The ESA Earth Observation Programme activities for the design, development and qualification of the mN-FEEP thruster

IEPC-2019-686

Presented at the 36th International Electric Propulsion Conference
University of Vienna • Vienna, Austria
September 15-20, 2019

L. Massotti[1], J. Gonzalez del Amo[2], J. Aschbacher[3], P. Silvestrin[4]
[1,2,4]*ESA/ESTEC*, The Netherlands; [3]*ESA/ESRIN*, Italy

A. Reissner[5]
ENPULSION GmbH, Austria

and

B. Seifert[6]
FOTEC GmbH, Austria

The EOP Directorate of the European Space Agency has contributed and supported the design of a variable specific impulse electrostatic thruster based on Field Emission Electric Propulsion (FEEP), in which a liquid propellant is electrostatically extracted and accelerated to high exhaust velocity. The core element of this propulsion technology is a passively fed, porous tungsten crown emitter consisting of 28 sharp needles. This emitter technology has been developed and qualified over more than a decade at FOTEC and the Austrian Institute of Technology under ten ESA/EOP contracts since 2005 targeting the *Next Generation Gravity Mission* (in brief, NGGM) needs, and it has recently been adapted for the use as main propulsion system in Nano- and Small-satellites. This paper summarizes the development efforts of the last decade and provides an assessment of the performance of this thruster technology, after extensive simulation and testing campaigns.

I. Introduction

Following the selection in 1999 of the first Earth Explorer Core mission, GOCE (*Gravity field and steady state Ocean Circulation Explorer*), the European Space Agency has initiated in 2003 activities on future gravity missions (*Next Generation Gravity Mission* or "NGGM") with a study on observation techniques for solid Earth missions, and has continued this effort in recent years through several scientific & system studies and technology development activities (an overview is given in Ref. 1). Whatever the number of the satellites and the preferred constellation for a proper temporal and spatial sampling of the Earth surface are, the distance variation between the satellites' centre of mass has to be measured to very high resolution by a distance metrology setup. Moreover, in order to retrieve the gravity signal from the laser metrology measurements, the non-gravitational forces need to be measured.

[1] EO System Engineer, EOP-ΦMP, Luca.Massotti@esa.int
[2] Head of the Electric Propulsion Section, TEC-MPE, Jose.Gonzalez.del.Amo@esa.int
[3] Director of Earth Observation Programmes, EOP, Josef.Aschbacher@esa.int
[4] Head of the Future Systems Department, EOP-Φ, Pierluigi.Silvestrin@esa.int
[5] Founder, CEO, reissner@enpulsion.com
[6] Head of Electric Propulsion, seifert@fotec.at

The proper operation of the accelerometers requires some level of drag compensation in order to reduce the linear and angular non-gravitational accelerations in view of the accelerometer limited dynamic range. As a consequence, the satellite would need loops to control the attitude, the orbit and the (loose) formation. In order to ensure the mission goals and a mature technology level for all the satellite subsystems, several specific activities were started for the technological development of the optical metrology, and for the breadboarding and testing of ion thrusters and Indium FEEP micro-thrusters for drag compensation and attitude control.

Liquid Metal Ion Sources (LMIS) have been developed since more than 30 years successfully in Austria and flown on numerous international space missions. Building on this heritage, the EOP Directorate has contributed and supported the design of a variable specific impulse electrostatic thruster based on Field Emission Electric Propulsion (FEEP), in which a liquid propellant is electrostatically extracted and accelerated to high exhaust velocity. The core element of this propulsion technology is a passively fed, porous tungsten crown emitter consisting of 28 sharp emitter tips, which features a dynamic thrust range from 1 µN to (potentially) 1 mN at 2000 s to 6000 s specific impulse. This emitter technology has been developed and qualified over more than a decade at FOTEC and the Austrian Institute of Technology under ten ESA/EOP contracts since 2005 targeting the NGGM needs, with more than 100 emitters tested and the successful completion of a 24,000 hrs endurance test showing no degradation in the emitter performance, and it has recently been adapted for the use as main propulsion system in Nano- and Small-satellites.

The paper describes the design evolution of the thruster, up to its final version as a commercial product of the IFM Nano (used in nano-sat up to 40 W) and Micro (targeting small-sat up to 400 W) Thruster. An overview will be given on the extended lifetime testing (up to 30000 hours), and related issues for the consolidation of the mN thruster performance (i.e. characterization, development of models, mass efficiency determination, extractor cleaning techniques for the lifetime test, etc.).

II. The Next Generation Gravity Mission concept

Soon after the selection of GOCE[5] as its first Earth Explorer (1999), ESA started preparatory studies towards a Next Generation Gravity Mission (NGGM). Since the mid-2000's, these studies have focused on a mission concept devoted to sustained observation and quantification of mass transport processes in the Earth system (encompassing AOHIS: Atmosphere – Ocean – Hydrology – Ice – Solid Earth) over a decade-long time span. The objective is achieved by monitoring the time variation of the Earth gravity field by Satellite-to-Satellite Tracking (SST) at high spatial and temporal resolution, using a heterodyne laser interferometer measuring the variation of the distance between two satellites flying in formation at low altitude. Such measurements of mass transport in the Earth system are unique to this type of mission and they complete science's picture of Global Change with otherwise unavailable data.

The NGGM mission and system design will build on the experience of GOCE, in particular for the design of the attitude and orbit controls, GRACE[6], for the concept of an SST mission with a direct inter-satellite metrological link between two spacecraft flying in low-Earth orbit, and GRACE-FO[7] for the laser interferometer instrument.

The science case and design concept of the new mission have been built up in parallel, linked studies of the user science institutes and the engineering teams. The tiny gravity signals associated to mass transport in the Earth system may be detected using "virtual gradiometers" formed by satellites orbiting in formation, with separation among members of tens to hundreds of km. Early studies addressed mission design options such as type of satellite formation, altitude, inter-satellite distance, number and accommodation of sensing instruments (accelerometers) per satellite, ranging technology. Design requirements and performance of formations such as "pearl-string", "pendulum" and "cartwheel" were studied. A notable outcome of this phase was the selection of the "Bender constellation"[2], consisting of two pairs of satellites flying in pearl-string formation in near-polar and mid-inclination orbits, as the best compromise between mission performance and implementation cost. The science studies[3,4] produced a detailed assessment of the impact on the gravity field solutions of ocean tide aliasing, which was found to cause larger errors than non-tidal signals, and developed ocean tide mitigation strategies to be applied in post-processing.

The NGGM mission will be devoted to measuring the time-variations of the gravity field with order of 100 km spatial resolution and weekly or better time resolution. The NGGM objectives will be achieved by employing an inter-satellite laser interferometer (282 THz, corresponding to 1064 nm wavelength) with expected performance in path length measurement error better than 10 nm/√Hz (NGGM threshold specification), provided the flight segment system performance matches that of the distance measurement (e.g. in measuring and/or suppressing disturbing accelerations from the external and internal environment).

