **NRT2c.2** - Hall effect thruster have a T/w between 10^{-3}-10^{-2}.

4.5 Nuclear Thermal Propulsion Analyses

Most data found about this type of SPS was on the NERVA/Rover program, which during the space race built and tested 20 reactors. At the other end of the race, Russia, at the time USSR, developed and tested the RD-0410. More recent developments were made in the 1990s by two different programs: Project Timberwind with three reactors designed, and Pratt & Whitney company designed three versions of theirs PW XNR 2000. The systems designed by both Project Timberwind and Pratt and Whitney were never built nor tested. The performance values of the PW XNR 2000 versions are a result of simulations. While Project Timberwind SPS values are relative to the design goals [57].

The gathered data about NTP systems is displayed in figure 4.15. Due to the very high, and theoretical, value of T/w, Project Timberwind systems were left out of the figures. These systems were designed to reach $I_{sp} = 1000$ s and a $T/w = 30$, across all three versions. The difference between the three versions was in the thrust delivered, with a T level of 45, 75 and 250 kN [40].

Figure 4.15 displays the vacuum I_{sp} as a function of the T/w, of NTP systems. The I_{sp} across all systems are very similar, ranging from 700 and 1000 s. There is a big difference in the T/w capabilities of the systems, especially if considering the Project Timberwind SPS. As we saw in section 2.3 the NTP systems are limited by the materials structural temperatures. Advances made in high-temperature

