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FINAL DESIGN OF A SPACE DEBRIS REMOVAL SYSTEM

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Submitted To:

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Executive Overview

Overview

In the fall semester of 1990, SPECS, Inc. of the University of Texas at Austin accepted the task of studying the orbital debris problem and designing a debris removal system. The debris problem has reached a stage at which the risk to satellites and spacecraft has become substantial in low Earth orbit (LEO). Our research uncovered that small particles posed little threat to spacecraft because shielding can effectively prevent these particles from damaging the spacecraft. The research also showed that, even though collision with a large debris could destroy the spacecraft, the large debris pose little danger because they can be tracked and maneuvered around. Additionally, there are many current designs to capture and remove large debris particles from the space environment have been proposed. From this analysis, the engineers at SPECS, Inc. have decided to concentrate on the removal of medium sized orbital debris, that is those pieces ranging from 1 cm to 50 cm in size.

Our current design incorporates a transfer vehicle and a netting vehicle to capture the medium size debris. The system is based near an operational space station located at 28.5 degrees inclination and 400 km altitude. The system uses ground based tracking to determine the location of a satellite breakup or debris cloud. This data is uploaded to the transfer vehicle and it proceeds to rendezvous with the debris at a lower altitude parking orbit. Next, the netting vehicle is deployed, tracks the targeted debris, and captures it. After expending the available nets, the netting vehicle returns to the transfer vehicle for a new netting module and continues to capture more debris in the target area. Once all the netting modules are expended, the transfer vehicle returns to the space station's orbit where it is resupplied with new netting modules from a space shuttle load. The new modules are launched by the shuttle from the ground and the expended modules are taken back to Earth for removal of the captured debris, refueling, and repacking of the nets. Once the netting modules are refurbished, they are then taken back into orbit for reuse. In a typical mission, the system has the ability to capture 50 pieces of orbital debris. One mission will take approximately six months and the system is designed to allow for a 30 degree inclination change on the outgoing and incoming trips of the transfer vehicle.

Transfer Vehicle

The transfer vehicle is the part of the debris removal system that moves the nets, netting vehicle, and netting modules close to the debris that is targeted for capture. A basic layout of the vehicle is shown in the following diagram.

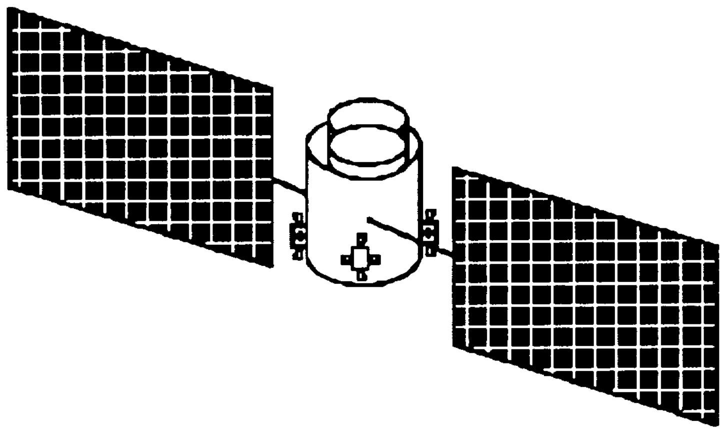


Figure 1 - Transfer Vehicle Layout

The transfer vehicle is capable of 30 degrees of inclination change on both legs of the trajectory. To accomplish the large inclination change without massive amounts of fuel, the transfer vehicle uses ion engines for thrust. This allows the fuel amount to be reduced to 10% of the amount that would be used if chemical engines were used. To provide the 35 kW of power that the 10 ion engines require, the transfer vehicle uses 2 high efficiency solar arrays. The vehicle also has batteries that will provide power while the vehicle is in the shadow of the Earth.

The transfer vehicle weighs approximately 8,000 kg. When it is fully loaded with the netting modules, propulsion module, and fuel, the transfer vehicle weighs 30,000 kg. Once the netting vehicle has captured the debris and returned to the transfer vehicle, the total mass of the transfer vehicle is about 21,000 kg. This reduction in weight is due to the fuel that is spent during the capture of the debris.

Control of the transfer vehicle is provide by control moment gyroscopes. The gyros will perform the fine attitude adjustments required as the vehicle rendezvous with the debris. For large maneuvers and momentum dumping, the vehicle also includes RCS thrusts similar to those used by the space shuttle.

Navigation of the transfer vehicle is done by a combination of onboard calculations and data from ground. Initially, the transfer vehicle receives data about the location of the debris and its location from external sources. From the data, the vehicle plots an intercept course. The vehicle proceeds along its trajectory and modifies it as new data is received about the location of the vehicle with respect to the debris.

The transfer vehicle receives this data from the command center located on Earth via a Ku-Band communications link through the TDRSS satellite. The transfer vehicle relays any commands to the netting vehicle with a V-Band communications system.

Netting Vehicle

The netting vehicle is responsible for gathering the debris and returning it to the transfer vehicle. The layout of the netting vehicle and the modules is shown in the following diagram.

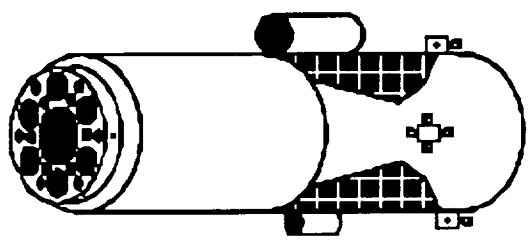


Figure 2 - Netting Vehicle Layout

Once in the debris orbit, the netting vehicle uses its onboard infrared (IR) tracking system to locate and target a piece of debris. Once the debris is targeted, the netting vehicle does a Hohmann transfer into a slightly different orbit. This allows the netting vehicle to close in on the debris piece. As the vehicle closes in on the debris to a distance of about 25 km, the tracking switches to a LADAR (LAser Detection and Ranging) system. The LADAR system provides more accurate ranging and location information to the netting vehicle as it approaches the debris. When the debris is within about 20 m of the debris, the vehicle will fire a net, capture the debris, and reel the net back into the netting module.

The netting vehicle will be controlled by ground or elsewhere with teleoperated controls. This will prevent the netting vehicle from having to have extensive artificial intelligence. The communication is relayed to and from the netting vehicle using V-Band link from the transfer vehicle through TDRSS. To provide the attitude adjustments, the vehicle will use control moment gyros in conjunction with RCS thrusters. The vehicle will also use

Hydrazine/Nitros Oxide fueled engines to provide the large orbital changes as the vehicle chases the debris.

Power is provided by surface mounted solar arrays. The arrays were surface mounted so that the area of the craft wasn't increased by the arrays. This is important because the smaller our craft, the less the chance of a collision with a debris particle. The array is also oversized by 25% to compensate for degradation due to debris impacts.

The total mass of the netting vehicle after it leaves the transfer vehicle is 8076.5 kg. Upon gathering all the debris and returning to the transfer vehicle, the mass is reduced to 5183 kg. This reduction in mass is caused by expending the fuel.

Table of Contents

Table of Figures

Table of Tables

1.0 Project Overview

The problem of orbital debris is a difficult problem to grasp. Most people have never seen pieces of orbital debris, much less witnessed some of its detrimental effects to the space environment. Accordingly, part of SPECS, Inc. design philosophy is to create an increased awareness of the past, present and future problems of orbital debris.

For this study, SPECS, Inc. has defined orbital debris as only those inactive objects in space resulting only from human factors. Therefore, the problem of orbital debris began in the 1950's when man launched the rocket into space [1.1,1]. Since this first launch into space, the problem has escalated because current technology requires multi-staged rockets to place payloads in space, and because these payloads have limited lifetimes. Thus, orbital debris consists of inactive payloads, spent upper stages and booster rockets, and other mission-related fragments.

Spacecraft anomaly reports can be examined to locate problems caused by orbital debris. In Appendix A, anomaly reports can be found on the following spacecraft: ISEE-1, ISEE-3, TDRS-1, TIROS-N, Voyager-2. The recorded anomalies range from contamination of thermal shielding to punctured pressure vessels. These anomalies were not fatal to the spacecraft's mission; however, they do indicate potential for major degradation. A better example of the damage orbital debris causes is illustrated by a recent space shuttle incident.

In June 1983, the space shuttle Challenger was struck by a piece of debris .two millimeters in size. The estimated damage was

$50,000 to the space shuttle. The untold damage was the danger to the lives of the five crew members aboard the Challenger.

Although the dangers of orbital debris had been expressed in the past, this incident caught the attention of engineers and scientists in the United States and around the world. Research projects with the sole purpose of solving and understanding the problem of orbital debris were created.

Orbital debris is a problem because collisions with tracked or untracked objects can cause severe damage to both space vehicles and personnel. Currently, the chances of catastrophic collisions are small, but the chances are increasing. Dr. Donald J. Kessler of the National Aeronautics Space Administration (NASA) has voiced one concern about the concentration of debris in Earth orbits. He theorizes that if the concentration of orbital debris becomes too high, there will be self-perpetuating collisions among the debris. This so-called Cascade, or Kessler, Effect could result in millions of small untrackable debris particles. [1.1,17].

SPECS, Inc. has initiated a program to clean up an important part of the orbital debris problem. Because Space Station Freedom (SSF) is scheduled to go into orbit by the end of the decade, we have concentrated our attention on eliminating debris near its orbit. The most feasible way of doing this is with an active system, a robotic spacecraft which collects the debris and removes it. The following sections presents the design process that was followed to accomplish this goal.

1.1 Project Objective and Scope

The project objective of SPECS, Inc. is to design a comprehensive orbital debris removal system that addresses man-made debris falling in the size range of one centimeter (cm) to fifty centimeters. To accomplish this objective, a project scope was created. The project scope encompasses four major areas: debris environment, mission scenario, design options, and a debris management philosophy.

A mission scenario that efficiently addresses the problem in a targeted debris environment has been developed. Feasible design options that enable the mission scenario to meet its objective have also been determined. Lastly, a debris management philosophy that encompasses both the short term goals of SPECS, Inc., as well as the long term goals that have yet to be implemented.

1.2 Defining the Debris Environment

No area of the space environment around the Earth has been excluded from the barrage of objects that have been launched during the last 30 years. Although the debris environment is huge, some areas of outer space are more populated than others. One of the initial tasks of SPECS, Inc. was to determine what types of debris contribute significantly to the debris problem, and where this debris is located.

1.2.1 Types of Debris

Orbital debris may be broadly classified as either trackable or untrackable debris. The North American Aerospace Defense Command (NORAD) presently tracks about 7,100 objects. Figure 1.2

displays the percentage breakdown of tracked objects in space [1.2,4].

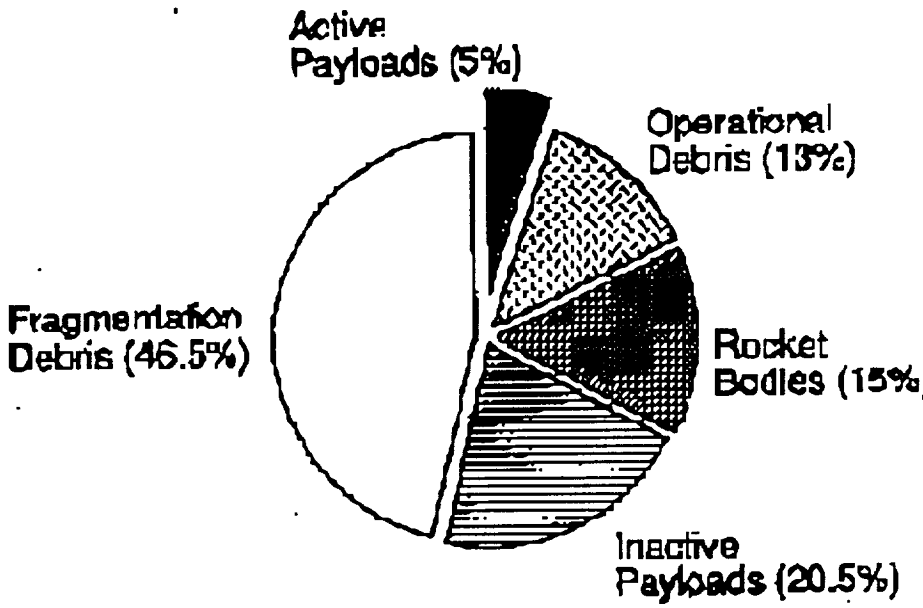


Figure 1.1 Tracked Debris Separated into Groups

It can be seen that fragmentation debris makes up about half of all the tracked objects. Besides the tracked objects, NORAD estimates that an additional 50,000 - 60,000 objects too small to track are present in low earth orbit [1.1,13].

1.2.2 Debris Location

A logical initial choice for targeting a debris area is the environment around the Space Station Freedom (SSF). Not only will this provide the SSF with a protective device against orbital debris, but it will also open up more options for possible scenarios. The question is whether enough orbital debris be available in the vincinity of the SSF. Figure 1.2 displays a global outlook of all the debris orbiting the earth [1.3,1.7]. From this figure, it can be seen that orbital debris exists in most orbits around Earth; however, the orbits containing the highest percentage of orbital debris are naturally the critical areas.

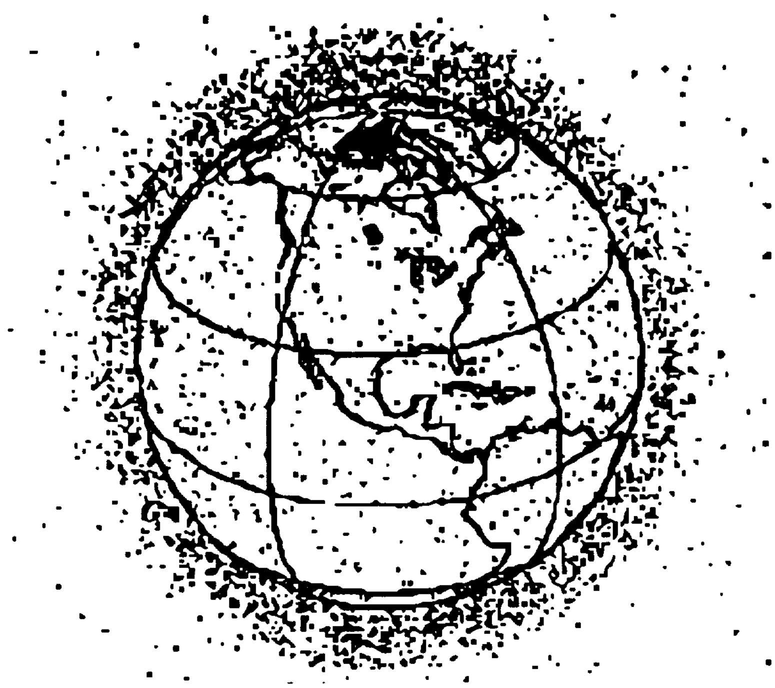


Figure 1.2 Global Outlook of the Debris Problem

Further, Figure 1.3 shows a large debris population for altitudes in low Earth orbit (LEO) [1.3,3.22]. A targeted altitude range from 200 kilometers (km) - 800 km was chosen because of the high percentage of debris found in this area.

In addition to the altitude, the inclination of the orbital debris was considered when targeting a region in outer space. In Figure 1.4,

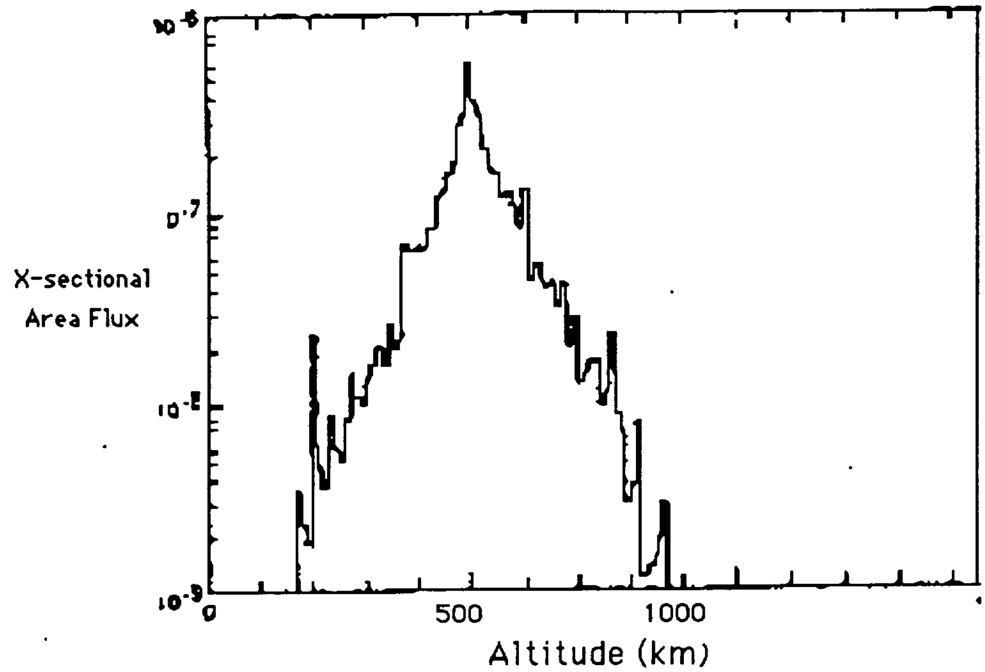


Figure 1.3 Area Flux for Large Debris at Given Altitudes

which displays the amount of orbital debris at certain altitudes and various inclinations, it may be seen that the inclinations around 28 degrees and 65 degrees have a large debris population [1.3,3.12]. These target areas for altitude and inclination match up ideally with the SSF's environment which will be located at 400 kilometers and at an inclination of 28.5 degrees.

1.2.3 Sizes of Orbital Debris

SPECS, Inc. defines a piece of orbital debris as large if its dimension is greater than fifty centimeters. Similarly, the dimensions for medium debris range from one to fifty centimeters, and small debris is anything smaller than one centimeter [1.2,5]. By

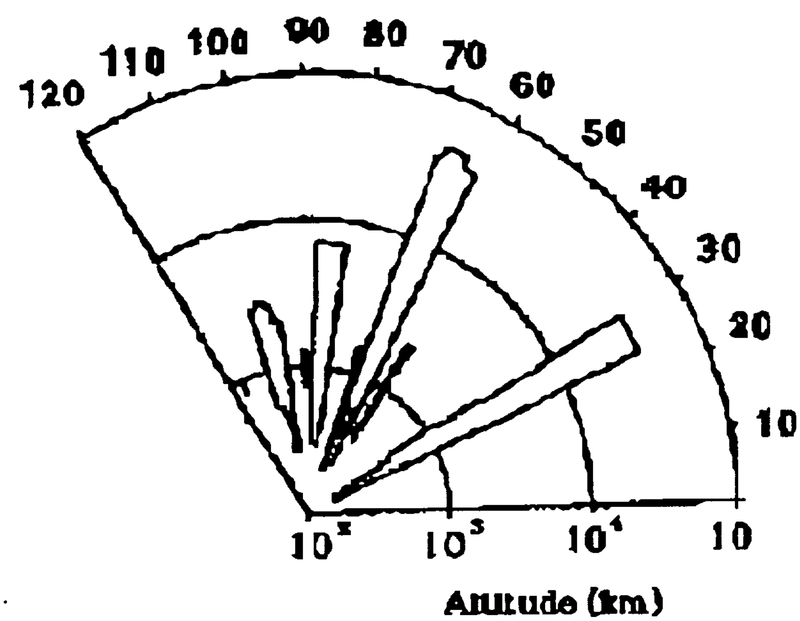


Figure 1.4 Debris at Given Altitudes and Inclinations

analyzing the dangers that each size group of orbital debris poses to the space environment, a logical choice for targeting a specific debris size was made.

Large debris pieces can be tracked from Earth. Presently NORAD is responsible for cataloging the positions and orbital elements of approximity 7,100 pieces of large debris [1.2,4]. Since the orbital elements of large debris pieces are known, they do not create a serious threat to a spacecraft capable of communicating with Earth. The spacecraft would have plenty of time (on the order of hours to days) to maneuver away from the object, thus, eliminating the threat of collision. Further, other designs are available that specifically address large debris.

When considering small debris, it was found that the technology of structural shielding can be used to alleviate most potential danger. Present technology enables spacecraft to be fortified with structural shielding that protects the spacecraft from debris hits of less than one centimeter. Since the number of small sized debris in outer space is approximately four billion, the idea of shielding against these small pieces seems to be the only sensible solution [1.2,4].

On the other hand, both tracking and shielding techniques are ineffective against medium size debris. NORAD has estimated that 17,500 pieces of medium-sized debris exist, and because they cannot be tracked or shielded against, they represent the most eminent danger to operational spacecraft [1.2,4].

Using the sizing decision matrix shown in Table 1.1, the three sizes were compared in the categories of existing protection, existing designs, and existing debris quantity (0 = lowest concern, 5 = highest concern). The medium-sized debris was recognized as the biggest threat to the space environment.

Table 1.1 Orbital Debris Sizing Matrix

DEBRIS SIZE	EXISTING PROTECTION	EXISTING DESIGNS	DEBRIS QUANTITY	TOTALS
LARGE (SIZE> 10cm)	3	3	1	7
MEDIUM (1cm<SIZE<10cm	5	5	3	13
SMALL (SIZE<1cm)	2	4	4	10

1.2.4 Targeted Debris Environment

After examining the different altitudes, inclinations and sizes of debris in the overall orbital debris environment, a specific area was focused on. The engineers at SPECS, Inc. felt that more progress on the orbital debris problem could be gained by concentrating on cleaning up debris clouds full of medium sized debris at altitudes ranging from 300 km - 1000 km at inclinations of 28.5° ± 30°. Most of the debris in this area resulted from satellite breakups. However, future breakups may shift the targeted area.

1.2.5 Dynamics of Satellite Breakup

The dynamics of a satellite break-up must be assessed since most of the debris in the targeted area is the result of such events. Figure 1.5 shows the history of a breakup in several stages [1.4,15-19].

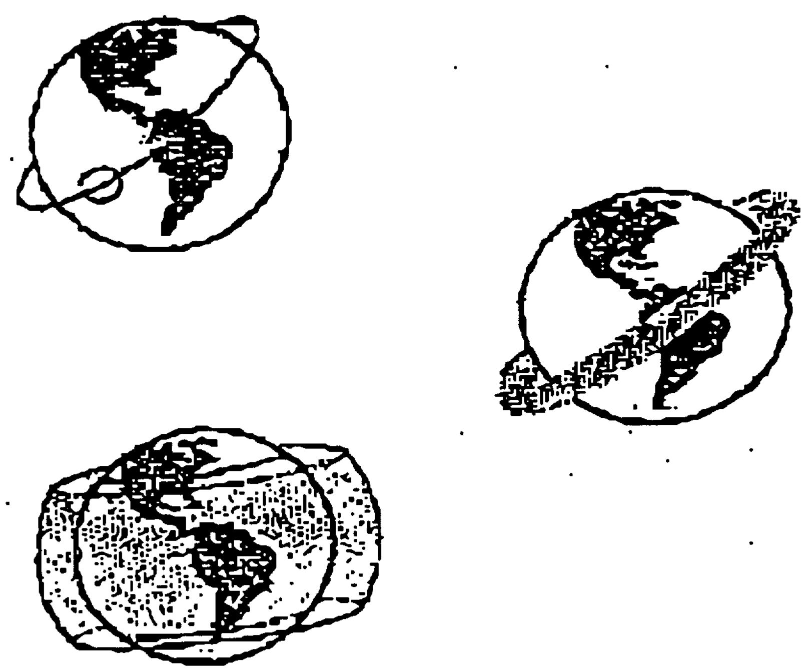


Figure 1.5 Evolution of a Satellite Breakup

A satellite breakup debris cloud initially forms an ellipsoid around the original location of the orbiting object. Due to differentials of the particles in their orbital period this ellipsoid evolves into an irregular, narrow torus encircling the Earth. This torus typically closes after several months to a year [1.5,223-241]. Further, the regression rates of the right ascension cause the torus to eventually dismantle into a band about the Earth. This low density band is limited in latitude only by the maximum inclination, and in altitude by the extremes of the cloud. This phase is reached several years after the event.

