Discussion of Micro-Newton Thruster Requirements for a Drag-Free Control System

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Abstract. The detection of gravitational waves is important both because of the information about the astrophysical sources and the confirmation of general relativity that it will provide. Space-borne gravitational wave missions such as LISA and GAMMA require Drag-Free Control (DFC) systems to control the motion of a constellation of spacecraft to high positional accuracy so that Michelson interferometers of vast scales can be constructed and used to detect gravitational waves. The spacecraft will continually experience forces and torques due to external disturbances such as atmospheric drag and solar radiation pressure, which will result in the spacecraft being perturbed. Therefore the development of a DFC system is essential to control these spacecraft to a high positional accuracy so as to stabilize them to a specified tolerance over a finite frequency range. In doing such, the signals of interest from compact astrophysical bodies, namely, short-period binary stars or coalescing supermassive black-hole binaries, can be measured. Similar technology is also necessary on other missions where high positional accuracy is required. Additionally these precision thrusters, if they are small enough and consume minimal power, can be very useful on many small satellite missions. Examples include small satellites that need accurate attitude or missions that involve formation flying with accurate positional control requirements. Prior to space-borne gravitational wave missions, a technology demonstrator such as the proposed ODIE, ELITE or SMART-2 missions is needed to test the feasibility of a drag-free spacecraft and the technology involved. This paper discusses some requirements of the associated hardware (e.g., accelerometers, etc.) needed to implement a DFC system for a technology demonstration mission, but primarily focuses on defining the requirements of the type of thruster required for implementing a DFC system. In particular, it presents an overview of current thruster technology and briefly discusses possible alternative technologies that could be applied to the development of a new micro-Newton thruster.

Introduction

Many upcoming scientific missions are extremely ambitious and thus require technological advances to be made in many of the subsystems and components that are currently available within the space industry in order for them to meet their requirements. One area that has recently been gaining attention is in the development of micro-Newton thrust propulsion systems. Up until recently most companies had focused their energy on developing large, high thrust generating propulsion systems. However, with the new philosophy regarding space exploration, smaller satellites are being developed, which in turn require smaller, cheaper and less powerful propulsion systems. Yet even though the cost budgets are smaller, the missions seem to be just as complex and intricate, with high-precision thruster requirements that are more challenging than before. This is particularly apparent in formation flying satellite missions where extremely high positional accuracy is required. For example, the LISA (Laser Interferometer Space Antenna) mission will detect miniscule strains in the space-time curvature due to gravitational waves. The mission requirements are to detect strain levels of $h \leq 10^{-23}$ over a frequency range of 3×10^{-4} Hz $\leq f \leq 3\times10^{-2}$ Hz, where h is the dimensionless amplitude of a gravitational wave (strain). In order to be able to achieve this the configuration of satellites (which form a giant interferometer) need to be separated by $\sim5\times10^{6}$ Km and have the change in their separation distances DL measured[4] to an accuracy of $\sim10^{-11}$ m and controlled[5] to an accuracy of $\sim10^{-8}$ m. Consequently, this mission requires the application of a propulsion system that can provide micro-Newton thrust levels with exceptional thrust controllability and that introduces minimum thrust noise into the system so that it does not interfere with the missions scientific measurements. The expected thruster capabilities[21] that companies like the Austrian Research Center anticipate will be necessary for this type of application and aim to achieve are summarized in Table 1.

Table 1: Thruster Technical Capabilities[21]

Technical Capability	Requirement
Thrust range	1 – 100 μN
Thrust Controllability	< 0.1 μN
Thrust Noise	< 0.1 $\mu N/Hz^{-0.5}$
Thrust Vector Stability	< 2°
Minimum Impulse Bit	< 0.001 μN.s
Specific Impulse	> 500 s

To date, no micro-Newton thrusters with the above technical specifications have been flight-qualified. However, the FEEP (Field Emission Electric Propulsion) thrusters are the most advanced in the micro-Newton thruster field and are the closest to achieving the above properties. For example, both the Austrian Research Center and Centrospazio group have recently demonstrated laboratory prototype FEEP thrusters (namely, an Indium ion thruster and a Cesium slit emitter thruster, respectively), which appear to meet the desired thrust range and thrust noise requirements. However, due to the huge leap in technology that these thrusters represent, their performance must be validated in space before they can be used in grand (expensive) missions like LISA. Thus there is a compelling need for technology type demonstrator missions like SMART2.

