AIAA 97-3058

Thruster Options for Microspacecraft: A Review and Evaluation of Existing Hardware and Emerging Technologies

Juergen Mueller"

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, CA 91109

State-of-the-art thruster technologies are reviewed and evaluated in view of potential microspacecraft applications. Microspacecraft are defined in this study as spacecraft with masses between 1 and 20 kg. Based on this review of existing technologies, future development needs for micropropulsion systems are defined and advanced new micropropulsion concepts especially designed with microspacecraft applications in mind are introduced. Of the state-of-the-art technologies, hydrazine thrusters and small solid rocket motors appear applicable to some microspacecraft. Cold gas systems may provide near-term solutions to attitude control, at the expense of leakage concerns and large and heavy propellant tankage. New thruster concepts, heavily relying on advanced microfabrication technologies have been designed and built at JPL, addressing some of the microspacecraft design challenges, and are introduced.

I. INTRODUCTION

Background and Significance

Within the National Aeronautics and Space Administration (NASA), a research and development initiative is currently underway to investigate the feasibility of microspacecraft in the 1-20 kg class'. The motivation behind this development is the desire to reduce launch masses in order to reduce mission costs and greatly increase launch rates. Launch costs for a typical interplanetary mission may be as high as **30%** of the overall mission cost, and these costs may be reduced significantly as a result of substantially reduced spacecraft masses.

In addition, microspacecraft mission scenarios may be envisioned where, rather than launching a single large spacecraft, the mission is accomplished by a fleet of several smaller microspacecraft, with the scientific payload distributed among the micro-craft to reduce mission risk. Loss of one microspacecraft would not eliminate the entire mission. A fleet of several small microspacecraft, possibly in connection with a larger "mother''-spacecraft, could also increase mission flexibility. For example, the smaller microspacecraft could be placed on different trajectories around the target planet and provide an almost instantaneous, global survey of the target. A "mother-craft" could also release smaller micro-craft to perform riskier portions of a mission. For example, a close-up investigation of Saturn's ring objects may be envisioned with a swarm of microspacecraft decending into Saturn's rings while the "mother-craft", providing high-data rate communication to earth via a large high-gain antenna, may cruise in a safe distance from the rings.

Building microspacecraft in the 1-20 kg class, however, will necessitate the miniaturization of every subsystem in order to maintain the high degree of onboard capability required to ensure an acceptable scientific return for the mission. One of the sub-systems that will be included in such a reduction in weight and size is propulsion. Although in the past many very small spacecraft have lacked propulsion systems altogether, future micro-spacecraft will likely require significant propulsion capability in order to provide a high degree of maneuverability and capability. In particular, interplanetary mission scenarios will require propulsion capability on microspacecraft for course corrections as well as attitude control to accurately point the spacecraft for observation or communication'. Attitude control in low Earth orbit is possibly achievable via other means, such as magnetic torquers, however propulsive capability is required in higher orbits, interplanetary space or around some other planet, either to directly control the spacecraft's attitude or to off-load momentum wheels[3]. In addition, very small spacecraft are often launched in a "piggy-back" configuration together with larger spacecraft to save launch costs. Propulsive

" Member of Technical Staff, Advanced Propulsion Technology Group, Member AIAA capability may be required onboard the microspacecraft to adjust its trajectory according to the desired mission objective[3].

In order to meet microspacecraft propulsion requirements, the use of lightweight, small sized, low-thrust and small impulse bit (I-bit) systems will be needed. It is the purpose of this paper to review and evaluate existing propulsion hardware and emerging micropropulsion technologies with respect to their applicability to microspacecraft. This paper is an extension of an earlier study performed by the author at the Jet Propulsion Laboratory (JPL), distributed JPL-internally in late 1995[4]. The JPL study contained company-discreet information and was therefore not available for public release. In the paper presented here, company-sensitive information has been eliminated and new information gained in the emerging field of micropropulsion since the earlier study has been added

Definitions

There exists a wide variety of opinions regarding the appropriate definition of what a microspacecraft is. Table 1 below gives a definition with respect to mass, size and power of the type of microspacecraft that are considered in this study. To simplify this discussion, three microspacecraft classes, Class I through III, have been defined. These microspacecraft classes distinguish themselves from each other by their mass, power and size ranges. These should be interpreted as approximate values.

Class I spacecraft, ranging in mass from about 5-20 kg, are characterized by the fact that they may still be able to use the smallest propulsion hardware available today, or currently under substantial development. This hardware will be conventionally integrated by interconnecting it via conventional feed lines. Development of some new propulsion hardware, taking miniaturization to new extremes and possibly incorporating advanced microfabrication techniques, such as MEMS Wicro-Electromechanical Systems) technologies, may also be required.

Figure 1 shows an example of a Class I-type spacecraft, designed and built at JPL and referred to as the JPL 2nd Generation Microspacecraft[1,5]. This craft, in its current design iteration, has a mass of around 7-8 kg. The 2nd Generation Microspacecraft is not designed for flight, but, rather, it is an evolutionary functional model of such a craft, with subsystem hardware constantly being upgraded to more “flight-like” versions[5]. Based on the 2nd Generation Microspacecraft design regime, power densities for microspacecraft have been estimated at 1 W/kg, resulting in 10 W and 20W onboard power for the two Class I type microspacecraft (10 kg and 20 kg) listed in Table 1. Low- and high-mass versions of a Class I microspacecraft will be considered.

Microspacecraft with masses between 1 and 5 kg have been categorized here as Class II microspacecraft. In the case of Class II microspacecraft, development of new, extremely miniaturized propulsion components, both for delta-v maneuvers and in particular for attitude control,

Table 1: Definition and Classifications of Microspacecraft for the Purpose of the Study

Micro S/C Class	S/C Mass (kg)	S/C Power (W)	S/C Dimension (m)	Comments
I	20	20	0.4	Use conventional components, possibly MEMS. Conventional integration (feed lines).
	10	10	0,3	Same as above.
II	1	1	0.1	MEMS components, high level of integration between components of each subsystem and possibly between subsystems.
III	<<1	<<1	0.03	All MEMS. Very high level of integration between all subsystems and within sub-systems required. Strong feasibility issues. Not considered in this

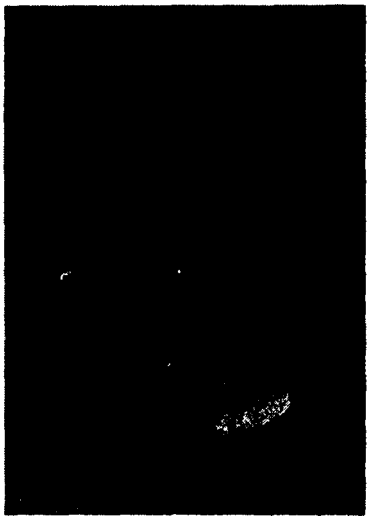


Fig. 1: JPL Second Generation Microspacecraft

where multiple clusters of small thrusters are used, will be required, These devices almost certainly will employ MEMS technologies in some fashion. Also, because of the severe volume constraints on such a spacecraft, a high level of integration will be required between different propulsion components, and between propulsion and other spacecraft subsystems. For example, in the case of MEMS based technologies, several propulsion components, such as thrusters and valves, plus the required control electronics, may be integrated onto a single chip, or a 3D-stack of chips. Integration approaches of this kind, not limited to propulsion however, are currently being addressed by JPL's Center for Integrated Space Microsystems (CISM). This requirement for an increased level of integration, in addition to an even more pronounced degree of miniaturization over the smallest availbale state-of-the-art propulsion hardware makes this Class II category of microspacecraft different in its design requirements from the Class I microspacecraft.

Even smaller microspacecraft with total masses of significantly less than 1 kg, categorized as Class III microspacecraft, have recently been studied at JPL[6], these however, will not be considered in this study since the design concepts discussed in Ref. 5 did not require propulsion. If propulsion needs should arise for Class III spacecraft, they will certainly require MEMS-based technologies". These propulsion systems would likely be based on significantly scaled down versions of MEMS-based Class II systems.

Scope of this Study

The goal of this study is to review current propulsion technology in view of its applicability to Class I and II microspacecraft, identify future technology needs and to outline potential future thruster technology currently emerging, aimed at meeting these needs, Microspacecraft mission scenarios may involve a variety of propulsive maneuvers, such as attitude control, course correction, delta-v maneuvers, orbit insertion or even landing and take-off from a distant planet. Depending on the maneuver and delta-v requirement, different propulsion technologies will be needed . This paper will focus on relatively low-thrust propulsion systems that can be integrated with a microspacecraft bus of either Class I or II type for the purposes of attitude control and delta-v maneuvers.

If take-off and landing operations are considered for microspacecraft, a class of propulsion devices very different from those to be considered here will be required, Given the large delta-v requirements associated with landing and take-off operations, even for relatively small payloads, fairly large chemical stages well exceeding the mass limits considered here may result. As a consequence, high-thrust, and thus high-flow, devices will be required. The need to sustain high propellant flow rates will not allow for miniaturization of propulsion components used in these applications significantly beyond sizes already available today (although significant research will have to be &voted to such areas as further component mass reduction and alternate propellant usage). These devices are not considered micropropulsion devices in the context of this study and are therefore not included in the following discussion.

Micropropulsion subsystems will not only consist of thrusters, but will also require miniature feed system components, such as valves, tanks, and pressure regulators, etc. In some of these areas considerable design challenges arise during miniaturization. In particular the moving parts in valves make miniaturization difficult. However, already the thruster material to be reviewed is so vast, that surveying miniature components in addition to thruster technologies could not be accommodated in this study. Evaluation of miniature components will, thus, not be included in this review.

IL REPRESENTATIVE MISSION REQUIREMENTS

Requirements for microspacecraft missions are difficult to predict accurately at this early stage of their development and will vary greatly given the multitude of conceivable missions, In this section, an attempt is made to present a class of representative mission requirements, based on our current understanding of these small spacecraft requirements, to serve as a basis for micropropulsion technology evaluation. Both attitude control and delta-v requirements (landing and take-off excluded, see Section I) have been considered. These mission requirements are based on estimates provided in a recent workshop on micropropulsion, held at NASA's Jet Propulsion Laboratory (JPL)[7]. As concrete future microspacecraft missions develop, this set of mission requirements will no doubt have to be modified.

Delta-v Requirements

Obviously, delta-v requirements do not depend on the size of the spacecraft and therefore requirements for four representative small spacecraft missions, currently under investigation at JPL, have been listed below in Tables 2 through 5 to serve as a reference. These missions include a small body (asteroid) rendezvous, an outer planet (Europa) orbiter, a spacecraft formation flight (DS-3) and an earth-observing cluster.

Inspecting Tables 2 and 3, the large delta-v requirements for deep space missions becomes immediately apparent. Electric propulsion applications result in even larger delta-v-requirements due to bum losses[8], which must be offset by the higher specific impulses and more fuel-efficient operation of electric engines. Use of electric propulsion may either lead to shorter trip times, or reduced spacecraft masses, or both. The benefit of using electric propulsion in regard to spacecraft mass reduction will likely be even more important for mass limited microspacecraft missions.

An additional requirement for chemical primary propulsion is the need to maintain large enough thrust-to-spacecraft weight ratios. Values around 0.1 -0.3 are typical. Too small a thrust-to-weight ratio will again lead to bum losses[8] and increase the required delta-v. Since in (he case of chemical engines this increased delta-v requirement cannot be easily offset by a sufficiently large specific impulse, too low a thrust for chemical primary propulsion maneuvers must bc avoided. Too large a thrust value, on the other hand, may generate accelerations too large to be tolerated by the spacecraft structure, in particular at times well into the mission, when portions of the spacecraft structures may be deployed.

Attitude Control Requirements

In order to estimate attitude control requirements the following assumptions were made': (1) fine pointing requirements are assumed, defined by the desire to stay within a 0.2-2 mrad deadband and ACS firings occuring no more frequently than one couple firing every 20-60 see; (2) Slew rates of 180°/minute required with one couple of thrusters firing. The spacecraft was assumed to be cubical in shape with the side of the cube being equal in length to the dimension listed in the forth column of Table 1. The resulting minimum impulse bit and minimum thrust requirements are listed in Table 6 for microspacecraft masses of 1, 10 and 20 kg. Very small impulse bit requirements can be noted. It should be pointed out in this context, that the fine pointing requirements given above are not extreme for today's spacecraft.

III. REVIEW OF THRUSTER TECHNOLOGIES

In this section, state-of-the-art thruster hardware, either available "off-the-shelf" or under significant development, will be reviewed and evaluated in view of microspacecraft applications. This section is structured into two main parts , focusing on primary and attitude control applications, respectively. Both primary and attitude control sections have been further sub-divided into chemical and electric thruster sections.

Primary Thrusters - Chemical

Bi-Propellant Engines

Bi-propellant engines are most commonly used for primary propulsion applications of conventional spacecraft today due to its relatively high specific impulse performances and considerable flight heritage, Advantages of hi-propellant engines over other chemical systems, such as mono-propellant thrusters, are their higher specific impulse performance, leading to lower fuel weights. Disadvantages are their relative complexity, both with respect to engine technology and the feed system. Since separate feed systems for fuel and oxidizer, as well as pressurants are required, component part count is high, leading to large propulsion system dry masses. Therefore, bi-propellnat systems are usually used on mission requiring large delta-v's (> 1,000 m/s) and large spacecraft,

Table 2: Delta-v Requirements for a Vesta Rendezvous Mission[7]

Delta-v Requirements (m/s) Primary		ACS***	Mission Duration (yrs)	
Chemical	Electric		Chemical	Electrical
3400”		2.5-25	4.3	
	7000**	**2.5-25**		**2.7**

* Assumes Mars Gravity Assist (MGA) trajectory
** Assumes direct trajectory
*** Assumes 250 slews (180) and 3-axis stabilization in the range of 1-20 kg S/C mass, thrust 0.5-10 mN.

