Injector design and test for a high power electrodeless plasma thruster

ROMAIN DELANOË

Master of Science Thesis
Stockholm, Sweden 2011
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Abstract

The HPEP (High Performance Electric Propulsion) thruster is expected to be the outcome of an innovative project initiated by the Swedish Space Corporation. It combines the concept of a 10 kW electrodeless plasma thruster designed by the Elwing Company and the ADN based monopropellant LMP-103S developed by ECAPS and used in the HPGP thrusters of the Prisma Satellites. Using a chemically energetic propellant in an EP thruster will allow mass and cost reduction by providing two propulsion systems sharing the same tank. This thruster will be suitable for the apogee raising manoeuvre of geostationary satellites; it will allow to carry more transponders and to obtain a better return on investment than with a classical apogee kick motor. This Master Thesis focuses on the design and test of the injector that will thermally decompose the liquid LMP-103S so it can enter in the plasma chamber in a gaseous state. The heating power required by the injector is calculated, which leads to a final design composed by a cartridge heater of 400 W inserted in a stainless steel cylinder. The liquid flows through seven other holes drilled around the heater. This injector is tested at both atmospheric and low pressure with deionized water. Results regarding the power required to vaporize water confirm the theoretical estimation. Steam flow without any liquid droplets is achieved in steady state at low pressure with a maximum temperature on the surface of the injector between 230°C and 260°C.
Sammanfattning

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From my double degree university KTH, I address a special acknowledgement to Nickolay Ivchenko who helped me to find proper equipments for tests when I needed it and who reviewed this report.

Last but not least, this work couldn’t have been done without the unrelenting support of my family, friends and of Aleksandra, my girlfriend and sambo.
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Chapter 1

Introduction

1.1 Project overview

This Master Thesis has been carried out from the 10th of January to the 8th of July 2011 at the Space System Division of the Swedish Space Corporation located in Solna, Sweden. This company designs and develops a wide range of space systems which includes among others satellites.

ECAPS is one of the subsidiaries of the Swedish Space Corporation. ECAPS develops rocket propellants less toxic and easier to handle than hydrazine but with the same applications: attitude and orbit control of satellites. Ten years of R&D have led to a monopropellant based on ammonium dinitramide (ADN): the LMP-103S. Together with this monopropellant, a suitable thruster has been developed. It is called the HPGP thruster, which stands for High Performance Green Propulsion.

The High Performance Electrical Propulsion (HPEP) project started with an assessment: during the transfer of a satellite from geostationary transfer orbit (GTO) to geostationary orbit (GEO), the electric power provided by the solar panels is nearly not used. This corresponds to approximately 10 kW of spare electric power that could be used with an Electric Propulsion (EP) thruster for the apogee raising of the satellite. If EP is considered for the apogee raising, the GEO satellite would still need several chemical thrusters for attitude control purposes, HPGP thrusters for example. Then, the main idea of the HPEP project is to use the same propellant than the one used in the chemical propulsion system in order to have only one tank onboard. Using the LMP-103S instead of Xenon in an EP thruster will allow mass and cost reduction by reunifying the propulsion systems.

A good candidate for the EP thruster is an electrodeless plasma thruster currently in development by the Elwing Company. Indeed, the use of a chemical propellant would make quick and irreversible damages to electrodes, that’s why an electrodeless design is preferred. As this thruster is planned to be used in the HPEP project, it is called in the followings “HPEP thruster”. Technical details about the thruster are presented in Chapter 3.

If one considers the theoretical performances of the HPEP thruster with LMP-103S, can an apogee raising with EP be economically viable compared to a standard apogee kick engine?
CHAPTER 1. INTRODUCTION

Indeed, an EP transfer is slow and is expecting to take several months compared to five days for a chemical transfer. This is the first question to be answered when developing a new technology and Chapter 2 brings the proper argumentation on this topic.

Once it has been proven that this project has a commercial interest, the Master Thesis focuses on its main task: the design and test of the part of the HPEP thruster that will turn into gas the liquid LMP-103S. This part is called “injector” and its design is the key point in the project. Its aim is to vaporize and partially break down the liquid propellant and then inject into the thruster chamber this gaseous flow. The optimization of the thrust to power ratio in Chapter 4 leads to a graphic ranking of chemical species in function of their ability to generate a lot of thrust with the HPEP concept for a given power value. The decomposition of the LMP-103S shall be oriented towards certain species and shall avoid the formation of certain others.

The design of the injector is based on thermodynamic investigations of the power required in order to vaporize the flow of liquid propellant. This is presented in detail in Chapter 5 which is followed by the test of the injector inside a vacuum chamber in Chapters 6 and 7.

1.2 Geostationary orbits

Geostationary orbits (GEO) are circular orbits at 35786 km altitude in which satellites are always over the same point on the Earth’s surface, right over the equator. The revolution period of a spacecraft orbiting at this altitude is exactly the same as the rotation period of the Earth [24]. For ground observers, a satellite following a geostationary orbit appears motionless, at a fixed position in the sky.

Figure 1.1: Operational geostationary satellites around the Earth [24]
In this way, antennas on the ground don’t need to track GEO satellites, they can just receive and transmit without moving. They are therefore cheaper than tracking antennas. GEO satellites have revolutionized telecommunications, television broadcasting, weather forecasting and intelligence gathering in the military sector. There are approximately 300 operational GEO satellites, and as shown in Figure 1.1, they cover most of the areas over the equator.

Decreasing the mass of GEO satellites’s propulsion system would lead to cheaper launches. This substantial cost saving would contribute to lower the prices of the services provided by GEO satellites. Developing more efficient propulsion systems is therefore in the interest of the prime contractor (Astrium, Thales Alenia Space, Boeing, etc), of the customer (Eutelsat, AsiaSat, Hispasat, etc) and of the final user (you and me).

1.3 Hohmann transfers from GTO to GEO

Geostationary satellites are first put on a geostationary transfer orbit (GTO) by the launch vehicle. The apogee of a GTO is situated at a geostationary altitude, at 35786 km, while its perigee is only at 220 km above the Earth. The perigee of a GTO is at a low Earth orbit (LEO) altitude in order to reduce the amount of propellant used by the launcher, but also in order to decrease the orbital lifetime of the last stage. Indeed, GTOs tend to be saturated due to all the geostationary satellites that have been launched, and a lot of space debris have been accumulated there.

GEOs are directly above the Earth’s equator; therefore these orbits have a 0° inclination. Most of the carrier rockets are not launched at the equator, see Table 1. This requires not only to increase the perigee of the GTO but also to change its inclination. Indeed, the inclination of the GTO is approximately equal to the one of the launch site. In order to simplify the calculation, it is assumed here that the launch pad is at the equator, and therefore no inclination change is needed. This is a possible launch scenario: the self-propelled Ocean Odyssey platform operated by the Sea Launch company is able to launch Zenith 3SL rockets wherever on the oceans, and especially at the equator [3].

<table>
<thead>
<tr>
<th>Country</th>
<th>Launch site</th>
<th>Inclination</th>
</tr>
</thead>
<tbody>
<tr>
<td>French Guiana</td>
<td>Guiana Space Centre, Kourou</td>
<td>5.2° N</td>
</tr>
<tr>
<td>USA</td>
<td>Kennedy Space Center, Florida</td>
<td>28.6° N</td>
</tr>
<tr>
<td>Kazakhstan</td>
<td>Baikonur Cosmodrome, Tyuratam</td>
<td>46.0° N</td>
</tr>
</tbody>
</table>

Table 1.1: Inclination of well-known launch pads [2]

In the following calculation, $\mu$ is the gravitational parameter and in the two-body problem Earth/satellite, it corresponds to:

$$\mu = G \left( m_{Earth} + m_{sat} \right) = G m_{Earth} = 3.987 \cdot 10^{14} \text{ N} \cdot \text{m}^2 \cdot \text{kg}^{-1}$$

Figure 1.2 represents the trajectory of a satellite with a speed $u$ from launch to GTO and then to GEO. $a_1$ and $a_2$ are the distances between the centre of the Earth and respectively
the perigee and the apogee of the GTO. Therefore, $a_2$ also corresponds to the radius of the GEO. If $r$ is the distance between the centre of the Earth and the satellite, the specific energy on the GTO is [1]:

$$E = \frac{u^2}{2} - \frac{\mu}{r} = -\frac{\mu}{a_1 + a_2}$$  \hspace{1cm} (1.2)

So the speed on the GTO is:

$$U = \sqrt{\frac{2\mu}{r} - \frac{2\mu}{a_1 + a_2}}$$  \hspace{1cm} (1.3)

Point 3 in Figure 1.2 is situated at the apogee of the GTO. At this point, $r = a_2$ and the speed is then:

$$U_{3_{\text{GTO}}} = \sqrt{\frac{2\mu}{a_2} - \frac{2\mu}{a_1 + a_2}}$$  \hspace{1cm} (1.4)

The apogee of the GTO is exactly on the GEO. However, the GEO is a circular orbit so its speed is different at this point:

$$U_{3_{\text{GEO}}} = \sqrt{\frac{\mu}{a_2}}$$  \hspace{1cm} (1.5)

In order to reach the GEO from the apogee of the GTO, an impulse shall be given to the satellite:

$$\Delta U_{GTO \rightarrow GEO} = U_{3_{\text{GEO}}} - U_{3_{\text{GTO}}} = \sqrt{\frac{\mu}{a_2}} - \sqrt{\frac{2\mu}{a_2} - \frac{2\mu}{a_1 + a_2}} = 1476 \text{ m} \cdot \text{s}^{-1}$$  \hspace{1cm} (1.6)

This impulse is given by an onboard apogee kick motor. It both corrects the orbit’s inclination and transforms the elliptical orbit into a circular one. An example of apogee engine is

![Figure 1.2: From launch to GEO using a GTO](image-url)
1.4. HPGP THRUSTER

S 400-15 from Astrium Space Transportation shown in Figure 1.3 [4]. It can deliver a thrust of 420 N with an $I_{sp}$ of 320 s. It is a bipropellant engine using monomethylhydrazine as a fuel and a mixture of nitrogen tetroxide with approximately 3% nitric oxide as an oxidiser. GEO satellites from the Eutelsat W series are for example using this kind of apogee kick motors.

![S 400-15 apogee engine from Astrium Space Transportation](image)

The average mass of geostationary telecommunication satellites launched nowadays is $m_{sat} = 6000$ kg. The mass of propellant required for delivering the $\Delta U_{GTO\rightarrow GEO}$ with the apogee engine is, with an $I_{sp}$ of 320 s [1]:

$$m_{prop} = \left(1 - e^{\frac{\Delta U_{GTO\rightarrow GEO}}{g_0 I_{sp}}} \right) m_{sat} = 2251 \text{ kg} \quad (1.7)$$

On can conclude that the dry mass of the satellite is then:

$$m_{dry} = m_{sat} - m_{prop} = 3749 \text{ kg} \quad (1.8)$$

1.4  HPGP thruster

Once a satellite is on its desired orbit, it still needs thrusters in order to change its orientation in space (attitude control) and to stay on the right orbit (station keeping). Indeed, several sorts of disturbing effects have a significant impact on satellites: non-spherical shape of the Earth, three-body interaction with the Moon, atmospheric drag, solar radiation pressure, etc. All these effects have an influence on the orbit and on the orientation of the spacecraft. For example, a common choice can be several monopropellant hydrazine thrusters for attitude control purposes.
The High Performance Green Propulsion (HPGP) is an alternative to traditional propulsion systems using hydrazine as a propellant. It was developed by ECAPS, a subsidiary of the Swedish Space Corporation. The general design of a HPGP thruster is presented in Figure 1.4.

![HPGP thruster design from ECAPS](image1)

The HPGP thrusters are as of 2011 available in a range going from 1 to 22 N. For example, the 1 N HPGP Rocket Engine is designed for attitude and orbit control of small-sized satellites. It supports operation in steady-state and pulse mode. It has a specific impulse of 233 s in vacuum [18]. This is 6% higher than the specific impulse of the 1 N hydrazine thruster from EADS Space Transportation, which has a specific impulse of 220 s in vacuum [19].

![The Prisma Satellites, Main on the right and Target on the left](image2)

The flight demonstration of the HPGP thrusters took place on the Prisma Satellites, project led by the Swedish Space Corporation. The mission configuration consists of a maneuverable spacecraft called “Main” and a simpler one called “Target”, as shown in Figure 1.5. The satellites were launched in a low earth orbit in June 2010 and are currently carrying out...
1.5 Composition and properties of the LMP-103S

The propellant of the HPGP thruster is based on ammonium dinitramide (ADN) and is known as the LMP-103S. A small sample is presented in Figure 1.7. It is less toxic and easier to handle than hydrazine. Indeed, in order to fuel a satellite with hydrazine, it is necessary to implement rigorous safety procedures. The fuelling personnel has for example to wear SCAPE suits designed to protect them from the toxic fuel, as shown in Figure 1.8. The handling procedures with the LMP-103S are simpler and only normal protective clothing for handling of chemicals is needed. The launch fuelling is therefore both easier and cheaper [15].

Unlike hydrazine, the LMP-103S is low-toxic and is not carcinogenic. This point increases safety and reduces the environmental impact. All these characteristics explain why the
CHAPTER 1. INTRODUCTION

Figure 1.8: Common hydrazine filling operations where engineers are dressed in SCAPE suits [20]

LMP-103S is also called “Green propellant” by its designers.

<table>
<thead>
<tr>
<th>Chemical species</th>
<th>Formula</th>
<th>% by mass</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADN</td>
<td>$\text{NH}_4^+\text{N(NO}_2\text{)}_2^-$</td>
<td>60-65</td>
</tr>
<tr>
<td>Methanol</td>
<td>$\text{CH}_3\text{OH}$</td>
<td>15-20</td>
</tr>
<tr>
<td>Ammonia</td>
<td>$\text{NH}_3$</td>
<td>3-6</td>
</tr>
<tr>
<td>Water</td>
<td>$\text{H}_2\text{O}$</td>
<td>Balanced</td>
</tr>
</tbody>
</table>

Table 1.2: Composition of the LMP-103S [16]

As detailed in Table 1.2, the LMP-103S is a mixture of ADN, methanol, ammonia and water. It has consequently a higher density than water: $1240 \text{ kg} \cdot \text{m}^{-3}$ at $20^\circ\text{C}$. It decomposes at a higher temperature than $120^\circ\text{C}$ [16].

![Figure 1.9: Ammonium Dinitramide molecular structure](image)

The ADN is an ammonium salt of dinitramic acid $\text{HN(NO}_2\text{)}_2$, called HDN. The Soviet Union discovered the ADN in the 1970s and used it from 1971 in various missile programs [23]. However, it was kept classified until it was discovered independently by the United States in 1989. The ADN is an inorganic oxidizer which has its potential use in solid rocket propellants. ADN delivers a higher $I_{sp}$ than the other oxidizers used, like ammonium perchlorate (AP) which is the most common one. AP has the advantage of all the studies that have been performed for many years on AP-based propellants. On the other hand, it generates chlorinated products which are absolutely not environmentally friendly. The ADN is then an ecological alternative to AP in solid propellants. Indeed, it has a high
burning rate and its combustion products are chlorine-free \[22\].

Though the scientific community has a major interest for the ADN concerning applications in solid propellants, it is used by the HPGP thrusters within a liquid propellant, the LMP-103S.

1.6 Electric Propulsion

The soviet rocket scientist K. Tsiolkovsky was clearly a visionary when he published the following statement in 1911 \[6\]:

“It is possible that in time we may use electricity to produce a large velocity for the particles ejected from a rocket device.”