A. Mission and System Requirements

The mission requirements, consolidated through the series of system studies, are summarized as follows.

- The instrument set on board each satellite of the NGGM constellation shall comprise a laser interferometer, two accelerometers, a GNSS receiver and retro-reflectors for laser ranging from the ground.
- The lifetime requirement shall be at least 7 years.
- A time series of measurements taken at constant altitude is preferred to a variable-altitude profile. The minimum design altitude will be selected around 340 km, a value which takes into account the resources needed for orbit maintenance and drag-free control over the complete lifetime.
- The altitude will be maintained, GOCE-like, within a band of about 100 m around a specified value which will be selected to realize a controlled longitude shift (and to control aliasing as mentioned below).
- The ESA satellite pair shall be conceived as one-half of a Bender pair, providing global coverage (via the polar pair) and extended coverage of the mid-latitudes (via both pairs). Current orbit design values are in Table 1.

Item	**Polar Pair**	**Inclined pair**
Altitude	340 km	355 km
Inclination	89°	70°
Right Ascension of the Ascending Node at epoch	Ω	Ω+90°

Table 1. Orbit definition

- The target spatial resolution shall be ~100 km (half-wavelength) at mission completion.
- The mission performance, stated in terms of geoid accuracy, shall reach 1 mm accuracy, at 3 days intervals with 500 km spatial resolution and at 10 days intervals with 150 km spatial resolution.
- The temporal resolution shall be such that ocean tide signals can be decoupled from signals from other Earth system constituents (atmosphere, ice, hydrology, and oceans), taking into account aliasing periods.
- For intersatellite distances in the range 70–100 km, the NGGM performance is relatively constant and lengths >100 km do not provide any benefit in terms of gravity field recovery. The current baseline is 100 km.

Figure 1 shows the measurement requirements (threshold and goal) of the fundamental observables of the mission: inter-satellite distance variation and projection of the differential non-gravitational acceleration along the satellite-to-satellite direction. Ultra-precise accelerometers such as those that made up the gradiometer of GOCE[5] can provide the required non-gravitational acceleration measurement performance.

The driving spacecraft system requirements concern the orbit, drag-free and attitude control subsystem (DFACS) and the on-board propulsion, addressed in the next chapter. A stringent temperature stability requirement applies in the compartment enveloping the optical bench with the temperature-sensitive items, including parts of the laser equipment and the accelerometers: $T(f) < 2\cdot [1+(5\cdot 10^{-3}/f)^2]^{1/2}$ mK/√Hz.

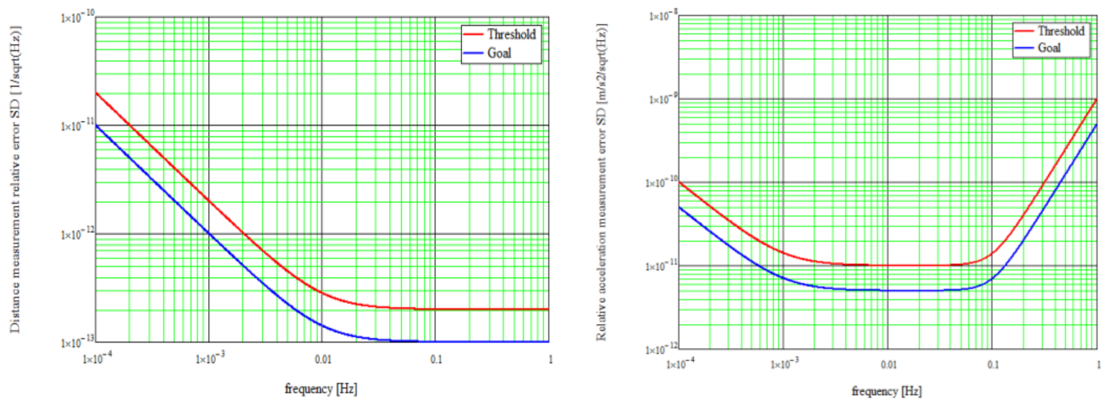

Figure 1. Instrument measurement performance requirements.
Left: Laser Metrology Instrument. Right: Accelerometer

B. Orbit and Spacecraft Design

As noted, the "Bender formation" emerged as the best compromise between gravity field coverage and satellite implementation. This consists of two pearl-string pairs, one in near-polar orbit (inclination between 88° and 90°) and another in medium inclination orbit (between 65° and 75°). The polar pair ensures complete geographic coverage while the combination of the two pairs provides sufficiently isotropic sampling of the gravity signal, East-West as well as North-South.

Later studies[4] addressed inter-satellite distance and the orbit definition of the two satellite pairs: altitude / repeat period, inclination, nodal spacing. The basic requirements of time-variable gravity measurements from space are: (a) orbit altitude as low as possible to maximize signal strength, (b) retrieval periods as short as possible to maximize the time resolution of the gravity field solutions, and (c) ground track coverage as dense as possible within the retrieval period for maximum spatial resolution. The orbit altitude is driven by satellite engineering constraints (configuration, drag cross section, propulsion type and allowed propellant load) and a suitable range was found around 340 km circular altitude; detailed solutions to realize specific repeat patterns, which depend on the semi-major axis, were sought around that value. The study identified a group of constellations with comparable performance; besides, it was found that the quality of the retrieved time-variable gravity field model was modulated over time due to the relative drift of the nodes of the two pairs. Eventually, the orbit parameters in Table 1 were selected. Specific repeat patterns are no longer required, but the longitude shift of 1.3° per 7-day cycle (the gravity retrieval period) guarantees that, despite the differential nodal drift, the relative pattern of the two sets of ground tracks remains constant.

The attitude and the environmental disturbing accelerations will be controlled within a measurement bandwidth between 1 mHz and 100 mHz according to the requirements in Table 2. The formation of the two satellites will tend to drift under the action of differential air drag and differential accelerometer bias and dedicated controls, acting below the measurement bandwidth, will ensure that the mean semi-major axis remains within 100 m of nominal and the relative distance between 90 and 100 km.