Table 3: Delta-v Requirements for a Europa Orbiter Mission’

Delta-v Requirements (m/s) Primary		ACS***	Mission Duration (yrs)	
Chemical	Electric		Chemical	Electrical
2500”		5-50	4.8	
945	5500**	5-50		5.8

*
** Assumes Solar Electric Venus Venus Gravity Assist (Se-VVG) and 345 ins/s chemical delta-v required for Jupiter Orbit Insertion (JOI) and 600 m/s for Earth Orbit Insertion (EOI).
*** Assume 500180° slews and 3-axis stabilization (S/C masses ranging between 1-20 kg and thrusts between 0.5 -10 mN). ACS requirements may be higher for low thrust trajectories.

Table 4: Delta-v Requirements for the Deep Space Interferometer Mission (DS-3)[7]

Delta-v Requirements (m/s) Primary	ACS	Mission Duration (yrs)
	100-300	0.5 -1.0

Table 5: Delta-v requirements for an Earth Observing Cluster’

Delta-v Requirements (m/s) Primary”		ACS**	Mission Duration (yrs)	
Chemical	Electric		Chemical	Electrical
500		5-50	5.0	
	550	5-50		5.0

* Primary delta-v assumes 200 m/s for non-Keplerian orbit, 250 m/s for NSSK, and 50-100 m/s for phasing.
** ACS assumes 500 slews of 180° and 3-axis stabilization (S/C masses ranging between 1 -20 kg and thrusts ranging between 0.5 -10 mN).

Table 6: Representative Attitude Control Requirements for Microspacecraft[7]

S/C Mass (kg)	S/C Dimension’ (m)	Ibit (mNs)	T_{min} (mN)
20	0.4	0.013	4.65
10	0.3	0.005	1.75
1	0.1	0.0002	0.06

* Assume cubical spacecraft shape

Consequently, most hi-propellant engines that have been built today provide fairly large thrust levels. Some smaller engines in the 5- 22 N (1-5 lbf) thrust range have been built or are under development, Up to this point, applications for these engines have been envisioned in the use of attitude control purposes of larger spacecraft in order to simplify the overall propulsion sytem, eliminating separate attitude control propellant tanks. Given these goals, considerable effort was devoted to fast thruster response times and short impulse bits[9-13].

Challenges encountered when building hi-propellant engines of such a small size, include combustion efficiency losses due to the potential of reduced mixing and vaporization, thermal control issues of chambers, nozzle throats and injector heads and related material issues, injector design issues and related accurate mixture ratio control issues, and possibly, spacecraft contamination issues due the potential of incomplete mixing and vaporization inside the thrust chamber.

Mixing and vaporization losses can occur in small engines due to the small chamber size. In general, good vaporization is obtained in longer chambers, at higher chamber pressures and for smaller injector orifice sizes[14], while better mixing is achieved in engines having high chamber-length-to--diameter ratios and a larger number of injector inlets[14]. In addition, engine size also plays a role 'n the mixing of the propellants[14]. Smaller engines have lower chamber flow Reynolds numbers and thus lead to less turbulent chambers, reducing mixing. The limitations imposed on chamber length and diameter has an immediate impact on the degree of miniaturization of a bi-propellant engine.

Thermal control of small hi-propellant engines is another key design issue. Film cooling, or boundary layer cooling (BLC), is often employed in hi-propellant engines to keep the chamber wall temperature within its thermal and structural design limits. Here, a fuel is injected close to the chamber wall. Since the propellant mixture is fuel rich, it does not burn completely and will shield the chamber walls from the heat output of the combustion reactions occurring closer to the center of the chamber. However, at the same time combustion efficiencies are reduced due to incomplete combustion. While for more conventionally sized engines about 15 - 30 % of the fuel is commonly used for film cooling, these values may reach up to 30-4090 for smaller engines in the 22-N class, causing performance losses[12] and possibly resulting in spacecraft contamination concerns due to the possibility that liquid fuel droplets may attach themselves to sensitive surfaces (optical lenses, solar cells, etc.).

Elimination of film cooling was done in the small bi-propellant attitude control engines developed by Rockwell for the Kinetic Energy Anti-Satellite (KE ASAT) program.''" [1] This increases combustion efficiency and decreases injector head complexity since no separate BLC holes are required, and should facilitate miniaturization, However, in order to survive the punishing thermal environment, high temperature chamber materials have to be employed. In the case of the KE ASAT technology[10,11], a carbon-silicon carbide chamber was used. Despite use of this high temperature material, engine bum durations in a single burn where limited to only a little over 20 seconds. Another chamber material under significant investigation is rhenium-iridium composite material. Rhenium is uses as the substrate material because of its high melting point (3453 K)[12] and coated with a iridium layer for chemical inertness. Iridium has a coefficient of thermal expansion (CTE) closely matching that of rhenium and a high melting point of 2727 K[12]. Using this chamber material, specific impulses in excess of 300 sec have been obtained in a 22 N thrust chamber over burn durations of a maximum of 350 seconds[12].

Injector design also requires careful attention in small hi-propellant rocket engine development. Due to the small flow cross sections encountered in small engines, flow rate control, and thus mixture rate control [15], as well as misalignment of impinging propellant jets[12] could lead to poor engine performance repeatability or engine reliability problems. In addition, thermal management of the injector head is important to ensure that heat diffusion from the hot chamber material to the injector head is minimized in order to prevent vaporization of propellants in the injector and not to exceed thermal limits of the injector material, which may be different from the high-temperature chamber material for machining reasons. Unlike-doublet injector types are favoured[14, 16] because of better mixing results and reduced heat load to the injector head by displacing the flame front away from the injector wall surfaces[16]. As mentioned above, more injector elements will lead to better mixing, however, limited engine size may limit the number of injector elements. In the case of the Rockwell engine discussed above, only a single unlike-doublet injector element is used[10,11] (combustion efficiencies are maintained at high levels due to the aforementioned elimination of the BLC layer).

Table 7 lists the smallest hi-propellant engine technology available today. Caution has to be exercised when referring to the data presented in this table. Not all engines listed are space-qualified at this point and some may still be experiencing problems in their respective development programs. Schwende et al. " point out in their 1993 paper that the 4 N engine experienced "anomalies" due

Table 7: Comparison of Small Bi-Propellant Engines

Thrust (N)	Manu-facturer	Type	Fuel/ Oxidizer	Isp (s)	Weight (kg)	Size (length* Max. Dia) (cm)	Comments	Ref
4	DASA	-	MMH/ MON-1	285	0.27		Reported temperature anomalies	13
4.45	Marquardt	R-21 R-2B	MMH/NTO	280	0.43	26.1 X <9	No known applications	17
10	DASA	-	MMH/ MON-1	290	0.3		Regenerative cooled throat in previous version. 34 flight units built.	13
10	Marquardt	R-52	MMH/NTO	295	-		No known applications	17
10	Royal Ordnance	LTT	MMH/NTO	-				18
22	Marquardt	R-6C/ R-6D	MMH or N_2H_4/NTO	289	'0.67	25x<13	Flight Applications for R-6C, no known application R-6D	16
22	Atlantic Research	A0809	MMH/NTO	290	0.55	21.7x5.4	Flight Applications	19
22	Aerojet	SSD	MMH/NTO	2 8 0	0.59	18.5x6.9	-	20
22	Aerojet	-	MMH/NTO	313	-		Rh/Ir chamber. In development	12
22	Royal Ordnance	Leros 20H	N_2H_4/MON	285	0.85	20.9x6.6	Under development	18
30	Rockwell	-	MMH/NTO	287	0.1		Max. 26 sec in single burn. Max. accumulative bum: 77 sec. 1.25 mixture ratio. Developed for BMDO	10, 11
156	Marquardt	Divert	N_2H_4/NTO	-	0.1		20 sec single burn demonstrated Developed for LEAP	9

to too high chamber wall temperatures. Also, although commonly referred to as examples for the degree of miniaturization achieved for hi-propellant engines, the KE ASAT engines have been tested only up to 26 seconds in single duration burns which is too short for interplanetary delta-v maneuvers. As mentioned above, the KE ASAT developments, as well as others, have focused on attitude control applications, rather than primary propulsion applications. As a result, pulsing performances rather than long duration burns were emphasized and led to the currently exhibited design performances. Almost all engines use nitrogen tetroxide (NTO) and monomethylhydrazine (MMH) as oxidizer and fuels, respectively, due to storage reasons, acceptable performance values and relatively benign mixture ratio sensitivities. Using a O/F mixture ratio of 1.6 results in equal propellant volumes for fuel and oxidizer, so that identical tanks can be used (reducing development cost and time) and the spacecraft will experience no center-of-gravity (e.g.) shifts during burns.

Given the engine data presented in Table 7, and using the representative mission requirements of Section II, propellant mass fractions of 0.7 and 0.58 can be computed for a spacecraft with delta-v requirements of 3400 m/s and 2500 m/s, respectively, at a specific impulse of 290 s (see Table 7). Although these values are not atypical for larger spacecraft, for a microspacecraft they may be too high given that component-mass-to-spacecraft-mass ratios are larger. For example, in the 3400 m/s case for a 20 kg spacecraft, merely 6 kg of available mass remains. This will have to include the entire dry mass of the propulsion system, structure plus all other subsystems. According to Table 7, one hi-propellant engine alone may take up about 5% of that mass, even using the lightest engines available.

Smaller delta-v requirements around 1000 m/s, on the other hand, would result in propellant mass fractions of 0.3 for a 290 s hi-propellant system. However, a hydrazine mono-propellant system with a specific impulse of 220 s (see below) would result in a propellant mass fraction of 0.37. In the case of a 10 or 20 kg Class I spacecraft, this difference would be a mere 0.7 or 1.4 kg in propellant mass, respectively. Given the lower component part count of a mono-propellant feed system, this higher propellant fraction can easily be offset by a simpler mono-propellant system.

It is, therefore, concluded that hi-propellant systems are not suitable for either high or low delta-v requirements onboard a microspacecraft due to too high dry weight of the system. Further, aggressive miniaturization may help, however, there exists considerable doubt that significantly smaller, yet reliable and space-qualifiable hi-propellant engine technology can be developed in view of the design challenges for small bi-propellant engines given above.

However, separate chemical stages for large delta-v maneuvers may make use of hi-propellant engines. An example of such a stage is given in Ref. 21, describing a Hydrazine (N_2H_4)/ Chlorine Pentafluoride (ClF_5) chemical upper stage, developed for the Lightweight Exo-Atmospheric Projectile (LEAP) program. Thrust levels provided by the LEAP stage are somewhat high (2056 N) for microspacecraft applications and there are concerns regarding the corrosivity and toxicity of ClF_5. Stages like these, or similar ones using more conventional propellants, however, may be required for orbit insertion maneuvers around distant planetary bodies, in particular when these bodies are lacking an atmosphere (no aerobraking possible) or when they are located too far from the sun (solar power levels too low to use solar electric propulsion). In addition, landing and take-off operations will require hi-propellant technology. However, those mission applications may require thrust levels well exceeding those obtainable with the engines listed in Table 7 due to high stage masses and large required vehicle accelerations to overcome the gravity of the respective planetary body.

Mono-Propellant Hydrazine Engines

Hydrazine mono-propellant thrusters combine engine technology substantially simpler than that of bi-propellant engines, relatively simple and low part-count feed systems, and high reliability with intermediate performance (specific impulses around 220 s for state-of-the art hydrazine thruster technology). In a hydrazine thruster, the propellant is passed through a catalyst bed and decomposed. The decomposition products are nitrogen, hydrogen and ammonia. The reaction takes place in two stages: hydrazine decomposes first through an exothermic reaction into ammonia and nitrogen. The ammonia then decomposes further through an endothermic reaction into hydrogen and nitrogen, however, leaving the overall reaction exothermic. The degree of ammonia decomposition depends on many factors, among them feed pressure, catalyst type and geometry. Shell 405 is the standard catalyst used in the US, consisting of 1.5 - 3 mm dia. alumina pellets coated with iridium. The catalyst pellets are contained within a mesh construction in a so called catalyst bed, Upon contact with the iridium surfaces, the hydrazine decomposition reaction is initiated.

Hydrazine thrusters have been used extensively on conventional spacecraft for attitude control as well as primary propulsion sources for intermediate to low delta-v requirements (about 1000 m/s or less). Of interest here are the smallest available hydrazine thrusters, in the 0.9 -4.45 N range. These engines are being manufactured in the US by - Primex (formerly Olin Aerospace/Rocket Research), Kaiser-Marquardt and TRW companies, and abroad by Daimler Benz Aerospace in Germany. Typical engine characteristics are listed in Table 8.

The engine sizes, weights and thrust levels should allow for relatively easy integration *on* a Class I

Table 8: State-of-the-Art US Hydrazine Thrusters

Thrust (N)	Manufacturer	Type	Isp (s)	Weight (kg)	Size (Lengthx Max. Dia) (cm)	Comments	Ref.
0.9	Primex	MR-103	210-220	0.33	14.8x 3.4	C/E, D and G models, Isp and thrust feed pressure dependent (340 - <100 psia). Considerable flight use.	22
0.9	Marquardt	KMHS 10	226	0.33	14.6x3.2	Isp and thrust feed pressure dependent, Flight use	17
2.2	Primex	MR-111E	213-224	0,33	16.9x3.8	Isp and thrust feed pressure dependent (370 -60 psia). Considerable flight use.	22
4.45	Primex	MR-111C	226-229	0.33	16.9x3.8	Isp and thrust feed pressure dependent (400-80 psia). Considerable flight use.	22
4.45	Marquardt	KMHS 17	230	0.38	20.3x3.2	lsp and thrust feed pressure dependent, Flight use.	17
5	lRW	MRE-1	220	0.82	15.2xN/A	Mass is for dual thruster module. Isp and thrust feed pressure dependent. Considerable flight use.	23
18	TRW	MRE-4	230	0.41	20.3x unknown	Isp and thrust feed pressure dependent, Considerable flight use.	23

microspacecraft bus, mounted along the axis of the bus for primary propulsion applications. All thrusters listed have seen considerable flight use and potentailly would require only minimal re-development for use as Class I main engines. Class H microspacecraft are too small to take advantage of these existing technologies.