This paper describes ongoing efforts to develop micro-Newton thrusters that can be utilized alone or within a drag-free control system by satellites of future missions where high positional accuracy is required. Examples of missions that require such technology are formation flying satellite missions, including gravitational wave missions like LISA (Laser Interferometer Space Antenna) and GAMMA (Gravitational Astronomy Multi-Miniprobe Array), and terrestrial planet searching missions like IRSI/Darwin (Space infrared interferometer) and TPF (Terrestrial Planet Finder), as well as NGST, which will demonstrate some of the technology required by the later missions. Micro-Newton thruster technology will also be useful for nano- and pico-satellite orbit control applications and for fine pointing attitude control of scientific spacecraft if the subsystems can be built small enough. Note that gravitational wave missions are extremely ambitious and impose the most stringent requirements on the attitude control system. Consequently, we will focus this paper on thruster ***types*** that are being developed for the gravitational wave application. However, many small spacecraft missions would serve as excellent opportunities to verify the performance of the thrusters under consideration. The less stringent requirements of these missions could easily be met even if the thruster performance is less than desired. **However, it should be remembered that similar thrusters are also needed on many small satellites in order to enable them to meet their own mission objectives.**

In this paper we give an overview of the state-of-the-art of propulsion systems that are currently available and discuss their attributes. We also define the requirements for the type of thruster required for implementing a DFC system. This paper also proposes possible technologies that could be applied to the development of a new micro-Newton thruster sufficiently compact for small satellite applications.

Gravitational Waves

The concept of gravitational waves was first suggested by Einstein's General theory of Relativity, in which they were visualized as small-scale ripples in the curvature of space-time, i.e., oscillatory distortions in the metric tensor g_{uv} (which describes the curvature of space-time). It is suggested that mass acts on space-time, resulting in its curvature, which is observed as a gravitational field. If a system's distribution of mass deviates from spherical symmetry and is non-uniformly accelerated, the result will be a disturbance in the surrounding space-time curvature, which propagates through space as a gravitational wave, carrying energy and angular momentum away from the system. Thus, gravitational waves should arise from the acceleration of mass in compact astrophysical bodies such as binary stars, supernovae or massive black holes in galactic nuclei. As a result the universe is expected to contain complex patterns of these ripples in the space-time curvature.

Gravitational waves should transmit the effects of the acceleration of masses with the speed of light *c*, in a similar manner to electromagnetic waves, which transmit the effects of the acceleration of charges. A fundamental difference between electromagnetic waves and gravitational waves is that the former is dipolar in nature and the latter is quadrupolar. The effect of gravitational waves can be seen more clearly by considering the relative separation in an array of test masses present in the distortion path. The wave would result in their separation distances expanding and contracting in anti-phase. Figure 1 illustrates this effect.

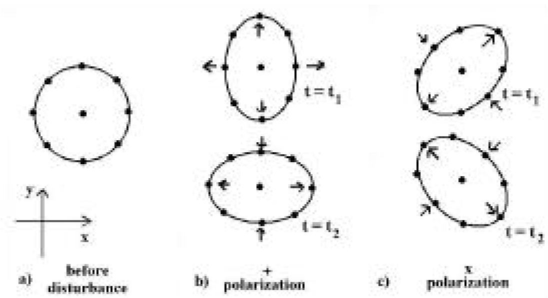

Figure 1: Distortion of test masses due to a gravitational wave, and its polarization states; a) before disturbance, b) after ‘+’ polarization, c) after ‘´ ’ polarization.

As yet, direct detection of gravitational radiation has not been achieved, although an indirect observation has been made. The decay in the orbital period of the pulsar PSR 1913+16 and its companion has been observed for decades. The predicted orbital decay rate, assuming the decay is due to the emission of gravitational radiation is in agreement with the observed rate to within a fraction of a percent[1, 6]. This result strongly suggests that the system is losing energy via the emission of gravitational radiation and thus supports the existence of gravitational waves. Consequently, several ambitious experiments, both Earth- and space-based, are underway that will attempt to detect gravitational waves directly.

Drag-Free Control (DFC) System

A DFC system consists of accelerometers, thrusters and a computer. The DFC system is used to stabilize a satellite that is continually being perturbed by external disturbances by counteracting these disturbances so that they do not induce motion into the system that would otherwise interfere with the scientific measurements. A DFC accelerometer is implemented by enclosing a proof (test) mass in a housing within the spacecraft so that the proof mass is isolated from the surrounding environment. The motion of the proof mass is therefore not disturbed by external surface forces such as atmospheric drag or solar radiation pressure, but is only determined by gravity. The spacecraft would be equipped with thrusters, specifically designed to allow it to maintain its position relative to the proof mass so that they do not come into contact with each other. Therefore, the spacecraft and the proof mass both behave as if they were not acted upon by external forces, and this state is described as being *drag-free*.

In the case of the LISA mission, each proof mass is a solid cube made out of a gold-platinum alloy (90% Au and 10% Pt)[5, 10], that is housed in an Ultra-Low-Expansion (ULE) glass chamber. Each proof mass is surrounded by electrodes that capacitively sense its position with respect to the chamber. This permits the motion of the proof mass relative to the satellite to be determined, as well as the degree of drag-free performance that is achieved. The thrusters would be used to provide forces and torques to cause the satellite to closely maintain its position relative to the proof mass. The DFC computer will be used to integrate all of the subsystems, interpret the attitude and positional data from these accelerometers and command the thrusters.

