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Space Station Propulsion Requirements Study

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PREFACE

The Space Station Propulsion Requirements Study, NASA Contract NAS 3-23353, was managed by the NASA Lewis Research Center (LeRC) and was performed by the Flight Technology organization of the Boeing Aerospace Company in Kent, Washington. The LeRC contract monitors were Martin E. Valgora and Richard M. Donovan.

This final report is organized into the following documents:

Volume 1: Technical Report
Volume 2: Executive Summary

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The Rocket Research Company, under the direction of William W. Smith, provided analyses of hydrazine, resistojet, and arcjet thrusters under a subcontract.

TABLE OF CONTENTS

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TABLE OF CONTENTS (Cont'd)

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1.0 INTRODUCTION

1.1 Study Objective

The primary objective of the work described herein is to define propulsion system requirements to support Low Earth Orbit (LEO) manned Space Station development and evolution over a wide range of potential capabilities and for a variety of STS servicing and Space Station operating strategies. The term Space Station and the overall Space Station configuration refers, for the purpose of this report, to a group of potential LEO spacecraft that support the overall Space Station mission. The group consisted of the central Space Station at 28.5-deg or 90-deg inclinations, unmanned free-flying spacecraft that are both tethered and untethered, a short-range servicing vehicle, and a longer range servicing vehicle capable of GEO payload transfers. The time phasing for preferred propulsion technology approaches is also investigated, as well as the high-leverage, state-of-the-art advancements needed, and the qualitative and quantitative benefits of these advancements on STS/Space Station operations. The time frame of propulsion technologies applicable to this study is the early 1990's to approximately the year 2000.

1.2 Scope of Work

The work described in this study consists of four primary tasks. These tasks, which are presented in figure 1-1, define propulsion system requirements for a wide range of Space Station configurations, growth options, and servicing strategies. For certain servicing options, an additional propulsion system is required for a vehicle to transfer crew members and/or supplies from the STS to the Space Station. The propulsion requirements for the additional propulsion systems are examined for each servicing option. For certain experiment requirements, isolated experimental platforms are required. The propulsion requirements of these free-flyers are also examined.

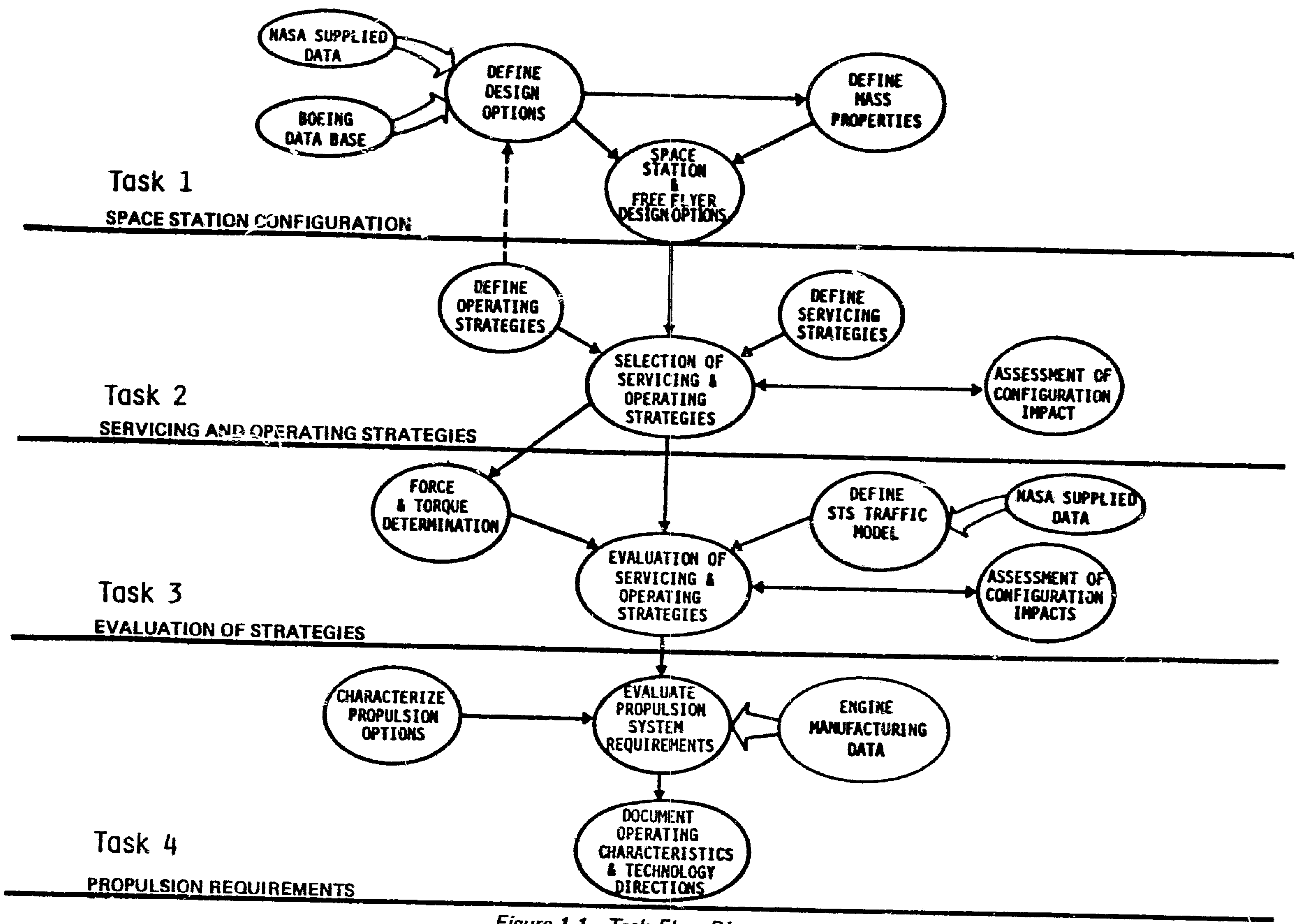

Figure 1-1. Task Flow Diagram

A parametric analysis is performed for a range of Space Station characteristics to determine the forces and torques acting on them and the resulting propulsion requirements for orbit maintenance, pointing control, and maneuvers in various LEO altitudes. Tradeoffs are performed for alternative servicing strategies, operating options, and candidate propulsion technologies to quantify their benefits and disadvantages in terms of propellant requirements, payload impacts, serviceability, maintainability, subsystem compatibility and safety.* These tradeoffs utilize the information from several mission models to evaluate the effects of variable traffic density on the benefits and result in a comparison of the alternative Space Station servicing and operating strategies from a propulsion standpoint.

1.3 Report Organization

There are seven major sections in this report, which encompass Tasks 1 through 4. The sections are briefly summarized by title and content as follows:

Section 1.0	Introduction	Overview and scope of the study.
Section 2.0	Configuration Selection	Possible station orientations, configuration families, design drivers, and recommended configuration designs.
Section 3.0	Configuration Effects	The principal environmental factors that affect Space Station design and propulsion requirements, station mass, aerodynamic and gravity-gradient forces, and momentum management.
Section 4.0	Servicing Strategies	Altitude selection, orbit decay influences, free-flyer servicing, and servicing vehicle capabilities.

*Cost trades are outside the scope of this study and are discussed only in a general sense with respect to propulsion options at the end of the report.

Section 5.0	Propulsion Requirements	Space Station, free-flyer, and servicing vehicle propulsion requirements, based on the recommended servicing strategies.
Section 6.0	Propulsion Systems Analysis	Propulsion system options, capabilities, and prospects for development.
Section 7.0	Recommendations	Selection of preferred propulsion system(s) and future Space Station development.

1.4 Space Station and Free-Flyer Propulsion Applications

The purpose of Space Station and free-flyer propulsion systems is to maintain orbital altitude over long periods of time and to assist in, or enable, attitude control, servicing operations, and emergency maneuvers as dictated by mission requirements and operational strategies. This study defined ranges of the resulting propulsion requirements by examining (1) potential Space Station configurations, (2) the effects of configuration designs on aerodynamic drag, attitude control torques, and control moments, and (3) Space Station and free-flyer servicing considerations including alternative operational strategies. Based on these requirements, the study examined a wide range of propulsion technologies to assess their applicability to this service and to summarize their advantages and disadvantages.

1.4.1 Normal Operations

The Space Station and free-flyer propulsion systems are primarily required to perform orbit maintenance. Attitude control is normally provided by momentum management devices, but the propulsion system is required for backup and desaturation. Orbit maintenance is required to counteract aerodynamic drag forces, which cause orbit decay. These forces vary considerably depending on atmospheric density and vehicle size. Only propulsive means for orbit maintenance are considered in this study. Other techniques, such as a combined Station reboost/Orbiter deboost using a tether, are not considered.

Space Station and free-flyer attitude must be maintained to ensure proper pointing of the solar arrays and to accommodate on-board experiments. Therefore, asymmetric forces causing aerodynamic torques and gravity-gradient effects must be overcome. These torques can be countered (1) propulsively by using a thruster on a moment arm from the center-of-mass, (2) by using torque rods that interact with the Earth's magnetic field to impart a torque to the vehicle, and (3) by using momentum management devices (MMD). The MMD's are especially useful for countering cyclic torque but will eventually become saturated if the torques are non-cyclic or have a non-cyclic component. Thrusters located on a moment arm or torque rods will be required to desaturate MMD's periodically.

Although one category of free-flyers investigated in this study is tethered to the station, other free-flyers require independent propulsion systems for orbit maintenance, attitude control, and servicing. The propulsion requirements for both normal and emergency operations for free-flyers are also analyzed in this study.

1.4.2 Emergency or Critical Operations

There are four emergency, or critical, situations that the Space Station will encounter: (1) disturbances due to docking the Orbiter with the station; (2) threatened collision with other orbiting objects, a runaway servicing vehicle, or an Orbiter; (3) the need to rescue an astronaut or a piece of equipment; and (4) safe end-of-life station disposal to a desired location on the Earth or to a higher altitude orbit. The Space Station propulsion system will be required for the first two situations. Rescue operations may be performed by the Orbiter, OMV, OTV, MMU, or a personal rescue system. The OMV may also be used for end-of-life station disposal. (See section 5.5 for a complete discussion of emergency or critical situations.)

1.5 Factors Determining Propellant Requirements

Propellant requirements are expressed in terms of mass, volume, and type. Propellant mass requirements depend on the free-flyer and station propulsion systems chosen, the specific impulse for the propulsion system utilized, and the total impulse necessary to accomplish propulsion requirements over the resupply interval.

Propellant storage volume requirements depend on propellant mass requirements and propellant density. Density, in turn, depends on propellant type and state (liquid or gas) and, in the case of a gas, storage pressure. If oxygen and hydrogen systems are used, water may be stored and electrolysized as required. If CO_2 effluent from the Environmental Control and Life Support System is used as it is produced, it could significantly reduce propellant storage and resupply requirements.

Propellant selection for the Space Station and free flyers is a complex issue influenced by numerous factors including thrust level, duty cycle, thrusting strategy, safety, reliability, system synergism, maintainability, commonality, and development risk.

2.0 REPRESENTATIVE CONFIGURATION SELECTIONS

2.1 Space Station Configurations

The configuration selection part of this Space Station Propulsion Requirements Study was conducted at a time when the NASA Headquarters Mission Analysis Studies were underway. At that time, the Space Station configuration data base included the "old" Phase B studies from the early 1970's, the Space Operations Center (SOC) and the Science and Applications Manned Space Platform (SAMSP) studies, plus concurrent Boeing IR&D configuration studies. The SOC and SAMSP design concepts are illustrated in figures 2-1 and 2-2, respectively. Both of these are variants of what is now called the "planar" concept. We began the study with a survey of the various station orientations and configurations. Although our study developed alternatives to the planar design, the fact that most prior Space Station studies considered only this design led us to develop a generic planar concept as our principal reference for analyzing propulsion requirements.

2.1.1 Space Station and Platform Orientations

The general placement of Space Station and platform components is based on the planned station orientation, the means for orienting the solar arrays to the Sun, and mission needs. Figure 2-3 illustrates the four station orientations investigated in this study. A fifth strategy of having no control over orientation was discarded as incompatible with Space Station/platform mission objectives.

In an inertial orientation, the Space Station attitude is fixed relative to inertial space. This strategy is ordinarily used as a means for Sun-tracking, but it can also be used as a target-tracking aid. (These functions often infer a flight mode that is not strictly inertial, but one that imposes similar propulsion requirements.) A constrained inertial orientation in which two of the vehicle's principal axes of inertia are in the orbit plane, offers relatively straightforward controllability for some configurations. This orientation permits inertial target-tracking devices

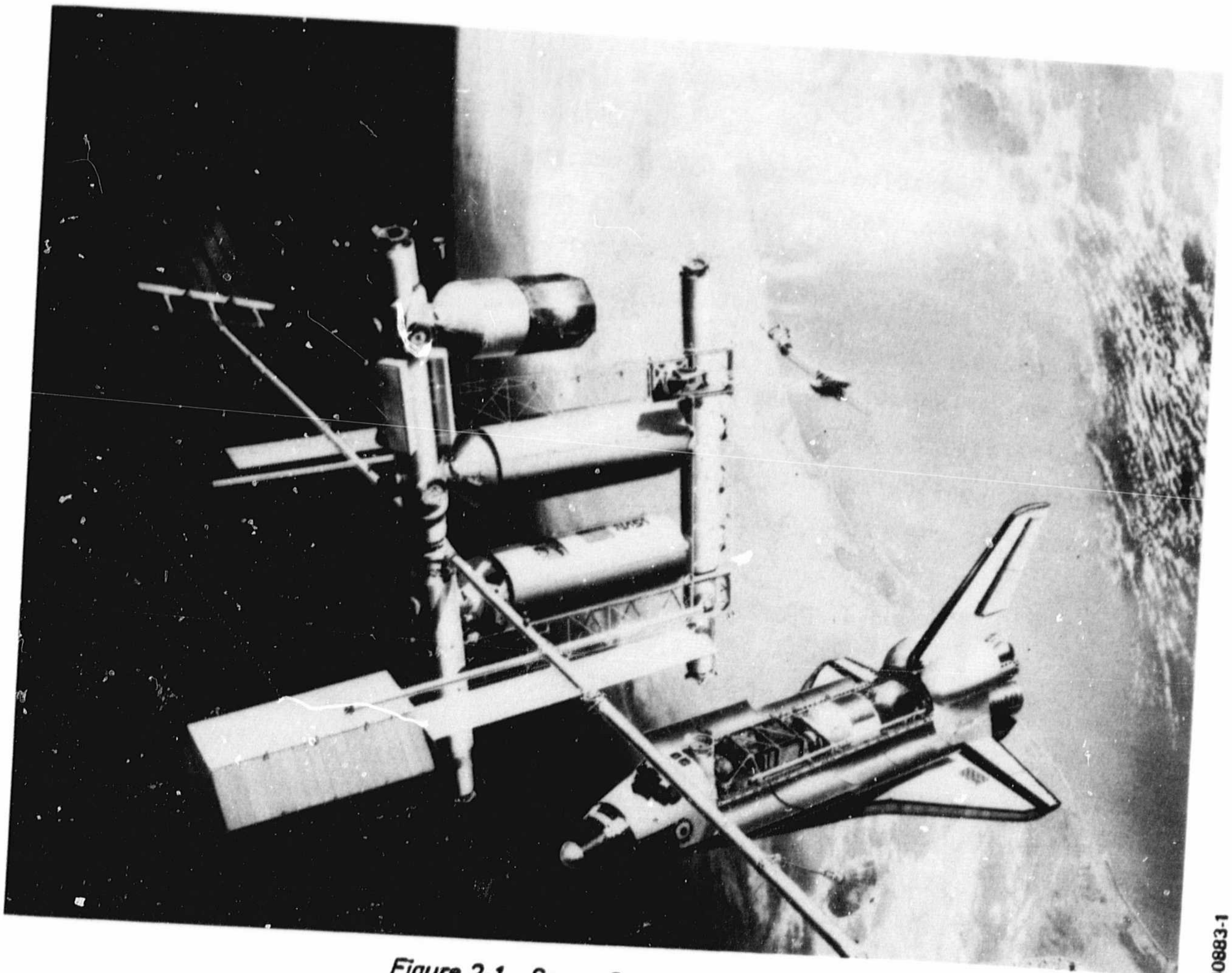

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Figure 2-1. Space Operations Center

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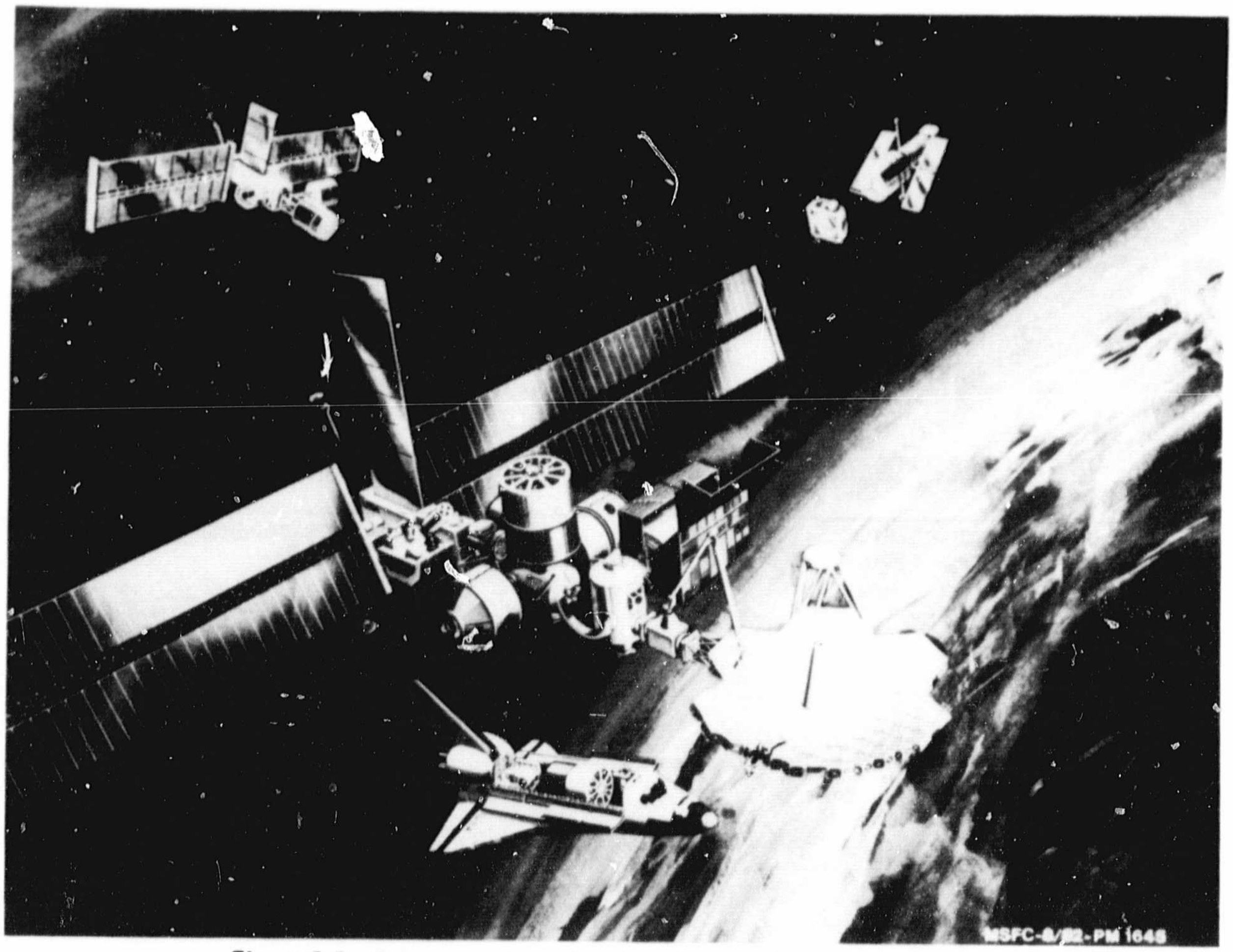

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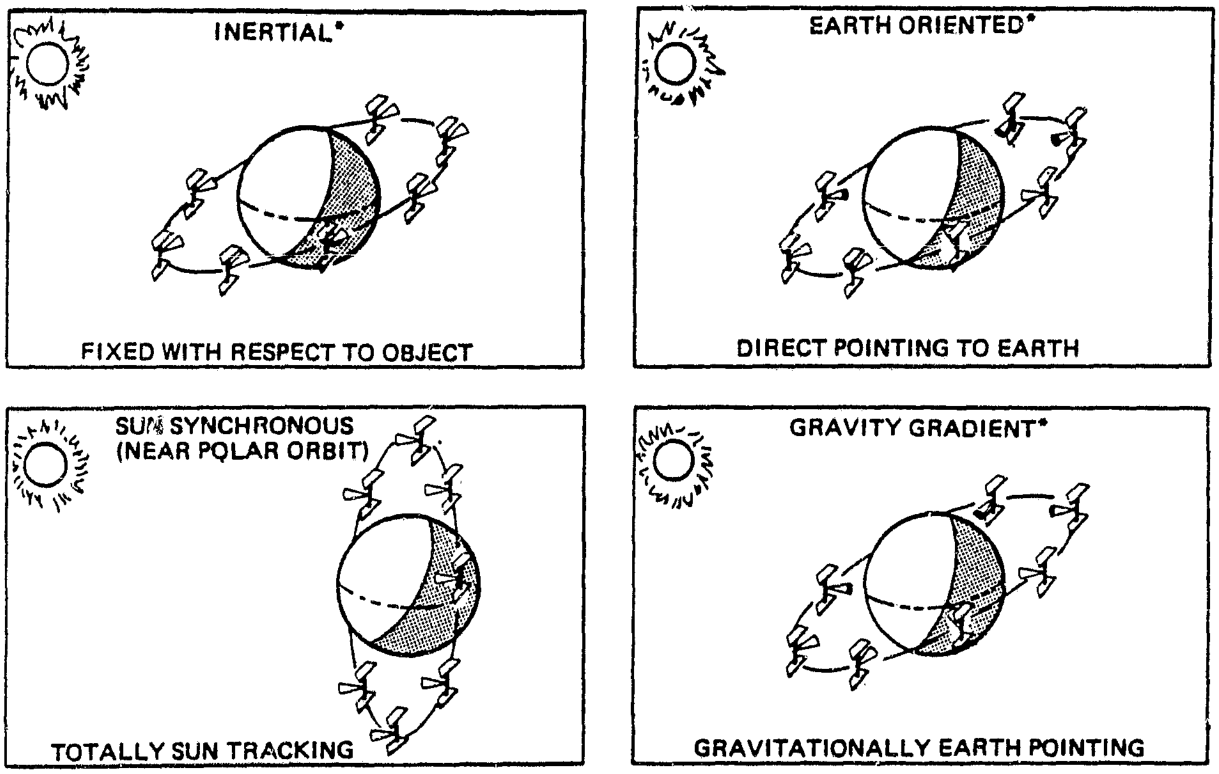
Figure 2-2. Science and Application Manned Space Platform



*LOW INCLINATION ORBIT, 28.5 deg

Figure 2-3. Station Orientations

to remain essentially fixed during any particular target-tracking session. However, an unconstrained inertial orientation, i.e., target-tracking by aiming the entire vehicle, poses difficult control problems. Skylab employed an inertial orientation mode that provided adequate Sun-tracking without solar array articulation. The JSC delta design shown in figure 2-4, which is a recent configuration option not analyzed in this study, employs a similar strategy. The Science and Applications Space Platform (SASP) concept developed by MSFC employed the inertial strategy to permit Sun-tracking with a single degree of freedom for seasonal variations ("beta-track") for the solar arrays. The beta-track is a slow oscillating motion that can be accommodated by flexible power cables; a rotary joint is not necessary. Rates and excursion angles depend on orbit parameters. Typical values are one cycle of $\pm$ 52 degrees amplitude for 60 days.

Earth-orientation indicates that a spacecraft maintains a fixed attitude with respect to the local vertical as it orbits the Earth. A strictly Earth-oriented spacecraft has fixed arrays that cannot track the Sun. A modified example of this is the JSC "Big T" configuration (figure 2-5), which provides a limited degree of Sun-tracking by tilting its large solar array platform. This design is not included in the current analysis because of its limited Sun-tracking ability. Most Earth-oriented configurations have articulated arrays that rotate in two axes: one axis moves very slowly as the beta angle changes with the season and orbit precession and the other axis rotates once per orbit. Stations that can be Earth-oriented are analyzed most fully in this study because a large portion of the missions require Earth pointing. As the section on environmental effects shows, torques and gravity-gradient effects are also more manageable in this orientation.

A gravity-gradient orientation is a special case of Earth orientation. The vehicle's inertial properties are arranged so that gravity-gradient forces maintain the desired Earth orientation. If solar array articulation is used for Sun-tracking, attitude-perturbing forces are introduced and must be offset. Figure 2-6 illustrates this design concept, which is similar to the "power tower" configuration. The "power tower" was identified by NASA as a reference configuration at the time of release of the RFP for Space

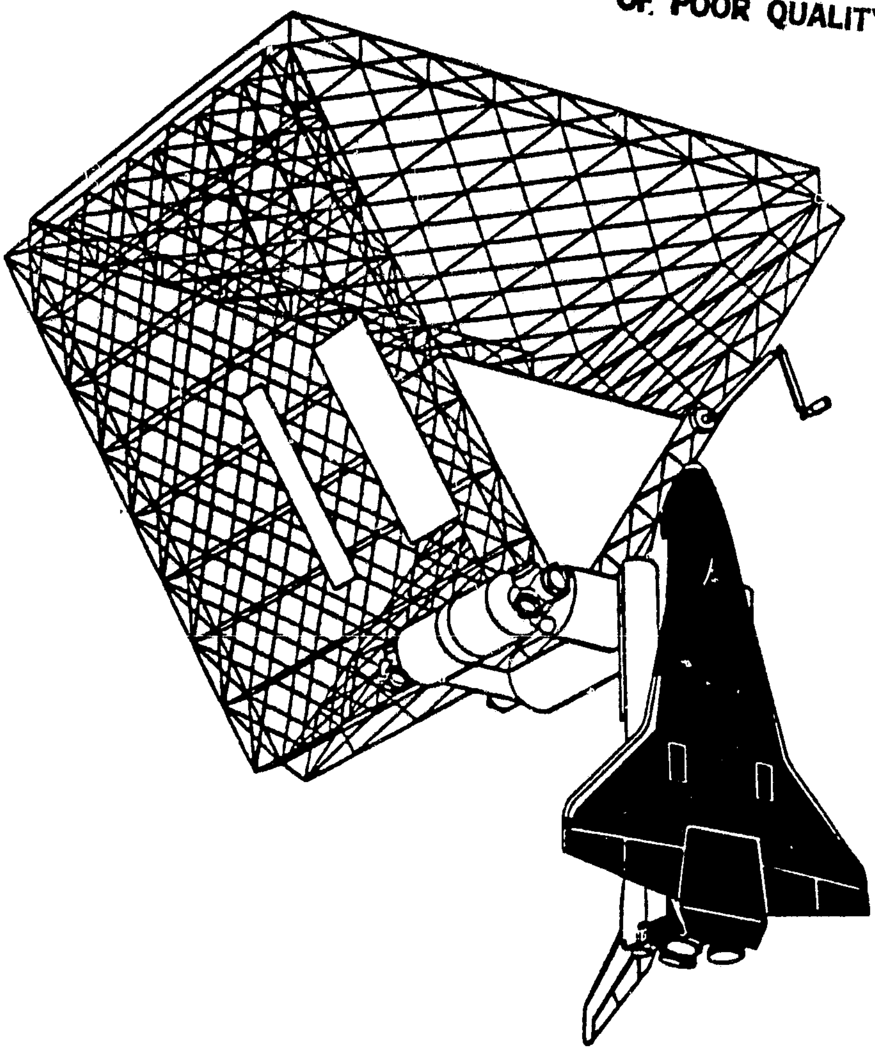


Figure 2-4. JSC Delta Configuration

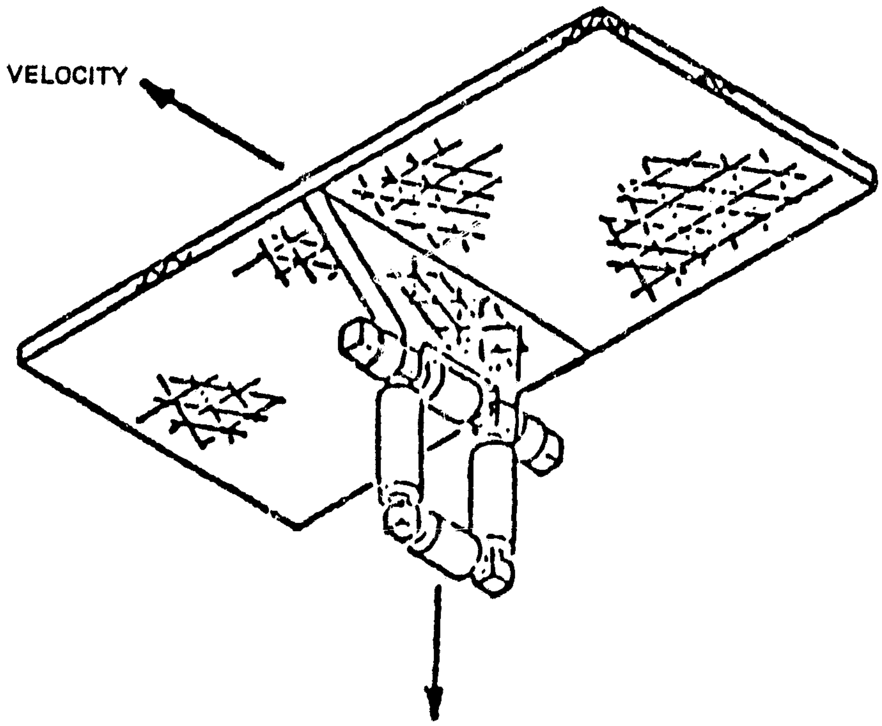


Figure 2-5. Streamlined Tee with Oversized Array

Station definition and preliminary design. Like the concept illustrated in Figure 2-6, it is designed to operate in a gravity-gradient stabilized flight mode. Solar array Sun-tracking, Earth oblateness, Earth-Sun-Moon triaxiality, and other gravitational anomalies cause a gravity-gradient-stabilized vehicle to have somewhat less precise pointing than the Earth-oriented configuration. A gravity-gradient Space Station may rely solely on gravity-gradient forces, and exhibit attitude perturbations as a result of perturbing forces. More often, this orientation means that the flight attitude is selected so as not to introduce any secular torques. In this case, the flight attitude will vary with changes in configuration, but the short-term attitude stability will be as good as for any other flight mode.

Sun-synchronous orbits, as shown in figure 2-3, are highly inclined to the Earth's equator so that Earth oblateness causes the orbit to precess synchronously with the Earth's motion about the Sun. The orbit line of nodes and the Sun's mean longitude maintain a constant angular relationship. If the longitude-node angle is near 90 deg, adequate Sun-tracking is afforded without array articulation. Figure 2-7 shows a Sun-synchronous configuration designed to operate over the terminator. Except for missions that require continuous solar observation, a Sun-synchronous orbit over the terminator offers few benefits. Most near-polar Earth-observation missions require a small meridian longitude line of nodes angle, which causes the spacecraft orbit plane to be nearly perpendicular to the terminator. Therefore, single-degree-of- freedom solar array articulation is appropriate. As far as propulsion requirements are concerned, the Sun-synchronous orbit over the terminator requires little orbit maintenance thrust because the Solar array is always aligned edgewise to the velocity vector.

2.1.2 Space Station Configuration Families

Figure 2-8 describes the logic that was used in this study to develop and narrow the range of Space Station options. The initial design assumptions, shown at the top of the diagram, were that the station be manned, powered by solar photovoltaic cells, and be delivered to orbit by the Space Trans-

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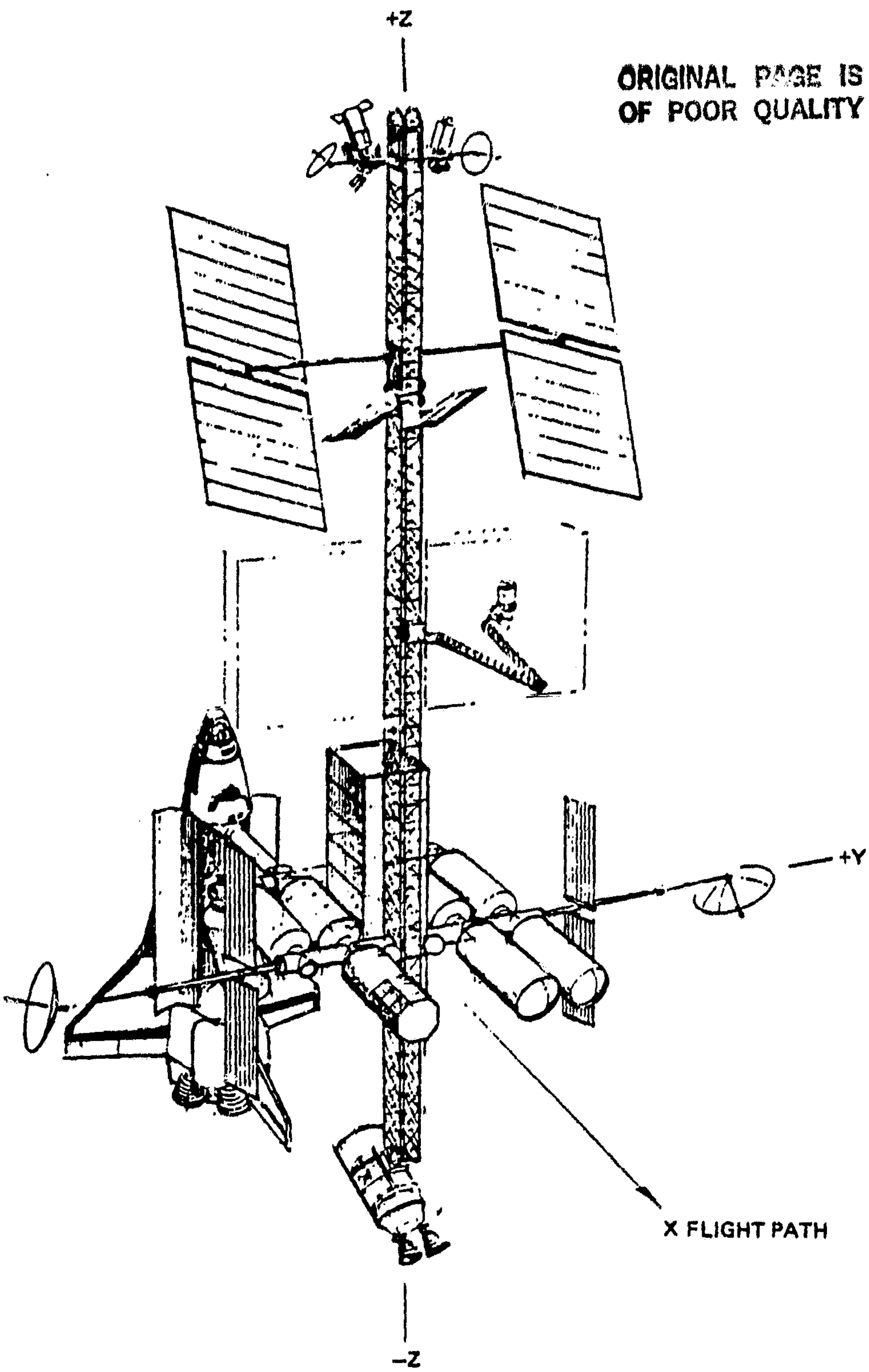

Figure 2-6. Gravity-Gradient Stabilized Station

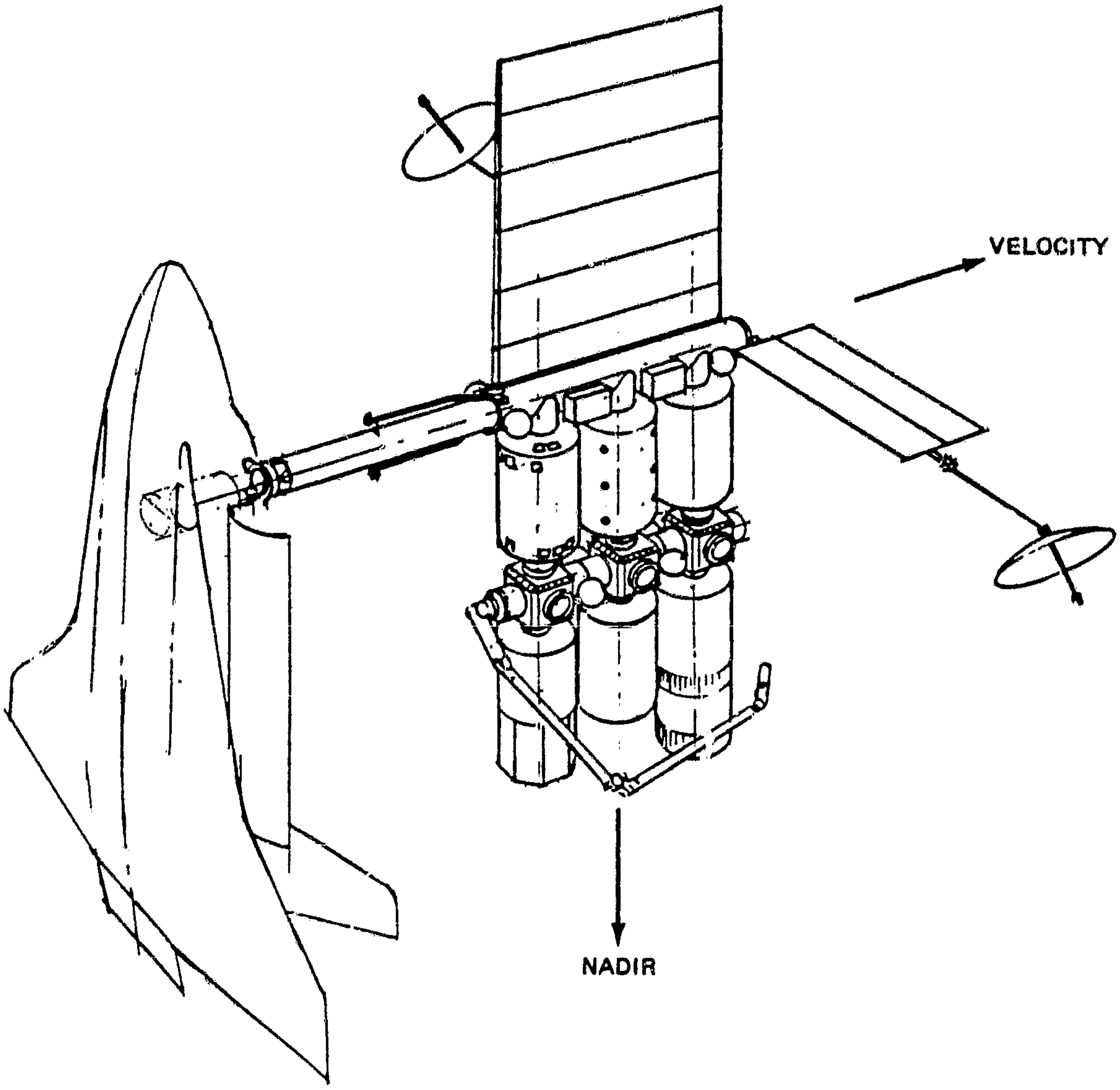

Figure 2-7. Sun Synchronous Station (Over Terminator)

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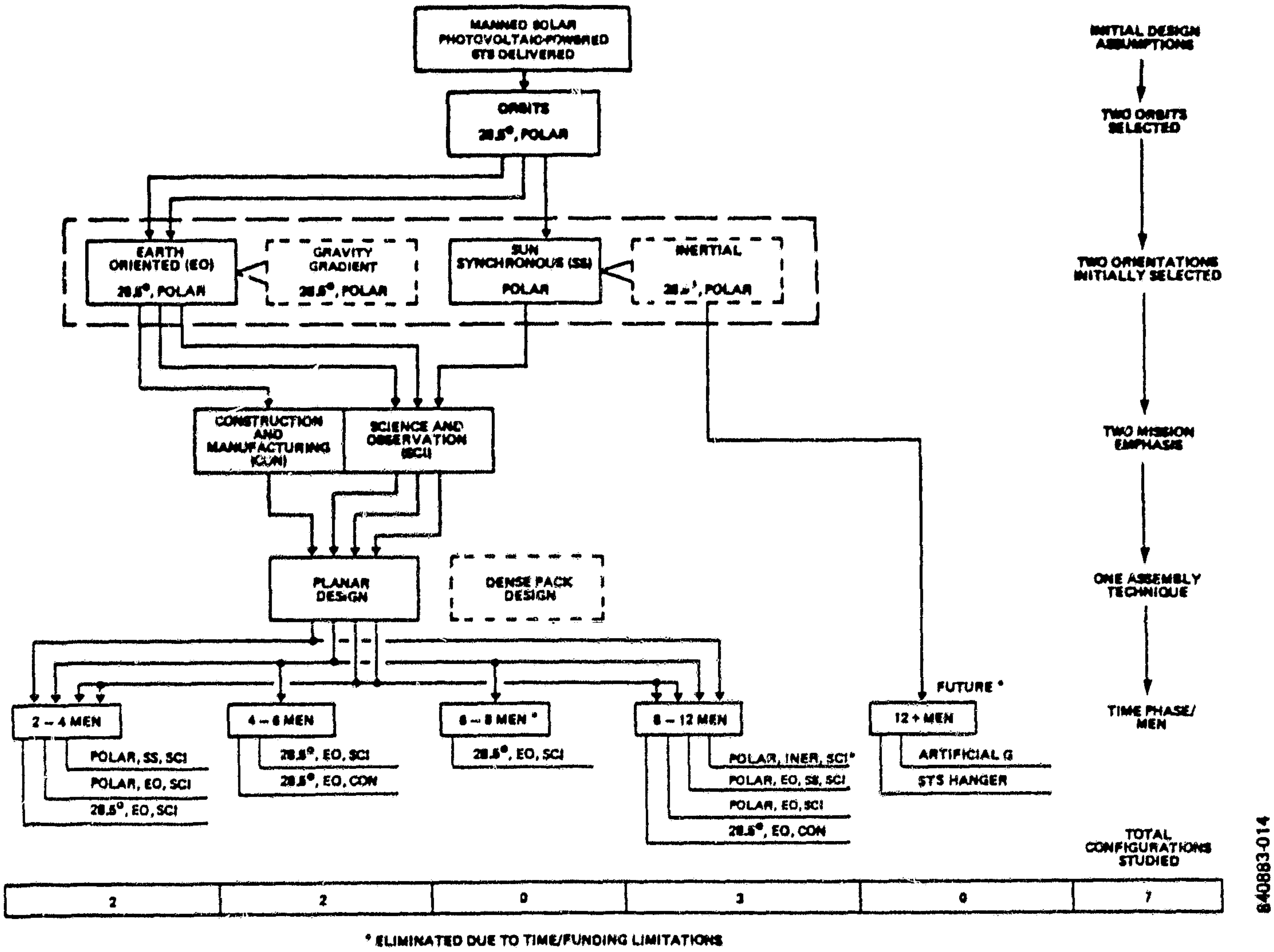


Figure 2-8. Configuration Selection Logic for Space Stations in Low Earth Orbit

portation System (STS). Only 28.5 deg and near-polar inclinations were chosen because they were the most probable in view of identified mission needs, and would thereby determine propulsion requirements. Higher inclination orbits differed primarily in terms of STS capability (see section 2.2.1) and beta tilt. Earth-oriented configurations at 28.5 deg and near-polar inclinations were given primary emphasis after the preliminary screening. We found no compelling mission needs to cause us to consider intermediate inclinations. Since all of our configurations included sun-tracking and therefore essentially inertial solar arrays, the data developed are applicable to inertial orientations also. We did not analyze passively controlled gravity-gradient-stable configurations, but the drag compensation parametric data in this report are appliable to them.

Space Station sizes are denoted here in terms of crew number and solar array size. For the two stations larger than 2 to 4 men that are deployed at a 28.5 deg inclination, both scientific and construction variants are defined. All other stations have only the scientific variant. The construction station has more platform area for construction and a larger servicing hangar, which facilitates EVA in a controlled thermal environment with uniform lighting and object containment. Figure 2-9 shows an example of a construction station core for an 8- to 12-man crew. The differences between the scientific and construction stations, in terms of factors affecting propulsion requirements, turn out to be small compared to the influences of the solar arrays and a docked Orbiter. Therefore, some of the analyses done in this study do not differentiate between the science and construction stations. Where propulsion requirements are significantly affected, the two types of station are delineated. Due to time and funding limitations, the 6- to 8-man station size (for both types) was dropped, as well as the future 12+ man station size.

