FEASIBILITY OF MICROSATELLITE ACTIVE DEBRIS REMOVAL
SYSTEMS

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ABSTRACT

Feasibility of Microsatellite Active Debris Removal Systems

Karsten James

Space debris has become an increasingly hazardous obstacle to continued space-flight operations. In an effort to mitigate this problem an investigation of the feasibility of a microsatellite active debris removal system was conducted. Through proposing a novel concept of operation, utilizing a grapple-and-tug system architecture, and by analyzing each resultant mission phase in the frame of a representative example, it was found that microsatellite scale systems are capable of fulfilling the active debris removal mission. Analysis of rendezvous, docking, control and deorbit mission requirements determined that the design of a grapple-and-tug system will be driven by sizing of the propellant required to deorbit the target vehicle. Further sensitivity analysis determined that target altitude and mass are critical factors in determining the capabilities of a microsatellite mission. Preliminary sizing demonstrated that hardware considerations for both satellite core and mission related activities do not impede microsatellite feasibility. Further investigation of microsatellite debris removal missions including detailed design analysis and engineering is suggested.

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Chapter 1

Introduction

Fifty-six years of space development, exploration and commercialization has left an increasingly prevalent and hazardous legacy of human space debris scattered in Earth orbit. This debris ranges in size from paint flecks and spent rocket propellant on the order of a millimeter to entire satellites and rocket bodies of tens of meters and hundreds of kilograms. This debris environment poses a real danger to current space operations and the projected growth of the problem threatens the viability of future missions. Due in part to the enormous domain of the problem, encompassing debris of multiple orders of magnitude in scale, and spanning all orbital regimes, no fully posed mitigation strategy has emerged. Through careful analysis of the problem and by narrowing the concept of operations a complete active debris removal solution is proposed.

1.1 Debris Environment

The NASA Orbital Debris Program Office has undertaken detailed studies of the current and projected human debris environment. In a paper titled *Collision*

Figure 1.1: Debris Environment Population Projections

Activities in the Future Orbital Debris Environment [14] a detailed breakdown of the history and future projections of the debris environment is developed as seen in Figure 1.1. This projection predicts the continued growth of the orbital debris problem with low earth orbital debris accounting for approximately half of the entire debris population and subject to higher levels of projected growth. Further findings expressed in the same paper indicate that fragments from on-orbit collisions of existing debris will come to dominate the debris environment. Figure 1.2 shows the projected growth of collision fragments in low earth orbit. Taken together these results indicate that the LEO regime will be most impacted by the future growth of the debris environment and that the primary driver of this growth will be on-orbit collisions between existent debris.

An additional paper by the Orbital Debris Program Office, *A Sensitivity Study on the Effectiveness of Active Debris Removal* [15], investigates the effect of active debris removal or ADR in the LEO environment. Projections of the debris environment given an objective removal rate of 5, 10 and 20 objects per year are

Figure 1.2: Debris Environment Population Composition Projection

shown in Figure 1.3. A removal rate of as few as 5 objects per year has the ability to greatly curtail the future growth of the debris environment. The decrease in the future growth projections is due primarily to the reduction in projected debris collisions, achieved by removing objects that pose a significant collision risk.

Figure 1.3: Sensitivity Study of Active Debris Removal Effectiveness

The space debris environment poses a serious challenge to be met by any mission attempting to mitigate the problem. Active debris removal presents a reasonable solution to curb the growth of the future debris environment by reducing on-orbit collisions and subsequent secondary collisions with fragment debris. However in order to have a substantive effect on debris populations a requisite number of target vehicles must be removed. Additionally systems must meet NASA's 25 year deorbit rule which states that any mission must deorbit within 25 years of completion of mission. Together the capability of achieving a net removal rate of 5 objects per year and meeting NASA's 25 year rule must be primary requirements of any ADR system for it to be effective.

Chapter 2

System and Mission Architecture

The scope of the orbital debris problem allows for a wide variety of potential mission and system architectures. However due to the variety inherent in these differing mission concepts, analysis of any potential ADR mission is highly dependent on the architecture selected. Architecture analysis is therefore an important aspect of any analysis of ADR missions. The need to achieve a net 5 object per year removal rate, meet NASA's 25 year deorbit rule, and examine a system that can be fully analyzed are primary design requirements that serve to drive the architecture selection decisions.

2.1 System Architecture Selection

Multiple methods of active debris removal have been proposed as the problem of space debris has grown more acute. Generally the systems proposed belong to one of two fundamental system architecture types. The first type is that of a small single use spacecraft, such as the tether system proposed by Kawamoto et al. in *Precise Numerical Simulations of Electrodynamic Tethers for an Active*

Debris Removal System [11]. The second primary type is that of a large spacecraft capable of deorbiting multiple target vehicles, such as the ion shepherd spacecraft as proposed by Bombardelli et al. in *Ion Beam Shepherd for Contactless Space Debris Removal* [2]. Both system architectures present benefits and pitfalls for any serious ADR mission. An analysis of the system architectures is highlighted in the trade study shown in Figure 2.1.

System Type	Single Use	Multiple Use
System Complexity	2	1
Impact	1	2
System Lifetime	2	1
Capability	1	2
Cost	2	1
Deployment	2	1
Total	10	8

Figure 2.1: Trade Study of System Architectures Types

The trade studies used for the system and mission architecture analysis provide a convenient way to organize and display the decisions that were made to focus the investigation. The n investigated items are rated from n to 1 with n being the best and the categories in which they are compared are equally weighted. A small single use system allows for a significantly reduced complexity when compared to a system that is required to maneuver to and deorbit multiple target vehicles. In addition the short mission lifetime of such a system can further reduce complexity by eliminating the need to account for long mission life effects of the space environment such as radiation and material degradation. These considerations together with the launch costs associated with flight hardware, significantly reduce the unit cost of a smaller system when compared with large complex al-

ternatives. Additionally a small enough system can be deployed as a secondary payload. A secondary payload capability both reduces the problems associated with the future of hardware used to deploy an ADR mission and greatly increases the potential launch opportunities for an prospective ADR mission.

Despite these advantages large systems do have several distinct advantages. Large systems have a much greater impact on the debris environment since they can remove multiple target vehicles over their lifetimes. A large system architecture also provides a better platform for expanded system capabilities which increases the flexibility of the system to deal with unexpected situations. Despite these advantages through the use of an effective concept of operations small systems can provide the required capability to complete the ADR mission.

Several methods of completing the ADR mission have been proposed including grapple-and-tug, particle pressure and drag device systems. Drag device systems include both solar sails and tethers. Both of which induce a drag like force on the target through either interaction with the earth's magnetic field or by increasing the effective area of the target, increasing the force due to atmospheric drag and solar radiation pressure. An example of a drag device system is the ElectroDynamic Debris Eliminator proposed in *Wholesale Debris Removal from LEO* [12]. Particle pressure systems use an emitter to shine a beam of either ions or photons at the target and use the pressure from these particles to affect a deorbit. An example of a particle pressure system is the ion shepherd spacecraft [2]. Grapple-and-tug systems rely on conventional propulsion and transfer maneuvers to deorbit the target. Propulsion options for grapple-and-tug systems include both impulsive and low thrust propulsion options. *Active Debris Removal Mission Design in Low Earth Orbit* [17] provides a detailed survey of currently proposed grapple-and-tug proposals.

