DEVELOPMENT OF THE MICROTHRUST BREADBOARD: A MINIATURIZED ELECTRIC PROPULSION SYSTEM FOR NANOSATELLITES

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Flavia Tata Nardini†, Berry Sanders and Michiel Straathof
TNO, Rijswijk, the Netherlands

Çağlar Ataman, Muriel Richard and Herbert Shea
Ecole Polytechnique Fédérale de Lausanne, Switzerland

Pelle Rangsten and Ana Salaverri
NanoSpace, Uppsala, Sweden

Charles Ryan and John Stark
Queen Mary University of London, U.K.

Richard Visée
SystematIC, Delft, the Netherlands

NOMENCLATURE

AU	Astronomical Unit
BBM	Breadboard Model
BBTS	Breadboard Test Setup
CDF	Concurrent Design Facility
CFA	Capillary Feeding Assembly
CPCB	Central Power & Control Board
EIFB	Electric Interface Board
EMI-BF_4	1-Ethyl-3-Methylimidazolium Tetrafluoroborate
EPFL	Ecole Polytechnique Fédérale de Lausanne
ESA (1)	European Space Agency
ESA (2)	Extractor Switch Assembly
FP7	Seventh Framework Programme
HV	High Voltage
INTA	Instituto Nacional de Técnica Aeroespacial
LV	Low Voltage
MEMS	Microelectromechanical System
MEX	Mars Express
MSA	Mechanical Structure Assembly
MT	MicroThrust
NEA	Near-Earth Asteroid
NEO	Near-Earth Object
OLFAR	Orbiting Low-Frequency Antennas for Radio Astronomy
PCS	Power & Control System
QMUL	Queen Mary University of London
SEM	Scanning Electron Microscope
SME	Small and Medium Enterprises
TBA	Thrust Board Assembly
THC	Thruster Chip
TMS	Thruster Module System
TNO	Nederlandse Organisatie voor Toegepast-Natuurwetenschappelijk Onderzoek
TOF	Time-of-Flight
TRL	Technology Readiness Level

†E-mail: flavia.tatanardini@tno.nl

ABSTRACT

Since 2008, the MicroThrust (MT) consortium consisting of EPFL, Nanospace, QMUL, SystematIC, and TNO has been working on the development of a MEMS based electric propulsion system [1]. Since 2010, the work has been performed as part of the European Union's FP7 programme with the goal to design, build and test an engineering model of such a system.

The engineering model shall show that this propulsion system can fit in a nano-satellite in terms of mass, volume and power consumption, while giving the satellite a very large ΔV capability (up to 5 km/s). These requirements can only be met by extreme miniaturization and integration of all components.

As a first step towards the engineering model, a laboratory breadboard model is currently under development. This paper starts with an analysis of a range of different mission scenarios that the MT could perform, to make sure that the mission requirements are clearly understood. It then describes the working principles and some of the design choices behind the different components of the breadboard.

The project is now well underway, and some of the breadboard components are starting to take shape. Testing of the complete breadboard system is scheduled for 2013.

INTRODUCTION

Over the past decade, the small satellite market has grown considerably. Since 2000, launch demand for < 100 kg satellites has increased by over 4% per year, with an expected growth of over 22% each year expected in the near future [2]. The main advantages of small satellite systems are their low cost and short development time. However, their small size also comes with limitations, such as the lack of a small and efficient propulsion system. Without a means of propulsion, satellites are severely restricted in the types of missions they can perform. As part of the European FP7 initiative, the MicroThrust (MT) consortium was established to design, develop and test such a propulsion system. The MT consortium consists of EPFL of Switzerland, NanoSpace of Sweden, QMUL of the U.K. and SystematIC and TNO of the Netherlands.

The availability of a micro propulsion system would facilitate low cost science missions for Earth observation, whilst also enabling exploration of the solar system. The MicroThrust team, consisting of partners from academia, research institutions, SME's and industry from four European countries, has developed a conceptual design of a very small, yet highly performing electrical propulsion system. The system is based on a MEMS colloid thruster and has a high degree of subsystem integration.

In order for the system to reach TRL 5, a laboratory breadboard model is being developed, which will demonstrate the functionality of the complete system. It includes a miniaturized 4 kV HV power supply, arrayed MEMS ion emitter chips, microfluidics for ionic liquid handling based on capillary forces, packaging, detailed mission analysis and experimentally validated operating points.

Using the MicroThrust system, satellites will be capable of performing missions ranging from Earth observation to interplanetary exploration. This paper will start with an analysis of the missions that the MT system could perform and ends with a detailed description of the breadboard and its components.

MISSION ANALYSIS

In parallel to the development of the MT breadboard system, a mission analysis is performed to address the FP7 needs for exploration missions.

