A Systems Study on How to Dispose
of Fleets of Small Satellites

by

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A systems study was conducted to determine the most mass-efficient method of achieving spacecraft disposal by atmospheric reentry. The focus of the study was on disposal of constellations of small microsatellites in low earth orbit. The impact of constellations of spacecraft on the orbital debris population will be substantial in the future. There is currently a trend in the space community toward constellations of satellites involving increasingly larger numbers of spacecraft. Advantages of distributed constellations of microsatellites over traditional single-satellite deployments include increased performance, redundancy, and the possibility of reduced overall cost. MEMS and the concept of the silicon satellite make the development of microsatellites feasible. Policy, tracking limitations, storage orbits, and natural orbit decay are important considerations with regards to spacecraft disposal. Various chemical and electrical propulsive technologies are compared against tethers and ballistic coefficient-altering techniques to determine the most mass-efficient method for disposal of spacecraft by atmospheric reentry within one year. It was determined that tethers and ballistic-coefficient altering methods are indeed the most efficient method of achieving atmospheric reentry of spacecraft within a certain range of initial altitude and mass. Tethers are the most mass-efficient method of spacecraft disposal for large satellites while decreasing the ballistic coefficient by deploying a balloon around the spacecraft or deploying a drag parachute is the most mass-efficient method for small spacecraft with a low initial altitude. The development of more advanced propulsion systems will be necessary to deorbit small spacecraft at high initial altitudes.

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Chapter 1

Introduction

1.1 Background and Motivation

Since the space age began in 1957 with the launch of Sputnik, approximately 20,000 metric tons of material have been lifted into orbit. Today, approximately 4,500 tons remain in the form of nearly 10,000 “resident space objects.” Debris in orbit, whether man-made or natural, presents a concern to space endeavors. The concern is that while the number of natural objects in orbit remains relatively stable, the quantity of man-made objects is increasing all the time. Currently, within 2000 km of earth, man-made debris is already a greater threat than natural debris [1]. Vladamir Lebedev, a Russian scientist, predicts that at our current rate of pollution, the path to space will be blocked within 150 years [2]. The possibility of increasing rates of orbital debris in the future is an even greater concern.

To a civilization that is increasingly dependent on our space infrastructure, the issue of space debris is a serious concern. A collision with a piece of space debris could damage or destroy a spacecraft such as the International Space Station [3]. Such a disaster would be devastating to current NASA efforts. Not only are current missions at risk but future missions are
affected by space debris as well. Mission planners must plan a mission to prevent or abate space debris and satellites must be designed to meet the risk of collision with debris.

This thesis explores the issue of orbital debris and discusses current ideas on debris mitigation. The trend toward distributed architectures of small satellites and their impact on orbital debris is also covered. The majority of the thesis is dedicated to presenting some ideas on more efficient and effective methods for satellite disposal at the end of life. The ultimate solution to orbital debris lies in effectively purging spacecraft from useful orbits when their lifetimes are complete.

1.1.1 Orbital Debris

Orbital debris is defined as “any non-operational object in space around earth [4].” This refuse comes in many forms: dead spacecraft, discarded rocket bodies, launch- and mission-related castoffs, remnants of satellite breakups, solid rocket exhaust, frayed surface materials, and even droplets from leaking nuclear reactors. Orbital debris also includes paint chips, explosive bolts, clamp bands, and springs. The most famous piece of orbital debris is probably Ed White’s lost glove. It is estimated that the glove reentered the atmosphere and burned up in approximately one month, but it was potentially the most lethal article of clothing ever produced. Of the more than 4800 spacecraft launched into orbit, 2400 remain. 75% of the remaining spacecraft are abandoned. These objects range in mass from one kg to approximately 20 tons [5]. U.S. Space Command currently tracks 8,782 objects orbiting earth. The breakdown of these objects is as follows [2]:

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Debris 6,079
Satellites 2,613
Space Probes 90

The nationality of these objects is also known:

U.S. origin 3,838
Russian origin 3,955

U.S. Space Command currently has the capability to track objects with a diameter greater than approximately 10 cm in low earth orbit (LEO). At geosynchronous orbit (GEO), an object must have a diameter of approximately one meter to be tracked. On the other hand, researchers estimate there are more than 100,000 objects between 1 and 10 cm in Earth orbit along with tens of millions of objects between one mm and one cm [6]. By far the greatest source of space debris larger than 0.1 mm is the breakup of satellites and rockets. Satellites and rocket bodies break up primarily due to the unintentional release of stored energy. Collisions will almost certainly be a significant source of break-up debris in the future. The causes for on-orbit fragmentation are presented in Figure 1.1 [4].
Space debris is not only dangerous because of the number of objects in orbit but because of the relative velocities of these objects. The natural particles that compose the rings of Saturn orbit the planet with small relative velocities. The small relative velocities are the result of these natural particles being formed by similar processes as the planet formed. Impacts between these particles are relatively sedate with little transferred energy. If the rings of Saturn resemble a school of fish swimming together, Earth’s artificial satellites resemble angry bees around a beehive, seeming to move randomly in all directions. The relative velocities are often huge.

So far, there has only been one confirmed collision between two tracked objects. In 1994, a piece of orbital debris from a French launch vehicle hit France’s Cerise satellite. The collision resulted in the severing of the gravity gradient boom. Collisions between spacecraft and untrackable objects are more widespread. A shuttle window was damaged by what was most likely a paint chip. In 1993, an Olympus communication satellite was lost after an
electronics failure that was attributed to a micrometeoroid impact [6]. It is almost certainly only a matter of time before collisions and orbital debris damage becomes more widespread. In LEO, the chances of collision between two tracked objects is greater than 50% in 10-15 years in the 800-1000 km altitude range. At GEO there is a 1 in 500 chance per year that a satellite will be struck by an uncontrolled, tracked object. These chances will increase by a factor of 1.5 in the next 10 years and by a factor of 3 over the next 30 years if current launching levels are maintained [6]. The probability of striking a piece of debris is given by the following equation:

\[ P_c = 1 - e^{-SPD \cdot AC \cdot T \cdot VREL} \]  

(1.1)

where SPD is the spatial density of debris objects (objects/m²), AC is the collisional cross-sectional area (m²), T is the mission duration (s), and VREL is the relative velocity (m/s) between satellite and debris population. Debris can also affect radio frequency interference, sensor entry/exit events, and radio frequency power impingement [4].

The loss of spacecraft to orbital debris could prove to be crippling to current space programs and society as a whole. Consider the recent loss of PanAmSat's Galaxy 4 spacecraft, a communication satellite. This loss cost paging service providers more than $6 million in rebates for down time, not to mention the cost of replacing the satellite and the inconvenience to physicians and other pager users [6]. Considerable effort has been invested in collision mitigation for the International Space Station (ISS). ISS is the size of two football fields and is thus at a significantly higher risk of collision than any other spacecraft that has been
constructed. Meteoroid and debris impact is currently ranked as one of the top 15 risks to the ISS program. The plan to protect ISS from impact is two-faceted: 1) shield ISS against impacts with particles up to 1 cm in diameter, and 2) move ISS out of the path of debris too large to shield against [3].

One concept that is being considered to protect spacecraft from orbital debris is space traffic control. NASA has requested that the Air Force track objects down to one centimeter in size in LEO. A recent Air Force report estimates the cost of the necessary upgrades at more than $400 million [6].

Currently, U.S. Space Command maintains the premier catalogue of orbiting objects. This catalogue is known as the resident space object (RSO) catalogue. The accuracy of the RSO depends on several factors. Time since last tracking measurement, atmospheric density variations, and any maneuvers or burns that may have occurred will all affect the accuracy of the RSO catalogue. High accuracy is needed to avoid unnecessary, fuel-consuming maneuvers. For example, a recent move of an operational satellite to avoid a potential collision cost one year’s worth of station keeping propellant.

The space debris concern is not limited to the debris currently in orbit. One current projection estimates that there will be 2,000 new objects launched into space in the next 10 years. These objects will be primarily privately-owned commercial communication satellites. Operational debris and on-orbit explosions are expected to increase the debris population to ever-larger numbers.
1.1.2 Orbital Debris Mitigation

Minimizing orbital debris has been U.S. national policy since February 1988. The current National Space Policy, signed in 1996, directs U.S. government agencies to minimize or reduce accumulation of orbital debris and also recognizes the necessity of such practices by the international community. The first government and industry workshop on orbital debris mitigation was held in January 1998 [7].

Orbital debris first appeared in the National Space Policy of 5 January, 1988 in which it is stated, “All space sectors will seek to minimize the creation of space debris.” In 16 November 1989, the following was added, “the U.S. government will encourage other space faring nations to adopt policies aimed at debris minimization [7].”

NASA has evaluated the risks posed by orbital debris since the early days of Gemini leading to a research program at Johnson Space Center. In 1981, a requirement was made for Delta second stages to be depleted of residual propellants at the end of mission. A requirement to conduct a formal assessment in accordance with NASA safety standards on each NASA program or project with respect to debris generation potential was made in 1997. This requirement stipulates that two reports are completed: one at the Preliminary Design Review and the other 45 days prior to the Critical Design Review [7].

The Department of Defense (DoD) was the first agency to adopt a formal policy on orbital debris. This policy stated in 4 February, 1987 that the department would seek to minimize the impact of space debris on its military operations. On 9 July, 1999, the DoD issued a
policy that stated, "Spacecraft disposal at end of life shall be planned for in programs involving on-orbit operations. Spacecraft disposal shall be accomplished by atmospheric reentry, direct retrieval, or maneuver to a storage orbit to minimize impact on future spacecraft [7]."

The U.S. Space Command has also released guidelines on orbital debris mitigation. Their stipulations state that LEO spacecraft should be designed with the goal to make an end of life maneuver to a disposal orbit from which natural decay will occur within 25 years. Spacecraft above 1600 km in altitude should be boosted to higher storage orbits. Semi-synchronous orbits (e.g. GPS) are to be boosted 500 km above operational altitudes while GEO spacecraft are to be maneuvered into high storage orbits 300 km above the nominal GEO altitude. The objective of these high altitude disposal orbits is a benign orbit with a lifetime of at least 1,000 years in which time it is assumed that mankind will devise more effective methods of disposal. In a recent survey of retired GEO spacecraft from 1997-1998, only one-fourth were placed in satisfactory disposal orbits [7].

Other emerging policies might require tanks to be vented and batteries discharged once placed in a disposal orbit. Satellite operators have not objected to these policies, but they want assurance that competitors are held to the same standards. One big issue involves determining how international operators will be held accountable.

Natural orbit decay exerts the primary cleansing force on orbital debris. This force is due to atmospheric drag and is discernable as high as 1500 km. For altitudes much above 1500 km,
the effect is negligible. It is apparent that natural orbital decay is insufficient in itself to solve the problem of orbital debris. Other methods must be developed and implemented to alleviate the problem.

1.1.3 Impact of Distributed Microsatellites on Orbital Debris

A concern mentioned earlier is the development of increasingly small satellites used in formation flying and distributed architectures. Distributed constellations are satellite constellations in which multiple satellites are used to accomplish a mission. Distributed constellations are more desirable than conventional single-satellite deployments if the distributed constellation provides improved performance or reduced cost. Some communications, sensing, and navigation missions have been developed that can only be accomplished with a distributed constellation. Such an architecture is being considered in the Space Systems Laboratory at MIT. The mission is called TechSat 21 and will be an Air Force radar mission. Between 40 and 280 total satellites may be utilized in the final architecture. Another mission that can only be accomplished with the use of a distributed array of satellites is a mission proposed by Goddard Space Flight Center. The mission would involve on the order of one hundred nanosatellites to get simultaneous measurements of the earth’s magnetic field [8]. Another idea is to use an array of nanosatellites to build a sparse aperture antenna. It is not currently possible to build a km-diameter satellite antenna any other way. The quality of the antenna pattern in terms of the main beam gain divided by the average side lobe gain is proportional to 1/N where N is the number of nanosatellite receiver elements. With N=1000, you can still get a 30-dB ratio which is adequate for many applications [9].
Using a distributed architecture may also reduce cost in some mission architectures by taking advantage of the redundancy built into a distributed architecture and accepting lowered requirements for individual satellite reliability [10]. Smaller satellite size is enabled by the continuing trend toward miniaturization in electronics and the development of micro-electromechanical systems (MEMS). The Aerospace Corporation has developed the concept of the “silicon satellite”. Silicon can simultaneously serve as electronics substrate, mechanical structure, heat sink, power generator, optic substrate, and radiation shield. The silicon satellite concept also enables a move toward increasingly small satellites. Possible future missions for small silicon satellites could be as follows: global communication or observation missions that use hundreds of nanosatellites in LEO to provide continuous earth missions and local cluster missions that utilize hundreds of nanosatellites in a sparse array configuration. A long-term future can be envisioned with thousands or millions of micro (1-100 kg), nano (1-1000 g), and pico (1-1000 mg) satellites in orbit around earth [11]. Obviously, without the implementation of effective satellite removal methods, the orbital debris situation will soon become crippling to space infrastructure.

NASA’s Johnson Space Center has done a lot of work in modeling the orbital debris situation as it currently exists and predicting how that might change in the future. Their model, “Evolve”, is one of several models used to predict the future proliferation of orbital debris [12]. A pair of plots presented here paints a grim picture of the current and future state of orbital debris. Figure 1.2 shows a plot of adjusted critical density and spatial density versus altitude [12].
Figure 1.2: Adjusted Critical Density and Spatial Density versus Altitude

Critical density is the point at which a region contains enough objects with sufficient mass that the rate of fragment production from collisions is greater than the rate at which objects are removed by natural processes. Once this critical density is reached, fragments from collisions will cause an ever-increasing number of new collisions. This is referred to as a "cascading" or "runaway" effect or as a "chain reaction" (although the time frame is typically on the order of decades or centuries). Once collisional cascading has begun, it cannot be stopped by a reduction in launch rate because it is self-sustaining. The collisional hazard in that orbital region may be too high for space operations for centuries to come. Adjusted critical density in figure 1.2 includes atmospheric drag as well as the size and inclination distribution of the current debris population. Altitudes where the spatial density of the debris population is greater than the adjusted critical density in figure 1.2 indicate a region where the debris population will increase exponentially in the future even if no additional objects are added in those altitude ranges in the future. The 900-km and 1500-km altitude regions of space already qualify as areas of exponential debris increase in the future. These regions,
even without further deployment in the future, will see a runaway effect in the debris population.