Of the three most commonly proposed methods of active debris removal for large targets, grapple-and-tug provides the most accessible solution. A trade study outlining this design decision is shown in Figure 2.2. Grapple-and-tug and drag devices both require the spacecraft to dock with the target vehicle however grapple-and-tug uses well proven technologies to affect a deorbit of the target while drag devices such as solar sails and tethers remain relatively unproven. Additionally grapple-and-tug systems only need to function until the primary deorbit burn is made. Both drag device and particle pressure systems require continued operation throughout the deorbit process in order to remove the target.

Systems that use particle pressure, either ions or photons, to affect a deorbit of the vehicle pose additional challenges of station keeping and target tracking as they deorbit the target. Additionally these systems have never been demonstrated in any flight operations and their deployment remains highly theoretical. Grapple-and-tug reduces risk and allows for the most complete system analysis though its use of mature technologies and its relatively simple operational procedures. Despite this a complete analysis of all aspects of a grapple-and-tug mission has yet to be performed. Completing a complete analysis is the primary goal of this analysis.

ADR Method	Grapple-and-Tug	Particle Pressure	Drag Device
Flight History	3	1	2
System Complexity	3	1	2
Target Dependence	2	3	2
System Lifetime	3	1	2
Development Uncertainty	3	1	2
Completeness	3	2	2
Total	21	10	14

Figure 2.2: Trade Study of Active Debris Removal Methods

2.2 Mission Development

The projections of the future orbital debris environment and the ability for active debris removal methods to substantively impact the future growth of debris populations are critical factors in the architecture selection of any ADR mission. Due to the projected debris environment discussed earlier, concentrating on active debris removal of high collision risk objects in LEO will yield the greatest benefit in mitigating the debris problem. Figure 2.3 outlines the motivation behind selection of the LEO operating regime for potential ADR activity. Additionally focusing on debris that poses a significant risk of fragmentary collisions narrows target candidates to objects such as spent rocket bodies and dead satellites. The fundamental mission of an effective ADR system will then be to actively remove target vehicles that pose a risk of an on-orbit collision in LEO.

Target Regime	LEO	MEO	GEO
Need	3	1	2
ADR Impact	3	2	1
Accessibility	3	1	2
Difficulty	3	2	3
Total	12	6	8

Figure 2.3: Trade Study of Orbital Regimes

Target Type	Rocket Body	Satellite
Population	1	2
ADR Impact	2	2
Accessibility	2	1
Uncertainty	2	1
Difficulty	2	1
Total	9	7

Figure 2.4: Trade Study of Potential Target Types

Narrowing target vehicle candidates to large objects in LEO and selecting a grapple-and-tug removal strategy reduces the number of mission parameters that need to be considered and introduces further system requirements. However the mission architecture is further refined by choosing to consider only rocket bodies as candidate targets. Figure 3.1 provides detail of the decision to narrow the candidates targets to rocket bodies. The inherent similarities between all rocket bodies reduce the effects and complexities of considering a specific target. Rocket bodies share similar geometries, on-orbit dynamics and features which make them good candidates for a detailed analysis. Different target vehicles can be quickly and easily considered without the need to model these specific geometries or features such as solar arrays and booms. Rocket bodies also present consistent and more predictable dynamics on orbit due to their simple geometry. Rocket

bodies also share a standard set of components such as engine nozzles which can be exploited for use as a place to dock. The prevalence of rocket bodies on orbit makes them good candidates for removal and removing them at a sufficient rate will positively impact the future debris environment [19].

The scope of the debris environment problem requires that considerations be made for specific mission and system architectures. Making architecture selections that retain a benefit to the debris environment evolution allows for the development of a concept of operations that can serve as a basis for analysis and feasibility assessment. A small grapple-and-tug system focused on deorbiting rocket bodies in LEO provides a feasible system architecture for an ADR mission that can meet the net removal of 5 objects per year requirement and allows for a complete analysis using existing technologies.

Chapter 3

Concept of Operations

The selection of the system and mission architecture informs the concept of operations or CONOPS which in turn drives system and subsystem requirements and serves as a basis for specific analysis tasks. The selection of a small, single use, system architecture allows for several unique CONOPS possibilities that serve to help satisfy the primary ADR mission requirement of deorbiting 5 targets a year. Microsatellites, spacecraft with a gross mass of under 100kg, are able to be flown as secondary payloads on a variety of currently operated launch vehicles [21]. However an expansion of secondary mission limitations to include fueled payloads will be needed prior to any ADR mission being flown as a secondary payload. This functionality creates the potential for nearly any launch to carry one or more secondary ADR missions. This capability mitigates a situation where the launch of an ADR mission itself contributes to the debris problem by introducing a new rocket body into the environment. While the primary payload launch vehicle will potentially remain in orbit the responsibility to meet existing debris mitigation guidelines will rest with the primary mission. Lewis et al. [13] conducted a sensitivities analysis of the impact of new launches associated with ADR missions

and found that ADR is still beneficial, despite increased launch activity, assuming that launch vehicles meet debris mitigation guidelines.

The microsatellite scale of the proposed system opens additional possibilities for launches. If multiple ADR spacecraft were treated as a primary payload it would be possible that potentially many target vehicles could be removed with the need for only one launch [21]. This type of mission profile could prove extremely beneficial even if the ability to meet debris mitigation guidelines on the primary launch vehicle does not exist. Flying standardized hardware for a variety of potential targets also allows the system to benefit from economies of scale, reducing unit cost for each successful target removal.

3.1 Mission Tasks

Regardless of the launch profile flown the concept of operations for a grapple-and-tug mission can be broken down into four general mission phases which will drive the design of the system: rendezvous, docking, control and deorbit. These mission phases will serve as the basis for a preliminary analysis of system delta-v requirements. The system delta-v requirements will serve as the principal metric of this survey and will be used to drive system sizing and feasibility assessments. Each of these phases is composed of several succinct tasks each of which can be analyzed independently to build up the system delta-v requirements.

The rendezvous mission phase maneuvers the spacecraft from its initial insertion orbit to a position where it can grapple the target object. In order to accomplish this two principal tasks must be accomplished. First the spacecraft must complete an orbital transfer from its insertion orbit to the orbit of the target vehicle. Secondly the spacecraft must complete a phasing maneuver to

match its position within the target orbit. A variety of orbital maneuvers may be needed to complete either or both of these fundamental tasks. These maneuvers may include both in and out of plane transfers as well as phasing maneuvers. The specific maneuvers required to complete a given ADR mission will be highly dependent on the specific mission profile flown and on the target and insertion vehicles.