Early in the development of configuration options, it was believed that the arrangement of the modules comprising the station core would have a significant influence on propulsion requirements. Two design options were considered: dense-pack and planar. The dense-pack modular design could expand in any direction to produce a symmetric package of modules in the densest possible arrangement. It was thought this would create less severe

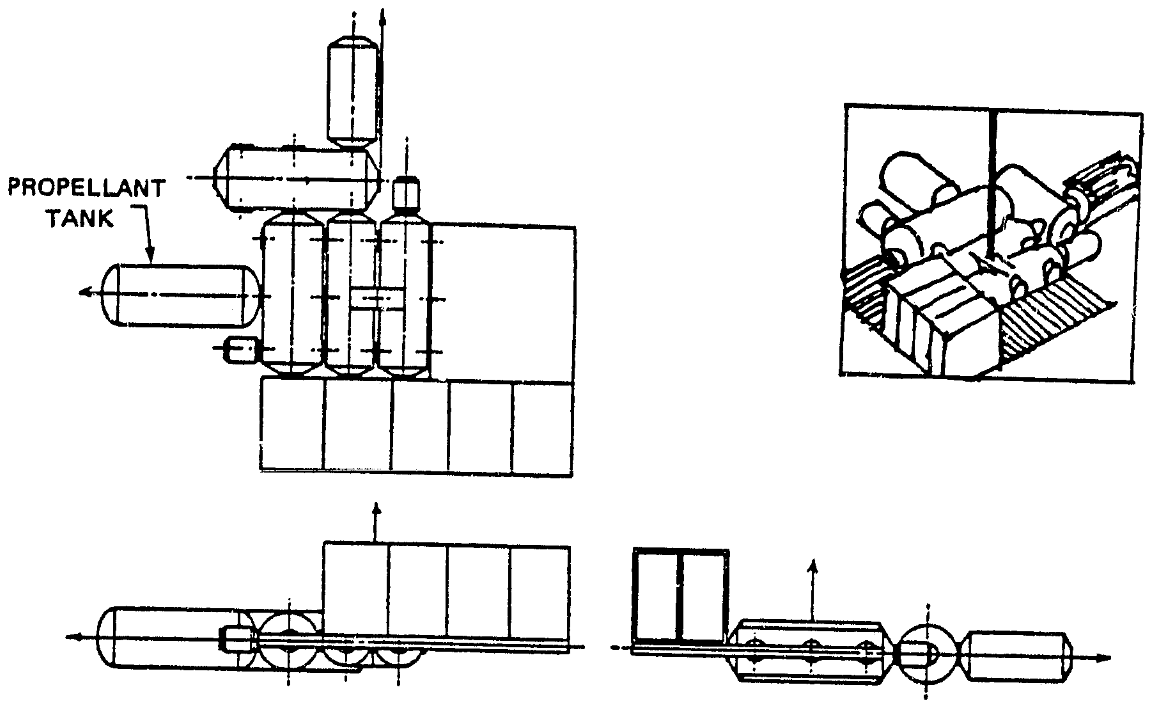
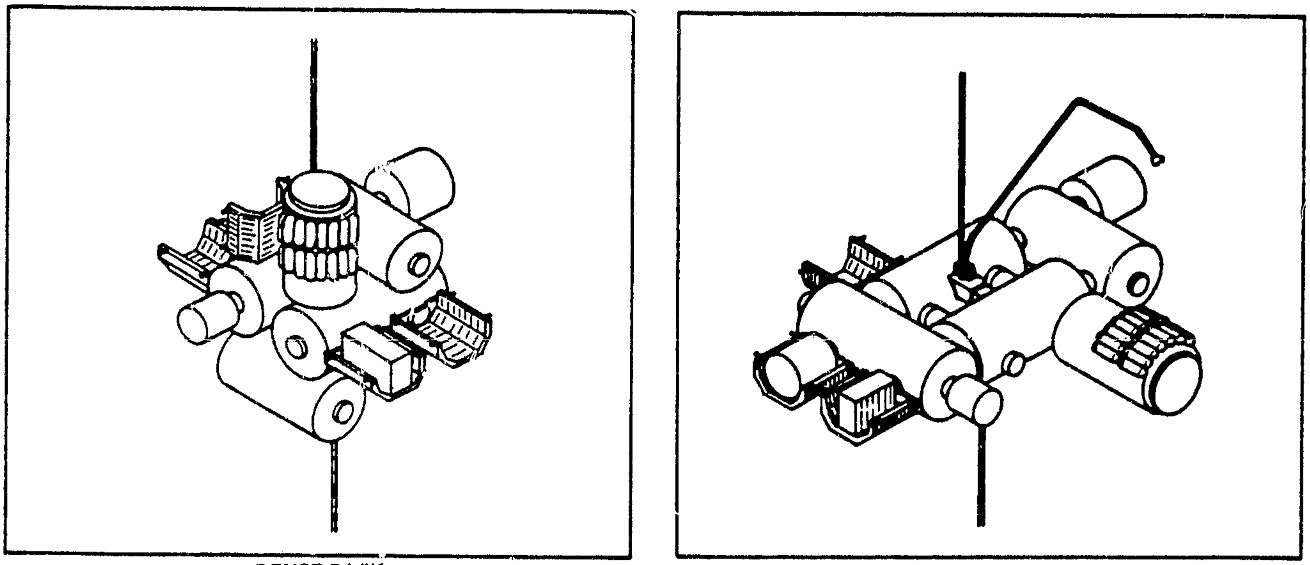


Figure 2-9. 8-12 Man, 28.5 Deg., Earth-Oriented, Construction/Manufacturing Station Core

attitude control requirements. However, the planar option was found to have better thermal and viewing capabilities from within the modules, was easier to assemble, was more compatible with onboard mobility systems, and could more easily incorporate two means of egress, which is a safety criterion. Therefore, the planar core arrangement was used in this study. The specific planar module arrangements for different crew sizes are illustrated in figures 2-10 and 2-11.

The impact of solar array configuration design on propulsion and attitude-control requirements was not fully realized at the outset of the study. However, after configuration effects were analyzed, as described in section 3.0, it became apparent that the aerodynamic and gravity-gradient balance of individual designs, in terms of torques and cross products of inertia, were key configuration considerations. It was found that propellant requirements for momentum management and torque cancellation were driven to extremes by solar array configurations that neglected balance. The two key design factors were found to be solar array design and Orbiter docking location and their effects on gravity-gradients. Drag was a key contributor to propulsion requirements for station altitudes below 450 km.

When this study was initiated, most Space Station configurations included a cantilevered array design, as shown in figure 2-12. The purpose of the cantilevered design was to keep the arrays as far away as possible from the core to minimize shadowing and prevent interference with core operations. The long axes of the solar array masts were perpendicular to the orbit plane and the arrays rotated around this axis relative to the Space Station body to track the Sun. The arrays also tilted $\pm$ 52 deg (for 28.5^0 orbit) with respect to that axis to provide beta tracking of the Sun. During the course of this study, it was found that this assymmetry with respect to the orbital plane with a high beta tilt created control problems due to cross products of inertia in the inertia tensor which resulted in large secular gravity-gradient torques. The cross-products-of-inertia contribution exceeded other control and propulsion requirements by roughly an order of magnitude and dominated the entire control issue.



DENSE PACK PLANAR

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Figure 2-10. Core Growth Options

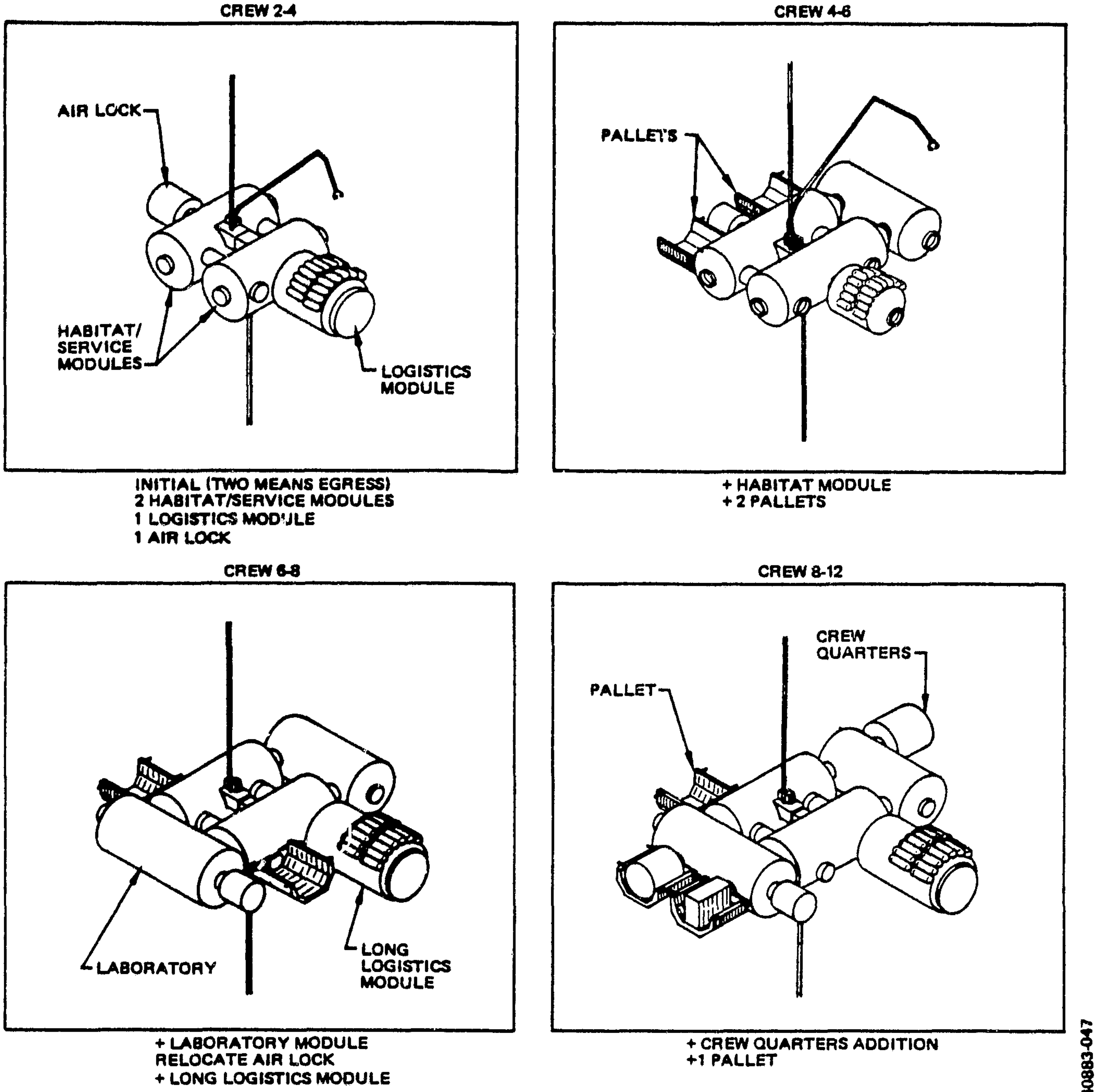


Figure 2-11. Planar Growth Option

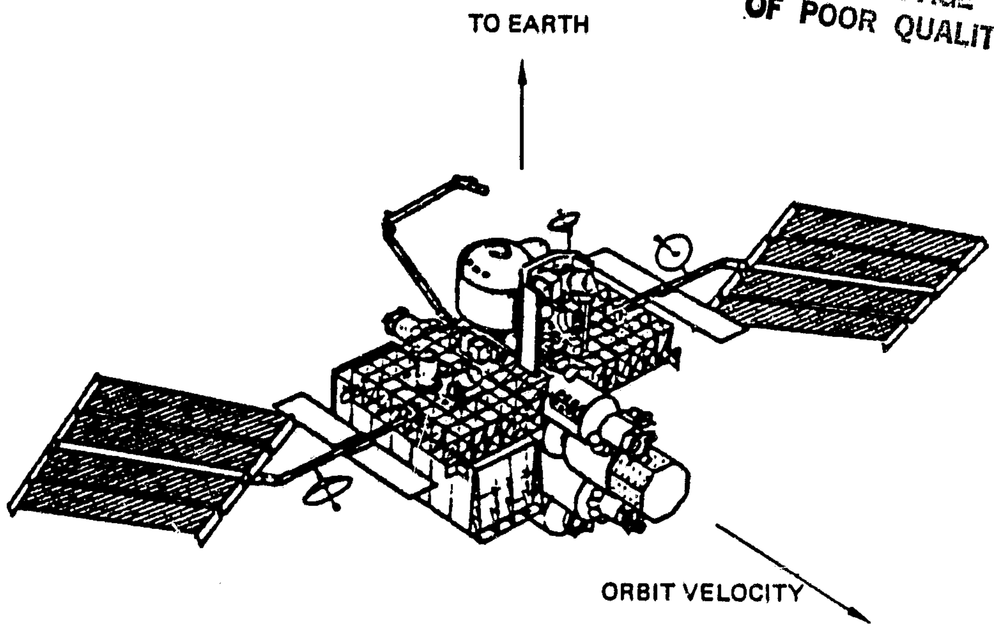


Figure 2-12. Earth-Oriented Cantilevered Array Station

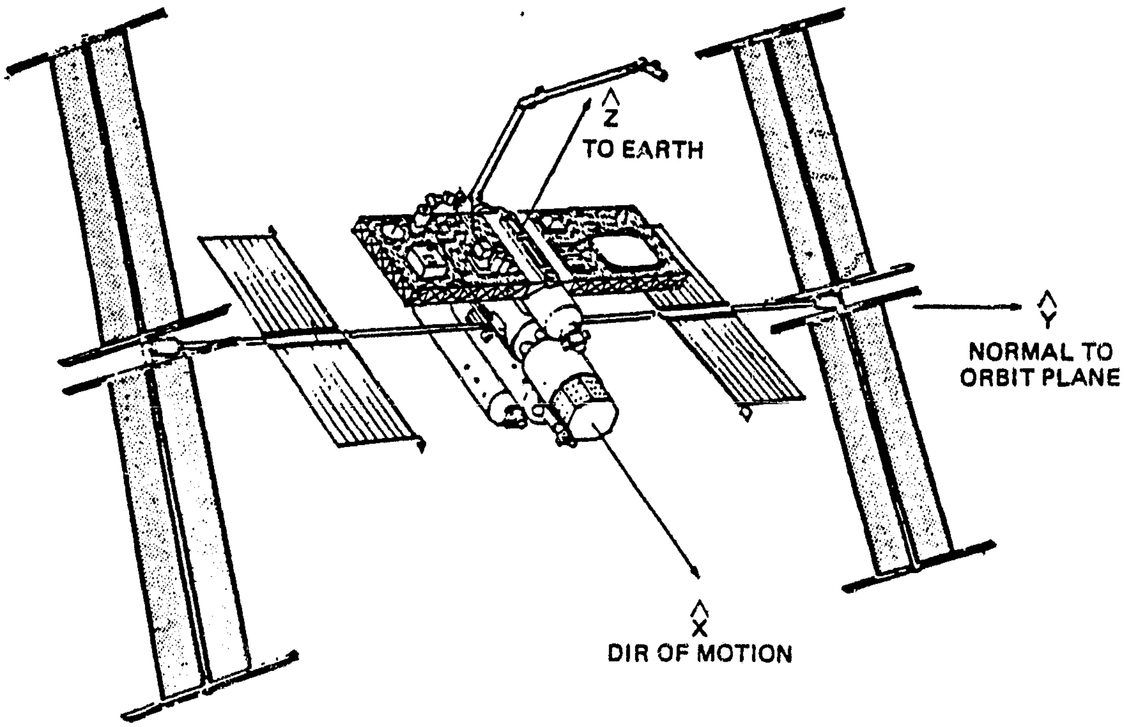


Figure 2-13. Balanced Solar Array Concept

The balanced solar array configuration, shown in figure 2-13, provides array symmetry with respect to the orbital plane. The rotational degrees of freedom are the same as for the cantilevered solar array configuration, but the rotational motion of the balanced solar arrays is more like a paddle wheel motion. The balanced array center of gravity is very close to the hinge point. The only contribution to inertial cross-products is due to array tilting, not motion of the center of gravity. This configuration reduces the cross products of inertia by roughly a factor of 10. The cantilevered array configuration is carried through much of the analysis in this report because it is so prevalent in the literature.

A special case of inertial orientation is one in which all or a portion of the Space Station spins as is the case for the concept illustrated in Figure 2-14. Spinning may be invoked to provide artificial g as in the illustration or to provide inertial "stiffness" through great angular momentum as in the "Hughes spinner" concept reviewed by the NASA Space Station Concept Development Group in the summer of 1983. Great angular momentum is present in either case and the station flies with its spin axis essentially inertially oriented. This configuration category was not evaluated in this contract.

2.1.3 Free-Flyers

Free-flyers are single or multipurpose experimental satellites that co-orbit with the Space Station and are tended by the servicing vehicles. Although it provides useful services to a payload, a Space Station can disturb the payload by causing induced gravity, vibration, thermal cycles, gas release, and electromagnetic interference. To obtain the benefit from Space Station presence but not be disturbed by it, free-flyers can be used as isolated platforms for experiments and observation. They will have independent pointing, power, safety, and acceleration requirements.

To ensure that a free-flyer remains in a relatively fixed position with respect to the Space Station, it must be in the same orbit as the station, either ahead of or behind it. (Other relative positions may be maintained by propulsion or by tethers. Required propellant consumption renders the

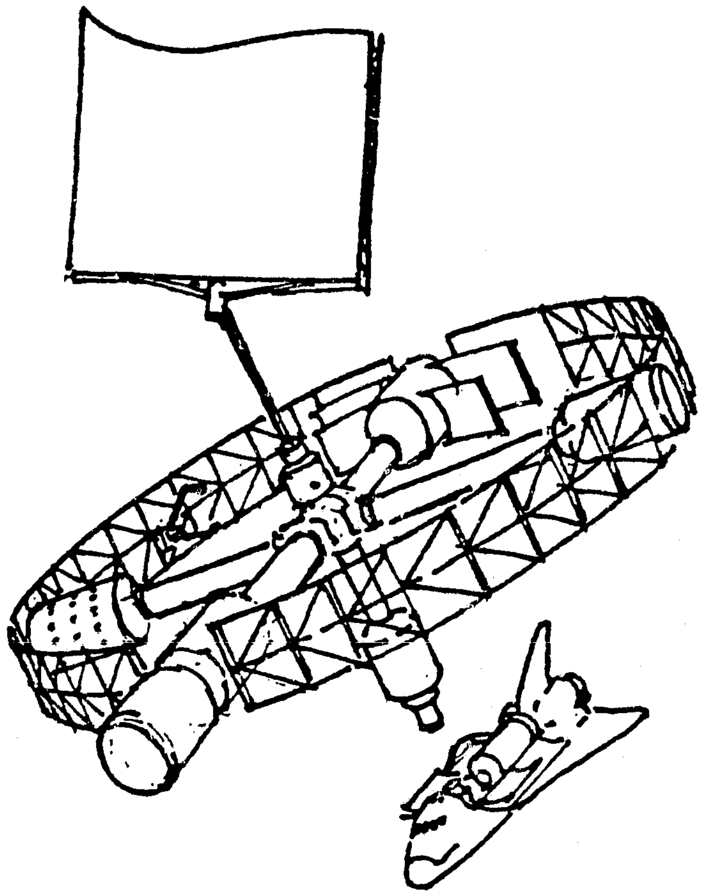


Figure 2-14. Future Station STS/Station Docking

propulsion alternative impractical. Tethers are discussed below.) Propulsion is also necessary to compensate differential orbit decay due to differences in ballistic coefficients. If there are a number of payloads that need to be near the station, they can be in slightly eccentric orbits that circle the station once per orbit. This study assumes the distance between the free-flyers and the station core can range from a few kilometers to the limit of direct line-of-sight and direct radio communication.

This study includes an analysis of the following potential free-flyer platforms that could accompany a Space Station: (1) a free-flying propellant farm; (2) a tethered propellant farm; (3) a slack-tethered power generation module(STPGM); and (4) a science and applications space platform (SASP). Specific information on configurations and operations follows in sections 2.2.2 Mission Requirements, and 2.3.7 Free-Flyer Configurations.

In addition, NASA has identified a number of free-flyers that could be supported by the Space Station. Table 2-1 lists these free-flyers, their deployment altitude, inclination, mass, servicing frequency and number of times deployed. We used these NASA free-flyers to generate an "unrestrained" traffic model to define STS and servicing vehicle usage and propellant requirements over the life of the station. Unrestrained in this sense means that the traffic model is not constrained by the Orbiter fleet size NASA currently projects. Instead, the traffic model generates the Orbiter fleet size and flight frequency based on free-flyer servicing projections. This is covered in greater detail in section 4.2.4.

2.2 Design Drivers

Factors that directly or indirectly affect Space Station propulsion requirements are:

a. STS capability, which affects Space Station module volume and mass, as well as orbit direction, inclination, altitude, and STS servicing frequency.

Table 2-1. NASA–Defined Free Flyers

NASA CODE	TITLE	SUBTITLE	ALT (KM)	INCL (DEG)	MASS (TONNES)	SERV FREQ (TIMES/YEAR)	NO. DEPLOYED (1991 - 2000)
SAAX0010	SOLAR CORONA	DIAGNOSTICS MISS	525.	28.5	1.	2.00	1
SAAX0012	SPACE TELESCOPE		600.	28.5	11.	2.00	1
SAAX0013	GAMMA RAY	OBSERVATORY	400.	28.5	14.	0.50	1
SAAX0014	XTE-XRAY TIMING	EXPLORER	400.	28.5	1.	0.50	1
SAAX0015	OPEN		525.	0.0	1.	0.00	—
SAAX0017	ADVANCED X-RAY	ASTROPHYS FACI	500.	28.5	10.	0.30	1
SAAX0018	VY LONG BASELINE	INTERFER DEMO	400.	57.0	1.	0.00	1
SAAX0019	FAR UV SPECTROSC	EXPLORER	500.	28.5	1.	0.00	1
SAAX0020	LARGE DEPLOYABLE	REFLECTOR	700.	28.5	27.	1.00	1
SAAX0022	SOLAR DYNAMICS	OBSERVATORY	400.	99.0	5.	0.00	1
SAAX0205	GOES FOLLOW-ON		35700.	0.0	1.	0.00	1
SAAX0501	EXPERIMENTAL GEO	PLATFORM (XGP)	35700.	0.0	5.	0.00	1
COMM1019	MULTI-LINEAR	ARRAY STEREO	900.	98.0	0.	0.50	1
COMM1023	STEREO SAR	MLA & CZCS	900.	98.0	2.	0.00	1
COMM1110	CENTAUR CLASS	COMM SAT-DEL	35700.	0.0	6.	1.00	9
COMM1115	IUS CLASS	COMM SAT-DEL	35700.	0.0	3.	4.00	37
COMM1116	PAM-A CLASS	COMSAT-DEL	35700.	0.0	1.	0.00	16
COMM1117	PAM-D CLASS	COMSAT-DEL	35700.	0.0	1.	3.00	23
COMM1124	CENTAUR CLASS	COM SAT-SERV	35700.	0.0	5.	0.00	7
COMM1125	IUS CLASS	COM SAT-SER	35700.	0.0	3.	0.00	8
COMM1126	PAM-A CLASS	COM SAT-SERV	35700.	0.0	1.	0.00	4
COMM1131	PAM-D CLASS	COM SAT-SERV	35700.	0.0	1.	0.00	1
TDMX2230	SPACE INTERFER	OMETER SYST TECH	400.	50.0	0.	3.00	3
TDMX2320	LOW THRUST	PROPULSION	525.	28.5	0.	6.00	2
TDMX2460	TELEPRESENCE	TECHNOLOGY	525.	28.5	1.	4.00	4

b. Mission requirements, which encompass the types of experiments, observations, and operations that will be performed.

c. Crew size, which will be dictated by the space available and workload.

d. Power levels, which will depend on experiment requirements, crew size, technology, funding, and power generation limits.

e. Servicing vehicle configurations, volume, and mass required for propellant storage, and

f. Free-flyer support requirements.

These six major factors are described in more detail in the following sections.

2.2.1 STS Capability

The Space Station must be deployed and serviced by the Space Transportation System (STS). STS servicing frequency is governed by fleet size and relaunch turnaround capability. The 90-day maximum servicing interval specified by NASA is used in this study.

There are two means for effecting a rendezvous between the station and the Orbiter (the STS without the external tank and solid rocket booster): nominal and direct insertion. Figure 2-15 illustrates the STS payload capability for each as a function of orbit altitude. This figure is a projection by JSC of STS capabilities in the 1990's. Performance assumptions are indicated (e.g., 109% SSME). Nominal insertion describes the technique used by all of the earlier STS launches. It is accomplished by terminating SSME thrusting short of attaining orbital velocity, discarding the external tank (ET), and achieving the final orbital velocity using the Orbital Maneuvering System (OMS). The disadvantage of nominal insertion is

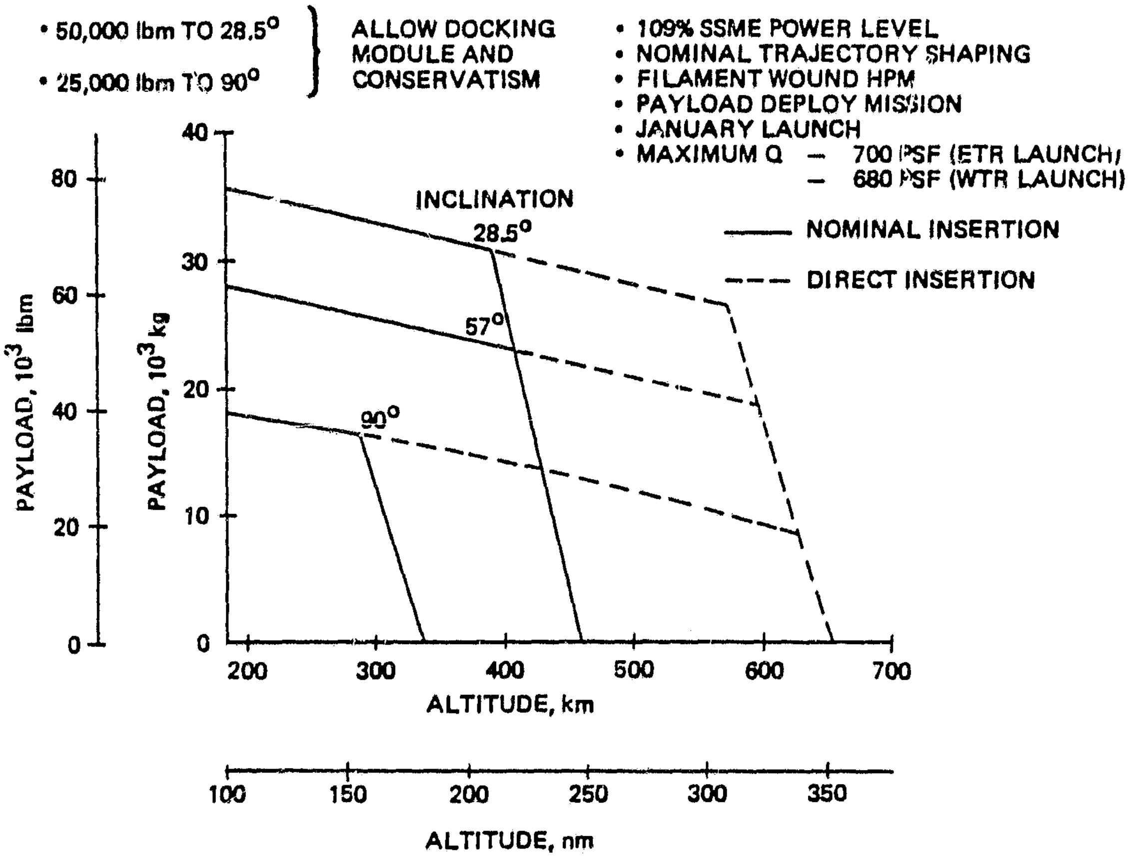


Figure 2-15. STS Delivery Altitude and Mass Capability

that payload capacity drops drastically above 400 km to zero at 460 km. With nominal insertion, SSME cutoff occurs at a slightly suborbital condition selected for external tank disposal. The first OMS burn immediately after tank jettison raises the insertion apogee to 160 km. A second burn circularizes at that altitude and subsequent burns are used to transfer to mission altitude and deorbit at the end of the mission. The sharp slope discontinuity of the performance curves in Figure 2-15 occurs at the OMS propellant capacity limit.

Direct insertion, on the other hand, continues SSME thrusting until transfer velocity to the mission orbit is attained, after which the ET is discarded. This conserves OMS propellant. STS payload capacity above 400 km is much greater with direct insertion than with nominal insertion, but there is a much larger "footprint" on the Earth's surface for ET impact. Direct insertion was used on the 12th STS flight and is now considered a baseline technique by NASA. It is consequently used as the baseline in this study. Since most of the STS payloads in support of the station may be volume, not mass, limited, direct insertion will usually permit launches with no payload constraints to the lower limit of the Van Allen belt (about 550 km). Advantages of a higher orbit are discussed in sections 3.0 and 4.0.

Payload volume is limited by Orbiter payload bay size. Figure 2-16 shows the length and diameter constraints on STS payloads. The 18.29m length shown is reduced to 14.6m when the docking module is included. This module is needed during assembly and for any crew transfer operations to or from the station. Figure 2-17 shows the available length, mass, and allowable center-of-gravity locations for an Orbiter equipped with a docking module. The module is assumed to be 3.5m in diameter and to weigh approximately 1800 kg (4,000 lbm).

2.2.2 Mission Requirements

Several types of scientific experiments may be conducted on the Space Station. Solar-scientific missions should be positioned on the Space Station to provide optimum solar viewing (i.e., the entire period that the

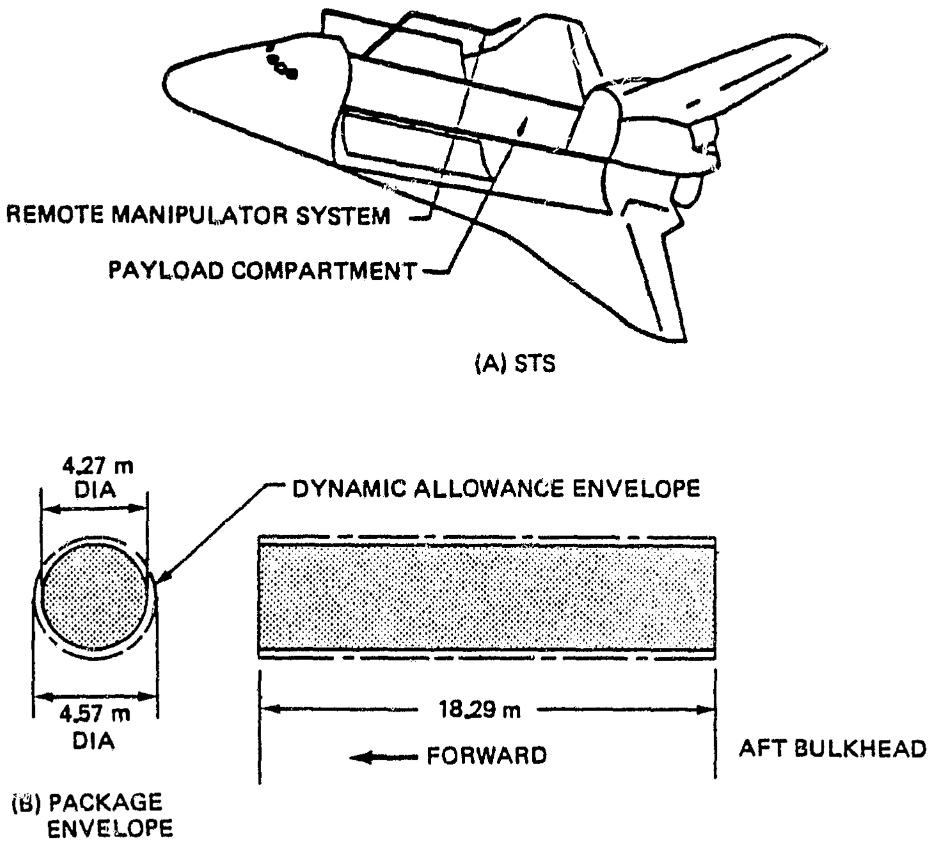
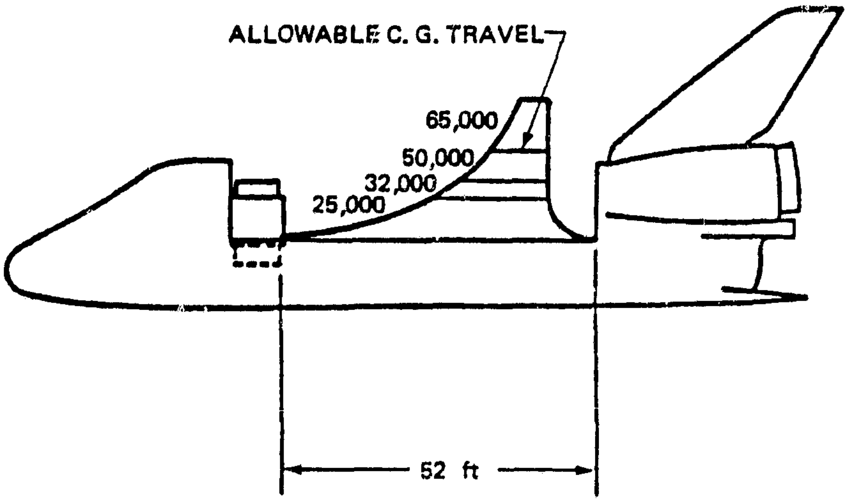


Figure 2-16. STS Volume Constraints



ORBITER WITH DOCKING MODULE

- ASSUME MODULES POSSESS UNIFORM DENSITY
- SINGLE MODULE (DOUBLE MODULES NEED DIFFERING DENSITIES)
- UNITS ARE lbm

Figure 2-17. Orbiter Payload Capacity With Docking Module station is not in the Earth's shadow). Other scientific payloads require stellar or Earth viewing, either periodically or continuously. Whether the science experiments are conducted in a laboratory module in the Space Station or on a free-flyer, these laboratories may have to be removed periodically and returned to Earth for updating and reconfiguration. Certain experiments will be particularly "g" sensitive, and may not be able to tolerate even the $< 10^{-4}$-g disturbance estimated from crew movement onboard the station, or a possibly higher g level associated with the thrust level chosen for orbit maintenance. Another design factor that impacts scientific missions is that sensitive instruments will have to be totally protected from sources of contamination. This is discussed more fully in section 6.0 on propulsion systems.

2.2.3 Crew Size

The number and timing of crew support for the Space Station is still being defined in terms of mission objectives. Figure 2-18 illustrates the projected number of crew members through the year 2005 assumed for this study. The initial station would house 2 to 4 crew members, and would subsequently increase to 6 to 8 by 1994, 8 to 12 by 1995, and would peak at 16 by the year 2000. The increase in crew size is based on the demand for materials processing and experimentation.

2.2.4 Power Levels

For this study, station power requirements were expressed in terms of array and net station power. The former is discussed in detail in section 2.3.5 under Solar Array Parametrics, and is the power the solar array must produce to provide the net bus station power. Net station power is the power that will be available for station operations. Power level requirements depend on mission requirements and crew size.

The projected net power requirements for the Space Station at a low inclination (28.5 deg) were assumed to begin at relatively low levels (20 to 25 kW), increasing to over 115 kW. Figure 2-19 shows the projected net power requirements by year through 2005 for both low and high inclinations.

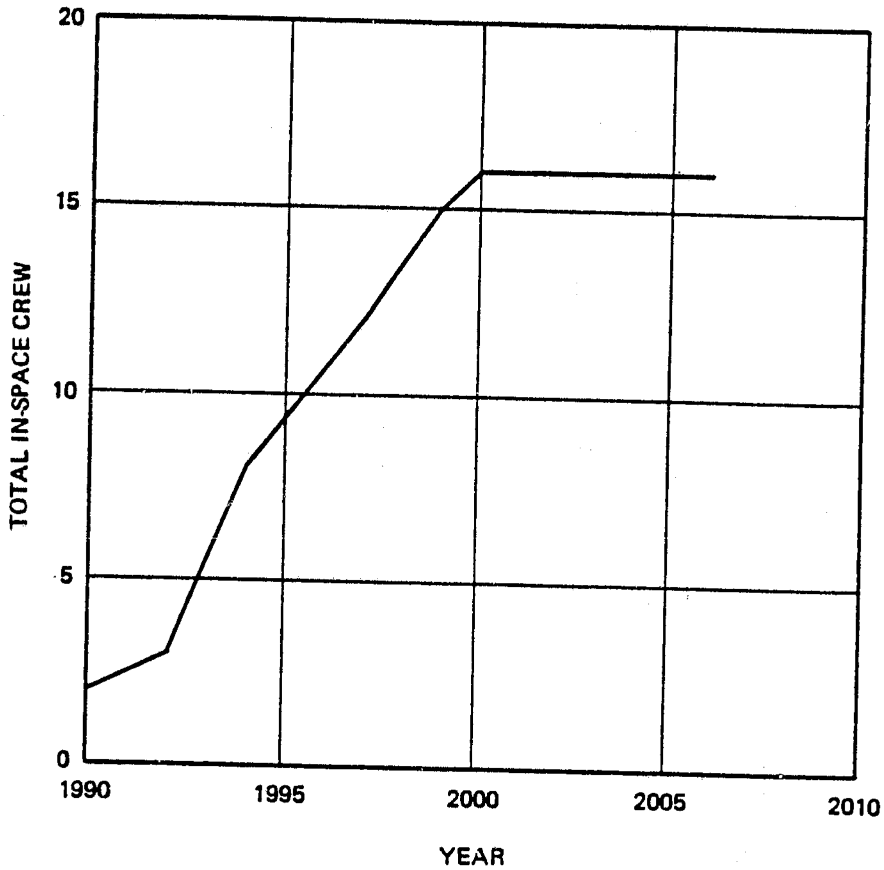


Figure 2-18. Space Station Crew Presence

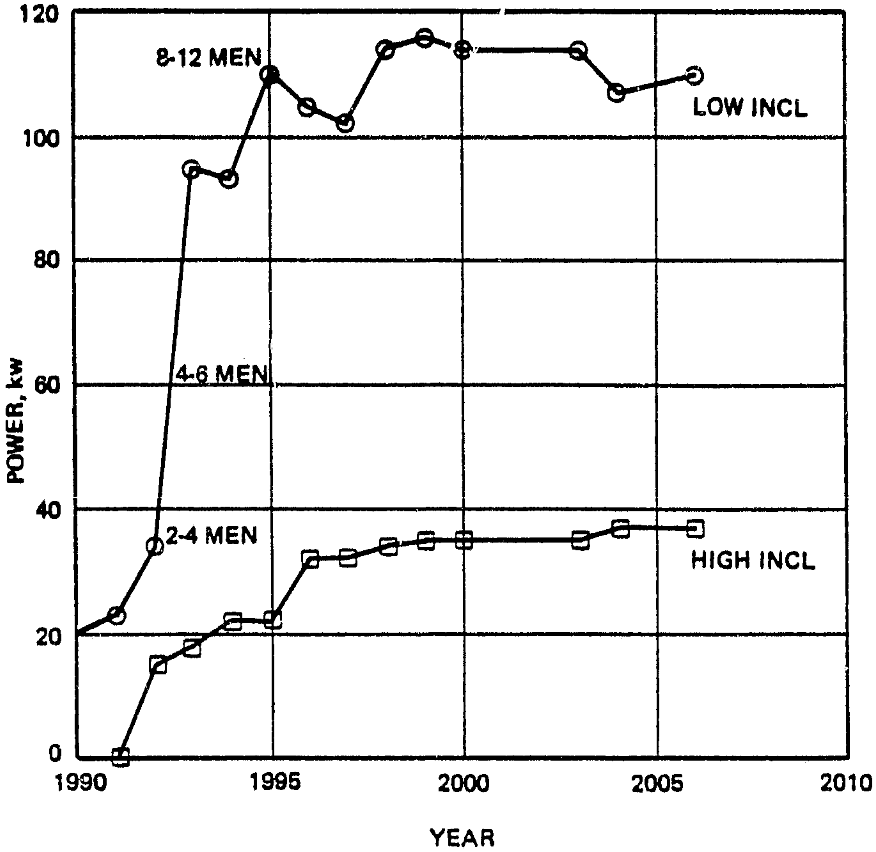


Figure 2-19. Projected Peak Power Load

The range of power levels estimated in this study are based on the results of the Boeing Space Station Mission Analysis Study (Contract NASw3680) for NASA Headquarters. Subsequent data from the NASA Space Station Mission Requirements Working Group (MRWG) indicate that power requirements will be in the 125 kW range, growing to over 200 kW by the year 2000. The range of array sizes considered in this study (see section 2.3.5) support station power levels up to about 200 kW, based on silicon solar cell planar technology.

2.2.5 Servicing Vehicles

Servicing vehicles deploy, retrieve, and service free-flyers associated with the Space Station. Two vehicles have been identified to perform these functions: an orbital maneuvering vehicle (OMV) that could handle small payloads and delta-V requirements, and an orbit transfer vehicle (OTV) that could handle larger loads and delta-V requirements. It is beyond the scope of this study to develop complete design and operating concepts, but vehicles that are representative of these options have been defined conceptually. A detailed discussion of servicing vehicle configurations and capabilities is contained in sections 2.3.7 and 4.2.3, respectively.

2.2.6 Free-Flyer Support Requirements

Free-flyer support and servicing requirements identified in this study consist of deployment, maintenance, and propellant servicing, which are addressed in more detail in section 4.2. Basically, free-flyer servicing is accomplished in one of three ways: (1) the OMV or OTV services the free-flyer in-situ; (2) the OMV or OTV retrieves the free-flyer, returns it to the station, and then redeploys the free-flyer; or (3) the free-flyer uses its own propulsion system to effect rendezvous with the station and, following servicing, redeploys itself.

2.3 Configuration Elements

The six major design drivers, the basic configuration families, and station orientations considered in this study were described in the previous section. This section provides the configuration and parametrics for the station core, solar array, radiator, free-flyer, and servicing vehicle options, all of which provide the basis for the demands made on the propulsion system.

2.3.1 Station Layout

Station orientation and mission requirements are the primary factors influencing station layout. Clearly, an infinite number of configuration variations could satisfy all requirements. This section provides a general discussion of the factors that led to the layouts selected for this study. Specific layouts selected and their rationale are developed in subsequent sections.

A basic premise for all station arrangements was that each normally manned module should have two separate means of egress to one or more manned modules for emergency escape in case of fire, loss of air pressure retention capability, the escape of noxious or poisonous gases, or a host of other unplanned events. Although dual-egress is not a Space Station design requirement, it is used as a safety criterion and it mainly affects selection of the pressurized module interconnect pattern. This, in itself, has little influence on propulsion requirements or implementation.

Another issue that is still unresolved is whether to position the core (i.e., pressurized modules and attached facilities and equipment) in an Earth-oriented or inertially-oriented mode. The principal source concepts for this study were the SOC and the SAMSP. The SOC was Earth-oriented and the SAMSP was normally inertially oriented. It was our judgement that mission needs and operational requirements tended to favor Earth-orientation. The Earth-oriented mode sometimes places one of the station's principal axes of inertia aligned with the local vertical. This strategy eliminates gravity-gradient torques if the station is truly Earth-oriented, and the propulsion requirements are dictated entirely by drag and aero torques. In this study, however, most Earth-oriented configurations include Sun-tracking solar arrays and the inertial attitude-control results of the study apply to this inertial component of an Earth-oriented station.

2.3.2 Module Descriptions

Based on the already identified preference for a planar core configuration, the Space Station core consists of habitat, laboratory, and logistics modules. The habitat module provides rest, recreation, and dining facilities for the crew. The laboratory module provides facilities for conducting various experiments that are performed within the station. The logistics module is used for resupply and contains consumables such as fuel, food, water, clothing, etc. The logistics module is regularly replaced during Orbiter resupply missions. Module design, which is based on STS payload carrying capability, provides for crew transfer, resupply or waste storage, and experimentation. These pressurized modules are typically 14 ft in diameter and up to 40 ft long. Additional modules, pallets, or payloads can be added by means of external attachment points. The module configurations are shown in figures 2-20 and 2-21.

2.3.3 Core Configuration: Dense-Pack Versus Planar

The method for building up the Space Station from separate modules will greatly affect operational, thermal, and propulsion requirements. Two methods of buildup were examined in this study: dense-pack and planar growth options. These were shown in figures 2-10 and 2-11. To expand on the points already made, the dense-pack option minimizes the difference between the principal axes of inertia, thereby limiting gravity-gradient torques and allowing any orientation to be maintained. The position of the solar arrays on long booms increases the magnitude of the overall inertia mix so much that any balancing achieved by the dense-pack configuration is outweighed by solar-array inertias. The disadvantages associated with the dense-pack design are: operational workspace requiring Earth or stellar observation would be limited; station assembly would be complicated by the number of interfaces and close proximity of modules; thermal management would be more difficult; and growth would be limited if the inner modules required view factors to space or had to be replaceable. Therefore, the planar core configuration was adopted, which posed none of these problems.

