A Systems Study of Propulsion Technologies for Orbit and Attitude Control of Microspacecraft

by

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Abstract

A systems study was conducted to determine which propulsion technologies have the most promise for future use on microspacecraft in order to enable enhanced orbit and attitude control that is currently not generally available on spacecraft of these scales. In addition, it identifies what technical hurdles need to be addressed to make the application of these technologies to these smaller spacecraft feasible.

Six missions were chosen to be representative of the types of missions that future advanced microspacecraft will perform, and propulsive requirements of each mission were analyzed. Concurrently, those propulsion technologies that seemed most likely to be used on future microspacecraft were identified. Since most of these technologies do not currently exist on the required small scale, a scaling analysis for each was performed, and mass models for the entire propulsion system, including thruster, propellant, tanks, valves, and power supply and conditioning equipment, were developed.

For each of the six missions, the technologies were matched to the propulsive requirements previously developed, and total propulsion system mass determined for each technology choice. The technologies with the lowest system mass were chosen as the appropriate technology for the given mission. Of these the most promising technologies for microspacecraft in general were selected, and the required technical hurdles identified.

Based on total system mass, many technologies appear promising for application to microspacecraft. For low thrust applications, particularly for fine position and attitude control, Pulsed Plasma Thrusters are the technology of choice. For impulsive orbit maintenance and control, advanced miniature chemical propulsion technologies, namely mini-hybrid motors and micro-pumped bipropellant rockets, are most appropriate. For missions with very high ΔV requirements Ion engines are preferred, and for missions with medium ΔV requirements that require low thrusts, Hall thrusters appear to be the best choice of technology.

Thesis Supervisor: Jack L. Kerrebrock
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Acknowledgments

Wow. I can't believe I'm actually writing this part. And though the official thesis due date of May 10th was a week ago, the so-called "hard" deadline is still more than a week away. One could argue that I'm about to turn something in *early*. Wonders never cease, I guess. Though it is my name on the title page, this thesis would not have been possible with the advice, friendship, and help of many people, a few of whom I'd like to thank.

First and foremost, a huge thanks to my advisor, Prof. Jack Kerrebrock, who was kind enough to actually listen when I walked into his office with some hair-brained ideas about microthrusters during my senior year, even more kind to decide that this might not be a bad idea for a thesis, and especially kind to agree to advise me on the project and help me get funding for it. However, I am most indebted to him singlehandedly deciding that it really made sense for me to do the whole thing in a year (and then convincing me to actually do it), because now we can get started on the really fun stuff—actually building something—even earlier! I thank you for your time, patience, never-ending enthusiasm, and great stories. I am greatly looking forward to a few more years of working with you to build the world's smallest rocket motor!

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To all my friends at Baker, thanks for all the free beer! (oh, and while I'm at it, all the good times and friendships as well!)

Last, but very far from least, a monstrous debt is owed to my great family. Anne, you are quite possibly the perfect sister. Thanks for coming to Brown so we can hang out more. Mom and Dad, my writing skills are insufficient to record my feelings here. Thanks for helping me to hang in there, for always believing in me, and for making this all possible. It's been said before, but you guys are truly the greatest!

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Chapter 1

Introduction

1.1 Microspacecraft Background and Motivation

Though many seem to think that the current craze over small spacecraft and the mantra of "smaller, faster, better, cheaper..." is a completely new idea, microspacecraft, which for the purposes of this study will be defined as those spacecraft with masses between 10 and 150 kg, have been around since the very beginnings of man's ventures into space. Sputnik had a mass of about 85 kg; Explorer 1 a mass of about 15 kg; and the mass of Vanguard 1 was less than 1.5 kg. This was not because people did not know how to build large spacecraft, but because the launch vehicles of the time were incapable of putting larger masses into orbit. However, as launch technology advanced, launch capacity did as well, and larger and larger spacecraft became possible. In the political and technical environment of the day, bigger was better, and so large, complex, many-function spacecraft were built.

Recently, due in a large part to federal budget constraints, the emphasis has switched almost entirely to cost. In the past few years, as spacecraft costs continued to increase dramatically, and launch costs remained high, there has been an increasing realization that small spacecraft provide somewhat of a reprieve from this problem. The most basic reason is that they are smaller, so the costs of building them and the costs of placing them into orbit, both of which generally scale with mass, are less.

An additional motivation for small spacecraft comes from the fact that they can be designed as a specialized platform for more specific functions. For example, in the old paradigm of large and complex spacecraft, a single spacecraft served as host to many different and often diverse functions. In some cases this lead to inefficiencies in design which were undesirable, but more importantly, if the spacecraft failed, all functions were lost. In the microspacecraft paradigm, these multitude of functions would be divided up among different spacecraft. This accomplishes two things: First, it allows the individual spacecraft to be less complex, smaller, and in some cases more efficiently designed. In many instances this leads to satellites that are less expensive. Secondly, since these smaller satellites are easier to produce, their development life cycles are smaller, and more can be produced. It is this mentality that has allowed NASA to plan on sending at least one spacecraft towards Mars during each of the Earth-Mars launch windows for the next few years.

It is the author's belief that there is an additional effect from the shift of focus from the large to the small and micro scales. By setting upper limits on cost and mass more explicitly, designers have been forced to be more ingenious and creative in their solutions. It is this "forced" creativity that has played a large part in making microspacecraft more capable over the past few years, and it is what will continue to drive this emerging industry.

It is worth noting, however, that the trend today seems to be repeating history. Designers are again placing more and more function, and thus more and more complexity, into the same sized box, only this time the box is smaller. It will be very interesting to see how this develops, and to see if the push for less complex spacecraft that originally spawned the micro-space industry will instead become a push toward more complex spacecraft, only smaller ones. The resolution of this issue lies in the realization that the true objective of the microspacecraft is a reduction in cost. If a given function can be provided at a lower cost in a smaller box, then this is sufficient motivation to do it. However, at some point in a given design, the added pain and cost of making something much smaller will not be outweighed by the corresponding reduction in launch, labor, or additional hardware costs, and it is at this "line in the sand" where the miniaturization must end for that particular design.

This project will determine if there are propulsion technologies that could be applied to micro-spacecraft that will further increase the capabilities of microspacecraft without crossing this "line in the sand" where the cost in terms of added complexity, added processing and handling time, added safety constraints, added mass, and everything else is outweighed by the additional functions and capabilities that this new technology would provide.

1.2 Control of Microsatellites

Most microsatellites up to the present have used passive orbit and attitude control methods. Gravity gradient stabilization or spin stabilization have been the predominate methods of attitude control employed, and most microsatellites are not designed with the means for re-boost or orbit correction, relying on their launch vehicle alone to place them in their final orbit. However, now that more and more demands are being placed on microsatellites, more capable and accurate control methods are required.

1.2.1 Orbit and Position Control

There are an increasing number of proposed missions for microsatellites, especially those deployed in constellations, that require orbit maintenance. This orbit maintenance generally falls into three categories: initial orbit insertion and correction, station-keeping and orbit maintenance, and re-phasing.

Initial Orbit or Trajectory Insertion

Launch vehicles attempt to deliver satellites to their nominal orbit, but often there is a small error in the insertion. For many missions, a perfect insertion into the nominal orbit is not necessary, provided it is within some given tolerance. However, for constellations of satellites, where it is very important to maintain a constant relative separation of satellites within a plane, it is vital that all the satellites end up in the same plane in nearly identical orbits. One possible way of insuring this is to add some propulsion capability to the satellite itself, which would allow it to make the small corrections necessary to place itself in the correct orbit. For many geostationary satellites and others in high orbits, the launch vehicle usually does not place the satellite in its final orbit. It places the satellite in a geostationary transfer orbit, and the satellite itself is responsible for the final circularization and fine-tuning of its orbit.

Station-keeping and Orbit Maintenance

Station-keeping and orbit maintenance are used to maintain both the satellite's orbit and its position within that orbit. These functions will be considered together, because the terms are often used interchangeably, and in many instances it is impossible to determine whether a given maneuver should be classified under station-keeping or orbit maintenance. As a basic definition, station-keeping is used to maintain a satellites position either relative to another satellite, or relative to an idealized location within an orbit. Orbit maintenance is used to maintain the shape, orientation, and location in space of the orbit itself. For example, maintaining sixty degree angular separations between six satellites in an orbital plane is an example of station-keeping, while keeping the orbit from decaying due to atmospheric drag and causing the satellite to re-enter would be an example of orbit maintenance. The vagueness comes, for example, in the case of what is termed North/ South Station-Keeping (NSSK), where the problem is manifested by a north/south wandering of the satellites position as viewed from on the earth, but is a result of a slowly increasing inclination of the orbit itself.

Orbit Re-phasing

Re-phasing is changing the relative position of a satellite within an orbit. An example would be moving a geostationary satellite from over Africa to over the Pacific Ocean. In constellations of satellites re-phasing would be used if one satellite were to fail, and the others needed to spread out slightly relative to each other to ensure a more uniform distribution within the orbit. Because many proposed uses for microsatellites involve employing a large number of them in a constellation, providing the propulsive capability for rephasing in the event of the failure or replacement of a single member of the constellation should prove very useful.

1.2.2 Attitude Control

The desire for more accurate attitude control methods can be seen with the increasing number of microsatellite designs that do contain some kind of active attitude control. Up to this point, this has usually been accomplished using magnetic torquers. However, there are some prospective missions for microspacecraft that require even more accurate attitude control than the torquers can deliver, so people are now beginning to propose three-axis stabilized satellites. Most of these proposals have focused on using reaction or momentum wheels as the primary actuator, as this is what is traditionally done in the larger spacecraft of today. However, this method has never been proven at such small scales, and it is possible that simple propulsive systems for attitude could prove advantageous in some situations.

1.3 Purpose of this study

As the demands on microspacecraft continue to increase, they will require advanced capabilities that are not currently available at their scale. Though there are many examples of these capabilities that need to be addressed, one of the most pressing in the author's opinion is that of controlling the orbit and attitude of the spacecraft. If propulsion technologies can be developed or identified that will allow this to be done with greater fidelity at reasonable cost, the usefulness of microspacecraft will be greatly enhanced.

This study will attempt to identify the most promising propulsion technologies that could be used to provide orbit and attitude control of microspacecraft. In addition it will attempt to identify what technical hurdles and roadblocks need to be cleared in order for these promising technologies to be applied successfully to microspacecraft. It is worth noting that there are a few separate but closely related issues which are not addressed in this study, though they certainly could and should be addressed in the future. The first is high-accuracy attitude control of small spacecraft using non-propulsive technologies. In larger satellites, reaction wheels have become the dominate actuator for fine three-axis attitude control systems. With sufficient miniaturization, a similar trend could develop in microsatellites. The second related issue that is not addressed is that of position and attitude determination. If there is no way to determine a spacecraft's position and orientation accurately, it does little good to attempt to control it. Though both of these issues are vital in the long-range success of microspacecraft, neither is within the scope of this study, and are left for future work and research within this field.

1.4 Methodology

The study began with a literature search, though this continued throughout the process. Next, some time was spent investigating the possible missions of future microspacecraft, as well as developing a way of classifying these missions. Because looking at all possible missions (or even a large subset of them) in a general sense proved to be much too broad, six representative missions were chosen to use in evaluating possible new technologies. For each mission, the mission requirements were translated into technical requirements for the orbit and attitude control systems.

Concurrently, possible propulsion technologies for spacecraft control were identified. These fell into three categories: technologies that had been used in microspacecraft; technologies that had not yet been used on the microspacecraft scale, but had been used in larger spacecraft; and technologies that had not been used at any scale. Though not a focus for this study, the already used technologies were included as "controls" to verify that newer technologies would, in fact, be worth the necessary investment. For those technologies that had not been used in microspacecraft, some very rough scaling and/or modeling was done to determine the characteristics necessary for them to be included in the mission trade studies.

Once the technologies had been identified, each mission was considered in turn, and technologies were evaluated in the context of that mission to determine which technologies were best suited for use in it. Once this part of the study was concluded, a group of technologies that were the most promising were selected for further investigation. Next, the technical hurdles of these promising technologies that remain to be cleared before they can be put to use in microspacecraft were identified and presented. Finally conclusions and recommendations for future work in this area were made.

Chapter 2

Introduction to Representative Missions

2.1 Introduction

At the beginning of the project, it was hoped that a general methodology could be developed which would allow one to know what the best technologies for orbit and attitude control would be for a given mission. In order to do this, a way of describing microspacecraft missions was required, and the concept of a six-dimensional mission-space was developed. Preliminary work in attempting to develop this generalized methodology re-validated the axiom that design can never be completely generalizable, and the attempt to create a generalized methodology was abandoned. Instead, the study was restricted to a limited number of missions which would be representative of the types of missions envisioned for future microspacecraft. This chapter will describe the processes involved in developing the concept of mission-space, the first attempts at a generalizable technology selection process, discuss how this broke down, and then introduce each of the six representative missions chosen.

2.2 Mission-Space

Because there are so many possible missions for microspacecraft, it was necessary to develop some kind of classification system for them that would help to show the kinds of groupings that microspacecraft most frequently fall into. To accomplish this, an attempt was made to identify which attributes of a given mission were most important in characterizing that mission. Six attributes were identified, which included the spacecraft's mass, orbit, positioning accuracy, pointing accuracy, lifetime, and satellite specific power. It is believed that these mission attributes are nearly independent of one another, and thus a given combination of them defines a certain class of missions. For this reason, these six attributes have been termed the dimensions of a six-dimensional "Mission-Space." Figure 2.1 presents a graphic representation of this space, along with two sample mission-points, to show how missions would be represented, with each point represented by a line snaking through the grid.

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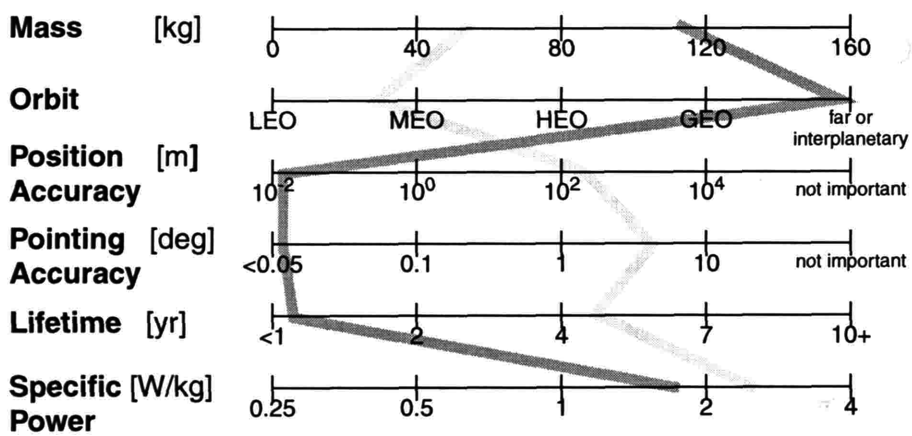


Figure 2.1: Mission Space

The mass dimension provides the overall scale of the spacecraft. Some might argue that mass is not something that should be taken as a "given" or as an input into the design process, but rather something that should be derived or determined as the process progresses. This would be a valid argument to a certain extent if the mission space was to be used to design an entire mission, but in this case, its function is to classify missions to help determine what technology should be used for control. In most design processes, a fairly accurate estimate of the overall mass of the spacecraft is in hand by the time decisions about control systems are being made.

The orbit dimension classifies the mission based on where in space it is to occur. The mission's location or orbit determines what types of disturbance forces and torques will be acting on the spacecraft, and these disturbances are critical in sizing and selecting a control system for the spacecraft.

The position accuracy dimension classifies a mission based on how accurately the spacecraft has to maintain its position. This translates into how fine the adjustments that orbit control system makes must be. The position accuracy can be measured relative to another spacecraft or to some idealized orbit or trajectory.

Similarly to the position accuracy dimension, the pointing accuracy dimension classifies a mission based on how accurately it must maintain its attitude. This attitude can be measured relative to an inertial direction in space or to some other body such as another spacecraft or Earth.

The lifetime dimension classifies the mission based on how long it must function. This has implications for the quality and rating of the parts used in the control systems, as well as the quantity of reaction mass used in the case of propulsive technologies. It also affects the sizing of solar arrays if they are used, as their output tends to degrade with time.

The specific power dimension classifies the mission based on the mission's power density, and determines how much power is available on the spacecraft. It is given as specific power rather than total power to ensure that this dimension and the mass dimension are as independent as possible.

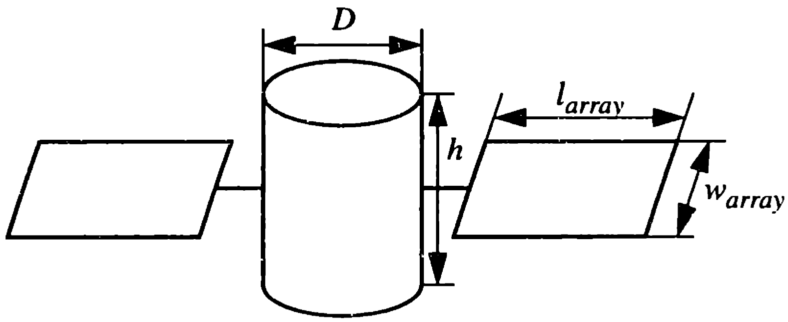
2.3 Ideal Methodology

Ideally, when designing a new microspace mission, one would see which point in mission space it corresponded to, and based on that point, would know which propulsion technology was the "best" for that mission. The ideal methodology would be the process that the designer would go through to arrive at this result. Its input would be the given point in mission-space, and its output, after following a particular set of steps, would be the ideal propulsion technology for that mission. As the reader will quickly realize, this is an rather idealistic approach, and would rarely work in practice. Design is not, nor can it ever become, a pure formulaic, flow-down process. There will always be factors other than those illustrated in the above mission-space that will play a role in determining which propulsion technology is the optimum choice for that specific mission development. But the study proceeded on the assumption that an attempt to systematize the process would provide some insight.

2.4 Attempts at Ideal Methodology

It was believed that if such an methodology could be developed for the mission-space as described above, it would provide a tool that could be used during the design phase, which, when given the dimensions of mission-space as inputs, would identify the most promising propulsion technologies for that particular mission. By eliminating some options and encouraging others, the job of the designer could be made easier. An attempt was initially made to develop this idealistic methodology.

This attempt involved creating a "generic" spacecraft with a constant density which could be scaled in size based on its mission-space mass dimension. Figure 2.2 illustrates this "generic" spacecraft. The size of its solar panels (if any) was governed by the specific power and mass dimensions. Based on the orbit dimension and the mass dimension, the



Scaling Summary:

$\rho = 250 \text{ kg/m}^3$
$h = 4/3D$

$w_{array} = D$
$l_{array} = f(\text{Power})$

Figure 2.2: Generic Satellite

torques and forces acting on the spacecraft could be calculated. Once the disturbances were understood, the thruster forces required to maintain the spacecraft within the position and attitude tolerances specified by their respective dimensions could be calculated. Finally, based on the lifetime dimension, the overall effective changes in velocity (ΔV's) could be determined. Once the thrusts, powers, and ΔV's are known, the overall system mass of the propulsion system could be estimated for each possible technology, using the information and scaling laws presented in Chapters 4 and 5. With the system masses, the different technologies could be compared in a manner similar to what ended up being done, and a map from mission space to ideal propulsive technology could be created.

2.5 Need for Representative Missions

As this methodology was being developed, it became clear that although the above steps made sense in theory, it was extremely difficult to implement them successfully in practice. The required thrusts depend heavily on the duty cycle of the thrusters, which is not taken into account in the mission-space, nor can it be considered a characteristic of a given technology. Further, the method proposed only takes into account the nulling of disturbances, but in reality, as was mentioned in the previous chapter, propulsion systems are often used for much more, including providing and fixing orbital insertions, changing the phase of the orbit, and providing other kinds of station-keeping than can be generally expressed in the model, such as non-keplarian orbits. In attitude control, thrusters can be used to provide disturbance nulling, as this model would suggest, but they can also be used for sluing the spacecraft at different rates to different orientations or to desaturate momentum or reaction wheels. Finally, each spacecraft that is actually designed and built requires a different combination of all of these capabilities. It became fairly clear that whatever product could be produced by this generalized model and methodology would be too simplistic to provide a reasonably valuable tool to the mission designer, and the attempt to develop such a methodology was abandoned.

Instead, the decision was made to select a number of representative microspacecraft missions with varied propulsion requirements Each would be evaluated and the most promising technologies for it identified. It was reasoned that if a technology proved useful to a number of different missions with varied propulsive requirements, it would prove useful to the microspacecraft field in general.

2.6 Brief Description of Representative Missions

Six representative missions were chosen in an attempt to provide a reasonable spectrum in the types of propulsion requirements that would be required by advanced microspacecraft in the fairly near future.

2.6.1 Separated Spacecraft Interferometry

This is a mission proposed by the Jet Propulsion Laboratory for the New Millennium series of missions. It would involve three small spacecraft, two collector spacecraft and one combiner spacecraft. Operating away from Earth, but in Earth's orbit around the sun, the two collector spacecraft would be separated by a baseline of 1-10 km with the combiner spacecraft in between them. The collectors would each point to the same location in space, and reflect the light they collected to the collector spacecraft where the two optical beams would be interfered. By looking at the interference patterns, much information about the object being imaged can be infered at significantly higher resolutions than would be possible otherwise thanks to the large baseline between the collectors. The propulsion requirements in this mission are primarily extremely accurate relative station-keeping and extremely fine attitude control. The system is currently baselined with cold gas thrusters for both functions, and the low specific impulse (I_{sp}) of such thrusters is one of the reasons for the relatively short mission lifetime of six months.

2.6.2 Future Global Positioning System

The NAVSTAR Global Positioning System has only been fully operational for about five years, but there is already much talk as to what will replace it. It is certainly possible that this mission could be accomplished on smaller satellites than it currently employes, though this will require significant advancement and miniaturization of atomic clocks. Assuming this could occur, this conceptual mission would be that of a small satellite for a GPS system. The primary propulsion requirement to be investigated is the capability of phasing to ensure that the most even coverage of the Earth possible is maintained in the event of a failure or loss of another satellite in the constellation. Another requirement is that of fixing any initial errors in the orbital insertion provided by the launch vehicle. In addition, the feasibility of providing propulsion onboard the satellite for the LEO-HEO transfer will be investigated.

2.6.3 Earth Observing Cluster

This mission is a conceptual one that makes use of a co-orbiting cluster of satellites to provide a larger aperture, which leads to a higher resolution in Earth Observing applications, without requiring the expense and mass of a full, solid aperture. This requires that the satellites orbit in such a way that they do not move relative to one another. This requirement would be fairly trivial if they all remained in the same orbital plane, but it is desirable to have some of the sparse aperture collectors out of the plane of the baseline orbit, so they form more of a circle than simply a line. These kinds of orbits are called non-Keplarian orbits, and require continuous thrusting to maintain them. This will be the dominate propulsion requirement considered, but basic attitude control and fixing of initial insertion errors will be looked at as well.

2.6.4 MiniMars Mission

This is another conceptual mission, to determine if it is feasible for a relatively small spacecraft to propel itself on interplanetary trajectories. Currently interplanetary spacecraft are usually inserted into their interplanetary trajectories by an additional stage of their launch vehicle. If this stage could be eliminated or reduced in size, either a smaller launch vehicle could be used or additional payload and function could be added to the spacecraft. The propulsion requirements to be investigated in this mission are almost exclusively for the interplanetary transfer aspect of the mission, though other possible uses while near Mars for the transfer engine and its required power will be looked at as well.

2.6.5 Next Generation Low Earth Orbit Communication System

This mission, also conceptual, is a future LEO-based world-wide communication system. Orbital Sciences Corporation is in the process of deploying such a system, but the only propulsive capabilities in the current satellite design is a cold gas thruster that is used to provide final precision orbital insertion. Once the orbit is finalized, the tanks are run dry to ensure that any propellant leaks over the satellite's lifetime do not cause additional disturbance forces and torques on the satellite. Any phasing that is required is accomplished through varying the exposed surface area of the satellites to produce a drag force on one satellite that is larger than the corresponding drag force on the others. It was felt that it would be useful to investigate the feasibility of providing additional propulsive capabilities on a small LEO communication system and to determine what advantages, if any, these additional capabilities would provide. The propulsion requirements that will be investigated for this mission are phasing to maintain even spacing in the orbital plane, fixing of initial orbital insertion, attitude control, and possibly drag makeup.

2.6.6 Low Altitude Earth Observation

This mission is based on an Earth observation satellite currently being developed by Draper Laboratory. They are satisfied with their planned method of attitude control for the spacecraft, but due to its relatively low altitude, are concerned with the propulsion requirements of drag makeup. Additional propulsive requirements to be investigated are initial orbital insertion correction and the possibility of a periodic phasing of the satellite within its orbit.

Chapter 3

Requirements of Representative Missions

3.1 Introduction

Once the representative missions had been selected, their specific requirements for propulsive systems had to be determined. This chapter will first discuss the four functional types of propulsion requirements and the technical basis for analyzing each of them. The balance of the chapter will discuss each representative mission in turn, concentrating on the process and results of the technical requirement derivation for each.