He already had in these early times an idea of electric propulsion (EP) in the space context. EP was defined later by R. J. Jahn in \[7\] as the followings:

“acceleration of gases for propulsion by electrical heating and/or by electric and magnetic body forces”

Different methods can be used in order to accelerate gases with electrical power. Electrical thrusters for satellites and spacecrafts can be divided into three main categories based on the acceleration method:

- **Electrothermal thrusters**: electricity is used to bring thermal energy to the gaseous propellant, which will be converted into kinetic energy by expanding the hot gas through a nozzle. This is the working principle of Resistojets and Arcjets \[8\]. They are fully operational and are part of the propulsion system of numerous satellites currently in orbit. They are often working with Hydrazine $N_2H_4$.

- **Electrostatic thrusters**: they use the Coulomb force to accelerate ions in the direction of a static electric field. In particular, this category contains Gridded Ion Thrusters (GIT) \[9\] and Hall Effect Thrusters (HET) \[10\]. These two thrusters use either Xenon or Argon and can be found on a range of operational satellites.

- **Electromagnetic thrusters**: in order to accelerate ions, they use either the Lorentz force or an electromagnetic field in which the electric field is not in the direction of the acceleration. This category includes Magnetoplasmadynamic Thrusters (MPD) \[11\], Pulsed Inductive Thrusters (PIT), Variable Specific Impulse Magnetoplasma Rocket (VASIMR) \[12\] and Electrodeless Plasma Thrusters, like the HPEP thruster presented in Chapter 3.

The performances of several of these thrusters are compared with the expected performances of the HPEP concept in Table 3.3.
CHAPTER 1. INTRODUCTION

1.7 One satellite, one tank and two propulsion systems

The EP technologies available today have either a low specific impulse or require a significant amount of electrical power for a given thrust level. In the latter case, the available thrust is in fact limited by the power system capabilities of the spacecraft. Due to the lower thrust of EP systems compared with chemical propulsion systems, EP has only been considered for missions where mass was a problem and where there was a sufficient amount of electric power available. EP technologies of today are expensive, and consequently missions using EP spend a significant part of their budget on the satellite bus where it could have been better spent on the scientific payload. Consequently, EP technologies have so far not been able to make a break-through in the commercial market. This is a vicious circle as the commercial market is thus not contributing to bringing the price of EP technologies down, which would benefit to the scientific community.

To break this circle, the High Performance Electrical Propulsion (HPEP) project was started in order to investigate the feasibility of an electric thruster combining a chemically energetic propellant with a high performance plasma chamber. Compared to most other EP systems that use an inert gas like Xenon as propellant, the HPEP thruster will use the same propellant as the HPGP thrusters: the LMP-103S. Since most EP spacecraft require, for attitude control purposes, a secondary propulsion system based on chemical thrusters, this will allow to reunify the propulsion systems on board. Today’s EP penalizes the mass budget of the spacecraft by requiring a complete separate set of equipments for the Xenon. The multiplication of components drives up the cost of the propulsion systems significantly. Ionizing the chemicals will allow mass and cost reduction by providing two propulsion systems sharing the same propellant tank.

The chemical propulsion system can then provide a high thrust when needed, for example in attitude control in the case of an emergency sun acquisition. On the other hand, the EP system can be used for all the major manoeuvres: orbit raising and north-south station keeping. It could in particular be used for the apogee raising manoeuvre of a geostationary satellite. Indeed, an average GEO communication satellite needs solar panels able to deliver 10 to 15 kW of electrical power in orbit for the transmission of data. During the transfer from GTO to GEO, this power is not used. Thus, it makes sense to use electrical propulsion for the apogee raising of a GEO satellite in order to use this available power.
Chapter 2

Low thrust transfer to GEO

2.1 GTO to GEO

If electric propulsion is now considered, the level of thrust is much lower than with chemical propulsion. The impulse takes then more time, and is continuous over a part of the orbit, or over the entire orbit. Consequently, the Hohmann transfer orbit cannot be used for the calculation of electric propulsion trajectories because it assumes that all the thrust is delivered in only two distinct points.

An electrically propelled satellite has been launched on a GTO and is aiming to reach a GEO. Its wet mass is $m_{sat} = 6000$ kg, like the satellite in Section 1.3, and it is also supposed that the satellite is launched at a $0^\circ$ inclination. Its propulsion system is constituted by one electrical thruster able to deliver a continuous $1$ N thrust with a specific impulse of $1000$ s. These characteristics are aimed by the HPEP thruster detailed on Chapter 3.

A simple analysis is to divide the problem into two parts: first, circularizing the orbit and then, increasing the orbit radius. In order to circularize an orbit, one needs to decrease its eccentricity $e$ to 0. The notion of eccentricity vector $\vec{e}$ can be introduced here in order to simplify the calculation. It is a vector that points from the center of the Earth towards the periapsis of the orbit, and its magnitude is the same than the orbital scalar eccentricity. The change of the eccentricity vector is for a change of tangential speed to the orbit $\Delta V_t$:

$$\Delta \vec{e}_t = \frac{2}{V} \Delta V_t \begin{pmatrix} \cos s \\ \sin s \end{pmatrix} \tag{2.1}$$

where $V$ is the speed of the satellite before the impulse and $s$ is the sidereal angle [26]. For a change of radial speed, the change of $\vec{e}$ becomes:

$$\Delta \vec{e}_r = \frac{\Delta V_r}{V} \begin{pmatrix} \sin s \\ -\cos s \end{pmatrix} \tag{2.2}$$

The effect of a constant thrust (in magnitude and in direction as shown in Figure 2.1) over one orbit can here be analyzed. On the Figure 2.1, one can identify four points of interest:
CHAPTER 2. LOW THRUST TRANSFER TO GEO

- On point 1, the thrust is purely tangential with respect to the \( \vec{r} \) vector. The induced change of the eccentricity vector is then:

\[
\Delta \vec{e}_t^1 = \frac{2}{V} \Delta V_1 \left( \begin{array}{c} 0 \\ 1 \end{array} \right)
\]  
(2.3)

- On point 2, the thrust is purely radial. The change of \( \vec{e} \) is:

\[
\Delta \vec{e}_t^2 = \frac{\Delta V_2}{V} \left( \begin{array}{c} 0 \\ 1 \end{array} \right)
\]  
(2.4)

- On point 3, the thrust is purely tangential. The change of \( \vec{e} \) is:

\[
\Delta \vec{e}_t^3 = \frac{-2}{V} \Delta V_3 \left( \begin{array}{c} 0 \\ -1 \end{array} \right) = \frac{2}{V} \Delta V_1 \left( \begin{array}{c} 0 \\ 1 \end{array} \right)
\]  
(2.5)

- On point 4, the thrust is purely radial. The change of \( \vec{e} \) is:

\[
\Delta \vec{e}_t^4 = \frac{-\Delta V_4}{V} \left( \begin{array}{c} 0 \\ -1 \end{array} \right) = \frac{\Delta V_4}{V} \left( \begin{array}{c} 0 \\ 1 \end{array} \right)
\]  
(2.6)

At this point, it is important to notice that \( \Delta \vec{e} \) is the same direction \( \left( \begin{array}{c} 0 \\ 1 \end{array} \right) \) for the four analyzed points. Soop shows in [26] that it’s the same for all the other points in the orbit. All over the initial elliptic trajectory, the constant thrust changes the eccentricity of
the ellipse in the same way, and all these changes are cumulative. By using the software \textit{TiraXOrbitaL} developed by Christophe Koppel from \textit{KoppoS Consulting Ind.}, it is possible to simulate such low thrust trajectories. This software needs a range of inputs concerning the space mission (initial mass, $I_{sp}$, thrust level and direction, initial orbit, etc.) and one can choose a criterion in order to stop the calculation. This condition can be “The orbit is circular”, “The apogee is at the GEO altitude”, etc.

A simulation with \textit{TiraXOrbitaL} is carried out by using a satellite that has the same characteristics than the one at the beginning of this section: $m_{sat} = 6000$ kg, initial GTO orbit with $0^\circ$ inclination, one thruster of 1 N with a specific impulse of 1000 s. In order to simplify the manoeuvre, one can divide it in two parts. By using the same axis orientation as in Figure 2.1, a long enough constant thrust towards $-\bar{x}$ first transforms the initial GTO orbit into a circular orbit. Then, a constant thrust in the direction $\bar{U}$ (same as the velocity vector) raises the radius of this circular orbit to the geostationary altitude.

![Figure 2.2: Part 1 of the GTO-GEO manoeuvre](image1)

![Figure 2.3: Part 2 of the GTO-GEO manoeuvre](image2)

<table>
<thead>
<tr>
<th>Initial orbit</th>
<th>GTO</th>
<th>circular, 18000 km altitude</th>
<th>GTO</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial mass (kg)</td>
<td>6000</td>
<td>4791</td>
<td>6000</td>
</tr>
<tr>
<td>Thrust level (N)</td>
<td>1</td>
<td>1</td>
<td>1</td>
</tr>
<tr>
<td>Thrust direction</td>
<td>$-\bar{x}$</td>
<td>$\bar{U}$</td>
<td>$-\bar{x}$ and $\bar{U}$</td>
</tr>
<tr>
<td>Criteria for termination</td>
<td>circular orbit</td>
<td>perigee at GEO altitude</td>
<td></td>
</tr>
<tr>
<td>Final orbit</td>
<td>circular, 18000 km altitude</td>
<td>GEO</td>
<td>GEO</td>
</tr>
<tr>
<td>Final mass (kg)</td>
<td>4791</td>
<td>4339</td>
<td>4339</td>
</tr>
<tr>
<td>Total time (days)</td>
<td>137</td>
<td>51</td>
<td>188</td>
</tr>
</tbody>
</table>

Table 2.1: Low thrust GTO-GEO manoeuvre
CHAPTER 2. LOW THRUST TRANSFER TO GEO

The results obtained are presented in Table 2.1. The total duration of this manoeuvre is 188 days, which is more than 6 months. This duration can be reduced by optimizing the direction of the thrust vector along the trajectory, but this analysis is going far beyond the scope of this work. However, there is always a main drawback for this low thrust GTO-GEO manoeuvre: the circularization of the orbit. With electric propulsion, it takes time and uses precious amounts of propellant. As detailed in Table 2.1, 137 days are spent on the change of eccentricity of the orbit, while the orbital raising takes only 51 days.

A problem that this kind of low thrust transfer rises is the impact of the radiations due to the Van Allen belt. The inner belt is situated between 100 and 10000 km of altitude, so the spacecraft is likely to pass through it several hundreds of times during the orbit raising. Radiations can damage solar cells, sensors and integrated circuits, so one should keep in mind that an appropriate shielding might be required.

2.2 CSO to GEO

In order to avoid an eccentricity change during the manoeuvre, it is required to launch the satellite in a circular subsynchronous orbit (CSO). In the case of an orbiting body around Earth, a subsynchronous orbit is an orbit which has a period lower than one sidereal day (23 hours, 56 minutes and 4.091 seconds). Then, the electric propulsion is used to increase the radius of the orbit, until reaching GEO. This scenario is possible only if the last stage of the launcher can be stopped and started again. Indeed, the Hohmann transfer laws require two impulses during the manoeuvre: the first one to escape LEO and the second one to enter in CSO. This will be possible for Ariane 5 ECB. This planned version of Ariane 5 will incorporate the new Vinci cryogenic engine fed with liquid hydrogen and liquid oxygen in its upper stage ESC-B [27].

If such a launcher is considered, one can calculate the radius of the CSO in which it can put a satellite by using the same amount of propellant than the one needed to put this satellite in GTO. Figure 2.4 illustrates these manoeuvres.

The speed on the GTO at the point A is:

\[ U_{A_{GTO}} = \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1 + a_3}} \]  

(2.7)

The speed on the LEO at the point A is:

\[ U_{A_{LEO}} = \sqrt{\frac{\mu}{a_1}} \]  

(2.8)

Consequently, the \( \Delta U \) that has to be provided by the launcher in order to go from LEO to GTO is:

\[ \Delta U_{LEO \rightarrow GTO} = U_{A_{GTO}} - U_{A_{LEO}} = \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1 + a_3}} - \sqrt{\frac{\mu}{a_1}} \]  

(2.9)

Besides, in order to bring the satellite in CSO, the launcher first has to go into a subsynchronous transfer orbit (STO) with a first impulse in A and then to enter in CSO with a
second impulse in B:

\[ \Delta U_{LEO\rightarrow CSO} = \Delta U_{LEO\rightarrow STO} + \Delta U_{STO\rightarrow CSO} \]  \hspace{1cm} (2.10)

By following the same method than between Equation 2.7 and 2.9, one obtain:

\[ \Delta U_{LEO\rightarrow STO} = U_{A_{STO}} - U_{A_{LEO}} = \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1 + a_2}} - \sqrt{\frac{\mu}{a_1}} \]  \hspace{1cm} (2.11)

\[ \Delta U_{LEO\rightarrow GTO} = U_{B_{CSO}} - U_{B_{STO}} = \sqrt{\frac{\mu}{a_2}} - \sqrt{\frac{2\mu}{a_2} - \frac{2\mu}{a_1 + a_2}} \]  \hspace{1cm} (2.12)

Then, the radius \( a_2 \) of the CSO can be obtained by solving the equation:

\[ \Delta U_{LEO\rightarrow GTO} = \Delta U_{LEO\rightarrow CSO} \]  \hspace{1cm} (2.13)

Which is equivalent to:

\[ \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1 + a_3}} = \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1 + a_2}} + \sqrt{\frac{\mu}{a_2}} - \sqrt{\frac{2\mu}{a_2} - \frac{2\mu}{a_1 + a_2}} \]  \hspace{1cm} (2.14)

It is found numerically that \( a_2 = 14565 \) km, so the altitude of the circular subsynchronous orbit is 8196 km. Another simulation is ran with TiraXYOrbitaL with the initial orbit corresponding to the CSO. The results are presented in Figure 2.5 and Table 2.2. It takes 135 days to go from a circular subsynchronous orbit to GEO with the electric propulsion. As it was taking 188 days to go from GTO to GEO, this second option is faster and uses also less propellant.
CHAPTER 2. LOW THRUST TRANSFER TO GEO

Figure 2.5: Low thrust CSO-GEO manoeuvre

<table>
<thead>
<tr>
<th></th>
<th>CSO to GEO</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial orbit</td>
<td>circular, 8196 km altitude</td>
</tr>
<tr>
<td>Initial mass (kg)</td>
<td>6000</td>
</tr>
<tr>
<td>Thrust level (N)</td>
<td>1</td>
</tr>
<tr>
<td>Thrust direction</td>
<td>$\vec{U}$</td>
</tr>
<tr>
<td>Criteria for termination</td>
<td>perigee at GEO altitude</td>
</tr>
<tr>
<td>Final orbit</td>
<td>GEO</td>
</tr>
<tr>
<td>Final mass (kg)</td>
<td>4815</td>
</tr>
<tr>
<td>Total time (days)</td>
<td>135</td>
</tr>
</tbody>
</table>

Table 2.2: Low thrust CSO-GEO manoeuvre

2.3 Supersynchronous orbit to GEO

Other methods of low thrust transfer to GEO have been investigated, and especially from a supersynchronous orbit. This is very well explained by Arnon Spitzer in [28], and illustrated in Figure 2.6. Similarly with the Section 2.1, a constant thrust vector will change the eccentricity of the orbit. If the initial orbit has proper characteristics, the constant thrust leads to a GEO. This initial orbit shall be supersynchronous, with an apogee twice higher than the GEO altitude and a perigee at LEO altitude.
2.4 Economic analysis

In this context, it is useful to investigate if the electrical propulsion (EP) can be an economically preferable alternative to the usual chemical apogee kick motor. If one considers to launch an average communication satellite of 6000 kg in GTO, it can be noticed in Table 2.3 that the cheapest solution would be to choose a Proton rocket for a cost of 60 M€.