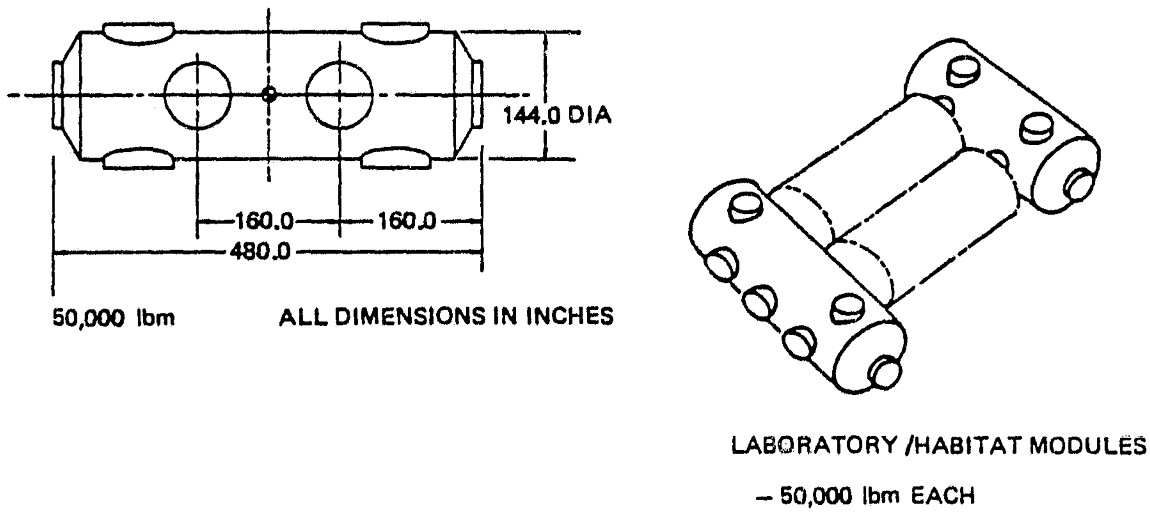


Figure 2-20. Habitat Module Design

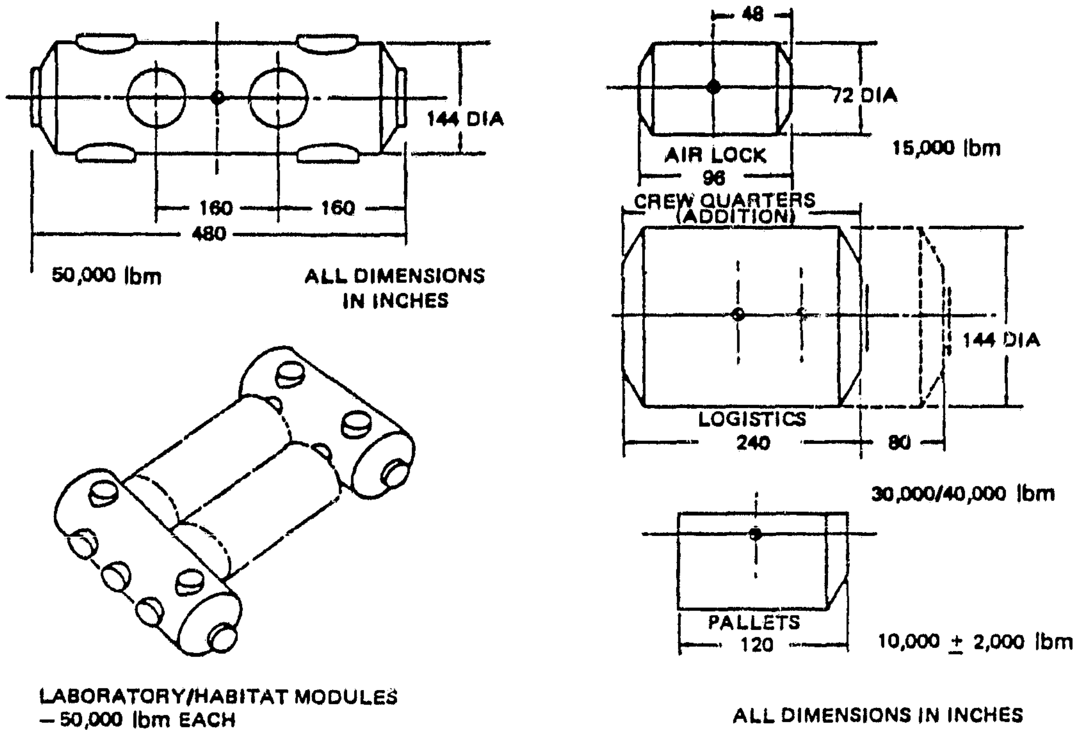


Figure 2-21. Space Station Kit of Parts

2.3.4 Planar Core Growth

The assumed Space Station core growth scenario from a 2- to 4-man station to an 8- to 12-man station is shown in figure 2-22. The initial (2- to 4-man) station consists of two habitat/service modules, a logistics module, and an airlock. A dedicated habitat module and two pallets (which contain experiment packages) are added to the station core to accommodate 4 to 6 men. The 6- to 8-man station size is achieved by adding a laboratory module, a larger logistics module, and relocating the airlock and one of the pallets. The largest core growth, intended for an 8- to 12-person crew, includes another habitat module and another pallet.

A wide variety of core growth scenarios could have been selected, but the foregoing method of buildup was adequate to determine related propulsion requirements.

2.3.5 Solar Array Parametrics

Solar array parametrics include the placement, operating mode, size, and geometry of the solar arrays. They affect or are affected by many other design drivers: (1) array placement must not interfere witn STS docking, free-flyers, or service vehicle operation; (2) arrays are a primary influence on inertia, balance, and drag; (3) power output from the arrays places limits on crew size and Space Station habitability, operations, and service missions; (4) radiator performance is dictated by array output; and (5) the effect of the solar arrays on orbit decay influences servicing strategies for the entire Space Station system.

As stated in section 2.1.2, before the findings of this study were known, virtually all Space Station configurations with Sun-tracking arrays used a cantilevered array arrangement. This study showed that cantilevered array caused excessive secular gravity-gradient torques and unacceptable control requirements. The balanced array configuration (shown in figure 2-13) was developed under Boeing IR&D to reduce the cross products of inertia. The balanced arrays differ from cantilevered arrays in the way they are hinged at the solar array mast. Instead of being hinged at the extreme end of the

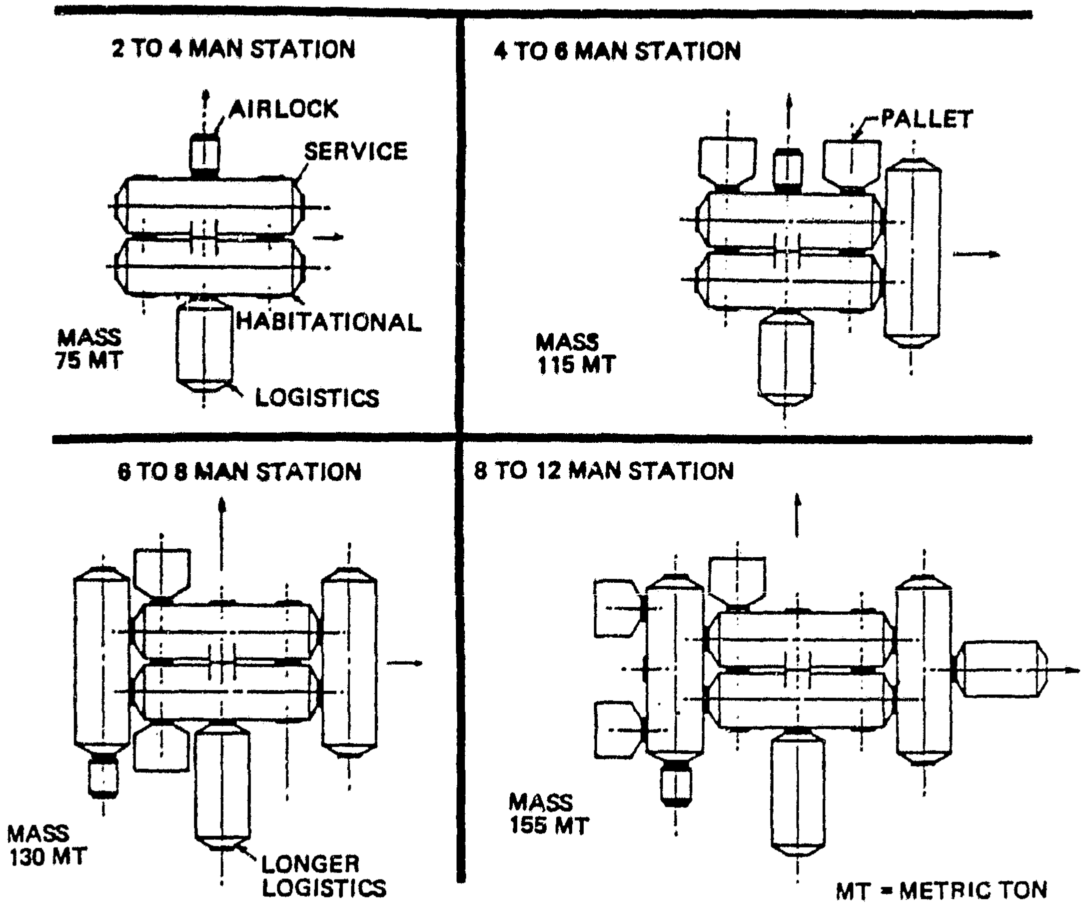


Figure 2-22. Space Station Core Growth Scenario

array, as is done for the cantilevered array, the balanced array hinge-point is at the array geometric center.

Solar array mass and power densities required for the Space Station have not been definitively specified. Table 2-2 lists the densities used to study various configurations. The more conservative baseline densities were provided by NASA-LeRC. Interim densities assumed that the array technology required for the Solar Electric Propulsion Stage (SEPS) will be developed. However, this technology and design approach may result in arrays that are too fragile to survive Space Station operating conditions in low Earth orbit.

Table 2-2. Comparative Solar Array Mass and Power Densities

	Solar Array	
Configuration	Power Density(W/m^2)	Mass Density(kg/m^2)
Baseline	105	2.5
Interim	129	1.56
Polar and Sun-synchronous	129	1.56

Projected power requirements for the initial station have continually increased since the inception of Space Station design. As of this report, power load estimates range from 35 kW to 140 kW, as shown in table 2-3.

Table 2-3 Power Requirements and Array Sizes

Crew Size	P_R, Power Requirements (kW)	P_{BOL}, Array Power (kW)	Solar Array Mass (kg) Baseline*	Interim**	Solar Array Area (m^2) Baseline*	Interim**
2-4	35	110	2613	1330	1046	853
4-6	70	210	5004	2540	2000	1628
8-12	140	420	9994	5080	4000	3256

*BOL/EOL = 1.33

**See Table 2.2 for power and mass density.

Average power delivered to the station core from the solar arrays is reduced by three factors:

a. Exposure to ionizing radiation degrades array power from the beginning-of-life (BOL) to the end-of-life (EOL) by as much as 25% over 10 years; or (EOL power)/(BOL power) = 0.75;

b. Only 60% of each orbit is in sunlight due to eclipsing (assuming a 28.5 deg inclination in LEO); therefore, the time in shadow, t_s, is: $t_s = 0.4 \times 90$ min = 36 min;

c. Of the energy used to charge batteries or other storage devices, only 60% to 70% is recovered.

These three factors are taken into consideration in sizing the array to meet station power requirements as shown in the following formula relating BOL array power to required power levels:

$$P_{BOL} = P_R \left[1 + \left(\frac{t_s}{(t_o - t_s)\eta_s} \right) \right] \left(\frac{BOL}{EOL} \right) \left(\frac{1}{\eta_{pd}} \right)$$

where

P_{BOL} = BOL array power

P_R = Power required

t_s = Time during which power is delivered from storage (time in eclipse)

t_o = Orbit period (approx. 90 minutes)

η_s = Storage turnaround efficiency (net energy out/energy in)

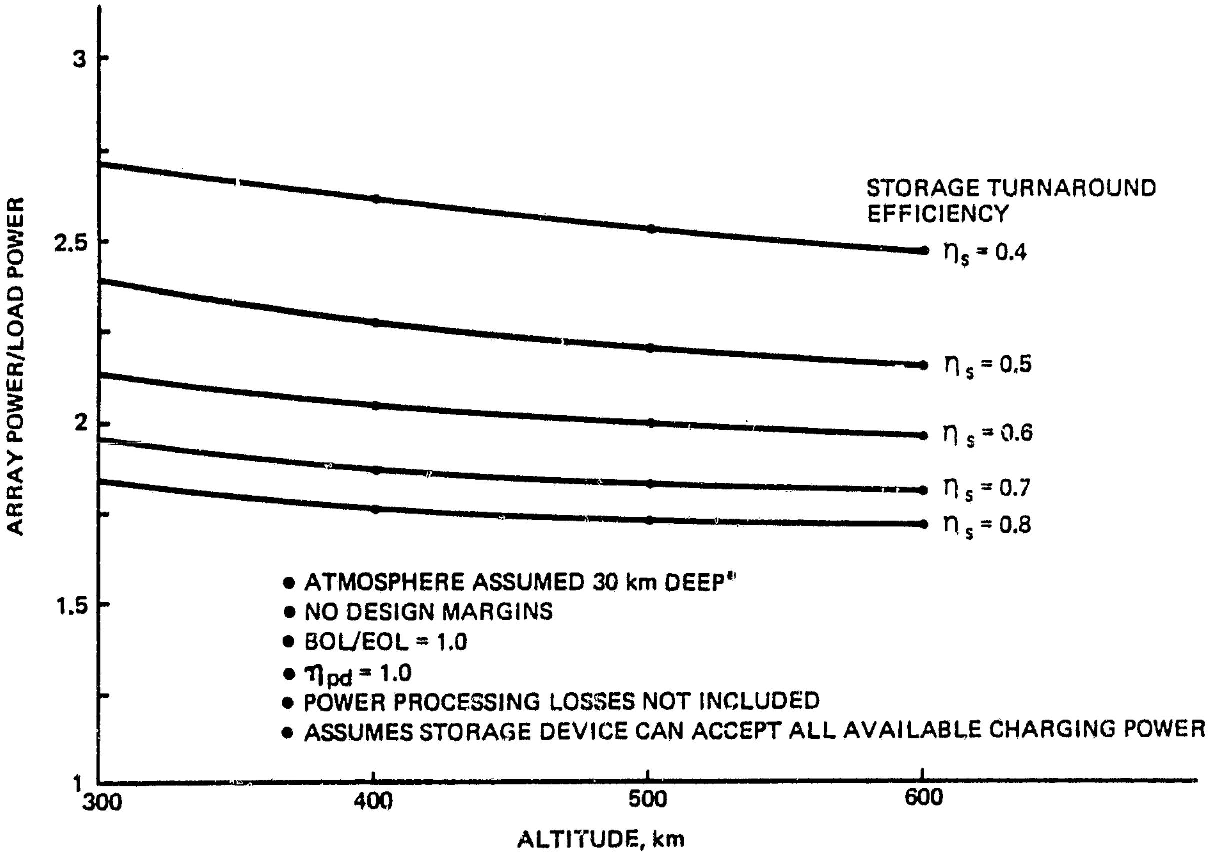
BOL/EOL = Beginning/End of Life power ratio

η_{pd} = Power distribution efficiency

Solar array size varies inversely with energy storage system turnaround efficiency and Space Station altitude. Higher altitudes lead to a requirement for slightly less array power because the occulted portion of the orbit is reduced. The principal influence on array sizing, however, is the efficiency of the energy storage system. The curves in figure 2-23 cover the efficiency range from 40% to 80% which encompasses the storage technologies considered for the Space Station. The data are based on Sun-tracking arrays.

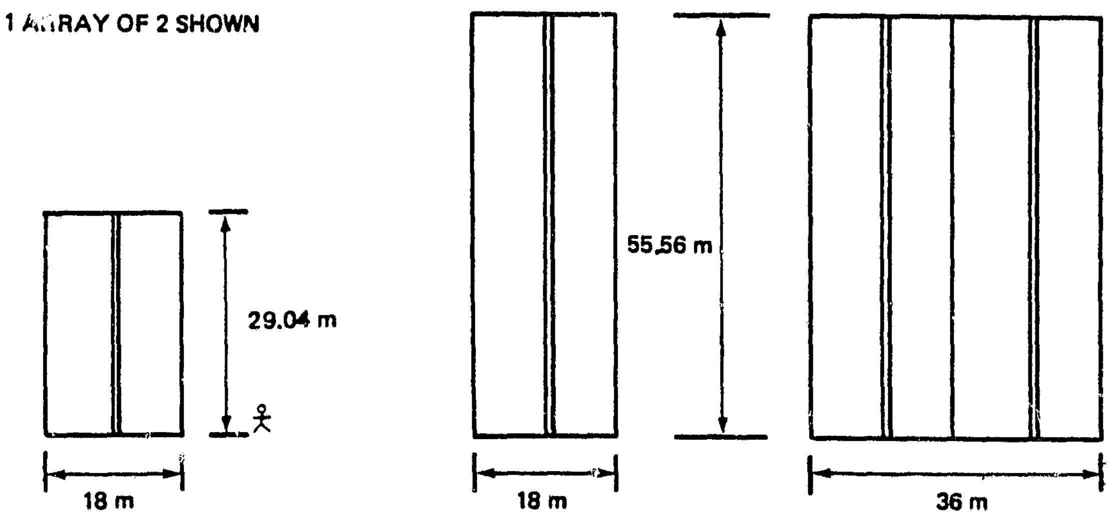
Array power sizing included a design margin, power distribution efficiency (η_{pd}), and BOL/EOL power ratio over a ten-year lifetime. The result is array beginning-of-life power capability approximately three times the net load requirement for the station. Array area sizing initially used a BOL power capability of about 130 watts/m^2, and was later baselined at a conservative 105 watts/m^2, as shown in Table 2.2 and used in Table 2.3.

Figure 2-24 shows the array dimensions and geometry that were used in with the baseline configuration. The rectangular array geometry enables the arrays to be efficiently packaged, transported by the STS, and assembled easily on-orbit. Array aspect ratio (ratio of length to width), array power requirements, and array power densities define array dimensions. Aspect ratio has a significant effect on inertia cross products, field of view, and shadowing effects. Appendix A discusses the relationship between aspect ratio and cross-products of inertia.



* OCCULT PERIOD BASED ON EARTH RADIUS OF 6,408 km COMPARED TO MEAN EQUATORIAL SURFACE RADIUS OF 6,378 km

Figure 2-23. Solar Array Power Sizing Factor for Energy Storage



CREW SIZE	2 TO 4	6 TO 8	8 TO 12
AREA	1,046 m^2	2,000 m^2	4,000 m^2
AVERAGE POWER,kw	36	75	150

Figure 2-24. Baseline Solar Array Sizes

2.3.6 Radiator Analysis

Heat sources to the Space Station are direct solar radiation, reflected solar radiation (from other station components such as the arrays and Earth albedo), Earth radiation, internal electrical power dissipation, and crew metabolic processes. Energy is lost from the station by radiation from the modules and by thermal radiators. This study assumes that solar heating, Earth radiation, and metabolic heating to the station core are approximately equal to radiation emitted from the station core. Therefore, for preliminary design purposes, the electrical power dissipated within the station must be rejected by the thermal radiators, which must be sized to reject this maximum station power dissipation level under worst case conditions.

A detailed radiator analysis was beyond the scope of this study and is of interest only to the extent that it affects station drag and mass properties. Radiator efficiencies, weights, and operating characteristics were taken from work done by Boeing under NASA Contract NAS8-34893. The reader is referred to the final report, Volume II of D190-27487-2, "Advanced Platform Systems Technology Study," dated April 1983, for a complete radiator analysis. For this study, the radiator is of the pumped two-phase type comprising a series of parallel runs of plumbing within which the two-phase fluid flows. The radiator operates at a constant temperature (50°F) because of the two-phase condensing nature of its operation.

2.3.6.1 Radiator Design Considerations

Virtually all of the spacecraft launched to date have used a body-mounted radiator system. Examples of these spacecraft include Skylab, Lunar Excursion Module, Apollo, Lunar Rover, Lunar Orbiter, Mariner-Venus-Mercury Probe, Mariner, and Voyager. The STS design includes a variation of this approach by having the radiator mounted on the inside of the payload bay doors. The Space Station, however, is designed to use boom-mounted panels for the following reasons:

a. Body-mounted radiators located on the pressurized modules have insufficient area to reject the Space Station heat load. Although boom-mounted radiators will be used, they must be augmented by a large thermal utility radiator panel.

b. Both radiation surfaces of the radiator panel can be used.

c. Modules can be added to the station without blocking the radiator.

d. STS, OTV, and other docking and proximity operations can be conducted without radiator interference.

e. Boom-mounted radiator panels can be gimballed to provide a minimum viewing angle with respect to the Sun.

f. Radiator changeout, if needed, may be more easily accomplished if the panels are boom-mounted and away from other adjacent structures.

Radiator sizing and weight trends are depicted in figures 2-25(a) and 2-25(b). Life requirements for the station, including the radiators, is 10 years, during which the solar absorptance (α_s) to infrared emissivity (ϵ_{ir}) ratio declines from 0.4 to 0.8 (see figure 2-25(b)).

A conservative estimate of the α_s/ϵ_{ir} ratio was assumed to be 0.8. It is readily apparent from figure 2-25 that, at 0.8 α_s/ϵ_{ir} ratio, a severe size and weight penalty will be paid if a heat-rejection system is chosen that does not use thermal storage. The beneficial effects of thermal storage increase as the radiator becomes exposed to the Sun a greater percentage of time, which can be seen by comparing curves 1, 2, 3, and 4 in figure 2-25(b). Figure 2-26 describes the North-South and East-West definition in figure 2-25(b). The steerable off-Sun configuration, curve 4, is affected the least, as would be expected. However, even a steerable radiator benefits from thermal storage due to reflections, Earth albedo, and increased Earth radiation on the sunlit side. Figure 2-27 depicts the

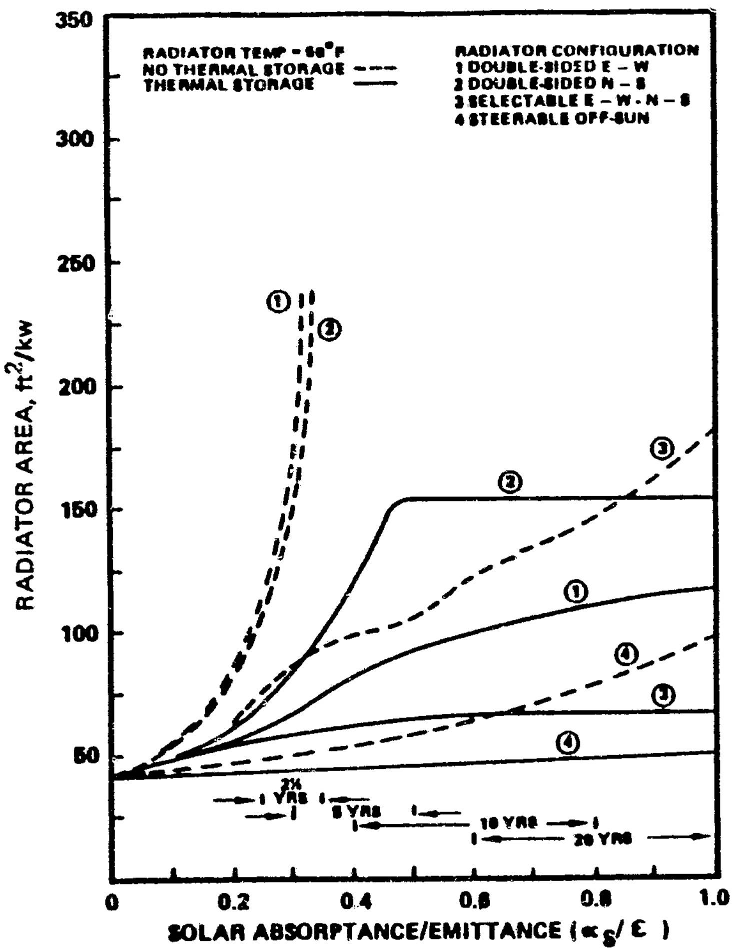


Figure 2-25(a). Radiator Sizing Criteria

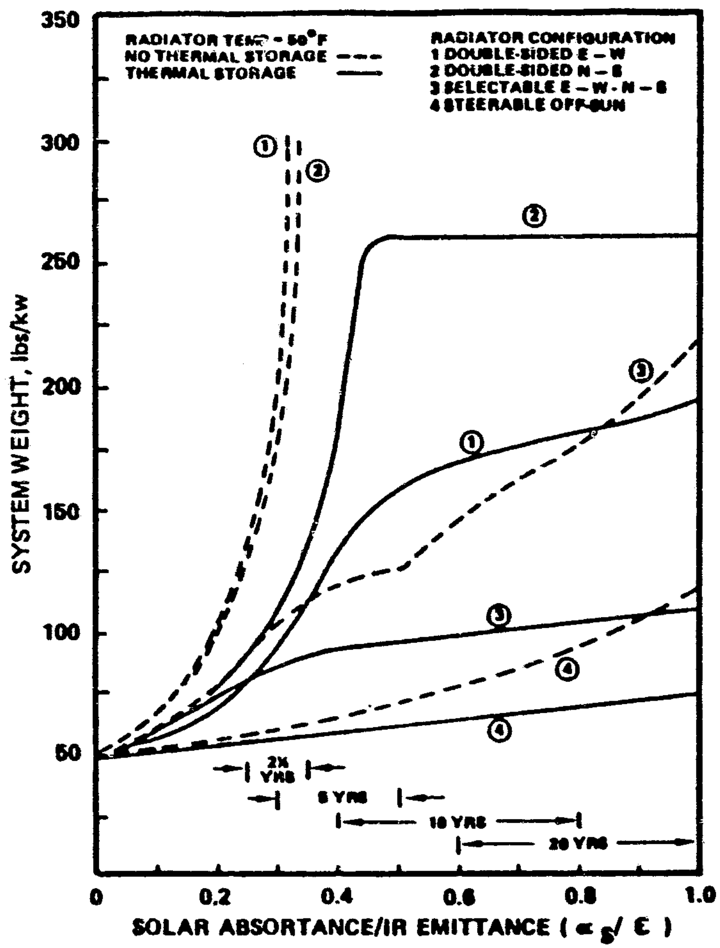


Figure 2-25(b). Radiator and Thermal Storage Weight

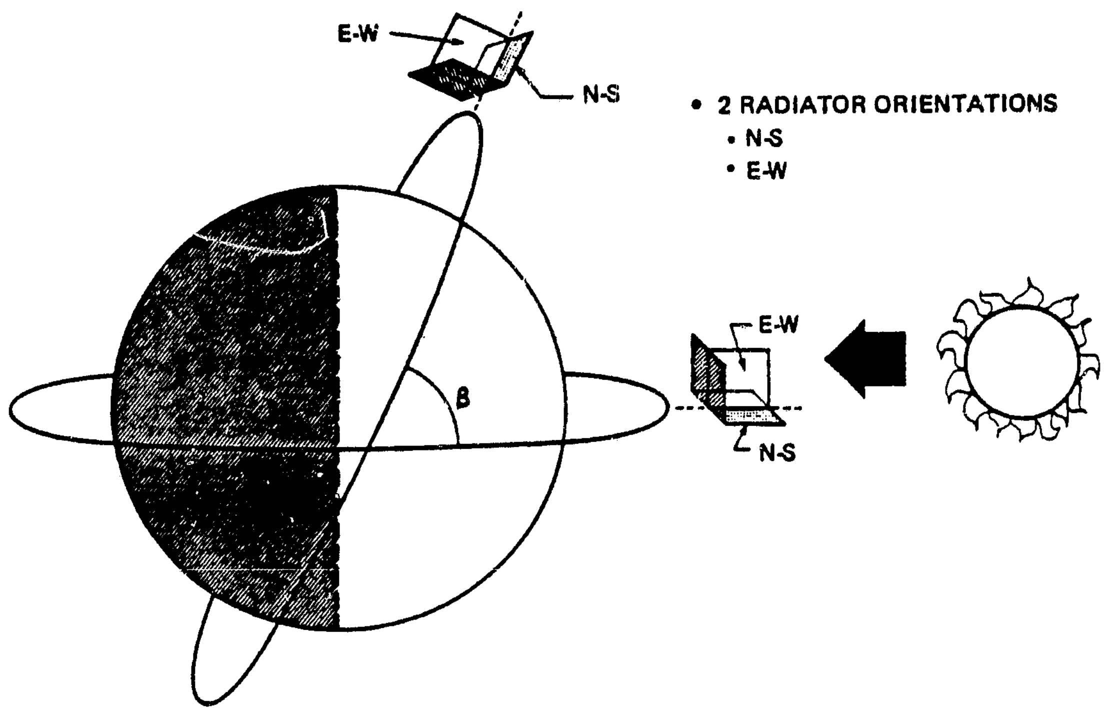


Figure 2-26. East-West and North-South Definitions

ORIGINAL PAGE IS
OF POOR QUALITY

• FIXED ORIENTATION
  • CURVES 1 AND 2, FIGURES 2-25 a & b

• SELECTABLE
  • CURVE 3, FIGURES 2-25 a & b

• STEERABLE OFF-SUN
  • CURVE 4, FIGURES 2-25 a & b

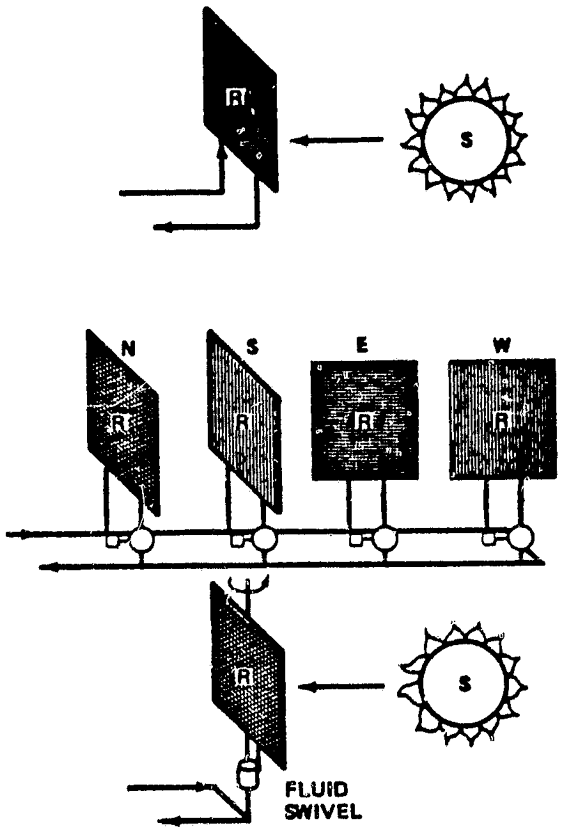


Figure 2-27. Radiator Option Schematics

various generic radiator concepts. The schematic for the selected thermal management system is shown in figure 2-28.

The fixed-orientation configurations, whose performance is shown by curves 1 and 2 in figure 2-25(a), require large radiator areas for space stations that need 100 kW or more of energy. The fully steerable radiator, curve 4 of figure 2-25(a), depends on a leak-free fluid swivel, which has yet to be developed. Therefore, a selectable radiator was initially considered. The initial configuration chosen was a cruciform radiator arrangement mounted on the boom between the station modules and the solar arrays (figure 2-29). This configuration was acceptable in terms of radiator size and impact on station mass properties, but caused a relatively high aerodynamic drag.

Therefore, a steerble design was sought that caused minimal drag but had better performance than the fixed-orientation design in figure 2-27. Figure 2-30 shows a design concept that meets these criteria. The radiator panel for this configuration is always parallel to the velocity vector because the panel is perpendicular to the boom which is perpendicular to the velocity vector. Because the outer boom segment is always perpendicular to the Sun's rays (to point the solar arrays at the Sun), the radiator panel is also always parallel to the Sun's rays. In essence, the result is steerable radiator performance without a fluid swivel. However, analysis revealed that both the cyclic and secular gravitygradient torque values are excessive for a cantilevered array configuration and are further aggravated by the radiator panels and split boom of the Figure 2-30 configuration. The use of the balanced-array configuration and the hinging only at the array geometric center overcomes moment and torque problems.

2.3.6.2 Selected Radiator Design

The final radiator configuration is compatible with the balanced solar array and causes only negligible drag due to constant panel alignment with the velocity vector. This configuration (figure 2-31) allows the radiator to pivot to partially compensate for the beta angle, while retaining the single degree of freedom required for simultaneous alignment with the velocity vector. This new configuration is hinged at the boom with the

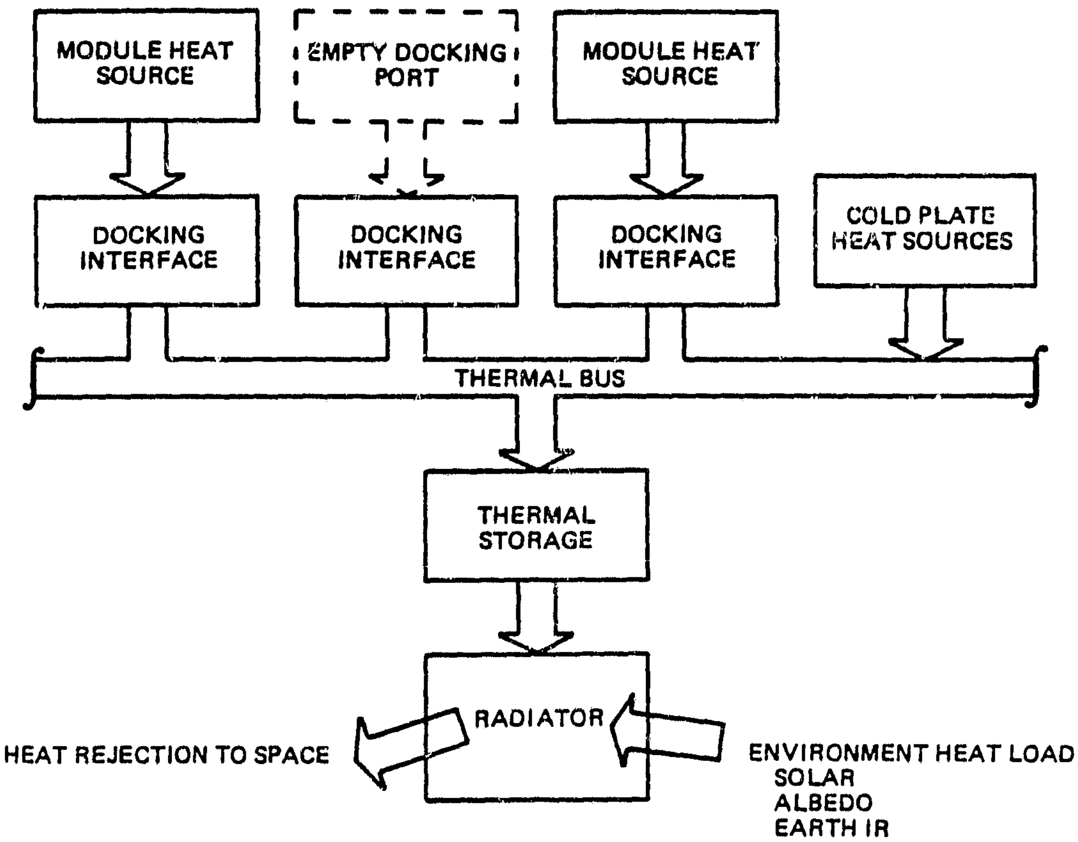


Figure 2-28. Thermal Management System

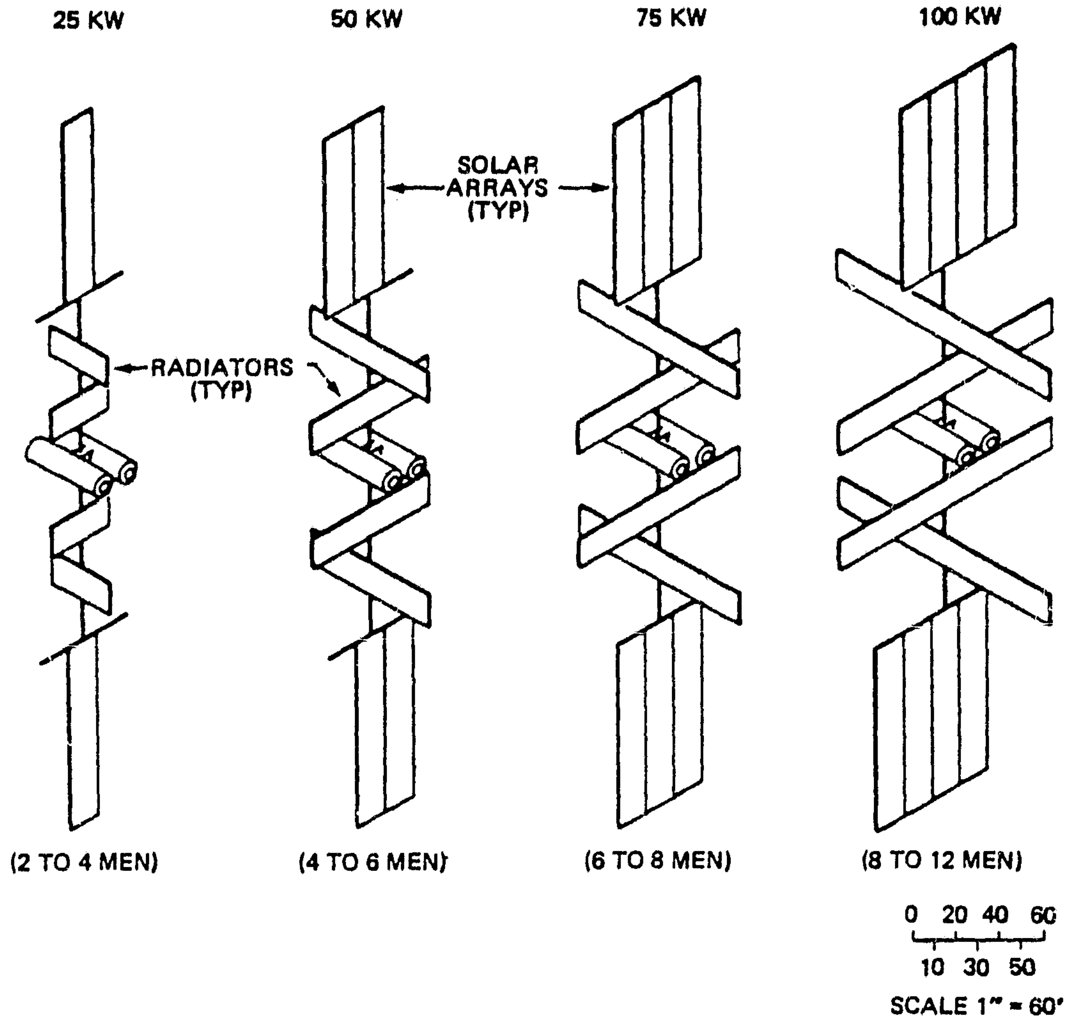


Figure 2-29. Initial Solar Array and Radiator Orientations

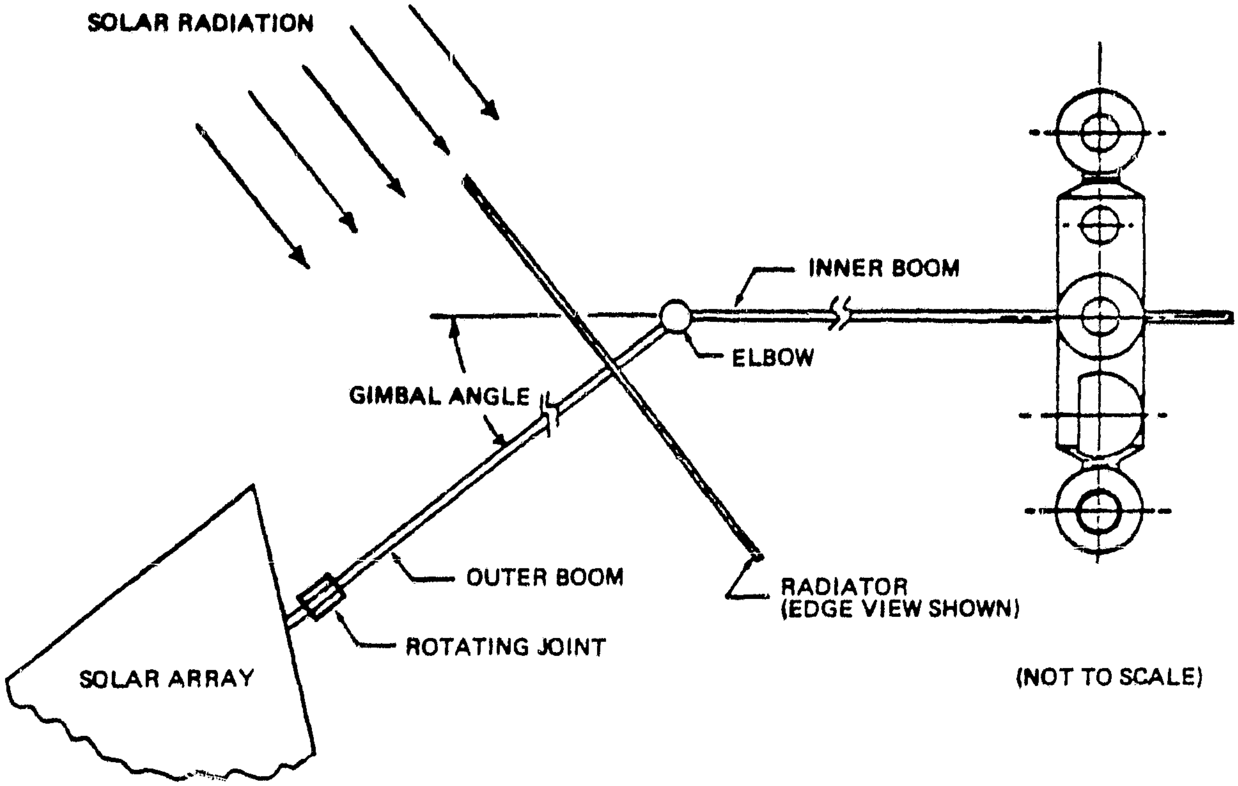


Figure 2-30. Bent-Boom Radiator Configuration

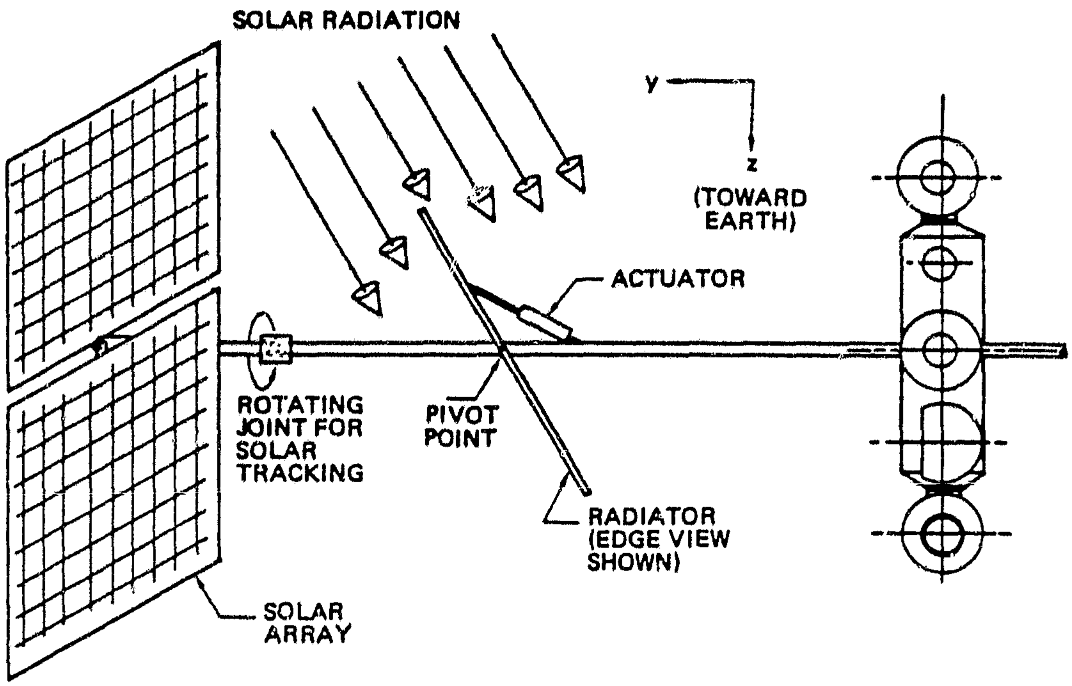


Figure 2-31. Selected Radiator Arrangement

hinge line parallel to the velocity vector. The radiator is pivoted on this hinge and driven by an actuator to reduce the area presented to the Sun.

Radiator weight, as opposed to heat rejection system weight, does not include the thermal storage material. This material is assumed to be included within the station modules and its weight is accounted for in their weights. Radiator panel weight, used separately for moment and torque calculations, is 5.86 kg/m^2 (1.2 lbm/ft^2) of radiating area. The radiator area factor from figure 2-25(a) is 6.5 m^2/kW (70 ft^2/kW).

The partially steerable radiator can be mounted to a boom or to an Earth-oriented Space Station core. The beta angle rotation places the radiator edge-on to the Sun when the vehicle is passing over the local noon meridian. The Sun elevation on the radiator is given by

$$\sin q = \sin \beta \cos \beta (1 - \cos \theta)$$

where θ is the orbit angle starting at zero over the noon meridian. This function peaks during the shadowed period of the orbit. For high beta angles, the solar input averaged over an orbit is 15% to 20% of a normally illuminated radiator. The adiabatic temperature for this radiator without selective coatings is about -40°C, compared to about 75°C for a normally illuminated radiator.

2.3.7 Servicing Vehicle Configurations

The two types of servicing vehicles that have been identified for free-flyer support are the Orbital Maneuvering Vehicle (OMV) and the Orbital Transfer Vehicle (OTV). The configurations shown in this section are taken from previous in-house studies. They are representative of vehicles that may be used and are not intended to imply recommendations of any particular vehicle design. Their capabilities are presented because they have a bearing on free-flyer propulsion requirements.

2.3.7.1 Orbital Maneuvering Vehicle (OMV)

Figure 2-32 depicts the selected OMV concept, which is taken from an in-house BAC study. The OMV can fly preprogrammed trajectories, as well as be controlled or reprogrammed from the aft flight deck or the ground. The lightweight airborne support equipment (ASE) cradle may be conveniently positioned along the payload bay length where it is attached by standard sill and keel fittings. The cradle supports the OMV during the launch and re-entry phases and houses the antennas, communication, video, and other avionics ASE necessary for vehicle man-in-loop control from the STS aft flight deck. The following is a summary of proposed OMV subsystem functions:

o <u>Viewing</u>

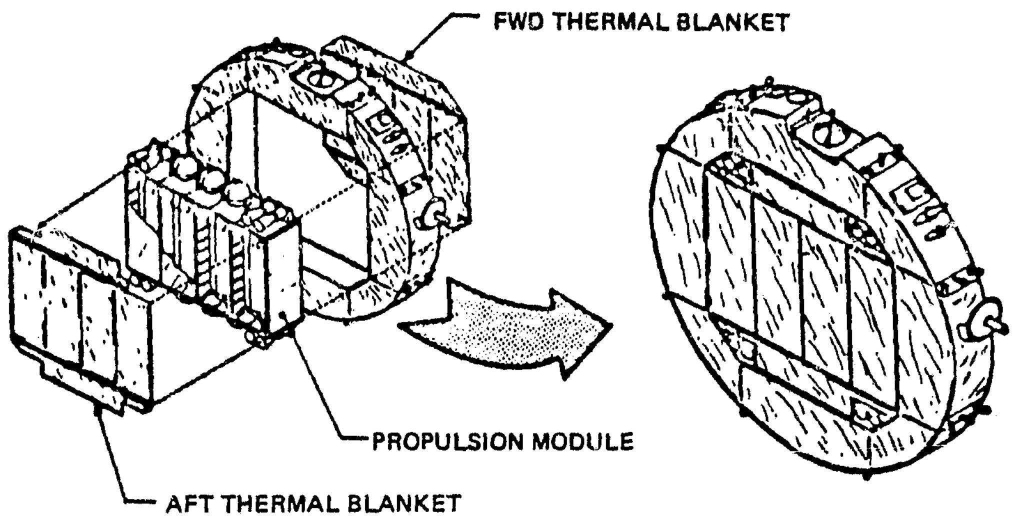
A viewing subsystem consisting of two monochrome television cameras: one rigidly mounted with a fixed lens, and one with a zoom lens having pan-tilt capability to provide scene information for observations.

o <u>Docking</u>

The viewing subsystem is also used in conjunction with a range and range-rate radar sensor, and a docking interface that uses the remote manipulator system and effector for docking to another spacecraft.

o <u>Guidance and Control</u>

A guidance and control subsystem performs the guidance,and navigation, and attitude control functions. An inertial reference unit, updated by star trackers, provides the OMV attitude, while navigation and ephemeris information are provided by a global positioning system receiver and processor. Control commands are computed by one of the dual redundant computers and routed to the main propulsion or reaction control subsystem via the valve drive electronics.