The rate at which these phases are reached is largely a function of the velocity imparted to the debris fragments upon breakup and the initial altitude and inclination of the original satellite.

1.3 General Project Requirements

In order to ensure the feasibility of efficiently meeting the project objective, mission requirements and performance parameters were instituted. The SPECS, Inc. mission design team has set the following requirements for the mission: the ability to reach targeted altitudes and inclinations, and the ability to capture a significant amount of debris

Performance parameters are listed below with a criticality rating next to them (0 = low criticality, 5 = high criticality).

- Fuel Budget — Criticality 5
- Power Requirements — Criticality 5
- Weight — Criticality 5

- Safety — Criticality 5
- Resupply/Maintenance — Criticality 4
- Lifetime — Criticality 4
- Effects on Environment — Criticality 4
- Cost — Criticality 3
- Design Complexity — Criticality 3
- Time Constraints — Criticality 1

1.4 Assumptions

Along with mission requirements, some general assumptions were made to ensure a workable mission.

- Satellite breakups will eventually exhibit torus qualities
- Tracking technology will accurately track orbit debris down to a size of one centimeter (cm)
- With the help of cameras, the geometric shape of the debris will be discernable

Other assumptions concerning the mission operations of the design will have to addressed after further research.

2.0 Design Approach

Initially, SPECS, Inc. considered all sizes of debris in assessing the debris problem; therefore, design scenarios were conceived for all types of debris located from low earth orbit to geosynchronous orbit. However, during the conceptual design phase, the scope of the problem was narrowed down to medium debris within a targeted region.

2.1 Design Options

SPECS, Inc. considered several designs to attack the problem of orbital debris. In considering designs to capture debris, the tumbling motion of the debris caused a problem when trying to grapple the debris directly. However, by using nets to capture debris the problem of spinning and tumbling debris is eliminated. SPECS, Inc. has designed an active netting system that uses Kevlar nets to capture pieces of medium-sized debris.

2.2 Primary Design

The active netting system shown in Figure 2.1 is composed of a Propulsion Module and a Netting Module. The Propulsion Module is used to perform the orbit transfers around the debris orbit and the netting module performs proximity maneuvers to reach the target debris. Each Netting Module contains several nets capable of capturing debris sizes ranging from 1 cm to 50 cm.

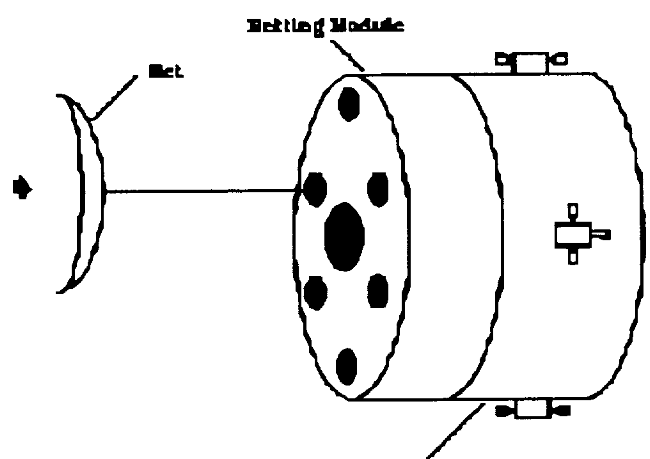


Figure 2.1 - Active Netting System

The active netting system will target areas of satellite breakups where a high density of debris exists. Details about the active netting system will be presented later in this document.

2.4 Design Philosophy

SPECS, Inc. has developed a debris management philosophy to assult the present and future problems caused by orbital debris. The objectives of SPECS, Inc. have been divided into near term, mid-term and long-term strategies:

- Near Term Strategy
    - Attack the medium sized fragmentation debris (1 cm to 50 cm)
    - Develop an active system using a netting device (or similar design)
    - Reduce the collision probability in the target altitude range
    - Single vehicle released from shuttle, space station or launched from Earth
    - Implement an international prevention policy on space debris
- Mid-term Strategy
    - Develop a network of active/passive devices
    - Launch an operational orbiting base
    - Perform area sweeps and explosion clean-ups
- Long term Objectives
    - Increase operational range to geosynchronous and transfer orbits
    - Expand system to remove the larger, tracked debris

SPECS, Inc. realizes that correcting the problem of orbital debris is very costly and that the immediate satisfaction of cleaning up a few debris clouds will not have a noticeable effect on the overall problem. However, SPECS, Inc. has initiated a start toward solving the orbital debris problem. Hopefully, other groups will join in helping complete the SPECS, Inc. debris management philosophy, thus, solving the problem of orbital debris and making the space environment safe for the people of Earth and those wishing to visit.

3.0 System Concept

3.1 Debris Removal System

The Debris Removal System will actively seek out each piece of debris and capture it. Because of the size of debris under consideration, SPECS, Inc. decided that a net shot from a main vehicle could be used to retrieve it. In order to capture as many pieces of debris as possible, the main vehicle was divided into two main components: the Transfer Vehicle (TV) and the Netting Vehicle (NV).

The Transfer Vehicle will carry the Netting Vehicle to a parking orbit near a debris torus; the Netting Vehicle will then use the TV as a temporary base while it seeks debris. The Netting Vehicle is also divided into two components : a Netting Module (NM) and a Propulsion Module (PM). Each Netting Module will contain several nets to capture debris, and the Transfer Vehicle will have several Netting Modules docked to it for later use.

3.1.1 Transfer Vehicle

The Transfer Vehicle will carry the Propulsion Module and several Netting Modules from the main base of operations to an orbit near the debris torus. The Propulsion and Netting Modules will be docked to the front of the Transfer Vehicle as shown in Figures 3.1 and 3.2.

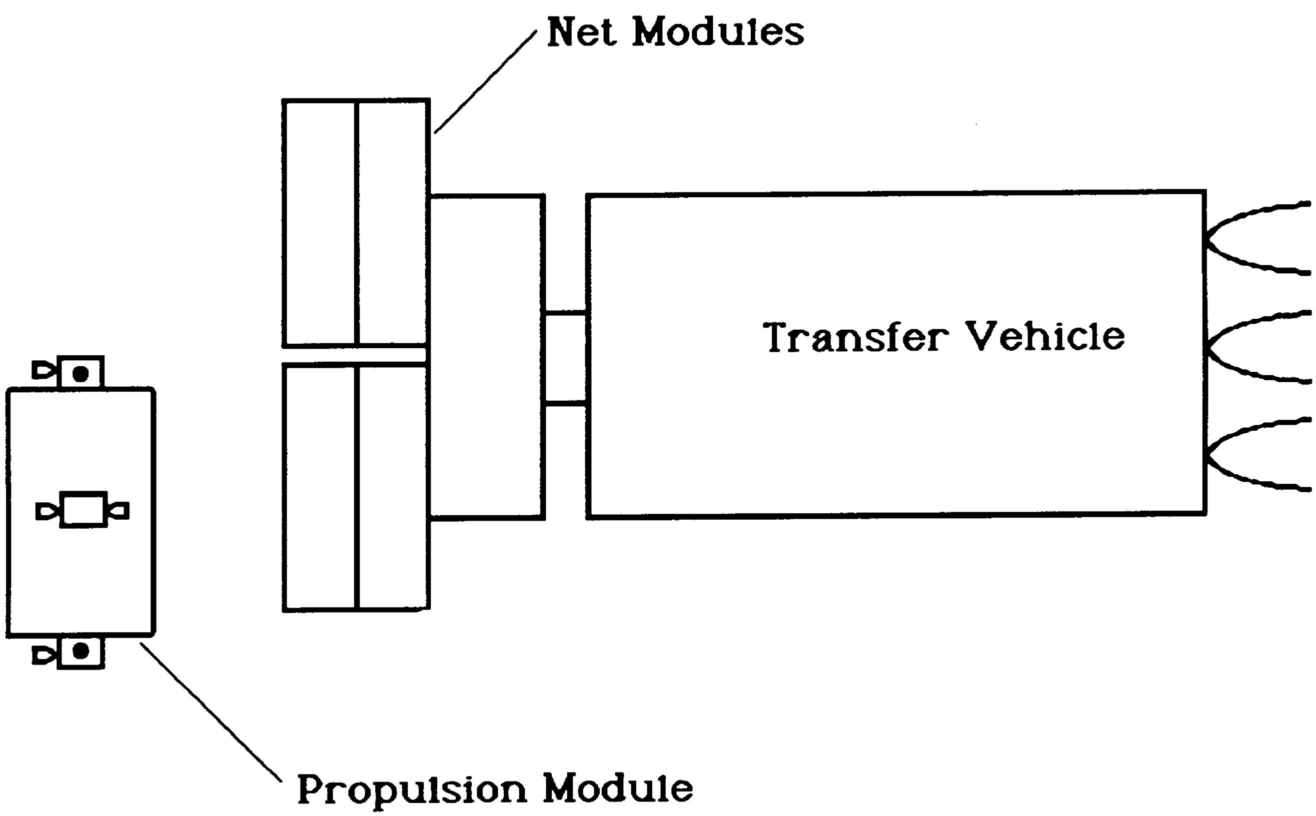


Figure 3.1 Conceptual Drawing of TV and Modules-Top View

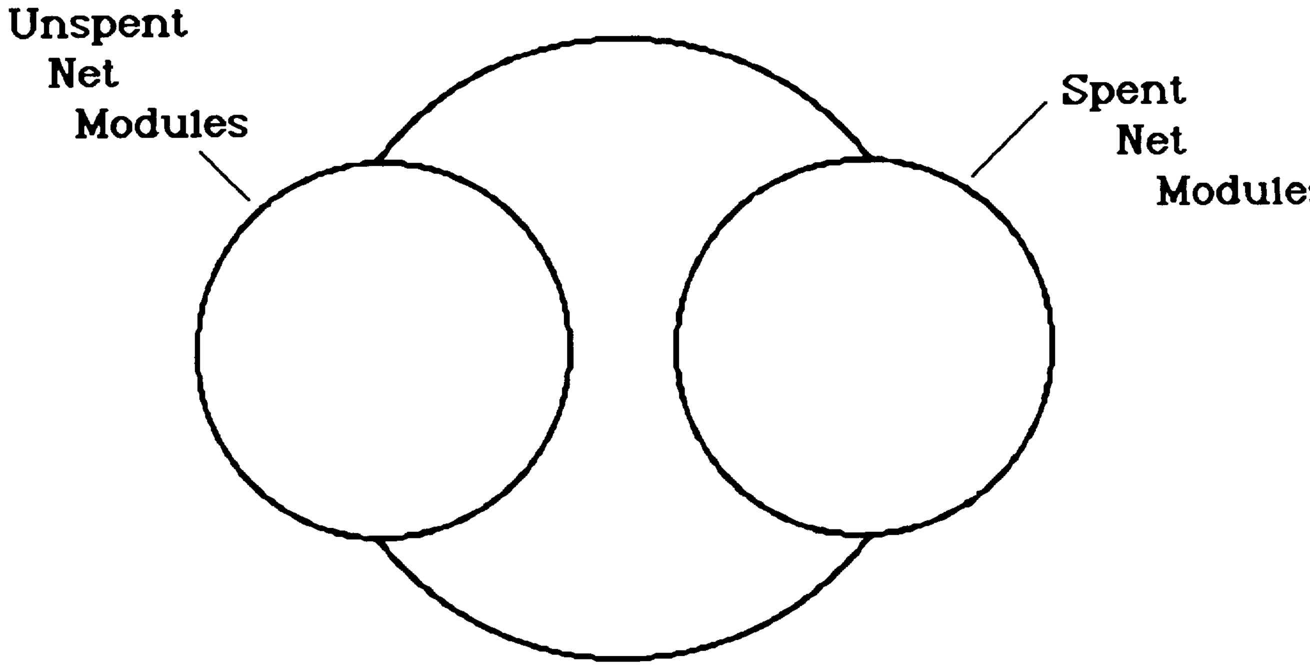


Figure 3.2 Conceptual Drawing of TV and Modules-Front View

After departing from the Transfer Vehicle, the Netting Vehicle will collect debrisin the prearranged atrget area. It will return when all the nets have been expended and dock with the Tansfer Vehicle as shown in Figure 3.3. The Propulsion Module will separate from the spent Netting Module, dock with a new one, and then the refurbished Netting Vehicle will leave for a new collection sweep.

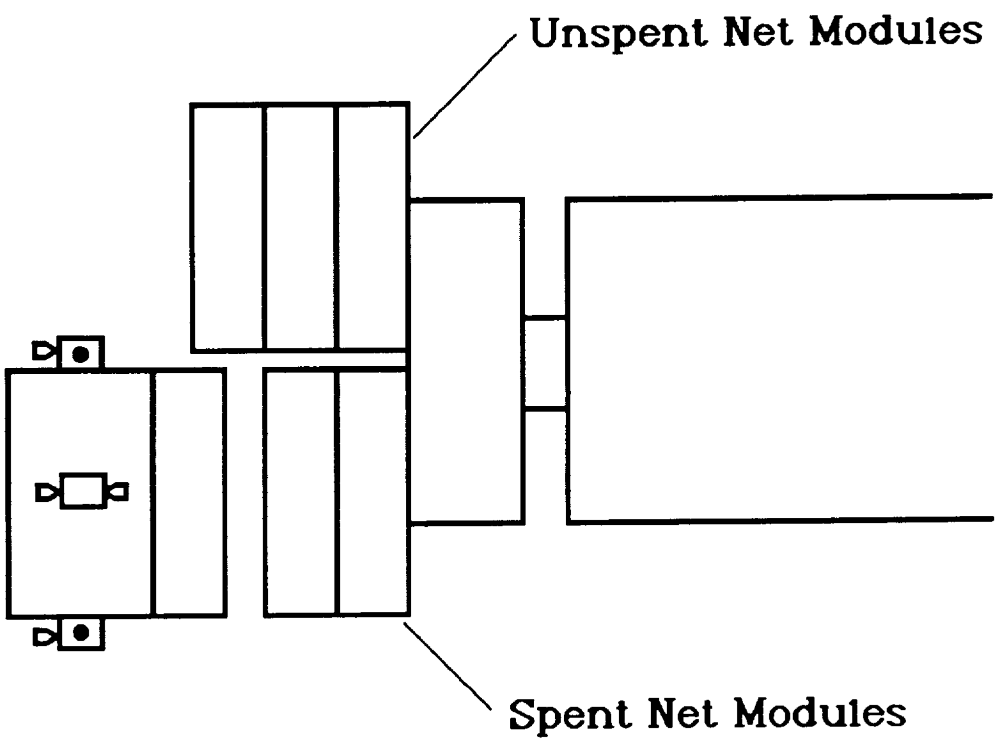


Figure 3.3 Docking with Transfer Vehicle

Because the Transfer Vehicle will have to stay in orbit for several months while the Netting Vehicle is collecting debris, it will need to be able to power itself for an extended period. It will also need to maintain a constant attitude, especially during docking, as well as a consistent orbit.

3.1.2 Netting Vehicle

Figure 3.4 shows how the Propulsion and Netting Modules will fit together to form the Netting Vehicle. As can be seen, there are several holes in the Netting Module, each of which will contain a separate net to capture a single piece of debris. During normal operating conditions, these holes will be covered, but when a net is being launched and retrieved, the cover will be retracted.

RCS (Reaction Control System) thrusters are shown on the Netting Vehicle in the figure. These, or control moment gyros, will be needed to make adjustments to the Netting Vehicle's orientation when the net is being retrieved so that it does not wrap around the spacecraft.

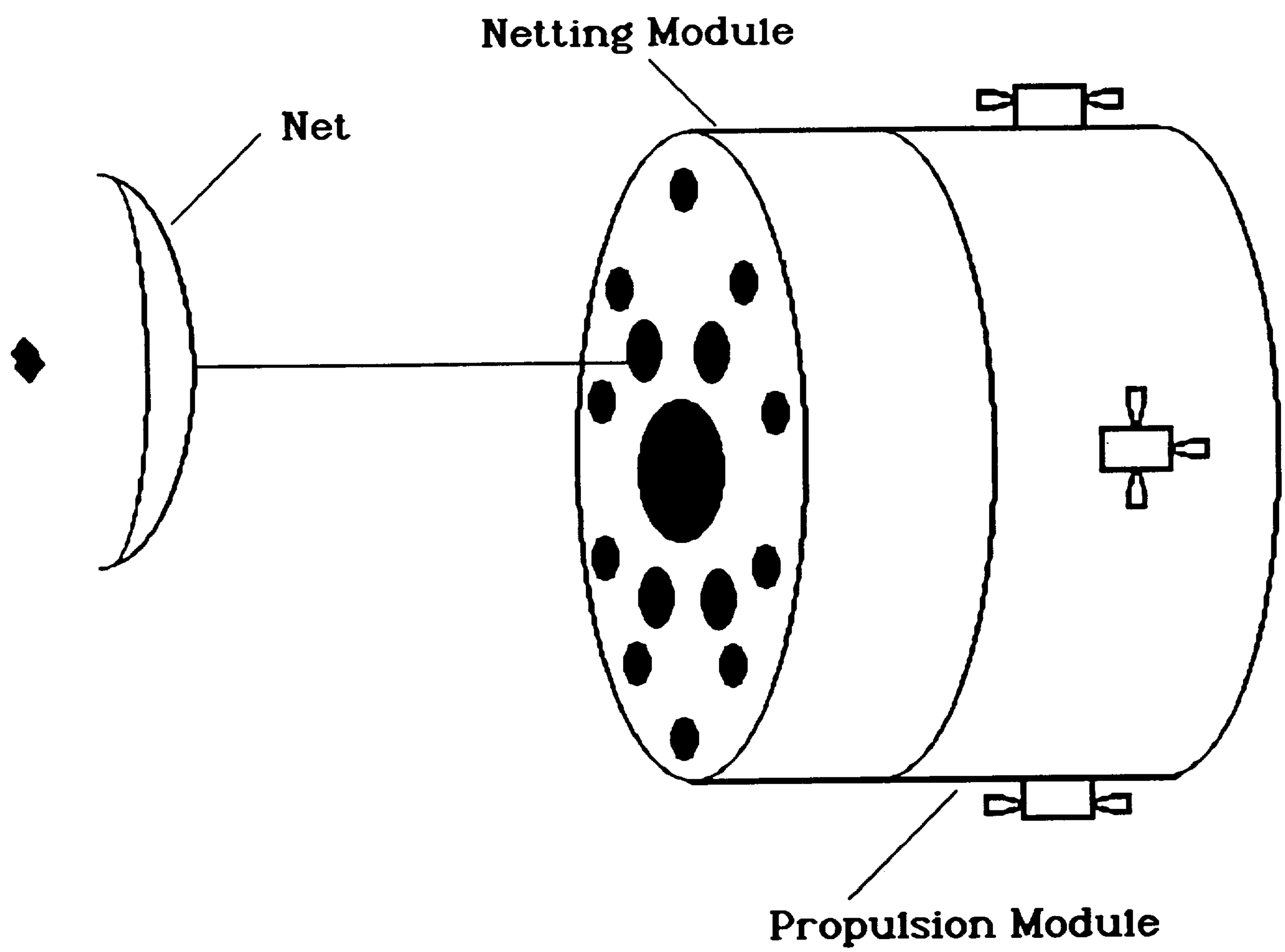


Figure 3.4 Conceptual Drawing of Netting Vehicle

Because the Netting Module is designed to capture several pieces of debris per mission, each mission will require a substantial amount of fuel. Considering that there will be several Netting Modules on the Transfer Vehicle, it would be very inefficient to store all the necessary fuel on the Propulsion Module. Instead, the fuel will be stored on each Netting Module, and, when the Propulsion Module docks with the Netting Module, a fuel link will be established between the tanks on the NM and the engine on the PM.

Since the subsystems requiring the most power will be located on the Propulsion Module, the power system will be located there as well. It will consist of a solar array mounted on the body of the spacecraft and a rechargeable battery. The power system will be linked to the Netting Module during docking in a similar manner to the fuel system so that the netting subsystem can be operated.

4.0 Mission Scenario

Based on the netting design, several possible mission scenarios for the debris removal system have been considered. All scenarios have been evaluated using the criteria listed under the General Project Requirements section. After considering and evaluating all reasonable mission options, the final scenario was chosen. Altenative mission options are also briefly discussed.

4.1 Mission Options

In designing the mission scenarios, several options have been considered. Alternatives for different stages in the scenarios can be seen from the logical structure in Figure 4.1. Arrows in the logical connections between the mission elements indicate closed loops. The processes within a loop can be repeated until it is necessary or intended to exit the loop. The connection between the highlighted mission options indicates the final mission scenario that was chosen.

4.2 Final Mission Scenario

4.2.1 Active Debris Removal System Launch

The Transfer Vehicle, the Netting Vehicle, and three Netting Modules will be launched on two Space Shuttle flights. The payload bay of the Shuttle has an area of 160 m^2 and is able to carry payloads 4.5 meters in diameter and 18 meters long. The maximum payload weight for the Space Shuttle when taking off is 29,500 kg. The shuttle specifications have been considered in the design process. The debris removal system will be placed in a 400 km altitude, circular parking orbit, after being unloaded from the cargo bay by the Remote Manipulator System (RMS) of the Shuttle.

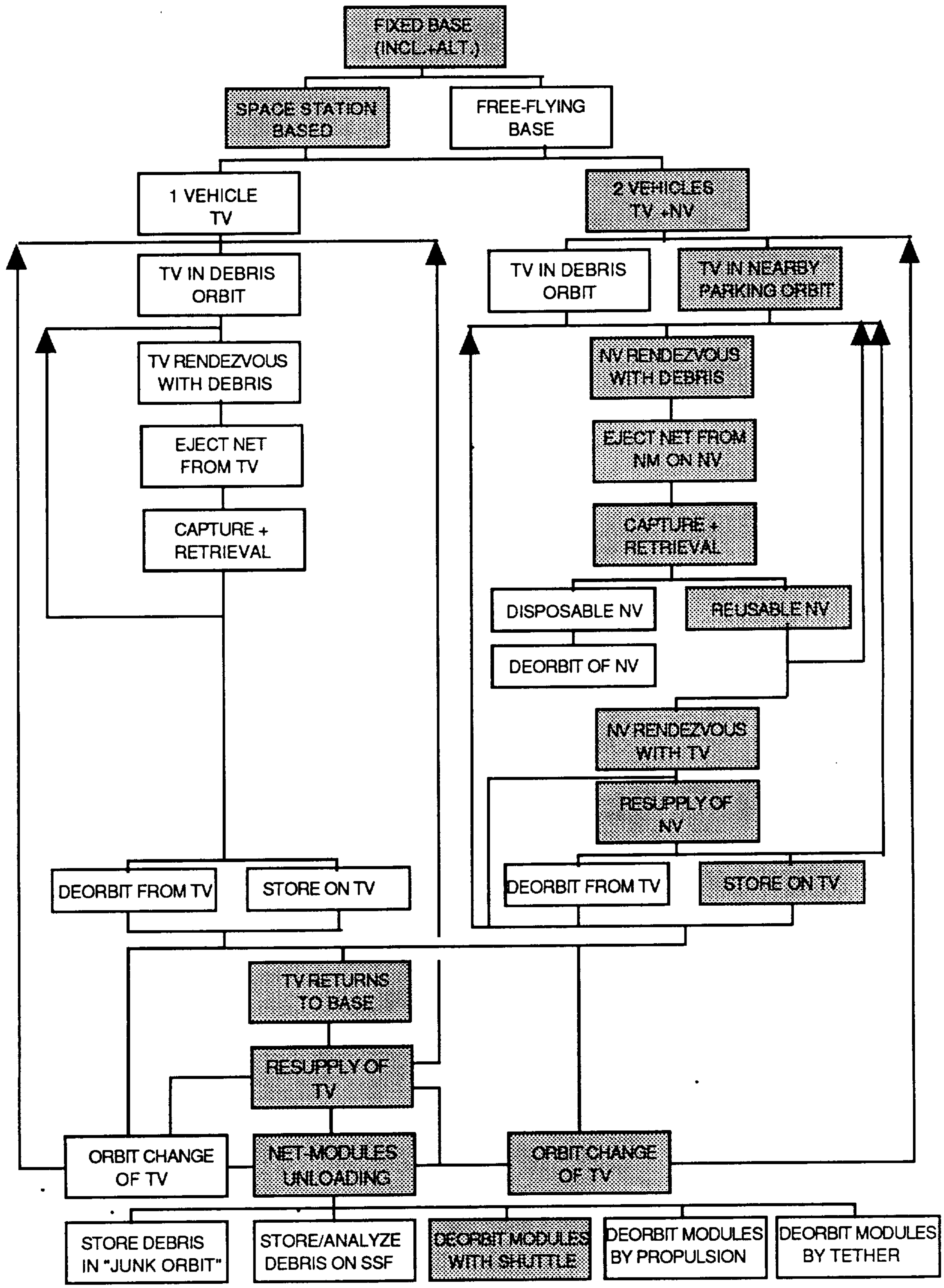


Figure 4.1 Mission Scenarios and Final Scenario

4.2.2 Orbital Transfer of the TV

The main engines of the Transfer Vehicle will be used to carry the Propulsion Module and three Netting Modules to the vicinity of the targeted area. The operational range of the debris removal system comprises orbits from 400 to about 1000 km altitude and an inclination of 28.5 ° ± 30°.

