

Design, Construction and Testing of a Radio Frequency Mini Ion Engine According to the Propulsion Requirements of the Next Generation Gravity Missions “NGGM”

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Abstract: The “Next-Generation Gravity Mission” (NGGM) concept, under study at the European Space Agency (ESA), will make use of the Low-Low Satellite-to-Satellite Tracking technique to monitor the temporal variations of the Earth gravity field over a long time span. One of the enabling technologies for the realization of the NGGM is the micro propulsion system. Granted of an ESA contract within the basic Technology Research Programme (TRP), an engineering model of such a thruster was designed, constructed and tested in TransMIT GmbH in Germany with subcontracts to the University of Southampton and CEC GmbH. The paper discusses the design specifications and delivers the test results.

Nomenclature

<i>RIT</i>	=	Radio Frequency Ion Thruster
<i>RF</i>	=	Radio Frequency (~1MHz)
<i>RFG</i>	=	Radio Frequency Generator
<i>PSCU</i>	=	Power Supply and Control Unit
<i>FCU</i>	=	Flow Control Unit
<i>F</i>	=	Thrust
<i>dm/dt</i>	=	Propellant mass flow
<i>U_{PHV}</i>	=	Positive High Voltage
<i>U_{NHV}</i>	=	Negative High Voltage
<i>P_{rfg}</i>	=	RF Power Consumption

I. Introduction

ESA’s Next-Generation Gravity Mission” (NGGM) concept will make use of the Low-Low Satellite-to-Satellite Tracking (LL-SST) technique to monitor the temporal variations of the Earth gravity field over a long time span,

like the GRACE mission, but with a much higher spatial resolution (comparable to that provided by the GOCE mission) and an improved temporal resolution.

One of the enabling technologies for the realization of the NGGM is the micro propulsion system. In particular, it allows:

- Satellite orbit maintenance at its operational altitude;
- Satellite loose formation control;
- Implementation of the drag-free control at level of each satellite;
- attitude control of each satellite;
- maintenance of the inter-satellite laser link.

The latest results from the NGGM preparatory studies have identified the preliminary requirements for the propulsion system in order to cope with the large variation of the drag forces encountered in a long duration mission.

The requirements contain the lifetime of the thruster, power consumption, overall Isp of the subsystem, on top of the specific thrust requirements like thrust range, dynamics, resolution, throttability and thrust noise. This typology of thruster will be applicable to other missions, either for Earth Observation or Space Science, which require precise attitude and orbit control by means of low-noise force actuators that are widely configurable. Examples include the high-resolution imaging from geosynchronous orbits. Concerning GOCE [1] and GRACE [3] follow-on missions, a thruster had to be designed to meet the requirements of the NGGM mission. The most challenging requirement is the very high thrust dynamic to cope with the large variation of the drag forces encountered in a long duration mission and the very high life time requirements. In particular a strong constrain is given by the low power consumption due to the limited solar cell arrays areas on NGGM spacecraft, that would also ensures the minimizing of the heat production in the thruster.

A multiphysics model is developed to predict the thruster performance values. The maximum thrust determines the diameter of the thruster. The very large thrust dynamics of the thruster dictates some restriction in the ion optics design, which leads -together with optimal discharge pressure- to higher amount of natural gas flow in lower thrusts, and therefore lower mass efficiency and Isp in this thrust range. A mechanical and a thermal model were developed to simulate the thermal and mechanical loads on the thruster during the start and during the operation time in orbit. Supported by an ESA TRP contract, an engineering model of such a thruster was designed, constructed and tested: this paper delivers the test results of the thruster that is manufactured and tested against major NGGM requirements.