3.2 Types of Propulsion Requirements

There are four types of propulsion requirements that correspond to the control capability issues for microsatellites that were first identified in Section 1.2. These are orbit transfer and fixing of initial orbit insertion, phasing, station-keeping and orbit maintenance, and attitude control. This section will present the technical basis for analyzing each of these.

3.2.1 Orbit Transfer and Final Orbit Insertion

In many cases, a spacecraft is deposited in a so-called parking orbit by its launch vehicle, and then it is up to the spacecraft to provide sufficient propulsion capability to move itself to its final orbit or to place itself onto its final trajectory. If the orbit or trajectory of the spacecraft is especially critical, such as in the case of geosynchronous or GPS satellites, the launch vehicle might not be able to inject the vehicle into the correct orbit precisely enough, and a correction may be required. In all of these cases, orbital transfers are involved, so they will be briefly discussed.

Impulsive Orbit Transfers

Impulsive Orbit Transfers are extensively discussed in many texts[1,*e.g.*], and will not be discussed in much detail. Hohmann transfers are in almost all cases the most economical means of changing orbits (in terms of minimizing ΔV requirements). The ΔV for a Hohmann transfer is given as:

$$\Delta\upsilon = \frac{\Delta V}{V_o} = \sqrt{\frac{2}{1+\rho}}(1-\rho) + \sqrt{\rho} - 1 \tag{3.1}$$

where V_o is the initial circular orbital velocity and ρ is the ratio of the initial orbital radius to the final orbital radius.[1]

Low Thrust Orbit Transfers

In the case of most non-chemical thrusters, the available thrust is insufficient to justify the impulsive approximation, and low thrust trajectories are necessary. These have been well discussed and optimized in the literature. A simple derivation is given in Appendix A, as it is important in the low thrust phasing discussed in the following section.

Appendix A, Equation A.4 gives the radius ratio as a function of time as:

$$\rho(t) = \frac{r_o}{r} = \left[1 + c\sqrt{\frac{r_o}{\mu}}\ln\left(1 - \frac{f_o}{c}t\right)\right]^2 \tag{3.2}$$

where r_o is the initial orbital radius, c the exhaust velocity, and f_o the initial acceleration.

1. Traditionally in orbital mechanics texts, ρ (or a similar quantity) is defined as the inverse of its definition here. This definition was used to ensure consistency with the phasing analysis in Sections 3.2.2 and A.3.2. It also leads to a slightly more compact notation, in the author's opinion.

Equation A.5 gives the required ΔV to transfer to a specific radius as:

$$\Delta \upsilon = \frac{\Delta V}{V_o} = 1 - \sqrt{\rho} \tag{3.3}$$

with ρ defined as above (ratio of initial to final radii).

Interplanetary Trajectories

For those missions where interplanetary trajectories are required, the appropriate trajectories are pieced together. For the impulsive case, the spacecraft would first transfer from Earth orbit to a hyperbolic trajectory so that its velocity at infinity with respect to the earth would be sufficient when added to Earth's velocity around the sun to place the craft on the correct orbit (transfer or final) in the Sun's reference frame. The process is repeated if a capture into orbit around another planet is required. In the case of low thrust interplanetary trajectories, escape velocity is achieved (the final radius of the transfer is infinity) at which point it is assumed that the spacecraft is in orbit around the sun at the same distance as Earth. A spiral trajectory is then begun around the sun until the required radius is achieved. (and then in the case of the Mars insertion, the spiral back down from escape velocity to final orbital velocity is performed.)

3.2.2 Rephasing within Orbit

Periodically it is necessary to change the relative phase of an orbit. This process would be used, for example, to move a geostationary satellite in its orbit so that its location as viewed from Earth moved from the Pacific to the Atlantic Oceans. In the case of a polar orbiting Earth Observing satellite, this process could be used to change the local time which it passed over a particular place on Earth. In the case of a constellation, phasing could be used to set up the constellation (ensure even spacing of satellites throughout an orbit in a given plane), or it could be used in the event of a loss of satellite to re-distribute the remaining satellites evenly around the orbit. Phasing can be considered a rendezvous operation between the phasing spacecraft and an imaginary target either ahead or behind by the correct phasing angle in the same orbit. Rendezvous is discussed extensively in Chobotov[1], as is impulsive phasing briefly. However, because phasing is a less well known maneuver than orbit transferring, it will be discussed in greater detail.

Impulsive Phasing

Phaseing an orbit using impulsive thrust is simply a matter of changing the period of the orbit the satellite is in. This can be done either by increasing or decreasing the semi-major axis of the orbit. For each orbit that the satellite remains in this intermediate orbit, it will drift a certain number of degrees relative to where it would have been had it remained in the original orbit. This is illustrated in Figure 3.1. The maneuver is accomplished through two thruster firings. The first changes the size of the orbit, and the second changes the intermediate orbit back to the original orbit, once a sufficient phase change has occurred.

The impulsive phasing maneuver begins with a ΔV applied at point 1 in the original orbit at radius r_o. This places the spacecraft into the intermediate ellipical orbit shown. After one revolution in the larger orbit, the spacecraft returns to point 1. Had it remained in its original orbit it would have been at point 2, thus creating a phase difference of $\Delta\theta$. If the spacecraft remains in the larger orbit for another revolution, the total phase difference will be $2\Delta\theta$, etc.

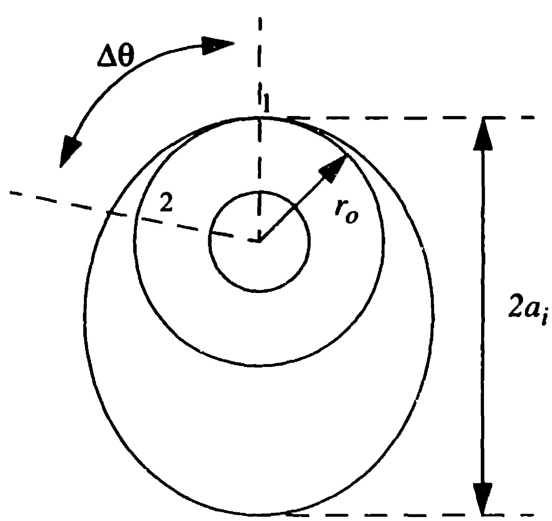


Figure 3.1: Impulsive Orbit Phasing

From the derivation in Appendix A, Equation A.9 gives the ratio of the initial radius to the semi-major axis of the intermediate orbit that is required to accomplish the required phase change in a given time as:

$$\rho = \left(1 - \frac{\Delta\theta}{2\pi\tau}\right)^{2/3} \tag{3.4}$$

where ρ is the ratio of the radius of the initial orbit to the semi-major axis of the intermediate orbit, and τ is the total time for the maneuver, given in numbers of revolutions in the original orbit.

The required change in velocity for the maneuver is given by Equation A.10 as:

$$\Delta\upsilon_{total} = 2\frac{\Delta V}{V_o} = 2[(\sqrt{2-\rho}) - 1] \tag{3.5}$$

Thus, for an impulsive phase change of a certain number of degrees in a given amount of time, Equation 3.4 gives the required change in semi-major axis, and Equation 3.5 gives the required change in velocity to accomplish the entire maneuver.

Low Thrust Phasing

In the case of electric or other low-thrust propulsion where impulsive maneuvers are not possible, the impulsive technique of placing the spacecraft into an orbit with an arbitrarily different period is not possible. Instead, the thrust must be applied continuously, which places the satellite on a spiral trajectory with a continuously changing "period." It remains in this spiral until it has built-up half the required phase difference with where its location would have been had it remained in the original orbit. At this point, the direction of thrust is reversed, and the satellite spirals back to the original orbit, arriving in it with the required phase difference. This process is illustrated in Figure 3.2.

Because there is no waiting at an intermediate orbit, the time required for the maneuver and the ratio of the initial to intermediate radii are no longer independent. As shown in

This example of a low-thrust rephasing maneuver begins with the thrust being applied in the direction of the current velocity at point 1. The satellite will follow a spiral-like trajectory (solid line) while thrusting. (In this figure, the satellite is denoted by the a-labels, while the position it would be in if it had remained in its original orbit is denoted by b-labels). By the time it reaches point 2a, it has already developed a phase difference with its original location. Once half of the desired phase difference has been reached, as at point 3, the direction of thrust is reversed, and the satellite spirals back down to the original orbit (dashed line), which it reaches at point 5a. The total chance of phase is the angle between 5a and 5b, shown as $\Delta\theta$.

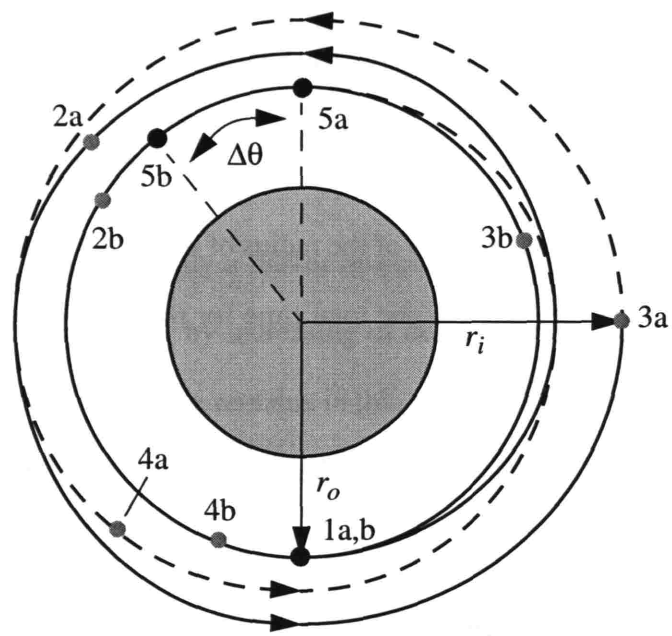


Figure 3.2: Low-thrust orbit rephasing

Appendix A, Equation A.23 gives the net phase change that a low thrust transfer from an initial orbit to an intermediate orbit and back in a total time (normalized by original orbit period) of τ as:

$$\Delta\theta = 2\pi\tau - \frac{1}{2\tilde{f}\rho_o^2}\left[1 - \left(1 - 2\pi\rho_o^2\tilde{f}\frac{\tau}{2}\right)^4\right] \qquad (3.6)$$

where $\tilde{f}$ is the constant acceleration in g's, and ρ_o is the ratio of the intial orbit radius to Earth's radius.

For a given $\Delta\theta$ and a given τ, one can determine the required acceleration to complete the specified maneuver in the specified time using Equation 3.6. Equation A.24 gives the effective ΔV required for the maneuver as:

$$\Delta\upsilon_{total} = \frac{\Delta V}{V_o} = \sqrt{\frac{r_o}{\mu}}ft = 2\pi\rho_o^2\tilde{f}\tau = 2(1-\sqrt{\rho}) \qquad (3.7)$$

3.2.3 Station-keeping and Orbit Maintenance

Station-keeping and orbit maintenance is perhaps tne most traditional and common use of propulsion capabilities on satellites today. As the real world never equates with the idealized formulas of Kepler and Newton, corrections must be made to keep spacecraft in the orbits and locations they are supposed to be in. This section will discuss the how the ΔV and thrust requirements are developed for the most common types of station-keeping and orbit maintenance.

Drag Makeup

One possible orbit maintenance requirement will be drag make-up. In this application, thrust is used to cancel the effects that aerodynamic drag has on satellites in Low Earth Orbit. This effect is principally to remove energy from the orbit and decrease the semi-major axis. Assuming that the orbital radius is never allowed to change very much, one can assume a constant density, and thus a constant specific drag force, given by:

$$f_{drag} = \frac{F_{drag}}{m} = \frac{1}{2}\rho_\infty v^2 \frac{C_D A}{m} \tag{3.8}$$

where ρ_∞ is the local atmospheric density, v is the orbital velocity, C_D the coefficient of drag, A the projected area, and m the mass of the spacecraft.

This force removes energy from the orbit in a manner quite similar to the way a low-thrust orbit transfer adds (or subtracts) energy to the orbit. Namely:

$$\frac{dE}{dt} = \frac{d}{dt}\left(-\frac{\mu}{2r}\right) = -f_{drag}v \tag{3.9}$$

Then, in time t, the changes in energy and in radius are given by:

$$\Delta E = f_{drag}vt \tag{3.10}$$

$$\frac{r_o}{r} = 1 + 2\frac{f_{drag}t}{v} \tag{3.11}$$

In the low-thrust case, the drag can either be cancelled by thrusting continuously with a magnitude equal to F_{drag}, or can be performed on a duty cycle basis, where the thrust is increased appropriately. For example, if thrusting is to occur ten percent of the time, the thrust will have to be ten times F_{drag}.[1] In the impulsive case, the orbit will have to be re-boosted periodically with a Hohmann transfer. The frequency of this re-boost is determined based on the allowable variation in orbital height. However, because the changes in radius are always very small, the ΔV required over the mission does not depend significantly on the thrusting scheme used.

Non-Keplarian Orbits

Non-Keplarian orbits are orbits that require significant external forces in addition to the gravity of a central body to maintain their shape. For example, an orbit that is parallel to but not in the equatorial plane of the earth is a non-Keplarian orbit. These kinds of orbit would be required if a cluster of satellites wanted to maintain its relative spacing, as in Figure 3.3. In that figure, Satellite a is in the Keplarian orbit, and the other three satellites are synchronized to it. Satellite b must thrust downwards continuously to maintain its orbit

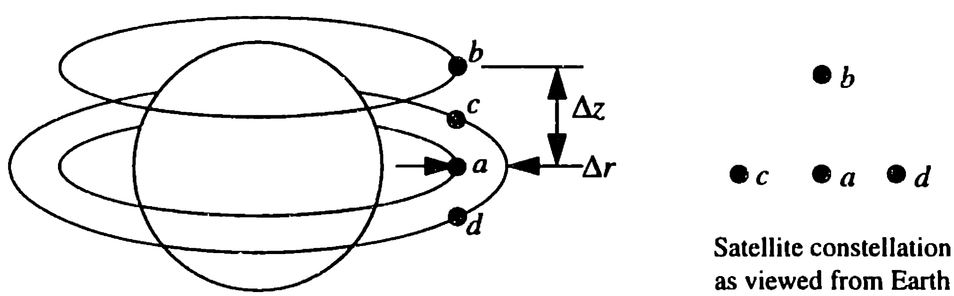


Figure 3.3: Non-Keplarian Orbits

1. Of course, it would be important to perform the burns in different parts of the orbit each time so that the net effect maintains the circular orbit and avoids raising its apogee significantly.

above the Earth's center of mass. Satellites c and d are in circular orbits, but in order to maintain a period that is equal to that of Satellite a, they must continuously thrust radially outward.

For small Δz and Δr, the accelerations required can be readily obtained from the Hill Equations, which are the linearized relative equations of motion of a satellite in the reference frame of another satellite in a circular orbit. From Chobotov[1, pg.177], they are given as:

$$\ddot{x} = f_x + 2\omega\dot{y} \tag{3.12}$$

$$\ddot{y} = f_y - 2\omega\dot{x} + 3\omega^2 y \tag{3.13}$$

$$\ddot{z} = f_z - \omega^2 z \tag{3.14}$$

where y is radially outwards, z is perpendicular to orbit plane, and x completes the right-handed coordinate system. The applied acceleration is made up of the three components of f, and ω is the angular velocity of the reference frame or satellite. In this case, since the satellites maintain position relative to one another, there are no changes in z or y with time, and using Δz for z and Δr for y as in Figure 3.3, we have the required accelerations to maintain the non-Keplarian orbits:

$$f_z = \omega^2 \Delta z \tag{3.15}$$

$$f_y = 3\omega^2 \Delta r \approx \frac{3\omega^2}{2r}(\Delta x)^2 \tag{3.16}$$

where Δx is the separation between satellites a and d, for example, and is related to Δr through r.

Since the required accelerations are constant, the ΔV required to maintain the non-keplarian orbits is simply the product of the required accelerations given above and the lifetime of the spacecraft.

3.2.4 Attitude Control

For spacecraft with very accurate pointing requirements, attitude control is primarily generating torques to cancel out any disturbance torques that the spacecraft experiences. Because disturbance torques are difficult to predict exactly, it is not usually possible to continuously torque the satellite at a low level to exactly cancel out the disturbance. Instead, the disturbance torques lead to an error in the angular orientation of the satellite. This error is sensed by a control system, and torques are commanded to keep the error within an allowable range, often termed a deadband. Though in general estimating the forces and total ΔV required for controlling attitude using thrusters is difficult and heavily dependent on the control methodology employed, as a first approximation, one can assume that the satellite is undergoing a constant disturbance torque, and that pointing error is allowed to accumulate until it reaches one side of the deadband. At this point, a correcting torque is applied to reverse the satellite's angular velocity, sending its pointing vector in the opposite direction. While the satellite crosses the deadband in this opposide direction, the disturbing torque will act on it to reduce its angular velocity until it reaches zero precisely at the opposite side of the deadband from where the torque was applied. At this point, the pointing vector will begin to drift back towards the original side of the deadband. Upon reaching this side again, the process will be repeated. This admittedly idealized process is illustrated for the translational equivalent in Appendix A, Figure A.1.

Appendix A contains a detailed analysis of the situation, the principle results of which are presented here. Equation A.32 gives the required thrust as:

$$F = \frac{T_{net}}{D\xi} \tag{3.17}$$

where T_{net} is the net disturbance torque experienced by the satellite, D is the distance between the pair of thrusters providing the couple (usually the diameter of the satellite), and ξ is the fraction of time that the thrusters are operating. From Equation A.34, the impulse that must be produced by each thruster during each cycle is given as:

$$I_{cycle} = \sqrt{\frac{8\Delta\theta_{max}T_{net}I_x}{D^2(1-\xi)}} \tag{3.18}$$

where $\Delta\theta_{max}$ is the maximum allowable pointing error and I_x the moment of inertia of the spacecraft about the appropriate axis. The total impulse that must be produced over the life of the satellite is independent of both duty cycle and maximum angular error and Equation A.35 gives it simply as:

$$I_{life} = 2\frac{T_{net}}{D}L \tag{3.19}$$

where L is the total lifetime of the spacecraft, and the factor of two stems from the fact that two thrusters are required to provide the restoring torque.

Application to Representative Missions

The rest of this chapter will be devoted to a discussion of how the specific technical requirements for each of the representative missions were developed, based on the methods and formulas developed in the previous sections.

3.3 Separated Spacecraft Interferometry

3.3.1 Mission Parameters

As was mentioned previously, this mission is made up of two collector spacecraft and one combiner spacecraft. The dry mass of each collector spacecraft is approximately 125 kg, and the dry mass of the combiner is 200 kg. The spacecraft are in Earth's orbit around the sun, but approximately 0.1 AU either ahead or behind Earth. There is a 0.05 degree attitude control requirement, and a 1 cm relative position control requirement. The combiner has a specific power of approximately 1.75 W/kg, and the collectors' specific power is approximately 1 W/kg. The current lifetime is expected to be 6 months. [2]

3.3.2 Derived Propulsion Technical Requirements

Orbit Transfer

Though orbit transfer is not a requirement of this mission, it is worth investigating the feasibility of having the spacecraft place themselves into the solar orbit. This requires a ΔV approximately equal to that needed for escape, which can be found from the above formulas for orbit transfer when the final radius is set to infinity, giving a ρ of zero. For an initial LEO parking orbit of 400 km, the impulsive ΔV required would be 3180 m/s, and for a low thrust transfer, the required ΔV would be 7670 m/s. Though using a high specific impulse low thrust engine would certainly reduce the propellant weight required to get the three spacecraft into their final solar orbit, the time required for such a transfer would easily exceed the desired operational lifetime of the satellite. This makes little sense, so for this reason and because there are launch vehicles that are currently capable of providing sufficient impulse to place the three spacecraft into the Earth escape orbit directly, in all likelihood the concept of providing this propulsive capability on the spacecraft themselves will be abandoned.

Station-Keeping and Attitude Control

Station-keeping and attitude control will be the primary propulsion requirements of this mission. Technically, the spacecraft are in non-keplarian orbits, as they are flying in formation around the sun, at slightly different radii. Using Equation 3.16, and assuming a 10 km baseline, the acceleration that the satellites must overcome is a scant 6 x 10^{-10} m/s^2. Though this is certainly very small, it cannot be ignored, because the relative station-keeping requirements are so tight. A 1 cm deviation will build up after little more than an hour if the satellites are allowed to drift. Thus they must be pulsed at least once per hour to ensure they maintain the correct relative separation between themselves. The other disturbance force acting on these satellites is the solar radiation pressure, which is in fact a few order of magnitudes larger than the non-keplarian accelerations. However, the force acting on each of the spacecraft is the same, assuming they all have the same projected area. Because the collector is more massive, it will experience a smaller acceleration than the two combiners. Since each spacecraft must experience identical accelerations to maintain their relative positions, the collector must counteract this difference in acceleration. This difference is given by:

$$\Delta a = a_{collector}\left(1 - \frac{m_{collector}}{m_{combiner}}\right) \tag{3.20}$$

where $a_{collector}$ is the acceleration due to solar pressure experienced by the collector and m the appropriate spacecraft mass. The mass ratio is approximately 1.6, so the net acceleration that must be corrected is 3.12×10^{-8} m/s^2. This acceleration will require a correcting pulse every 13 minutes to ensure that the 1 cm relative position maintenance requirement is met.

These pulse rates and accelerations can be used to determine the minimum impulse bit requirements for each spacecraft. For the collectors, the impulse of each pulse must be approximately 0.9 mNs, and for the collector, each impulse must be 10 mNs in magnitude.

However, the actual analysis of attitude and position control involves much more detail, and depends on the frequency and types of maneuvers that the constellation as a whole must be able to produce. This analysis has been done by JPL, and their results will be utilized here as the overall ΔV requirement for the mission. For the six month mission, the combiner requires a ΔV of 50 m/s and the collectors each require a ΔV of 78 m/s. This is based simply on the quantity of propellant available for the mission and the given specific impulse of the baselined cold gas nitrogen thrusters of 65 sec. [2]

3.3.3 Summary

Table 3.1: Requirements Summary for SSI Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Orbit Transfer	Earth Escape	3180 m/s	7670 m/s		
Station-keep & Att. Cntrl	1 cm 0.05 deg				
Collector Combiner		78 m/s 50 m/s	78 m/s 50 m/s	75 nN 6.2 μN	0.9 mNs 10 mNs

3.4 Future Global Positioning System

3.4.1 Mission Parameters[1]

The idealized future GPS satellite that will be considered here has an operational mass of 100 kg, which is about one tenth of the mass of current NAVSTAR GPS satellites. It will be assumed that its attitude control accuracy requirement will be 0.5 deg, the same as on the current generation of satellites. The total power requirement will be estimated as 350 watts, about half of the current requirement, for an overall specific power of 3.5 W/kg. This compares with a current specific power of approximately 0.5 W/kg. The same 12 hour orbit of 20184 km altitude that the current satellites are in will be assumed as well. The current satellites are designed for a 7.5 year lifetime, but the satellites considered here will have a nominal lifetime of 5 years to ease the individual expense and allow for more frequent upgrading of the overall system as advances in technology allow.

1. Most of the information on the current GPS satellites used throught this section is from [3].

3.4.2 Derived Propulsion Technical Requirements

Orbit Transfer

The current GPS satellites have a solid propellant kick stage that remains a part of the satellite throughout its life. This capability will be investigated for this future satellite as well. If an impulsive thrusting scheme is used, the ΔV is given by Equation 3.1. In this case, $1/\rho$ is 3.977, so the total ΔV required for the Hohmann transfer is 3460 m/s. Using Equation 3.3, the total ΔV required for a low thrust transfer is 3850 m/s. In the low thrust case, the initial acceleration sets the transfer time, but there is not a definite requirement on this transfer time, so other requirements will be allowed to set the thrust requirement in the case of low thrust propulsion.

If the determination is made that the satellites should be delivered to their final orbits by a non-integrated last stage or transfer vehicle, then the capability to fix any insertion errors will be important, as the GPS orbits must be extremely precise. To fix an insertion error of 100km, the required ΔV of approximately 80 m/s is essentially the same for either an impulsive or a low thrust maneuver.

Phasing

The capability of rephasing any given satellite in its orbit will be important to ensure that the spacing of satellites remains as even as possible, especially in the event of a single satellite failure. For this study, the phasing requirement will be given as one 180 deg rephasing during the satellites life, to occur in less than two weeks. Though it is unlikely that a given satellite would ever have to complete a rephasing of a full 180 deg, this capability will allow for a larger number of smaller rephasings, and should be more than sufficient to account for any in-orbit station-keeping that is required to maintain even separations. A 10 day maneuver time at the 12 hour orbit gives $\tau = 20$, so for the impulsive maneuver, Equations 3.4 and 3.5 give a ΔV of approximately 65 m/s for the rephasing. For the low thrust case, Equation 3.6 yields a required acceleration of approximately 15 μg's, for a required thrust of 15 mN. Equation 3.7 gives a total required ΔV of 128 m/s for the maneuver, almost twice the impulsive case.