One area of improvement in the use of state-of-the-art hydrazine thrusters as Class I main engines may be found in the valve area. Currently, a considerable weight fraction of a small hydrazine thruster is taken up by the thruster valve (greater than 50$% for the small engines considered here). This fact may open an opportunity for further weight reductions. Since the smallest hydrazine thrusters have been used mainly for attitude control purposes where fast valve action is essential (on the order of 15 ms on/oft_), these valves could possibly be replaced by slower valves, since primary propulsion applications of these thrusters, as envisioned here for microspacecraft, seldom would require engine pulses that short. Slower valves, depending on design, may require less electromagnetic force action to open the valve, which might reduce electromagnet masses.

A disadvantage of hydrazine propellant is its toxicity and flammability and resulting ground handling and propellant loading procedures. These procedures are obviously well established due to extensive hydrazine thruster use on conventional spacecraft, but will significantly contribute to the cost of small spacecraft. In addition, as was pointed out in the preceding section, a hydrazine propulsion system onboard a microspacecraft is only practical if small or intermediate delta-v maneuvers are required (i.e. < 1000 m/s). In these cases, mono-propellant systems will have an advantage over hi-propellant systems due to reduced system complexity, smaller component part count and, thus, smaller volume requirements. If higher delta-v's are required (see Section II), mono-propellant systems become increasingly heavy due to large propellant requirements. In these cases, hi-propellant engines, likely to be mounted on a separate kick-stage, or electric thruster options (see below) should be preferred. Thus, limited Class I microspacecraft use of existing hydrazine thruster technology for primary propulsion applications appears reasonable .

HAN-based Mono-Propellant Thrusters

Recently, so called HAN/TEAN thrusters[4,24] have received attention. This thruster is of the mono-propellant type, using mixtures of HAN (Hydroxylammonium Nitrate - NH_3OH+NO_3.), TEAN (Triethanolammonium Nitrate - $(HOCH_2)_3HNOH+NO_3$.) and water as a propellant. HAN/DEHAN mixtures have also been studied, consisting of HAN, water and DEHAN (Diethylhydroxylammonium Nitrate - $(CH_3CH_2)HNOH+NO_3$.)[24]. HAN is an oxygen rich component and TEAN or DEHAN are fuel rich components. Due to the water additive, both components can coexist in a mixture without detonation, as long as the water content is maintained at 10% or above[25]. Exposing the mixture to a catalyst causes a chemical reaction and exothermic decomposition of the components into $C0_2$, N_2 and H_2O[4,24].

Mixtures of these propellants have been studied for use as liquid gun propellants by the US Army and have been categorized according to their composition. LP1 846, for example, consists of 60.8% HAN, 19.2% TEAN and 20% water, while LP1 845 consists of 63.2% HAN, 20% TEAN and 16.8% water[24], i.e. has a lower water content than LP 1846. The amount of water in the mixture greatly influences the decomposition temperature and, thus, the available specific impulse. Increasing the water content will lower the flame temperature. Jankowski[24] quotes flame temperatures, based on numerical calculations, of 2022 K for LP1 846 and 2125 K for LP 1845, having the lower water content. These values result into theoretical specific impulses, assuming a specific impulse efficiency of 92%, of 233 and 239 sec[24]. Tests performed with HAN-bawd propellants with different, not specified additives have resulted in specific impulses of 270 sec at a flame temperature of 2500 K[24].

The flame temperatures of HAN/TEAN combinations arc very high and approach values found in small hi-propellant engines. Thus, many of the thermal design challenges found in the construction of small bi-propellant chambers would have to be overcome when using high-performing HAN/TEAN mixtures with low water content. Engine lifetime restrictions may thus result if high performance is required.

However, as mentioned above, flame temperatures may be lowered at the expense of specific impulse performance if the water content is raised. Even though not providing a significant performance advantage over existing hydrazine mono-propellant thrusters in those cases anymore, HAN/TEAN thrusters still offer advantages due to the low toxicity of both the NAN/TEAN propellant as well as its reaction products, high storage densities (about 40% higher densities than that of hydrazine) and lower environmental temperature handling capabilities. While hydrazine freezes at about O C, HAN/TEAN mixtures may be used at temperatures as low as about -33 C[24]. At this point, the viscosity of HAN/TEAN mixtures increases and propellant feeding will no longer be possible using conventional feed system technologies[24].

Higher storage densities and lower environmental temperature handling capabilities of HAN/TEAN propellants are beneficial for microspacecraft, since they allow for smaller and lighter storage tanks and the elimination of, or reduction in power for, tank and line heaters, reducing overall power requirements for the spacecraft. Thus, HAN/TEAN thrusters may find Class I microspacecraft applications for those reasons. Reduction in engine size to meet Class II requirements may not be possible due to the high heat loads to be expected in a HAN/TEAN decomposition chamber. Considerable development work will be required to bring current HAN/TEAN thruster concepts to flight status, with reaction chamber thermal design issues being one of the most challenging steps in the development.

Other Mono-Propellant Thrusters

Hydrogen peroxide (H_2O_2) thrusters are considered from time to time as an alternative to more conventional mono-propellant systems[25,26]. Hydrogen peroxide, when subjected to a suitable catalyst, decomposes into water and oxygen in an exothermic reaction. Although hydrogen peroxide has been used in flight applications as a mono-propellant in the past[8,26-29], it is no longer in use due to propellant storability issues[8,26-29]. Hydrogen peroxide slowly decomposes when heated or exposed to a catalyst. Almost any organic substance can serve as such a catalyst[29]. If slow decomposition occurs in propellant tanks, as has been observed in the past[8], tank pressure increases result over time and propellant is lost due to its slow conversion into its reaction products inside the propellant tank.

Solid Rocket Motors

Solid rocket motors are frequently used in kick-stages for orbit raising or orbit insertion of spacecraft, beginning with the Explorer 1 spacecraft’ and leading to the more recent Pioneer-Venus[x]’, Magellan[8,30] and Galileo missions, as well as numerous commercial missions (orbit raising). In solid motors, fuel (typically aluminum powder), oxidizer (typically ammonium perchlorate - NH_4ClO_4) and an organic binder (typically Hydroxyl-terminated Polybutadiene - HTPB) are combined into a composite to form the solid propellant[8,30]. The advantages of solid rocket motors are their compact size combined with a relatively high specific impulse performance - less than that of bi-propellant systems but higher than that of mono-propellant systems[29]. For obvious reasons, solid motors also do not suffer from propellant leakage concerns. Propellant sublimation by exposure to space through an open nozzle was a concern for the use of solid motors for deep-space applications, but has been found to have no impact on motor performance after 10 - 15 months of in-space storage[8,30]. In the case of the Magellan mission, a Thiokol STAR 48B motor was tired for Venus orbit insertion after 462 days in space[30]. A Thiokol STAR 24 motor was fired after 6.5 months in space for the Venus orbit insertion of the Pioneer probe.[30]

Disadvantages of solid motors are that they are generally not restartable, and therefore do not allow for orbit trimming. If several delta-v burns are required it is necessary to stack multiple stages leading to system complexities and higher propulsion system dry masses. The issue of orbit trimming is of particular importance for solid motors since exact prediction of delivered total impulse is difficult to estimate due to uncertainties in the expected grain temperature, the exact propellant composition, and the amount of inert material consumed[8]. Thus, a separate small liquid system may have to provide for the orbit trimming maneuvers A separate liquid system may also be required for despin of the satellite. Solid motors for space applications are usually not equipped with thrust vectoring capability. Although some larger motors have been tested with such a capability [32], these nozzle gimbal systems may be too heavy and complex for very small motors, such as those required for microspacecraft appl i cat ions. Thus, spacecraft generally are spin-stabilized before motor firings, and despin may be required after separation from the stage depending on the mission,.

Table 9 shows some of the smallest solid motors available today[30-33]. In addition to the companies listed in Table 9, Pacific Scientific, Inc. is also building small rocket engines for missile divert purposes[34]. As can be seen by inspecting Table 9, envelopes and masses of the smallest available motors fit within the Class I category of microspacecraft and specific impulse performances are quite good. However, thrust levels are much higher than desired and burn times are relatively short, which would lead to very large microspacecraft accelerations. For example, assuming a 20 kg overall spacecraft weight (incl. motor), using a Thiokol STAR 6B motor results in accelerations at the beginning of the burn of about 13 g’s and at the end of the burn around 18 g’s. The delta-v that can be achieved with this motor for a 20 kg spacecraft would be 963 m/s. A similar calculation for a 10 kg spacecraft equipped with the STAR 5A motor would lead to an initial acceleration of 1.7 g’s and an acceleration just prior to burn out of about 2.2 g’s, and a delta-v of 641 m/s. Achievable delta-v’s are limited by the reduced propellant mass fractions found typical for smaller motors.

The high thrust forces and short burn times are a result of the intended design applications for most of these small motors, i.e. stage separation or use as missile divert engines (missile attitude control). In both cases it is essential to provide a relatively large thrust in a short amount of time. In the case of microspacecraft applications, this could lead to limitations of solid motor use due to the requirement of being able to fire only in a stowed vehicle configuration (no deployments) and possibly costly re-qualification of spacecraft components to account for these high accelerations.

An exception to the fast burning, high-thrust small solid motors shown in Table 9 is the STAR 5A. Even though accelerations in the example given above are still quite high, those values may be much more tolerable. The longer burn time and smaller thrust of this motor were achieved by using an end burner propellant grain. This grain type was extensively used in the past in the so called JATO[8] (Jet Assisted take-Off) units used in the 1940s to assist in the take-off of aircraft from short runways or assisting aircraft requiring additional thrust for heavy-lift capability. This grain type may be the grain of choice for microspacecraft applications. Even longer, lower-thrust burns could be accomplished if the length-to-diameter ratio of the motor case could be increased.

Solid rocket motors may thus present an interesting alternative to more complex liquid systems where mission profiles are simple, require only single burns and intermediate delta-v values (< 1000 m/s), a small liquid system is onboard the microspacecraft for orbit trimming or if the required accuracy of the actually delivered delta-v is not too high. For example, in the Saturn-ring explorer scenario discussed in Section 1[2], quite sizable delta-v changes may be achieved for the microspacecraft probes using small solid motors. Since multiple probes are being used anyway to

Table 9: State-of-the-Art Small Solid Rocket Motors

Manufacturer	Type	Loaded Weight (kg)	Propellant Weight (kg)	Size (LxD) (cm)	Burn Time"	Isp (s)	Thrust"' (N)
Thiokol[32]	STAR 5A"'	4.7	2.3	22.5x13	32	250	169
Thiokol[32]	STAR 5C	4.5	2.1	34x12	2.8	266	1953
Thiokol[32]	STAR 5CB	4.5	2.1	34x12	2.67	270	2041
Thiokol[32]	'STAR 6B	10.3	6.1	40x18.6	5.9	273	2513
Atlantic Research'[1]		0.4			1	-	222
Atlantic Research'		0.5			1	-	311
Atlantic Research]'		1.6					952

· Bum time, 10% thrust at ignition, 90% thrust at shut-down
"" Bum time averaged
""' End Burner

account for the potential loss of some, total impulse and delivered delta-v uncertainties for individual probes may be acceptable.

The absence of leakage concerns and the ability to compactly package solid motors will be attractive for microspacecraft applications Existing motor hardware appears to fit the envelope of Class I spacecraft, although longer bum times, lower thrust values and, thus, lower vehicle accelerations should be aimed for. The benefits of using small solid rocket motors would be even more pronounced for smaller microspacecraft, such as types falling into the Class II-category. Here, compactness plays an even greater role than for Class I craft. Class 11 application of solid motors would require further miniaturization of solid motor technology and dedicated full development programs to achieve the desired reductions in size, weight and thrust.

Hybrid Rocket Motors

In a hybrid rocket motor a solid fuel is combined with a liquid or gaseous oxidizer, which is stored in a separate propellant tank and fed into the motor case[27,35,36]. As a result of this separation between solid fuel and liquid oxidizer, hybrid rockets exhibit some interesting properties. Hybrid rockets are restartable, relatively safe when compared with solid motors, offer appreciable specific impulse performances up to around 300 s when using storable propellants, and still offer a higher degree of compactness than hi-propellant systems. Hybrid rockets, at first glance, may thus seem to be an attractive cross between high performing hi-propellant engines and compact solid motor technology.

One of the disadvantages of hybrid motors, in particular when viewed in terms of space applications (microspacecraft or otherwise) with long mission durations, is the limited choice of suitable storable propellant combinations available today. Typically, Hydroxyl-terminated Polybutadiene (HTPB) is being used as fuel and liquid oxygen (LO_2) or hydrogen peroxide (H_2O_2) are used as oxidizers[27,28,3 5-37]. Although both oxidizers may be suitable for launch applications, they are not storable over long periods of time because they are either cryogenic (LO_2) or may slowly decompose over time in the propellant tank before use (H_2O_2 - see above). Among storable oxidizer options, nitrogen tetroxide (NTO) was used, but required a separate ignition source27 Chlorine-fluorides, such as ClF_3 and ClF_5, have also been used as oxidants. These substances, however, are highly toxic and corrosive.

Work on hybrid rocket motors started in the 1930's to 40's in both Germany and the US, with some of the early work in Germany performed by Hermann Oberth[35,36]. Development has continued on an on-and-off basis over the years. Focus was placed mainly on launch applications, leading to the development of the H-500 (312,000 N thrust) and the H-250F (1,000,000 N thrust) engines developed by the recently failed AMROC company[35]. Both of the latter two motors used HTPB and $L0_2$ as propellants. Extensive research on hybrid rockets has also been performed at various university research laboratories around the world[28]. This work was performed on smaller test devices. The smallest quoted thrust value for a hybrid rocket engine is found in Sellers et al. *"at 10N.