Figure 1.3 projects the debris population in LEO into the future based on five scenarios [12]:

- **"Business as usual"**
- Cutting the number of explosions of rocket bodies on orbit in half
- Completely preventing all future explosions of rocket bodies on orbit
- Preventing all future explosions as well as removing all future payloads and spacecraft from orbit after their mission is completed
- Doing the above as well as removing approximately 3000 old rocket bodies and payloads

The following figure shows these scenarios in order from top to bottom.

![Figure 1.3: Projection of Orbital Debris Population in the Future](image)

**Figure 1.3: Projection of Orbital Debris Population in the Future**
The top three curves indicate the beginning of an exponential increase (or runaway) of orbital debris in the future. The fourth curve, although not an exponential increase, still shows a gradual increase in the debris population in the future. The only curve that provides for a similar orbital debris population in 50 years as today is the bottom one which involves removing objects already in orbit as well as any objects deployed in the future.

Using the probability of collision equation introduced earlier, it is possible to predict the future proliferation of orbital debris if large numbers of distributed micro-satellites are launched into orbit and are not disposed of at the end of mission life. Figure 1.4 displays the collisions per year of tracked objects versus time for a future in which 1000 satellites are deployed per year and a future in which 10,000 satellites are deployed per year.

![Number of Collisions per Year in LEO](image)

*Figure 1.4: Number of Collisions per Year Based on Deployment Rate*
The plot considers spacecraft in the altitude range from 200 km to 2000 km. It also starts by considering only objects with diameters greater than 10 cm (and thus are trackable). An average cross-sectional area of 1 m² was used for each object for ease of calculation. Each collision is assumed to occur between two tracked objects and to result in a total of 4 objects from the 2 that collided (very conservative assumptions).

Even though the number of collisions seems small initially in figure 1.4, the plot indicates continuous increases in the number of collisions between tracked objects in the future. With a deployment rate of 10,000 satellites per year into LEO, there will be almost 1000 collisions between tracked objects per year within 100 years. The collision rate of satellites assuming 1000 deployments per year develops at a slower rate but becomes extreme nonetheless in the distant future.

A future in which 1000 to 10,000 (and possibly more) microsatellites are deployed every year may not be unlikely. It may seem farfetched now, but systems utilizing larger constellations are being developed. TechSat 21, an Air Force radar mission that will be discussed in more detail later, has been modeled at MIT. For a case using small 60-kg satellites in formation-flying clusters of 16 satellites each, the total constellation size is optimized at just over 250 total satellites. The metric used in producing figure 1.5 is total cost of the mission over a 10-year lifetime versus percentage of time that a satellite is in view for any given location on earth. This means that you get the most for your money by using a constellation with approximately 250 total satellites. As technology continues to improve, constellation sizes will probably be optimal at increasingly larger numbers of satellites.
Figure 1.5: Optimization of TechSat 21 Constellation Size

Figure 1.4 assumes an even distribution of satellites in all low earth orbit (LEO) altitudes between 200 km and 2000 km. An even distribution is unlikely in reality. More likely would be a higher concentration of satellites in certain optimal altitudes for various mission types. In these more highly concentrated altitudes, the collision rate will be much more frequent. Figure 1.6 is a plot in which 1000 or 10,000 satellites are deployed per year into a 200 km band in LEO (between 800-1000 km for instance).

Figure 1.6: Number of Collisions per Year in a 200 km band
In this plot, for 10,000 deployments per year, there will be more than 10 collisions per year within the first 10 years, more than 1000 collisions per year after 100 years, and more than 100,000 collisions per year after 1000 years. Once again, the collision frequency for 1000 deployments per year develops at a slower, but still significant, rate.

Two factors are not accounted for in the previous plots. First, there will be a constant purging of satellites due to natural orbital decay. Although natural orbital decay will be the over-riding factor in very low orbits (below 300 km), it will not be effective much above 600 km. The other factor is the fragmentation of satellites on impact with other spacecraft and pieces of space debris. Depending on the materials involved, it may be possible for two colliding spacecraft to disintegrate into scores or hundreds of pieces on impact with other objects at their high orbital velocities. Some of the resulting pieces may be trackable while others may not. As more collisions occur, more space debris will be added to the orbiting minefield, and these pieces of debris will cause more collisions again, and the process will continue. An assumption made that may counter-act the low fragmentation rate used in figures 1.4 and 1.6 is the fact that a cross-sectional area of 1 m² was used. If constellations involving thousands of satellites become reality, they will undoubtedly be much smaller than 1 m² in cross-sectional area.

Not all authorities are in agreement about the future impact that orbital debris will have. The National Research Council published “Orbital Debris: A Technical Assessment” as a result of investigation into the issue [12]. A conclusion of the work is that more research must be done to determine the degree to which orbital debris will affect future space missions. At
one point the authors say, “Some studies show that untrackable populations have no detectable effect on the evolution of the future LEO debris population” and go on to point out that “Models of collisions are based on only two in-space collisions and a few ground tests.” The authors also state that “If the only additions to the future debris population were rocket bodies, nonfunctional spacecraft, mission-related debris, and the products of explosions and surface deterioration, the space object population would likely continue its roughly linear growth.” The authors also point out that “Collisions between space objects, however, threaten to add a potentially large and exponentially growing number of new objects to this population. Several models of the future debris population suggest that collisionsal cascading is likely to occur in Earth orbit.” But in making their recommendation that more research must be done, the authors write “Because of the numerous uncertainties involved in models of the debris environment, it is premature to suggest exactly when collisional growth will begin to occur, it may already be under way, or it may not begin for several decades.”

1.2 Outline

The purpose of writing this thesis is not only to explore the current orbital debris problem, but also to go into more detail about the advantages of distributed architectures of small satellites. In the next chapter, the advantages of distribution are discussed. The role played by MEMS and microelectronics in enabling increasingly smaller satellites is also covered.

The bulk of the thesis is reserved for a discussion of how the orbital debris problem can be effectively mitigated in the future. It is only by designing spacecraft with inherent means of disposal at the end of their mission lifetime that the orbital debris problem can be solved.
The spacecraft disposal portion of the thesis discusses all current methods of satellite disposal as well as policy, tracking limitations, and the use of storage orbits. Future and unexplored technologies are considered as well. Finally, these technologies and methods are compared to determine the most effective disposal method based on a satellite’s mass and altitude at the end of mission life. The method of comparison, which will be discussed in more detail throughout the rest of the thesis, involves determining what method can deorbit a satellite within one year from a given altitude with the least amount of added mass to the spacecraft. One year is used as the time basis because it is believed that deorbit times of one year will be more reasonable in a future of increased satellite deployment as opposed to the 25-year time horizon allowed now. Since mass is such a significant factor in the cost of launching a spacecraft, it is used as the primary attribute of comparison.
Chapter 2

Advantages of Distributed Microsatellites

2.1 Distribution

The term “distributed satellite systems” refers to the coordinated operation of numerous satellites in performing a specific function. This definition is broad and encompasses a wide range of applications in all aerospace sectors. The advantages offered by distributed satellite systems include improvements in performance, cost, and survivability compared with single-satellite deployments. These improvements make the implementation of these systems attractive, and almost certainly, inevitable. With distributed satellite systems, the vision of what can be achieved in space is no longer bound by what an individual satellite can accomplish. Rather, the functionality is spread over a number of cooperating satellites. This functionality enables and expands the utility of small satellites and microsatellites (which will be discussed in the next section) [13].

In the near future, the world will see the continued development of four types of space-based systems that will be available on a worldwide basis:

- Global positioning (GPS) and navigation services
• Global communication services
• Information transfer services
• Global reconnaissance services

These four mission types can only be accomplished with distributed constellations of satellites. For instance, the current GPS constellation includes 24 satellites while the Iridium global communication system involves 66 satellites. Future, improved versions may include significantly more spacecraft. There are other potential future missions that can only be accomplished with distributed constellations of satellites. An example mission is a constellation of satellites used for continuous measurement of the earth’s magnetosphere. This mission could easily involve thousands (or perhaps millions) of small satellites.

The move toward distribution in satellite systems is analogous to the paradigm shift in the scientific computing market. This market started with large mainframe computers but moved to powerful workstations because more functionality was offered for the price. Now, the market is moving to distributed sets of workstations to handle larger problems that cannot be attacked by a single workstation.

A distributed satellite system can have two different definitions:

• A system of many satellites that are distributed in space to satisfy global demand.
• A system of satellites that gives multifold coverage of target regions.

The most important trait of distributed satellite systems and the characteristic common to both definitions is the use of more than one satellite to satisfy a global demand.
All current and envisioned satellite applications involve providing some kind of service in communications, sensing, and navigation. The common theme in these missions is that the satellite system must essentially perform the task of collection and dissemination of information. With this idea in mind it is possible to develop parameters to compare satellite systems. In order for a distributed architecture to make sense, it must offer either reduced cost or improved performance compared with traditional single deployments. There are four parameters that can be used to compare satellites from a performance standpoint:

- **Signal isolation** measures the system’s ability to isolate signals from different sources.
- **Information rate** is the rate at which information is transferred through the system.
- **Information integrity** is the error performance of the system.
- **Information availability** is a measure of the probability that the information is being transferred between the correct origin-destination pairs at the correct rate and with the desired integrity.

These four parameters characterize the performance of a satellite system. The system cost is the total resource expenditure required to build, launch, and operate the satellite system over the system’s lifetime. This cost includes the baseline development cost, construction, launch, and operating cost, as well as the expected costs of failure. The expected costs arise from the probability that failures will occur that could compromise the mission. All of the reasons supporting the use of distribution relate in some way to improving the performance characteristics or to reducing the baseline or failure compensation costs. The following subsections detail these reasons.
2.1.1 Signal Isolation Improvements

By separating resources spatially over a large area, the geometry of the signal collection is different for each detector. This geometry can assist in the separation of the different signals due to field of view (FOV) changes, different times of flight, or different frequency or phase of the received signals.

This improvement can be realized in a spacecraft array. The angular resolution, $\theta_r$ (arcseconds), of any aperture scales with the overall aperture dimension, expressed in wavelength, $\lambda$ (m):

$$\theta_r = \frac{\lambda}{D}$$

where $D$ (m) is the size of the aperture. The concept of the spacecraft array involves forming a large aperture (large $D$) from a set of satellites with each satellite acting as a single radiator element. The signal to noise ratio (SNR) of a sparse array is identical to a filled aperture of the same physical area. That is, a sparse array of $N$ elements, each of area $A$, will achieve the same SNR as a filled aperture of area $NA$. However, the resolution of sparse arrays can be much larger than that of an equivalent filled aperture. This resolution is the result of the enlarged overall aperture dimension that results from splitting and separating the aperture into elements.
2.1.2 Information Rate Improvements

There are several applications for which even the largest, highest-power satellite buses available today are too small. For instance, an Air Force performance requirement for a space-based radar system to replace the aging Airborne Warning and Control System (AWACS) aircraft forced designs featuring massive satellites beyond the capabilities of today’s technology (30 kW of RF power and 12,300 m² of aperture) [13]. The solution to these requirements for high information rates lies in multiplying the capabilities of several smaller satellites, such that their combined operation satisfies the overall mission requirements. A distributed version of the space-based radar satisfies the mean-time-to-detection requirement using a group of small satellites working in collaboration. For these missions, a mean time to detection of 10 s is considered reasonable. In a distributed architecture, several satellites search the same area independently, each with a mean detection time that is actually longer than 10 s. The cumulative detection rate (detections/time) of the whole group is the sum of the rates from each satellite:

\[
R_s = \sum_{sats} R_{\text{sat}}
\]  

(2-2)

Where \( R_{\text{sat}} \) is the detection rate for each satellite. The combined performance from several lower-performance components can therefore satisfy the desired mean-time-to-detection requirement for the system.
2.1.3 Information Integrity Improvements

The error performance of data collection and transfer systems is a critical issue in their design and operation. The probability of error for a single measurement is the likelihood that the interfering noise power exceeds some threshold. The noise can arise from several sources:

- Thermal noise from resistive heating of electrical components in the receiver
- Noisy radiation sources in the FOV of the instrument
- Jamming from unfriendly systems
- Interaction with the transmission medium (rain, bulk scatters)
- Background clutter

The random statistics of these noise sources is the basic reason supporting the use of distribution to improve the error performance. Errors made in the interpretation of signals are likely isolated to specific detectors or satellites. So, the overall integrity of the compounded information from several instruments is improved.

2.1.4 Information Availability Improvements

If carefully designed, a distributed architecture can often lead to improved performance of the satellite system by increasing the availability of system operations. A system is defined to be unavailable if it cannot collect and disseminate information between known and identified source/sink pairs at the required rate and integrity. Losses of availability can result from component failures, from signal attenuation due to blockage or weather, or from random performance fluctuations. There are several methods by which distributed
architectures can lead to increased availability through reductions in the variance of performance:

- Improved coverage of the demand
- Reduced impact of component failures

Clearly, a distributed architecture is the only option for applications requiring simultaneous sampling at all demand locations. For instance, a single satellite cannot serve the entire globe in a mobile communication system. Another example is the separated spacecraft interferometer. Optical interferometers collect light at widely separated apertures and direct this light to a central combining location, where the two light beams are interfered. Fringes produced by the interference provide magnitude and phase information from which a synthesized image can be generated. The collector spacecraft sample the distant starlight at several different baselines (separation and orientation) in order to construct the image. The locations of the sampling points define a distributed demand.

Availability considerations were instrumental in the design of the current GPS configuration. Important performance objectives that impacted the design of the GPS system are as follows:

- High accuracy, real-time position, velocity, and time for military users on a variety of platforms, some of which have high dynamics, e.g. high-performance aircraft. “High accuracy” implies 10-m three-dimensional root-mean-square (rms) position accuracy or better. The velocity accuracy requires errors to be less than 0.1 m/s.
- Good accuracy to civilian users. The objective for civil user position accuracy is 100 m or better in three dimensions.
- Worldwide all-weather operation, 24 h a day.
- Resistance to intentional (jamming) or unintentional interference for all users, with enhanced jamming resistance for military users.
- Affordable, reliable user equipment. This eliminates the possibility of requiring high-accuracy clocks or directional antennas on user equipment.

These requirements effectively shaped the architecture of the GPS system into a distributed architecture with 24 satellites in 12-h orbits.