Control domain	Item	Requirement
Formation	Altitude control range	100 m
	Satellite-to-satellite distance	100 km + 0% -10%
Drag-free control	Measurement bandwidth	1 ÷ 100 mHz
	Linear acceleration	$\leq 10^{-6}$ m/s²
	Linear acceleration ASD	$\leq 5\cdot10^{-9}$ m/s²/Hz$^{1/2}$
	Angular acceleration	$\leq 10^{-6}$ rad/s²
	Angular acceleration ASD	$\leq 10^{-8}$ rad/s²/Hz$^{1/2}$

Attitude Control	Satellite-to-satellite pointing	$\leq 2\cdot10^{-5}$ rad
	Satellite-to-satellite pointing error ASD	$\leq 10^{-5}$ rad/Hz^{½} ($1 \leq f < 10$ mHz) $\leq 2\cdot10^{-6}$ rad/Hz^{½} ($10 \leq f \leq 100$ mHz)

Table 2. Orbit, drag-free and attitude control requirements (ASD = amplitude spectral density)

The spacecraft propulsion enacting the DFACS and orbit control functions is the main challenge of the spacecraft design. The thrust range and modulation capability imposed by the mission, coupled with the lifetime requirement, can only be satisfied by electric propulsion. Electric propulsion trades propellant mass for electric power, and ~1 kW-level power generation is a demanding task in a mission which needs to keep the drag cross section small (below 1m²) with high seasonal and orbital variation of the solar aspect angle. Moreover, high specific impulse is only available over a limited thrust range and thrust demand varies by a factor ~3 to 5 both at orbit frequency and with epoch over the 11-yr solar flux cycle. This leads to using two different thruster types for different tasks, dubbed Drag Compensation Thruster (DCT) and Fine Control Thruster (FCT). Two DCT thrusters, operating in cold redundancy, provide for in-track drag control, orbit maintenance, formation acquisition and orbit maneuvers at thrust levels variable between 1 and 10 mN. Ten FCT thrusters, operating in the 50 µN to 1 mN range, take care of attitude control, pointing of the laser beam (thus avoiding the steering mirrors used in GRACE-FO) and cross-axis drag free control. The set is tolerant of one failure without any impact on performance, and allows graceful degradation after another failure. Requirements have been defined for each type of thruster, in terms of thrust throttling range, specific impulse, total impulse, response time, noise and beam divergence, under ceiling requirements on total propellant mass and total power demand. Thrusters meeting these requirements have already been demonstrated.

The spacecraft will be built on the heritage of platforms used in the ESA Copernicus program. The main modifications are in the solar array - to install the large panels, including deployed wings, needed by the electric propulsion - and in the internal layout - to make room for the interferometer core near the center of mass and for the laser beam baffles which run through the spacecraft body.

Figure 2 shows the launch configuration of an NGGM pair under the VEGA-C faring, with the two spacecraft aligned vertically on opposite sides of the launch vehicle dispenser, and the orbit configuration with deployed solar array. Four star-tracker heads view through the top solar panel, in a position minimizing the distance to the accelerometers and to the optical bench inside the spacecraft. Four heads are needed to compensate the variation of the occultation of the sky by the Earth, as the spacecraft is rotated seasonally around roll for the solar array to follow the sun; two trackers are active at any given time, as in GOCE.

The internal layout is dictated by the requirement that the optical reference for the inter-satellite distance measurement by the laser interferometer shall be placed in the center of mass and the two accelerometers close to and symmetric with respect to the center of mass. The temperature-sensitive parts of the payload including the accelerometer sensor heads, the optical bench assembly and the retro-reflector are accommodated in a dedicated temperature-controlled enclosure occupying the central volume of the spacecraft. The payload and service equipment boxes are accommodated on either side of the central bay, grouped according to function.

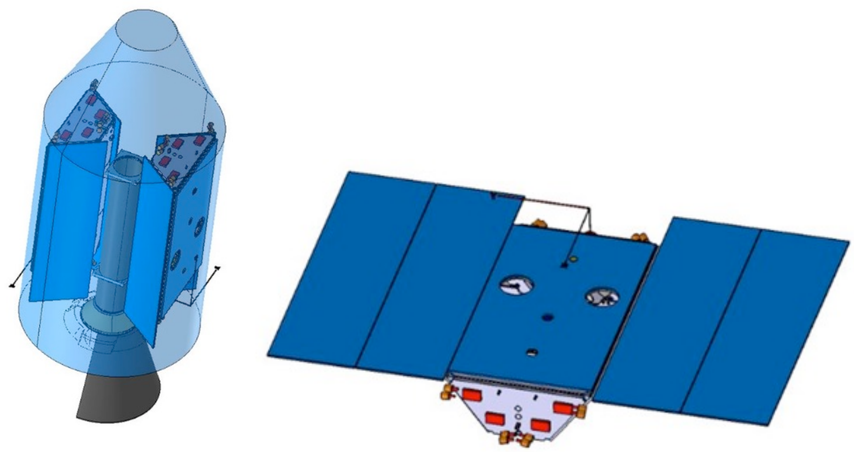

Figure 2. NGGM dual launch configuration under the VEGA-C fairing, and configuration in orbit.

III. The Indium FEEP Technology

Field emission of ions from a liquid is an effect that is closely tied to the presence of strong electric fields[8]. To achieve the electric field necessary for ion extraction, the fundamental structure on which field emission takes place is typically shaped like a needle, to utilize the field-enhancing effect at the tip. An important application of this effect is the so-called 'Liquid Metal Ion Source' (LMIS), using the process of field emission to ionize a thin film of liquid metal covering a needle, which is biased to a few kV with respect to a counter electrode. The thusly created ions are then accelerated by the strong electric fields and can be used for ion implantation in semiconductor industry or micromachining in a focused ion beam.

The emission sites can be solid needles, capillaries, or porous needles, all having in common that the liquid metal is transported to a very sharp tip via capillary forces. Applying an electric field to this sharp tip causes the formation of the so-called Taylor cone[9]. At the tip of the Taylor cone, neutral atoms of the liquid metal evaporate from the surface and due to the strong electric field one or more electrons tunnel back to the surface, changing the formerly neutral atom to a positively charged ion.

The sharper the emission site, the smaller is the base of the Taylor cone, leading to a higher efficiency of the thruster at any given ion current. Figure 3 shows three different ways of manufacturing small emission sites. On the left, a solid needle is covered with a film of indium. This technology is particularly attractive in terms of performance, but is also very prone to contamination or any effects that could compromise or disrupt the indium film. The capillary emitter shown in the center of Figure 3 has the advantage that it is very resistant to contamination and the manufacturing is very reliable, but the exit of the capillary cannot be manufactured as small as a needle tip, leading to a slightly lower specific impulse if used as a thruster.

This principle of creating positive ions and accelerating them by the very same electric field can also be used to generate thrust. When a liquid metal ion source is used in this fashion, it is termed 'Field Emission Electric Propulsion' (FEEP). Due to the feasible precision at which the voltage between the needle and the extraction electrode can be controlled, the generated thrust can be controlled with unmatched accuracy. In addition to precise throttling, FEEP thrusters are capable of generating thrust over a wide range, from the sub-µN level to several tens of µN per emission site.