The objective of the mission analysis is to ensure that real mission requirements are understood and implemented in the design of the MT propulsion system. The goals set for this activity were to perform an initial mission survey (needs), understand nano-satellite constraints, and derive preliminary propulsion system specifications. These tasks involved the EPFL Space Center (analysis) with inputs from EPFL LMTS (thruster design support), TNO (system support) and QMUL (thruster performance support). To achieve these goals, several tasks were laid out. First the definition of mission destinations and characteristics, second the understanding of satellite constraints, third the analysis and fourth the derivation of the requirements.

Mission Destinations

A list of most promising exploration missions was set up, based on flown electric propulsion missions and identified interests from ESA and small satellite mission publications. It was found however that there isn't a wide breadth of exploration missions proposed for nano-satellites as their main show stoppers are propulsion and power requirements. The approach has since been to investigate the possibilities and propose several destinations as targets of interest. Thus, a preliminary set of destinations was chosen to cover the most likely pertinent scientific data needs. This set includes:

1) A mission to the Moon;

2) A mission to a Near-Earth Object;

3) A mission to a Lagrange point;

4) A mission to Mars.

The Moon is an obvious destination for small satellite missions. A good example of such mission is the OLFAR mission proposed by 3 institutions in the Netherlands [3]. In this proposal, an array of individual nano-satellites would create a network of radio telescopes based around the Moon. The nano-satellites would be utilizing 3-Unit CubeSat platforms.

A second destination of choice is a near-Earth object. A near-Earth object (NEO) is a Solar System object whose orbit brings it into close proximity with the Earth. All NEOs have a perihelion distance less than 1.3 AU. A typical example of a NEA mission is the proposed MarcoPolo-R mission [4]. MarcoPolo-R has been selected for the assessment study phase of ESA M3 missions in February 2011. Although this mission calls for a rather large spacecraft, it shows the interest of the science community for NEOs. Precursor missions could be flown to these destinations with nano-satellites.

The Earth-Sun Lagrange points are also potential candidate mission destinations. The L1 point is suitable for Sun observations while the L2 point is suitable for cosmology studies.

Finally, a nano-satellite mission to Mars has been evaluated under the ESA NEOMEx studies. One of these studies, called MiniMEX [5, 6] was performed by ESA to assess the benefits of miniaturized technologies. The goal of MiniMEX was to demonstrate that the Mars Express (MEX) science objectives can be met in full by using a microsystem-based spacecraft and that such an approach to the spacecraft design may even result in the ability to fly more payloads. The propulsion system on MiniMEX assumed chemical propulsion delivering a total ΔV of 1.57 km/s. The operations were split in Mars capture maneuvers, mid-transit corrections, and Mars orbital maneuvers. Other missions to Mars have been proposed that use electric propulsion and allow encounters with Phobos and Deimos [7].

It is believed that such a suite of mission destinations is appropriate for the development of the MicroThrust propulsion system. Note that most of these destinations have aphelion below or at around 1.7 AU.

For the analysis presented in this paper, the approach has been to summarize the trajectory characteristics of 4 flown electric propulsion missions [8 – 13], and of one to fly in 2014 [14, 15]. The results are provided in Table 1. For further use in a first order approximation, two parameters are important: 1) the initial acceleration levels, which will provide a magnitude of the thrust needed to make low-thrust trajectories to the target feasible; 2) the trajectory ΔV, which is a measure of the energy that the propulsion system needs to reach the final destination. These two parameters are highlighted in Table 1.

Note that for these missions, a propellant mass fraction of about 15%-35% is found, together with a typical propulsion system dry mass of 10% (does not include the solar array mass).

Table 1: Mission characteristics of 4 flown missions and one still to fly (BepiColombo).

Mission		DS-1	SMART-1	DAWN	HAYABUSA	BEPI COLOMBO
Mission target		Main-belt asteroid	Moon	Main-belt asteroid	Near-Earth asteroid	Mercury
Target semi-major axis	AU	Br: 2.3, Bo: 3.6	-	V: 2.36, C: 2.8	1.3	0.4
Target peri- & apohelion	AU	Br: 1.3 - 3.4 Bo: 1.3 - 5.8	-	V: 2.1 - 2.6 C: 2.5 - 3	0.9 - 1.7	0.3 - 0.5
Launch mass	kg	486.3	369	1240	510	4200
Dry mass	kg	412.3	287	790	444	
M_propellant	kg	74	82	450	66	560
I_sp, initial	s	3174	1640	3174	3200	4300
P_in ppu	W	2110	1200		~2400	10600
Max. thrust	mN	78	68	78	83	290
Trajectory DV	**km/s**	**4.3**	**4.0**	**~14**	**~4**	**5.8**
Initial calculated acc.	**mm/s2**	**0.16**	**0.18**	**0.06**	**0.16**	**0.07**
Time-of-Flight	yrs	1.9	0.57	~8	~3 - 4	6.6
Initial trajectory state		C3 slightly positive	GTO	C3 slightly positive	Positive C3	Positive C3
Final orbit		Asteroid encounter + comet	Moon polar	Vesta + Ceres orbits	Sample return from Itokawa back to Earth	Mercury
Propellant mass fraction		**15%**	**22%**	**36%**	**13%**	**13%**
Prop. system dry mass	kg	48	29	129		1390
Prop. system dry mass fraction		**10%**	**8%**	**10%**		**33% (separate module)**