The Transfer Vehicle will spiral up its orbit using its electric propusion system to reach the target orbit, which has been determined from ground tracking by the control center. All the major inclination changes will also be performed by the Transfer Vehicle. The Tranfer Vehicle will go into a parking orbit with a slightly lower (50 to 100 km) semi-major axis than the actual debris orbit. Once the vehicle has reached this position in the parking orbit and its propulsion system is turned off, the solar panels will be retracted so that they are aligned along the side of the Transfer Vehicle. In a gravity gradient stabilized position it will orient one panel towards the sun. From there it will release the Propulsion Module attacted to one Netting Module to go and capture the debris.

4.2.3 Rendezvous and Debris Capture

The Netting Vehicle will use a Hohmann transfer to move from the Transfer Vehicle orbit to the debris orbit and rendezvous with a piece of debris. Onboard sensors will be used to track the debris when the debris is within a few km of the vehicle. According to the detected size of the piece targeted, an adequate net will be ejected by a spring mechanism to catch the debris when the debris is within a range of 50 m. The net will be closed and reeled back into the

Netting Module by an attached chord. The Netting Vehicle will then target another piece of debris and go into a drift orbit to rendezvous with it. This procedure can be repeated until all the nets of the module have been used.

4.2.4 Netting Vehicle Resupply

The Netting Vehicle will then return to the Transfer Vehicle for resupply. It will dock with the Transfer Vehicle to unload the spent Netting Module. Another docking procedure will provide the Propusion Module with a new Netting Module allowing the Netting Vehicle to leave for another collection sweep.

4.2.5 Further Orbit Changes

After the resupply procedure the NettingVehicle can return to the same debris orbit in order to capture further debris, or the Transfer Vehicle can take the Propulsion Module and the Netting Modules to another target orbit. This procedure is repeated until all the Netting Modules are filled up with debris. The Transfer Vehicle will then return to the Space Station with all the spent Netting Modules and the Propulsion Module attached. If the TV is going to return to the same or a similar orbit the Propulsion Module can remain in a parking orbit to wait for the return of the Transfer Vehicle and new Netting Modules. This procedure will save propellant.

4.2.6 Resupply Base on SSF

For maintenance and resupply reasons, the active debris removal system will be based on the Space Station Freedom. The Space Station is assumed operational by the year 2000.

For these operations, the Transfer Vehicle will first fly into a certain area within the proximity of the Space Station. Using EVA astronauts or the robotics on the Space Station ("Canadarm" mobile servicing system), any maintenance that the Transfer Vehicle needs will be performed. The Transfer Vehicle will be resupplied with three new Netting Modules, which have been launched via the Space Shuttle. Spent Netting Modules will be placed in the shuttle payload bay for return to earth. If the shuttle is not available, the spent Netting Modules will be attached to the Space Station truss at a predefined area, where they are stored until they can be deployed to the shuttle payload bay.

This resupply option seems to be reliable: the shuttle will supply the Space Station frequently, and space for the Netting Modules will be available with the logistics modules for the Space Station. For the reentry purposes, the maximum payload reentry weight for the shuttle (23,000 kg) must be considered.

4.3 Discussion of Alternative Missions

Because of difficullty in the resupply and maintenance sequences, a free-flying base was considered to be more complex and less reliable than the Space Station based system: additional robotics on this base would be necessary, and an additional docking maneuver of the Shuttle would be required. Nevertheless, if the system proves to be effective, SPECS, Inc. considers expanding the active debris removal system to different inclinations using a free-flying resupply base with onboard robotics.

Using only one vehicle for the orbit transfers and the debris capture operations was not considered to be efficient because of the large amount of fuel needed.

The option of transferring the Transfer Vehicle directly into the debris orbit was excluded due to safety aspects. The collision probability with debris is relatively high and poses a high risk, especially for the large solar panels.

Deorbiting the spent Netting Modules from the debris orbit by either a disposable Propulsion Module or an additional deorbit device has been excluded due to cost and mass. It was decided that a reusable system would be cheaper in the long run, and would also limit the production of further debris.

For comparable reasons, deorbiting the Netting Modules by special deorbit devices from the Space Station was considered less effective than the deorbiting with the Shuttle. On the other hand, this scenario will be strongly dependent on the ability Shuttle flights to send used Netting Modules to Earth in the Space Shuttle. Therefore, the storage of the spent modules in a safe orbit that could be tracked from Earth is still a viable option.

The deorbit of the modules by a tethered deployment and release has also been considered since this system is being designed for the Space Station to use with reentry capsules. If the tether system is operational by the year considered to launch the debris removal system, this option can be reconsidered.

The final option of storing and analyzing the debris on the Space Station seems to be feasible, but safety aspects as well as the minimization of Extra Vehicular Activities (EVA) have to be considered.

5.0 Subsystem Design

The list of mission requirements that were defined in the General Project Requirements section of this report formed the foundation for all subsystem selections. These subsystems include:

- Propulsion
- Power
- Thermal Control
- Communications
- Data Processing System (DPS)
- Tracking
- Guidance, Navigation, & Control (GNC)
- Netting
- Structural Materials
- Fuel Requirements

5.1 Propulsion

5.1.1 Transfer Vehicle

Electric propulsion was selected as the method of transportation for the Transfer Vehicle. The decision to select electric propulsion was made under a list of specific propulsion requirements as shown in Table 5.1. In comparison with chemical propulsion, electric provides greater efficiency, lower fuel costs, greater operating times, and a lower chamber pressure for easier fuel storage [5.7]. NASA prohibition of H_2O_2 in the Shuttle Bay limits chemical propulsion choice to monopropellants [5.5]. The low-thrust option is a viable choice for the Debris Removal System since time is not a critical factor.

Table 5.1 Propulsion Requirements

Table5.1 Propulsion Requirements

- **Clean Exhaust**
- **Storable Fuel**
- **Fuel Production Cost**
- **Total Propellant Mass**
- **Time Of Flight**
- **Thruster Efficiency**
- **Operating Time**
- **Feasilbility**

Selection of the specific type of electric thruster was made through a comparison of four different propulsion systems. Table 5.2 is a compiled list of several important performance characteristics on each of the four thruster classes.

With a low input power, good thruster efficiency, and a fairly high ISP, an electro-static Xenon ion propulsion system has been selected as the primary propulsion unit for the Transfer Vehicle. A total of 10 ion thrusters will be used to provide the continuous thrusting that the Transfer Vehicle needs to reach the target orbit, which is about 50 km. below the debris torus. The ten engines will

Table 5.2 Transfer Vehicle Electric Propulsion Options

Table 5.2 Transfer Vehicle Electric Propulsion Options

Types	ISP (s)	Thruster Efficiency	Thrust (N)	Power (kw)	Life (hrs.)
Arcject (NH3)	968	37%	2.4	30.0	750
Ion (Xe)	3600	70%	0.4	3.5	10,000
Ion (A)	7000	80%	5.4	50.0	10,000
MPD Thruster	5000	30%	100.0	500.0	1,000
[Reference 5.7,9,10,11,13]					

continue thrusting even during shadow times through power from the regenerative MnO_2-H_2 batteries. Table 5.3 is a mass and power breakdown of the electro-static Xenon propulsion system onboard the Transfer Vehicle.

Table 5.3 Electrostatic Thruster System

Table 5.3 Electrostatic Thruster System

System Configuration:		Total Mass Breakdown:	
•Total Mass:	2550 kg.	•Thrusters	1000 kg.
•Total Power:	35 kw.	•PPU	120 kg.
•Total Volume:	5 m^3*	•Radiator	630 kg.
		•Cradle Mass	800 kg.

*Does not include fuel storage or power supply volumes.

The 10 ion thrusters onboard the Transfer Vehicle will generate only 4 N of thrust. The thrust acceleration of the Transfer

Vehicle will be about .0002 m/s^2 for a system mass of about 21,000 kg. With such a low acceleration, concern arises as to whether the thrust acceleration can overcome perturbations such as J_2, drag, or solar pressure. Appendix C contains a number of computations done to determine the Transfer Vehicle's acceleration due to these perturbations.

At a station orbit of 400 km, the drag perturbation was about 4,300 times smaller than the thrust acceleration.

The computed solar pressure acceleration on the vehicle was 5,000 times smaller than the thrust acceleration.

The largest computed perturbation experienced by the Transfer Vehicle was due to J_2 which was about 8 times smaller than the thrust acceleration. The large perturbation from J_2 is misleading since for orbital transfer, thrust is directed so that only eccentricity and semi-major axis change. J_2 applies a secular change in the nodal and periapsis orbital elements, but only a periodic change in the eccentricity and semi-major axis [5.1,14].

Figure 5.1 is a functional diagram of the ion thruster operational configuration [5.11]. The Xenon is stored as a liquid at -111 degrees Celsius and at a pressure slightly less than one atmosphere. The Xenon needs to be stored as a liquid since gaseous Xenon has a very low storage density of about 5 kg/m^3 as compared to 3520 kg/m^3 for liquid Xenon [5.3]. A vaporizer is employed to heat the Xenon into gaseous form before it reaches the electric induction chamber.

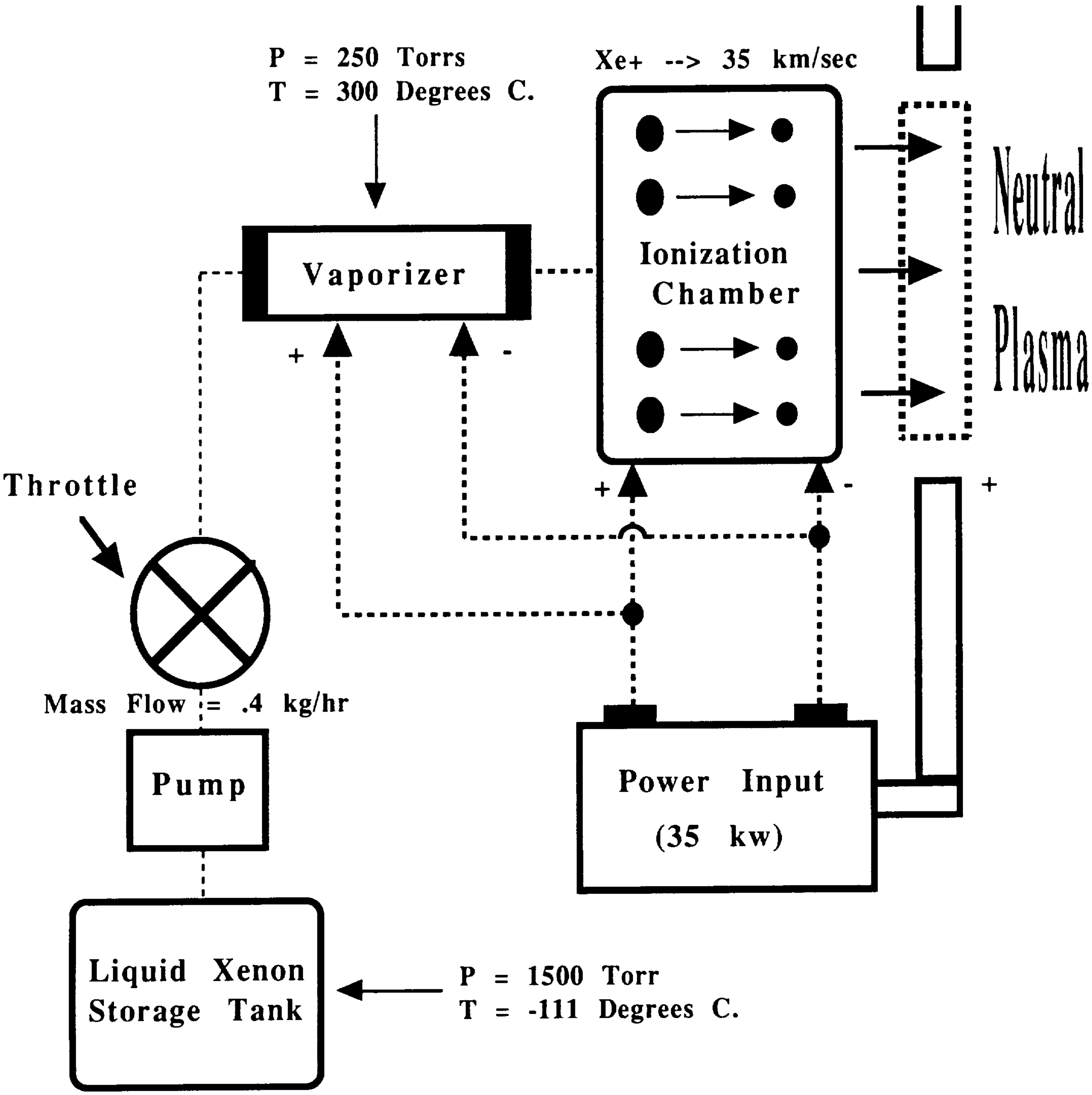


Figure 5.1 System Breakdown

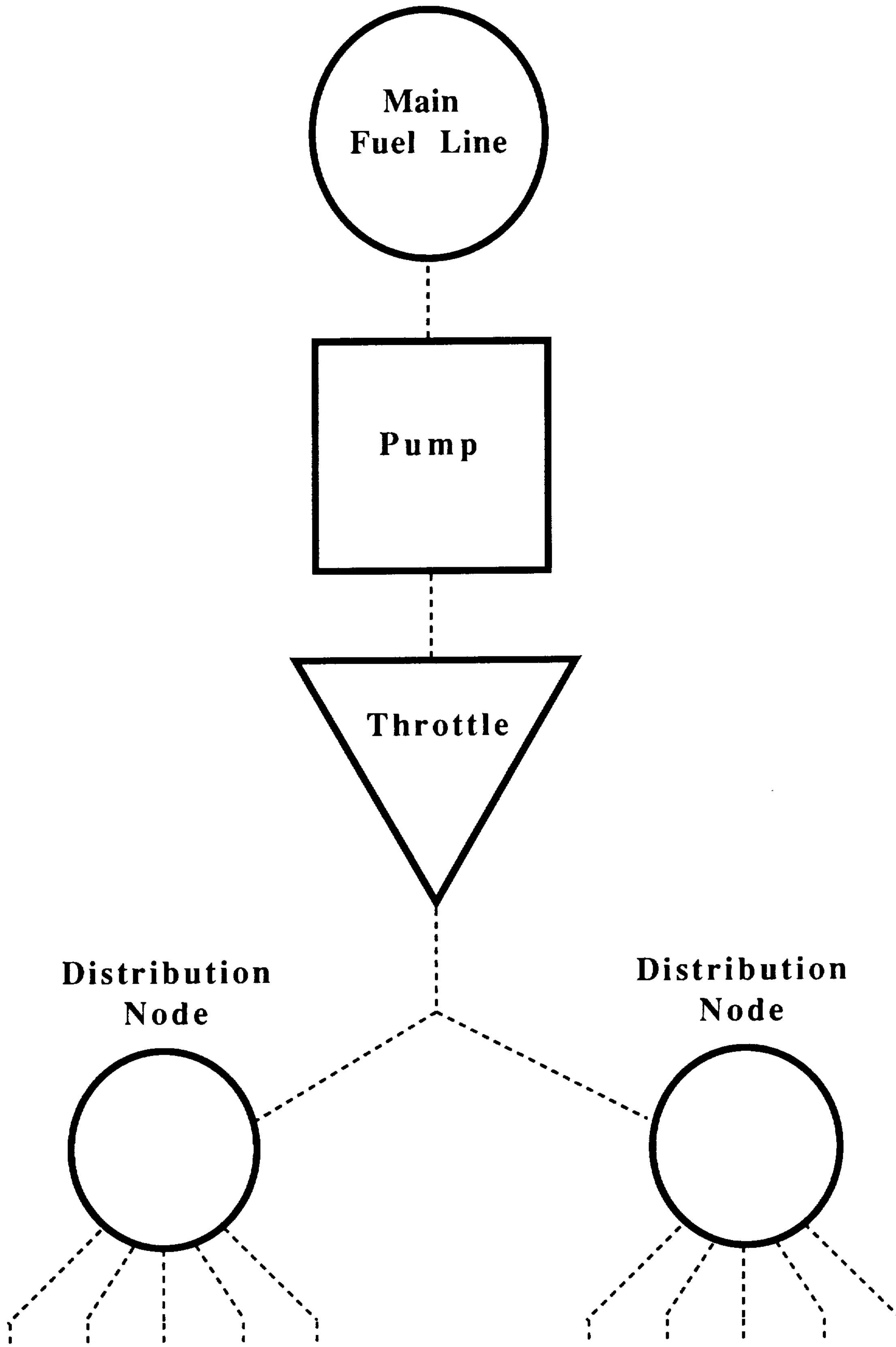


Figure 5.2 Fuel Flow Configuration

5.1.2 Netting Vehicle Propulsion

Selection of the Netting Vehicle propulsion system required a different set of propulsion criteria, as shown in Table 5.4.

Table 5.4 Netting Vehicle Propulsion Requirements

Table 5.4 Netting Vehicle Propulsion Requirements
• High Thrust Availability • Low Power Consumption • Throttle Engine Capability • Storable Propellant with highest Isp possible

A bi-propellant, Hydrazine-N_2O_4, has been selected to fuel the engine on the Propulsion Module [5.2,5]. The Primary Engine of the Space Shuttle's RCS has been chosen for the Netting Vehicle. Table 5.5 is a list of the engine/ propellant characteristics utilized in the Netting Vehicle.

Table 5.5 Primary Engine Characteristics

Table 5.5 Primary Engine Characteristics	
Propellant:	Hydrazine/N2O4
Chamber Pressure:	7.5 Atms.
Oxidizer/Fuel Ratio:	1.6 : 1
Thrust (Vacuum):	3870 N
ISP (Vacuum):	318 sec.
Restart Lifetime:	› 20,000 times
Gimbal:	6 dgs. / pitch - yaw
Width:	1.168 m.
Length:	1.958 m.

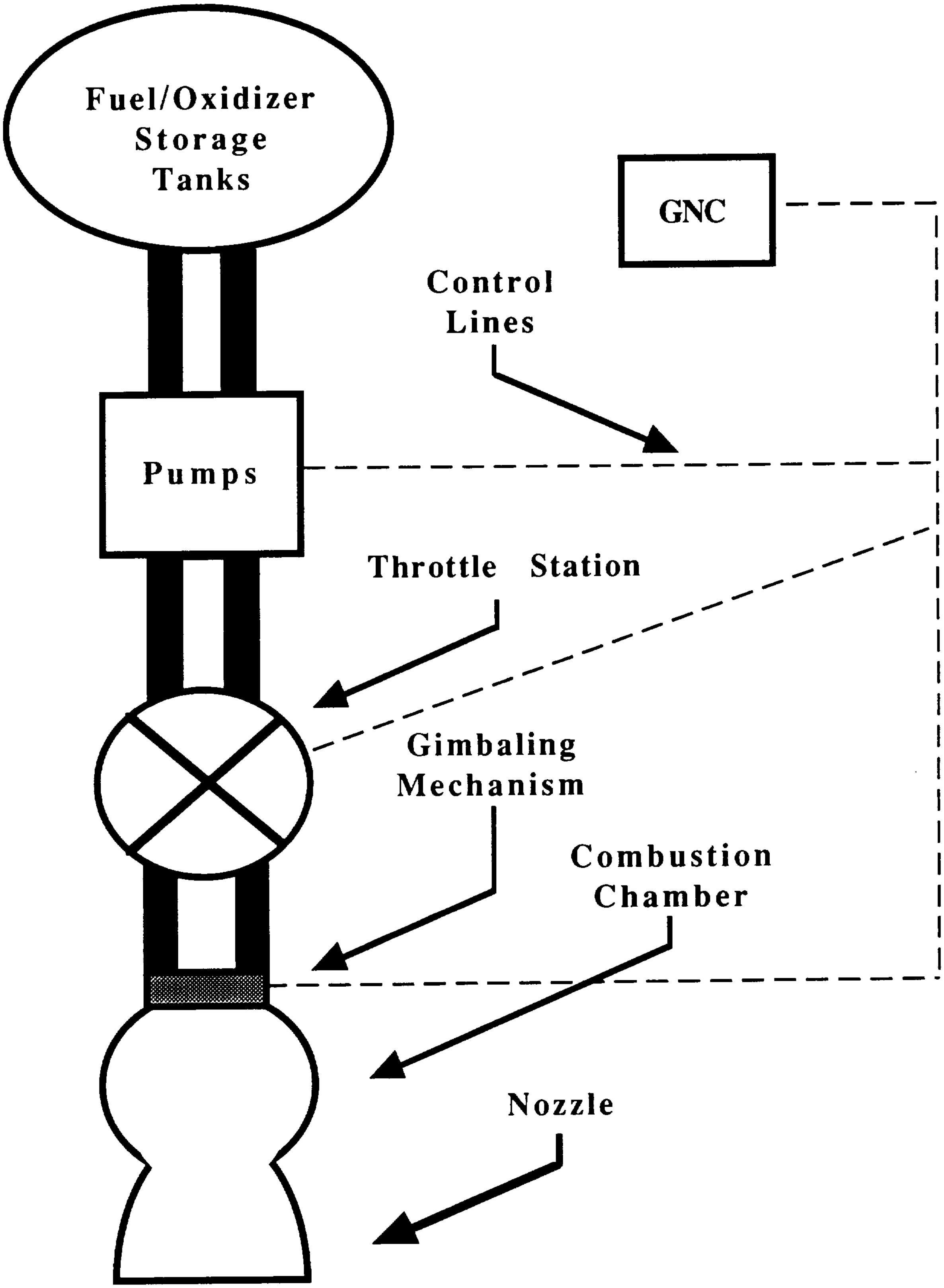


Figure 5.3 Netting Vehicle Propulsion System

Figure 5.3 illustrates the functional diagram of the Netting Vehicle's primary propulsion system with connections to GNC as well as the gimbaling mechanisms.

5.2 Power

5.2.1 Transfer Vehicle Power Supply

The main power consumer of the Transfer Vehicle system is the propulsion subsystem. Much of the power that is produced will be used by the Transfer Vehicle's electric ion thrusters. Since solar photovoltaics are a clean source of energy with no mechanical moving parts, this power source was chosen to supply the needed power. The power requirements of each subsystem on the Transfer Vehicle are shown in Table 5.6.