The docking mission phase moves the spacecraft from the approach position to the point where it has completed a hard dock on the target vehicle. In order to achieve a dock three primary tasks must be accomplished. First the spacecraft must match the orbital motion of the target vehicle. Secondly the spacecraft must match the relative motion of the target vehicle and close to the target vehicle in order to dock. Finally the spacecraft must attach itself to the target vehicle so that it may impart the requisite delta-v to deorbit the target. This task must be completed before the difference in orbital motion between the spacecraft and the target cause them to drift apart. Since orbital effects are neglected for this portion of the analysis the orbital model is not needed however the dimensions of the target and system are relevant to the delta-v analysis.

The control mission phase requires the spacecraft to gain positive attitude control of the target. Following this the spacecraft must orient itself and the target so that it can perform a deorbit maneuver. This aspect of the mission assumes rigid body dynamics of the combined spacecraft-target system.

The final mission phase that will drive the design of the system is the deorbit phase. This phase will consider an impulse deorbit maneuver that will deorbit the target object. This task requires a complete orbital model to understand the effectiveness of deorbit maneuvers and estimate the amount of fuel required to make them. Additionally a sensitivities analysis of the deorbit phase of the

mission is conducted to examine how varying target parameters impact system requirements.

These four mission phases allow for a good estimation of the delta-v requirements of a single use ADR mission. The eight tasks outlined in the mission phases serve as closed analysis problems and represent all critical mission elements of an ADR mission of the selected architecture. These tasks will be analyzed assuming a small system architecture, the use of a launch vehicle to provide an initial in plane orbit, an in-plane tumbling rocket body as a target, and a single impulsive maneuver to deorbit the target. The delta-v analysis in addition to a careful survey of required mission hardware will form a solid base from which a feasibility assessment of the single use mission architecture can be made.

Mission Phase	Mission Tasks
Rendezvous	Match orbit Phase
Docking	Match orbital motion Match relative motion Grapple
Control	Stop latent motion Orient rocket body
Deorbit	Impart delta-v to deorbit

Figure 3.1: CONOPS Summary

Chapter 4

Model Development

In order to conduct an analysis of the proposed concept of operations and allow the maximum flexibility in analyzing specific mission elements across a variety of conditions and target vehicles a model of the LEO system was written in MATLAB. The model was used for analysis of the rendezvous, docking and deorbit mission phases in order to determine the delta-v requirements which were subsequently used to drive the sizing of the spacecraft system. Creating a custom model allowed for a higher degree of flexibility than using an off the shelf model and allowed the model to be tailored to the problem. The subsections of the model were verified using the industry standard Systems Tool Kit software.

4.1 Central Potential

Initial orbits of potential target vehicles were obtained using published radar tracking data in the form of two line orbital elements or TLEs. The classical orbital elements given in the TLE are the inclination i , the right ascension of ascending node Ω , the eccentricity e , the argument of perigee ω , the mean anomaly

Figure 4.1: The Classical Orbital Elements [3]

n , and the mean motion M_e . The inclination, right ascension of ascending node, eccentricity, and argument of perigee are all defined by the geometry of the orbit as seen in Figure 4.1. The mean anomaly is a non-dimensional measure of the angular distance from the point of an object's perigee to its location and the mean motion is a measure of the number of orbits made by an object in one earth day.

Using the classical orbital elements a position and velocity vector in earth centered inertial or ECI coordinates can be determined. The ECI coordinate system is an inertial Cartesian system with its origin at the center of the earth and the X direction in the direction of the first point of Aries as shown in Figure 4.2. One additional transformation is made to earth centered earth fixed or ECEF coordinates, a co-rotating reference frame that remains fixed with respect to the earth's surface and defines x' as the direction from the center of the earth to the intersection of the equator and prime meridian, also shown in Figure 4.2.

Figure 4.2: The Earth Centered Inertial and Earth Centered Earth Fixed Coordinate Systems [3]

The equations of motion for the model are based around the familiar two body point potential problem. The gravitational force exerted on an object near the earth can be written using the familiar,

$$F = -\frac{Gm_1m_2}{r^2}\hat{\mathbf{u}}_r, \quad (4.1)$$

where G is Newton's gravitational constant, m_1 is the mass of object 1, m_2 is the mass of object 2, r is the distance between the center of mass of objects 1 and 2, and $\hat{\mathbf{u}}_r$ is the unit vector from object 2 to 1. Assuming the mass of the earth is much greater than the mass of the target object and substituting Newton's second law,

$$F = \ddot{\mathbf{r}}_2 m_2, \quad (4.2)$$

and

$$\mu = Gm_1 \quad \hat{\mathbf{u}}_r = \frac{\vec{r}}{r}, \quad (4.3)$$

Figure 4.3: Orbit of the Alouette 2 rocket body about the Earth.

yields,

$$\ddot{\vec{r}} = -\frac{\mu}{r^3}\vec{r}, \quad (4.4)$$

the governing equation of motion for the system [3].

Breaking equation 4.4 into components in the ECEF frame we can easily derive the equations of motion in ECEF,

$$\begin{aligned} \ddot{x} &= -\frac{\mu}{r^3}x \\ \ddot{y} &= -\frac{\mu}{r^3}y \\ \ddot{z} &= -\frac{\mu}{r^3}z \end{aligned} \quad (4.5)$$

By iterating the initial position and velocity using equations 4.5 the position and velocity vector of the target object at every time can be generated over any desired time span. Figure 4.3 shows an iteration of the position of a rocket body in low earth orbit over one orbital period.

4.2 Perturbations

In addition to modeling the standard gravitational potential of the Earth several of the largest perturbative effects to the simple central potential were considered as well. These effects can be used to aid in deorbiting target objects, lessening the delta-v requirements on potential debris removal systems.

Drag is by far the largest magnitude perturbation within the LEO regime. Drag at orbital altitudes is caused by the residual atmosphere of the Earth. Due to the high velocities associated with orbiting objects the quartic drag term dominates linear drag resulting in the drag equation for a body of mass m ,

$$a_D = -\frac{1}{2}\rho v^2 C_D A \frac{1}{m} \hat{\mathbf{v}}. \quad (4.6)$$

The coefficient of drag C_D is taken to be 2.2 for objects with uncertain aerodynamic characteristics, the A and velocity v of the object is dependent on its cross sectional area in the velocity direction and relative velocity between the object and the medium. The density of the atmosphere ρ is highly dependent on the altitude of the object and requires a working atmospheric model to determine.