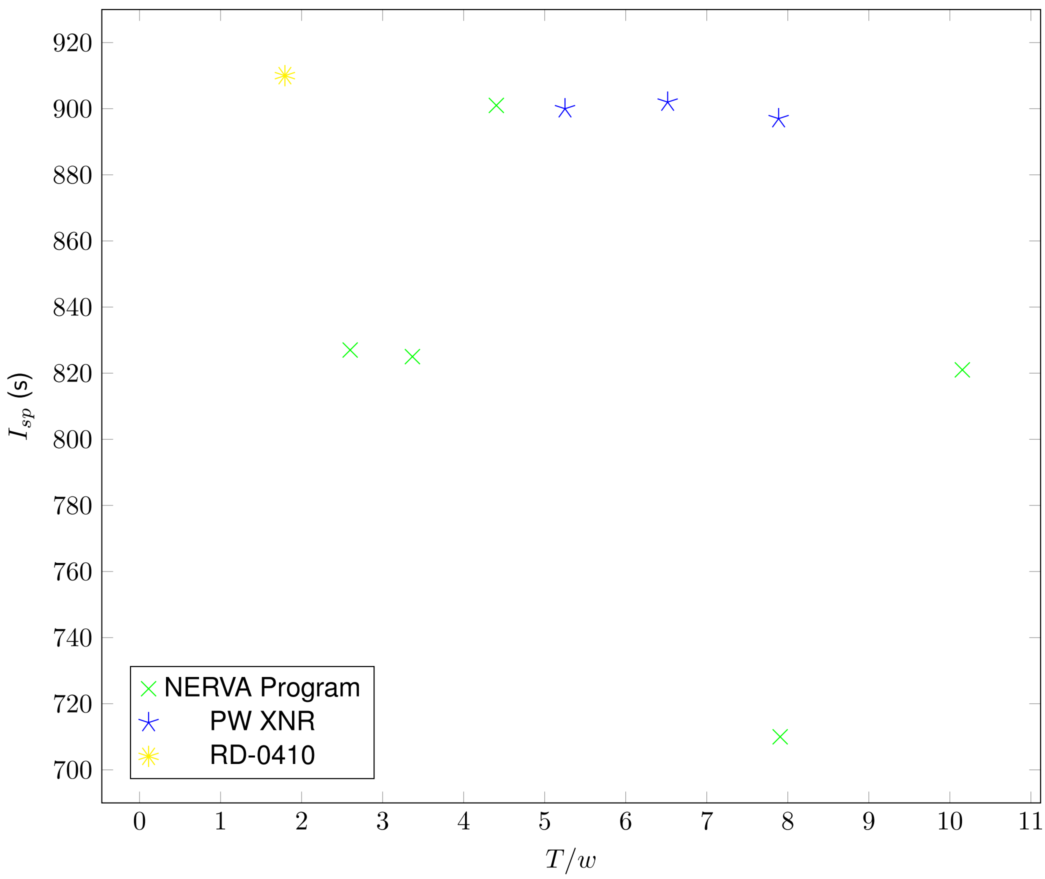

Figure 4.15: Vacuum I_{sp} plotted as a function of T/w, of NTP

metals since the space race era, greatly reduced the weigh of the SPS, explaining the difference in T/w possibilities. NTP are the only systems off all non-chemical technologies capable of delivering a thrust-to-weight ratio of the propulsion system above one.

From figure 4.15 the NRT relative to thrust-to-weight ratio and the specific impulse of NTP can be extrapolated:

- **NRT3 - Nuclear Thermal Propulsion System** using a nuclear solid core usually provide I_{sp} around 700-1000 s and a T/w up to 30.

4.6 Advanced Propulsion System Analyses

Figure 4.16 displays the vacuum specific impulse as a function of the vacuum thrust, of advanced propulsion technologies. Data relative to the thrust-to-weight ratio is essential to understand the capabilities of these concepts, however it is difficult to identify relevant and accessible literature.

There seems to be no tendency between the thrust and specific impulse across all technologies displayed in figure 4.16. The highest I_{sp} is achieved by the two different fusion reactor concepts, inertial confinement fusion (ICF) and magnetic confinement fusion (MCF). ICF systems specific impulse range between 10^4 s to 10^6 s, while MCF are around 10^6 s. This is two orders of magnitude greater than any EP system described in the accessed literature. Advanced fission technologies have relatively lower I_{sp}

capabilities, around 10^3 s to 10^4 s, comparable to the values obtained of electrostatic and electromagnetic systems.

Regarding the thrust capabilities, all technologies have thrust capabilities comparable to those of chemical and NTP systems. Advanced fission technologies thrust ranges from 10^3 N to 10^7 N. Out of advanced fission, external-pulsed plasma propulsion (EPPP) and fission fragment propulsion (FFP) have the highest values of thrust, around 10^7 N. Concepts using a liquid core have a T level off 10^3 N, or if using a gaseous core the level is higher between 10^5 N and 10^6 N. Fusion concepts thrust level varies largely, MCF concepts thrust ranges from 1 N to 10^6 N, while ICF T levels are between 10^6 N to 10^7 N.

From figure 4.16 the NRT relative to the specific impulse of advanced propulsion technologies can be extrapolated:

- **NRT4 - Advanced Propulsion Systems** vacuum I_{sp} ranges between 2000 s and 10^6 s.
- **NRT4a** - Advanced fission SPS have I_{sp} around 2000 s to 10^4 s.
- **NRT4a.1** - Systems using a gas core have I_{sp} around 2000 s.
- **NRT4a.2** - Fission fragment propulsion are capable of providing a I_{sp} of 6600 s.
- **NRT4a.3** - External-pulsed plasma systems are capable of providing I_{sp} between 10^3 s and 10^4 s.
- **NRT4a.4** - Systems using a liquid core have I_{sp} around 2000 s.
- **NRT4b** - Fusion systems have I_{sp} that can go from 10^4 s to 10^6 s.
- **NRT4b.1** - Magnetic confinement fusion concepts I_{sp} ranges from 10^4 s to 10^6 s.
- **NRT4b.2** - Inertial confinement fusion specific impulse is around 10^6 s.

All NRT relative to I_{sp} and T/w are summarized in table 4.3.