Table 5.6 TV Power Requirements

Communications	280 W
DPS	50 W
GNC	260 W
Propulsion	35 kW
Structures	---
Thermal	13 W
Tracking	---
Total	35.6 kW

Because the solar photovoltaic power system can not operate in the Earth's shadow, batteries will be needed to power the vehicle during this time. For one cycle, a 1.5 hr. orbit , the shadow time is 36 min. [5.14]; therefore, the battery charge time is about 58 min. To power each system during each cycle, various sections of a power system are needed. As shown in Figure 5.4, mass and power estimates of several power components are required.

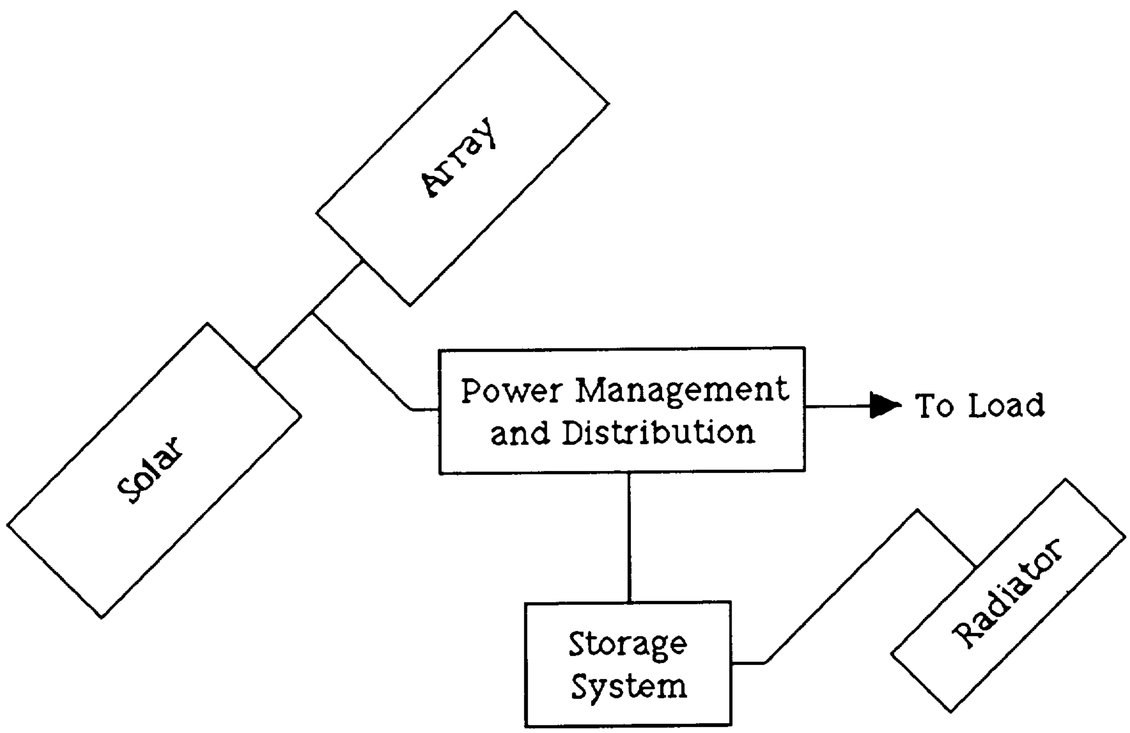


Figure 5.4 Components of a Photovoltaic Space Power System [5.15]

During sunlight times, the arrays will need to power each subsystem and charge the batteries, which is a total of about 73 kW. Using a specific power of 100 W/kg [5.16], the arrays will weigh approximately 900 kg. Note that this value includes both blanket and structural weights [5.17] and estimates of the radiator and the PMAD system. Current array technology includes solar cells that have a reduced sensitivity to radiation. By 1997 cells should be available that are completely tolerant to radiation [5.18]. The array size is needed to better define the appearance of the Transfer Vehicle and determine the mass. For the required system power, the two arrays must have an area of 76.2 m^2 with dimensions of about 5.0 m x 15.24 m, with one array on each side of the Transfer Vehicle. This array size is calculated assuming that 35% efficient GaAs solar cells will be available by the time that the vehicle is in operation.

Several batteries were acceptable for the Transfer Vehicle's power requirements, but the one with the best combination of characteristics is a new development for high-cycle life LEO, rechargeable MnO_2-H_2 cells [5.19]. This battery has a high specific energy, used in determining the mass of the battery, as shown in Table 5.7 ompared to the widely used NiH_2 battery.

Table 5.7 Battery Cell Comparison

Property	Ni-H_2	MnO_2-H_2
Specific Energy (Wh/kg)	22.2	32.6
(Wh/l)	33.1	33.5
Efficiency	82%	78%
Maximum DOD*	80%	85%
Cycle Life	10,000	25,000

*Depth of Discharge

For LEO applications the cycle life-one orbit with a charge and discharge of the battery-is necessarily large. The efficiency of this battery is comparable to the widely used $Ni\text{-}H_2$ battery. Also of interest is the large depth-of-discharge, 85%, which is the maximum amount of energy available to be drawn from the battery per cycle. Using the specific energy of the $MnO_2\text{-}H_2$ battery, a weight of about 741 kg is found with a volume of 0.721 m^3.

5.2.2 Netting Vehicle Power Supply

The subsystems of the Netting Vehicle will be powered with the same type of photovoltaic array and battery selected for the Transfer Vehicle. The power requirements of the Netting Vehicle are much smaller than the Transfer Vehicle as shown in Table 5.8.

Table 5.8 NV Power Requirements

Communications	107 W
DPS	50 W
GNC	260 W
Propulsion	12 W
Structures	---
Thermal	13 W
Tracking	---
Total	561 W

Similar calculations were made to find the necessary power to charge the battery and run each subsystem during sun times. The $MnO_2\text{-}H_2$ battery weighs 13 kg and has a volume of 0.01265 m^3. The array weight is calculated as 14 kg. The required area of the array is 2.4 m^2 which is small enough to mount the solar cells directly on the propulsion module of the Netting Vehicle. In this way the solar arrays will be less exposed to debris impacts.

5.3 Thermal Subsystem

5.3.1 Transfer Vehicle

The Thermal subsystem for the TV requires both active and passive cooling networks. The passive system consists of radiation paint and heat dissipation plates. External equipment, such as the antennas, are mounted on these plates and the excess heat is dissipated through them. The inner environment is controlled via an active cooling system, which circulates Freon through the inner volume to maintain the temperature. Pumps and heat exchangers are used to perform the circulation. Table 5.9 contains the weight, volume, and power requirements for the TV thermal subsystem.

Table 5.9 - Thermal Subsystem Characteristics

Characteristic	NM	PM	TV
Weight (kg)	436.0	235.0	303.0
Power (W)	7.0	5.0	13.0
Volume (m^3)	3.95	.85	2.75

5.3.2 Netting Vehicle

The NV thermal subsystem can be divided into the Netting Module (NM) network and the Propulsion Module (PM) network. The thermal control system for the NM is responsible for cooling the outer structure and maintaining the fuel temperature requirements. The outer hull is protected from solar radiation by special paint. Thermal dissipation plates passively bleed off the excess heat generated by the externally attached instruments (OMNIs and tracking devices). The internal NM environment requires an active cooling system to maintain the temperature of the stored fuel. This thermal network uses a Freon cooling loop with a pump to circulate the fluid and heat exchangers to regulate the temperature. The PM thermal network also uses the radiation paint and thermal dissipation plates to passively control the outer hull temperature. The inner environment is regulated by an active system similar to the system used on the NM. In addition to controlling the fuel temperature, the PM network controls the guidance, navigation and control subsystem, the data processing subsystem, and the communication computer temperatures. However, the PM network is not as extensive as the NM network because less fuel is stored on the PM as compared to the NM. Table 5.9 contains the weight, volume and power characteristics of the Netting Module and Propulsion Module thermal subsystems.

5.4 Communications

5.4.1 Subsystem Requirements

The design of the vehicle and the operational sequence levy certain requirements on the communications subsystem. The separation of the Netting Vehicle (NV) and the Transfer Vehicle (TV) require that a communications link be established between the two to provide transfer of data. In addition, the remote command center needs to be able to control the NV via another link. The communications link developed for the debris removal system must be compatible with the STS Orbiters because the resupply sequence will be conducted with these vehicles. During resupply it may be necessary for command of the TV to be handed over to the STS. Thus, the communications subsystem on-board the TV must be capable of communicating with the STS Payload Interrogator (PI). Also, the TV must be able to communicate with the command center. Nominal operations will utilized the telecommunications satellite, TDRSS, to relay the signal to the center. This command center will be located on the ground or in the Space Station Freedom (SSF). In contingency operations the TV must be capable of a direct communications link with the command center or an appropriate vehicle (STS or SSF). Although this capability will be limited by the orbital positions of the two endpoints, given enough time a direct link could eventually be established. Based on these three general requirements and types of data required in each case, the following specific requirements have been developed for the communications subsystem.

Transfer Vehicle

- The TV will receive command data via TDRSS from the external control center.
- The TV will transmit data and video via TDRSS to the external control center.
- The TV will transmit command data direct to the NV.
- The TV will receive data and video from the direct from the NV.

Netting Vehicle

- The NV will receive command data direct from the TV.
- The NV will transmit data and video direct to the TV.

These requirements provide the basic subsystem outline needed by the TV and NV. From these guideline a subsystem was developed based on the subsystems designed for the Orbiters (STS), the Space Station Freedom (SSF), and the Orbital Maneuvering Vehicle (OMV).

5.4.2 Design Approach

In developing the design criteria it was necessary to examine the communications subsystems of the TDRSS satellites, the OMV, the STS and the SSF. These programs provided compatibility guidelines and sample communications subsystems that aided in the design of the TV and NV subsystems.

The TDRSS satellite is capable of receiving and transmitting digital and analog data, audio and video over three frequency bands: S-Band (1.7 to 2.3 GHz), C- Band (4 to 6 GHz), and Ku-Band (12 to 14 GHz) [5.20,559]. Because of this operational range, the link between the TV and the TDRSS satellite is limited to these three bands.

The STS orbiter communicates in either the S-Band, Ku-Band, and in Ultrahigh Frequency band (UHF). The S-Band system includes a phase modulation (PM) system and a Frequency Modulation (FM) system. The S-band PM system is used to communicate from the Orbiter to the ground via TDRSS or ground stations. The S-band FM system can only transmit information directly to the ground stations in contingency operations or during Department of Defense (DOD) missions. The S-Band system is the means the Orbiter communicates with detached payloads. The primary communications system for the Orbiter is the Ku-Band system. This system transmits and receives information to the ground via the TDRSS satellite. Because the TDRSS has a problem locking on to the narrow beam of the Ku-Band signal, the S-Band is used to establish antenna lock with the TDRSS and then the link is handed over to the Ku-Band system. The UHF system is the means the Orbiter communicates with the EVA astronauts. This system is a voice link only [5.20,573-598].

Based on this system definition it is clear the Orbiter would communicate directly with the TV using the S-Band FM link. Therefore, to support contingency operations, the TV must be able to receive an S-Band signal from the Orbiter. Additionally, the capability must exist to command the NV directly from either STS or SSF, should communications between the TV and the NV be lost. Thus, the Orbiter would communicate to the NV using the S-Band FM link, and the NV must be able to receive. the transmission. Any communications with the TV via TDRSS would employ the Ku-Band system.

The SSF communications system is similar to the Orbiter communications system. Ku-Band is the primary means of communication between SSF and the control center through the TDRSS. The Ku-Band system is also capable of direct communication between the SSF and a vehicle with the line of sight. Any direct communications within a proximity of 1 km will be completed using the UHF system. SSF has an S-Band capability that could be used for direct link in contingency operations. Because of the similarities between the Orbiter and the SSF communications systems, the TV and NV capabilities outline in the STS section remain unchanged.

As a possible sample design for the TV and NV, the OMV communications systems was examined. The OMV will communicate to the TDRSS satellites,the SSF, the Orbiter, the Deep Space Network (DSN) and the Ground Spaceflight Tracking and Data Network (GSTDN) via and S-Band RF link [5.21,21]. These compatibility requirements influenced the requirements for the TV system. Like the OMV, the TV will be capable of communicating with the Orbiter, TDRSS, and SSF. Because the TV will not be travelling out of Earth orbit there is no need to communicate with DSN. Additionally, because GSTDN is being phased out by NASA in favor of the TDRSS constellations, this requirement was also unnecessary.

5.4.3 Subsystem Design

From an evaluation of these designs and the resulting system requirements, the communications subsystem design shown in Figure 5.5 was developed. The TV will communicate through TDRSS using a Ku-Band system as its primary method. An S-Band link will be employed to establish signal lock with the TDRSS and to serve as a backup system should the Ku-Band system fail. The S-Band capability is also needed on the TV to communicate with the STS and the SSF. The TV will communicate with the NV via a V-Band (46 to 56 GHz) system. V- Band was chosen over an S-Band or Ku-Band system to prevent interference in the signals due to the crowded and overused band. This choice was weighed against the addition of another antenna and found valid. Again the S-Band will be used to acquire signal lock and then the V-Band will take over. The S-Band system will then become the backup communications link between the TV and the NV.

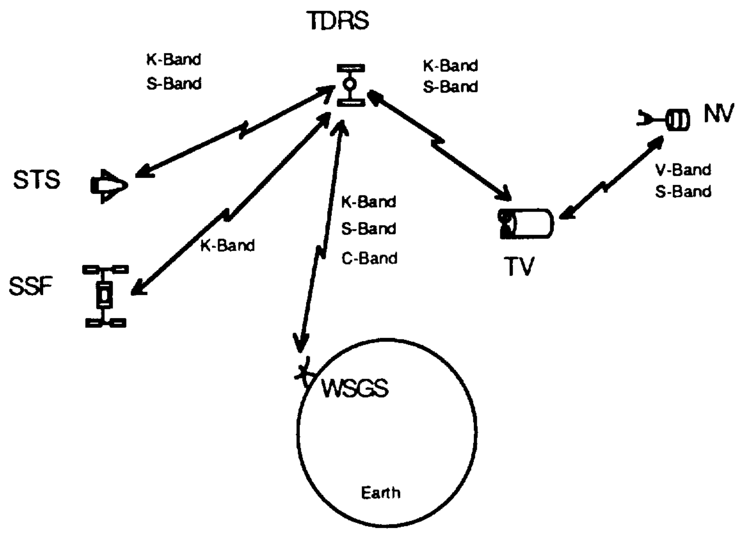


Figure 5.5 Communications Subsystem for the TV and NV

The TV and the NV will be required to receive command uplink from the external command center. The NV will transmit both data and video from its internal computers and external cameras back to the TV. The TV will then transmit the data and video to the control center.

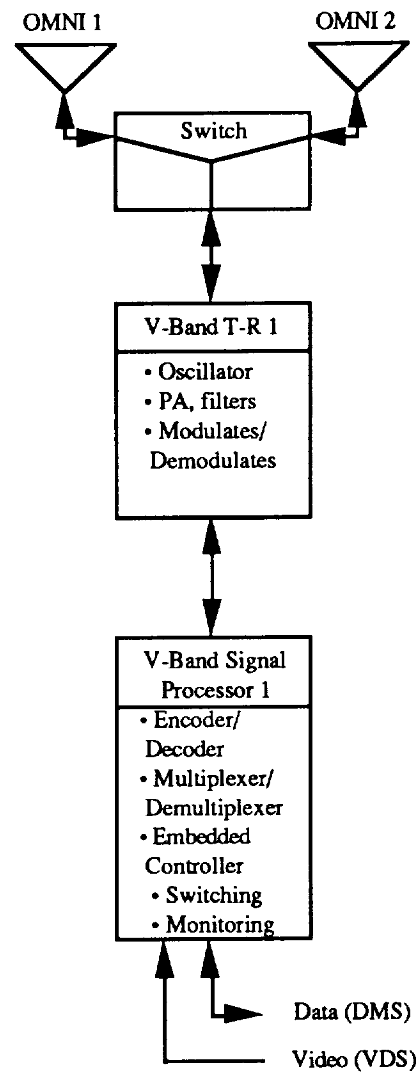
5.4.4 Netting Vehicle

The Netting Vehicle (NV) will have both V-Band and S-Band communications capability. The V-Band network will be the primary means of communication between the NV and the Transfer Vehicle (TV). The S-Band network will be used in contingency operations. The NV communications subsystem will be responsible for transmitting both data and video to the TV and receiving command data from the TV.

5.4.4.1 V-Band Network

A schematic of the V-Band network is shown in Figure 5.6. Two low power, low gain, hemispherical omnidirectional antennas (OMNIs) will receive and transmit the NV signals. These antennas provide sufficient gain for proximity zone operations between the NV and the TV. The OMNIs also have a wide TX/RX range; thus, with two OMNI antennas mounted on opposite sides of the NV, communications will be virtually independent of attitude. Additionally, OMNIs have a smaller surface area than the parabolic antennas, which is desirable for NV operations in high density debris zones.

The remaining components of the V-Band network are the Switch, the Transmitter-Receiver, and the Signal Processor. The V-Band Switch is an electrically driven switch that alternates between the two OMNIs when commanded by the V-Band Signal Processor (VSP). The V-Band Transmitter-Receiver (VT-R) performs the modulation and demodulation of the inbound/outbound signals. The VT-R contains the crystal oscillators that regulate the carrier frequency, the power amplifier that steps-up or steps-down the signal, and the filters. The "brain" of the V-Band communications network is the VSP. This unit is receives input data from the Video Distribution Subsystem (VDS) and the Data Management Subsystem (DMS). The data is first encoded and multiplexed, then sent to the VT-R for transmission. Additionally, the VSP receives signal data



V-Band Communications Network

Figure 5.6 V-Band Communications Network for the NV

from the VT-R, demultiplexes and decodes the data, then, relays it to the DMS for processing. The VSP also controls and monitors the VT-R and the Switch. All switching commands are initiated by the VSP software upon receipt of the command from the DMS. Fault detection is also performed by the VSP. Upon detection of a fault, the VSP software notifies the DMS. Table 5.10 shows the power, weight, and volume characteristics of the V-Band communications network.

The V-Band network is the primary means of communication between the NV and the TV. The network is single fault tolerant: a failure of one of the units will disable the entire string. Upon failure of the V-Band network the NV can utilize the S-Band network for communication.

5.4.4.2 S-Band Network

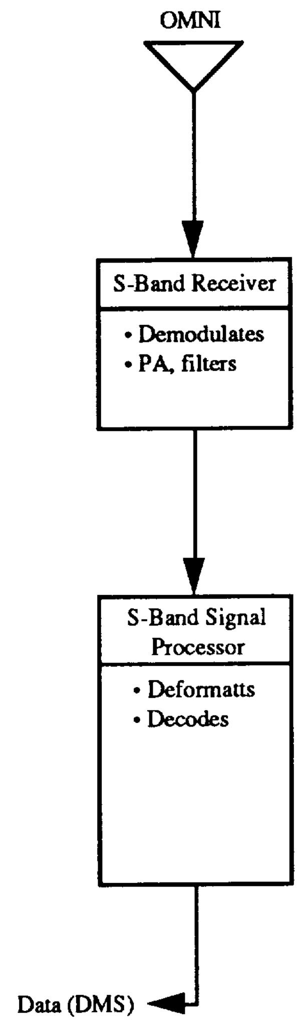
The S-band network has receive only capability for contingency operations. A diagram of the S-Band communications network is shown in Figure 5.7.

Again, an OMNI antenna was chosen to provide maximum coverage. Only one S-Band OMNI will be located on the NV, thus, inhibiting communications to certain attitudes. Incoming command data is sent to the S-Band Receiver for demodulation. The data is then sent to the S-Band Signal Processor (SSP) for demultiplexing and decoding. The resulting command data is shipped to the DMS for processing.

The S-Band network is the secondary means of communication for the NV. For contingency operations only command data can be received by the NV. Most likely, this data will direct the NV to return to the TV from repair. For this reason the S-Band network is significantly scaled down when compared to the V-Band network.

Table 5.10 NV Communications Subsystem Characteristics

ORU	Power (W)	Weight (kg)	Volume (m^3)
V-Switch	2 (SS) 25 (Switching)	.907	.0018
V T-R	50	15.87	.0183
VSP	30	8.16	.0117
S Receiver	35	2.27	.0018
SSP	30	15.87	.0117
OMNI	N/A	20.41	.0006
TOTALS	**107 (V-Band)** **65 (S-Band)**	**140**	**.0457**



S-Band Communications Network

Figure 5.7 S-Band Communications Network for the NV

5.4.5 Transfer Vehicle

The Transfer Vehicle (TV) has three communications networks on-board: Ku-Band, V-Band, and S-Band. Communications between the TV and the Control Center (CC) is accomplished by a Ku-Band link via the TDRSS communications satellite. The V-Band network is used as the primary means of communications between the TV and the NV. The S-Band network provides the secondary link between the

TV and the NV and serves as the backup network for the Ku-Band network. Additionally, the S-Band is used to acquire TDRSS for the Ku-Band network. Estimates of the power, weight and volume of the TV communications subsystem are provided in Table 5.11.

5.4.5.1 Ku-Band Network

The Ku-Band network provides the communications link between the TV and the CC via TDRSS. A diagram of the Ku-Band network is shown in Figure 5.8. Because of the large distances the signal must travel, 3 foot diameter, high gain, parabolic antennas were chosen for the Ku-Band network. These antennas are directional and have pointing capability through a two axis gimballing mechanism.

The antennas are controlled by an Antenna Controller (ACON), which regulates the motion of the antennas. The ACON is connected to each of the antennas by an electrically driven switch. All gimballing commands are issued via the ACON upon request from the Ku-Band Signal Processor (KSP). The ACON also monitors the gimbals for failures and performs a small degree of fault detection on the switch, gimbal motors, and itself.

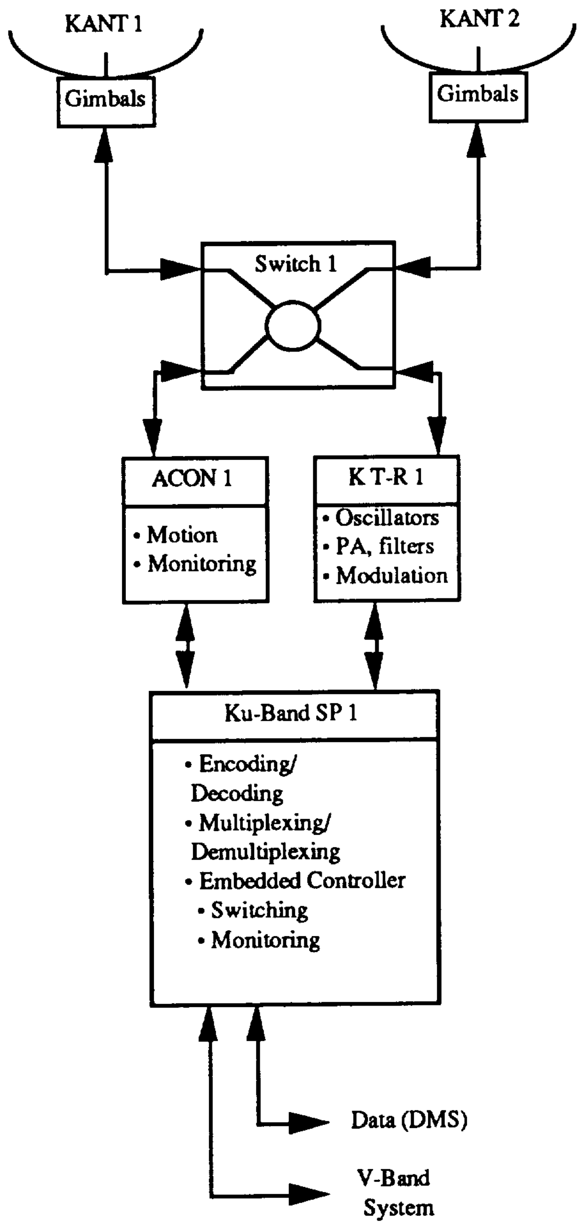
The remaining components, the Ku-Band Transmitter-Receiver (KT-R) and the KSP, are functionally identical to the ST-R and the SSP discussed for the NV. However, these units will be specifically designed for the Ku-Band frequency range. The KSP receives inputs from and outputs data to the TV DMS and the V-Band network.

