DESIGN OF AN UNMANNED, REUSABLE VEHICLE TO DE-ORBIT DEBRIS IN EARTH ORBIT

Submitted to:

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November 6, 1990

Mr. James Aliberti
Mail Stop PT-PMO
National Aeronautics and Space Administration
Kennedy Space Center Florida 32899

Dear Mr. Aliberti:

Attached is our final report for the design of an unmanned, reusable vehicle to de-orbit debris in earth orbit. The design team selected a Chain-and-Bar Shot (CABS) method as the most feasible concept for de-orbiting space debris. The CABS method consists of a weighted net fired from a rail-gun, and, upon exit from the muzzle, utilizes centrifugal force to open the net and capture the debris mass. The net imparts sufficient kinetic energy to de-orbit the debris mass in one step.

This report contains background on the space debris problem, a complete description of the alternative designs investigated, a decision matrix to select the design solution, and the final design solution. The design team also presents flowcharts describing the operational sequence of the de-orbiting procedure.

It has been a pleasure working on this project for NASA/USRA, and we look forward to seeing your representative at our design presentation. The presentation will take place on Tuesday, November 27, 1990, at 2:00 p.m. in the Engineering Teaching Center II, Room 4.110, on the campus of The University of Texas at Austin. A catered luncheon will precede the presentation.

Sincerely,

Shahed Aziz

Timothy W. Cunningham

Michelle M. Moore-McCassey

Michelle M. Moore-McCassey, Team Leader

Acknowledgements

The design team members would like to thank the National Aeronautics and Space Administration and the Universities Space Research Association (NASA/USRA) for sponsoring this project.

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Abstract

Design of an Unmanned, Reusable Vehicle to De-orbit Debris in Earth Orbit

The space debris problem is becoming more important because as orbital missions increase, the amount of debris increases. It was the design team's objective to present alternative designs and a problem solution for a de-orbiting vehicle that will alleviate the debris problem by reducing the amount of large space debris in earth orbit. For a senior design project sponsored by the University of Texas Mechanical Engineering Design Project Program, NASA/USRA asked the design team to design an unmanned, reusable vehicle to de-orbit debris in earth orbit. The design team also will construct a model to demonstrate the system configuration and key operating features.

The alternative designs for the unmanned, reusable vehicle were developed in three stages: selection of project requirements and success criteria, formulation of a specification list, and the creation of alternatives that would satisfy the standards set forth by the design team and their sponsor.

The design team selected a Chain-and-Bar Shot method for de-orbiting debris in earth orbit. The De-orbiting Vehicle (DOV) uses the NASA Orbital Maneuvering Vehicle (OMV) as the propulsion and command modules with the de-orbiting module attached to the front.

KEY WORDS: DE-ORBITING VEHICLE (DOV), CHAIN-AND-BAR SHOT (CABS), SPACE DEBRIS MASS, ORBITAL MANEUVERING VEHICLE (OMV), ELECTROMAGNETIC RAIL-GUN

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Introduction

This project is sponsored by the National Aeronautics and Space Administration (NASA) in cooperation with the Universities Space Research Associates (USRA). As a part of the "Mission Planet Earth" project, NASA and USRA have been examining the problems created by orbital debris. It is estimated there are millions, perhaps billions, of debris masses of different sizes in earth orbit . Of these debris, approximately 7000 are at least 10 centimeters in diameter and are continuously tracked by the United States Air Force's North American Air Defense Command (NORAD) [11]. Figure 1 shows a representation of the earth's space debris population. The objective of this project is to

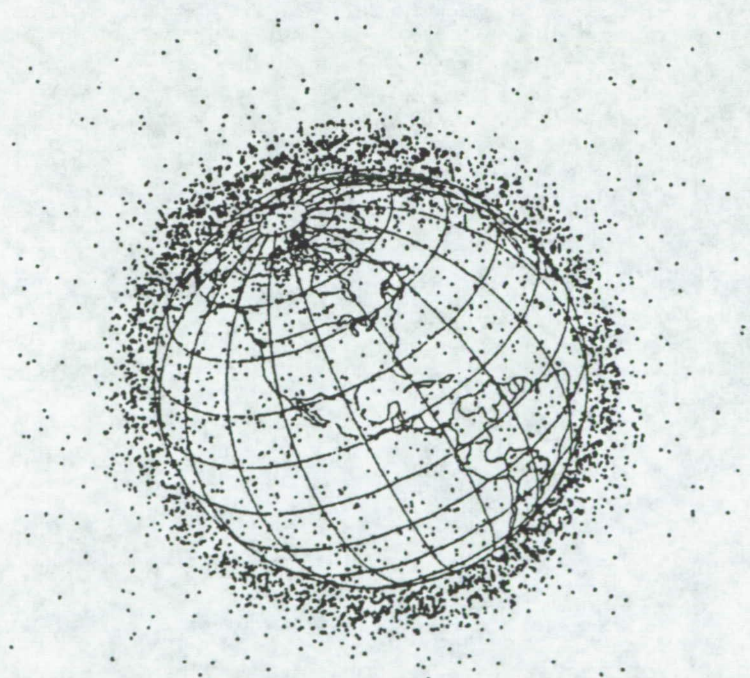


Figure 1. REPRESENTATION OF DEBRIS IN EARTH ORBIT.
Figure taken from "Evolution of the Artificial Earth Satellite Environment", by Nicolas L. Johnson, Teledyne Brown Engineering, Colorado Springs, Colorado, 1987.

design a De-orbiting Vehicle (DOV) to remove the largest debris masses from earth orbit.

The three members of the design team are mechanical engineering seniors at The University of Texas at Austin. Involvement with this project came about through the Mechanical Engineering Design Projects Course, ME466K.

The purpose of this report is to present a solution for de-orbiting large debris currently in earth orbit. This report presents background on orbital debris, outlines the project requirements, defines the design criteria, examines different design solutions, presents the project solution, and discusses conclusions and recommendations for the problem of orbital debris.

1.1 Background

The space debris issue continues to take on more importance as NASA plans more space missions, including the building of the Space Station Freedom. Solving the space debris problem becomes more important as the frequency of missions increases. The following sections describe the space debris population, the resulting problems, and a previously proposed concept.

1.1.1 Space Debris. All of the approximately 7000 objects currently tracked by NORAD are at least 10 centimeters in diameter and only five percent of these are operating vehicles. Of the remaining objects, nearly half are debris from space collisions. The remaining objects consist of abandoned satellites, discarded upper stages of rockets from previous missions, and other mission related objects (see Figure 2). The contributors are all countries involved in space exploration. Beyond these 7000 objects, NASA estimates that more than one billion smaller particles (micro-meteorites) are orbiting undetected but

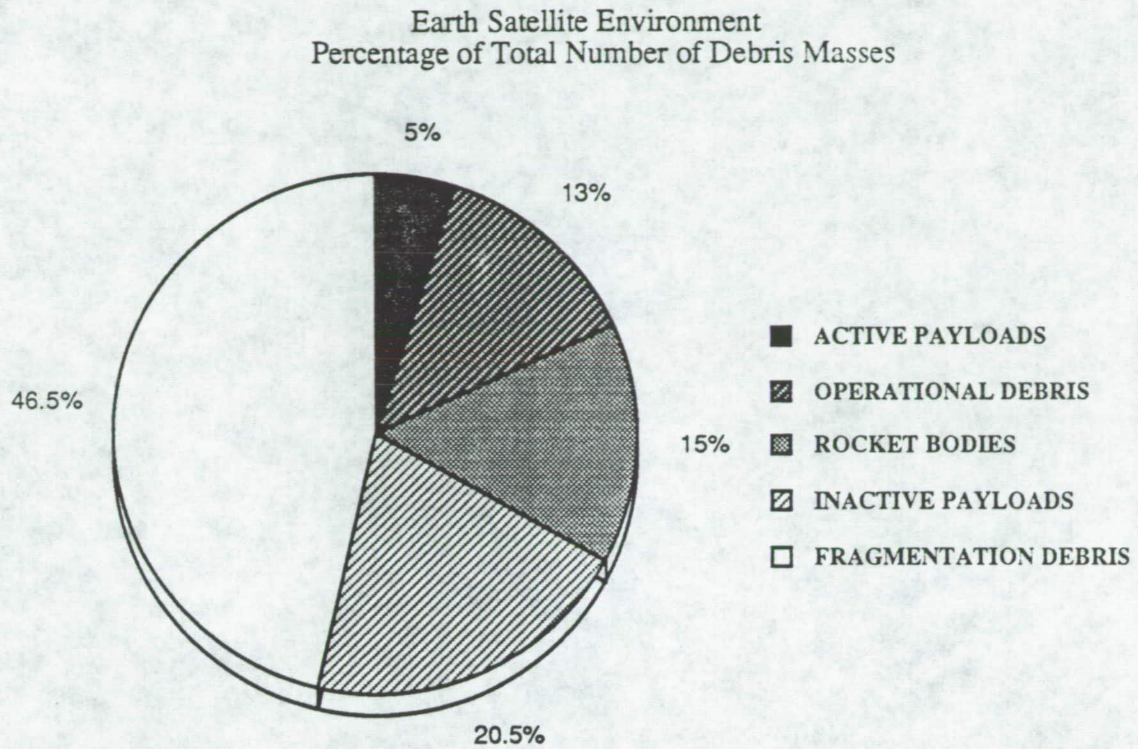


Figure 2. CATEGORIES OF SPACE DEBRIS.
Data taken from "Evolution of the Artificial Earth Satellite Environment", by Nicolas L. Johnson, Teledyne Brown Engineering, Colorado Springs, Colorado, 1987.

are equally dangerous to other active missions [11].

Two major problems result from space debris. First, impact with space debris, which move on the order of 10 kilometers per second, can cause serious if not fatal damage to a vehicle. This requires considerable and expensive attention to the design of spacecraft to ensure survivability. One example is the Space Station Freedom; originally designed with many windows, the design has been altered to include almost no windows for fear of a catastrophic collision with space debris [3]. Design for survivability increases in importance as the threat of space debris grows.

The second problem concerns mission planning. When launching any vehicle, NASA must program around the known trajectories of all objects being tracked. Having to avoid all debris causes complicated planning and critical time constraints to safely launch a spacecraft.

NASA considers the most realistic method of combating space debris is the elimination of as much useless mass as possible from earth orbit. The "Cascade Effect" or "Kessler Effect" states that the amount of debris propagates according to the amount of mass. This is, in effect, a progression of collisions between small and large debris, resulting in more debris and in turn, more collisions [11]. Therefore, NASA seeks a system to de-orbit the largest bodies to decrease the greatest amount of mass possible. The design team will configure a DOV to eliminate large debris masses (on the order of 2000 kilograms).

1.1.2 The Grappling Net Concept. A University of Texas Mechanical Engineering graduate project to solve the space debris problem was completed by Mr. Richard Connell in the spring of 1990. His design solution is known as the Grappling Net Concept. An unmanned reusable DOV traps an object in space by the use of a net attached to a tether. The net de-tumbles the object to control it (see Figure 3). The DOV then fires its engine to spin itself and the grappled debris (see Figure 4). By releasing the object as it moves toward the earth, the DOV gains the energy lost by the object as it is sent to lower orbit to burn-up in the atmosphere [10].

The Grappling Net concept has two advantages: it allows the DOV to obtain energy from the process, resulting in less fuel consumption; and the system is relatively compact, minimizing mass and cost. While these advantages are compelling, there are two potential problems concerning the spinning motion and the grappling net. First, the spin and subsequent release of the object will result in an unstable DOV, requiring re-stabilization. In turn, the re-stabilization process adds to the fuel consumed by the Orbital Maneuvering

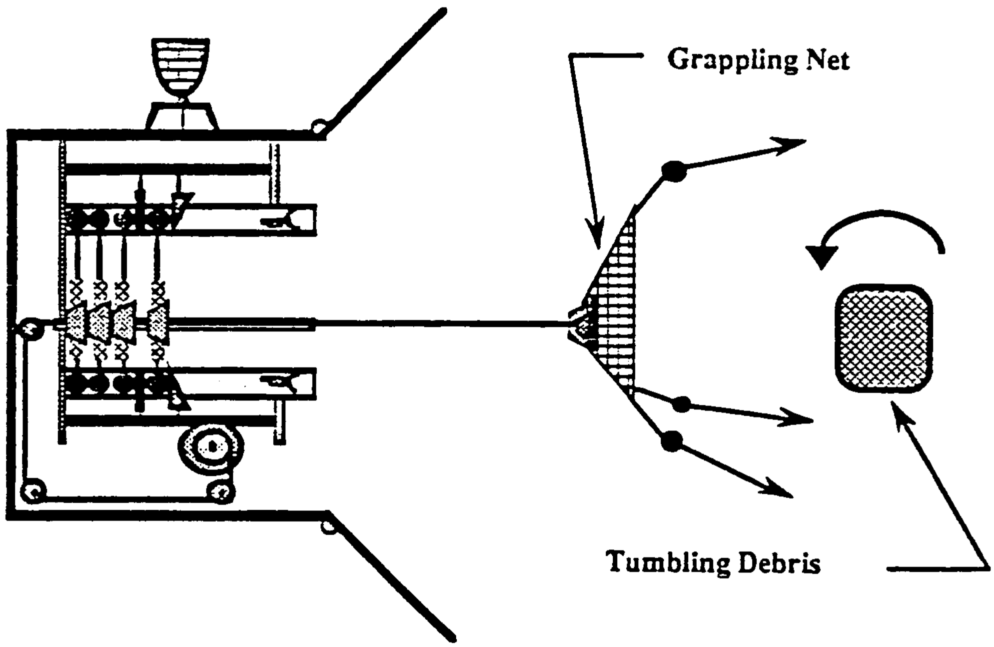


Figure 3. GRAPPLING NET DEPLOYMENT.
Figure taken from "Design of a Vehicle for De-orbiting Space Debris in Earth Orbits", Richard Connell, The University of Texas at Austin Mechanical Engineering Department, May 7, 1990.

Thrusters (OMTs). Second, the net has to be designed to accommodate several sizes of objects and would be lost every time a debris mass was de-orbited; these two problems make the net a very complicated mechanism. Therefore, the DOV would have to be equipped with a number of grappling nets and periodically resupplied to maintain operations.

This alternative serves as background information but will not be considered as one of the design team's alternative designs [3].

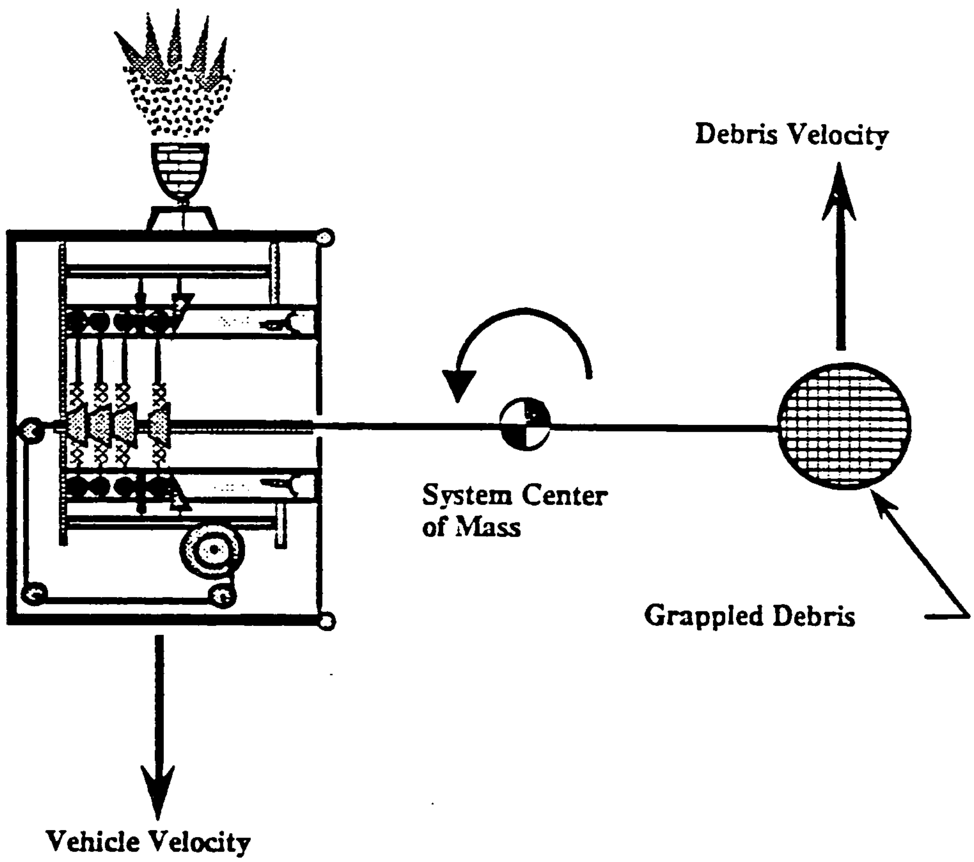


Figure 4. GRAPPLING DOV FIRING ENGINE TO SPIN DEBRIS TO EARTH. Figure taken from "Design of a Vehicle for De-orbiting Space Debris in Earth Orbits", Richard Connell, The University of Texas at Austin Mechanical Engineering Department, May 7, 1990.

1.2 Project Requirements

The design team identified a number of design criteria for the project. The first three are specific objectives to research a "static deployment" method of de-orbiting debris, the method suggested by Mr. Connell (discussed fully in Section 2.1). The final two objectives are the deliverables the design team will produce.

1. Grappling of Tumbling Debris: The first task in the static deployment is to gain control and capture of the debris mass. The previous concept called for the use of a grappling net; this will be considered among our alternatives.
2. Tether Attachment/Detachment to the Debris Mass: The second design requirement is the development of a new concept for attaching and safely detaching the tether from the DOV to the debris. Mr. Connell advised the team to discard the Grappling Net Concept because of the complexity in operation as well as the mass penalty incurred by the carriage of extra grappling nets. The previous method of detaching the tether at the base of the grappling net could also lead to problems with the uncontrolled movement of the tether.
3. Method of Supplying the Impulse to the Debris: The static deployment method is a new method of de-orbiting the space debris utilizing an energy gain from the process and attempting to control instability. The previous de-orbiting method involved imparting a momentum to the debris by spinning it around the center of rotation of the two connected bodies (the DOV and the debris mass) [10]. However, after detachment of the debris, the DOV remains at a higher energy orbit as well as an unstable one. Restabilization of the DOV's orbit would require expended fuel for the OMTs. Minimization of OMT fuel expenditure is necessary to increase the energy efficiency of the DOV. NASA tasked this design team to solve the problem by the new method. We must determine a method of giving the debris mass the "initial push" toward earth to begin the de-orbiting process.
4. Vehicle Configuration: The configuration of the DOV itself needs further refinement. The vehicle is required to be reusable by the replenishment of consumables (fuel, tether, etc.) in orbit. Therefore, the maximum possible modularization of the vehicle configuration is required. Modularization should make replenishment and repair operations by astronauts wearing spacesuits in a

micro-gravity environment as simple as possible.

5. Model of the DOV: The design team's final task will be to construct a scaled model of the final vehicle configuration.

With these design criteria in consideration, the design team proposes to define a de-orbiting concept as well as a more refined configuration for the DOV itself. To this end, five "success criteria" have been defined for the DOV concept chosen. The successful DOV concept should be :

1. reusable,
2. energy efficient,
3. an effective "de-orbiter,"
4. simple, and
5. economical in terms of mass and size.

1.3 Specifications

The project requirements and success criteria were used to formulate the specifications for the design project. A full list of the specifications, divided into eight categories, can be found in Appendix A. In the list of specifications, the letters "D" and "W" represented "demands", or required items, and "wishes", or desired items, respectively. The following discussion highlights some of these specifications.

First, modular construction for the DOV has been specified. This satisfies the ergonomic requirement for simple replacement of the consumables by an astronaut wearing an EVA (Extra-Vehicular Activities) suit. Modularization is the key factor in the design of the DOV configuration.

One of the most important design considerations is the gain of useful kinetic or potential energy from the de-orbiting operation. Of course, the DOV is also required to supply enough energy to de-orbit the debris to successfully complete the mission.

Assumptions were made about the type of signals NASA ground controllers will use to control the DOV. First, it is expected that the DOV will either follow a pre-programmed mission profile or be actively controlled by a human operator using remote control. For replenishment operations, active control will be obtained from either the Space Shuttle or the Space Station Freedom; in this case, astronauts will maneuver the DOV close enough for retrieval by the robotic arm. Finally, for terminal homing once the target debris is acquired, the DOV will have its own Doppler radar; this will allow precise positioning of the DOV with respect to the target. The most important signal will be the final "go/no go" decision by the ground controller before the de-orbiting operation can begin.

The final specifications concern the transportation of the DOV. The DOV should be designed for either manned or unmanned insertion into earth orbit, and should not contribute to the debris problem in either the transport or de-orbiting phases. The last operational requirement is for multi-mission capability (reusability) and multiple de-orbiting operations during each mission.

1.4 Design Proposal

This section outlines the design proposal submitted September 20, 1990. It discusses the work the team expected to accomplish.

The design team concentrated on the solution of the orbital debris problem by designing an unmanned satellite, or DOV, to remove the larger (in the neighborhood of 2000 kilograms) pieces of debris from earth orbit. The de-orbiting module was designed to be reusable by having the ability to have the consumables (fuel and supplies related to the de-orbiting procedure) replenished periodically. The DOV has multi-mission capability between replenishment procedures.

The team also set up the configuration and placement of the different subsystems. Attention was paid to the subsystems to ensure the use of current technology in the DOV. However, the team did not perform a cost analysis of the system, as this is considered beyond the scope of this project [3].

The most important limitation of this project was the DOV design team's lack of access to the manufacturing facilities required to build a "proof-of-concept" prototype of the design selected. Ideally, a design concept can be "proven" by the testing of a prototype under realistic conditions. However, the design team did not have access to the production facilities or the large infrastructure required for such an undertaking.

The design team is also unable to test a working scale model. The testing of such a model will require access to artificial micro-gravity facilities. At the moment, the design team doesn't have access to the NASA-USAF Boeing KC-135 aircraft used for this procedure. However, it is strongly suggested that such a simulation be carried out in the KC-135 (see Section 5). In this case a scale model of only the de-orbiting part of the DOV, especially the CABS projectile itself, will have to be used to prove the validity of the concept.

1.5 Solution Methodology

The first phase of the design project was a thorough review of the background of the project. This involved a review of the orbital debris problem as well as the orbital mechanics involved in the removal of orbital debris. The design team also conducted a literature search for background material on the design of unmanned space vehicles. Extensive literature is available defining the space debris itself. A Congressional Office of Technology Assessment Report published September 1990 gives a thorough and up to date definition of the problem [18].

The next phase involved the generation of alternative solutions to the space debris problem. Based on the five criteria outlined in Section 1.2, the design team took a morphological approach where all possible solutions to the problem specific to the team's task were considered. These design alternatives were then examined with respect to the design criteria and narrowed down to five design alternatives. The best alternative was then chosen using a decision matrix and utilizing the success criteria defined by the team.

After the selection of the most viable problem solution concept, the design team concentrated on the configuration of the DOV itself. Special attention was paid to the modularization and mass minimization of the design. Although the design team is unable to construct a working prototype of the design, a scale model of the DOV will be constructed prior to the oral presentation of the project scheduled for November 27, 1990.