LISA Mission Overview

This is the seventh cornerstone mission that has been proposed to the European Space Agency (ESA) for their ‘Horizon 2000 Plus’ programme. LISA is a mission to detect astrophysical gravitational radiation in the low frequency band (10^{-4} Hz to 1 Hz)[7, 8]. The LISA constellation will consist of three satellites each containing two optical benches. They will be deployed with one satellite at each of the vertices of a slowly rotating equilateral triangle. Thus the satellites will form a giant Michelson interferometer. Each satellite will have a proposed mass budget of 265 kg and power budget of 200 W. The orbital deployment[4, 8] is designed to maintain the triangular formation, with the triangle appearing to rotate about the center once per year. The center of the triangular formation will be located in the ecliptic plane 1 AU (150×10^6 km) from the sun and about 20° behind the Earth. The plane of the triangular configuration will be tilted out of the ecliptic plane by 60° (see Figure 2).

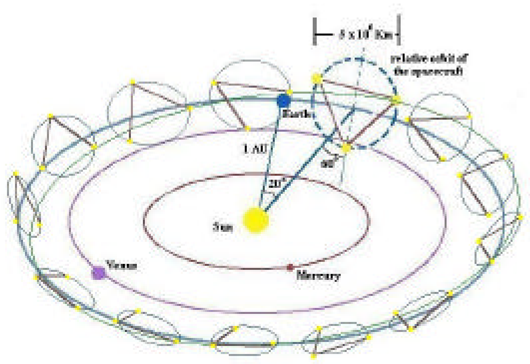

Figure 2: LISA's initial orbit configuration and evolution over 1 year.

Although LISA is one of several proposed missions intended to detect gravitational waves, it is the most ambitious with regards to the sensitivity level it aims to achieve. One way this sensitivity can be achieved is by having very long interferometer arm lengths. The LISA arm lengths are ~ 5×10^6 km. Figure 3 shows a schematic of this configuration.

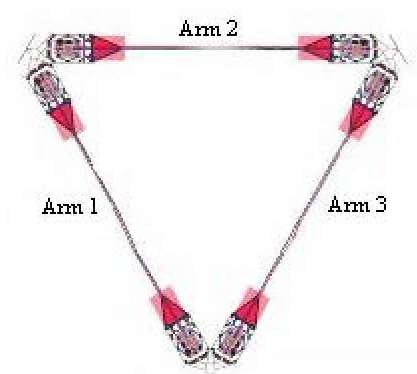

Figure 3: Satellite placement for the LISA interferometer.

Each pair of arms act like a one-bounce Michelson interferometer, i.e. the lasers are locked onto the front faces of the various proof masses (the proof mass faces act in the same way as mirrors in a conventional interferometer) and laser beams are sent back and forth along the interferometer arms and their phases monitored such that any changes in distance between the proof masses can be measured. Figure 4 shows the detail of a pair of optical benches.

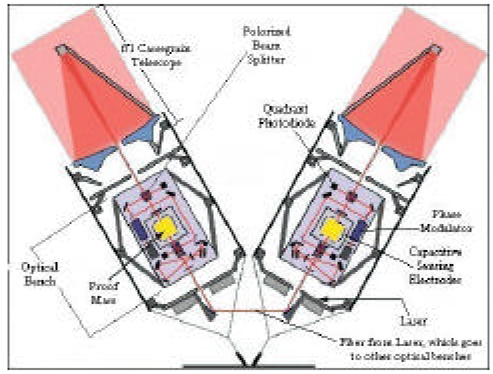

Figure 4: Pair of optical benches. [Figure adapted from 3]

With the LISA configuration, instead of having a separate beam splitter, each of the satellites effectively forms two independent Michelson interferometers, which provides redundancy. This results because the optical benches on each of the satellites are identical and therefore enable the satellites to act either as the central beam splitter or as one of the end mirrors. The change in path length ΔL that would result from a passing gravitational wave is related to the amplitude of the gravitational wave (strain) h[11], by

$$h = \Delta L / L_o, \qquad (1)$$

where L_o is the unperturbed path length. The amplitude of the measured phase difference can be amplified by increasing the interferometer arm lengths, although this also limits the maximum detectable frequency. Using pairs of instruments within the spacecraft allows for redundancy of components, increases the probability of detection and will enable the polarization of the gravitational waves to be determined.

The proposed launch date for LISA is around 2017. Although LISA has been approved by ESA for a future program, its exact launch date is still uncertain, and it seems most likely that it will go ahead as a joint ESA/NASA mission.

Propulsion Systems

Propulsion System Requirements

LISA's science objectives require that its proof masses are isolated from external disturbances so that any changes in their separation distance due to perturbations are minimal compared to changes due to gravitational waves. The science objectives[4] also require that the sensor instrumentation can

of both the cost effectiveness and the greater number of opportunities. Small satellites would also make a perfect platform from which to characterize the performance of the other hardware components involved in a DFC system and to provide knowledge to help further the design of more advanced components. This should ultimately lead to designs that will meet the most stringent of performance requirements and be suitable for use on the most ambitious future missions.