II. The Next Generation Gravity Mission concept

Following the selection in 1999 of the first Earth Explorer Core mission, GOCE [1] (Gravity field and steady-state Ocean Circulation Explorer), the European Space Agency has initiated in 2003 activities on future gravity missions with a study on observation techniques for solid Earth missions [2], and has continued this effort in recent years through several system studies and technology development activities. In particular, the new science goals were derived within the study [2], which also covered the major characteristics of the sensor systems and defined scenarios with mission outlines and anticipated performance for the determination of the gravity field, improving upon the GRACE [3] (Gravity Recovery and Climate Experiment) and GOCE missions. The most promising identified scenarios addressed a Gradiometer mission, targeting the higher spatial frequencies of the spherical harmonics, and a Satellite-to-Satellite Tracking (SST) mission for the determination of the low spatial frequencies to higher accuracies than its predecessors.

The following system studies have suggested that such a mission ought to be based on Low-Low Satellite-to-Satellite Tracking (ll-SST). This technique exploits the satellites themselves as “proof masses” immersed in the Earth gravity field (as shown in Fig. 1). The satellites fly in a loose formation in which they are free to move under the action of the gravity field within a measurement band with a typical lower bound <1 mHz. Since the altitude of the satellite must be low (around 350 km or less) to increase the sensitivity to the higher harmonics of the Earth gravity field, the relative motion between the satellites will be perturbed by aerodynamic forces (i.e. air drag) too. The distance variation between their centres of mass (produced by both gravitational and non-gravitational forces together) is measured at high resolution by a distance metrology set-up. The achievement of the mission objectives calls for a relative error in the distance measurement of the order of $10\text{-}13$ m/ $\sqrt{\text{Hz}}$, namely 10 nm/ $\sqrt{\text{Hz}}$ for a typical inter-satellite distance of 100 km.

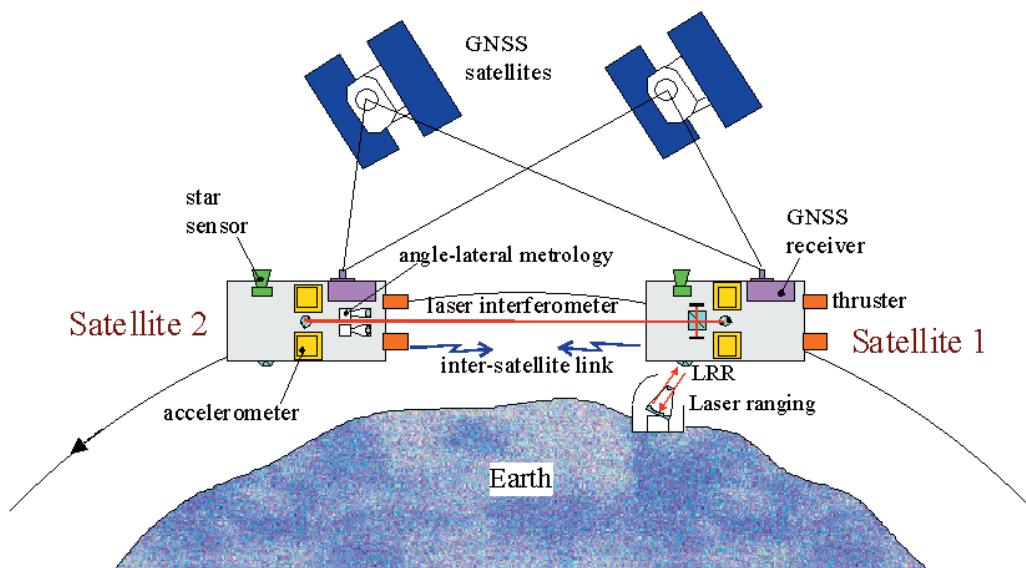


Figure 1: ISST principle for the Next Generation Gravity Mission (courtesy by TAS Turin, Italy).

All these activities have received precious inputs from the in-flight lessons learned either from GOCE or the US-German mission GRACE. Others activities were initiated to improve the technology of the Attitude and Orbit Control System (AOCS, e.g. tests on magnetic bearing reaction wheels and miniaturized ion thrusters), this area being clearly a driver for the mission design.