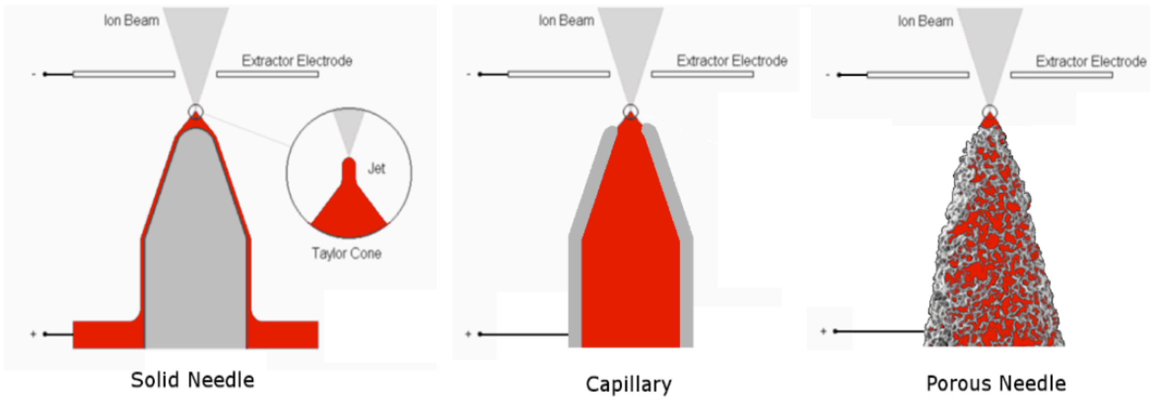

Figure 3. Different types of emitter technologies for LMIS

A. The Porous Tungsten Crown Emitter

In 2008, the development of the so-called porous tungsten crown emitter has been started, which constitutes a porous matrix with 28 needles for field emission[9] (see crown A39 in Figure 4). Apart from multiple emission sites, the most important new feature is the porous tungsten matrix which enables internal flow of the liquid metal (similar to a capillary emitter) to a very sharp tip (similar to a solid needle). It therefore combines the advantages of both the capillary and the solid needle emitter.

When high voltage is applied to the crown emitter, the needles whose onset voltage lies below this voltage emit ions. At high thrust levels, most of the needles are contributing to the total current, whereas at low thrust levels only a couple of needles (whose electrical impedance is lowest) are emitting ions.

The porous tungsten crown emitter is manufactured in several steps. These include: powder injection molding, sintering, electrochemical etching of the needle tips and wetting with indium. An accurate control of the process parameter in all of these steps is required to arrive at a final geometry that has 28 needles with the desired porosity and sharpness. Nevertheless, the desired feature size is so small (tip radii of 1 µm circa), that stochastic processes in the manufacturing chain start to play an important role. This inherent characteristic of the crown emitter is in fact an advantage: an ideal crown that only features very sharp needles of 0.5 µm would have a low starting voltage, but a high impedance. An ideal crown with only blunt needles of 3 µm on the other hand would have low impedance, but high starting voltage. Details on the manufacturing processes and statistical distribution of the needle tip radii up to 60 emitters, together with firing homogeneity tests (an example is given in Figure 5), can be found in Ref. 9 and in the ESA activities of Ref. 10.

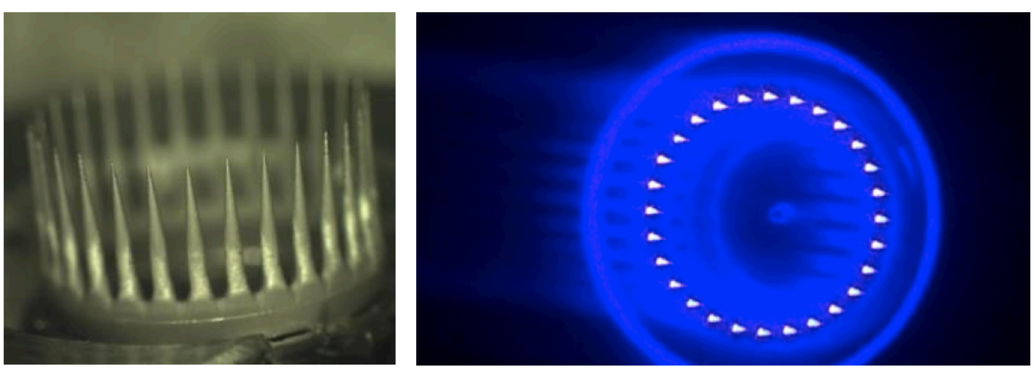

Figure 4. Porous tungsten crown emitter A39 after wetting with indium (left), during high-thrust operation (right).