Given that attention in hybrid rocket engine development was focused mostly on large launch motors, where oxidizer storage issues play a lesser role than for in-space applications, it is uncertain whether hybrid rockets may find space applications for interplanetary missions . In particular microspacecraft applications, requiring small motor technology comparable in size to solid motor technology given in Table 9, remain uncertain. The use of hybrid engines for these applications would require a full development program, starting at the point of propellant selection. If successful, hybrid engine technology could fill a useful gap between high performing, yet complex bi-propellant engines and compact and simple, yet relatively inflexible solid motor technology.

Primary Propulsion - Electric

Ion Engines

In an ion engine, the propellant (typically xenon) is ionized in a plasma discharge. Ions are extracted from the plasma via electrostatic forces and accelerated across an electric potential difference of about 1 kV. In the process, xenon ions achieve a velocity of about 30,000 m/s, corresponding to a specific impulse of about 3,000 sec.

An ion propulsion subsystem consists of several components that all will have to be miniaturized for microspacecraft applications. These are the thruster itself, the power conditioning unit providing the required voltages to the engine, and the feed system. Within the thruster assembly, critical components include the cathode for certain engine types, the accelerator grid system and the neutralizer, (used to neutralize the ion beam to avoid charging the spacecraft). Different types of ion engines are being developed . DC electron bombardment types use an electron current emitted from a hollow cathode inside the engine body to ionize the propellant gas by causing collisions between the electrons and the propellant gas atoms. RF (radio-frequency) electron bombardment engines use electrons accelerated in an inductive coupled RF field to cause propellant ionization.

Advantages of ion engines are their large specific impulses which translate into significant propellant and spacecraft mass reductions. This fact is of particular importance for mass constrained microspacecraft, especially for interplanetary missions which have large delta-v requirements. Using ion engine technology may lead to lighter overall spacecraft masses and shorter mission trip times when compared with chemical hi-propellant systems. In addition, xenon propellant, when stored at about 2,000 psia pressure, takes on a supercritical state with a density about twice that of water. Reduced propellant requirements due to the high specific impulse of the engine and high density will allow for compact propellant storage.

On the other hand, propellant mass reductions due to higher specific impulses will have to be traded off with electric power requirements. Power requirements drive power conditioning unit and power supply masses. In order to reduce overall system wet masses, an optimum operating point must be selected for the engine, allowing for both significant propellant savings and low power system masses. For typical interplanetary mission requirements and current power system technology, this optimum is typically found around 3,000s specific impulse.

Table 10 lists the smallest available ion engine technology today [38-41]. As can be seen, all current ion engine systems are too large for use on a microspacecraft, both with respect to mass (incl. PPU) and power requirements. These engines may, however, be used on separate electrical stages just like the hi-propellant thruster technology discussed above, and provide large delta-v changes for Class I - type microspacecraft in that configuration. In the case of electric stages, a dedicated power supply will have to be provided.

Table 10: State-of-the-Art Small Ion Engines

Manufacturer	Hughes[40]	DASA[38,39]	JPL[41]
Discharge Type	DC	RF	DC
Thrust (mN)	17.8	5-15	21-31
Isp (s)	2585	3000	2500-3900
Power (W)	**439**	**240 (5 mN - 600 (15 mN)**	500-900
Thruster Mass (kg)	**5.0**	**1.6**	2.5
PPU Mass (kg)	**6.8**	**8.0** (PPU) 1.3 (RF generator)	
Beam Diameter (cm)	**13**	10	15

In order to integrate ion engine technology onboard a microspacecraft bus within the mass margins provided in Section II, new technologies will have to be developed. Thruster sizes will have to be reduced Challenges to be overcome here will be to maintain plasmas in small discharge chambers, where electron wall losses may be high. Cathodes and neutralizers will have to be miniaturized and cold cathode technology may be explored. Micromachined grid systems, allowing for electrostatic beam steering, eliminating heavy engine gimbals (see Section V) and miniaturized power processing units will be needed. Miniature ion engine technology may possibly be based on hollow cathode technology, currently used for conventional engine cathodes and neutralizers. Thus, miniature ion engines have to be considered as very advanced micropropulsion concepts that still have to overcome many feasibility concerns before they can be seriously considered for microspacecraft applications.

Hall Thrusters

Hall thrusters[42,43] are electrostatic propulsion devices which use xenon propellant. Plasma generation and ion beam acceleration are different from those found in ion engines and lead to a more compact thruster technology. In a Hall thruster[42,43], electrons emitted from a hollow cathode external to the thruster are accelerated towards a positive anode located upstream and inside an annular discharge chamber. On their way to the anode, the electrons cross a radial magnetic field extending across the annular chamber. Due to Lorentz-force action, the electrons gyrate around the magnetic field lines, and drift azimuthally through the annular channel, colliding with propellant gas atoms (xenon) and ionizing them. The ions are accelerated away from the engine by the same electric field that attracted the electrons. The ion beam is neutralized by additional electrons streaming off the cathode.

Due to the high electron density in the magnetic field region, a dense ion beam can be formed, overcoming space charge limitation effects found in ion engines, Hall thrusters are thus more compact for the same delivered thrust level than ion engines. On the other hand, Hall thruster typically deliver specific impulses around 1500 - 2000 s, making them more suitable for near-earth missions (orbit transfer, repositioning, etc.), rather than interplanetary flights. A high-Isp Hall thruster would be an attractive alternative to ion engines.

Current Hall engine technology is far too heavy and power consuming to be used within the microspacecraft design envelope. However, efforts are underway to miniaturize the technology for small satellite applications[43]. Design goals are a power level of 50 W at a thrust of 5 mN and a specific impulse of 1600s. The device is estimated to be about 4 mm in diameter. Given the smaller channel dimensions, larger magnetic fields have to be provided to achieve smaller electron gyration radii. Required estimated field strengths are 0.5 T. For even smaller devices, magnetic field strengths would have to be increased further, Currently, samarium-cobalt permanent magnets are able to deliver about 1 T at their surface, and thus further miniaturization beyond the engine size outlined above may be difficult to achieve.

Power levels even for the miniaturized Hall thruster version currently being studied at MIT[43] are quite high for the microspacecraft considered in this study, but may be fit within the upper Class I category if power requirements can be relaxed and a dedicated power supply is provided for the thruster. However, as mentioned, a Hall thruster device appears unsuitable for interplanetary applications requiring large delta-v's due to their lower specific impulse capability when compared with ion engines. In addition, as for ion engine applications, substantial reductions in power processing unit weight and size will have to be made.

Field Emission Thrusters

Field emission thrusters[44-46], or Field Emission Electric Propulsion (FEEP) devices as they are commonly being referred to, have traditionally been envisioned for spacecraft attitude control, providing ultra-fine control or drag make-up to establish virtual-drag-free environments on scientific missions. However, FEEP devices, due to their already small size and low available thrust levels, may also be able to provide primary propulsion for microspacecraft.

In a FEEP device, thrust is generated through electrostatic forces as in ion engines or Hall thrusters, but the ionization mechanism is different yet again. In a FEEP thruster, a liquid metal propellant (Cs) is fed by capillary forces through a small channel. The channel ends forming sharp edges that are located opposite a negative electrode, separated by a small gap (about 1 mm) from the channel tip. The channel structure itself carries a positive potential. An electric field develops between the two electrodes and the free surface of the liquid CS metal column near the tip of the channel deforms, forming cusps which protrude from the surface of the liquid. As the liquid forms ever sharper cone structures due to the action of the electric field, the local electric field strength near these cusps increases. Once a local electric field strength of about 10^6 V/cm is reached, electrons are ripped off the Cs metal atoms. The electrons are collected through the liquid metal column and the channel walls, and the positive ions are accelerated away from the liquid columns through a gap in the negative electrode by the same electric field that created them.

Accelerating voltages between the two electrodes typically reach values of around 10,000 V, resulting in specific impulse values up to 8,000 s[48]. Rectangular, slit-shaped channel geometries are most frequently investigated in the laboratory . Depending on slit width and voltages applied, thrust levels between 10^{-6} and 10“3 N have been achieved. Required power levels for the thruster itself as low as 60W/mN have been achieved. In addition to thruster power, power has to be provided to a neutralizer. The neutralizer is required to prevent spacecraft charging while emitting the positive ion beam. Different neutralizer concepts have been tested, among them a hollow cathode type and a tungsten filament configuration[47]. In the hollow cathode type, which requires less power than the tungsten filament type, a Cs compound is heated, creating a Cs vapor that is ionized in a hollow cathode discharge, providing the electrons required for beam neutralization. This neutralizer type delivers 0.1 mA per Watt of electric power, including heater power.

Advantages of FEEP devices for microspacecraft primary propulsion applications are its high-specific impulse and compact size and propellant storage (liquid metal). Disadvantages are potential contamination issues due to the use of Cs propellant, and high voltage and power requirements, requiring dedicated power processing units adding to the propulsion system weight. FEEP thrusters also require a fairly narrow operating temperature range to avoid solidification of the Cs propellant,

Table 11 lists two FEEP thrusters under development at Centrospazio in Italy, currently (he only provider of FEEP thrusters. As can be noted, thrust specific power requirements are about 90W/mN for the smaller model and 75 W/mN for the larger thruster. Given that the neutralizer requires 1 W per 0.1 mA, and , according to Petagna et al.[49] , an emitter current of 7mA/mN is required, the power required per mN of thrust is (90W/mN + 10W/mA*7mA/mN), i.e. 160W/mN or 145 W/mN in case of the larger thruster. These power requirements will limit achievable thrust values with a FEEP thruster on board a 20 W Class I spacecraft to about 0.125 -0.14 mN. At thrust values this low, thrust-to-spacecraft mass ratios will be around 0.006 mN/kg. The Europa orbiter mission currently being studied at JPL uses solar elelctric propulsion (SEP) and has thrust-to-weight ratios of around 0.23 mN/kg at the beginning of the mission, dropping to about 0.03 mN/kg at the end of the mission[50]. These values are significantly

Table 1 I: State-of-the-Art FEEP Thrusters

Manufacturer	Centrospazio	Centrospazio
Propellant	Cs	Cs
Thrust (μN)	100	800
Isp (s)	8000	8000
Power (W)	9	60
Thruster Mass (kg)	0.45	3.5
PPU Mass (kg)	2.9	4.1
Thruster Size (cm)	1.2x1 .2x0.8	2.5x2 .5x1.5
PPU Size (cm)	1.2x I. 0x0.6	2.5x1 .3x1,5
Total Impulse (Ns)	160	60,000

higher than those obtainable with a state-of-the art FEEP system onboard a microspacecraft.

Using FEEP thrusters for microspacecraft primary applications will therefore require higher power levels than those assumed in this study in Section 11. In addition, the potential Cs contamination issues will have to be studied and evaluated for each mission. Total impulse capabilities appear low for the smaller of the two thrusters listed in Table 11 (corresponding to about 450 hrs of run time assuming steady-state nominal thrust values), however, are quite high for the larger thruster (corresponding to over 20,000 hrs run time assuming nominal steady-state thrust conditions). The latter would be sufficient for electric primary propulsion applications. PPU masses appear compatible with the microspacecraft mass allocations.

Future work in field emission thruster technology is focusing on the use of microfabricated emitter arrays[45] consisting of a series of micro-"volcano" structures on a wafer. The significance of these arrays is that ions can be produced at much lower voltages, reducing power requirements. Since extractable currents from each micro-emitter are much lower than those obtained with conventionally machined emitters, arrays of many of these emitters will be required to operate in parallel.

Colloid Thrusters

Colloid thrusters were studied extensively during the late 1960s and early 1970s for spacecraft attitude control and drag-makeup, but due to their failure to produce high enough thrusts at reasonable power levels fell out of favor. With the advent of microspacecraft designs, a potential application for microspacecraft primary propulsion may have arrived.

In a colloid thruster, thrust is produced by electrostatically accelerating fine liquid droplets ejected from a capillary[51,52]. A strong electric field applied between the sharp-edged exit of the capillary and an external electrode causes charge separation inside the liquid propellant, which in most cases is doped with an additive to increase its electric conductivity. Through a combination of hydrodynamic instabilities, causing jet break-up into small liquid droplets, and the action of the applied field acting on the conductive liquid, charged droplets are extracted from the capillary at high velocities, producing thrust.

Depending on the propellant used, either positive or negative liquid droplets can be produced, Most applications studied in the past used glycerol doped with sodium iodine to produce positive droplets and glycerol doped with 2 - 10 % sulfuric acid to produce negative droplets. The ability to produce both positive and negative droplets was termed a "hi-polar" thruster by Perel et al.[51]. Its significance is that it can potentially be self-neutralizing, provided the same amount of current can be drawn from each set of capillaries, eliminating the need for a separate neutralizer.

Several trade-offs have to be made in the design of a successful colloid thruster to optimize performance. In order to obtain good performance, the specific charge, measured in Coulomb per droplet mass, has to be high in order to obtain high specific impulses at reasonable applied voltages. The colloid thruster is an electric thruster, and as such additional propulsion system masses associated with the power supply or conditioning will have to be offset by sufficient propellant mass savings that can only be obtained through a high enough specific impulse. Specific impulse in a colloid thruster is also determined by the so called specific charge efficiency, which measures the distribution of specific charge in a droplet stream. A more "peaked" specific charge distribution will lead to higher specific impulses and higher propulsion system efficiencies.

Specific charge efficiency in turn depends on several parameters, such as electric field strengths (higher electric field strengths reduce the efficiency since they create higher charged particles in addition to lower charged ones), flow rate (lower mass flow rates result in higher specific charge efficiencies), fluid conductivity (a higher conductivity leads to lower specific charge efficiencies since it becomes easier to produce more charges in a more conductive stream) and capillary tip design (which affects the specific charge efficiency mostly through its effect on the local electric field strength near the tip)[51]. Some of these design considerations work against each other. A large potential drop caused by a strong electric field, for example, will accelerate the droplets to a higher exhaust velocity (which would help to raise the specific impulse), however, it will also decrease the specific charge efficiency (which will lower the specific impulse again). Low flow rates, aiding in obtaining higher specific charge efficiencies and, thus, specific impulses, will also reduce thrust.