2.1.5 Reducing the Cost

The advent of small satellite technology (which will be discussed in section 2.2) has ushered in a new era of satellite engineering that minimizes costs by risk management rather than risk avoidance. Should distributed satellites systems really proliferate, they will achieve low costs by lowering the requirements on individual satellite reliability, taking advantage of the redundancy built into the architecture. Distributed systems also offer the possibility of being able to ramp up the investment gradually, in order to match the development of the market. Only those satellites needed to satisfy the early market are initially deployed. If and when additional demand develops, the constellation can be augmented. The cost of constructing and launching these additional satellites is incurred later in the system lifetime. Because of the time value of money, the delayed expenditure can result in significant cost savings.

A design study at MIT showed that distributed systems appear to yield the greatest cost savings under two conditions:
• When the components being distributed make up a large fraction of the system cost. It is prudent to distribute the highest cost components among many satellites. As the old saying goes, “don’t put all your eggs in one basket!”

• When the component being distributed drives the replacement schedule of the spacecraft within the system.

Savings are realized in the following ways:

• Replacements on average occur later, resulting in larger savings from discounting to constant dollars.

• There are fewer replacements of overall components.

• The cost of replacing a single module in a distributed system is much less than that associated with the replacement of the entire satellite in a traditional design.

### 2.1.6 Distributed Satellite Conclusions

Distributed architectures are enabling for small satellite designs by expanding their useful range of applications to include high rate and resolution sensing and communications. The capabilities of many small satellites are combined to satisfy mission requirements. Some major advantages of distributed constellations are as follows:

• Improved resolution, corresponding to the large baselines that are possible with widely separated antennas on separate spacecraft within a cluster.

• Higher net rate of information transfer, achieved by combining the capacities of several satellites in order to satisfy the local and global demand.

• Improved availability through redundancy and path diversity. Frequently, the cost of adding a given level of redundancy is less for a distributed architecture.
• Improved availability through a reduced variance in the coverage of target regions. This variance reduces the need to “over design” and provides more opportunities for a favorable viewing geometry.

• Lower failure compensation costs, due to the separation of important system components among many satellites; only components that break need replacement.

And perhaps the most important advantage of distributed satellite systems is that they permit new missions that cannot be done with traditional single-satellite deployments.

2.2 Microsatellites

The term “microsatellite” or “microspacecraft” in this thesis is typically intended to mean any small satellite. As has been mentioned before, different categories of small satellites have already been coined based on mass. These categories include micro (1-100 kg), nano (1-1000 g), and pico (1-1000 mg) satellites. Unless stated differently, in this thesis, “microsatellite” or “microspacecraft” means any satellite with a mass less than 100 kg.

The trend toward clusters of microsatellites is highlighted in a recent September 2000 article in Smithsonian Air and Space magazine [14]. The article, titled “Swarm: The newest satellites are small, they’re clever, and they travel in packs”, is about the efforts of the Aerospace Corporation’s Center for Microtechnology. This center works on the implementation of nanotechnologies and MEMS into spacecraft design. The center was founded in 1998 by Ernie Robinson, Henry Helvajian, and Siegfried Janson.
An upcoming test of microsatellites will be part of the MightlySat 2.1 mission. This $15 million experimental Air Force satellite will be about the size and shape of a microwave oven and will weigh approximately 120 kg. While this satellite itself is small, it is one of MightySat’s payloads that are truly lilliputian. Mighty Sat 2.1 will carry two smaller passengers: picosats. These picosats will weigh 10 ounces apiece and will be the smallest operational satellites ever sent into orbit. After deployment, these satellites will transmit and receive signals from earth and to each other while being tethered together by 30 meters of polyethylene and gold line.

The Defense Advanced Research Projects Agency (DARPA) contributed $535,000 to the Aerospace Corporation for the picosat program. And the Clinton administration plans to spend $495 million (an $83 million spending jump) on a host of nanotechnologies in 2001. NASA’s plans for microsatellites are aimed toward sending a swarm of 50 to 100 satellites, each weighing less than 10 kg, to monitor Earth’s magnetotail. The constellation of microsatellites could warn other satellites of solar storms and dangerous swirls of electrified plasma in Earth’s wake. Scientists hope to launch those spacecraft in 2008 or 2010 under a project called Magnetosail Constellation DRACO.

Eventually, constellations of microsatellites could save billions of dollars in launch costs by replacing some hugely expensive military, reconnaissance, and commercial communications satellites. Currently, launching a kg into orbit costs more than $20,000 (or $10,000 per pound). As Siegfried Janson says on this topic, “If you buy a computer at the store and it costs $10,000 a pound to bring it home, you will end up buying something like a Palm Pilot

Jason Andringa

MIT Space Systems Laboratory
or smaller. That’s what we’re working on. We’re trying to deliver the satellite equivalent of the Palm Pilot.”

2.2.1 MEMS

At the heart of the push toward smaller, microsatellites, is the evolution of microelectromechanical systems (MEMS) into commercial off-the-shelf components (COTS). The drivers, government investment and strong market forces, have historically focused on terrestrial applications, but applications to aerospace abound. MEMS offers a capability for mass-producing small, reliable, intelligent instruments at reduced cost by reducing the number of piece-parts, eliminating manual-assembly steps, and controlling material variability. These features, along with reduced mass and power requirements, are what make aerospace engineers eager to apply these technologies to space systems.

The Mars Pathfinder mission successfully demonstrated what could be accomplished with microspacecraft that used many COTS components. The “New Millennium” and “X-2000” efforts will utilize more MEMS technology to ultimately produce 10-kg-class interplanetary spacecraft.

Japan has made strong efforts in nanotechnology. The Japanese Government has identified micromachine technology as a cornerstone technology for the 21st century. As an example, the Toyota/Nippon Denso microcar is as tiny as a long-grain rice (7mm). The microcar is a .0001 scale replica of the Toyota Motor Corporation’s first automobile, the 1936 Model AA sedan. The minute vehicle has 24 parts, including tires, wheels, axles, headlights and
taillights, and hubcaps that carry the company name. Made of five parts, the electromagnetic motor is only 1 mm in diameter and can propel the car at speeds of up to 10 cm/s [15]. For space systems, there are applications for micromachines, micromotors, and microrobots.

Following is a list of the MEMS technology that will be useful in the near term for space systems [15]:

- **Command and Control Systems**
  - “MEMtronics” for ultraradiation hard and temperature-intensive digital logic
  - On-chip thermal switches for latchup isolation and reset

- **Inertial Guidance Systems**
  - Microgyros (rate sensors)
  - Microaccelerometers
  - Micromirrors and microoptics for FOGs (fiber-optic gyros)

- **Attitude determination and control systems**
  - Micromachined sun and Earth sensors
  - Micromachined magnetometers
  - Microthrusters

- **Power systems**
  - MEMtronic blocking diodes
  - MEMtronic switches for active solar cell array reconfiguration
  - Microthermoelectric generators

- **Propulsion systems**
  - Micromachined pressure sensors
- Micromachined chemical sensors (leak detection)
- Arrays of single-shot thrusters ("digital propulsion")
- Continuous microthrusters (cold gas, combustible solid, resistojet, and ion engine)
- Pulsed microthrusters (charged droplet, water electrolysis, and pulsed plasma)

- Thermal control systems
  - Micro heat pipes
  - Microradiators
  - Thermal switches

- Communications and radar systems
  - Very high-bandwidth, low-power, low-resistance radio frequency (RF) switches
  - Micromirrors and micro-optics for laser communications
  - Micromechanical variable capacitors, inductors, and oscillators

- Space environment sensors
  - Micromachined magnetometers
  - Gravity-gradient monitors (nano-g accelerometers)

- Distributed semiautonomous sensors
  - Multiparameter-sensor ASIM with accelerometers and chemical sensors

- Interconnects and packaging
  - Interconnects and packaging designed for ease of repairability (e.g. active "Velcro")
  - Field programmable interconnect structures
“Smart” interconnects for positive-feedback

2.2.2 Silicon Satellites

Figure 2.1 is a rendering of a silicon satellite [16]. The concept of a silicon satellite is a new paradigm for space system design, construction, testing, architecture, and deployment. The idea is to create integrated circuits for command and data handling, communications, power conversion and control, on-board sensors, attitude sensors, and attitude control devices on 1 to 4 mm-thick silicon substrates that simultaneously provide structure, radiation shielding, and thermal control. Silicon compares favorably with aluminum in terms of thermal conductivity, radiation-shielding ability, and mass density, and yet it is stronger than steel. While diamond is better on almost all counts, silicon is readily available. Batteries and solar cells for these silicon microsatellites will still need to be fabricated using conventional materials. The satellite shown in Figure 2.1 is essentially a stacked multiwafer package. First generation silicon satellites could have dimensions of 10 to 30 cm, but could be larger for more complex configurations using additional nonsilicon mechanical structure.

Figure 2.1: Silicon Satellite
Some of the benefits of batch-fabricated silicon satellites are as follows [15]:

- Radically increased functionality per unit mass
- Ability to produce 10,000 or more units for “throw-away” and dispersed satellite missions
- Decreased material variability and increased reliability because of rigid process control
- Rapid prototype production capability using electronic circuit, sensor, and MEMS design libraries with existing (and future) computer-assisted design (CAD)/CAM tools and semiconductor foundries
- Reduced number of piece parts
- Ability to tailor designs in CAD/CAM to fabricate mission-specific units

Using only solar radiation and depending on the overall configuration, microsatellites with masses in the 1 mg to 100 kg range can produce power levels on the order of 1-100 W [15]. Thermal control is also an issue for these minuscule satellites. Simple lumped parameter models have shown that passive thermal control is possible for nearly spherical microsatellites. When the dimensions drop below 2 cm, the temperature extremes exceed typical electronics and battery limits.

Silicon microsatellites with 10 cm dimensions and larger with micromachined attitude sensors and micropropulsion for attitude and orbit control could perform useful missions with on-orbit lifetimes of 1 to 5 years. Possible mission applications include communications relay, cloud cover monitoring, geolocation, and space environment monitoring. Mission applications can be grouped into three categories [15]:
Disposal missions that use satellites for a short period of time followed by deorbit

- Global coverage missions that use hundreds of silicon satellites in LEO to provide continuous Earth coverage for communications or Earth observation

- Local cluster missions that utilize hundreds of silicon satellites in a sparse array configuration to provide a large effective aperture

An example of a "disposable" mission might be tiny picosats similar to the ones that will be deployed from MightySat 2.1. The satellites would collect data in the vicinity of a "mothership" and transmit information back to the mothership. Lifetimes would be short for disposable satellites. Figure 2.2 shows a rendering of a formation-flying cluster of satellites that could be used in distributed architectures of silicon satellites [16]. Formation Flying, an idea that has been researched at MIT, involves two or more satellites actively controlling their orientation with respect to one another.

![Formation Flying Cluster of Silicon Satellites](image)

**Figure 2.2: Formation Flying Cluster of Silicon Satellites**

Both NASA and the DoD have recognized the potential of using microengineered systems in space applications. The application of MEMS technology and the development of silicon
satellites could result in revolutionary changes in current and future space systems. It may also result in drastically increased numbers of satellites in orbit around earth. Without effective methods of satellite disposal, an orbital mine field will truly be created. For the remainder of this thesis, disposal of satellites at the end of mission lifetime will be considered.
Chapter 3

Spacecraft Disposal by Thrusters and Tethers

The best way to assure that orbital debris does not continue to increase indefinitely is to design spacecraft with an inherent means of efficient disposal at the end of useful life. This thesis focuses on disposal of LEO spacecraft with end of life altitudes between 150 km and 2000 km.

Before continuing any further, the equation of motion for a satellite and the ballistic coefficient of a satellite must be introduced. An understanding of these equations is necessary before it is possible to investigate spacecraft disposal methods. These equations will be revisited throughout the thesis. The effect of perturbations on the orbit of a spacecraft is given by the following equation [17]:

\[ a = \frac{2 \cdot a^2}{\mu^2 \cdot (1 - e^2)^{\frac{1}{2}}} \cdot \left( R \cdot e \cdot \sin(f) + T \cdot (1 + e \cdot \cos(f)) \right) \]  

where \( a = \) semi-major axis (m)

\( \mu = \) gravitational parameter \( (3.986 \times 10^{14} \text{ m}^3/\text{s}^2) \)
\( e = \text{eccentricity} \)
\( R = \text{radial force per unit mass (N/kg)} \)
\( T = \text{tangential force per unit mass (N/kg)} \) and
\( f = \text{true anomaly (rad)} \).

The effect of atmospheric drag can be obtained with the assumption that \( R = 0 \) for satellites in low earth orbit [17]:

\[
\frac{\Delta a}{\mu^2} = 2 \cdot \frac{a^3}{a^2} \cdot T \cdot (1 + e \cdot \cos(f)) = -2 \cdot \frac{a^3}{\mu^2} \cdot \left( \frac{\rho \cdot C_d \cdot A \cdot v^2}{2 \cdot m} \right)
\] (3-2)

Where the orbit is nearly circular, then the velocity, \( v = (\mu/a)^{1/2} \). The error associated with this assumption is significantly less than 1% until the altitude descends below 100 km. Also, when the rate of change of the semi-major axis (a) is small (\( \Delta a/a \ll 1 \)), the rate at which a satellite will deorbit per revolution is given as follows [4]:

\[
\frac{\Delta a}{\text{rev}} = -2 \cdot \pi \cdot \frac{1}{\beta} \cdot \rho \cdot a^2
\] (3-3)

This equation gives the amount of change in the semi-major axis, a (m), per revolution. For circular orbits, subtracting the radius of the earth from the semi-major axis produces the satellite’s altitude. By integrating the change in altitude over a number of revolutions, the deorbit rate is determined. The rate at which a satellite deorbits is directly proportional to the density of the environment, \( \rho \) (kg/m\(^3\)), and inversely proportional to the satellite’s ballistic coefficient, \( \beta \). Since there is very little mankind can do about the density of the space
environment around a spacecraft (nor would mankind want to), the ballistic coefficient offers opportunity to alter the rate at which a spacecraft deorbits. The ballistic coefficient is given by the following equation [4]:

\[ \beta = \frac{m}{C_D A} \]  \hspace{1cm} (3-4)

A satellite's ballistic coefficient, \( \beta \) (kg/m\(^2\)), is proportional to the mass, \( m \) (kg), of the spacecraft and inversely proportional to the drag coefficient, \( C_D \), and the area, \( A \) (m\(^2\)), presented by the satellite in the direction of travel. The preceding equations are used in generating several of the plots in this paper.