- DIMENSIONS
  - LENGTH 0.71 m
  - DIAMETER 4.37 m
- MASS (MT) (1 MT = 1,000 kg)
  - DRY 1.6
  - PROP 2.6 (N_2O_4/MMH)
  - GROSS 4.2
  - MASS FRACTION 0.61
  - PAYLOAD DELIVERY OR RETRIEVAL

Figure 2-32. Orbital Maneuvering Vehicle Concept

- o Propulsion

The propulsion subsystem includes eight throttleable main thrusters (111 to 556 in. or 25 to 125 lb_f) and 24 67-in. (15 lb_f) RCS thrusters. A common, pressure-regulated monopropellant hydrazine feed system is used for both types. A cold gas reaction control system is available as an add-on component for proximity maneuvers near sensitive spacecraft. The variable thrust main engines provide pitch and yaw control during main burn.

- o Power

The electrical power and distibution subsystem provides for the storage, distribution, regulation, and control of the electrical power. Silver-zinc (AgZn) batteries store the electrical power. Solar arrays and rechargeable nickle-cadmium batteries are available for missions longer than 24 hours.

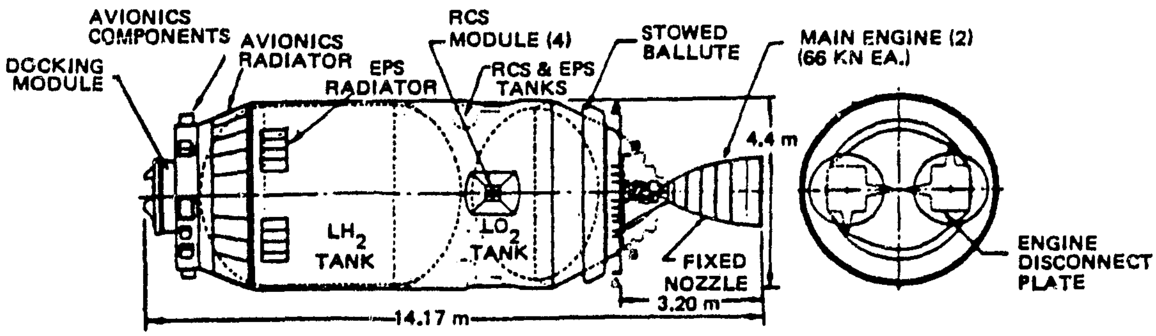
- o Environmental

Thermal louvers and multilayer insulation blankets provide passive OMV thermal control to the maximum possible extent. Electrical resistance heaters are installed on propellant tanks and lines to prevent propellant freezing under certain orbital conditions.

2.3.7.2 Orbital Transfer Vehicle (OTV)

The OTV is a high-performance, space-based vehicle that uses liquid hydrogen (LH_2) and liquid oxygen (LO_2) as propellants. The configuration shown in figure 2-33 and used in this study is taken from the Future OTV study performed by BAC.[1] The viewing, docking, environmental control, and guidance and control functions are similar to that described for the OMV.

1 Eldon ?. Davis, "Future Orbital Transfer Vehicle Technology Study," NASA CR 3536, Contract NAS1-16088, May, 1982.



- MASS (MT) (1 MT = 1,000 kg)
  - DRY 3.6
  - PROP 32.5
  - GROSS 37.7
  - MASS FRACTION 0.8638
  - PAYLOAD TO GEO (ΔV = 4,300 m/s)
    - ROUND TRIP 5.0
    - DELIVERY (0.2g) 13.0

Figure 2-33. Space-Based OTV Concept

Hydrazine thrusters provide attitude control. Electrical power is provided by fuel cells using super-critical hydrogen and oxygen (from separate tanks located in the OTV intertank area).

The OTV is initially carried to orbit in the STS payload bay without propellant or payloads. Payload, fluids and spares for the OTV are delivered to the Space Station and OTV by the STS. Before each flight, the OTV is serviced, and payloads, and consumables and flight programs are loaded. Flight operations for a typical LEO-to-geosynchronous Earth-orbit-(GEO)-transfer involve a total delta-V of 4,300 m/s. Once back at the base, the OTV is housed in a hangar that protects it from space debris and serves as a maintenance facility. OTV housekeeping needs (power, thermal, and data links), are provided by the Space Station.

2.3.8 Free-Flyer Configurations

Four free-flyers, which are representative of the various types that may be serviced by the Space Station, have been analyzed in this study: (1) a free-flying propellant farm; (2) a tethered propellant farm; (3) a slack-tethered power generation module (STPGM); and (4) a space application and science platform (SASP). These free-flyers will either be in the same orbit or be at a slightly differing orbit inclination and eccentricity so that they appear to orbit around the Space Station. Section 5.7 describes the propulsion requirements associated with these different free-flyers.

2.3.8.1 Free-Flying Propellant Farm

Propellant for the servicing vehicles (OMV and/or OTV) be stored in a free flyer to increase safety, prevent contamination due to leakage, keep propellant slosh isolated, provide thermal isolation, and reduce Space Station center-of-mass variations. The design criteria used in this study for the propellant farms were the following:

a. Components must fit within the STS cargo bay.

b. There must be sufficient capacity for 3 OTV and 10 OMV refuelings.

c. There will be no on-orbit pressure-vessel fabrication.

d. Pressure vessels must be made from structures that exist at the time of "farm" fabrication.

e. Simplified on-orbit assembly will utilize flat panels that fit within STS cargo bay.

f. Propellant farm must be protected from space debris originating from natural and human sources.

The free-flying propellant farm is shown in figure 2-34. The large box-shaped section contains four LH_2 tanks that are identical to those used on the space-based OTV. Three of these tanks contain LH_2 (for three OTV refuelings) and the fourth contains LO_2 sufficient for three OTV refuelings. The capacity of three OTV's is 97,978 kg (216,000 lbm) of propellant. At a mixture ratio of five parts LO_2 to one part LH_2 (by weight), the LO_2 requirement is 81,648 kg (180,000 lbm). This mass of LO_2 occupies 71.56 m^3 (2527 ft^3), or about one OTV LH_2 tank volume. The arrangement of the LH_2 and LO_2 tanks is depicted at the bottom of figure 2-35. The free-flying farm (untethered) has the LO_2 tank adjacent to the triangular section. For simplicity, and because the OMV propellant selection is unknown, it is assumed here that the propellant will be hydrazine. Hydrazine sufficient for 10 OMV refuelings occupies 23.34 m^3 (824 ft^3). Because one OTV LO_2 tank volume is 24.1 m^3 (850 ft^3), an OTV LO_2 tank is selected for the hydrazine storage. Figure 2-35 shows that the hydrazine tank is located within the triangular section extending to the right of the box-like section. One concern associated with hydrazine is its relatively high freezing point (35°F). The LO_2 tank is located adjacent to the hydrazine tank to minimize heat loss from the hydrazine (the LO_2 storage temperature is -297°F while the LH_2 storage temperature is -423°F).

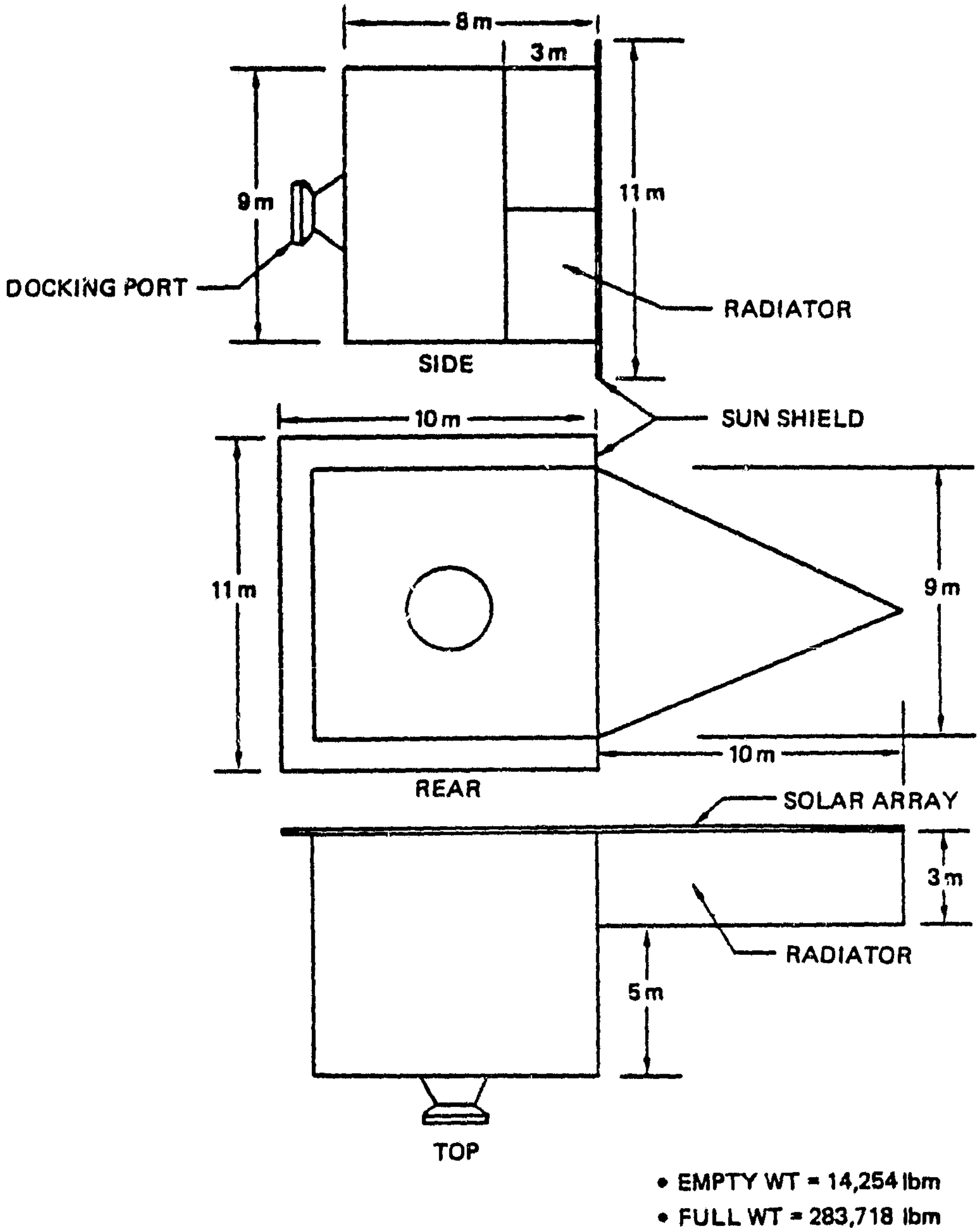


Figure 2-34. Free-Flying Propellant Farm Configuration

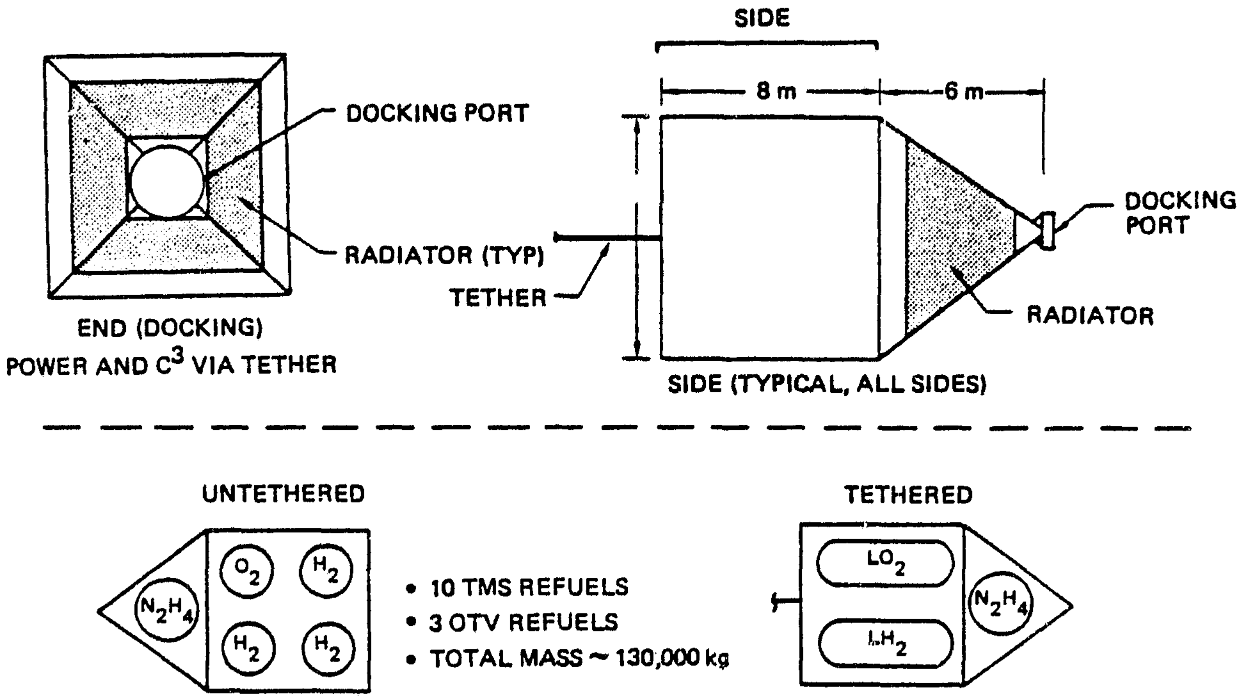


Figure 2-35. Tethered Propellant Farm Configuration

The free-flying propellant farm is Sun-oriented. A Sun shield, featuring a flexible optical surface reflector (FOSR) for low-energy absorption, is used over the entire surface of the rectangular 10m x 11m face at one end of the box. FOSR has a solar absorptivity to infrared emissivity (α_s/ϵ_{ir}) ratio of about 0.1 and provides good thermal isolation for the box section that contains the cryogens. The Sun-facing side of the triangular section containing the hydrazine tank is covered with a 5-kW solar array to generate power for command, control, and communications functions, heaters for hydrazine freezing prevention, electronic and electrical equipment, pumps, and instrumentation. The heat emitted by the backside of the solar array will help maintain the hydrazine above freezing and keep the electronic equipment warm. The flat sides of the triangular section are radiator panels whose relatively warm surfaces will also help keep the hydrazine tank warm. All other surfaces will be covered with FOSR or a paint with a favorable α_s/ϵ_{ir} ratio.

Electrical and electronic equipment is located within the triangular section to the extent possible. The flat panels that make up the exterior of the farm provide meteoroid protection for the hardware inside.

A docking mechanism that incorporates umbilicals for C^3 and propellant transfer is located on the side away from the Sun. Attitude control and orbit makeup is achieved by small thrusters that utilize the boiloff as propellant. These thrusters could be the GO_2/GH_2 bipropellant type, simple GO_2 or GH_2 blowdown type, or may be GO_2 or GH_2 resistojets, depending on performance requirements. There will probably be very little, if any, O_2 boiloff since the H_2 boiloff can be used to reduce the heat leak into the O_2 tank to virtually zero. An analysis of the optimum approach requires a detailed propellant farm analysis, which is beyond the scope of this study.

A weight breakdown for the free-flying propellant farm is shown in Table 2-4.

2.3.8.2 Tethered Propellant Farm

The tethered farm, shown at the top of figure 2-35, consists of a boxlike portion, which is very similar to that described in the previous section for the free-flying farm, and a pyramidal section attached to the box portion. The hydrazine tank and the electrical and electronic equipment are located within this pyramidal area. A radiator is located on the sides of the pyramidal section. All surfaces, except for the radiator, will be covered with FOSR. The radiator area is larger for the tethered farm than for the free-flyer because the surface isn't always shielded from solar heating. There is no solar array because power is obtained from the station via the tether, as are C^3 functions. The docking mechanism is located at the apex of the pyramid with the propellant transfer umbilicals. The tethered farm will be gravity-gradient stabilized and, therefore,

Table 2-4. Weight Breakdown for Free-Flyer Propellant Farm

ITEM	MASS (kg)	MASS (lbm)
Tankage		
LH_2/LO_2	1,415	3,120
N_2H_4	163	360
Docking Mechanism	249	550
Meteoroid Protection	2,624	5,785
Structure	2,177	4,800
Solar Array	54	120
Radiator	145	320
EMPTY WEIGHT	6,466	14,255
Propellant		
LO_2	81,648	180,000
LH_2	16,330	36,000
N_2H_4 (hydrazine)	24,252	53,465
PROPELLANT WEIGHT	122,229	269,465
TOTAL FULL WEIGHT	128,695	283,720

attitude control may be required during docking or other disturbances and would logically be provided through the use of LO_2 and/or LH_2 boiloff, as discussed for the free-flying farm. Control of tethered vehicles is beyond the scope of this study. However, a tethered propellant tank farm would most likely be reeled in, except during propellant transfer operations. When reeled in, the center-of-gravity of the station could be near the station core. Net gravity forces at the core will be small enough to permit zero-gravity laboratory operations and the core-based propulsion system could effect reboost without exciting tether motions.

A weight breakdown for the tethered propellant farm is shown in Table 2-5.

Table 2-5. Weight Breakdown for Tethered Propellant Farm

Item	Mass (kg)	Mass (lbm)
Tankage		
LH_2/LO_2	1,415	3,120
N_2H_4 (hydrazine)	163	360
Docking Mechanism	249	550
Meteoroid Protection	2,313	5,100
Structure	2,177	4,800
Solar Array	54	120
Radiator	184	405
EMPTY WEIGHT	6,139	13,535
Propellant		
LO_2	81,648	180,000
LH_2	16,330	36,000
N_2H_4 (hydrazine)	24,252	53,465
PROPELLANT WEIGHT	122,229	269,465
TOTAL FULL WEIGHT	128,369	283,000

Martin Marietta is currently conducting a tethered propellant farm study for JSC under contract NAS9-17059.

2.3.8.3 Science and Applications Space Platform

The science and applications space platform (SASP) defined by the Marshall Space Flight Center is a free-flying vehicle that can be adapted to a wide variety of payloads. Figure 2-36(a) shows SASP and some of the projected configuration options. The SASP is assumed to be in the same orbit but ahead of the station by 10 to 50 km. The SASP design provides a highly modular system for:

a. Low-cost initial use with extended-duration spacelab payloads.

b. Conservative escalation of mission capability.

c. Flexible adaptation to the wide variety of payload sizes, groups, and orbits being planned.

The SASP design is also intended to simplify payload integration, increase the flexibility of platform use, and optimize the platform, power system, and payload functions.

This long duration, multi-payload, free-flight vehicle is designed to carry a wide variety of payloads and may accommodate certain overloaded mission support elements, such as data relay satellites. Payloads that will particularly benefit from the SASP include payloads:

a. that have similar orbit altitude and inclination requirements;

b. whose budgets preclude investment in dedicated free-flyers;

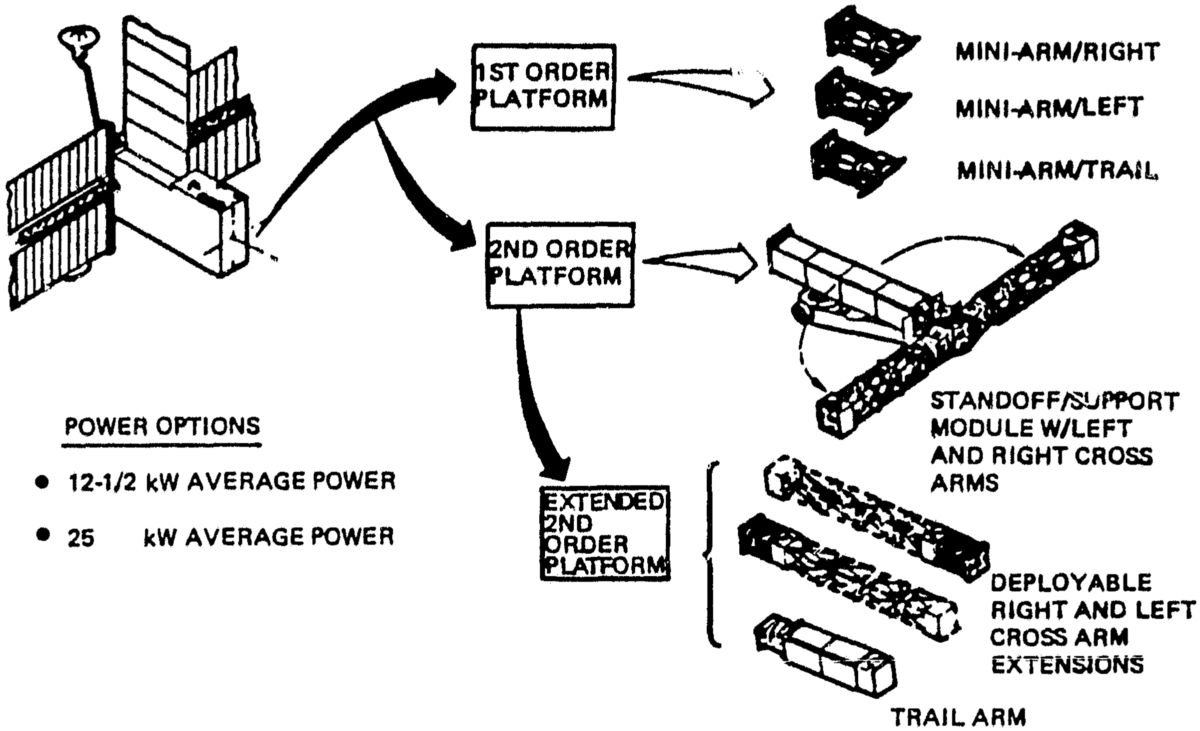
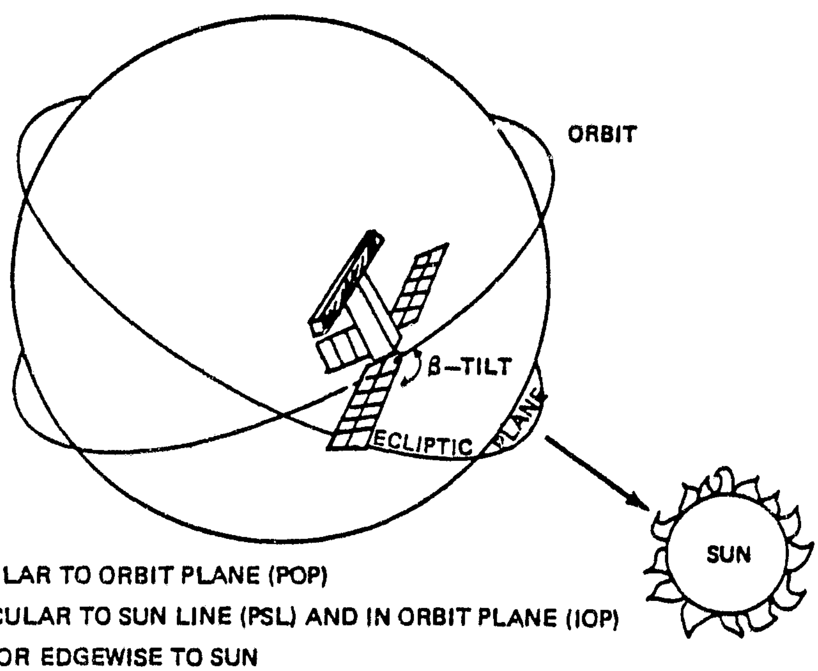


Figure 2-36(a). SASP Configuration Options



- BODY PERPENDICULAR TO ORBIT PLANE (POP)
- ARRAY PERPENDICULAR TO SUN LINE (PSL) AND IN ORBIT PLANE (IOP)
- THERMAL RADIATOR EDGEWISE TO SUN

Figure 2-36(b). SASP Flight Mode

c. that have previously flown on Spacelab pallets for a short duration in the STS sortie mode and would benefit from the advantages SASP can offer; and

d. whose flight durations range from a few months to a few years, or those requiring periodic return to Earth, on-orbit modification, maintenance or replenishment. In these instances, the costs of dedicated spacecraft and multi-rendezvous STS services would be prohibitive for separately flown payloads.

e. which, when grouped for maximum synergism, are of significant size and constitute a multi-STS delivery operation and, thus, require a centralized orbit rendezvous, assembly, and resource facility.

In general, the SASP satisfies payload needs by centralizing resouces, being available as a "rental" facility for long- or short-term users, and enabling the STS to support a number of payloads at once.

Although the SASP has a broad range of potential uses, it is not generally thought of as a vehicle for payloads that have unique orbits or that would have untenable interfaces with the SASP by virtue of physical or operating features or sensitivities.

Most of the missions for the SASP require inertial pointing (of the payload) or have no pointing preference. Accordingly, an orientation that does not require two degrees of freedom with an electrical slip ring for Sun tracking is favored. Several orientations were investigated in the SASP studies; the one depicted in figure 2-36(b) is representative.

In this orientation, the body of the spacecraft is perpendicular to the orbit plane (X-POP). The solar array is in the orbit plane and the vehicle is rolled around its X-axis so that the solar array is perpendicular to the Sun line (Y-PSL). In this attitude, the thermal radiator is always edge-on to the Sun and a single-degree-of-freedom beta-tilt is sufficient to track the Sun.

Because the SASP is a relatively simple, regular configuration, the inertial principal axes are closely aligned with the body axes. The orientation described places two of the principal axes in or nearly in the orbit plane, thereby nearly eliminating secular gravity-gradient torques. Cyclic torques about the X-axis occur and are readily controlled by momentum management devices.

2.3.8.4 Slack-Tethered Power Generation Module

Solar arrays significantly affect Space Station operations because of their size, loction and orientation. They are relatively large items located on long booms on either side of the station and are Sun-oriented, while the station core (for the cases studied here) is Earth-oriented. The relative movement of these two major (and massive) hardware items has significant impact on station propulsion requirements.

To overcome these difficulties, the power generation system could be isolated from the rest of the station. Transmitting power to the station might be done with microwaves or via a transmission line. The propulsion requirements of a free-flying power generation module using microwave transmission would not differ significantly from those of the SASP discussed previously.

The power generation module would have a much smaller ballistic coefficient than the core vehicle. Such tethered combinations experience large aero drag torques and may have two stable attitudes depending on atmospheric density, as shown in figure 2-37. To avoid these and other complications, we elected to examine a slack-tethered power generation module. A slack, as opposed to a taut, transmission line (tether) prevents disturbances such as docking, attitude change maneuvers, and gravity-gradient forces from being transmitted from the station to the free-flyer. A system incorporating a taut tether, i.e., the tethered propellant farm previously discussed in this section, does transmit disturbances through the tether.

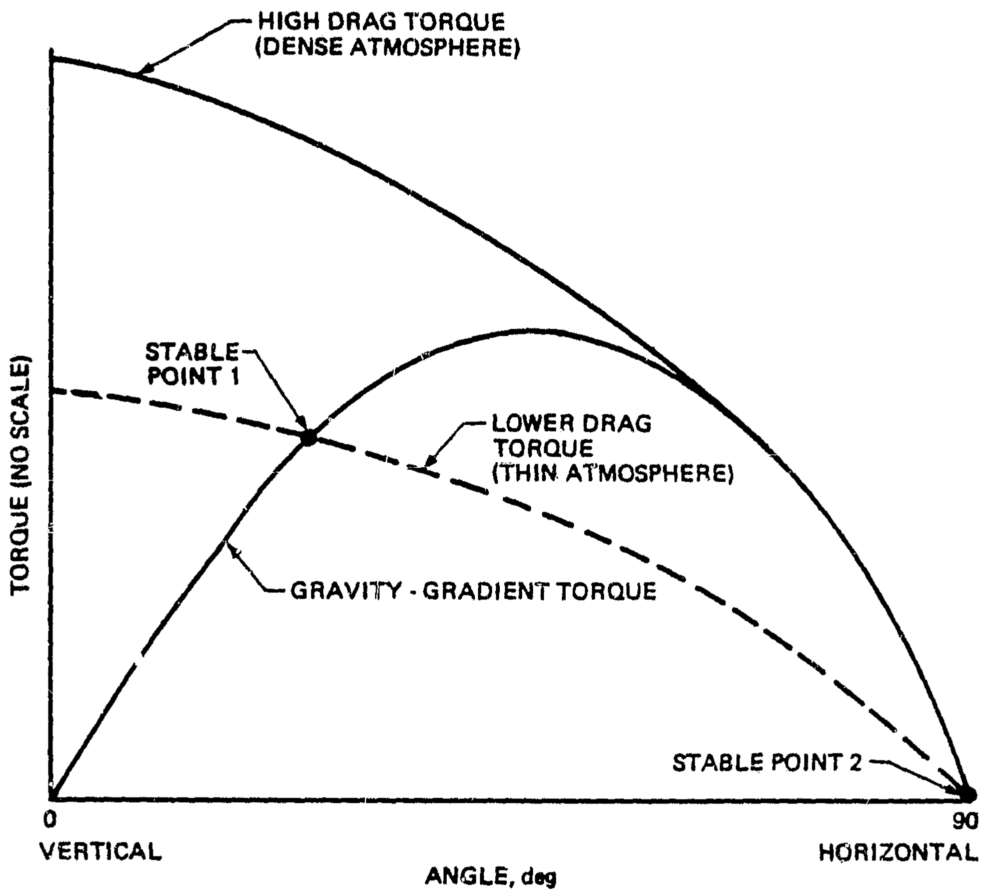


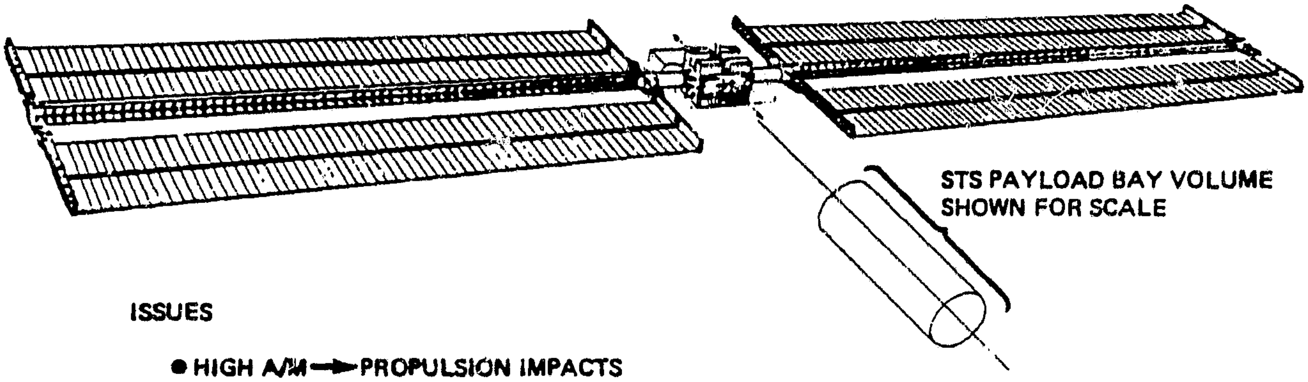
Figure 2-37. Stable Points for Taut-Tethered Subplatform with Lesser Ballistic Coefficient Than Main Platforms

Additionally, gravity-gradients cause a body force to exist at either end of the tether. However, an analysis of forces that are transmitted through either a taut or slack tether is beyond the scope of this study.

The slack-tethered power generation module (STPGM), shown in figure 2-38, is envisioned to consist primarily of two large solar arrays on booms with a central power module for power conversion and conditioning. The tether is attached at the power module and a gap is provided between the arrays to facilitate tether movement as the station and the STPGM move relative to each other. An evaluation of this movement is beyond the scope of the current study. A small radiator, to dissipate power lost in the processing equipment, extends from the power module. A docking mechanism to facilitate servicing is also provided.

<u>Transmission Line Losses for the STPGM</u>. Solar arrays generate power at relatively low voltage levels (200V is typical). Higher voltage levels can be achieved, but the low-density plasma in LEO causes arcing at voltages above about 200V. Electrical power (P) delivered through a conductor can be expressed as $P = EI$. It is apparent that, for a given power level, a low voltage (E) requires a high current (I). However, because power loss due to resistance (R) can be expressed as $P = I^2R$, there is an advantage to reducing the current, and therefore, increasing the voltage. Of course, resistance must be minimized, but for the STPGM application, weight, size, and stiffness impose limits for conductor design.

Figure 2-39 illustrates the effect of voltage on line losses for a given conductor. It shows that an order of magnitude reduction in line loss can occur if the voltage can be raised from 200V to 600V. Therefore, the system requires a transformer to step up the voltage from the array for transmission to the station via the slack tether and then to step it down again for use at the station. However, since transformers work only with alternating current, whereas the array produces direct current, a dc to ac inverter is also required. Figure 2-40 depicts a system concept for accomplishing the dc to ac conversion and the required voltage changes that were adapted from a Westinghouse design. Westinghouse Electric (and, presumably, other manufacturers) produces various inverter types and sizes,



ISSUES

- HIGH A/M → PROPULSION IMPACTS
- MAX ALLOWED TETHER LENGTH
  - LINE LOSSES & PLASMA INTERACTIONS

Figure 2-38. Slack-Tethered Remote Power Generation Module

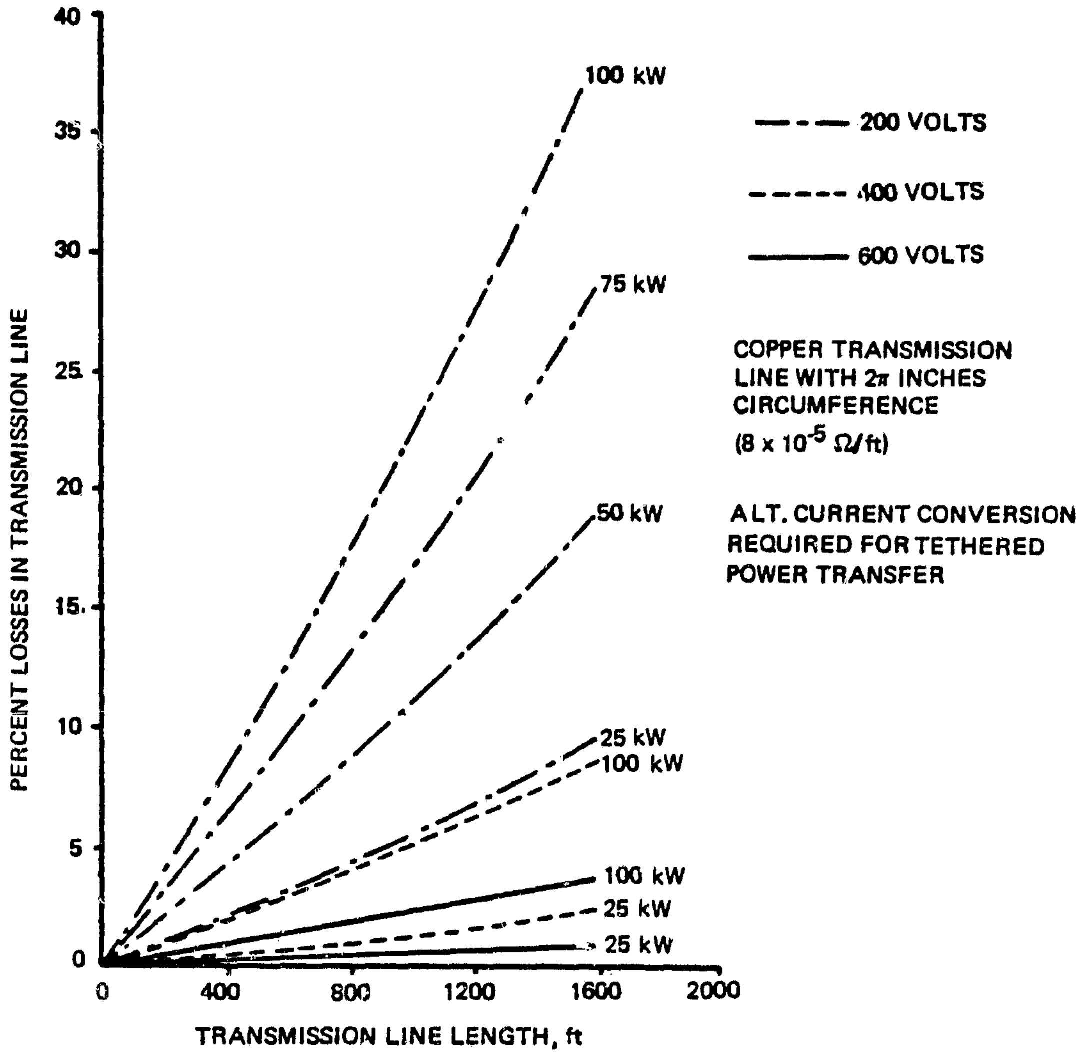


Figure 2-39. Influence of Voltage on Transmission Line Losses

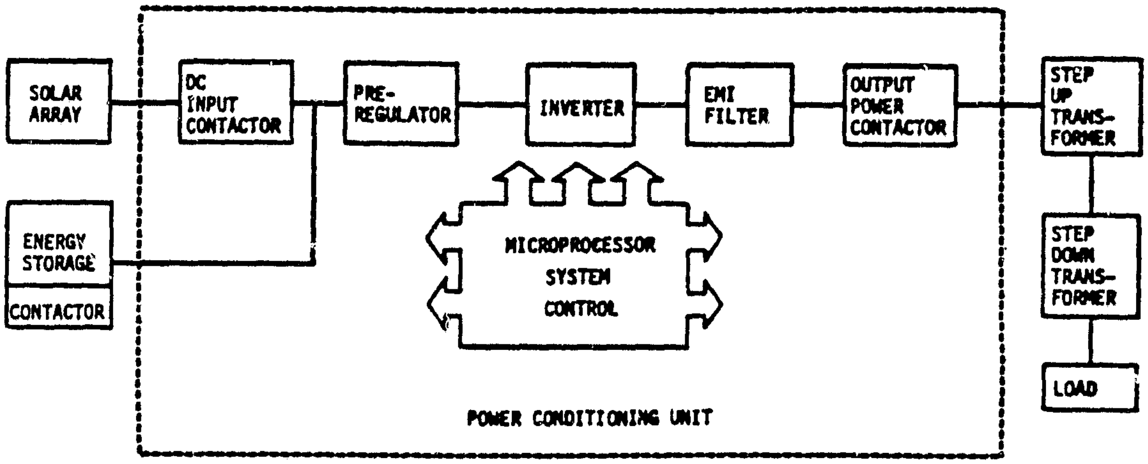


Figure 2-40. Use of Inverter to Limit Line Losses

as listed in Table 2-6.

Further development of this concept is beyond the scope of this study. It is assumed that a viable system is within or close to the state of the art.

STPGM Tether Design. The tether must be highly flexible to minimize the forces on both the station and the STPGM, which will affect stationkeeping requirements. The selected tether concept is a flat copper strip 7.6 cm (3 in) wide and 0.38 cm (0.15 in) thick. This tether will resemble a ribbon and is highly flexible. Electrical insulation, with a low solar absorptivity and emissivity coating, will cover the copper strip. The necessary conductors for C^3 will be embedded in the insulation.

In summary, a tether allows for the transfer of data, power, and possibly thermal working fluids. On the other hand, untethered satellites can: maintain orbits widely separated from the Space Station, thereby eliminating disturbances and contamination; obtain greater pointing accuracy; and have a range of orientations beyond that allowable for the station.

2.4 Configuration Summary

The configuration analysis surveyed a range of Space Station and platform configuration options, and selected certain ones as representative of those characteristics that generate propulsion requirements. These representative options were used for the balance of the study.

The solar array configuration has a predominant effect on Space Station propulsion requirements through its contributions to aerodynamic drag and gravity-gradient torques. We elected to concentrate on articulated Sun-tracking arrays even though a number of innovative concepts that did not Sun-track were being actively investigated by other studies during this period. It was our judgement that the mass and cost penalties incurred by not Sun-tracking outweighed any advantages.

Table 2-6. Typical Westinghouse Inverters

Continuous Kva	Number & Type Power Stages	Continuous Induction	Speed Range Synchronous	Dimensions-Inches Height	Depth	Width	Weight* lbm
50	4/35	4.1	3.1	90	30	30	1200
60	4/70	4.1	3.1	90	30	30	1300
75	4/70	4.1	3.1	90	30	30	1400
100	6/70	6.1	5.1	90	30	30	1700
125	8/70	8.1	7.1	90	30	60	2000
150	10/70	10.1	9.1	90	30	60	2300
175	10/70	10.1	9.1	90	30	60	2600
200	12/70	12.1	11.1	90	30	60	2900
250	7/150	7.1	6.1	90	30	90	3600
300	9/150	9.1	8.1	90	30	120	4700
350	10/150	10.1	9.1	90	30	120	5000
400	12/150	12.1	11.1	90	30	150	6000

*Done for 60 Hz system—this weight would be significantly reduced if a higher frequency was used.

During the period of this study, new insights into Space Station needs and requirements were being developed by NASA and its contractors. Two of importance to Space Station propulsion dealt with power levels and shuttle direct insertion.

The Space Station mission requirements working group in May of 1983 identified requirements for power to Space Station users in excess of 60 kW at the beginning of Space Station operations, and more than twice that at the end of a ten-year operational period. Earlier studies had assigned much smaller values to user power needs; the net result was that Space Station solar array area estimates roughly doubled.

Also during this period, "direct insertion" was identified as a viable flight mode for the space shuttle. Whereas earlier Space Station studies had concentrated on mission altitudes in the range of 350 km to 400 km, direct insertion readily permits altitudes of 500 km or more. Inasmuch as there are valid mission reasons to adopt the higher altitude, the various Space Station studies quickly did so. This change in altitude reduces the importance of drag enough that attitude control factors are much more significant in setting propulsion requirements.

Most of the configurations selected for detailed study were Earth-oriented and located in a 28.5^{0} inclination orbit. Station sizes corresponded to 2-4, 4-6, and 8-12 crew members. Solar array sizes from 1000 to 4000 square meters were considered, covering the power range of interest. High-inclination options were also considered because they tend to have much lower power needs and are constrained to lower altitudes because of the reduced performance capability of the space shuttle at high inclinations.

There are several free-flyers described that are representative of those that might be developed and offer alternatives to certain Space Station design elements: two tethered variants (the tethered propellant farm and the slack-tethered power generation module), a compact and low area-to-mass ratio vehicle (the free-flying propellant farm), and the high area-to-mass ratio vehicle (SASP). The Space Station "system" also includes servicing vehicles that perform a range of activities, from transferring payloads between the STS (Orbiter), Space Station, and/or free-flyers, to performing rescue operations. The OMV and OTV servicing vehicles can effectively accomplish short- and long-range missions, respectively. Although these vehicles are perfectly satisfactory for a wide variety of missions, the baseline OMV has excess capability for those missions in the immediate vicinity of the station.

Therefore, these configuration "elements" form the basis for the remaining analysis, which encompasses environmental effects, servicing strategies, and propulsion/propellant requirements.

3.0 CONFIGURATION/ENVIRONMENTAL EFFECTS

This section evaluates how environmental factors, such as aerodynamic torques, drag, and gravity-gradient forces, influence configuration design. Propulsive and non-propulsive means for countering these effects are also introduced, which are discussed in more detail in section 5.0.

3.1 Mass Properties

The term mass properties refers primarily to products of inertia, rather than to mass in the normal usage. The mass properties are derived from the distribution of mass of station components. It was shown during the study of the various Earth-oriented Space Station configurations that, in many cases, the core configuation has a relatively small role in determining the overall mass properties of the station. The solar array arrangement and the Orbiter docking configuration assume much greater roles in affecting station mass properties.