Satellite Constraints and Models

Similarly, a survey of nano- and micro-satellites, existing and in concept was performed to derive reasonable power, mass and volume resources. Contacts were established with two nano-satellite builders, INTA (Spain) and OHB (Germany) for further design investigations. As these existing small satellites were not designed for exploration missions, extrapolations had to be made based on existing data.

Thus, to address the need for robotic exploration, a set of satellite "representative" models has been established in order to provide constraints, restraint the trade space and inject reality into the analysis assumptions. The satellite models established for the analysis are the CubeSat 3U (10x10x30 cm^3), NanoSat 8 kg (20x20x20 cm^3), MicroSat 27 kg (30x30x30 cm^3) and MicroSat 64 kg (40x40x40 cm^3). These models are based on the survey of developed or flown satellites and their resources. However, as none of the flown satellites were intended for space exploration, the models have been adapted for this new mission aim. For instance, it was assumed that the MicroSats have deployable wing solar arrays, and that the CubeSat 3U and NanoSat 8 kg have 2 deployable side panels (similar to Delfi-C3). Without an increase in the solar array area compared to the volume of the cube itself, there would not be enough power to propel the satellites to exploration destinations. All 4 satellites have the ability to point towards the Sun and stay pointed for large amounts of time. Table 2 summarizes the main satellite resources. At this point, they are simplified models, and further investigations will refine them.

Table 2: Satellite models and power assumptions for the analysis.

Parameter	CubeSat 3U	NanoSat 8 kg	MicroSat 27 kg	MicroSat 64 kg
Mass (kg)	3	8	27	64
Dimensions (cm3)	10 x 10 x 30	20 x 20 x 20	30 x 30 x 30	40 x 40 x 40
Solar panel dimensions (cm2)	10 x 30 + 2 deployable side panels 10 x 30	20 x 20 + 2 deployable side panels 20 x 40	2 wings 30 x 60	2 wings 40 x 80
Power produced at 1 AU (approximate 19% EOL efficiency) (W)	~ 12 average	54	94	168
Power available to propulsion system, 80% of 1 AU power (W)	~ 9.5	42	75	134

Analysis Results

Based on the findings in Table 1, and as a first approximation for this preliminary round of mission requirements, a target ΔV of 5 km/s was chosen. Similarly, a target initial acceleration of 0.1 mm/s^2 was assumed.

The propellant mass for a total ΔV of 5 km/s can be calculated as a function of specific impuls (I_{sp}). To keep the overall mass of the propulsion system low, a target total propulsion wet mass of less than 25% of the launch mass is set (see for reference Table 1). Also taking into account an estimated total dry propulsion mass of 100 g per 1 kg of launch mass (10% mass fraction). The minimum I_{sp} found is around 3000 s. Note that for an I_{sp} of 2000 s, the propulsion wet system mass fraction becomes 32% instead of 25%.

Another aspect of interplanetary flight that has a significant impact on the design of the propulsion module is the power profile during flight. For most interplanetary missions, the power generated by the satellite during the mission varies, to a first order, as $1/R^2$ (R being the distance between the spacecraft and the Sun in AU).

It is important to notice that such flight profiles imply a decrease of a factor of 3 in the available power. The propulsion system shall accommodate this decrease with as little impact on the performance as possible.

Furthermore, using the satellite models established earlier, and the power available to the whole propulsion system, one can infer the thrust needed per kg of launch mass. Assuming:

- An initial acceleration of 0.1 mm/s^2 (initial target);
- An initial thrust of 0.1 mN per 1 kg of launch mass (or 100 µN/kg of launch mass);
- A power processing efficiency of 0.5 (target bipolar efficiency);

- Low voltage control electronics utilizing 100 mW per 1 kg of launch mass;
- A total emitters efficiency of 0.5 (ideal vs. measured thrust out of one emitter).

The thrust/total emitter power ratio is then calculated and shown in Table 3.

Thus a thrust/power ratio above 0.05 µN/mW (50 µN/W) is desirable from a mission performance standpoint. For clarity, the power entering these calculations is the total input power to the propulsion system.

A thrust/emitter power ratio above 0.2 µN/mW (200 µN/W) is desirable from a mission performance standpoint.