The objectives of a “Next-Generation Gravity Mission” (NGGM), as identified in the science workshops held in 2007[4] and 2009[5], are to provide the temporal variations of the Earth’s gravity field over a time span of several years with high spatial resolution (i.e. ~100 km, comparable to that provided by GOCE) and higher temporal resolution than GRACE, which is limited to ~1 month intervals between successive gravity field maps. Such a mission will significantly improve our understanding of mass change of ice sheets and glaciers, continental water cycles, ocean masses dynamics, solid-earth deformations and other geophysical phenomena through observations of the mass transportation within the Earth system, as derived from the consequent temporal variations of the gravity field.

From this basis, in 2009 the “Assessment of a Next Generation Gravity Mission for Monitoring the Variations of the Earth’s Gravity” [6-7] has been run through two parallel studies, one led by ThalesAlenia Space (Turin, Italy) and another one by Astrium GmbH (Friedrichshafen, Germany). Several mission aspects were analysed, covering orbit geometry, altitude, inter-satellite distance, number and accommodation of sensing instruments (i.e. accelerometers) per satellite, laser ranging technology, satellite formation and lifetime. After the system studies, further technological development studies have been started within the ESA Technology Research Programme, concerning the laser metrology design (i.e. laser head and laser stabilization unit), miniaturized Field Emission Electric Propulsion (FEEP) and miniaturised Radio-frequency Ion Thruster (mini-RIT).

The low-thrust electric propulsion with variable thrust is one of the key technologies for accomplishing the challenging mission requirements of the NGGM. In particular, it shall allow:

- satellite orbit maintenance;
- satellite loose formation control;
- implementation of the individual drag-free control;
- attitude control of each satellite;
- laser beam pointing control.

III. NGGM Requirements

The future Next-Generation Gravity Mission concept, as well as other Earth Observation missions and Science missions, will require fine attitude and orbit control by means of low-noise force actuators. Such missions will benefit and, in some case, will be enabled by electric propulsion systems capable of combining high specific impulse and ultra fine controllability. For the broad range of applications, more than one thruster typology is required, depending also on the satellite configuration and on the type of formation geometry.

For NGGM, in case of the simplest in-line formation, in which the satellites chase each other along the same orbit and experience a main drag force mainly along the velocity vector, the need of two types of thrusters has been identified (main thrusters and miniaturized thrusters for lateral controllability, in brief “lateral” ones). The criticalities of these thrusters are the thrust dynamic range (a value > 40 is needed to cope with the large variation of the drag forces encountered in a long duration mission, especially in periods of high solar activity), the specific power for minimizing the solar panel surface, the specific impulse for reducing propellant consumption (the current estimate leads to about 50 kg of Xenon for a 10 year mission) and the lifetime.

Table 1 summarizes the propulsion requirements for the lateral thrusters as identified in the last system studies [6-7].

Parameter	Unit	Collinear Lateral Thrusters
Minimum Thrust	mN	0.05 (0*)
Maximum Thrust	mN	< 3
Thrust Resolution	μN	0.5
Thrust Noise		$< 1\mu\text{N}/\sqrt{\text{Hz}}$ above 0.08Hz
Rise/Fall Time	ms	< 50
Slew Rate	mN/s	> 0.5
Update command rate	Hz	10
Thrust non linearity		$< 2\%$
Lifetime	yr	> 10
Specific Power	W/mN	< 40
* Thrust has to be turned off completely if thruster is not operating		