Table 5.11 TV Communications Subsystem Characteristics

ORU	Power (W)	Weight (kg)	Volume (m^3)
Switch	2 (SS) 25 (Switching)	.907	.0036
V T-R	120	15.87	.0184
VSP	30	8.16	.0117
ST-R	35	2.27	.0018
SSP	30	15.87	.0117
OMNI	N/A	20.4	.0006
KANT	15	27.2	1.601
ACON	30	15.87	.0082
KT-R	120	15.87	.0184
KSP	30	15.87	.0117
TOTALS	**107 (V-Band)** **65 (S-Band)** **220 (Ku-Band)**	**154.2**	**1.687**

5.4.5.2 V-Band Network

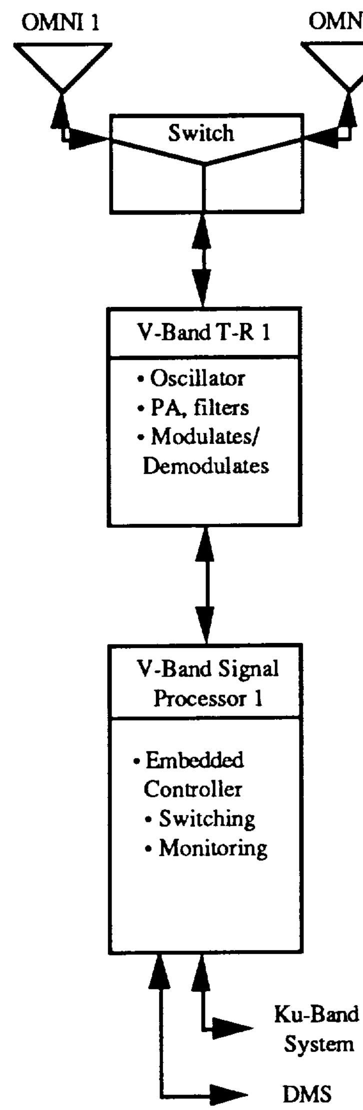
The V-Band communications network on-board the TV is similar to the network used on the NV. A schematic of the V-Band network is shown in Figure 5.9. The only component that differs from the NV components is the VSP. Because the TV operates as a



Ku-Band Communications Network

Figure 5.8 Ku-Band Communications Network for the TV

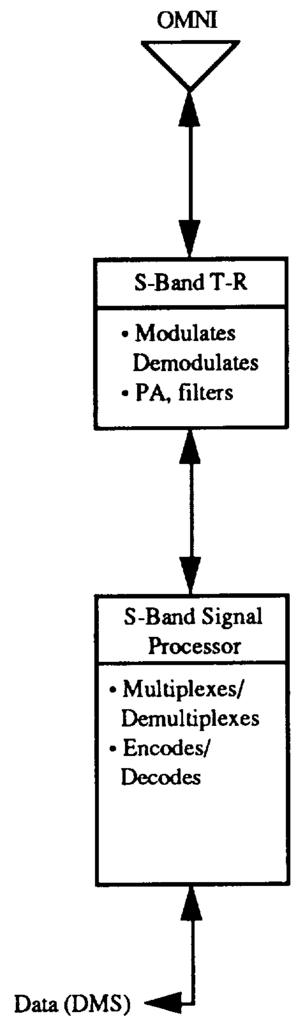
relay station between the NV and the CC, there is no need to decode and demultiplex the incoming data stream from the NV. The data is simply shipped through the V-Band network to the Ku-Band network and on to the CC. Similarly, outbound command data from the Ku-



V-Band Communications Network

Figure 5.9 V-Band Communications Network for the TV

Band network is transmitted through the VSP with no encoding or multiplexing required. Essentially, the VSP only provides switching and monitoring functions.



S-Band Communications Network

Figure 5.10 S-Band Communications Network for the TV

5.4.5.3 S-Band Network

The S-Band network for the TV is more extensive than the S-Band network used on the NV. Figure 5.10 shows a schematic of the S-Band network for the TV. The TV S-Band has the capability to receive and transmit signals. The transmit capability was necessary to perform the .S acquisition of signal function for the Ku-Band network. After acquisition of signal, the S-Band network will hand over to the Ku-Band network for data transfer.

The components of the S-Band network consist of a single high power, low gain hemispherical OMNI antenna, an S-Band Transmitter-Receiver (ST-R), and an S-Band Signal Processor (SSP). The OMNI antenna will need higher power than the NV OMNIs because of the distance the signal must travel. The ST-R modulates and demodulates the S-Band signal. The SSP performs the encoding , decoding, multiplexing, and demultiplexing of the in-bound and out-bound signal. The S-Band network receive input and outputs data to the DMS subsystem.

5.5 Data Processing

Both the TV and the NV will have isolated Data Processing Subsystems (DPSs). These subsystems shall support communication, GNC, tracking, and control and monitoring of the vehicles. Any instrumentation data will be processed in this subsystem. All formatting and preparing of the data to be transmitted to the Control Center will be handled by the DPS. Basically, the DPS constitutes the "brain" of the two vehicles.

The DPS for each vehicle will consist of 2 redundant computers loaded with identical software. The computers will be state-of-the-art to provide maximum processing capability. All processing of commands and data will be conducted by the DPS. Characteristics of the DPS subsystem are provided in Table 5.12.

Table 5.12 Characteristics of the DPS for the NV and TV

ORU		Weight (kg)	Power (W)	Volume (in^3)
NV	Computers (2)	35	100	1440
TV	Computers (2)	35	100	1440

5.6 Tracking and Detection Subsystem

The Debris Removal System must be able to accurately locate and track the orbital debris particles before it can remove them. The current tracking system employed by NORAD allows particles greater than 10 cm in diameter to be tracked in LEO [5.22]. This presents a problem, because some of our target debris, those less than 10 cm, cannot be tracked from Earth. For the DRS it was assumed that in the near future there will be a ground based tracking system that can track particles as small as 1 cm in LEO. SPECS, Inc. believes this assumption is reasonable because new ground tracking systems that meet this requirement are under consideration and are within the ability of current technology. The biggest change that will be performed is upgrading the computers that will keep track of the additional tens of thousands of particles[5.23]

The Netting Vehicle will employ a combination active/passive system to track orbital debris in order to estimate a rendezvous. Established ground based radar will detect a breakup to guide the deployment of the DRS to the target trajectory. Once the Transfer

Vehicle is established 50-100 kilometers below the debris torus (i.e.. semi major axis 50 km less), the Netting Vehicle will detach from the Transfer Vehicle and enter the debris torus. Then, using low power passive sensors to track the piece of debris, the Netting Vehicle will compute a rendezvous trajectory. After the vehicle is approaching the debris, the active sensors would determine the size and the distance to the debris. Capture would then be possible with the enhanced tracking and sizing data.

For our tracking system, we have ruled out the possibility of using an active radar tracking system because of their narrow field of view and their very large power requirements. The tracking systems that were considered for our Netting Vehicle were an infrared, an optical or a LADAR (LAser Detection and Ranging) tracking system. Experiments performed at MIT's Lincoln Laboratory have shown that optical sensors, using a small telescope and a low light video detector, can detect particles as small as 1 cm at a distance of 500 km [5.22]. Another system considered is the infrared tracking sensor. This sensor tracks the debris particles by detecting the IR radiation given off by the particle due to solar heating. This system is able to detect particles that are 2 cm in diameter at a distance of 1900 km. The system then detects how the particles are moving in the field of view to determine their location and the direction of their velocity. This system is considered to be practical for tracking debris in space and testing of an IR collision avoidance system for the Space Station is scheduled to be conducted on the Space Shuttle in 1991 [5.24]. Finally, the third tracking system is the LADAR system. This system uses pulses of laser light to detect the debris and accurately measure its size and distance from the spacecraft. The LADAR system is able to resolve the size of a particle to a microradian at a range accuracy of 0.1 m at 25 km [5.25-5.28].

To satisfy our requirements for the detection system on our Netting Vehicle, SPECS, Inc. has decided to use a combination tracking system. The tracking system will use both a passive IR tracking sensor and an active LADAR system as shown in Figure 5.11. The confidence by NASA in the IR collision avoidance system and the wide field of view it provides led us to select this system for our Netting Vehicle. This system will be used initially to locate the debris and maneuver the Netting Vehicle toward the debris. Once the vehicle is approaching the debris, the LADAR system will be used to determine the exact size of the debris and the distance to the debris.

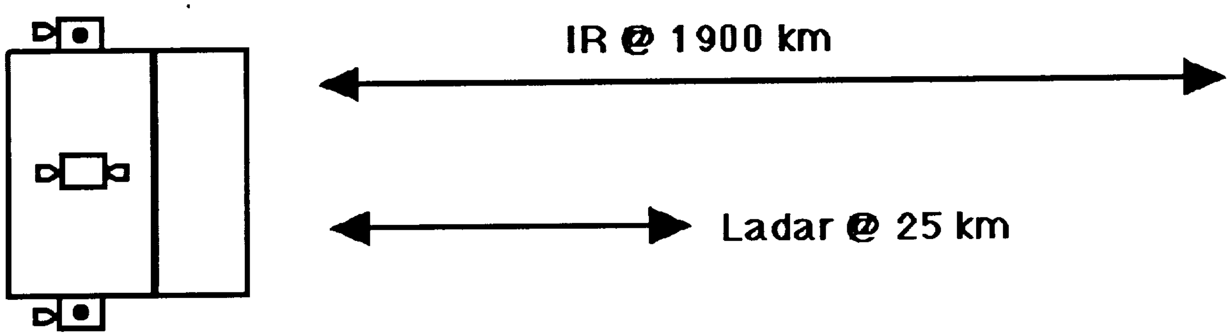


Figure 5.11 Tracking Range Characteristics

With this data, the Netting Vehicle will continue to close in on the debris and will fire a net and capture the debris once it is in the range of the net.

The reason that these two systems were chosen was because they each offset the other's weaknesses. The main disadvantage of the IR system is that it is unable to determine the size of the debris piece, however it is able to detect the particle far away. Another disadvantage is that it is difficult to accurately determine the range of the particle with the IR system. On the other hand, the LADAR system is able to accurately determine the size and the range of the debris piece once it is within 25 kilometers of the Netting Vehicle. However, it is unable to detect the particles at large distances. Therefore, we have chosen these two systems so that we are able to detect the particles at long ranges and measure the size and distance to great accuracy once the Netting Vehicle has closed in on the particle.

The tracking system will require about 60 Watts of power for the LADAR system and 10 Watts for the IR system. The systems will be mounted on the propulsion module of the Netting Vehicle. The LADAR system will weigh about 30 kg and the IR system will weight about 100 kg. Most of the IR sensor's weight is due to the large optical light collector used to focus the IR radiation on the sensors. This gives a total system weight of 130 km and a power consumption of 70 Watts. These systems also require a substantial amount of computational power. The data from the sensors will have to be interpreted by the on-board computers so that the proper maneuvers can be performed to rendezvous with the debris particle. Finally, the use of the on-board computers to determine the particle location instead of ground computers will reduce the amount of data that has to be transmitted between the vehicle and ground.

5.7 Guidance, Navigation, and Control

5.7.1 Guidance and Navigation

Guidance and navigation may be conveniently broken into two main tasks. The first consists of determining the position of the center of mass of the spacecraft, while the second consists of determining the spacecraft's inertial orientation, or attitude. In both tasks, the quantities of interest, such as position, velocity, angular velocities, and angular measurements, are generally determined using some form of "inertial measurement unit" (or IMU).

The IMUs for both the transfer vehicle and the netting vehicle were designed using the IMUs aboard the shuttle as a general guideline, since this system has proven itself reliable in the past and is not overly restrictive in terms of power and weight. Each IMU consists of three orthogonal accelerometers and two two-degree of freedom gyroscopes. Used in conjunction with an intergrating algorithm, the IMUs provide sufficient information to determine the inertial quantities of interest mentioned above [5.29, 196]. Three IMUs are utilized in each vehicle, as in the shuttle, to provide redundant information. Furthermore, using power requirements for a typical rate-gyro, each IMU was estimated to consume approximately 47 watts and to have a mass of 10 kilograms [5.29, 200].

Due to measurement drift and basic inaccuracies, the IMUs must periodically be updated by other sources. In determining the position of the center of mass of a spacecraft, either global positioning satellites (GPS) or the Tracking and Data Relay Satellite

System (TDRSS) may be utilized. In either case, onboard computers can be used to analyze the time delays and the doppler shifts of radio signals sent to the spacecraft from a ground station through a TDRS. Given a sufficient number of time delay and doppler shift measurements (i.e., range and range-rate information), and given dynamic models for both the spacecraft and the TDRS, the position and velocity of the spacecraft's center of mass may be calculated. Of course, it is typically necessary to provide error modeling, in addition to dynamic modeling, to filter out random noise. The concept of using TDRSS for the on-board tracking of near-earth satellites is extensively discussed by Shank in his article "Automated Orbit Determination Using Tracking and Data Relay Satellite (TDRS) Data" [5.30, 1-21].

The decision was made to utilize TDRSS in navigation because TDRSS is also used for the design's communication purposes. Further, onboard computers are anticipated to handle much of the navigation work to minimize ground support. Moreover, TDRSS is capable of providing communications and tracking for over 85% of the orbits under 5000 km in altitude [5.29, 288].

In addition to the center of mass position information, the attitude information provided by the IMUs must also be periodically updated. This updating may be accomplished by using appropriate sensors (described below) and an on-board computer. If the position vector of the center of mass of a spacecraft is known, it turns out that knowing the unit vectors to two non-collinear bodies (the Earth and Sun at an appropriate time, for example) uniquely determines the attitude of the spacecraft [5.31, 140]. These unit vectors may, in turn, be obtained from Earth, sun, or star sensors.

Sun sensors have the advantage that, for near-Earth orbits, the inertial displacement vector from the spacecraft to the sun is virtually constant over several orbit revolutions, thereby providing a direction that is fixed in inertial space for a time duration of interest (for example, the time it takes to perform an angular momentum change) [5.29, 155]. A further advantage of sun sensors is that, because of the sun's brightness, they tend to be relatively inexpensive, reliable, and consume small amounts of power [5.29, 155].

Earth sensors generally consist of a scanning mechanism, an optical system, a radiance detector, and signal processing electronics. The principal drawback to Earth sensors is that significant uncertainties can arise due to the presence of the atmosphere on the horizon [5.29, 167]. However, for near-Earth applications, they have the advantage that the Earth is always in view and cannot be confused with other luminous sources.

Star sensors are generally the most accurate sensors, but the drawback with these sensors is that they tend to be heavier, more expensive, and consume more power than other sensors. They also require preprocessed position data on the star being tracked as well as extensive star maps and computer software for data reduction [5.29, 186].

Magnetometers are used to detect the direction of the Earth's magnetic field in body-fixed coordinates. Then, knowledge of the Earth's magnetic field and the position of the center of mass gives attitude information. Magnetometers have the advantage of being lightweight, require only a small amount of power, and can operate through a wide range of temperatures. However, they often cannot be used with confidence in determining the attitude of the spacecraft because the Earth's magnetic field is poorly known in many regions [5.29, 180-181].

The criteria for choosing the sensors was, in decreasing order of importance. accuracy, power, weight, and expense. The importance placed on the accuracy was due to the extensive docking and debris capture anticipated. Further, as a result of the accuracy requirement, magnetometers were not used. Each vehicle, however, makes use of all the other three sensors. Even though it requires only two sensors operating at one time to theoretically determine the spacecraft's orientation, three will be used for redundancy and for use while in shadow. Also, four star and digital sun sensors will be aboard so as to encompass a large field of view, even though only one of each will operate at any given time. The weight of these sensors and the power they consume (per vehicle) were estimated to be, respectively, 25 kg and 20W [5.29, 177-190].

5.7.2 Vehicle Control

The basic control mechanisms of both vehicles will be control moment gyros (CMGs) and RCS thrusters. The primary disadvantage associated with CMGs is that they tend to be large and consume considerable power. For instance, some of the larger CMG systems weigh in excess of 600 lbs [5.29, 201]. Another disadvantage of CMGs is that undesirable momentum configurations invariably arise during the process of cancelling secular disturbance torques; as a result, CMGs are usually accompanied by an RCS system for periodic momentum dumping [5.29, 200]. However, CMGs offer the capability for fine tune attitude ajustments, as required in docking and debris retrieval, and they will not blow the debris away as an RCS might. Equally important, if an RCS was used exclusively, the amount of fuel required by the large transfer vehicle over many months and possibly years would definitely limit the mission. For example, shuttle missions, which are relatively short, can require over 3600 kg of fuel and oxidizer for its RCS [5.32, 297]. Lastly, based on representative CMG systems, the CMGs for both the transfer and netting vehicles were estimated to have a mass of 175 kg and to consume 100 W of power [5.29, 200].

A RCS is necessary to supplement the CMGs and provide small adjustments in the position of the center of mass. The dry weight of the RCS of the transfer vehicle was roughly estimated using the dry weight of the RCS of the Orbital Maneuvering Vehicle (OMV) as a guide, because both vehicles perform similar tasks and are of roughly the same mass. The dry weight RCS estimates for the netting vehicle were obtained by scaling the dry weight RCS estimates of the transfer vehicle down to 25%. The OMV RCS consists of 28 hydrazine thrusters weighing 5.45 kg apeice and with a thrust of 15 lbs [5.33, 30, Appendix 1].

The RCS fuel requirements were difficult to estimate because, as of now, it is not known exactly how large a role the RCS will play in relation to the CMGs. It is anticipated that with the CMGs providing virtually all the attitude control and with the possible aid of the ion engines for fine-tuning the position of the center of mass, the role of the RCS will be minimized. For calculation purposes, upper limits for the combined fuel and oxidizer masses for the Transfer and Netting Vehicles were speculated to be 1500 kg and 400 kg, respectively. A summary of each component of the GNC subsystem with its corresponding weight, power, and volume estimates is given in the Table 5.13. (The volume of the RCS systems include fuel volume estimates based on the bulk density of hydrazine and nitrous oxide being 1200 kg/m^3.)

Table 5.13 NV and TV Weight and Power for GNC

	Mass (kg)	Power (W)	Volume (m^3)
Sensors	25	20	.5
IMUs	30	140	1.0
CMGs	175	100	1.0
RCS (dry)	TV-165, NV-41	*****	TV-4.6, NV-1.5 *
Total TV (dry)	395	260	7.1
Total NV (dry)	271	260	4.0

* Includes fuel volume estimates.

In addition to the active control systems mentioned above, the moments of inertia and the nominal orientation of the Transfer Vehicle will be designed for gravity gradient stabilization. This is done because, while on the transfer orbit, the vehicle must spin at approximately one rev per orbital period so that the ion engines can be pointed in the appropriate direction at all times. (The ion engines do not provide ideal delta v's; but rather, operate continuously throughout the transfer.) A spin rate of one revolution per orbital period is ideal for gravity gradient stabilization. The criteria for this stabilization is that the pitch moment of inertia should be greater than the roll moment of inertia which in turn should be greater than the yaw moment of inertia [5.31, 203]. (This is assuming that the principle moments of inertia are alligned about these three axis.)

Finally, the idea of passive control of the Transfer Vehicle using a dual spinner was rejected due to the size and non-axisymmetric nature of the Transfer Vehicle. The theory available on stability requirements for dual spinners deals largely with axisymmetric bodies [5.31, 175-188]. Further, the size of the Transfer Vehicle would correspondingly require a large spinner, and this extra mass would restrict the design in terms of getting the vehicles into space and in terms of the extra fuel required for the transfer orbit. Finally, the structure of the Transfer Vehicle was designed mainly with the idea that the ion engines working together would only provide about four Newtons of thrust. Therefore, the structure as a whole is quite light and correspondingly very flexible. It would not withstand the stresses induced by a huge, fast-spinning mass, and even if it could, the resulting vibrations would be unacceptable.

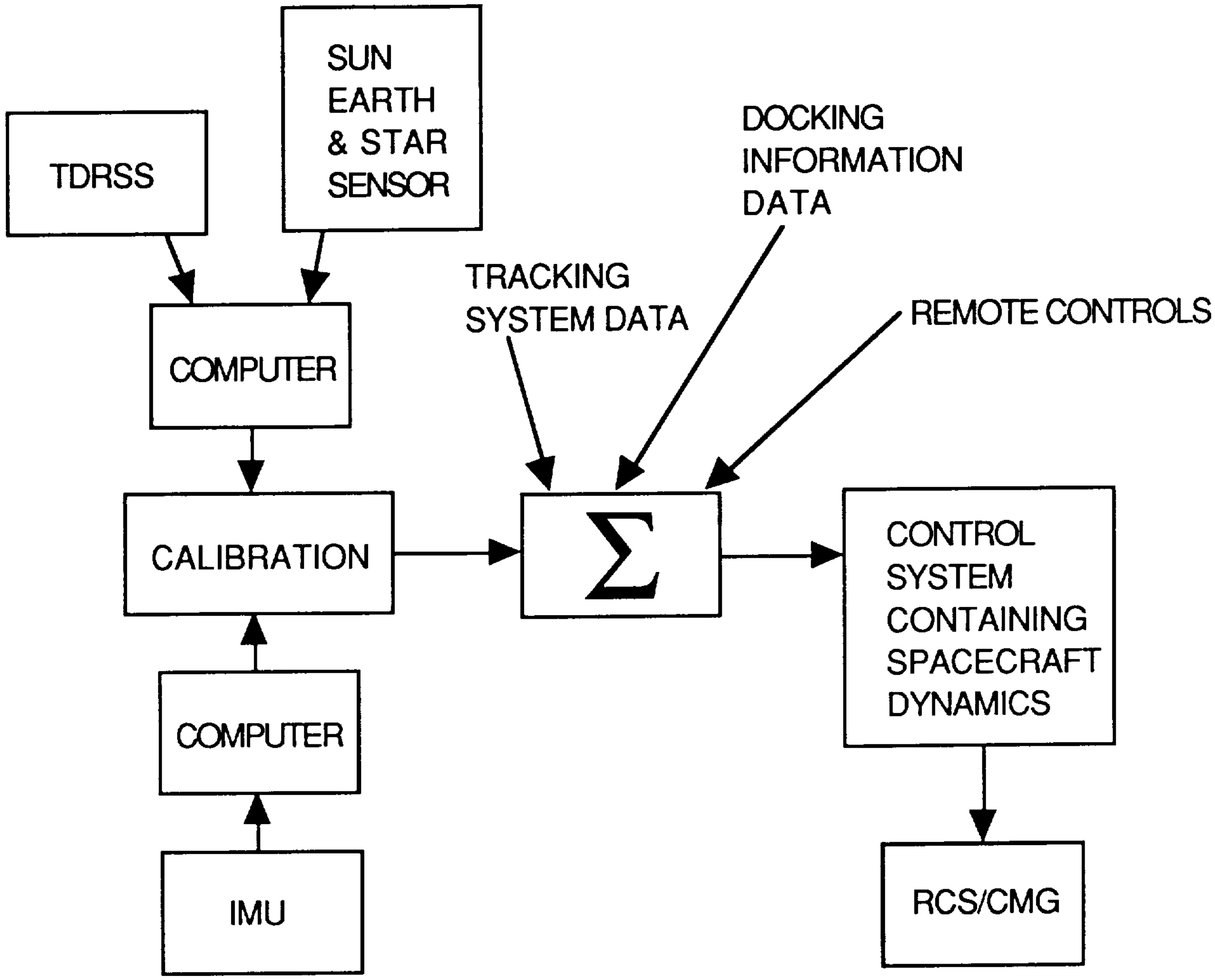


Figure 5.12 GNC Integration System

5.8 Netting Subsystem

The netting subsystem is composed of four parts: the nets, the launching system, the retrieval system, and the storage volume.