Note that these results are at this point independent on the I_{sp}. Also note that trajectories with lower initial accelerations are also possible, and thus there is margin included in these requirements.

Table 3: Minimum performance requirements for various satellite models.

Parameter	CubeSat 3U 3 kg	NanoSat 8 kg	MicroSat 27 kg	MicroSat 64 kg
Power available to propulsion system (W)	9.5	42	75	134
Minimum thrust/power ratio (µN/W)	31	19	38	48
Minimum thrust/emitter power ratio (µN/W)	120	70	140	190

Target requirements:

- Th/Pw > 50 µN/W_total
- Th/Pw > 200 µN/W_emitter

An estimation of the required overall lifetime of the propulsion system can be inferred from the propellant mass fraction. At 3000 s I_{sp}, the propellant mass fraction is 15%, which leads to about 470 g of propellant for a ΔV of 5 km/s in a 3U CubeSat. The burn time (time during which the thruster is operating) required is then about 13,000 hrs, corresponding to an impulse of 1300 Ns. Although very large, this lifetime can be shared between several thruster heads in a module.

Furthermore, the MT propulsion system is by design highly modular. The approach taken in the mission analysis is that the MT module in its smallest configuration should fit within a 3U CubeSat. Then, as the mass of the satellite increases, the number of usable and operational thruster heads (chips) should increase up to its maximum per module. If more thrust is required, then more modules are added. Making sure that the power demand is still acceptable is part of this analysis. The number of thruster heads per module (configuration) is tightly linked to the thrust performance of the MEMS colloid thrusters, and target requirements were specified by the analysis.

Summary of Requirements

The main derived and high level mission and system requirements can thus be summarized as such:

- The MicroThrust propulsion system shall be designed for exploration mission destinations including the Moon, NEO, Sun-Earth Lagrange points and Mars. The total ΔV capability of the MicroThrust propulsion system shall be greater than or equal to 5 km/s.
- The design of the propulsion system shall accommodate a decrease of input power by a factor of 3 during the duration of the mission without major impact on its I_{sp} performance.
- The propulsion specific impulse at full power shall be greater than or equal to 3000 s for interplanetary exploration missions.
- The produced thrust to power ratio of the MicroThrust propulsion system shall be greater than 50 µN/W when considering the power as the total input power to the propulsion system.
- The lifetime of the propulsion system shall be greater than or equal to 13,000 hrs over the range of considered I_{sp} values. This is equivalent to a total impulse of at least 1300 Ns over the same range.
- The MT propulsion system in its smallest configuration shall fit the constraints of a 3U CubeSat. The available thruster area should cover a square area that is no more than 88 x 88 mm^2, as specified in Fig. 1.

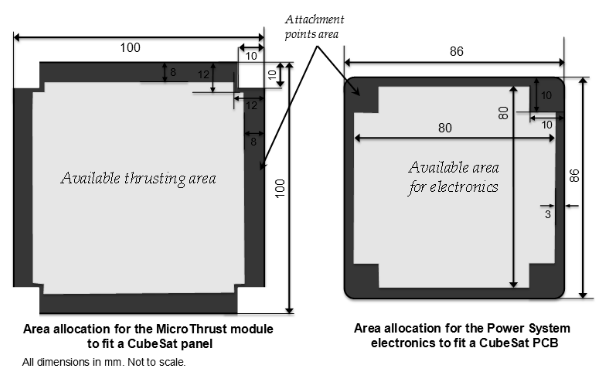

Figure 1: Mechanical interface requirements to fit a CubeSat 3U.

BREADBOARD SYSTEM DESIGN

A laboratory breadboard model of the MicroThrust propulsion system is currently under development. This model will allow the MT system to reach TRL 5 and get one step closer to actually help performing the missions that were outlined in the previous section. The current section will elaborate on the different subsystems of the breadboard (see Fig. 2) and explain some of the design choices that have been made.

The subsystems that will be looked at in more detail are the Thruster Module System (TMS), the Thruster Chip (THC), the Power & Control Supply (PCS) and the Breadboard Test Setup (BBTS). Within the MicroThrust consortium, NanoSpace is responsible for the TMS, EPFL designs and builds the THC, the PCS is designed and built by SystematIC, QMUL is responsible for the BBTS and TNO provides systems engineering support.