3.1 Policy, Tracking Limitations, and Storage Orbits

As mentioned earlier, guidelines have been proposed by U.S. Space Command with regards to spacecraft disposal. LEO spacecraft above 1600 km in altitude should be boosted to a higher storage orbit. The idea is that high LEO spacecraft can be boosted into higher orbits that are comparatively unused and therefore should not disrupt future launches. But it is only a matter of time before fragmentations and orbital decay will take their toll. Eventually, space debris in storage orbits will reenter useful orbits. Utilizing storage orbits buys time, but does not offer a long-term solution to the orbital debris problem. U.S. Space Command also stipulates that spacecraft with end of life altitudes below 1600 km should be placed in a disposal orbit that will result in natural orbital decay and atmospheric reentry within 25 years. For this research, it has been assumed that a natural orbital decay within 25 years will not be sufficient in the future. With the possibility of increased numbers of satellites in the
future, it is more reasonable to assume that satellites should be deorbited within a maximum of one year after the mission lifetime is completed.

Another issue to consider is the tracking limitations of U.S. Space Command. Currently, U.S. Space Command can track objects 10 cm in diameter in LEO. The figure increases to objects with 1 m diameter in GEO. So, any object with dimensions smaller than 10 cm in LEO is untrackable and therefore even more dangerous than a larger object that is trackable. An asset such as the International Space Station can be maneuvered to avoid a trackable object while it is defenseless against a smaller, but still very destructive untrackable object. NASA has requested that the Air Force track objects down to one centimeter in size in LEO. A recent Air Force report estimates the cost of the necessary upgrades at more than $400 million [6]. While the limitation remains at 10 cm, it is unadvisable and dangerous to deploy any spacecraft with dimensions smaller than 10 cm. It is also crucial that satellites are not fragmented into smaller pieces with dimensions smaller than 10 cm.

### 3.2 Natural Orbit Decay

Natural orbital decay can cleanse very low altitude orbits within one year without the aid of propulsive maneuvers. Spacecraft with end of life altitudes between 170 km and 250 km will deorbit unaided within one year. The time it takes for a satellite to deorbit unaided is primarily affected by two factors: the solar cycle and the ballistic coefficient of the spacecraft. The solar cycle oscillates between solar maximum and solar minimum on an 11-year timescale. Spacecraft will deorbit significantly faster at solar maximum because the density of the neutral environment, and particularly atomic oxygen, is increased at solar max.
Figure 3.1 is a chart indicating the relative deorbit rates of spacecraft at solar maximum and solar minimum. The rate of orbital decay was determined by a loop that calculated the change in the semi-major axis using equation 3-3 and subtracting that value from the semi-major axis of the satellite. The loop continued until the altitude of the spacecraft was 100 km or lower. The chart depicts 10 kg satellites with an initial altitude of 175 km and average density. The average density of spacecraft launched between the years of 1978-1984 was 79 kg/m$^3$ [4]. In this thesis, a satellite with this density will be referred to as "average."

![Figure 3.1 Deorbit Rates at Solar Maximum and Solar Minimum](image)

The atmospheric densities used in producing this plot are from the MSIS 1986 atmospheric model. For solar minimum, the solar flux value, $F_{10.7}$, was chosen such that 10% of all measured data are less than the minimum value used ($65.8 \times 10^{-22}$ W m$^{-2}$ Hz$^{-1}$). For solar maximum, the $F_{10.7}$ value is such that 10% of all measured values are above it ($189.0 \times 10^{-22}$ W m$^{-2}$ Hz$^{-1}$).
W m\textsuperscript{-2} Hz\textsuperscript{-1}). Using these values for F\textsubscript{10.7} in determining the solar minimum condition and solar maximum condition is carried out throughout this thesis.

The ballistic coefficient of the spacecraft also plays a significant role in the deorbit rate of a spacecraft. A small satellite will deorbit more rapidly than a large satellite with equal densities and equivalent shape because mass varies as the cube of the satellite’s side dimension while area varies as the square of the side dimension. Aerial density is defined as mass divided by area. So if satellite “A” has a greater aerial density but equal mass to satellite “B”, the area presented in the direction of travel by A is less than B. Aerial density is directly related to the ballistic coefficient of a satellite. A 1000 kg satellite with average density will deorbit within one year from approximately 175 km at solar minimum. A 1 kg satellite with the same density will deorbit within one year from an altitude of 220 km at solar minimum. Solid silicon has a density of 2330 kg/m\textsuperscript{3}. So, if a satellite can be built out of nearly solid silicon, it’s density will be approximately 30 times greater than the average spacecraft launched between 1978 and 1984. A plot indicating the relative deorbit rates of a silicon and average satellite at solar minimum is presented in figure 3.2. The satellites begin with an initial altitude of 175 km and have a mass of 10 kg. Assuming the satellites are approximately cubical, the average density satellite will have dimensions of 0.79 m and the silicon satellite will have dimensions of 0.54 m. The discrepancy in area presented in the direction of flow accounts for the difference in deorbit rate.
3.3 Thruster Disposal

In addition to natural orbital decay, there are currently 2 other ways to dispose of a satellite at the end of its lifetime:

- Thruster transfer to deorbit or storage orbit
- Shuttle retrieval or refurbishment

Shuttle retrieval is obviously a very expensive method and is reserved for retrieval or refurbishment of very large spacecraft such as the Hubble Space Telescope and the International Space Station. Work is being done in the MIT Space Systems Lab on the possibility of extending refueling, refurbishment, and retrieval services to smaller spacecraft by way of orbiting spacecraft called “servicers.” But the possibility of servicing smaller
satellites is undoubtedly several years in the future if it proves economically feasible at all and will most likely never apply to microsatellites.

In the final currently available option, thrusters are used to supply the velocity change to transfer the spacecraft to deorbit or to transfer the spacecraft to a storage orbit. Using thrusters is currently the dominant method for disposing of satellites and will probably remain the dominant method in the near future. Possible thrusting systems include numerous propellant options: cold gas, solid, monopropellant and bipropellant liquid, various electric thrusters such as ion, hall, and field emission electric, and pulsed plasma thrusters. For the remainder of this section, currently available thruster systems and thrusters using future technologies will be explored as possible methods of spacecraft disposal.

3.3.1 Current Thruster Technologies

For LEO spacecraft that already have a thruster system for attitude control, orbit transfer, phasing, or station keeping, the most effective method of disposal is probably adding enough propellant to perform a deorbit burn at the end of the useful mission. For relatively large spacecraft (spacecraft with masses greater than 1000 kg) an electromagnetic tether, which will be covered later, will probably provide the most effective method. In this section, current thruster technologies used on spacecraft will be covered. London’s A Systems Study of Propulsion Technologies for Orbit and Attitude Control of Microspacecraft, is the primary source for information regarding propulsion choices [19]. Application of current thruster technologies to microspacecraft will be emphasized. Chemical systems will be discussed first followed by electrical systems.
3.3.1.1 Chemical Technologies

Cold Gas Thrusters

A cold gas thruster is probably the simplest kind of engine in terms of operation. A reservoir of gas is held at high pressure and pulses of the gas are allowed to expand through the converging-diverging nozzle. Its performance can be estimated using the traditional adiabatic expansion relations presented in most propulsion texts. The primary advantages of a cold gas system are simplicity and the capacity to produce extremely small thrusts and impulse bits. Therefore, cold gas systems are very useful in missions requiring precise station keeping and attitude control and not as efficient in the large ΔV maneuvers needed for deorbit. They have very low $I_{sp}$ (around 60 to 70 sec.) and there is a much higher tendency for the valves to leak because it is an all-gas system. Table 3.1 is a summary of cold gas systems [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>60 sec.</td>
</tr>
<tr>
<td>Propellant</td>
<td>GN2, GHe, etc.</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valves</td>
</tr>
<tr>
<td>Advantages</td>
<td>simplicity, power only required for valves, very low impulse bits possible</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>low $I_{sp}$, high tank pressures, valves tend to leak</td>
</tr>
</tbody>
</table>

Table 3.1: Cold Gas Summary

Solid Thrusters

Two types of solid propellants have been traditionally identified: double-based propellants and composite propellants. Double-based propellants are those that are based mainly on
combinations of nitroglycerin and nitrocellulose. This forms a rigid, homogeneous structure. Composite propellants consist of a polymeric matrix in which small particles of an oxidizer and fuel are distributed. Micropropulsion applications that would benefit from a conventionally designed solid-propulsion system are those that would only required a single, short burn time, have a large thrust of a high velocity requirement, or one that requires an extremely simple one-package propulsion device. Solid systems are a reasonable choice to be used purely for spacecraft disposal on spacecraft with attitude determination and control systems. Several small solid rocket motors, down to 0.4 kg, have been built and demonstrated [20]. Even smaller, hobby-sized solid motors for model rockets have been in use for decades. Table 3.2 is a summary of solid propulsion systems [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>260 sec.</td>
</tr>
<tr>
<td>Propellant</td>
<td>nitroglycerin and nitrocellulose, ammonia perchlorate and aluminum</td>
</tr>
<tr>
<td>Additional components required</td>
<td>nothing</td>
</tr>
<tr>
<td>Advantages</td>
<td>simplicity, higher density and thrust over liquid monopropellants, long-term storability</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>lower specific impulse than bipropellants, no restart or refill capability, potential for catastrophic failure</td>
</tr>
</tbody>
</table>

Table 3.2: Summary of Solid Propulsion Systems

Hydrazine Thrusters

Hydrazine thrusters are perhaps the most common type of thruster in use today. In these thrusters, the hydrazine first encounters a catalyst bed where it is decomposed, releasing energy that is absorbed by the decomposition products. The hot gas then expands through a
nozzle to produce the thrust. 200 seconds is a typical specific impulse. Because these thrusters typically operate in blow-down mode, the chamber pressure and thrust decreases over the lifetime of the mission. It is unclear whether these systems can scale to much smaller sizes than currently exist without significant losses in performance. If ways can be found to manufacture even smaller thrusters than the currently offered 1 to 5 N thrust range, 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. Currently, MEMS technology is limited to using silicon as the working material. Unfortunately, hydrazine tends to dissolve pure silicon, so the application of hydrazine systems will have to wait until the development of silicon carbide MEMS, which is currently being developed. Table 3.3 is a summary of hydrazine thrusters [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>210 sec.</td>
</tr>
<tr>
<td>Propellant</td>
<td>hydrazine</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valve</td>
</tr>
<tr>
<td>Advantages</td>
<td>simplicity, power only required for valves, large knowledge base</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>low Isp, difficult to scale smaller, hard to make very small impulse bits</td>
</tr>
</tbody>
</table>

**Table 3.3: Hydrazine Thruster Summary**

**Hybrid Motors**

In Hybrid rockets, the oxidizer is a liquid and the fuel is a solid (the reverse is the case in reverse hybrid motors). Hybrid rockets have been considered for applications to launch vehicles most frequently, but a group at the University of Surrey has recently suggested their
use in small satellites [19]. The proposed system uses hydrogen peroxide (HTP) as the oxidizer, and polyethylene (PE) as the fuel. The HTP is sent through a catalyst bed, where it decomposes into hot oxygen and steam that enter the combustion chamber lined with solid PE. The heat of the decomposed products is sufficient to cause the initiation of combustion that continues for as long as there is a supply of the decomposed HTP. The flow of HTP can be throttled or stopped with an upstream valve, so the hybrid motor can be as well. Also, because the decomposed gases are hot enough to initiate combustion independently, such a system does not require an igniter, increasing simplicity. 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 3.4 [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>280 sec.</td>
</tr>
<tr>
<td>Propellant</td>
<td>HTP/PE</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valve</td>
</tr>
<tr>
<td>Advantages</td>
<td>simplicity, higher $I_{sp}$, power only required for valves, no ignition, safe propellants, restartable</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>hard to make very small impulse bits</td>
</tr>
</tbody>
</table>

Table 3.4: Hybrid Motor Summary

Bipropellant Engines

Bipropellant engines provide significantly higher specific impulse (300 sec.) than monopropellant engines at the expense of requiring two tanks for the propellant, one for the fuel and one for the oxidizer. Two sets of valves, lines, and other supporting equipment are required as well. Bipropellant engines, including work done in developing micro-
bipropellant engines, will be covered in greater detail in the next section. For now, a schematic of a micro-bipropellant engine is provided in figure 3.3 and a summary is given in table 3.5 [19].

![Figure 3.3: Micro-Bipropellant Engine Concept](image)

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>300 sec.</td>
</tr>
<tr>
<td>Propellant</td>
<td>LOX/RP-1</td>
</tr>
<tr>
<td>Additional components required</td>
<td>2 tanks</td>
</tr>
<tr>
<td>Advantages</td>
<td>low tank pressure, high Isp, restartable, high T/W ratio</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>never done before, cryogenic oxidizer, power required for pumps</td>
</tr>
</tbody>
</table>

Table 3.5: Summary of Bipropellant Systems
3.3.1.2 Electric Propulsion

Resistojets, Electrothermal Hydrazine Thrusters, Arcjets

These three technologies are basically a cross between electrical 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 that requires an input power of 1.8 kW to produce approximately 0.2 N of thrust at 500 sec specific impulse. To be complete, these technologies are summarized in table 3.6 [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Resistojet</th>
<th>EHT</th>
<th>Arcjet</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>250 sec</td>
<td>300 sec</td>
<td>500 sec</td>
</tr>
<tr>
<td>Propellant</td>
<td>many</td>
<td>Hydrazine</td>
<td>many, Hydrazine</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valve, power conditioning unit</td>
<td>tank, valve, power conditioning unit</td>
<td>tank, valve, power conditioning unit</td>
</tr>
<tr>
<td>Advantages</td>
<td>simplicity, higher Isp than cold gas thrusters</td>
<td>higher Isp than hydrazine monopropellant</td>
<td>high Isp</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>very high power requirement, added mass of PCU</td>
<td>high power requirement, added mass of PCU</td>
<td>very high power requirement, added mass of PCU</td>
</tr>
</tbody>
</table>

Table 3.6: Summary of Resistojets, EHTs, and Arcjets

Ion Thrusters

The concept of an ion thruster is to create heavy ions in an ionization chamber, and then accelerate these ions electrostatically to very high exit velocities. Figure 3.4 is a schematic on an ion engine [19].
The acceleration is created by applying a large potential 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 an ion reaches the accelerator grid it is attracted by the large negative potential on the outer grid and accelerated to the exhaust velocity. 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. Table 3.7 is a summary of ion thrusters [19].
<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>3000 sec</td>
</tr>
<tr>
<td>Efficiency</td>
<td>65%</td>
</tr>
<tr>
<td>Propellant</td>
<td>Xenon gas</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valve, power conditioning unit, power supply</td>
</tr>
<tr>
<td>Advantages</td>
<td>very high Isp</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>low thrust per area, high magnetic field required at small sizes</td>
</tr>
</tbody>
</table>

Table 3.7: Ion Thruster Summary

Hall-Effect Thrusters

The principle of operation of a Hall thruster is similar to an ion engine in that it uses an applied potential to accelerate ions to a high exhaust velocity. Figure 3.5 is a schematic of a Hall thruster [19].