In addition to these performance considerations, careful propellant selection will need to be made to ensure proper thruster function and long lifetime. High solvation capability (to take up dopants), low vapor pressure (to avoid crystallization of dopants on capillary walls near tip, potentially clogging the system), low freezing point (to avoid clogging) and low corrosivity (to ensure long thruster lifetime) are key parameters in the selection of the propellant[51]. The glycerol propellants discussed above were found to have a somehwat high vaporization pressure, but were chosen for its superior ability to dissolve dopants[51]. Platinum capillaries have been used because of their resistance to corrosion[51].

Based on the design considerations above, colloid thrusters have been designed and operated with platinum capillaries having ID's of about 200 microns, using sodium iodine and sulfuric acid doped glycerol propellants, producing thrusts between 0.2 -0.5 mN at power levels of about 4.4 W/mN, requiring voltages of +4.4 kV and -5.8 kV, depending on droplet polarity[51]. Specific impulses were estimated between 450 - 700 sec[51]. In other cases, specific impulses of up to 1350 sec at thrust levels of 0.55 mN were obtained (power levels were not reported in that case)[52].

Of all micro-electric primary propulsion options reviewed so far, colloid thrusters are quite possibly the most suited for microspacecraft primary propulsion applications at this stage of micro-electric propulsion development. Power requirements are much lower than those of FEEP devices and fit well within Class I constraints. Thrust levels of 0.5 mN can easily be achieved using only about 2 W of power in a self-neutralizing, hi-polar array[51]. Thruster specific masses (excluding power supply and conditioning) have been estimated at 0.2 - 0.5 kg/W[50], and would also fit within Class I microspacecraft envelopes. However, specific impulse values are somehwat low for interplanetary applications. Thus, future work should focus on obtaining higher specific impulse values.

Attitude Control - Chemical

Cold Gas Thrusters

Cold gas thrusters represent the smallest rocket engine technology available today. Cold gas systems are valued for their low system complexity, small I-bit and thrust capability and the fact that, when using benign propellants (e.g. N,), they present no spacecraft contamination problems. Sometimes high reliability is also referred to as one of the advantages. However, valve leakage problems have resulted in repeated losses of spacecraft due to premature depletion of propellant through valve internal leaks.

This leakage problem is a result of a combination of small amount of microscopic contaminants (to be found in even the cleanest propulsion system) and high pressure propellant storage. On the contamination side, the propellant tank is one of the major contaminant sources (microscopic metal flakes left over from fabrication, etc.). These contaminants may locate themselves on valve seats, carried along with the propellant flow, and subsequently prevent the valve from sealing completely. Even though the remaining opening across the valve seat may be so small that for a liquid propellant application it would pose no problem due to the higher liquid viscosity, in cold gas systems, where the propellant is stored at very high pressures, propellant may escape even through these microscopic openings.

Another disadvantage of cold gas systems are their low specific impulse performances, unless very light gases (H_2, He) are used. Neither hydrogen or helium is commonly used, however, since storage problems and large and heavy tankage would result due to low gas densities, and additional leakage concerns would have to be considered with these light gases.

Table 12 gives a list of typical cold gas performance values, based on data found in Refs. 8 and 26. Of the gases listed, nitrogen is by far the gas most frequently used as a cold gas, due to a combination of reasonable propellant storage density, performance and lack of contamination concerns.

Table 13 lists some of the smallest cold gas thrusters available today. The Moog 58x125 thruster is merely 4.3 cm in length and 1.4 cm max. dia., including fitting. The fitting accounts for roughly half of the size of the total envelope. Size, mass and power requirements fit well within the Class I microspacecraft envelope. However, even a cold gas thruster this size may perform only marginally with respect to the impulse bit requirements (compare data in Table 13 with comments in Section II).

Using the data found in Tables 12 and 13, required leak rates for microspacecraft can be estimated and current cold gas thruster performances be evaluated in this regard. Using the information provided in Section II, a total "delta-v" requirement of 50 m/s is assumed. For a specific impulse of 70 sec (N_2) and an assumed microspacecraft mass of 10 kg (Class I), the required attitude control propellant mass is 0.7 kg of nitrogen. At a storage density of 0.28 g/cm^3 for nitrogen at 3500 psia and O C, a tank volume of about 2500 cm^3 is required. Taking into account the possibility of propellant leakage, assume that 10% more propellant is loaded onto the spacecraft, now requiring a tank volume of 2750 cm^3 at the same storage pressure. Assuming a spherical tank, this translates into an inner tank diameter of roughly 37 cm. This tank size is slightly larger than the

Table 12: Cold Gas Propellant Performances

Propellant	Molecular Weight (Kg/Kmol)	Density (3500 psia O C) (g/cm')	Isp* (Theoretical) (s)	Isp* (Measured) (s)
Hydrogen	2.0	0.02	296	272
Helium	4.0	0.04	179	165
Neon	20.4	0.19	82	75
Nitrogen	28.0	0.28	80	73
Argon	39.9	0.44	57	52
Krypton	83.8	1.08	39	37
Xenon	131.3	2.74***	31	28
Freon 12	121		46**	37
Freon 14	88	0.96	55	45
Methane	16	0.19	114	105
Ammonia	17	liquid	105	96
Nitrous Oxide	44		67"	61
Carbon Dioxide	44	liquid	67	61

* at 25 C. Assume exoansion to zero pressure in case of theoretical value.

** at 38 C (560 R) and area ratio of 100.

*** Likely stored at lower pressure values (2000 psia) to maximize propellant to tank weight ratio.

Table 13: Small Cold Gas Thrusters

Manufacturer	Moog[55]	Moog[56]	Marquardt[7]
Type	**58x125**	**58x115**	**-**
Thrust (N)	0.0045	2.89	4.5
Ibit (Ns)	10^{-4}	-	
Isp (s)	65 (N,)	-	-
Pressure (kPa)	34.5	1460	8840
Open Response (ins)	0.94	3.5 (spec)	<1.1
Power (pull-in) (W)	**2.4**	**30**	**-**
Weight (g)	7.34	13	5.4

envelope assumed for a 10-kg spacecraft, but is within the right range. The tank, however, will dominate the spacecraft design layout completely. Assuming further that all of the additional 10% of the propellant may be lost over the course of the mission (corresponding to 250 cm^3 at 3500 psia storage pressure or almost 59,000 see, assuming zero compressibility of nitrogen in this rough estimate), maximum allowable leak rates for a 2 year mission would be 9 x 10”^{4} see/s and for a 3 year mission 6 x 10”^{4} see/s. These small leak rate requirements are a consequence of the small spacecraft size. Since smaller spacecraft carry smaller onboard propellant supplies for the same attitude control requirements, less propellant can be afforded to be lost due to leakage and valve leak rates for microspacecraft consequently have to be even lower than for bigger spacecraft.

Recent leak tests with cold gas thrusters have shown leak rates lower than the ones calculated above[53], however, at thruster operating pressures which where a mere 34.5 kPa, or about one third of an atmosphere. The low leak rates calculated above will have to be maintained throughout the entire feed system, however, in particular across components that are exposed to the full tank pressure. A cold gas thruster valve recently developed by Marquardt[7] is subjected to pressures of about 8900 kPa (1300 psig) and maintains leak rates of <2.8 x 102 see/s GHe[7], corresponding to roughly 1 x 10^{2} see/s GN_2, assuming a 1/ $\sqrt{M}$ (M being the molecular weight) dependence for the leak rate. These values are far higher than the estimated leak rate requirements given above.

The leak rate requirements could be relaxed if more propellant reserves were carried. Performing the calculations above for the same case of a 10 kg microspacecraft requiring a 50 m/s attitude control budget over the course of the mission, but raising the propellant margin to 50% would yield maximum allowable leak rates of 4 x 10”^{3} see/s in the case of a 2 year mission and 2.5 x 10^{-3} see/s for a 3 year mission. While these leak rates approach obtainable values, it has to be noted that these rates will have to be maintained over the course of the entire mission, even after the valve has been subjected to many cycles and substantial propellant flow, carrying contaminants through the valve. In addition, raising the propellant reserves from 10% to 50% leads to a bigger required tank diameter of 41 cm ID compared to 37 cm before. These tank designs will completely dominate the spacecraft design layout and consume a substantial portion of the overall spacecraft mass and volume.

An interesting alternative to conventional cold gas propulsion using high pressure gas tanks is the use of ammonia as a propellant. As pointed out by Nakazono[54], ammonia has a vapor pressure of 33 psia (224 kPa) at -18 C. Thus, even without tank heaters, sufficient pressure could be provided to an ammonia cold gas thruster merely using the boil-off of the propellant. As can be seen from Table 12, specific impulses obtainable with ammonia are higher than those achievable with nitrogen. The ammonia system would allow for liquid storage, reducing tank size and mass and propellant leakage concerns. Depending on available vaporization rates (dependent on tank temperature), propellant flow rates may be limited[39].

It may therefore be concluded that cold gas propulsion systems using high-pressure gas supplies do not appear to be a viable option for microspacecraft unless attitude control requirements can be substantially reduced, valve leak rates can be lowered by at least one order of magnitude, and severe mass and volume constraints on the remainder of the microspacecraft, caused by large and heavy propellant tanks, can be tolerated. Cold gas systems based on liquid storage of ammonia, on the other hand, appear as a very attractive option for microspacecraft attitude control. In either case, obtainable input bits have to be reduced further, even **for Class 1 microspacecraft applications. This requires the development of either faster valves or smaller nozzle throat areas. Fabricating nozzle throat diameters smaller than the ones obtainable today may require the exploration of new** technologies, such as MEMS (see Section V).

Warm Gas Thrusters

Cold gas technology discussed above could be adapted to warm gas thruster concepts. In this ease hydrazine propellant is decomposed in a separated gas generator, consisting in essence of a Shell 405 catalyst bed, and gaseous hydrazine decomposition products are then fed to a plenum and finally through a cold gas thruster[8]. Such a system would not require separate propellant tanks if a conventional hydrazine thruster was used as the microspacecraft primary propulsion device. In addition, several cold gas thrusters in existence today already claim compatibility with hydrazine decomposition products. A separate heater would be required to heat the catalyst bed in order to allow for a sufficient number of starts, just as in conventional hydrazine thruster technology. By feeding the decomposition products into a plenum, from which they can be drawn to the various attitude control thruster clusters upon demand would lower the required number of catalyst starts[54].

A hydrazine warm gas system is a very attractive option for microspacecraft, in particular when a hydrazine propellant supply is already onboard for primary propulsion purposes. Relatively high performance, comparable to the ammonia cold gas system described above, can be combined with compact propellant storage and the relatively near term availability of the required propulsion components, drawing upon cold gas heritage. As in the case of cold gas systems, either faster valves or smaller nozzle orifices are required to lower impulse bits to the requirements for both Class I and Class II microspacecraft applications.

Tri-propellant Thrusters

In a tri-propellant thruster, a propellant mixture of hydrogen, oxygen and an inert gas, such as helium or, more commonly for propellant storage reasons, nitrogen is used[7,57]. The propellants are stored fully mixed, no separate tanks are required. The addition of the inert gas to the mixture renders the mixture non-combustible, until exposed to a suitable catalyst. Different catalysts are being studied, typically based on nobel metal compounds[27,58], although details on catalyst compositions are mostly treated as proprietary information. Thruster performances range between 70 s and 140 s of specific impulse[57]. A recent design by French[7] is aimed at 125 sec Isp and a thrust of about 2N. However, this design was not specifically intended for microspacecraft use.

Tri-propellant systems, even though able to deliver higher performance than cold gas systems, suffer the same disadvantage of high-pressure propellant storage and the associated leakage problems as cold gas systems. In addition, even though required propellant masses can be reduced over conventional cold gas systems due to the higher specific impulse performance, required high-pressure propellant tanks will likely continue to dominate the spacecraft design. Thus, the advantages gained with the use of tri-propellant systems over cold gas systems onboard microspacecraft may be limited.

Hydrazine Mono-Propellant Thrusters

Hydrazinc mono-propellant thrusters readily available today (see Table 8) are far too large and heavy to be used in attitude control clusters around a microspacecraft. The development of new, miniature hydrazine thrusters is required. A research initiative to study the feasibility of miniature hydrazine thrusters was recently begun at JPL, but it is too early to present results from this acitivity.

Bi-propellant Thrusters

Bi-propellant thrusters were not considered for microspacecraft attitude control. Currently available engine technology is far too big and heavy for use in microspacecraft attitude control clusters. Bi-propellant engines have been considered for attitude control purposes on larger spacecraft to allow for easier integration into primary propulsion systems, eliminating separate attitude propellant supplies. It was already concluded, however, that bi-propellant systems are probably not suitable for onboard microspacecraft primary propulsion. In addition, even though hi-propellant thrusters do offer higher performances, reducing propellant requirements, required propellant masses for attitude control are usually small, not providing an opportunity for large spacecraft mass reductions. Even if a substantial reduction of the typically small attitude control propellant mass could be achieved, it would likely be offset by the higher dry mass of a hi-propellant system due to increased component parts count when compared with much less complex cold or warm gas, or even miniature hydrazine systems. Finally, in view of the survey of state-of-the-art miniature hi-propellant technology given above, there exists considerable doubt whether further substantial reductions in thruster size can be made while still being able to provide reliable, space-qualifiable engine technology.

Attitude Control - Electric

Pulsed Plasma Thrusters (PPTs)

In a pulsed plasma thruster, electrical power is used to ablate, ionize and electromagnetically accelerate atoms and molecules from a bar of solid Teflon[50,59]. The Teflon bar is pushed against a retaining lid between two electrodes by means of a negator spring. The electrodes are connected to a capacitor, which is unable to discharge because the vacuum and solid Teflon bar between the electrodes do not provide a conductive path. A spark plug located near the solid Teflon surface is fired, removing a portion of the Teflon and ionizing it. In the process, the Teflon bar is pushed forward toward the lid and brought in position for the next pulse. The capacitor now discharges through the ionized Teflon gas. The particles inside the Teflon discharge consist of a variety of molecular fluorocarbons, which are ionized in the process and give rise to a current flow between the electrodes. This discharge current generates a strong magnetic field surrounding it. Lorentz forces acting on the discharge current as a result of this magnetic field push the Teflon plasma out of the thruster at high exhaust velocities of about 10- 20 km/s. Non-ionized particles are expanded from the thruster due to Joule-heating.