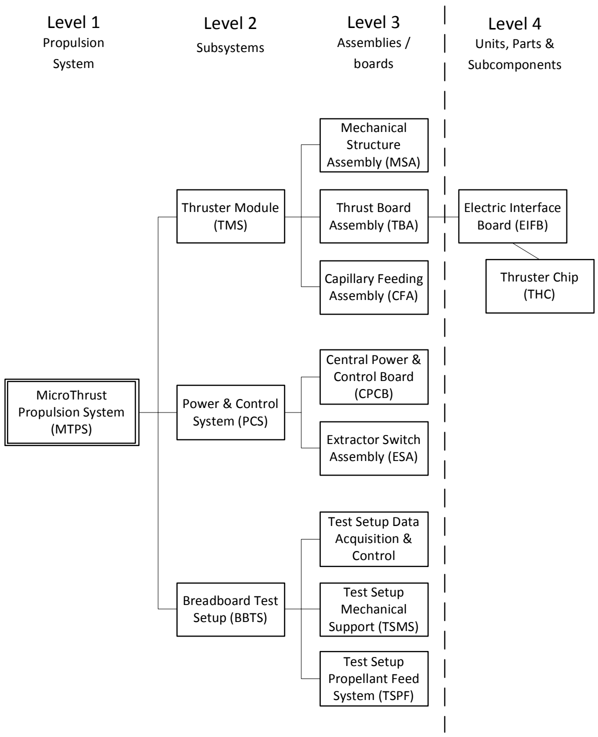

Figure 2: Breadboard system breakdown.

Thruster Module System (TMS)

The key drivers for the preliminary design have been miniaturisation and a high degree of integration between the components. MEMS technology is the enabling technology which offers a distinct improvement in terms of miniaturisation of propulsion systems, however the integration and interaction between MEMS components and more macroscopic parts need to be overcome. The design work also included the idea of a modular thruster system, here called Thruster Module System (TMS), which allows it to be readily adapted to a wide range of mission, with one or several TMS per spacecraft. The TMS consists of three different assemblies (according to Fig. 2 above); the MSA, the TBA and the CFA. All parts in the TMS are there to provide mechanical, electrical and fluidic interfaces to the key component: the Thruster Chip (THC). The THC is the most critical, sophisticated and interesting part of the TMS and will be described in more detail in a subsequent section below.

In the plastic prototype shown in Fig. 3, four MEMS-fabricated thruster chips are visible on top, which are fed with propellant via the CFA (not visible in Fig. 3) and assembled between the mechanical housing parts (MSA). The thruster chips are mounted in a special Electrical Interface Board (EIFB), which in turn is clamped between the housing.

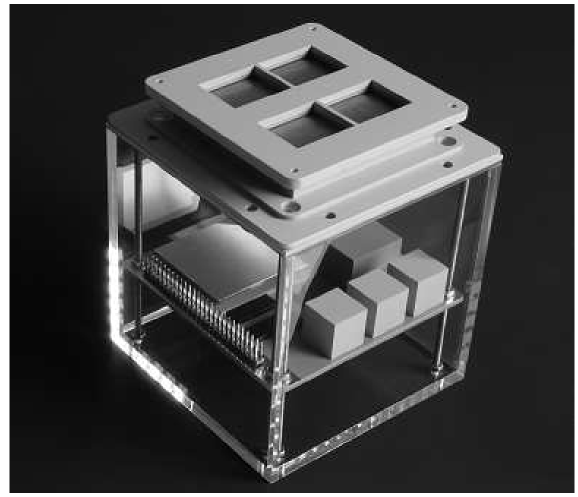

Figure 3: Prototype of Breadboard Thruster Module.

Thruster Chip (THC)

The thruster chip is the main thrust producing element and consists of arrays of microfabricated electrospray emitters with integrated extractor and accelerator electrodes. The internal structure of a single emitter in the array is depicted Fig. 4. The functional components of the thruster chip are located on two different wafers, which are stacked vertically using a polymer separation layer. On the lower wafer, the 5 μm inner diameter capillaries and the local liquid reservoirs are implemented via standard silicon processing techniques. The capillaries stand off the silicon surface and each face an individual annular extractor and accelerator electrodes, which are implemented on a common glass substrate. Through this layout, in combination with the high structural uniformity of the emitter arrays, which is a typical advantage of microfabrication techniques, homogeneous spray characteristics across the array can be guaranteed.

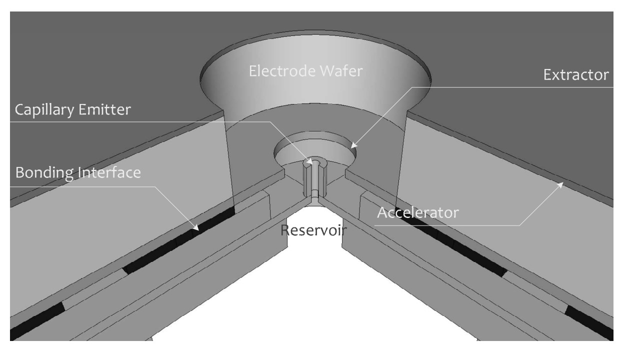

Figure 4: 3D depiction of a single pixel of the thruster array.