<table>
<thead>
<tr>
<th>Rocket</th>
<th>Country</th>
<th>Mass to GTO (kg)</th>
<th>Cost (M€)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ariane 5 ECA</td>
<td>Europe</td>
<td>9600 (two payloads)</td>
<td>150</td>
</tr>
<tr>
<td>Long March 3B</td>
<td>China</td>
<td>5100</td>
<td>45</td>
</tr>
<tr>
<td>Zenit 3SL</td>
<td>Ukraine</td>
<td>5200</td>
<td>45</td>
</tr>
<tr>
<td>Proton</td>
<td>Russia</td>
<td>6140</td>
<td>60</td>
</tr>
<tr>
<td>Atlas 5</td>
<td>USA</td>
<td>6650</td>
<td>90</td>
</tr>
<tr>
<td>Falcon 9</td>
<td>USA</td>
<td>4540</td>
<td>40</td>
</tr>
</tbody>
</table>

Table 2.3: Cost to GTO for medium to heavy lift launchers [25] [29] (1 USD = 0.70 € in April 2011)

In order to transmit data, communication satellites have several channels onboard called transponders. Recent ones have around 40 transponders. Of course, the more transponders satellites have, the more money they bring in. Then, two options are available in order to take advantage of the electric propulsion:

- The first one is to design an EP satellite with the same dry mass \( m_{\text{dry}} = 3749 \text{ kg} \) calculated in Section 1.3, so with the same number of transponders. This satellite needs less propellant than with an apogee kick motor to reach GEO, so its wet mass is less than 6000 kg. Consequently, a Zenit 3SL or a Falcon 9 can be chosen and the launch is cheaper, see Table 2.3.

- The second one is to aim at the same wet mass, 6000 kg, but then its dry mass is larger than 3749 kg. Therefore, it is possible to install more transponders on the satellite so it can bring in more money yearly.
The fact is that economies of scale are made when the satellite is bigger. This is described in [30]: “The cost per transponder decreases sharply as the number of transponders per satellite is increased”. Indeed, if a satellite has a bigger transponder capacity, the overall operating cost per transponder drops. Furthermore, the initial cost per transponder that has to be put on the satellite development and insurance also decreases. The trend for the design of geostationary communication satellites is clearly to grow bigger and bigger, until the full capacity of launchers is reached. In this context, it seems that the second option (increasing the satellite capacity for the same launch mass) is likely to have a commercial value in the satellite market of tomorrow.

In Section 1.3, it has been calculated that a satellite with a launch mass of 6000 kg using a chemical apogee kick motor has a dry mass in GEO of \( m_{\text{chem}}^{\text{dry}} = 3749 \) kg. It has been seen in Section 2.2 that if the satellite has a HPEP thruster for the transfer, it can have a dry mass of \( m_{\text{HPEP}}^{\text{dry}} = 4815 \) kg in GEO. Its capacity is increased by 1066 kg. If \( n \) is the number of transponder on the satellite, one can assume that \( n \) is proportional to the dry mass \( m_{\text{dry}} \):

\[
n_{\text{HPEP}} = n_{\text{chem}} \left( 1 + \frac{m_{\text{HPEP}}^{\text{dry}} - m_{\text{chem}}^{\text{dry}}}{m_{\text{chem}}^{\text{dry}}} \right) = 40 \left( 1 + \frac{4815 - 3749}{3749} \right) = 51.37
\]  

(2.15)

Of course, the number of transponders should be an integer, so \( n_{\text{HPEP}} = 51 \). As it has 11 more transponders, it’s reasonable to assume that it is more expensive. An average GEO communication satellite has manufacturing cost of 120 M\( \)€. Due to the economies of scale, the cost increase should be less than proportional. For a first assumption, the cost \( C \) of the HPEP satellite in M\( \)€ can be calculated with:

\[
C_{\text{HPEP}} = C_{\text{chem}} \left( 1 + \frac{1}{2} \frac{n_{\text{HPEP}} - n_{\text{chem}}}{n_{\text{chem}}} \right) = 120 \left( 1 + \frac{1}{2} \frac{51 - 40}{40} \right) = 136.5
\]  

(2.16)

The financial term of “Net Present Value” (NPV) is here introduced with the intention of comparing the different technologies for the apogee raising of GEO satellites. The NPV of a series of cash flows is the sum of the present values of the individual cash flows. In the case of a communication company that wants to put one GEO satellite into orbit, the first cash flow is negative. It corresponds to the initial investment: the cost of the satellite and of the launch. Depending on the time it takes for the satellite to reach GEO, it needs a ground station support during the transfer. Once on GEO, the cash flows become positive. An average GEO satellite of 6000 kg with 40 transponders brings in 200 M\( \)€ per year. One can deduce that each transponder brings in 5 M\( \)€ per year. Eventually, discounted cash flows have to be considered in order to take into account the time value of money which is decreasing. All the cash flows are discounted down to their present value by using the discount rate \( i \). The NPV is then the sum of all the terms:

\[
\frac{R_t}{(1 + i)^t}
\]

(2.17)

where \( t \) is the time of the cash flow and \( R_t \) is the net value of the cash flow at the time \( t \). The cash inflows and outflows are calculated for each month following launch until ten years after. Ten years is considered as the average lifetime for GEO satellites. Costs used are presented in Table 2.4 and the obtained NPV are shown in Figure 2.7.
2.4. ECONOMIC ANALYSIS

Table 2.4: Data used for the calculation of the cash inflows/outflows and of the NPV

<table>
<thead>
<tr>
<th></th>
<th>Cost launch 6000 kg with Proton</th>
<th>60 M€</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Cost ground station support per year</td>
<td>5 M€</td>
</tr>
<tr>
<td></td>
<td>Gain transponder per year</td>
<td>5 M€</td>
</tr>
<tr>
<td></td>
<td>Discount rate</td>
<td>5%</td>
</tr>
<tr>
<td>Transfer with apogee kick motor</td>
<td>Cost satellite</td>
<td>120 M€</td>
</tr>
<tr>
<td></td>
<td>Dry mass</td>
<td>3749 kg</td>
</tr>
<tr>
<td></td>
<td>Number of transponders</td>
<td>40</td>
</tr>
<tr>
<td></td>
<td>Transfer time</td>
<td>5 days</td>
</tr>
<tr>
<td>Transfer with HPEP</td>
<td>Cost satellite</td>
<td>136.5 M€</td>
</tr>
<tr>
<td></td>
<td>Dry mass</td>
<td>4815 kg</td>
</tr>
<tr>
<td></td>
<td>Number of transponders</td>
<td>51</td>
</tr>
<tr>
<td></td>
<td>Transfer time</td>
<td>4.5 months</td>
</tr>
</tbody>
</table>

Figure 2.7: NPV compared for two GEO satellites with the same launch mass (6000 kg) but with different transfer technologies to GEO

Two years after launch, the satellite with HPEP transfer has brought in more money than the other one. After ten years, the difference is up to 330 M€ on HPEP’s advantage. In finance, it is useful to consider the return on investment (ROI). It corresponds to the ratio of the money gained or lost relative to the money invested. The initial investment for the satellite with kick apogee motor transfer takes in account the cost of the satellite and of the launch. After ten years, its ROI is up to 777%. For the satellite with HPEP transfer, 4.5 months of ground station support are also considered in the initial investment, but it has more transponders so it brings in more money. Its ROI is up to 870% after ten years, 93% more than for the other satellite. Consequently, it makes sense for a communication company to invest in a satellite with the HPEP technology.
Chapter 3

HPEP thruster

3.1 Working principle

Electric Propulsion (EP) thrusters of nowadays have several technological and physical limitations [31] [32]:

- Electrothermal thrusters use electricity to bring thermal energy to the gaseous propellant, so their materials should be able to stand high temperatures. In order to achieve higher specific impulses, they have to heat the gas to higher temperatures, and this cannot be withstood by nowadays materials. Besides, they have a low conversion efficiency from the thermal to the kinetic energy.

- Electrostatic thrusters accelerate only ions, so they use electrodes in order to neutralize the exhaust gases. Electrodes are subjected to erosion, and so are the walls of the spacecraft close to the thruster. What’s more, their thrust density is limited, which means that they deliver a low thrust per unit of surface at the exit of the thruster.

- Electromagnetic thrusters need high electrical power and therefore a heavy power processing unit. For example, magnetoplasmadynamic thrusters need hundreds of kilowatts in order to reach optimum performances. Their cathodes are also subjected to erosion.

With the intention of improving EP technologies, the Elwing Company has been working for more than ten years on the development of a high power electrodeless plasma thruster. In order to get rid of the main disadvantages of current EP thrusters, their concept has been designed with:

- Two distinct stages, one for the ionization and one for the acceleration, physically separated. This aims to a higher overall efficiency by being able to optimize independently the two stages. Indeed, EP thrusters usually have only one stage for both ionizing and accelerating the plasma, so this stage cannot be efficiently optimized.

- An ionization stage using the electron cyclotron resonance (ECR), an efficient electrodeless ionization method. More information on the ECR is presented in Section 3.2.
• An acceleration stage with no theoretical limitation on the thrust density: the plasma is accelerated by the ponderomotive force, see Section 3.3. This creates bulk plasma acceleration where both ions and electrons are accelerated. Therefore, there is no need for a neutralizer which prevents the erosion of the spacecraft structure.

• A power required of 10 kW. This is a high power, but however not too high to be unrealistic. Indeed, GEO satellites have solar panels able to deliver 10 to 15 kW of power for the transmission of data. This electrical power is not used when the satellite is performing the orbit raising manoeuvre from GTO to GEO.

• A wider range of operation than usual EP thrusters so the spacecraft can perform different kinds of manoeuvre with the same thruster. For example, having an EP thruster able to deliver a high thrust (1 N) at medium $I_{sp}$ (1000 s) but also a low thrust (300 mN and lower) at high $I_{sp}$ (more than 4000 s) is definitely interesting.

A schematic overview of the thruster is presented in Section 3.5.

3.2 Electron cyclotron resonance

Different methods are used in electrical propulsion in order to ionize a gas. For example, gridded ion thrusters use electron bombardment for the creation of ions. Hall effect thrusters make use of their particular structure to lead high speed electrons into the anode where a neutral gas is ionized. The electrodeless plasma thruster from the Elwing Company uses a completely different method for the ionization process: it produces ions with the electron cyclotron resonance (ECR).

When electrons are in a static and uniform magnetic field $B$, they are subject to a circular motion due to the Lorentz force. They turn with the angular frequency $\omega_{ce}$:

$$\omega_{ce} = \frac{eB}{m}$$

(3.1)

where $m$ is the mass of one electron and $e$ is the elementary charge. This $\omega_{ce}$ is called the electron cyclotron frequency. If the static and uniform magnetic field is combined with a high-frequency electromagnetic field of frequency $\omega_{ce}$, electrons will turn faster and faster. In presence of a neutral gas, high speed electrons will eventually hit an atom. If electrons have a big enough energy, the atom will lose one electron and become an ion. Thus, the neutral gas will progressively be ionized.

This ionization process is electrodeless, so it is more durable than other methods with electrodes. ECR is used as the main part of the ionization stage of the thruster. It needs a microwave generator for the high-frequency electromagnetic field and permanent or electro magnets for the static magnetic field. However, instead of having an uniform magnetic field, it has been chosen to create a so called “leaking magnetic bottle”. Indeed, charged particles like electrons tend to rebound on areas of high magnetic field intensity if their velocity component parallel to the magnetic field lines is not big enough (“magnetic mirror”). Thus, two regions of static high magnetic field around an area of high-frequency electromagnetic field can prevent low energy particles to leave this “bottle”. Nevertheless, high energy particles can leave this area and go to the next stage: the acceleration zone. The structure of the thruster is of course optimized so that only ions or electrons that have enough energy leave the ionization stage and go to the next one.
3.3 Ponderomotive force

The main concept behind the electrodeless plasma thruster is the use of the ponderomotive force. In order to get rid of the neutralizer at the outlet, a neutral plasma is required, so as many electrons as ions should exit the thruster. Therefore, they should be both accelerated towards the same direction. If a charged particle is situated in a region of high frequency electromagnetic field, the force exerted on the particle is called the ponderomotive force $\vec{F}$. This force is expressed by:

$$\vec{F} = -\frac{q^2}{4m\omega^2} \nabla E^2$$  \hspace{1cm} (3.2)

where $E$ is the magnitude of the electric field, $q$ is the electrical charge of the particle, $m$ is the particle mass and $\omega$ is the frequency of the electromagnetic wave \[32\]. This region of high frequency electromagnetic field could for example be created by microwaves generated by a magnetron. As $\vec{F}$ is proportional to the square of the charge of the particle, both electrons and ions are accelerated towards the same direction. $\vec{F}$ is also inversely proportional to the mass of the particle, so it has a bigger effect on electrons than ions. However, due to the ambipolar field created by the quick acceleration of the electrons, the speed of ions and electrons equalizes. It can be seen as if electrons were “dragging” ions. This phenomenon is called the ambipolar diffusion. If the plasma frequency $\omega_p$ is defined as the following:

$$\omega_p^2 = \frac{n_e q^2}{m \varepsilon_0}$$  \hspace{1cm} (3.3)

where $n_e$ is the plasma electron density and $\varepsilon_0$ is the vacuum electrical permittivity, the ponderomotive force can be rewritten:

$$\vec{F} = \frac{\omega_p^2}{2\omega^2} n_e \nabla \varepsilon_0 E^2$$  \hspace{1cm} (3.4)

It is interesting to notice that the term $\frac{\omega_p^2}{2\omega^2} \nabla \varepsilon_0 E^2$ corresponds to the electromagnetic energy density stored in an electric field of magnitude $E$. Thus, it is clear that the ponderomotive force requires a strong electromagnetic energy density gradient. For a plasma, it is convenient to consider a “density of force” $\vec{F}_d$ and not only the force applied on one particle:

$$\vec{F}_d = n_e \vec{F} = \frac{\omega_p^2}{2\omega^2 n_e} \nabla \varepsilon_0 E^2$$  \hspace{1cm} (3.5)

The minus sign indicates that the plasma is pushed in the direction of the lowest electromagnetic energy density. $\vec{F}_d$ can be written as deriving from a potential $\Psi$:

$$\vec{F}_d = -\nabla \Psi$$  \hspace{1cm} (3.6)

$$\Psi = \frac{\omega_p^2 \varepsilon_0 E^2}{2\omega^2}$$  \hspace{1cm} (3.7)

A high EM energy density area could be used to design an elementary plasma thruster. The plasma would be accelerated from a high EM energy density area to a lower one. However, such a device would use only one gradient of EM energy density, while as shown in Figure 3.1, high EM energy density areas have two gradients if the device is linear. The problem encountered here is that the two accelerating areas accelerate in opposite
CHAPTER 3. HPEP THRUSTER

Figure 3.1: A high electromagnetic energy density area creates two zones with opposite ponderomotive force. Inspired from [33].