Station-Keeping

As was mentioned above, most of the station keeping requirements will be covered by the ΔV allotted for rephasing. It will be important to pick the location of the ascending nodes in such a way as to minimize the perturbation effects of the sun and moon[1].

Attitude Control

To determine the attitude control requirements, it is important to understand both the mass properties of the satellite and the torques acting on it. To model the mass properties, a generic satellite is assumed, similar to the one discussed briefly in Section 2.4. A cube is assumed for the satellite body, containing most of the mass, with a solar array on either side. Assuming a satellite density of 250 kg/m^3, the center cube will have a volume of approximately 0.4 m^3, or sides of 0.75 m. To provide the 350 W required at end of life (EOL), a beginning of life (BOL) power of approximately 400 W is required. [3] Based on an efficiency of 14%, 190 W/m^2 is provided, meaning 2.1 m^2 of array area is required. This translates into two arrays 0.75 m wide by 1.4 m long. This is convenient, since they could be folded in half and still fit on a side of the cube, providing a compact launch arrangement. With a array specific power of 65 W/kg, each array will have a mass of 3 kg. Using this information, the three principal moments of inertia can be calculated. They are approximately 9.1 kg m^2 along the array axis, and 23.5 kg m^2 along either axis perpendicular to the axis of the arrays. Using the simplified disturbance torque equations presented in [4], the sum of torques acting on the vehicle at any given time is approximately 2×10^{-6} N m. In order to prevent this net torque from causing a pointing error greater than the 0.5

1. Chobotov (Ref [1], Section 11.5) provides more information about these "frozen" orbits.

deg allowed, it must be counteracted at least every 9.5 minutes, with an impulse of approximately 1.5 mN s from each thruster providing the couple that produces the counter acting torque. This corresponds to a total impulse requirement of 840 N s over the 5 year life of the GPS satellite. This corresponds to a surprisingly small effective ΔV requirement of 8.4 m/s over the life of the satellite. To ensure that all eventualities are covered, and to allow for more complicated attitude maneuvers, this will be increased by a factor of ten, giving a ΔV requirement for attitude control of 84 m/s over the life of the satellite.

3.4.3 Summary

Table 3.2: Requirements Summary for Future GPS Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Orbit Transfer[a]	LEO to HEO	3460 m/s	3850 m/s		
Orbit Transfer	fix 100m err.	80 m/s	80 m/s		
Phasing	180deg,10 day	65 m/s	128 m/s	15 mN	
Attitude Ctrl	0.5 deg	84 m/s	84 m/s	20 nN	1.5 mNs
Total		229 m/s	292 m/s	15 mN	1.5 mNs

a. Optional Requirement

3.5 Earth Observing Cluster

3.5.1 Mission Parameters

The Earth Observation Cluster (EOC) takes the concept of interferometry and points it toward Earth. By using a cluster of satellites to make up an effective aperture, higher resolution observation is feasible. One of the more promising applications of this technique is providing higher resolution infrared detectors. This mission will explore using a cluster of satellites to provide a 1 m resolution in the IR at GEO. This resolution requires an effective aperture diameter of 400 m [5]. The mass of each satellite will be assumed to be 80 kg. A relative position maintenance accuracy of 1 cm is assumed, based on the requirements of the SSI mission above. The lifetime of such systems are estimated at 5 years.

3.5.2 Derived Propulsive Technical Requirements

Orbit Transfer

The orbit transfer function is not really applicable to this mission, as the satellites will be placed into GEO by a launch vehicle, and any final orbital corrections can be absorbed in to the initial set-up of the non-keplarian orbits.

Phasing

Though re-phasing the orbit is not a vital requirement, it would be valuable to have this capability, to allow the cluster to be moved from its normal position over the equator to some other position to allow for better observation of this area. A requirement similar to the GPS and other phasing requirements will be adopted, although the time constraint will be relaxed somewhat. The requirement of one 180 deg maneuver to occur in no more that three weeks leads to an impulsive ΔV of 50 m/s for the rephasing, or a ΔV of 97 m/s at an acceleration of 5.5 μg's for the low thrust maneuver. The 5.5 μg's of acceleration translates into a thrust of 4.4 mN.

Station-Keeping

In order to maintain the relative separation between satellites that provides the sparse aperture, most of the satellites must be in non-keplarian orbits. Those satellites above or below the equatorial plane will require the largest accelerations to maintain their positions. The required aperture diameter is 400 m, which corresponds to a Δz of 200 m. Equation 3.15 gives the required continuous acceleration as 1.06×10^{-6} m/s^2. This corresponds to a continuous thrust of 85 μN and a total ΔV of 166 m/s over the five year mission life. If the thrust is not applied continuously, the satellite will begin to drift away from its correct position, requiring a thruster impulse every 4.6 minutes to keep its position error less than 1 cm in magnitude. This impulse must be approximately 23 mN s. To ensure that any other necessary station-keeping maneuvers can be accomplished, an additional ΔV margin of approximately 35 m/s will be added to the non-keplarian requirement, making the required ΔV 200 m/s.

In order for the cluster to remain in the same position over the equator, it will need to perform North-South station-keeping to counteract the sun and moon disturbances that tend to change the inclination of GEO orbits by approximately 0.9 deg per year. Following the analysis presented in [14], the impulsive yearly ΔV requirement is approximately 50 m/s. The ΔV requirement for correcting this inclination change using low thrust depends on the duration of a given burn, and the required duration depends on the frequency of thrusting and the thrust applied. If a thrust of 4 mN is assumed to be compatible with the phasing requirement, and an average burn duration of about 25 minutes is asummed (10 degrees of the orbit), one burn will be required each orbit (or each day), with its location alternating each orbit between the ascending and descending nodes. In this case, beacuse the thrusting is almost all happening near the line of nodes, the required ΔV is essentially the same as the impulsive case, 50 m/s. Over five years, the required ΔV for North-South sation-keeping is then 250 m/s.

Attitude Control

The ΔV required for attitude control to maintain the individual satellites of the cluster pointed in the correct direction will be fairly small compared to the ΔV required for station-keeping, and can be considered included in the extra ΔV of 35 m/s added to the station-keeping ΔV budget. However, to to be doubly sure that attitude control requirements can be met and to add some additional margin to the station-keeping budget, an additional 20 m/s of ΔV will be added to the requirement.

3.5.3 Summary

Table 3.3: Requirements Summary for IR EOC Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Phasing	180deg,21 day	50 m/s	97 m/s	4.4 mN	
Station-keep	1 cm in non-kep Δz=200m	200 m/s	200 m/s	85 μN	23 mNs
Station-keep	NSSK	250 m/s	250 m/s	4mN	
Att. Cntrl	margin	20 m/s	20 m/s		
Total		520 m/s	567 m/s	4.4 mN	23 mNs

3.6 MiniMars Mission

3.6.1 Mission Parameters

The MiniMars mission is a conceptual mission devised for this study. The final mass inserted into Mars orbit will be 70 kg, at a Mars orbital height of 500 km. Though the primary propulsion needs will be a LEO to LMO (Low Mars Orbit) transfer, the feasibility of providing other propulsive abilities once in Mars orbit will be investigated as well.

3.6.2 Derived Propulsive Technical Requirements

Orbit Transfer

Assuming that the spacecraft is placed in a LEO of 300 km by its launch vehicle, the first step is to escape from Earth's gravity well. In the impulsive case, this escape maneuver is combined with the first ΔV required to get into the heliocentric transfer orbit. The required impulsive ΔV is 3590 m/s and the required low thrust ΔV is 7726 m/s. Once earth's gravitational pull has been escaped, there is no further maneuver required in the impulsive case as it was on a hyperbolic escape trajectory and is already in the required Hohmann transfer orbit to Mars. In the low-thrust case, the spacecraft must now begin its spiral around the sun, and the required ΔV for this segment is 5652 m/s. Upon arriving at Mars, the spacecraft must be captured into its final LMO orbit. In the impulsive case, this can be done directly with a single ΔV of 2090 m/s at the periapsis of its hyperbolic trajectory as seen in Mars' coordinate frame. The low thrust version must spiral down to its final orbit, requiring a ΔV of 3316 m/s. Thus the total required ΔV is 5660 m/s in the impulsive case and 16700 m/s in the low thrust case. If the launch vehicle could place the spacecraft on an parabolic escape trajectory, then the required ΔV's would be 5013 m/s and 8968 m/s respectively.

Phasing, Station-Keeping, and Attitude Control

Because these other types of propulsive maneuvers are not the driving requirement in this mission, they will be looked at later in the study, once the main propulsion system is more fully developed. The reasoning for this is that the requirements for these types of maneuvers would be quite similar to requirements for missions in LEO. The difference here will be to see if the propulsion technologies that make sense for the primary propulsion requirement could be used for additional propulsive maneuvers of these types once in Mars orbit.

3.6.3 Summary

Table 3.4: Requirements Summary for MiniMars Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Orbit transfer	LEO to LMO	5660 m/s	16700 m/s		
Orbit transfer[a]	EET[b] to LMO	5013 m/s	8968 m/s		

a. Alternate requirement if original unfeasible.
b. Earth Escape Trajectory

3.7 Next Generation Low Earth Orbit Communication System

3.7.1 Mission Parameters

The next generation LEO communication system is a mission based on the current Orbcomm system now being developed and operated by Orbital Communications Corporation. The mission is assumed to be a constellation of satellites orbiting at an altitude of 800 km. The mass of each satellite is 50 kg, which is somewhat larger than the current

Orbcomm satellites. However, it is envisioned that this future LEO communications satellite will have significantly more capabilities in terms of data throughput and number of connections than current satellites. The assumption made is that a slightly larger satellite will be used to provide significantly more power and function. The total power will be 225 W, for a specific power of 4.5 W/kg, as compared to Orbcomm's current 3.8 W/kg [6]. To help reduce required power for the transmitters, a more accurate attitude control system is desired. A pointing accuracy of one degree will be assumed as a requirement for this mission. The lifetime of the satellites will be six years.

3.7.2 Derived Propulsive Technical Requirements

Orbit Transfer

The only orbit transfer requirement for this mission is to fix any possible orbit insertion errors of the launch vehicle. Assuming a nominal orbit of 800 km, and a possible error of 50 km, the necessary ΔV to correct this is approximately 26 m/s in both the impulsive and low thrust regimes.

Phasing

To ensure even spacing of the constellation, a rephasing requirement will be imposed on this mission as it was on the future GPS mission discussed above. Again a requirement of one 180 deg phase shift during the lifetime of the satellite will be made. One week will be the maximum allowable time for the transfer to occur. For an impulsive maneuvers, the total required ΔV for the rephasing is 25 m/s. For the low-thrust maneuver, an acceleration of 8.4 μg's is required. This translates into a total ΔV of 50 m/s and a thrust requirement of 5 mN.

Station-Keeping

Most of the required station keeping will be in the form of small rephasings, and will come out of the rephasing ΔV budget developed above. Though the orbit is high enough to not require a very active drag makeup, some drag makeup capability is desired. In active solar conditions, the required ΔV per year to make up drag losses is approximately 7.9 m/s per year. However, during average solar conditions, drag makeup only requires 0.5 m/s per year. If the six year lifetime is assumed to be made up of two active years, two average years, and two years somewhere in between, the total drag makeup ΔV requirement will be 30 m/s.

Attitude Control

The attitude control requirement is to maintain the satellite's antenna to within one degree of earth pointing. In order to evaluate the propulsion requirements for this, the mass properties of the satellite must be estimated. A shape almost identical to the satellites in the current Orbcomm constellation is assumed, with a deployable Earth-pointing antenna and two ear-like solar arrays. Based on this configuration, estimates of the three principal moments of inertia and the locations of the center of gravity and center of pressure were made. Based on these mass and area properties, the disturbance torques acting on the satellite were estimated. The largest disturbance torque is the gravity gradient torque, but this only occurs when the antenna is not nadir-pointing. It causes an oscillation around the nadir vector with a period of approximately one hour. Since the objective is to keep the satellite nadir pointing, this torque will generally be small. However, torques from solar pressure and from aerodynamic drag will be non-zero with the satellite in its desired attitude. They will tend to cause a change in attitude, eventually leading to these unwanted gravity gradient oscillations. For these reasons, it is the solar pressure and aerodynamic torques that the attitude control system must counteract. The worst-case net torque experienced is approximately 17.5 μNm, which will cause the satellite to have a pointing error of one degree approximately every 10 minutes. The impulse required of each of two thrusters creating a couple to counteract this torque each cycle is approximately 11 mNs, and the overall impulse required over the six year mission life is 6025 Ns. This leads to an effective ΔV requirement for attitude control of approximately 120 m/s.

3.7.3 Summary

Table 3.5: Requirements Summary for Future LEO Communication Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Orbit Transfer	fix 50km err.	26 m/s	26 m/s		
Phasing	180deg,7 day	25 m/s	50 m/s	5 mN	
Station-keep	drag makeup	30 m/s	30 m/s	12 µN	
Attitude Ctrl	1 deg	120 m/s	120 m/s	16 µN	11 mNs
Total		201 m/s	226 m/s	5 mN	11 mNs

3.8 Low Altitude Earth Observation

3.8.1 Overall Mission Parameters

Many feel that Earth observation and remote sensing will be one of the next large-scale business opportunities for satellite makers. The goal of providing one meter resolution images of the earth in the visible spectrum is often mentioned, and this mission is based on a satellite in development at Draper Laboratories that will provide this function. Its mass is 40 kg, and it provides this high resolution with a satellite that is so small by flying close to the Earth. Its nominal orbit is 400 km. At this low altitude, the effects of atmospheric drag are very significant, and in order to have a satellite lifetime of three years, significant drag makeup is required. In addition, a precision injection into the proper orbit is necessary as a few kilometers can make a very large difference in drag makeup requirements. Because the orbit is sun-synchronous, the satellite will always appear over a given location at the same local time. The capability for rephasing the orbit to allow viewing points at different local times is also desired. Draper is developing a three-axis control system for this mission, so propulsion is not required for attitude control. [7]

3.8.2 Derived Propulsive Technical Requirements

Orbit Transfer

The only orbit transfer requirement for this mission is to fix any possible orbit insertion errors of the launch vehicle. Assuming a nominal orbit of 400 km, and a possible error of 50 km, the necessary ΔV to fix this is approximately 30 m/s in both the impulsive and low thrust regimes.

Phasing

Though not a definite requirement, it would be desirable to rephase the orbit periodically if this proves feasible. Ideally, a performance similar to that of the LEO communication constellation satellite above would be desirable. Two 180 deg phasings in two weeks each will be the nominal requirement, though changing the time constraint will be permitted if that is required to make the maneuver possible. For an impulsive rephasing, each 180 degree maneuver will require a ΔV of 12 m/s. For the low thrust version, the ΔV requirement will be 23 m/s, and the acceleration requirement will be 20 μg's, or a thrust level of 0.8 mN.

Station-Keeping

The propulsion driver of this mission will the drag-makeup requirement. Because it is operating at a low orbit to improve its resolution of Earth, there are significant aerodynamic drag forces. At an orbit of 400 km, a ΔV of approximately 400 m/s will be required to cancel out the drag experienced by the satellite over its three year life time. This assumes one year of maximum solar conditions, one year of average solar conditions, and one year where the solar conditions are somewhere in between. The maximum drag force experienced is approximately 0.25 mN.

3.8.3 Summary

Table 3.6: Requirements Summary for Low Altitude EOS Mission

Type	Requirement	Impulsive ΔV	Low thrust ΔV	time-ave thrust	Impulse
Orbit Transfer	fix 50km err.	30 m/s	30 m/s		
Phasing	180deg,14 day (2x)	24 m/s	46 m/s	0.8 mN	
Station-keep	drag makeup	400 m/s	400 m/s	0.25 mN	
Total		454 m/s	476 m/s	0.8 mN	

3.9 Summary of Requirements

Table 3.7: Summary and Comparison of Mission Requirements

Mission	ΔV and Thrust Requirements			
	Insertion imp / low-T ΔV [m/s]	**Phasing** imp / low-T ΔV -- / low-T F	**Station-keep**	**Attitude Control**
New Millenium Interferometer (JPL)	Earth escape (?) 3180 / 7670 (other setup in station-keeping)	--	1 cm $\Delta V = 50, 78$ m/s[a] $I = 0.9, 10$ mNs	0.05 deg included in station-keeping
Future GPS	fix 100m error 80 / 80 LEO to HEO 3460 / 3850	180 deg, 10 days 65m/s / 128m/s -- / 15mN	maintain relative angular separation covered by re-phasing reqmt	0.5 deg $\Delta V = 84$ m/s $I = 1.5$ mNs
Earth Observation Cluster	included in station-keeping	180 deg, 21 day 50m/s / 97m/s -- / 4.4mN	non-keplarian orbit, Δz=200m ΔV=200m/s I=23mNs F=85μN NSSK: 250m/s	? included in station-keeping, but add 20m/s to be sure.
MiniMars	LEO to LMO 5660 / 16700 EET[b] to LMO 5013 / 8968	TBD	TBD	TBD
Future LEO Comm System	fix 50km error 26 / 26	180 deg, 7 day 25m/s / 50m/s -- / 5mN	most in rephasing requirement drag makeup ΔV=48 m/s	1 deg $\Delta V = 120$ m/s $I = 11$ mNs
Low Alt Earth Observation (Draper)	fix 50km error 30 / 30	2x 180deg, 14 day 24 m/s / 46m/s -- / 0.8mN	drag makeup ΔV=400 m/s F=0.25mN	they have it covered

a. ΔV of 50 m/s for combiner, 78 m/s for collectors; I (min. Impulse per correction cycle) of 10 mNs for combiner, 0.9 mNs for collectors.
b. Earth Escape Trajectory

Chapter 4

Scaling of Propulsion Technologies

4.1 Introduction

In order to translate the specific mission requirements into the required masses of the propulsion system, the various propulsion systems to be considered need to be understood. This is the role of this chapter. The first segment of the chapter discusses chemical propulsion technologies, which is followed by a discussion of electrical propulsion technologies. The last segment discusses the additional components that are required in propulsion systems, such as tanks, valves, and power supply and conditioning systems. The segments dealing with chemical and electrical propulsion begin with a general discussion of the particular type of propulsion, and focus on what happens in a general sense as these technologies are applied to the small spacecraft being considered. In each case this general discussion is followed by a more detailed analysis of each specific technology.

4.2 Chemical Propulsion Scaling Arguments

For a chemical rocket, the thrust can be expressed as:

$$F = P_c A_t c_F = \dot{m} c \tag{4.1}$$

where P_c is the chamber pressure; A_t the throat area; c_F the coefficient of thrust, a function of the propellant and geometry; $\dot{m}$ the mass flow rate; and c the exhaust velocity, also a function of the propellant and geometry. Thus for a given propellant and geometry,

$$F \propto \dot{m} \propto P_c \cdot L^2 \tag{4.2}$$

where L is some characteristic length.

However, when performance is considered, one of the usual parameters discussed is the thrust to weight ratio of the propulsion system. Looking at the entire system, this can be divided into two parts: the thrust to weight ratio of the engine itself, and the thrust to weight ratio of the tanks that hold the propellant, which often dominate the overall system mass. For a pressure vessel, the wall thickness can be shown to be proportional to a characteristic linear dimension of the vessel and to the pressure it contains:

$$t_w \propto P \cdot l \tag{4.3}$$

where P is the pressure contained in the vessel and l a characteristic linear dimension of the vessel. The mass of the pressure vessel is proportional to its surface area multiplied by its wall thickness, and its surface area varies with the square of its characteristic linear dimension. Thus the mass of the vessel varies with the product of the pressure it contains and the cube of its linear dimension:

$$m_{vessel} \propto P \cdot l^3 \propto P \cdot V_{vessel} \tag{4.4}$$

In the case of a rocket motor which can be treated as a pressure vessel as a first approximation, the characteristic dimension is L, and the pressure is Pc. Thus, if the motor scales geometrically, the thrust to weight ratio of the motor itself can be shown to scale as:

$$\frac{F}{m_{eng}} \propto \frac{P_c \cdot L^2}{P_c \cdot L^3} \propto \frac{1}{L} \tag{4.5}$$

The thrust to weight ratio of the engine increases as the engine gets smaller. This assumes that there are no limiting factors that occur as the engine gets smaller geometrically, which is not necessarily the case, as will be discussed shortly.

In the case of a propellant tank, a more traditional pressure vessel, the volume is proportional to the mass it contains, which is in turn proportional to the mass flow rate for a given length mission. This means that for a given tank geometry, the tank mass per unit propellant mass is invariant to size, and that the thrust to tank mass ratio scales as:

$$\frac{F}{m_{\text{tank}}} \propto \frac{F}{P_t \cdot \dot{m}} \propto \frac{P_c \cdot L^2}{P_t(P_c \cdot L^2)} \propto \frac{1}{P_t} \tag{4.6}$$

where P_t is the pressure in the tank. Most space propulsion systems are pressure fed, and thus the tank pressure must be equal to or greater than the chamber pressure. Thus, the tank pressure would scale with the chamber pressure, and F/m_{tank} would scale as $1/P_c$.

The above argument would seem to imply that reducing the thrust by a factor of 100 while maintaining a constant chamber pressure would increase the thrust to weight ratio of the engine by a factor of 10, and maintain the same thrust to tank weight ratio. This is not necessarily the case. If the motor is reduced in size geometrically, its characteristic chamber length, L^*, defined as the ratio of the chamber's volume to the throat area, will decrease as well. In the case of bipropellant engines where mixing of propellants must occur prior to combustion, it is generally believed that for a given propellant combination, chamber shape, and injector type, L^* must remain invariant to scale in order for combustion efficiency to be maintained. Allowing it to decrease may lead to incomplete combustion and additional thrust losses. If L^* is held invariant, then the thrust to mass ratio of the engine would remain invariant to scale as well. However, as was noted earlier, chamber mass is generally a small fraction of the total system mass, so losing some of the scaling advantage based on Equation 4.5 should not be a driving effect. The other performance penalty that may occur as engines are made smaller is from frictional losses. These losses are a function of the Reynolds number (*Re*) of the flow, and increase as *Re* decreases. *Re* can be shown to scale as:

$$Re \propto \frac{\dot{m}}{L} \propto P_c \cdot L \tag{4.7}$$

In order to keep the fractional losses from changing as scale is changed, *Re* must be held constant, and therefore the chamber pressure must scale inversely with length. If this is true, then in the case of pressure fed space engines,

$$\frac{F}{m_{\text{tank}}} \propto \frac{1}{P_t} \propto \frac{1}{P_c} \propto L \tag{4.8}$$

As the scale of the thruster is decreased, the thrust to tank weight ratio will go down proportionally. The reason for this can be seen by investigating the denominator in Equation 4.6. With $P_t \propto P_c$, the mass of the tank is proportional to the square of the Reynolds number. Therefore, if Re is held constant to maintain performance, the mass of the tanks will remain constant as well. Because the tanks make up the majority of the mass of the propulsion system in many cases, this is clearly not a good consequence, and in reality some losses in thrust are allowed to prevent this eventuality. In fact, the magnitude of the losses is quite small unless Re becomes very small, at which point some increase in chamber pressure could be warranted to preserve most of the performance, though in almost no case would maintaining Re invariant through a significant scaling make much sense.

If a higher performance chemical propulsion system is desired (with performance in this case measured in terms of thrust to system weight ratios), the above scaling arguments point in only a few directions. The most dramatic effect would be to decouple the tank and chamber pressures. Of course this requires the use of a pump, something that has never been done on the scale being considered. More about this idea and why new technologies may now make this feasible is discussed below in Section 4.3.5.

Another strategy would be to remove some of the tank mass by getting rid of one or more tanks altogether. This is one motivation for a hybrid engine, where some of the propellant is already in the thrust chamber and does not require a separate tank. Of course, the chamber must be larger to accommodate this extra propellant, so a trade-off exists.

Thirdly, the weight of the required propellant, and therefore the weight of the tank required to hold it, can be decreased by using higher specific impulse propellants. This is another reason to investigate the feasibility of hybrid and bipropellant options discussed below.

Finally, the engines themselves can be made smaller and used in parallel. There is clearly a trade-off here as well as the added losses from shrinking them may at some point outweigh the gains in thrust to weight ratios. In addition, as was already mentioned, in most cases, the mass of the engines themselves is rather insignificant when compared to the entire propulsion system.

4.3 Specific Chemical Propulsion Technologies

At this point, each of the chemical propulsion options being considered will be discussed. The primary advantages and disadvantages of each system will be identified, and rough models for estimating propulsion system mass will be discussed.

4.3.1 Cold Gas Thrusters

A cold gas thruster is quite possibly the simplest kind of engine in terms of its operating principles. A reservoir of gas is held at high pressure, and small pulses of it are allowed to expand through a converging-diverging nozzle. Its performance can generally be estimated using the traditional adiabatic expansion relations presented in most propulsion texts. Its primary advantages are its simplicity and capability of producing extremely small thrusts and impulse bits, useful for missions where very precise station-keeping or attitude control is required, such as in the NMI SSI mission, for which they are currently baselined.