The operation principle of an electrospray thruster and the microfabricated thruster structure are depicted in Fig. 5. In the most basic configuration, an electrospray thruster consists of a capillary filled with a conductive ionic liquid (EMI-BF_4 in this case) and a hollow extractor electrode placed at close proximity of the capillary tip. With high potential difference in the order of a kV applied between the liquid and the extractor, the tip meniscus collapses into a Taylor cone, leading to the emission of high energy droplets and/or ions from the cone apex. The acceleration of these massive particles exerts a reaction force on the spacecraft that thrusts it in the opposite direction. A secondary electrode stage provides further acceleration of these particles, and also acts as an ion lens limiting the beam width and spread.

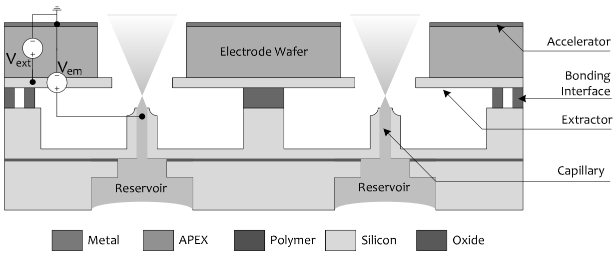

Figure 5: Cross-section view of the thruster chip and the driving scheme. The extractor and the accelerator electrodes are fabricated on a common glass wafer with electrode layers on each face. This joint electrode stack can be integrated with the capillary wafer using a conventional bonding process.

As a first step to the fabrication of the structure depicted above, thruster chips with integrated capillaries and the extractor electrodes are fabricated. Through this device, the capillary fabrication and the bonding process are successfully demonstrated. The fabrication process for this device is detailed elsewhere [16]. SEM images of this preliminary device are shown in Fig. 6.

The fabricated thruster chips with integrated emitters and extractor electrodes were successfully operated in pumpless liquid delivery configuration in a dedicated test chamber using conventional laboratory-use high voltage power supplies and switches. The fundamental achievements in terms of device performance were the stable bipolar operation and ionic-mode operation with passive liquid delivery [16], which are demonstrated using single emitters and 19 emitter arrays. This is a major step towards a pumpless thruster system, which could significantly simplify thruster design and flight procedures. Despite its obvious advantages, the bipolar operation mode introduces certain challenges in terms of power supply and control, which are addressed in the following section.

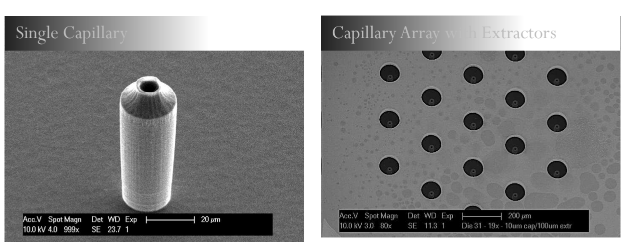

Figure 6: SEM picture of a single capillary and the thruster array with integrated extractor electrodes. The emitter inner diameter is 5 µm, and the extractors are 150 µm. Both components are fabricated on silicon wafers, and integrated via wafer level polymer bonding. As the next step, the electrode wafers will be fabricated on a glass wafer with integrated accelerator electrodes.

Power & Control System (PCS)

This section will explain a number of aspects of the PCS, such as the consequences of bipolar operation, the system requirements, and the breadboard specifications and design.

Bipolar Operation and the Consequences for the Power System

In the power and control system, bipolar operation is applied to the thruster module to prevent electrochemical degradation due to the deposition of the charged particles within the propellant system onto a conductive surface, and optimally (fully) use the propellant.

In electric propulsion systems the net flow of charge needs to be neutral in order to prevent charging of the S/C. Traditionally positive ions are neutralized by neutralizer electrons. In the MT thruster design, bipolar operation is applied, where positive and negative ions are subsequently expelled. Bipolar switching of the thruster at a high frequency to prevent S/C charging is not feasible at acceptable power efficiency, and also limited to the formation time of the electrospray. As a solution to this problem a dual bipolar thruster is designed, where synchronously one thruster expels positively charged ions and the other thruster expels negatively charged ions. Both thrusters are controlled to expel the same amount of charged particles per time (flow). Impulse variation due to a difference in mass of both ion types is compensated for by independent control of the positive and negative emitter voltages.

This dual bipolar thruster concept is depicted in Fig. 7. On the right side a bipolar thruster module can be seen that includes two bipolar colloid thrusters that are driven with opposite polarity. In the thruster module an extractor switch assembly (ESA) independently switches and controls a number of extractor grids.

The thruster module is connected to a central power and control board (CPCB) through HV lines and LV control lines. In the CPCB the interface to the spacecraft's 12 V supply and control bus, a LV control assembly, HV supplies, a HV switch matrix and current sensing of the HV lines can be found. The current sense capability allows accurate monitoring of positive and negative particle flows.