Table 1 Propulsion requirements for the lateral thrusters of the NGGM

IV. Thruster Modelling

A. Fundamentals of Radio Frequency Ion Thrusters

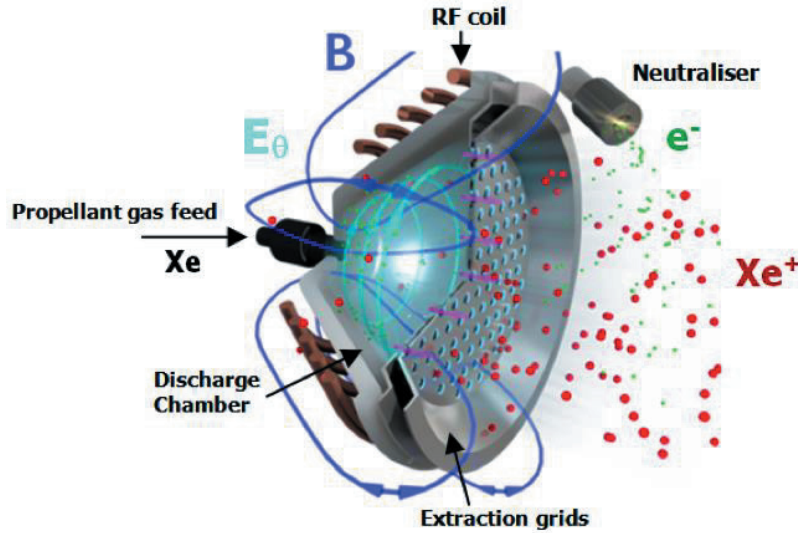


Figure 2: Operating principle of a Radio Frequency Ion Thruster⁸.

B. Ionization

The discharge chamber of a radio frequency ion thruster consists of a dielectric vessel surrounded by a conducting coil (termed RF coil or induction coil) to which an oscillating current (typically at a few MHz) is applied to generate a RF field. A gas feed system supplies propellant to be ionised; typically the inert gas xenon is used but RF ion thrusters have also been operated on other propellants such as mercury, caesium, argon, nitrogen and oxygen. The RF magnetic field penetrates the vessel inducing an azimuthal electric field E_{θ} according to Faraday's Law:

$$\nabla \times \vec{E} = -\frac{\partial \vec{B}}{\partial t}$$

The induced electric field acts to accelerate free electrons present within the discharge chamber to sufficient energies for electron-impact ionizations of the neutral propellant to occur, leading to a self-sustaining discharge. No internal cathodes are required to supply electrons to maintain the discharge, as is required for Kaufman ion thrusters. A weakly ionized, non-equilibrium plasma is formed, with electron temperature considerably higher than that of the ions and neutral propellant, whose temperatures approximate that of a gas in thermal equilibrium with the walls of the discharge chamber. A stable discharge requires the power absorbed by the plasma to balance the power dissipated.

C. Ion Extraction and acceleration

The ions generated within the discharge are extracted, focussed and accelerated by a set of two (sometimes three) multi-aperture grid electrodes which are biased to high potentials.

A plasma sheath forms upstream to the surface of the first grid (as depicted in Figure 3), which is biased to a high positive potential typically between 1-2 kV with respect to 'ground' (or common potential of the spacecraft). This grid is often referred to as the screen grid. The second grid is located in close proximity to the first and is termed the accelerator grid.

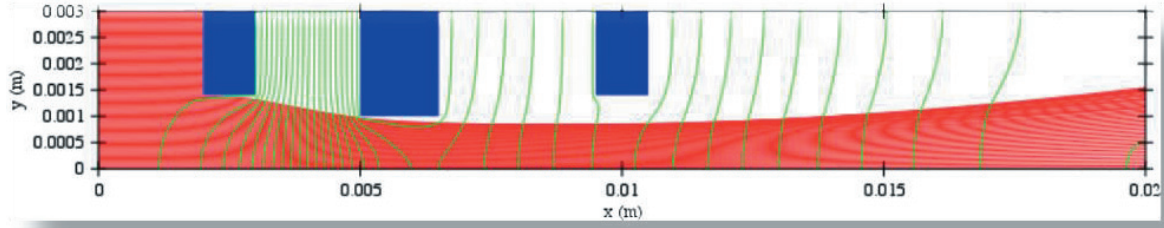


Figure 3: Schematics indicating ion extraction through a grid aperture.