However, there are significant drawbacks to cold gas systems. The first is their low I_{sp} of around 60 to 70 sec. For any mission with significant ΔV requirements, the propellant weight will quickly begin to dominate. An issue that may be even more problematic is that of valves. Because it is an all gas system, there is a much higher tendency for valves leaking, as stopping a gas is significantly more difficult than stopping a liquid since the molecules are more mobile in a gas.

To model these systems, the mass of the thruster itself will be ignored, as it is simply a chamber and a nozzle. In fact, the 12 thrusters baselined in the NMI SSI mission are less than one percent of the total propulsion system mass. The dominate masses are the tank and valves required for the rest of the system, and mass models for these are presented in Sections 4.6.4 and 4.6.5 below. Table 4.1 presents a summary of cold gas systems.

Table 4.1: Cold Gas Summary

Attribute	Value
Specific Impulse	60 sec
Propellant	GN_2, GHe, etc.
Additional components required:	tank, valves
Advantages:	simplicity, power only required for valves, very low impulse bits feasible
Disadvantages:	low I_{sp}, high tank pressures, valves tend to leak

4.3.2 Hydrazine Thrusters

Hydrazine monopropellant thrusters are perhaps the most common type of thruster in use today. They are relatively simple devices, with the hydrazine first encountering a catalyst bed where it is decomposed, releasing energy that is absorbed by the decomposition products. This hot gas then expands through a nozzle to produce the thrust. Typical specific impulses are near 200 sec, and most thrusters operate in a blow-down mode, where the tank pressure is not regulated and held constant, leading to the chamber pressure and thus thrust decreasing over the life of the mission.

It is unclear whether such systems can scale to much smaller sizes than currently exist without significant losses in performance, as the products currently offered in the smaller thrust ranges (1 to 5 N) are approaching the limits of traditional manufacturing technology. If ways can be found to manufacture even smaller thrusters, the advantages of their use would be the extensive knowledge base currently available, and their relatively high specific impulse when compared to cold gas systems. Micro-Electrical and Mechanical Systems (MEMS) technology (see Section 4.3.5 below) might have some promise in this area, but currently MEMS technology is limited to using silicon as the working material. Unfortunately, hydrazine tends to dissolve pure silicon, so the application of MEMS technology to hydrazine systems will have to wait until the development of silicon carbide MEMS, which is currently being developed.[8]

Assuming that some kind of technology becomes available to allow the construction of micro-hydrazine thrusters, the scaling of the systems should roughly follow the relations presented above. It will be worthwhile to see how these ideally scaled versions of a hydrazine thruster fair when compared to other systems being discussed. Olin Aerospace's

MR-103C/E, a 0.2 lbf (0.9 N) thruster will serve as the basis of the scaling. Since the thrust scales as the square of the characteristic length, the mass will scale as:

$$m_{HYD} = m_{HYDo}\left(\frac{F_{req}}{F_o}\right)^{3/2} \tag{4.9}$$

where m_{HYDo} is the mass of the reference hydrazine thruster, 130 g, F_{req} is the required thrust, and F_o is the thrust of the reference hydrazine thruster, 0.9 N. A valve is required for each thruster, and the system requires a tank for storing the hydrazine. Table 4.2 summarizes the attributes of hydrazine thrusters.

Table 4.2: Hydrazine Thruster Summary

Attribute	Value
Specific Impulse	210 sec
Propellant	hydrazine
Additional components required:	tank, valves
Advantages:	simplicity, power only required for valves, large knowledge base
Disadvantages:	low I_{sp}, difficult to scale smaller, hard to make very small impulse bits.

4.3.3 Hybrid Motors

Hybrid rockets, where the oxidizer is a liquid and the fuel a solid, have most frequently been considered for applications to launch vehicles. However, they have recently been suggested for use in small satellites by a group at the University of Surrey. The proposed system uses hydrogen peroxide (HTP) as the oxidizer, and polyethylene (PE) as the fuel. A hybrid motor utilizing the mentioned propellants was built and tested, and is being considered for use on future small satellites built by the University of Surrey [9]. The HTP is sent through a catalyst bed, where it decomposes and into hot oxygen and steam, which enter the combustion chamber, lined with the solid PE. The heat of the decomposed products is sufficient to cause the initiation of combustion, which continues for as long as there is a supply of the decomposed HTP. Because the flow of HTP can be throttled or stopped with an upstream valve, the hybrid motor can be as well. In addition, because the decomposed gases are hot enough to initiate combustion independently, such a system does not require an ignitor, increasing simplicity. For the application of this technology to this study, the hybrid system will only be considered for impulsive operations, and the total mass will be modeled as the sum of the masses of the solid fuel, oxidizer, oxidizer tank, oxidizer valve, and engine housing. The engine housing mass will be estimated by assuming a center-burning cylindrical propellant grain shape with a height twice its diameter, and with a core diameter one third of the motor diameter. The motor will be assumed to be a cylindrical pressure vessel and its mass estimated in a manner similar to that of a tank.

The I_{sp} of such a system is reported as approximately 280 sec, with an oxidizer to fuel mass ratio of 8:1. The hybrid motor characteristics are summarized in Table 4.3:

Table 4.3: Hybrid Motor Summary

Attribute	Value
Specific Impulse	280 sec
Propellant	HTP/PE
Additional components required:	Tank, valve
Advantages:	simplicity, higher I_{sp}, power only required for valves, no ignition, safe propellants, restartable
Disadvantages:	hard to make very small impulse bits.

4.3.4 Reverse Hybrid Motors

Although the "reverse" hybrid concept is not developed further here, it is presented as an outgrowth of the hybrid idea presented above. There is nothing sacred about having a liquid (or gaseous) oxidizer and a solid fuel in a hybrid motor. In fact, the reverse might be feasible as well. Liquid hydrazine could be used as the fuel, decomposed to a high temperature gas in a traditional and well-understood catalyst bed and allowed to burn with a typical solid propellant oxidizer, such as Ammoniumperchlorate (AP). Preliminary calculations indicate that the specific impulse achievable with such a system could significantly exceed that of the HTP/EP system discussed above. However, some additional discussion of the idea with a member of the solid propulsion community lead to the concern that the regression rate of the AP might be insufficient to support combustion [10]. There was insufficient time to investigate further, so the idea is left for future investigations, and is not included in this study.

4.3.5 Bipropellant Engines

Bipropellant engines provide significantly higher specific impulse than monopropellant engines at the expense of requiring two tanks for the propellant, one for the fuel and one for the oxidizer, as well as two sets of valves, lines and other supporting equipment. If the systems are operated in blowdown or pressure-regulated modes where the tank pressures must exceed the chamber pressures, no advantages in terms of thrust to system weight ratios come from scaling, and in many cases, the increase in specific impulse and thus decrease in total propellant weight will not make up for the mass of the additional tank and other hardware. Thus traditional blowdown or regulated bipropellant systems are not considered here.

However, as was shown above in Section 4.2, if the tank pressure and chamber pressure could be de-coupled, allowing lower tank pressures, there would be significant gains in thrust to system mass ratio, as the thrust to tank mass ratio scales inversely with tank pressure. (Equation 4.6). This is what happens in high-performance launch vehicles where the pressure rise from tank to chamber is performed by turbopumps. Pumps have never been used on the scales of vehicles being considered in this study, as creating pumps with sufficient efficiencies at that small scale was considered infeasible.

This may no longer be the case. There is currently a large on-going effort at the Gas Turbine Laboratory at MIT to produce a micro gas turbine utilizing MEMS technologies [8]. This shirt-button sized device will produce an estimated 50 W of power, yielding a fairly impressive power density. There is no reason that this same technology can not be applied to turbomachinery for micro rocket motors. For sufficiently small thrust devices, the traditional turbine part of the turbopump could be removed and the pump driven electrically. If a higher performance or larger mass flow rate (and thus larger thrust) was desired, then the more traditional turbopump could be used. MEMS technology utilizes the manufacturing techniques used in creating computer chips to build micro-machines and electronics that are integrated on the same "chip."

To further evaluate this concept, some discussion of scaling is warranted. The stress in a rotating turbomachine is generally proportional to the square of the tip speed of the machine's rotor. For a given material, the stress experienced should remain independent of scale, and thus the tip speed should remain constant when scaled. This requirement ensures that the pressure rise across the pump remains constant, since

$$\frac{\Delta P}{\rho} \sim (\omega r)^2 \tag{4.10}$$

where ΔP is the pressure rise across the pump, ρ the density of the fluid, and ωr the tip speed of the rotor (ω is the angular velocity and r the radius of rotor). Since the pressure rise remains constant, the chamber pressure also remains constant with scale. Thus, from

Equation 4.2, the mass flow rate scales as the square of the characteristic length of the engine. The flow area of the pump will scale with the square of the pump's characteristic length. If the radial flow velocity u is assumed to always be a fixed fraction of the tip speed and thus constant, then the mass flow rate through the pump will scale with the square of the characteristic length of the pump. Since the mass flow through either pump must remain a constant fraction of the total mass flow through the engine to ensure an invariant mixture ratio, the pump characteristic length must scale as the engine characteristic length. Since this is true, the mass of the pump (which will scale with the cube of the pump characteristic length) will also scale with the cube of the engine characteristic length. In other words:

$$\frac{F}{m_{pump}} \propto \frac{L^2}{l^3} \propto \frac{L^2}{L^3} \propto \frac{1}{L} \tag{4.11}$$

where m_{pump} is the mass of the pump, L the characteristic dimension of the engine, and l the characteristic dimension of the pump. The thrust to pump weight ratio will increase as the engine is made smaller. This assumes a geometrical scaling, and in reality the transition is from a traditional centrifugal pump to an effectively two dimensional planar pump. It is unlikely that the improvement will be as high as predicted by the above scaling relation, but the trends should certainly be valid.

A very rough conceptual design for a LOX/RP-1 4.5 N bipropellant micro-rocket follows. Because MEMS technology is currently limited to silicon as the only material, a turbine-driven pump will not be attempted, as the high temperature capabilities of silicon carbide would be preferred for use in a turbine. Instead the pumps will be driven electrically. The maximum tip speed for these microturbines is approximately 500 m/s [8]. Conservatively, a tip speed of 200 m/s is assumed, yielding a maximum ideal pressure rise [$\rho(\omega r)^2$] of 450 atm for LOX and 340 atm for RP-1. Choosing a chamber pressure (and thus approximate ΔP) of 200 atm seems quite feasible. If a 4.5 N thrust is desired, assuming an exit velocity of 3000 m/s yields a required mass flow rate of 1.5 g/s. If the two pumps are assumed to be identical (not a great assumption, but it will suffice for this illustration), and of similar dimensions to the rotors being designed for the micro-gas turbine (blade height of 200 μm; rotor radius of 2 mm), the radial flow velocity out of the pump will need to be approximately 0.3 m/s, entirely reasonable when compared to a tip speed of 200 m/s. The power required to operate each pump, assuming a 50% overall efficiency, is approximately 30 W. A total required power of 60 W is actually quite small, considering that the useful thrust power produced is 6.75 kW! If a thrust coefficient of 1.5 is assumed, the throat area of the nozzle required would be 0.15 mm^2, which corresponds to a square 390 μm on a side. This is precisely the scale that MEMS technology deals with. It appears feasible that such a device would work. For an idea of what this kind of device would weigh, a planar schematic is depicted in Figure 4.1, with the large circles representing the pumps and the small circles valves. A solid block of silicon with these dimensions would have a mass of just over 5 g, and thus an thrust to weight ratio of approximately 90. This is not spectacular, as the thrust to weight ratio of the NK-33, a first stage Russian booster engine using the same propellants, is approximately 125 [11], but considering the large amount of wasted volume shown in the figure which could eventually be removed, it is actually quite reasonable. For another comparison, the thrust to weight ratio of a hydrazine thruster with a similar thrust is approximately 1.4. None of this takes into account the additional mass savings from allowing lower pressure in the propellant tanks.

There are some caveats to be made, however. Using such a thruster for any significant period requires storing cryogenic liquid oxygen for that significant period, which is not a simple task, especially when storage times are measured in years. However, there are other

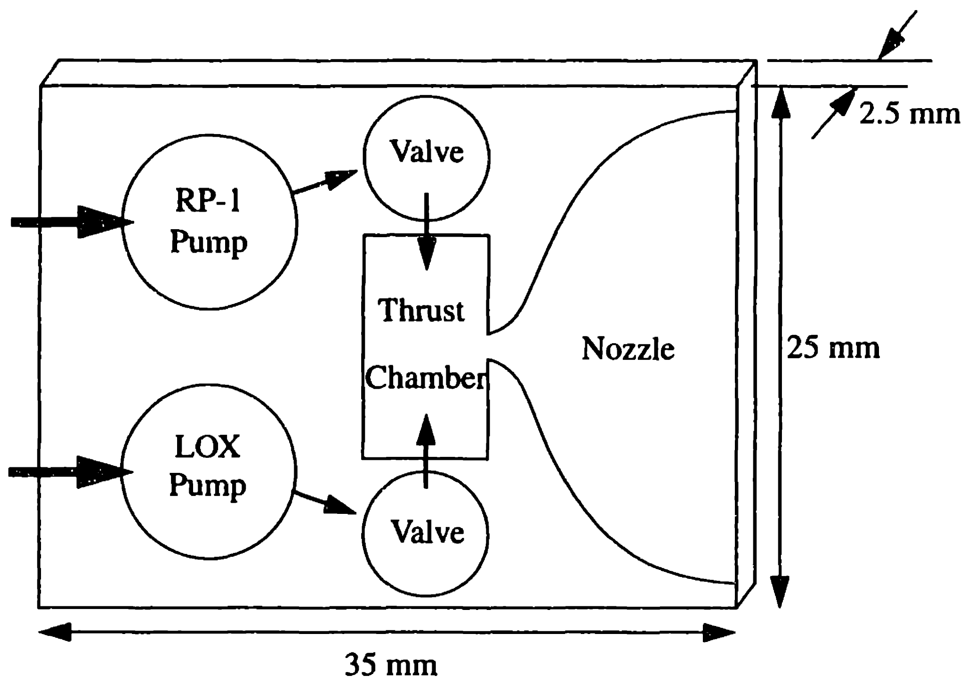


Figure 4.1: Micro-rocket Concept Layout

non-cryogenic propulsion combinations, such as HTP/RP-X2 [11] or N_2O_4/JP-4 [12], which have been proposed as possible solutions to this issue that allow for similar (or even better) I_{sp} but are non-cryogenic, and thus much more easily stored for long periods. At this time, it is unclear what compatability issues these propellants will have with silicon, but addressing this is left for future work. The scaling analysis further assumes that the losses associated with the smaller nozzles will not become overly significant, something that is not yet certain. In addition, there is the issue of contamination. Because the flow areas are so small, even the smallest particle could provide a potential blockage problem.

Nevertheless, this appears to be a very promising technology, especially for those missions where a fairly high ΔV is required in a particular time constraint. The time constraint effectively eliminates the higher I_{sp} electric propulsion devices that would usually provide large gains in fulfilling the high ΔV requirement, and leaves bipropellant options with the highest I_{sp} of feasible options. An excellent example of this is the LEO to GEO transfer or circularization required of many communications satellites. It is quite feasible that a system such as this (or a few of them stacked together) could provide a significant improvement in I_{sp} and thus in propellant weight fraction of the communications satellite while still providing a reasonable transfer time from LEO to GEO. Systems such as this could also prove very useful in quickly accelerating small payloads to escape velocities.

For the purposes of this study, this technology will be evaluated for use in the various impulsive maneuver requirements, with the engine mass remaining constant (although negligible) at 5 g, and the propellant and its tanks providing the balance of the system weight. The model used for tank mass is discussed in Section 4.6.4, and the conceptual bipropellant system is summarized in Table 4.4.

:

Table 4.4: Bipropellant Engine Summary

Attribute	Value
Specific Impulse	300 sec
Propellant	LOX/RP-1
Additional components required:	2 tanks
Advantages:	low tank pressure, high I_{sp}, restartable, high T/W ratio
Disadvantages:	never done before; cyrogenic oxidizer; power required for pumps

4.4 Electric Propulsion General Scaling Arguments

Unfortunately, since electric propulsion devices have very different operating principles, it is nearly impossible to provide general scaling arguments as were presented above for chemical systems. However, it is instructive to look at electric propulsion from the perspective of the entire system. For electric propulsion systems, thrust can be written:

$$F = \dot{m}c = \frac{2P\eta}{c} \tag{4.12}$$

where $\dot{m}$ is the mass flow rate, c the specific impulse, P the required power, and η the thruster efficiency. Assuming a constant specific impulse and efficiency, the thrust will be proportional to the mass flow rate and to the required power.

For most applications of propulsion in space, the desired result is an acceleration of the vehicle. For this reason, the required thrust of the propulsion system is proportional to the mass, which scales as the cube of a characteristic linear dimension of the spacecraft. As we have seen, the required power scales with the thrust, and thus it too will scale as the cube of the linear dimension. However, power is generally produced via solar panels in proportion to the area exposed to the sun. Thus the available power scales as the square of the linear dimension. For this reason, the fraction of available power required to produce a given acceleration of a spacecraft scales as the characteristic linear dimension:

$$\frac{P_{req}}{P_{avail}} \propto \frac{mass}{Area} \propto \frac{L^3}{L^2} \propto L \tag{4.13}$$

Thus a smaller fraction of total power will be required to produce the same acceleration in smaller spacecraft. This widely-known relation is often cited as a reason for using notoriously power-hungry electric propulsion on smaller spacecraft, provided the electric thrusters can be scaled to the small sizes required.

However, not all applications of propulsion are based on providing a constant acceleration. For drag makeup, the drag force that must be counteracted depends on the frontal area, which scales as the square of the linear dimension. In this case, the ratio of required power to available power will be independent of spacecraft scale, and the argument for electric propulsion in these cases is not as clear. The same argument applies for using thrusters to counteract the torques generated by drag and by solar pressure, as they are produced by forces that scale as the square of the linear dimension acting over distances that scale as the linear dimension. The thrusters must produce opposing torques by providing a thrust that acts over a distance which scales as the linear dimension. This thrust must then scale with the square of the linear dimension as above.

In addition, there is a third special case that is unique. The torque due to the gravity gradient effect scales with the difference between spacecraft principle moments of inertia. Since the moments of inertia scale as mass multiplied by the square of a linear dimension, the disturbance torque scales as the linear dimension to the fifth power. The thrusts that act to produce a counteracting torque must then scale as the linear dimension to the fourth power, and in a manner similar to above we have:

$$\frac{P_{req_{gg}}}{P_{avail}} \propto \frac{F}{L^2} \propto \frac{L^4}{L^2} \propto L^2 \qquad (4.14)$$

The ratio of required power to available power goes as the characteristic length squared, an even better result for small spacecraft than in the constant acceleration regime above. For this application, electric propulsion would make even more sense, using the traditional logic. However, it is worth noting that the primary reason for this improved performance is that the gravity gradient torques simply get smaller at a much faster rate than the spacecraft does. Other disturbance torques and forces do not go away so conveniently as the size of the spacecraft decreases, and will in most cases end up dominating the gravity gradient torques.

4.5 Specific Electric Propulsion Technologies

This section will discuss each electric propulsion technology considered in turn. The principle of operation will be identified and explained, any scaling arguments required will be discussed or developed, and a model to determine the propulsion system mass will be identified.

4.5.1 Resistojets, Electrothermal Hydrazine Thrusters, Arcjets

These three technologies, which are becoming more prevalent in the large communications satellite industry are basically crosses between electric and chemical propulsion. They are not included in this study due to their high power requirements, even at fairly low thrust levels. For example, Olin Aerospace is marketing a "low power" arcjet which requires an input power of 1.8 kW to produce approximately 0.2 N of thrust at 500 sec specific impulse. For completeness, the technologies are summarized below in Table 4.5

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Table 4.5: Summary of Resistojets, EHTs and Arcjets

Attribute	Resistojet	EHT	Arcjet
Specific Impulse	250 sec	300 sec	500 sec
Propellant	many	Hydrazine	many, Hydrazine common
Additional components required:	tank, valve, power conditioning unit	tank, valve, power conditioning unit	tank, valve, power conditioning unit
Advantages:	simplicity, higher I_{sp} than cold gas thrusters	higher I_{sp} than hydrazine monopropellant	high I_{sp}
Disadvantages:	very high power requirement, added mass of PCU	high power requirement, added mass of PCU	very high power requirement, added mass of PCU

4.5.2 Ion Thrusters

The basic concept of ion thruster operation is to create heavy ions in an ionization chamber, and then to accelerate these ions electrostatically to very high exit velocities.

This is illustrated in Figure 4.2. The acceleration is created by applying a large potential

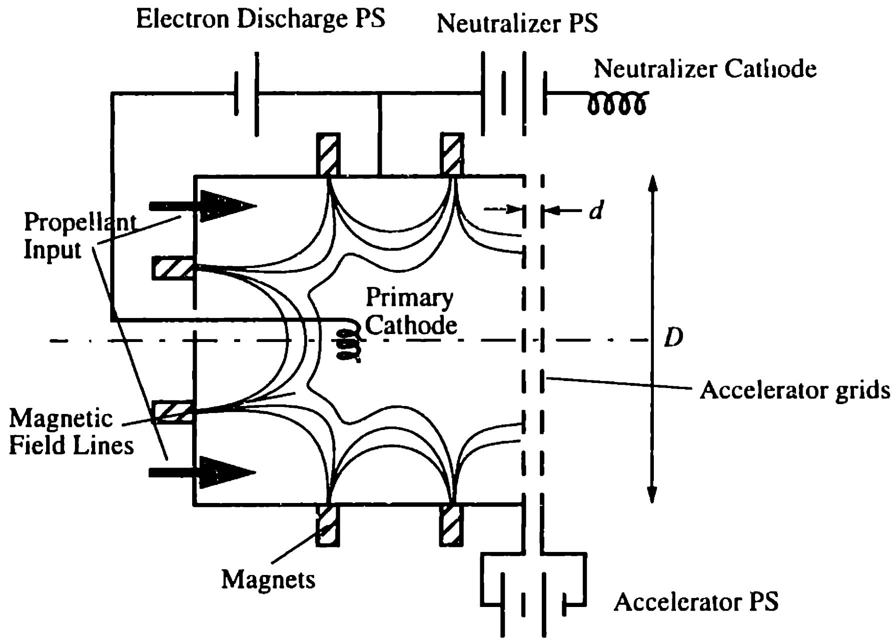

Figure 4.2: Ion Engine Schematic

difference between two grids, with the ions being accelerated towards the outer grid. The ions are created in the body of the thruster via electron bombardment, in the case illustrated here. Electrons are emitted from the primary cathode where they are contained by the magnetic fields to prevent them from immediately flowing to the anode (here the outer surface of the ionization chamber) Gas is passed through this swarm of electrons, and a certain fraction becomes ionized. Once a ion reaches the accelerator grid, it is attracted by the large negative potential on the outer grid and accelerated to the exhaust velocity, c. Eventually the electrons make their way to the walls of the chamber and are collected. Most are emitted to neutralize the ion beam at the neutralizer cathode, and the rest sent back to the primary cathode to repeat the process.

The net voltage required across the accelerator to achieve a certain exhaust velocity is given approximately by:

$$V = \frac{m_i c^2}{2q} \tag{4.15}$$

where m_i is the mass of an ion, c the exhaust velocity, and q the charge of the ion. For a specific impulse of 3000 sec ($c \sim 30\,000$ m/s), and assuming singly-ionized Xenon as the working gas, yields a required voltage of approximately 600 V. Since this voltage is applied between two grids, the electric field will increase if the thruster is made smaller. At some point, the small quanity of gas in the accelerator gap will break down, and the thruster will no longer be able to operate. If an upper limit of the electric field is set as 10^8 V/m, then the minimum separation distance is on the order of 10 μm. Of course, on the traditional scales of ion thrusters with diameters of 10-50 cm, it is nearly impossible to maintain such a small separation distance without contact between the grids, and thus a larger grid spacing of approximately 1 mm is used. For a given grid spacing and voltage, it can be shown that only a certain current of ions can pass through a unit area of the grid. This is called the space charge limited current and given by the Child-Langmuir Law:

$$j_B = \frac{J_B}{A} = \frac{4\sqrt{2}}{9}\varepsilon_o\sqrt{\frac{m_i}{q}}\left(\frac{V^{3/2}}{d^2}\right) \tag{4.16}$$

where j_B is the beam (ion) current per unit area, J_B the total beam current, A the exit or grid area of the thruster, ε_0 the permittivity of vacuum, V the applied accelerating voltage, and d the grid spacing. Based on this, the thrust can be written as:

$$F = \frac{8}{9}\varepsilon_o\left(\frac{V}{d}\right)^2 A \tag{4.17}$$

Thus, if the voltage is held constant to preserve specific impulse, and the thruster is scaled geometrically, the thrust will be constant. The electric field in the gap ($\sim V/d$) will be scaling as $1/L$, and at some point the breakdown field will be reached. However, though the thrust per unit area (and thus per mass of thruster) will be going up, the required power will remain constant (if the efficiency can be held invariant) or increase, if there are efficiency penalties to the scaling procedure.