![Figure 3.5: Hall Thruster Schematic](image)
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. Table 3.8 is a summary of Hall Thrusters [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>1600 sec.</td>
</tr>
<tr>
<td>Efficiency</td>
<td>50%</td>
</tr>
<tr>
<td>Propellant</td>
<td>Xenon gas</td>
</tr>
<tr>
<td>Additional components required</td>
<td>tank, valve, power conditioning unit, power supply</td>
</tr>
<tr>
<td>Advantages</td>
<td>very high Isp</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>low thrust per area, high magnetic field required at small sizes</td>
</tr>
</tbody>
</table>

Table 3.8: Hall Thruster Summary

**Pulsed Plasma Thruster**

Conceptually, the pulsed plasma thruster (PPT) is a fairly simple device. First developed in the mid 1960’s, they have been used for both attitude control and station keeping. They are
considered useful because they can produce very small and repeatable impulse bits. Also, 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 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. Figure 3.6 is a schematic of a PPT and table 3.9 is a summary [19].

Figure 3.6: PPT Schematic

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>1000 sec.</td>
</tr>
<tr>
<td>Efficiency</td>
<td>15%</td>
</tr>
<tr>
<td>Propellant</td>
<td>solid Teflon</td>
</tr>
<tr>
<td>Additional components</td>
<td>power conditioning unit, power supply</td>
</tr>
<tr>
<td>Advantages</td>
<td>very high Isp compared to chemical, no tank or valve requirements, simplicity, small and repeatable impulse bits</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>low efficiency</td>
</tr>
</tbody>
</table>

Table 3.9: PPT Summary
Field Emission Electric Propulsion

Field Emission Electric Propulsion (FEEP) is basically an ion engine where the ionization and acceleration are both provided by the same electric field generated by a plate at an extremely high negative potential. The FEEP concept is schematically illustrated in Figure 3.7 [19].

![FEEP Schematic](image)

**Figure 3.7: FEEP Schematic**

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. 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 the 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. FEEP thrusters are perhaps ideally suited to missions with extremely low thrust requirements. A summary of FEEP thrusters is presented in Table 3.9 [19].

<table>
<thead>
<tr>
<th>Attribute</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Impulse</td>
<td>10,000 sec.</td>
</tr>
<tr>
<td>Efficiency</td>
<td>95%</td>
</tr>
<tr>
<td>Propellant</td>
<td>liquid Cesium</td>
</tr>
<tr>
<td>Additional components required</td>
<td>power conditioning unit, power supply</td>
</tr>
<tr>
<td>Advantages</td>
<td>extremely high Isp, limited tank or valve requirements, simplicity, extremely small and repeatable impulse bits, extremely low thrust</td>
</tr>
<tr>
<td>Disadvantages</td>
<td>for some applications Isp too high and thrust too low</td>
</tr>
</tbody>
</table>

Table 3.10: FEEP Thruster Summary

3.3.1.3 Scaling of Other Components

Before moving on, some general scaling parameters will be provided. In traditional space propulsion systems, the actual thrusters usually make up a fairly small proportion of the total propulsion system mass when compared to the other parts of the system. In this study, the other components are classified as power supply, power conditioning, propellant, propellant tanks, and valves. Each of these will be briefly discussed.

Power Supply

Solar voltaic cells are the principal power supply in space. For this study, solar arrays will be modeled as having a specific power of 70 W/kg. Solar voltaics are improving rapidly, so 70
$W$/kg is probably a conservative figure. The mass, $m_{PS}$ (kg), of the power supply system is then given as:

$$m_{PS} = \frac{P_{req}}{\alpha_{PS}}$$  \hspace{1cm} (3-5)

where $P_{req}$ (W) is the required power and $\alpha_{PS}$ is the specific power parameter, 70 $W$/kg.

**Power Conditioning Equipment**

Particularly for electrical propulsion, the power conditioning unit (PCU) often dominates the power system and weighs far more than the arrays required to produce the power in the first place. With current technology, a minimum mass of 40 g can be used for a power conditioning unit that processes no power (just the box with the various cards, connections, and other components). A scaling law is given as:

$$m_{PC} = m_{pc0} + \frac{P_{req}}{\alpha_{PC}}$$  \hspace{1cm} (3-6)

where $m_{PC0}$ is the minimum mass of 0.040 kg, $P_{req}$ (W) is the power that is processed, and $\alpha_{PC}$ is the scaling parameter, which depends on the type of thruster being used. A table of the scaling parameters is given as table 3.10 [19].
<table>
<thead>
<tr>
<th>Thruster Type</th>
<th>$\alpha_{PC}$ [W/kg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ion</td>
<td>65</td>
</tr>
<tr>
<td>Hall</td>
<td>100</td>
</tr>
<tr>
<td>PPT</td>
<td>150</td>
</tr>
<tr>
<td>FEEP</td>
<td>1.25</td>
</tr>
</tbody>
</table>

Table 3.11: PCU Scaling Parameters for Electric Thrusters

Propellant

The required mass of the propellant is determined based on the specific impulse of the thruster system and the required $\Delta V$ that the propellants must provide. This leads to the rocket equation:

\[
m_{prop} = m_0 \left(1 - e^{\frac{\Delta V}{c}}\right) = m_0 \left(1 - e^{\frac{\Delta V}{I_{sp} g_0}}\right)
\]  

(3-7)

where $m_0$ (kg) is the initial mass of the satellite, and $c$ (m/s) the thruster exhaust velocity that is equal to the product of $I_{sp}$ (s) and $g_0$, where $g_0$ (m/s$^2$) is the acceleration of gravity at the Earth's surface.

Propellant Tanks

Propellant tanks scale based on their volume and the pressure that they must contain. For each case, the volume of the tank must be determined from it's propellant density and mass, and the pressure required is based on the type of thruster being used. Assuming spherical tanks, their thickness is given by:
where \( P \) (Pa, N/m\(^2\)) is pressure contained in the tank, \( r \) (m) its radius, \( \sigma \) (N) the working strength of the material, and \( FOS \) a factor of safety. The tank mass is simply:

\[
M_{\text{tank}} = \frac{Pr}{2\sigma} FOS
\]

(3-8)

\[
m_{\text{tank}} = 4\pi r^2 t_{\text{tank}} \rho_{\text{tank}}
\]

(3-9)

where \( t_{\text{tank}} \) is calculated above and \( \rho_{\text{tank}} \) (kg/m\(^3\)) is the density of the tank material.

Valves

In order to prevent leaks, valves are often designed with significant internal redundancy through placing many in series or parallel. 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 thrusters. This gives:

\[
m_{\text{valve}} = m_{\text{valve}_0} \left(\frac{\dot{m}}{m_0}\right)^{3/2}
\]

(3-10)

where \( m_{\text{valve}_0} \) is the mass of the reference valve, 0.2 kg, \( \dot{m} \) is the desired mass flow rate, and \( m_0 \) is the mass flow rate of the reference valve, 2.4 g/s.
3.3.2 Promising Future Propulsion Technologies

Now that the most promising propulsion systems for microsatellites have been introduced, this section will expand upon the issues of scaling to small sizes. For each technology, the most appropriate microsatellite niche is identified as well as speculation on the technical hurdles for the technology.

Micro Ion Engines

Ion thrusters seem to be most 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 at approximately 400 m/s for thrusts of about 3mN. This cutoff value actually increases fairly quickly as the required thrust levels increase. 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 [19]. For extremely high ΔV requirements, such as a microspacecraft mission to Mars, 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.

The largest technical issue that must be overcome in making ion engines smaller is probably a manufacturing issue. It is difficult to manufacture devices as complex as ion engines on the scale required. Reproducing the complex magnetic field patterns at the smaller scales required could also be extremely difficult.
**Micro Hall Thrusters**

Hall thrusters seem to be most appropriate for medium $\Delta V$ missions that require thrusts on the order of 3 to 20 mN. A definition for medium $\Delta V$ missions might be about 150 m/s to approximately 450 m/s. When cost and manufacturability are taken into account, the domain of Hall thrusters extends even higher in $\Delta V$ due to the much higher complexity of Ion engines when compared to Hall thrusters.

The technical hurdles to producing a micro Hall thruster system are basically the same as those for the Ion thruster. However, the hurdles should be somewhat lower. Although the magnetic field required is probably higher in magnitude than in a comparable ion engine, the complex field shape is not required.

**Pulsed Plasma Thrusters**

Pulsed Plasma thrusters (PPTs) are well suited to fine positioning and attitude control where very small impulse bits are required. They are also suited to medium $\Delta V$ missions where there is no (or very low) minimum thrust requirement. The fairly high specific impulse of PPTs lead to low propellant mass, and the solid propellant eliminates tanks, valves, lines, and pressure regulators, greatly simplifying the propulsion system. These advantages also allow them to be placed more remotely and strategically on a spacecraft, without the need to transfer propellant via a line from a central tank. Their energy density is low so they require more power per unit thrust than either Hall or ion thrusters.
There are a few technical issues that need to be addressed prior to PPTs 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. Further improvements in capacitor energy storage density would also be very beneficial, as the vast majority of the thruster mass is the capacitor that is discharged in each pulse. Also unclear is the best way to fabricate these small thrusters.

**Field Emission Electric Propulsion**

FEEP seems most promising for very large ΔV missions that permit (or require) low thrust. 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.

The technical hurdles to FEEP are listed above: reduction in mass, particularly the mass of the power conditioning electronics. If the electronics could be integrated into the system as part of MEMS manufacturing, one can imagine a rather simply produced, compact, and modular thruster that produces extremely low thrusts requiring very small amounts of propellant.

**Micro Hybrid Motors**

Small hybrid motors appear to have significant promise in the fairly near term for microsatellite 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.
There seems to be no technical hurdles to the development of these systems. Development programs appear to be progressing quite successfully, though no flight tests have yet occurred.

**Micro Bipropellant Engines**

A micro bipropellant engine has the same domain of applicability as the hybrid rockets discussed previously, as it has similar if not better performance in most cases.

An actual working prototype of a micro bipropellant engine was developed recently in MIT’s Gas Turbine Laboratory. MEMS technology was relied upon heavily in the fabrication of the system. High precision micromachining in silicon or silicon carbide makes high chamber pressure micro-bipropellant rocket engines a technical reality. Test results from an engine using oxygen and methane as propellants support the feasibility of the concept by producing a maximum thrust of 1 N at a chamber pressure of 12 atm. These values correspond to a thrust to weight ratio of 85:1. The working engine is approximately equal in size to a postage stamp. A schematic of the thruster is shown in figure 3.8 [18].
Domains of Applicability of Micro Propulsion

These technologies tend to map into domains of applicability if the two parameters of thrust and mission $\Delta V$ are considered. Figure 3.9 is a graphical representation of the domains of applicability [19].
A two-way arrow is drawn at the top of the diagram to represent the ΔV required just for deorbit of spacecraft from LEO. The left side of the arrow represents the necessary ΔV for very low LEO orbits while the right side of the arrow is the ΔV required for spacecraft with orbit altitudes of 1600 km. Pulsed Plasma Thrusters appear to be the 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 micro hybrid motors or micro bipropellant rockets are the technologies of choice.

3.3.3 Enabling Technologies for Micro Propulsion Systems

How can micromachining techniques enable the building of propellant tanks, propellant lines and valves for small spacecraft? One solution is to bond several layers so that shallow surface cavities become tubes and deep cavities become propellant tanks. Figure 3.10 shows the basic concept in which three layers are bonded to form a propellant tank, associated plumbing, and two simple expansion nozzles [15]. Multiple thrusters and propellant feed systems can be produced on the same substrate.
Another approach, funded by DARPA and executed by TRW, Inc., Aerospace, and the California Institute of Technology, is to construct an array of single-shot microthrusters [15]. In its simplest form, this “digital propulsion concept uses individually sealed microcavities containing propellant, an internal heating resistor, and a micromachined silicon or silicon nitride burst disk as shown in figure 3.11 [15].
Each microcavity provides an impulse when the contained propellant is ignited and the gases exhausted. The diaphragm is designed to burst at a preset pressure. For additional thrust, the exhaust gases are made to flow through a converging/diverging nozzle. Micromachining enables the fabrication of thousands of similar microthrusters so that hundreds of complex propulsion maneuvers can be performed.

Nozzles for microthrusters have also been fabricated using deep reactive ion etching of silicon or other materials. Figure 3.12 illustrates a top view and an end-on view of different micronozzles [21].

![Microfabricated Nozzles](image)

**Figure 3.12: Microfabricated Nozzles**

Throat "diameters" varying from 12 to 30 µm have been fabricated with expansion ratios (exit plane area divided by the throat cross-sectional area) ranging from 5 to 20 [21].

Electric propulsion has not been left behind with regard to enabling technologies. Miniaturized liquid metal ion sources fabricated using microfabrication technology were
promoted by J. Mitterauer during the early 1990s. These devices are mechanically similar to a Spindt "microvolcano" shown schematically in figure 3.13 [21].

The Spindt microvolcano relies on solid electrodes, not on conductive fluids, that are electrostatically shaped into sharp tips. It can directly field-ionize gases. Figure 3.14 shows electrostatic potential contours in 10 V steps for a potential drop of 100 V between the upper and lower electrodes [21].
Electric fields of ~5 x 10^8 V/m (50 V per 0.1 μm) are developed near the rim of the volcano orifice. These fields are high enough to generate ions by field-ionization and can readily produce molecular ions without fragmentation. In general, electric fields of ~10^9 V/m will produce significant electron emission from metals, and ~10^10 V/m will enable efficient field-ionization [21].