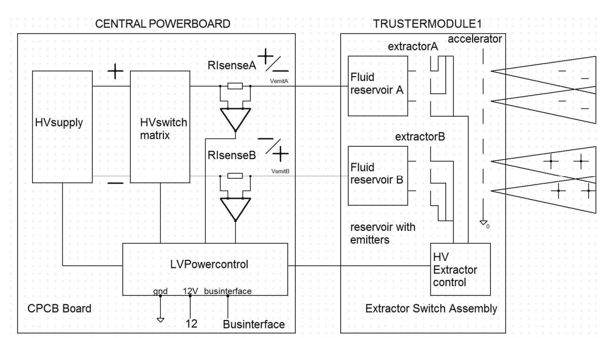

Figure 7: Dual bipolar thrusters with CPCB and ESA board.

Requirements for the Power and Control

This section lists the general requirements and design solutions for the power and control architecture:

- Dual thruster with bipolar operation.

The dual bipolar thruster concept adds significant complexity to the design and poses challenges to the low mass and low volume requirements. In the design a single positive and single negative HV bus are available for the thruster modules in the spacecraft. Each thruster module has an individual extractor grid switch and control assembly (ESA) that allows individual control of each thruster module and up to N_{extr} extractors in such a module.

- Thruster voltage is in the ±4 kV range. Low power operation and power conversion efficiency are key parameters.

Due to power budget and mass requirements, efficient HV boost conversion is a key design challenge. In order to accommodate high efficiency over a broad range of loads the converters implement a power save control algorithm well known in LV DCDC conversion systems (pulse skip and standby mode).

- Power range supports various missions 1-5 W/42 W/134 W.

In the breadboard design, in line with the MicroThrust idea of a low mass low volume thruster, focus is on a 1-5 W HV supply.

- The architecture will be constructed in a way to allow extension of the number of supplies in the propulsion system and actual control ranges to allow use of the concept in different missions.

This architecture covers a dual bipolar setup and is scalable in power to accommodate various missions. The architecture also covers static bipolar operation, in which case the thruster operation is bipolar, but tank voltage is not switched or even unipolar operation, in which case the HV part is significantly simplified. The design has a single CPCB with the ability to drive multiple thruster modules with their own ESA control board. In this way the MicroThrust modular approach is implemented by design and offers flexibility to extend the number of thruster modules as well as the power range.

- Control of supply start-up, regulation, fault handling and redundancy are to be included.

Actual thrust settings are under control of the central MCU of the spacecraft and not part of the power and control board. Extractor grid control and error handling is done locally by the MicroThrust CPCB and its local state machine.

- A strategy towards space qualification must be addressed.

Design guidelines for radiation tolerance will be followed; relevant qualification related tests are foreseen in the project.

Specifications of the Power & Control System Breadboard

The breadboard proves functionality of the electronic control for a configuration of two extractor grids of a dual bipolar thruster. The circuit covers all critical electrical functional blocks and allows scaling of the design to mission requirements.

Table 4: Breadboard electrical system specifications.

Emitter voltage	±3.8 kV with ripple < 4 V and 100 μA (0.4 W)
Current sensor accuracy	< 1% of maximum current
Bipolar operation frequency	< 0.1 Hz
Target power conversion efficiency in bipolar operation	> 50% at 0.4 W
$\|V_{emitter}-V_{extractor}\|$ voltage	0-1 kV with ripple < 4 V
$V_{extractor}$ voltage accuracy	< 0.1%
Accelerator voltage	0 V
Configuration	1 thruster module with 2 thrusters and 2 controlled extractor grids per thruster
S/C power supply voltage	12 V
S/C control interface	I^2C

Breadboard Design

The various circuit concepts are validated in hardware. The board design that includes a single bipolar supply, HV switch matrix and single channel extractor control is shown in Fig. 8.

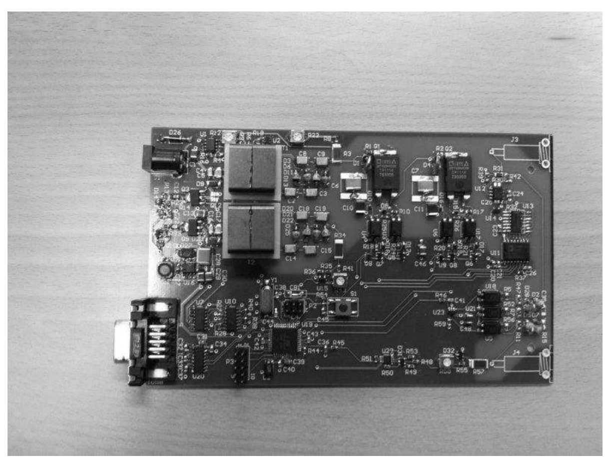

Figure 8: Initial hardware design.