Since there are two separate processes involved in ion thrusters, it is worthwhile to separate them for scaling discussions. The first is the ion creation process, which governs the efficiency of the device, and the second is the acceleration, which governs the specific impulse and the thrust of the device. Most ion thrusters now use electron bombardment ionization to create the ions in the ionization chamber. If the efficiency is desired to remain invariant as the size of the chamber becomes smaller, then the objective should be to ensure that "the physics" do not change. To do this, a parallel discussion to that presented in [13] is followed. Looking at the Brophy Model [14], one notices that the inputs used to determine efficiency are functions of geometry, which will remain constant if the thruster is scaled geometrically, and the energy of the Maxwellian (secondary) electrons, assuming the various operating voltages are held constant. In order for the efficiency to remain constant, the electron temperature (as well as ion and neutral temperatures) must also remain constant. Beginning with this assumption, the required scaling of the various parameters to ensure that this is the case can be worked out. To ensure the same frequency of collisions in the scaled thruster, the various mean free paths, λ, must scale with the characteristic thruster length, L. This requires that:

$$\lambda \sim \frac{1}{nQ} \propto L \tag{4.18}$$

where n is the number density of a given species, and Q the relevant collisional cross section. Since Q is generally dependent only on temperature, it remains constant, and thus the number densities must scale inversely with the characteristic length.

In order for the electron containment characteristics of the scaled thruster to be similar to the original, the electron gyro radius must also scale with the characteristic thruster length. This means:

$$r_L = \frac{c_e m_e}{eB} \propto L \tag{4.19}$$

where r_L is the gyro radius, c_e the mean electron speed, m_e the mass of an electron, e the charge of the electron, and B the magnitude of the local magnetic field. Since c_e is a function of electron temperature which is to be maintained constant, the magnetic field must scale inversely with the characteristic length. Because B must scale as $1/L$, as the size of the thruster decreases, the required magnetic field for containment of the electrons will increase as well. This will eventually present a lower limit on the feasible scaling size, as it will become impossible to generate a magnetic field large enough to effectively contain the electrons, and a lower electron temperature will be necessitated. Most ion engine designs currently use permanent magnets that produce a magnetic field on the order of 1000 Gauss.[14] The largest magnetic field that can be produced by a permanent magnet is of the order of 1 Telsa (10 000 Gauss), so a scaling in size by a factor of 10 should be feasible.

The mass flow rate is made up of the ions and neutrals, and thus scales as:

$$\dot{m} \sim nvA \propto nL^2 \propto L \tag{4.20}$$

where A is the exit area, and v the velocity of the appropriate particle, which depends only on temperature, and thus is invariant with scale. Since the beam current scales directly with mass flow rate, it will also scale with the characteristic length. The power is simply the product of the current and applied constant voltage, so it too scales with the linear dimension.

It would appear feasible to maintain the efficiency of ion engines as they are scaled smaller, provided a sufficiently larger magnetic field can be produced. Turning to the acceleration portion of the thruster, it is necessary to see how the scaling requirements shown above impact the thrust of the device.

Thrust is the product of the mass flow rate and the exhaust velocity, and since the exhaust velocity remains invariant and we have seen above that the mass flow must scale with the characteristic linear dimension (Equation 4.20), the thrust must also scale as the linear dimension. This counteracts what was implied by Equation 4.17, which shows that for a pure geometrical scaling the thrust should remain constant. In order for the thrust to scale as the linear dimension, the ratio between the thruster diameter and gap thickness must change. This implies:

$$F \propto \left(\frac{D}{d}\right)^2 \propto \frac{L^2}{d^2} \propto L \tag{4.21}$$

$$d \propto \sqrt{L} \tag{4.22}$$

where D is the diameter of the engine. Thus in order to provide the correct thrust to ensure that the efficiency and specific impulse remain constant after scaling, the ratio of gap width to thruster diameter must scale inversely with the square root of the characteristic linear dimension. The gap width will get larger relatively as the thruster is scaled downwards. However, this effect will not have a very significant effect in terms of mass. If a 10 cm thruster with a gap width of 0.5 mm was to be scaled down by a factor of 10, the ratio of gap width to diameter would increase by a factor of about 3.2, but it would still only be about 0.016.

To model the mass of the thruster, we will use the scaling laws developed here, but ignore the effects of the non-geometric scaling of the gap width. We use the dimensions and power requirement of the Hughes 13 cm engine [15] to begin the model. The Hughes thruster has a mass of 5.0 kg, and requires a power of approximately 440 W. If the thruster was to be scaled geometrically by a factor of 8.8 to yield an operating current of 50 W[1], a mass on the order of 5 g could be assumed. However, at this small size, the magnets will have to be relatively larger and additional support structure will be required. Looking at the example of the scaled Hall thruster presented below and in [16], the eventual mass scaling factor was not L^3 but approximately $L^{1.4}$ due to the larger magnets required. This scaling factor would lead to an Ion engine with a mass of approximately 240 g. However, as the ion engine does not require the full magnetic circuit necessary in the Hall thruster, the error in scaling factor should not be so extreme. The baseline mass for the 50 W ion engine will be taken as 120 g, which corresponds to a mass scaling as $L^{1.7}$. The 50 W engine will be taken as a minimum size, with an engine requiring additional power for more thrust scaling as:

$$m_{ION} = m_{ION_0}\left(\frac{P_{req}}{P_o}\right)^{1.7} \tag{4.23}$$

where m_{IONo} is the mass of the reference thruster, 120 g, P_{req} is the required power, and P_o is the reference power, 50 W. Table 4.6 summarizes the qualities of an ion thruster:

Table 4.6: Ion Thruster Summary

Attribute	Value
Specific Impulse	3000 sec
Efficiency	65%
Propellant	Xenon gas
Additional components required:	tank, valve, power conditioning unit, power supply

1. The power evtually became 45W, as the scaling factor of 8.8 was used on the Hughes' thruster's nominal 18mN thrust to yield a thrust of 2 mN for the miniature ion engine, and then the required power backed out using the assumed efficiency of 65%.

Table 4.6: Ion Thruster Summary

Attribute	Value
Advantages:	very high I_{sp}
Disadvantages:	low thrust per area, high magnetic field required at small sizes

4.5.3 Hall-Effect Thrusters

A schematic of a Hall Thruster is presented in Figure 4.3. Its principle of operation is similar to an ion engine, in that it uses an applied potential to accelerate ions to a high exhaust velocity. The cathode emits electrons at low potential, and they attempt to go to the anode, which also serves as the injector of the propellant gas. The electrons are contained somewhat by a radial magnetic field across the channel gap that is part of a toroidal magnetic circuit. While the electrons are trapped in the magnetic field, they encounter the neutral atoms of the propellant gas flowing down the channel and ionize some fraction of them. Once ionized, the ions are immediately accelerated out the channel to high velocity by the electric field set up by the negative potential of the cathode. Eventually, the electrons escape the magnetic field containment and make their way to the anode. In traditionally sized Hall thrusters, the magnetic field is set up through a few sets of coils around the iron magnetic circuit at various points, but in order to provide a large enough magnetic field on a smaller scale, permanent magnets are used.

Hall thrusters that are used today operate on 0.5 kW to 1 kW levels of power [17]. To be useful on the microspace scale, they must be scaled down significantly. As this scaling occurs, it is desired to preserve the physics, specific impulse, and efficiency as invariant. The required analysis has been performed elsewhere [13], and has the following consequences: voltage is held constant to preserve specific impulse. Power and thrust scale as the characteristic linear dimension, L. In order to maintain the same relative electron gyro

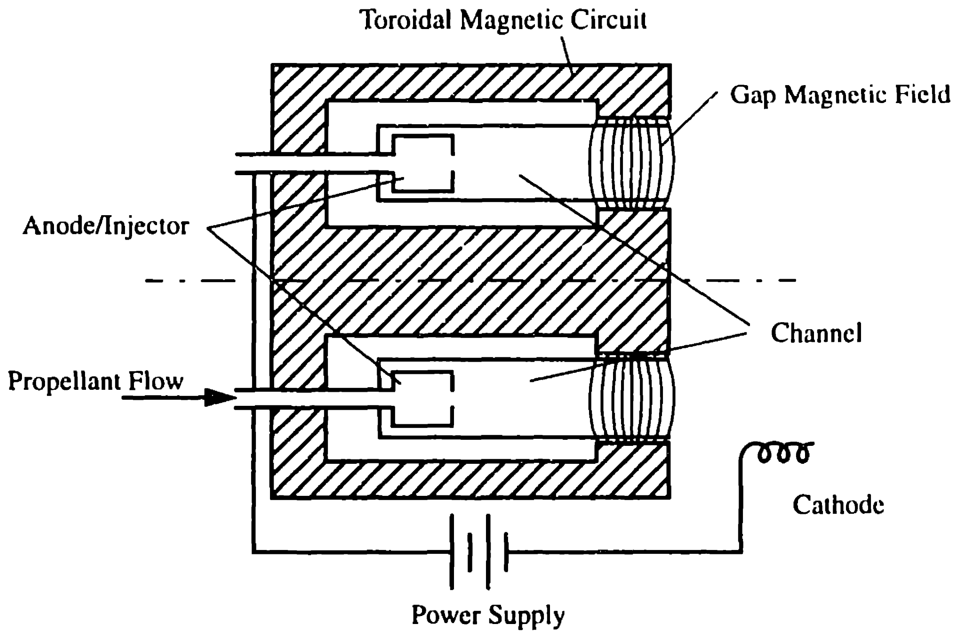

Figure 4.3: Hall Thruster Schematic

radius in the smaller channels, the magnetic field must increase as the thruster gets smaller; it scales as $1/L$. This presents the primary difficulty in producing these small Hall thrusters, as the traditional magnetic field generation technique of using a solenoid is no longer sufficient. Permanent rare-earth magnets are used instead. A colleague is currently developing a 50 W Hall thruster [16], and the mass model used here is based on his thruster design. The I_{sp} of the 50 W design is 1600 sec, and the efficiency is 50%. This yields a thrust of 3 mN. Based somewhat on fabrication issues (the channel diameter is only 3.6 mm), but mostly on the difficulty of creating a larger magnetic field in the gap, this will be considered a minimum size, and if a higher thrust is required, the thruster will be scaled up from this reference. The estimated mass of the 50 W design is 40 g. However, if the geometrical scaling laws were to be believed, the mass should have been $m_0(P/P_0)^3$, or 0.6 g. The difference is that at the small scale the permanent magnet and magnetic circuit dominate the mass of the thruster. The scaling factor appears to be closer to $L^{1.4}$.

Thus, to more accurately model the mass of the thruster, the following relation will be used:

$$m_{HALL} = m_{HALLo}\left(\frac{P_{req}}{P_o}\right)^{1.4} \tag{4.24}$$

where m_{HALLo} is the reference mass of 40 g, P_o the reference power of 50 W, and P_{req} the desired power of the thruster.

Table 4.7: Hall Thruster Summary

Attribute	Value
Specific Impulse	1600 sec
Efficiency	50%
Propellant	Xenon gas
Diagonal components required:	tank, valve, power conditioning unit, power suppiy
Advantages:	high I_{sp}
Disadvantages:	high magnetic field required at small sizes

4.5.4 Pulsed Plasma Thruster

The pulsed plasma thruster (PPT) is a fairly simple device conceptually. It was first developed in the mid 1960's and first flown on the LES-6 satellite[19,20,21]. They have been used for both attitude control and station-keeping, and are considered useful because they can produce a very small and repeatable impulse bit. In addition, they use a solid propellant, so a propulsion system utilizing them does not require tanks or valves. To create the pulse, an arc is initiated across the face of a solid block of teflon, ablating and ionizing a very small amount of the teflon. The ionized portion is accelerated through a self-induced magnetic field out the nozzle, and the ablated portion expands adiabatically out the nozzle in the same way. A schematic of the thruster is presented in Figure 4.4. Since

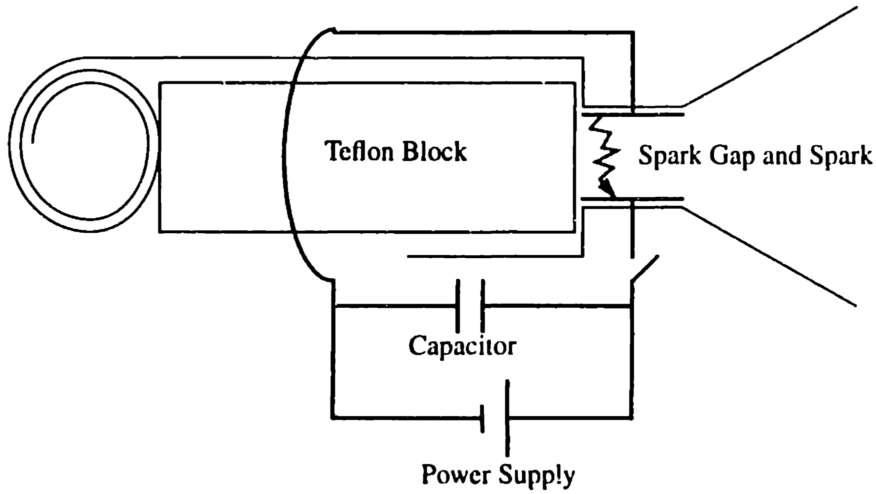

Figure 4.4: PPT Thruster Schematic

this thruster is pulsed, the power delivered is stored in a capacitor bank until the discharge is initiated. The impulse bit delivered per firing can be seen to be:

$$I_{bit} = \frac{2\eta E}{c} \tag{4.25}$$

where E is the energy stored in the capacitor and discharged during each pulse, η is the thruster efficiency, assumed here to be 15%, and c the exhaust velocity, assumed here to be 10 000 m/s. To scale this thruster, it is assumed that to preserve performance the energy density, or the ratio of energy discharged by the capacitor to the face area of the teflon block, should remain constant. Thus, $E \propto L^2$. Since the energy per pulse and the impulse bit both scale linearly with each other for constant specific impulse and efficiency, the characteristic length must scale as the square root of the impulse bit. (or of the time averaged thrust, for that matter). For the reference engine, an energy per pulse of 10 J is used, as given in []. This energy was discharged over a one square inch face of teflon, so the reference engine must be created around this dimension. Most of the mass of the thruster is the capacitor. Assuming a specific energy of 50 J/kg, this mass can be calculated. The rest of the mass of the engine is the housing and miscellaneous electronics. For the one square inch bar, this housing is modeled as three hollow aluminum cubes, each with a side length of 3 cm and thickness of 1 mm, for a total mass of 50 g. The capacitor has a mass of 200 g. Thus the total mass of the PPT thruster is modeled as:

$$m_{PPT} = m_{housing} + m_{capacitor} = m_{PPTo}\left(\frac{E}{E_o}\right)^{3/2} + \frac{E}{\alpha_{capacitor}} \tag{4.26}$$

where m_{PPTo} is the reference mass of the thruster housing, 50 g; E is the energy per pulse, found from the required impulse bit via Equation 4.25; E_o the reference pulse energy, 10 J; and $\alpha_{capacitor}$ the specific energy of the capacitor, 50 J/kg. Table 4.8 summarizes PPT characteristics.

Table 4.8: PPT Summary

Attribute	Value
Specific Impulse	1000 sec
Efficiency	15%
Propellant	solid Teflon
Diagonal components required:	power conditioning unit, power supply
Advantages:	high I_{sp} compared to chem, no tank or valve requirements, simplicity, small & repeatable impulse bits
Disadvantages:	low efficiency

4.5.5 Field Emission Electric Propulsion[1]

Field Emission Electric Propulsion (FEEP) was originally developed in Europe, and much of the research and development of the systems continues to be concentrated there. FEEP is basically an ion engine where the ionization and acceleration are both [illegible]

1. Unless otherwise specified, the information for this section was taken from [22].

by the same electric field generated by a plate at an extremely high negative potential. The FEEP concept is schematically illustrated in Figure 4.5. Cesium (Cs), a liquid metal, is used as the propellant. It is kept in a reservoir, and comes to the narrow slit opening at the emitter tip via capillary action. At this point, there is a meniscus of sorts at the tip, and surface tension forces are sufficient to prevent any liquid from flowing out in the absence of an electric field. As the potential difference between the plate and tip increase, the electric field goes up accordingly. At a sufficiently high electric field, cusps will begin to form along the tip, and the local electric field will increase even more. Once the field has reached approximately 10^9 V/m, the atoms at the tips of the cusps will be spontaneously ionized by field emission, and once ionized accelerated away by the negative potential of the accelerator. A total applied voltage of approximately 10 kV is required, leading to a specific impulse on the order of 10 000 sec. Specific power is quoted as approximately 50 W/mN, so for a thrust of 10 μN, the power required is only 0.5 W. However, as mentioned above, the power must be delivered at multi-kV level voltages, which leads to relatively large power conditioning equipment. Impulse bits on the order of 10^{-8} Ns are feasible.

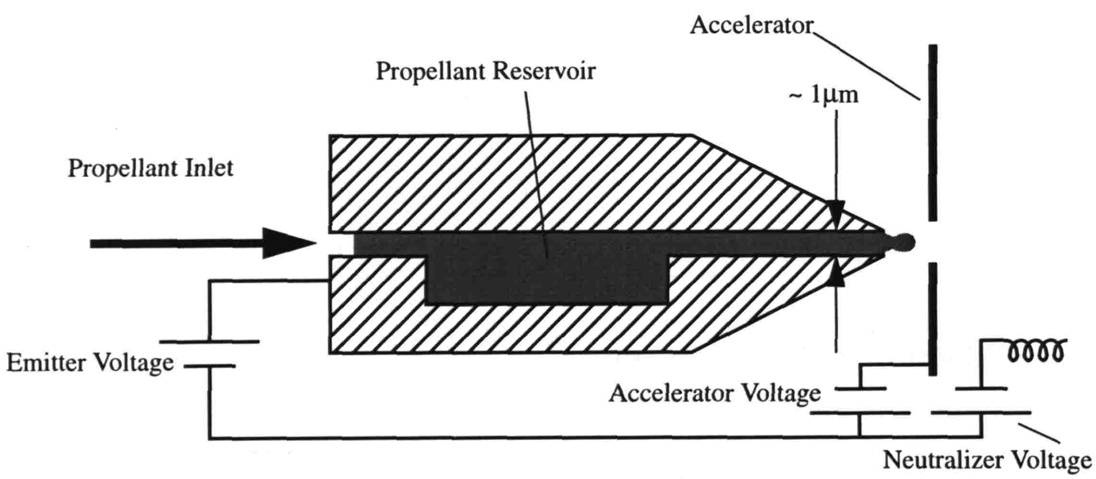

Figure 4.5: Schematic of FEEP Concept. Adapted from [22]

FEEP thrusters are perhaps ideally suited to missions with extremely low thrust requirements. The particular application mentioned repeatedly in the literature is to drag free satellites where the satellite continually "chases" a free flying test mass inside of itself, with the FEEP thrusters providing the small impulses required to cancel out any disturbance observed, providing a completely disturbance-free trajectory. The extremely high specific impulse means that the propellant mass is usually negligible, and often can be completely contained in the emitter reservoir. Reference [22] provides a system mass estimate for the devices which will be used in this study without further scaling, as the thrusts and power requirements seem appropriate. Each thruster has a mass of 375 g, and each power conditioning unit has a mass of 1 kg. The power required by each thruster is 1.2 W, so this leads to a α_{ps} (see Section 4.6.2 below) of 1.25 W/kg, which is rather horrendous, but considering the low powers involved, does not lead to power supply masses that are overly excessive. Other tankage and valves will not be necessary, as the small quantity of propellant needed should be able to fit into the thruster reservoirs. A summary of FEEP thrusters is presented in Table 4.9 below.

Table 4.9: FEEP Thruster Summary

Attribute	Value
Specific Impulse	10 000 sec
Efficiency	95%
Propellant	liquid Cesium
Diagonal components required:	power conditioning unit, power supply
Advantages:	extremely high I_{sp}, limited tank and valve requirements, simplicity, extremely small & repeatable impulse bits, extremely low thrust
Disadvantages:	for some applications, I_{sp} too high & thrust too low

4.6 Scaling of Other Components

As has been mentioned previously, in traditional space propulsion systems, the thrusters usually make up a fairly small proportion of the total propulsion system mass when compared to the other parts of the system. For the purposes of this study the other parts of the system have been classified as power supply, power conditioning, propellant, propellant tanks, and valves. Each of these will be discussed briefly in the sections that follow, and models for their masses explained and developed.

4.6.1 Power Supply

The principle power supply in space is solar voltaic cells. For the purposes of this study solar arrays will be modeled as having a specific power of 70 W/kg. This is based on an attempt to find a consensus among a number of sources, and assuming a slight improvement in performance in the near future. Though today one can find solar power systems with specific powers higher than 70 W/kg, the cost is often substantial, and as the primary objective of microspacecraft is to reduce cost, it was felt that this would be a conservative value for specific power that could be had at a reasonable price. The weight of the power supply system is then given as:

$$m_{PS} = \frac{P_{req}}{\alpha_{PS}} \tag{4.27}$$

where P_{req} is the required power, and α_{PS} is the specific power parameter, 70 W/kg.

4.6.2 Power Conditioning Equipment

For electric propulsion particularly, it is the power conditioning unit (PCU) that often dominates the power system and weighs far more than the arrays required to produce the power in the first place. It will be assumed that the mass of this also scales linearly with power processed, though because it is usually a box on a given spacecraft that must have various cables attached to it, a minimum mass will be set. Assuming an aluminum hollow cube as the minimum box, a minimum side length of 5 cm is chosen. With a wall thickness of 0.5 mm, the box has a mass of 20 g. Adding another 20 g for the various cards, connectors, and other components that must be inside, the mass for a power conditioning unit that processes no power is 40 g. For the other point of the scaling law, a representative PCU from an application of the particular type of thruster is chosen. This translates into a scaling law given as:

$$m_{PC} = m_{PC_o} + \frac{P_{req}}{\alpha_{PC}} \tag{4.28}$$

where m_{PC_o} is the minimum mass of 0.040 kg, P_{req} is the power that is processed, and α_{PC} is the scaling parameter, which depends on the type of thruster being used. A table of the scaling parameters used is presented below. In all cases, the efficiency of the power conditioning unit is taken to be 85%.:

Table 4.10: PCU Scaling Parameters for Electric Thrusters

Thruster type	α_{PC} [W/kg]
Ion	65
Hall	100
PPT	150
FEEP	1.25

4.6.3 Propellant

The required mass of propellant is determined based on the specific impulse of the thruster system and the required ΔV that the propellants must provide. This is governed by the rocket equation, given as:

$$m_{prop} = m_o\left(1 - e^{-\frac{\Delta V}{c}}\right) = m_o\left(1 - e^{-\frac{\Delta V}{I_{sp}g_o}}\right) \tag{4.29}$$

where m_o is the initial mass of the satellite, and c the thruster exhaust velocity which is equal to the product of I_{sp} and g_o, where g_o is the acceleration of gravity at the Earth's surface.

4.6.4 Propellant Tanks

Propellant tanks scale based on their volume and the pressure that they must contain. For each case, the volume of the tank will be determined from its the propellant density and mass, and the pressure required is based on the type of thruster being used. In the case of tanks that are not highly pressurized, a minimum gauge should be specified. Assuming spherical tanks, their thickness is given by:

$$t_{tank} = \frac{Pr}{2\sigma}FOS \tag{4.30}$$

where P is pressure contained in tank, r its radius, σ the working strength of the material, and FOS a factor of safety. The tank mass is simply:

$$m_{tank} = 4\pi r^2 t_{tank}\rho_{tank} \tag{4.31}$$

where t_{tank} is the larger of t_{min}, the minimum gauge thickness, or t_{tank}, calculated from Equation 4.30, and ρ_{tank} is the density of the tank material. Composite materials have both excellent strength to weight ratios and are particularly suited to handling the tensile loads found in pressure vessels. For these reasons they will be the material baselined in this model. If the propellant is incompatible with the composite material, then a stainless steel tank will be used.

4.6.5 Valves

Valves are one of the more troubling parts of the propulsion system, and in order to minimize leaks they are often designed with significant internal redundancy through placing many in series and in parallel. Much study could be dedicated to valves, but for this effort, a simple scaling law will be used. The valves will be assumed to scale with mass flow rate, with the baseline being the valve system used in one of Olin Aerospace's low thrust hydrazine thruster.[23] This gives:

$$m_{VALVE} = m_{VALVEo}\left(\frac{\dot{m}}{\dot{m}_o}\right)^{3/2} \tag{4.32}$$

where m_{VALVEo} is the mass of the reference valve, 0.2 kg, $\dot{m}$ is the desired mass flow rate, and $\dot{m}_o$ is the mass flow rate of the reference valve, 2.4 g/s. However, since valves are generally purchased off the shelf and are not specifically designed for each mission, they do not necessarily scale with mass flow. For example, if Olin's Low Power Arcjet, which uses the same valve system as the hydrazine thruster, had been used as the scaling point, the scaled mass would have been approximately 10 times larger for the same actual mass flow. For this reason, in those cases where valves are required, three cases will be evaluated. The first case will use the same valve mass of 0.2 kg independent of flow rate, the second will use a valve mass of 10 g, based on a micro-valve concept currently in development by Mirada Scientific Controls[24], and the third will use the mass obtained from the scaling law above. This will help determine how important a role the valves play in the overall propulsion system, and more importantly, how sizable the effect of reducing valve mass would be.