The nets will be made from Kevlar, a high strength composite material. The nets will be spinning when they are launched by a simple compressed spring system, and there will be four masses on the perimater to open the net with centrifugal forces. A Kevlar net 1 meter (m) in diameter, 1 millimeter (mm) thick and with four 0.23 kg masses on the perimeter will have a mass of 1.92 kg. 2.5 kg was used to include an extra amount of mass for the launching system. A 2 meter diameter net and launching system will have a mass of 6.0 kg. A 3 meter diameter net and launching system will have a mass of 11.5 kg.

After the net has captured the debris (see Section 6.2 for more details on launching the net and capturing the debris), the net and debris will be retrieved by a tether connected between the net and the Netting Module. The tether will be wound up by a winch in the Netting Module. The netting winch should have a mass of approximately 50 kg, a volume of 0.0063 m^3, and a power requirement of 78 W (based on small automobile winch as a model). There will be only one winch per Netting Module, with a separate cable for each net. These cables will be able to be deployed, braked, and retrieved independently. The mass of the cables is expected to be no more than 16 kg (calculations based on 20 steel cables 2 mm in diameter and 100 m long).

The sum of the cross-sectional areas of the storage volumes will not exceed 75% of the area on the front face of the Netting Module in order to ensure structural rigidity. Three sizes of storage volumes were considered:

- A 20 cm diameter, 50 cm long cylinder
  Could safely hold a plate 14cm x 14cm or smaller
  Would use a 1 m diameter net

- A 40 cm diameter, 60 cm long cylinder
  Could safely hold a plate 28cm x 28cm or smaller
  Would use a 2 m diameter net

- A 90 cm diameter, 110 cm long cylinder
  Could safely hold a plate 63cm x 63cm or smaller
  Would use a 3 m diameter net

The dimensions of a plate that could safely fit in each cylinder was taken by assuming that the greatest possible length that could fit across the cylinder would be a plate with a length the size of the diameter. The length of the sides were chosen by considering the worst case: the plate could be turned so that its diagonal is being pulled across the cylinder. The sizes for safety are therefore the diameter of the cylinder divided by the square root of 2.

Furthermore, three different Netting Module configurations were examined:

- NM20 - has **75** 20cm holes
  Total Storage Volume - 0.94 m^3
  Mass of nets and launching systems - 187.5 kg

- NM20/40 - has **18** 20cm holes, **9** 40cm holes
  Total Storage Volume - 0.9 m^3
  Mass of nets and launching systems - 99 kg

- NM20/40/90 - has **12** 20cm holes, **6** 40cm holes, **1** 90cm holes
  Total Storage Volume - 1.3 m^3
  Mass of nets and launching systems - 77.5 kg

5.9 Structural Materials

The Netting Module, Propulsion Module, and Transfer Vehicle will be made of aluminum, a proven material in space flights. Composites were considered, but they were judged to be too expensive for our system. An estimate of the structural mass was made by assuming that each of the vehicles was a cylinder closed at both ends with a skin thickness of 2 centimeters. A 10% factor was added to this figure to take into account the internal support structure.

The subsystems for the Netting Module (including a fuel allocation volume) will require approximately 7.91 cubic meters (m^3) of volume each (see Table 5.14 for a summary of subsystem volume requirements for all vehicles). A 2.52 meter long cylinder with a 2 meter diameter will satisfy this requirement. The docking collar, located at the back of the Netting Module, was assumed to be a hollow cylinder 0.5 meter long with a diameter of 2 meter and a skin thickness of 4 centimeters. With these dimensions, the unloaded Netting Module will have a mass of approximately 1653 kg.

The Propulsion Module will need to have 9.163 m^3 of space. A 2.93 meter long cylinder with a 2 meter diameter will satisfy this requirement. It will provide 0.042 m^3 extra space and will have a structural mass of 1466.8 kg.

The Transfer Vehicle will need to contain 21.021 m^3 of subsystem components, so a 3 meter long cylinder with a 3 meter diameter will be used (the diameter needs to be this large to fit the ten engines inside). It will provide 0.2 m^3 extra space and will have a structural mass of 1933.9 kg.

Since this will not be a manned mission and no nuclear reactor will be on board, there will be no need for heavy radiation shielding. Radiation shielding paint will suffice to protect the computer and navigation subsystems from solar and cosmic radiation.

Table 5.14 Summary of Volume Requirements (m^3)
Vehicle

NM Configuration

Subsystem	20	20/40	20/40/90	PM	TV
Structure	******	******	******	******	******
Netting	0.95	0.91	1.31	******	******
Propulsion	******	******	******	3.450	8.200
Power	******	******	******	0.012[1]	0.721[1]
Thermal	3.95	3.95	3.95	0.850	2.750
Tracking	******	******	******	0.780[2]	*******
Comm.	******	******	******	0.047	0.086
GNC	******	******	******	4.000	7.100
DPS	******	******	******	0.024	0.024
Fuel	3.01	3.05	2.65	included in GNC	2.140
Total	7.91	7.91	7.91	9.163	21.021

1 space for batteries only

2 space for sun, star, and earth sensors only; LADAR and IR sensors are mounted on body

However, because the Netting Vehicle and the Transfer Vehicle will be in or near relatively dense concentrations of debris, debris impact shielding will be needed. We have decided to use a new, lightweight ceramic fabric called Nextel that is being manufactured by 3M [5.34]. Nextel has been tested by Johnson Space Center engineers to see if it would stop particles travelling at velocities higher than 3 km/s known as hypervelocities. A shield composed of

4 layers of Nextel and a thin aluminum plate has successfully stopped a 1 cm sphere of aluminum travelling at hypervelocities [structures.1].

The debris shield will be composed of 4 sheets of Nextel, each with a surface density of 0.123 g/cm^2 (4.92 kg/m^2) [structures.1]. The sheets will have to be spaced three inches apart and the skin of the spacecraft will take the place of the aluminum plate (the plate in the NASA test was 80 mil, or 0.203 cm thick). This shield should stop particles with a diameter less than 1 cm, the small debris our system is not targeting.

The mass of the shielding required to cover the front of the Netting Module and the perimeters of the Propulsion Module, Netting Module, and Transfer Vehicle is approximately 359.6 kg. This includes a 10% overestimate to take into account the structure that will be needed to support the sheets of Nextel.

5.10 Fuel Requirements

The masses of the other subsystems, as well as their volumes, played an important role in the calculation of the mass of the fuel needed. The calculations used the ideal rocket sizing equation

$$\text{mass of fuel} = (\text{mass of spacecraft})\text{x}(1 - e^{-dv/g*Isp})$$

where

dv = velocity change required to change spacecraft's orbit

g = the acceleration due to gravity

Isp = the specific impulse of the fuel

and the following assumptions

- Mass of Netting Module is
  NM20 - 2,432.5kg
  NM20/40 - 2,350.4 kg
  NM20/40/90 - 2,332.6

- Mass of Propulsion Module is 2672.0 kg

- The Netting Module completely fills its nets with maximum size debris for each hole (masses for 2 cm thick aluminum plates)
  14cm x 14cm plate - mass of 1.06 kg
  28cm x 28cm plate - mass of 4.23 kg
  63cm x 63cm plate - mass of 21.43 kg

- Fuel is Hydrazine-Nitrous Oxide mixture
  Isp = 318 seconds
  density = 1200 kg/m^3 [5.35]

- The delta v needed to capture each piece of debris (Data obtained from Himawari 1 rocket booster breakup in July 1977. See Appendix C)
  delta v = 15 m/s

- The Netting Vehicle collects the smallest pieces of debris first, then moves to larger pieces

A program (a listing is included as Appendix D) was written to iterate the amount of fuel needed for each of the Netting Module configurations to collect all the debris they can hold. The program added an extra 10% at the end to take into account proximity operations when capturing the debris. The NM20 configuration would require 2,972 kg of fuel and 2.48m^3 of storage space. The NM20/40 configuration requires 796.1 kg to perform its mission, and the fuel will take up a volume of 0.66 m^3. The NM20/40/90 configuration required 552.2 kg of fuel and 0.46 m^3 of volume.

All three configurations can therefore be used, although the structure length of the 20/40 and 20/40/90 configurations can be reduced. To maintain an extra volume of approximately 0.5 m^3, the lengths of the NM20/40 and NM20/40/90 structure can be reduced to 2.22 m, reducing the structural mass by 102 kg.

Similar calculations were performed to calculate the fuel needed for the fully loaded Debris Removal System to go from the Space Station to the parking orbit. We included a 30 degrees wedge angle or 30 degrees inclination change. The total fuel mass needed for the Transfer Vehicle was 3400 kg.

A complete summary of the vehicle masses, using this new data, is included as Table 5.15.

Table 5.15 Summary of Vehicle Masses

(All values in kg)

Vehicle

NM Configuration

Subsystem	20	20/40	20/40/90	PM	TV
Structure	1653.0	1551.0	1551.0	1466.8	1933.9
Netting	253.5	165.0	143.5	******	******
Propulsion	******	******	******	180.0	2550.0•
Power	******	******	******	23.4•1	1453.0•2
Thermal	436.0	436.0	436.0	235.0	303.0
Tracking	******	******	******	130.0	*******
Comm.	******	******	******	80.0	140.0
GNC	******	******	******	271.0	1076.0[3]
DPS	******	******	******	mass included in GNC, tracking, & comm.	
Shielding	90.0	79.0	79.0	116.6	153.0
Fuel	2972.0	796.1	552.2	170.0	3400.0
Total - Dry	2432.5	2231.0	2209.5	2502.0	7608.9
Total - Fueled	5404.5	3027.1	2761.7	2672.0	11008.9

• assumes 1995 technology

1 Battery - 13 kg
Solar Array - 10.4 kg (assumes 35% effeiciency)

2 Battery - 741 kg
Solar Array - 712 kg (assumes 35% efficiency)

3 includes mass of RCS fuel

6.0 System Integration

6.1 Debris Removal System

The final dimensions and configurations of the Propulsion Module, Netting Module, and Transfer Vehicle are shown in Figures 6.1, 6.2, and 6.3. In order that the complete system can be transported into space with one shuttle flight, the NM20 configuration will not be used initially. However, it may be used for later missions since it can collect a greater amount of debris.

The DRS will consist of three Netting Modules, one Propulsion Module, and the Transfer Vehicle, as shown in Figure 6.4. The total pre-launch mass of a system with two NM20/40/90 modules and one NM20/40 module to collect 65 pieces of debris is **22,231.4 kg**. After all the pieces of debris have been collected, the mass of the system will be approximately **18,637.1 kg**. The mass that will need to be returned to Earth in the space shuttle (the three unfueled Netting Modules) will be approximately **6,750 kg**.

6.2 Debris Retrieval

The most important part in capturing a piece of debris is knowing where it is. The Netting Vehicle will first use the onboard IR sensor to estimate a trajectory for the particle when the distance is less than 2000 kilometers, and later it will use the LADAR sensor when it is within 25 kilometers. Using the information derived from these sensors, the Netting Vehicle will attempt to get as close to the debris as possible to facillitate capture.

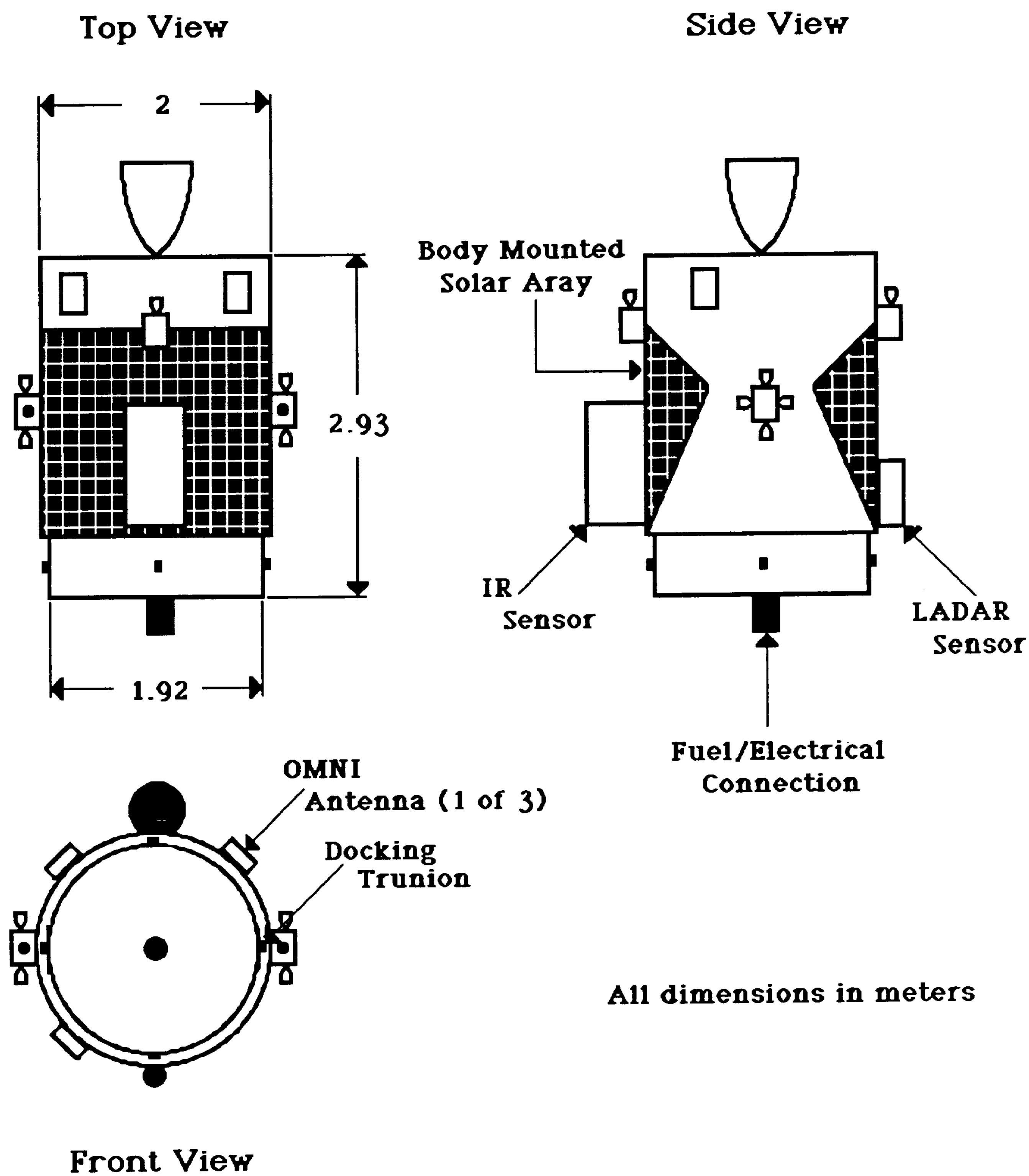


Figure 6.1 Propulsion Module

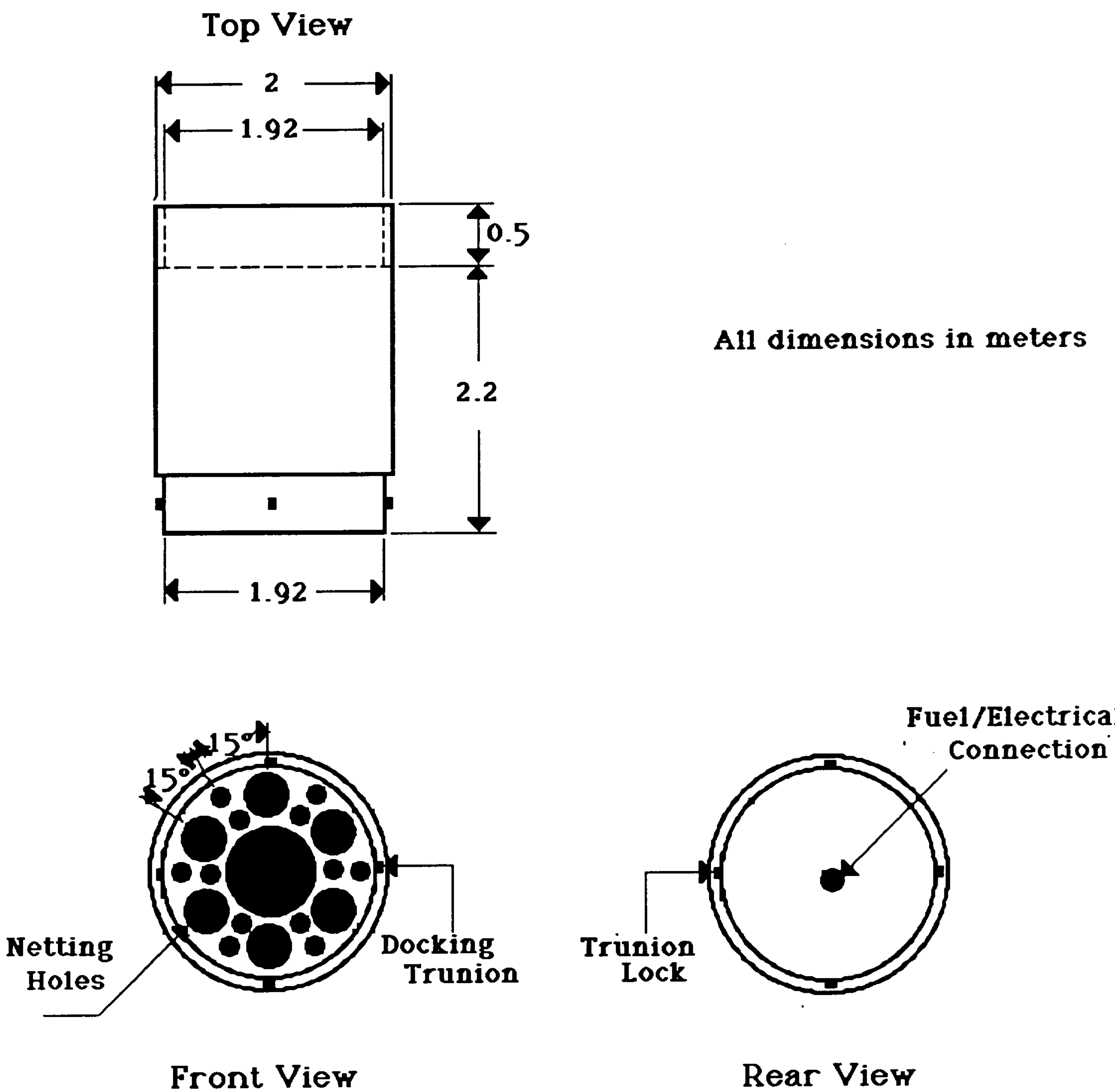


Figure 6.2 Netting Module (NM20/40/90 Configuration)