The accelerator grid is biased to a large negative potential, typically between -100V and -500V with respect to ground (or common potential). Ions that diffuse from the bulk plasma within the discharge chamber to the plasma (sheath) edge upstream of the screen grid are accelerated across the plasma sheath and either impinge on the upstream face of the screen grid or pass through the apertures, where they are then accelerated to high velocity due to the high potential difference between the screen and accelerator grid. The beam divergence is a function of the geometrical parameters of the “Ion Optics System” like the grid thicknesses and the distances between the grids and also on voltages and the plasma parameters. An intelligent design of the ion optics system ensures the good performance of the thruster and determines the beam divergence.

D. Thrust and Isp

As stated previously, the thrust generated is equivalent to the net force acting on the grids due to the extraction and acceleration of the ions. Thrust is proportional to the rate of change of momentum of the expelled propellant, dependent on ion beam current, ion mass and velocity of the ions (ion velocity being proportional to the square root of the beam voltage). Thrust can be approximated by:

$$F = \eta_d \cdot I_{beam} \cdot \sqrt{2 \frac{m_i}{q_i} (U_+ + V_p)} = 50.87 \mu N \cdot \frac{I_{beam}}{mA} \cdot \sqrt{\frac{U_+ + V_p}{kV}} \quad \text{Eq.1}$$

where η_d is a thrust correction factor due to beam divergence, m_i is the mean ion mass, which for xenon is $m_i = 131.3$ AMU, q_i is the mean ion charge $q_i = 1.01e$ (assuming <1% ratio of singly-to-doubly charged ions), U_+ is the positive high potential applied to the screen grid and V_p is the plasma potential, which can be assumed to be approx. 20-25V.

$$I_{sp} = \frac{\eta_m}{g_o} \cdot \sqrt{2 \frac{q_i}{m_i} (U_+ + V_p)} = 3927.6s \cdot \eta_m \cdot \sqrt{\frac{U_+ + V_p}{kV}} \quad \text{Eq. 2}$$

E. Thruster Power

The power of the thruster is a summation of the ionisation power, which is necessary to produce and extract enough current and the extraction/acceleration power, that accelerate the ion current to the necessary kinetic energy. The electric power necessary for the ionisation comes in the form of radio frequency EM field from the Radio Frequency Generator and its amount is a function of ion current and the size of the thruster. A calculation of the RF power needs a large amount of the EM simulation of the thruster. The necessary electric field power for acceleration of the ions will be delivered by the Positive High Voltage (PHV) power Supply. The amount of the DC power of the PHV can be easily calculated as the multiplication of the current and the voltage of the PHV. The negative High Voltage has only a minor role in the calculation of the total power, as its current and voltage is only 1-2 % of the PHV current and voltage.

$$P_{RFG} \propto f(I_{beam}, D)$$

Eq. 1

$$P_{Beam} = I_{Beam} \times V_{PHV}$$

Where D is the diameter of the thruster.

The size of the thruster or its diameter determines the maximum current, i.e. the maximum thrust of the thruster. A larger thruster needs however a higher RF power as the surface contacting with the plasma would be higher.

F. MODELLING

A comprehensive radio frequency (RF) gridded – ion thruster model is developed in order to simulate the plasma, electromagnetic and thermal aspects of the thruster⁹. This was achieved by representing different areas of physics by a different sub-model: 2D ion optics, 3D neutral gas, Boltzmann electron transport, 0D plasma, RF circuit, 2D electromagnetic and 3D thermal. All the sub – models were coupled together for a consistent solution. Such a holistic approach, apart from other parameters, enables one to determine the power transferred to the plasma, the coupling between the plasma and the coil, the extraction beam current and ultimately the thruster efficiency. Therefore, a thorough optimization analysis can be performed. Furthermore, the thruster performance can be simulated as in the actual experiment where all different parameters/physics are connected. For instance, the thruster temperature affects the electrical conductivity of the thruster components, and this influences the electromagnetic heating losses which, in turn, change the coil current and so on. By comparing the RIT 3.5 experimental data to the model results, it was shown that the model can predict the input power and current to the radio – frequency generator at different operational conditions with an error less than 10%, similar accuracy was achieved in predicting the temperature distribution. Figure 4 shows the simulated thruster performance when the input mass flow rate \dot{m}_{in} is kept constant, but the input power is varied to achieve a particular mass utilization efficiency. This means that as the input power to the radio – frequency generator is increased for the same mass flow rate, the ion density rises and, thus, the beam current increases, resulting in a higher mass utilization efficiency. Nevertheless, the required power increases exponentially with increasing mass utilization efficiency.