Chapter 5

Technology Matching and Selection

5.1 Introduction

This chapter describes the process that was used to match the propulsive technologies described in Chapter 4 with each of the missions described in Chapter 3. The chapter begins with a discussion of the methodology and processes used to perform the matchings. This is followed by sections devoted to each mission in which the results of the matching process are presented. The chapter concludes with a discussion of the overall results of the matching process, and an identification of those technologies that seem most promising for future use in microspacecraft.

5.2 Matching Process and Methodology

The matching process is conducted on each mission in turn. It is an effort to choose the combination of possible propulsive technologies that minimizes total propulsion system mass. For each mission, each possible propulsive technology is applied to the mission propulsive requirements, and based on these requirements and the mass models for that propulsive technology discussed in Chapter 4, a system mass is estimated. However, in most cases, no one propulsive technology can meet all the requirements of the mission, so the technology is evaluated only for the specific requirements that it can meet, and then some combination of technologies is ultimately used. For example, the PPT technology is well suited to the small impulse, low thrust requirements of attitude control, whereas ion and Hall thrusters are more suited to the continuous, relatively higher thrust requirements of orbit transfer or orbit phasing. As another example, FEEP is a very low thrust system that in general cannot be made to meet the minimum thrust requirements of rephasing. When a combination of technologies is necessary as in these examples, each set of technology combinations that could feasibly meet the requirements is evaluated, and the set with the lowest combined system mass chosen.

The process for determining the system mass for a given thruster technology is illustrated in Figure 5.1. Based on the mission requirements and the thruster technology being considered, the ΔV and required thrust for that technology is determined.The ΔV and

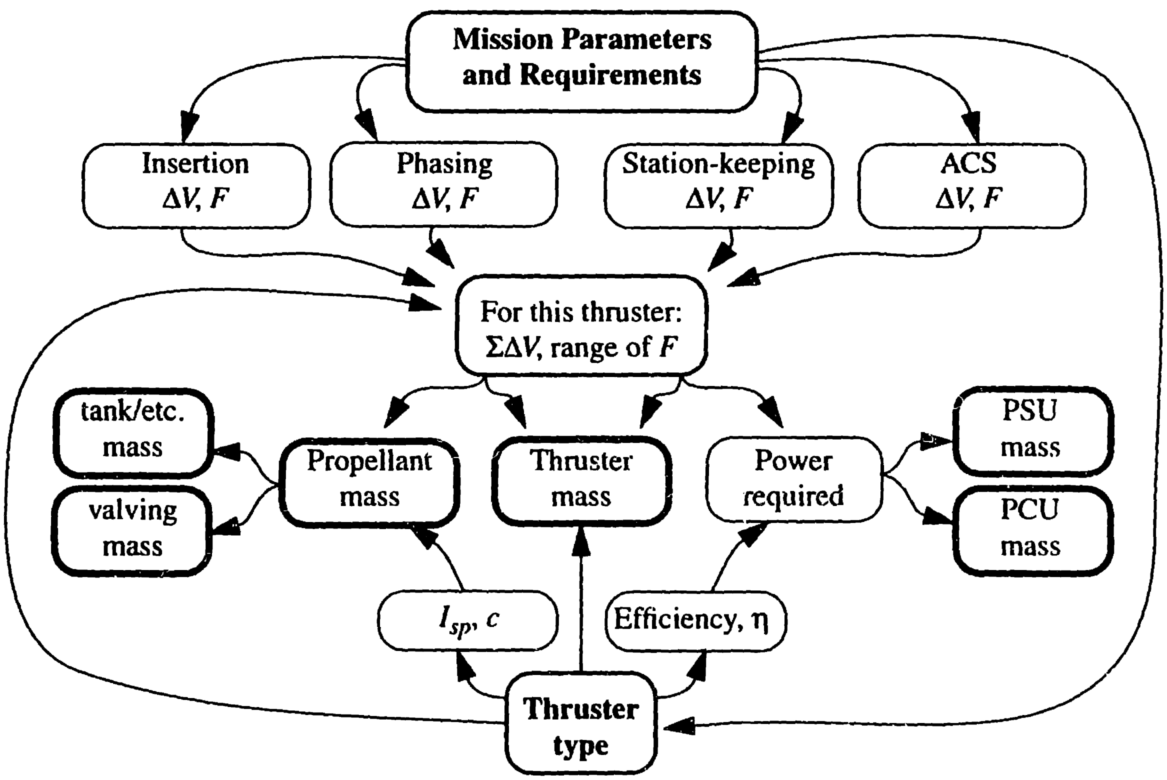

Figure 5.1: Schematic illustrating process for matching Missions and Technologies

thruster characteristics (I_{sp}) determine propellant mass required. Propellant mass and type determines tank mass and valve mass. The thruster efficiency, specific impulse and required thrust determine the power requirement if any, and the power requirement determines the mass of the arrays required to supply that power and the mass of the equipment needed to condition and regulate that power. The final step is to identify the number of each component required in the whole system, and then sum the masses to determine a total propulsion system mass.

5.3 Separated Spacecraft Interferometry

The first mission to be considered is the New Millennium Separated Spacecraft Interferometry (NMI SSI) mission. The only requirements of this mission are station-keeping and attitude control, for a ΔV of 78 m/s for the collectors and 50 m/s for the combiner. The minimum continuous thrust was calculated to be 6.2 μN for the combiner and 75 nN for the collectors, with the maximum allowable impulse about 10 mNs for the collector and 1 mNs for the combiner. Because the ΔV requirements are based on [], where the total mass of propellant was the same in the combiners and collector, this will continue to be the case, and only one of the three vehicles needs to be considered. This also makes sense from a cost standpoint, as the same propulsion system will be suitable for all three of the vehicles.

Ion Thrusters

Ion thrusters are considered first. The minimum size ion engine that was determined to be feasible in Section 4.5.2 was a thruster with 2 mN of thrust that required 45 W and had a mass of 100 g. It is questionable if it will be able to be operated in a manner to provide the minimum impulse bit of 1 mNs, which would imply that it must be able to be turned on for only half a second at a time, but this will be assumed to be possible for the time being. Since the thruster is to provide both translation and rotation of the vehicle independently, a total of 12 thrusters will be required. It is unlikely that all 12 thrusters must ever be fired simultaneously, so the PSU and PCU will be sized for eight thrusters. The advanced small valves are assumed, though it is worth noting that if the 200 g valves that are more common today were used, the system mass would increase by nearly 2.5 kg. The system mass for an Ion thruster system would be 12.75 kg.

Hall Thrusters

Next Hall thrusters are considered. The minimum size for them is 40 g, each producing 3 mN of thrust, and requiring 50 W of power. The same concern over minimum impulse bit size mentioned for Ion engines applies, but will not be addressed further. Again 12 thrusters are required, and again the power system is sized for eight being operational at a time. The total system mass is approximately 11 kg.

Pulsed Plasma Thrusters

PPT's are much more suited to the requirements of this mission, as they are inherently very low thrust devices. There does not appear to be a minimum size, and thus they are scaled as discussed in the previous chapter. For this application, they are sized to producing a 30 μNs impulse every second, for a time averaged thrust of 30 μN. As in the case of the ion and Hall thrusters, 12 thrusters are required. Because the thrusts are so much lower, it is possible that all the individual thrusters could be called on to fire at once, so the power system is sized for all 12 thrusters. The total system mass is not quite 2 kg.

Field Emission Electric Propulsion

FEEP is also well inclined to this mission, as it produces thrusts that are on the same order as the PPT chosen above. In addition, its high efficiency (~ 95%) means that it produces this same thrust at a specific impulse ten times higher than PPT's, but requiring nearly the same amount of power. However, in its current incarnation, its mass is too large to make it practical except on the longest missions where its extremely high I_{sp} (and thus low propellant usage) would make up for its large "fixed" mass in the thruster and power conditioning equipment. The system mass is about 12.75 kg.

Cold Gas Thrusters

Cold Gas thrusters are currently baselined for the NMI SSI mission. Based on the model presented in the previous chapter, the mass of a cold gas system is just the propellant, the tank, and the valves. The total cold gas system mass as calculated is about 24.5 kg. Thus, all forms of electric propulsion considered present quite an improvement over the baselined system.

Chemical Propulsion

Other kinds of chemical propulsion were not considered for this mission, as the thrusts and impulses that the mission required were too small. Even if the chemical systems could have produced thrusts small enough, their propellant mass would have exceeded the total system mass for the PPT system.

Technology Selection

Pulsed Plasma Thrusters are the best choice for this mission. They provide a system mass that is almost six times less than the Hall thruster system which was the second best system in terms of system mass. In addition, if the mission were to be extended beyond its current six month duration, the only additional requirement would be propellant. To increase the mission time by a factor of four to two years, an additional 3 kg of propellant would be required, bringing the system mass to approximately 5 kg, which is still nearly five times less than the mass of the cold gas system as determined by this study for the current 6 month mission. Table 5.1 shows the mass breakdown of the selected system, and Figure 5.2 shows a comparison of the considered technologies. A complete mass breakdown for each system considered is available in Appendix B, Section B.2.

Table 5.1: NMI SSI PPT System Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 020**
Thruster	12	21	252
Hardware Total			**252**
Power Supply	12	14	168
Power Conditioning	12	46	552
Power Total		**60**	**720**
System Total			**1 992**

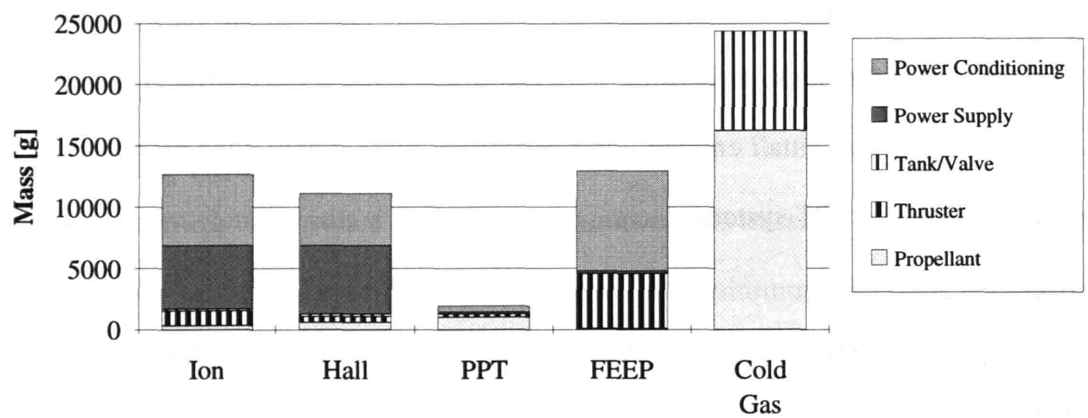

Figure 5.2: System Mass Comparison for NMI SSI Mission

5.4 Future Global Positioning System

The next mission to be considered is the future GPS mission. The mission requirements can be separated into two groups quite naturally. The first is orbital maneuvers, namely insertion correction and re-phasing, and the second is attitude control. The first group requires a relatively higher thrust (15mN minimum for phasing), and in general only requires thrusting in one direction at once, so many thrusters are not required. Attitude control, on the other hand, requires a time-averaged thrust of only 20 nN, and does require 12 thrusters. It makes sense to treat the two groups of requirements separately.

For the orbital maneuvers, ion and Hall thrusters will be considered first. Their minimum thrusts are too large for the attitude control application, as demonstrated in the NMI SSI mission above, but they may prove useful for the phasing and orbit correction requirements. For these orbital maneuvers, in the low thrust case, the total required ΔV for this mission is approximately 200 m/s. The minimum thrust requirement is 15 mN. The various chemical systems will also be considered for the orbital maneuvers. Since they can provide sufficient thrust to be considered impulsive, the impulsive ΔV requirement of 145 m/s will be used.

Ion Thrusters

The minimum thrust of these engines is 2 mN, so providing a thrust of 15 mN will not be difficult. An 16 mN ion thruster will have a mass of 4.2 kg, according to the model presented in Section 4.5.2. However, two 8 mN ion thrusters will have a total mass of 2.44 kg, and produce the same total thrust when using the same total power. This is due to the scaling laws that were presented earlier. Based on this argument, it would make sense to produce any required thrust by some number of 2 mN thrusters acting in parallel, each with a mass of 100 g. Of course, at some point the added weight of the extra valves, lines, wires and other support equipment will outweigh the mass saved in the thrusters. So though it does not make sense to take this to an extreme, in general two thrusters will be better than one. In this case, after looking at all the options of producing 16 mN of thrust, the decision was made to use 5 thrusters that each produced 3.2 mN of thrust. The system mass is approximately 13 kg.

Hall Thrusters

For Hall thrusters, the same arguments apply, and 5 thrusters are again chosen to produce the required 16 mN of thrust, with each one producing 3.2 mN. The total mass of the system is approximately 8.2 kg, which is a significant improvement over the ion thruster system. This improvement is due to the fact that both the hardware and power mass required for the hall thrusters is less than for ion engines for a given thrust. This will always be true, and only once the required ΔV is high enough will the higher I_{sp} of the ion engine play a large enough role for an ion thruster based propulsion system to overtake the Hall thruster based system and have a smaller total system mass.

Hydrazine

Of the total required impulsive ΔV of 145 m/s, the largest single impulsive ΔV that must be performed at any given time is 40 m/s. If an impulse maneuver is considered to last 10 minutes at most (about 1.5% of an orbit), the required thrust level would be 7 N. Similar scaling arguments to those discussed in the Hall and ion thrusters apply, and it makes more sense to produce this thrust with more than one thruster. Two hydrazine thrusters, each producing 3.5 N, are chosen. The valve mass used is that scaled from current designs, and the total system mass is approximately 9.5 kg.

Hybrid Motor

The hybrid motor system is also in the impulsive class. One 7 N motor will be used to meet the orbit maneuvering requirements. The total system mass is about 6.25 kg.

Pumped micro-Bipropellant Rocket

The conceptual micro bipropellant rocket would also qualify as an impulsive option for the orbit maneuvering requirements of this mission. Two of the 4.5 N engines would be needed, although the feed could be from the same tank. Total system mass is about 5.4 kg. The micro-bipropellant rocket is the best choice in terms of total system mass for the orbital maneuvering requirements. There are a few points worth noting however. First, the pumped micro rocket is probably the farthest from existence of the systems that were considered. Second, it nominally requires 60 W of power for each motor to drive the pumps. It is envisioned that these burns would occur when the satellite had excess power, and thus providing the extra power for this application should not be a problem. Thirdly, the tank masses in this case reflect the minimum gauge imposed. Since they are no longer highly pressurized, the pressure requirement no longer determines their thickness. If the minimum gauge could be reduced, the mass of the tanks would decrease further, as would the system mass. However, the tank mass is currently only about 10% of system mass, which is dominated by the propellant mass. Finally, the mass of the oxidizer tank has been artificially inflated by a factor of three to account for the cyrogenic issue. In all likelihood, if this technology was to be applied to long term space missions, a storable propellant would be used. This would decrease the hardware mass, but may increase the propellant mass somewhat since the specific impulse could decrease.

Attitude Control - PPT, FEEP, and Cold Gas

The other part of this mission is the attitude control requirement. The ΔV requirement is 85 m/s, with a maximum impulse bit of 1.5 mNs. PPT, FEEP, and cold gas systems should be able to handle these requirements fairly easily due to their low thrust levels, but again, the fixed weight of the FEEP system and the high propellant weight of the cold gas system will lead to the PPT system having the lower total system mass by a significant margin. For all three systems a thrust of 25 μN is assumed. The PPT system has an overall mass of about 1.75 kg, the FEEP system is approximately 12.5 kg, and the cold gas system is approximately 19 kg.

Technology Selection

An orbit maneuvering system to provide the insertion fixing and the rephasing must be combined with an attitude control system to provide the attitude control. The PPT ACS is clearly superior to the other two possibilities, and Figure 5.3 presents the total system mass when the PPT ACS is combined with each of the five orbit maneuvering options. The best choice is the PPT ACS combined with the micro-bipropellant engine, with the hybrid motor system having only a slightly larger mass. Of the electrical systems, the ion thruster based system has a total mass larger than that of the hydrazine based one, almost entirely due to the large power requirements. The complete mass breakdowns for each technology are given in Appendix B, Section B.3, and the breakdown for the chosen set of technologies is given in Table 5.2. The hybrid motor is shown in the table as it is a near term technology, when compared to the micro-bipropellant engine.

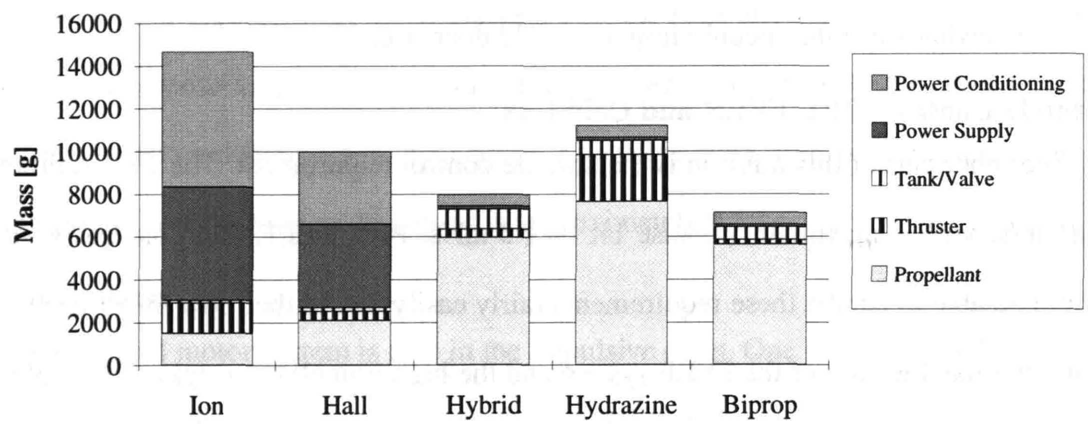

Figure 5.3: System Mass Comparison for Future GPS Mission
(each system includes PPT Attitude Control)

Table 5.2: Future GPS Hybrid/PPT System Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
PPT Propellant	12	70.8	850
Hybrid Propellant	1		5 140
Propellant Total			**5 990**
Hybrid Motor	1	170	170
PPT Thruster	12	18	215
Tank	1	750	750
Valve	1	185	185
Hardware Total			**1 320**
PPT Power Supply	12	12	145
PPT Power Conditioning	12	46	550
Power Total		**58**	**695**
System Total			**8 005**

5.5 Infrared Earth Observing Cluster

The requirements for the IR Earth Observing Cluster mission can also be divided into two groups. The first group contains the orbit phasing and station-keeping requirements, and the second is the attitude control requirements. For the attitude control requirement of a 50 m/s ΔV at very low thrust, PPT thrusters are the only reasonable alternative, as the previous looks at the attitude control systems have determined. They are assumed for the ACS role, with that segment of the system having a total mass of just over 1 kg. However, the choice for the station-keeping and rephasing requirement is less clear, and will be investigated more thoroughly.

Ion Thrusters

Ion thrusters can be used to provide the station-keeping requirements for the mission. The minimum thrust of 4.4 mN is provided through two ion engines, but since the North/

South Station Keeping requires a thrusting that alternates in direction (orbit normal and anti-orbital normal) a total of four thrusters are required, although only two will operate at any given time. For this reason, only two power systems are required. For the supply of power, it is assumed that in most cases the batteries can be used to provide all or at least half of the approximate 100 W required to operate the thrusters. Thus, only one unit of power supply will be required. The total system mass for the ion thruster configuration is about 4.5 kg.

Hall Thrusters

Hall thrusters are also considered for this application, with similar assumptions as to number of thrusters and number of required power systems. The total system mass for the Hall thruster alternative is about 5 kg.

Pulsed Plasma Thrusters

PPT is also considered for the job of primary propulsion. Unfortunately, it is not possible to generate the required thrust with this technology to meet the time constraint in the rephasing requirement, and thus if it were to be ultimately chosen, this requirement would have to be relaxed. Again four thrusters are required, but only two power supplies are utilized, as more than two thrusters would not be firing at once. Because the thrust is lower than the ion or Hall thrusters, the ΔV requirement for NSSK is larger, and the thrusting to correct that must occur continuously. The total system mass is about 6 kg.

Technology Selection

The ion thruster is the lowest mass system to perform the station-keeping and phasing requirements, and combined with the PPT attitude control system is the lowest overall system mass for this mission. Figure 5.4 presents a comparison of the different systems and Table 5.3 presents a more detailed breakdown of the final combination system chosen, and a complete breakdown of all systems considered in in Appendix B, Section B.4. In this application, the ΔV requirement was sufficiently high to allow the high I_{sp} of the ion system to make up for its large fixed mass primarily in the form of power conditioning equipment.,

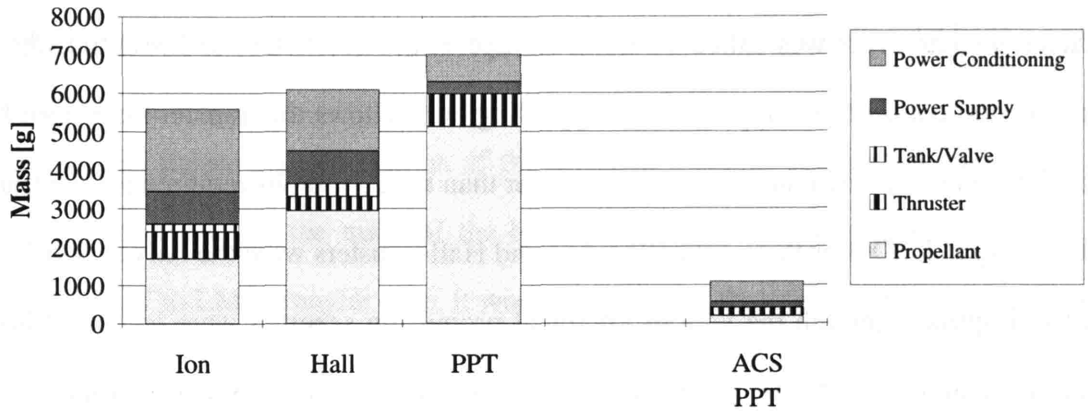

Figure 5.4: System Mass Comparison for IR EOS Cluster
(each system on left includes PPT ACS shown on right)

Table 5.3: IR EOC Ion/PPT System Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
PPT Propellant	12	70.8	205
Ion Propellant	1		1 500
Propellant Total			**1 705**
PPT thruster	12	18	215
Ion thruster	4	120	480
Tank	1	160	160
Valve	4	10	40
Hardware Total			**895**
PPT Power Supply	12	12	145
PPT Power Conditioning	12	45	540
Ion Power Supply	1	715	715
Ion Power Conditioning	2	805	1610
Power Total			**3 010**
System Total			**8 005**

5.6 MiniMars

For the MiniMars mission, two scenarios were evaluated. The first was a completely electric propulsion transfer from LEO to Low Mars Orbit (LMO). The second was an electric propulsion transfer beginning once Earth's gravity well had been escaped by the launch vehicle. This was called an Earth Escape Trajectory (EET) to LMO transfer. It is this second concept that appears most appealing, as it allows the transfer time from Earth to Mars to be approximately 15 months, rather than the 33 months required for the transfer beginning at LEO. For this application, ion and Hall thrusters were the only seriously considered options, though the various chemical propulsion schemes were looked at briefly, before realizing that the propellant mass required became excessive quickly. The extremely low thrusts available from PPT and FEEP prevented them from being considered as the missions would have taken much too long.

Ion Thrusters

Two 9 mN ion engines were chosen as the thrusters, which provided a transfer time from EET to LMO of approximately 15 months. The total system mass is approximately 43 kg, which is quite reasonable to deliver a 70 kg satellite. Has the full LEO to LMO transfer been attempted, it would have taken 33 months, and the system mass would have been about 75 kg.

Hall Thrusters

One 21 mN Hall thruster was chosen to provide the same 15 month transfer time. The system mass was approximately 70 kg, and if the full 33 month transfer had been attempted, the total system mass would have been 160 kg. Though this sounds large, it still represents a LEO to LMO transfer with a payload fraction of 30%.

Chemical Thrusters

Chemical thrusters were briefly looked at, but their relatively low specific impulse meant that the propellant required even for the EET to LMO transfer was enormous when compared to the electric thrusters. For the hybrid motor, the propellant mass was 365 kg, for the bipropellant it was 314 kg, and for a hydrazine thruster it would have been 727 kg.