1.6 General DOV Configuration

The modular concept of design is used in the configuration. This concept separates the DOV into different sections or modules by function. Therefore, the influence of the de-orbiting concept selected is independent of the overall configuration of the DOV. The DOV is designed with three modules: the propulsion, the command, and the de-orbit modules. Figures 5 and 6 show this modular configuration.

1.6.1 Propulsion Module. The propulsion module houses the liquid-fueled main engine, the fuel supply, and the orbital maneuvering system for the DOV. The main engine and associated fuel tanks will be in one replaceable unit. The DOV is required to be reusable by replenishing the consumables in earth orbit (see Appendix A, Specification 6.1). Therefore the main engine may be easily replaced by one operator. The command antenna for the control link with the human operator is also located on the body of the propulsion module.

1.6.2 Command. The command module houses the control systems, the track/scan unit, and the power supply unit for the DOV (see Figures 5 and 6). The operations of the DOV are controlled by the command module through a digital databus that interconnects all the modules and their subsystems. The track/scan unit will be a pulse-

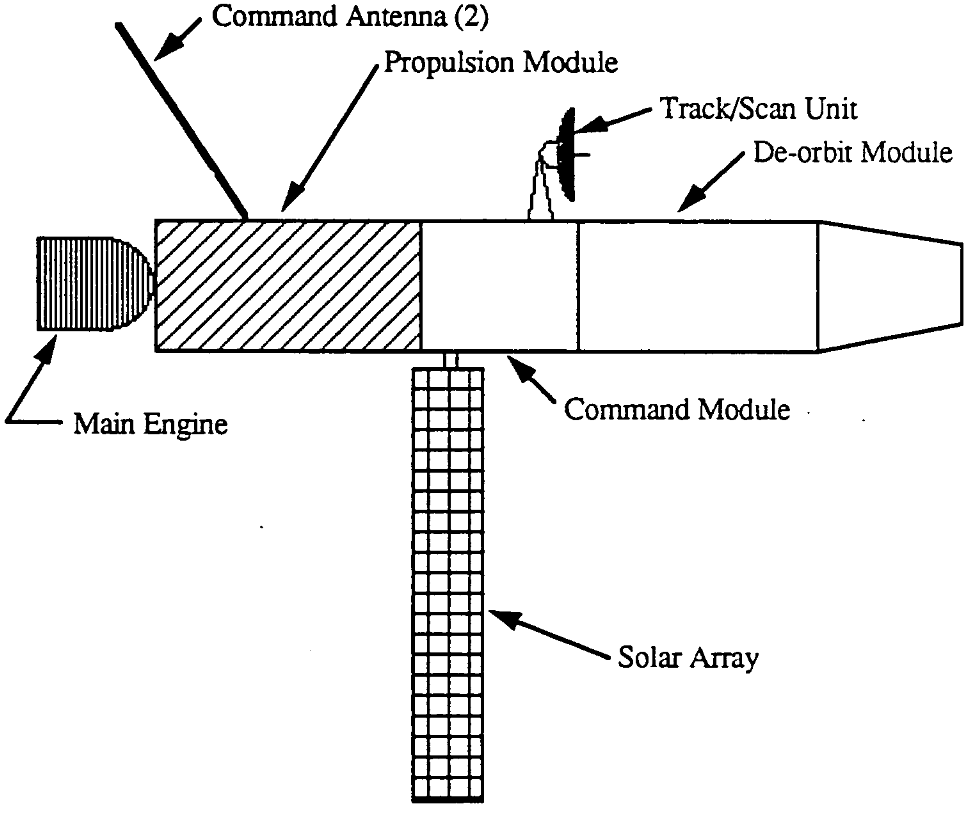


Figure 5. SIDE VIEW OF GENERAL DOV WITH SOLAR ARRAY

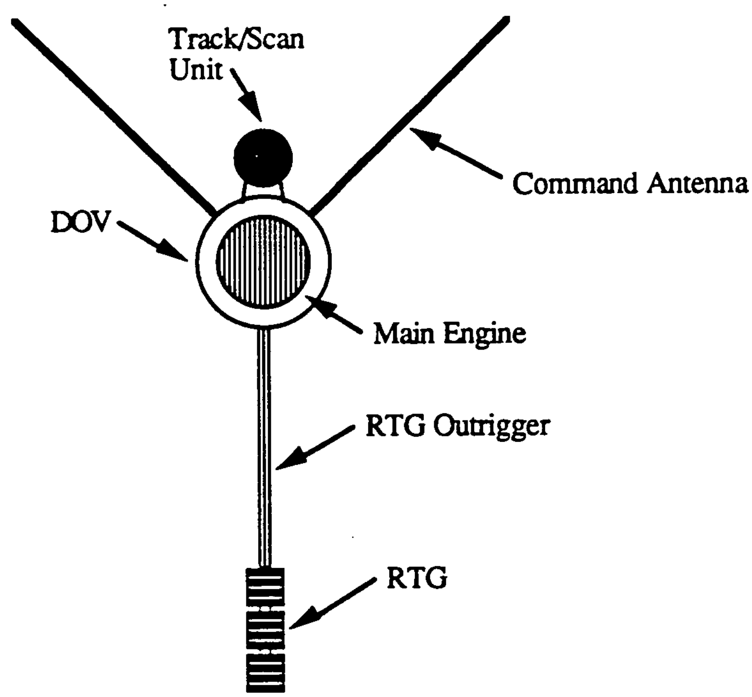


Figure 6. REAR VIEW OF GENERAL DOV WITH RTG

Doppler radar similar to that used in the General Dynamics F-16 fighter-bomber for the tracking mode, and a Television/ Infra-Red (TV/IR) for the scan mode system similar to the USAF Low Altitude Navigation and Targeting using Infra-Red at Night (LANTIRN) system [12]. The track/scan unit will be down-linked to the human operator through the command antenna.

The command module will also house the power supply. The power supply alternatives were a solar array, a fuel cell, or a Radioisotope Thermoelectric Generator (RTG). The solar array would have been a deployable type as used on most earth orbiting satellites. Solar arrays are reliable although they are relatively inefficient compared to RTGs or fuel cells [6]. Also, they require large surface areas and need to be constantly adjusted for solar orientation. Figure 5 shows a deployed solar array on the DOV configuration.

Fuel cells use the reaction between oxygen and hydrogen to create thermoelectric energy, with water as a reaction byproduct. This type of power pack has been used extensively in the United States Space program since the Project Mercury. They are reliable and safe but are heavy because the reactants must be carried. These fuel cells are transportable within the DOV.

RTGs convert the heat produced by the decay of plutonium-238 dioxide into electrical power. Heat is produced by the decay of plutonium-238 dioxide isotopes. The heat in turn generates a current in a bimetallic thermocouple. RTGs contain no moving parts, and, unlike nuclear reactors or nuclear weapons, do not use fission. RTGs have been used by NASA for more than two decades on twenty-three missions including the Apollo, Viking, Pioneer, and Voyager spacecraft [16]. They do, however, cause electro-magnetic interference and have to be housed on an outrigger (see Figure 6). Also, there is considerable public antipathy against using power sources that use radiation in any form. At present any satellite using RTGs requires a Presidential directive before it can be approved for launching [6].

These three options for electrical supply were considered during the final configuration portion of the design (see Section 3.2.2).

1.6.3 De-orbit Module. The de-orbit module would have the consumables and the associated equipment for the de-orbit concept selected. The power as well as the command inputs required to operate the de-orbiting module would be transmitted by the operator through the command module.

Alternative Designs

This section discusses five alternatives for the design team's project of designing an unmanned, reusable vehicle to de-orbit debris in earth orbit. The alternatives are:

1. Static Deployment and Tether,
2. Capture and Retro-fire,
3. Orbital Maneuver and Tether,
4. Kinetic Energy Projectiles (flechette and chain-and-bar shot), and
5. Kinetic Energy with Balloon Attachment.

The Static Deployment method, which Mr. Connell asked the design team to investigate, involves the use of a long tether to lower the debris to the earth's atmosphere. The second and third alternatives also use the idea of tethering. The Capture and Retro-fire method uses a short tether and braking with the DOV main engine to slow the velocity of the debris so it will enter an orbit terminating in the earth's atmosphere. A combination of the first two alternatives is the Orbital Maneuver and Tether concept. This method involves capturing the debris, slowing its velocity so it falls into an elliptical orbit close to the atmosphere, then tethering the debris until it reaches atmospheric level at approximately 80 kilometers above the earth's surface.

The remaining two alternatives involve the concept of changing kinetic energy (KE) by slowing the velocity of the debris mass. Two types of KE projectiles are proposed for de-orbiting debris at all altitudes: the flechette, a spear-like object used to impart KE to the debris, and the chain-and-bar shot, which functions as a casting net and provides the momentum required to slow the debris during the de-orbiting process. For low altitudes, a

KE projectile with a balloon attachment is described. This concept increases the surface area of the debris, resulting in increased atmospheric drag force on it. This induced drag magnifies the orbital decay that is characteristic of all satellites in Low Earth Orbit (LEO).

The following sections describe the alternatives. Solutions to the three main problems the design team addressed - grappling the debris, attaching the tether to gain control, and providing the impulse necessary to de-orbit the debris - are discussed for each alternative. The advantages and disadvantages are also discussed for each concept.

2.1 Static Deployment and Tether

The static deployment and tether concept involves the tethered sub-satellite (TSS) concept in reverse. In the TSS concept, a small sub-satellite is "suspended" at a very low orbit from another satellite (or the Space Shuttle) at a higher orbit. The atmospheric drag effects are expected to slow down the satellite and reduce its orbital altitude into the atmosphere. However, the momentum of the larger satellite in higher orbit prevents this unwanted de-orbiting [1].

The Static Deployment concept uses the high-orbit vehicle to slow the tethered mass, "dragging" it into the atmosphere. Once the debris has been grappled with a tethered net, the DOV pays out the tether until the debris reaches the earth's atmospheric altitude at approximately 80 kilometers above the earth's surface The calculations supporting this alternative can be found in Appendix B.

In the first stage of the procedure, the DOV will rendezvous with the debris mass in orbit. The DOV will then gain control of the debris mass by grappling it by deploying its capturing device. A number of options are being examined for grappling the debris. One option for the net is the "butterfly" type net. This net will have an inflatable rim that will be deflated for storage in the DOV until it is needed. Each rim and net combination will have a tether attached to it. For grappling, the rim and net combination will be deployed outside the DOV, with the rim inflated to the size needed to hold the debris mass (see Figure 7). Finally, the DOV will maneuver to catch the debris in the deployed net. The butterfly net is considered feasible because it grapples and controls the debris mass in one step, attaching the debris to the DOV by a tethered net.

The tether, attached to the net, connects the debris mass to the DOV when it is grappled. The tether will then be detached by severing it near the DOV. The tether is considered a consumable item in this process since it will be discarded during each de-orbiting procedure.

The de-orbiting procedure requires a relative velocity between the debris mass and the DOV to initiate separation between them. In the absence of atmospheric drag or any other external forces, an impulse is required to initiate this motion. This impulse requires sufficient magnitude to move a 2000 kilogram debris mass away from the DOV and toward

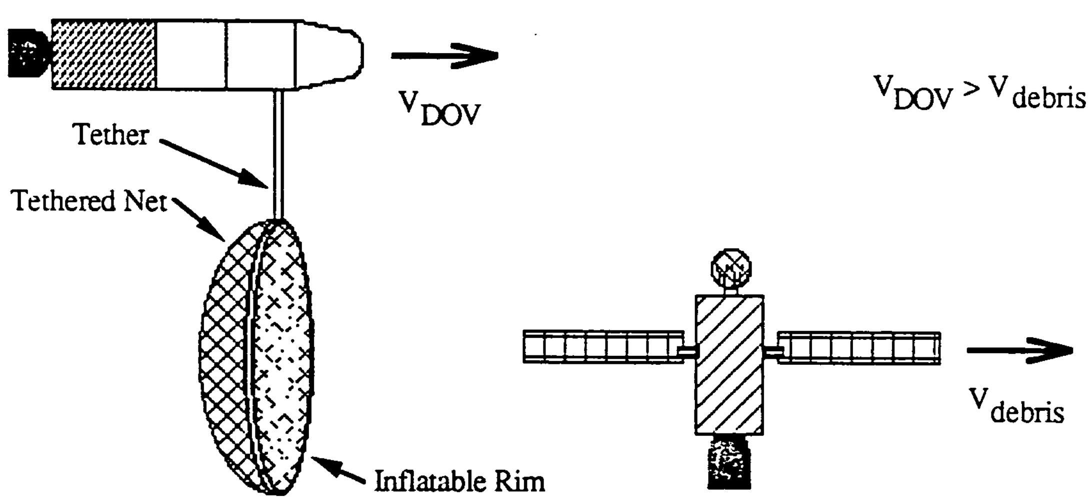


Figure 7. GRAPPLING USING A TETHERED 'BUTTERFLY NET' (not to scale)

earth. However, if an impulse is delivered over a very short period of time, the impact effect may cause break-up of the debris mass, thereby exacerbating the space debris problem [13]. After the impulse is provided, the debris moves to a lower orbit. The DOV and debris mass are a connected system. Therefore, as the debris mass moves down, the DOV moves to a higher orbit to maintain the center of mass at the center of the system at the original (debris mass') altitude [4]. Also, at the lower altitude, the debris mass tends to speed up. However, the tether connection and the DOV mass progressively slows down the debris mass [4]. The relative positions of the DOV and debris mass are shown in Figure 8.

The most important advantage of the Static Deployment and Tether concept is the potential energy gained by the DOV from the process [14]. This is caused by the height gained by the DOV during the relative motion of the DOV and debris system. This energy gain was considered one of the main criteria for a successful DOV concept. However, this concept is operationally unrealistic in terms of the mass and volume of tether required as well as the problems of controlling the tether length. Long tethers tend to act as strings and propagate disturbances. Also the loads upon the tether are not uniform due to the relative movement of the bodies, the debris mass and the DOV, as well as the gravity gradient along very long tethers [14].

Mr. Connell suggested the team use a model debris mass of 2000 kilograms orbiting in a circular orbit at 800 kilometers. A 6 millimeter diameter tether made of aramid fibers (commercially known as Kevlar 69) was calculated to have sufficient strength to control the model 2000 kilogram debris mass (see Appendix B). However, the tether required to de-orbit the debris mass from an 800 kilometer orbit will have a mass of 33,446.41 kilograms. That is an unacceptable penalty of operation as NASA has no launch capacity to lift such heavy loads. Because of this penalty, the Static Deployment and Tether

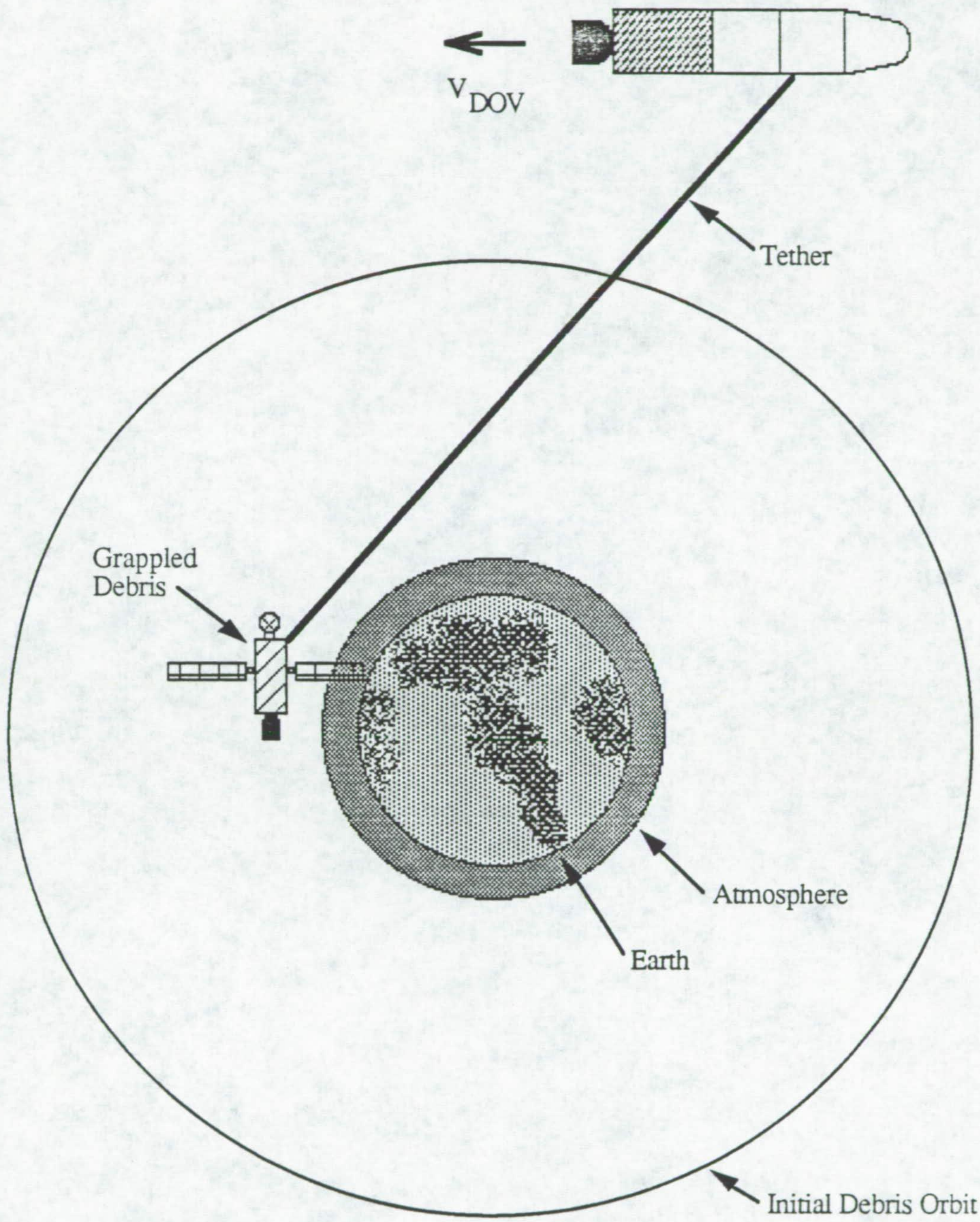


Figure 8. SCHEMATIC OF STATIC DEPLOYMENT WITH TETHER (not to scale)

concept is considered unrealistic at present technological levels.

The Static Deployment and Tether concept requires substantial technological advances before it can be efficiently utilized. The TSS experiments planned for the Space Shuttle are expected to offer more insight into the dynamic control problems of tethered satellites [1]. Also, advances in material sciences are necessary to yield lighter, stronger, monofilament fibers with sufficient strength to withstand the operational loads of de-orbiting before this concept will be feasible.

2.2 Capture and Retro-fire

The capture and retro-fire alternative utilizes the Hohmann Transfer concept discussed in Appendix C. The DOV grapples the debris mass with a tethered net, as with the previous alternative (see Figure 7). The DOV then retro-fires the main engine to slow down the combined DOV-and-debris mass system. The DOV releases the debris mass into an elliptical orbit (concept described in Appendix C) terminating in the atmosphere, where burn-up will occur. Figure 9 shows the schematic of the necessary velocity changes.

The rendezvous and grappling stage of this concept is similar to that used in the static deployment method. As mentioned previously, the grappling and tether attachment procedures are accomplished in a single operation. In this case, the tethered net first captures the debris mass and then controls the movement and rotation of the debris mass in relation to the DOV by use of the tethering mechanism.

The tether can be detached at either the DOV end or the debris mass end. The latter will be more complicated because the cutting mechanism, and possibly a power link, will have to be deployed with the net. Also, several kilometers of loose tether material attached to a DOV moving in earth orbit will create considerable control problems, considering the cut tether will be "whipping" as it is retrieved by the DOV [4].

The impulse to slow the debris is provided by the main engine on the DOV. In

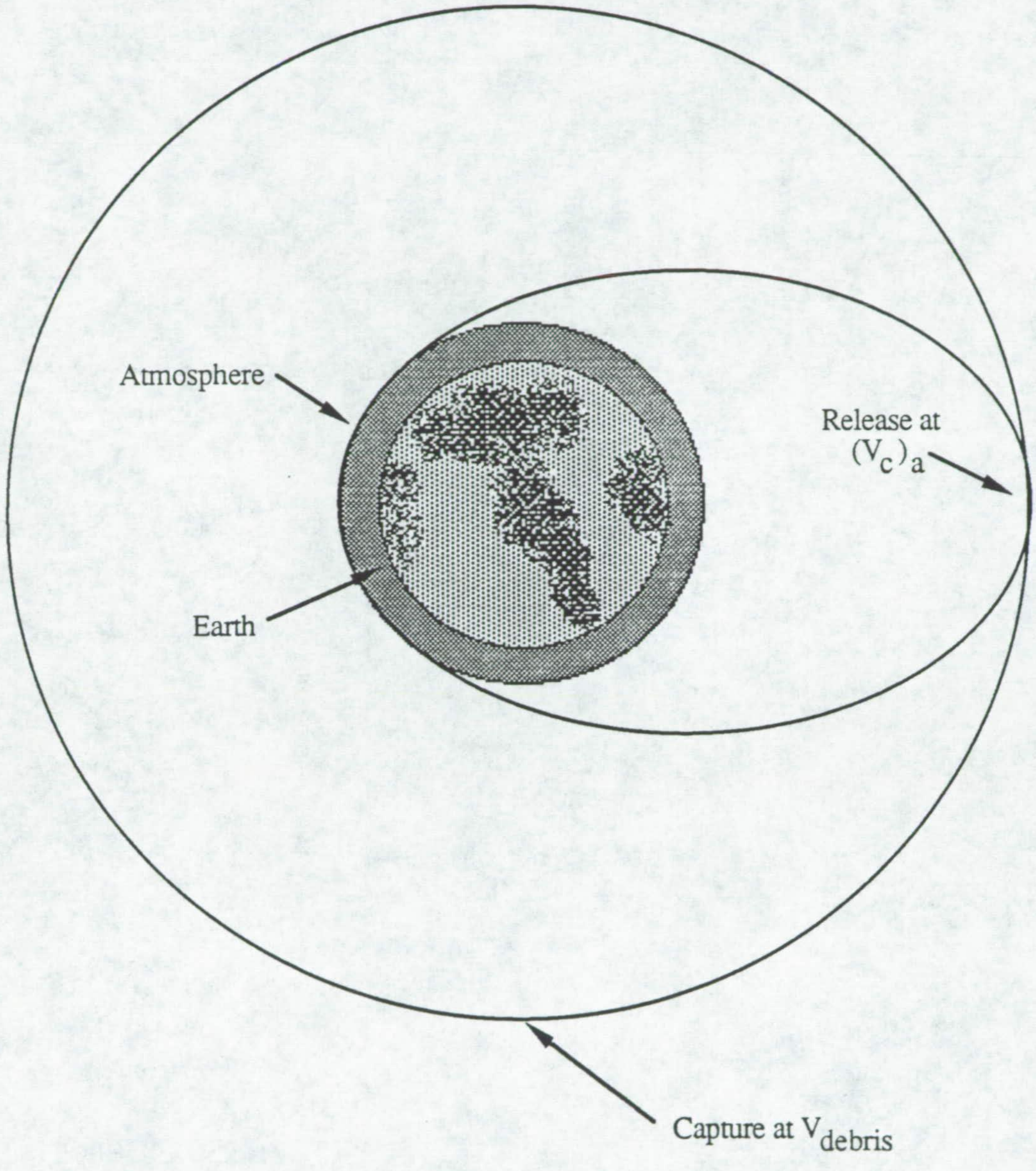


Figure 9. SCHEMATIC OF CAPTURE AND RETRO-FIRE

this case, the DOV's main engine is oriented opposite to the direction of motion and acts as a "retro-rocket". This is similar to the de-orbiting procedure used in manned missions. The magnitude of this impulse will be sufficient enough to lower the altitude of the DOV and debris mass combination such that, when the debris mass is detached from the DOV, its transfer orbit will reach perigee in the atmosphere (at approximately 80 kilometers altitude), while the DOV continues in the original orbit (see Figure 9). The atmospheric drag effects will then cause the debris mass orbit to degenerate further and cause the debris to burn-up.