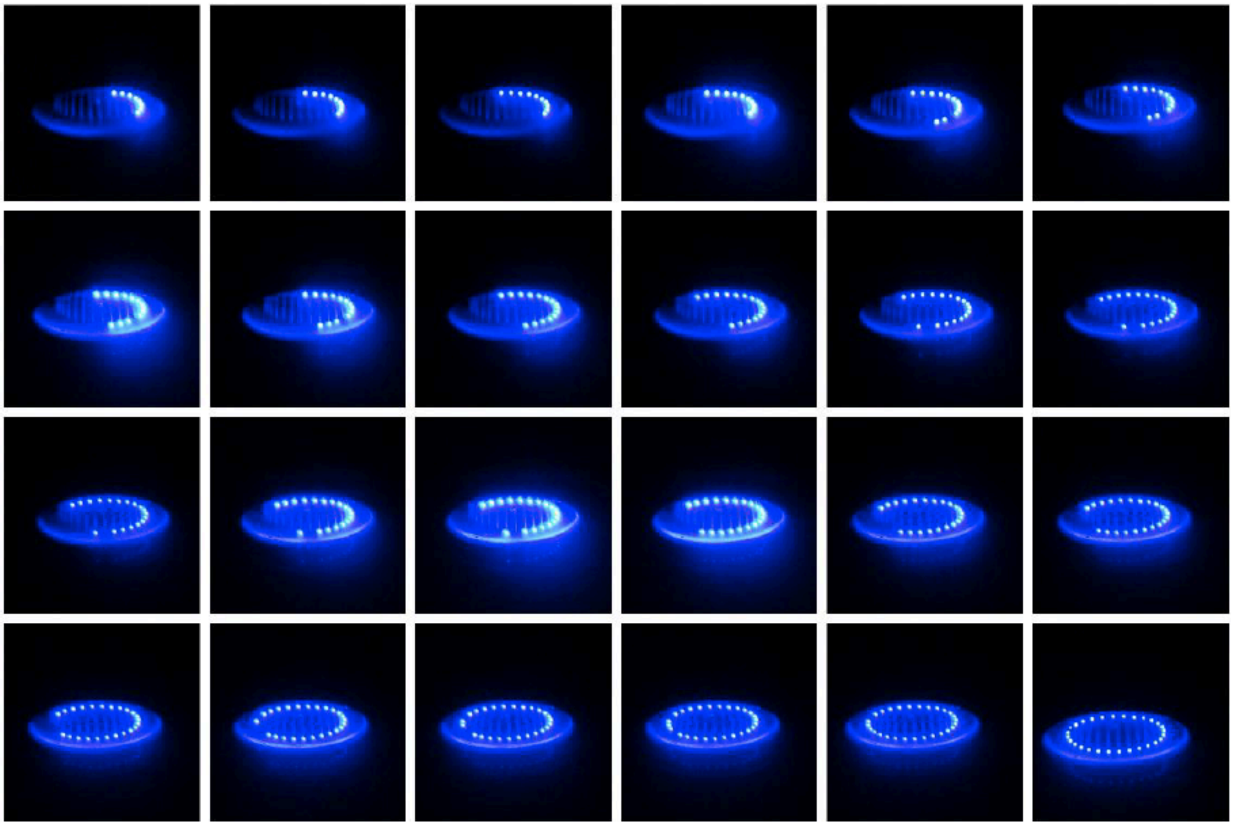

Figure 5. Firing homogeneity test. Firing needles stimulate their neighbors to fire as well, leading to the typical intermediary arc-like pattern (from top left to bottom right).

B. Thruster Modules

The high flexibility of the porous tungsten crown emitters introduces a variety of possible applications. So far, two different thruster modules have been designed, manufactured and have undergone extensive testing, ad depicted in Figure 6. The **IFM 350** thruster was developed within preparatory studies of the ESA Next Generation Gravity Mission (NGGM), in Ref. 11. With advanced beam focusing capability and additional spacing between high-voltage components, the module in Fig. 6-left is intended to be used for high reliability, long-term operation.

In 2014, a new thruster family has been designed and tested, building on the heritage from the development of the IFM 350 thruster. The so called **IFM Nano thruster** (in Fig. 6-right) uses the same porous tungsten crown emitter technology without the need for miniaturization, while a new compact power supply was developed in order to provide a complete propulsion system of less than 1 dm^3 volume (< 1 U) and less than 1 kg mass. This envelope includes all sub-systems (propellant reservoir, emitter and Power Processing Unit (PPU)). The thruster was demonstrated in orbit, for the first time ever, in January 2018.

Figure 7 shows the modular design concept of the IFM Nano thruster on his commercial versions, using parts qualified not only for space but also for automotive. Each module includes the emitter itself, the propellant reservoir with integrated heater, the power processing unit with high voltage electronics and two redundant neutralizers. As the thruster uses indium as propellant, the metal is solid at room temperature and also during launch. Once in orbit, the propellant is heated above its melting point of 156.6 °C and becomes liquid. Avoiding any liquid and reactive propellants as well as pressurized tanks during integration and launch significantly simplifies all the corresponding procedures. Electrical interfaces between the IFM Nano thruster module and the spacecraft are reduced to a low voltage bus (12 V and 28 V) and a digital communication interface (RS-422 or RS-485).

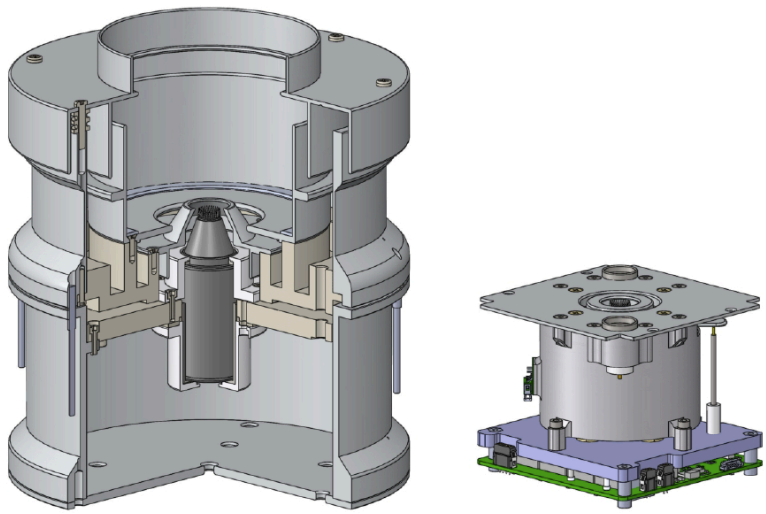

Figure 6. The IFM 350 (left, for NGGM) and the IFM Nano (right, for New Space market) thruster modules, using the *same* porous tungsten crown emitter.

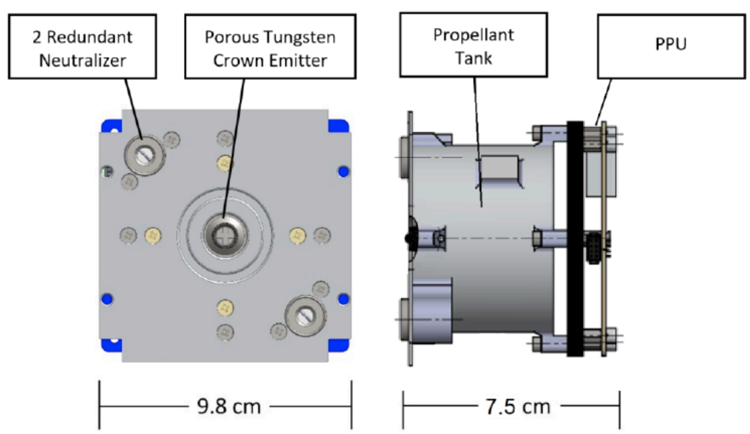

Figure 7. Highly integrated design concept of the IFM Nano thruster family

C. Power supply for NGGM and Characterization of Cathode Emitter

Additionally, in the activity in Ref. 11, other important activities were performed for optimizing the design of the IFM 350 thruster, targeting the challenging requirements for the lateral and attitude control of the NGGM mission. In brief, the following tasks were executed:

- specification of requirements for a NGGM Power Supply
- assessment of Electromagnetic Compatibility and Interference
- characterization of a cathode emitter for mN-FEEP
- simulation of minimum energy surface of propellant in the tank
  trade-off of tank size
- design review of the emitter module
- experimental performance evaluation of the thruster module.

Since the power per thrust ratio *PTR* of a FEEP thruster exclusively depends on the emitter voltage, the requirement regarding the specific power can be reformulated in an emitter voltage requirement:

$$PTR = \frac{VI}{fI\sqrt{2V\frac{m}{e}}} = \frac{1}{f}\sqrt{\frac{e}{2m}V} \tag{1}$$

where *f* is the total thrust correction, as a product of the divergence and multiple charged species terms.

Based on the value of the emitter voltage corresponding to a PTR of 40 W/mN, as per the requirements, the highest voltage which the power supply needs to be able to output during continuous operation is in the range of 3000 V under ideal circumstances. In this configuration, the emitter voltage would be adjustable between 0 - 3000 V and the emission current, and thus thrust, is controlled via the extractor bias.