From the standpoint of fabricating a practical ion thruster based on the above concepts, the most important criteria is developing microstructure gaps or tips that can generate very high electric fields. An approach being considered at Aerospace is the nanotexturing of surfaces whereby natural gaps, facets, or separations can be exploited for field ionization.

3.4 Tethers for Satellite Disposal

The idea of using tethers for satellite orbital transfer has received a considerable amount of attention in the past few years. A tether system is conceivable for orbital transfer in one of three architectures. These three architectures will be referred to as system #1, system #2, and system #3. The three systems are presented in Figure 3.15.

Figure 3.15: Possible Tether Configurations
These architectures are by no means drawn to scale. In an actual tether system, the tether would be much longer relative to the size of the satellite. The first system is one in which the satellite is lowered by raising a mass which is connected by the system. The center of mass of the system will remain at the original altitude. When the tether is severed at some point, the lower mass (satellite) will have a lower velocity than the circular velocity for the orbit that it is in, and it will acquire an elliptical orbit with a perigee at some lower altitude and an apogee at the original altitude before the tether was snapped. One immediate disadvantage of this system is that the top mass assumes a higher elliptical orbit after the tether is severed and remains in orbit. Also, the tether length would have to be unreasonably long to deorbit the satellite from any appreciable altitude. For these reasons, architecture #1 is not a viable architecture for spacecraft disposal. The second system is one in which a "drag sail" is lowered on a tether to an altitude in which the atmospheric drag is high enough to produce a rapid deorbit for the drag sail and the satellite. The problem with this architecture is that the dynamics of the system is difficult to predict when the sail is lowered into a significantly higher density environment. Also, the tether would have to be reasonably long which would possibly interfere with satellites in lower orbits. The third system utilizes an electromagnetic tether connected to a plasma contactor. In the presence of the earth’s magnetic field, a current will be established along the tether that produces EM forces that will decelerate the system and cause it to deorbit more rapidly. This architecture has the added advantage that the EM tether can supply power to the satellite during the mission lifetime [22].
By doing a systems analysis, it is possible to compare an electromagnetic tether system to various propulsion systems. The Harvard-Smithsonian Center for Astrophysics in Cambridge has done significant work in the area of electromagnetic tethers [23]. A tether system is capable of deorbiting a satellite within a year from altitudes up to 1600 km. Currently, tether systems have been developed which include the tether itself, a deployment mechanism, and the plasma contactor at the end of the tether. The minimum mass of the system with current technology is approximately 50-70 kg. The size of the tether scales with altitude, but the rate at which the tether mass increases is significantly less than the rate at which the mass of liquid and solid propellants increase. Tether systems prove to be the best deorbiting alternative for satellites that are approximately 1000 kg and larger.

3.4.1 Electromagnetic Tethers

A more complete understanding of the workings of an electromagnetic tether is in order. Gravity gradient forces are fundamental to tether applications. For the purposes of this discussion, it will be sufficient to describe the motion of a simple “dumbbell” configuration, composed of two masses (possibly with large mass variation) connected by a tether. Figure 3.16 shows the forces acting on this system at orbital velocity [22].
When it is oriented such that there is a vertical separation between the two masses, the upper mass experiences a larger centrifugal than gravitational force, and the lower mass experiences a larger gravitational than centrifugal force. The result of this is a force couple applied to the system, forcing it into a vertical orientation. Displacing the system from vertical produces restoring forces acting on the system that act to return the system to a vertical orientation. The restoring forces acting on the system are shown in figure 3.17 [22].
The gravitational and centrifugal forces (accelerations) are equal and balanced at only one place: the system’s center of gravity. The center of gravity (or mass) is in free fall as it orbits the earth, but the two end masses are not. They are constrained by the tether to orbit with the same angular velocity as the center of gravity. Without the tether, the upper mass would move at a slower speed and the lower mass would move at a higher speed. The tether, therefore, speeds up the upper mass and slows down the lower mass. This is why the upper mass experiences a larger centrifugal than gravitational acceleration, and why the lower mass experiences a larger gravitational than centrifugal acceleration. The resulting upward acceleration of the upper mass and downward acceleration of the lower mass give rise to the balancing tether tension. They also produce the restoring forces when the system is deflected from a vertical orientation. The masses experience this tension as artificial gravity.
Electromagnetic (or electrodynamic) tether systems can be designed to produce several useful effects by interacting with magnetic fields. They can be designed to produce either electrical power (and drag) or thrust. The can also be designed to alternatively produce electrical power and thrust. In addition, they can be designed to produce ULF/ELF/VFR electromagnetic signals in the upper atmosphere, and shape-stability for orbiting satellite constellations.

Consider a vertical, gravity-gradient stabilized, insulated, conducting tether, which is terminated at both ends by plasma contactors. A typical configuration is shown in figure 3.18 [22].
As the system orbits Earth, it cuts across the geomagnetic field at about 8 km/s (for LEO spacecraft). An electromotive force (emf) is induced across the length of the tether. This emf is given by the equation:

\[ V = \int (\mathbf{v} \times \mathbf{B}) \cdot d\mathbf{l} \]  \hspace{1cm} (3-11)

where \( V \) = induced emf across the tether length (volts)
\( \mathbf{v} \) = tether velocity relative to the geomagnetic field (m/s)
\( \mathbf{B} \) = magnetic field strength (webers/m\(^2\)), and
\( d\mathbf{l} \) = differential element of tether length

For the case where the tether is straight and perpendicular to the magnetic field lines everywhere along its length, \( L \) (m), the equation for the emf simplifies to:

\[ V = (\mathbf{v} \times \mathbf{B}) \cdot \mathbf{L} \]  \hspace{1cm} (3-12)

The equation for the induced emf across the tether in this special case can also be written as:

\[ V = L \cdot \mathbf{v} \cdot \mathbf{B} \cdot \sin \theta \]  \hspace{1cm} (3-13)

where \( \theta \) = angle (deg) between \( \mathbf{v} \) and \( \mathbf{B} \).
From these equations, it can be seen that equatorial and low-inclination orbits will produce the largest emfs, since the maximum emf is produced when the tether velocity and the magnetic field are perpendicular to each other.

The emf acts to create a potential difference across the tether by making the upper end of the tether positive with respect to the lower end. In order to produce a current from this potential difference, the tether ends must make electrical contact with the earth's plasma environment. Plasma contactors at the tether ends provide this contact, establishing a current loop (a so-called "phantom loop") through the tether, external plasma, and ionosphere. Although processes in the plasma and ionosphere are not clearly understood at this time, it is believed that the current path is like that shown in figure 3.18. The collection of electrons from the plasma at the top end of the tether and their emission from the bottom end creates a net-positive charge cloud (or region) at the top end, and a net-negative charge cloud at the bottom end. The excess free charges are constrained to move along the geomagnetic field lines intercepted by the tether ends until they reach the vicinity of the E region of the lower ionosphere where there are sufficient collisions with neutral particles to allow electrons to migrate across the field lines and complete the circuit.

To optimize the ionosphere's ability to sustain a tether current, the tether current density at each end must not exceed the external ionospheric current density. Plasma contactors must effectively spread the tether current over a large enough area to reduce the current densities to the necessary levels. Three basic tether system configurations, using three types of plasma contactors, have been considered. They are:
• A passive large-area conductor at both tether ends
• A passive large-area conductor at the upper end and an electron gun at the lower end
• A plasma-generating hollow cathode at both ends

As mentioned before, electromagnetic tether systems can be used to generate thrust or drag (with drag being the goal in this study). Consider the gravity-gradient-stabilized system in Earth orbit, for example. Its motion through the geomagnetic field induces an emf across the tether. When the current generated by this emf is allowed to flow through the tether, a force is exerted on the tether by the geomagnetic field given by:

\[ \vec{F} = \int (I \cdot d\vec{l}) \times \vec{B} \]  \hspace{1cm} (3-14)

where \( \vec{F} = \text{force exerted on the tether by the magnetic field (newtons)} \)

\( I = \text{tether current (amps)} \)

\( d\vec{l} = \text{differential element of tether length} \)

\( \vec{B} = \text{magnetic field strength (Wb/m}^2\text{)} \)

This equation simplifies, for the case of a straight tether, to:

\[ \vec{F} = I\vec{L} \times \vec{B} \]  \hspace{1cm} (3-15)

and can also be written as:

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\[ F = ILB \cdot \sin \theta \] (3-16)

where \( \theta \) = angle (deg) between \( \vec{L} \) and \( \vec{B} \) [22].

3.4.2 Current Electromagnetic Tether Work

Electromagnetic tether work that is being done at the Smithsonian Astrophysical Observatory is the Propulsive Small Expendable Deployment System (ProSEDS) [24]. As presently planned, ProSEDS will be carried into a 400 km orbit as a Delta-III secondary payload. The tether deployer used to deploy the tether upward will be made of two parts as shown in figure 3.19 [24].

![Figure 3.19 Schematic of ProSEDS](image)

Figure 3.19 Schematic of ProSEDS
First, a 10 km nonconductive ballast tether with an end mass will be deployed. This will serve to pull off the remaining 5 km segment of bare metallic tether, which will be used to collect electrons. A hollow cathode will maintain electrical connection with the plasma at the Delta platform. The goals are to demonstrate high current collection by a bare tether and significantly accelerated orbital decay, due to the magnetic drag. A parametric analysis of the reentry time for different values of the average tether current and for different starting altitudes was carried out. The parametric computations were carried out with the following assumptions:

- Launch in August 2000 = F10.7 \approx 155 (50\% \text{ probability})
  = F10.7 \approx 240 (98\% \text{ probability})
- Tether = 5-km wire, 10-km
- Delta mass = 900 kg
- ProSEDS mass = 100 kg

As of the writing of this thesis, ProSEDS has not been launched. But figure 3.20 is a prediction of the deorbit rate of the tether and Delta system from a starting altitude of 400 km based on the above assumptions [24].
A company called Tethers Unlimited, Inc. is currently developing a tether system called the Terminator Tether. The system is being developed with NASA’s Marshall Space Flight Center under a Small Business Innovation Research (SBIR) agreement [25]. Significant information on the use of tethers for spacecraft disposal (as well as the use of tethers to accomplish other space missions) can be found at the home page of Tethers Unlimited, Inc [26].

Figure 3.20 Reentry Time for a Tether and Delta Stage System
Chapter 4

Alternate Methods of Spacecraft Disposal

4.1 Altering the Ballistic Coefficient

The ballistic coefficient of a satellite determines the rate at which the orbit of a satellite will decay. The ballistic coefficient and its impact on deorbit rate have been discussed previously, but will be revisited again. The rate at which a satellite will deorbit from circular orbits for which the semi-major axis changes in small increments is given by the following equation:

\[
\frac{\Delta a}{\text{rev}} = -2 \cdot \pi \cdot \frac{1}{\beta} \cdot \rho \cdot a^2
\]  

(4-1)

The equation gives the amount of change in the semi-major axis (a) per revolution. Subtracting the radius of the earth from the semi-major axis produces the satellite’s altitude for a circular orbit. By integrating the change in altitude over a number of revolutions, the deorbit rate is determined. The rate at which a satellite deorbits is directly proportional to the density of the environment (\(\rho\)) and inversely proportional to the satellite’s ballistic
coefficient ($\beta$). The ballistic coefficient offers opportunity to alter the rate at which a spacecraft deorbits. The ballistic coefficient is given by the following equation:

$$\beta = \frac{m}{C_D \cdot A}$$

(4-2)

A satellite’s ballistic coefficient is proportional to the mass ($m$) of the spacecraft and inversely proportional to the drag coefficient ($C_D$) and the area ($A$) presented by the satellite in the direction of travel.

To cause a satellite to deorbit more rapidly, it is desirable to decrease the ballistic coefficient. Decreasing the ballistic coefficient can be accomplished by one (or a combination) of the following three options:

- Reduce the mass
- Increase the drag coefficient
- Increase the area

These options will be explored in that order in this section of the thesis.

### 4.2 Reducing Satellite Mass

The first option considered is to decrease the satellite’s mass while maintaining a proportionally high effective area. Several methods can be conceived of to accomplish the breakup of a satellite. One idea is the concept of a self-consuming satellite. An area of research focusing on the idea of a self-consuming satellite has been initiated by the Air Force Research Laboratory [27]. A self-consuming satellite would be one in which propellant...
(such as Teflon for a pulsed-plasma thruster) is also used as connecting structure for various components of the spacecraft. As the Teflon is consumed for propulsive maneuvers, unused components of the spacecraft would be separated. Another idea is the concept of a biodegradable satellite. A biodegradable satellite would be one in which proteins or sugars are used as structure. These materials could then be converted into propellants (ammonia, methane, methanol, ethanol) [21]. As the structure is converted, unused components of the spacecraft could be discarded.

Various simple spacecraft shapes can be explored to determine if it would be possible to lower the ballistic coefficient of the shape by changing the dimensions of the shape. The idea is to lower the mass to area ratio (m/A) of the subsequent object. For flat plates, the m/A ratio is as follows:

$$\frac{m}{A} = \frac{\rho_m \cdot A \cdot t}{A} = \rho_m \cdot t$$  \hspace{1cm} (4-3)

where \( \rho_m \) = density of the material (kg/m\(^3\))

\( t \) = thickness of the material (m)

Assuming the density of the plate remains the same, the ballistic coefficient will be unaffected unless the thickness of the plate is altered while the same effective area is maintained. This means that the plate would have to be "sliced" into thinner plates with the same surface area to decrease the ballistic coefficient and increase decay rate. There are several problems with this scenario. First, satellites rarely start life or spend any part of their
life as a flat plate. Also, if a satellite is broken into thinner slices, there is no easy way to orient the flat plate that results so that the largest area is facing into the direction of travel.

A cube is a more realistic simple shape to analyze with respect to spacecraft. A cube with six sides of equivalent thickness, \( t \), was assumed. First, a hollow cube can be considered. If the hollow cube is broken up in to six equivalent sides, the resulting \( m/A \) ratio is as follows:

\[
\frac{m}{A} = \frac{6 \cdot \rho_m \cdot A \cdot t}{A} = 6 \cdot \rho_m \cdot t
\]  

(4-4)

This indicates that the six individual plates will deorbit 6 times more rapidly than the initial box as long as the effective area of the plates is equal to that of the cube. But the problem is that there is no way of assuring that the effective area of the plates in the direction of flow will be the same as the original cube without having an attitude control system associated with each of the cubes.