The way to solve it is to add a magnetic field. The ponderomotive force changes when a static magnetic field \( B \) appears, and becomes slightly more complicated. If \( \mu \) is the magnetic moment and \( \omega_{ce} \) is the electron cyclotron frequency, the potential \( \Psi \) becomes [33]:

\[
\Psi = \frac{\omega_p^2}{2\omega(\omega - \omega_{ce})} \frac{\varepsilon_0 E^2}{2} + \mu B
\]  

(3.8)

\[
\omega_{ce} = \frac{|qB|}{m}
\]  

(3.9)

Therefore, the ponderomotive force becomes:

\[
\vec{F}_d = -\nabla \Psi = -\nabla \left( \frac{\omega_p^2}{2\omega(\omega - \omega_{ce})} \frac{\varepsilon_0 E^2}{2} + \mu B \right)
\]  

(3.10)

It is considered that only the static magnetic field \( B \) and the magnitude \( E \) of the oscillating electromagnetic field are space dependent. However, \( \omega_{ce} \) depends of \( B \) so it is also space dependant. All the other parameters in the right side of Equation (3.10) are constant. \( \vec{F}_d \) becomes:

\[
\vec{F}_d = \frac{-\omega_p^2}{2\omega(\omega - \omega_{ce})} \nabla \varepsilon_0 E^2 \ expulsion - \frac{-\omega_p^2}{2\omega(\omega - \omega_{ce})} \nabla \frac{\varepsilon_0 E^2}{\omega - \omega_{ce}} - \nabla \left( \frac{1}{\omega - \omega_{ce}} \right) - \mu \nabla B
\]  

(3.11)

If one assumes that the EM energy density and \( B \) are only varying along the \( x \) axis, Equation (3.12) can be rewritten as a scalar expression:

\[
\vec{F}_d = \frac{-\omega_p^2}{2\omega(\omega - \omega_{ce})} \frac{\partial \varepsilon_0 E^2}{\partial x} - \frac{\omega_p^2}{2\omega(\omega - \omega_{ce})} \frac{\varepsilon_0 E^2}{2} \frac{\partial \omega_{ce}}{\partial x} - \mu \frac{\partial B}{\partial x}
\]  

(3.13)

A useful parameter can be defined here: the resonant magnetic field \( B_{res} = \left| \frac{\omega m}{q} \right| \). Indeed:

\[
B = B_{res} = \frac{\omega m}{q} \Rightarrow \omega_{ce} = \omega
\]  

(3.14)
3.3. PONDEROMOTIVE FORCE

Before reading Table 3.1, it is important to understand that:

- $\nabla \varepsilon_0 \varepsilon_0 E^2$ (or $\partial \varepsilon_0 E^2 / \partial x$) is the gradient of electromagnetic energy density

- Since $\omega_{ce} = \frac{|qB|}{mc}$, the sign of $\frac{\partial \omega_{ce}}{\partial x}$ is the same than the one of $\frac{\partial B}{\partial x}$

- $\omega - \omega_{ce} > 0$ if $B < B_{res}$ and $\omega - \omega_{ce} < 0$ if $B > B_{res}$

<table>
<thead>
<tr>
<th>case</th>
<th>$\partial \varepsilon_0 E^2 / \partial x$</th>
<th>$B - B_{res}$</th>
<th>$\partial B / \partial x$</th>
<th>$F_1$</th>
<th>$F_2$</th>
<th>$F_3$</th>
<th>$F_d$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>?</td>
</tr>
<tr>
<td>2</td>
<td>+</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
<tr>
<td>3</td>
<td>+</td>
<td>-</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>4</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>?</td>
</tr>
<tr>
<td>5</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>6</td>
<td>-</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>?</td>
</tr>
<tr>
<td>7</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>?</td>
</tr>
<tr>
<td>8</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
</tbody>
</table>

Table 3.1: Sign of the scalar ponderomotive force $F_d$ along the $x$ axis

Table 3.1 shows that only two cases could lead to a positive value for the ponderomotive force: cases 2 and 8. In the first one, the gradient of EM energy density is positive, $B > B_{res}$ and the gradient of magnetic field intensity is negative. In the second one, the gradient of EM energy density is negative, $B < B_{res}$ and the gradient of magnetic field intensity is again negative. Therefore, it is clear that a high EM energy intensity area can create two acceleration zones accelerating towards the same direction if:

- It is combined with a static magnetic field

- This magnetic field has a decreasing intensity along the $x$ axis

- Around the middle of the high EM energy zone, the intensity of the magnetic field is equal to $B_{res}$

This particular configuration of static and oscillating field is presented in Figure 3.2, and is used in the thruster of the Elwing Company in order to accelerate a plasma. Compared to Figure 3.1, the direction of the ponderomotive force on the left has been reversed by the static magnetic field, so the two accelerating areas are now acting towards the same direction.
3.4 Waveguides

In order to ionize the gas and to accelerate the plasma, the HPEP thruster needs two areas of high frequency electromagnetic field. This oscillating electromagnetic field is created by two microwave generators, one for each zone. Common microwave generators operate at 2.45 GHz. Then, the microwaves enter in a cavity resonator, which can be viewed as a waveguide where the entrance and exit are blocked. Of course, the dimensions of the resonator are calculated so that its cutoff frequency is lower than the desired frequency, 2.45 GHz for example.

The cutoff frequency $\omega_{cut}$ of a rectangular waveguide of dimensions $a \times b$ depends on the mode numbers $n$ and $m$ and of the speed of light $c$, as written in Equation (3.15). $\omega_{cut}$ is the lowest frequency that can propagate in a waveguide for a given mode.

$$\omega_{cut} = c \sqrt{\left(\frac{n\pi}{a}\right)^2 + \left(\frac{m\pi}{b}\right)^2}$$  \hspace{1cm} (3.15)

If the waveguide has a square section, $a = b$ and $\omega_{cut}$ becomes:

$$\omega_{cut} = \frac{c\pi}{a} \sqrt{n^2 + m^2}$$  \hspace{1cm} (3.16)

This leads to the side of the waveguide:

$$a = \frac{c\pi}{\omega_{cut}} \sqrt{n^2 + m^2}$$  \hspace{1cm} (3.17)

The dominant mode in such a waveguide is for a transverse electric wave with $n = 1$ and $m = 0$ (or the opposite). This gives to this mode the name $\text{TE}_{10}$. Therefore, in order to lead 2.45 GHz microwaves, the side length of a waveguide with a square section should measure at least 6.12 cm. If it is the case, this waveguide will only be able to lead the $\text{TE}_{10}$ mode.

Figure 3.2: A high electromagnetic energy density area combined with an appropriate static magnetic field creates two accelerating zones acting towards the same direction. Inspired from [33].
3.4. WAVEGUIDES

As previously said, a cavity resonator can be seen as a waveguide blocked at both ends. Besides, it should have a specific length in order to generate resonance. Indeed, the length of a way and back $2d$ in the resonator of length $d$ should be a multiple of the wavelength $\lambda$:

$$2d = N\lambda, \quad N \in \{1, 2, 3...\}$$  \hspace{1cm} (3.18)

Consequently, the resonant frequencies of a cavity resonator are:

$$f = \frac{Nc}{2d}, \quad N \in \{1, 2, 3...\}$$  \hspace{1cm} (3.19)

For a given resonator, all these frequencies are the normal modes, the lowest one corresponding to $N = 1$ being the fundamental frequency.

The Elwing Company has until now considered two different designs for the applicators, parts that connect the resonators and the thruster chamber in order to create these two areas of high frequency electromagnetic field. The first design is presented in Figure 3.3. The yellow structure is the resonator and the blue cylinder is the thruster chamber, which goes through the resonator. The thruster chamber has to be in a material transparent to microwaves, quartz for example.

![Figure 3.3: Applicator design n°1: resonator in yellow, thruster chamber in blue and field pattern with the red arrows [34]](image1)

![Figure 3.4: Applicator design n°2: resonator and applicator in light blue, thruster chamber in dark blue and field pattern with the green arrows [34]](image2)
In the second design presented in Figure 3.4, the light blue rectangular parallelepiped is the resonator and the dark blue cylinder is the thruster chamber. The light blue cylinder around the thrust chamber is the applicator, it is connected to the resonator by an opening at the bottom.

3.5 Schematic overview

![Diagram](image)

Figure 3.5: Schematic overview of the HPEP thruster working with the LMP-103S injector

An overview of the working principle of the HPEP thruster is presented in Figure 3.5. The injector, whose design is presented in Chapter 5, vaporizes the liquid propellant. The gas enters in the first high EM energy density area: the ionization stage. Thanks to the static magnetic field, the electron cyclotron resonance takes place and ionizes the gas: it becomes a plasma. Electrons are trapped into the region of high EM energy density due to the
two areas of high magnetic field intensity around. They are rebounding on these areas and eventually hit and ionize an atom or a molecule of the gas.

Charged particles then enter in the second high EM energy density area: the acceleration stage. In this region, the intensity of the static magnetic field decreases. Two zones where the ponderomotive force applies on the plasma are then created, exactly where the two gradients of high EM energy density are, as explained in Figure 3.2. The plasma is accelerated and leaves the thruster at high speed, creating the thrust. Both ions and electrons are leaving the thruster, that’s why it is considered to be a “plasma thruster” and not an “ion thruster”.

3.6 Performances

Prototypes based on the HPEP concept have been successfully tested by Elwing at Princeton University [35] and independently examined by the French aerospace laboratory Onera [36]. Until now, they have been tried with several gases: Argon, Xenon, Hydrogen, Helium, Nitrogen, Oxygen and Air. As often with electric propulsion, best results are with Xenon (high molar mass, low ionization energy), but for costs reasons, the prototypes were running with Argon for most of the measurements.

The independent report written by Onera in 2008 about tests of a HPEP prototype has not been made public, but some of its important results can be published here. Onera carried out tests with Argon and they obtained the results presented in Table 3.2. The ionization power and the acceleration power correspond to the effective power values delivered to the plasma and not to the input power in the electrical circuit. More recent measurements carried out by Elwing at Princeton University with a better thrust balance show that Onera was a bit too optimistic in its report. Indeed, it seems that the thrust to power ratio for Argon is more around $T/P = 60 \text{ mN} \cdot \text{kW}^{-1}$ for the same $I_{sp}$ value.

<table>
<thead>
<tr>
<th>Ionization power</th>
<th>71 W</th>
</tr>
</thead>
<tbody>
<tr>
<td>Acceleration power</td>
<td>153 W</td>
</tr>
<tr>
<td>Trust</td>
<td>20 mN</td>
</tr>
<tr>
<td>Specific impulse</td>
<td>2000 s</td>
</tr>
<tr>
<td>Thrust/Power</td>
<td>89 mN · kW$^{-1}$</td>
</tr>
<tr>
<td>Ionization efficiency</td>
<td>0.6</td>
</tr>
</tbody>
</table>

Table 3.2: Test of a prototype based on the HPEP concept with Argon performed by Onera in 2008

During these tests, the power range was limited by the microwave generators used. This range shall be extended in the future towards higher power in order to reach the 10 kW magnitude. Until now, no test has been carried out at 10 kW, so the performances of the HPEP thruster presented in Table 3.3 are an extrapolation of the data gathered at lower power. Theses performances are compared with other electric propulsion engines that are fully operational and are onboard orbiting satellites: resistojet, arcject, gridded ion thruster (GIT) and Hall effect thruster (HET). Compared to the others, the HPEP thruster
is working in a wide range of $I_{sp}$ and is reaching high thrust (1 N) with good $I_{sp}$ (1000 s). Major applications could then become possible not only for satellite station-keeping but also for interplanetary probes.

<table>
<thead>
<tr>
<th>Engine type</th>
<th>Propellant</th>
<th>$I_{sp}$ (s)</th>
<th>$T$ (N)</th>
<th>$P$ (W)</th>
<th>$T/P$ (mN·kW$^{-1}$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Resistojet</td>
<td>Hydrazine</td>
<td>280-310</td>
<td>0.2-0.8</td>
<td>500-1500</td>
<td>400-500</td>
</tr>
<tr>
<td>Arcjet</td>
<td>Hydrazine</td>
<td>450-600</td>
<td>0.2-0.25</td>
<td>300-2000</td>
<td>125-650</td>
</tr>
<tr>
<td>GIT NSTAR</td>
<td>Xenon</td>
<td>3100</td>
<td>0.093</td>
<td>2300-2500</td>
<td>37-40</td>
</tr>
<tr>
<td>HET SPT-100</td>
<td>Xenon</td>
<td>1600</td>
<td>0.083</td>
<td>1350</td>
<td>61</td>
</tr>
<tr>
<td>HPEP (expected)</td>
<td>LMP-103S</td>
<td>1000-5000</td>
<td>0.3-1</td>
<td>10000</td>
<td>30-100</td>
</tr>
</tbody>
</table>

Table 3.3: Comparison between current operational EP technologies and the HPEP thruster [5]
Chapter 4

Thrust to power ratio

4.1 Derivation

A key concept in the HPEP thruster is the thrust to power ratio $T/P$. It answers the question “How much thrust does the thruster deliver for a given power input?” For each atom or molecule in the thruster, the gain of energy is first made in the ionization stage and then in the acceleration stage. The ionization energy $E_{ion}$ is the energy required for one atom or molecule in order to lose one electron. However, like with every ionization method, the ionization efficiency $\alpha_{ion}$ is not equal to 1. The energy used to generate microwaves in the ECR is not entirely used to ionize the gas. The thruster needs the energy $E_{ion}/\alpha_{ion}$ to effectively ionize each atom or molecule, with $0 \leq \alpha_{ion} \leq 1$.

In the acceleration stage, a gain of energy is made in the form of kinetic energy. At the exit of the thruster, the speed of the plasma is $V_e$. Thus, each atom or molecule of mass $m$ has gained $\frac{1}{2}mV_e^2$ in kinetic energy. At this point, one can make the assumption that all energy in the acceleration stage goes into the directed acceleration of plasma. This means that all the losses are neglected: heating of electrons, radiation losses, losses to the walls, efficiency of conversion of electric power to microwaves, ohmic losses in the waveguides and the resonator, etc. Therefore, the total energy $E$ spent in the thruster per atom or molecule is equal to:

$$E = \frac{E_{ion}}{\alpha_{ion}} + \frac{1}{2}mV_e^2$$  \hspace{1cm} (4.1)

The effective exhaust velocity $V_e$ of a thruster is linked to the specific impulse $I_{sp}$ and the acceleration at the Earth’s surface $g_0$ by:

$$V_e = g_0 I_{sp}$$  \hspace{1cm} (4.2)

Consequently, the expression of the energy $E$ becomes:

$$E = \frac{E_{ion}}{\alpha_{ion}} + \frac{1}{2}mg_0^2 I_{sp}^2$$  \hspace{1cm} (4.3)

The power $P$ used by the thruster can then be deduced by using the mass flow rate $\dot{m}$ of the gas:

$$P = E \frac{\dot{m}}{m} = \frac{E_{ion}}{\alpha_{ion}} \frac{\dot{m}}{m} + \frac{1}{2} \dot{m}g_0^2 I_{sp}^2$$  \hspace{1cm} (4.4)
CHAPTER 4. THRUST TO POWER RATIO

The thrust $T$ is related to $V_e$ and $\dot{m}$ with:

$$ T = V_e \dot{m} = g_0 I_{sp} \dot{m} $$  \hspace{1cm} (4.5)

Therefore, the thrust to power ratio can be expressed by:

$$ \frac{T}{P} = \frac{g_0 I_{sp}}{\alpha_{ion} m} + \frac{g_0^2 I_{sp}^2}{2 \dot{m}} $$  \hspace{1cm} (4.6)

Equation (4.6) clearly shows that the thrust to power ratio depends on:

- The gas used as a propellant. For each atom or molecule, it corresponds a mass $m$ and an ionization energy $E_{ion}$.
- The ionization efficiency $\alpha_{ion}$.
- The specific impulse. If the gas is chosen and the ionization efficiency determined, the thrust to power ratio can be seen as a function of the $I_{sp}$, as presented in Sections 4.2 and 4.3.