For *initial startup*, however, experience has shown that a larger difference in voltages between emitter and extractor is required. The additional force necessary to initiate emission is related to an increased surface tension from the coverage of Indium with a thin oxide layer after exposure to atmosphere. Furthermore, upon initial ignition, the liquid metal first needs to be pulled out of the pores, where it retracts when the thruster is deactivated and cold; this adds to the initial voltage requirement. In practice, it is advisable to have the capability of generating a voltage difference of 20 kV between emitter and extractor in order to reliably ignite the thruster, in particular after storage in air. In order to have sufficient voltage reserves on the emitter side, it is suggested to have a maximum voltage capability of at least +10 kV on the power supply of the emitter. This high voltage level would only have to be applied for short durations (i.e. tenths of seconds, repeatedly) to provide sufficient pulling force to the liquid propellant in order to deform it into the characteristic cone-shaped tip necessary for ion emission.

During this assessment, it was found that designing the power supply with these reserves regarding emitter voltage Min voltage of 0 V, Max voltage of +10 kV (burst), +3 kV (continuous, optimal case) also provides a safety margin to cope with events that may only appear after prolonged use (i.e. several thousand hours), such as a potential change in firing characteristics.

Another important task was the survey on available electron source technologies for neutralizing the ionized beam, with European and worldwide suppliers. The different technologies were rated and compared with respect to criterions like their vacuum requirements, their efficiency, and their cost: it becomes clear that different applications require different cathodes. For life time testing of Ion emitter, a very interesting cathode is the *dispenser cathode*, since it can emit high currents of electrons for tens of thousands of hours in good vacuum environments. It consists of a strongly-bounded, continuous metallic phase of a refractory meal (or metals), interspersed uniformly with the emitting material (the impregnant). The porous metal matrix acts as a reservoir from which the emitting material can diffuse to the surface, maintain an active layer and, consequently, provide a low work-function surface for the thermionic emission of electrons. In LEO environment, the best choice is cathodes very stable in bad vacuum conditions and having a very low work function, which makes them the most efficient technology available, but with a limited lifetime of a few hundred hours.

Another activity -under ESA contract in Ref. 12- was the investigation of storage requirements: having a large number of emitters available, they were stored in different environmental conditions (e.g. in high humidity) and then they were tested, in order to see how they perform after (at least) 6-month storage time. Meanwhile, the start-up performances were consolidated under the activity in Ref. 15, where the reproducibility of the wetting procedure was assessed, together with a preliminary analytical model to predict the crown emitter performance starting from the distribution of the needle sharpness.

IV. Endurance Testing of the IFM Module

The above-mentioned crown A39 was selected for further endurance testing, up to end of life. Several ESA contracts have been supporting this extensive testing, as briefly explained hereafter.

A. 10000 Hours Testing

Under ESA Contract in Ref. 13, the crown emitter A39 has undergone over 10000 hrs of lifetime testing. The publication in Ref. 14 summarizes the main results of the laboratory tests, from which Figure 8 shows the current and voltage evolution of the mN-FEEP thruster. While at the beginning of the test the impedance started at a high value of 4 MΩ, the value decreased in time to 2.7 MΩ after a period of 4,000 h. At 4200 hrs, a short circuit between the extractor and the emitter occurred due to the shrinking of the extractor. After removing the indium and restarting the thruster, the initial impedance of 4 MΩ could be re-established. This shows that there was no degradation of the emitter itself after 5000 hrs of testing. The same behavior was observed between 5000 and 7500 hrs of testing, followed by a return to the initial impedance at 7500 hrs, after the extractor was refurbished again.

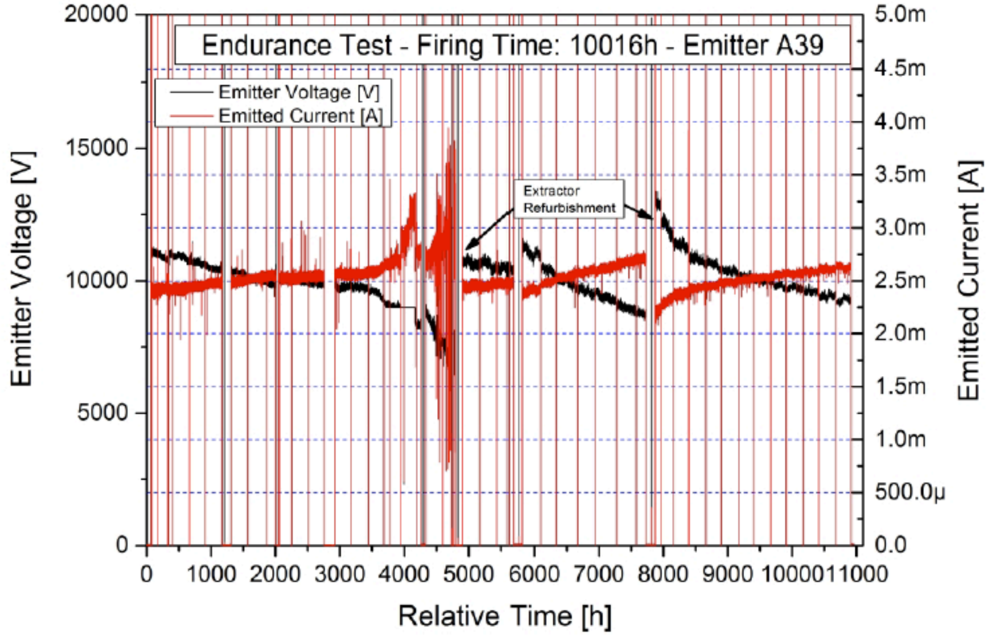

Figure 8. Current and Voltage Evolution over 10000 hrs of testing

The specific impulse Isp and mass efficiency evolution were monitored during the lifetime testing, as shown in Fig. 9. It can be seen that the mass efficiency increases due to the lower voltages as the extractor diameter decreases. Note that the mass efficiency after 10000 hrs is not reduced with respect to the initial values and is still well above 30%.