Using small satellites as platforms to test and validate new thruster technology is very effective and efficient. This is because their smaller mass accentuates the effect of the inherent noise of the thruster system utilized, as well as that of any system instabilities. Thus, the greater effect of any non-ideal performance on the attitude and position of a small satellite should make it easier to quantify, characterize and qualify the thrusters than if they were tested on larger spacecraft. Examples of some micro-satellite missions that have been proposed and that could be utilized for this purpose include ODIE, ELITE and SMART2.

Pulsed Plasma Thruster (PPT)

Pulsed plasma thrusters are one of the earliest forms of electric propulsion. Their technology and application being first conceived, researched and developed in the 1950's with the first PPT being flown on the Soviet Zond-2 spacecraft in 1964[13]. Pulsed plasma thrusters are classified as electromagnetic type thrusters as they rely upon the production of plasma, which is then accelerated to its exhaust velocity by an electromagnetic field to generate the desired thrust. A schematic showing the basic components of a PPT is shown in Figure 6.

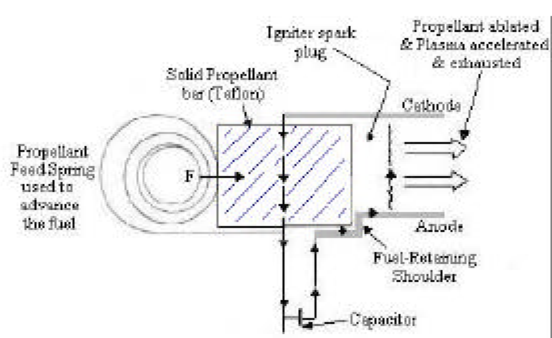

Figure 6: Components of a Pulsed plasma thruster.

It can be seen from Figure 6 that the main elements of a PPT propulsion system are; a bar of *solid propellant*, a *propellant feed spring*, a *fuel retaining shoulder*, an *igniter*, a *capacitor* and two *electrodes*. Teflon is typically used as the solid propellant as tests have shown it to provide better performance than other tested propellants such as polypropylene, Kynar, Delrin, etc.[13] Additionally, Teflon is inert, making it suitable for long missions that require extended fuel storage. Its use also eliminates many of the safety issues regarding its handling and leakage, which are common among other propellants. The operation of the PPT involves the propellant being pushed by the propellant feed spring against a fuel retaining shoulder toward an igniter plug and between two electrodes (an anode and a cathode). The capacitor is charged and applies ~2 kV across the exposed propellant surface. The resulting spark from the igniter initiates the creation of a small amount of plasma, which in turn triggers an electrical discharge across the propellants surface. The heat from this causes the propellant to ablate and self-induces a magnetic field. The propellant is then accelerated away by electromagnetic and pressure forces at speeds[13] up to 15 km.s^{-1}.

Interest in pulsed plasma thrusters was recently renewed in the 1990's due to their simple design, short development time, low cost, robustness and capability of providing tiny thrust impulses. All of these characteristics are important for enabling fine attitude and position control of satellites, and are necessary for formation flying. However, investigations into the exhaust from PPT thrusters requires further analysis to address concerns regarding electromagnetic interference (EMI) and contamination issues. In particular, since the LISA mission makes use of optical instrumentation, which would be prone to contamination by Carbon deposits from the thruster exhaust.

A summary of the performance parameters of two typical PPT Propulsion systems is given in Table 3.

Table 3: PPT Propulsion System Properties

Propulsion System Type	PPT [13, 15, 16, 30, 31, 32] (Teflon)	DAWGSTAR PPT [42, 45] (Teflon)
Thruster Operation	Pulsed	Pulsed
Thrust Range [mN]	< 2000	< 112
Thrust Noise [mN/Hz$^{-0.5}$]		
Specific Impulse [s]	1000 - 1500	500
Minimum Impulse bit [N.s]	1e^{-5} - 1 e^{-3}	5.6e^{-5}
Thrust to Power ratio [mN/W]	15	8.3
Mass [kg]	5.83	3.59

Field Emission Electric Propulsion (FEEP) Thruster

FEEP thrusters are classified as electrostatic type thrusters as they involve the use of a high voltage electrostatic field to accelerate ions to large exhaust velocities in order to generate the desired thrust. A schematic showing the basic components and layout of a FEEP thruster is shown in Figure 7.

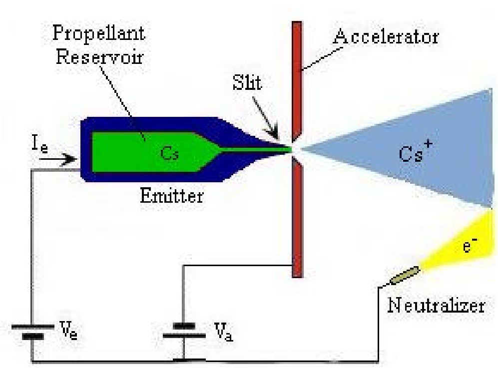

Figure 7: Components of a FEEP thruster.