The planar, Earth-oriented station configurations carried forward in this study (in three distinct growth stages) have Earth-oriented cores and arrays that track the Sun. The solar arrays have two axes of rotation with respect to the core as seen from Figure 2-13. The rotation of solar panels about the Y body axis accounts for the orbital angle, referred to as theta in table 3-1. The rotation of the solar panels out of the orbit plane allows solar tracking. This Sun angle is referred to as beta and varies seasonally from -52 deg to +52 deg (for a 28.5 deg inclination orbit plane). Appendix A provides a detailed analysis of the solar array motion. The inertia tensors for the arrays without the core listed in table 3-1 are the result of this analysis. Due to the solar panel rotation, the products and cross products of inertia for the solar panels are not constant when they are referenced to the station core body frame. For a given solar array area, the inertia terms are a function of the mass, the array aspect ratio (ratio of length to width), the Sun angle (beta), and the orbital angle (theta). Table 3-1 lists the baseline solar array inertia tensors as a function of the Sun and orbital angles for both the cantilevered and

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OF POOR QUALITY

Table 3-1. Baseline Solar Array Inertia Tensors (1,000 kg-m²)

AREA (m²)	φ DEG	BALANCED ARRAY β = 0.0 DEG	BALANCED ARRAY β = 45.0 DEG	BALANCED ARRAY β = 52.0 DEG	CANTILEVERED ARRAY β = 0.0 DEG	CANTILEVERED ARRAY β = 45.0 DEG	CANTILEVERED ARRAY β = 52.0 DEG
1,046	0	254 0 0	219 35 0	210 34 0	254 0 0	325 557 0	342 600 0
		0 183 0	35 219 0	34 227 0	0 3,206 0	557 2,791 0	600 2,666 0
		0 0 70	0 0 70	0 0 70	0 0 70	0 0 70	0 0 70
	45	162 0 91	144 24 74	140 24 70	162 0 91	250 762 180	272 825 201
		0 183 0	24 219 24	24 227 24	0 6,228 0	762 5,362 762	825 5,106 825
		91 0 162	74 24 144	70 24 140	91 0 162	180 762 250	201 825 272
	90	70 0 0	70 0 0	70 0 0	70 0 0	70 0 0	70 0 0
		0 183 0	0 219 35	0 227 34	0 12,273 0	0 10,506 2,122	0 9,982 2,299
		0 0 254	0 35 219	0 34 210	0 0 254	0 2,122 642	0 2,299 736
2,000	0	1,424 0 0	1,356 67 0	1,340 65 0	1,424 0 0	1,569 1,066 0	1,591 1,149 0
		0 1,289 0	67 1,356 0	65 1,372 0	0 7,073 0	1,066 6,279 0	1,149 6,040 0
		0 0 135	0 0 135	0 0 135	0 0 135	0 0 135	0 0 135
	45	779 0 644	745 47 610	737 46 602	779 0 644	948 1,460 813	989 1,579 854
		0 1,289 0	47 1,356 47	46 1,372 46	0 12,858 0	1,460 11,201 1,460	1,579 10,707 1,579
		644 0 779	610 47 745	602 46 737	644 0 779	813 1,460 948	854 1,579 989
	90	135 0 0	135 0 0	135 0 0	135 0 0	135 0 0	135 0 0
		0 1,289 0	0 1,356 67	0 1,372 65	0 24,427 0	0 21,046 4,062	0 20,042 4,401
		0 0 1,424	0 67 1,356	0 65 1,340	0 0 1,424	0 4,062 2,167	0 4,401 2,347
4,000	0	3,659 0 0	3,118 540 0	2,987 524 0	3,659 0 0	4,739 5,346 0	5,001 5,646 0
		0 2,578 0	540 3,118 0	524 3,249 0	0 21,082 0	5,346 17,363 0	5,646 16,278 0
		0 0 1,080	0 0 1,080	0 0 1,080	0 0 1,080	0 0 1,080	0 0 1,080
	45	2,369 0 1,289	2,099 382 1,018	2,034 370 953	2,369 0 1,289	3,721 7,178 2,640	4,047 7,614 2,967
		0 2,578 0	382 3,118 382	370 3,249 370	0 39,587 0	7,178 31,609 7,178	7,614 29,308 7,614
		1,289 0 2,369	1,018 382 2,099	953 370 2,034	1,289 0 2,369	2,640 7,178 3,721	2,967 7,614 4,047
	90	1,080 0 0	1,080 0 0	1,080 0 0	1,080 0 0	1,080 0 0	1,080 0 0
		0 2,578 0	0 3,118 540	0 3,249 524	0 76,597 0	0 60,100 19,763	0 55,367 21,012
		0 0 3,659	0 540 3,118	0 524 2,987	0 0 3,659	0 19,763 9,603	0 21,012 11,041

840883-32

balanced arrangements. The area, mass, and aspect ratios for the baseline configurations are listed in table 3-2. (Appendix A provides solar array mass properties as a function of aspect ratio). As mentioned in section 2.3.5, the baseline array mass and power densities are more conservative than those assumed for the interim, Sun-synchronous, and polar configurations. The power requirements are also higher for the baseline configuration. Baseline radiator areas are 477, 911, and 1822 m^2 for the 2- to 4-, 4- to 6-, and 8- to 12-man stations, respectively.

The advantages of the balanced array configurations are evident from table 3-1. The cross products of inertia for worst case theta and beta angles, are in many cases, considerably more than an order of magnitude larger for

Table 3-2. Baseline Solar Array Characteristics

Number Men	2 to 4	4 to 6	8 to 12
Area (m^2)	1046	2000	4,000
Mass (kg)	2613	5000	10,000
Aspect Ratio	1.01	3.08	1.54
BOL Power (kW)	110	210	420

the cantilevered arrays. As discussed in Appendix A and section 3.3 (Gravity Gradients), a reduction in the cross products of inertia has a significant effect on reducing gravity-gradient torques and momenta.

Table 3-3(a) lists the inertia tensors for the baseline planar core configurations. These tensors are constant since they are referenced to the station core. Two sets of inertia tensors are shown for each of the core configurations. The first tensor is for the Space Station only, while the second includes a docked Orbiter and orbital maneuvering vehicle (OMV). This table graphically illustrates the importance of Orbiter docking on

Table 3-3(a). Baseline Planar Core Mass Properties

CREW SIZE (PEOPLE)	UNDOCKED				DOCKED (OMV AND ORBITER)			
	MASS (kg)	INERTIA TENSOR (1,000 kg-m2)			MASS (kg)	INERTIA TENSOR (1,000 kg-m2)		
2-4	68,534	1,072 0 0	0 1,751 0	0 0 1,673	190,642	18,608 -6,106 -1,689	-6,106 17,952 -1,493	-1,689 -1,493 15,753
4-6	102,804	3,489 0 19	0 3,376 0	19 0 3,284	224,912	22,484 0 -12,878	0 40,397 0	-12,878 0 23,052
8-12	153,450	12,310 0 326	0 9,823 0	326 0 7,447	275,558	35,272 0 -10,666	0 44,460 -465	-10,666 -465 20,083

840883-33

Table 3-3(b). Mass Properties for Baseline Earth-Oriented, 28.5° Inclination Space Station

CREW SIZE STATION TYPE AND ARRAY SIZE	DOCKING CONFIGURATION	STATION MASS (KG)	AVERAGE AREA*/ MASS (M^2/KG)	CP – CG OFFSET (M)	
				Y	Z
2 TO 4 MAN SCIENTIFIC 1046m²	NO DOCKED VEHICLES	77,718	0.0098	0	0.79
	OMV AND ORBITER	199,828	0.0063	−3.87	−0.41
4 TO 6 MAN SCIENTIFIC 2000m²	NO DOCKED VEHICLES	113,126	0.0126	0	−0.10
	OMV AND ORBITER	236,233	0.0073	0	3.91
4 TO 6 MAN CONSTRUCTION 2000m²	NO DOCKED VEHICLES	120,120	0.0129	−0.17	−1.29
	OMV AND ORBITER	278,587	0.0066	−0.34	3.25
8 TO 12 MAN SCIENTIFIC 4000m²	NO DOCKED VEHICLES	166,132	0.0168	0	1.65
	OMV AND ORBITER	288,239	0.0107	0	−3.66
8 TO 12 MAN CONTRUCTION 4000m²	NO DOCKED VEHICLES	178,545	0.0163	0.01	−4.01
	OMV, OTV, AND ORBITER	338,301	0.0094	−0.28	−5.88

* BASED ON AVERAGE AREA PROJECTED IN DIRECTION OF TRAVEL OVER THE ENTIRE ORBIT

C - 2

mass properties. The station core without the docked Orbiter is well balanced, whereas the station core with the Orbiter is an unbalanced configuration, thereby causing attitude maintenance problems.

To determine the mass properties of a baseline Space Station configuration, it is necessary to add the inertia tensors in tables 3-1 and 3-3(a). Note that the undocked core cross products of inertia are negligible compared to the solar array cross products. These tables, used in conjunction with data in section 3.3, provide an estimate of momentum and propellant requirements due to gravity-gradient effects. Additional mass properties for the baseline stations are shown in table 3-3(b).

Mass properties for the Earth-oriented, 28.5 deg, interim configuration with and without various docked vehicles, are shown in table 3-4. These configurations used the cantilevered array exclusively. The station cores for the interim configuration are identical to those in the baseline configuration. The differences between them are in the power level, solar array efficiency, and radiator design. Table 3-5 shows the mass properties of the polar and Sun-synchronous stations, with and without the Orbiter docked. The mass properties of free-flyers defined in this study are shown in table 3-6.

3.2 Aerodynamic Force Effects

There are two primary disturbances that affect the pointing and orbital stability of the Space Station: aerodynamic effects and gravity-gradient torques. The relative magnitude of these disturbances is a function of altitude, orientation, configuration, and atmospheric density. An example of aerodynamic and gravity-gradient torques as affected by altitude for cantilevered arrays is shown in figure 3-1. The figure was plotted for one of our selected configurations as noted. Aerodynamic torque is, of course, linearly dependent on the offset distance between center of pressure (CP) and center of gravity (CG), and hence quite sensitive to configuration details. The main point to be made here is the great reduction in significance of aerodynamic drag at the higher altitudes. Other disturbances that perturb the orbit or affect station pointing

Table 3-4. *Mass Properties for the Interim Configuration Earth-Oriented, 28.5-deg Inclination Space Station*

CREW SIZE, PURPOSE & POWER LEVEL AT BOL	ARRAY AREA (M^2)	RADIATOR TYPE AND WETTED AREA (M^2)	DOCKING CONFIG-URATION	STATION MASS (KG)	CP–CM OFFSET (M)		INERTIA TENSOR (1000 KG–M^2)		
					Y	Z			
2 TO 4 MAN SCIENTIFIC 25 KW	193.6	CRUCIFORM 325.3	UNDOCKED	60,978	-2.71	0	1457 0 0	0 748 13.7	0 13.7 754
			WITH OMV	66,480	-2.61	0	1482 22.3 0	22.3 1204 13.7	0 13.7 1211
			WITH ORBITER	178,912	-0.07	-7.09	17994 1230 -4499	1230 19507 1659	-4499 1659 5687
			WITH OMV AND ORBITER	184,414	0.01	-1.74	18376 1454 -5119	1454 21057 1773	-5119 1773 6963
2 TO 4 MAN SCIENTIFIC 60 KW	464.7	PLANAR 260.2	UNDOCKED	75,777	-1.0	0	1786 51 0	51 2009 21	0 21 1752
			WITH OMV AND ORBITER	197,884	-0.1	-4.16	1946 1918 -6427	1918 24550 2022	-6427 2022 9575
4 TO 6 MAN SCIENTIFIC 50 KW	387.2	CRUCIFORM 650.6	UNDOCKED	94,061	-0.73	0	2589 -83.5 0	-83.5 2789 79	0 79 2479
			WITH OMV AND	129,933	-2.05	1.52	13868 -194 8.7	-194 16323 -340	8.7 -340 7215
4 TO 6 MAN SCIENTIFIC 110 KW	851.9	PLANAR 477.1	UNDOCKED	109,252	0.99	0	3391 415 0	415 4925 65	0 65 4245
			WITH OMV AND	231,359	-5.03	-9.09	32406 2761 -8906	2761 32841 2123	-8906 2123 16004

840683-19

Table 3-4. *Mass Properties for the Interim Configuration Earth-Oriented, 28.5-deg Inclination Space Station (Continued)*

CREW SIZE PURPOSE & POWER LEVEL AT BOL	ARRAY AREA (M^2)	RADIATOR TYPE AND WETTED AREA (M^2)	DOCKING CONFIG-URATION	STATION MASS (KG)	CP–CM OFFSET (M)		INERTIA TENSOR (1000 KG–M^2)		
					Y	Z			
4 TO 8 MAN CONSTRUCTION 50 KW	387.2	CRUCIFORM 850.8	UNDOCKED	86,632	0.87	0.14	2027 -383 -61	-383 2562 62	-61 82 1616
			WITH OMV AND ORBITER	188,739	7.18	-13.44	21855 -1904 1882	-1904 21224 3539	1882 3539 6575
4 TO 6 MAN CONSTRUCTION 110 KW	851.9	PLANAR 477.1	UNDOCKED	116,247	4.98	1.22	3537 483 -41	483 4676 82	-41 82 4072
			WITH OMV OTV, AND ORBITER	276,047	3.72	5.09	28211 -4229 6118	-4229 32190 4125	6118 4215 16159
6 TO 8 MAN SCIENTIFIC 75 KW	580.9	CRUCIFORM 975.8	UNDOCKED	122,615	-2.24	0	3683 10.5 0	10.5 5445 41.2	0 41.2 4781
			WITH OMV AND ORBITER	217,497	0.31	-2.23	21629 763 -7235	763 26759 1152	-7235 1152 11343
8 TO 12 MAN SCIENTIFIC 100 KW	774.5	CRUCIFORM 1301.1	UNDOCKED	142,083	-1.66	0	4505 15.6 0	15.6 9882 129	0 129 8359
			WITH OMV	147,585	-2.15	0	4600 367 5936	367 10006 7190	0 129 19949
			WITH ORBITER	260,017	2.56	-4.06	29714 -4755 5936	-4461 33005 7190	5813 7190 19949
			WITH OMV AND ORBITER	365,519	-5.86	-3.98	30750 -4461 5813	-4461 33281 7551	5813 7551 20934

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Table 3-4 . Mass Properties for the Interim Configuration Earth-Oriented, 28.5-deg Inclination Space Station (Continued)

CREW SIZE PURPOSE & POWER LEVEL AT BOL	ARRAY AREA (M^2)	RADIATOR TYPE AND WETTED AREA (M^2)	DOCKING CONFIG-URATION	STATION MASS (KG)	CP–CM OFFSET (M)		INERTIA TENSOR (1000 KG–M^2)		
					Y	Z			
8 TO 12 MAN SCIENTIFIC 240 KW	1858.7	PLANAR 1040.9	UNDOCKED	159,048	3.39	0	7780 868 0	866 12627 294	0 294 10199
			WITH OMV AND ORBITER	281,155	3.08	-2.53	36289 -3723 5950	-3723 37148 8732	5950 8732 23365
8 TO 12 MAN CONSTRUCTION 100 KW	774.5	CRUCIFORM 1301.1	UNDOCKED	172,236	11.01	0.09	11366 -21.5 -57.4	-21.5 9473 87.3	57.4 87.3 14635
			WITH ORBITER	290,170	10.5	-7.22	31476 -737 10394	-737 41513 720	10394 720 28380
			WITH OTV AND ORBITER	327,864	8.54	-7.41	35533 -4498 4207	-4498 35138 1591	4207 1591 25599
8 TO 12 MAN CONSTRUCTION 240 KW	1858.7	PLANAR 1040.9	UNDOCKED	171,461	6.48	0.98	9392 -184 -45	-1846 12708 284	-452 284 11600
			WITH OMV, OTV, AND ORBITER	331,216	11.66	-21.77	23569 -3561 4102	-3561 29425 2947	4102 2947 16568

840883-21

Table 3-5. Mass Properties for Earth-Oriented Polar and Sun-Synchronous Stations

CREW SIZE PURPOSE & POWER LEVEL AT BOL	ARRAY AREA (M^2)	RADIATOR TYPE AND WETTED AREA (M^2)	DOCKING CONFIG-URATION	STATION MASS (KG)	CP–CM OFFSET (M)		INERTIA TENSOR (1000 KG–M^2)		
					Y	Z			
2 TO 4 MAN POLAR SCIENCE 25 KW	198.6	CRUCIFORM 325.3	UNDOCKED	48,958	0.1	0.1	1197 -7.5 0	-7.5 1507 0	0 0 1219
			WITH ORBITER	157,820	7.98	-10.95	9409 178 -667	178 7353 3287	-667 3287 4883
8 TO 12 MAN POLAR SCIENCE 100 KW	774.5	CRUCIFORM 1301.1	UNDOCKED	72,684	-1.25	0	3781 217 0	217 4555 -131	0 -131 2582
			WITH ORBITER	181,546	-9.09	-11.92	13608 -250 -1112	-250 11803 -4070	-1112 -4070 6598
2 TO 4 MAN SUN SYNCHRONOUS SCIENCE, 25 KW	193.6	PLANAR 325.3	UNDOCKED	83,118	0	0	1453 0 -839	0 936 0	-839 0 14066
			WITH ORBITER	150,264	0	-12.7	28035 22 -8945	22 1150 -210	-8945 -210 16596
8 TO 12 MAN SUN SYNCHRONOUS SCIENCE, 100 KW	774.5	PLANAR 1301.1	UNDOCKED	162,246	0	0	4025 0 0	0 3489 0	0 0 37423
			WITH ORBITER	228,391	0	-13.08	30606 22 -8105	22 3702 -210	-8105 -210 39952

840683-17

Table 3-6. Free Flyer Mass Properties

CREW SIZE PURPOSE & POWER LEVEL AT BOL	ARRAY AREA (M^2)	RADIATOR TYPE AND WETTED AREA (M^2)	ORIENTATION	MASS (KG)	CP–CM OFFSET (M)		INERTIA TENSOR (1000 KG–M^2)		
					Y	Z			
PROPELLANT STORAGE 6 KW	35	INTEGRAL 24.6	INERTIAL	129,811 (FULL)	1.02	-0.01	2006 -106.7 220.5	-106.7 2259 435.7	220.5 435.7 728.4
PROPELLANT STORAGE POWER FROM STATION VIA TETHER	NA	INTEGRAL 156	STATION-ORIENTED DUE TO TETHER	128,452 (FULL)	-2.63	3.04	1520 156.7 -261.1	156.7 1520 260.4	-261.1 260.4 658.3
POWER GENERATION MODULE 160 KW	936.8	PLANAR 25	INERTIAL	3,720	0	0	91.1 0 0	0 284.6 0	0 0 375.7
SASP 33.3 KW	237	PLANAR 79.7	EARTH-ORIENTED	8,780	0	0.84	89.5 0 -9.1	0 131.6 0	9.1 0 116.5
SASP 66.7 KW	473	PLANAR 159.4	EARTH-ORIENTED	14,731	0	1.09	534.4 0 -0.82	0 407.6 0	0.82 0 389.8

840683-22

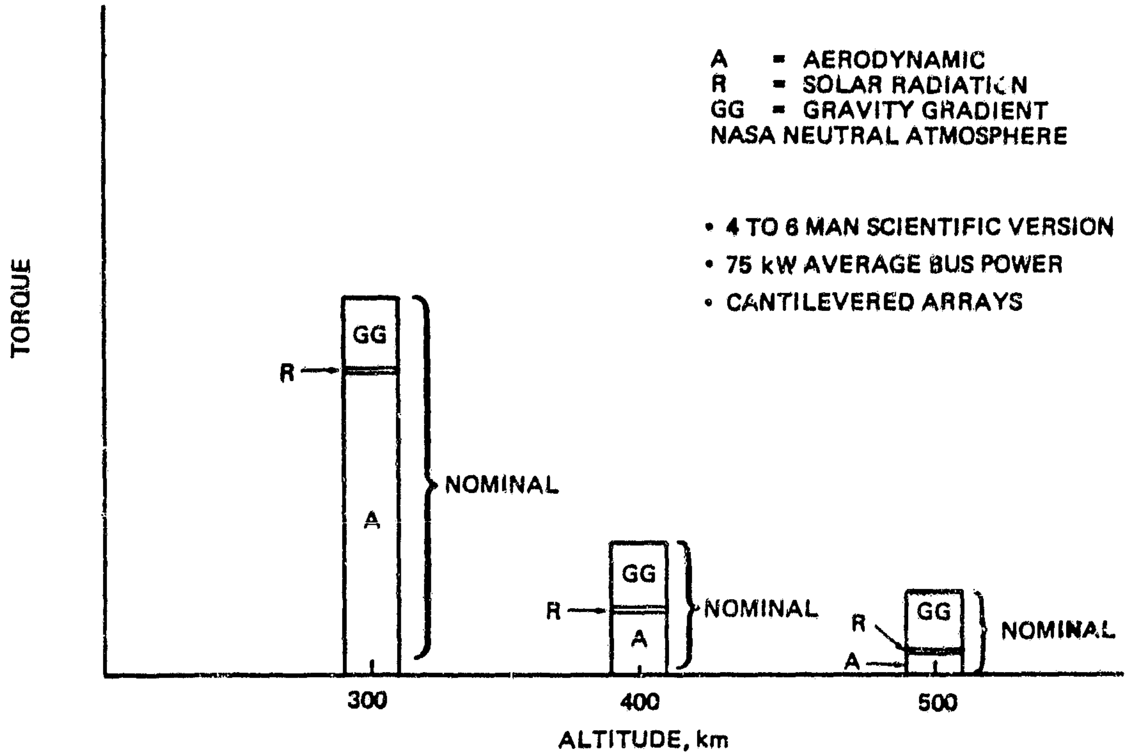

Figure 3-1. Relative Environmental Torque Magnitudes

include Earth triaxiality, lunar and solar gravitational harmonics, solar pressure, and magnetic torques. Magnetic torques can be used to counter aerodynamic and gravity-gradient torques (see section 3.4). The other disturances are small, for the configurations and altitudes studied, relative to aerodynamic and gravity-gradient effects, and have been deleted from further analysis. Gravitational harmonics are discussed in the section on orbit decay (3.2.3).

There are two separate phenomena that are caused by aerodynamic influences: aerodynamic torques and aerodynamic drag. These effects are proportional to atmospheric density, which is a factor up to about 1,000 km. The determination of atmospheric density and the assumptions made about orbit decay, momentum sizing, and propellant resupply, as well as total impulse requirements, are critical in determining propulsion, servicing, operating, and even configuration requirements.

3.2.1 Air Density at Low Earth Orbit

Atmospheric density is difficult to estimate accurately. The density at a given altitude may vary by a factor of five from the sunlit to the dark side of the orbit and by a factor of two depending on the latitude and longitude. Density will vary with the season by as much as a factor of four, being higher at a given altitude during the summer months for that hemipshere. In this study, these effects have been averaged over periods varying from months to years to estimate resupply requirements. The seasonal variation is significant in determining the worst case density for the 90-day maximum resupply interval.

However, the seasonal variation effect on atmospheric density is small when compared to solar activity which fluctates on an 11-year cycle and changes atmospheric density by several orders of magnitude at a given altitude. Solar flux has a direct impact on atmospheric density because it is the source of heat and molecular energy that causes the atmosphere to expand or contract. Solar activity levels follow a number of short- and long-term cycles, the most prominent being the previously mentioned 11-year solar cycle associated with Sunspot activity. Solar activity varies significantly throughout this cycle. The indeterminate effect of both seasonal and solar influence on atmospheric density make a 90-day decay profile difficult to predict.

Figures 3-2 shows the Sunspot index for solar cycles from 1750 to the present and indicates that the monthly average values can vary widely, especially during periods of high solar activity. The effect of extreme ultraviolet (EUV) radiation on atmospheric density has been treated at length elsewhere[1] and it was shown that the EUV radiation emitted by the Sun correlates well with the 10.7-cm long-wavelength radio noise emitted from the Sun which can be measured on Earth. The intensity of the 10.7-cm solar flux varies monthly during the 11-year solar cycle as shown in figure 3-3.

The methods used to predict future solar activity do not attempt to predict the kind of monthly fluctuations shown in figure 3-3, but try to predict solar flux values smoothed over several months. The major current method uses the mean monthly solar flux values smoothed over a 13-month period. The prediction method seeks to predict monthly values for the smoothed flux for about 17 years in the future.

Figure 3-4 shows the average and extreme values of the 10.7-cm solar flux for the past known solar cycles. The extremes are referred to as the +2 sigma (97.7 percentile) and the -2 sigma (2.3 percentile). The most recent solar predictions, which extend to the year 2000, are shown in figure 3-5. It is important to note that the differences between +2 sigma and the -2 sigma values in figures 3-4 and 3-5 represent a 95% probability range for the smoothed solar flux. In particular, the +2 sigma or 97.7 percentile value gives no information regarding any short-term higher values of the solar flux.

[1] L. G. Jacchia, "New Static Models of the Thermosphere and Exosphere with Empirical Temperature Profiles," Smithsonian Astropohysical Special Report, No. 313, Washington D.C., 1970.

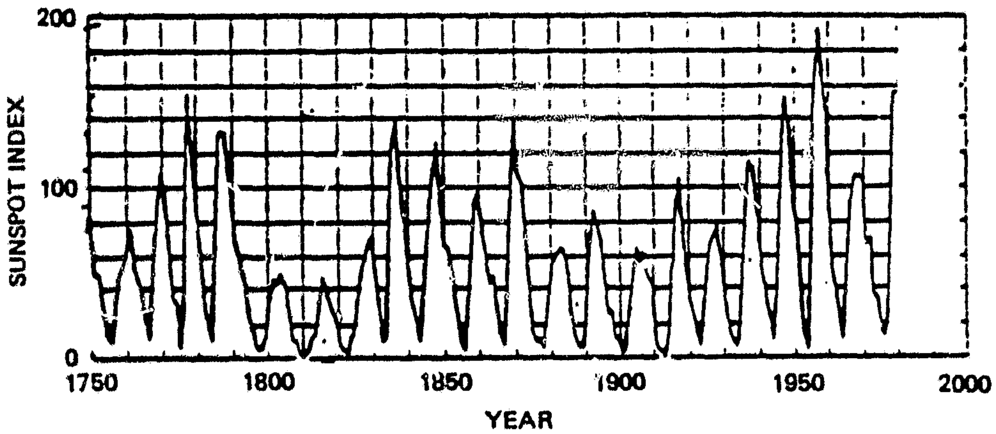

Figure 3-2. Sunspot Number Chronology Derived From Direct Measurements and Historical Documents. Solar Cycles 1 to 21 (1756 to 1976).

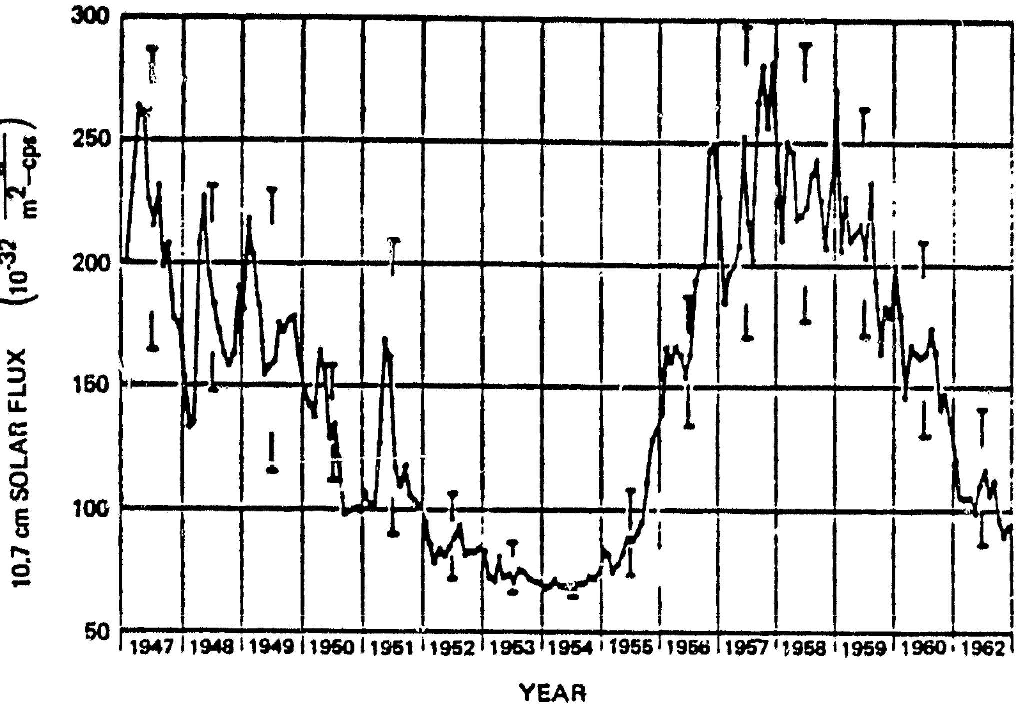

Figure 3-3. *Variation of Monthly Averages of 10.7-cm Solar Flux According to Measurements by the National Research Council of Canada. (Bars indicate the maximum and minimum fluctuations in June and July of each year.)*

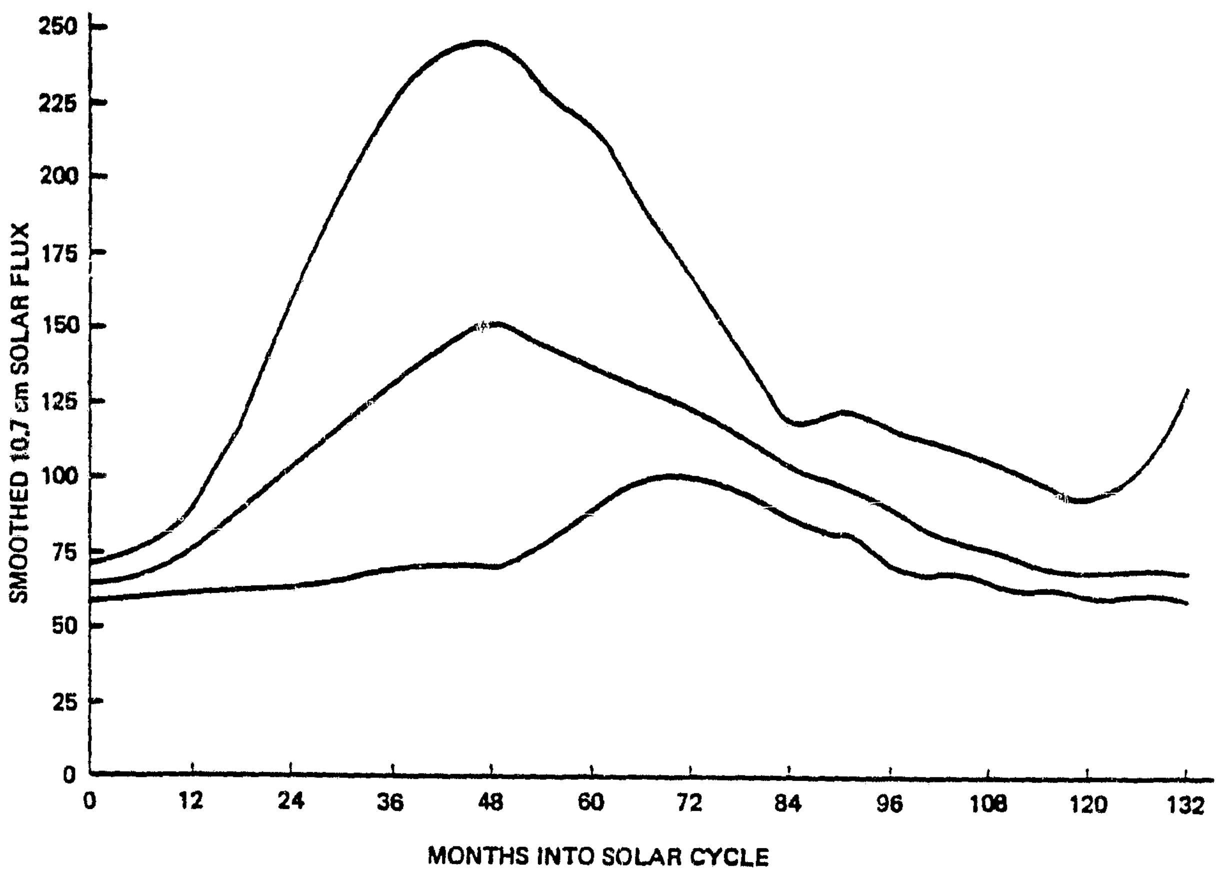

Figure 3-4. Extreme and Mean Values of the Smoothed 10.7-cm Solar Flux for Solar Cycles 1 to 21.

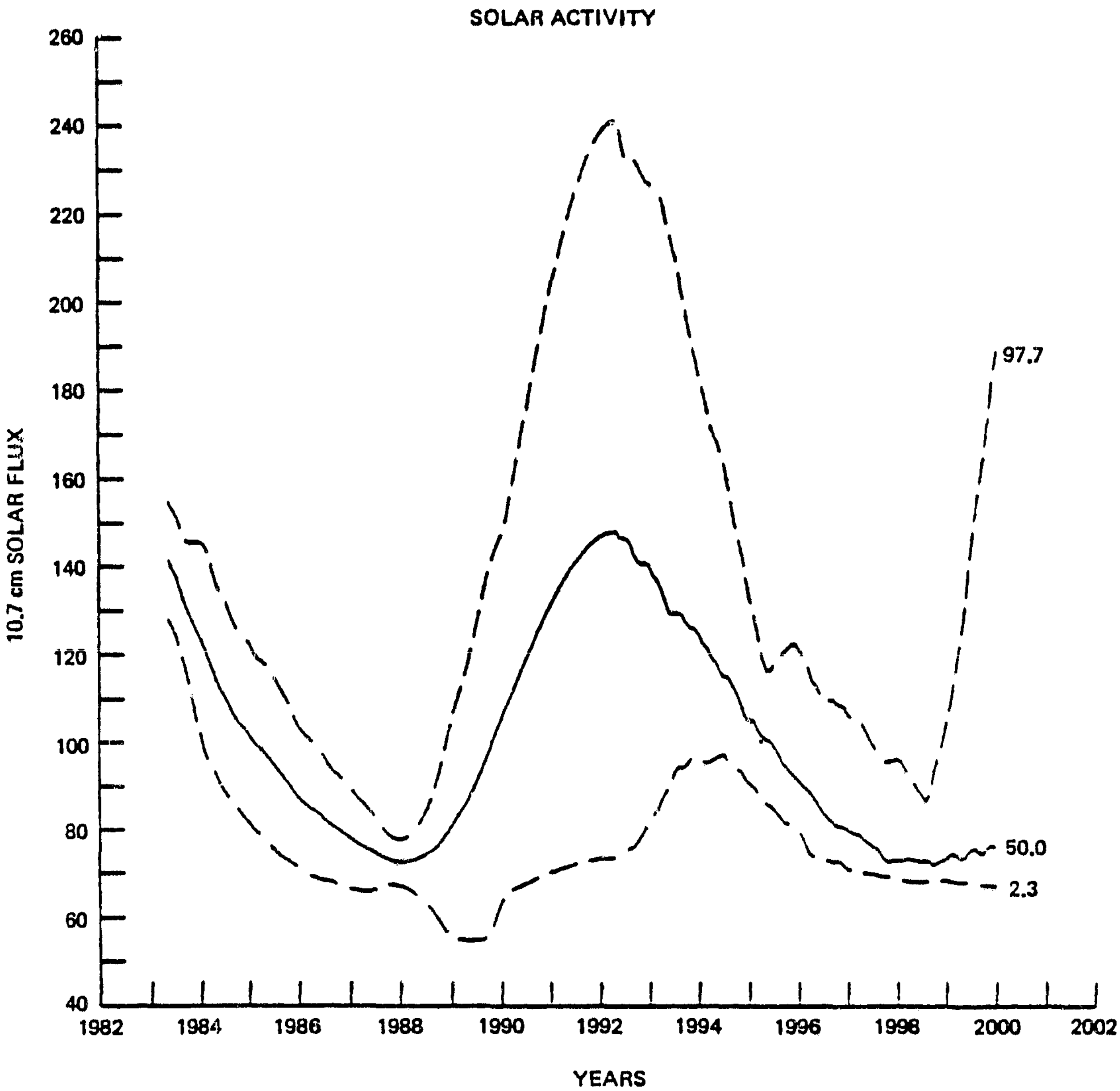

Figure 3-5. Predicted 13-Month Smoothed 10.7-Cm Solar Flux (97.7, 50, and 2.3 Percentile Values are Shown)

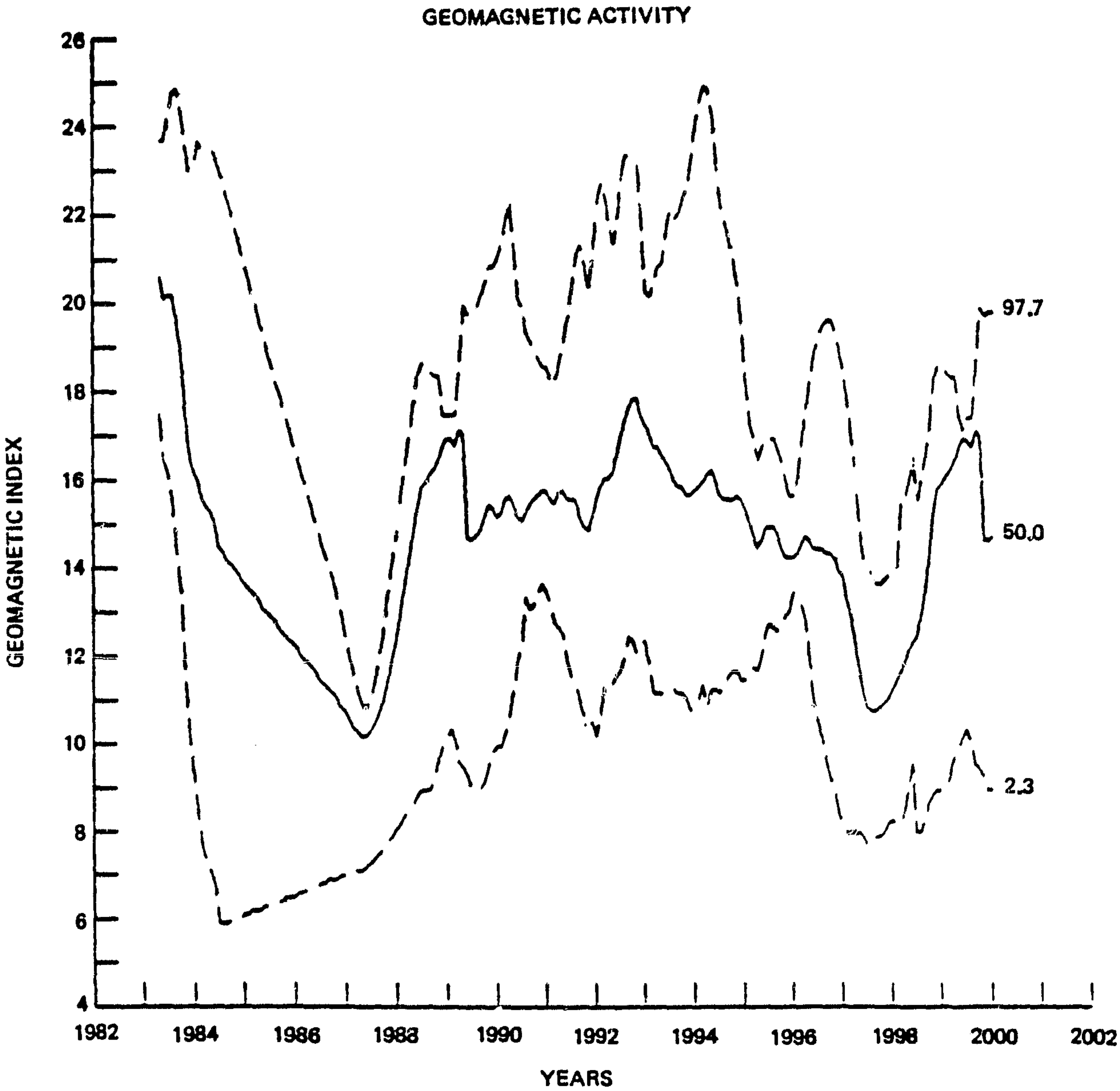

Figure 3-6. Predicted 13-Month Smoothed Geomagnetic Index (97.7, 50, and 2.3 Percentile Values are Shown)

Another source of atmospheric heating is geomagnetic activity. Geomagnetic activity is caused by the interaction between the Earth's atmosphere and charged particles emitted by the Sun. The geomagnetic index is a measure of geomagnetic activity. Predictions for the geomagnetic index, also smoothed over 13 months, are shown in figure 3-6.

To account for these large variations, a statistical approach was taken that uses a static model of the exosphere called the Jacchia Atmosphere. This model uses the following inputs: altitude, 10.7-cm solar flux, geomagnetic index, time of year, time of day, longitude, and latitude. The last three inputs are averaged out when, as is the case for the curent study, a duration of at least two weeks is considered. Four values of the solar flux and geomagnetic index were selected for June of 1991 from the Marshall Space flight Center (MSFC) predictions. These assumed solar flux and geomagnetic values corresponded to +2 sigma, and -2 sigma values for the time period, for the short-time maximum, which used extreme values suggested for Space Shuttle studies. These four atmospheric models are shown in figure 3-7.

The NASA neutral model is a high-solar-activity model, with a value of 230 for the mean 10.7-cm solar flux, and a geomagnetic index of 20.3. The short-time maximum model uses a 10.7-cm solar flux of 250 and a geomagnetic index of 40. These conditions occur only for a few days during an extremely large magnetic storm. The minimum model uses figures of 73.3 for the 10.7-cm solar flux, and 10.9 for the geomagnetic index.

The atmosphere densities shown in Figure 3-7 are used as the baseline in this study and are assumed to remain constant over the station lifetime with no accounting made for the 11-year solar cycle.

Atmospheric density assumptions were used to calculate various propulsion and momentum management requirements. These assumptions are listed in Table 3-7.

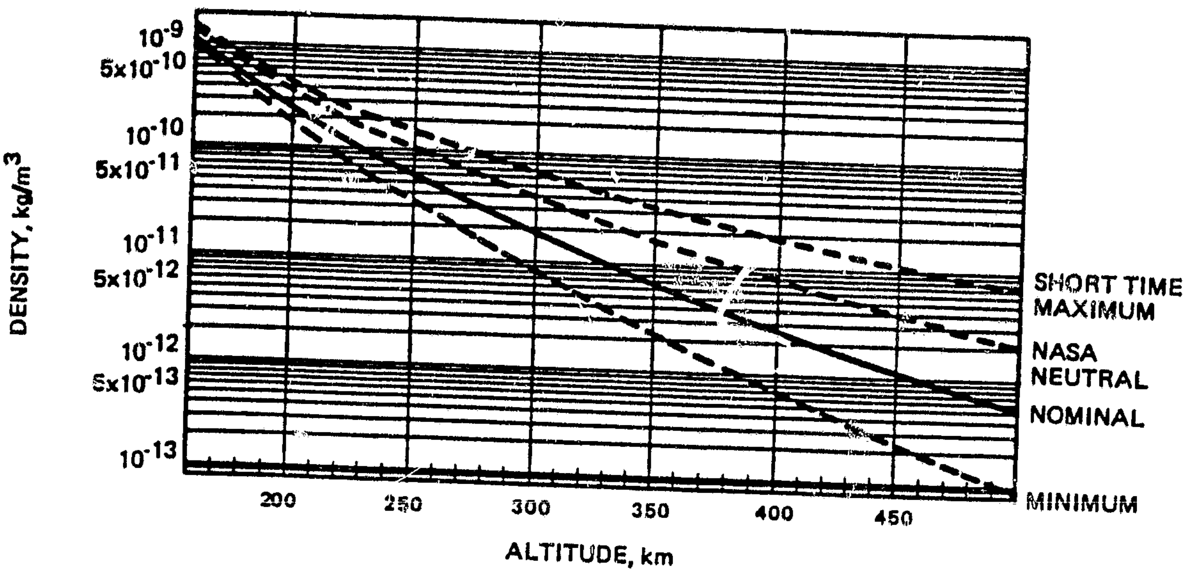

Figure 3-7. Atmosphere Density Models

Table 3-7. Atmospheric Density Selection

Requirement	Period of Interest	Model
Total Mission Impulse	10 years	Nominal
90 Day Resupply	90 days	Neutral
Momentum Sizing	1 Orbit	Neutral
Thruster Torque or Mag. Torque sizing	1 Orbit	Neutral
Orbit Decay without Propulsion	90 days	Neutral
Control Authority	1 day	Short-time Maximum

3.2.2 Aerodynamic Drag and Torques

Aerodynamic drag is determined from the following equation:

$$\text{Drag} = \frac{1}{2}(\rho C_D A V^2)$$

Where:

C_D = drag coefficient

A = effective cross sectional area

ρ = atmospheric density

V = velocity

At Space Station operating altitudes, the mean-free path between the molecules making up the highly rarefied atmosphere is large compared to the vehicle dimensions. Drag is caused by the station striking these particles as it moves through this atmosphere at orbital velocities, causing the particles to bounce off the structure or to be momentarily absorbed by the surface and then re-emitted. Since the station surface roughness is large compared to the molecule diameter, the rebound or re-emission direction of these molecules has little relationship to the original direction. There are two significant manifestations of this type of flow that pertain to Space Station propulsion: (1) the drag coefficient, C_D, based on analyses of satellite orbit decay histories, has a value of 2.2; and (2) the only force experienced by the station while traveling through free-molecule flow is opposite the direction of travel; i.e., there is no lift force regardless of shape or orientation of the surfaces.[1,2]

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[1] A. W. Wilhite, J. P. Arrington, and R. S. McCandless, Performance Aerodynamics of Aero-Assisted Orbital Transfer Vehicles," AIAA-84-0406, January 9, 1984.

[2] D. G. Andrews, R. T. Savage, and S. W. Paris, "Technology Identification for Aeroconfigured Orbital Transfer Vehicles," Volume II, Technical Results, AFWAL TR-83-3090, October 1983.

Aerodynamic torque is caused by a condition where the effective drag force acts through a point, known as the center-of-pressure, which is usually offset from the center-of-mass. Since the station momentum acts through the center-of-mass, this gives rise to the term CP-CM offset. Thus, the magnitude of the aerodynamic torque is given by the drag force multiplied by the CP-CM moment arm. The following sections discuss aerodynamic drag and torque in more detail.

3.2.2.1 Aerodynamic Drag

Table 3-8 shows the aerodynamic drag experienced by the baseline stations with and without the drag contribution from a docked Orbiter at the altitudes studied. The drag data shown are average values based on the average frontal area of the station. Since the solar arrays are Sun-pointing, their area, projected along the velocity vector, changes cyclically. Figure 3-8 illustrates these effects. At point A, the arrays are "flat" to the wind and the array area is reduced by cosine of the beta angle only. At point B, the arrays have remained essentially fixed in inertial space while the body has rotated between the arrays. Effective array area at point B has been reduced to zero. Figure 3-9 illustrates the array angle of attack history from which an average 47.5 deg angle of attack was determined.