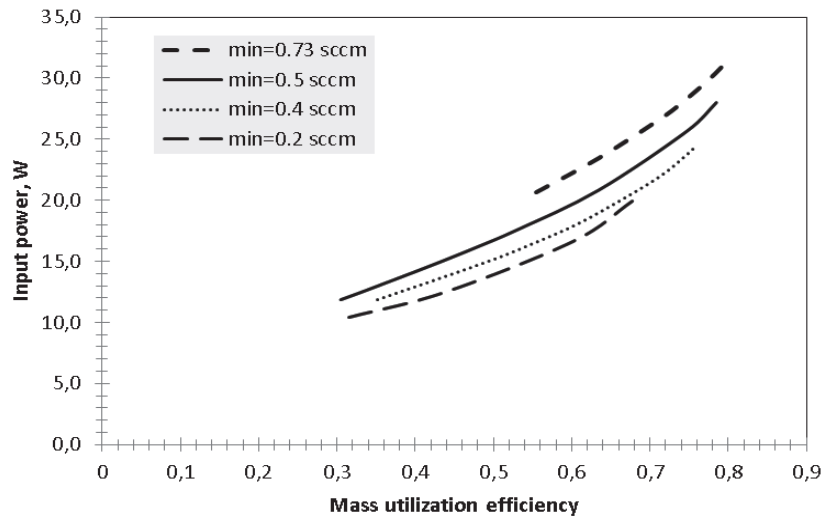


Figure 4. Input power as a function of the mass utilization efficiency for different mass flow rates.
The different input mass flow rates are represented by the “min” variable.

A thermal model is developed to distinguish the critical components in the RIT 3.5 design. Figure 5 depicts the simulated temperature distribution inside the RIT 3.5 thruster at 18 W of input power without the plasma. The model is cross checked with the results of the test. In the thruster nine thermosensors are installed, which register the temperature of the different components in the thruster. The thruster can be operated to its maximum thrust without any thermal problems.

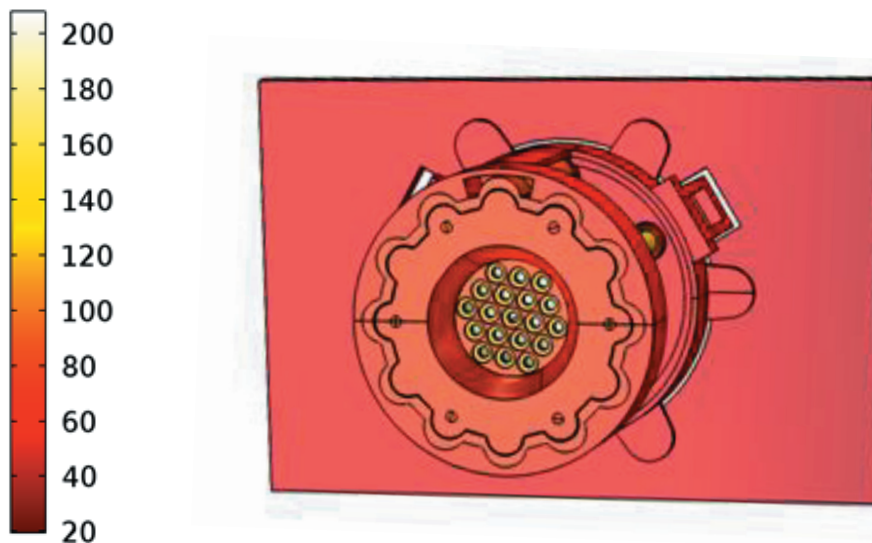


Figure 5. Temperature distribution in RIT 3.5 thruster for 18 W of input power without plasma.
Temperatures are in°C; the thruster outer case is hidden for the visualization purposes.