Assuming a thickness, \( t \), approximately \( 1/20^{th} \) of the height and width of the plate (5 cm for a plate with dimensions of 1 m\(^2\)) and assuming that the plate is oriented as in figure 4.1 and is rotating about the axis through the plate as illustrated in figure 4.1, the average area presented to the flow can be determined.
The average area presented in the direction of travel by the resulting plates in this example will be approximately 0.06 m². It can be shown that the individual plates will deorbit at only 36% of the rate of the original cube.

If it is assumed that the cube is a representation of a basically solid satellite (accounting for the internal components), an option is to break the cube in half as illustrated in figure 4.2.

Figure 4.1: Worst-Case Orientation of Resulting Plates

Figure 4.2: Breaking a Cube into Two Triangular Pieces
It is possible to compute and compare the relative m/A ratios of the resulting pieces to the original object. It turns out that the average m/A ratio of the triangular pieces is approximately 56% of the original object. This is true for any cube of equal dimensions and uniform density. So, the triangular pieces have a ballistic coefficient of approximately one half of the original object and will deorbit approximately twice as fast.

A final option with respect to solid cubes is to break the original object into numerous smaller pieces as illustrated in figure 4.3.

By breaking the original cubical object into approximately 125 smaller pieces, it can be shown that the m/A ratio of the smaller pieces is approximately 20% of the original object for cubes with uniform density. Therefore, the smaller pieces will deorbit approximately 5 times faster than the original object. The rate at which the smaller cubes will deorbit is generalized as follows:

Figure 4.3: Breakup of Cube into Numerous Smaller Pieces

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\[
\text{deorbit rate}_{\text{late}} = \sqrt[3]{\text{number of pieces}}
\]

It is also possible to determine the result of decomposing a satellite with an original spherical or cylindrical shape. For a sphere, once again assuming that the density of the object remains the same, the \(m/A\) ratio is determined as follows:

\[
\frac{m}{A} = \frac{\frac{4}{3} \cdot \pi \cdot r^3 \cdot \rho_m}{\pi \cdot r^2} = \frac{4}{3} \cdot r \cdot \rho_m
\]

where \(r = \text{radius of the sphere}\).

By reducing the radius of the sphere, the ballistic coefficient will be reduced and the satellite will reenter more rapidly.

The \(m/A\) ratio of a cylinder can be determined as well:

\[
\frac{m}{A} = \frac{\pi \cdot r^2 \cdot h \cdot \rho_m}{2 \cdot r \cdot h} = \frac{\pi \cdot \rho_m \cdot r}{2}
\]

where \(r = \text{radius of the cylinder}\)

\(h = \text{height of the cylinder}\).

Here again, the \(m/A\) ratio (and thus the ballistic coefficient) is lowered by reducing the radius of the cylinder.
By analyzing plates, cubes, spheres, and cylinders, it has been shown that it is possible to alter the ballistic coefficients of the shapes by altering dimensions. For cubes, it is advantageous to break the cube into smaller cubes. Reducing the radius of a sphere or cylinder will decrease the ballistic coefficient proportionally. So if an architecture can be determined to lower the radius of a spherical or cylindrical satellite after the conclusion of its useful lifetime, it will reenter and burn up in the atmosphere more rapidly. These conclusions are summarized in Table 4.1.

<table>
<thead>
<tr>
<th>Satellite Shape</th>
<th>Method of Breaking Up</th>
<th>Decrease in $m/A$ ratio or Deorbit Rate</th>
</tr>
</thead>
<tbody>
<tr>
<td>Plate</td>
<td>not worthwhile, ballistic coefficient unaffected</td>
<td>$\frac{m}{A} = \frac{\rho_m \cdot A \cdot t}{A} = \rho_m \cdot t$</td>
</tr>
<tr>
<td>Cube</td>
<td>break into numerous smaller pieces</td>
<td>$\frac{deorbit _rate_{little}}{deorbit _rate_{big}} = \sqrt[3]{\text{number of pieces}}$</td>
</tr>
<tr>
<td>Sphere</td>
<td>reduce size (radius)</td>
<td>$\frac{m}{A} = \frac{4 \cdot \pi \cdot r^3 \cdot \rho_m}{3 \cdot \pi \cdot r^3} = \frac{4}{3} \cdot r \cdot \rho_m$</td>
</tr>
<tr>
<td>Cylinder</td>
<td>reduce size (radius)</td>
<td>$\frac{m}{A} = \frac{\pi \cdot r^2 \cdot h \cdot \rho_m}{2 \cdot r \cdot h} = \frac{\pi \cdot \rho_m \cdot r}{2}$</td>
</tr>
</tbody>
</table>

**Table 4.1: Comparison of Decrease in $m/A$ Ratio by Breaking Up Spacecraft**

There are several difficulties associated with the idea of breaking up a satellite into smaller pieces. The first problem is that to accomplish any appreciable decrease in the ballistic coefficient, the spacecraft must be broken into many pieces. These smaller pieces, for satellites that are already small to begin with, would soon be smaller than the 10-cm dimensions that can be tracked by U.S. Space Command. In that case a more significant
problem is created than the one that is potentially solved. For instance, breaking a cubical satellite into 125 smaller cubical pieces, decreases the satellite's ballistic coefficient only by a factor of 5. Another problem is the challenge of developing the self-consuming or biodegradable concept to the point where it is truly an effective method. Coupling these difficulties together makes the idea of breaking a satellite up into smaller pieces challenging.

4.3 Increasing the Coefficient of Drag

Increasing the drag coefficient of a satellite is another option in the pursuit of decreasing a satellite's ballistic coefficient. Table 3.11 is a table of the ballistic coefficients of satellites that have flown in space in the past [4].

<table>
<thead>
<tr>
<th>Satellite</th>
<th>Shape</th>
<th>Maximum Cross-Sectional Drag Coefficient</th>
<th>Minimum Cross-Sectional Drag Coefficient</th>
</tr>
</thead>
<tbody>
<tr>
<td>Oscar-1</td>
<td>box</td>
<td>4</td>
<td>2</td>
</tr>
<tr>
<td>Intercos.-16</td>
<td>cylinder</td>
<td>2.67</td>
<td>2.1</td>
</tr>
<tr>
<td>Viking</td>
<td>octagon</td>
<td>4</td>
<td>2.6</td>
</tr>
<tr>
<td>Explorer-11</td>
<td>octagon</td>
<td>2.83</td>
<td>2.6</td>
</tr>
<tr>
<td>Explorer-17</td>
<td>sphere</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td>Hubble</td>
<td>cylinder</td>
<td>4</td>
<td>3.33</td>
</tr>
<tr>
<td>OSO-7</td>
<td>9-sided</td>
<td>3.67</td>
<td>2.9</td>
</tr>
<tr>
<td>OSO-8</td>
<td>cylinder</td>
<td>4</td>
<td>3.76</td>
</tr>
<tr>
<td>Pegasus-3</td>
<td>cylinder</td>
<td>4</td>
<td>3.3</td>
</tr>
<tr>
<td>Landsat-1</td>
<td>cylinder</td>
<td>4</td>
<td>3.4</td>
</tr>
<tr>
<td>ERS-1</td>
<td>box</td>
<td>4</td>
<td>4</td>
</tr>
<tr>
<td>LDEF-1</td>
<td>12-face</td>
<td>4</td>
<td>2.67</td>
</tr>
<tr>
<td>HEAO-2</td>
<td>hexagon</td>
<td>4</td>
<td>2.83</td>
</tr>
<tr>
<td>Vanguard-2</td>
<td>sphere</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td>Skylab</td>
<td>cylinder</td>
<td>4</td>
<td>3.5</td>
</tr>
<tr>
<td>Echo-1</td>
<td>sphere</td>
<td>2</td>
<td>2</td>
</tr>
</tbody>
</table>

Table 4.2: Drag Coefficients of Various Satellites
It is apparent that the ballistic coefficient varies only by a factor of 2 regardless of the spacecraft’s shape or orientation. It may be possible to alter the drag coefficient of a satellite slightly by using exotic materials as well. The remaining option to be considered with regards to decreasing the ballistic coefficient of spacecraft is to increase the satellite’s area.

4.4 Increasing the Effective Area

While the ideas of reducing the relative mass of a satellite or increasing the drag coefficient do not seem to be promising concepts, increasing the area to mass ratio presented by the satellite in the direction of travel may hold more potential. The following plot, Figure 4.4, displays the deorbit rate for two 1 kg satellites at solar minimum.

![Figure 4.4: Deorbit Comparison of Two 1-kg Spacecraft](image)

**Figure 4.4: Deorbit Comparison of Two 1-kg Spacecraft**
Both satellites begin with an initial altitude of 225 km. These plots were produced assuming an average solar condition and average F_{10.7} value ($118.7 \times 10^{22} \text{ W m}^{-2} \text{ Hz}^{-1}$). The equation of motion was integrated over the lifetime of the spacecraft as in previous plots. The only difference between the two satellites is that one satellite presents an effective area in the direction of travel five times greater than the other satellite. It is obvious that the satellite with the larger area to mass ratio deorbits much more rapidly. It deorbits approximately six times faster from the initial orbit of 225 km.

Plots can also be produced to determine from what altitude a spacecraft will deorbit within one year. A plot, figure 4.5, was produced for an “average” density satellite over a range of masses. The area below the lines represents altitudes from which the satellite will deorbit within one year for an unaffected satellite, a satellite with an area increase of 2 times, and an area increase of 5 times. For an average density, spherical satellite, a mass of 1 kg represents a satellite with a diameter of approximately 30 cm.

![Graph showing altitude deorbit for different areas and masses.](image)

**Figure 4.5: Altitude Below which Average Density Satellites Deorbit within One Year**

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Because silicon satellites are approximately 30 times denser than an average satellite, the area of the satellite must be increased by a greater amount to produce the same results. Figure 4.6 is a plot of mission class in which a silicon satellite will deorbit within a year for an unaffected satellite, a satellite with an area increase of 5 times, and a satellite with an area increase of 100 times. For a silicon, spherical satellite, a mass of 1 kg represents a satellite with a diameter of approximately 10 cm.

![Figure 4.6: Altitude Below which Silicon Satellites Deorbit within One Year](image)

The three preceding plots indicate that increasing the effective area presented by satellites into the flow direction can significantly alter the ballistic coefficient and thus the deorbit rate of spacecraft. The next issue is how to accomplish the increase in a satellite’s effective area.
Meyer and Chao compared drag enhancement devices to thrusters in a recent article in the Journal of Spacecraft and Rockets [28]. They look at four options for atmospheric disposal of spacecraft within 25 years: chemical propulsion, low-thrust (electric) propulsion, drag enhancement devices (balloon), and a combination of chemical propulsion followed by balloon deployment. They compared the different options on the basis of weight added to the spacecraft to perform atmospheric disposal within 25 years. Their conclusion is that the low thrust option requires significantly less additional weight than the other options. They also found that a balloon is competitive with chemical propulsion for low initial altitudes with regard to weight. The combination of chemical propulsion and balloon requires less added weight than either option alone for spacecraft with high mass-to-area ratios. Figure 4.7 displays the option selected by the authors on the basis of ballistic coefficient (ballistic coefficient, $\beta$, as defined by Meyer and Chao is actually $1/\beta$ as defined in this thesis) and initial altitude [28].

![Figure 4.7: Regions where the Explored Disposal Methods are the Most Weight Efficient](image)

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In the next chapter similar charts are presented based on data from the work done in this thesis. One difference is that this thesis plots the data on a scale of initial altitude versus satellite mass. Initial altitude versus mass is used to allow quick reference by mission designers. Several other differences exist in the approach used by this work as opposed to Meyer and Chao. This thesis looks at disposal of spacecraft within a time span of one year as opposed to 25 years. It is believed that a 1-year disposal horizon is more reasonable in an effort to decrease the orbital debris population. Another difference is that this thesis focuses on disposal of small, microspacecraft where Meyer does not focus on a particular size of spacecraft.

As mentioned before, the ballistic coefficient, $\beta$, in Meyer and Chao is $1/\beta$ as presented in this thesis. A cubical, 10 kg silicon satellite would have a ballistic coefficient of approximately 0.005 m$^2$/kg as defined by Meyer and Chao. For this ballistic coefficient, Meyer and Chao conclude that the best method of disposal by atmospheric reentry is to use a balloon up to an altitude of approximately 920 km. Using a balloon in collaboration with chemical thrusting raises the useful altitude of balloons to over 1100 km. For small dense satellites, Meyer and Chao indicate a large range of LEO altitudes for which the use of balloons are the most efficient method of disposal.

An average density satellite as defined in this thesis would have a ballistic coefficient of approximately 0.023 m$^2$/kg for a 100 kg satellite. A satellite with this ballistic coefficient falls on the boundary between missions where the most efficient method is "balloon or
chemical” versus balloon alone. So, it is clear that Meyer and Chao conclude that there is a range of mission classes in which the use of balloons to achieve atmospheric reentry is the most efficient method.

4.5 Method of Increasing the Effective Area

There are a number of possibilities by which the effective cross-sectional area of a satellite can be increased at the end of life to increase deorbit rate. Three methods will be explored in this study. First, deployment of a “drag parachute” will be considered. Then, inflation of a balloon around the spacecraft will be explored. Finally, the use of solidifying foams is a possibility to be considered.

4.5.1 Deployment of a Drag Parachute

The most obvious possibility is to deploy or release a “drag parachute” behind the spacecraft that will become rigidized either by material properties or pressurization. The drag parachute could be connected to the satellite by Kevlar (or a similar material) straps or tubes. Such an idea is sketched in Figure 4.8.
Figure 4.8: Schematic of a Satellite and Deployed Drag Parachute

The material used for the drag parachute must be light and able to withstand atomic oxygen erosion. A material developed by Triton Systems, Inc. shows promise for such applications. The material is called TOR (Triton Oxygen Resistant) and appears in tests to be approximately 20 times more AO resistant than Kapton. Tumbling of the satellite is a concern for this type of architecture. If the satellite tumbles, it might be possible for the parachute to get tangled around the satellite. A possible remedy would be pressurizing the connecting tubes between the satellite and drag parachute and the tubes around the perimeter of the parachute. In the vacuum of space, very little pressure would be needed to pressurize the narrow tubes so that the parachute would “inflate.” The pressurization would prevent the parachute from wrapping around the spacecraft.
Another concern is that the satellite and drag parachute will not attain the desired orientation as illustrated in figure 4.7. It is desirable for the parachute to present the maximum amount of effective cross-sectional area into the direction of travel. There are two forces acting on the satellite/parachute system: atmospheric drag and gravity gradient. These forces act on the system as illustrated in figure 4.9. Atmospheric drag will cause the system to be oriented as desired (and as illustrated in figure 4.8). The gravity gradient will cause the system to be oriented in such a way that drag and deorbit rate are minimized. Figure 4.8 shows the system in the worst possible orientation for increasing deorbit rate.