Initial electronic design of the breadboard indicates that the 8 cm by 8 cm breadboard CPCB is almost fully occupied with components. The mass is estimated to be 96 gram. The 8 cm by 8 cm ESA breadboard has a mass of 51 gram. Extending the thruster with more extractor grid control circuits or multiple thruster modules (the MicroThrust modular approach) will further increase ESA circuitry and one board per thruster module is mandatory.

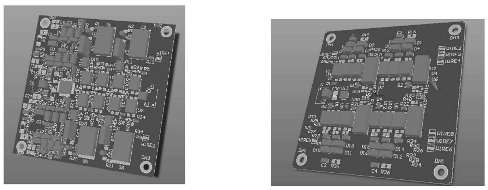

Figure 9: CPCB and ESA breadboard design.

Breadboard Test Setup (BBTS)

The breadboard will consist of four thruster chips, connected to two separate propellant tanks. Therefore two thruster chips will be connected to each propellant tank, allowing for the testing of emitter to extractor short-circuiting. The thruster chips will be housed within the Thruster Module system (TMS), as described earlier. The PCS units, consisting of the CPCB and ESA boards, will then be mounted to the rear of the TMS using four threaded studding poles. These poles then provide the method to mount the whole breadboard testing assembly within the thruster chamber, by a simple flange with a slot that fits into the associated flange within the vacuum chamber.

At this stage of breadboard test planning, testing the breadboard will initially be completed vertically within the 400 mm diam., 400 mm length vacuum chamber available at Queen Mary, University of London (QMUL). The reason for this is the simplicity and uniformity with the vacuum chamber setup currently available, but it will also allow for the mounting of the breadboard model (BBM) to a linear manipulator, as illustrated in Fig. 10. This linear manipulator now procured by QMUL will result in the motorised vertical positioning of the full breadboard model, allowing for accurate analysis of the BBM plume angle amongst other things.

It is currently conceived that the BBM will be positioned onto the linear manipulator (or to a vacuum flange if the manipulator is not needed for some testing) by feeding it through a side port of a smaller vacuum chamber attached to the top of the larger chamber. This smaller chamber has multiple ports, allowing for ease of access and multiple feedthroughs for propellant, electrical connections, etc, and is available at QMUL.

Within the main chamber there will be available multiple types of diagnostic equipment, which can be mounted to a movable stage with a translation length of 280 mm attached to the bottom face of the cylindrical vacuum chamber. These diagnostic equipment include a Faraday cup, a Retarding Potential Analyser, a Time-of-Flight (TOF) system (all available at QMUL), or other diagnostic equipment under consideration. The test campaign is currently scheduled for 2013.

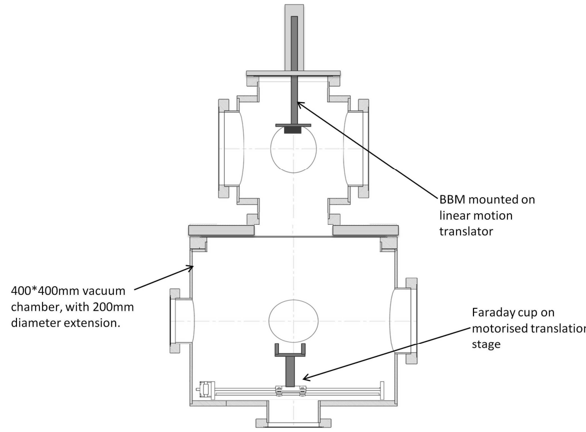

Figure 10: Simple schematic of possible breadboard model mounting setup within chamber available at QMUL.

CONCLUSIONS

From the mission analysis, it was determined that the MicroThrust electric propulsion system shall be designed for exploration mission destinations including the Moon, Near-Earth Objects, Sun-Earth Lagrange points, and Mars, on board of small platforms (i.e. from 64 kg down to a CubeSat). The specific impulse of the system should be at least 3000 s with a thrust to power ratio greater than 50 μN/W.

MicroThrust electric propulsion system preliminary design is under development by a team of 5 European parties within the FP7 framework and in order to prove that the system is capable of fulfilling the system and mission requirements, a breadboard model is currently being developed. Most system components have already been defined and some are already being built; among the others, a miniaturized HV power control and supply system, a module able to allocate and feed thruster chips operating in bipolar mode and MEMS thruster chips. Testing of the complete breadboard system is scheduled for 2013 at QMUL.

Once operational, the MicroThrust technology will enable low-cost exploration missions to a wide range of destinations, that used to require very high budgets to reach. Industry is starting to see the potential of the MT system and talks are already underway about implementing it in future missions.

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