The NASA neutral atmosphere was used to calculate the drag data for figure 3-10. Table 3-9 provides multiplication factors for converting the drag data from a neutral atmosphere to minimum, nominal, or short-time maximum atmospheres. The data shown in figure 3-10 for the 4- to 6- and 8- to 12man stations are for the construction variant. The scientific version has slightly lower drag. Table 3-8 provides a numerical listing of the same data as figure 3-10 except that the drag for the 4- to 6- and 8- to 12-man science versions are also included.

3.2.2.2 Aerodynamic Torques

The torques caused by aerodynamic drag and CP-CM offsets vary with altitude, angle of progression in the orbit, atmospheric density, and

Table 3-8. Average Aerodynamic Drag Versus Altitude for Earth-Oriented, 28.5 deg Inclination Stations

NUMBER OF MEN		PURPOSE	ORBITER DOCKED ?	DRAG, lb_f			
				400 km	450 km	500 km	525 km
2 – 4	••	SCIENCE	NO	0.087	0.042	0.022	0.016
2 – 4	••	SCIENCE	YES	0.119	0.058	0.030	0.022
4 – 6	•••	SCIENCE	NO	0.162	0.079	0.041	0.029
4 – 6	•••	SCIENCE	YES	0.194	0.095	0.049	0.035
4 – 6	•••	CONSTRUCTION	NO	0.175	0.085	0.044	0.032
4 – 6	•••	CONSTRUCTION	YES	0.208	0.101	0.052	0.037
8 – 12	+	SCIENCE	NO	0.318	0.155	0.079	0.057
8 – 12	+	SCIENCE	YES	0.351	0.171	0.088	0.063
8 – 12	+	CONSTRUCTION	NO	0.331	0.161	0.083	0.060
8 – 12	+	CONSTRUCTION	YES	0.363	0.177	0.091	0.065

• AVERAGE FOR ENTIRE ORBIT USING A NASA NEUTRAL ATMOSPHERE

•• 110 kW, 1046 m^2 SOLAR ARRAY

••• 210 kW, 2000 m^2 SOLAR ARRAY

+ 420 kW, 4000 m^2 SOLAR ARRAY

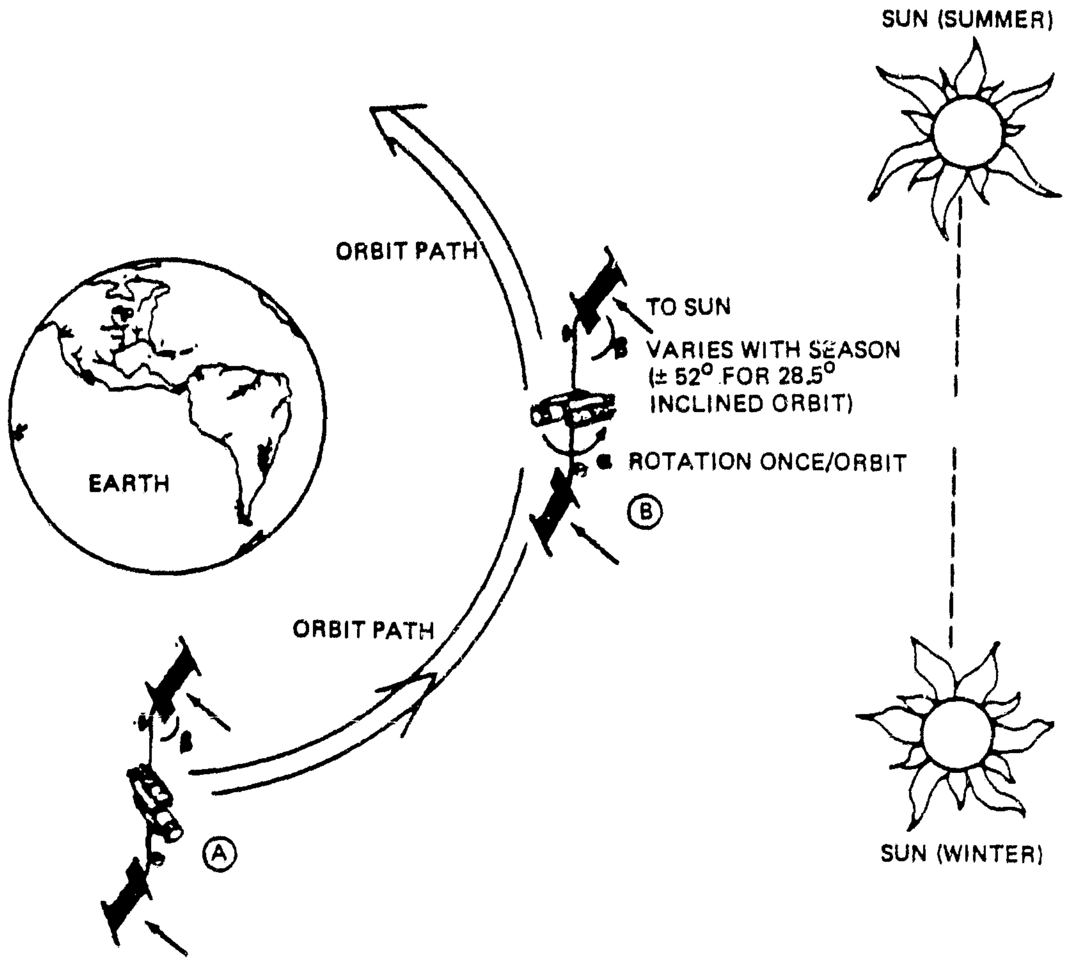


Figure 3-8. Earth-Oriented Solar Array Angle Variation

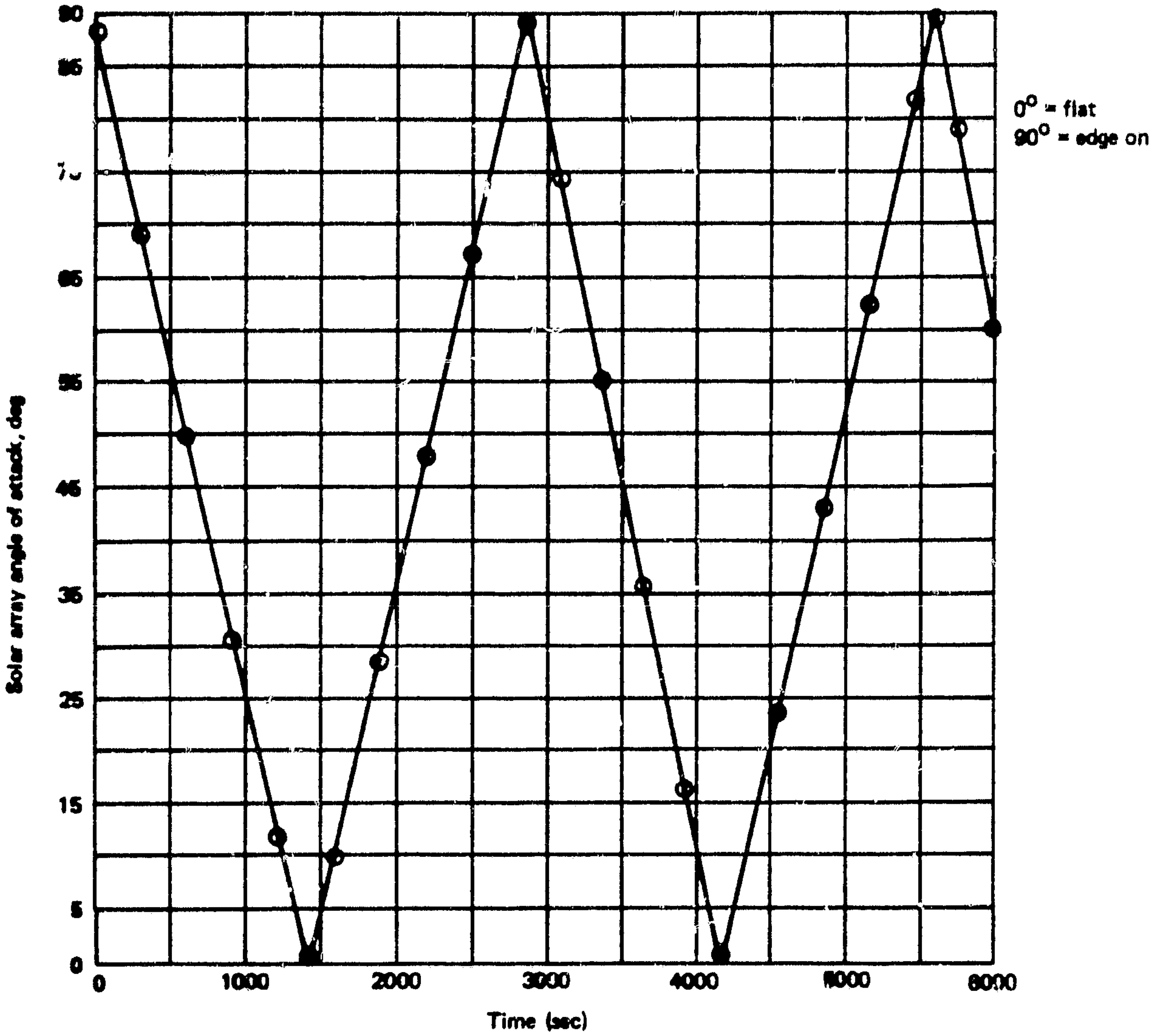


Figure 3-9. Solar Array Angle of Attack

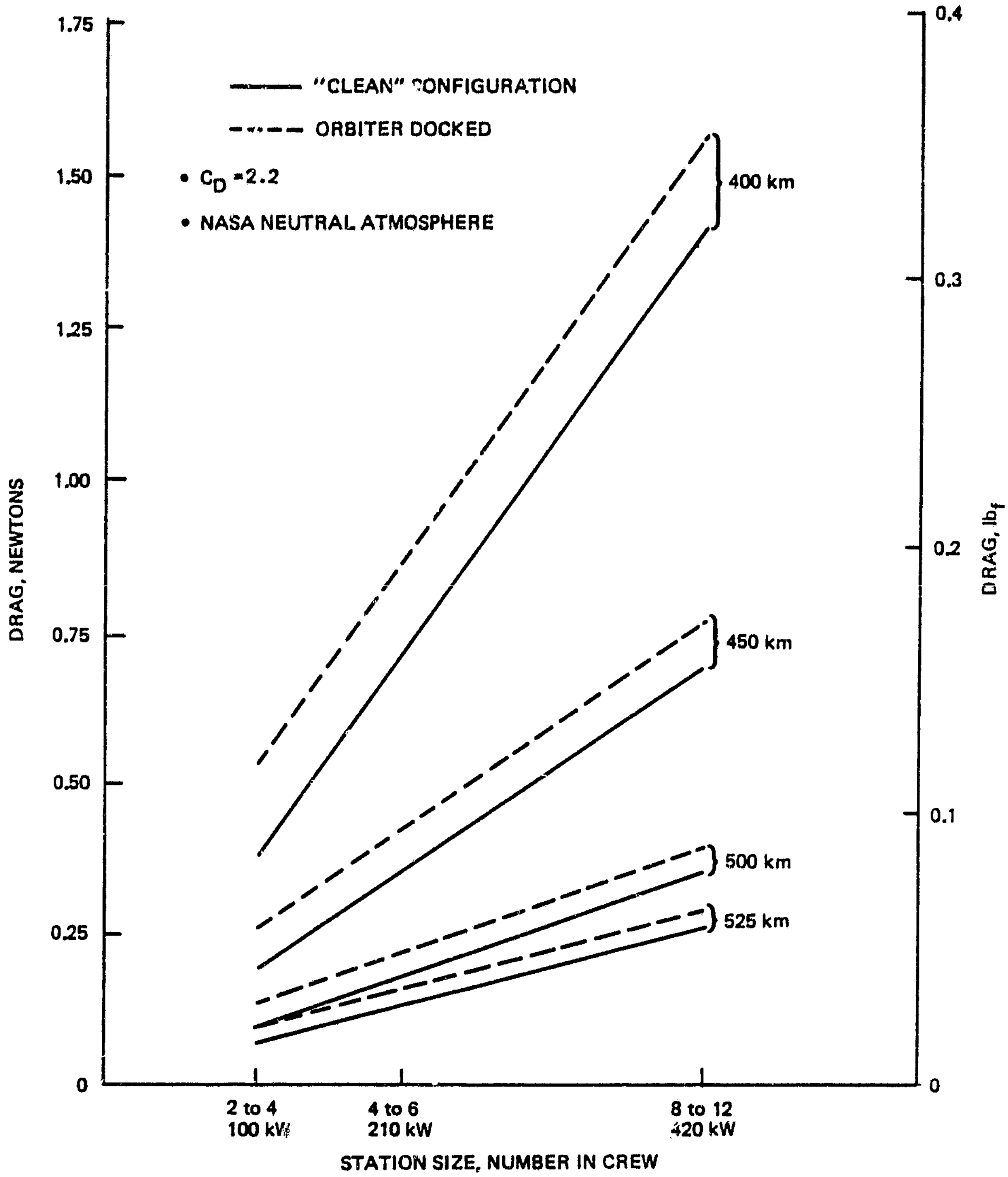


Figure 3-10. Aerodynamic Drag as Affected by Altitude

Table 3-9. Drag Conversion Factors to Change From NASA-Neutral Atmosphere

Altitude (km)	Multiplier to convert neutral atmosphere to:		
	Minimum	Nominal	Short-time max
400	0.0875	0.35	2.5
450	0.0567	0.3077	2.82
500	0.04	0.25	3.5
525	0.035	0.221	3.63

season. The magnitude of the torque is greatly increased for vehicles with a docked Orbiter, depending on the docking location. Drag increases by 20% to 40% for small stations and around 10% for larger stations when the Orbiter is docked. To summarize these variations, three sets of torque values were calculated (see table 3-10). A maximum torque value for a beta angle of zero, using the NASA neutral atmosphere, will size the thruster, magnetic torque rod, or CMG torque requirement. To size the 90-day resupply and the on-orbit maximum momentum requirements, a short-time average torque value was generated. In this set of torque values, beta was again worst case at 0 deg, and the neutral atmosphere was employed. The drag value used in the torque calculations was an average value determined in the manner described in section 3.2.2.1.

The third set of torques was used to calculate total impulse requirements for the 10-year mission. This set used the nominal atmosphere, an average solar array angle of attack, and an average beta angle of 26 deg. Beta angle will vary from zero to 52°. The maximum angle, 52°, results from the sum of the orbit inclination (28.5°) and the Earth's tilt (23.5°). The 26° Beta angle used here is simply the average between 0 and 52°. The beta angle was reduced to account for long-term seasonal and nodal regression effects. The aerodynamic force on the station was estimated for each of the above assumptions. The short-time average values of force are shown in table 3-8 as a function of altitude for each Earth-oriented vehicle.

Table 3-10 lists the values for each set of torques in the Y (pitch) and Z (yaw) body axis. The pitch axis torques are generally larger than yaw axis torques because of the greater moment arm along the Z axis for most designs. A summary of the effects of area multiplied by moment arm on aerodynamic torque is shown in figures 3-11 and 3-12 for pitch and yaw, respectively.

The thrust levels required to counter torques or desaturate momentum mangement devices depend on the moment arms and thrusting duration. The CP-CM moment arms range from 0.1m to 4m; therefore, it is apparent that if orbit maintenance thrusters are used to counter torques directly, thrust levels as low as 0.1 lb_f or lower with a similar moment arm would be

Table 3-10. Aerodynamic Torque for Baseline Stations at 525 km

CREW SIZE	SOLAR ARRAY AREA (m^2)	PURPOSE	ORBITER DOCKED?	TORQUE (N-M): MAXIMUM ($\beta = 0^\circ$, NASA NEUTRAL)		TORQUE (N-M): SHORT-TIME AVG ($\beta = 0^\circ$, AVG AREA, NASA NEUTRAL)		TORQUE (N-M): LONG-TERM AVG ($\beta = 26^\circ$ AVG AREA, NASA NOMINAL)	
				Y	Z	Y	Z	Y	Z
2 – 4	1046	SCIENCE	NO	.079	0	.065	0	.012	0
2 – 4	1046	SCIENCE	YES	.062	.489	.039	.370	.009	.082
4 – 6	2000	SCIENCE	NO	.019	0	.013	0	.003	0
4 – 6	2000	SCIENCE	YES	.838	0	.611	0	.130	0
4 – 6	2000	CONSTRUCTION	NO	.257	.034	.182	.023	.040	.006
4 – 6	2000	CONSTRUCTION	YES	.729	.077	.541	.067	.120	.013
8 – 12	4000	SCIENCE	NO	.613	0	.421	0	.093	0
8 – 12	4000	SCIENCE	YES	1.460	0	1.030	0	.228	0
8 – 12	4000	CONSTRUCTION	NO	1.530	.004	1.084	.003	.236	.001
8 – 12	4000	CONSTRUCTION	YES	2.700	.114	1.720	.082	.379	.018

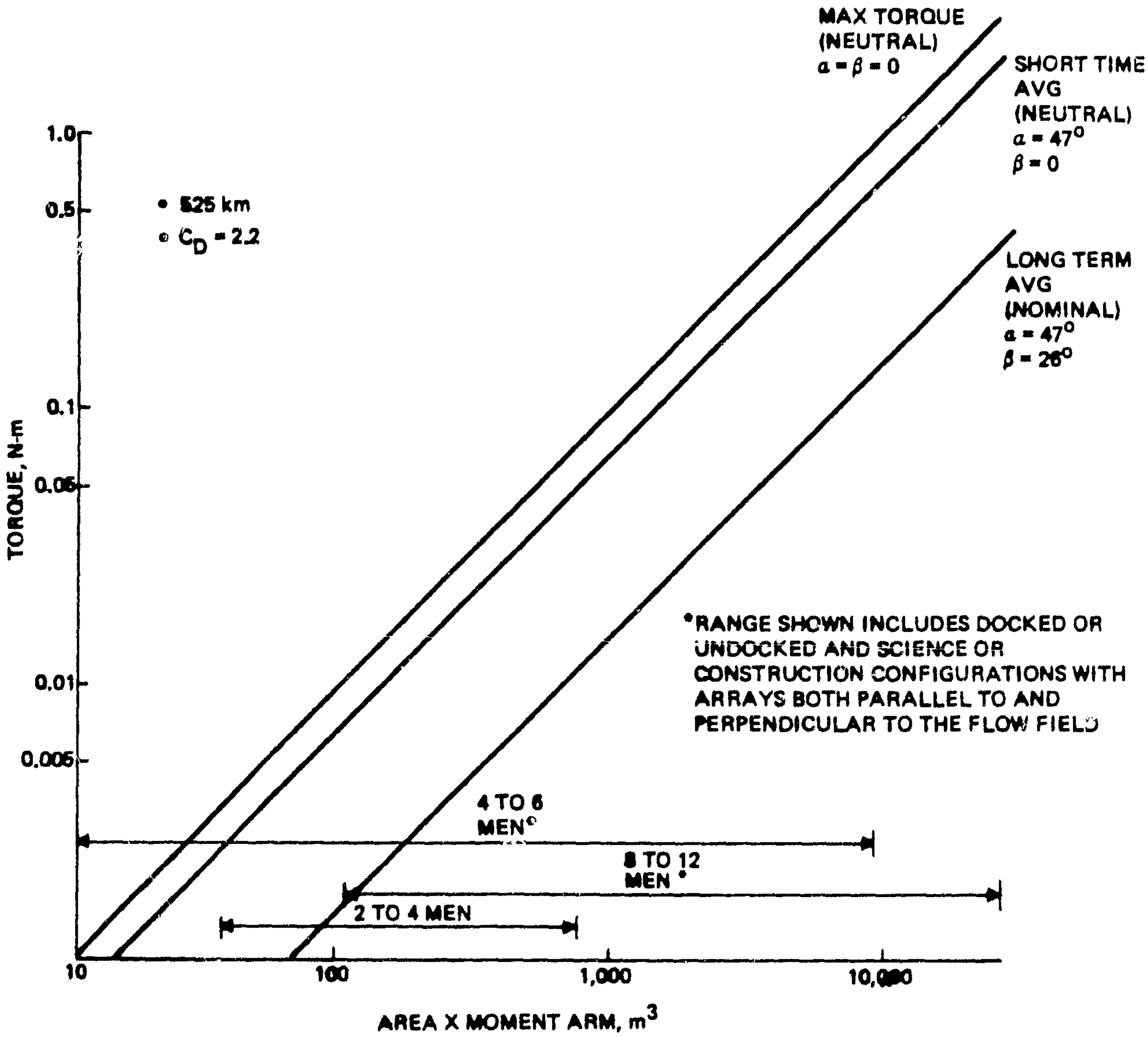


Figure 3-11. Pitch (Y-Body) Axis Aero Torque

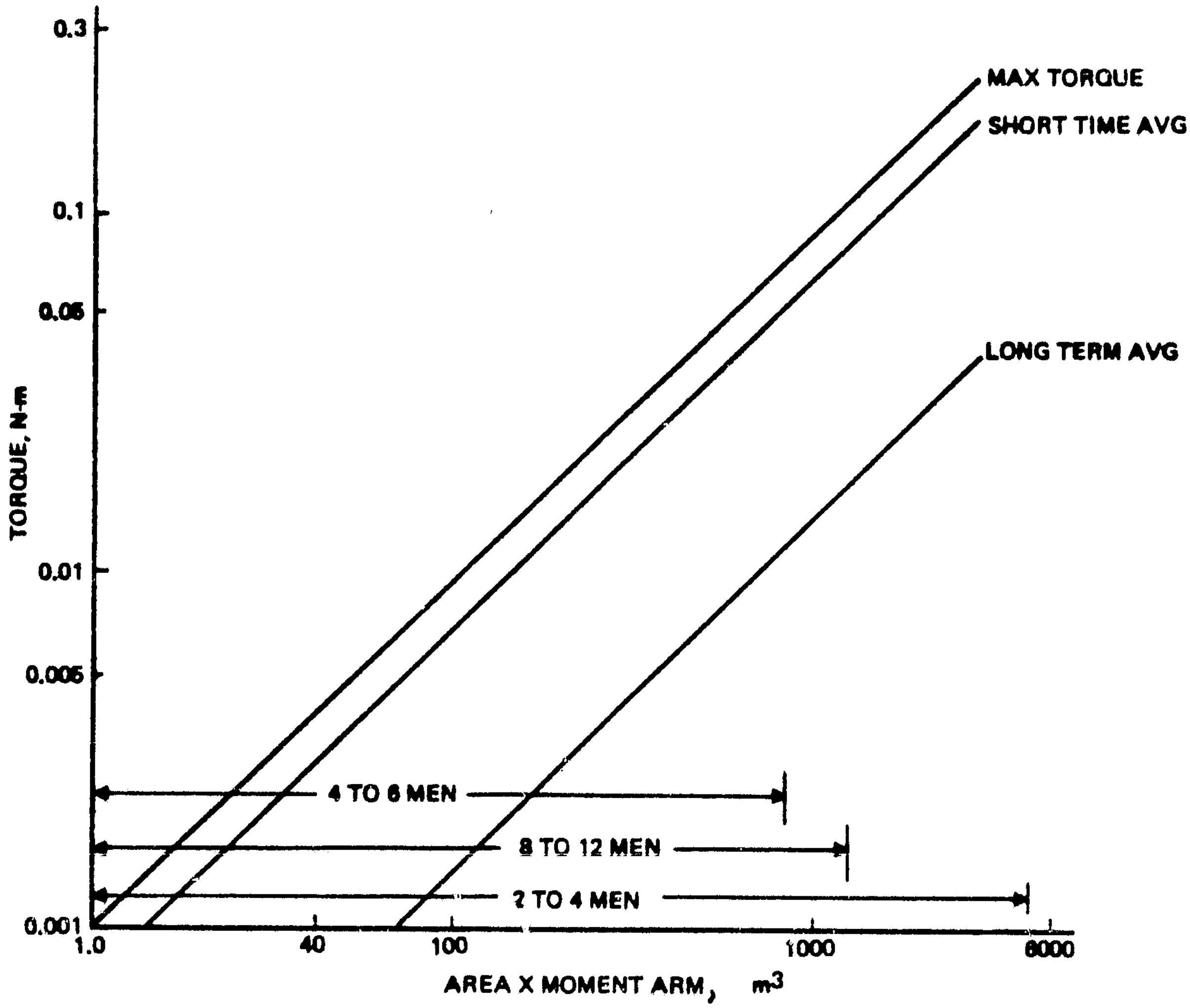


Figure 3-12. Yaw (Z Body) Axis Aero Torque

required. However, if these torques are absorbed by a momentum management device, the thrust level used for simultaneous orbit maintenance and momentum device desaturation can be any level, as long as the effective moment arm and the firing duration provide the necessary desaturation. For instance, if 2000 N-m-s of desaturation is required, the thrust level is 50N and the moment arm is 10m, it will be necessary to thrust for 4 sec on that moment arm. This, of course, imposes no limitation on orbit maintenance thrust duration but merely thrust duration over which the aggregate thrust vector is offset from the CM by 10m.

3.2.3 Orbit Decay

Whenever aerodynamic drag is not counteracted with a propulsive force, the Space Station will lose altitude; i.e., the orbit will decay. The magnitude of the aerodynamic drag force is a function of atmospheric density, orbital velocity, and the projected station area in the direction of travel, as discussed in sections 3.2.1 and 3.2.2, respectively. Orbit decay rate and orbit lifetime are functions of "ballistic coefficient," expressed as M/C_DA, where C_D is the drag coefficient. The A/M ratio is used in this study as a standard of comparison because C_D assumes a constant value of 2.2 (as discussed in section 3.2.2) at Space Station operating altitudes. Table 3-3(b) shows the A/M ratios for the baseline station configurations examined in this study.

Orbital life for an Earth satellite ends when the drag forces increase to the point that the trajectory is no longer essentially elliptical and entry into the atmosphere is imminent. Space Station orbit lifetimes were calculated using the Long-Term Earth Satellite Orbit Prediction Program (LTESOP). The results are presented in figures 3-13 and 3-14, which show the effect of A/M ratio on orbit lifetime for stations at 400, 450, 500, and 525 km for the nominal and neutral atmospheres, respectively. LTESOP uses a Jacchia model of the Earth's atmosphere, which includes such effects as diurnal and semiannual variations in density in addition to the seasonal and solar cycle effects. Since solar activity is the major determinant of atmospheric density at altitudes greater than 100 km, studies were done for two different but constant levels of solar activity. The nominal atmo-

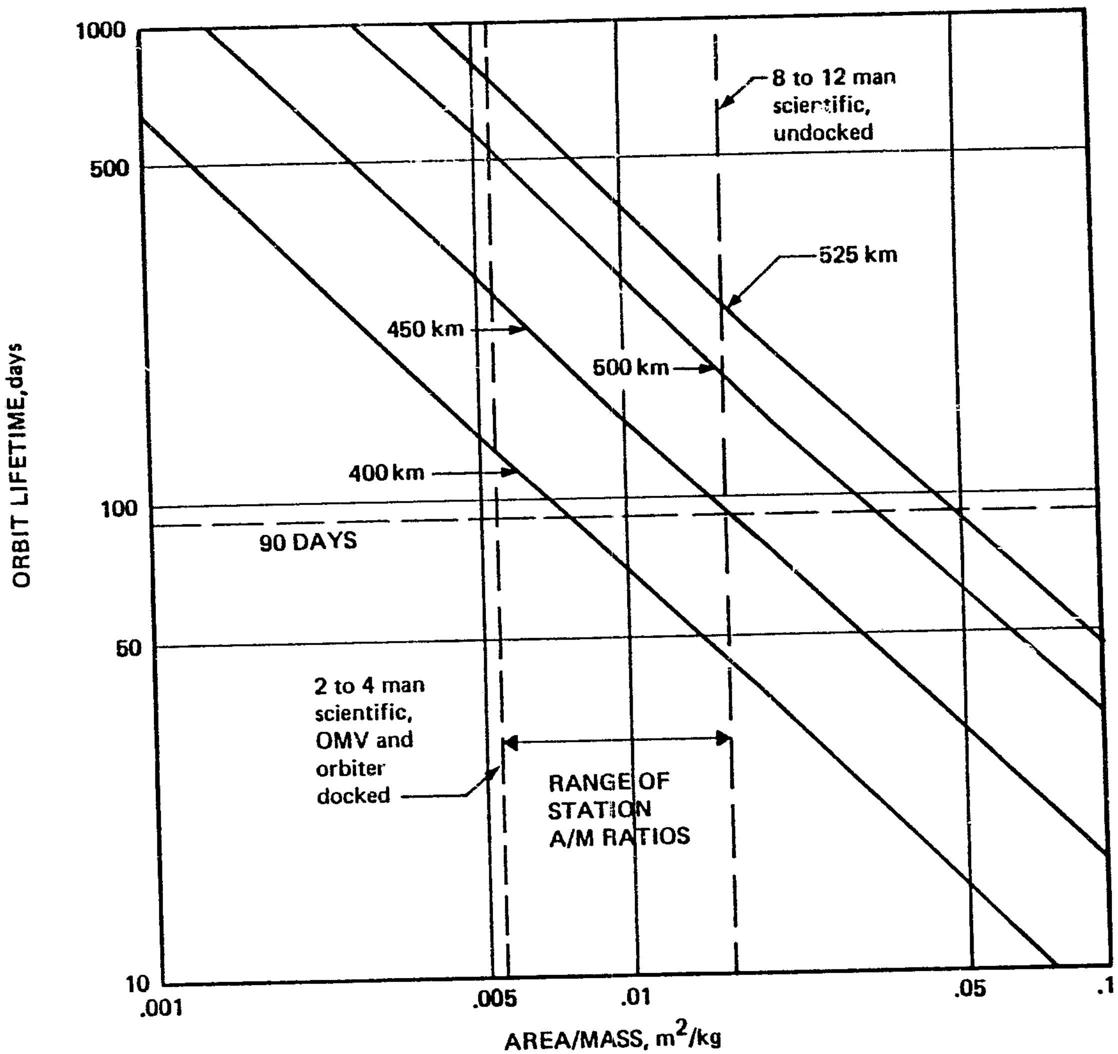


Figure 3-13. Space Station Orbit Lifetime, Neutral Atmosphere, F10.7 = 230., AP = 20.3

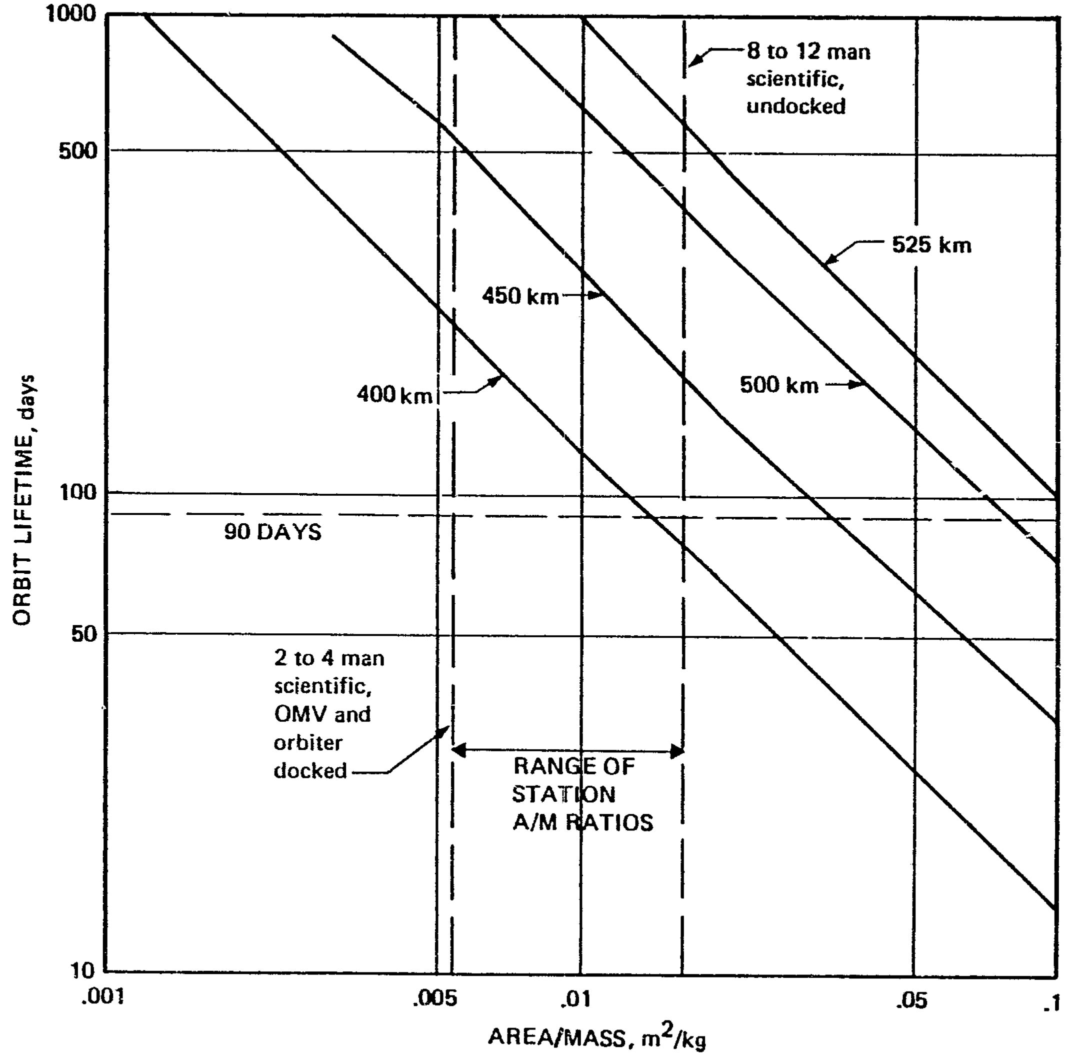


Figure 3-14. Space Station Orbit Lifetime, Nominal Atmosphere, F10.7 = 158.7, AP = 12.

sphere had a moderate solar activity level with a solar flux of (F10.7) of 158.7 and a geomagnetic index (a_p) of 12. The neutral atmosphere had a high level of solar activity with F10.7 = 230. and a_p = 20.3.

All cases were run with a Space Station drag coefficient of 2.2, an orbit inclination of 28.5 deg, and an initial date of January 1. The geo-potential model used tesseral and zonal harmonics through the 4th order. The gravitational perturbations had an effect on the orbit lifetime, as did the starting date (due to the semiannual variation in the atmosphere), although gravity perturbations do not cause an average decrease in orbit altitude as does aerodynamic drag. Since the variation in lifetime for a particular altitude and area-to-mass ratio is generally on the order of 5%, the lifetimes shown in figures 3-13 and 3-14 should be considered to have a $\pm$ 5% uncertainty.

Space Station area-to-weight ratios, orbit altitude, and solar activity levels all have large effects on orbit lifetime. Solar activity levels follow a regular 11-year cycle and can exhibit large, short-term variations within this cycle. Therefore, a fairly high solar activity level has been used for orbit altitude selection in this study.

Orbit decay was based on 1) a Space Station which does not use thrusting to maintain altitude or rate of altitude descent, and 2) an assumed requirement that the initial orbit altitude does not decay so as to re-enter the earth's atmosphere for 90 days. It was further assumed that this criterion must be met without losing power, which would be caused by array feathering or altering the station configuration by means of solar array jettison. Obviously, the solar arrays cause the bulk of the station drag in view of their large area. Estimates of the orbit lifetime effects resulting from procedures that would eliminate solar array drag can be made from figures 3-13 and 3-14. For example, array feathering decreases the A/M ratio from 0.0098 to 0.0007 for the 2- to 4-man station without docked vehicles. This causes a lifetime increase at 400 km from about 70 to at least 600 days. Jettisoning the arrays would cause a smaller lifetime increase because the change in area from the feathered condition is slight, while the array mass would be lost.

Figure 3-15 illustrates a typical time-varying altitude decay profile for astation operating in a neutral atmosphere initially at 300, 400, and 500 km. It is seen that the 90-day lifetime criterion is not met by the 300 and 400 km cases. Figure 3-16 illustrates similar effects of atmospheric density on orbit decay history for a somewhat different configuration from an initial altitude of 500 km. The roughly sinusoidal variation that is superimposed on the minimum density curve and the "wobble" in the other curves are due to Earth-Sun-Moon gravitational harmonic effects. Figure 3-14 shows that, in a nominal atmosphere, all but the lowest A/M ratio station configurations could survive for at least 90 days at an orbit as low as 400 km if orbit maintenance propulsion was lost. However, the lifetime in a neutral atmosphere is diminished to 45 days for the configuration that has an A/M ratio of 0.0168. In a neutral atmosphere, the lowest altitude at which all stations can survive for 90 days is 450 km, as shown in figure 3-13.

Figure 3-17 demonstrates the effect of both altitude and solar array on station orbit decay. In this case, a 2- to 4-man station was fitted with array sizes corresponding to 110 kW (1046 m^2), 210 kW (2000 m^2), and 420 kW (4000 m^2). The figure shows that for the assumed atmosphere, all stations decay within 80 days for a 400 km initial altitude. At a 525 km initial altitude, the 420 kW array station decays rapidly after 80 days.

3.3 Gravity Gradient Effects

The Space Station's mass properties and orientations are the dominant factors that affect gravity-gradient torques. Depending on the particular orientation, these torques can cause cyclic and secular momenta. Momentum management devices (MMD) can be sized to torque the maximum cyclic momenta. MMD's used to store secular or non-periodic momenta must be desaturated at regular intervals. Magnetic torque rods or a propulsion subsystem can be used to counter the torques, thereby eliminating the momentum buildup and reducing the need for MMD's. The primary means for countering secular torque could be a combination of torque rods and CMG's. The propulsion system would be used as a backup in the event of a primary system failure, or as discussed in section 3.2.2, secular yaw and pitch torques could be

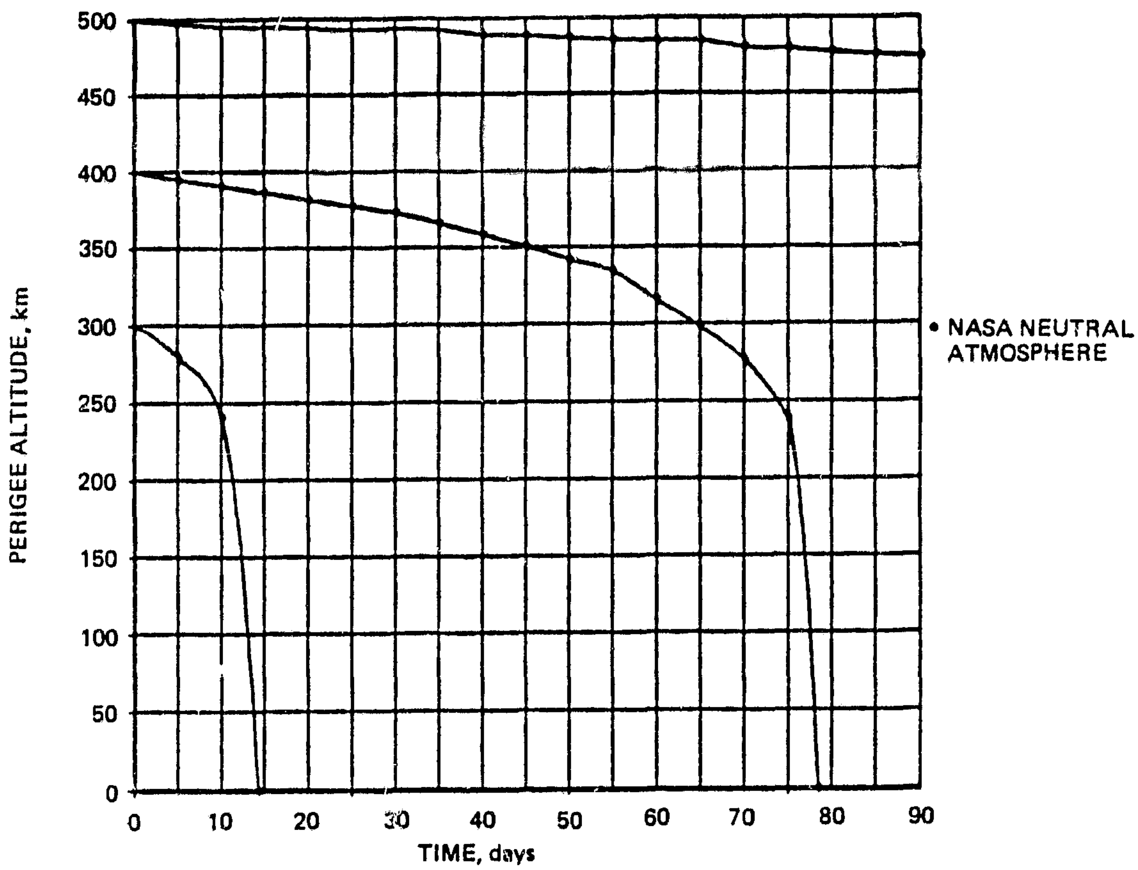


Figure 3-15. 4- to 6-Man Station Orbit Decay

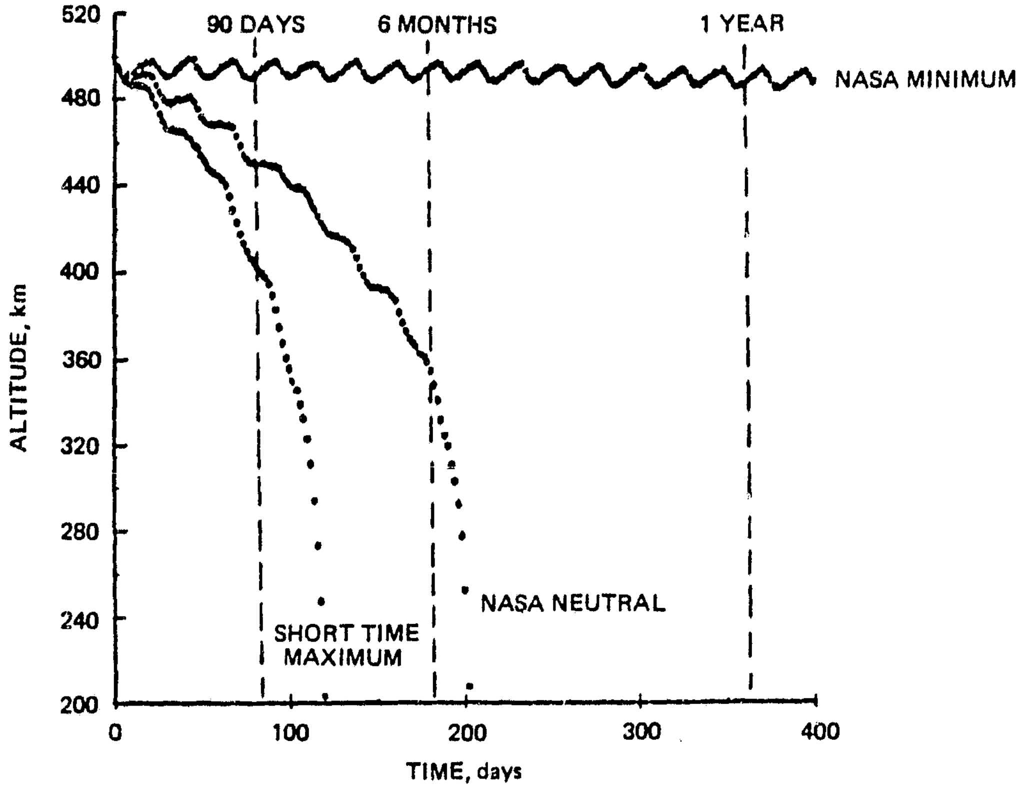


Figure 3-16. Typical Atmospheric Modeling Impacts

(Data Based on "Space Operations Center Technology Identification Support Study, Final Report," Boeing, D180-26495-7, July 1981.)

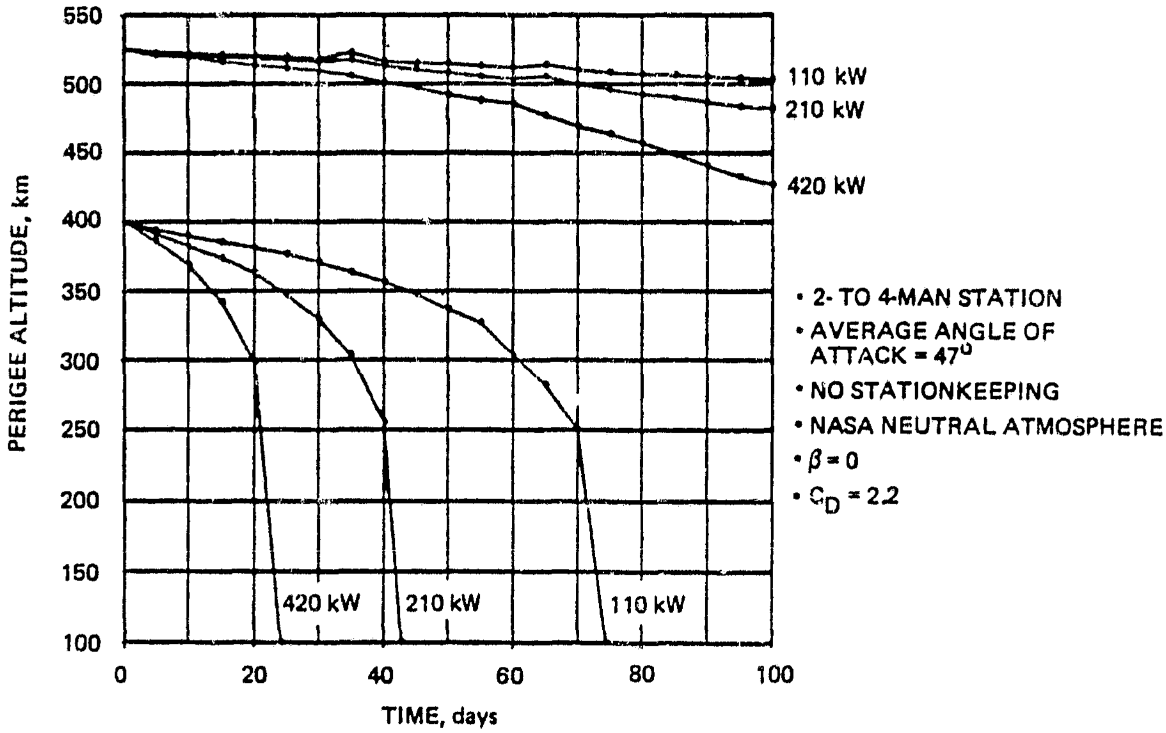


Figure 3-17. Influence of Solar Array Area and Altitude on Orbit Decay Time

countered by offset thrusting of the orbit maintenance thrusters.