![Direction of Travel Diagram](image)

**Figure 4.9: Forces on Satellite/Parachute System**

The spacecraft is assumed to be cubical with 10 cm sides. The drag parachute is assumed to be 50 cm² with pressurized tubes 0.5 cm in diameter. The distance between the drag parachute and satellite is 20 cm in this example. It is possible to determine whether...
atmospheric drag is sufficient to rotate the spacecraft system into the desired orientation. The force of drag \( D \) will be stronger on the parachute than on the satellite while the gravity gradient force \( G \) will be greater on the satellite than on the drag parachute. \( \Delta D \) (the rotating force) and \( \Delta G \) (the restoring force) can be calculated at various altitudes. The calculations in table 4.2 assume that the system is initially in the orientation depicted in figure 4.9.

<table>
<thead>
<tr>
<th>Altitude (km)</th>
<th>( \Delta Gravity ) (m/s^2)</th>
<th>( \Delta Drag ) (m/s^2)</th>
<th>( \Delta Drag / \Delta Gravity )</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>7.33 ( \times 10^{-7} )</td>
<td>0.1136</td>
<td>155,000</td>
</tr>
<tr>
<td>200</td>
<td>7.00 ( \times 10^{-7} )</td>
<td>4.32 ( \times 10^{-5} )</td>
<td>61.7</td>
</tr>
<tr>
<td>300</td>
<td>6.69 ( \times 10^{-7} )</td>
<td>1.96 ( \times 10^{-6} )</td>
<td>2.6</td>
</tr>
<tr>
<td>400</td>
<td>6.40 ( \times 10^{-7} )</td>
<td>1.73 ( \times 10^{-7} )</td>
<td>0.27</td>
</tr>
</tbody>
</table>

Table 4.3: Differential Accelerations Due to Gravity Gradient and Drag

The table illustrates that drag forces are stronger than gravity gradient on the system up to approximately 340 km. These calculations are conservative. In reality, the drag parachute will most likely be larger relative to the satellite (especially for silicon satellites). Also, these calculations assume solar minimum conditions and thus minimum drag. Therefore, the spacecraft will rotate into the desired orientation over a period of time.

4.5.2 Inflate a Balloon Around the Spacecraft

Another idea for increasing the area of the satellite is to inflate a balloon around of behind the satellite. The immediate advantage that a balloon has over the drag parachute idea is that there will be no concern over the orientation of the system if the balloon is circular. Random rotations of the system will not diminish the effective area presented by the balloon in the
direction of flow. Figure 4.10 shows how a balloon may be deployed behind a spacecraft. A pressure of 0.1 psi is sufficient to assure that the balloon will maintain its shape, assuming some leakage, down to an altitude of 120 km [29].

Figure 4.10: Deployment of a Balloon Behind a Spacecraft

Balloons in space are not a new idea. In the 1960s the United States deployed several balloon satellites. Ironically, they were used to make measurements of the Earth’s atmospheric density by measuring the effect of atmospheric drag on the balloon’s orbit. Another use for balloons in space is to shroud warheads inside a balloon as a countermeasure to anti-ballistic missile defense. The thermal effects of a warhead inside a balloon were studied as part of a technical assessment of the proposed national missile defense system. In this context, a series of balloons could be deployed in space in a ballistic missile attack in which most of the balloons are decoys meant to fool the defense system while other balloons contain one or more warheads [29].
4.5.3 Use of Solidifying Foams

Releasing or deploying a foam that would solidify around the satellite is another option. The use of solidifying foam would also eliminate the tumbling concern. An idea for the deployment of the foam might be similar to the "digital propulsion" idea in that there could be numerous small cavities of compressed material. The cavities would release the foam making the spacecraft look like a porcupine with puffy quills. These foam quills would serve to dramatically increase the area presented by the spacecraft in the direction of flow.

4.6 Retrieval or Refurbishment

Another issue that has not been considered very extensively in this thesis is what to do with spacecraft when they fail. When a spacecraft runs low on fuel or a components fails, typically 99% of a satellite’s capabilities are still operational. Therefore an alternative to disposal could be to use automated on-orbit servicing in order to replenish consumables and replace or upgrade satellite parts. Re-using existing resources on failed satellites would reduce the rate of orbital debris creation.

However, satellite refurbishment would not eliminate the need for efficient disposal methods. Even if a spacecraft can be refurbished, there will always be a physical limit on the useful life of space platforms. In addition, on-orbit servicing is likely to produce new types of space debris such as debris from the “servicer” launch, servicer spacecraft, and failed parts removed from satellites. Finally, on-orbit servicing will most likely only be implemented for space missions in which it is cheaper to refurbish than replace (such as missions to the Hubble Space Telescope and ISS). There is likely to be a minimum bound, in terms of
spacecraft mass or cost, under which refurbishment is not worthwhile, leaving a whole category of spacecraft (particularly small spacecraft) that need to be disposed of at the first failure.
Chapter 5

Most Efficient Methods of Spacecraft Disposal

In this chapter, satellite disposal methods are compared on a mass basis. This chapter is the culmination of the groundwork that has been developed in the previous chapters. A method of disposal is considered to be the most efficient method if it can accomplish spacecraft disposal from LEO by atmospheric reentry within one year using the least amount of added mass to the spacecraft. A solar minimum condition ($F_{10.7} = 65.8 \times 10^{-22} \text{ W m}^{-2} \text{ Hz}^{-1}$) was used for all calculations making the plots conservative for natural orbital decay, drag parachutes, and tethers. Figures 5.1 and 5.2 detail the most efficient methods of disposal for average density and silicon satellites. The plots allow a mission designer to determine the most efficient disposal method based on the end-of-life altitude and mass of the spacecraft. The mass scale for silicon satellites is smaller because it is believed the silicon satellites will enable a smaller class of satellites. It is believed that silicon satellites will be useful in the 100-kg class and smaller.

Figure 5.1 is a plot of the best disposal method for average density spacecraft. Average density spacecraft have been defined as those that have a density of 79 kg/m$^3$, the average density of spacecraft launched between the years of 1978-1984.

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Figure 5.1: Most Efficient Disposal Options for Average Density Spacecraft

The plot includes mission types where natural orbital decay, boosting to a storage orbit using thrusters or tethers, deorbiting using thrusters or tethers, and deploying a drag parachute or balloon for disposal all make the most sense as a method of spacecraft disposal depending on the end-of-life altitude and mass of the spacecraft. Boosting to a storage orbit should be considered as a temporary solution and not a long-term method of spacecraft disposal. It is noted that with current tracking capabilities, average density spacecraft with dimensions less than 10 cm (approximately .1 kg) should not be deployed as they will be untrackable.

Like Meyer and Chao, this research discovered a range of missions where the use of a drag enhancement device such as a drag parachute or balloon proved to be the most mass-efficient
method of disposal. But unlike Meyer and Chao, this work considered a time horizon of disposal of one year as opposed to 25 years. Plots in this work assume a solar minimum condition. With that assumption, the altitude below which a satellite will deorbit within one year is determined to be 310 km. During solar maximum, a drag parachute is the most efficient method of disposal up to an altitude of approximately 350 km. It is also noted that at solar maximum the altitude from which a spacecraft will deorbit unaided by natural orbital decay within one year is approximately 40 km higher as well. A boundary is included between a drag parachute or balloon composed of material that is currently available and a minimum thickness drag parachute or balloon composed of material completely resistant to atomic oxygen. In this research, the material considered for the drag parachute was the same as for a balloon.

Both chemical and electric propulsion systems are included in the “current thruster” and “next generation thruster” category. Decisions on what thrusting system to use should be based on the thrust and total mission ΔV needed for the entire mission as outlined in chapter 3. A boundary is included between thrusters that are currently available and those that must be developed. The reason that current thrusters cannot be used efficiently on small microsatellites is that the propulsion system would make up a prohibitively large percentage of the spacecraft mass. Work being done to scale thrusting systems to smaller sizes was considered in chapter 3.

This research determined that the use of electromagnetic tethers is the most efficient method of disposal for all spacecraft with masses greater than approximately 1000 kg. Because such
a large percentage of the weight associated with an electromagnetic tether system is the deployer, the added mass needed to scale a tether system to larger sizes is much less than the mass needed to add extra propellant for a thrusting system.

Determining the most mass-efficient disposal method was accomplished by modeling the various methods of disposal in a spreadsheet program. Mass must be added to spacecraft in the form of hardware, fuel, and necessary systems to accomplish deorbit. These masses were computed based on assumptions detailed in previous chapters. After computing the added mass required to deorbit a spacecraft within one year from LEO altitudes, the mass added was compared for the different methods as a percentage of the total spacecraft mass. The method that added the lowest percentage of additional weight to the spacecraft was considered to be the most efficient method. It is believed that plotting the results on a scale of end-of-life altitude versus mass would be most beneficial to a mission designer who can now determine the most mass-efficient method of disposal quickly.

Figure 5.2 is a plot of the best disposal method for satellites constructed exclusively of silicon.
Figure 5.2: Most Efficient Disposal Options for Silicon Satellites

The mass scale in this plot is smaller than in figure 5.1 because the silicon satellite concept enables smaller satellites and will most likely not be used in the construction of large spacecraft. The altitude scale is also smaller to show greater detail. For silicon satellites, like average density spacecraft, a satellite with dimensions less than 10 cm (or approximately 3 kg for silicon satellites) is untrackable and should not be deployed.

There is currently no efficient method for deorbiting silicon satellites above approximately 310 km in altitude. Current propulsion systems would make up a disproportional percentage of the total mass of the spacecraft. Continued development of micropropulsion technology is necessary to provide a method of disposing silicon satellites at higher LEO altitudes.
A drag parachute or balloon is the most effective method of deorbiting a silicon satellite below 310 km in altitude. Above approximately 310 km, the mass of the drag parachute system would total more than 10% of the total mass of the spacecraft. Drag parachutes can be used above 310 km, but the mass of the system will increase rapidly for deorbit within one year. Figure 5.2 assumes a solar minimum condition. At solar maximum, the altitudes below which a drag parachute or natural orbital decay will deorbit a spacecraft within one year can be raised approximately 40 km. Meyer and Chao’s work indicates that balloons are the most efficient method of disposal for spacecraft of this class up to an altitude of over 900 km. They come to that conclusion because they look at a 25-year time horizon that would include more than two complete solar cycles. Once again, a boundary is included in figure 5.2 between a drag parachute composed of material currently available and a material that would be completely impervious to atomic oxygen erosion.
Chapter 6

Conclusions

6.1 The Need for Spacecraft Disposal

In more than four decades of launching spacecraft into orbit, many tons of man-made materials have been added to the natural debris already in orbit. As the human species relies more and more on our space infrastructure for communication, navigation, earth sensing, weather tracking, observation, and science, orbital debris will become increasingly threatening.

Throughout this thesis, it has been demonstrated that orbital debris is a problem that will only increase in the magnitude with time unless something is done about it. A trend toward distributed constellations of satellites will magnify the proliferation of orbital debris unless satellites are designed with end of life disposal in mind.

To prevent a continued increase in the orbital debris population, three things must be done. First, spacecraft must be designed to minimize the inadvertent release of material into space. Ed White’s glove was one example of this while bolts, paint chips, and loose pieces of metal are a more pressing concern. Secondly, all explosions of rocket bodies must be eliminated.
Progress has been made in this area, but the complete elimination of on-orbit explosions by all space faring nations is necessary. Finally, spacecraft must be designed with a means for disposal at the conclusion of their useful lifetime.

In this thesis, it was demonstrated that a variety of methods of disposal prove to be the most efficient method based on a satellite’s end-of-life altitude and mass. For LEO spacecraft with propulsion systems already on board for reasons such as station keeping and attitude control, the best method of disposal is most likely adding additional propellant to achieve reentry at the end of the mission. For very heavy spacecraft (such as International Space Station) and very low altitude missions, it is better to use tethers and natural orbital decay, respectively.

Although thrusters are still the dominant method of achieving disposal by atmospheric reentry, the use of tethers and drag enhancing devices are the most efficient method in certain mission classes. Tethers are the most efficient method of disposing spacecraft with masses greater than 1000 kg. The use of a drag enhancement device such as a balloon or drag parachute is the most efficient method for disposal of low LEO missions.

For small microspacecraft, there is currently no efficient method for disposing spacecraft above 310 km in altitude within one year. Current thruster technologies simply do not scale to the small sizes needed to make them a viable option from the point of view of mass added to the mission. Thruster systems must be developed to enable efficient disposal of small microspacecraft.
Also, spacecraft with dimensions smaller than 10 cm in LEO are untrackable by U.S. Space Command. Missions should not be deployed with spacecraft in this size range as they are an even greater threat to future space missions than larger trackable objects.

6.2 Suggestions for Continued Research

It has been pointed out in this thesis that the state and future of orbital debris is a topic upon which there is much disagreement. There is little consensus in the aerospace community as to the current threat of orbital debris or to the extant that that threat will increase in the future as the result of continued space launches. More work and observation must be done to more fully understand the situation.

Another area of suggested continued research is the area on the plots in the previous chapter labeled “Next Generation Thruster.” Developing propulsion systems on the small scales required for microspacecraft is a challenge that must be addressed if microspacecraft are going to have the capabilities of station keeping, attitude control, orbit transfer, and disposal by atmospheric reentry. Promising ideas and the development of MEMS have been highlighted in this thesis as the beginning of the quest to develop propulsion technologies for small spacecraft.

Much more work must also be done on developing the concept of drag enhancing devices such as a drag parachute or balloon. Building prototypes and testing prototypes of drag parachutes and balloons in space will be necessary before these methods will be relied upon
as viable methods of spacecraft disposal.
References

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16. Significant information on silicon satellites can be found on the Aerospace Corporation’s web site (www.aero.org/home.html).
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27. Information on self-consuming satellites can be found at the home page of the Air Force Research Laboratory (www.afrl.af.mil).