The main components of a FEEP thruster are; an *emitter*, an *accelerator* and a *neutralizer*. The propellant (which is usually in a solid form) is stored in a reservoir inside the emitter and is heated so that it liquefies ready for firing. The liquid propellant then flows towards the emission slit via a capillary suction effect. Once in this ready state the FEEP thrusters operate by creating ions directly from the surface of a liquid metal (e.g. Cesium or Indium) by subjecting it to a strong electric field (8 to 15 kV). This causes the liquid metal to distort until the surface layer of the atoms are ionized and accelerated away at speeds[25] greater than 50 km.s^{-1}.

The electric field is formed by applying a positive potential to an emitter, and a negative potential to an accelerator. The neutralizer distributes an electron beam across the ionized thrust exhaust path, thereby neutralizing the exhaust.

Cesium has a low melting point of approximately 29°C and was chosen by the Centrospazio development group because of this and because it has a low work function (2.14 eV) and high atomic weight (≈ 2.207×10^{-25} kg). The thrust generated by a FEEP thruster is directly proportional to the applied voltage, and is also a function of the length of the slit in the accelerator plate through which the propellant exhaust is emitted. Slit lengths[22, 31] ranging from 1 mm to 15 cm have been investigated to date. The slit generally has a separation gap[22] of about 1 µm. The resulting thrust can range from 0.1 to >100 µN.

Note that FEEP thrusters appear to be a good choice for missions like LISA but present problems for small satellites because of the high operational voltages they require.

A summary of the performance parameters of two typical FEEP Propulsion systems is given in Table 4.

Table 4: FEEP Propulsion System Properties

Propulsion System Type	Slit Emitter FEEP [12, 15, 28, 29, 32, 43] (Cesium)	FEEP [12, 21, 38, 39] (Indium)
Thruster Operation	Continuous	Continuous
Thrust Range [mN]	0.1-1200	1-100
Thrust Noise [mN/Hz$^{-0.5}$]	0.1	< 0.1
Specific Impulse [s]	7000-11000	10000
Minimum Impulse bit [N.s]	1e^{-9}	<10e^{-9}
Thrust to Power ratio [mN/W]	16 -20	15
Mass [kg]	2.2	2.5

Colloid Thruster

Colloid thrusters were another of the earliest forms of electric propulsion. Their technology and application was first proposed in the 1960's with continued research into the early 70's. However, interest in them soon waned because high thrust requirements meant the overall size would increase significantly since much greater voltages would be needed to operate them. In contrast to pulsed plasma thrusters, which were flown early in their development time, colloid thrusters have not been flown until very recently. This renewed interest in colloid thrusters has come as a result of technology advances. A schematic showing the basic components of a colloid thruster is shown below in Figure 8.

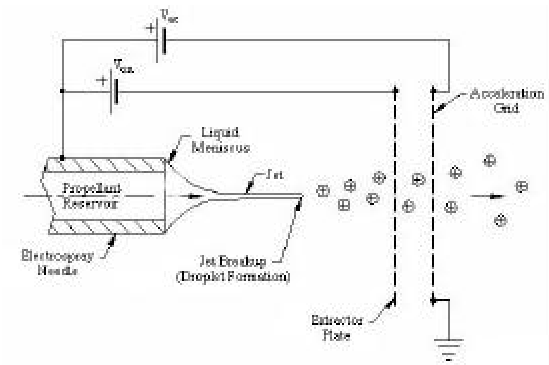

Figure 8: Components of a Colloid thruster. [Figure reproduced from 34]

The main components of a colloid thruster are; an *electrospray needle*, an *extractor plate*, an *acceleration grid* and a *neutralizer*. The operating principle of a colloid thruster is similar to that of a FEEP thruster, in that it also uses a high-voltage electrostatic field to accelerate ions to generate a thrust. The propellant used is generally a conductive substance that is in a liquid form. Examples of propellants used[31] are glycerol, sodium iodide or lithium chloride. The propellant is pumped through the electrospray needle at a high potential (~ 5 to 10 kV)[30] and the force from the negatively charged extractor plate deforms the propellant fluid such that it forms a stream of positively charged droplets, which are then accelerated away at high speeds via the acceleration grid. Note that the exiting positively charged colloid beam needs to be neutralized to avoid charging of the satellite. This can be achieved either by applying a similar colloid beam of reverse polarity or by using a low-power electron field emission cathode to inject a beam of electrons.

A summary of the performance parameters of a typical Colloid Propulsion system is given in Table 5.

Table 5: Colloid Propulsion System Properties

Propulsion System Type	Colloid [12, 30, 40, 44]
Thruster Operation	Continuous
Thrust Range [μN]	0.5 – 25, 500
Thrust Noise [μN/Hz$^{-0.5}$]	0.01
Specific Impulse [s]	500 – 1500
Minimum Impulse bit [N.s]	0.5e^{-6}
Thrust to Power ratio [μN/W]	10
Mass [kg]	0.5

Hall Thruster

Like the FEEP and Colloid thrusters, the Hall thruster is also classified as electrostatic type thruster as it involves the use of a high-voltage electrostatic field to accelerate ions to large exhaust velocities. The development of Hall thrusters is being pursued widely in Russia, Europe, the U.S. and Japan. They have been used in orbit for a significant number of years, one of their first applications being in 1974 for orbit control of Meteor-18. Hall thrusters generally lead in technology with respect to operating in the milli-Newton thrust range but require further development for application to the micro-Newton level. A schematic showing the basic components of a Hall thruster is shown below in Figure 9.