3.3.1 Effects of Orientation on Gravity-Gradient Torques

This section emphasizes the effect orientation has on the gravity-gradient torque. The four orientations discussed in this section are outlined below.

(1) Fixed inertial. The orientation of the station is fixed with respect to an inertial reference frame. The Y-body axis is perpendicular to the orbit plane. The delta configuration would have a fixed inertial orienation.

(2) Earth oriented. The Z-body axis is always pointed toward the Earth. The Y-body axis is perpendicular to the orbit plane. The baseline Space Station core uses this mode.

(3) Balanced array. This orientation describes solar array panels that are considered fixed with respect to an inertial reference frame. Seasonal rotation of the panels out of the orbit plane to account for the Sun-tracking angle, beta, is considered negligible. The panels rotate about the Y-body axis of the Earth-oriented reference frame described above. The array pivot point is located at the geometrical center of the panels. In the following discussion and figures (3-18 through 3-20), the solar panel reference frame is denoted by a subscript 'p' to differentiate this reference frame from the station core body frame. The panel reference frame is defined for the flat rectangular solar panels such that the Zp-axis is the axis about which the panels rotate to track the Sun (beta angle) and the Yp-axis is perpendicular to the panel (See figure A-1 in Appendix A). For baseline solar arrays, the products of inertia in the panel reference frame are listed in section 3.2.2, in table 3-11.

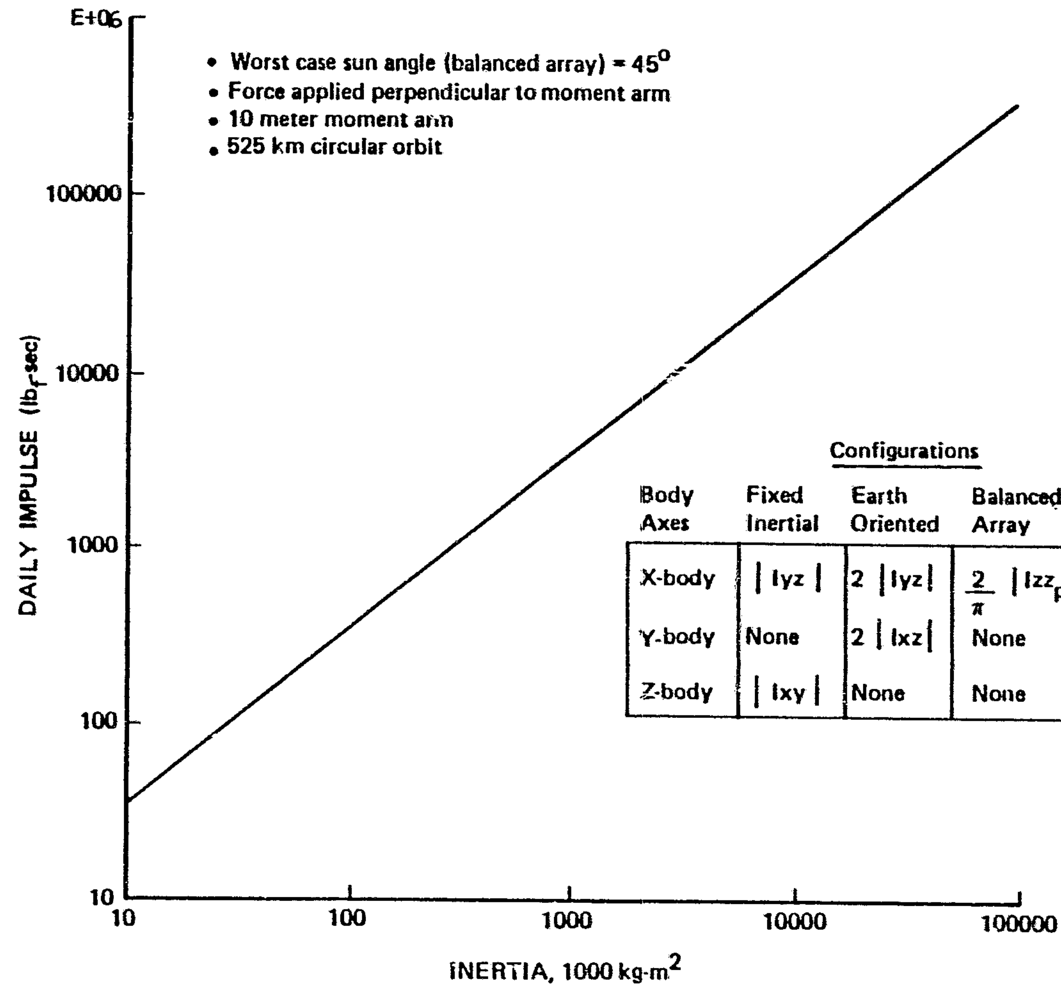
(4) Gravity gradient. In this orientation the station is allowed to assume a stable gravitational altitude. There is no attempt to maintain a fixed attitude.

Figures 3-18 through 3-20 display certain gravity-gradient effects as a function of the station's (or solar panel's) inertia tensor. The cross products of inertia are substituted into the appropriate relation denoted in the box on each figure. This defines a "generalized" inertia term, which is represented on the abscissa axis. In all of the orientations listed, except for the balanced array, the inertia tensor used should be referenced to the station body frame. The balanced array inertia tensor is transformed into the panel reference frame described above. However, the quantities represented on the ordinate axis are all referenced to the station body reference frame.

Figure 3-18 displays the daily impulse needed to counter the gravity gradient torques resulting in secular momenta. This assumes that a propulsion system is used to counter the torques, as opposed to a system using MMD's. The force is assumed to be applied perpendicular to the body axis 10 meters from the body origin.

Note that for the balanced array, only a roll torque must be countered. The section dealing with mass properties (section 3.1) demonstrated that for the baseline planar core, the I_{yz} cross product is small. Hence, only the roll torque produced by the balanced array significantly contributes to the secular momenta buildup. To reduce the roll torque in the balanced array orientation, it is necessary to reduce the Z panel axis principle product of inertia. This can be accomplished by reducing the length of the rectangular panel in the X panel axis (refer to Appendix A). If the aspect ratio is defined as the ratio of the length of the panel along the Z-axis to the length of the panel in the X-axis, then increasing the aspect ratio will reduce the Z panel principle axis inertia term and, consequently, the station roll torque.

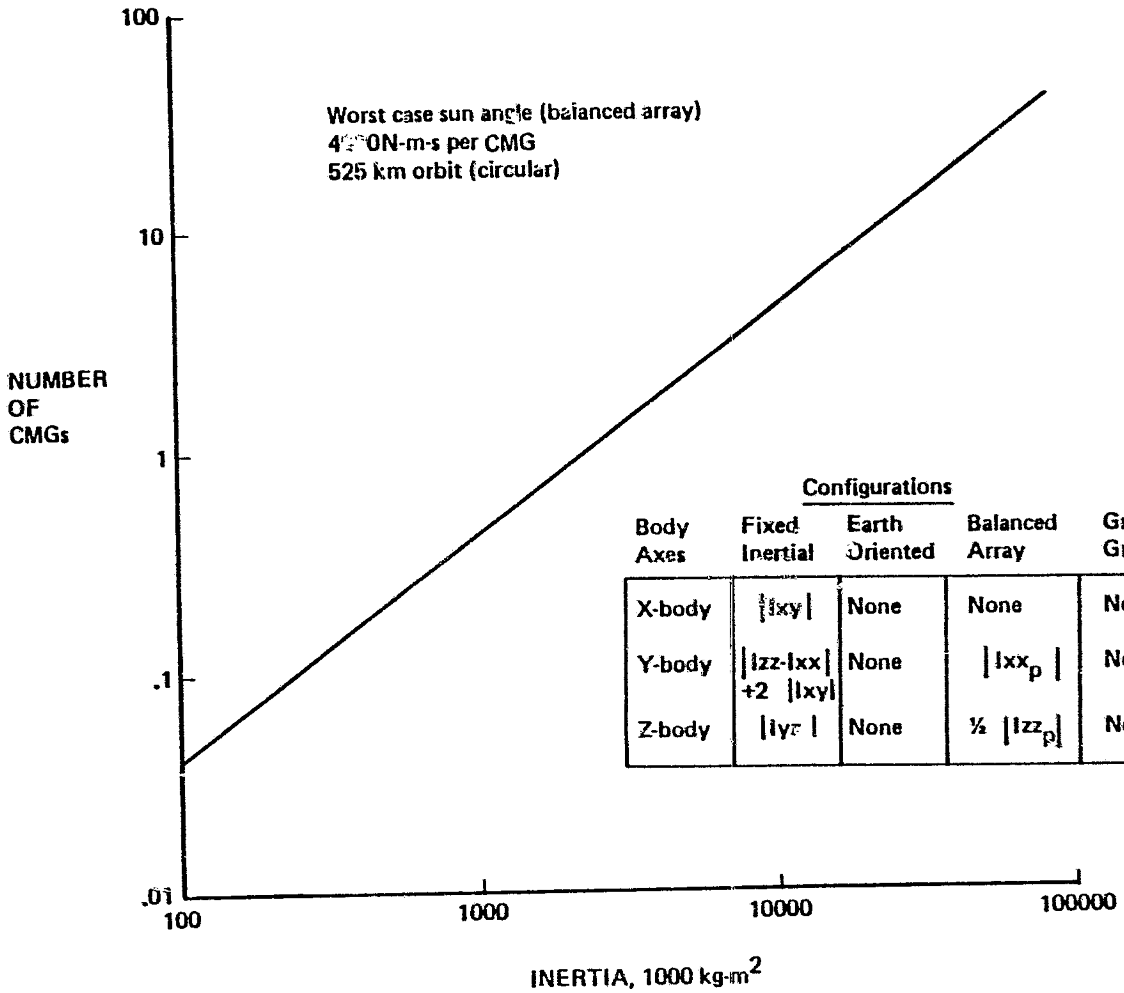
Figure 3-19 provides data enabling the sizing of the momentum management system to absorb the maximum cyclic momenta. The ordinate axis represents the number of control moment gyros (CMG's) needed per axis. Each CMG is assumed to be of the advanced Skylab class, which is capable of absorbing 4000 N-m-s. Referring to the previous discussion on aspect ratio, an increase in the aspect ratio implies a larger length in the Z panel axis,



Configurations

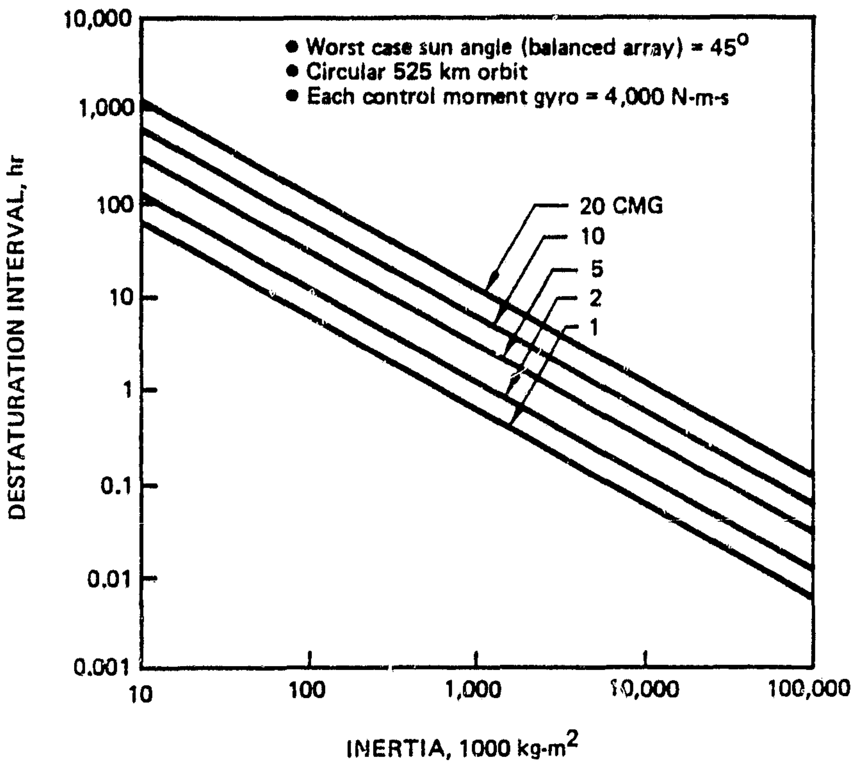
Body Axes	Fixed Inertial	Earth Oriented	Balanced Array	Gravity Gradient
X-body	$\lvert Iyz \rvert$	$2 \lvert Iyz \rvert$	$\frac{2}{\pi} \lvert Izz_p \rvert$	None
Y-body	None	$2 \lvert Ixz \rvert$	None	None
Z-body	$\lvert Ixy \rvert$	None	None	None

Figure 3-18. Daily Impulse per Axis Due to Gravity Gradient Torques Resulting in Secular Momenta



Body Axes	Configurations: Fixed Inertial	Earth Oriented	Balanced Array	Gravity Gradient
X-body	$\lvert Ixy \rvert$	None	None	None
Y-body	$\lvert Izz-Ixx \rvert + 2\,\lvert Ixy \rvert$	None	$\lvert Ixx_p \rvert$	None
Z-body	$\lvert Iyz \rvert$	None	$\frac{1}{2}\,\lvert Izz_p \rvert$	None

Figure 3-19. CMG Requirements per Axis Due to Gravity Gradient Torques Resulting in Cyclic Momenta



Configurations

Body Axes	Fixed Inertial	Earth Oriented	Balanced Array	Gravity Gradient
X-body	$\lvert Iyz \rvert$	$2 \lvert Iyz \rvert$	$\frac{1}{2} \lvert Izz_p \rvert$	None
Y-body	None	$2 \lvert Ixz \rvert$	None	None
Z-body	$\lvert Ixy \rvert$	None	$\frac{1}{2} \lvert Izz_p \rvert$	None

Figure 3-20. Desaturation Interval per Axis Due to Gravity Gradient Torques Resulting in Secular Momenta

which results in a larger X principle product of inertia. Figure 3-19 also shows that this, in turn, leads to larger pitch CMG requirements for the balanced array orientation.

The series of curves in figure 3-20 represents the interval of time it takes to saturate the CMG's. Each curve assumes a particular number of CMG's available per axis.

3.3.2 Baseline Gravity-Gradient Torques and Momenta

The previous section examined the effect of orientations on gravitygradient torques and momenta. This section discusses gravity-gradient effects for the Earth-oriented baseline configuration. Appendix A provides an analysis of the solar array dynamics for the baseline configuration and how the dynamics affect the station gravity-gradient torques and momenta. The following discussion and figures are largely a result of that analysis.

The section dealing with mass properties (section 3.1) shows that for the well-balanced undocked station core, the cross products of inertia are small compared to the solar array cross products. The gravity-gradient torques for the Earth-oriented baseline configuration are directly proportional to the cross products of inertia (see section 3.3.1). Thus, the solar array cross products of inertia are the dominant factors affecting the gravity-gradient torques and momenta.

The baseline configuration was developed before it was realized that the aspect ratio plays a significant role in determining the solar array cross products of inertia. Both the previous section and Appendix A discuss the solar panel aspect ratio and its relation to the station cross products of inertia. It should be noted that an increase in the aspect ratio would yield lower secular momenta and larger cyclic momenta. This is an important design criteria that should be considered.

Table 3-11 lists the baseline solar array properties. The principle moments of inertia refer to the panel reference frame described in the last section. The panel reference frame (PRF) coordinate system is shown in figure A-1 of Appendix A. The moments of inertia of the solar panels in the PRF system are constant and defined in equation 18 of Appendix A. The secular and cycle momenta in the station body axis system (Earth-Oriented) are summarized in Tables A-3 and A-4 of Appendix A.

Table 3-11. Baseline Solar Array Properties

Area(m^2)	1046	2000	4000
Mass (kg)	2613	5000	10000
BOL* Power (kW)	76	150	300
Aspect Ratio	1.01	3.08	1.54
Dimensions (m x m)	29.05x18.00	55.56x18.00	55.56x36.00
IXXp (1000 kg-m^2)	368	2572	5145
IYYp (1000 kg-m^2)	509	2842	7305
IZZp (1000 kg-m^2)	141	270	2160

*Beginning of Life

For the baseline configurations, the cyclic and secular components of the torque are listed in table 3-12. The maximum torque as a function of solar array area is plotted in figure 3-21 and the maximum momenta are plotted in figure 3-22.

Appendix A concludes that the balanced solar array arrangement offers considerable advantages over the cantilevered arrangement. Figure 3-21 shows that the roll torque of the balanced array is about one order of magnitude lower than for the cantilevered array. This yields much lower secular momenta (roll and yaw) for the balanced array.

Table 3-12. Cyclic and Secular Components of Torque

Disturbance		Body Axis	
Torque	Roll (X)	Pitch (Y)	Yaw (Z)
Gravity gradient	Secular	Cyclic	Negligible
Aerodynamic	Negligible	Secular	Cyclic

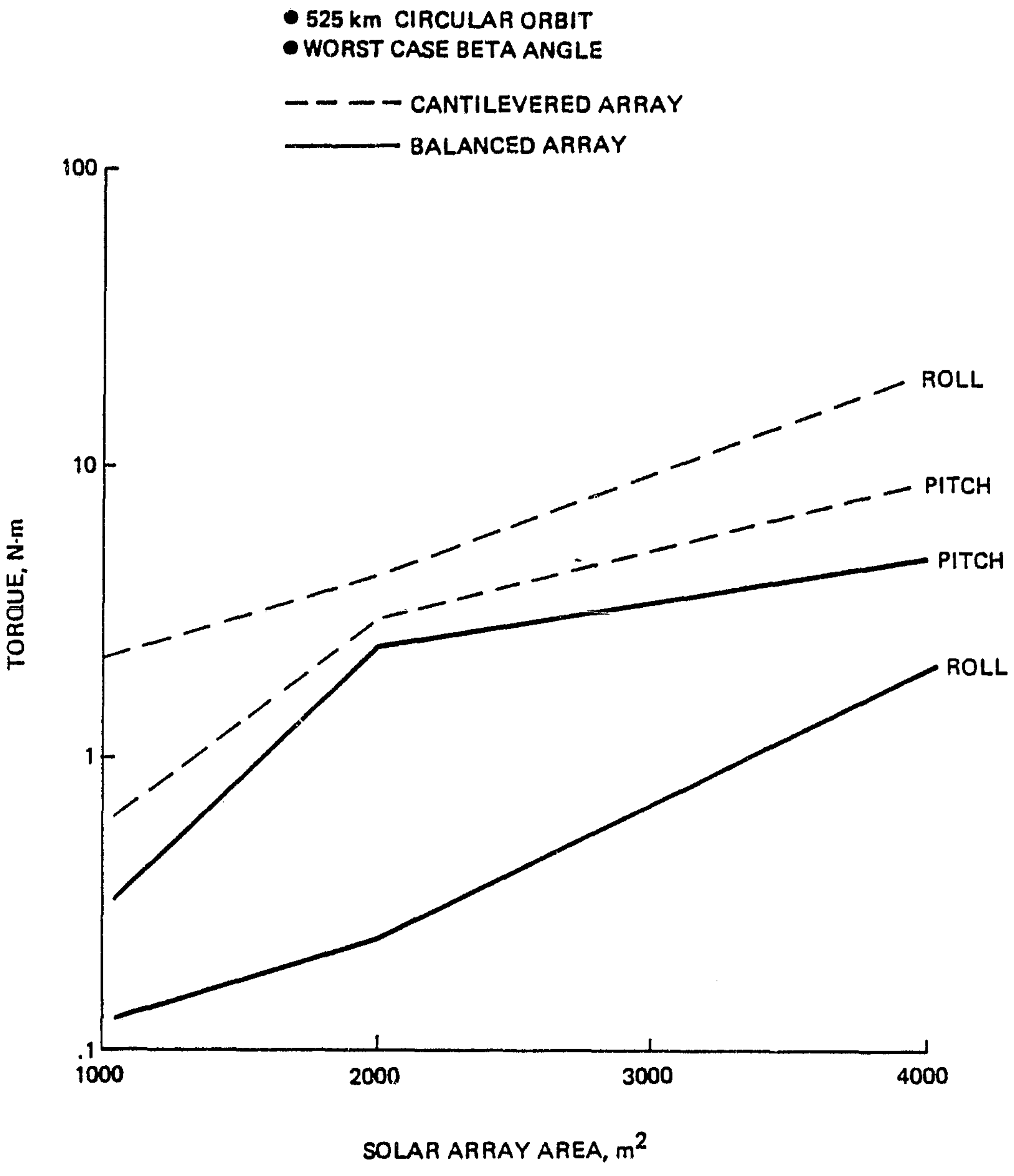


Figure 3-21. Maximum Gravity Gradient Torques by Axis (Baseline Aspect Ratio)

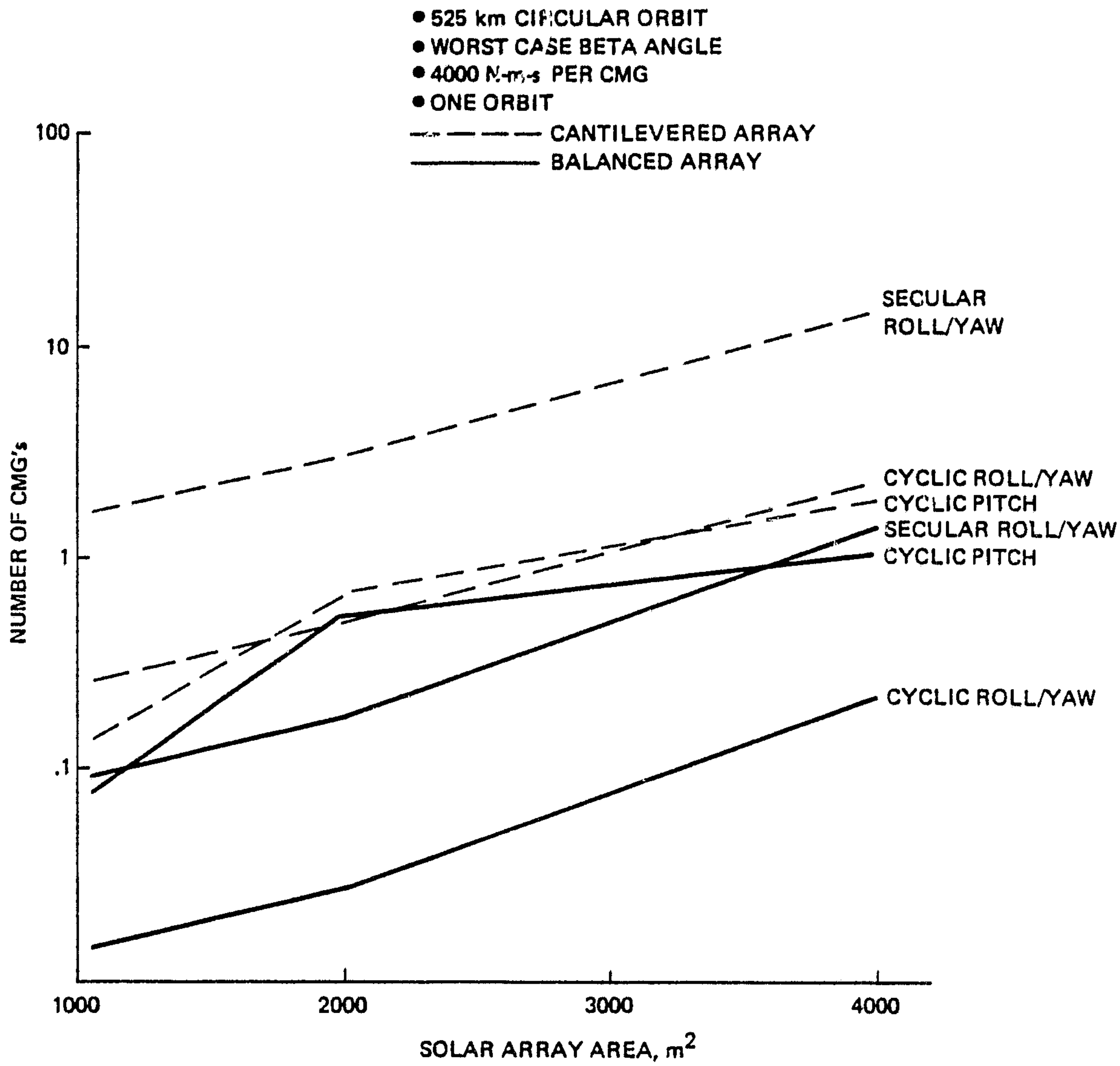


Figure 3-22. Momentum or CMG Requirements Per Axis (Baseline Array Aspect Ratio)

3.4 Configuration/Environmental Effects Summary

The solar arrays are the configuration element that causes the largest force, torque, and momentum effects on the Space Station. Since the solar arrays make up about 95% of the average station cross-sectional area (without the Orbiter), they determine the drag forces. However, since the arrays are symmetrical about the array mast, aerodynamic torques are minimized (as compared to a cantilevered array). Because there is no aerodynamic lift produced in the free- molecule flow environment of the Space Station, there are no forces or torques due to lift. The manner in which the solar array is supported has a significant effect on the torques due to gravity-gradient by as much as an order of magnitude. Gravity-gradient torques are minimized if a balanced, rather than cantilevered, solar array configuration is used. The balanced array configuration can utilize momentum management devices (CMG's) and torque rods can be used for CMG desaturation and torque cancellation.

The drag characteristics of the station are significantly increased when the Orbiter is docked. The torque effects of this drag depend on the location and orientation of the Orbiter docking module. The change in torque could vary from a small decrease to a factor of eight increase for the configurations studied. Therefore, the Orbiter must be docked in a gravity-gradient stable location.

Based on the analyses of aerodynamic effects on the Space Station, it is concluded that (1) static models of density-altitude profiles are suitable for most, if not all, preliminary design studies; (2) the NASA neutral atmosphere should be used for propulsion system capacity and resupply analyses; (3) the short-term maximum atmosphere should be used for control authority studies; and (4) the nominal atmosphere should be used for long-term studies (one year duration or more).

4.0 SPACE STATION SERVICING STRATEGIES

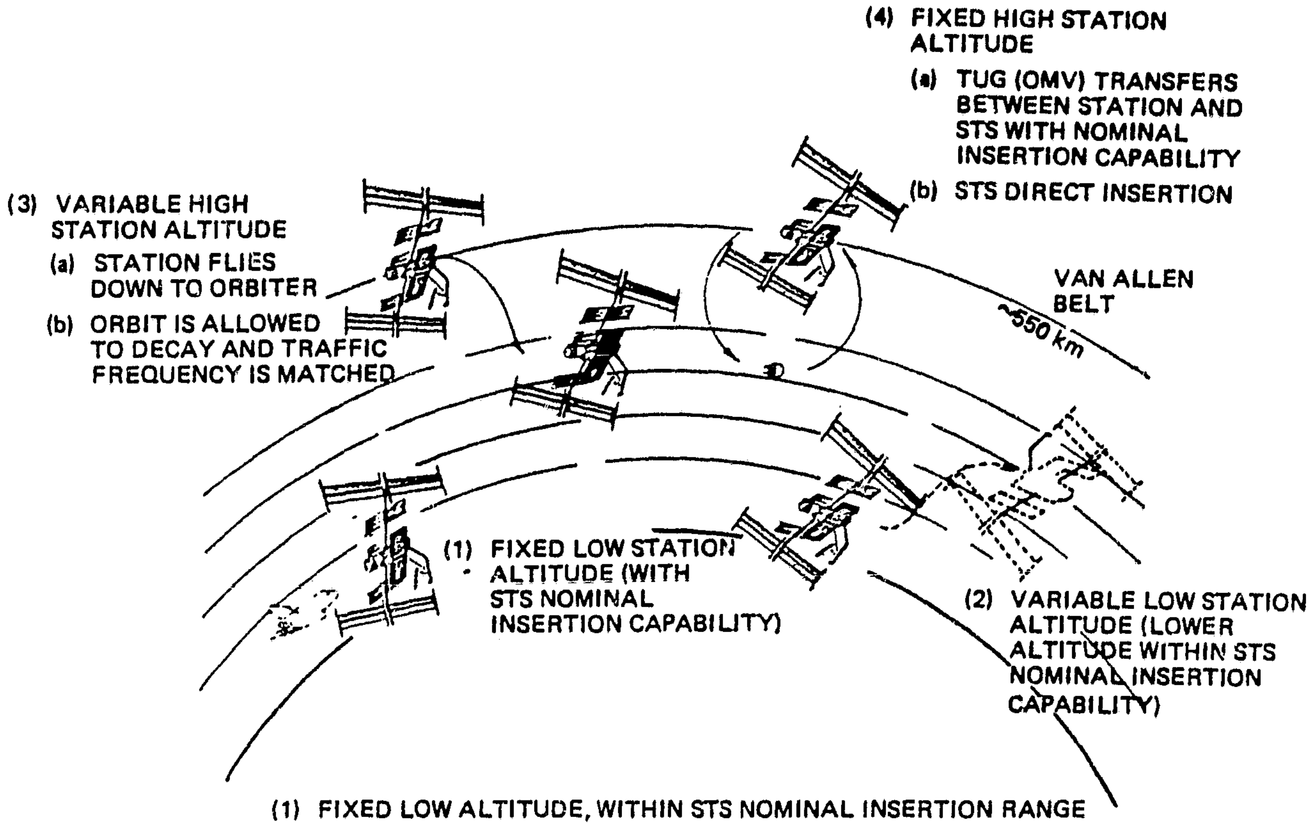
4.1 Space Station Servicing

This section examines servicing strategies for the Space Station and associated free-flyers. The servicing options depend on: Space Station orbiting altitude; propellant requirements for the station, free-flyers, and servicing vehicles; STS and servicing vehicle capabilities; and environmental factors.

Six primary servicing strategies have been identified for a low-inclination (28.5-deg) Space Station and these are shown in figure 4-1. A seventh strategy is also described in this section, which is based on servicing a 90-deg station from a 28.5-deg station by using an OMV.

4.1.1 Servicing Options

The servicing options examined in this study include low, high, and variable orbit altitudes. The following definitions are used for these altitudes in this study. The low altitude is accessible by STS nominal insertion and has an upper limit of 390 km. As shown in Figure 2-15, above that altitude, STS nominal insertion payload capacity decreases rapidly, being near zero at 460 km. The high altitude has a range between 450 and 525 km. The lower limit is based on 90-day lifetime requirement (if the propulsion system fails), as shown in section 3.2.3. The upper limit is restricted by the Van Allen Radiation belts, which have a lower boundary of about 550 km. The variable altitude requires that the Space Station change altitude to rendezvous with the Orbiter. The low variable altitude is accessible by STS nominal insertion and permits the Space Station to vary its orbiting altitude as atmospheric conditions change. When there are periods of high Sunspot activity, which increase atmospheric density, the station must be boosted to a higher orbit. Likewise, when there is low Sunspot activity, the station can be deboosted into a lower orbit. In this



(1) FIXED LOW ALTITUDE, WITHIN STS NOMINAL INSERTION RANGE

(2) VARIABLE LOW ALTITUDE, LOWER ALTITUDE WITHIN STS NOMINAL INSERTION RANGE

(3) VARIABLE HIGH ALTITUDE
   (a) STATION DEBOOSTS TO MEET THE STS
   (b) ORBIT IS ALLOWED TO DECAY TO MEET THE STS RANGE AND TRAFFIC FREQUENCY IS MATCHED

(4) FIXED HIGH ALTITUDE
   (a) TUG (OMV) TRANSFER BETWEEN STATION AND STS, WHICH FLIES TO NOMINAL INSERTION RANGE
   (b) DIRECT INSERTION USED TO HIGH ORBIT

Figure 4-1. Space Stationing Servicing Options manner, it can maintain a constant-density or a constant lifetime altitude if the propulsion system fails.

4.1.1.1 Nominal Insertion to a Fixed Low Altitude ($\leq$ 390 km)

This servicing option, number (1) in figure 4-1, places the Space Station in a fixed low altitude--up to 390 km--which is within the STS nominal insertion range. The 390 km altitude enables the STS to carry maximum payloads to the station without requiring servicing vehicles to transfer payloads from the STS to the station. However, as the section on orbit decay (3.2.3) showed, the relatively high atmospheric density at altitudes below 450 km prevents the 8- to 12-man station from satisfying the requirement for a 90-day safe orbit lifetime in the event the propulsion system is inoperative. The smaller stations have lower altitudes for a 90-day lifetime as shown in Figure 3-13.

4.1.1.2 Variable Low altitude (390 to 450 km)

In this strategy, the Space Station varies its operating altitude based on atmospheric density and maintains a constant-density or constant- lifetime altitude. This option, depicted as number (2) in figure 4-1, was recommended by the earlier Space Operations Center studies by Boeing and Rockwell. This altitude can be at or below 390 km in the extreme low-density conditions, and can utilize STS nominal insertion during those times. Conversely, there will be times when the station must operate at higher altitudes (i.e., well above 450 km) to meet the 90-day lifetime requirement.

There are several disadvantages associated with operating at a variable or constant-density low altitude:

(1) The servicing altitude would be constantly changing, which would complicate STS mission planning;

(2) The co-orbiting free-flyer constellation would have to be maneuvered in concert with the station;

(3) Boosting the station to a higher altitudes to follow atmospheric density changes would increase propellant usage;

(4) Sensitive low-gravity experiments could be disrupted during altitude changes; and

(5) At low alitudes, there are increased atomic oxygen and airglow concerns (see section 4.1.5).

4.1.1.3 Variable High Altitude (450 to 525 km)

In this servicing strategy, the Space Station remains in a high orbit until rendezvous with the Orbiter. At that time, it descends to the 450 km lower limit of the high altitude range. To reiterate the basis for this 450 km lower limit, if the propulsion system failed below this altitude, the station could re-enter the Earth's atmosphere before the next STS 90-day servicing mission could be accomplished. It is estimated that the station would have to descend to the lower orbit for 14 out of 90 days to be serviced by the STS.

The disadvantages of this flydown servicing strategy, which is shown as number (3a) in figure 4-1, far outweigh the advantages.

(1) Large propellant requirements for deboost and reboost between orbiting altitudes.

(2) Experiments would be disrupted during altitude changes.

(3) The complexity of having to match STS launch schedules with station altitude changes cause launch and operating delays.

(4) The nodal regression between the Space Station at the STS nominal insertion rendezvous altitude and the station's co-orbiting free-flyers at the high alitude would be about half a degree per day. The delta-V requirement to regain formation flight with the free-flyers would be approximately 33 m/s a day, based on half a degree

per day differential nodal regression and half a degree plane change per degree differential nodal regression.

4.1.1.4 Orbit Decay from High Altitude

This is a variation on the variable high-altitude option and is identified as number (3b) in figure 4-1. The Space Station maintains a high altitude but incorporates an orbit decay cycle to rendezvous with the STS at a somewhat lower altitude. The station is reboosted after the STS is disengaged. The primary advantage of this option is that no propellant would be used for deboost.

Again, the disadvantages outweigh the advantages.

(1) Since atmospheric density varies dramatically due to Sunspot activity and there is currently a high uncertainty rate ($\pm$25%) in predicting even very short-term atmospheric density, it would be extremely difficult to match traffic frequency with orbit decay rates.

(2) Large thrusters or a long burn time with small thrusters would be required for reboost to high orbit.

(3) Some experiments may be adversely affected by not having a repeating orbit.

(4) Co-orbiting free-flyers would either have to be left in high orbit and realigned after station reboost, or would have to accompany the station during orbit decay and reboost.

4.1.1.5 Fixed High Altitude, Remote Servicing

This servicing strategy involves keeping the Space Station at a fixed high altitude, ranging from about 450 to 525 km, near the lower limit of the Van Allen Radiation belt. The Space Station is placed in high orbit by direct insertion and the OMV servicing vehicle transfers payloads between the STS and the Space Station. This strategy is identified as number (4a) in figure 4-1.

The advantages of the high, fixed altitude orbit with this remote servicing strategy are:

(1) Low station-keeping propellant usage due to decreased atmospheric drag and subsequent orbit decay, and minimal reboost requirements from servicing operations.

(2) Constant free-flyer/Space Station formation.

(3) No contamination from the STS.

(4) Minimal experiment disruption.

The disadvantage associated with this remotely serviced fixed high-altitude strategy is that a servicing vehicle (OMV) must be used to transfer crew and payloads between the Orbiter and Space Station, thereby increasing OMV propellant requirements considerably.

4.1.1.6 Fixed High Altitude, Servicing via Direct Insertion

This servicing strategy has the Space Station at a fixed altitude, again near the lower limit of the Van Allen Radiation belt. A discussion of methodology for maintaining a given altitude or dispersion about that altitude is found in Section 5.1.1. This strategy, identified as number (4b) in figure 4-1, utilizes STS direct insertion to the high altitude (450 to 525 km). The advantages associated with this servicing strategy are similar to those of the other high altitude strategies in that the propellant usage is less due to the decreased atmospheric drag. There are also minimal reboost requirements because there is not a variable altitude involved. The other advantages are the constant free-flyer/Space Station formation, minimal experiment disruption due to the constant altitude, and the direct cargo delivery from the Orbiter to the Space Station at the high altitude. A disadvantage is that the shuttle must launch on time since a phasing orbit is not used. Also, all of the servicing strategies that involve Orbiter docking directly with the Space Station, as opposed to using a transfer vehicle such as the OMV, cause potential experiment disruption from the docking operation. The surrounding environment can suffer contamination effects during STS docking, which poses potential EVA concerns. (This is discussed in more detail in section 5.0).

4.1.1.7 Two-Station Case

A seventh servicing strategy involving two stations was analyzed, assuming one was at a 90-deg and one at a 28.5-deg orbit inclination. The 28.5-deg station could use one of the previously discussed servicing options and the 90-deg station would be subsequently serviced from the lower station using an OTV or OMV servicing vehicle.

The advantages of the two-station servicing operation are:

(1) The payload limitations imposed by a 90-deg STS launch would be eliminated.

(2) Both stations could be serviced by a single STS launch, one directly and one indirectly.

(3) The 90-deg station would be free of STS contamination.

The disadvantages of this strategy are:

(1) The 90-deg station requires the support of a 28.5-deg station.

(2) The servicing vehicle must have a large delta-V capability. Even if a three-impulse aero-assisted orbit change technique is used, the delta-V's are so large that direct servicing of the 90-deg station by a 90-deg STS launch may be more sensible.

(3) Payloads destined for the 90-deg station must be designed for mating and demating with servicing vehicles.

(4) Two on-orbit transfers are required for each payload.

4.1.1.8 Servicing Option Summary

The servicing strategies broadly fall into two groups: variable or fixed altitude. The advantages and disadvantages are briefly reviewed as follows. The effects of flying the Space Station at different altitudes include: a high degree of thrust capability and therefore propellant usage; disruption of station/free-flyer formation; irregular orbit decay influences (thereby complicating mission and service planning, and experiments); and disruption of other operations. The advantages associated with variable altitudes are a possible increase in STS payload capacity and potential propellant savings due to maintaining a constant-density orbit.

The advantages of a fixed altitude depend on the actual altitude chosen. Low altitudes, while increasing STS payload capacity require more frequent thrust activity because of the more rapid orbit decay. Servicing vehicles would not be necessary to transfer payloads from the STS to the Space Station. However, at a fixed low altitude, free-flyers would also require a greater propulsive capacity to maintain orbit. The fixed high altitude, on the other hand (up to 525 km), requires the least amount of propellant, maintains free-flyer formation, poses little disruption to mission experiments, and can be serviced either by the OMV from the STS via nominal insertion or through STS direct insertion.

Section 5.0 of this report compares the propulsion requirements for the servicing strategies that are retained from this analysis.

4.1.2 STS Performance Influences

The STS performance capabilities were discussed in section 2.2.1 and were shown graphically in figure 2-15. The impact on station servicing depends on the inclination and altitude of the station to be serviced, and the insertion technique. Direct insertion was successfully used for the recent Solar Maximum repair mission, and it is used as the baseline in this study. Figure

2-15 showed that there is a payload penalty of only about 3,000 kg from 400 km to 560 km for the 28.5 deg inclination station if direct insertion is used. Since most payloads are volume rather than weight limited, the weight limitation is relatively insignificant.

In conclusion, STS performance will have little influence on servicing strategies if direct insertion is employed. If nominal insertion is used however, it becomes mandatory to use an OMV-type vehicle to service the fixed high alitude station.

4.1.3 Orbiter Fleet Size Influences

This study assumes that Orbiter fleet size will be adequate to meet Space Station and free-flyer servicing requirements, in addition to non-Space Station-related activities. No constraints were placed on servicing strategies or propulsion system selections due to potential fleet size limitations or unavailability. The traffic model, which is discussed in section 4.2, provides estimates of fleet size, number of flights per year, and other factors that will determine Orbiter traffic and shows that a fleet of at least eight Orbiters could be required by the year 2000 to satisfy mission requirements. The traffic model incorporated potential commercial, defense, and research budget estimates to project high, median, and low mission activity levels.

4.1.4 Atomic Oxygen Influences

The Space Station will be exposed to dissociated, or "atomic," oxygen in the upper atmosphere. In fact, atomic oxygen is the predominant atmospheric constituent at some altitudes. Normal molecular O_2 is photo-dissociated in the upper atmosphere. Because of the low atomic weight of the dissociated atom, it preferentially stratifies to Space Station altitudes.

Atomic oxygen is a vigorous, highly reactive oxidizer. Apparently, the only materials that are immune to atomic oxygen attack are those that form a hard, impervious oxide coating, such as aluminum. Atomic oxygen and its reactions with spaceraft effluents are believed to be responsible for the infrared and airglow effects that can adversely affect the use of sensitive optical instruments.

Atomic oxygen density decreases as atmospheric density decreases. Consequently, the deleterious effects of atomic oxygen are reduced by about an order of magnitude from about 400 km to about 500 to 550 km. Even at the higher altitude, atomic oxygen could cause certain materials to erode.

4.2 Free-Flyer Servicing

Free-flyer servicing covers the entire mission cycle, from checkout and orbital deployment, to subsequent on-orbit support and removal of the spacecraft from orbit. On-orbit support includes the examination, maintenance, and repair of basic subsystem and mission-peculiar equipment, resupply of consumables, and reconfiguration of experiments. Free-flyer servicing also encompasses temporary on-orbit storage of free-flyers awaiting repair, end-of-mission retrieval, Earth return, or controlled re-entry disposal.

4.2.1 Differential Drag Considerations

Free-flyers that co-orbit with a Space Station will, in general, have different drag characteristics than the Space Station. Differential drag, and the changes in relative orbit location it causes, must be considered in selecting (1) an orbit makeup strategy for the Space Station, (2) an orbit makeup strategy for co-orbiting free-flyers, and (3) a propulsion system for servicing operations. If two spacecraft, initially co-orbital, experience differential drag and do not compensate for it, they will become separated (1) in altitude by the difference in orbit decay; (2) along the orbit track because the free-flyer will move faster at lower altitudes ; and (3) in plane, because of differential nodal regression caused by the progressive difference in altitude.

4.2.1.1 Analytical Model

To study this phenomena, an orbital computer simulation was employed for two different free-flyers. This simulation model contained a Jacchia dynamic atmospheric density model, effects due to the Sun and Moon, and harmonics of the Earth's gravitational field through the fourth order. The SASP and advanced X-ray astronomy facility (AXAF), another specialized free-flyer, were used as the Space Station co-orbiting free-flyers. These free-flyers were chosen on the basis that they represented a fairly wide range of ballistic coefficients (approximately 21 to 190 kg/m^2). For comparing the different orbit decay rates, it was assumed that the Space Station maintained its initial altitude by using continuous orbit makeup. The NASA neutral and minimum atmospheres were used for comparison. These models represent design (maximum) and average-density profiles, respectively. They are generated by entering appropriate fixed parameters into the Jacchia atmosphere model.

The results of these simulations are shown in figures 4-2 and 4-3.

4.2.1.2 Results

As figures 4-2 and 4-3 show, the along-track separation develops more rapidly than the other separations. The sinusoidal effects in the along-track separation are due to the fact that once the free-flyers and station become 180 deg out of phase, they once again approach each other (i.e., the free-flyer "laps" the station).

If the same average altitude is maintained, the plane differences very nearly cancel out. A possible maneuvering strategy for a co-orbiting free-flyer needing periodic service is illustrated in figure 4-4. The orbit of the free-flyer experiencing the greatest decay rate is reboosted once per service interval. The reboost occurs halfway between intervals so that as the service time approaches, the free-flyer approaches the Space Station with a low closing velocity. Terminal maneuvering can then be used to effect rendezvous and capture. A computer simulation of this strategy is shown in figure 4-5.