This operation is much more realistic than the previous concept because the tether used in this case is much shorter. An additional advantage is the combination of the grappling and tether attachment steps, resulting in a simpler process. Finally, the concept of the Hohmann Transfer Orbit (Appendix C) is a proven orbital mechanics concept.

The retro-fire concept using the main engine means that the DOV effectively acts as a brake to slow down the debris mass. Braking is considered one of the most energy inefficient processes [9]. Therefore, fuel usage for the main engines is expected to be very high. This high fuel consumption, as well as the complication of the tethered net design, is expected to seriously constrain the configuration of the DOV.

2.3 Orbital Maneuver and Tether

The third alternative combines the Static Deployment and Capture and Retro-fire concepts in an attempt to utilize the advantages of both. In this case, orbital maneuvering is used first to move the debris mass to low-earth orbit (LEO). The debris mass is then lowered to the 80 kilometer altitude using the tether method discussed in Section 2.1. This concept offers a more realistic method of static deployment by tethering the debris from a lower orbit, thereby shortening the required tether lengths. As in the second alternative, the Hohmann Transfer is utilized to move the debris to the lower orbit (see Figure 10).

The grappling and tether attachment/detachment methods for this de-orbiting

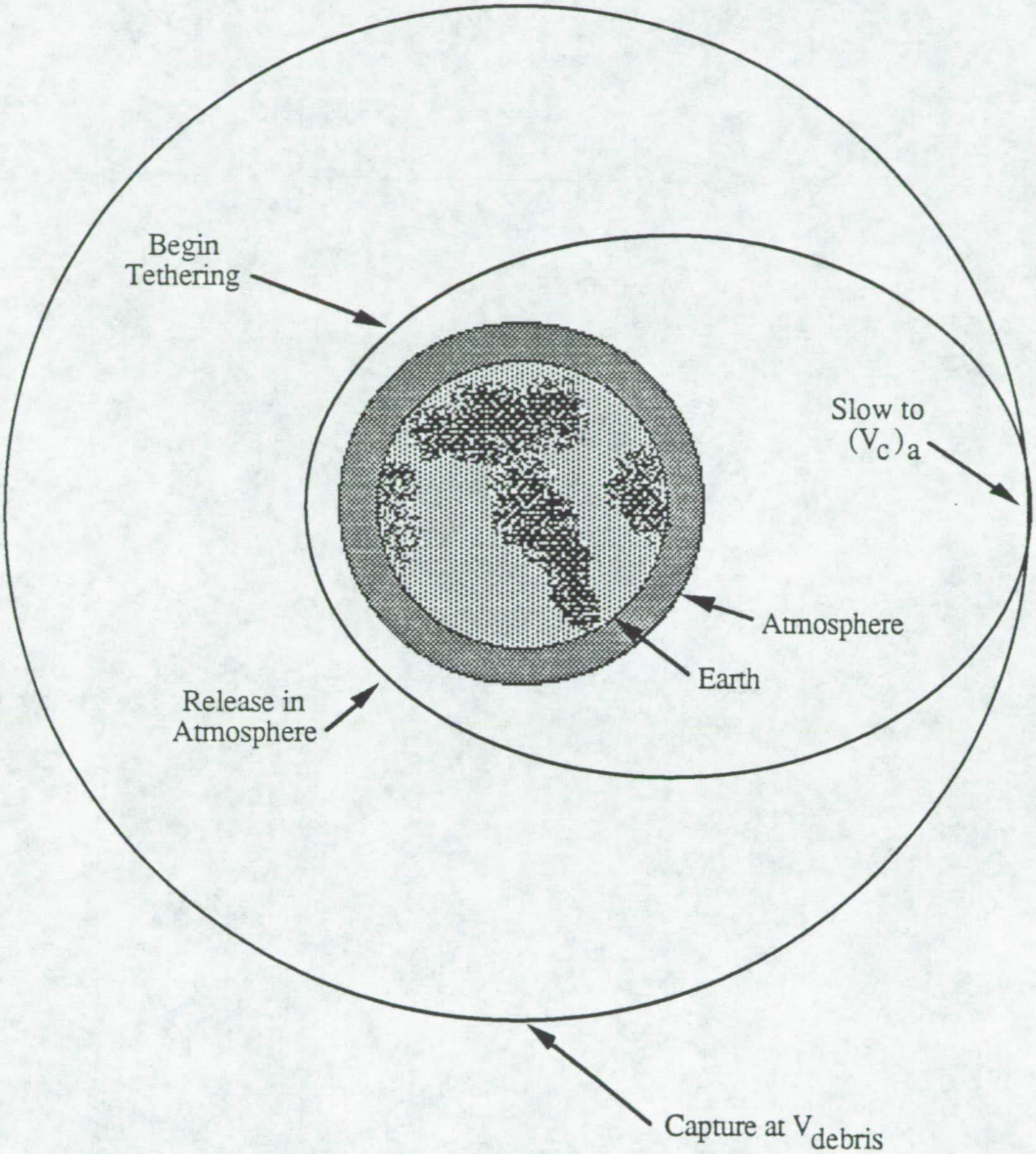


Figure 10. SCHEMATIC OF ORBITAL MANEUVER AND TETHER METHOD

process are similar to those discussed in the first two alternatives. Also, this method has the advantages of simplicity of operation as the tether will already be attached to the net.

This concept requires extensive use of energy. The retro-fire of the main DOV engines is necessary to place the DOV and debris in the transfer orbit (see Figure 10). The

DOV accompanies the debris mass in a connected system to the elliptical orbit low enough (at altitudes of 100 to 200 kilometers at perigee) to use a shorter tether. The debris mass is then moved away from the DOV by the application of an impulse.

This alternative utilizes the potential energy gain of the TSS concept without the mass penalties involved in using long tethers. Also, the shorter tether lengths employed are expected to reduce the associated control problems [4].

Unfortunately, these advantages are still not expected to overcome the disadvantages of poor fuel efficiency and high mass [4]. Though less impulse is necessary for the retro-fire to go to an altitude of 200 kilometers instead of 80, the magnitude of the velocity is only about 10 percent less than the velocity change for 80 kilometers. Meanwhile, it is still necessary to carry all of the tethering material and mechanisms, adding considerable mass to the relatively simpler capture and retro-fire method. Finally, this alternative requires more steps in the procedure, including capture, retro-fire, tethering, and releasing the debris (see Figure 10). This is an added complication without any significant payoffs in terms of material savings by using a shorter tether.

2.4 Kinetic Energy Projectiles

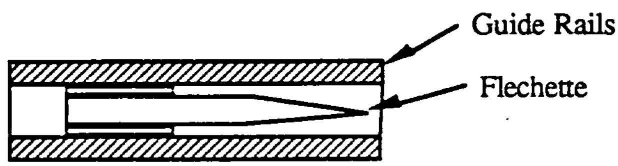
The kinetic energy projectile method employs the negative KE imparted to a debris mass by a projectile fired by the DOV to de-orbit it. In this case, a high velocity is imparted to the projectile to attain a high kinetic energy without requiring a very high mass. In this method, the problems of docking with and grappling a tumbling debris mass are solved by avoiding docking completely.

The DOV is programmed with the orbit of the target debris and rendezvous with it using the onboard inertial guidance system. At closer ranges, typically 5 to 10 kilometers from the target debris mass, the DOV will initiate an autonomous tracking mode and close in on the target debris mass using the search and tracking system. This track and scan unit will consist of the F-16 radar and a LANTIRN pod (see Section 1.6.2). The image of this debris mass will be transmitted by an integral command link to the human controllers at an orbiting station or on earth. There is a finite time lag induced by the distance between the operator and the DOV. Therefore, the rendezvous and firing procedure will have to be carried out in stages over a longer period of time. At close range, the operator can use his/her judgement or earth-based computational facilities to determine the exact tumbling characteristics of the mass. Also, a very close approximation of the mass can be obtained using the NORAD satellite catalog and the Royal Aircraft Establishment (RAE) Satellite tables [8].

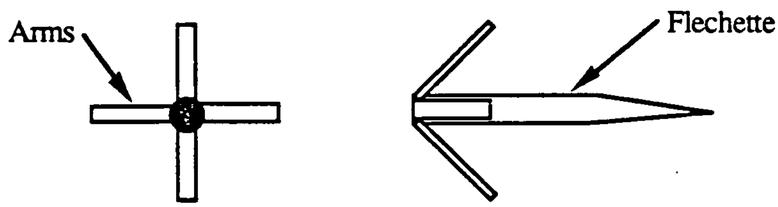
Once the exact point of impact has been chosen, the projectile is fired either by direct operator command or by a programmed sequence initiated by the human operator. A debris mass in earth orbit has a specific kinetic energy that is a function of the mass, orbital velocity, and orbital altitude. This KE in turn influences the magnitude of the required change in KE to de-orbit that particular debris mass. The orbital velocity and altitude will be calculated using the Doppler Radar/IR image and, with the mass approximation used to calculate the velocity required for the projectile. Because the projectile is fired and "sticks" into the target, the collision can be modelled as a momentum transfer with the projectile velocity as the only unknown if the mass of the target is known.

Two different types of projectiles are being considered for use in the kinetic energy method of de-orbiting debris. One uses a "spearing" action similar to that used in APFSDS (Armor-Piercing, Fin-Stabilized, Discarding Sabot) tank guns, and the other uses the "Chain-and-Bar Shot" (CABS) concept used in sixteenth century sailing "men o' war". However, unlike their predecessors, these projectiles impart kinetic energy instead of a destructive impact to their targets.

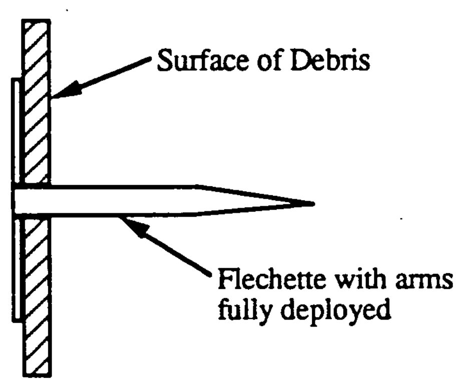
<u>2.4.1 The Flechette.</u> The APFSDS tank projectile uses kinetic energy to destroy a target. A very high powder charge is used to propel the projectile inside a sabot down a barrel. The projectile is tubular and shaped like an arrow (see Figure 11). The sabot is an insert that guides the assembly along the barrel. As soon as the projectile leaves the barrel, the sabot halves are discarded and the fin-stabilized "arrow" speeds at a very high velocity



a. Flechette in guide rails



b. Rear and side views as arms deploy



c. Flechette after piercing surface of debris mass

Figure 11. OPERATION OF THE FLECHETTE

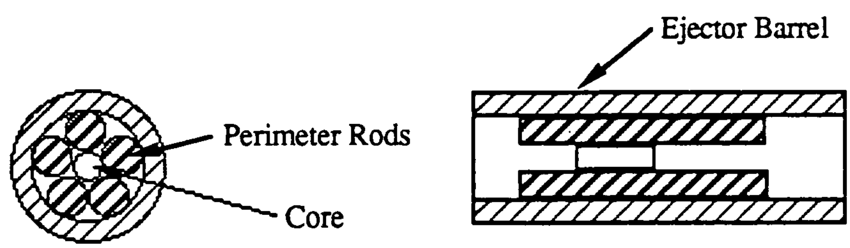
toward the target and destroys the target on impact. In this case the relatively small and low mass arrow gains all of its energy from the high velocities it attains. Therefore, the size and mass of the projectile can be minimized for stability and accuracy.

The flechette uses the APFSDS concept of imparting a high KE to a low-mass projectile. In the de-orbiting concept, however, the emphasis is on imparting the necessary kinetic energy to the target debris mass without causing any damage to it. Damage may be caused by the heat and possible structural destruction induced by impact. This exacerbates the debris problem by creating even more debris [13].

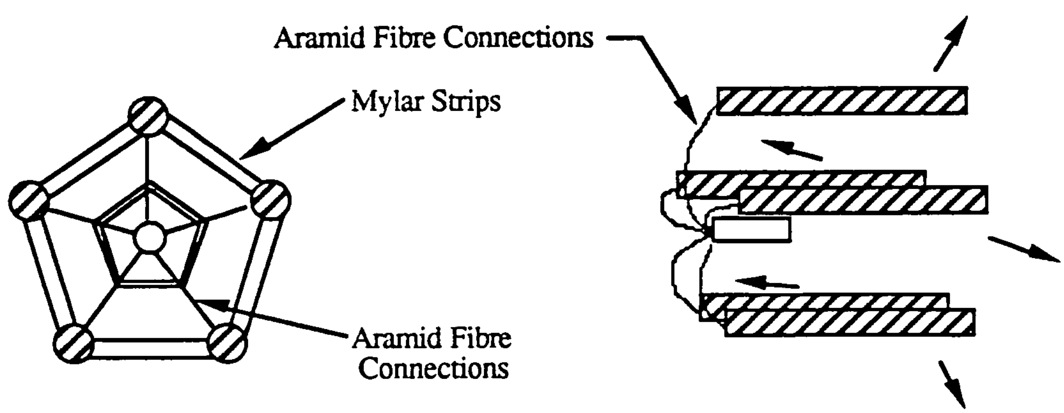
The flechette will be sharp-tipped and circular in cross section to aid in the penetration of the skin of the debris mass. The flechette will be constructed of steel, making the flechette harder than the aluminum used in the construction of the majority of spacecraft; the projectile must be harder to ensure its survival and preventing it from acting as a "fragmentation grenade" as in the APFSDS system. Most debris objects are either spent (and hollow) upper stages or inoperative satellites, increasing the possibility of the flechette going through the debris mass. There will be four arms near the end of the flechette to stop it at the surface of the debris mass. These arms will prevent the flechette from going through the body of the debris mass itself (see Figure 11). The pads will also transfer the momentum of the projectile more evenly onto the debris mass, lowering the probability of debris braking off due to impact.

2.4.2 Chain-and-bar shot. The CABS idea was used in the sixteenth and seventeenth centuries by naval "men o' war" with devastating effect. Rods were tied together with chains, bundled into a cylindrical shape and fired from a cannon. Once out of the barrel, the rods spread due to centrifugal force and formed a lethal chain moving at fairly high velocities. This had a scything effect on the target, cutting down rigging, masts, and the rest of the superstructure.

The CABS concept for de-orbiting debris, however, only utilizes the centrifugal effect on the components of the projectile rather than the "scything" effect employed in the original idea. A number of perimeter rods will be grouped around a core rod. These rods and the core will fit together as one projectile inside the barrel of the ejector or "gun" (see Figure 12a). The ejector provides the impulse to impart the necessary velocity to the projectile. The barrel of the ejector has grooves and the CABS projectile fits along the grooves. When the projectile is accelerated, the grooves cause a spinning motion of the projectile. When the projectile leaves the barrel, the rods are no longer constrained to stay



a. Chain-and-bar-shot projectile in ejector barrel



b. Core and rods seperated during flight

Figure 12. OPERATION OF THE CHAIN-AND-BAR SHOT

around the core, and the centrifugal force due to rotation causes the rods to fly apart (see Figure 12b). The lines connecting each perimeter rod to the core rod cause the perimeter rods to spread out in a circular pattern around the core. There are strips of mylar or a similar light-weight material at intervals along the lines to keep the perimeter rods and core as a single unit. These strips maintain a constant "spread" of the perimeter rods to ensure a sufficiently large capture area to increase the chances of hitting large and small debris.

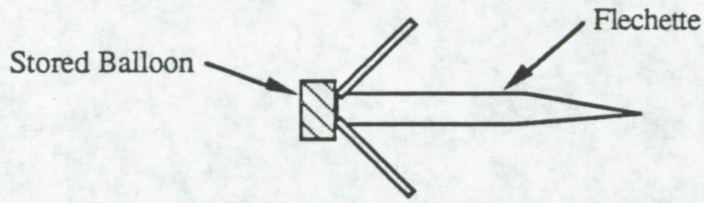
2.5 KE with Balloon Attachment

The Balloon Attachment method uses the drag effect of the atmospheric periphery on low earth-orbiting satellites. This method was proposed by Andrew J. Petro and David L. Talent [17] as a solution to the space debris problem in LEO. This method can be used in combination with the KE flechette method; in this case, the DOV will have a dual supply of both KE Flechettes and KE Balloon Attachment type projectiles in it. The flechettes will be used to de-orbit masses in higher (above 500 kilometers) altitudes while the balloon attachment will be used at lower altitudes where it will be effective [17].

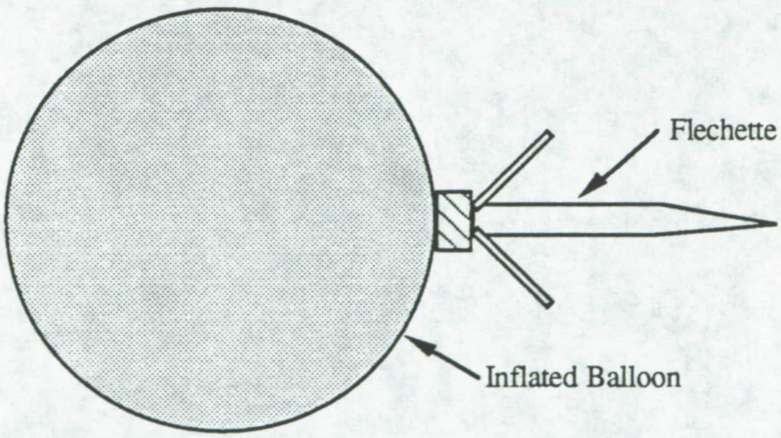
The DOV will rendezvous with and de-orbit debris masses in high earth orbit in the procedure described in the previous section, using KE to de-orbit those masses. However, the procedure will be modified for debris masses in LEO. In this case, the DOV will get close (within 5 kilometers) to the debris mass in LEO and gradually match the mass' velocity. Using the on-board sensors (discussed in Section 1.6.2), the remote operator will then decide the exact target location for the projectile and initiate the firing sequence. The projectile will then be fired with sufficient velocity to lodge on the surface of the debris mass.

The projectile itself will be shaped like a flechette (see Section 2.4), with an attachment on it. This attachment will be a deflated balloon with an integral inflation mechanism. Once the projectile is lodged onto the debris mass, a command signal will activate the inflating mechanism and inflate the balloon (see Figure 13).

The inflated balloon attached to the debris mass in LEO will effectively increase the surface area of the debris and the drag effects of the atmosphere, thus gradually decreasing the debris mass' velocity [17]. The atmosphere extends to approximately 80 kilometers above the earth's surface, but the drag effects can be measured up to 500 kilometers above



a. Flechette prior to balloon deployment



b. Flechette with inflated balloon (not to scale)

Figure 13. FLECHETTE WITH BALLOON ATTACHMENT

earth orbit. This drag effect with the area increase caused by the balloon is expected to decrease the debris mass' orbital life by 90 to 95 percent. The drag effect is also expected to cause heating of the balloon material as well as the gas inside it. However, the heating problem can be controlled by using mylarized material similar to that used in ECHO 1 balloons [17].

The possible KE Balloon and KE Flechette combination is versatile because it can de-orbit debris in LEO (with a balloon projectile) or high orbit (with a flechette projectile). With this method, virtually any debris mass in the mass range of 150 to 2,000 kilograms can be effectively de-orbited. However, this method is also complicated and more massive because of the need to carry effectively two different systems. One possible solution may be the use of two different types of DOVs: one in LEO using the Balloon Attachment, with the other at higher altitudes using the KE Flechette. The DOV is required to have a modular configuration (see Section 1.6), therefore the DOV can be configured for the two different methods by changing the hardware to match the de-orbiting mission profile.

Design Solution

This section presents the final design solution developed for an unmanned, reusable vehicle to de-orbit debris in earth orbit. The alternative designs were first evaluated using a decision matrix (see Appendix D). This evaluation led to the selection of the CABS concept of de-orbiting debris masses.

Following the presentation of this evaluation, the configuration of the DOV is described in detail. The operating procedure of the DOV concludes the section; this procedure has three stages: orbital insertion, the de-orbiting procedure, and replenishment of the consumables.

3.1 Evaluation of Alternative Designs

The Alternative designs were evaluated using a decision matrix developed by the design team (see Appendix D). The decision criteria used were based on the requirements and success criteria established by the team at the beginning of the project. A weighting factor was also assigned to each of the criteria based on the relative importance of that particular criteria to the others. Each alternative was graded on a scale from 1 to 10 for each decision criteria . The grade was then multiplied by the weighting factor to obtain the score for that criteria. These numbers were then used to generate the decision matrix shown in Appendix D.

<u>3.1.1 Fuel/Power Efficiency.</u> Efficiency of the de-orbiting concept in terms of fuel and power consumption was one of the team's primary design objectives and the weighting

factor of 0.20 reflects the importance of fuel and power efficiency. As noted in the alternative designs (see Section 2), some of the de-orbiting concepts consume large amounts of fuel. Fuel consumption calculations were based on the approximate mass for each de-orbiting method. The consumption calculations for each alternative design were then used to rank the alternatives for fuel and power efficiency for the decision matrix. The team determined the KE - Flechette and KE - CABS de-orbiting methods to be the most efficient in terms of fuel and power consumption.

3.1.2 Life of Consumables. The frequency of replenishing the consumables affects the number of de-orbiting operations that can be conducted per mission. Assumptions were made for the number of operations that each alternative design can achieve in one mission. These assumptions were based on the size of the consumables and general mass estimates of those consumables. Since the life of the consumables is a key factor in the efficiency of the DOV, it was assigned a weighting factor of 0.15. The designs were ranked for the life of consumables category. Since the KE - Flechette method's consumables are the smallest and lightest, it was assumed this method would complete the most de-orbiting operations per DOV mission.

3.1.3 Mass Minimization. This category was another important requirement for the design of a successful DOV. A weighting factor of 0.20 was assigned to mass minimization to reflect the importance of this criteria. The design team addressed mass minimization with respect to the additional equipment required for each de-orbiting method. According to this analysis, the Orbital Maneuver and Tether method required considerably more equipment than the other alternatives. Then the alternative designs were ranked for the mass minimization category.

3.1.4 Simplicity. The key concept of modularity was addressed in terms of simplicity. The replenishment of the consumables is required to be completed in one step ("magazine" form) to simplify the procedure. This requirement also applies to the repair and maintenance of the DOV. Since the Orbital Maneuvering and Tether method involves the most equipment, it was determined to be the most difficult to repair and maintain. The importance of simplicity is reflected in the weighting factor of 0.175. Most alternatives received a grade in the region of five since the equipment maintenance in space is considerably harder than on earth.