While many atmospheric models of varying complexity exist, the exponential atmospheric model was primarily used to determine the density of the atmosphere for a given altitude. The exponential model provides a reasonable estimate for atmospheric density under average conditions [22]. The density of the atmosphere in the exponential model is governed by,

$$\rho = \rho_0 e^{\frac{-(z-h_0)}{H}}, \quad (4.7)$$

where h_0 , H and ρ_0 are defined for a series of reference altitudes. Adding the drag acceleration term in equation 4.6 to equation 4.4 yields the modified equation of

motion,

$$\ddot{\vec{r}} = -\frac{\mu}{r^3}\vec{r} - \frac{1}{2}\rho v^2 C_D A \frac{1}{m} \hat{\mathbf{v}}. \quad (4.8)$$

Figure 4.4 shows the impact that drag has on an orbiting body. The International Space Station which orbits at a mean altitude of 360km decays by approximately 100 meters a day due to drag and can be used as an additional test the validity of the drag model.

Figure 4.4: International Space Station apogee altitude over one day.

The second largest perturbative effect on objects in low earth orbit is caused by the oblateness of the Earth. The rotation of the Earth causes a bulging of the Earth's equipotential surface near the equator. This bulge causes the radius vector pointing from the Earth's center of mass to diverge from the normal vector of the Earth's equipotential surface. This difference perturbs the orbit of objects as they pass latitudinal over the Earth's surface. This perturbation is given as a disturbing acceleration on the familiar central potential acceleration,

$$\ddot{\vec{r}} = -\frac{\mu}{r^2} \hat{\mathbf{u}}_r + \vec{p}. \quad (4.9)$$

This disturbing acceleration can be resolved into the ECEF coordinates,

$$\begin{aligned} p_x &= -\frac{3J_2\mu R_e^2 r_x}{2r^5} \left[3r_k - \frac{5r_z^2}{r^2} \right] \\ p_y &= -\frac{3J_2\mu R_e^2 r_y}{2r^5} \left[3r_k - \frac{5r_z^2}{r^2} \right]. \\ p_z &= -\frac{3J_2\mu R_e^2 r_z}{2r^5} \left[3 - \frac{5r_z^2}{r^2} \right] \end{aligned} \quad (4.10)$$

These accelerations are dependent on the second zonal harmonic J_2 which is a non-dimensional parameter dependent on the central body that is being orbited [22]. Adding this acceleration to equation 4.8 gives the complete equation of motion for the model,

$$\ddot{\vec{r}} = -\frac{\mu}{r^3}\vec{r} - \frac{1}{2}\rho v^2 C_D A \frac{1}{m} \hat{\mathbf{v}} + \vec{p}. \quad (4.11)$$

The model was iterated forward using Cowell's method and an explicit Runge-Kutta scheme.

4.3 Relative Motion

In order to conduct analysis for rendezvous maneuvers the developed orbital model is recast in a co-moving local vertical local horizon or LVLH frame centered on the target vehicle. Here the δx direction is given in the direction of the target's radius and the δy direction is given in the target's velocity direction. Figure 4.5 gives a representation of this frame with regards to the internal frame, the target A, and the spacecraft B. The equations of relative motion between the spacecraft and the target can be linearized when $\frac{\delta r}{R} \ll 1$ as is the case with a spacecraft and target vehicle in earth orbit. The equations of relative motion when linearized

Figure 4.5: Local Vertical Local Horizon Coordinate System [3]

and resolved into the LVLH frame can be written as,

$$\begin{aligned}
 \delta \ddot{x} &= -\left(\frac{h^2}{R^4} + \frac{2\mu}{R^3}\right)\delta x - \frac{2(V \cdot R)h}{R^4}\delta y + \frac{2h}{R^2}\delta \dot{y} \\
 \delta \ddot{y} &= \left(\frac{h^2}{R^4} - \frac{\mu}{R^3}\right)\delta y + \frac{2(V \cdot R)h}{R^4}\delta x - \frac{2h}{R^2}\delta \dot{x} , \\
 \delta \ddot{z} &= -\frac{\mu\delta z}{R^3}
 \end{aligned} \tag{4.12}$$

where R and V are the familiar inertial radius and velocity vectors respectively propagated forward with respect to all applicable perturbations [4].

Finally for maneuvers on short time scales and close to or attached to the target, orbital motion effects are small and the system is considered from the point of view of a barycentric, co-moving Cartesian coordinate system centered on the barycenter of the target vehicle.

Chapter 5

Mission Analysis

In order to determine delta-v requirements for the ADR mission a representative test case is used to develop a complete delta-v budget and determine the feasibility of a microsatellite scale system and provide a cursory sizing estimate of the completed system. This test case highlights the various analysis tasks consider for determining the feasibility of an ADR mission but is not intended to present an optimal or final design even for the selected target.

The representative target considered is a Thor Agena D SLV-2 rocket body currently located in a 800km, 99° inclination, near circular orbit. The Agena has a ballistic coefficient of, $48.6 \frac{kg}{m^2}$, and a mass of 670kg [23]. Launched in 1964 this target demonstrates the longevity of debris objects and is representative of rocket bodies found throughout LEO. As seen in Figure 5.1 and Figure 5.2 from *A Sensitivity Study on the Effectiveness of Active Debris Removal* [15] the 800km altitude and 100° inclination regimes are local maximum of the debris environment population. This makes removing high collision risks objects a priority for ADR missions. The mission spacecraft is initially placed in a 600km circular orbit in plane with the target object.

Figure 5.2: LEO Debris Population by Inclination.

Figure 5.1: LEO Debris Population by Altitude.

5.1 Rendezvous

In order to rendezvous with the target object a minimum of two burns are required to both match altitude and phase. To accomplish this, the spacecraft must wait a maximum of the synodic period given by,

$$T_{syn} = \frac{T_t T_{s/c}}{T_t - T_{s/c}}, \quad (5.1)$$

before transferring to the target orbit. In this case, the wait time is 1.7 days. Since this is an acceptably short amount of time and transfer windows between LEO orbits will not exceed more than several days the phasing and transfer maneuvers can be combined as a two burn impulsive maneuver [3]. This maneuver is made using Hohmann transfers which results in the most fuel efficient transfer possible. The Hohmann transfers are determined by finding the semi-major axis of the parking, transfer and target orbits using,

$$a = \frac{1}{2}(r_a + r_p). \quad (5.2)$$

From the semi-major axis the velocity of each orbit at each burn on the transfer orbit can be determined from,

$$v = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}}. \quad (5.3)$$

The delta-v required to complete the transfer is computed by finding the difference in velocity between the orbits at the two co-located points. Figure 5.3 shows the transfer maneuvers used to meet the target object in the representative test case.

Computing the delta-v for this particular case gives the result,

$$\Delta v = 98.8 \frac{m}{s}. \quad (5.4)$$

Figure 5.3: Spacecraft to Target Transfer and Phase. Red: spacecraft. Blue: target.

This delta-v is relatively small as the orbits are fairly close and the periodicity of the orbits was used to phase in lieu of a phasing maneuver. However a quick worst case scenario calculation using three burns to both transfer and phase a full half orbit yields,

$$\Delta v = 298.4 \frac{m}{s}. \quad (5.5)$$

It is expected that there will be a great deal of variability in the maneuvers required to rendezvous with the target object. The maneuvers required will depend greatly on the initial insertion orbit and the orbit of the target and may include plane changing maneuvers in addition to in-plane maneuvers. However delta-v costs of rendezvous can be mitigated through careful selection of burn times and by combining maneuvers. Regardless delta-v values on the order of several hundred meters per second are not beyond reason. Despite this, as will be shown later, since the spacecraft maneuvering is relatively small the amount of fuel required to attain this delta-v is not significant when compared to the fuel

mass required to deorbit a target vehicle.