Table 4.3: A summarize of the NRT identified

NRT	Type of SPS	I_{sp}, s	T/w
NRT1	Chemical	260-470	10-170
NRT1a	Solid Propellant	260-300	10-120
NRT1b	Bipropellant	300-470	20-170
NRT1b.1	Storable	310-350	20-170
NRT1b.2	LOX/LH_2	410-470	30-80
NRT1b.3	LOX/RP-1	300-360	60-150
NRT1b.4	LOX/CH_4	310-380	30-80
NRT2	Electric	300-14000	10^{-5}-10^{-1}
NRT2a	Electrothermal	300-600	10^{-3}-10^{-1}
NRT2a.1	Resistojets	300	5×10^{-1}
NRT2a.2	Arcjets	500-600	10^{-3}-10^{-2}
NRT2b	Electrostatic	2500-4500	10^{-4}-10^{-3}
NRT2c	Electromagnetic	1000-4000	10^{-5}-10^{-3}
NRT2c.1	PPT	1150	3×10^{-5}
NRT2c.2	HET	1000-3000	10^{-3}-10^{-2}
NRT2c.3	MPD	850	-
NRT2c.4	VASIMR	4000	-
NRT3	NTP	700-1000	1-30
NRT4	Advanced Propulsion Systems	2000-10^6	-
NRT4a	Advanced Fission	2000-10^4	-
NRT4a.1	Gas Core	2000	-
NRT4a.2	FFP	6600	-
NRT4a.3	EPPP	10^3-10^4	-
NRT4a.4	Liquid Core	2000	-
NRT4b	Fusion	10^4-10^6	-
NRT4b.1	MCF	10^4-10^6	-
NRT4b.2	ICF	10^6	-

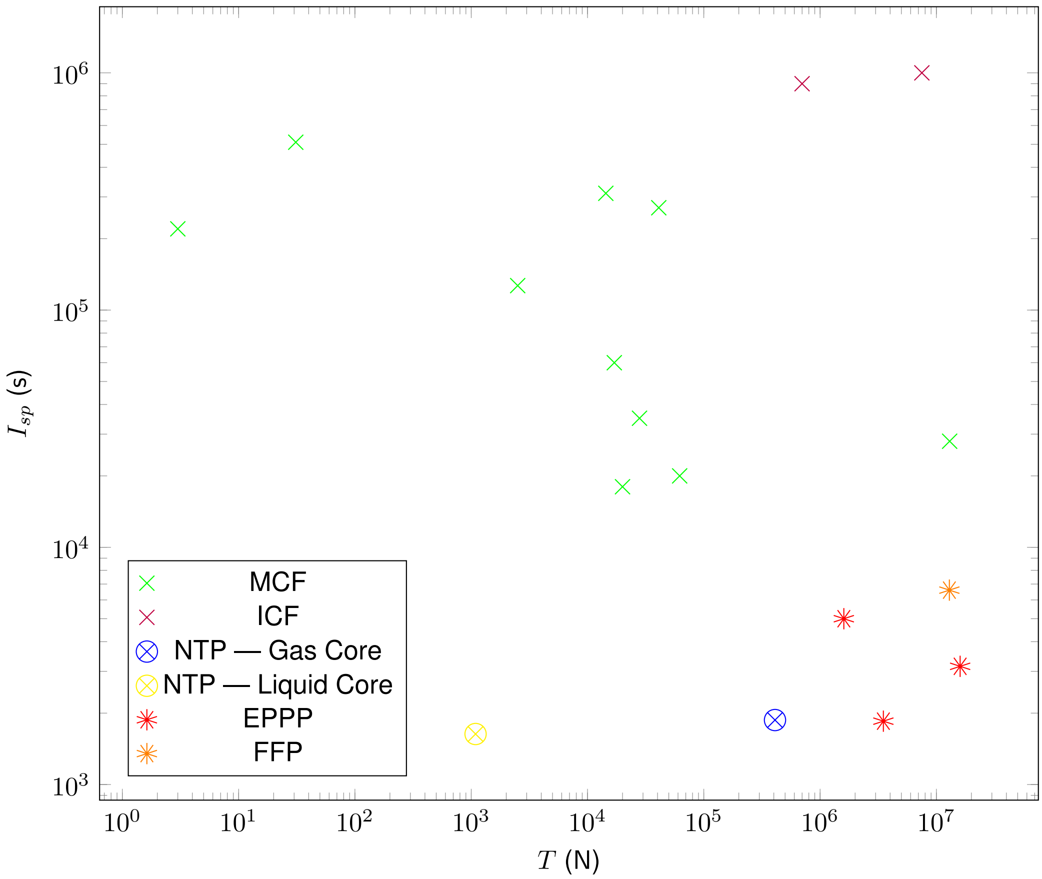

Figure 4.16: Vacuum I_{sp} plotted as a function of T, of Advanced Propulsion Systems