Thrusts generated per pulse are on the order of 10s to 100s of micro-newtons. However, because the pulse length is on the order of milliseconds, the capacitor can be charged and discharged several times per second, thus creating an accumulative thrust in the micronewton to millinewton range. Pulsing frequencies between 1 Hz and 6 Hz are common. The ability of a PPT to produce very small thrust levels per pulse allows for the possibility to deliver very small impulse bits on the order of less than 1 mNs. Future designs are aimed at providing merely 10s of μNs. PPT thrusters have been developed since the 1950s and have flown on several US satellites. State-of-the art thruster are able to provide specific impulses in the range of 800- 1500 s, thrusts of 220- 1100 μN, and efficiencies between 5 - 15 % at a wet mass of 5 kg.

Current PPT designs are therefore too large for microspacecraft attitude control. Future designs are being contemplated and are predicted to have wet masses as low as 0.5 kg. Providing 12 of these thrusters for attitude control on a microspacecraft will lead to a total (wet) system mass of 6 kg. These mass values are compatible with Class I mass guidelines. With minimum impulse bits for the future , 0.5 kg thrusters predicted at 10- 100 μNs, PPT are also able to provide the minimum impulse bit requirements in the upper Class I microspacecraft category. However, minimum thrust requirements for attitude control of Class I spacecraft, ranging between 1.75 - 4.5 mN for the larger spacecraft masses, can only be met at high power levels. According to Ref. 59, a 1.75 mN thrust requires a power level of 120 W and pulsing frequencies between 3 - 6 Hz, depending on capacitor size. These values far exceed the power levels that will be available for attitude control on any of the microspacecraft considered here. PPTs can thus only be used on Class I craft if slew rate requirements can be reduced substantially. Current and predicted PPT technologies are too large and heavy for Class II spacecraft,

Field Emission Thrusters

Field emission, or FEEP, thrusters, surveyed above, are reviewed here in terms of their applicability as attitude control thrusters on microspacecraft. Large PPU masses and high power requirements may prohibit their use in such a function. Following the example given in the cold gas thruster section above, for a typical Class I microspacecraft mission, 0.7 kg of nitrogen gas would be required to meet a 50 m/s delta-v budget. If a FEEP system with a specific impulse of 8,000 s was used, the required propellant mass would be r-educed to 12 g. While this is a substantial reduction, the required PPU mass is, according to Table 11, is 2.9 kg for the smaller of the two FEEP thrusters considered. Since Cs propellant can be stored in its liquid state, whereas nitrogen gas would have to be stored in high-pressure tanks, tank weight reduction will benefit the FEEP system. According to recent tank data[60], a 3,244 cm^3 tank (roughly the size required for nitrogen storage in the example above), capable of maintaining a maximum expected operating pressure (MEOP) of 10,000 psia (far more than required for a storage pressure of 3,500 psia as assumed), weighs around 1.8 kg (these data are based on a cylindrical tank). Even neglecting tank masses for the FEEP system, the cold gas system will still be lighter than the FEEP system by about 0.4 kg (although volume requirements will be higher due to the large tank volume). In addition, the FEEP system will not be able to provide the minimum thrust for the assumed slew rate requirements within the power constraints expected to be found on a microspacecraft. FEEP systems may find applications as attitude control devices for microspacecraft if slew rate requirements can be relaxed significantly and very small impulse bits are required,

Colloid Thrusters

Much the same comments as made with respect to the applicability of a field emission systems to microspacecraft attitude control can be made for colloid systems. Minimum thrust requirements cannot be met , and additional power processing unit masses may offset propellant reductions gained over cold gas systems through the use of the higher specific impulse colloid thruster. Since current colloid thrusters provide lower specific impulses and require less power, the comparison between colloid and cold gas systems may be shifted somewhat more in favor of the colloid system than was the case for the FEEP system.

Resistojets

State-of-the art resistojet technology[22] is far too heavy, and requires far too much power (in excess of **350 W) to be useful** for microspacecraft attitude control. However, some work on small water resistojets was performed[61-63]. Work focused on measuring small nozzle performances using water vapor. No heater power requirements were reported.

Water is not the most suitable propellant for resistojet use to its high heat of vaporization, even though it does simplify laboratory testing due to its lack of safety and toxicity concerns. Table 14, based on data from Ref. 26, list several relevant properties of candidate propellants for a resistojet system. Of the propellants listed, ammonia and water immediately stand out due to their low molecular weight, resulting in high specific impulse performance. Of these two propellants, ammonia requires less heat to vaporize and is thus the propellant of choice.

Since no small resistojet technology exists, it is not clear how well this technology is suited for microspacecraft use. Liquid storage of propellants will reduce system weights over high-pressure cold gas storage systems . Comparing ammonia resistojets with the ammonia cold gas thruster discussed above, the resistojet should enable higher duty cycles and longer burns, making resistojet technology more versatile. The disadvantage is their need for a separate power supply, adding to the system weight. Work is currently underway at JPL and the Aerospace Corporation to develop micro-resistojet technology using MEMS technologies (see Section IV).

IV. FUTURE DEVELOPMENT NEEDS AND EMERGING TECHNOLOGIES

Based on the thruster survey performed in Section III, technology needs for microspacecraft propulsion can be identified. Several emerging micropropulsion technologies are being introduced. These emerging technologies are the first propulsion components designed specifically with microspacecraft applications in mind.

Identification of Technology Needs

The results of the survey conducted in Section III are summarized in a matrix, shown in Table 15. Only state-of-the-art technologies, or those under significant development, are listed in this matrix. The technologies listed were evaluated in view of their application to Class I and II microspacecraft attitude control and primary propulsion applications by grouping them into three categories: those technologies that appear applicable to the task ("yes" category), those that do not ("no" category), and those that fall somewhat in between ("maybe"). The latter category serves to classify technologies that may fulfill some mission requirements, but not others, or those that could be made to fulfill all requirements for the specific

Table 14: Properties of Candidate Resistojets propellants

Propellant	Formula	Molecular Weight (kg/Kmol)	Liquid Density (g/cm’)	Heat of Vaporization (kJ/kg)
Ammonia	NH,	17.0	0.6	1159.7
Propane	C_3H_8	44.1	0.49	339.3
Ethyl Chloride	C_2H_5Cl	64.5	0.92	388.1
Butane	C_4H_{10}	58.1	0.57	360.2
Freon 12	CCl_2F_2	120.9	0.98	141.8
Water	H_20	18.0	1.0	**2442.5**
Hydrogen Fluoride	HF	20.1	0.99	1505.9
Methanol	C_2H_3OH	44.0	0.79	1099.3
Methyl Chloride	CH_30H	51.0	0.91	376.5
Ethane	C_2H_8	30.0	0.56	313.7
Ethyl Methyl Ether	$C2H_5OCH_3$	60.0	0.8	350.9
Mono Methyl Amine	CH_3NH_2	31.0	0.77	873.8

Table 15: Matrix of Applicability Status of State-of-the-Art Technologies, or Technologies under Significant Development, to Microspacecraft

Technologies (State-of-the-Art Only)	Class I Primary	Class I ACS	Class 11 Primary	Class 11 ACS
Bi-Propellant	No	No	No	No
Hydrazine	Yes	No	No	No
HAN/TEAN	Maybe	No	No	No
Peroxide	No	No	No	No
Solid	Yes	No	Maybe	No
Hybrid	Maybe	No	No	No
Cold Gas	No	Maybe	No	No
Ammonia Cold Gas	No	Maybe	No	No
Warm Gas	No	Maybe	No	No
Tri-Propellant	No	Maybe	No	No
Ion	No	No	No	No
Hall	No	No	No	No
FEEP	Maybe	Maybe	No	No
Colloid	Maybe	Maybe	No	No
PPT	No	Maybe	No	No
Resistojet	No	No	No	No

microspacecraft category and task with only minor technology advances required.

Inspecting Table 15, the lack of suitable thruster technology for microspacecraft applications becomes obvious. This is not too surprising a result, given that most of the technologies reviewed here were developed for spacecraft much larger than the spacecraft considered here. Some technologies stand out, however, and may appear applicable to microspacecraft even in, or close to, their current design stage. For primary Class I applications, the smallest available hydrazine attitude control thrusters appear as a suitable and reliable thruster option if only small to intermediate delta-v's are required. Solid motors also may provide low to intermediate delta-v capability within their current design limits, although thrust values may need to be reduced further. However, no thruster options providing large delta-v and compatible with the microspacecraft assumptions are currently available. Both hi-propellant chemical as well as ion in their current manifestations would require separate propulsion stages due to high system masses and volumes resulting in larger launch masses. Possible exceptions are FEEP and colloid thrusters. While FEEP thrusters still require too much power, and colloid thrusters deliver fairly low specific impulses, they may be marginally suitable at this point of their development, but will likley require further development. PPT thrusters do not have the total impulse capability and also require too high power levels for the minimum thrust levels needed.

For Class I attitude control applications, cold gas and warm gas systems currently offer the greatest near-term potential. In both cases, however, delivered impulse bits are too high. Impulse bits maybe reduced through faster valve technologies or increased flow restriction through the use of smaller nozzle throats or separate flow restrictors. The last option may be the easiest to achieve, but fabricating small orifices with low tolerances and protecting them from clogging through contaminants may require the use of new technologies (MEMS).

PPTs, FEEPs and colloid thrusters may be applicable for Class I spacecraft attitude control only when slew rate requirements can be lowered dramatically. Given that current microspacecraft mission requirements may still undergo considerable evaluation and may vary from mission to mission, these three technologies have been listed as possible candidate technologies. If slew rate requirements will not change significantly, large reduction in power requirements and PPU specific masses will be needed.

For the Class II category, virtually no state--of-the-arr propulsion technologies appear suitable. Once again, the reason for this is that current hardware was designed for larger spacecraft and currently available components may even exceed the size of a typical Class 11 spacecraft. The only primary propulsion device that may be applicable to Class II spacecraft using state-of-the-art technology (yet still requiring a full development program) may be solid motors. For attitude control, no current propulsion hardware appears suitable, either because the size of available components do not allow for the distributed mounting of a dozen or so thruster around a Class II bus, or because delivered minimum impulse bits are too high.

Therefore, high-specific impulse primary propulsion for both spacecraft categories, as well as low I-bit attitude control thruster technology for Class I, but in particular Class II, are identified as major propulsion technology needs for the microspacecraft considered here. Class H attitude control applications will also require extreme miniaturization.

Emerging Technologies

The increasing interest in microspacecraft for deep-space missions has led to the exploration of technologies suitable for these spacecraft at JPL, NASA's lead center for robotic interplanetary space exploration and one of the first potential users of these technologies. Several new micropropulsion technologies were proposed in the course of these activities. Three of these emerging micropropulsion technologies are discussed here. Among them are two micro-attitude control phase-change thrusters (vaporizing liquid or resistojet and subliming solid) and micro-ion engines for high delta-v primary applications. All three technologies rely heavily on MEMS-fabrication techniques, allowing for the potential of order-of-magnitude reductions in component mass and size, possibly suitable for both Class I as well as Class 11 applications. Associated with each technology considered here, however, are significant feasibility issues that will have to be thoroughly investigated in coming years. Thus, the thruster technologies introduced here have to be viewed as very advanced micropropulsion options.

Vaporizing Liquid Micro-Thruster

The vaporizing liquid micro-thruster, or micro-resistojet, is a concept that was proposed by Leifer and Mueller , as well as Janson at the Aerospace Corportaion, a couple of years ago[7,64]. In this thruster concept, described in detail in a companion paper[65], a suitable liquid (ammonia or hydrazine) is heated via a thin-film heat exchanger, micro-fabricated onto a silicon substrate. A conceptual sketch and picture of the assembled JPL-device are shown in Figs. 2 and 3. Two identical silicon wafers, featuring thin-film heaters and micro-nozzles, are bonded to a Pyrex spacer that, in its final assembly, is sandwiched between the two silicon wafers. The liquid propellant, pressure-fed through one of the openings machined into one of the silicon wafers, enters the thruster and is vaporized as it flows between the heater elements. Propellant vapor is then exhausted through the second nozzle. A recess machined into the silicon underneath the heater creates thermal chokes near the heater edges, reducing conductive losses to the. structure. Silicon has a very high thermal conductivity, providing a bigger challenge for the thermal design of this device. However, no other material can currently be micromachined to the degree and flexibility that silicon can, and silicon was thus chosen as the substrate material.

A key aspect of this design is its simplicity. It does not contain any complex moving parts (such as MEMS-pumps, turbines, etc.) which could decrease the

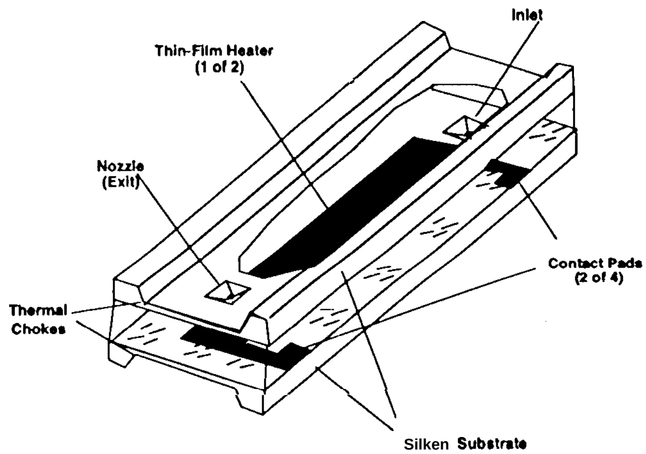


Fig. 2: Concept of the Vaporizing Liquid Thruster

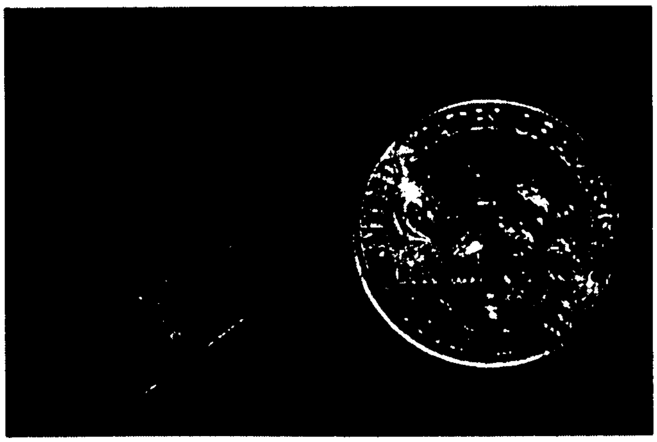


Fig. 3: Vaporizing Liquid Micro-Thruster Chip

reliability of the device. Performance targets are 0.5 to several mN thrust (to meet minimum thrust requirements for microspacecraft attitude control), 50% efficiency and power requirements of a few W or less, The latter condition, dictated by available assumed power levels onboard a microspacecraft, will limit performance. Specific impulse values around 100 to 150 sec have been estimated depending on propellant and heater temperature obtainable. However, given that delta-v budgets for attitude control are fairly low (see Section H), high performance is not necessarily required.