5.2 Docking

Docking with the spacecraft is composed of two fundamental tasks that must be completed in order for a successful grapple to be made. The first task is to match the orbital motion of the target vehicle. The second task is to match the relative motion of the target vehicle.

5.2.1 Obital Motion Analysis

Matching the orbital motion of the target vehicle requires the use the LVLH frame and the associated equations of relative motion. A two impulse rendezvous trajectory is considered beginning from a position 30km in trail of the target. As shown by Goodman et al. [6] and Fehse [4] this position is a typical point from which to begin relative motion considerations during rendezvous operations.

To compute the necessary impulsive maneuvers required to match the relative orbital motion of the spacecraft and the target the target is assumed to occupy a circular orbit. This assumption is made only to compute the required two-impulse maneuver, the maneuver itself is propagated with no circular orbit assumption. However as can be seen in Figure 5.5 the error associated with this assumption for the considered representative target as with other objects in near circular orbits is small. The LVLH frame of a target in a circular orbit represents a special case which is renamed the Clohessy-Wiltshire frame. The Clohessy-Wilshire frame allows the time evolution of the state vectors to be written with respect to the

Clohessey-Wilshire matrices defined as follows [3]:

$$[\phi_{rr}(t)] = \begin{bmatrix} 4 - 3 \cos nt & 0 & 0 \\ 6 \sin nt - nt & 1 & 0 \\ 0 & 0 & \cos nt \end{bmatrix}, \quad (5.6)$$

$$[\phi_{rv}(t)] = \begin{bmatrix} \frac{1}{n} \sin nt & \frac{2}{n}(1 - \cos nt) & 0 \\ \frac{2}{n}(\cos nt - 1) & \frac{1}{n}(4 \sin nt - 3nt) & 0 \\ 0 & 0 & \frac{1}{n} \sin nt \end{bmatrix}, \quad (5.7)$$

$$[\phi_{vr}(t)] = \begin{bmatrix} 3n \sin nt & 0 & 0 \\ 6n(\cos nt - 1) & 0 & 0 \\ 0 & 0 & \cos nt \end{bmatrix}, \quad (5.8)$$

$$[\phi_{vv}(t)] = \begin{bmatrix} \cos nt & 2 \sin nt & 0 \\ -2 \sin nt & 4 \cos nt - 3 & 0 \\ 0 & 0 & \cos nt \end{bmatrix}. \quad (5.9)$$

Figure 5.4 shows an example of a two-impulse rendezvous maneuver within the LVLH frame. Given a desired end state of no relative motion and the spacecraft co-located with the target vehicle the velocities required at the beginning and end of the rendezvous denoted by,

$$[\delta v_0^+] = [\delta u_0^+ \hat{\mathbf{i}} \quad \delta v_0^+ \hat{\mathbf{j}} \quad \delta w_0^+ \hat{\mathbf{k}}] \quad (5.10)$$

and

$$[\delta v_f^-] = [\delta u_f^- \hat{\mathbf{i}} \quad \delta v_f^- \hat{\mathbf{j}} \quad \delta w_f^- \hat{\mathbf{k}}] \quad (5.11)$$

respectively can be solved for. The equations are given by,

$$[\delta v_0^+] = -[\phi_{rv}(t)]^{-1}[\phi_{rr}(t)][\delta r_0], \quad (5.12)$$

and

$$[\delta v_f^-] = -([\phi_{vr}(t)] - [\phi_{vv}(t)][\phi_{rv}(t)]^{-1}[\phi_{rr}(t)])(\delta r_0). \quad (5.13)$$

Figure 5.4: General Spacecraft to Target Two-Impulse Maneuver [3]

The delta-v required to complete the maneuver is simply,

$$\Delta v = \delta v_0^+ - \delta v_0^- - \delta v_f^-, \quad (5.14)$$

since δv_f^+ is required to be zero to match the target motion.

Using this solution to the two-impulse rendezvous problem and propagating the trajectory forward in the standard LVLH frame a rendezvous trajectory from the spacecraft's initial position to the target vehicle is developed. Solving the representative test case from a 30km trail orbit for an eight hour rendezvous results in the rendezvous trajectory seen in Figure 5.5. The delta-v required to accomplish this rendezvous is,

$$\Delta v = 84.4 \frac{m}{s}. \quad (5.15)$$

This maneuver constitutes what Fehse considers the far range rendezvous problem and in normal operations is typically conducted through the use of multiple smaller maneuvers of an open-loop type where the accuracy of each preceding

Figure 5.5: Spacecraft to Target Two-Impulse Maneuver

maneuver is directly evaluated before the next maneuver is performed. However an approximation of a single maneuver can appropriately represent this procedure as the resulted delta-v is in good agreement with similar maneuvers detailed in the NASA technical report titled *History of Space Shuttle Rendezvous* [5]. For the given situation we can reasonable assume that the maneuver is sufficient to place the spacecraft within a few hundred meters of the target established in a position to match the relative motion of the rocket body.

Figure 5.6: Spacecraft to Target Divergence

Upon the completion of the two-impulse rendezvous maneuver the spacecraft and target have maneuvered into effectively the same orbit. As shown in Figure 5.6, obtained using the linearized relative motion equations, the residual orbital motion between the spacecraft and the target is small and can be easily mitigated by small station keeping maneuvers or conglomerated into the delta-v cost of docking maneuvers. For small time scales and for sufficiently small separation distances, here considered to be 15 minutes and 255m respectively, orbital motion is dominated by commanded docking maneuvers and its effect on the relative motion of the target and spacecraft is correspondingly small. The analysis by Kasai et al. [10] of the ETS-7 rendezvous, docking and space robotics experiments

confirms the validity of this assumption. The ETS-7 mission analysis assumed a fixed reference trajectory within 520m of the target satellite with a Clohessy-Wilshire analysis considered for maneuvers beyond 520m.

5.2.2 Relative Motion Analysis

With the orbital motion effectively minimized through the use of a two-impulse Clohessy-Wilshire rendezvous the relative motion of the target body remains to be dealt with. The selection of rocket bodies as the primary mission target helps reduce the complexity of the capture portion of an ADR mission drastically. As demonstrated by Praly et al. in *Study on the Eddy Current Damping of the Spin Dynamics of Space Debris from the Ariane Launcher Upper Stages* [20] natural damping forces in LEO, primarily eddy effects of the earth's magnetic field, drastically reduce body rates on rocket body objects in LEO. An analysis conducted on Ariane upper stages concluded that spin rates are damped onto the order of 2° per second, as seen in Figure 5.7. In addition simulation and analysis suggests that the spin of the target vehicle is likely to decay into a planar flat spin along it's transverse axis, shown in Figure 5.8, as is expected considering the moments of inertia of cylindrical objects. Praly concluded that the target vehicles matching the typical geometries of rocket bodies, thin-walled cylinders, that have spent several years on orbit will exhibit a low magnitude planar spin. While greatly simplifying the dynamics of the capture problem a treatment of the required maneuvers to effectively dock with the target is needed.