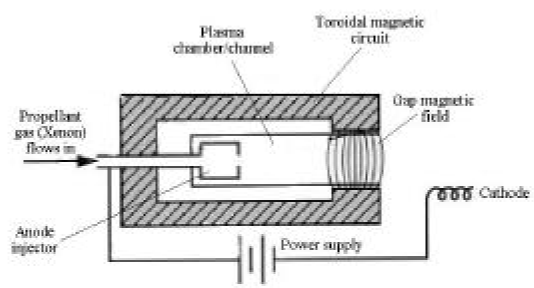

Figure 9: Components of a Hall thruster.

The main components of a Hall thruster are a *toroidal magnetic circuit*, an *anode injector*, a *plasma chamber/channel* and a *cathode*. Hall thrusters are gridless accelerators that use the forces on charges in crossed electric and magnetic fields. Electrons are emitted from a cathode at a negative potential and are initially attracted toward the anode. However, many of the electrons become trapped in the magnetic field within the chamber. A propellant gas (generally Xenon) is injected through the anode into the plasma chamber where it is ionized by the trapped electrons. Once the propellant is ionized, it is immediately accelerated out of the chamber to a high velocity by the electric field, which is formed by the negative potential of the cathode.

A summary of the performance parameters of a typical Hall Propulsion system is given in Table 6.

Table 6: Hall Propulsion System Properties

Propulsion System Type	Hall [30, 32, 34, 41, 44] (Xenon)
Thruster Operation	Continuous
Thrust Range [μN]	> 4e^{+3}
Thrust Noise [μN/Hz$^{-0.5}$]	Unknown
Specific Impulse [s]	> 1200
Minimum Impulse bit [N.s]	1e^{-3}
Thrust to Power ratio [μN/W]	60
Mass [kg]	0.9

Micro Electro-Mechanical System (MEMS) Thruster

Micro Electro-Mechanical propulsion systems involve the use of advanced semiconductor manufacturing techniques to produce a chip, which comprise of an array of tiny cells that contain propellant and that are fired to generate thrust. Research into this technology was first pursued in 1992, by the Aerospace Corporation and has received continued interest and development by TRW, Caltech and the Aerospace Corporation. MEMS were first flown on the MightySat 1 mission in 1998 and most recently had their first in-flight functionality test aboard the Scorpius sub-orbital sounding rocket in 2001 where it fired more than 20 times at 1 second intervals and each thruster (cell) produced a 10^{-4} N.s of impulse[35, 36]. Schematics showing the basic components of a MEMs Thruster and size are shown in Figures 10 & 11.

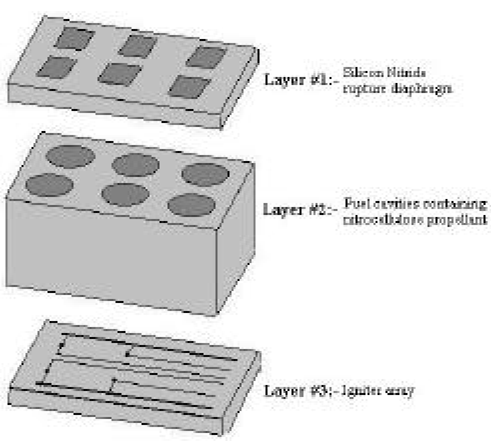

Figure 10: Components of a MEMS thruster array.

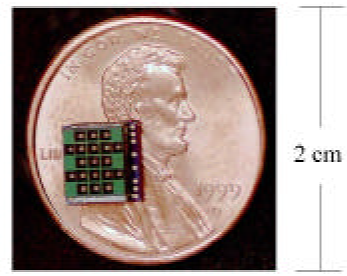

Figure 11: Photo illustrating the size of one of TRW's prototype MEMS propulsion system. [Photo courtesy of The Aerospace Corporation[35].]

The MEMS propulsions system is a multi-layered chip made out of silicon and glass that contains an array of cells in which the nitrocellulose propellant is stored. The cells have 3 layers; the bottom layer of the cell is the igniter array, which consists of polysilicon resistors and also contains lead styphanate. The middle layer is the fuel cavity that consists of cylindrical chambers, which contain the propellant and the top layer is the silicon nitride rupture diaphragm, which seals in the cells propellant. The MEMS thrusters are detonated by current pulses. The heat from the resistor ignites the lead styphanate, which in turn ignites the nitrocellulose propellant stored in the fuel cavity. The pressure of this propellant subsequently increases as a result of this action, and when it exceeds the burst pressure of the diaphragm it explodes through the upper layer and expels, generating a thrust. Note that each cell is an individual thruster that is ignited separately. Thus the thrust delivered in discrete levels.