3.1.5 Number of Operations Required. The number of steps required in the de-orbiting operation are also an indicator of the relative efficiency of a particular concept. Therefore, in terms of the number of operations, the KE concepts ranked higher than the tethering concepts because each contains only one step (firing the projectile). However, as a consideration for a successful design, this criteria received a weighting factor of 0.1375.

3.1.6 Secondary Debris Propagation. The minimization of secondary debris propagation during the de-orbiting procedure is considered important as it coincides with the project's objective of minimizing orbital debris. A weighting factor of 0.1375 was assigned to this criteria. The concepts which employ nets to capture the debris have the lowest probability of secondary debris propagation due to the low impact magnitudes involved and were given a much higher grade than the methods involving momentum transfer through impact

3.1.7 Design Solution Selection. Once the six design criteria were chosen and evaluated, a decision matrix was generated to see which alternative design would be the most likely to follow through to the problem solution. The decision matrix in Appendix D shows the CABS method for de-orbiting debris resulted in the highest score and was chosen as the design for the problem solution. Although the gradient between the designs is not very large, the design team feels that the CABS method will prove to be the most effective method for attaining the objectives of the problem solution.

3.2 De-orbiting Vehicle Configuration

The DOV utilizes the concept of modularity in the design of its overall configuration (see Section 1.6). Therefore, the vehicle is divided into three modules according to their function. These modules are the de-orbiting, command, and propulsion modules (see Figure 14). The DOV uses the NASA-TRW Orbital Maneuvering Vehicle (OMV) as the Command-Propulsion modules. This design decision also satisfies the NASA requirement of maximum reusability and modularity of components (See Appendix A, Specification 1.2).

The OMV is designed to perform as a utility tug to boost Shuttle-launched satellites into higher orbits up to 1250 miles beyond that of the Space Shuttle [2]. The prime contractor for the OMV, TRW Federal Systems Division, has also designed the OMV to accept a wide variety of operational modules. Therefore, the design team decided to configure the de-orbiting module as a unit that can be attached to the OMV Command-Propulsion unit to "transform" it into a DOV. As a matter of fact, debris de-orbiting with the attachment of a specialized module was considered in the original specifications of the OMV [18]. The following sections describe the power and command modules of the OMV, and the de-orbiting module.

<u>3.2.1 Propulsion Module.</u> The propulsion module is primarily a sub-frame

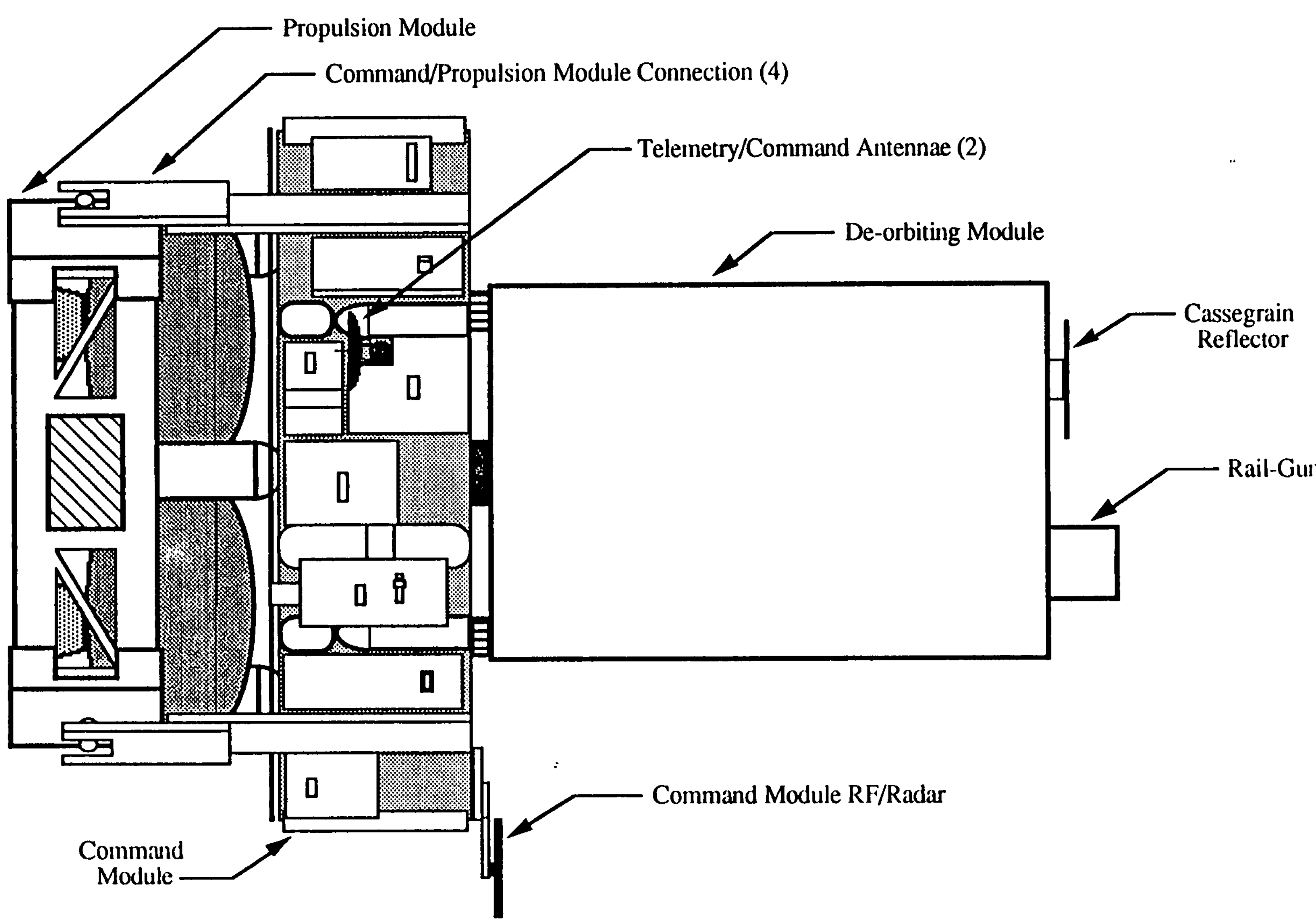
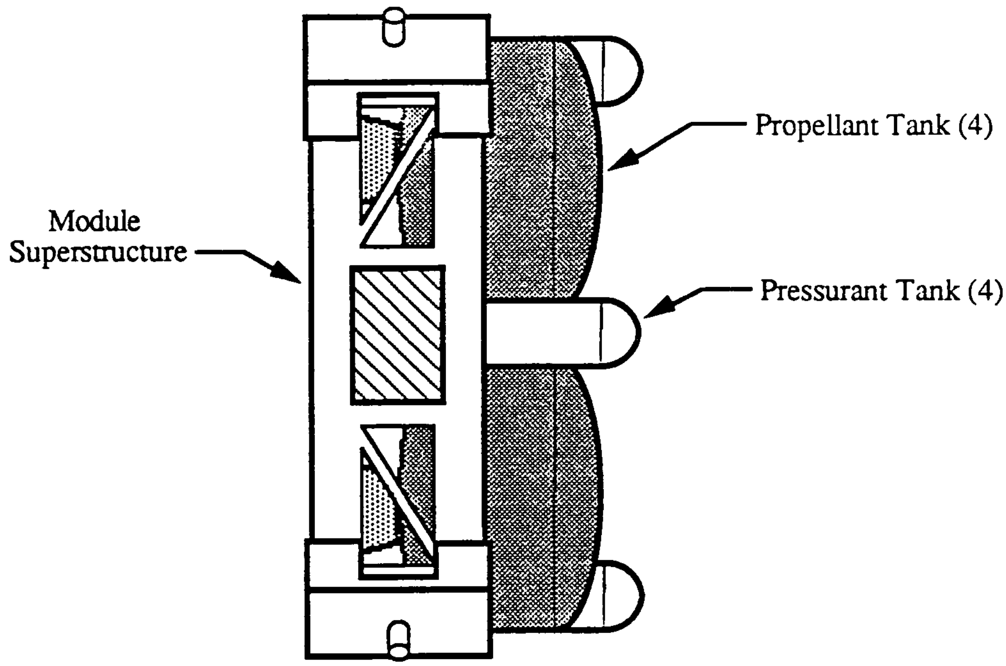


Figure 14. DOV CONFIGURATION

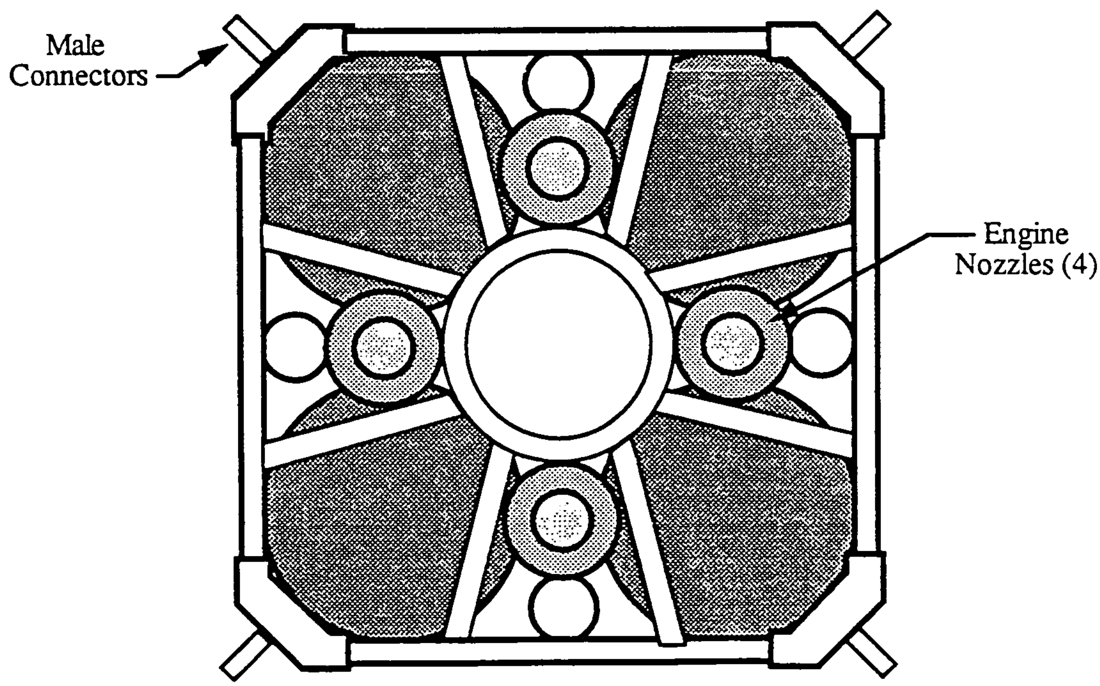
housing four liquid-fueled thrusters and their propellant tanks (See Figure 15). The four bi-propellant engines have variable thrust ranging from 13-130 pounds [2]. The engine nozzles are arranged in a cross-shaped arrangement at the rear face of the propulsion module with the propellant tanks arrayed right behind them. Pressurized nitrogen Reaction Control Thrusters (RCTs) are used for altitude control as well as low-level thrusters during docking operations. All the control systems are triplex redundant with additional mechanical triplex redundancy for the propellant feed lines [5]. The pressurized tanks (see Figure 15a) on the propulsion modules contain high-pressure helium used to maintain pressure in the propellant fuel lines.

The fuel and pressurized can be replenished by either "topping up" the tanks when the system is docked with the Space Station or simply by replacing the depleted propulsion module with a fully replenished one. The four main thrusters can be used separately or together to give a wide range of propulsion settings and configurations. However, this versatility has the attendant risk of an accidentally unstable propulsion configuration. To avoid this risk all propulsion configuration commands from the Ground Control Center (GCC) are automatically double-checked for stability before execution. The propulsion module is attached to the command module by four electro-mechanically operated "male-female" type latches. The command signals from the Command module are transmitted through a digital data-bus connection.

3.2.2 Command Module. The command module houses the control systems, command-data links, data storage systems, and fuel cells for the entire vehicle. The propulsion and de-orbiting modules receive their electrical power and system commands from the command module through the digital data-bus system. The system commands are generated by the operator at the Ground Control Center (GCC) and transmitted to the DOV via the Command/Data link.



a. Side view



b. Rear view

Figure 15. PROPULSION MODULE

The command module is relatively short in comparison to its diameter (see Figure 14). The module was designed deliberately like this to facilitate placement of all subsystems on the perimeter. Using a system similar to that used in the USAF's Line Replaceable Unit (LRU) concept, all the subsystems are designed to be quickly replaced in case of a malfunction. Placement of the subsystems modules in the perimeter considerably simplifies this replacement procedure. The fuel cells are based inside, closer to the center of the command module.

Two Telemetry-Command antenna situated on the module's perimeter (see Figure 14) are used to maintain a constant two-way communications ink with the GCC. The GCC is planned to be situated at NASA's Johnson Space Center, the Space Shuttle, or the Space Station. It is configured like a standard aircraft cockpit with stations for a pilot, co-pilot, and two flight engineers (see Figure 16). These operators constantly monitor the DOV's systems and effectively fly it using the Telemetery-Command systems. In case of a communication failure, however, the telemetery data can be stored onboard for subsequent retrieval and the DOV flown on automatic "fail-safe" mode. In fact, the DOV can fly a pre-programmed mission profile (see Appendix G, Figure G1). Therefore, most of the communications equipment can be shut down during transit to the rendezvous point to conserve power. There is also a pulse-Doppler range-finding radar antenna attached to the rim of the command module. This unit is used only when the vehicle is operating as an OMV without the de-orbiting module attached. When the vehicle operates as a DOV, it will be "flown" by the GCC using the de-orbiting module's pulse-Doppler radar and TV/IR system.

3.2.3 De-orbiting Module. The de-orbiting module is the "front-end" of the DOV and contains the sensors, the CABS projectiles, and the ejector for the CABS projectiles. This module is designed to be replaced as a single unit during replenishment procedures. It

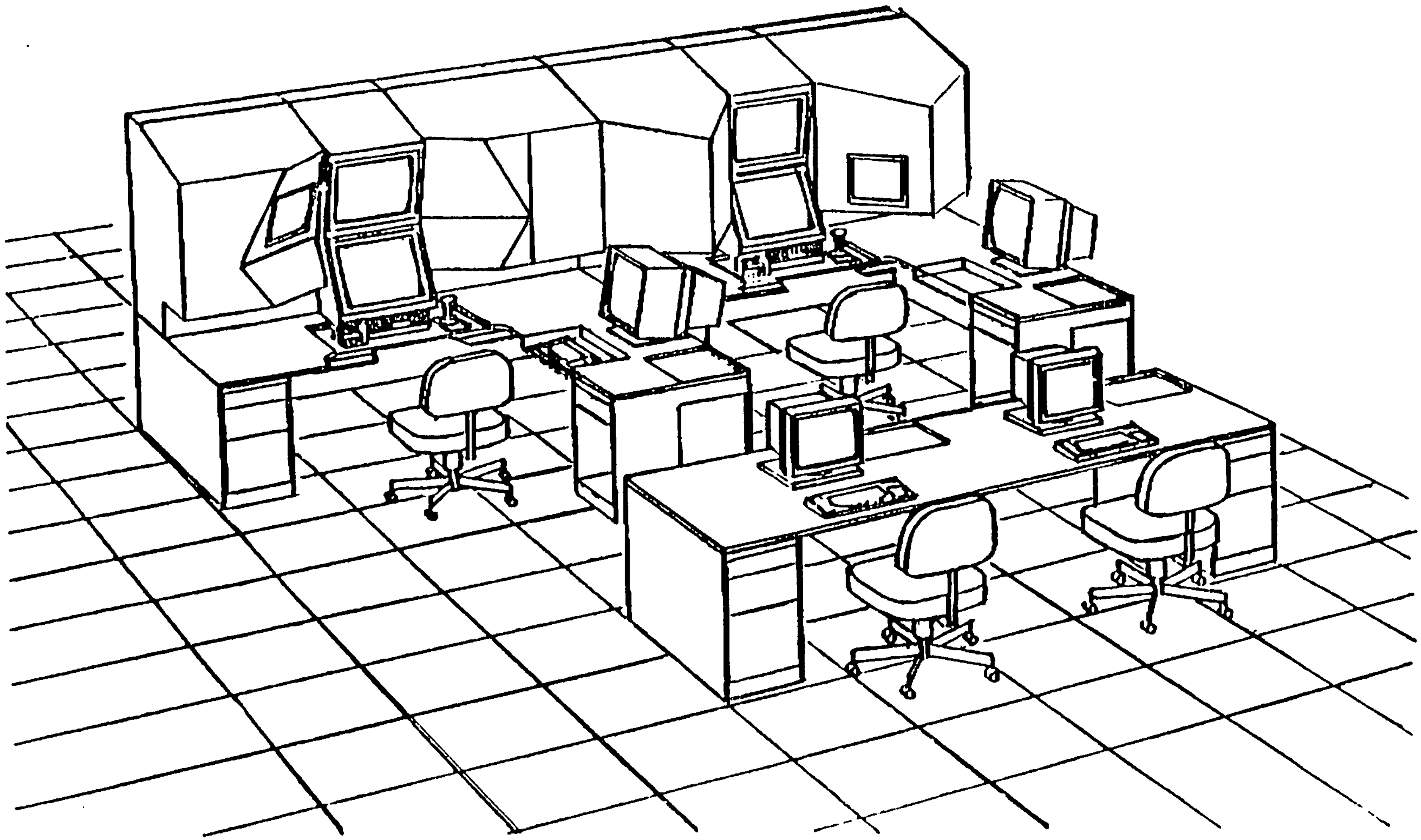


Figure 16. GROUND CONTROL CONSOLE (Illustration from OMV briefing to NASA JSC by TRW Federal Systems Division)

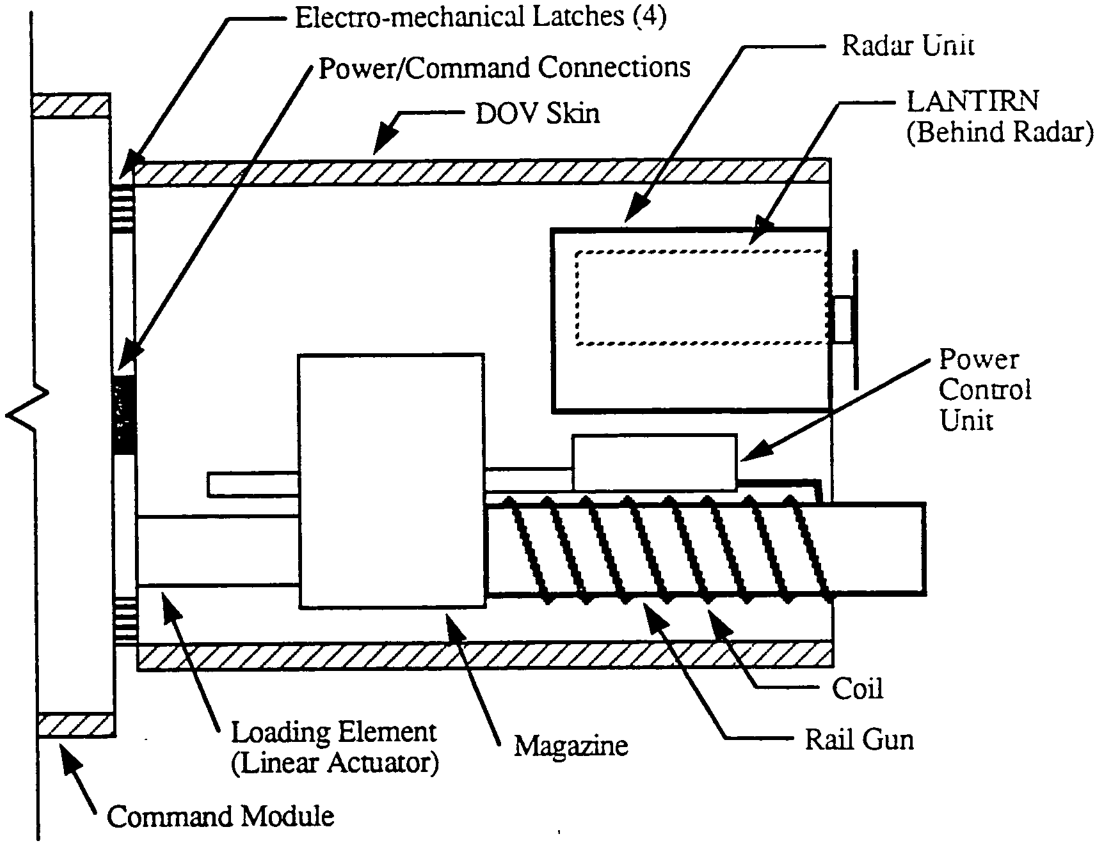
is connected to the front of the command module with electromagnetically operated "latches" that can be released by radio command simplifying the replenishment process for the astronaut (see Figure 17). The de-orbiting module will also have an integral attachment point for the robotic arm that will be used to maneuver the module into the payload bay of the Space Shuttle. The following section describes the de-orbiting module and the associated components, i.e., the CABS projectile, the magazine, the ejector, and the sensor system.

As discussed earlier (see section 2.4.2), the CABS projectile is a system of rods attached to the core rod with flexible lines. The perimeter rods in this case are not circular in cross-section but "wedged" (see Figure 18). Six of these rods fit tightly together around the core. The flat face of the core faces backwards inside the ejector barrel. The aramid fibers (commercially known as Kevlar 69) connect each of the perimeter rods to the core at the "front" of the core rod. The "structural integrity" of these projectiles is maintained by the cylindrical housing (similar to a revolver chamber) on the magazine during storage and loading and by the polished inner surface of the ejector barrel during the firing sequence.

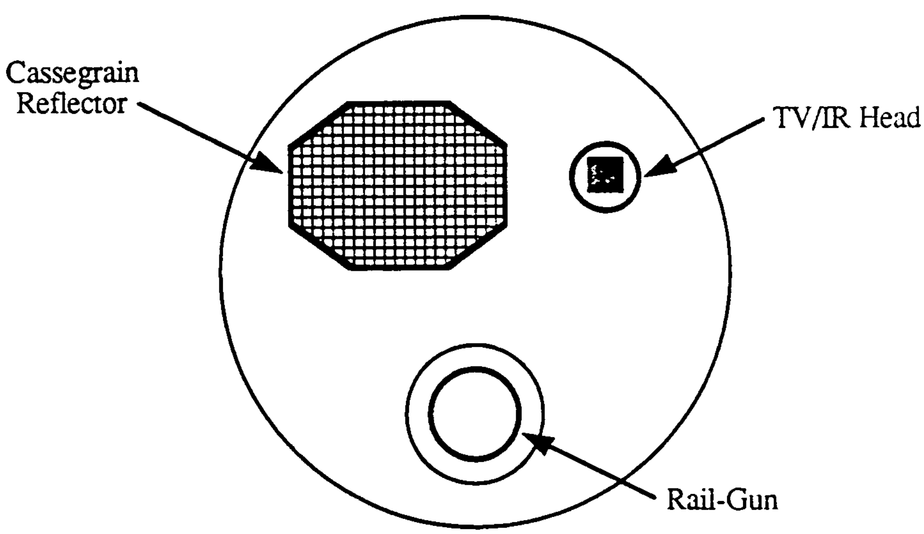
The CABS projectiles will be stored in a removable "drum-magazine" (see Figure 19). Each magazine will have an integral motor with hollow cylinders attached to the outside of the motor. In this case the rotor of the motor will be stationary and act as the axis of rotation for the whole unit. The CABS projectiles will be housed in the hollow cylinders and arrayed around the motor.