Figure 5.7: Rocket Body Spin Rate Decay

As demonstrated by Ma et al. [16] computing an optimal capture trajectory for a tumbling spacecraft is non-trivial. However to assess the feasibility of an ADR mission an optimized solution is not required, only an approximation for the delta-v cost incurred matching the relative motion of the target is needed. To do this an approximation for the capture trajectory of a rocket body was developed. The target was assumed to be spinning in plane at a rate of 2° per second.

Figure 5.8: Axial and Transverse Spin States of Rocket Body

Figure 5.9: Geometry of Simple Discretized Relative Motion

The approximation was created by discretizing the motion of the target and spacecraft into n elements. At the beginning and end of each element the spacecraft performs an impulsive maneuver to a trajectory tangent with the motion of the desired attachment point on the target vehicle. The element is flown at a velocity that matches the time of flight of the spacecraft along the element with the time for the target to rotate through the angle, θ , subtended by the element. Figure 5.9 gives a representation of the geometry of the problem. Since rocket bodies are considered for this analysis the attachment point is taken to be the nozzle of the rocket, located at one end of the rocket body. For sufficiently small elements the motion of the spacecraft very closely approximates both the position and velocity of the end of the rocket body.

In order to find the requisite trajectory for each impulsive maneuver the circle traced by the rocket body's nozzle is concentrically placed with a second circle, offset a distance d from the target, inscribed within an n -sided regular polygon. The angular change in the trajectory α needed to move the spacecraft to the next

Figure 5.10: Geometry of Discretized Relative Motion with Closure

tangent trajectory is therefore given by,

$$\alpha = \pi - \left(1 - \frac{2}{n}\right)\pi. \quad (5.16)$$

This trajectory allows for a good approximation of the relative motion of the target. However it does not yet consider the need to close with the target object.

The ability to model closure with the target can be considered as a super position of the established discretization of the target's motion with a discretized closure with the target vehicle. The geometry of the problem is correspondingly modified to include β the angle required to match the imposed distance of closure δ per element. Figure 5.10 shows the modified geometry required to both match the motion of the target vehicle and close with the target vehicle over time t . Since the radius r_n of each point, the angular position of the spacecraft ϕ , and the angle subtended by the next element θ are all known the trajectory needed to fly from point A to point B can be solved for directly.

The solution for the trajectory gives the direction β needed to sufficiently close with the next element. Continuing to meet the condition that time of flight of each element is equal to the time it takes for the nozzle of the rocket body

to subtend it's sector θ continuous to approximate the motion of the target. The final stage of this closing leaves the target at the target position with a small relative velocity to the target. Figure 5.11 shows how this representation is utilized to approximate a capture trajectory a delta-v requirement for the capture maneuver can be obtained. For the representative case with an initial distance of 255m and an intercept time of 5 minutes this yields a delta-v of,

$$\Delta v = 102.6 \frac{m}{s}. \quad (5.17)$$

Taken together the relative motion analysis and orbital motion analysis provide a reasonable, conservative, estimation for the delta-v required to close with and capture the target vehicle. However the method described still requires a means to physically facilitate the capture of the target.

Figure 5.11: Discretized Relative Motion of Representative Case

5.2.3 Grapple

One of the largest challenges facing any ADR mission is the fact that the vast majority of target objects were never intended to be part of an on orbit rendezvous. As a result nearly all potential targets lack a dedicated attachment point or other means by which a conventional docking can be performed. A number of solutions have been proposed ranging from inflatable grapples to harpoon like devices. However the selection of rocket bodies as the primary target considered allows for a simplification of the problem.

Figure 5.12 shows existing off the shelf hardware that is specifically designed to capture rocket bodies [9]. The mechanism uses the nozzle of the rocket body to create a hard dock analogous to connections provided by existing docking mechanisms used on various spacecraft. This grapple method creates a secure way of attaching the target vehicle, allowing the spacecraft to efficiently impart the required delta-v on the now connected target-spacecraft system. Since the attachment device is gimbaled the system easily translates to a probe-and-drouge docking system which is described in detail by Fehse [4]. Additionally work by the NASA Orbital Debris Program Office on the optical signature of tumbling rocket bodies shows that an optical system should have little difficulty identifying the nozzle in order to dock [7].

In solving the problem of docking with the target vehicle all major components of the ADR system's mission to grapple the target vehicle have been accounted for and as will be shown they pose no undue mass or delta-v requirement that would preclude a small system architecture from accomplishing the ADR mission.

Figure 5.12: Grapple Mechanism [9]

5.3 Control

Before any maneuvers to deorbit the target vehicle can take place the target must be oriented correctly so that the deorbit burn can be made as efficiently as possible. To do this two tasks must be accomplished. First the spacecraft must stop any residual motion of the target vehicle. Secondly the spacecraft must orient itself and the target so that the spacecraft's deorbit motor is pointing the velocity direction. Having made the assumption of a planar rotation of 2° per second about the transverse axis that assumption will be sustained for this portion of the analysis.

While Praly et al. [20] have clearly demonstrated that there are external torques acting on rocket bodies in LEO the magnitude of these are sufficiently small such that for impulsive maneuvers on small time scales they can be con-

sidered negligible. The target spacecraft system is therefore taken as spinning in plane as before under no external torques [3]. The moment of inertia of a cylindrical shell about its transverse axis is given by,

$$I_x = \frac{1}{2}mr^2 + \frac{1}{12}ml^2. \quad (5.18)$$

The angular momentum of the target vehicle is given simply as,

$$L_x = (\frac{1}{2}mr^2 + \frac{1}{12}ml^2)\omega_x \hat{\mathbf{i}} = 105.2 \text{ Nm.s.} \quad (5.19)$$