Like the PPT thrusters the MEMs demonstrate desirable qualities such as robustness, low power usage[37] (≈ 10 mWatts), radiation tolerance and low unit cost. All of these attributes are important and preferred when considering any technology for application to nano- and pico-satellites or formation-flying missions. Other advantages of MEMs that make their use desirable compared with conventional thrusters is that they have no moving parts, they can utilize a variety of propellants and can be scaled in size easily. They can also be fully integrated into the structure of the satellite[35].

A disadvantage of using MEMs is that the thruster elements are discrete and expendable[38]. This obviously would impact the MEMs propulsion system's degree of control over a period of time.

Consequently, this needs further investigation to see whether it would be possible to create a propulsion system of this type that could meet the stringent control requirements that are needed for missions like LISA throughout their desired mission lifetime (3 years for prime science, 10 year goal).

A summary of the performance parameters of two typical MEMs Propulsion systems is given in Table 7.

Table 7: MEMs Propulsion System Properties

Propulsion System Type	MEMs [30, 35, 36, 37, 46] (Chemical)	MEMs [30, 35, 36, 37, 46] (Electrical)
Thruster Operation	Pulsed	Pulsed
Thrust Range [μN]	1-100000	> 100000
Thrust Noise [$\mu N/Hz^{-0.5}$]	~15%	
Specific Impulse [s]	100 -300	1000 - 2000
Minimum Impulse bit [N.s]	$1e^{-9}$	$1e^{-9}$
Thrust to Power ratio [μN/W]	1000	> 1000
Mass [kg]	> $2.4e^{-3}$	

Discussion

The PPT type thruster system has many attributes as discussed briefly earlier. It has a simplistic design, which lends itself to quick development, low cost and reliability. The robustness and technology of this system has also been proven in space on numerous occasions (examples of flights include: Zond-2 1964, LES-6 1968, TIP, NOVA 1976, and MightySatII.1). Also the PPT system has a moderate specific impulse I_{sp}, which aids with minimizing the propellant mass and it produces small and repeatable impulse bits. Disadvantages of this system include it having a fixed system mass, which results because of the use of a capacitor. There are also concerns regarding contamination issues from the Teflon plume, which is particularly of interest since LISA utilizes a number of optical components. Also the system has a fairly low thrust to power ratio *T/P*, which results in high power requirements. Other issues include the difficulty involved in characterizing the response of the PPT thruster system. This results because the system involves complex interactions from both electromagnetic and electrothermal processes, which generate a complex discharge flowfield.

The FEEP thruster system also has many attributes and is currently the most advanced with regards to meeting the LISA mission thruster requirements. The advantages of this system include its high specific impulse I_{sp}, which leads to low propellant mass and contributes to the systems overall low mass. It produces extremely small and repeatable impulse bits and the indium ion emitter technology has also been tested in space on several occasions (MIR 1991, GEOTAIL 1992 and EQUATOR-S 1997). Note however, the technology was used for controlling the spacecraft's potential and not as thrusters. Although the proposed thruster system configuration that aims to meet many of LISA's requirements has not been flight qualified yet, they have demonstrated these qualities experimentally and are probably close to being flight qualified. Disadvantages with the system include it having a low thrust to power ratio *T/P*, which gives rise to high power requirements. The system also requires very high voltages ≈ 10 kV to initiate ion extraction and accelerate the ions. With this system there are again concerns regarding contamination from the exhaust plume. Plus the thruster lifetime is unknown and thus needs further investigation. Limitations are also introduced into the system by the cathode emitter, which has a limited lifetime[32] (≈ 28,000 Hrs). Also the cesium thruster has not yet been flown in space.

The Colloid thruster system also holds great promise it too having a simplistic design, which enables quick development, low cost and reliability. The use of inert propellants such as glycerol lends itself to long missions, which require extended fuel storage and eliminates many of the safety issues regarding its handling on the ground. It has a moderate specific impulse I_{sp}, which aids in minimizing the propellant mass and the systems overall volume. Both of which are criteria needed for application on nano and pico-satellites. Colloid thruster technology is also versatile with regards to its configuration and capabilities, in that it can provide a broad range of specific impulse and thrust. Disadvantages of the system include it having a low thrust to power ratio *T/P*, which results in high power requirements and currently no μNewton level colloid thruster systems have been flown in space. This system also requires the use of an electron emission cathode to neutralize the beam to avoid charging of the satellite. Therefore this introduces lifetime limitations into the system, since a cathode emitter has a limited lifetime[32] (≈ 28,000 Hrs). This system also requires very high voltages (several tens to hundreds of kV) to reach the desired specific impulse (around 1000 s) due to the low charge/mass ratio of the droplets extracted.