4.2 Dependence on the ionization efficiency

For a given gas and a given ionization efficiency, the thrust to power ratio can be graphically represented as a function of the specific impulse $I_{sp}$ thanks to Equation (4.6). Figure 4.1 illustrates this with Xenon for different ionization efficiencies going from 5 to 100%. The molar mass and ionization energy used in Equation (4.6) for Xenon are presented in Table 4.2.

Figure 4.1: Thrust to power ratio compared for different ionization efficiencies for Xenon
Xenon is chosen as an example because it is often used in electrical propulsion. Indeed, it has a high molar mass and a low ionization energy, which makes it a suitable gas. From Figure 4.1, one can see that improving the ionization efficiency is the key to a higher thrust to power ratio. It shows that with an ionization efficiency $\alpha_{\text{ion}}$ of 20%, the HPEP thruster can deliver a thrust of 1.06 N by using 10 kW of electrical power. If $\alpha_{\text{ion}}$ goes up to 50%, the thrust is then 1.67 N, and if $\alpha_{\text{ion}} = 100\%$, the thrust is 2.36 N.

### 4.3 Dependence on the gas

A HPEP thruster using the LMP-103S as propellant is now considered. The LMP-103S is a liquid at atmospheric pressure and temperature, so it needs first to be vaporized before entering in the thruster. This is the purpose of the injector whose design is presented in Chapter 5. However, before designing this injector, it is required to know how far the decomposition of the propellant should go. For example, it can be interesting to know if a total combustion of the LMP-103S is desirable or if this is something that should be avoided.

<table>
<thead>
<tr>
<th>Chemical species</th>
<th>Formula</th>
<th>% by volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>Water</td>
<td>H$_2$O</td>
<td>50</td>
</tr>
<tr>
<td>Dinitrogen</td>
<td>N$_2$</td>
<td>25</td>
</tr>
<tr>
<td>Dihydrogen</td>
<td>H$_2$</td>
<td>15</td>
</tr>
<tr>
<td>Carbon monoxide</td>
<td>CO</td>
<td>5</td>
</tr>
<tr>
<td>Carbon dioxide</td>
<td>CO$_2$</td>
<td>5</td>
</tr>
</tbody>
</table>

Table 4.1: Composition of the exhaust gases of a HPGP thruster using LMP-103S as propellant

Table 4.1 shows the composition of the exhaust gases of a HPGP thruster, after total combustion of the LMP-103S. Figure 4.2 represents the thrust to power ratio of the gases in this composition for an ionization efficiency of 60%. The molar mass and ionization energy used to draw these curves are presented in Table 4.2.

<table>
<thead>
<tr>
<th>Chemical species</th>
<th>Formula</th>
<th>Molar mass (g · mol$^{-1}$)</th>
<th>Ionization energy (eV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Xenon</td>
<td>Xe</td>
<td>131.293</td>
<td>12.130</td>
</tr>
<tr>
<td>Water</td>
<td>H$_2$O</td>
<td>18.015</td>
<td>12.621</td>
</tr>
<tr>
<td>Dinitrogen</td>
<td>N$_2$</td>
<td>2.016</td>
<td>15.426</td>
</tr>
<tr>
<td>Dihydrogen</td>
<td>H$_2$</td>
<td>2.016</td>
<td>15.426</td>
</tr>
<tr>
<td>Carbon monoxide</td>
<td>CO</td>
<td>28.010</td>
<td>14.014</td>
</tr>
<tr>
<td>Carbon dioxide</td>
<td>CO$_2$</td>
<td>44.010</td>
<td>13.773</td>
</tr>
<tr>
<td>Ammonia</td>
<td>NH$_3$</td>
<td>17.031</td>
<td>10.070</td>
</tr>
<tr>
<td>Dinitrogen</td>
<td>N$_2$</td>
<td>28.013</td>
<td>15.581</td>
</tr>
<tr>
<td>Nitric oxide</td>
<td>NO</td>
<td>30.006</td>
<td>9.264</td>
</tr>
<tr>
<td>Nitrous oxide</td>
<td>N$_2$O</td>
<td>44.013</td>
<td>12.886</td>
</tr>
<tr>
<td>Nitrogen dioxide</td>
<td>NO$_2$</td>
<td>46.006</td>
<td>9.586</td>
</tr>
</tbody>
</table>

Table 4.2: Molar mass and ionization energy of different gases [37]

From Figure 4.2, it is clear that the exhaust gases of the HPGP thruster are not suitable for an injection in the HPEP thruster. They would give a low thrust to power ratio, mainly due
to their high concentration in \( \text{H}_2 \) and \( \text{H}_2\text{O} \). Indeed, \( \text{H}_2 \) is a bad gas for electric propulsion because it has a low molar mass and high ionization energy.

Figure 4.2: Thrust to power ratio compared for the exhaust gases of a HPGP thruster, calculated with 60% ionization efficiency. The curve for Xenon is given for a comparison purpose.

Figure 4.2 shows also that some gases present in the decomposition of the LMP-103S like \( \text{CO}_2 \) give a high thrust to power ratio. An extensive study of the literature about thermal decomposition of ADN is here required in order to understand its complexity [39] [40].

Figure 4.3: Thrust to power ratio compared for gases likely to appear during the decomposition of ADN, calculated with 60% ionization efficiency. The curves for Xenon and the HPGP plume are given for a comparison purpose.
4.4 THERMAL EXPANSION

It reveals that intermediate species like NH$_3$, CO, CO$_2$, N$_2$, NO, N$_2$O and NO$_2$ are likely to appear during the decomposition of ADN. Figure 4.3 represents the thrust to power ratio of these chemical species and compares them with Xenon and the exhaust gases of a HPGP thruster. They all give a higher thrust to power ratio than the HPGP plume. CO$_2$, NO and N$_2$O give very good results: if these three gases are injected in the HPEP thruster, the thrust delivered would be around 1 N by using 10 kW of electrical power, with $\alpha_{ion} = 60\%$. These are matching the expected performances of the HPEP thruster.

Consequently, the injector should on priority decompose the LMP-103S towards species like CO$_2$, NO and N$_2$O and avoid the formation of H$_2$ and H$_2$O. If the decomposition is not total inside the injector, chemical reactions are expected to occur at the exhaust of the injector. The gas composition just at the exit is then not likely to be the same than the one a few centimeters further. The challenge in the integration of the injector inside the HPEP thruster is to know where to place it with respect to the first resonant cavity. Indeed, the chemical species with the best thrust to power ratio shall be present where the ECR takes place.

However, molecular ionization is not the only process likely to happen during the ECR. Indeed, electron impacts on molecules can also generate impact dissociation and dissociative ionization. Therefore, only real tests of the injector with a HPEP prototype can determine where in the thruster the injector shall be in order to obtain the highest thrust.

4.4 Thermal expansion

One parameter is not considered in the thrust to power ratio analysis of Section 4.1: the temperature of the gas used as a propellant. If the gas is not chemically energetic, the temperature effect is not significant. This is the case with for example inert gases like Xenon or Argon. However, the LMP-103S is a very chemically energetic liquid, and that’s why it is used as a chemical propellant in the HPGP thruster. Therefore, its reaction of decomposition is exothermic and is likely to release a lot of heat. As these warm gases expand due to the increase of temperature, they gain kinetic energy. The overall thrust of the HPEP thruster is then increased by this gain of kinetic energy, and the question is to know if this growth is significant or not.

As explained in [41], the average kinetic energy $\langle E \rangle$ of molecules in a monatomic ideal gas is linked to the temperature $T$ by the equation presented below, where $m$ is the mass of one molecule, $\langle v^2 \rangle$ is the average square speed of molecules and $k_B = 1.381 \cdot 10^{-23}$ J $\cdot$ K$^{-1}$ is the Boltzmann constant.

$$\langle E \rangle = \frac{1}{2}m\langle v^2 \rangle = \frac{3}{2}k_BT$$ (4.7)

If $\langle E_x \rangle$, $\langle E_y \rangle$ and $\langle E_z \rangle$ are the average kinetic energies along the $x$, $y$ and $z$ axis, the assumption of molecular chaos in the gas gives:

$$\langle E_x \rangle = \langle E_y \rangle = \langle E_z \rangle = \frac{1}{3}\langle E \rangle$$ (4.8)

It is then possible to express the average kinetic energy along the $x$ axis in function of $T$:

$$\langle E_x \rangle = \frac{1}{2}m\langle v_x^2 \rangle = \frac{1}{2}k_BT$$ (4.9)
In the thruster, gas molecules can rebound on walls, but they cannot rebound on the open exit. If \( x \) is the axis of the thruster, the gas moves towards the exit at the average speed \( \langle v_x \rangle \):

\[
\langle v_x \rangle = \sqrt{\frac{k_B T}{m}}
\]  

(4.10)

Of course, since molecules rebound on walls, the average speed of the gas along the \( y \) and \( z \) axis is:

\[
\langle v_y \rangle = \langle v_z \rangle = 0
\]  

(4.11)

In order to check if the speed increase \( \langle v_x \rangle \) is significant, a numerical estimation is required. If the gas considered is Xenon and its temperature 1000°C, its expansion in the HPEP thruster leads to a gain of speed of:

\[
\langle v_x \rangle = 284 \text{ m} \cdot \text{s}^{-1}
\]  

(4.12)

Consequently, the thrust to power ratio expressed in Equation (4.6) then becomes the one in equation below.

\[
\frac{T}{P} = \frac{E_{\text{ion}}}{\alpha_{\text{ion}} m} + \frac{1}{2} \left( g_0 I_{\text{sp}} - \langle v_x \rangle \right)^2
\]  

(4.13)

This is plotted in Figure 4.4 and compared with the previous results without thermal expansion.

![Figure 4.4: Thrust to power ratio compared for Xenon with and without thermal expansion, calculated with 60% ionization efficiency](image)

At the maximum of the curves, for \( I_{\text{sp}} = 560 \text{ s} \), the red curve is 5.4% higher than the blue one. The increase in the thrust to power ratio due to the thermal expansion is small but not negligible. Therefore, the heat released during the decomposition of the LMP-103S will have a positive effect on the performances of the thruster.
Chapter 5

Injector design

5.1 Specifications

In order to use the liquid LMP-103S in the HPEP thruster, it first needs to be vaporized. For this purpose, an injector has to be designed. It shall perform the followings:

- To inject into the thruster chamber a gaseous flow
- To vaporize and partially break down the propellant in case its standard state is liquid
- To work at least with water, LMP-103S and its simulant (mixture of Ammonium Nitrate and water)

As for its performance requirements, it shall deliver a mass flow rate between 20 and 200 mg $\cdot$ s$^{-1}$. It should be able to work within a temperature range going from 10 to 1700°C.

The injector shall have an external diameter lower than 30 mm so it can enter in the plasma chamber. The curve radius of each point of its surface needs to be higher than 1 mm for metallic parts in order to avoid point discharge due to the microwaves. The material of the injector can be either:

- Dielectric: no risk of point discharge due to the microwaves, but the microwave energy will go inside the injector. It can possibly cause damages to the electric circuit of the heating element or initiate decomposition at an undesired location.
- Metallic: protection of the electronics against the microwaves but can cause some point discharge and can strongly modify the electromagnetic field topology in the thruster chamber. Even if the chosen material is a metal, the microwave energy will be present inside the injector, from the exhaust until a few times the exhaust diameter.

Last but not least, as the heating element is likely to need replacement, a way to change it quickly has to be considered.

5.2 Thermal losses

Before calculating the power required by the injector in order to perform the tests, a rough estimation of the thermal losses has to be made.
5.2.1 Radiation

The injector is not designed yet but its shape has to be guessed in order to estimate radiative losses. Let’s consider that it is a stainless steel cylinder of length \( l = 100 \) mm and of radius \( r = 12 \) mm. Its operating surface temperature is estimated to be on average \( T_{\text{inj}} = 150^\circ \text{C} \). The power radiated by this cylinder is:

\[
P_{\text{rad}} = \epsilon \sigma S_{\text{inj}} T_{\text{inj}}^4
\]

where \( S_{\text{inj}} \) is the area of the injector, \( \sigma = 5.67 \times 10^{-8} \) W · m⁻² · K⁻⁴ is the Stefan-Boltzmann constant and \( \epsilon \) is the material’s emissivity. For stainless steel, \( \epsilon = 0.32 \). The surface of the injector is equal to:

\[
S_{\text{inj}} = 2\pi r^2 + 2\pi rl = 8.44 \times 10^{-3} \text{ m}^2
\]

Therefore, the power radiated by the injector is equal to \( P_{\text{rad}} = 4.9 \) W.

5.2.2 Conduction

During the tests, the injector will be installed inside a vacuum chamber. Thermal insulation will be placed between the injector and its support, so conductive losses can be neglected there. It will be connected to the propellant reservoir by a stainless steal pipe. Inside the vacuum chamber, this pipe is estimated to be \( L = 50 \) cm long and 1/4” wide (external diameter). Standard pipes of 1/4” have an external radius of \( R = 3.2 \) mm and a wall thickness of \( t = 0.89 \) mm. Therefore, the area of a pipe section is:

\[
S_{\text{pipe}} = \pi R^2 - \pi (R - t)^2 = \pi (2Rt - t^2) = 1.54 \times 10^{-5} \text{ m}^2
\]

According to Fourier’s law, the heat flux density going through the stainless steal pipe of thermal conductivity \( k = 26 \) W · m⁻¹ · K⁻¹ is:

\[
\vec{q} = -k \vec{\nabla}T
\]

\( \vec{\nabla}T \) represents the temperature gradient in the pipe. This heat transfer problem can be analyzed in only one dimension. If one consider that the temperature of the vacuum chamber is \( T_{\text{chamber}} = 20^\circ \text{C} \), this heat flux density becomes:

\[
q = k \frac{T_{\text{inj}} - T_{\text{chamber}}}{L} = 0.1 \text{ W}
\]

Consequently, the conductive power through this pipe is:

\[
P_{\text{cond}} = k S_{\text{pipe}} \frac{T_{\text{inj}} - T_{\text{chamber}}}{L} = 0.1 \text{ W}
\]

The contribution of conduction in the thermal losses is approximately 50 times less than the one of radiation.