Since torque is the time derivative of angular momentum the force needed to bring the target to rest can easily be calculated as,

$$F = \frac{L_x}{tr \sin \theta}. \quad (5.20)$$

Solving for the firing time of a 20N thruster given, the radius of the target vehicle, 3.55m, and the angle between the radius vector and the force, θ is 90° , the time needed to stop the target's rotation is,

$$t = 1.48s. \quad (5.21)$$

Such a maneuver is well within the capability of many commercially available thrusters such as the ASTRIUM 20N thruster with a component mass of 0.395 kg. The fuel required can be easily determined given the thrusters specific impulse, thrust and burn time as,

$$\Delta m = \frac{F}{I_{sp}}t = 0.09kg. \quad (5.22)$$

A worst case scenario is one where the plane of the tumble is exactly perpendicular to the orbital plane. This scenario requires that after the motion of the target vehicle has been stopped an additional 90° rotation is needed to orient the deorbit motor into the velocity direction. In all cases the first maneuver can be

chosen such that the target vehicle deorbit motor lies in the orbital plane. This reduces the total number of impulsive maneuvers required to three, one to stop the motion of the target and a second pair to orient the target correctly. A 90° rotation of the target over 1 minute will require a angular velocity of,

$$\omega = 0.026 \frac{1}{s}. \quad (5.23)$$

Solving for the burn time for a 20N thruster yields,

$$t = \frac{I\omega}{Fr \sin \theta} = 1.1s. \quad (5.24)$$

A second burn of the same magnitude can be made to correctly orient the target. Again this is well within the capability of available thrusters and will require only,

$$\Delta m = \frac{F}{Isp}t = 0.19kg, \quad (5.25)$$

of fuel. Arresting the motion of the target vehicle and orienting it to effect a deorbit is in no way cost or mass intensive. Controlling the spacecraft poses no undue problems for any type of ADR mission.

5.4 Deorbit

With the target-spacecraft system correctly oriented an analysis of the deorbit can be conducted. The deorbit maneuver consists of a simple 1 burn Hohmann transfer, used to lower the perigee altitude of the target. A perigee of 200km or less is widely considered to be untenable due to atmospheric drag and will serve as the target altitude for the deorbit maneuver. The delta-v required to complete the maneuver can be easily found using equation 5.3 to find the difference in velocity between the transfer and target orbits at apogee. Figure 5.13 shows the

evolution of the target's orbital altitude after the completion of the deorbit burn. Drag decays the target until the target is circularized and then passes below 100km altitude at which point it will deorbit within hours. The delta-v required to deorbit the target is,

$$\Delta v = 154.6 \frac{m}{s}. \quad (5.26)$$

While this is commensurate with the delta-v required for other phases of the it will require much more fuel to achieve than the other required maneuvers due to the need to maneuver not only the mass of the spacecraft but also the mass of the target vehicle as well.

Figure 5.13: Deorbit of the Target

The mass required to achieve a given delta-v is given by the rocket equation,

$$\Delta m = m(1 - e^{(-\frac{\Delta v}{I_{sp} g_0})}). \quad (5.27)$$

Here I_{sp} is the specific impulse of the propellant seen in Figure 5.14, g_0 is the standard gravitational acceleration and m is the mass of the rocket.

Propellant	Isp (s)
Cold Gas	50
Monopropellant hydrazine	230
Solid	290

Figure 5.14: Specific Impulse of Various Propellants

Assuming the ability to use a solid rocket motor as the primary deorbit mechanism the mass required in fuel to deorbit the target can be easily calculated from equation 5.27. Assuming monopropellant hydrazine thrusters are used for all other maneuvers the mass budget for this representative mission can be developed. Figure 5.15 shows the fuel mass requirement for each mission task with the aforementioned propellant choices.

Figure 5.15: Fuel Mass Requirement of Each Mission Task

As Figure 5.15 clearly demonstrates the deorbit task is by far the primary

driver of the system mass. Furthermore the entire mission fuel requirement is 44.7 kg. This falls well within the capability of a microsatellite scale platform. However since the target vehicle mass and altitude prove the deorbit fuel consumption which in turn drives the fuel mass requirement for the system a careful sensitivities analysis of the effect of these parameters on the system mass budget as well as other design sensitivities should be considered. The representative case considered demonstrates the viability of using a micro satellite platform in the LEO regime to conduct an ADR mission. However the sensitivity of the mission viability to target dependent parameters is an important aspect of a more general feasibility for microsatellites to meet the requirements of the ADR mission.

5.5 Sizing

While the sizing of the system is primarily driven by the fuel mass other required components do contribute to the overall system size. Each task requires components that are necessary for the successful completion of the mission. All of these additional components must be considered to ensure the feasibility of a microsatellite design. The requisite components for a ADR mission can be grouped generally into seven primary subsystems: propulsion, control systems, processing, power, communications, structure and mission subsystems. Each of these subsystems requires particular components in order to carry out their function and contribute to a successful flight. Sizing these components can be done by picking representative off the shelf hardware to achieve an estimate for the total system mass. *Space Mission Engineering: The New SMAD* [8] and cubesatshop.com [1] were used to help pick components and determine what components were needed for mission completion. In particular the Firesat II

example was used as a reference to size subsystem mass requirements. Figure 5.16 shows the requisite subsystem mass percentages, the major needed components and an estimate for the percent of the target launch mass each subsystem will require.

Subsystem	Required Components	Mass (%)
Propulsion	Fuel, Tanks, Motor	50
Controls	Attitude determination, Telemetry, Thrusters	5 ± 1
Processing	Processing hardware	3 ± 1
Power	Batteries, Solar panels	13 ± 3
Structure	Structure	20 ± 2
Mission	Docking adapter, Rendezvous laser radar	5 ± 2
Other	Thermal, Communications, Misc.	4 ± 2
Total		100%

Figure 5.16: System Mass Sizing

The relatively short mission duration the effects of the space environment do not have to be exhaustively mitigated for, allowing for the selection of lighter, less robust hardware. In addition a short mission lifetime removes concerns of battery wear and other long term considerations when selecting hardware. The narrow scope of the mission also allows for more flexibility when choosing communications and processing hardware. As expected the fuel and other propulsion components require the majority of the system mass budget. Structure is taken to be approximately ten percent of the total system mass, although clever use of solid fuel and other components could reduce the dependence on an independent structure. Mission specific hardware such as the docking mechanism and a rendezvous laser radar fill out the remaining system mass. While careful design may allow for an optimization of component masses Figure 5.16 presents a conservative mass budget for an ADR mission. The resultant size is well within the design

limits of a microsatellite platform and verifies the feasibility of a microsatellite ADR mission.

Chapter 6

System Performance

While the representative analysis proves the feasibility of a microsatellite platform as a general system concept and as a solution for a specific target, a wider analysis of the effectiveness of a small system architecture is required. In order to prove the usefulness of a microsatellite platform the performance of such a system is evaluated over a range of target and mission parameters.

As was shown in Figure 5.15 the mass budget of the spacecraft is driven by the mass required to deorbit the target vehicle. The mass of fuel required to completely deorbit a target is highly dependent on the target's altitude and mass. Solving equation 5.27 after plugging in equation 5.3 for v_i allows the mass to completely deorbit a target vehicle to be written as,

$$\Delta m = m(1 - e^{(-\frac{\sqrt{\frac{2\mu}{r}} - \frac{\mu}{a}} - \sqrt{\frac{\mu}{200}})}{I_{sp}g_0}). \quad (6.1)$$

When equation 6.1 is evaluated over a range of altitudes and target vehicle masses a sensitivity analysis of the fuel required to deorbit a target is generated. Figure 6.1 shows the results of this sensitivities analysis assuming circular initial orbits.