The magazine will rotate each of the projectiles into position for loading into the ejector. An electrical linear actuator with a "plunger" attachment (see Figure 20) will push each projectile into the ejector. The cylinder and the barrel will maintain the integrity of the projectile during this transitional phase. After each projectile is fired, the magazine will rotate another projectile into position and prepare it for loading.

The ejector is an electromagnetic rail-gun using a moving magnetic field to

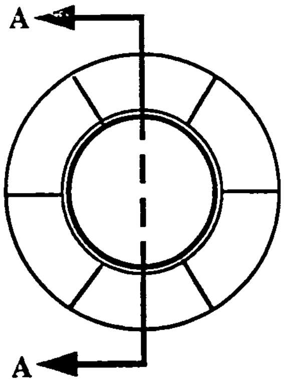


a. Side section view

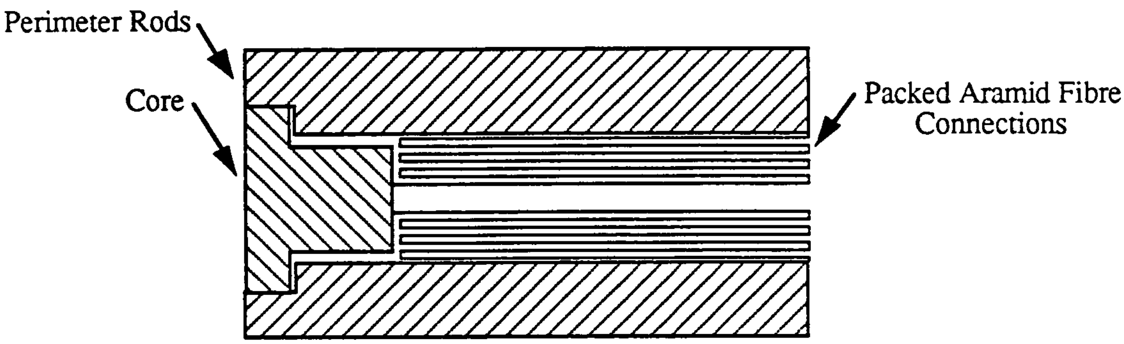


b. Front View

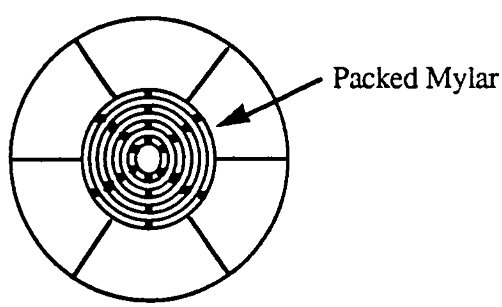
Figure 17. DE-ORBITING MODULE CONFIGURATION



a. Left end view of projectile

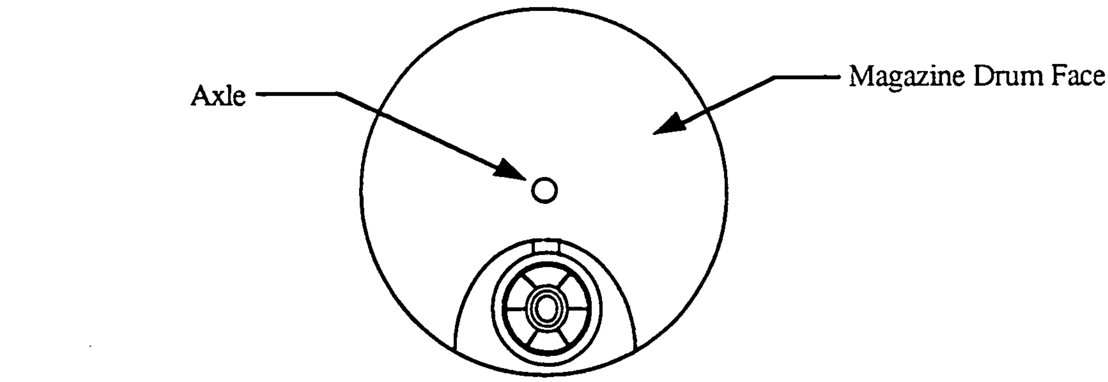


b. Side section (A-A) view of projectile

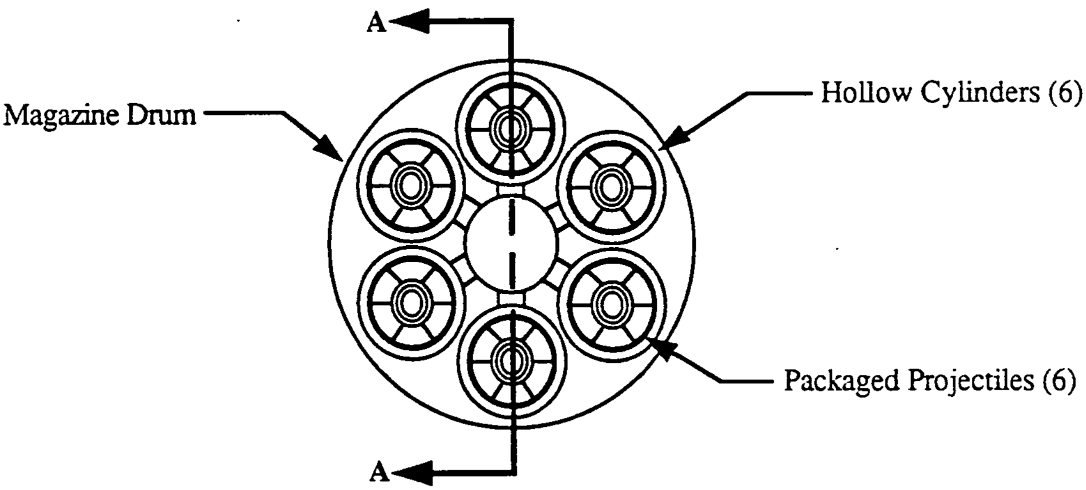


c. Right end view of projectile

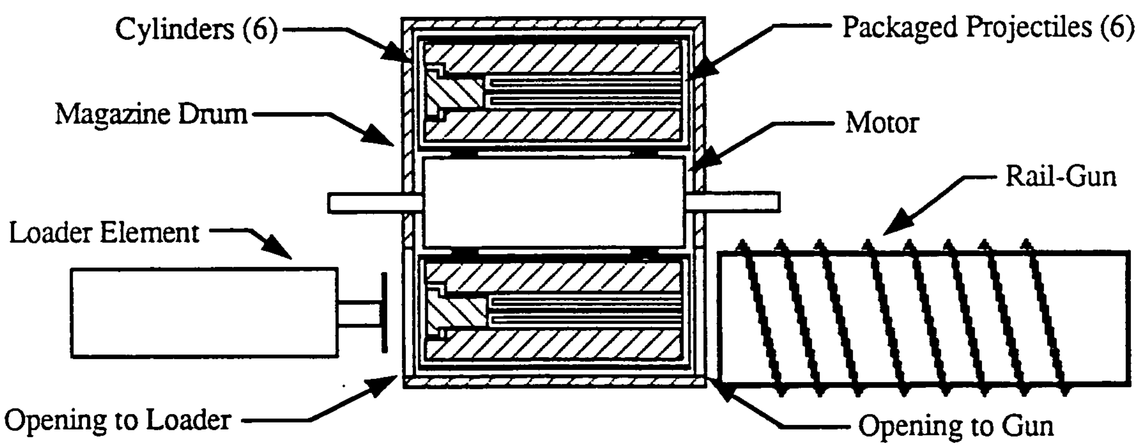
Figure 18. PACKAGED CABS PROJECTILE



a. Front view of loaded magazine drum



b. Front view of loaded magazine (with front cover removed)



c. side section (A-A) view of magazine, loader, and rail-gun

Figure 19. CABS MAGAZINE CONFIGURATION

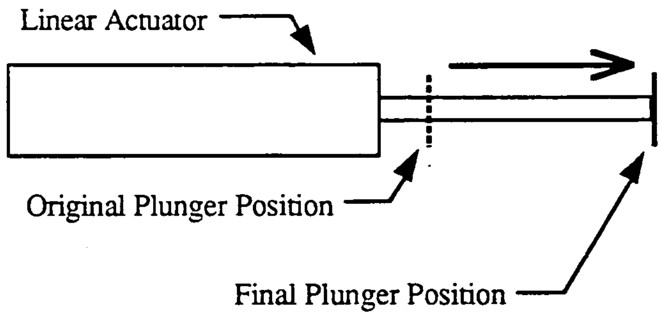


Figure 20. LINEAR ACTUATOR FOR CABS PROJECTILE

accelerate a projectile to very high velocities. As the name suggests the gun uses a pair of rails to guide its projectile. However its technically feasible to use the same procedure in a barrel [7]. The projectile is made of a ferrous material (in this case steel) and the magnetic field accelerates it forward without any actual physical contact. In this case, the magnetic field also "rotates" around the longitudinal axis of the gun to provide the spin to the CABS projectile. When the CABS projectile leaves the barrel, the centrifugal forces due to the spin causes the perimeter rods to "fling" outwards and deploy around the core in a radial pattern (see Figure 21). Therefore, unlike the original idea (see Section 2.4.2), the barrel does not require any rifling and can be given the smoothest possible finish to minimize friction losses.

The sensor system is a combination Doppler Radar and TV/IR system. The Doppler Radar will be used to track the target debris at ranges between 10 and 100 kilometers and then the TV/IR system will be used for ranges less and 1000 meters. The Doppler Radar is a high-resolution and high pulse-rate frequency unit similar to the one used in the General Dynamics F-16 fighter-bomber (see Appendix E) [12]. The planar array (in this case a cassegrain reflector) has two degrees of motion and is mounted on the front face of the de-orbiting module. The barrel of the rail-gun is extended ahead of the face of the radar reflector to avoid the possibility of reflector damage by the CABS

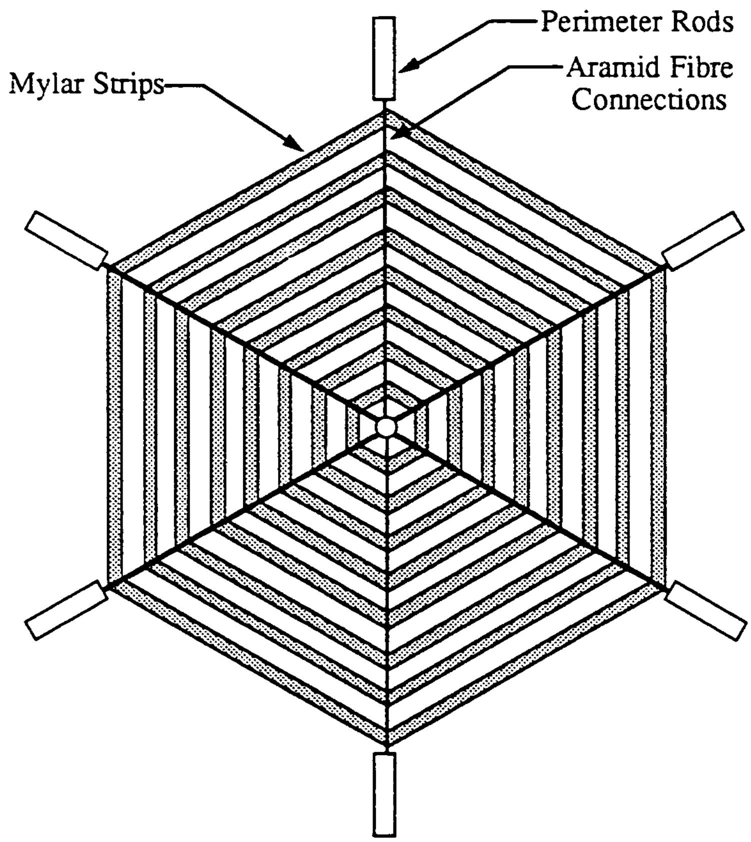


Figure 21. DEPLOYED CABS PROJECTILE

projectile.

The TV/IR sensor head is mounted next to the radar reflector on the front face of the de-orbiting module. This system uses the same lens for the TV/IR systems, electronically switching systems by operator command (see Appendix F). The lens is mounted on a fully stabilized two-axis-of-motion swiveling head. During transit, when the TV/IR system is not required, the sensitive lens is covered by an electrically actuated "iris".

The entire de-orbiting module is attached to the front of the command module by electro-mechanical latches. These latches can be disconnected by radio command when the de-orbiting module has to be removed. During replenishment operations, the de-orbiting module is therefore disconnected and maneuvered into the Space Shuttle payload bay or to the Space Station. The outside of this module is also provided with attachment points for the remote manipulator or to facilitate secure attachment.

The de-orbiting module requires power for its onboard system as well as a link to the system controller in the command module. These quick-release connections will allow quick and easy disconnection of the de-orbiting module from the DOV. Also, two different types of power connections are required: a "low power" connection for the sensor systems, and a "high-power" one for the electromagnetic rail-gun. The rail-gun will actually be connected to a capacitor that will "charge-up" between firings and then provide the very high electrical voltage required for the actual firing [7].

3.3 DOV Operating Procedure

The DOV operates in three stages: orbital insertion, de-orbiting, and replenishment and repair. Orbital insertion is the actual rendezvous of the DOV with the target debris mass and consists of the orbital altitude and plane changes necessary for the DOV to rendezvous with the target debris mass. The actual de-orbiting process, including acquisition of the target and release of the CABS projectile, is initiated when the DOV rendezvous with the target. The final step is the replenishment of and/or repair of the DOV by astronauts from the Space Shuttle or Space Station Freedom.

The following three sections describe these operating procedures. A flowchart showing the sequence of the steps within each procedure is included in Appendix G. This flowchart can be used for development of the control software and operational checklists.

3.3.1 Orbital Insertion. Orbital insertion is the transportation of the DOV to the altitude and inclination of the target debris. The initial orbital insertion by the DOV will be a launch from earth using either an expendable booster or the Space Shuttle. After each replenishment procedure in the DOV's life cycle, the DOV will use the main engine to reach the new target rendezvous.

The original specifications required the DOV to be configured to allow initial insertion by either an expendable booster or the Space Shuttle. The orbits most heavily populated by United States and Soviet space vehicles are at inclinations of 28.5 and 74 degrees, respectively [13]. An expendable booster will be able to directly insert the DOV into either region [4].

After the initial insertion, the DOV will have to fire its main thrusters to transfer to the target's orbit. Both the Space Shuttle and Space Station Freedom (the two vehicles to be used for replenishment) are planned for missions in LEO and 28.5 degrees inclination [4]. This procedure will require expenditure of extra fuel. However, it will be required after each resupply operation in any case (see Section 3.3.3).

3.3.2 De-orbiting Procedure. The de-orbiting procedure begins with the rendezvous of the DOV and target debris. The DOV approaches the debris using an inertial navigation system (INS) and the pre-programmed rendezvous coordinates until it reaches the range of a short-range radar (see Figure G.1, Appendix G). Doppler radars can achieve good image resolution by utilizing high pulse-repetition frequencies, but at the penalty of shorter operating ranges. In addition, these radars tend to use considerably less power. As shown in Appendix G, the procedure to orient the radar is iterative where the calculated range-to-target is continuously updated to initiate the radar "power-on" sequence.

Once within visual range, the DOV activates the TV/IR system. As the DOV closes within visual range of the target, the GCC operators, using the TV/IR system, begin acquiring firing parameters. Those parameters, fully discussed in Appendix G (see Figure G2), as well as the controller determine the mass of the debris, the velocity the CABS projectile must achieve, and the probability of a successful procedure. The controller uses all data acquired to make the "go/no go" decision, determining whether to fire or abort. If the first shot taken by the DOV is unsuccessful and does not de-orbit the target, the DOV can "go around" for a second shot. The controller makes this decision as the DOV continues tracking the target.

After each de-orbiting procedure a full systems and status check of the DOV is conducted. This automatic checklist procedure, known as a Built-In-Test (BIT), is conducted with the data transmitted to the controller via the telemetery link for evaluation. If the status is positive, the DOV is programmed for the next mission. If the status is negative, the GCC controllers program the DOV to to rendezvous at the resupply point for replenishment and/or repair. Also, in case of a telemetery/command link malfunction the DOV is programmed to return immediately to the mother ship.

3.3.3 Replenishment. The DOV will always maintain sufficient fuel status to travel from its present location to the replenishment point. Therefore, when replenishment procedures are initiated, the DOV transfers orbit to rendezvous with the "mother ship". The mother ship may be the Space Shuttle, an orbiting space station, or even a robotic "supply-ship". This too will require prudent scheduling and programming to ensure the mother ship will be available when the DOV is expected to require replenishment. However, if the mother ship is unavailable,e the DOV is designed to be put into a parking orbit until it can be retrieved.

During replenishment, the de-orbiting module of the DOV will be replaced as one entire unit. The connections between the DOV and de-orbiting module will be released by radio command. A Remote Manipulating System can then be used to replace it with a fully stocked module pre-loaded with a full complement of projectiles. Decisions on repairs to be undertaken are also made at this stage. Actually any sub-system that is unserviceable will just be replaced. The DOV then undergoes another system check and, if fit, is programmed for the next mission. However, if the DOV is still beyond repair, it will be deactivated. This can be accomplished by either returning it to earth for extensive refit inside the Space Shuttle payload bay or it may de-orbit itself (see Specification 5.2 in Appendix A).

Evaluation of the Design Solution

In addition to the requirements of this project, the design team determined five "success criteria" to select the successful design solution (see Section 1.2). These criteria concern the reusability, energy efficiency, de-orbiting effectiveness, simplicity, and economy of mass and size. This section discusses the design team's evaluation of the final configuration selected for the DOV based on these five criteria.

4.1 Reusability

The DOV meets the design specification for reusability in terms of multi-mission and multi-operation capabilities (see Appendix A, Specification 8.2). The de-orbiting module is designed to be replaced with a fully-loaded de-orbiting module in orbit every time its consumables are depleted. Also, the OMV, which acts as the command and propulsion module for the DOV, is designed for a service life of twelve years with in-orbit replacement of its consumables [2]. The service life can be extended by replacing malfunctioning subsystems of the OMV/Command and Propulsion Module using the LRU concept. In addition, the OMV system is designed to enable mid-life system upgrades to improve or extend its capabilities even further. Therefore, these design features confer the maximum possible reusability on the DOV as required by the design specifications.

Also, the DOV is designed to complete several de-orbiting operations during one mission. Prudent mission planning to target debris masses with planes and inclinations close to each other allow the DOV to efficiently utilize all the consumables with minimum possible fuel expenditures. The process efficiency is important because the debris de-orbiting program has no commercial benefits, only the reduction of damage possibilities to expensive satellites and space stations due to orbital collisions. Appendix H contains a discussion of a typical de-orbiting mission.

An additional advantage obtained from the modular design concept is the potential for multiple missions and application of the DOV. The DOV is essentially a special applications de-orbiting module attached to a general purpose OMV. In this configuration (as mentioned in Section 3.2) the OMV operates as the Command and Propulsion modules of the DOV. The vehicle can be reconfigured for different missions by replacing the de-orbiting module with a grappling or docking unit the OMV/DOV can be converted into an automated "space-tug" [2]. Therefore, the command and propulsion modules of the DOV can be used as the command and propulsion modules of any other system.

4.2 Energy Efficiency

A major weakness of the DOV design selected is the energy expenditure required for de-orbiting operations. The energy expenditure is in three different categories: the main engine required for OMV rendezvous with the target debris, the electrical energy required for the rail-gun operation, and the electrical energy required to operate the DOV systems.

The DOV is designed to travel to and from the Space Shuttle or the Space Station to its target debris area to conduct de-orbiting operations. Typically, this will involve altitude changes from 300 to 800 kilometers and plane changes from 20 to 90 degrees inclinations. These maneuvers will require large amounts of specific impulse and consequently, fuel consumption. The DOV therefore has to carry 60 to 70 percent of its mass as fuel, which in turn requires more fuel to transport the mass to rendezvous.

The electro-magnetic rail-gun requires very high voltages over an extremely short period of time. Although the high voltages can be supplied by capacitor discharge, the capacitor requires electrical energy to be charged. The DOV utilizes fuel cells to supply the electrical energy. These fuel cells tend to be heavier than RTGs or solar arrays and require replenishment of their reactants. This requirement puts a limit on the DOV's operational time-span as well as an extra mass penalty.

The sensor, command, control, and data-relay systems in the DOV require electrical power, putting an additional strain on the power supply. However, the DOV may spend up to 70 percent of its time in transit between de-orbiting and replenishment procedures. Non-essential electrical systems like the radar and TV/IR systems can be turned off during this period to minimize electrical energy expenditure.

4.3 De-orbiting Effectiveness

The effectiveness of the CABS projectile cannot be determined until realistic tests can be conducted. The stability of the CABS projectile is one aspect of the concept that requires zero-gravity testing for validation of the system. Also, there is insufficient data to determine the probability of secondary debris propagation during de-orbiting operations. A testing procedure to obtain the data is recommended in Section 5.1. However, the CABS de-orbit concept can successfully de-orbit orbital debris in terms of the change in velocity required. The calculations are shown in Appendix I.

Appendix I discusses the debris the de-orbiting concept in terms of momentum and velocities required to de-orbit debris. The concept does de-orbit our target range of debris successfully.

4.4 Simplicity

Simplicity of the DOV design selected can only be determined in relative terms. Although the DOV configuration is complex in absolute terms because of the complexity and number of critical subsystems, the entire DOV is much simpler than its alternatives. For example, the radar, TV/IR system, controller, command/data link systems, and other critical DOV components are technologically complex. However, the integrated system including these subsystems is considerably simpler than the Static Deployment and Tether method (see Section 2). Another alternative suggested for a de-orbiting method is an orbital vehicle like the Space Shuttle that actively transports the debris masses out of earth orbit. In comparison to this alternative, the DOV is a simple method of de-orbiting debris masses in earth orbit.

4.5 Economy of Mass and Size

Economy of mass and size are considered very important because of their direct correlation to launch costs. Launch costs are directly related to the mass of the vehicle. Although the sub-systems of the DOV are designed to be small, efficient, and light-weight, the main thruster propellants, fuel-cell reactants, and CABS projectiles have relatively greater mass and require transportation into orbit. Therefore the mass minimization criteria has to be sacrificed for operational requirements.

The DOV was originally required to have a diameter of 3.7 meters or less to enable orbital insertion by unmanned launch systems like the Atlas or Ariane-4 systems. However, the OMV, which acts as the DOV's command and propulsion systems, was designed for orbital insertion by the Space Shuttle. It has a diameter of five meters and is too large for any western unmanned launch system. Only the Soviet Proton and Energia unmanned launch systems are capable of inserting such large diameter payloads into orbit [8]. Once the DOV has been inserted into earth orbit by the Space Shuttle, its consumables can be lifted into orbit by unmanned boosters. For example, the propellant and fuel-cell reactants can be transported separately by unmanned boosters into earth orbit and then transferred to the DOV during the replenishment procedures. Also, the de-orbiting module itself is designed to fit into the Atlas and Ariane-4 boosters and can be inserted into orbit by itself. It can then be attached to the DOV during replenishment procedures.