5.2.3 Convection

According to Newton’s law of cooling, the convective power lost by the injector at a temperature \( T_{\text{inj}} = 150^\circ \text{C} \) is:

\[
P_{\text{conv}} = h S_{\text{inj}} (T_{\text{inj}} - T_{\text{air}})
\]
where $h$ is the heat transfer coefficient, $S_{\text{inj}}$ is the external area of the injector and $T_{\text{air}} = 20^\circ$C is the temperature of the air around the injector. For air at atmospheric pressure, the heat transfer coefficient is $h = 10$ W·m$^{-2}$·K$^{-1}$ [38]. The convective power lost at atmospheric pressure is then:

$$P_{\text{conv}}^{\text{atm}} = 11.0 \text{ W}$$ (5.8)

Inside the vacuum chamber, the pressure is expected to be 42 mbar (calculated in Section 6.2), 24 times less than the atmospheric pressure. The convective power lost at 42 mbar can then be roughly estimated to be:

$$P_{\text{conv}}^{42 \text{ mbar}} = \frac{P_{\text{conv}}^{\text{atm}}}{24} = 0.5 \text{ W}$$ (5.9)

### 5.2.4 Total

The power lost by the injector due to thermal losses is therefore equal to:

$$P_{\text{losses}} = P_{\text{rad}} + P_{\text{cond}} + P_{\text{conv}}$$ (5.10)

At atmospheric pressure:

$$P_{\text{losses}}^{\text{atm}} = P_{\text{rad}} + P_{\text{cond}} + P_{\text{conv}}^{\text{atm}} = 16.0 \text{ W}$$ (5.11)

At 42 mbar inside the vacuum chamber:

$$P_{\text{losses}}^{42 \text{ mbar}} = P_{\text{rad}} + P_{\text{cond}} + P_{\text{conv}}^{42 \text{ mbar}} = 5.5 \text{ W}$$ (5.12)

### 5.3 Power required

The injector shall decompose the LMP-103S. It is a highly explosive liquid, so for safety reasons, the first tests will be carried out with water. LMP-103S is also an expensive propellant, so tests after water will then be performed with a simulant: a solution of Ammonium Nitrate in water. It is a good simulant regarding the density, the viscosity and the vapor pressure. Consequently, the injector should also be able to vaporize water and to vaporize and decompose the simulant.

One should keep in mind that the injector is made to operate in the thruster chamber. The pressure is low there, in the millibar range. Due to the vapor pressure, the lower the pressure is, the lower the boiling point is for a liquid. For example, water boils at 30$^\circ$C at 42 mbar while it boils at 100$^\circ$C at 1.013 bar. The first step can be to calculate the power required by the injector in order to vaporize water in both atmospheric and low pressure. The value of this low pressure is determined by the pumping flow rate of the vacuum pump and by the quality of the vacuum chamber used during the tests. It is calculated in Section 6.2 and is equal to 42 mbar.
5.3.1 To vaporize water

The power $P$ required to vaporize a liquid at an initial temperature $T_{in}$ flowing at a mass flow rate $\dot{m}$ is expressed by:

$$P = \dot{m} \left( \frac{C_{p,m}}{M} (T_{vap} - T_{in}) + \frac{\Delta_{vap}H}{M} \right) + P_{losses}$$ (5.13)

where $C_{p,m}$ is the molar heat capacity of the liquid, $\Delta_{vap}H$ its molar vaporization enthalpy, $T_{vap}$ its boiling temperature and $M$ its molar mass. $P_{losses}$ corresponds to the thermal losses calculated in Section 5.2. In the case of water, the useful data are listed in Table 5.1. However, before starting the calculation of $P$, the mass flow rate $\dot{m}$ has to be determined. The expected performances of the HPEP thruster are a thrust of $T = 1$ N and a specific impulse of $I_{sp} = 1000$ s. The mass flow rate is then equal to:

$$\dot{m} = \frac{T}{g_0 I_{sp}} = 0.1019 \text{ g} \cdot \text{s}^{-1}$$ (5.14)

In order to simplify the calculations and the experiments, this mass flow rate is settled to the value of $\dot{m} = 0.1 \text{ g} \cdot \text{s}^{-1}$.

In order to simplify the calculations and the experiments, this mass flow rate is settled to the value of $\dot{m} = 0.1 \text{ g} \cdot \text{s}^{-1}$.

<table>
<thead>
<tr>
<th>Molar mass $M$ (g · mol$^{-1}$)</th>
<th>Water at 1.013 bar</th>
<th>Water at 42 mbar</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boiling point $T_{vap}$ (°C)</td>
<td>100</td>
<td>30</td>
</tr>
<tr>
<td>Molar vaporization enthalpy $\Delta_{vap}H$ (kJ · mol$^{-1}$)</td>
<td>40.657</td>
<td>43.776</td>
</tr>
<tr>
<td>Molar heat capacity $C_{p,m}$ (kJ · mol$^{-1}$ · K$^{-1}$)</td>
<td>75.38 (liquid)</td>
<td>75.28 (liquid)</td>
</tr>
<tr>
<td>Power needed to reach vaporization $P$ (W)</td>
<td>275</td>
<td>253</td>
</tr>
</tbody>
</table>

Table 5.1: Power required to vaporize a flow of 0.1 g · s$^{-1}$ of water at different pressures, with $T_{in} = 20$°C [37]

It is interesting to notice in Table 5.1 that even though the boiling temperature of water at 42 mbar is low compared with the one of water at atmospheric pressure, the power $P$ required to reach vaporization at 42 mbar is only 8% below the one needed at atmospheric pressure. This is due to the vaporization enthalpy $\Delta_{vap}H$. In Equation (5.13), this term contributes to 82% of $P$ at atmospheric pressure and 96% of $P$ at 42 mbar.

5.3.2 To vaporize and decompose a water solution of Ammonium Nitrate

The same estimation can now be made for the water solution of Ammonium Nitrate $\text{NH}_4\text{NO}_3$. As the power needed to vaporize a liquid is higher at atmospheric pressure than at low pressure, the calculation is made only at atmospheric one. This makes the calculation simpler and gives a safety margin. The boiling point of a water solution is expected to be slightly higher than the one of pure water. When the solution reaches this temperature, the water vaporizes, leaving crystals of $\text{NH}_4\text{NO}_3$. If one keeps on heating, the Ammonium Nitrate decomposes at 210°C. The decomposition can follow the two reactions:
5.3. POWER REQUIRED

\[
\begin{align*}
\text{NH}_4\text{NO}_3 \,(s) & \rightarrow \text{N}_2\text{O}_4 \,(g) + 2\text{H}_2\text{O}_2 \,(g) \quad (5.15) \\
\text{NH}_4\text{NO}_3 \,(s) & \rightarrow \text{NH}_3 \,(g) + 2\text{HNO}_3 \,(g) \quad (5.16)
\end{align*}
\]

According to [42], 98% of the Ammonium Nitrate follows Reaction (5.15) and 2% follows Reaction (5.16). Gaseous Nitric Acid HNO\(_3\) is likely to appear during the decomposition; this is an important point to remember while choosing the equipment since it can damage the vacuum pump. As Reaction (5.15) is predominant, Reaction (5.16) is neglected is the followings. The standard enthalpy of formation \(\Delta_f H^0\) of compounds in Reaction (5.15) are given in Table 5.2.

\[
\Delta_f H^0 \,(\text{kJ} \cdot \text{mol}^{-1}) & \quad \text{NH}_4\text{NO}_3 \,(s) \quad \text{N}_2\text{O}_4 \,(g) \quad \text{H}_2\text{O}_2 \,(g) \\
-365.6 & \quad 81.6 & \quad -241.8
\]

Table 5.2: Standard enthalpy of formation of compounds in Reaction (5.15) [37]

The decomposition reaction of the Ammonium Nitrate has then the following enthalpy:

\[
\Delta H = \left[ \Delta_f H^0(\text{N}_2\text{O}_4 \,(g)) + 2\Delta_f H^0(\text{H}_2\text{O}_2 \,(g)) \right] - \left[ \Delta_f H^0(\text{NH}_4\text{NO}_3 \,(s)) \right] = -36.4 \, \text{kJ} \cdot \text{mol}^{-1} \quad (5.17)
\]

The enthalpy of reaction is negative therefore the reaction is exothermic. When \(\text{NH}_4\text{NO}_3 \,(s)\) reaches 210°C, it decomposes with no need of additional energy.

<table>
<thead>
<tr>
<th>Mass fraction in the simulant (f) (%)</th>
<th>Water</th>
<th>NH(_4)NO(_3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Molar mass (M) ,(g \cdot \text{mol}^{-1})</td>
<td>70</td>
<td>30</td>
</tr>
<tr>
<td>Boiling or decomposition point (T_{\text{vap}}) or (T_{\text{dec}}) ,(°C)</td>
<td>18.015</td>
<td>80.052</td>
</tr>
<tr>
<td>Molar enthalpy of vaporization or decomposition ,(kJ \cdot \text{mol}^{-1})</td>
<td>100</td>
<td>210</td>
</tr>
<tr>
<td>Molar heat capacity (C_{p,m}) ,(kJ \cdot \text{mol}^{-1} \cdot \text{K}^{-1})</td>
<td>40.657</td>
<td>&lt; 0</td>
</tr>
<tr>
<td></td>
<td>75.38 (liquid)</td>
<td>139.3 (crystal)</td>
</tr>
<tr>
<td></td>
<td>33.6 (gas)</td>
<td></td>
</tr>
</tbody>
</table>

Table 5.3: Physical and thermodynamic properties of water and Ammonium Nitrate at 1.013 bar [37]

The useful data used for the calculation of the power needed to vaporize and decompose the simulant are presented in Table 5.3. With \(\dot{m} = 0.1 \, \text{g} \cdot \text{s}^{-1}\) and \(T_{\text{in}} = 20\,\text{°C}\), this power \(P\) corresponds to the sum of:

- The power required to heat the liquid water from \(T_{\text{in}}\) to \(T_{\text{vap}}\) = 120°C (higher boiling point for water when NH\(_4\)NO\(_3\) is dissolved in it), with \(f_{\text{H}_2\text{O}}\) the mass fraction of water in the simulant:

\[
P_1 = f_{\text{H}_2\text{O}} \dot{m} \frac{\text{C}_{\text{p,m}}}{M_{\text{H}_2\text{O}}} (T_{\text{vap}} - T_{\text{in}}) = 33 \, \text{W} \quad (5.18)
\]

- The power required to vaporize the water at \(T_{\text{vap}}\) = 120°C:

\[
P_2 = f_{\text{H}_2\text{O}} \dot{m} \frac{\Delta_{\text{vap}} H_{\text{H}_2\text{O}}}{M_{\text{H}_2\text{O}}} = 175 \, \text{W} \quad (5.19)
\]
• The power required to heat the Ammonium Nitrate (AN) from $T_{in}$ to $T_{AN\text{ dec}} = 210^\circ C$, with $f_{AN}$ the mass fraction of Ammonium Nitrate in the simulant:

$$P_3 = f_{AN} \dot{m} \frac{C_{p,\text{AN}}}{M_{AN}} (T_{AN\text{ dec}} - T_{in}) = 11 \text{ W}$$

(5.20)

• The power required to heat the gaseous water from $T_{\text{H}_2\text{O vap}} = 120^\circ C$ to $T_{AN\text{ dec}} = 210^\circ C$:

$$P_4 = f_{H_2O} \dot{m} \frac{C_{p,\text{H}_2\text{O}(g)}}{M_{\text{H}_2\text{O}}} (T_{AN\text{ dec}} - T_{\text{H}_2\text{O vap}}) = 13 \text{ W}$$

(5.21)

• The thermal losses:

$$P_{\text{atm losses}} = 16 \text{ W}$$

(5.22)

Consequently, the power required to turn into gas the simulant is $P = P_1 + P_2 + P_3 + P_4 + P_{\text{atm losses}} = 225 \text{ W}$. This is lower than the 275 W required to vaporize the same mass flow rate of pure water (Table 5.1).

5.3.3 To decompose the LMP-103S

This research study becomes more complicated with the LMP-103S, as it is a mixture of ADN, methanol, ammonia and water in one liquid and one gaseous phase. However, a very rough estimation of the power required to decompose it can be made. The calculation is once again made only at atmospheric pressure. According to [17], the specific heat capacity of the LMP-103S is around 2.4 J.g^{-1}.K^{-1} between 20 and 70$^\circ$C. Its decomposition temperature is trickier to find since the decomposition rate increases with temperature. The LMP-103S starts to decompose slowly around 120-130$^\circ$C [17] [43], but how fast the decomposition goes is not specified. According to ECAPS, 350$^\circ$C are required in order to initiate the combustion of the LMP-103S in their HPGP thrusters. Without more information about the decomposition of the propellant, the value of $T_{LMP\text{ dec}} = 350^\circ$C is chosen in order to carry out the rough estimation on the power required. It is clear that this value is very likely to be too high, but it gives a safety margin.

<table>
<thead>
<tr>
<th>Mass fraction $f$ (%)</th>
<th>ADN</th>
<th>Methanol</th>
<th>Ammonia</th>
<th>Water</th>
</tr>
</thead>
<tbody>
<tr>
<td>$M$ (g·mol^{-1})</td>
<td>124.06</td>
<td>32.04</td>
<td>17.03</td>
<td>18.015</td>
</tr>
<tr>
<td>$\Delta_{vap} H$ (kJ·mol^{-1})</td>
<td>155.4</td>
<td>35.30</td>
<td>23.35</td>
<td>40.66</td>
</tr>
</tbody>
</table>

Table 5.4: Physical and thermodynamic properties of components of LMP-103S at 1.013 bar [37]

The useful data used for the calculation of the power needed to decompose the LMP-103S are presented in Table 5.4. With $\dot{m} = 0.1 \text{ g} \cdot \text{s}^{-1}$ and $T_{in} = 20^\circ$C, this power $P$ corresponds to the sum of:

• The power required to heat the liquid LMP-103S from $T_{in}$ to $T_{LMP\text{ dec}} = 350^\circ$C:

$$P_1 = \dot{m} C_{p,\text{LMP}} (T_{LMP\text{ dec}} - T_{in}) = 87 \text{ W}$$

(5.23)
• The power required to vaporize or decompose all the species present in the LMP-103S at $T_{\text{dec}}^{\text{LMP}} = 350^\circ \text{C}$:

$$P_2 = f_{\text{ADN}} \dot{m} \frac{\Delta_{\text{vap}}H_{\text{ADN}}}{M_{\text{ADN}}} = 90 \text{ W}$$  \hspace{1cm} (5.24)

$$P_3 = f_{\text{CH}_3\text{OH}} \dot{m} \frac{\Delta_{\text{vap}}H_{\text{CH}_3\text{OH}}}{M_{\text{CH}_3\text{OH}}} = 24 \text{ W}$$  \hspace{1cm} (5.25)

$$P_4 = f_{\text{NH}_3} \dot{m} \frac{\Delta_{\text{vap}}H_{\text{NH}_3}}{M_{\text{NH}_3}} = 8 \text{ W}$$  \hspace{1cm} (5.26)

$$P_5 = f_{\text{H}_2\text{O}} \dot{m} \frac{\Delta_{\text{vap}}H_{\text{H}_2\text{O}}}{M_{\text{H}_2\text{O}}} = 25 \text{ W}$$  \hspace{1cm} (5.27)

• The thermal losses:

$$P_{\text{atm losses}} = 16 \text{ W}$$  \hspace{1cm} (5.28)

Consequently, the power required to turn into gas the LMP-103S is $P = P_1 + P_2 + P_3 + P_4 + P_5 + P_{\text{atm losses}} = 227 \text{ W}$.

5.3.4 Summary

From Table 5.5, it is clear that both the LMP-103S and its simulant require less power than water to be turned into gas at the same mass flow rate, and this no matter if the pressure is atmospheric or low. Consequently, the injector shall be able to deliver at least 275 W.