Figure 6.1: Fuel Mass Sensitivity Analysis

A careful examination of Figure 6.1 shows that a fuel mass of under 80kg allows for operations within a significant portion of LEO. Particularly the region between 800 and 1000 km can be serviced directly for target vehicles up to 1400kg. As shown in Figure 5.1 this regime constitutes the bulk of the LEO debris population and is ideal territory for an ADR mission. Additionally Figure 6.2 also from *A Sensitivity Study on the Effectiveness of Active Debris Removal* [15] demonstrates the usefulness of such a capability in targeting worst offender populations.

Figure 6.2: LEO Debris Population by Mass

Despite the ability for a small architecture system to service a large portion of the LEO debris population there will be some target vehicles that lie outside the capability of the system to directly deorbit. However the proposed system may still be able to drastically reduce the lifetime of these objects and allow them to meet the 25 year rule. Figure 6.3 shows the capability of systems with 50kg, and 75kg of fuel payload respectively to deorbit targets of 800kg, 1200kg and 1600kg at various altitudes.

Figure 6.4 characterizes the lifetime of rocket bodies in low earth circular orbits with ballistic coefficients of $0.01 \frac{kg}{m^2}$, $0.1 \frac{kg}{m^2}$, and $1 \frac{kg}{m^2}$ under exponential drag [18]. Between 100km and 700km the lifetime of satellite increases exponentially with altitude as the atmosphere rapidly dissipates. In spite of this the capability to remove targets to altitudes between 200km and 600km may still allow them to meet the 25 year deorbit rule. The deployment of drag enhancing

Figure 6.3: System Performance

packages in addition to a grapple-and-tug provides even further benefits and may allow for target altitudes as high as 800km.

Figure 6.4: Rocket Body Lifetimes in LEO

Combining the data from Figures 6.4 and 6.3 allows the impact of sub-optimal fuel payloads or sub-optimal deorbit burns to be seen. Figure 6.5 shows the impact on sub-optimal fuel payloads on a representative case nearly identical to the one considered previously. A rocket body in a 800km circular orbit. The lifetime reduction achieved by the spacecraft quickly becomes ineffective as the perigee altitude increases above 250km. This highlights the need for accuracy in the deorbit burn in order to achieve an effective result.

Figure 6.5: Lifetime vs Fuel Mass for a rocket body in a 800km circular orbit

A microsatellite scale system can effectively operate across a wide swath of LEO deorbiting targets on the order of 1600kg from altitudes as high as 1100km. This allows a grapple-and-tug platform to choose from a large number of possible target candidates greatly increasing the chance of a successful mission and a meaningful ADR contribution. Even targets outside the performance envelope of the design could see benefits in orbital lifetimes.

Chapter 7

Conclusion

The space debris environment poses a fundamental challenge to the future of human spaceflight. While many solutions and system designs should be explored a microsatellite ADR mission is a feasible, realistic, and achievable mission concept. A small single use architecture allows for flights as a secondary payload greatly increasing the launch opportunities for ADR missions. Additionally small architectures could allow a primary launch with multiple payloads to remove several different targets with one flight; effectively completing the required target removal rate with one mission. Microsatellite systems also have the potential to lower cost, and decrease the risk associated with attempting an ADR mission.

In conjunction with a small system architecture grapple-and-tug provides the most proven and reliable means to deorbit target vehicles of currently proposed systems. Through the use of a careful CONOPS selection, mission complexities can be drastically reduced, allowing for a higher performance more focused design. Although rendezvous, docking and control of the target vehicle may be fuel and capability intensive the driving factor of system size is the fuel payload required to accomplish the deorbit impulse maneuver. The fuel payload is sensitive to

both the target mass and target altitude but a system mass of under 100kg still provides ample opportunities for conducting ADR operations in LEO.

By selecting rocket bodies as the preferred target candidates many of the problems facing the grapple-and-tug concept are simplified. Rocket bodies allow for the ability to create a hard-dock like attachment through the use of engine nozzles as pseudo docking drogues. In addition rocket body dynamics are more uniform across the spectrum of possible targets due to the geometrical similarities of rocket bodies and their interaction with damping forces in the LEO environment. The planar nature of rocket body relative motion allows for an optimum close rendezvous flight path to be geometrically approximated. No aspect of the mission profile poses undue challenges to the ability of the selected architecture to successfully deorbit target vehicles as part of a larger ADR effort.

The representative mission considered clearly demonstrated the feasibility of a microsatellite ADR mission. Advances and increased interest in small scale satellite operations have developed technologies, practices and flight hardware well suited to a microsatellite mission. These advances allow a short duration, focused mission to cut away much of the existing hardware overhead and focus on fulfilling mission requirements. The ability for a microsatellite mission to devote nearly all of its design towards fulfilling mission requirements coupled with its basis in flight proven hardware and technologies gives any potential microsatellite ADR mission a decided edge over other alternatives. The nature of the debris problem requires prompt and decisive action. A microsatellite ADR mission has the potential to immediately begin affecting a solution.

Chapter 8

Future Work

While a microsatellite ADR mission is certainly feasible there is still much work to be done before the benefits of such a system could be brought to fruition. Several different avenues of investigation should be promptly explored to take advantage of the opportunities provided by microsatellite platforms. Foremost is the need to investigate alternate propulsion methods for a grapple-and-tug system. This analysis only treated the impulsive maneuver case assuming the use of a conventional solid propellant motor to deorbit the target spacecraft and use of monopellant thrusters to perform other orbital maneuvers. Numerous low-thrust methods exist and should be investigated to see if an additional reduction in scale or complexity can be achieved. Possible low thrust alternatives include the use of electric propulsion or the use of solar sails to conduct low thrust deorbit maneuvers.

In addition to the use of low thrust maneuvers a further examination of how debris lifetimes are effected by solar activity could help reduce the burdens placed on ADR systems. If a parking orbits could be established at a relatively high altitudes that utilize density increases during solar maximums to meeting the 25

year deorbit rule ADR systems could be made smaller or go after larger or higher targets without having to increase their own system mass. Higher parking orbits would also mitigate problems caused by orbits that regularly cross the orbit of the International Space Station which have the possibility for necessitating additional maneuvering of station. Additional investigation of debris dynamics would also aid future active debris removal missions going after unused satellites and other poorly behaved target vehicles.

In addition to investigations concerning the debris environment and mission parameters a serious engineering effort should be made to design a microsatellite scale ADR mission. There have been no attempts at any sort of ADR mitigation despite the pressing and growing space debris problem. A serious engineering effort to tackle the practical problems of the ADR mission following to proposed CONOPS could likely improve on estimated performance established for a microsatellite ADR mission. Such a performance could have immediate and tangible benefits for the prognosis of the debris environment. Each of the mission tasks outlined deserves a careful, thorough design process in order to solve and optimize the real world problems associated with this complicated yet vital mission. Microsatellite ADR is a very real possibility and serious resources should be devoted to solving this serious problem.

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