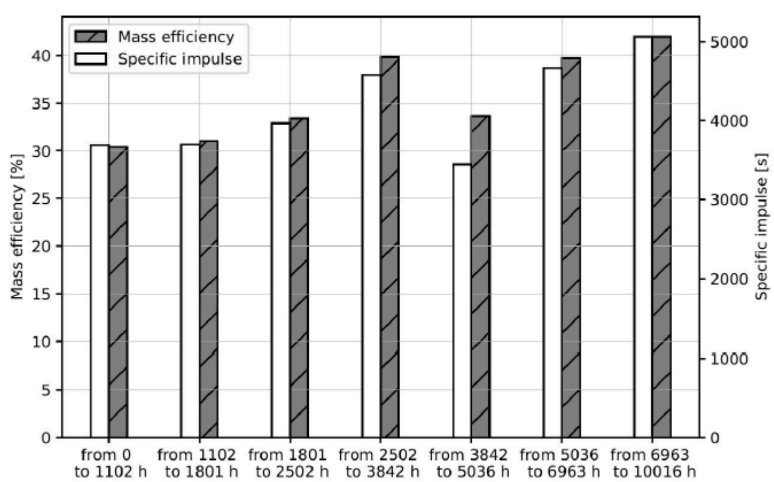

Figure 9. Evolution of Isp and Mass Efficiency over 10000 hrs of testing

Under observation it was the evolution of the extractor diameter during the endurance testing. From Fig. 10, it is evident that the extractor diameter is steadily shrinking due to indium deposition up to 2500 hrs. Between 2500 and 3842 hrs, the shrinking of the extractor starts to become uneven: the reason is that the distance between the extractor and the emitter becomes small enough so that a slight deviation has a large effect on the distribution of the ion current among the needles. This, in turn, results in an increased current and droplet emission from the needles closest to shrinking extractor rim: the effect can clearly be seen almost 1000 hrs before an actual short circuit. After the short circuit, the extractor was replaced by a new one and it is recognizable that indium starts to be deposited at about the same rate as at the beginning (see, for more details, Ref. 13 & 14).

If no countermeasures are taken, this deposition of indium is limiting the lifetime of the thruster. However, contrary to the erosion processes being observed in other thruster technologies, this deposition of indium does not damage any part of the thruster electrodes and is completely reversible. Several countermeasures are under investigations to treat this deposition: the results of this investigation are object of a patent and future publications.

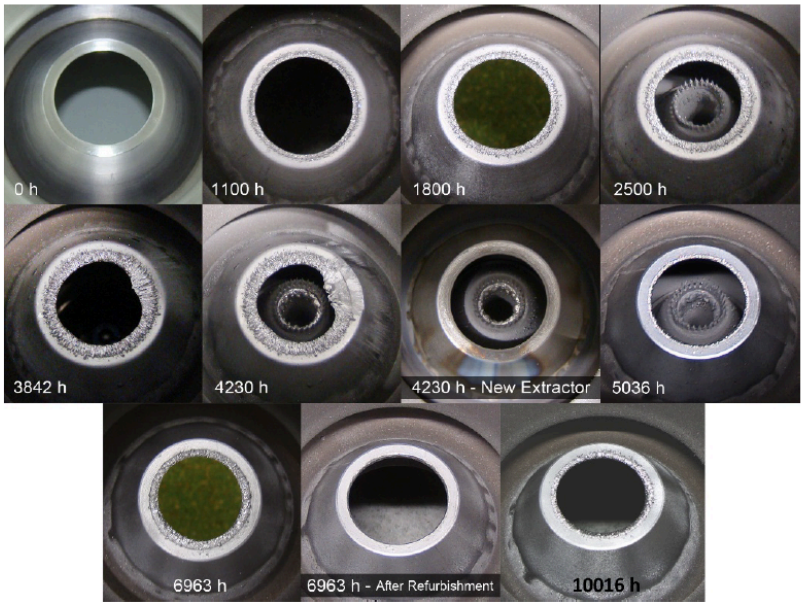

Figure 10. Change of the Extractor Diameter as a result of indium Deposition.

B. 24000 Hours Testing

The porous liquid indium ion emitter with label A39, was tested up to 10000 hrs at a thrust level of 350 μN without any significant performance degradation. But several important findings have emerged from the endurance test:

- the emitter has showed no clear signs of performance degradation over time: in fact, after the endurance test the entire emitter was still firing at good performance and without a clear indication on the lifetime limiting factors, which makes it a promising candidate for a propulsion system of long-term, continuous firing and high total impulse missions;
- shrinking of the extractor diameter due to the "slow" Indium deposition (treated in the section above);
- the needles of the emitter were found to exhibit a slow change in tip shape (as in Fig.11), but with no measurable impact on performance. The process behind this deformation is unknown, but the erosion rate slowed down over time;

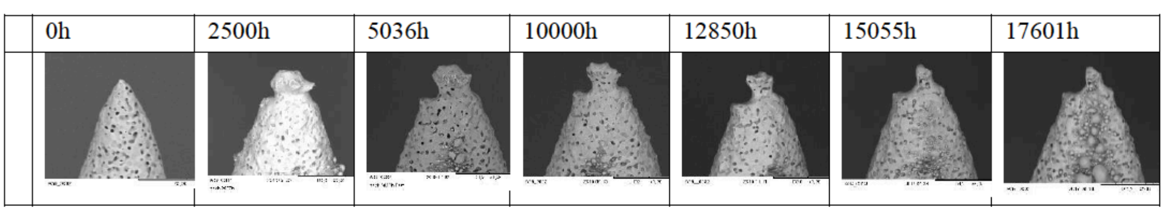

Figure 11. Tip deformation over time

- the impedance is significantly increased due to the formation of a contamination layer on the tips, at each restart after that the emitter is taken out and exposed to air. This effect is relevant only for terrestrial application.

Therefore, this activity[16] has aimed at finding the limiting factors of this propulsion system, extending the endurance test (and related inspections and analysis) firstly to 16000 hrs, and then to 24000 hrs at the beginning of May 2019. The processing and analysis of the test data are undergoing, but the preliminary promising results are shown in Figure 12: as it can be seen, at the end of the test, the emitter voltage at nominal thrust has highly increased, reaching the 18 kV, but with a general positive effect on a stable (or slightly increased) mass efficiency and very high Isp. At the end of the test, the number of firing needs was less than half, around 11-12.

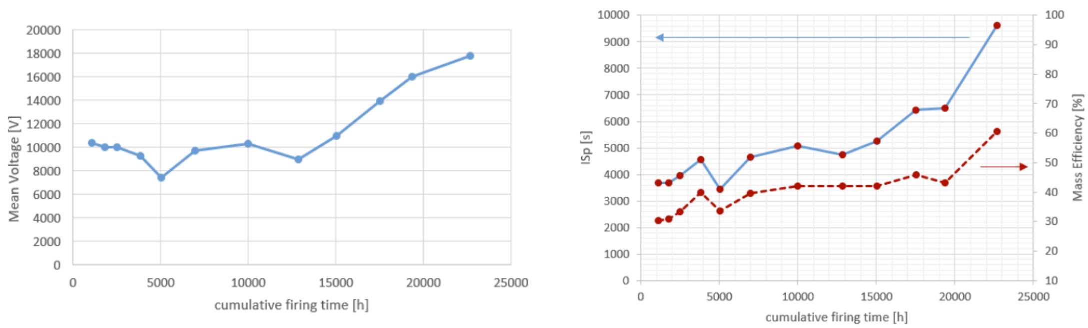

Figure 12. Extractor voltage, mass efficiency and Isp evolution of the A39 crown during the 24000 hrs endurance test.