Technology Selection

Figure 5.5 presents a comparison of the different systems considered, and Table 5.4 gives the breakdown of the mass of the best system, the ion thruster. Both are for the 15 month EET to LMO transfer case. It would appear that there are some significant gains

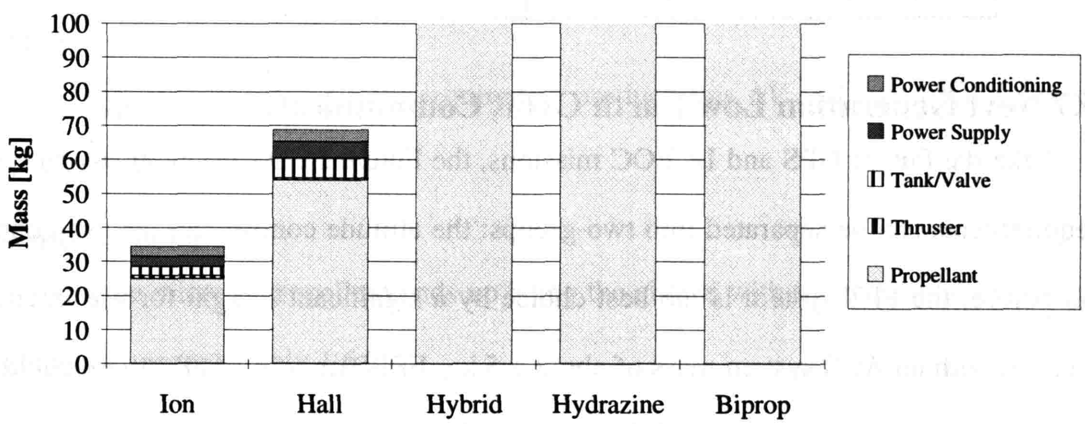

Figure 5.5: System Mass Comparison for MiniMars Mission (systems shown provide propulsion for EET-LMO transfer)

to be made, especially in launch vehicle size, if such a scheme was used without a very large penalty in terms of transfer time. The breakdown of each system considered is given in Appendix B, Section B.5.

Table 5.4: MiniMars Ion Thruster Mass Breakdown (15 month EET-LMO)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**25 000**
Thruster	2	1 500	3 000
Tank	1	2 700	2 700
Valve	2	10	20
Hardware Total			**5 720**
Power Supply	2	2 900	5 800
Power Conditioning	2	3 200	6 400
Power Total		**6 100**	**12 200**
System Total			**42 920**

5.7 Next Generation Low Earth Orbit Communication System

Like the Future GPS and IR EOC missions, the Future LEO Communications system requirements can be separated into two groups: the attitude control and the orbit control. As before, the PPT system is the best choice by a significant margin for use in attitude control, with an ACS system mass of about 1.5 kg. FEEP or cold gas thrusters could also provide the attitude control, but FEEP's currently large thruster and power system mass makes it too large, and the low I_{sp} of the cold gas thrusters makes the mass of its propellant, and thus mass of the tank large as well.

Ion Thrusters

For the low thrust devices, the required ΔV for orbit control is 106 m/s. Ion thrusters were first considered for the orbit control requirement. The minimum thrust requirement for the phasing maneuver was 5 mN, so three of the minimum size 2 mN ion thrusters were chosen. This leads to a mass of 4.7 kg for the ion thruster-based orbit control system.

Hall Thrusters

Hall thrusters were also considered. Since their minimum size for this study is 3 mN, two are required to produce the required thrust. This leads to an orbit control system mass of about 3 kg.

Hybrid Motor

For impulsive devices, the required ΔV for orbit control is 81 m/s. To ensure that the thrusters are effectively impulsive, a thrust of 2.5 N is chosen. This thrust allows a typical impulsive ΔV for this mission of 13 m/s to be accomplished in about 4 minutes, which is about 4.5% of time spent in one orbit. The hybrid motor system to produce this thrust and ΔV would have a mass of 1.75 kg.

Hydrazine

The hydrazine thruster system required to produce this ΔV and thrust has a mass of approximately 2.75 kg.

Micro-Bipropellant

Again, the micro-bipropellant ends up being the choice with the minimum overall system mass of approximately 1.6 kg.

Technology Selection

The micro-bipropellant rocket, though the lowest overall mass, continues to be very close to the hybrid system, which is closer to development and actual use. Thus, once again, the promise of the micro bipropellant engine will be noted, and the mass breakdown for the hybrid rocket-based system presented in Table 5.5, and the mass breakdown for each system considered is presented in Appendix B, Section B.6. A comparison of the total propulsion system mass (both orbit and attitude control) is presented in Figure 5.6.

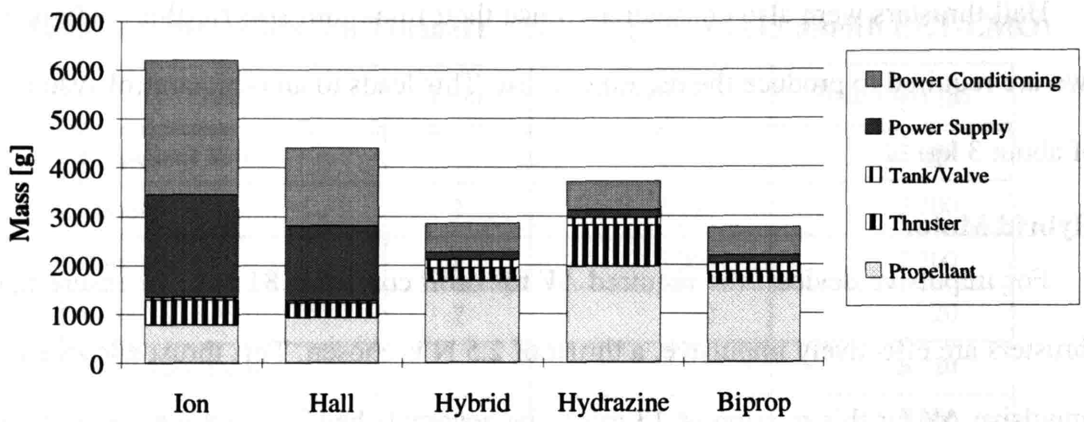

Figure 5.6: System Mass Comparison for Future LEO Comm. System
(each system includes PPT Attitude Control)

Table 5.5: Future LEO Comm. System Hybrid/PPT System Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
PPT Propellant	12	50.8	610
Hybrid Propellant	1		1 450
Propellant Total			**2 060**
Hybrid Motor	1	50	50
PPT Thruster	12	18	215
Tank	1	210	210
Valve	1	40	40
Hardware Total			**515**
PPT Power Supply	12	12	145
PPT Power Conditioning	12	46	550
Power Total		**58**	**695**
System Total			**3 270**

5.8 Low Altitude Earth Observation

The low altitude EOS mission does not have any attitude control requirements, and thus only requires an orbit control system. The ΔV requirements are 476 m/s for low thrust systems with a minimum thrust of approximately 1 mN, and 454 m/s for impulsive systems.

Ion engines

Since the minimum thrust of the Ion engines being considered is 2 mN, only one thruster will be required. This leads to a system mass of approximately 2.2 kg.

Hall thrusters

Likewise, since the minimum thrust of a Hall thruster considered in this study is 3 mN, one will be required to produce the required thrust. This leads to a total system mass of about 2.6 kg.

PPT system

PPT's could in theory be made large enough to produce 1 mN of thrust, but they are better suited to lower thrusts. In this case, two 0.5 mN PPT's will be used. Thus, only in those rare occasions when phasing is necessary will both be required, and for the rest of the time they can alternate in performing the drag makeup function and provide a higher reliability through redundancy. Unfortunately the system mass of the PPT system is almost 1 kg larger than that of the Hall system, at 3.5 kg.

Hydrazine

Hydrazine thrusters are currently baselined as the primary propulsion system for drag makeup on the satellite being designed and built by Draper Laboratory on which the mission considered in this study is based. As such, the expected lifetime is limited to approximately six months[7]. The mission being considered in this study has a nominal lifetime of three years. For a comparison, the mass of a hydrazine system to perform the same function was calculated and is 8.8 kg, about four times larger than the total mass of the ion thruster-based system.

Technology Selection

The ion system has the least mass of the three systems considered. A comparison of the system masses is given in Figure 5.7. However, it is important to note that the Hall thruster based system is only 400 g heavier. Considering the uncertainties in the study, this is not an extremely large difference, and both would make a considerable improvement to a chemical thruster based system, especially for low altitude missions with high drag makeup requirements such as this one. A mass breakdown of the ion system is presented in Table 5.6, and a complete breakdown of each technology considered for this mission is presented in Appendix B, Section B.7

Table 5.6: Low Altitude EOS Ion Thruster Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**650**
Thruster	1	100	100
Tank	1	70	70
Valve	1	10	10
Hardware Total			**180**
Power Supply	1	645	645
Power Conditioning	1	735	735
Power Total		**1 380**	**1 380**
System Total			**2 210**

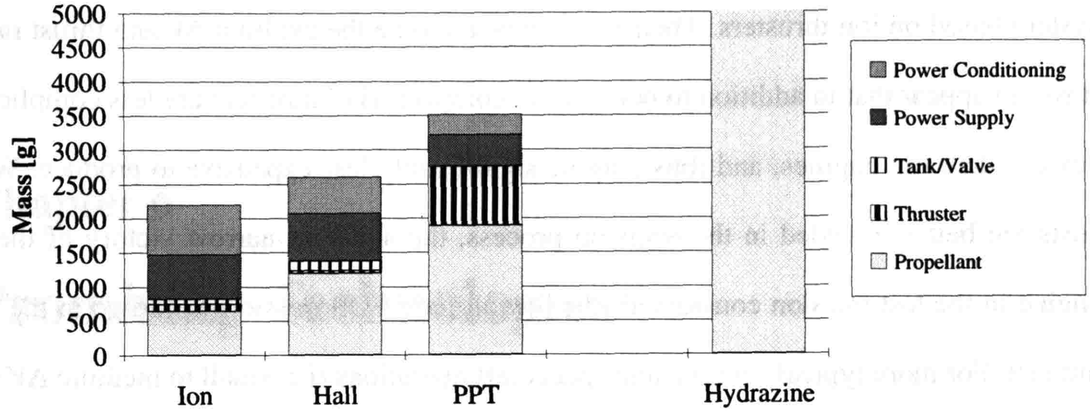

Figure 5.7: System Mass Comparison for Low Altitude EOS Mission

5.9 Identification of Most Promising Technologies

Based on the results presented in the previous six sections, an attempt can be made to identify those technologies that will be most promising for use in microspacecraft in general, and in particular in missions similar to those discussed in this study. In fact, there appear to be many promising technologies, as most of those discussed would appear to have a niche where they seem to make the most sense for use. They will be identified here, and then discussed in more detail in the following chapter.

PPT seem to have definite applications in the small thrust regime. They are a fairly simple conceptual device which does not appear to have many barriers to scaling. Though FEEP did not "win" in any of the missions discussed here, this is primarily due to the fact that in its current incarnation, both the emitter and power conditioner are extremely heavy. If these deficiencies could be remedied, FEEP would become much more useful. Ion engines are particularly effective for high ΔV missions, such as the MiniMars mission. By making them smaller, the thrust to thruster weight ratio seems to improve dramatically, and improve overall thruster performance. Hall thrusters also did not "win" any of the missions considered, but in many instances a system based on them would be lighter than a system based on ion thrusters. Their niche appears to be the medium ΔV and thrust range. It would appear that in addition to being more compact, Hall thrusters are less complicated devices than ion engines, and thus may be significantly less expensive to produce. When costs are better included in the decision process, the apparent narrow victory of the ion engine in the last mission considered (the low altitude EOS mission) could go to the Hall thruster. For more typical satellite and spacecraft operations (i.e. small to medium ΔV's), it would appear that traditional chemical approaches hold a fair amount of promise in the form of small hybrid motors and micro-bipropellant rockets. Because they lack the large power supplies and conditioners required by electric thrusters, they can often come out ahead, as was seen repeatedly in this study. Additional effort should definitely be invested in further developing these technologies.

Chapter 6

Promising Technologies

6.1 Introduction

Having identified those technologies that seem most promising for application to microspacecraft, this chapter will attempt to expand on these issues somewhat. For each technology, a niche domain of application will be identified to show when and where this technology is most likely to be successfully applied. Additional discussions and some speculation as to what "technical hurdles" need to be surpassed in order to apply the technology successfully to microspacecraft will be made. The chapter will conclude with a brief discussion of the conclusions reached in this study and some recommendations of future work that can be done in this area.

6.2 Miniature Ion Engines

6.2.1 Domain of Applicability

Ion thrusters seem to be advantageous principally in missions with very high ΔV requirements thanks to their high I_{sp}. In comparing them to Hall thrusters the mission ΔV cutoff where the propellant savings of Ion engines seem to overcome the smaller power supply and conditioning equipment of Hall thrusters is approximately 400 m/s for thrusts of about 3 mN. This cutoff value actually increases fairly quickly as the required thrust levels go up, as in general a larger ion thruster and power supply is required to produce the same amount of thrust as a Hall thruster. In fact if the thrust level is 5 mN, the mission ΔV

cutoff where the total system mass of an Ion engine system becomes less than that of a Hall thruster is about 600 m/s. For extremely high ΔV requirements, such as the MiniMars mission discussed in this study, ion engines are clearly the propulsion technology of choice, provided the thrust can be made high enough so that the transfer time is not excessive. As was seen in this study, one way to reduce a large portion of the time required for the interplanetary transfer is to choose the launch vehicle so that it is capable of placing the spacecraft on an Earth Escape trajectory. In the example discussed here, this reduced the travel time and propulsion system mass by more than a factor of two. Of course, the mass of the last stage of the launch vehicle will be significantly larger than otherwise necessary, but considering that small spacecraft are being considered, this is not an unreasonable proposition.

6.2.2 Technical Hurdles

The largest technical hurdle that must be overcome in making ion engines smaller is probably a manufacturing issue. It is quite difficult to manufacture devices as complex as ion engines on the scale required. However, it is possible that MEMS technology could play a role in making this miniaturizing possible. The 2 mN ion engine considered the minimum size in this study was scaled from the Hughes 13 cm ion by a factor of 8.8. The diameter of the thruster considered here would then be about 1.5 cm, which is on the upper end of the scale at which the MEMS technology of manufacturing is feasible. However, producing the magnetic field to contain the ions is probably not feasible with MEMS techniques, since it requires the use of fairly large pieces of rare-earth magnets, a far cry from the traditional MEMS materials of Silicon and some metal plating. Thus, combination of MEMS and traditional fabrication techniques would in all likelihood be required. Producing the magnetic field is another technical hurdle. As was seen in the scaling discussion in

Chapter 4, to preserve the physics and thus the efficiency of the thruster as it is made smaller, the magnetic field must increase inversely with the characteristic length of the thruster. Producing a magnetic field of the required magnitude will probably not be overly difficult, but reproducing the complex field patterns at the smaller scales required could be extremely difficult. It is possible that relaxing the constraint that the "physics stays the same" and allowing smaller magnetic fields, though it would impact the efficiency, could lead to even smaller thrusters. It is unclear what effect this would have on the entire system, though since the thruster mass is generally small compared to the power equipment, any decrease in efficiency could magnify itself detrimentally into additional power conditioning and supply mass. In addition, the possibility of using other ionization techniques besides electron bombardment (such as contact ionization) may make sense at this smaller scale. These are all issues that can and should be addressed in future work in this area.

6.3 Miniature Hall Thrusters

6.3.1 Domain of Applicability

Hall thrusters seem to be most appropriate for use in missions with medium ΔV requirements that require thrusts on the order of 3 to 20 mN. Of course, as medium is a very relative and vague term, it is important to define this further. For the application to future microspacecraft, medium ΔV requirements seem to range from about 150 m/s to approximately 450 m/s. As was pointed out above, this varies somewhat depending on the thrust required. For higher thrusts, the domain of applicability extends to higher ΔV requirements. It is the author's opinion that when costs and manufacturablity are eventually taken into account, the Hall thruster's domain will extend to even higher ΔV's. This is due to the much higher complexity of an Ion engine system when compared to a Hall thruster system, both of the thruster, and of the supporting power conditioning equipment.

Additional analysis to support this position has not been done to date, and it remains simply an educated speculation.

6.3.2 Technical Hurdles

The technical hurdles required to produce a miniature Hall thruster are basically the same as those for the Ion thruster discussed above. However, the hurdles should be somewhat lower in each case. To an extent this has been demonstrated by Khyams[16], who is in the process of constructing the 50 W thruster considered as the minimum size in this study, although to date testing has not yet begun. Although the magnetic field required is probably higher in magnitude than in a comparable ion engine, a complex field shape is not required, as it is desired to be basically radial across the gap (See Figure 4.3, page 90), and thus it should be easier to produce.

6.4 Pulsed Plasma Thrusters

6.4.1 Domain of Applicability

PPT's seem to be extremely well suited to fine position and attitude control where very small impulse bits are required. In addition, they are suited to medium ΔV missions where there is no (or a very low) minimum thrust requirement, Their fairly high specific impulse leads to a low propellant mass, and the solid propellant eliminates tanks, valves, lines, and pressure regulators, greatly simplifying the propulsion system. This also allows them to be placed more remotely and strategically on a spacecraft, without the need to transfer propellant via a line from a central tank. Though their energy efficiency is low, and they thus require more power per unit of thrust than either Hall or ion thrusters, the thrusts that PPT's are ideally suited for are so small that this effect is not that important. It is the author's opinion that the capability of providing small thrusts will become increasingly important, and not simply in the world of microspacecraft. Large flexible space structures like long trusses being considered for non-separated interferometry will require actuators to control them, and the corrective forces involved will be extremely small.

At low enough thrust levels and at mission ΔV requirements of 200 to 300 m/s, FEEP with its very large specific impulse will begin to make more sense than PPT's in terms of total system mass, but it is fairly rare for ΔV requirements for attitude or fine position control to be this high, except for extremely long-life missions.

6.4.2 Technical Hurdles

There are a few technical issues that need to be addressed prior to PPT's becoming more commonplace, but none seem especially difficult to overcome. The electronics required for their operation need to be miniaturized and ideally integrated more closely with the thruster itself. Any further improvements in capacitor energy storage density would also be very beneficial, as the vast majority of the thruster mass is the capacitor which is discharged in each pulse. Some fabrication issues also exist, as it is unclear as to how best to make these smaller thrusters. As no barriers to scaling appear to exist, it maybe possible to take the thrusters to MEMS technology level, and potentially integrate the electronics onto the same "chip," although again a mix of more traditional materials and techniques would be required to integrate the solid teflon fuel rod into the assembly. Most basically, additional investigation into PPT's must be carried out. There has not been much research done in this area in the recent past, and most of what was done has centered on making them produce larger impulse bits rather than the smaller ones that microspacecraft missions call for.

6.5 Field Emission Electric Propulsion

6.5.1 Domain of Applicability

As was discussed above, currently FEEP seems most promising for very large ΔV missions that require (or can permit) low thrusts. However, if the unit mass of an emitter and the power conditioning electronics could be reduced significantly, FEEP systems would begin to be mass-effective for even lower ΔV missions. In their current incarnation they apparently do make sense for larger spacecraft that require very small forces, such as "drag-free" missions [22].

6.5.2 Technical Hurdles

The technical hurdles that need to be overcome to make FEEP more useful to small spacecraft are precisely those mentioned above: reduction in mass, particularly the mass power conditioning electronics. Based on an admittedly very limited knowledge and understanding of the technology, the author sees no real reason why the emitter needs to be as massive as it is reported to be. As can be seen in Figure 4.5 (page 90), the thruster is an extremely simple device, consisting basically of a reservoir of fluid connected to a very thin slit, with an accelerator electrode a small distance from the slit exit. Providing the materials are compatible with the propellant, such a device could be fabricated via MEMS technology, which may provide some significant savings in the mass of the emitter. The other more massive part of the system is the electronics that must provide a voltage differential on the order of 10kV between the emitter and accelerator. Certainly any reduction in this mass would be very welcome, and if it the electronics could integrated into the system as part of the MEMS manufacturing, one can imagine a rather simply produced, compact, and modular thruster that produces extremely low thrusts requiring very small amounts of propellant.

6.6 Small Hybrid Motors

6.6.1 Domain of Applicability

Small hybrid motors appear to have significant promise in the fairly near term for small satellite propulsion. They appear to make the most sense for application to missions where impulsive maneuvers are possible, and where mission ΔV is less than about 300 m/s. This means that they would typically be applied to orbital maneuvers, such as orbital insertion or rephasing, and perhaps drag makeup for missions that are not at extremely low altitudes. This can be seen in their selection over Ion and Hall thrusters for these types of tasks in the Future GPS and Future LEO Communication missions.

6.6.2 Technical Hurdles

There do not appear to be any technical hurdles to the development of these systems. The University of Surrey has a development program ongoing[9], which appears to be progressing quite successfully, though no flight tests have yet occurred.

6.7 Micro-Bipropellant Rockets

6.7.1 Domain of Applicability

The micro-bipropellant turbopumped rocket first discussed in Section 4.3.5 on page 70 has the same domain of applicability as the hybrid rockets discussed above, as it has similar if slightly better performance in most cases, according to this study. In addition, it would in all likelihood have some major applications outside of the purely microspacecraft arena. If a few of the planar engines were stacked on top of each other, a reasonable thrust could be produced, and they could possibly be used for transfer stages for geosynchronous satellites Though they would not be the nearly impulsive transfers provided by the solid motors traditionally used now, the transfer times should be significantly less than those envisioned for purely electric transfer vehicles utilizing very high I_{sp} thrusters like ion engines, and the mass savings thanks to the higher I_{sp} of the bipropellant engines would be significant.

6.7.2 Technical Hurdles

This technology probably has the most technical hurdles, simply because it does not exist on any scale today save the engines used in launch vehicles. The launch vehicle engines produce thrusts that are five orders of magnitude larger than those being considered for this application. However, some of the technical hurdles seem to be lowering, thanks to similar on-going research. MEMS technology has been mentioned fairly extensively in this chapter, but it is only this concept that is completely dependent on it. The only reason that these micro-turbopumped rockets may be feasible is that MEMS techniques may allow the relative tolerances available in manufacturing in the macro world of space shuttle main engines to be maintained at the micro scale. Thus, even though there are many sub-hurdles that make it up, the only real technical hurdle for this technology is the rather daunting one of actually developing and building such a rocket, as it has quite simply never been done before. This rather exciting hurdle is what the author intends to dedicate his doctorate research to clearing successfully.

6.8 Conclusions and Future Work

This study has attempted to determine which propulsive technologies make the most sense for use on microspacecraft. To evaluate this, six missions were selected to represent likely missions for microspacecraft in the future. In addition a number of possible propulsion technologies were identified and models for each developed to predict total propulsion system mass (including thruster, propellant, tanks, valves, and power supply and conditioning equipment) as a function of mission parameters and propulsive requirements. Each of the applicable technologies was examined for each mission, and system masses estimated. Based on the lowest system masses, the most promising technologies were identified.

It appears that there are many promising propulsion technologies for use in microspacecraft. These technologies tend to map into domains of applicability if the two parameters of thrust and mission ΔV are considered. An attempt to represent these domains graphically is presented in Figure 6.1.

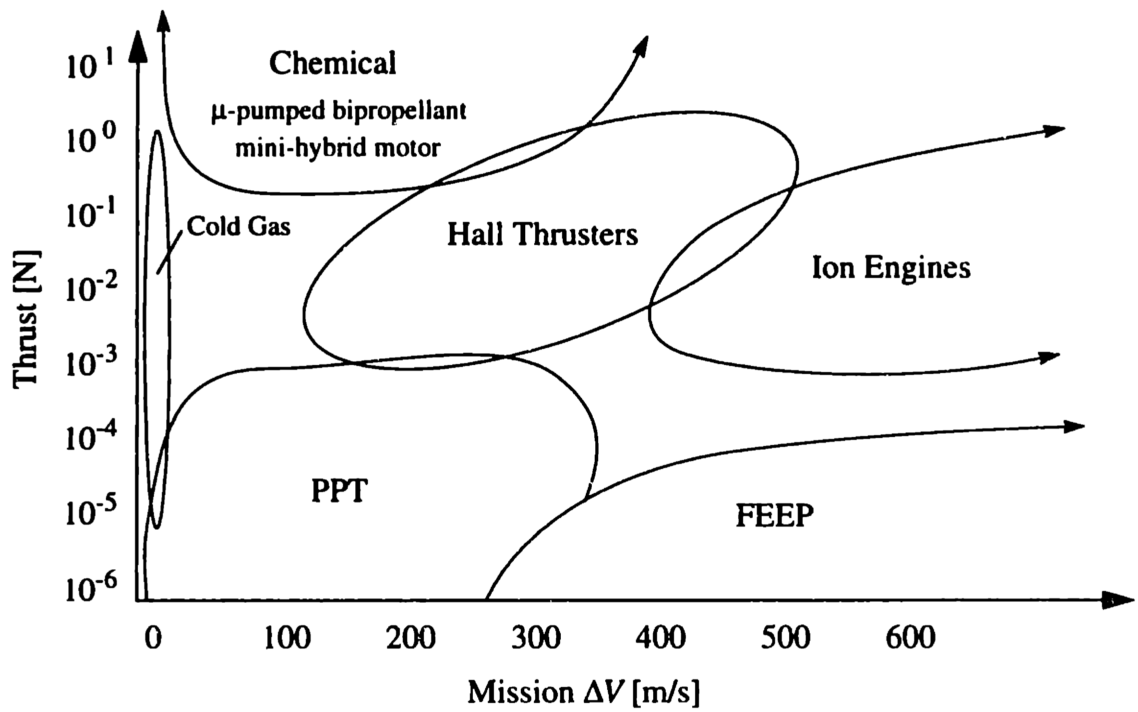

Figure 6.1: Domains of Applicability for Propulsion Technologies

Basically, Pulsed Plasma Thrusters appear to be most promising for very low thrust applications (approximately 1 to 50 μN) in missions with low to medium ΔV requirements (20 to 300 m/s). For low thrusts, but higher ΔV requirements (300 m/s and larger), Field Emission Electric Propulsion appears to be most qualified. In the realm of medium thrust (approximately 2 to 50 mN), the Hall thruster and ion engine are the contenders, with the

Hall thruster being a better choice for ΔV requirements of up to around 500 m/s, and the ion engine becoming a better choice for ΔV's larger than that. For low and even some medium ΔV requirements (up to about 300 m/s) where impulsive thrusting is possible, chemical systems, specifically a mini hybrid rocket and a micro-turbopumped bipropellant rocket are a the technologies of choice.