4.6 Conclusions

The design team concludes that the DOV configuration using the CABS method of de-orbiting debris masses in earth orbit satisfies most of the five success criteria.

Using the OMV as its command and propulsion modules, the DOV is essentially a special applications module for the OMV. This maximizes the potential for reusability of the system.

The major weakness of this system is the amount of propellant required by the DOV for its operations. However, this is more a result of the inherent in-efficiency of present space propulsion systems than of the design of the DOV. Also, fuel cells are used to supply electrical energy to the de-orbiting module to maintain commonality with the OMV systems.

Although the preliminary calculations indicate the CABS concept will be a successful de-orbiting procedure in terms of momentum and KE, the actual effect of the impact inherent in the procedure is still unknown. Also, it is not yet known if the CABS projectile will be deployed in the manner theoretically expected of it.

The team believes that, although there is not a quantitative method of determining it, the DOV concept selected is relatively simple. This is evident from a general comparison between the concept selected and some of the alternatives to it.

Finally, although the team had to compromise on its original specification regarding the size of the DOV, the DOV does satisfy the success criteria for mass and size. The original requirement for the DOV was for orbital insertion by an unmanned booster. However, this is no longer feasible as the OMV, which is the command and propulsion module for the DOV, is designed to be launched by the Space Shuttle. Also, the mass has a definite lower limit dictated by the mass of the OMV and the mass of the CABS projectile, this leaves very little room for drastic mass minimization.

Recommendations for Further Work

The design team has an number of recommendations for the DOV design as well as for the control of orbital debris. The team recommends further development work on the DOV configuration as well as realistic simulations of the CABS concept. These recommendations are followed by a general discussion of the overall strategy required to control the debris population in earth orbit.

5.1 Development Work on the DOV

1. The exact dynamic behavior of the CABS projectile during the firing sequence is still unknown. Specifically, it is not yet known whether the CABS perimeter rods will deploy in a radial pattern or randomly oscillate around the core rod. The exact orientation of the rods during impact may affect the way the momentum is transferred to the target debris mass.

The team suggests zero-gravity simulation of the CABS firing procedure using a scaled model of the projectile. This testing procedure will determine the behavior of the projectile as it leaves the rail-gun barrel.

2. The KE de-orbiting procedure involves the risk of further debris propagation due to impact. Debris masses, especially inactive satellites, may have appendages like antennae, dish reflectors, or solar panels. Because of the high velocities involves, impact of the CABS projectile on these debris masses may cause these appendages to break off. Also, there is a risk of explosive impact because of the high KE created by the high velocities as well as the presence of volatile materials in some debris masses. However, the exact risk factors involved are still unknown.

3. The team suggests extensive impact simulation of the firing procedure using

representative target debris masses of different sizes, shapes, and masses. The simulation results can be used as a database for the calculation of the debris propagation risk factors involved in a particular debris de-orbiting operation. The DOV operator can use these risk factors to make the final "go/no go" decision for a procedure.

4. The electro-magnetic rail-gun concept has already been proven in "bench-tests" [7]. This gun was also planned for utilization in the United States Department of Defense Strategic Defense Initiative (SDI) program, albeit in a much larger scale. Therefore, although scaled analogies can be used to prove the efficacy of the rail-gun in the de-orbiting concept, a unit has yet to be designed for the DOV. Also, the gun in the CABS concept uses a tube to accelerate the projectile and impart the spin to it. The design will have to address the questions of power consumption, installation, dynamic reactions during the firing sequence, space requirements, and control systems.

The design team recommends further work on electro-magnetic rail-guns to meet the requirements for utilization in the DOV.

5. In accordance with the project proposal (see Appendix A, Specification 3.1), the team did not investigate either the structural design or the materials to be used on the DOV. The structural design of the DOV will require a more detailed analysis of the dynamic loading on each component of the DOV over the whole performance envelope. Also, although aluminium alloys are used extensively in aerospace structures, the dynamic analysis is needed to determine the exact alloys to be used. Therefore, the team suggests that a detailed dynamic analysis of the DOV over the entire performance envelope. The structure of the DOV can then be designed and the material selected.

5.2 Controlling the Overall Debris Problem

The active removal of debris masses from the earth orbit will not solve the debris problem. Active removal, using the DOV can only remove the debris that are already there but continuing space activity will keep on adding more debris even as the removal process is underway. Furthermore, the "Cascade Effect" [14] of debris propagation almost guarantees a geometric growth rate of the orbital debris problem Therefore, the team has the following recommendations for the control of debris in earth orbit:

1. At present, there are no laws or conventions governing the control the debris population. As a matter of fact, there is even no accepted legal definition of space debris [18].

Therefore, the team recommends that conventions regarding the control and removal of space debris should be formulated with the major space-faring nations like the USA, the USSR, UK, France, Japan, India etc. as signatories. However, their mere presence itself is a danger to other satellites in orbit.

2. One of the major sources of space debris are discarded upper stages of rockets. These rockets also have a tendency to explode because of the presence of excess propellants. At present the US Atlas and French Ariane 4 launchers have procedures for venting propellants.

The team recommends the formulation of protocols requiring all present and future launch system operators to install fuel venting procedures on their upper stage rockets and automatic de-orbiting packages on the upper stages of their rockets.

3. Another major source of orbital debris is derelict payloads and incidental debris. Incidental debris is debris masses generated during launch and orbital operations. They include payload fairings, mating collars, and residue from the explosive bolts used in the stage separation procedures used to separate satellite payloads from upper stages.

The team recommends that stage separation procedures be redesigned to minimize incidental debris propagation, this would include connecting the explosive bolts to larger bodies by wires. Also, the derelict payloads and payload fairings should have the provision to de-orbit themselves. In LEO, this provision would involve the attachment of relatively simple and cheap drag increasing devices like balloon (see Section 2.5). For payloads in Geostationary Earth Orbit (GEO), it is suggested that altitude control systems be provided with sufficient fuel to bring them down to a lower orbit and thus de-orbit them.

4. Smaller debris particles, even paint chips as small as one centimeter in diameter can cause appreciable damage to satellites as a result of hyper-velocity impacts. There are two major contributors to this type of debris: paint chips from satellite surfaces, and aluminum -oxide particles from the operation of solid-fueled rocket motors [3].

The team recommends that either more resilient or more easily degradable paint materials be used on satellites to reduce the risk of damage due to impact. Also, use of solid fuel boosters, especially of the type that leave metallic oxide particles as residue, should be curtailed.

References

1. Bainum, Peter M. (ed.), Tethers in Space, Advances in Astronautical Sciences Series, Volume 62, (AIAA), 1983.

2. Brooks, Denise C., “OMV: Space Missions Workhorse”, Design News, November 23, 1987.

3. Conversation with Mr. Richard Connell, Project contact engineer, September 6, 1990.

4. Conversation with Dr. Wallace Fowler, Professor, Department of Aerospace Engineering, University of Texas at Austin, October 2, 1990.

5. Conversation with Mr. William Gerstenmaier, Director, Space Shuttle and Space Station Assembly and Operations, National Aeronautics and Space Administration , Johnson Space Center, Houston, Texas, October 29, 1990.

6. Conversation with Mr. Hoppy Price, Engineer, Jet Propulsion Laboratories, Pasadena, California, November 26, 1990.

7. Conversation with Dr. Grady Rylander, Professor, Department of Mechanical Engineering, University of Texas at Austin, November 2, 1990.

8. Conversation with Dr. Harlan Smith, Director, University of Texas McDonnell Space Observatory, October 7, 1990.

9. Conversation with Dr. Kristin Wood, Professor, Department of Mechanical Engineering, University of Texas at Austin, October 4, 1990.

10. "Design of a Vehicle for De-orbiting Space Debris in Earth Orbits", Richard Connell, The University of Texas at Austin Mechanical Engineering Department, May 7, 1990.

11 "Final Design for a Comprehensive Orbital Debris Management Program”, The University of Texas at Austin Aerospace Engineering Department, May 4, 1990.

12. Hogg, Ian V. and Gunston, Bill, World Encyclopedia of Fighter Aircraft, (London, England, Salamander Press), 1987.

13. Johnson, Nicholas L., "Evolution of the Artificial Earth Satellite Environment”, (Colorado Springs, Colorado: Teledyne Brown Engineering, AIAA), 1987.

14. Kessler, Donald J., Orbital Debris: Proceedings of a Workshop Held in Houston, Texas, AIAA Proceedings, (American Institute of Aeronautics and Astronautics), 1987.

15. King-Hele, D.G., et al, The RAE Table of Earth Satellites, 1957-1982, (New York: John Wiley & Sons, Inc.), 1983.

16 “NASA Faces Second RTG Battle”, Spaceflight, Vol. 32, September 1990.

17. Petro, Andrew J. and Talent, David L., "Removal of Orbital Debris", Orbital Debris from Upper Stage Breakup, (AIAA), 1989.

18. U.S. Congress OTA, "Orbiting Debris: A Space Debris Environmental Problem - (Backgound Paper)", OTA-BP-ISC-72 [Washington, D.C.: U.S. Government Printing Office, September 1990].

Appendices

Appendix A: Specifications

Appendix A consists of the design team's List of Specifications. This lists the project requirements by eight categories. In addition, each requirement is coded either "D" or "W", or "Demand" and "Wish", respectively. A demand is a required element of the design. Wishes are additional elements the team will try to include. This code appears in the second column of the table.

Specification Sheet

NASA/USRA DOV Project		**Specifications for De-orbiting Vehicle (DOV)**	Issued on 09/18/90 Page 1
Changes	D/W*	Requirements	
		1. Geometry	
	W	1. The diameter of the DOV should be limited to allow launch on any sized booster available	
	D	2. Modular construction	
	D	3. Prefer longer length over diameter	
11-03-90	D	4. Mating capability with the Space Shuttle or SS Freedom	
	D	5. Provision of command-link antenna	
		2. Kinematic, forces, and energy	
	D	1. Has to gain energy from de-orbiting operation	
	W	2. System should have minimum possible inertia	
	D	3. Should have three-axis maneuverability	
	D	4. Should have sufficient energy, potential or kinetic, to de-orbit debris, amount determined by the debris mass' altitude and speed	
	D	5. Minimize energy expenditure	
	D	6. Provide radiation and IR shielding	
	D	7. Shielding from micrometeorite impact	
	W	8. Consumables should be easily replenishable	
	W	9. Fuel should outlast other consumables	
		3. Materials	
	D	1. Structural materials (superstructure, skin, etc.) not a driving factor in this phase of the design	
	W	2. Materials which must be specified should be "off-the-shelf"	
		4. Signals	
	D	1. Provision for programmed mission profile for de-orbiting phase and active control for Space Shuttle rendezvous	
	D	2. Should have integral Doppler radar on the DOV for terminal homing	
	D	3. Final "go/no go" authority from ground control	
		5. Safety	
10-27-90	D	1. Power packs should not use a radiation source.	
	D	2. Automatic self-destruct (de-orbit itself) in case DOV goes "rogue"	
	D	3. Liquid fuel for DOV main engine	

<table>
<tr><td colspan="2">NASA/USRA
DOV Project</td><td>Specifications
for De-orbiting Vehicle (DOV)</td><td>Issued on 09/18/90
Page 2</td></tr>
<tr><td>Changes</td><td>D/W</td><td colspan="2">Requirements</td></tr>
<tr><td></td><td>
D

D

W

D

D</td><td colspan="2">6. <u>Ergonomics</u>
1. Replenishables should be in "modular magazine" form to aid refurbishing operations

7. <u>Transport</u>
1. DOV should have integral shroud (during insertion phase) to avoid exacerbating the debris problem
2. DOV should not require manned orbital insertion

8. <u>Operation</u>
1. De-orbiting operation should not break-up target to increase debris population
2. Multi-mission capability with multiple de-orbiting operations per mission capability</td></tr>
</table>

* - D = Demand, W = Wish

Appendix B: Calculations for the Static Deployment Concept

Two factors control the application of tethers in space: the strength and the mass of the tether used.

Strength of Tethers Because of the length of the tethers to be used in space, it is important to minimize the total mass of the tether by using materials such as aramid fibers (Kevlar 69) which have high strength to weight ratio. The tether cannot be made too thin because of the risk of being cut by a micro-meteoritic impact. A long tether can have a large total surface area so that the risk of a micrometeorite impact can be significant. The tethers planned for use on the TSS project have a diameter of about two millimeters and are made of a braided construction which should be able to survive most impact so as to have a reasonable expected lifetime.

If the diameter of a tether is kept constant, there is a maximum attainable length in near earth orbit beyond which the tether will break under its own weight because of the gravity gradient forces of the system. This stress occurs at the orbital center of the system. The gravity gradient forces are directed away form the orbital center both in the upward and downward directions. At any point along the tether, the tether must be strong enough to support all the parts of the system in the direction away from the orbital center. Tethers of indefinite length can be put in orbit by tapering the tether so as to maintain a constant stress per unit cross sectional area. The difference in tension between the ends of a segment of tether is equal to the net force on the segment due to gravitational and centrifugal forces [1].

The tapering requirement for long tethers places an operational penalty on the use of the tethered concept. Each tether of various extremely long lengths and small diameters (ranging from two to six millimeters) would have to be specifically tapered for the particular mission it is used in.

Mass Calculations

Assume debris mass = m (kg) = 2000 kg

Assume acceleration due to gravity = $g(m/sec^2) = 9.8\ m/sec^2$

Tensile Strength = TS = $2.7 \times 10^9\ N/m^2$

$$F = mg \tag{B.1}$$

also $TS = F_{max}/A_{max}$ (B.2)

therefore $A_{max} = F/TS = mg/TS = 7.26 \times 10^{-6}\ m^2$

Now $A = \pi d^2/4$ (B.3)

therefore $d = \sqrt{\frac{4 \times (7.26 \times 10^{-6})}{\pi}}\ m = 3.04\ mm$

using a factor of safety of 2

$d = 6.08 \times 10^{-1}$ cm

therefore $A_{max} = 0.2903\ cm^2$

Now the density of kevlar, $\rho = 1440\ kg/m^3$

and weight/length = $A_{max} \times \rho$ (B.4)

and w = weight/length x l , where l = length of tether

This calculation was used to generate the plot of Figure B1

Graph of Mass of Kevlar 69 Required

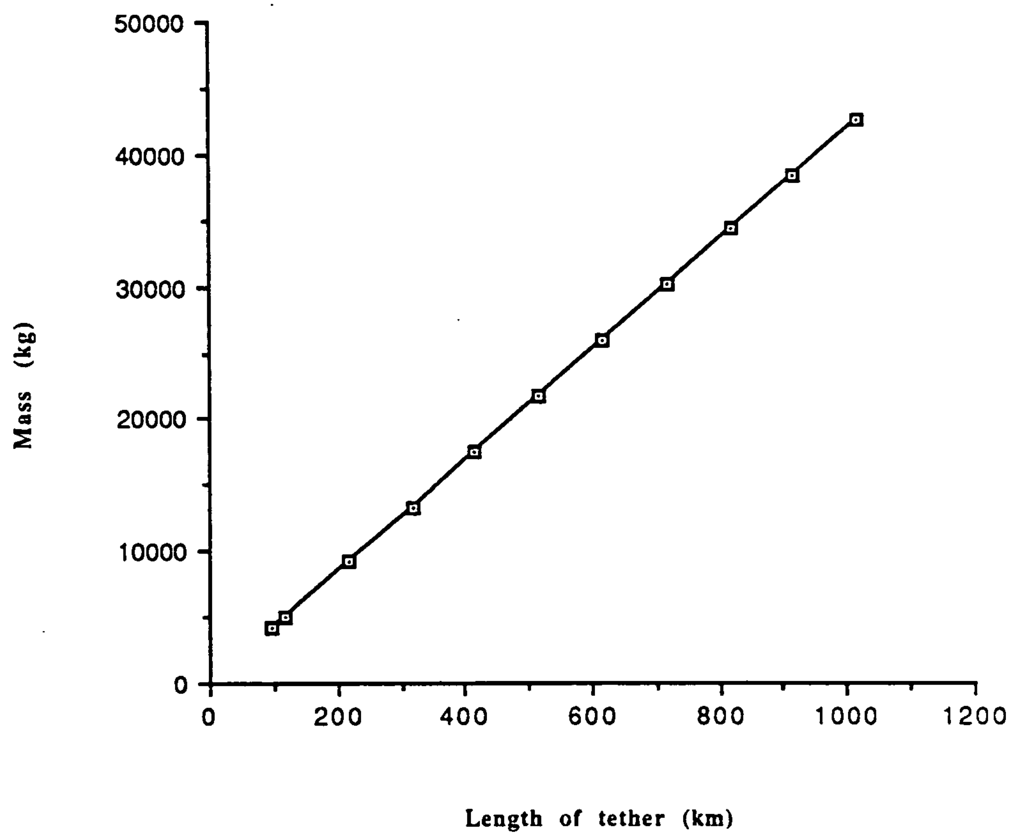


Figure B1. TETHER MASS AND LENGTH RELATIONSHIP

Appendix C: Orbital Mechanics - The Hohmann Transfer [6]

The concept of the Hohmann Transfer enables us to compute the velocity changes necessary to change a circular orbit to another circular orbit of a different radius. This utilizes a change in velocity at one point to move the satellite to a transfer elliptical orbit. An additional velocity change, executed at a point half-way around the ellipse, places the satellite in a new circular orbit.

First, terminology used to describe elliptical orbits must be understood. A satellite in an elliptical orbit centers the ellipse around two foci. The planet being orbited is located at one focus. The point on the ellipse furthest from this focus is called apogee. The perigee is the point closest to the planet (see Figure C1).

If the satellite being maneuvered is to be moved from a larger orbit to a smaller orbit, it follows the pattern pictured in Figure C1. The satellite travels in the larger circle at velocity V_{c1}. This velocity is found by

$$v = (\mu/r)^{0.5} \tag{C.1}$$

where

$$\mu = G(m_1+m_2) \tag{C.2}$$

The gravitational constant is noted by G and m_1 and m_2 represent the masses of the earth and satellite. The radius r is the sum of the earth's radius and the distance between the satellite and the earth's surface.

The velocity is decreased at a point which becomes apogee of the transfer elliptical orbit. The change in velocity is

$$\Delta v = \left(\frac{\mu}{r_1}\right)^{0.5} - \left\{\mu\left[\left(\frac{2}{r_1}\right) - \left(\frac{1}{a}\right)\right]\right\}^{0.5} \tag{C.3}$$

where

$$a = \frac{(r_1 + r_2)}{2} \tag{C.4}$$

and the new velocity is denoted by $(V_c)_a$. An additional velocity decrease at perigee of the

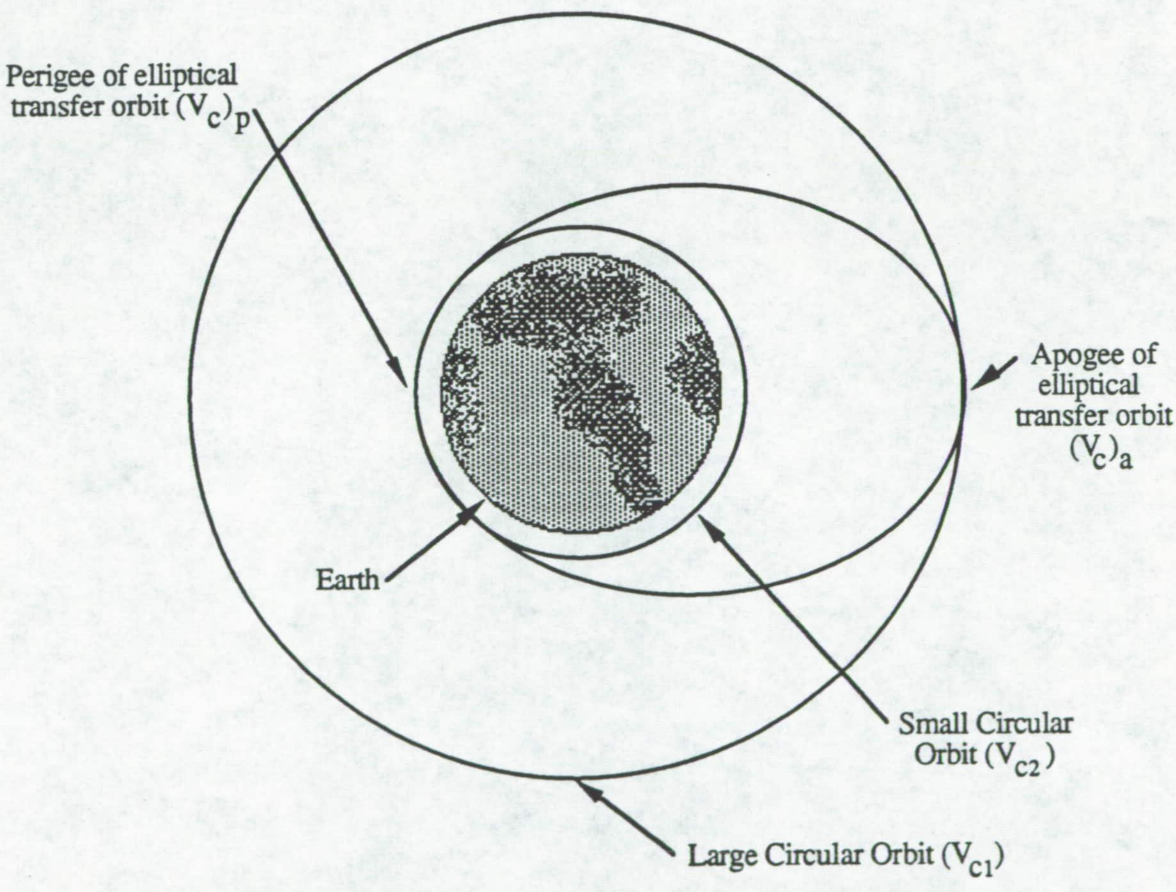


Figure C1. THE HOHMANN TRANSFER ORBIT

new circular orbit (of radius r_2) is defined by

$$\Delta v = - \left\{\mu\left[\left(\frac{2}{r_2}\right) - \left(\frac{1}{a}\right)\right]\right\}^{0.5} - \left(\frac{\mu}{r_2}\right)^{0.5} \tag{C.5}$$

changing the velocity from $(V_c)_p$ to V_{c2}. The satellite now resides in a new, smaller circular orbit.

Appendix D: Decision Matrix

The table on the following page shows the numerical breakdown of the alternatives comparison. The scores for each alternative based on each design criteria (on a 1 to 10 scale) appear on the left hand side of each column. These are multiplied by the weighting factors to obtain the numbers to the right of each column. The values are summed to show the score for each alternative.

The Static Deployment and Tether alternative was judged infeasible by our study, so it was not considered in the Decision Matrix.