Advantages of the Hall thruster system include it having a moderate specific impulse and high thrust to power ratio *T/P*, which give rise to reasonable propellant mass and power requirements. Its technology has been space proven on a number of times (Meteor-18 1974, Kosmos, Luch 1982 and Gals, Express 1994) and there is a low risk of contamination from its plume. Disadvantages include; it has a fixed mass and complexity due to its tankage and valves, which is added to by the power supply unit (PPU). The PPU unit tends to be more complex and thus larger and heavier than others due to the added control requirements needed to control the flow rate and accommodate the plasma fluctuations. Also like the FEEP and Colloid propulsion systems, the lifetime of this system is again limited due to its use of a cathode emitter. Another draw back of the hall propulsion system is that its design is currently geared toward mNewton operation ranges, and requires relatively high power.

Advantages for the MEMs thruster system include its robustness due to it not having any moving parts. They have low power usage, good radiation tolerance and low unit cost. It can utilize a variety of propellants and can be scaled easily according to the thrust range requirements. Also its versatile composition enables it to be integrated into the body of the satellite. They have also been flown in space, and although their flight was predominantly successful not all thruster cells detonated (20% failed)[35]. Therefore the systems main disadvantage is that, this is relatively new technology and requires further investigations to characterize them and determine failure rates etc.

It can be seen when comparing the various thruster property tables with the requirements for the LISA mission that none of the thruster systems have demonstrated all of these requirements yet. However, they all seem to have good potential with regards to obtaining the requirements with further development. Continued interest and growth in the research and development of these thrusters is clearly shown. Thus it seems inevitable that the use of one these types of thruster systems will be applied to LISA and steady growth in this field will ensure that these thruster systems will be widely used on other missions with similar requirements. However, it can be seen that the Indium FEEP thrusters are the closest to achieving all of the desired requirements. Although both the colloid and Cesium slit emitter FEEP thruster systems are in close contention with the Indium FEEP thruster, as they too have demonstrated that they can meet the thrust noise requirement of $< 0.1\ \mu N/Hz^{-0.5}$ and thus the positional control requirement, which are probably the most critical. Since these requirements define whether the assigned acceleration budget can be met or not. This in turn determines if the science measurements can be made.

Continued development into making these technologies smaller, lighter, less power demanding and efficient will also ensure that they will be suitable for application to small satellites as well and probably will be used widely on micro, nano and pico-satellites in the near future.

From the foregoing, it appears that PPT propulsion systems may be well suited for application to missions like LISA that require micro-Newton thrust, as their capability domain includes most of the domain of the LISA mission. However, PPT systems do not presently achieve the smallest thrust levels of importance to LISA. Also, the thrust noise of PPT propulsion systems has not yet been characterized, and to date PPT propulsion systems have been designed for pulsed operation only, which introduces more noise into the system than continuous operation. Thus it is not clear that a PPT propulsion system can meet all of the performance requirements of the LISA mission. However, if the PPT system could be adapted to operate in a continuous manner, it would be a more appealing option for gravitational wave mission applications. Consequently, this aspect should be researched further.

Cold-gas thrusters and FEEP thrusters are also close to the desired performance domain. Therefore is seems logical that further investigations into the designs of these thrusters should be performed, to see if improvements can be made that will enable them to meet these requirements.

Continued research into developing μN thruster systems that operate in a continuous manner is also preferred (over pulsed thruster systems) when considering applications to small satellites (e.g. micro, nano and pico-satellites) since pulsed systems would induce larger perturbations on the spacecraft, which could interfere with the mission objectives.

Conclusions

This paper gives an overview of the status of currently available thruster technology. It highlights the technology that demonstrates the most potential for application within DFC systems or other missions that require high positional control or station keeping.

This work also discusses some of the weaknesses that are apparent in the various technologies. Further investigations into available technology from other fields are underway to assess the suitability for application to thruster designs. Some of the areas of technology that seem promising include continuous-thrust plasma and the application of lasers.

This paper also emphasizes the fact that small satellites seem to be the ideal candidates for testing, validating and ultimately qualifying such elaborate and exceptional precision thruster systems.

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Biography

Rachel Leach graduated from Leicester University in 1997, and completed her Ph.D in July 2001 at the University of Birmingham in Physics and Astronomy. Her Ph.D research was primarily focused on the development of hardware for future space-borne gravitational wave missions. In particular, her work concentrated on modeling responses of dynamic systems, which require extremely low drag (pico-g satellites) and formation flying satellites, which require high precision position control. She is currently working for Design_Net Engineering in Lakewood, Colorado as the assistant project manager of software and electronics development for the Low Temperature Micro-Gravity Physics Experiment Facility, which will be flown as a facility class experiment on the ISS. Her current research efforts are also focused on the design and development of a new class of micro-Newton thrusters.

Kerry L. Neal obtained a B.A. in physics from the University of Colorado in 1973, and later went on to obtain a B.S. and M.S. in electrical engineering from the University of Colorado in 1984 and 1987 respectively. Mr. Neal has broad interests and experience in designing and constructing space hardware. He has done engineering design work on ATS-6, ISEE-1 TIROS-N, PDP, APEX, SeaStar, FUSE, COS, and numerous other programs. He is currently working for Design_Net Engineering in Lakewood, Colorado as the lead electrical engineer for the Low Temperature Micro-Gravity Physics Experiment Facility.

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