G. Representative Mechanical and Thermal Environment and Requirements for NGGM

Figure 6 shows the RIT3.5 thruster. The thrusters for NGGM is designed in order to withstand peak loads arising from the following environments:

- **Ground**

On Ground Handling and Hoisting Operations
 (considered enveloped by Flight environment induced loads).

- **Flight**

Induced environments: Sinusoidal Loads, Random Mechanical Loads, Shock Loads.

These environments are applicable during the following mission phases:

- Lift-off
- Ascent
- On-Orbit
- Transport and Handling

- **Thermal**

The thermal environment considered in combination with the above loads is RT (20°C).

A preliminary mechanical environment is defined.

The sinusoidal and random vibration environment of GOCE, LISA Pathfinder and ARTES are considered.

The shock environment of the following launchers are considered:

- Vega
- Soyuz
- Ariane V

- Dnepr•Rocket



Figure 6. RIT 3.5 thruster

V. Experimental results

The most challenging part of the thruster requirements for NGGM was the very large dynamic range of the thrust. The minimum thrust was required to be around $50\mu\text{N}$ and the maximum around $2500\mu\text{N}$. In the tests a minimum thrust of $45\mu\text{N}$ and a maximum thrust of $2500\mu\text{N}$ were achieved. The power consumption at maximum thrust is around 90 Watts, which allows a power to thrust ratio below 40W/mN at bus level. The figure 7 shows the preliminary results of the thrust range tests: power consumption of the thruster (including the losses in power supplies) for different thrust levels between $45\mu\text{N}$ - $2500\mu\text{N}$

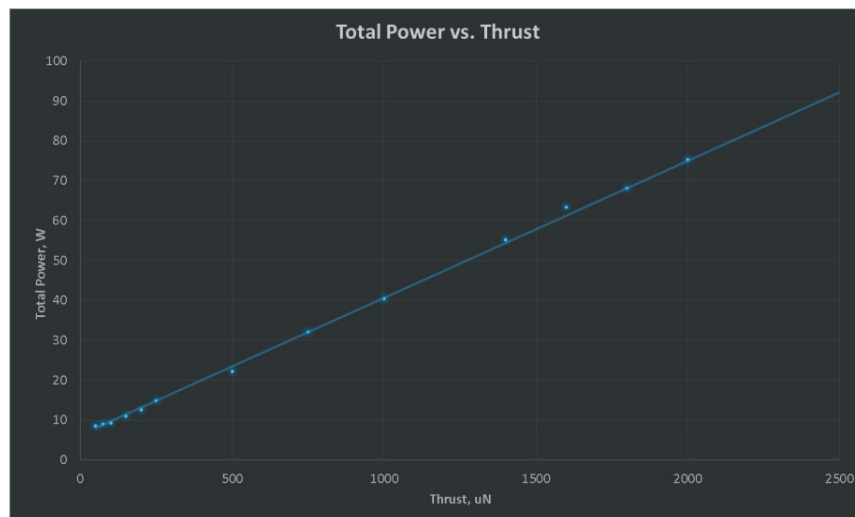


Figure 7. The power consumption of the thruster (including the losses in power supplies) for different thrust levels between $45\mu\text{N}$ - $2500\mu\text{N}$

Figure 8 shows the variation of the thruster Isp for different thrusts. An Isp between 300s to 3800s depending on the thrust were achieved.