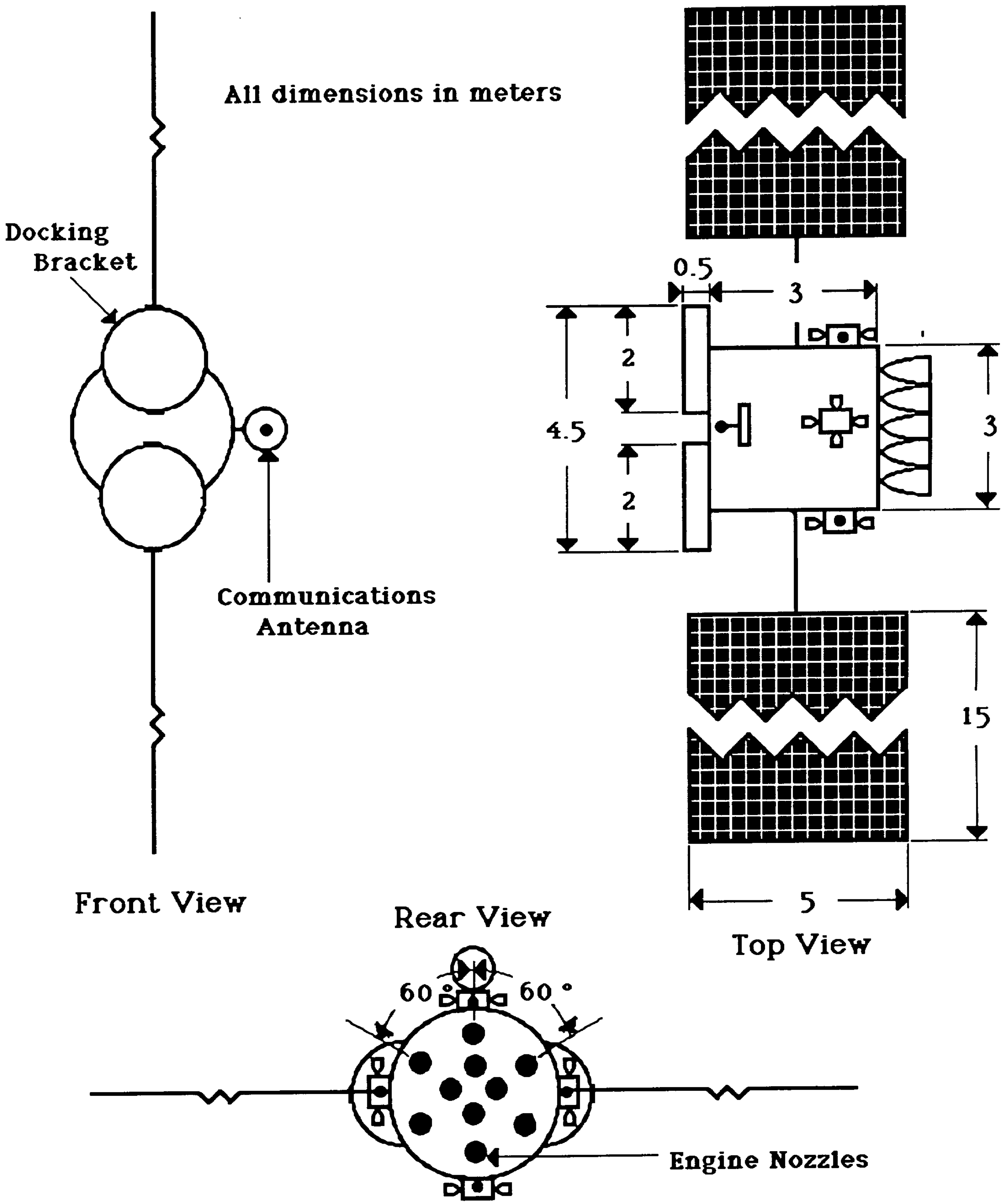


Figure 6.3 Transfer Vehicle

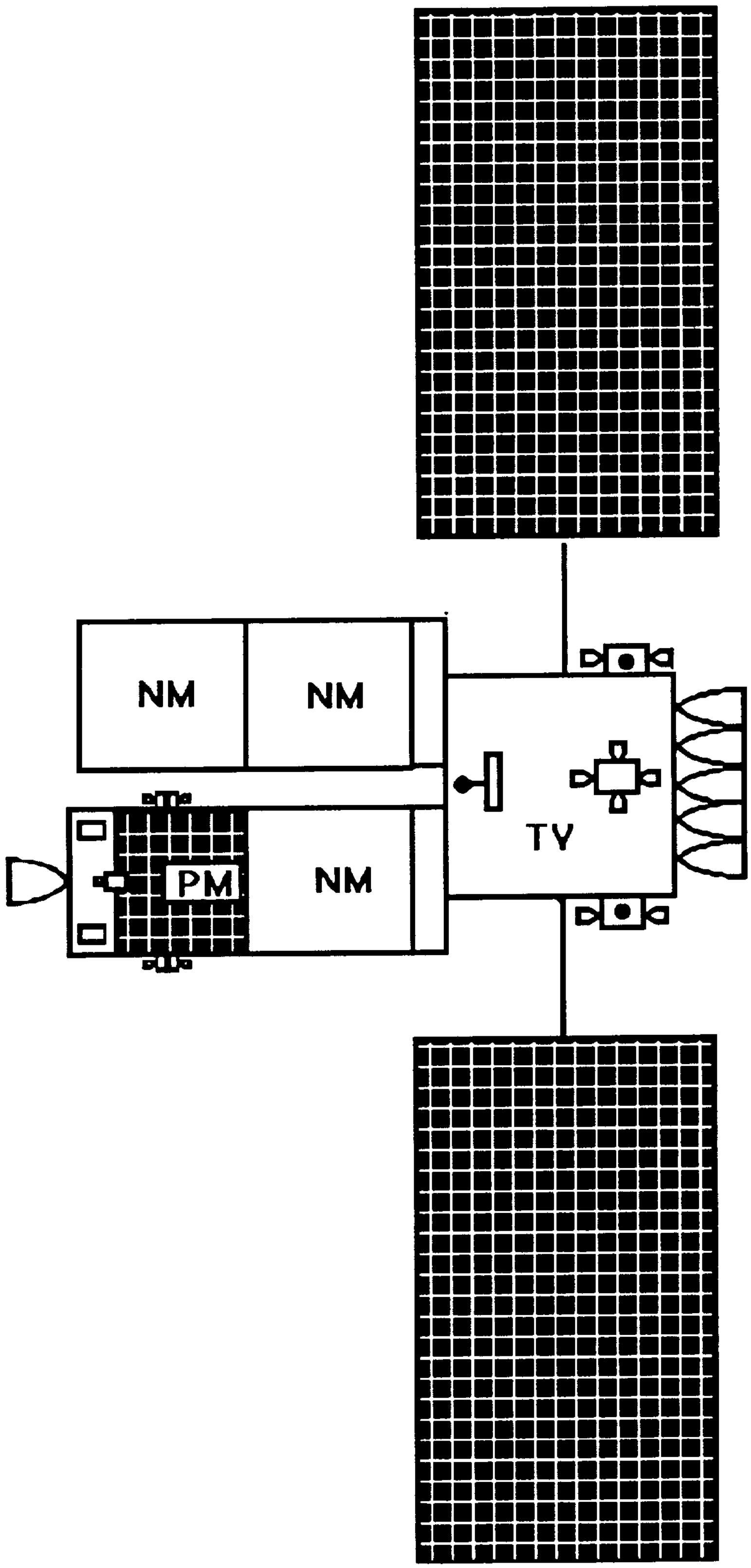


Figure 6.4 Debris Removal System

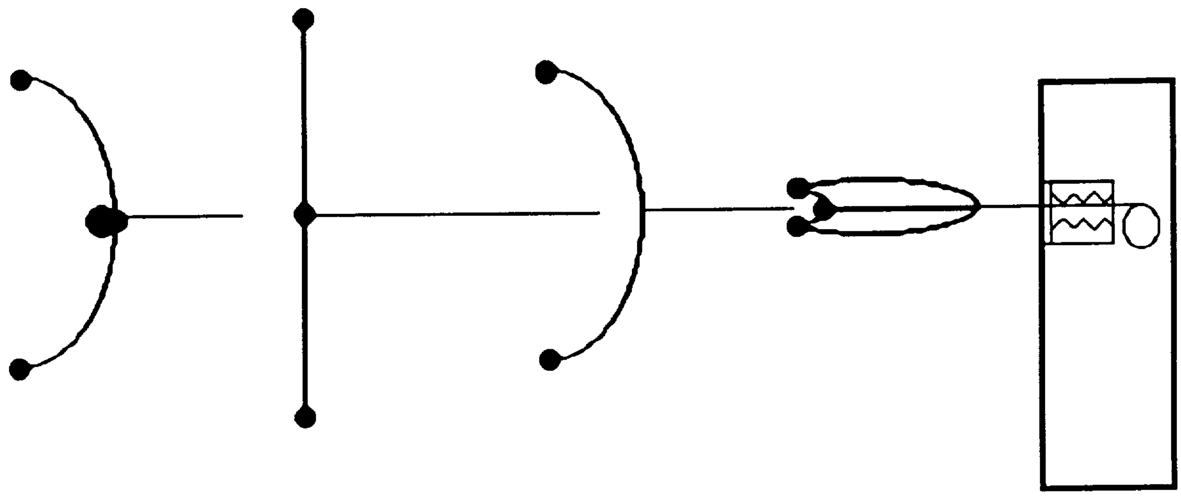


Figure 6.5 Deploying the Net

While the Netting Vehicle is doing this, it is also interpreting the sensor information in order to approximate the size of the debris. This is very important, since after the debris is netted, it will be reeled back into the Netting Module for storage. For example, suppose the sensors estimate the size of the debris to be 14.5 cm. This size is just above the upper limit for storage in the 20 cm cylinder because the debris could be rectangular with a longer diagonal that could impinge on the hole when it is pulled in. The debris still might fit in one of the 20 cm cylinders, but it would be safer to store it in one of the 50 cm cylinders. Therefore, a net from one of the 50 cm cylinders would be launched at the the debris.

The dynamics of this launch is shown in Figure 6.5. The net is launched by a spring system and is connected to the Netting Module by a tether. The net is spun when it is launched so that the masses on the perimeter will open it. This spin is generated in the launch cylinder (assumed to be 10 cm in diameter and 20 cm long) because the end masses are in slots that spiral along the length of the tube, like rifling in a gun barrel.

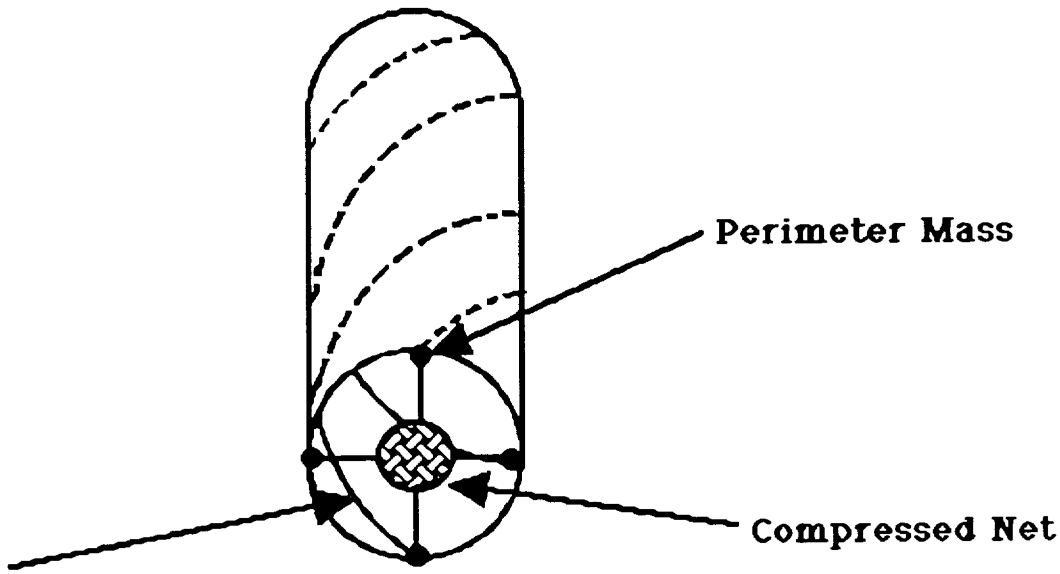


Figure 6.6 Launching Tube

This launch cylinder is located at the back of each storage cylinder and along the midline, and the perimeter mases will fit in grooves in the wall as shown in Figure 6.6. If the net is spinning at 1.6 revolutions per second when it leaves the tube and is travelling forward at a speed of 1.1 m/sec, there is no problem with the masses hitting the inside walls of the storage cylinder. The net will fully open some 1 to 5 meters from the Netting Module, depending on which cylinder is is launched from.

Once the net has hit the debris, the net will be closed so that the debris is contained inside when the net is reeled into the Netting Module. The net will be closed with a mechanical pulley that is activated by braking the tether. When a collision has been detected (by a small accelerometer on the net), or when the net has reached the end of its tether, the tether will be braked. The masses on the perimeter will continue to move forward until the tension in the tether is redistributed via pulleys and cables to pull them together like a cinch.

After the debris has been captured and contained in a net, the net will be reeled back into the storage cylinder. Because the Netting Vehicle will not be able to approach the debris without some small relative velocities, control moment gyros and RCS thrusters will be used to rotate the Netting Vehicle during the retrieval so that the net does not wrap around the vehicle.

6.3 Docking

In order for this Debris Removal System to work, the Propulsion Module will have to dock with the Netting Modules, the Netting Modules will have to dock with each other for storage, and all of them will have to dock with the Transfer Vehicle. The docking between the Propulsion Module and the Netting Module will have to establish connections for the fuel and power interfaces.

NASA's proposed Orbital Maneuvering Vehicle uses four trunions on the perimeter of its propulsion module to connect with the flight vehicle [6.1] (see Figure 6.7). Our docking mechanism will be similar, but will have umbilicals for electrical connections and a fluid connection to transfer fuel between the Netting Modules and the Propulsion Module. All the connections will be controlled by the Propulsion Module, since they will not be needed for Netting Module to Netting Module couplings or Netting Module to Transfer Vehicle couplings.

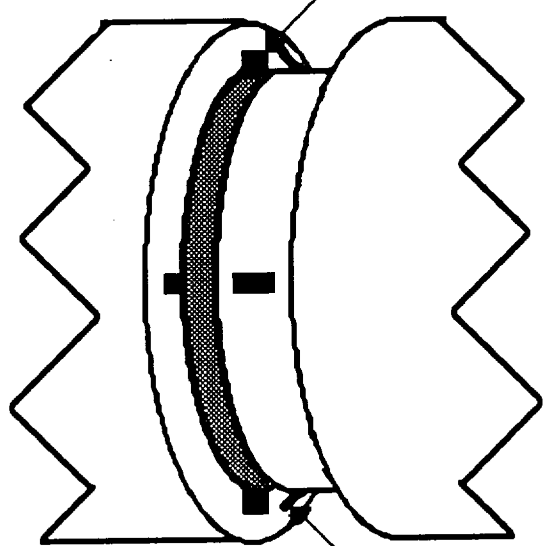


Figure 6.7 Docking Mechanism

7.0 Debris Prevention Concepts

In this section we will discuss concepts for prevention of orbital debris. Since a detailed discussion on this topic was included in the spring design report of the orbital debris working group [7.1,60-72] we refer to this part of the report.

This section will therefore contain a short overview of debris prevention techniques and design alterations. This relates to the fact that modification of mission hardware and space practices to prevent orbital debris is far more economical than a complex and costly mission for active debris removal.

7.1 Self Disposal of Spacecraft

By deorbiting payloads or inserting them into higher orbits or Earth escape trajectories, further contamination of the space environment can be prevented. This can be achieved by a series of passive or active devices and methods.

7.1.1 Drag Devices

The effect of atmospheric drag on a satellite can be increased by deploying a large ballon which increases the effective area of the satellite without significantly increasing its mass. For objects orbiting below 800 kilometers, a ballon with a diameter of about 15 meters can reduce the orbital lifetime of the satellite from several years to several weeks. This proposed deorbit device would be included as part of the mission payload and would have to safely reamain inert for a period of up to many years. The ballon could be inflated after a rocket or satellite completes its mission. The main advantage of the drag device concept is that it is simple, passive, and the satellite does not need to maintain any specific orientation and no attitude control system is needed [7.2,4-5].

7.1.2 Solar Sails

Solar sails might be an option for disposal of objects in very high orbits. Solar sails are a relatively passive system and they require no propellant storage or engines. They might be used for moving satellites in geosynchronous orbit into higher orbits or to send the satellites onto Earth escape trajectories. However, deployment and control of the solar sail might present significant technical challenges.

7.1.3 Deorbit Engine

Another method for self-disposal is the addition of a seperate system for deorbit at the end of the operational lifetime. Deorbit with a conventional propulsion system is an approach which would be effective for all orbital altitudes (for circular orbits above 25,000 kilometers, an escape from Earth orbit is less costly than a deorbit maneuver). Such a system would naturally increase the payload wight, but is is still much less expensive than active retrieval. For altitudes below 700 kilometers drag devices appear to be a lower-mass alternative to propulsion packages [7.3,5].

7.1.4 Additional Fuel

Upper stages and satellites can be designed for self-disposal using its own propulsion system for a controlled deorbit and ocean impact or orbit raising. Adding a small percentage of fuel would enable the station keeping motors to act as deorbit engines once the useful life of the spacecraft has ended. This method requires no additional engines or other devices and is therefore relatively cost efficient. This policy has already been adopted by a number of space agencies for their geostationary satellites. At the end of the lifetime of a satellite the remaining station keeping fuel is used to boost the satellite several hundred kilometers above geostationary altitude into a "Graveyard Orbit" which does not interferre with the geostationary ring, thereby reducing the collision probability significantly.

7.2 Subsystem Redesign

By modifying current spacecraft subsystems and components the production of additional space debris can be widely prevented. By minimizing the risk of future orbital breakups by hardware redesign and mission design alterations, extremely costly active removal procedures can be reduced.

7.2.1 Rocket Redesign

Main contributors to orbital debris have been breakups of upper stages. One main design change is the arrangement for the depletion of all pressurized propellants and reduction of gas pressures. Therefore, experimental restarts should be made standard, i.e., hold the engine on long enough to assure that as much fuel and oxidizer as possible is vented from the tanks. Leakage of the tanks due to structural fatigue (repeated expansions and contractions as the vehicle goes in and out of the eclipse) has to be considered[7.3].

7.2.2 Seperation Mechanism Redesign

Currently most launch vehicles are referred to as "dirty" rockets because they use explosive stage connecting bolts to separate rocket stages and payloads. In order to provide a clean stage separation the related mechanisms need to be redesigned. Such a mechanical release system is currently being developed at Johnson Space Center, Houston [7.4].

7.2.3 Increased Use of Reusable Hardware

The design philosophy applied in the design of future space systems needs to take into account the risks and costs associated with a growing debris hazard. Generally, because of the high cost of launching space hardware, all launch vehicle and spacecraft elements are jettisoned or abandoned as soon as they are no longer needed or when critical systems fail. The "expendable" philosophy is beginning to change: single-use satellites could be replaced by multi-purpose platforms which can be repaired and upgraded periodically (modular design!). Reusable orbital maneuvering and transfer vehicles could replace the expendable upper stages which litter the orbital environment.

7.2.4 Improved Shielding

Advanced shielding concepts applied to future spacecraft design can greatly minimize the creation of secondary debris caused by meteorite and space debris impacts. A multi-wall structure such as a multi-layer bumper system can significantly reduce the amount of secondary debris created by the impact. All shielded surfaces would then act as debris "sinks", rather than debris "sources".

7.2.5 Redesign of Protective Coating

Another main source of orbital debris is microparticles from paints and protective coatings. Alternative durable bonding agents could reduce degradation of those elements by atomic oxygen and the harsh thermal effects in space, in order not to cause paint and coating to fleck.

8.0 Management Proposal

8.1 Management Structure

The management structure adopted by SPECS, Inc. combines a general Program Manager, a Technical Manager, and three subgroup leaders into an organizational support structure designed to facilitate the engineering process. Figure 8.1 shows a diagram of the complete management structure.

The Program Manager oversees all aspects of the project at a high level of involvement. The administrative decisions and coordination effort fall into the Program Manager's responsibility. The Program Manager also works closely with the Technical Manager on developing realistic long term goals and design milestones.

The Technical Manager coordinates the design effort and provides a common point of contact between the three subgroup leaders and the Program Manager. Weekly status reports are collected, combined and distributed by the Technical Manager to aid in communication within the group. The Technical Manager must work directly with the three subgroup leaders to develop intermediate design goals that progress toward the long term milestones.

The subgroup leaders are responsible for directing the engineer's design philosophy and integrating each individual's effort into a workable product. The subgroup leaders provide a means of communication between the separate subgroups when cross-information is required.

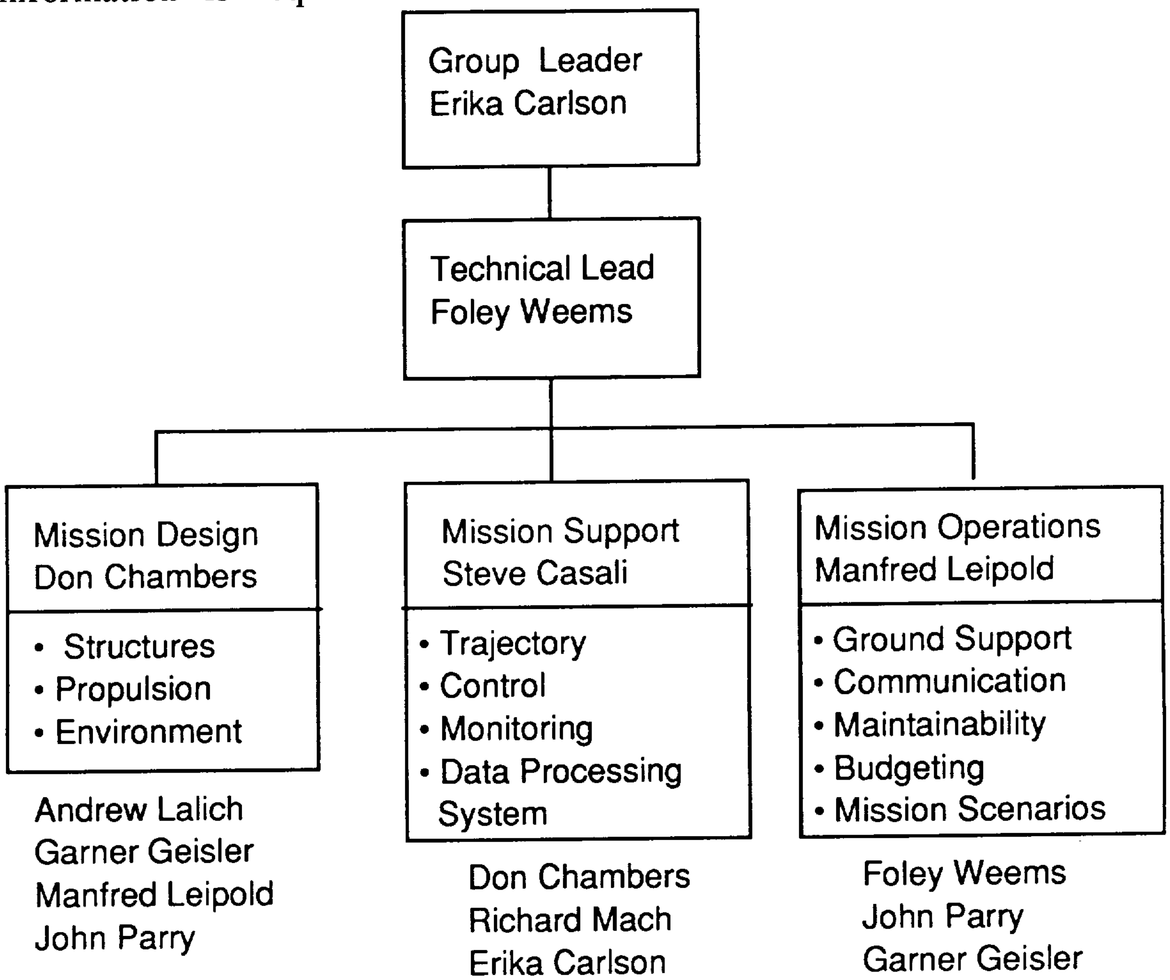


Figure 8.1 SPECS, Inc. Organization Structure

SPECS, Inc. consists of nine members that are dually responsible for the engineering tasks and the management responsibilities. As a result, communication between the group members is facilitated. Most of the group members belong to two or more subgroups. Any problems or requests for information that arise are quickly transmitted to the management and the other subgroups.

8.2 Subgroup Responsibilities

The organizational structure of SPECS, Inc. divides the design effort into three subgroups: Mission Design, Mission Support and Mission Operations. Each subgroup concentrates on particular aspects of the overall project.

The Mission Design subgroup focuses on the structural and mechanical development of the primary and secondary designs. All research and design of the propulsion, environmental, and electrical systems and any robotic development is the responsibility of this subgroup.

The Mission Support team handles these critical aspects affecting the design and its operation. Trajectory analysis and the dynamics and control of the vehicles developed in this area. Additionally, any data processing systems, commanding, monitoring and instrumentation requirements are identified by this subgroup.

Mission Operations develops the mission scenarios the design must perform. Any ground support required for the mission is developed in this area. Communication, maintainability, safety, and mission planning considerations are also handles by the Mission Operations team.

8.3 Task Development

A project timeline that displays the major milestones of the design effort was developed to aid in meeting the project deadline. Figure 8.2 illustrates the project schedule. The critical paths of the design process were identified to help control the development of the project. Figure 8.3 depicts the PERT/CPM critical path chart.lank page for timeline

Figure 8.4 describes the problem solving method SPECs, Inc. employs. Problems are detected by an individual or a subgroup and evaluated according to criticality. Minor problems will be solved internal to the subgroup. Research on the item will proceed at the subgroup level. Again, the item will evaluated to determine if the entire group must become involved. The item can either be discussed and solved at the subgroup level, with a presentation of the solution to the full working group for education, or the item can be referred to the full group for a discussion and solution.

8.4 Workload Considerations

Because of the size of SPECS, Inc., each engineer is involved in several tasks. To keep track of individual workloads, manpower utilization charts are collected and updated weekly by the Project Manager. As an estimate of the total man-hours required for the project, it is assumed each engineer devote 12 hours a week toward the project, and each manager contributes 15 hours weekly.

Blank page for timeline

Blank page for Pert/CPM

Figure 8.5 displays the resulting manpower estimate for the total project. The total effort required for the completion of the project is 1722 man-hours. This estimate will be compared to the actual man-hours to guard against over and under working the engineers.

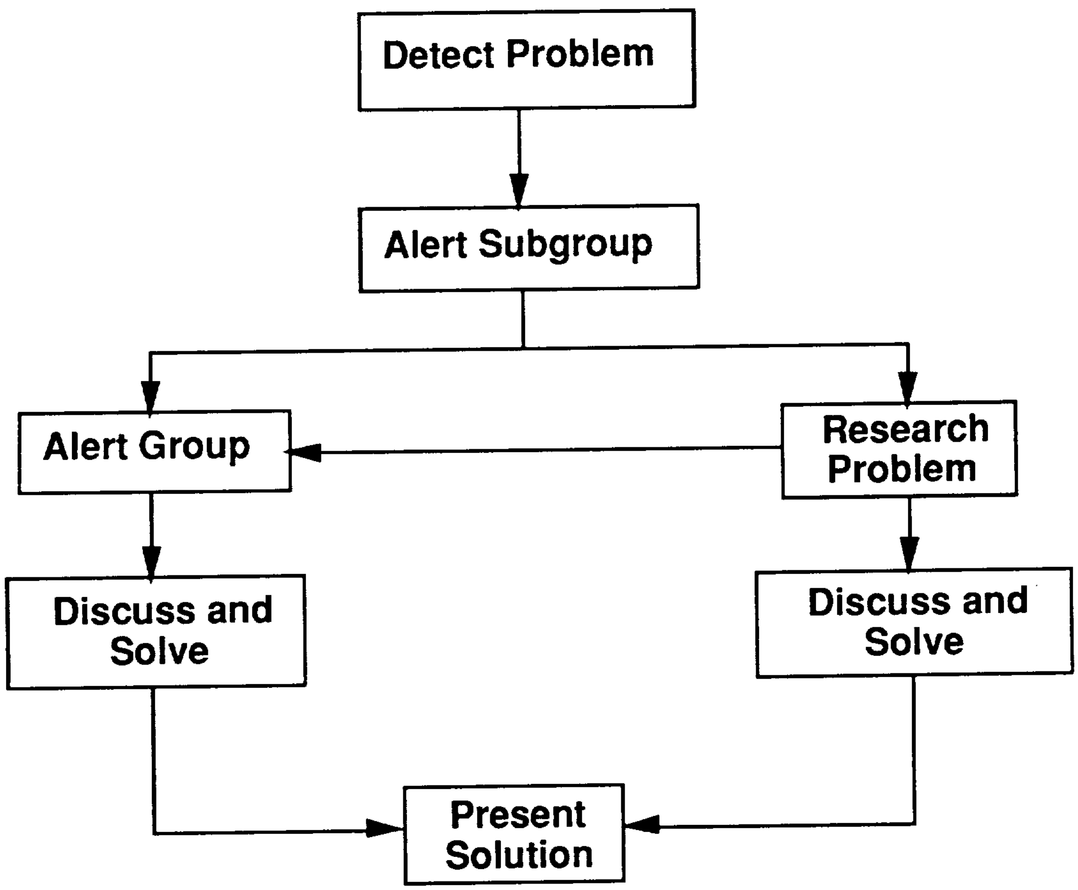


Figure 8.4 Problem Solving with SPECS, Inc.