Impulse. bits will largely be determined by the valve technology to be used. Presently, no suitable MEMS valve technology exists to be interfaced with this thruster, but miniature conventional valves, approaching in both size and mass of fully packaged MEMS valve technology, may be used initially. With open-to-close times on the order of a few ms, impulse bits may range as low as 10^{-5} Ns or less. Work on integrating MEMS technology with conventional valve technology is currently being explored at Marotta Scientific Controls, Inc under an SBIR program.

Several prototype devices (see Fig. 3) have been assembled and are awaiting testing this summer. Focus of the initial work is to optimize heater designs, search for design options to reduce power losses to the structure, determine minimum heater lengths, test the reliability of the electric contacting through thruster cycling and gain initial thrust performance data. Work on micro-resistojets is also underway at the Aerospace Corporation'. These devices are similar in function, but feature slightly different heater designs'.

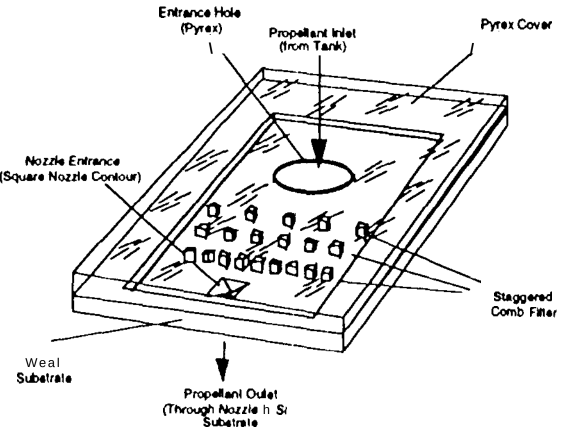
Subliming Solid Micro-Thruster

Subliming solid thruster concepts are not new and substantial development work was performed with these thrusters in the 1960's. Main contributors to this field were Rocket Research[wa] (now Primex), Lockheed[w-»] (now Lockheed-Martin), and NASA Goddard[75,76]. Some work was also performed at Aerospace Industries", the Lewis Research Center'* and the Martin-Marietta company[79] (now Lockheed-Martin also). In the subliming solid thruster concept, a solid propellant is chosen with a high sublimation pressure, such as ammonium hydrosulfide (NH_4HS) or ammonium carbamate ($NH_4CO_2NH_2$). Upon heating, gas pressure builds up inside the propellant tank and the vapor is vented through a valve and nozzle to produce thrust.

The simplicity of this design, and the solid storability of the propellant appear to easily lend themselves to miniaturization. Based on the 1960's work in this area, a subliming solid micro-thruster concept was proposed by the author using MEMS technology [no.] A conceptual drawing of the device is shown in Fig. 4. The chip consists of two layers, one made from silicon the other from Pyrex. The silicon portion of the chip was machined at Sandia Nat'] Labs. Propellant vapor enters the chip from the tank through the circular Pyrex hole and flows along a recess machined into the silicon towards the nozzle orifice. On its way, the propellant passes through a micromachined comb filter. This filter will prevent solid particles from drifting out of the tank and towards the nozzle in the zero-g environment of space, preventing nozzle blockage.

The micro-nozzle has a throat area of about 50x 50 µm and is shown in Fig. 5. The square nozzle shape is a result of the anisotropic etch used in the fabrication of this device, resulting in preferential etching of some crystal planes over others. The wall surfaces of the nozzle are composed of { 111} planes, etching slowest in the fabrication process. Using this technique, simple converging-diverging nozzle shapes can easily be fabricated. Nozzle shapes are not optimized in this design, since performance optimization is not a goal with the current, first generation of chips, and future design iterations may explore different contours.

A mock-up of the thruster concept is shown in Fig. 6, consisting of a tank, a valve and the thruster chip assembly. The valve shown in Fig. 6 is a MEMS-type valve based on thermopneumatic action, designed by Redwood Microsystems, Inc. This valve has leakage issues and currently only serves as a place holder. It should be pointed out, however, that due to the solid propellant storage and relatively low vapor pressures inside the tank, leak rate requirements for valves may be relaxed significantly over those required for cold gas systems.



Fig, 4: Subliming Solid Thruster Concept

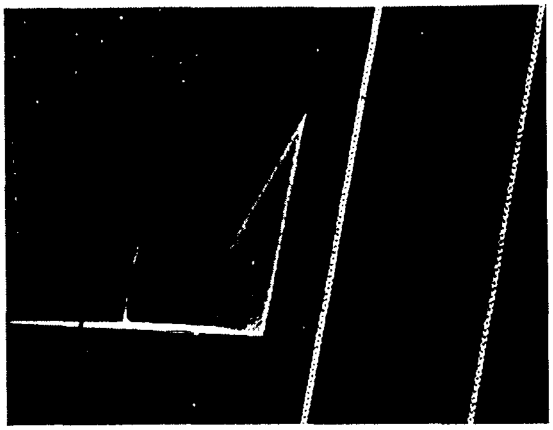


Fig. 5: Subliming Solid Micro-Thruster Nozzle and Filter

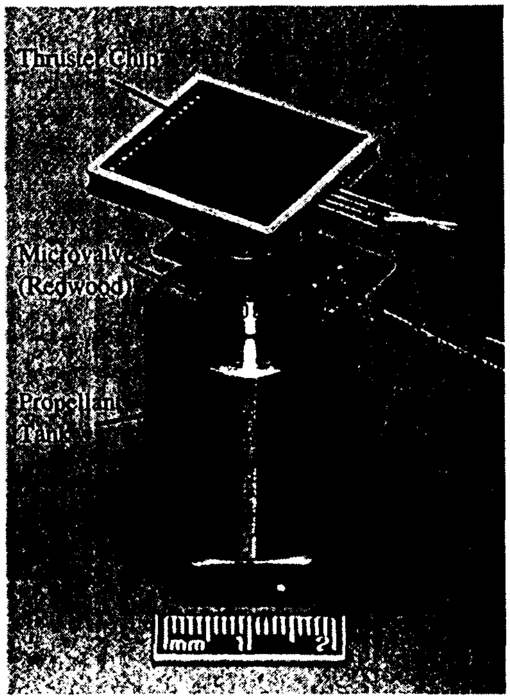


Fig. 6: Mock-up 01' a Subliming Solid Micro-thruster System

Propellant condensation may occur as propellant vapors are exposed to the cold surfaces. In the event that propellant condensation leads to clogging of the filter, a heater was thin- film deposited just underneath the filter on the opposite side of the silicon substrate (facing up in Fig. 6). Similar heaters may be required around the valve to prevent recondensation cm the valve scat, preventing sealing. Valve heating will necessitate the selection of a different valve design than the one shown in Fig. 6 (with no thermal activation as in the case of the Redwood valve shown in Fig. 6 allowed).

Advantages of the subliming solid micro-thruster concepts are its compact size and propellant storage, light weight tankage due to low pressure requirements, relative immunity to leak rate concerns and suitable projected performances. Thrust values of about 0.5 mN up to several mN are targeted with this design. If the chip can be interfaced with fast acting valves very low impulse bits would result. Disadvantages of the system are its low performances (50- 75 s Isp estimated) and toxicity issues associated with the propellants, requiring special handling during testing and propellant loading.

Micro-Ion Engines

High specific impulse micropropulsion devices will be required in order to achieve high-delta-v capability for microspacecraft. MEMS-based devices have been studied[81,82], however, were found to have problems associated with electron wall losses. In addition, for the thrust and power levels required, conventionally machined thrusters may be used'. MEMS technology, however, does hold promise for use in various ion engine components. MEMS accelerator grid system technology is currently being studied at JPL. Using this technology, ion beams may be steered electrostatically by designing a special third, or decelerator grid, of a three-grid system, allowing for the application different electric potentials to different grid section (see Fig. 7).

In order to fabricate MEMS-scale grids, electric breakdown voltages of MEMS materials, such as silicon oxide, will have to be tested. Insulator materials, such as

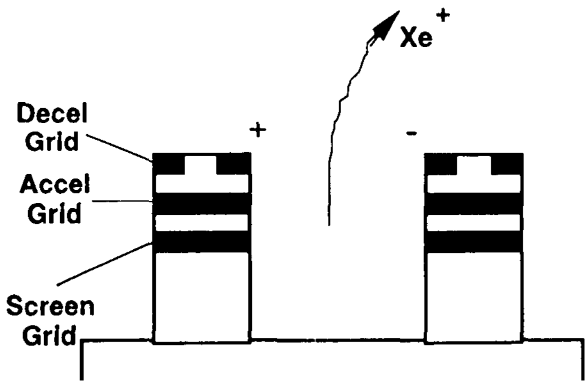


Fig. 7: Concept of Electrostatic Gimabling

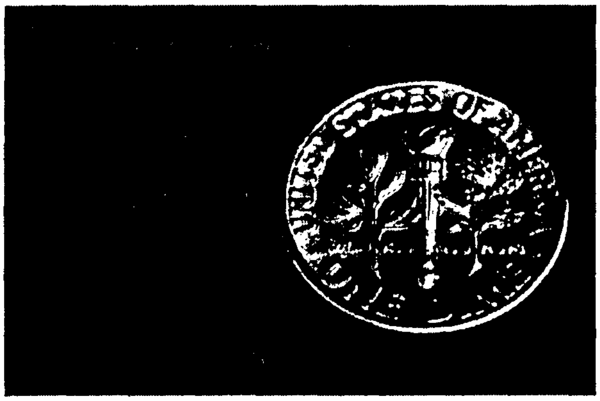
silicon oxide, will be needed as grid insulator materials since free standing grids may not be practical at these dimensions. Figure 8 shows a test chip recently completed at JPL to test silicon oxide breakdown strengths. The oxide layer is deposited between two electrodes (a polysilicon layer underneath the oxide layer, accessible through a via etched through the oxide, and an aluminum layer on top of the oxide). Applying voltage to the two contact pads shown in Fig. 8 will allow silicon oxide breakdown voltages to be tested. An integral heater allows for breakdown tests at various temperatures.

Other areas in the development of miniature ion engine technology that require attention in the future arc micro power processing units and micro flow control and feed systems. In the latter area, micro-ion engine systems (and conventional systems due to their low flow rate requirements) are likely to benefit from the integration of MEMS technologies. Another SBIR contract performed by Marotta Scientific Controls, Inc addresses the issues of microflow control on a chip in a device termed a "micro gas rheostat. "

V. CONCLUSIONS

Existing thruster technologies were reviewed in the view of potential applications for microspacecraft, defines as spacecraft with masses of 1 -20 kg. Based on this review, technology needs were defined and several emerging, advanced micropropulsion concepts specifically designed for microspacecraft applications were introduced.

Only a few of the currently existing thruster technologies appear applicable for spacecraft of the size considered here. For primary propulsion applications, small hydrazine thrusters and solid motors may provide intermediate to low delta-v capability. Thrust values of solid motors may have to be reduced further in order to avoid excessive spacecraft accelerations. FEEP and colloid thruster may possibly be used as primary propulsion devices if power requirements can be lowered in the case of FEEP devices, and specific impulse can be raised for colloid thrusters.



Fig, 8: Grid Breakdown Test Chip

For attitude control functions, currently available cold gas systems approach in performance the requirements imposed by microspacecraft designs with respect to minimum thrust and impulse bit values, Impulse bits, however, will have to be lowered even further beyond the values obtainable with todays smallest thrusters for Class II microspacecraft. In addition, significant leakage concerns exist for cold gas systems and the required high-pressure storage tanks will completely dominate microspacecraft design with respect to both size and mass, even for relatively benign attitude control requirements. Ammonia cold gas thrusters or hydrazine warm gas systems may provide fairly near-term solutions to the propellant storage and leakage issues. For spacecraft with masses of less than 5 kg, virtually no suitable propulsion hardware exists, neither for primary propulsion, nor attitude control. In many cases, existing thrusters are larger than the spacecraft in question.

Future technology needs for both primary and attitude control propulsion will be required, in particular for spacecraft below 5 kg in mass. Howevet, even for larger spacecraft (5 - 20 kg), significant improvements for high-specific impulse primary propulsion will need to be made in order to reduce propellant masses , thus aiding in keeping microspacecraft small. Also, further reductions with respect to I-bit performance are required.

New advanced micropropulsion technologies introduced include both attitude control thrusters of the vaporizing liquid (resistojet) and subliming solid thruster types, and micro-ion engines for primary propulsion applications. All devices make heavy use of advanced microfabrication techniques, such as silicon micromachining. Several devices *were* designed and built over the course of the past year and represent the latest technology advances in the micropropulsion area, specifically aimed at addressing microspacecraft technology needs.