Future Work

There is much future work that can be done in this area. Most of the concepts discussed have yet to be demonstrated at the scales considered. The various technical hurdles discussed in the previous sections should be addressed, particularly the fabrication issues, including how MEMS techniques can be combined with traditional techniques to produce smaller thrusters cost-effectively. Power supply and conditioning equipment mass continue to make up a large portion of the mass of electric propulsion systems, and any success in making these devices more efficient and smaller would be extremely useful. Similarly, efforts to reduce the size or mass of valves and tanks should be pursued. Provided the current trends towards microspacecraft continue, there should be much exciting work in micro-propulsion systems to improve the performance and capabilities of the microspacecraft.

Appendix A

Orbital Mechanics Derivations

A.1 Introduction

This appendix presents the orbital mechanics derivations discussed in Chapter 3 in some additional detail.

A.2 Orbit Transfer

A.2.1 Low thrust orbit transfer[1]

The simplifying assumptions are that the thrust is sufficiently low that the orbit always approximates a circle, and that the thrust is constant and always applied along the velocity vector, which is tangential to the radial direction. Equating thrust power and rate of change of orbital energy gives:

$$\frac{d}{dt}\left(-\frac{\mu}{2r}\right) = \frac{F}{m}v = \frac{F}{m}\sqrt{\frac{\mu}{r}} \tag{A.1}$$

where F is the constant thrust; m the decreasing mass of the spacecraft; and r its orbital radius. F/m can be writen as a function of time, assuming a constant specific impulse:

$$\frac{F}{m} = \frac{F}{m_o - \dot{m}t} = \frac{F}{m_o - \frac{F}{c}t} = \frac{f_o}{1 - \frac{f_o}{c}t} \tag{A.2}$$

where f_o is the thrust divided by the initial mass, and c is the exhaust velocity.

1. This derivation is based on my personal notes from 16.512, Rocket Propulsion, as taught by Prof. Jack Kerrebrock, Fall 1995.

Substituting and then integrating the result, we have:

$$\sqrt{r}d\left(\frac{1}{r}\right) = \frac{-2f_o}{\sqrt{\mu}}\frac{dt}{1-\frac{f_o}{c}t} \tag{A.3}$$

$$\frac{r}{r_o} = \left[1 + c\sqrt{\frac{r_o}{\mu}}\ln\left(1-\frac{f_o}{c}t\right)\right]^{-2} \tag{A.4}$$

where r_o is the initial orbital radius.

It can be shown that the ΔV required to transfer to a specific radius is:

$$\Delta\upsilon = \frac{\Delta V}{V_o} = 1-\sqrt{\rho} \tag{A.5}$$

where ρ is defined as the ratio of initial to final radii.

A.3 Orbit Phasing

A.3.1 Impulsive Phasing

Recalling Figure 3.1, the phase change per revolution in the intermediate orbit can be written as:

$$\frac{\Delta\theta}{\tau_i} = \omega_o(P_o - P_i) \tag{A.6}$$

where $\Delta\theta$ is the total phase change angle; τ_i the number of orbits spent in the intermediate orbit; ω_o the angular velocity of the original orbit; and P_o and P_i the periods of the original and intermediate orbits, respectively.

Further non-dimensionalizing has:

$$\frac{\Delta\theta}{\tau}\left(\frac{P_i}{P_o}\right) = 2\pi\left(\frac{P_i}{P_o} - 1\right) \tag{A.7}$$

$$\Delta\theta = 2\pi\tau(1 - \rho^{3/2}) \tag{A.8}$$

where τ is now the time spent in the phasing manuever normalized by the period of the original orbit, and ρ ($= r_o/a_i$) is the ratio of the original radius to the intermediate semi-major axis.

For a desired total phase change angle in a given time, the necessary semimajor axis ratio can be seen to be:

$$\rho = \left(1 - \frac{\Delta\theta}{2\pi\tau}\right)^{2/3} \tag{A.9}$$

The change in velocity required to get to this new orbit is:

$$\Delta V = V_i - V_o = \sqrt{\mu\left(\frac{2}{r_o} - \frac{1}{a_i}\right)} - \sqrt{\frac{\mu}{r_o}} \tag{A.10}$$

$$\Delta V = \sqrt{\frac{\mu}{r_o}}[(\sqrt{2-\rho}) - 1] \tag{A.11}$$

Non-dimensionalizing, and remebering that the same ΔV must be performed to enter and to leave the intermediate orbit, we have:

$$\Delta\upsilon_{total} = 2\frac{\Delta V}{V_o} = 2[(\sqrt{2-\rho}) - 1] \tag{A.12}$$

A.3.2 Low Thrust Phasing

As in the orbit changing case (Section A.2.1 above), but with the added assumption of constant acceleration throughout the maneuver, we have:

$$r = r_o\left[1 - \sqrt{\frac{r_o}{\mu}}ft\right]^{-2} \tag{A.13}$$

where *ro* is the initial orbit radius; *f* is the thrust per unit mass of the spacecraft, and *r* is the radius of the spacecraft's orbit after a time *t*.

Since the change in angular momentum is equal to the applied torque, we have:

$$\frac{d}{dt}(r^2\omega) = fr \tag{A.14}$$

Separating variables, simplifying and substituting *dr*, which can be optained from Equation A.13, we have:

$$\frac{d\omega}{\omega} = \left(\frac{1}{r\omega} - 4\sqrt{\frac{r}{\mu}}\right)f\,dt = -3\sqrt{\frac{r}{\mu}}f\,dt \tag{A.15}$$

Noting that $r\omega = v$, and substituting r(t) from Equation A.13, we have:

$$\frac{d\omega}{\omega} = -3\sqrt{\frac{r_o}{\mu}}f\frac{dt}{1 - \sqrt{\frac{r_o}{\mu}}ft} \tag{A.16}$$

Integrating yields:

$$\ln\left(\frac{\omega}{\omega_o}\right) = 3\ln\left(1 - \sqrt{\frac{r_o}{\mu}}ft\right)$$

or

$$\omega = \omega_o\left(1 - \sqrt{\frac{r_o}{\mu}}ft\right)^3 \tag{A.17}$$

Integrating again yields:

$$\theta = \frac{\omega_o}{4f}\sqrt{\frac{r_o}{\mu}}\left[1 - \left(1 - \sqrt{\frac{r_o}{\mu}}ft\right)^4\right] \tag{A.18}$$

Non-dimensionalizing Equations A.13 and A.18 yield:

$$\theta = \frac{1}{4\bar{f}\rho_o^2}[1-(1-2\pi\rho_o^2\bar{f}\tau)^4] \tag{A.19}$$

$$\rho = (1-2\pi\rho_o^2\bar{f}\tau)^2 \tag{A.20}$$

where the non-dimensional quantities are defined as:

$$\bar{f} = \frac{f}{g} = f\frac{R_E^2}{\mu} \qquad \tau = \frac{t}{P_o} \qquad \rho_o = \frac{r_o}{R_E} \tag{A.21}$$

Solving Equation A.20 for τ and substituing that into Equation A.19 yields:

$$\theta = \frac{1}{4\bar{f}\rho_o^2}[1-\rho^2] \tag{A.22}$$

This is the total angular displacement that occurs when the orbit radius is changed under a constant low acceleration by a factor of ρ. Because there is no waiting at the intermediate orbit as in the impulsive case, τ and ρ are no longer independent, since the total time of the manuever is exactly twice the time required to transfer to the intermediate orbit. Instead, it is the acceleration that must be chosen to provide a given phase change in a given amount of time.

The net phase change that a low thrust transfer from an intial orbit to an intermediate orbit and back in a total time (normalized by original orbit period) of τ, is given by:

$$\Delta\theta = 2\pi\tau - \frac{1}{2\bar{f}\rho_o^2}\left[1-\left(1-2\pi\rho_o^2\bar{f}\frac{\tau}{2}\right)^4\right] \tag{A.23}$$

So for a given $\Delta\theta$ and a given τ, one can determine the required acceleration to complete the specified manuever in the specified time using Equation A.23. The effective ΔV required for the manuever is simply ft, which can be non-dimensionalized as:

$$\Delta\upsilon_{total} = \frac{\Delta V}{V_o} = \sqrt{\frac{r_o}{\mu}}ft = 2\pi\rho_o^2\bar{f}\tau = 2(1-\sqrt{\rho}) \tag{A.24}$$

A.4 Precision Station-Keeping

This section will discuss the idealized process of counteracting a constant disturbance force, such as that caused by solar pressure with a periodic restoring force provided by a propulsion system. Figure A.1 provides an illustration of the process being considered. The position as a function of time is labeled *x(t)*, and the velocity as a function of time is labeled *v(t)*. Each cycle lasts for a time Δt, and the correcting thrust is provided for some fraction ξ (the duty cycle) of that time.

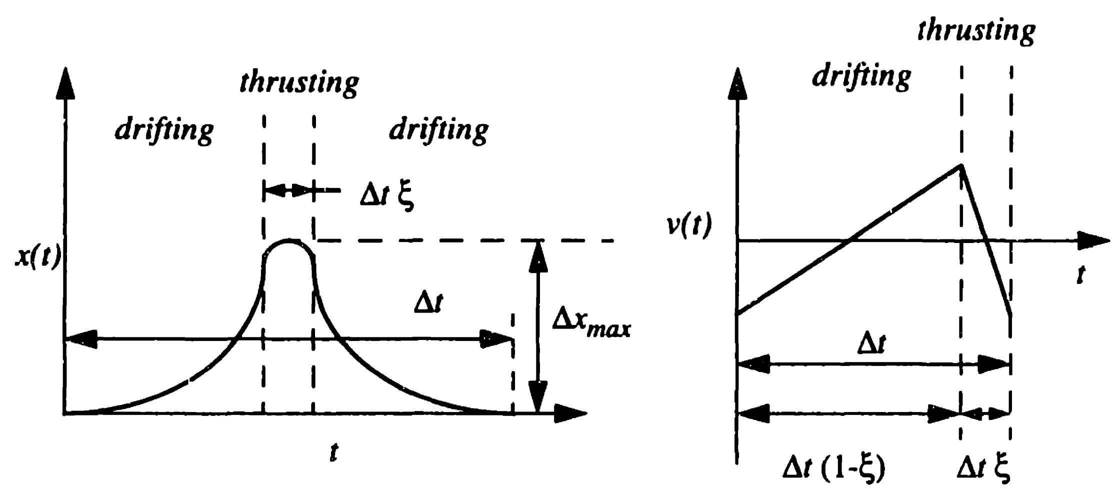

Figure A.1: Illustration of position and velocity during one correction cycie for a spacecraft undergoing a constant acceleration.

Since the objective is to keep the spacecraft within a position tolerace of Δx_{max} around the nominal position, it must be true that

$$a\Delta t(1-\xi) = (a_F - a)\Delta t\xi \tag{A.25}$$

where a is the disturbing acceleration and a_F the acceleration produced by the thruster, or that:

$$a_F = \frac{1}{\xi}a \tag{A.26}$$

Δx_{max} can be written as:

$$\Delta x_{max} = \frac{1}{2}a\left(\frac{\Delta t}{2}\right)^2(1-\xi)^2 + \frac{1}{2}(a_F - a)\left(\frac{\Delta t}{2}\right)^2\xi^2 \quad \text{(A.27)}$$

and remembering Equation A.26, this can be solved for Δt to show the required time between firings:

$$\Delta t = \sqrt{\frac{8\Delta x_{max}}{a(1-\xi)}} \quad \text{(A.28)}$$

Since propulsion requirements are expressed as ΔV's, the ΔV required per cycle is $a_F\Delta t\xi$, or:

$$\Delta V_{cycle} = a_F\Delta t\xi = \frac{a}{\xi}\xi\sqrt{\frac{8\Delta x_{max}}{a(1-\xi)}} = \sqrt{\frac{8\Delta x_{max}a}{(1-\xi)}} \quad \text{(A.29)}$$

The impulse required to produce this ΔV each cycle is:

$$I_{cylce} = m\Delta V_{cycle} = \sqrt{\frac{8\Delta x_{max}F_{dist}m}{(1-\xi)}} \quad \text{(A.30)}$$

where m is the mass of the spacecraft and F_{dist} is the disturbance force.

The total impusle required over the lifetime of the spacecraft can be see to be:

$$I_{life} = I_{cycle}\frac{L}{\Delta t} = maL = F_{dist}L \quad \text{(A.31)}$$

where L is the lifetime.

A.5 Precision Attitude Control

A similar analysis can be performed for a body undergoing a constant torque, with a correcting torque being periodically applied via a pair of thrusters opperating a distance D appart. This analysis is completely analogous to the one performed in the previous section, so only the results will be presented here.

The required thrust of each thruster is (analagous to the a_F):

$$F = \frac{T_{net}}{D\xi} \tag{A.32}$$

where T_{net} is the net disturbance torque experienced, and ξ is defined as before.

The time of each correction cycle is:

$$\Delta t = \sqrt{\frac{8\Delta\theta_{max} I}{T_{net}(1-\xi)}} \tag{A.33}$$

where I is the appropriate moment of intertia of the spacecraft.

The impulse required of each thruster during each cycle is:

$$I_{cycle} = \sqrt{\frac{8\Delta\theta_{max} T_{net} I}{D^2(1-\xi)}} \tag{A.34}$$

Finally, the total impulse required for the spacecraft's lifetime is:

$$I_{life} = 2\frac{T_{net}}{D}L \tag{A.35}$$

where L is the lifetime as above, and the factor of two arises from the fact that two thrusters are required to produce the couple.

Appendix B

Detailed Breakdowns of System Masses

B.1 Introduction

This appendix presents the detailed system mass breakdowns for each technology considered in each mission.

B.2 Separated Spacecraft Interferometry

Table B.1: NMI SSI Ion Thruster Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total		**340**	**340**
Thruster	12	100	1 200
Tank	1	40	40
Valve	12	10	120
Hardware Total			**1 360**
Power Supply	8	645	5 160
Power Conditioning	8	735	5 880
Power Total		**1 380**	**11 040**
System Total			**12 740**

Table B.2: NMI SSI Hall Thruster Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total		**630**	**630**
Thruster	12	40	480
Tank	1	70	70
Valve	12	10	120
Hardware Total			**670**
Power Supply	8	700	5 600
Power Conditioning	8	530	4 240
Power Total		**1 230**	**9 840**
System Total			**11 140**

Table B.3: NMI SSI PPT Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 020**
Thruster	12	21	252
Hardware Total			**252**
Power Supply	12	14	168
Power Conditioning	12	46	552
Power Total		**60**	**720**
System Total			**1 992**

Table B.4: NMI SSI FEEP Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**102**
Thruster	12	375	4 500
Hardware Total			**4 500**

Table B.4: NMI SSI FEEP Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Power Supply	8	18	144
Power Conditioning	8	1 000	8 000
Power Total		**1 018**	**8 144**
System Total			**12 746**

Table B.5: NMI SSI Cold Gas Mass Breakdown

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**16 300**
Thruster	12	negligible	
Tank	1	8 000	8 000
Valve	12	10	120
Hardware Total			**8 120**
System Total			**24 420**

B.3 Future Global Positioning System

Table B.6: Future GPS Ion Thruster Mass Breakdown (OCS[a])

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total		**680**	**680**
Thruster	5	235	1 175
Tank	1	75	75
Valve	5	10	50
Hardware Total			**1 300**
Power Supply	5	1 035	5 175
Power Conditioning	5	1 155	5 775
Power Total		**2 190**	**10 950**
System Total			**12 930**

a. Orbit Control System

Table B.7: Future GPS Hall Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total		**1 260**	**1 260**
Thruster	5	43	215
Tank	1	135	135
Valve	5	10	50
Hardware Total			**400**
Power Supply	5	750	3 750
Power Conditioning	5	565	2 825
Power Total		**1 315**	**6 575**
System Total			**8 235**

Table B.8: Future GPS Hydrazine Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**6 800**
Thruster	2	1 000	2 000
Tank	1	430	430
Valve	2	120	240
Hardware Total			**2 670**
System Total			**9 470**

Table B.9: Future GPS Hybrid Motor Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**5 140**
Thruster	1	170	170
Tank	1	750	750
Valve	1	185	185

Table B.9: Future GPS Hybrid Motor Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Hardware Total			**1 105**
System Total			**6 245**

Table B.10: Future GPS Micro-Biprop. Engine Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**4 800**
Thruster	2	5	10
Oxidizer Tank	1	475	475
Fuel Tank	1	100	100
Hardware Total			**585**
System Total			**5 385**

Table B.11: Future GPS PPT Mass Breakdown (ACS[a])

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**850**
Thruster	12	18	215
Hardware Total			**215**
Power Supply	12	12	145
Power Conditioning	12	46	550
Power Total		**58**	**695**
System Total			**1 760**

a. Attitude Control System

Table B.12: Future GPS FEEP Mass Breakdown (ACS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**85**
Thruster	12	375	4 500
Hardware Total			**4 500**
Power Supply	8	18	145
Power Conditioning	8	1 070	8 560
Power Total		**1 088**	**8 705**
System Total			**8 790**

Table B.13: Future GPS Cold Gas Mass Breakdown (ACS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**12 500**
Thruster	12	negligable	
Tank	1	6 300	6 300
Valve	12	10	120
Hardware Total			**6 420**
System Total			**18 920**

B.4 IR Earth Observing Cluster

Table B.14: EOC Ion Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 500**
Thruster	4	120	480
Tank	1	160	160
Valve	4	10	40

Table B.14: EOC Ion Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Hardware Total			**680**
Power Supply	1	715	715
Power Conditioning	2	805	1 610
Power Total		**1 520**	**2 325**
System Total			**4 505**

Table B.15: EOC Hall Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**2 750**
Thruster	4	40	160
Tank	1	295	295
Valve	4	10	40
Hardware Total			**495**
Power Supply	1	700	700
Power Conditioning	2	530	1 060
Power Total		**1 230**	**1 760**
System Total			**5 005**

Table B.16: EOC PPT Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**4 940**
Thruster	4	158	630
Hardware Total			**630**
Power Supply	2	93	185
Power Conditioning	2	83	165
Power Total		**175**	**350**

Table B.16: EOC PPT Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
System Total			**5 920**

Table B.17: EOC PPT Thruster Mass Breakdown (ACS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**205**
Thruster	12	18	210
Hardware Total			**210**
Power Supply	12	12	145
Power Conditioning	12	45	540
Power Total		**57**	**685**
System Total			**1 110**

B.5 MiniMars Mission

Table B.18: MiniMars Ion Thruster Mass Breakdown (15 month EET-LMO)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**25 000**
Thruster	2	1 500	3 000
Tank	1	2 700	2 700
Valve	2	10	20
Hardware Total			**5 720**
Power Supply	2	2 900	5 800
Power Conditioning	2	3 200	6 400
Power Total		**6 100**	**12 200**
System Total			**42 920**

Table B.19: MiniMars Hall Thruster Mass Breakdown (15 month EET-LMO)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**54 000**
Thruster	2	225	450
Tank	1	5 800	5 800
Valve	2	10	20
Hardware Total			**6 270**
Power Supply	2	2 450	4 900
Power Conditioning	2	1 755	3 510
Power Total		**4 205**	**8 410**
System Total			**68 680**

Table B.20: MiniMars Ion Thruster Mass Breakdown (33 month LEO-LMO)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**53 500**
Thruster	2	1 500	3 000
Tank	1	5 700	5 700
Valve	2	10	20
Hardware Total			**8 720**
Power Supply	2	2 900	5 800
Power Conditioning	2	3 200	6 400
Power Total		**6 100**	**12 200**
System Total			**74 420**

Table B.21: MiniMars Hall Thruster Mass Breakdown (33 month LEO-LMO)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**132 000**
Thruster	2	270	540
Tank	1	14 250	14 250
Valve	2	10	20
Hardware Total			**14 790**
Power Supply	2	2 800	5 600
Power Conditioning	2	1 950	3 900
Power Total		**4 750**	**9 500**
System Total			**156 290**

B.6 Next Generation LEO Communication System

Table B.22: LEO Comm. System Ion Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**180**
Thruster	3	100	300
Tank	1	35	20
Valve	3	10	30
Hardware Total			**350**
Power Supply	3	645	1 935
Power Conditioning	3	735	2 205
Power Total		**1 380**	**4 140**
System Total			**4 670**

Table B.23: LEO Comm. System Hall Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**335**
Thruster	2	40	80
Tank	1	65	65
Valve	2	10	20
Hardware Total			**165**
Power Supply	2	700	1 400
Power Conditioning	2	530	1 060
Power Total		**1 230**	**2 460**
System Total			**2 960**

Table B.24: LEO Comm. System PPT Mass Breakdown (ACS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**610**
Thruster	12	18	215
Hardware Total			**215**
Power Supply	12	12	145
Power Conditioning	12	45	540
Power Total		**57**	**685**
System Total			**1 510**

Table B.25: LEO Comm. System PPT Mass Breakdown (ACS + Drag)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**760**
Thruster	13	18	235
Hardware Total			**235**

Table B.25: LEO Comm. System PPT Mass Breakdown (ACS + Drag)

Item	Units	Unit Mass [g]	Total Mass [g]
Power Supply	13	12	155
Power Conditioning	13	45	585
Power Total		**57**	**740**
System Total			**2 500**

Table B.26: LEO Comm. System Hydrazine Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 220**
Thruster	1	600	600
Tank	1	85	85
Valve	1	70	70
Hardware Total			**755**
System Total			**1 975**

Table B.27: LEO Comm. System Hybrid Motor Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**920**
Thruster	1	35	35
Tank	1	135	135
Valve	1	40	40
Hardware Total			**210**
System Total			**1 130**

Table B.28: LEO Comm. System Micro-Biprop. Engine Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**860**
Thruster	1	5	5
Oxidizer Tank	1	150	150
Fuel Tank	1	30	30
Hardware Total			**185**
System Total			**1 045**

B.7 Low Altitude Earth Observation Satellite

Table B.29: Low Altitude EOS Ion Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**650**
Thruster	1	100	100
Tank	1	70	70
Valve	1	10	10
Hardware Total			**180**
Power Supply	1	645	645
Power Conditioning	1	735	735
Power Total		**1 380**	**1 380**
System Total			**2 210**

Table B.30: Low Altitude EOS Hall Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 200**
Thruster	1	40	40

Table B.30: Low Altitude EOS Hall Thruster Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Tank	1	130	130
Valve	1	10	10
Hardware Total			**180**
Power Supply	1	700	700
Power Conditioning	1	530	530
Power Total		**1 230**	**1 230**
System Total			**2 610**

Table B.31: Low Altitude EOS PPT Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 900**
Thruster	2	430	860
Hardware Total			**860**
Power Supply	2	230	460
Power Conditioning	2	150	300
Power Total		**380**	**760**
System Total			**3 980**

Table B.32: Low Altitude EOS PPT Mass Breakdown (Drag only)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**1 600**
Thruster	1	430	430
Hardware Total			**430**
Power Supply	1	230	230
Power Conditioning	1	150	150
Power Total		**380**	**380**

Table B.32: Low Altitude EOS PPT Mass Breakdown (Drag only)

Item	Units	Unit Mass [g]	Total Mass [g]
System Total			**2 410**

Table B.33: Low Altitude EOS Hydrazine Mass Breakdown (OCS)

Item	Units	Unit Mass [g]	Total Mass [g]
Propellant Total			**7 900**
Thruster	1	430	430
Tank	1	500	500
Valve	1	50	50
Hardware Total			**980**
System Total			**8 880**

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