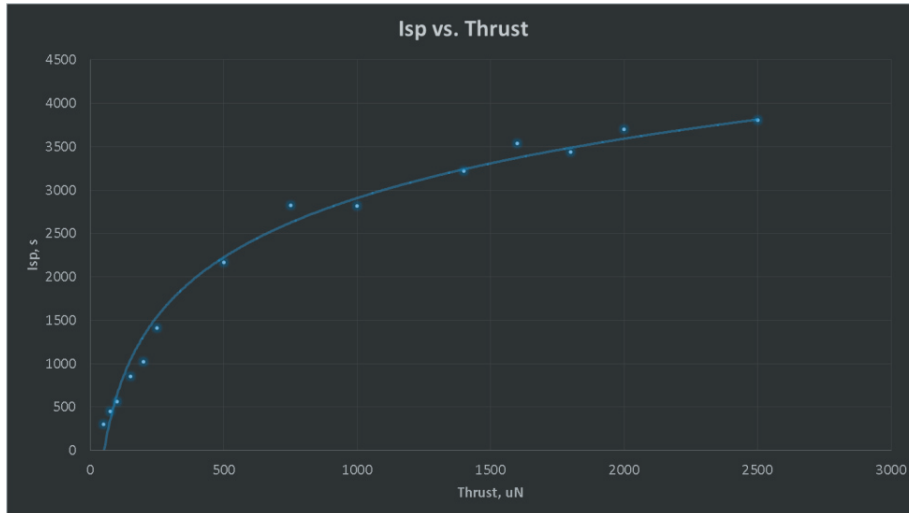


Figure 8. The Isp of the thruster for different thrust levels between 45 μ N-2500 μ N

The thrust stepping and resolution is tested for different thrust levels. A stepping of under 0.2 μ N could be achieved. Figure 9 shows the thrust stepping for 2mN thrust.

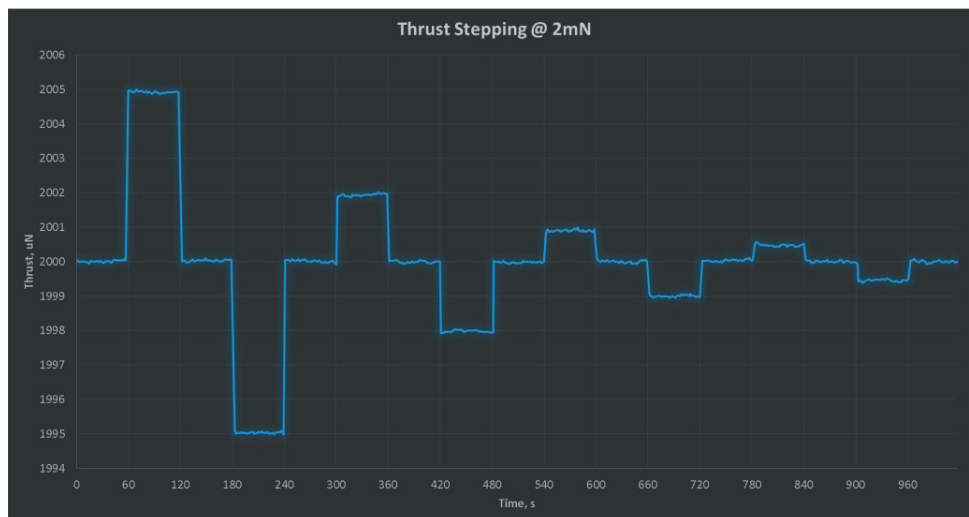


Figure 8. The thrust stepping by 2mN thrust

It is to mention that the tests are still going on and the best performance of the thruster may not yet have been achieved.

VI. Conclusion

A Mini Ion Engine is designed, modelled and constructed according to the requirements of the Next Generation Gravity Mission concept. The experimental results show a very good agreement between the experimental results with the predicted results from the modelling and good compliance with the NGGM requirements.

Acknowledgments

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- ⁹Mantas Dobkevicius, Johann Müller and Davar Feili, "Comprehensive Radio Frequency (RF) Ion Thruster Electromagnetic and Thermal Modelling," 30th ISTS, 34th IEPC and 6th NSAT Joint Conference, Kobe-Hyogo, Japan, July 4 – 10, 2015.