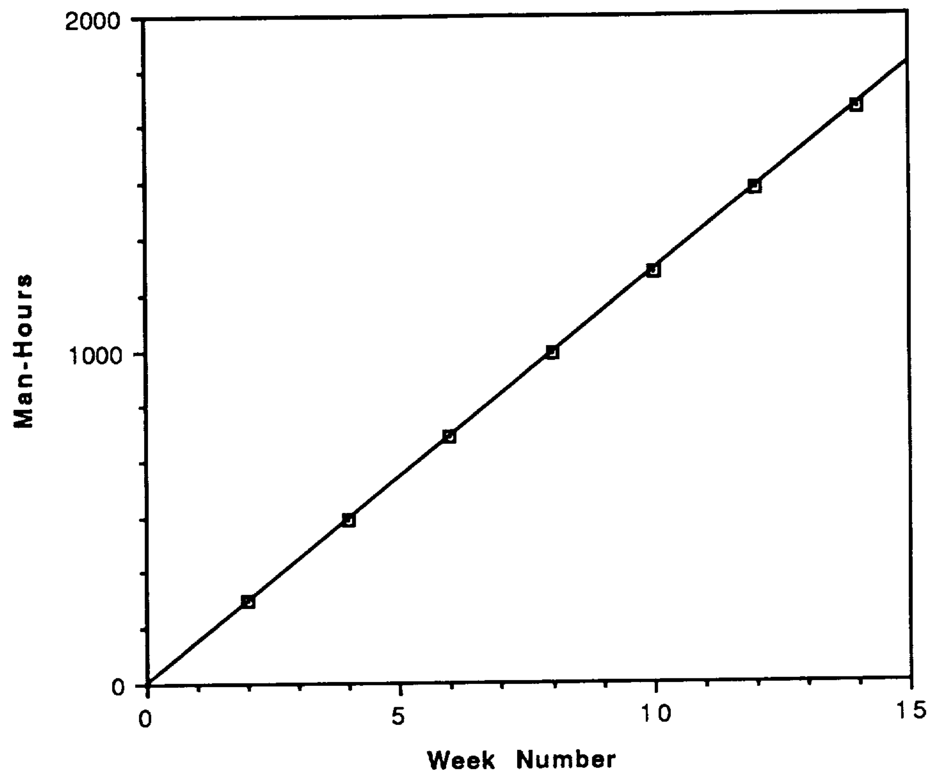


Figure 8.5 Manpower Estimates for SPECS, Inc.

9.0 Cost Proposal

9.1 Personnel Cost Estimate

Pay scales were derived from the Request for Proposal as follows: Engineers, $17.00/hr; Sub-Leaders, $20.00/hr; Technical Lead, $22.00/hr; project manager, $25.00/hr; and technical consultants, $75.00/hr.

Table 9.1 Formulation of Projected Costs

Weekly breakdown	
1 project manager @ $25/hr:	375.00
1 technical lead @ $22/hr:	330.00
3 sub leaders @ $20/hr:	720.00
9 engineers @ $17/hr	1530.00
5 hours of consulting	375.00
total weekly personnel cost estimate:	$ 3330.00
Projected cost for 14 weeks:	$ 46620.00
plus 10% error estimate	4662.00
TOTAL ESTIMATE	$ 51282.00

9.2 Material and Hardware Costs

The material and hardware cost estimates are based on expenses to date and those of previous design groups. Government furnished equipment (GFE) consists of computer hardware, software, and mainframe computer time. A table of anticipated costs follows in the table below

Table 9.2 Anticipated Hardware Costs

	PROPOSED
Macintosh software and peripherals:	$ 2300.00
IBM PC-AT software and peripherals:	500.00
CDC computer mainframe time:	50.00
modeling of design:	200.00
photocopies @ $.05/each:	35.00
transparencies @ $.70/each:	70.00
miscellaneous supplies:	80.00
SUBTOTAL	$ 3235.00
plus 10% error estimate	323.50
Total Estimate	$ 3558.50

ESTIMATED TOTAL COST

	PROPOSED
personnel cost:	$ 51282.00
material and hardware cost	3558.50
GRAND TOTAL	$ 54,840.50

COST TO DATE (12/3/90) $ 35,756.13

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1.5 Chobotov, V. A., *"Dynamics of Orbiting Debris Clouds and Resulting Collision Hazard to Spacecraft"*, Space safety and Rescue 1986-1987, Science and technology Series, Vol. 70, pp. 223-241

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PFNO: A00920 SPACECRAFT: ISEE LAUNCH: 08/12/78
DATE: 04/15/81 FLIGHT: 3 STATUS: UP

SUBSYSTEM : INST-WIDENBCK
TIER LEVEL 1 : PRESSURE VESSEL
TIER LEVEL 2 :
TIER LEVEL 3 :

MISSION IMPACT : 2 - POTENTIAL FOR MAJOR
POSSIBLE CAUSES : E
ENVIRONMENT CODE : M

OCCURENCE RATE: 4 - SLOW DEGRADATION
DURATION: 4 - TOTAL LOSS (NO IMPROVEMENT

IMMEDIATE RESPONSE : D -
LONG-TERM SOLUTION : -

POSSIBLE CAUSES:

HARDWARE DESIGN
MANUFACTURING
WORKMANSHIP
PART FAILURE
MATERIALS
INDUCED FAILURE
✓ ENVIRONMENTAL

OPERATING TIME
HUMAN/OPERATOR ERROR
PROCEDURAL DESIGN
OTHER
UNKNOWN
UNDEFINED

SYMPTOM : LEAK IN GAS SYSTEM CAUSED COMPLETE LOSS OF GAS IN DRIFT CHAMBER OVER A PERIOD OF ONE HALF HOUR. THIS CAUSES LOSS OF TRAJECTORY MEASUREMENT CAPABILITY.
COMMENT:

CAUSE : NOT KNOWN FOR SURE- PROBABLY DUE TO MICROMETEOROIDS OF SUFFICIENT SIZE & VELOCITY TO PUNCTURE THE 0.13MM BERYLLIUM-COPPER PRESSURE VESSEL WINDOW.

RECOVERY: NONE POSSIBLE.

CORR.ACT: USE DIFF.DESIGN:NO-GAS SYSTEMS OR BETTER SHIELDING OF GAS TNK

GENERAL :

OUR NOTE:

PFNO: A00682 SPACECRAFT: ISEE LAUNCH: 10/22/77
DATE: 08/01/78 FLIGHT: 1 STATUS: UD

SUBSYSTEM : INST-HVESTADT
TIER LEVEL 1 : PROPORTIONL CNTR
TIER LEVEL 2 : LO-ENERGY DETCTR
TIER LEVEL 3 :

MISSION IMPACT : 2 - POTENTIAL FOR MAJOR
POSSIBLE CAUSES : E
ENVIRONMENT CODE : MB

OCCURENCE RATE: 2 - INTERMITTENT
DURATION: 4 - TOTAL LOSS (NO IMPROVEMENT

IMMEDIATE RESPONSE : D -
LONG-TERM SOLUTION : -

POSSIBLE CAUSES:

HARDWARE DESIGN
MANUFACTURING
WORKMANSHIP
PART FAILURE
MATERIALS
INDUCED FAILURE
✓ ENVIRONMENTAL

OPERATING TIME
HUMAN/OPERATOR ERROR
PROCEDURAL DESIGN
OTHER
UNKNOWN
UNDEFINED

SYMPTOM : SUDDEN LOSS OF GAS PRESSURE IN ONE OF 3 LOW ENERGY DETECTORS.
COMMENT:

CAUSE : PROBABLY DUE TO PUNCTURING OF THIN WINDOW(FRONT) BY MICRO-METEORITE.

RECOVERY: NONE POSSIBLE.

CORR.ACT:

GENERAL :

OUR NOTE:

PFNO: A00932 SPACECRAFT: TDRS LAUNCH: 04/04/83
DATE: 04/09/85 FLIGHT: 1 STATUS: UD

SUBSYSTEM : TLM & DH
TIER LEVEL 1 : LCP/RCP SWITCH
TIER LEVEL 2 : SA2 ANTENNA COMP
TIER LEVEL 3 :

MISSION IMPACT : 2 - POTENTIAL FOR MAJOR
POSSIBLE CAUSES : D
ENVIRONMENT CODE : L

OCCURENCE RATE: 5 - SYSTEMATIC
DURATION: -

IMMEDIATE RESPONSE : C -
LONG-TERM SOLUTION : -

POSSIBLE CAUSES:

- ✓ HARDWARE DESIGN
- MANUFACTURING
- WORKMANSHIP
- PART FAILURE
- MATERIALS
- INDUCED FAILURE
- ENVIRONMENTAL
- OPERATING TIME
- HUMAN/OPERATOR ERROR
- PROCEDURAL DESIGN
- OTHER
- UNKNOWN
- UNDEFINED

SYMPTOM : CONTAMINATES ARE SUSPECTED TO BE WITHIN CLOSE PROXIMITY TO SWITCH. THIS CONDITION MAY CAUSE THE SWITCH TO BECOME STUCK, RESULTING IN LOSS OF KSA2 SERVICES.
COMMENT:

CAUSE : CONTAMINATES(PARTICLES) IN VICINITY OF SWITCH. (CONTINUED USE OF SWITCH MAY CAUSE PARTICLES TO MIGRATE & DECREASE KSA OUTPUT.)

RECOVERY: RESTRICTED OPERATION OF WAVEGUIDE SWITCH.

CORR.ACT:

GENERAL :

OUR NOTE:

PFNO: 0011
DATE: 10/15/78

SPACECRAFT: TIROS
FLIGHT: N

LAUNCH: 10/13/78
STATUS: UD

SUBSYSTEM : THERMAL
TIER LEVEL 1 :
TIER LEVEL 2 : *
TIER LEVEL 3 : *

MISSION IMPACT : 1 - MINOR OR NONE
POSSIBLE CAUSES : E
ENVIRONMENT CODE : L

OCCURENCE RATE: -
DURATION: -

IMMEDIATE RESPONSE : D -
LONG-TERM SOLUTION : * -

POSSIBLE CAUSES:

HARDWARE DESIGN
MANUFACTURING
WORKMANSHIP
PART FAILURE
MATERIALS
INDUCED FAILURE
√ ENVIRONMENTAL

OPERATING TIME
HUMAN/OPERATOR ERROR
PROCEDURAL DESIGN
OTHER
UNKNOWN
UNDEFINED

SYMPTOM : THE TEMPERATURE OF THE HYDRAZINE COMPONENTS IS WARMER THAN PREDICTED.

CAUSE :

RECOVERY: THE WARMER TEMPERATURE OF THE HYDRAZINE TANKS AND LINES IS CAUSED BY CONTAMINATION OF THE THERMAL COATINGS.

CORR.ACT:

GENERAL :

OUR NOTE:

PFNO: 41013 SPACECRAFT: VOYAGER LAUNCH: 08/20/77
DATE: 08/21/77 FLIGHT: 2 STATUS: UD

SUBSYSTEM : ARTICULATION & CONTROL SUBSYSTEM
TIER LEVEL 1 : *
TIER LEVEL 2 : *
TIER LEVEL 3 : *

MISSION IMPACT : 2 - POTENTIAL FOR MAJOR
POSSIBLE CAUSES : S
ENVIRONMENT CODE : M

OCCURENCE RATE: 2 - INTERMITTENT
DURATION: -

IMMEDIATE RESPONSE : A -
LONG-TERM SOLUTION : A -

POSSIBLE CAUSES:

HARDWARE DESIGN
MANUFACTURING
WORKMANSHIP
PART FAILURE
MATERIALS
INDUCED FAILURE
ENVIRONMENTAL

OPERATING TIME
HUMAN/OPERATOR ERROR
PROCEDURAL DESIGN
OTHER
UNKNOWN
UNDEFINED

SYMPTOM : PARTICLES IN THE FIELD OF VIEW OF THE TRACKER CAUSED LARGE EXCURSIONS IN THE PITCH AND YAW AXES WHILE UNDER CELESTIAL CONTROL IN THOSE AXES. PROBLEM REFERRED TO AS THE 'BUMP IN THE NIGHT'!

CAUSE : WHEN THE CR240 ROUTINE IS EXECUTED THE PITCH AND YAW S.S. BIASES ARE I NCREMENTED. THIS CAN HAPPEN EVERY .24 SEC. THEIR REMOVAL IS ONLY EVERY 1.2 SECONDS SO LARGE ERROR BIASES CAN ACCUMULATE. THE BRIGHT PARTICULE S CAUSE THE ERROR TO ACCUMULATE FASTER THAN EXPECTED. ALSO SEE PFR'S 3 7399,40411,40683

RECOVERY: NONE

CORR.ACT: THE SOFTWARE WAS REWRITTEN. INSTEAD OF INCREMENTING THE PITCH AND YAW BIASES TO FORCE A RELOAD OF THE S/S D/A CONVERTERS, THE CONVERTERS ARE . RELOADED DIRECTLY IN CR240 USING THE CORRECT VALUE (NO INCREMENT) OF THE S/S BIASES. THE CORRECTIVE S/W PATCH WAS LOADED TO BOTH FLIGHT S/C

GENERAL :

OUR NOTE:

Appendix B Other Design Options Considered

B.1.1 SPInning DEbris Remover (SPIDER)

The SPIDER design incorporates a vehicle similar to the Orbital Maneuvering Vehicle (OMV), and in fact it is being considered whether it should be made into a modular attachment for the OMV. The SPIDER vehicle will actively track large pieces of debris (inactive satellites, spent rocket stages, non-operational hardware) in the targeted region and send the debris into the atmosphere. The SPIDER will be equipped with three robotic arms for collecting large debris, that with diameters greater than 1 meter (m).

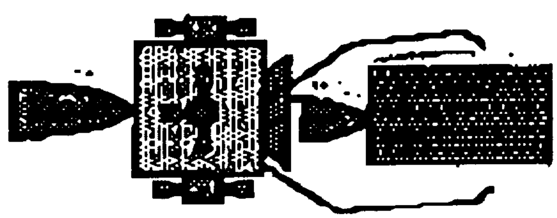

Figure B.1 SPinning DEbris Remover

Since the large debris will probably be spinning about a major axis, we foresee the SPIDER attaching itself to the debris by spinning at the same rate as the debris and then clamping on to it with its three robotic arms. After it has despun the device, the SPIDER will either place a small thruster device on the debris, or will remain attached itself. In either case, the debris will be slowed down by a thruster firing so that its orbit will decay into the atmosphere. If the SPIDER is reused, it will be able to detach itself from the debris and return to the Space Station, or a similar base, for maintenance and refueling.

B.1.2 Tethering

Tethering is a concept that has been extensively researched in the last ten years [B.1:]. The principle of using tethers to eliminate orbital debris is to redistribute the orbital momentum of the debris. Fuel is saved when the energy from the faster moving debris is used to increase the velocity of the spacecraft, thus eliminating the need for a propulsive maneuver, while at the same time slowing the debris down to a reentry orbit [B.2:].

Figure B.2 Tethering Principle

Calculations have shown that a tether design would be very efficient when working with large debris in low earth orbit (in the 200 km to 700 km altitude range), eliminating up to 50 kilograms (kg) of fuel for each deorbit mission. The possibility of using a tethering device in the SPIDER has been considered.

B.1.3 Netting

Using nets to handle large and small debris would eliminate the need for despinning the debris. It could also be used to capture tumbling debris. However, because of the problem of the net tearing as well as potential danger to the spacecraft deploying the net, we do not believe it is feasible to net large objects.

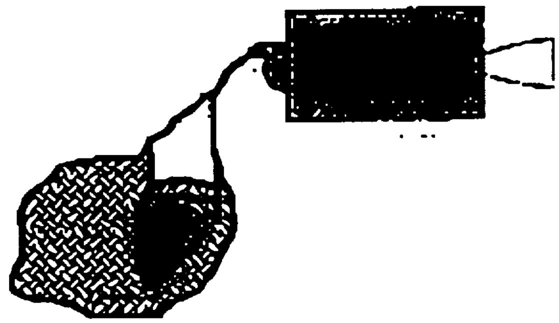

Figure B.3 Netting Design

However, this appears to be the best idea for capturing medium sized objects, and SPECS, Inc. believes workable nets could be fashioned out of current high strength fabrics like Kevlar. Again, the possibility of using this with the SPIDER has been considered, principally to collect any medium sized debris in the area, and any debris that may be created by the SPIDER attaching to the object.

References

B.1 Tethers in Space Handbook, First Edition, NASA Office of Advanced Programs, January, 1985

B.2 Colombo, G., *"The Use of Tethers for Payload Orbital Transfer"*, NASA Contract NAS8-33691, Vol. II, March, 19282

B.3 Carroll, J. A., *"Guidebook for Analysis of Tether Applications"*, Contract RH-394049, Martin Marietta Corporation, March, 1985

Appendix C Calculation of Perturbative Accelerations

1. **Thrust Acceleration Magnitude:**
   - a_t = 4 Newtons / 21,000 Kilograms = **2e-4 m/s^2**

2. **Solar Pressure Perturbative Acceleration Magnitude:**
   - Compute Total Surface Area To Sun (Assume 50%) [A_t]
     - Solar Arrays - A_{sa} = 2 * (5 * 14) = 140 m^2
     - Transfer Vehicle Body - A_{tvb} = 2 * (.5 * 2) + (3 * 3) = 11 m^2
     - Netting Modules - A_{nm} = 4 * (2 * 2.7) = 21.6 m^2
     - Propulsion Module - A_{pm} = 2.93 * 2 = 5.86 m^2
     - Total Surface Area: $A_t = A_{sa} + A_{tvb} + A_{nm} + A_{pm}$ = 178.46 m^2
   - Total Area (cm) A = 1.7846e+6 cm. • Mass (gm) M = 2.1 e+7 gm
   - f = -4.5e-5 * A/M = 3.82414e-6 m/s^2 • a_{sp} = **3.8241e-8 m/s^2**

3. **Atmospheric Drag Perturbative Acceleration Magnitude:**
   - A/M = .085 • Cd = 2.0
   - Compute state r and v vectors at Space Station orbit.
   - H = 400 kilometers is the lowest altitude for the Transfer Vehicle.
   - Space Station Orbital Elements: a=6778.145 km., e=0
     i=28.5, Ω=0, w=0, M=0
   - Computed State Vectors: R = (-4864.9, 4555.153, -1281.306)km.
   - w_{earth} = 7.252e-5 rad/sec V = (-5.7619, 5.395, -1.5175) km./sec.
   - r_a = (-4.7034, 5.488, -1.5175) • ρ = 1e-12 kg/m^3 (est.?)
   - v_a = 7.3853 km./sec. • a_{drag} = **4.638e-8 m/s^2**

4. **J_2 Perturbative Acceleration Magnitude**
   - Debris Torus Orbital Elements: a=7540.645 km., e=.1
     i=29, Ω=300, w=200, M=0
   - Computed State Vectors: R = (-4864.9, 4555.153, -1281.306)km.
   - a_{J2} = **2.5621e-5 m/s^2** V = (-5.7619, 5.395, -1.5175) km./sec.

Thrust Magnitude Comparisons to Perturbative Accelerations

1. *Solar Pressure:* **5000 times**
2. *Atmosphere Drag:* **4300 times**
3. *Oblateness:* **8 times**

Appendix D Himawari 1 Rocket Booster Explosion

Satellite Data

Type: Delta Second Stage (2914)
Owner: US
Launch Date: 14.44 Jul 1977
Dry Mass (kg): 900 (approx.)
Main Body: Cylinder-Nozzle; 1.2 m by 5.8 m
Major Appendages: Mini-skirt; 2.4 m by .3 m
Attitude Control: None at time of the event
Energy Sources: On-board propellants, range safety devices

Event Data

Date: 14 Jul 1977
Time: 1612 GMT
Altitude: 1450 km
Location: 14N, 249E (dsc)
Assessed Cause: Propulsion-related

Post-Event Elements

Epoch: 77197.57445278
Right Ascension: 262.0317
Inclination: 29.0493
Eccentricity: .0973469
Arg. of Perigee: 66.7255
Mean Anomaly: 303.2693
Mean Motion Dot/2: .00007335
Mean Motion Dot Dot/6: .0
Bstar: .0

Cataloged Debris Cloud Data

Debris Cataloged: 168
Debris in Orbit: 93
Maximum delta P: 937 min*
Maximum delta I: 3.0 deg*

*Based on uncataloged debris data

Comments

This was the fifth Delta Second Stage to experience a severe fragmentation. It is also the only one which was not in a sun-synchronous orbit, which had performed a depletion burn, and which fragmented on the day of launch. This rocket body did perform its mission successfully, carrying the third stage and the payload into a low Earth orbit. The energy for the breakup is assessed to have been the 40 kg of propellants (mainly oxidizer) remaining after the depletion burn. The elements above are the first available after the event.

Reference

Gabbard, J.R.; Explosion of Satellite 10704 and other Delta Second Stage Rockets; Technical Memorandum 81-5; DCS Plans, Headquarters NORAD/ADCOM; Colorado Springs; May,1981.

Appendix E Fuel Calculation Program Listing

```
      PROGRAM FUELCOST
         REAL MNMDRY,MPM,MNV,MASS,M20,M40,M90,ISP,MFUEL,N20,N40,N90
      CHARACTER*1 ANSWER
100      PRINT *,'Input number of 20 cm holes'
      READ *,N20
      PRINT *,'Input number of 40 cm holes'
      READ *,N40
       PRINT *,'Input number of 90 cm holes'
      READ *,N90
      PRINT *,'Input mass of Netting Module (in kg)'
      READ *,MNMDRY
      PRINT *,'Input mass of Propulsion Module (in kg)'
      READ *,MPM
      MNV = MNMDRY + MPM
C
C input delta-v to get to debris (assume hydrazine-N2O2 w/ Isp = 318 s
C
      G = 9.8
       PRINT *,'Input delta-v to get each piece of debris (in m/sec)'
      READ *,DV
      ISP = 318.0
      DENSITY = 1200.0
      M20 = 1.06
      M40 = 4.23
      M90 = 21.43
      MASS = MNV + N20*M20 + N40*M40 + N90*M90
      RATIO = 1 - EXP(-DV/(ISP*G))
      DO 10 X = 1.0,N90
         MASS = MASS + MASS*RATIO
         MASS = MASS - M90
10    CONTINUE
      DO 20 X = 1.0,N40
         MASS = MASS + MASS*RATIO
         MASS = MASS - M40
20    CONTINUE
      DO 30 X = 1.0,N20
         MASS = MASS + MASS*RATIO
         MASS = MASS - M20
30    CONTINUE
       MASS = MASS + MASS*RATIO
       MFUEL = MASS - MNV
C
c take into account proximity ops. by adding 10% of fuel mass
c
       MFUEL = MFUEL + 0.1*MFUEL
       VFUEL = MFUEL/DENSITY
        PRINT *,'The mass of the fuel needed is ',MFUEL,' kg'
        PRINT *,'The volume of the fuel needed is ',VFUEL,' cubic ',
    t          'meters'
       PRINT *,'Would you like to try another configuration (y or n)'
      READ *, ANSWER
       IF (ANSWER.EQ.'Y') THEN
        GO TO 100
      ELSE
        STOP
      END IF
```