C. Extension of the Endurance Testing until EOL

The porous liquid indium ion emitter with the label A39 was tested for a cumulative time of more than 24000 hrs at a thrust level of 350 µN, but without reaching an EOL event within the project core contract in Ref. 16. Thus it is proposed to extend this lifetime up to 30000 hrs or until an EOL event, whichever will occur first. At the time of writing, the activity is in preparation.

V. Application to EOP Future Science Mission: NGGM

A low-thrust electric propulsion system with variable thrust is one of the key technologies for the realization of the NGGM mission. Summarizing the mission layout in Section 2, such EP system can enable:

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

Electric propulsion systems capable of combining high specific impulse and ultra-precise controllability have been identified as enabling technology for such a mission. This is considered to be also the case for future science missions like LISA, which seek to make use of the possibility of controlled formation flight of several spacecraft. In Table 3 the compliance of the Indium mN-FEEP thruster with the preliminary requirements of the NGGM mission for lateral controllability and attitude control, is presented, showing to be a promising back-up technology for the mission.

Parameter	Requirement for lat/att. thrusters	Compliancy of the IFM 350
Minimum Thrust	0.05 mN	Fully Compliant
Maximum Thrust	< 3 mN	Fully Compliant
Thrust Resolution	0.5 µN	Emitter Compliant [*]
Thrust Noise	<1µN/√Hz above 0.08Hz	Emitter Compliant [*]
Rise/Fall Time	< 50 ms	Emitter Compliant [*]
Slew Rate	> 0.5 mN/s	Emitter Compliant [*]
Update command rate	10 Hz	Emitter Compliant [*]
Thrust non-linearity	< 2%	Emitter Compliant [*]
Lifetime	> 10 yrs	24000 hrs demonstrated
Specific Power	< 40 W/mN	Fully Compliant

[*] *Full Compliancy depends on the performance of dedicated power electronics that has not yet been manufactured. Nevertheless, compliancy has already been demonstrated with LISAPathFinder In-FEEP power electronics.*

Table 3. Compliance of the Indium mN-FEEP thruster with the NGGM requirements.

VI. Conclusion

The EOP Directorate of the European Space Agency has contributed and supported the design of a variable specific impulse electrostatic thruster based on Field Emission Electric Propulsion (FEEP). This emitter technology has been developed and qualified over more than a decade at FOTEC and the Austrian Institute of Technology under ten ESA/EOP contracts since 2005 targeting the *Next Generation Gravity Mission* needs, and it has recently been adapted for the use as main propulsion system in Nano- and Small-satellites. This paper has illustrated the NGGM mission concept, together with the thruster development activities, the assessment of the performance of this thruster technology and its suitability as a back-up propulsion for lateral and attitude control of the NGGM satellites, after extensive simulation and testing campaigns.

References

[1]The ESA Earth Observation Programmes Activities for the Preparation of the Next Generation Gravity Mission, *AIAA Guidance, Navigation, and Control Conference 2013*, Boston, Massachusetts, 19-22 August 2013, DOI: 10.2514/6.2013-4637

[2]Bender, P.L., Wiese, D.N., Nerem, R.S., "*A Possible Dual-GRACE Mission With 90 Degree And 63 Degree Inclination Orbits*", ESA SP-654, Proceedings of the 3rd International Symposium on Formation Flying, Missions and Technologies, ESA-ESTEC, Noordwijk, The Netherlands, 23-25 April 2008

[3]Reubelt, T., Sneeuw, N., Iran Pour, S., Pail, R., Gruber, T., Murböck, M., Daras, I., Visser, P., Texeira de Encarnação, J., van Dam, T., Weigelt, M., Cesare, S., Cornara, S., "*The ESA project SC4MGV Assessment of Satellite Constellations for Monitoring the Variations in Earth's Gravity Field – overview, objectives and first results*", Proc. 14th EGU General Assembly, Geophysical Research Abstracts 16 (2014)

[4]Daras I., Visser, P., Sneeuw, N., van Dam, T., Pail, R., Gruber, T., Tabibi, S., Chen, Q., Liu, W., Tourian, M., Engels, J., Saemian, P., Siemes, C., Haagmans, R., "*Impact of orbit design choices on the gravity field retrieval of Next Generation Gravity Missions - Insights on the ESA-ADDCON project*", Proc. 19th EGU General Assembly, Geophysical Research Abstracts 19 (2017)

[5]Floberghagen, R., Fehringer, M., Lamarre, D., Muzi, D., Frommknecht, B., Steiger, C., Piñeiro, J., da Costa, A., "*Mission design, operation and exploitation of the gravity field and steady-state ocean circulation explorer mission*", J. Geodesy 85, 749–758 (2011).

[6]Tapley, B. D., Watkins, M. M., Flechtner, F., Reigber, C., et al., "*Contributions of GRACE to understanding climate change*", Nature Climate Change 9, 358–369 (2019).

[7]Kornfeld, R. P., Arnold, B. W., Gross, M. A., Dahya, N. T., Klipstein, W. M., Gath, P. F., Bettadpur, S., "*GRACE-FO: The Gravity Recovery and Climate Experiment Follow-On Mission*", J. Spacecraft and Rockets 56, 931–951 (2019).

[8]G. Taylor, "*Disintegration of Water Drops in an Electric Field*" Proc. R. Soc. Math. Phys. Eng. Sci., vol. 280, no. 1382, pp. 383–397, Jul. 1964.

[9]I. Vasiljevich, "*Design, development and testing of a highly integrated and up-scalable FEEP-multi-emitter using indium as propellant*" PhD Thesis, TU Vienna, Vienna, Austria, 2010.

[10]"Consolidation of microPIM Technology for mN-FEEP Thruster", Final report, ESA Contract No. 22418/09/NL/JA, May 2011.

[11]"System Level Study of the mN-FEEP for NGGM and other science missions", Final report, ESA Contract No. 4000105680/12/NL/Cbi, April 2014.

[12]"Investigation of Storage Requirements", CCN Final report, ESA Contract No. 4000109581/14/NL/FF/LF, 2016

[13]"Extension of the Lifetime Testing of the mN-FEEP technology for NGGM", ESA Contract No. 4000121521/17/NL/FF/mg, 2017.

[14]Reissener A., Buldrini N., Seifert B., Hörbe T., Plesescu F., Gonzalez del Amo J., Massotti L., "*10000 h Lifetime Testing of the mN-FEEP Thruster*", *Proc. of* AIAA JPC (Propulsion Energy Conference) 2016, Salt Lake City, UT, 25-27 July 2016

[15]"Consolidation of the mN-FEEP Start-Up Performance", Final Report, ESA Contract No. 4000109581/NL/FF/lf, September 2016.

[16]"Continued Lifetime Testing to 24000 h and Contamination Mitigation", CCN1 to ESA Contract No. 4000121521/17/NL/FF/mg, June 2018.