VI, ACKNOWLEDGMENTS

The author would like to thank Ross Jones from JPL and the NASA Advanced Propulsion Concept program for supporting this work. He also wishes to thank William Tang, Wen Li, Indrani Chakraborty, Lilac Muller, Thomas

George, Andrew Wallace, Russell Lawton, Dave Bame, Morgan Parker and John Blandino for their contributions to the JPL micropropulsion activities, and John Brophy for reviewing this paper.

The work described in the paper was conducted at the Jet Propulsion Laboratory under contract with the National Aeronautics and Space Administration.

VII. REFERENCES

'Collins, D., Kukkonen, C., and Venneri, S., "Miniature, Low-Cost, Highly Autonomous Spacecraft - A Focus for the New Millennium", IAF Paper 95- U.2.06, Oslo, Norway, Oc(. 1995.

'West, J., Pers. Comm., JPL, Fall 1995.

'Fleeter, R., "Microspacecraft", The Edge City Press, Reston, VA, 1995.

'Mueller, J., "Propulsion Options for Microspacecraft", JPL Internal Document, JPL-D-13444, Dec. 4, 1995.

'Jones, R., "JPL Microspacecraft Technology Development (MTD) Program", JPL Internal Document, May 31, 1996.

'West, J., "Microelectromechanical Systems (MEMS)/Nanotechnology Studies", JPL Internal Document, JPL-D-1 3302, January 12, 1996.

'Proceedings, JPL Micropropulsion Workshop, April 7-9, 1997, to be published.

'Brown, C. D., "Spacecraft Propulsion", AIAA Education Series, 1995.

[9]Ruttle, D. and Fitzsimmons, M., "Development of Miniature 35-lbf Fast Response Bipropellant Divert Thruster", AIAA Paper 93-2585, Monterey, CA, June 1993.

'[0]Hodge, K. F., Allen, K. A., and Hemmings, B., "Development and Test of the ASAT Bipropellant Attitude Control System (ACS) Engine", AIAA Paper 93-2587, Monterey, CA, June 1993.

[11]Craddock, J. and Janeski, B., "Design and Development of the Army KE ASAT ACS Thruster, AIAA Paper 93-1959, Monterey, CA, June 1993.

[12]Rosenberg, S.D. and Schoenmann, L., "New Generation of High-Performance Engines for Spacecraft Propulsion", *JPP*, Vol.10, No. 1, Jan-Feb. 1994. pp.-40-46.

1[1]Schwende, M. A., Schulte, G., Dargies, E., Gotzig, U. and Scharli-Weinert, E., "New Generation of Low-Thrust Bi-Propellant Engines in Qualification Process", AIAA Paper 93-2120, Monterey, CA, June 1993.

[14]Chirivella, J., "Analysis of the Transient Behavior of the 4-Newton Attitude Control Thruster of the Comet Rendezvous Asteroid Flyby (CRAF) Spacecraft", JPL Internal Document, JPL P.O. 958214, Nov. 25, 1988.

"Sutherland, G.S. and Maes, M. E., "A Review of Microrocket Technology: 10^{-6} to 1 lbf Thrust", J.Spacecraft *and Rockets*, Vol. 3, No. 8, August 1966, pp. 1153 - 1165.

[16]Wichmann, H. and Fitzsimmons, M., "Miniature High performance Delta V Engine", AIAA Paper 93-2582, Monterey, CA, June 1993.

'[7]Kaiser-Marquardt, Company Information.

"Royal Ordnance, Company Information.

I"Atlantic Research, Company Information.

[20]Aerojet, Company information.

"Bryant, K., Knight, C., and Hurtz, R., "Planetary Lander Vehicles Utilizing LEAP Technology", AfAA Paper 94-2748, Indianapolis, IN, June 1994.

'* Primex, Company Information.

"TRW, Company Information.

[24] Jankowsky, R. S., HAN-Based Monopropellant Assessment for Spacecraft", AIAA Paper 96-2863, Lake Buena Vista, FL, July, 1996.

[25]deGroot, W.A. and Oleson, S. R., "Chemical Microthruster Options", AIAA Paper 96-2868, Lake Buena Vista, FL, July 1996.

[26]Grossmann, L, Jones, I. R., and Lee, D. H., "Auxiliary Propulsion Survey, Part III: Survey of Secondary Propulsion and Passive Attitude Control Systems for Spacecraft", AFAPL-TR-68-67, Part Ill.

*'Sutton, G. P., "Rocket Propulsion Elements", 6th cd., Wiley, 1992.

'*Sellers, J. J., Meerman, M., Paul, M., and Sweeting, M., "A Low-Cost Propulsion Option for Small Satellites", *Journal of the British Interplanetary Society,* Vol. 48, pp. 129-138, 1995.

[29]Heister, S., "Solid Rocket Motors", in "Space Propulsion Analysis and Design", edited by Humble, R. W., Henry, G. N., and Larson, W. J., Space Technology Series, McGraw-Hill, 1995, pp. 295.

[30]McGrath, D., 'The History of Thiokol STAR™ Motor Missions", AIAA Paper 95-3129, San Diego, CA, July 1995.

[31]McGrath, D., "Motors with Movable Nozzles", AIAA 95-3020, San Diego, CA, July 1995.

[32]Thiokol Space Motors Catalogue.

"Atlantic Research Solid Motor Catalogue.

[34]Todd, M., Personal Comm., Pacific Research, Oct. 13, 1995.

[35]Altman, D. and Humble, R., "Hybrid Rocket Propulsion Systems", in "Space Propulsion Analysis and Design", edited by Humble, R. W., Henry, G. N., and Larson, W. J., Space Technology Series, McGraw-Hill, 1995, pp. 365.

[36]Helmy, A. M., "Chronicle Review of the Hybrid Rocket Combustion", AIAA Paper 94-2881, Indianapolis, IN, June 1994.

[37]Wernimont, E.J. and Meyer, S. E., "Hydrogen Peroxide Hybrid Rocket Engine Performance Investigation", AIAA 94-3147, Indianapolis, IN, June 1994.

[38]"RITA Ion Thruster Assembly", DASA Company Brochure.

[39]Bassner, H., Berg, H.P. and Kukis, R., "The Design of RITA Electric Propulsion System for Sat 2 (Artemis)", AIAA Paper 90-2539, Orlando, FL, July 1990.

[40]Beattie, J. R., Williams, J. D., and Robson, R.R., "Flight Qualification of an 18-mN Xenon Ion Thruster", IEPC Paper 93-106, Seattle, WA, Sept. 1993.

[41]Brophy, J.R, Pless, L. C., Mueller, J., and Anderson, J.R., "Operating Characteristics of a 15-cm-dia. Ion Engine for Small Planetary Spacecraft", IEPC Paper 93-110, Seattle, WA, Sept. 1993.

[42] Brophy, J.R, "Stationary Plasma Thruster Evaluation in Russia", JPL Internal Document, JPL Publication 92-4, March 15, 1992.

[43] Khayms, V. and Martinez-Sanchez, M., "Design of a Miniaturized Hall Thruster for Microsatellites", AMA Paper 96-3291, Lake Buena Vista, FL, July 1996.

[44] Marcuccio, S., Ciucci, A., Oest, H., Genovese, A., and Andrenucci, M., "Flight Demonstration Opportunities for FEEP", AIAA Paper 96-2724, Lake Buena Vista, FL, July 1996.

[45] Mitterauer, J., "Miniaturized Liquid Metal Ion Sources (M ILMS)", *IEEE Transactions* on *Plasma Science,* Vol. 19, No.5, Oct.1991, pp. 790-799.

[46] Mitterauer, J., "Prospects of Liquid Metal Ion Thrusters for Electric Propulsion", IEPC Paper 9I-105, Viareggio, Italy, Oct. 1991.

[47] Genovese, A., Marcuccio, S., Petracchi, P., and Andrenucci, M., "Neutralization Tests of a mN FEEP Thruster", AIAA Paper 96-2725, Lake Buena Vista, FL, July 1996.

'*''Micro Thrust FEEP Systems", Centrospazio Company Brochure.

[50] Blandino, J. J., Pers. Comm., June 26, 1997.

[51] Perel, J., Bates, T., Mahoney, J., Moore, RD., and Yahiku, A. Y., "Research on Charged Particle Bipolar Thruster", AIAA Paper 67-728, Colorado Springs, CO, Sept. 1967.

[52] Hubermann, M.N. and Rosen, S. G., "Advanced High-Thrust Colloid Sources", *J. Spacecraft,* Vol. 11, No. 7, July 1974, pp. 475-480.

'[1] Strand, L., Toews, H., Schwartz, K., and Milewski, R., "Extended Duty Cycle Testing of Spacecraft Propulsion Miniaturized Components", AIAA Paper 95-2810, San Diego, CA, July 1995.

[54] Nakazono, B., "Second-Generation Microspacecraft Propulsion Analysis and Alternatives", JPL Internal Document, JPL-D-14038, Nov. 8, 1996.

[55] Morash, D.H. and Strand, L., "Miniature Propulsion Components for the Pluto Fast Flyby Spacecraft", AIAA Paper 94-3374, Indianapolis, IN, June 1994.

'[6] Bzibziak, R., "Miniature Cold Gas Thrusters, AIAA Paper 92-3256, Nashville, TN, July 1992.

"Barber, H. E., Falkenstein, G. L., Buell, C, A., and Gurnitz, R. N., "Microthrusters Employing Catalytically Reacted N_2-O_2-H_2 Gas Mixtures, Tridyne", *J.Spacecraft and Rockets,* Vol. *8, No. 2,* February 1971, pp. 111 - 116.

"Mueller, J.M. and Mcfarlane, J. S., "Design of Tridyne Pressurization Systems for Liquid Oxygen Polybutadiene Hybrid Rocket Motors", AIAA Paper 91-2406, Sacramento, CA, June 1991.

[59] "Pulsed Plasma Thruster", Primex Company Information.

[60] Lincoln Composites Company Information.

[61] Miksch, R.S. and Heller, K. G., **"Design and Development of a Vaporjet Attitude-Control** System for Space Vehicles", ATL-D-648, Contract AF 33(616)-7044, Wright-Patterson AFB, Dec. 1961.

[62] Tinling, B. E., "Measured Steady-State Performance of Water Vapor Jets for Use in Space Vehicle Attitude Control Systems", NASA TN-D- 1302, NASA, Washington, May 1962.

[63] Kanning, G., "Measured Performance of Water Vapor Jets for Space Vehicle Attitude Control Systems", NASA TN D-3561, NASA, Washington, 1966.

'Mueller, J., Leifer, S. D., Muller, L, and George, T., "Liquid Vaporizing Thrusters", *NASA Tech Brief,* Vol. 21, No. 5, 1997, pp. 73.

"Mueller, J., Tang, W., Wallace, A., Li, W., Bame, D., Chakraborty, 1., and Lawton, R., "Design, Analysis and Fabrication of a Vaporizing Liquid Micro-Thruster, AIAA Paper 97-3054, Seattle, WA, July 1997.

"'Subliming Solid Control Rocket", Final Report, NASA CR-711, Contract NAS 5-3599, Rocket Research Corp., June 1965.

[67] "Development of the Subliming Solid Control Rocket Phase 11", NASA CR-712, Contract NAS 5-9070, Rocket Research Corp., Seattle WA, March 1967.

[6]* ''Application of the Valveless Subliming Control Rocket to the NRL Gravity Gradient Satellite", RRC-66-R-59, Contract Nonr-5123(00)(X), Naval Research Laboratories, May 4, 1966.

[69] Hardt, A. P., Foley, W. M., and Brandon, R. L., "The Chemistry of Subliming Solids for Micro Thrust Engines", *Astronautica Acts,* Vol. 11, No. 5, 1965, pp. 340-347.

'"Owens, W. L., Jr., "Design Aspects of Subliming Solid Reaction Control Systems", AIAA Paper 68-516, Atlantic City, NJ, June 1968.

[71] Kindsvater, H. M., "Simplified Space Mechanisms Using Sublimng Solids", Proceedings of the First Aerospace Mechanisms Symposium, University of Santa Clara, Santa Clara, CA, May 19-20, 1966.

[72] Owens, W. L., "An Experimental Study of Superheated Subliming Solid Thruster Performance", AIAA Paper 70-210, 8th Aerospace Sciences Meeting, Jan. 1970.

[73] Hardt, A. P., Foley, W. M., and Brandon, R. L., 'The Chemistry of Subliming Solids for Micro-Thrust Engines", AlAA Paper 65-595, Colorado Springs, CO, June 1965.

[74] Kindsvater, H. M., "Design Criteria for Subliming Solid Applications", ICPRG/AIAA Solid Propulsion Conference, Anaheim, CA, June 6-8, 1967.

'*Forsythe, R. W., "Impulse and Thrust Stand Test of a Subliming Solid Micropropulsion System", NASA TN D-3245, NASA Goddard Space Flight Center, March 1966.

[76] Federline, F., "Development of the Subliming Solid Control Rocket - Contributions from the Explorer XXXV (AIMP-E) Satellite Program", Document X-723-68-227, NASA Goddard Space Flight Center, June 1968.

[77] Greer, H. and Griep, D. J., "Dynamic Performance of a Subliming Solid Reaction Jet", Report No. TR-1001 (2230-33)-1, Aerospace Industries, Dec. 1966.

[78] Berkopic, F. D., "Performance of Two Subliming Solid Propellant Thruster Systems for Attitude Control of Spacecraft", NASA TN D-3841, Lewis Research Center, 1967.

[79] Maycock, N. and Pai Vemeker, V. R., "A Photochemical Microrocket for Attitude Control", *J. Spacecraft and Rockets,* Vol. 6, No. 3, March 1969.

[80] Mueller, J, Muller, L., and George, T., "Subliming Solid Micro-Thruster for Microspacecraft", New Technology Report NPO- 19926/9525, JPL Technology Utilization Office, April 22, 1996.

[81]Brophy, J. R., Polk, J. E., Blandino, J., and Mueller, J., "Micromachined Ion Accelerators", *NASA Tech Briefs*, Vol. 20, No. 2, February 1996, pp. 36-37.

'*Hastings, D, Progress Report on the MIT Microthruster Contract, Jan. 1997.