<table>
<thead>
<tr>
<th>Power needed to reach vaporization $P$ (W)</th>
<th>Water at 1.013 bar</th>
<th>Water at 42 mbar</th>
<th>Solution of Ammonium Nitrate at 1.013 bar</th>
<th>LMP-103S at 1.013 bar</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>275</td>
<td>253</td>
<td>225</td>
<td>227</td>
</tr>
</tbody>
</table>

Table 5.5: Power required to vaporize a flow of 0.1 g·s$^{-1}$ and with $T_{\text{in}} = 20^\circ \text{C}$

5.4 Brainstorming

Two different kinds of heater were considered for the design of the injector: rope or tape heaters and cartridge heaters.

5.4.1 Rope and tape heaters

Rope and tape heaters can be wrapped around a tube and heat the fluid that is flowing inside, as shown in Figure 5.1. The company *Omega* supplies such heaters. According to models, they can deliver between 60 and 3000 W. If this option is to be chosen, one pipe can be wrapped by a heater of 300 W, and the liquid propellant would flow inside this pipe.
The problem encountered with these heaters is their length. Indeed, the tape heater of 300 W is 122 cm long, which leads to a long injector. An alternative solution would be to consider several tubes in parallel, with one heater wrapped around each tube, and with the total power around 300 W. This would allow to have a shorter injector.

![Figure 5.1: Rope heater on the left and tape heaters on the right, from Omega.](image)

5.4.2 Cartridge heaters

Cartridge heaters offer a high watt density and *Omega* supplies them from 75 to 400 W for an outer diameter of 6.4 mm, see Figure 5.2. A cartridge heater of 300 or 400 W can be inserted in a block of metal. In this block, one or several holes for the fluid are drilled. It would lead to a shorter injector than with a tape heater since a 400 W cartridge heater is only 7.62 cm long.

![Figure 5.2: Cartridge heater of 6.4 mm diameter from Omega.](image)

5.5 Final design

Eventually, the chosen heating element is one cartridge heater of 400 W. Its high watt density enables the injector to be short and its 400 W give a good range to explore the behaviour of the vaporization and decomposition of the propellant between 0 and 400 W. In order to have a better heat exchange between the heater and the fluid, the fluid should flow slowly through holes of small diameters.

The final design of the injector consists of three parts. First, there is the main cylinder through which several holes are drilled, see Figure 5.3. The cartridge heater fits into the hole in the middle of the cylinder. At the end of this hole, a perpendicular hole is drilled to allow the wires of the heater to exit the injector without being in contact with the propellant. Seven small holes for the fluid are drilled around the hole of the heater. As shown in
Figure 5.3: Sectional views of the main part of the injector: a cylinder with seven holes for the propellant and one hole for the cartridge heater (mm)

Figure 5.4: Main cylinder manufactured in stainless steel

Figure 5.3, the free space for the eighth hole is actually not available due to the perpendicular hole for the wires. A picture of this part manufactured in stainless steel is presented in Figure 5.4.
Then, a cap makes the connection between the main cylinder and the propellant inlet, see Figure 5.5. Due to standard pipes, the propellant will flow to the injector through 1/4” (6.4 mm) diameter tubes. Therefore, the inlet of the cap is similar to a 1/4” tube. The shape of this cap makes it easy to weld it on the main cylinder. Its dimensions are calculated in order to reduce the amount of propellant at the same place. Indeed the LMP-103S is highly explosive, so any accumulation of the propellant in the injector is potentially dangerous and could lead to an explosion.

Figure 5.5: Sectional views of the part connecting the cylinder and the propellant inlet: the cap (mm)

Figure 5.6: Sectional view of the injector: the main cylinder in orange, the cap in blue and the cartridge heater in red (mm)
The main cylinder and the cap are manufactured in the workshop of the *Swedish Space Corporation*. Due to the compatibility of the LMP-103S with stainless steel [17], they are made in this material. The cartridge heater is ordered at *Omega*. All these three parts are presented together in Figure 5.6, they constitute the injector. Due to its wires, the cartridge heater is a bit tricky to insert into the main cylinder. As presented in the three pictures in Figure 5.7, a wire is first inserted through the two perpendicular holes and are bound to the wires of the heater (left). This wire is then dragged through the hole (middle). Eventually, the cartridge heater is well placed inside the main cylinder (right).

![Figure 5.7: Insertion of the cartridge heater in the main cylinder](image)

A picture of the manufactured injector is shown in Figure 5.8. The main cylinder and the cap are welded together. On the right side of the picture, the cap is welded to a 1/4” diameter pipe, at the end of which is placed a VCR connector. VCR connectors are well known for their reliability and low leak properties; they are supplied by the company *Swagelok*. On the top left, one can notice that an additional tube has been welded to the main cylinder. It is actually connected to one of the seven holes where the propellant flows. It will be used to measure the pressure at the end of the injector. A support is screwed under the injector. Two ceramic parts thermally insulate the injector from what the support is screwed on (a vacuum chamber for example).

![Figure 5.8: Manufactured injector](image)
An interesting data to obtain here is the speed of the propellant in the seven holes of the injector. If $m = 0.1 \text{ g} \cdot \text{s}^{-1}$ is the mass flow rate, the flow rate is equal to $\dot{m}/\rho$ with $\rho$ corresponding to the density of the fluid. For LMP-103S, $\rho = 1217 \text{ kg} \cdot \text{m}^{-3}$ at 50°C. The flow rate is then equal to:

$$\frac{\dot{m}}{\rho} = 7\pi r^2 \cdot v$$

(5.29)

with $7\pi r^2$ corresponding to the total surface of the sections of the seven holes and $v$ to the speed of the propellant. Each one of the seven holes has a radius of $r = 1 \text{ mm}$. The speed of the LMP-103S in the injector is then $v = 3.7 \text{ mm} \cdot \text{s}^{-1}$.
Chapter 6

Test setup

6.1 Overview

![Diagram of the test setup for the injector](image)

Table 6.1: Input and output parameters of the test setup

<table>
<thead>
<tr>
<th>Input parameters</th>
<th>Output parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flow rate of the HPLC pump</td>
<td>Temperature on the injector</td>
</tr>
<tr>
<td>Heating power on the cartridge heater</td>
<td>Vaporization or not of the propellant</td>
</tr>
<tr>
<td>Type of propellant</td>
<td>Pressure in the chamber and at the end of the injector</td>
</tr>
</tbody>
</table>

Now that the injector is manufactured, it has to be tested. In vacuum, the pressure inside the thruster chamber is in the range $10^{-3} - 10^{-4}$ bar. Consequently, in order to create similar conditions during the tests, the injector has to be placed inside a vacuum chamber.
As explained in the diagram in Figure 6.1, the propellant flows from the reservoir to the injector thanks to the HPLC pump. More details about the HPLC pump and the orifice that follows are given in Section 6.4. Thermocouples (TC) measure the temperature on the surface of the injector, one pressure transducer (PT) measures the pressure at the end of the injector and the vacuum gauge (VG) indicates the level of vacuum inside the chamber. The input and output parameters of the test setup are presented in Table 6.1. An explained picture of the test setup is presented in Figure 6.2.

![Figure 6.2: Overview of the test setup for the injector](image-url)
6.2 Vacuum pump

The vacuum pump available in house for this test setup is an *Alcatel 2012A Dual Stage* shown in Figure 6.3. It has a pumping speed of \( \dot{V} = 15 \text{ m}^3 \cdot \text{h}^{-1} \) and is able to reach the ultimate pressure of \( 2 \cdot 10^{-7} \text{ bar} \). Of course, this ultimate pressure can be reached only if the quality of the vacuum chamber and of all the seals is very good. The injector ejects gases which are being sucked by the vacuum pump. However, several kinds of gases can damage the vacuum pump; it is the case of the water vapor. In order to get rid of the water vapor between the vacuum chamber and the vacuum pump, a cryotrap is installed, Figure 6.4. Its inner container is filled with liquid nitrogen \((-196^\circ \text{C})\) and the exhaust gases flow through its outer container. When the water vapor enters in the cryotrap, it condenses on the very cold external surface of the inner container and stays trapped in the outer one in a liquid (or solid) state. In this way, there is no water vapor going through the vacuum pump.

In Figure 6.4, one can notice white parts at the bottom and at the top of the cryotrap. The cryotrap is so cold that it even condenses the water present in the air, and it freezes on the external surface. The white parts are covered with ice.

![Figure 6.3: Vacuum pump Alcatel 2012A](image)

![Figure 6.4: Cryotrap](image)

If the injector ejects a mass flow rate \( \dot{m} \) of gas which molar mass is \( M \), a pressure equilibrium will eventually be reached in the vacuum chamber when the vacuum pump is working. This pressure \( P \) is an useful parameter to obtain in order to know if the vacuum pump is pumping fast enough. If it is assumed that the expelled gas is an ideal gas, the ideal gas law for a flow rate is:

\[
P \dot{V} = \frac{\dot{m} R T}{M} \tag{6.1}
\]

\( R = 8.314 \text{ J} \cdot \text{K}^{-1} \cdot \text{mol}^{-1} \) is the gas constant and \( T \) is the temperature of the gas in the chamber. The equilibrium pressure \( P \) is then given in the equation below with the assumption of \( T = 100^\circ \text{C} \), \( \dot{m} = 0.1 \text{ g} \cdot \text{s}^{-1} \) and \( M = 18.015 \text{ g} \cdot \text{mol}^{-1} \) (water).

\[
P = \frac{\dot{m} R T}{M \dot{V}} = 42 \text{ mbar} \tag{6.2}
\]

The pressure \( P = 42 \text{ mbar} \) obtained is relatively large compared to the pressure in the thruster chamber \((0.1 - 1 \text{ mbar})\). However, in order to have 1 mbar in the vacuum chamber with the same mass flow rate, the vacuum pump should have a pumping speed of...
632 m$^3 \cdot$ h$^{-1}$, which is more than 40 times the pumping speed of the Alcatel 2012A. As such a big pump is not available at the Swedish Space Corporation, the experiments will be carried out with the Alcatel 2012A, assuming that $P = 42$ mbar is a good enough level of vacuum for this kind of test.

6.3 Vacuum chamber installation

![Vacuum chamber in the EP lab](image)

Nitric acid is likely to appear during the decomposition of the simulant of the LMP-103S. This can damage the vacuum pump but also the vacuum chamber. For this reason, an old “no risk” vacuum chamber is used during the tests, as shown in Figure 6.5. The chamber is a cylinder, its inner diameter is 55 cm and it is 80 cm long. It has two lids in order to close it: a window has been fixed on the front lid (Figure 6.2) and the flanges for the connections are placed on the back lid (Figure 6.8).

![Six holes on the injector for six TC](image)

![TC screwed on the injector inside the vacuum chamber](image)

Six holes are drilled on the side of the injector in order to connect thermocouples (TC), Figure 6.6. Indeed, it is expected that the temperature on the surface changes in function of the
Figure 6.8: Flanges and connections on the back side of the vacuum chamber

Due to the low quality of the vacuum chamber and of the seals, the lowest pressure able to be reached in the vacuum chamber with the Alcatel 2012A vacuum pump is 0.02 bar.

6.4 HPLC pump

Let’s assume that the propellant reservoir is directly connected to the injector inlet, with only pipes in between. As the pressure in the vacuum chamber is much lower than the atmospheric one, the vacuum sucks in the liquid directly from the propellant reservoir. The flow rate is then constant, too high and impossible to regulate. The solution is to place a small orifice in between the reservoir and the injector, as shown in Figure 6.1. With the orifice, the flow rate is smaller, and how small depends on the diameter of the hole. Then, the regulation of the flow rate can be made only with addition of a pump delivering high pressurized flow.
The device used to flow the liquid from the propellant reservoir to the injector is an HPLC pump \textit{Shimadzu LC-10AD}, Figure 6.10. HPLC stands for “high-pressure liquid chromatography”. This is a pump used in liquid chromatography because of its ability to pressurize the flow up to 220 bar. In spite of its original application, its performances are matching the requirements of this project \cite{44}:

- Constant flow delivery from 0 to 10 mL $\cdot$ min$^{-1}$
- Maximum pressure between 1 and 220 bar
6.5 Orifice plate

The easiest way to add an orifice in the fluid line is to drill a hole in a thin circular plate and insert it inside a VCR connector between the HPLC pump and the injector. The important parameter that has to be calculated is the area of the hole in this plate.

![Image](image_url)

Figure 6.11: Behaviour of the flow through an orifice plate

An orifice plate as presented in Figure 6.11 introduces losses in the flow. The Bernoulli equation for viscous incompressible flows claims that in the case of an orifice plate [45]:

\[
\left(P_3 + \frac{1}{2} \rho V_3^2\right) - \left(P_1 + \frac{1}{2} \rho V_1^2\right) = - \xi \frac{1}{2} \rho V_1^2 \tag{6.3}
\]

with \(P\), \(V\) and \(\rho\) the pressure, the speed and the density of the fluid at the specified points. \(A_1\) and \(A_3\) are the area of the pipe, \(A_2\) is the area of the orifice. \(\xi\) is a parameter which depends of the characteristics of the pipe and of the orifice. The continuity equation says that the flow rate \(Q\) equals:

\[
Q = A_1 V_1 = A_3 V_3 \tag{6.4}
\]

As \(A_1 = A_3\), one obtains that \(V_1 = V_3\). Eventually, Equation (6.3) leads to:

\[
P_3 - P_1 = - \xi \frac{1}{2} \rho Q^2 A_1^2 \tag{6.5}
\]

\[
\xi = \frac{2(P_1 - P_3)A_1^2}{\rho Q^2} \tag{6.6}
\]

According to [45], the parameter \(\xi\) is function of \(A_1\) and \(A_2\):

\[
\xi = \left(1 + 0.707 \sqrt{1 - \frac{A_2}{A_1}} - \frac{A_2}{A_1}\right)^2 \left(\frac{A_1}{A_2}\right)^2 \tag{6.7}
\]

With \(\Delta P = P_1 - P_3\), Equations (6.6) and (6.7) give:

\[
\frac{2 A_1^2}{\rho Q^2} \Delta P = \left(1 + 0.707 \sqrt{1 - \frac{A_2}{A_1}} - \frac{A_2}{A_1}\right)^2 \left(\frac{A_1}{A_2}\right)^2 \tag{6.8}
\]

Here, the only unknown parameter is \(A_2\), the area of the orifice. If water is considered, \(\rho = 1000 \text{ kg} \cdot \text{m}^{-3}\). \(A_1\) correspond to the inner area inside a VCR connetor, so \(A_1 = 16.8 \text{ mm}^2\). The liquid shall be able to flow between 30 and 150 mg · s\(^{-1}\), so approximately between \(Q = 2\) and \(Q = 9 \text{ mL} \cdot \text{min}^{-1}\). The pressure difference \(\Delta P\) should be below 30 bar for safety reasons and above 2 bar to be able to push the liquid. With Equation (6.8) and \(d\) the diameter of the circular orifice:
• $Q = 9 \text{ mL} \cdot \text{min}^{-1}$ and $\Delta P < 30 \text{ bar}$ give $d \geq 0.067 \text{ mm}$

• $Q = 2 \text{ mL} \cdot \text{min}^{-1}$ and $\Delta P > 2 \text{ bar}$ give $d \leq 0.070 \text{ mm}$

Consequently, the orifice should be a circular hole of approximate diameter $d \approx 0.07 \text{ mm}$. This is a tiny hole. The smallest holes that can be drilled at the Swedish Space Corporation have a diameter of 0.2 mm. A hole of 0.2 mm diameter is made in a thin circular plate (Figure 6.12) and as expected, the flow rate is still too high because the hole is too big. The orifice size is then shrunk successively with several hammer blows, as shown in Figure 6.13. Eventually, the correct diameter is achieved, the orifice plate is tested and it works perfectly together with the HPLC pump.