Chapter 5

Conclusions

Each Space Propulsion System (SPS) is designed for a different objective, the capabilities of each one depends not only of its main function but also according to the technology employed. In this work we aimed to identify trends and maturity level of different SPS technologies, and compare them to the typical values from the accessed literature. In order to achieve this, first we explored the space propulsion fundamentals: the impact a SPS has on the vehicle, how to separate and categorize different types of SPS, and which characteristics have an impact on their performance. With this knowledge we surveyed the literature, to construct a database of SPS and their attributes. With this database, we identified and analysed several trends relative to different types of technologies (NRT — summarized in table 4.3) of SPS to compare them with supposedly typical values from the literature (RT — summarized in table 4.1).

We analysed the NRT relative to specific impulse of chemical propulsion, with the RT from the literature. Here, it was detected a discrepancy relative to solid propellant systems, where the values of NRT1a are larger than those of RT1a by a factor of 1.5. This discrepancy can be explained due to a lack of data relative to military application in the database. For the different types of liquid propellant combinations, RT1b.1-RT1b.4 give representative values of specific impulse for contemporary chemical systems, with a chamber pressure $p_c = 6895$ kPa and a nozzle area ratio $A_e/A_t = 81$. In figures 4.4 and 4.5 we plotted I_{sp} as a function of p_c and A_e/A_t, respectively. The identified range of values for all liquid propellant combinations of chemical propulsion systems, are in the spectrum given by the RT.

As for the T/w ratio of chemical propulsion, no RT were found relative to different propellant combinations. Sutton refers in [6, p. 39] the typical values for solid and liquid chemical propellant systems are $T/w < 200$. From figures 4.6-4.9 we identified NRT, relative to different propellant combinations of chemical SPS. Storable propellants have the highest T/w of chemical SPS gathered in the database, with $T/w < 170$. Systems using the propellant combination LOX/RP-1 achieve values of $T/w < 150$. As for LOX/CH_4 and LOX/LH_2 propellant combinations, both have similar capabilities, $T/w < 80$.

Electric propulsion has a lower maturity level compared to chemical propulsion. The identification of trends is harder in this type of SPS because there is a lack of information relative to the majority of EP technologies, with the exception of ion and hall effect thrusters. This was most impactful on the evaluation of the T/w ratio. Also, the I_{sp} and T/w performance depends on the electrical power input available P_{in}. The values identified in NRT2 are relative to power levels up to a few tens of kW, with the progressive increase of the available electrical power levels in space, the maximum values of the EP are bound to increase.

Nuclear Thermal Propulsion (NTP), is a technology with a lower maturity level (TRL $\leq$ 6) in comparison to Chemical Propulsion and some types of Electrical Propulsion (ion and hall effect thrusters). The values identified in NRT3 are very similar to the values of RT3. The analysed systems have a specific impulse between 700 s and 1000 s with a $T/w < 30$. The performance of this technology surpasses chemical propulsion capabilities, but since it never had the same amount of investment (as a result of the dangers associated with using nuclear fuels), its capabilities were never utilized. If this technology reaches the level of maturity and security of chemical propulsion, it's expected to be a viable alternative to chemical propulsion [16].

Regarding Advanced Propulsion Technologies, the extrapolated NRT4 values are theoretical, as are the RT4 identified in literature. These values have to be thought as expected values, since these technologies development and maturity are very low (TRL $<$ 3). As a result, it is difficult to analyse them. However, we can recognize that a development in one of these concepts, and if the theoretical values become a reality, would enable missions that, with the today's existing technical capabilities, are unreachable.

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