Table D.1: DECISION MATRIX FOR ALTERNATIVE DESIGNS

Decision Criteria	Weighting Factor	Alternative Design									
		#2		#3		#4a		#4b		#5	
Fuel/Power Efficiency	0.20	3.0	0.60	4.0	0.80	7.5	1.50	7.5	1.50	8.0	1.60
Life of Consumables	0.15	5.0	0.75	3.0	0.45	9.0	1.35	7.0	1.05	8.0	1.20
Mass Minimization	0.20	7.5	1.50	3.0	0.60	6.0	1.20	5.0	1.00	5.0	1.00
Simplicity	0.175	4.0	0.70	3.0	0.53	6.0	1.05	6.0	1.05	5.0	0.88
Number of Opertations Required	0.1375	5.0	0.69	3.0	0.41	8.0	1.10	8.0	1.10	5.0	0.69
Secondary Debris Propagation	0.1375	8.0	1.10	8.0	1.10	1.0	0.14	8.0	1.10	2.0	0.28
TOTALS	**1.00**		**5.34**		**3.89**		**6.34**		**6.80**		**5.65**

Alternative Design #2: Capture and Retro-fire

Alternative Design #3: Orbital Maneuver and Tether

Alternative Design #4a: Kinetic Energy Projectile - Flechette

Alternative Design #4b: Kinetic Energy Projectile - Chain-and-Bar Shot

Alternative Design #5: Kinetic Energy Projectile with Balloon Attachment

Appendix E: The F-16 Radar [5]

The primary target-detection sensor of the F-16 is the Westinghouse APG-66 radar. In order to carry out a radar-controlled interception, an aircraft requires data on the bearing of the target and its range. Bearing can be measured by means of a highly-directional antenna giving good angular discrimination, but range data can most easily be obtained by pulsing the radar transmitter on and off again at a rate known as the pulse-repetition frequency (PRF). In the simpler types of radar equipment, sufficient time is allowed for one pulse to travel out to the target, be reflected, and return to the radar before the next pulse is transmitted. Engineers describe such radars as low-PRF sets.

By the 1960s a new source of microwave power known as a travelling wave tube (TWT), along with the use of digital signal processors, allowed the creation of pulse-Doppler radars with a good look-down performance. The use of stable and coherent (phase-related) pulses from a TWT allows the radar to measure the Doppler shift in the radar returns from the target -- the tiny change in frequency caused by target motion relative to the signal source. Using this technique, the relative velocity of the target against the terrain background allowed the wanted target signal to be extracted from the massive background returns. This technique is known as pulse-Doppler radar.

TWT transmitters cannot match the high levels of power available from the magnetron transmitters used in low-PRF radars, so the designers were forced to use high PRFs in order to illuminate the target with sufficient power. Since each pulse would be transmitted before the previous pulse has completed the round trip out to a distant target and back, each pulse had to be electronically 'labelled' by a low-frequency modulation at the time of transmission.

The range data obtained by processing the labelled pulses is of low accuracy, and high PRFs are also poor at detecting targets whose closure rate is low. In the 1970s, therefore, designers of airborne radar turned to medium PRFs. These allow traditional methods of ranging to be used at most combat ranges, while still allowing pulse-Doppler techniques to be used for look-down operation.

Since the PRFs best suited to range measurement are different to those effective against low closing-rate targets, a practical design of medium-PRF set has to switch rapidly from one PRF value to another. This made the design of hardware able to carry out pulse-Doppler signal processing virtually impossible. the solution lay in the use of software-controlled digital signal processing. By making the characteristics of the filter dependent on a computer program (software) rather than physical components (hardware), the designers could contrive near-instantaneous re-configuration of the filter to match each PRF waveform used by the radar. The first radars to use medium PRFs and digital signal processors were the Hughes APG-65 in the F-15 Eagle and the L.M. Ericsson PS-46/A in the Viggen JA37 intercepter.

The APG-66 used in the F-16 A/B is a medium-PRF radar (typically 10 to 15 kilo-Hertz). It operates in the I/J band and incorporates a 'flat-plate' planar array antenna (see Figure E1). Sixteen operating frequencies are available within the band, and the pilot may select between any four. Total weight is 296 pounds (134 kilograms), and the set occupies a volume of 3.6 cubic feet (0.1 cubic meter). A mean time between failures of 97 hours has been demonstrated.

Continuously-computed release point (CCRP) attacks use the set's ground mapping modes. Real-beam ground mapping gives a plan position indicator (PPI) display at 10, 20, 40, or 80 kilometer range, and scan widths of plus or minus 10, 30, or 60 degrees. Several auxiliary methods of presenting imagery may be used in these PPI modes. If Freeze mode is selected, the radar carries out a final scan, the image of which is 'held' on the display, following which the radar transmitter is turned off so that the aircraft cannot be detected by passive means. A moving symbol on the display continues to indicate aircraft motion.

Highest definition of ground features is given by a special Doppler beam sharpened mode. Usable when the set is ground mapping at ranges of 10 of 20 kilometers, this

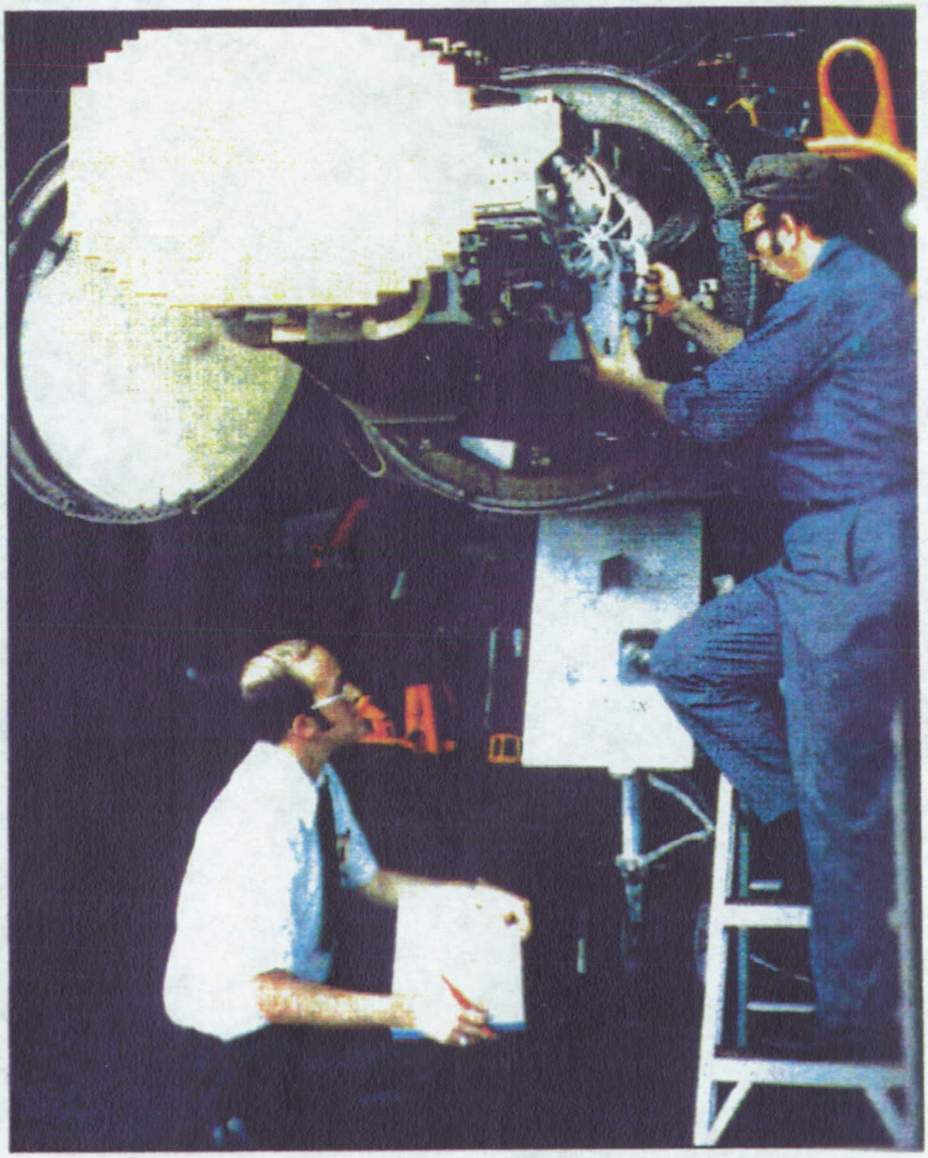


Figure E1. PHOTOGRAPH OF F-16 RADAR
Figure taken from World Encyclopedia of Fighter Aircraft, Hogg, Ian V. and Gunston, Bill, (London, England, Salamander Press), 1987.

provides a further magnification factor of eight over that in expanded-beam real map mode.

Development of an effective pulse-Doppler radar of such small size was a formidable technical undertaking, so it was hardly surprising that several problems were experienced during early tests, particularly in look-down mode. Pulse-Doppler radars measure the Doppler shift created by target velocity in order to discriminate between genuine targets and ground clutter. This involves defining a threshold velocity - a speed at which targets must be moving in order to be accepted as valid. Vehicles on West German autobahns often move at speeds of 100 miles per hour (160 kilometers per hour) or more, and were sometimes registered as low-level targets.

Earlier radars presented a direct radar picture to the operator, who could to some degree use his own skill and experience in deciding which targets were real. Sets such as the APG-66 reduce all radar data to digital form, and present the pilot with a synthetically generated image made up of predefined symbols.

Appendix F: LANTIRN System [5]

In the late 1980s, the APG-66 will be backed up by the Martin Marietta LANTIRN (Low-Altitude Navigation and Targeting Infra-Red for Night) system. This equipment will allow the pilot of a single-seat aircraft to fly sorties by day or night and in adverse weather. It can provide terrain-following radar and FLIR (forward-looking infra-red) imagery for navigation; automatically acquire, identify and categorize tank targets, passing target information to the aircraft's fire-control system so that Maverick missiles may be launched against several targets in a single pass; and can acquire and track fixed ground targets using FLIR or visual techniques and then designate them for attack using a build-in laser.

The basic installation comprises two avionics pods containing the sensors for navigation and target acquisition tracking respectively (see Figure F1). Martin Marietta is prime contractor for both. On the F-16 the pods will be carried on hard points under the inlet. They can operate autonomously, so an aircraft could fly into action with only one should this meet the requirements of the mission. Although the program was formally launched in 1980, it was suspended just over a year later, and reshaped to reduce the technical risks involved.

The navigation pod is 12 inches (30.5 centimeters) in diameter, 78 inches (198 centimeters) long and weighs about 430 lb (195 kilograms). Main subsystems are a Ku-band terrain-following radar, wide field-of-view FLIR, pod computer and the associated power supply. Sophisticated signal processing is used to give the radar a wide azimuth coverage, allowing high-rate turns at low level in order to avoid or confuse the defences. This should give greater survivability than earlier-generation equipments which simply issued pitch commands to the pilot. The latter may have allowed him to avoid the terrain ahead but exposed the aircraft to ground fire during this 'pull-up' maneuver.

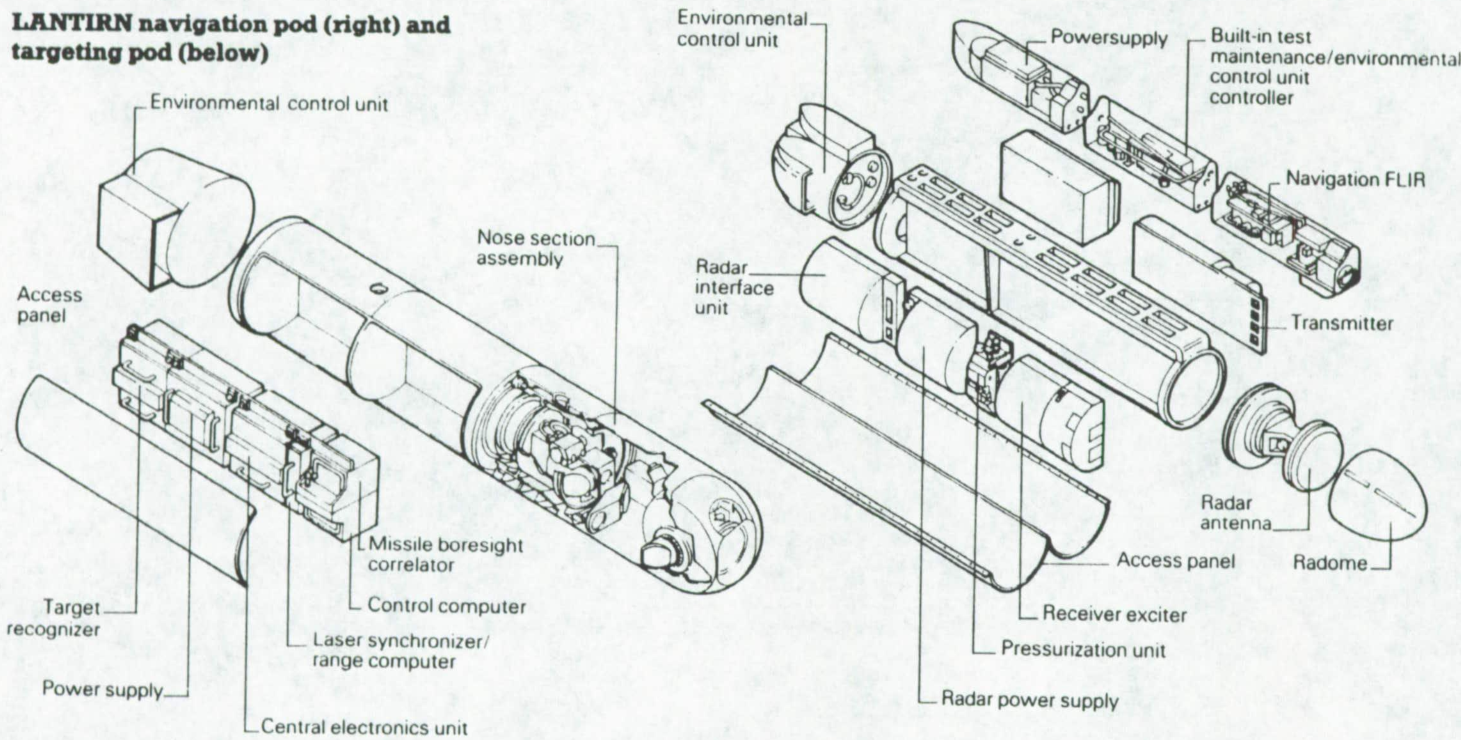


Figure F1. BLOWN-UP VIEW OF LANTIRN PODS
Figure taken from <u>World Encyclopedia of Fighter Aircraft</u>, Hogg, Ian V, and Gunston, Bill, (London, England, Salamander Press), 1987.

FLIR field-of-view is 28 degrees in azimuth and 21 degrees in elevation. The resulting wide-angle imagery ma be superimposed on the outside scene by means of the HUD. In darkness or bad weather the HUD provides an image of TV-like quality and sufficient width to allow the pilot to look in the direction of his turn in order to 'preview' the terrain. A number of sample images are shown in Figure F2. These images, taken at night with no ambient light show the kind of detail available with the LANTIRN system.

The larger targeting pod has a movable nose section containing a FLIR sensor, laser transmitter/receiver and stabilization system able to compensate for aircraft movements and vibration. A fixed center section houses the tracker electronics and signal-processing systems and the boresight correlator used to pass target data to the aircraft's air-to-ground weapons. Environmental control of these systems and the nose-section sensors is handled by equipment in the aft section of the pod.

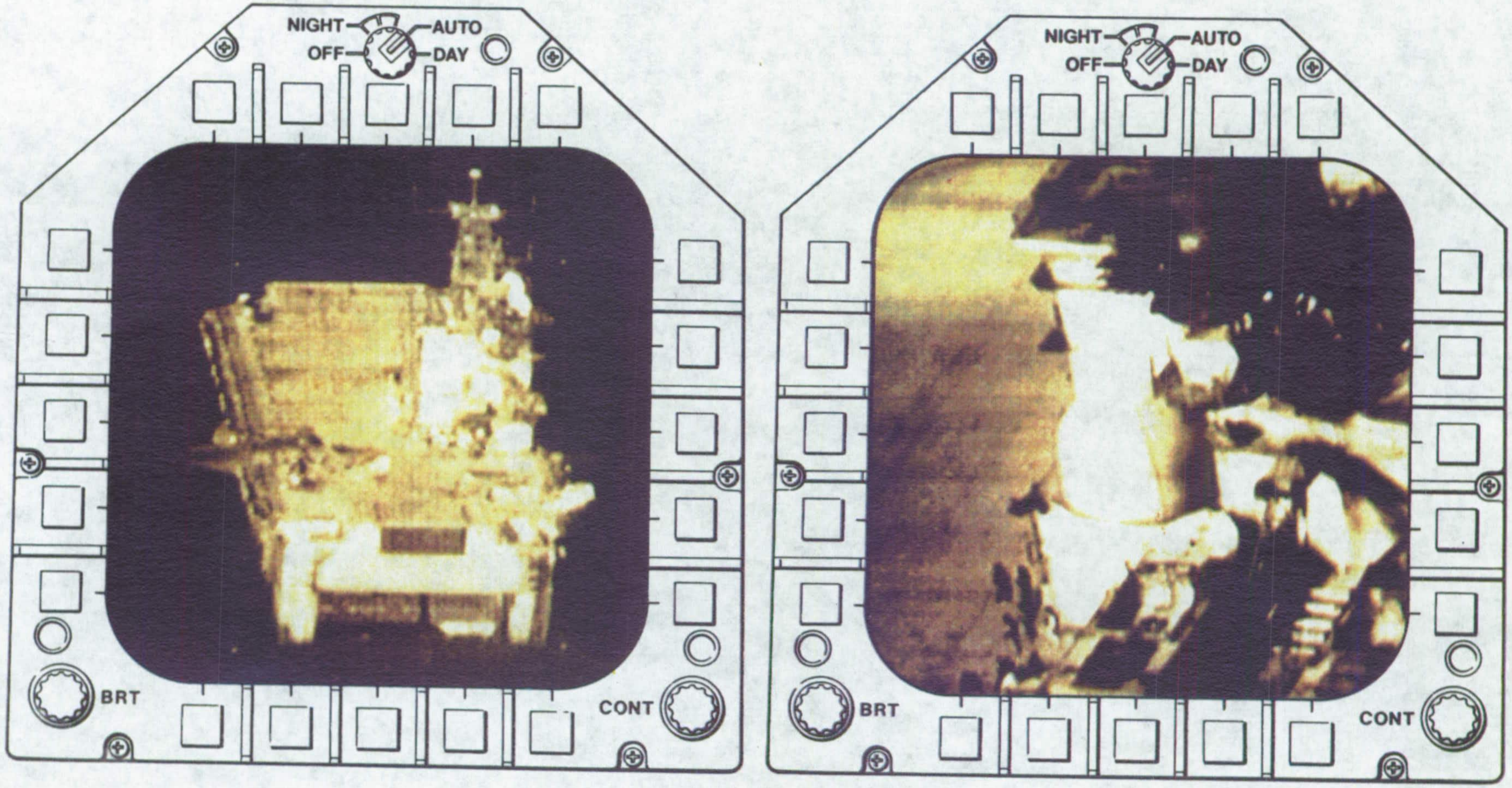


Figure F2. SAMPLE LANTIRN IMAGES
Figure taken from <u>World Encyclopedia of Fighter Aircraft</u>, Hogg, Ian V, and Gunston, Bill, (London, England, Salamander Press), 1987.

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Appendix G: Operation Procedures

Appendix G discusses the operating procedures of the de-orbiting process via two flowcharts: Figure G1 is a flowchart representing the de-orbiting operating procedure and the Targeting Parameters are shown in flowchart form in Figure G2.

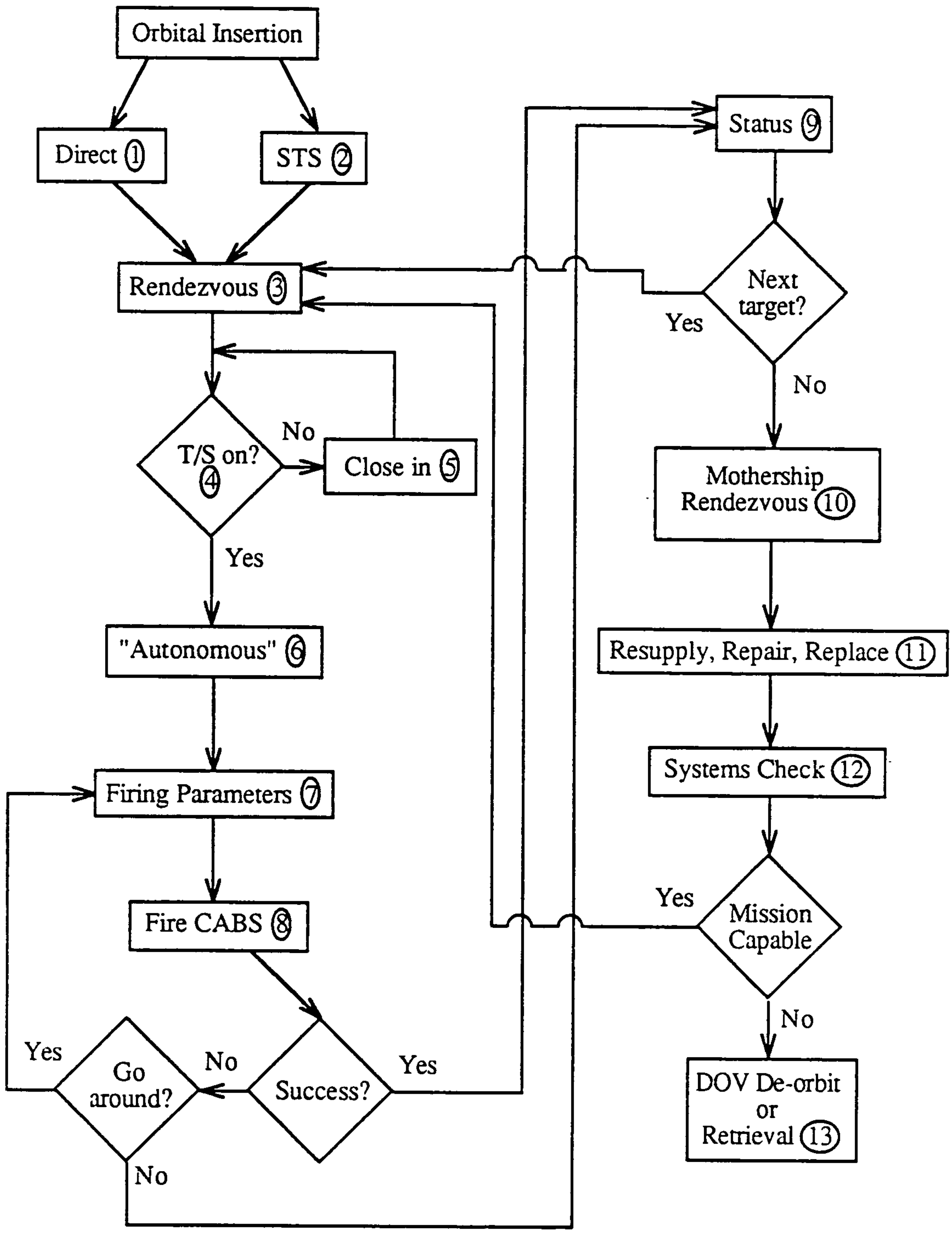


Figure G1: OPERATING PROCEDURE FLOWCHART (See Notes 1 - 13)

Notes :

1. Expendable boosters are used for direct insertion into the target debris mass' orbital plane. This method can be used for the first deployment only. It is a relatively simpler method than using the STS for insertion. Also, because of the lack of a human crew (with their accompanying life-support systems and safety restrictions), this method is cheaper than the STS insertion.

2. The Space Shuttle (STS) can only take the DOV to LEO and an inclinations of 28.5 degrees [2]. Therefore, a further orbital transfer, with the requisite plane changes to the higher inclinations of the target debris masses, is required to boost the DOV up to the operational area. However, this boost procedure would be required anyway after every replenishment procedure as the Space Shuttle orbits are the ones where most manned missions are.