![Figure 6.12: Orifice plate with a 0.2 mm diameter hole](image)

![Figure 6.13: Shrunk hole after several hammer blows](image)

6.6 Labview code

![Figure 6.14: Block diagram of the Labview VI used during the tests of the injector - Part 1](image)
Figure 6.15: Block diagram of the Labview VI used during the tests of the injector - Part 2
Figure 6.16: Front panel of the Labview VI used during the tests of the injector
Labview is a software from National Instrument using a visual programming language. It is particularly convenient to use in laboratories because it automates procedures between measuring devices and computers. Its graphical programming “drag and drop” characteristic makes it easier to use than other programming languages. A Labview program is called a Virtual Instrument (VI). Each VI has a block diagram and a front panel. The block diagram represents graphically all the connections between the inputs, outputs and variables. The front panel is the user interface, and represents the VI as a real instrument. Even if Labview is user-friendly, it takes time to understand all its features and to create a VI that works. More details about Labview are not given here; it is assumed that the reader has the background knowledge to understand a VI. If not, a lot of good tutorials can be found on the Internet, for example [46].

Labview is used in the test setup to manage all the measuring devices and save data into a text file. Figure 6.14 represents the first part of the block diagram. It sends and gathers data to and from the Fluke 45, the multimeter used to measure the heating power on the cartridge heater. The other part of the block diagram is presented in Figure 6.15. It gets data from the thermocouples and the pressure transducers and displays them nicely on the front panel, Figure 6.16. The front panel represents graphically the evolution with time of the temperature on the injector (6 thermocouples) and of the pressure in the chamber and at the end of the injector. A scheme of the injector allows to have a quick look on the temperature gradient on it, thanks to a temperature ↔ colour VI.

The HPLC pump is not connected to the interface and to Labview, but the user can manually set the flow rate so it can be saved in the text file. The two thermocouples TC7 and TC8 are placed at the exhaust of the injector and aim the measure the temperature of the plume. An extra thermocouple is placed inside the liquid nitrogen reservoir in the cryotrap. Thus, the user knows when to refill it (approximately every 5 minutes).
Chapter 7

Test of the injector with deionized water

7.1 Tests at atmospheric pressure

The first tests with deionized water are carried out at atmospheric pressure with an opened vacuum chamber, as presented in Figure 7.1. The HPLC pump is set at 6 mL · min$^{-1}$, the equivalent of 100 mg · s$^{-1}$ for water. A first experiment is to observe the thermal behaviour of the injector while it is heated at the maximal power, 450 W. The temperature profile of the injector is shown in Figure 7.2, as it appears on the front panel of Labview. All the temperatures below 50°C are represented in blue and all the ones above 250°C are in red. The range 50 - 250°C is illustrated by a color gradient going from blue to red.

In Figure 7.2, liquid water is flowing through injector 1. Water vapour is ejected in injectors 2 to 5. However, the flow of vapour doesn’t seem to be optimized, there are a lot a liquid droplets in the steam flow. It can also be seen on the walls of the vacuum chamber, they are all wet. When the temperature at the end of the injector is higher than $T_{C6} = 260°C$, 61
the flow is even worse. From what can be seen in injectors 6 to 8, the temperature is so high that vaporization is likely to take place in the first half of the injector. Some parts of the flow are still liquid while there is vaporization happening upstream. This pushes the liquid water through the exit and it doesn’t have time to vaporize. The flow mainly consists of warm liquid droplets, even though the end of the injector is very warm ($T_C6 > 260^\circ C$).

Figure 7.2: Evolution of the temperature profile of the injector while being continuously heated at 450 W with $Q = 6 \text{ mL} \cdot \text{min}^{-1}$ of water

It is also noticed that the water doesn’t flow continuously through the seven holes. On the opposite, it flows irregularly through only one or two holes at a time, during a small duration (less than a second). It is understood that in each hole, the water vaporizes while
there is still some liquid downstream. The liquid and gaseous water are ejected through
the hole. The empty hole is refilled by some liquid water and the same phenomenon occurs
again and again. The injector is operating in an “unwanted pulse mode”. The irregular flow
of water is shown in the three pictures in Figure 7.3.

![Figure 7.3: Running of the injector at atmospheric pressure. Irregular flow of water through the seven holes.](image)

Now, it is interesting to figure out if the injector can work in a steady state: constant flow
rate, constant heating power and constant temperature on its surface. Figure 7.4 shows one
typical test. The HPLC pump is set at 6 mL · min$^{-1}$. The heating power $P$ is started at its
maximum: 450 W. One minute later, when $TC6 = 130^\circ$C, the heating power $P$ is decreased
down to 230 W. After two more minutes, the six TC values are stable between 96$^\circ$C (for
$TC1$) and 145$^\circ$C (for $TC5$). The same procedure is repeated, but this time $P = 250$ W.
During the next 10 min, the six TC values are stable between 96$^\circ$C (for $TC1$) and 154$^\circ$C
(for $TC5$). During these two first tests, the injector is operating in steady state.

Once again, the same procedure is repeated three times, with $P = 270$ W, $P = 260$ W and
$P = 255$ W. In the three cases, the evolution of $TC5$ and $TC6$ over 300$^\circ$C shows that $P$
is too high to reach thermal equilibrium. During the last three tests, the injector is not
working in steady state. This shows that if the injector is operated at atmospheric pressure
with $Q = 6$ mL · min$^{-1}$, the highest temperature possible on the injector during a steady
state is $TC5 = 154^\circ$C. This is lower than the 210$^\circ$C required to decompose Ammonium
Nitrate which is part of the simulant of the LMP-103S. Therefore, it is believed that this
CHAPTER 7. TEST OF THE INJECTOR WITH DEIONIZED WATER

The injector cannot decompose the simulant in a steady state at atmospheric pressure. Tests with the Ammonium Nitrate solution shall not be made in these conditions.

Another hypothesis explaining the “unwanted pulse mode” is the effect of Earth’s gravity on the fluid. Indeed, as the injector is horizontal, the liquid tends to enter first in the lower holes, which create an asymmetrical distribution of the fluid in the seven holes. In order to
7.2. TESTS AT 20 MBAR

verify if it could be the case, the vacuum chamber is inclined of 90°, as shown in Figure 7.5. In this case, the injector is vertical, ejecting upward. Tests at atmospheric pressure are performed again, and the flow is as bad as before. There are still a lot of droplets and the water doesn’t flow continuously through the seven holes. Earth’s gravity is therefore not the main reason of the problems encountered at atmospheric pressure.

![Figure 7.5: Vacuum chamber inclined of 90°](image)

7.2 Tests at 20 mbar

Tests at low pressure can now start. The vacuum chamber is closed, the vacuum pump starts and the lowest level of vacuum possible with this setup is reached within 5 min: 0.02 bar. The cryotrap has to be filled with liquid nitrogen before the beginning of the experiments. As it has a temperature of −196°C, one should be very cautious during this handling. The liquid nitrogen stored in a Dewar flask is carefully poured in the cryotrap as shown in Figure 7.6. During the experiments the liquid nitrogen vaporizes quite fast, so the cryotrap has to be filled in every five minutes.

During the tests, the aim was to find a proper way to bring the injector into a long lasting thermal equilibrium while it ejects a perfect steam flow. It was found that if the temperature $T_C6$ at the end of the injector is higher that 280°C, the steam flow contains liquid droplets. As previously observed, water vapor created at the beginning of the injector pushes some remaining liquid water through the exit.

It was observed that the steam flow is good (continuous steam flow with a visual absence of droplets) if $160°C < T_C6 < 260°C$. However, one has to keep in mind that the injector will also be tested with a simulant of the LMP-103S, the water solution of Ammonium Nitrate. It decomposes around 210°C. In order to avoid the formation of $\text{NH}_4\text{NO}_3$ crystals in the injector, the temperature at its end shall be higher than 210°C. If the temperature at the end of the injector is lower than 210°C, the formation of $\text{NH}_4\text{NO}_3$ crystals can block off the seven holes, which would stop the flow, increase the pressure and heat the Ammonium Nitrate solution upstream. In the worst case scenario, it could lead to an explosion. Thus,
in order to avoid the formation of $\text{NH}_4\text{NO}_3$ crystals, a reasonable margin is to set the lower limit at $230^\circ\text{C}$. Consequently, the tests were aimed to obtain the followings:

- Constant flow rate
- Constant heating power
- $230^\circ\text{C} < T_C6 < 260^\circ\text{C}$
- Steam flow without any droplets

In Figure 7.7, the injector is ejecting $100 \text{ mg} \cdot \text{s}^{-1}$ of water vapor. As the water is totally in a gaseous state and as the flow is small, nothing can be seen on the picture at the end of the injector.
Figure 7.8 shows one typical test with water. The pump is started after 1 min and flows continuously 6 mL · min\(^{-1}\) (equivalent to 100 mg · s\(^{-1}\) for water). At the same time, the heating power \(P\) is also started at its maximum: 450 W. One minute later, when \(TC6 = 130^\circ C\), the heating power is decreased down to 252 W. After several minutes, the evolution of \(TC6\) and \(TC5\) shows that \(P\) is not enough. The same procedure is repeated, but this time \(P=260\) W. During the next 10 min, \(TC6\) and \(TC5\) are increasing, showing that 260 W are too much.

Once again, the same routine is repeated, with \(P=255\) W. The heating power is then slightly manually regulated **between 249 and 255 W**. The results are clear: thermal equilibrium of the injector is achieved and the steam flow doesn’t contain any droplets. Furthermore, as previously requested, \(TC6 = 240^\circ C\), so \(230^\circ C < TC6 < 260^\circ C\). During the last 15 min, the injector is working exactly as it is supposed to work with water.

This experimental range of 249 - 255 W has to be compared with the theoretical one. Indeed, it has been calculated in Table 5.1 that 253 W are required to vaporize 100 mg · s\(^{-1}\) of water initially at 20\(^\circ\)C. Therefore, this value is exactly in the experimental range of 249 - 255 W. Consequently, this confirms the calculation of the power required by the injector and of the 5.5 W of thermal losses at low pressure.

During the tests, the pressure in the vacuum chamber increases from 0.02 to 0.04 bar due to the steam. The last pressure value is very close to the 42 mbar calculated in Section 6.2, which corresponds to the theoretical pressure equilibrium between the pumping speed and the gas flow rate.

It is also noticed in Figure 7.8 that the pressure graph is not so relevant for the analysis of the data. Due to the gas expansion, the pressure at the end of the injector was expected to be much higher than the one in the chamber; it is clearly not the case. As there is no apparent difference between the two pressure measurements, the pipe linked to the pressure transducer is disconnected from the end of the injector. Indeed, thermal losses were expected to occur from the injector to the pipe connected to the PT. However, further tests don’t strengthen this theory, as the results obtained are identical with and without the pipe.
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Figure 7.8: Test of the injector at low pressure with water
7.3 Temperature of the plume

The thermocouples $TC_7$ and $TC_8$ are especially situated to study the temperature of the plume. As shown in Figure 7.3, $TC_7$ is placed exactly in the middle of the exhaust gases of the injector and $TC_8$ is on the side, not directly inside the plume. Figure 7.9 represents the evolution of $TC_7$ and $TC_8$ during one test at atmospheric pressure. While the temperature of the air around the injector is always close to $20^\circ C$, the temperature of the plume is on average $75^\circ C$ when the heating power is on. When the power stops, $TC_7$ quickly goes down to $20^\circ C$.

This is very different to what happens at low pressure in Figure 7.10. In this case, both $TC_7$ and $TC_8$ are around $30^\circ C$. This is explained by the fact that the boiling temperature of water in these conditions (42 mbar) is $30^\circ C$. Having the same reasoning for tests at atmospheric pressure, it can seem strange that the temperature of the plume is $75^\circ C$ and not $100^\circ C$. The reason is that the plume is not composed only by gaseous water at $100^\circ C$ but also by liquid droplets at lower temperature.

7.4 Vacuum pump problem

After one week of tests under vacuum, a major problem appeared. The pump was spewing out oil through the air exhaust. This is particularly disturbing because the air exhaust of the vacuum pump is connected to the ventilation system (pink pipe in Figures 6.2 and 6.4). In order to prevent damages to the vacuum pump and to the ventilation system, it was decided to stop the tests. This is thought to be because of some the water vapour entering in the pump and damaging its working principle. Indeed, the cryotrap is supposed to condense all the water vapour, but a fraction of it might still go into the pump.

Tests have unfortunately to stop here in the scope of this Master Thesis work. However, they will keep on later on by SSC. A new vacuum pump has been ordered, it is a diaphragm pump chemically resistant for which water vapour is not a problem. Therefore, the upcoming tests will be performed as soon as the new pump is delivered.
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Figure 7.9: Evolution of $TC_7$ and $TC_8$ during a test at atmospheric pressure with water

Figure 7.10: Evolution of $TC_7$ and $TC_8$ during a test at 30 mbar with water
Chapter 8

Conclusion

8.1 Overall impression

I have been carrying this Master Thesis from the 10\textsuperscript{th} of January to the 8\textsuperscript{th} of July 2011 at the Swedish Space Corporation. I highly liked this challenging project that was very diversified. Indeed, I performed a lot of different tasks: orbital trajectory calculations, economic analysis, understanding of the plasma physics, heat transfer investigations, chemical comprehension of a decomposition process, technical drawing, Labview programming, order of various pieces of equipment, setting-up and performing experimental tests, etc. I found it very demanding regarding the varied knowledge and skills required, which enables me to fully appreciate my French background of “Ingénieur généraliste”.

8.2 Upcoming tests

As soon as the new pump is delivered, tests with water can be finalized. This involves to try the injector with different flow rates (50 and 150 mg \( \cdot \) s\(^{-1}\)) and to investigate the heating power needed in order to reach thermal equilibrium with a good steam flow. Are these powers different from the 249 - 255 W required at 100 mg \( \cdot \) s\(^{-1}\)? Or maybe it is just the temperature profile of the injector that is different?

The next step will be to test the injector with the simulant of the LMP-103S: a water solution of Ammonium Nitrate. It is very important to avoid the formation of NH\(_4\)NO\(_3\) crystals inside the injector. Therefore, the tests will first start with a solution at low concentration (10\% by mass) during a short time (5 s). Only after a cautious inspection of the injector in between two tests, the concentration of the solution and the duration of the tests will be increased. The last test aimed is with high concentration (50\%) during a long period (15 min). During the tests, the temperature at the end of the injector will have to be \( TC6 > 210^\circ C \) in order to decompose properly all the NH\(_4\)NO\(_3\). The power required to obtain thermal equilibrium will be compared with the 249 - 255 W needed for water at 100 mg \( \cdot \) s\(^{-1}\).

Eventually, the tests will be done with the LMP-103S, following the same procedure as for the simulant (test, cautious inspection and increased duration).
The LMP-103S and its simulant are decomposed into chemicals that can cause damages to the vacuum pump, for example nitric acid. Therefore, the new chemically resistant diaphragm pump that has been ordered will be very valuable for the upcoming tests.

At last, it is of course planned that the HPEP thruster will be tested in a bigger vacuum chamber with the designed injector, using the LMP-103S as a propellant.

As for me, I will keep on working on this project and hopefully on many others at the Swedish Space Corporation as a Space Propulsion Engineer.
Bibliography


[16] ECAPS, Monopropellant LMP-103S, 2006