3. The rendezvous procedure is conducted on inertial guidance. A mission profile is programmed into the DOV's control system giving it the orbital plane, velocity, and altitude of the target debris. This mission profile is programmed via the command link from earth or the Space Station and is updated after every de-orbiting procedure.

4. The inertial guidance system will keep track of the computed range to the target debris. When this range is within the track/scan (T/S) unit, the T/S unit is turned on.

5. If the target is still not within range, the DOV continues to close in on the target using the computed target data.

6. When the T/S unit is turned on, the image is down-linked to a human operator through the command-link. The operator will first use the radar data to make any necessary course or altitude changes for the DOV. At closer ranges, the TV/IR system of the T/S unit can be used by the operator to actually "see" the target.

7. This stage is where all the preliminary targeting calculations are conducted. To save the power and space that would be required for onboard target computing by the DOV, the DOV's T/S unit is used as a "sensor" at this stage by the human operator. Therefore, the data from this "sensor" can then be used to calculate he actual firing parameters. This procedure will be as follows (notes from Figure G2):

i) The target data (i.e. the velocity, orbital altitude, attitude or motion, inclination, perturbations, and orbital decay) are obtained by the T/S unit and relayed through the command-link to the human operator.

ii) When the target is within visual range (using the TV/IR system) the operator can visually determine whether the target is stable, i.e. if it is tumbling or not.

iii) If the target is tumbling, then the target has to be studied visually (and perhaps using computer modelling) to determine the most efficient method of shooting at the target.

iv) If the target is stable then the mass of the target can be estimated by observing the target's orbital decay rate and the atmospheric density during that period [5]. Actually, this step can be completed by ground-based tracking systems.

v) The mass, orbital altitude, and velocity of the target are used to calculate the velocity change required to reduce the target orbital altitude to 80 kilometers.

vi) The velocity change required is used to calculate the velocity required for the CABS projectile. This velocity is calculated by using momentum transfer equations. The rail-gun can then be programmed to impart that velocity to the CABS projectile during the firing procedure.

vii) The rail-gun requires a high amount of power over a short period of time. Therefore, it is hooked up to a capacitor which is discharged during the firing procedure and is recharged over a longer period of time. Before the firing procedure is initiated, the charge status of the capacitor is checked. If is is insufficient, then the firing procedure has to be delayed.

viii) The final command to initiate the firing sequence has to be given by the human operator. This ensures safety and success as well reduces the computer space required for automatic safety checks. At this stage, the operator has the option of either initiating the firing procedure or aborting it.

ix) In case of an abort, the next step in the procedure is the status check of the system (see note 9).

8. The CABS projectile is fired at the target and the effectiveness is observed. At this stage the operator can use both DOV as well as ground-based sensor systems.

9. The status of the system has to be checked before the DOV can go on to another de-orbiting procedure. At this stage the supply of the CABS projectiles and the fuel and power supply status are checked. Also, Built-In-Tests (BIT) of the DOV systems and programs are conducted to detect any possible malfunctions. If all the systems and status are "green", the DOV is programmed for rendezvous with the next target.

10. If the system status is "down" then the DOV is "turned" around for rendezvous with the "mother ship". This "mother ship" may be the Space Shuttle, the Space Station, or even a robotic resupply vehicle [4].

11. When the DOV rendezvous with the "mother ship" the fuel, fuel-cell reactants, and

CABS projectiles are replenished. The DOV may suffer micro-meteorite damage (as do all satellites in orbit). Therefore, minor repairs of the DOV skin may be conducted if required. Also, the subsystems (e.g. controllers, motors)may be replaced if they are unserviceable. This step could be facilitated by designing the subsystems as Line Replaceable Units (LRUs). This design procedure is used for military aircraft whereby any subsystem can be easily removed and replaced [6].

12. A systems check is then conducted to determine if the DOV is "mission-capable", or ready for another mission.

13. If the DOV is "mission-incapable" then it may be either de-orbited itself (if it is unsalvageable), or retrieved by the Space Shuttle to be repaired on earth.

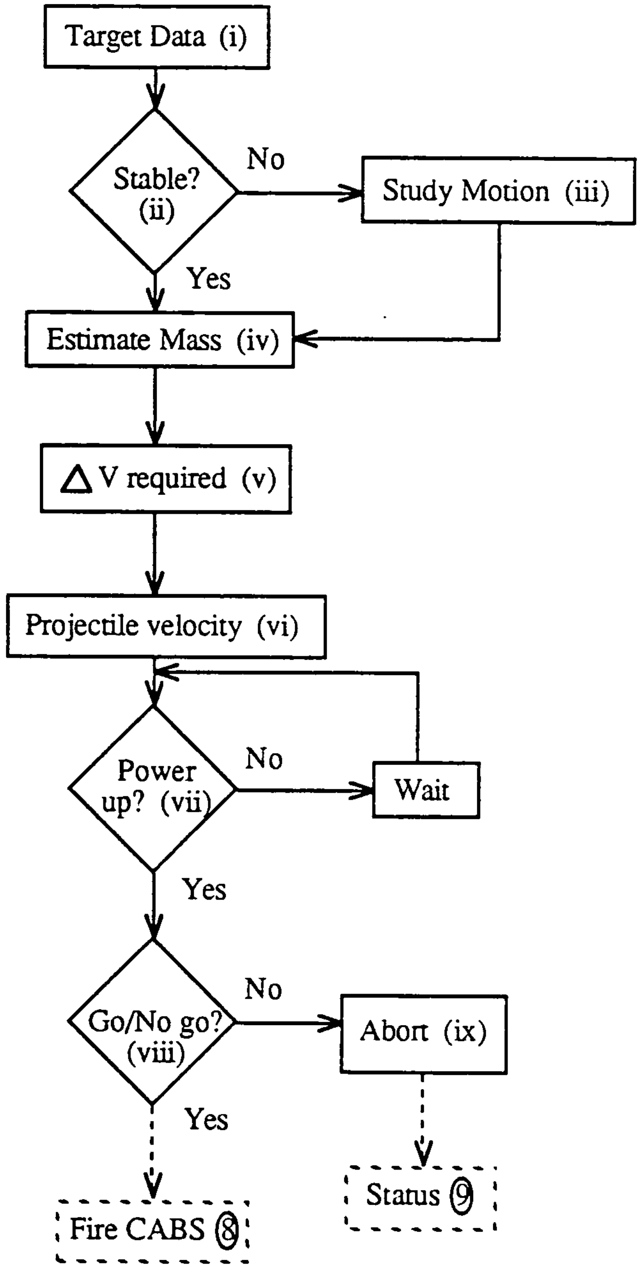


Figure G2: TARGETING PARAMETERS FLOWCHART (See Notes i - ix)

Appendix H - Sample Mission Profile and Mission Planning

Mission Planning

These calculations were completed to estimate propellant consumption for one type of mission profile. This profile included targets requiring large plane changes for the DOV to maneuver through. The targets illustrates a true worst-case scenario, because plane changes require large fuel expenditures in addition to the required altitude changes [2]. This profile was developed to de-orbit four debris masses at planes of 74 degrees inclination or higher.

Due to a lack of specific data on the OMV at publication time, these results represent only a rough estimation. Estimates were required for the DOV mass and the specific impulse of the propulsion module. In addition, assumptions were made for the fuel cell consumption for each de-orbiting operation.

The design team estimated the DOV mass to be 4000 kilograms, based on rough estimations and data from the rocket motor catalogs, a specific impulse of 450 seconds was assumed for each of the four engines, totalling a specific impulse, I_{sp}, of 1800 seconds. It was assumed that twice the propellant would be used for each possible projectile (6 x 100 kilograms x 2) totalling 1200 kilograms. An additional 1000 kilograms of fuel propellant was added for electrical usage, totaling 2200 kilograms at the start. Finally, 300 kilograms of propellant was assumed to be used for each firing.

Based on the previous assumptions, the 4000 kilogram DOV will have 400 kilograms reactant remaining left in the fuel cells after four de-orbiting operations. The amount of unusable propellant, or the fuel which must always remain, was assumed to be needed to change from 800 kilometers to 350 kilometers (mother ship rendezvous altitude) both at 28.5 degrees inclination. This was calculated to be 256.8 kilograms, rounded up to

300 kilograms unusable propellant.

Subsequently, an iterative process was completed using orbital transfer equations (see Appendix C) and the rocket equation.

$$\Delta v = c \ln \left(\frac{m_i}{m_f} \right) \qquad \text{(H.1)}$$

Here, c is the specific impulse times the earth's gravitational constant, and m_i and m_f are the DOV's initial and final masses respectively [2]. The results are displayed in Table H1 and discussed in the section entitled "Prudent Mission Planning".

Table H1
Sample Mission Profile

Rendezvous Vehicle [5] (kg)	Mass (km)	Altitude (degrees)	Inclination (kg)	Propellant Expended (kg)	DOV (kg)	Expendables
Mother ship	-----	350	285.00	-----	14649.45	-----
Cosmos 1364 (10000 years)	40	1522	74.02	3963.85	10685.60	400
Cosmos 1354 (100 years)	2200	799	74.05	194.10	10089.50	400
Titan 3d Capsule (70 years)	60	704	95.99	1456.00	8224.50	400
Landsat 4 (80 years)	1938	692	98.26	130.00	7693.80	400
Mother ship	-----	350	28.50	2593.80	4700.00	-----

Prudent Mission Planning

The sample mission profile introduces important concepts affecting the effectiveness and efficiency of the DOV. This profile was constructed to show high propellant expenditures and illustrates an important point concerning methods of improving performance.

The DOV has its greatest mass at the beginning of the mission because it carries all its propellant and consumables at this time. For this reason, the largest plane change was executed at the end. The smallest possible altitude and inclination changes should be executed first.

The profile also shows just how much more propellant is used in executing plane changes. For this reason, missions should be planned to one plane or planes close to each other to avoid jumping back and forth and wasting propellant.

Finally, the DOV shouldn't be wasted chasing targets out of its feasible range. As discussed in Appendix I, large debris at high altitudes may prove to be outside the feasible target range. Smart planning will avoid wasting propellant for a rendezvous with a target the DOV will not be able to de-orbit.

Appendix I - Range of Debris DOV Can De-orbit

Using the equations discussed in Appendix C, the orbital velocities can be calculates for various debris masses. Figure I1 shows the results of these calculations. Because the equations in Appendix C are influenced by the mass of the earth, the mass of the debris satellite has no effect on the orbital velocity. Instead, the altitude determines the velocity.

Equation C.3 was used to find the velocity change necessary to de-orbit various debris masses. This velocity change is necessary to place the mass in an elliptical orbit reaching perigee in the atmosphere. Again, the mass of the debris does not effect these results (see Figure I2).

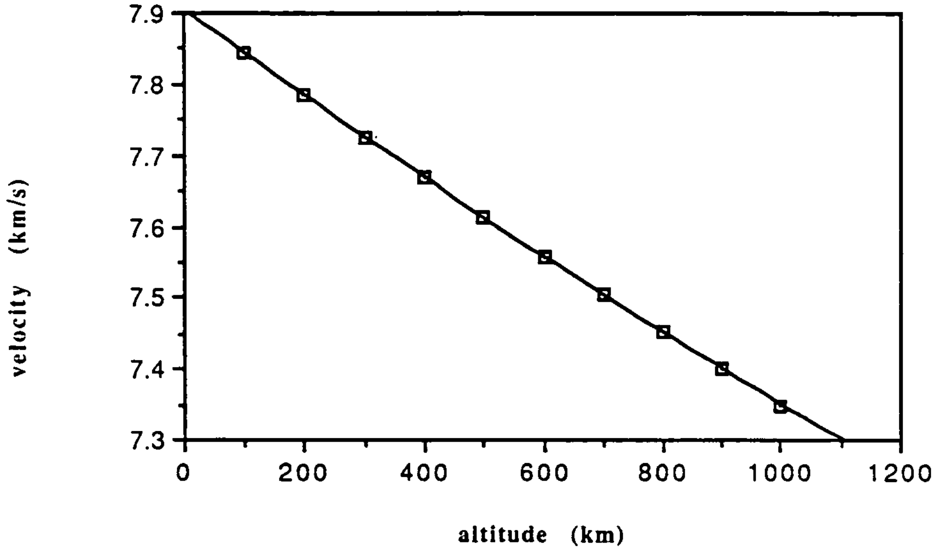
The mass of the debris does come into consideration when calculating the speed at which the projectile must be fired. Using conservation of momentum, the projectile speed was calculated for two projectile masses (50 and 100 kilograms), three debris masses (500, 2000, and 3500 kilograms), and altitudes ranging from 150 to 1500 kilometers (see Figure I3).

The results give us an estimate of the range of debris the DOV will de-orbit. The team estimated the maximum speed at which the rail-gun will fire the projectile to be on the order of five kilometers per second [3] This gives us our limitation on the range of the debris. Figure I3 shows a 100 kilogram CABS projectile can de-orbit:

1. A 3500 kilogram mass at 500 kilometers.
2. A 2000 kilogram mass at 1000 kilometers.
3. Debris less than 500 kilograms at altitudes above 2000 kilometers.

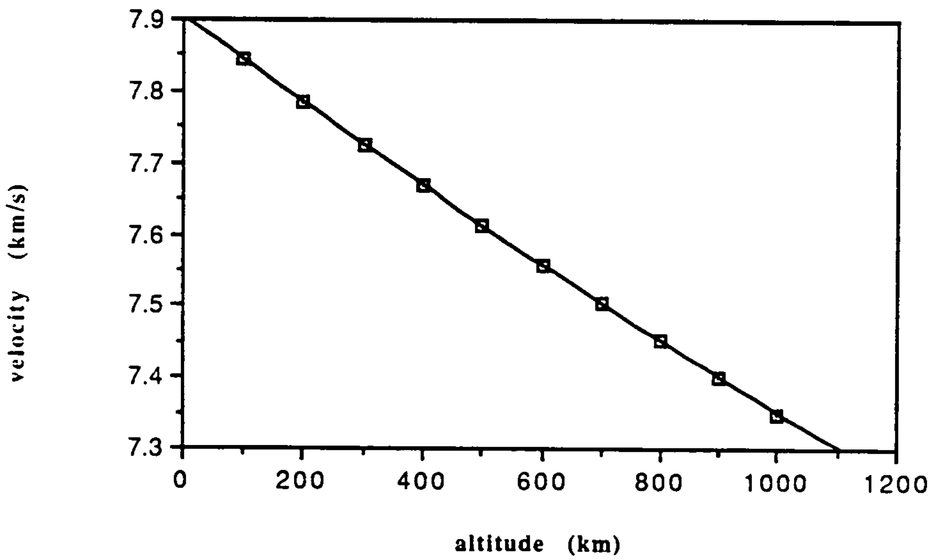
The DOV conceivably de-orbits debris in the range Mr. Connell suggested the team aim for (2000 kilograms at 800 kilometers altitude). The CABS concept has a major weakness in de-orbiting large masses above 500 kilometers. The use of more massive projectiles can help overcome this limitation. Impact studies and rail-gun tests will determine if this is a viable solution.

Velocities of a 150kg Debris Mass



a. 150 kilogram debris mass

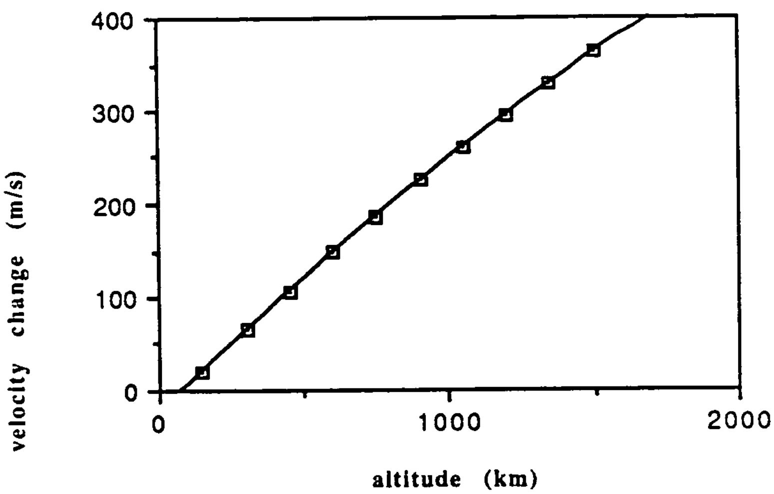
Velocities of a 3500kg Debris Mass



b. 3500 kilogram debris mass

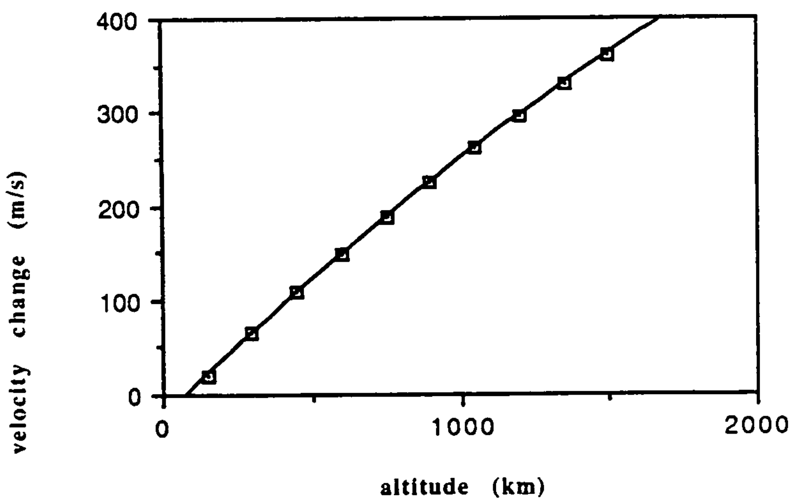
Figure I1. ORBITAL VELOCITIES OF SPACE DEBRIS

Velocity Change to De-orbit a 150kg Mass



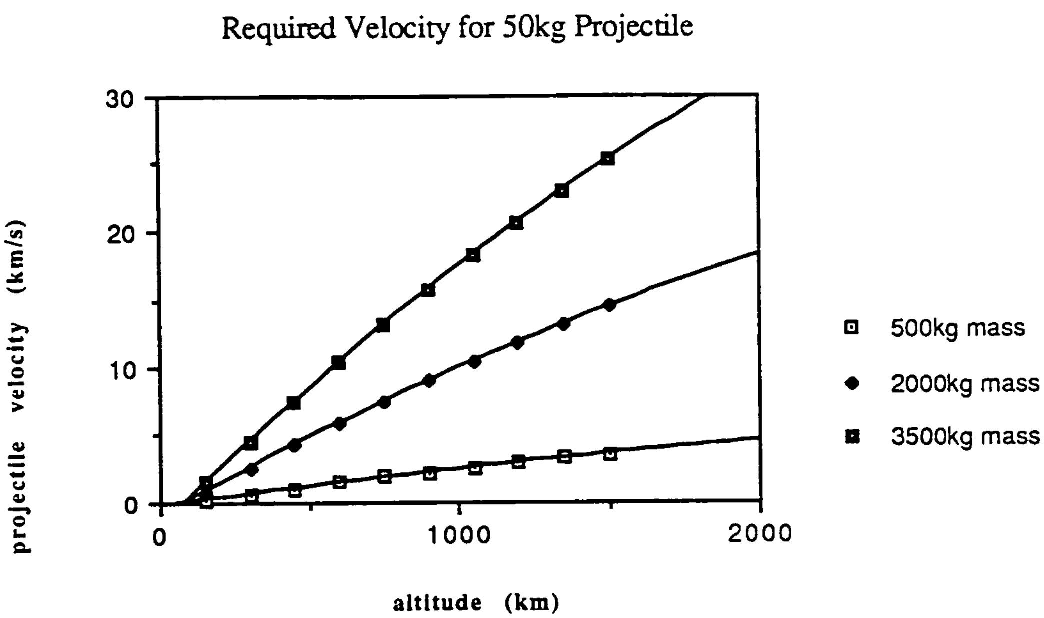
a. 150 kilogram debris mass

Velocity Change to De-orbit a 3500kg Mass

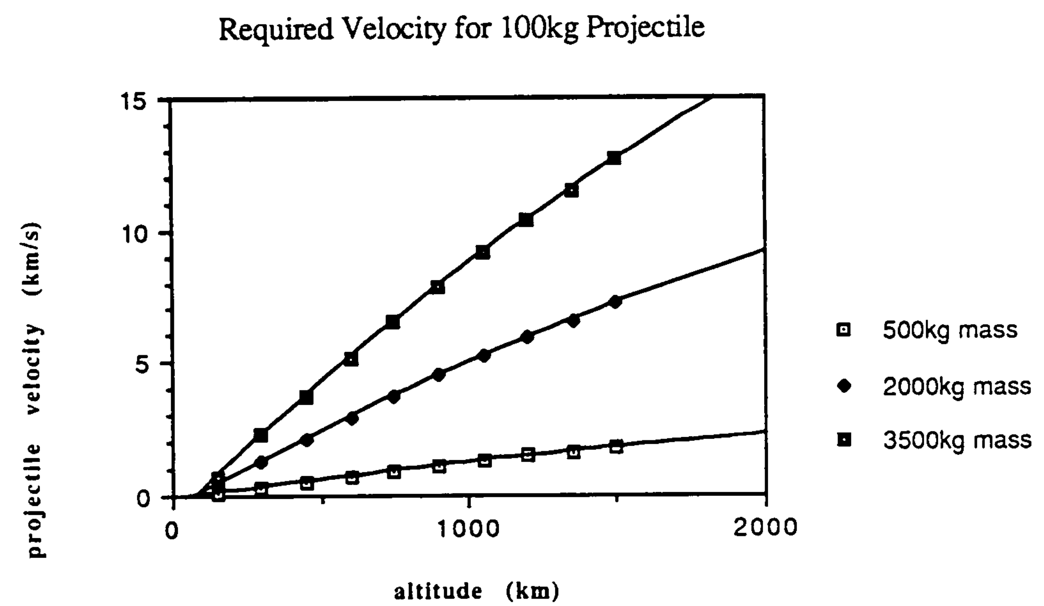


b. 3500 kilogram debris mass

Figure I2. VELOCITY CHANGES REQUIRED TO DE-ORBIT DEBRIS



a. 50 kilogram projectile velocities



b. 100 kilogram projectile velocities

Figure I3. REQUIRED VELOCITIES OF CABS PROJECTILES

Appendix J - References for Appendices

1. Bainum, Peter M. (ed.), Tethers in Space, Advances in Astronautical Sciences Series, Volume 62, (AIAA), 1983.

2. Conversation with Dr. Wallace Fowler, Professor, Department of Aerospace Engineering, University of Texas at Austin, October 2, 1990.

3. Conversation with Dr. Grady Rylander, Professor, Department of Mechanical Engineering, University of Texas at Austin, November 2, 1990.

4. Conversation with Dr. Kristin Wood, Professor, Department of Mechanical Engineering, University of Texas at Austin, October 4, 1990.

5. King-Hele, D.G., et al, The RAE Table of Earth Satellites, 1957-1982, (New York: John Wiley & Sons, Inc.), 1983.

6. Hogg, Ian v. and Gunston, Bill, World Encyclopedia of Fighter Aircraft, (London, England, Salamander Press), 1987.

7. Szebehely, Victor G., Adventures in Celestial Mechanics ,University of Texas Press, Austin, Texas, 1989.