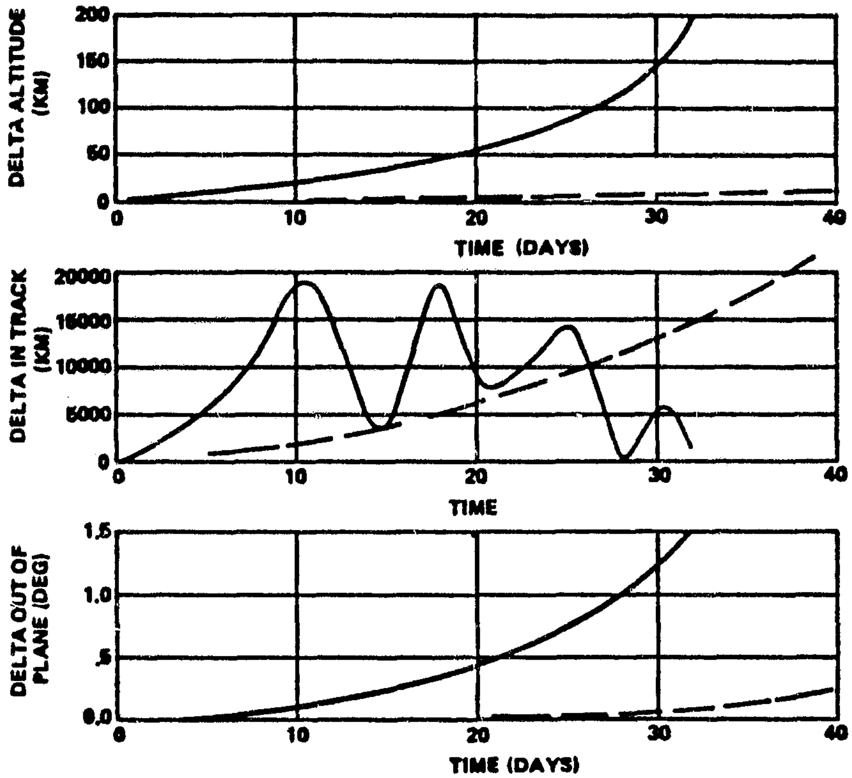


Figure 4-2. Differential Drag Between Operational Station With Drag Makeup and SASP Without Drag Makeup

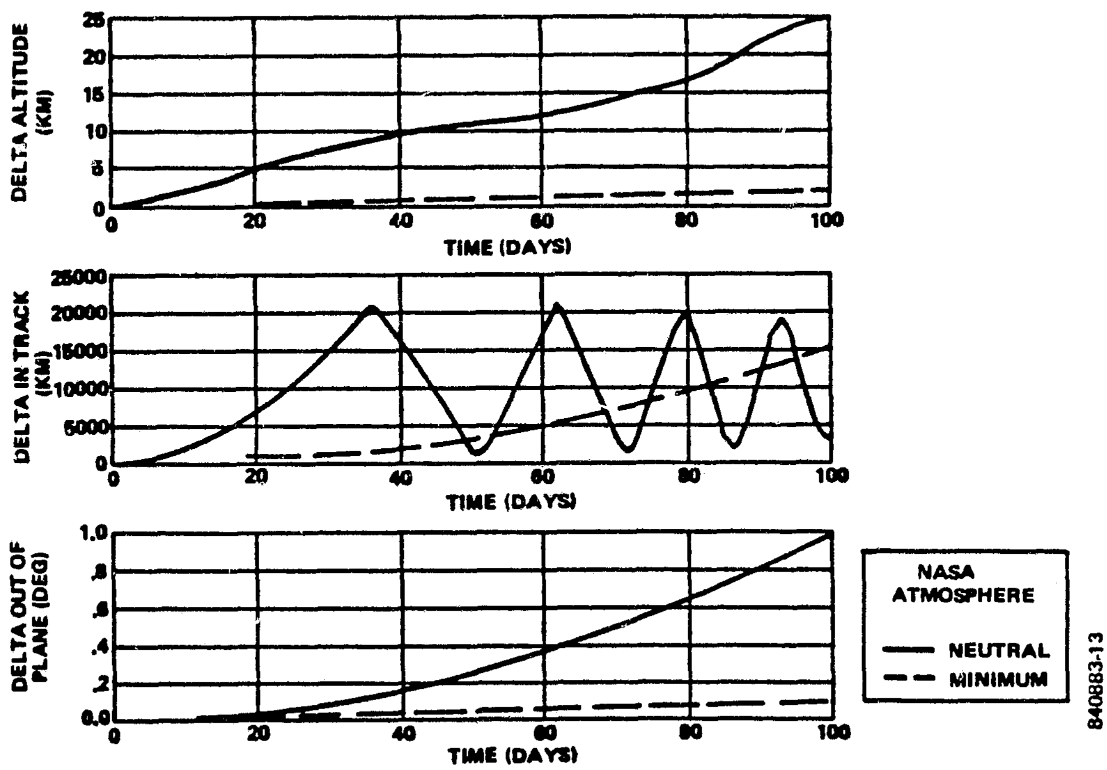


Figure 4-3. Differential Drag Between Operational Station With Drag Makeup and AXAF Without Drag Makeup

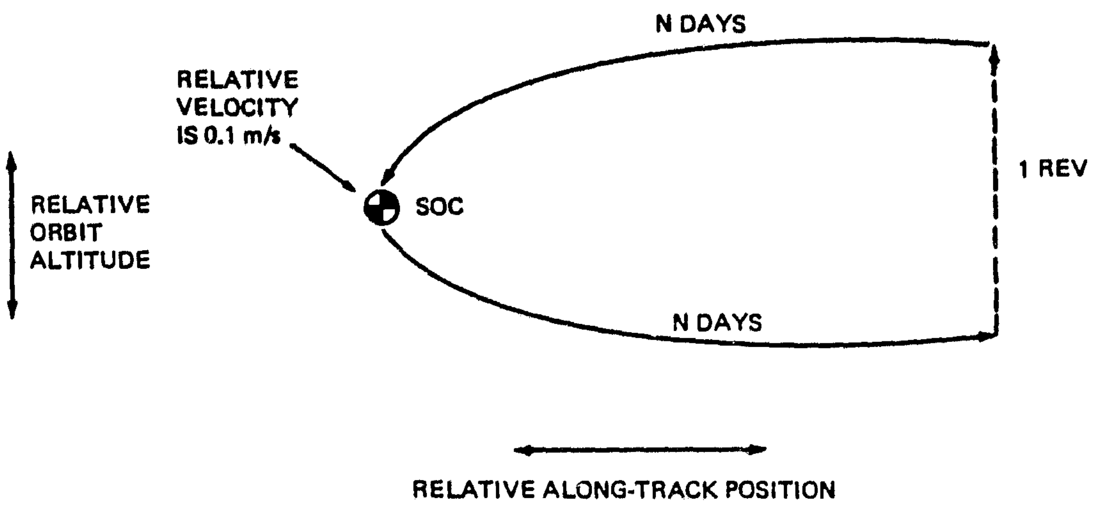


Figure 4-4. Differential Drag Orbit Makeup Strategy

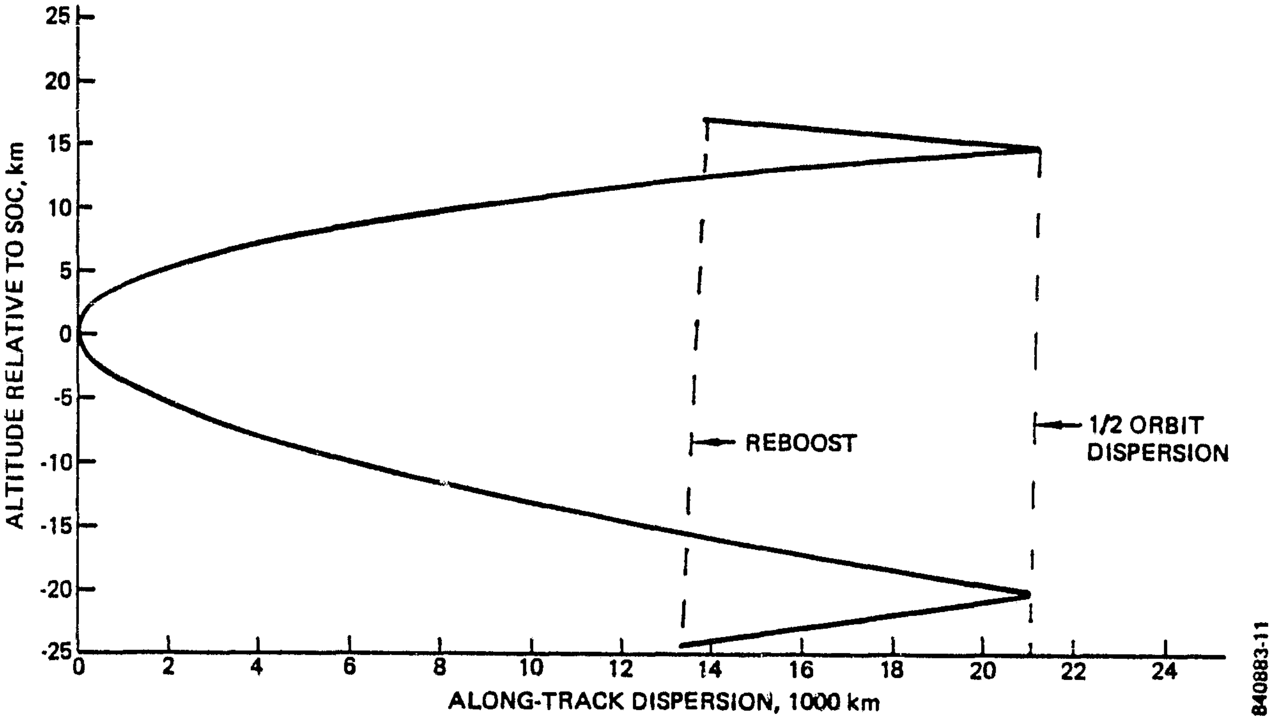


Figure 4-5. Altitude Versus Along-Track Dispersion from Station

Free-flyer propulsion and servicing strategies are interrelated. It is apparent that the free-flyer can be maintained either near the station or allowed to drift. Relative drift periods up to a few weeks long are nearly the same as formation flying in terms of the delta-V. However, extended drifting periods cause differences in orbit plane to buildup, creating additional delta-V and propellant requirements to return the free-flyer to the station (except at those infrequent times when in-plane phasing occurs naturally).

4.2.2 Functional Analysis of Free-Flyer Servicing

There are two types of servicing operations for free-flyers: those performed at the Space Station and those performed in-situ remotely from the station. Remote servicing would be performed on free-flyers that are too large to fit into the OMV or OTV to be brought to the Space Station, or those satellites that would impose prohibitive propulsion requirements to transport to the Space Station.

Figure 4-6 shows the following functional modes of free-flyer servicing at the Space Station:

(1) Payloads attached to and operated on the Space Station.

(2) On-orbit free-flyers without propulsion.

(3) On-orbit free-flyers with propulsion.

(4) Free-flyers prepared and assembled at the Space Station and launched for co-orbiting flight or transfer to another operating orbit.

The types of servicing operations performed on the Space Station are listed in Table 4-1 and are keyed to the respective missions. Many of the co-orbiting free-flyer services are the same as those required for attached payloads. While most of these servicing operations could be performed with the STS, the Space Station can deliver servicing on demand and can offer other services.

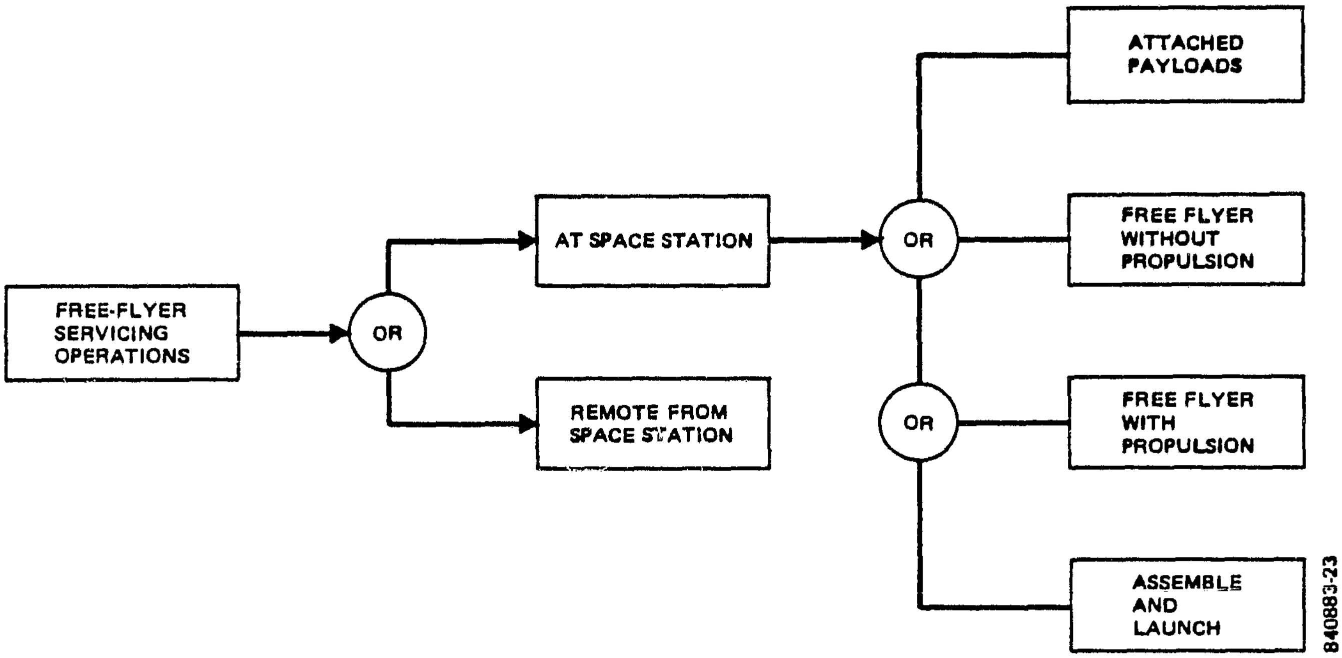


Figure 4-6. Free-Flyer Servicing Operations

Table 4-1: Free-Flyer Service Missions

	LOCATION			LAUNCH	
SERVICE OPERATIONS	ATTACHED PAYLOADS	CO-ORBITING	REMOTELY ACCESSIBLE	LOW ENERGY ORBIT	HIGH ENERGY ORBIT
EXAMINATION	•	•	•		
RETRIEVAL		•			
MAINTENANCE/REPAIR	•	•	•		
RESUPPLY	•	•	•		
RECONFIGURATION	•	•	•		
ON-ORBIT ASSEMBLY				•	•
MATE UPPER STAGES					•
TEST & CHECKOUT	•	•	•	•	•
ON-ORBIT STORAGE		•		•	•
DEPLOY		•		•	•

- GO-NO-GO FOR DEPLOYMENT, SERVICING VERIFICATION/ EFFECTIVENESS IS FREE FLYER USER DECISION
- FREE FLYER DEPLOYMENT VIA SPACE STATION COMMAND
- FREE FLYER SEPARATION ΔV DURING DEPLOYMENT IMPARTED BY STATION EQUIPMENT WHERE PRACTICAL
- STATION SAFETY CONSIDERATIONS
  - FF HOT RCS FIRINGS .... > 200 FT SEPARATION
  - LIQUID ROCKET ENGINE FIRINGS....... > 2700 FT SEPARATION
  - SOLID ROCKET ENGINE FIRINGS .... ADEQUATE SEPARATION REQUIRED TO ASSURE STATION EXIT OF HAZARD ENVELOPE
- CLOSE PROXIMITY OPERATIONS
  - TERMINAL ACQUISITION OF S/C WILL BE CONTROLLED BY STATION
  - "CLEAN" VEHICLE PROVIDES CLOSURE ΔV
- STATUS MONITORING, CHECKOUT, ACTIVATION/DEACTIVATION OF FF'S IS USER-CONTROLLED (FF COMM VIA STATION S-BAND OR FF'S SYSTEM VIA TDRS)
- MINIMIZE STATION STATUS/CHECKOUT INVOLVEMENT
  - POWER (AS REQUIRED)
  - OVERALL HEALTH (EXTENT TBD, STANDARDIZED FOR ALL FREE FLYERS)
- EVA IS ACCEPTABLE SERVICE MODE

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Figure 4-7. Assumptions-Servicing Scenarios

These services include on-orbit assembly of large systems, mating of large upper stages, on-orbit storage of free-flyer hardware if predeployment test and checkout fails, and the continuous availability of these services. Figure 4-7 gives the free-flyer servicing assumptions used in this study. Free-flyers co-orbiting in close proximity to the Space Station can be serviced easily and at frequent intervals. This could be the preferred operational mode for missions that require frequent service but are separated from the Space Station to avoid contamination. A good example is a space processing facility that needs a low-g environment. Certain optical instrument missions will also utilize this mode to avoid outgassing and similar contamination problems associated with the Space Station. These free-flyers would have to be retrieved and serviced by an OMV.

4.2.2.1 Space Station-Attached Payloads

There are some missions, derived from the Spacelab, that could be on Space Station-attached free-flyers. In this case, the services that would be provided are power, communications, maintenance and instrument changes. This type of servicing mission is not relevant to the current propulsion study however, and is not evaluated in any detail.

4.2.2.2 Free-Flyers Without Propulsion

Free-flyers without a delta-V capability, either because they lack a propulsion system or their fuel is depleted, must be transported to the Space Station for servicing. In this case, the Space Station dispatches a vehicle, such as the OMV or OTV (depending on propulsion needs) to retrieve the free-flyer. Figure 4-8 shows the primary servicing functions for free-flyers that must be returned to the Space Station. After the free-flyer is berthed to the Space Station, the free-flyer can be repaired, resupplied, reconfigured, checked out, and returned to its operational orbit. Free-flyers such as the space telescope, long-duration exposure facility, advanced X-ray astrophysics facility, and materials-processing free-flyers are likely candidates.

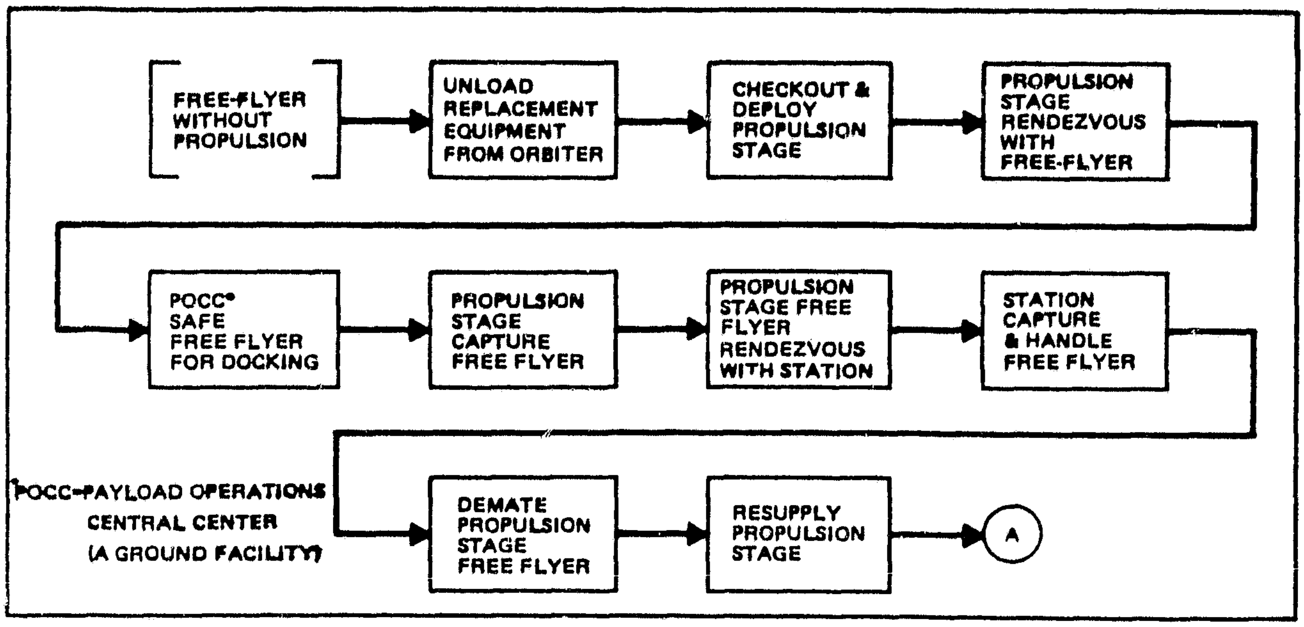

Figure 4-8. Servicing Functions for Free-Flyers Without Propulsion

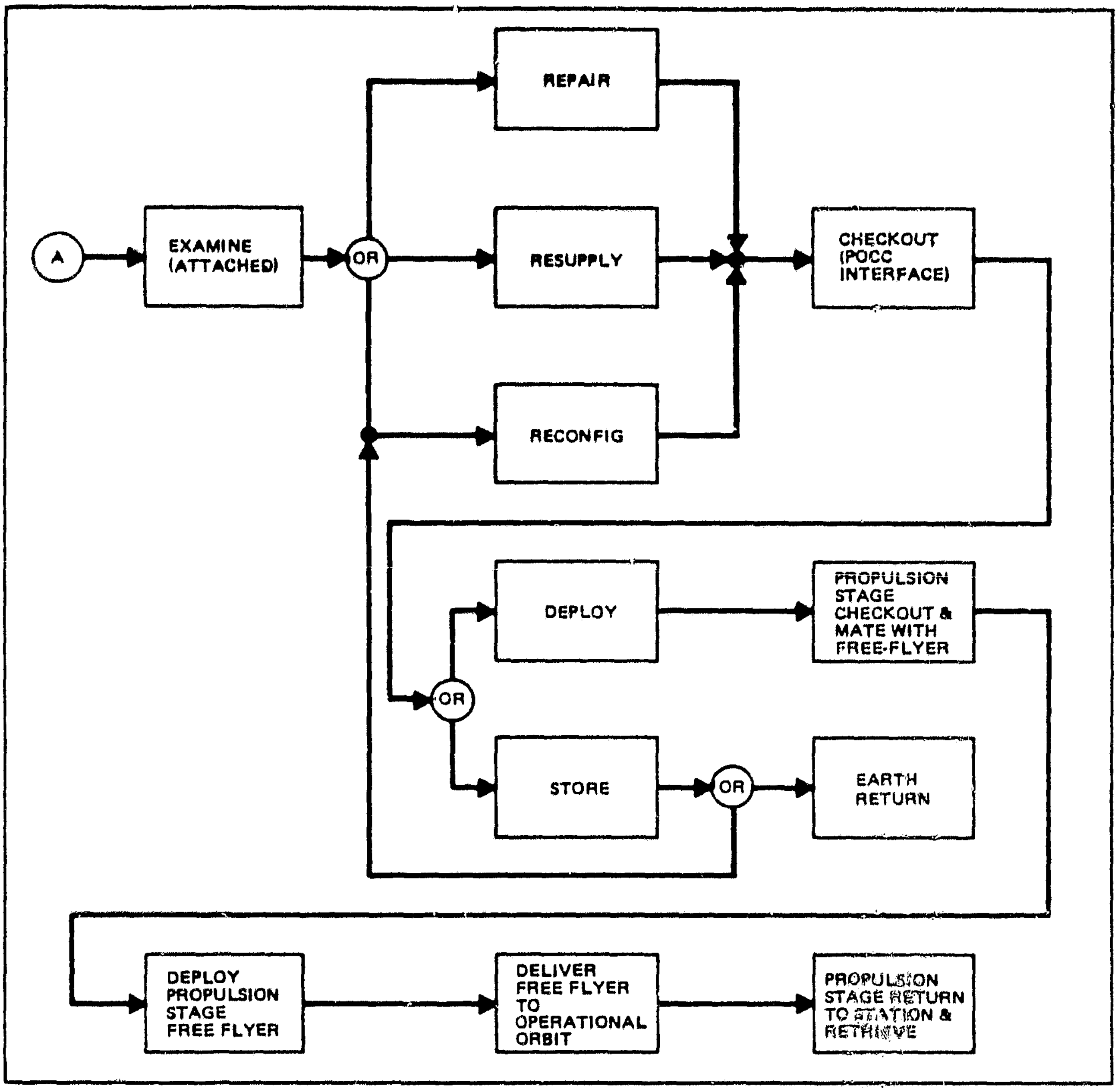

Figure 4-8. Servicing Functions for Free-Flyers Without Propulsion (Continued)

4.2.2.3 Free-Flyers With Propulsion

Free-flyers that have a propulsion system are maneuvered to the vicinity of the Space Station by a payload operations control center, and either retrieved by an OMV or berthed directly to the Space Station. The propulsion requirements for this category of free-flyers are discussed in section 5.0 of this report.

4.2.2.4 Free-Flyers Assembled and Launched from the Space Station

The launched free-flyers are subdivided into two energy-orbit categories: low-energy orbits (up to 2,000 km) and high-energy orbits (above 2,000 km). The assembly and launch mode (figure 4-9 consists of free-flyers, such as the GEO communications platform, that are delivered to the Space Station by the STS for subsequent launch. Free-flyers are launched at the appropriate time into a near co-orbiting operational location or launched with a propulsion stage to transport them to an operational location. In this scenario, an appropriate propulsion stage is checked out and attached to the free-flyer prior to launching operations.

4.2.2.5 Strategies for Retrieving Co-orbiting Free-Flyers

Figure 4-10 depicts three strategies for retrieving co-orbiting free-flyers for maintenance, resupply, and reconfiguration at the Space Station. The requirements imposed on the Space Station will vary according to the proximity or relative position of each co-orbiting free-flyer to the Space Station and the free-flyer's orbit-adjust capabilities. In the first retrieval scenario, the free-flyer occupies the same orbit (altitude and inclination) as the Space Station. In this situation, the free-flyer can either be controlled from the Space Station or operated by ground control. When free-flyers return to the Space Station and are berthing or operating in close proximity, they should be controlled by the Space Station for on-orbit safety. Because it is impractical to maneuver the Space Station toward the free-flyer for terminal acquisition, final free-flyer retrieval is expected to be accomplished by an OMV that can be readily deployed and controlled from the Space Station.

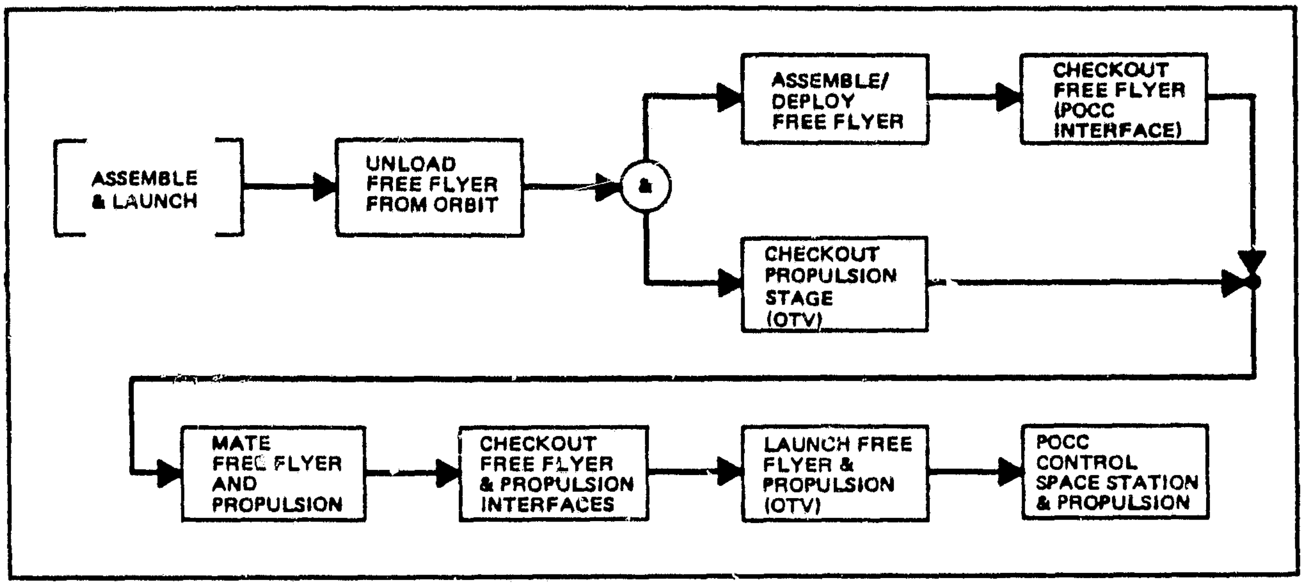

Figure 4-9. Free-Flyer Assembly and Launch Services

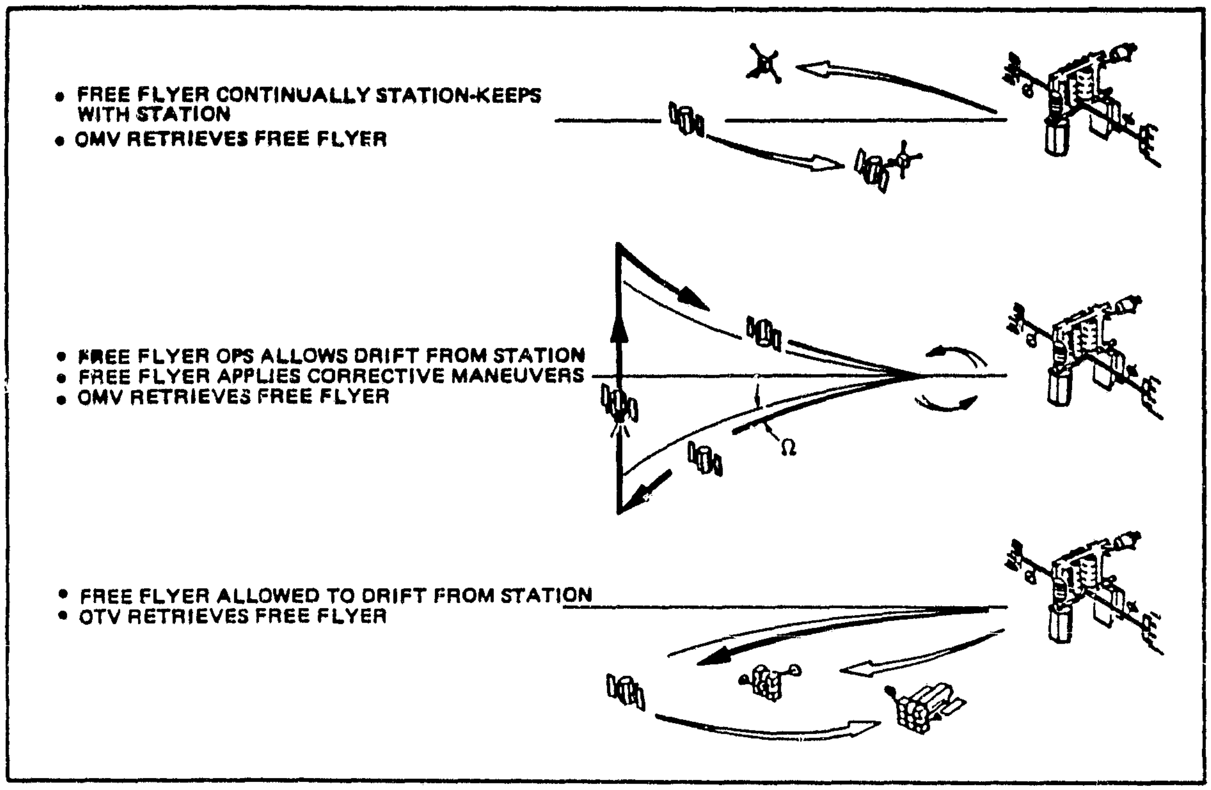

Figure 4-10. Strategies for Retrieving Co-Orbiting Free-Flyers

Some free-flyers will not actively maintain an orbit with the Space Station but will be allowed to decay in altitude and drift out of plane. If the free-flyer has an orbital maneuvering system, as shown in the second scenario, it can be used to adjust its altitude so that it will drift back toward the Space Station when it is time for maintenance. A Space Station-controlled OMV can then retrieve these free-flyers. Without an orbital maneuvering system, it will continue to drift out of plane from the Space Station, as shown in the third scenario. The latter free-flyer must be retrieved by a more capable Space Station-based vehicle, such as the OTV, which must rendezvous with the free-flyer, dock, and transport it back to the Space Station.

4.2.3 OTV and OMV Servicing Capabilities

This section discusses the capabilities of the OTV and OMV to accomplish on-orbit servicing, which involves inclination, altitude, or phasing change, or any combination of these changes.

4.2.3.1 Phasing Changes

A free-flyer in the same orbit as the station, but at some distance ahead or behind the station, is said to be at a phase angle from the station. Transfer from the station to the free-flyer involves an initial delta-V and then an identical but opposite delta-V for rendezvous. Returning to the station involves an essentially identical scenario. The delta-V imparted to the servicing vehicle primarly affects the time required to travel a given distance. Figure 4-11 illustrates the time-distance-delta-V relationship. In most cases, the time allowed for rendezvous and retrieval will be adequate to permit these operations to be carried out with a total delta-V on the order of 100 m/s.

4.2.3.2 Orbit Altitude and Inclination Change

Figure 4-12 shows Hohmann transfer (minimum energy) delta-V requirements for in-plane transfers from 400 km to 7000 km.

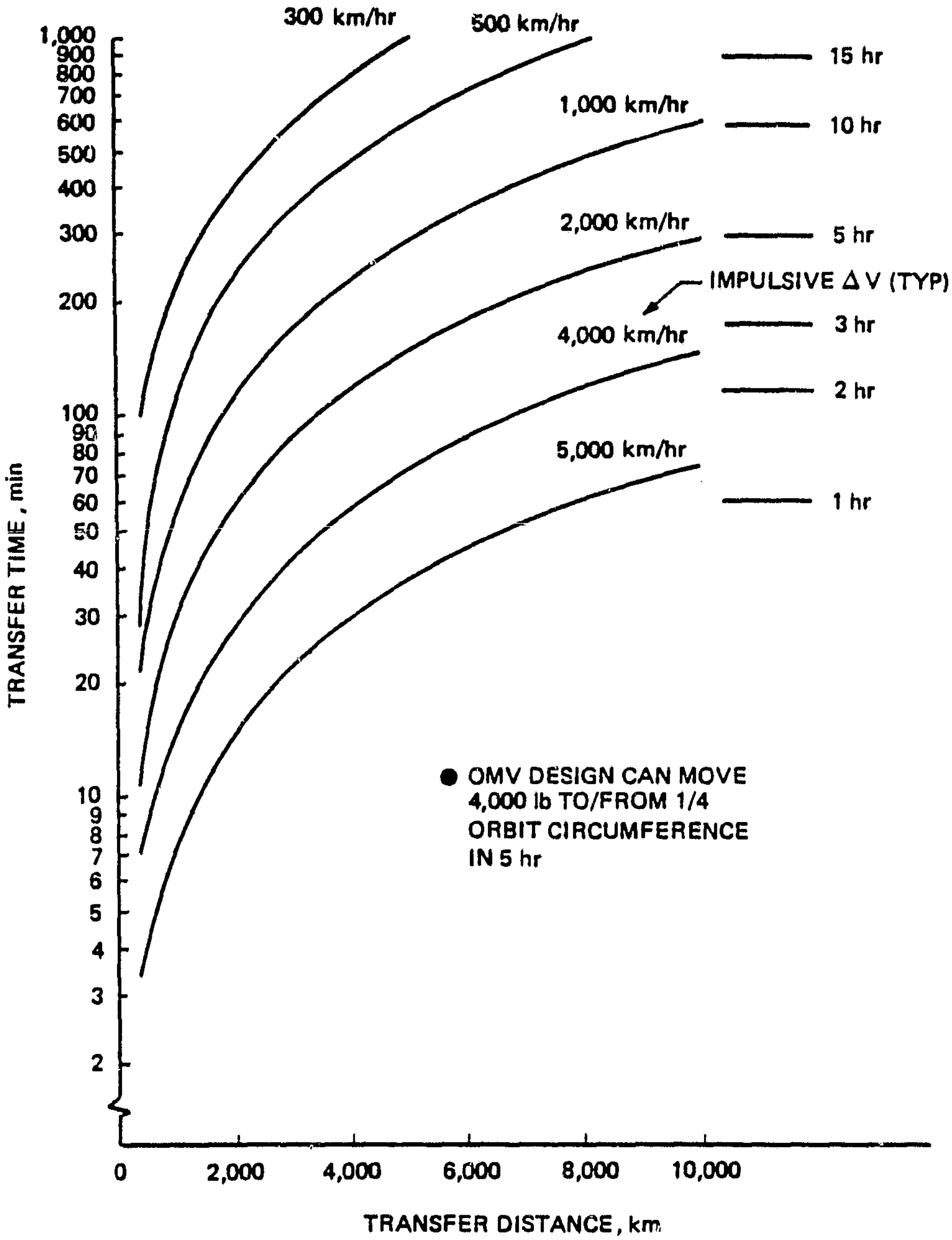

Figure 4-11. Time Required for Phase-Change Transfer

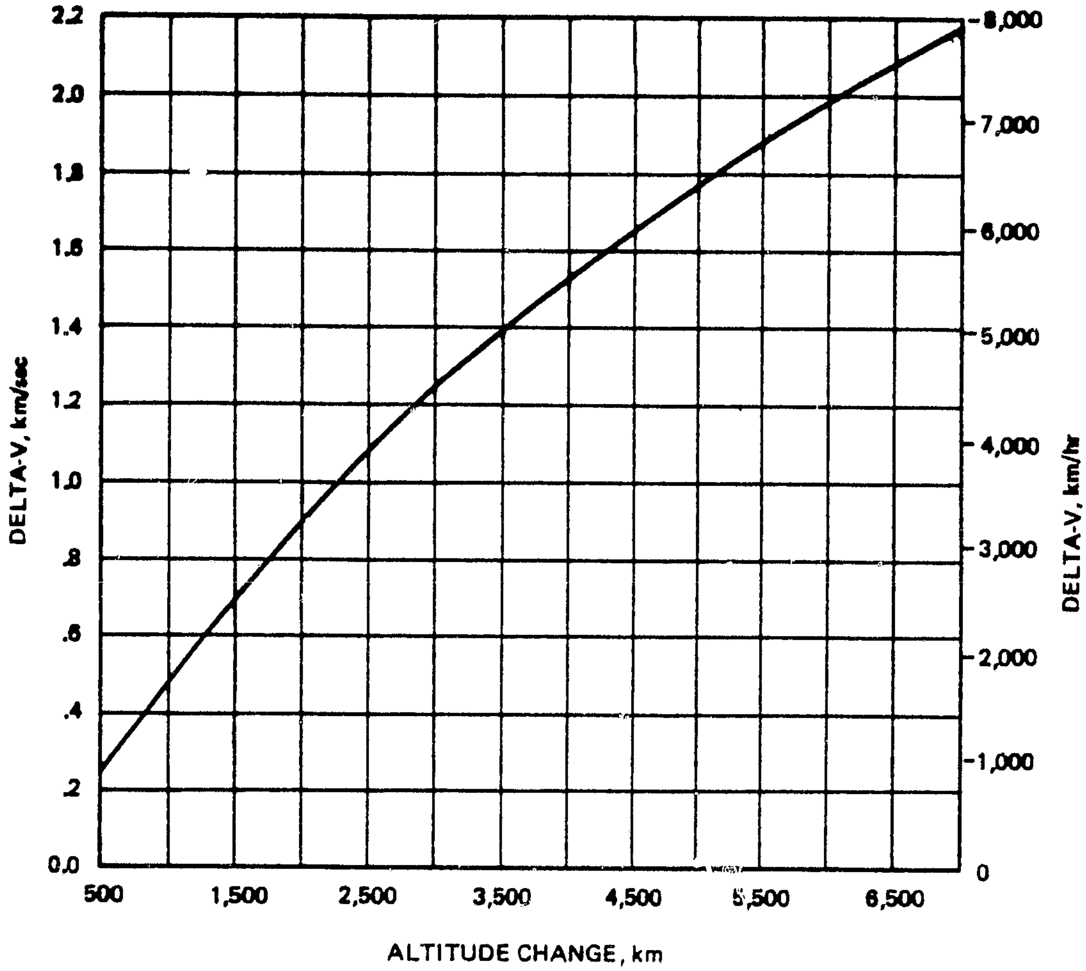

Figure 4-12. Hohmann Transfer Delta-V Required for Orbit Altitude Change

The delta-V required for an inclination change with no altitude change can be expressed by:

$$\Delta V = 2\ V_c \sin\left(\frac{\theta}{2}\right)$$

where: V_c = orbital velocity
θ = inclination change

For V_c = 27,974 km/hr (25,500 ft/s) some typical values are:

θ(deg)	ΔV km/hr	(ft/s)
10	4,876	4,445
20	9,715	8,856
30	14,480	13,200
40	19,135	17,443
50	23,645	21,554
60	27,974	25,500 = V_c

If plane and altitude changes are both required, the optimal propulsive strategy is to perform them simultaneously. Figures 4-13 through 4-16[1] show the optimal delta-V and perigee plane changes for these maneuvers for plane changes up to 30 degrees. Representative O_2/H_2 OTV propellant requirements are superimposed on the figures. The delta-V's are for one-way operations (transfer from a 500-km orbit to a higher orbit), while the propellant requirements are for round-trip operations.

The dashed lines illustrate the propellant required to (1) place a 2000g payload at the higher orbit and return with no payload, (2) go to the higher orbit with no payload and return with a 2000g payload, or (3) carry a manned module to the higher orbit and return with the module. Figure 4-17[1] illustrates curves of constant propellant space-based OTV requirements for plane, altitude increase, or a combination thereof for placing or retrieving the 2000g payload or for a manned module round trip.

1. Figures adapted from Boeing Document D180-26495-4, "Space Operations Center System Analysis, Final report", Vol. IV, Book 1 of 2, July 1981.

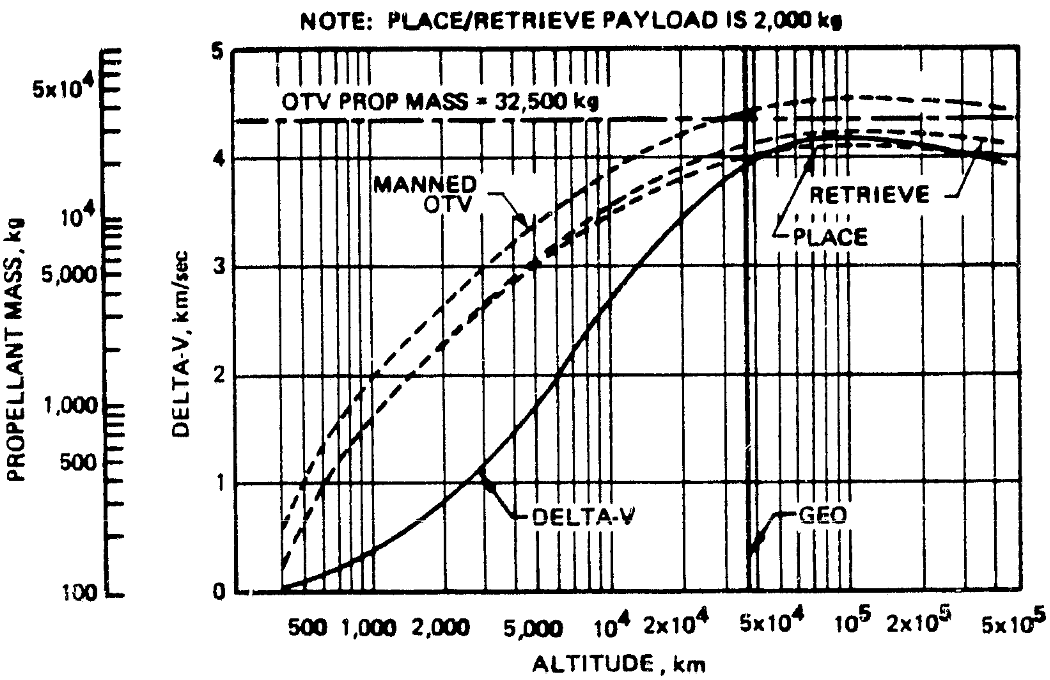

Figure 4-13. OTV Free-Flyer Access From Space Station With No Plane Change

NOTE: PLACE/RETRIEVE PAYLOAD IS 2,000 kg

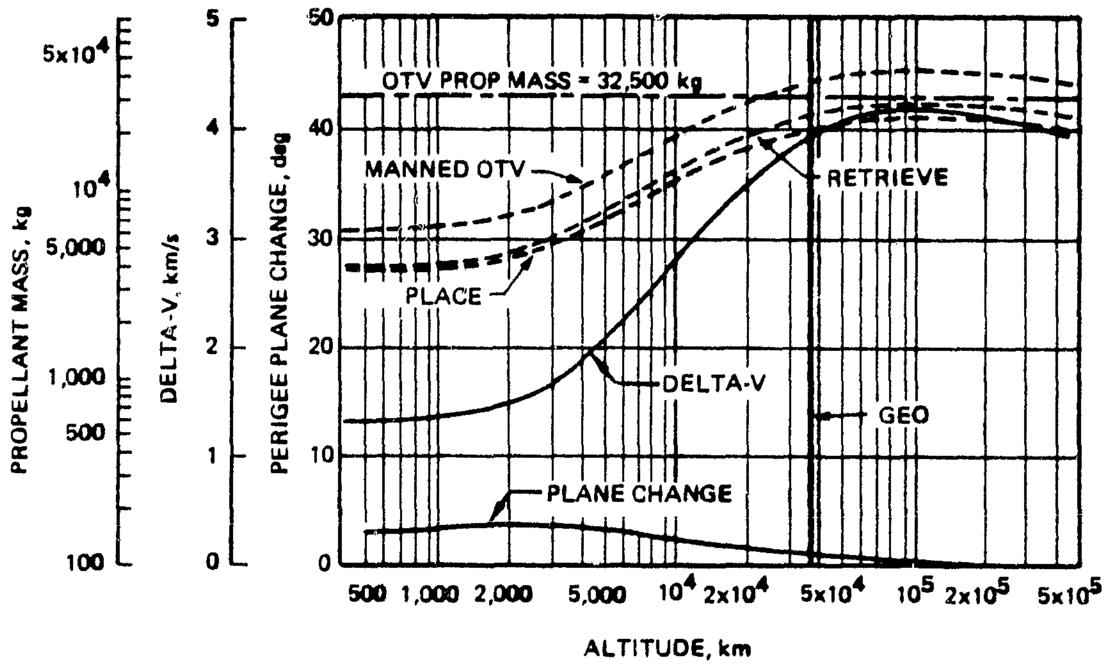

Figure 4-14. OTV Free-Flyer Access From Space Station With 10-Degree Plane Change

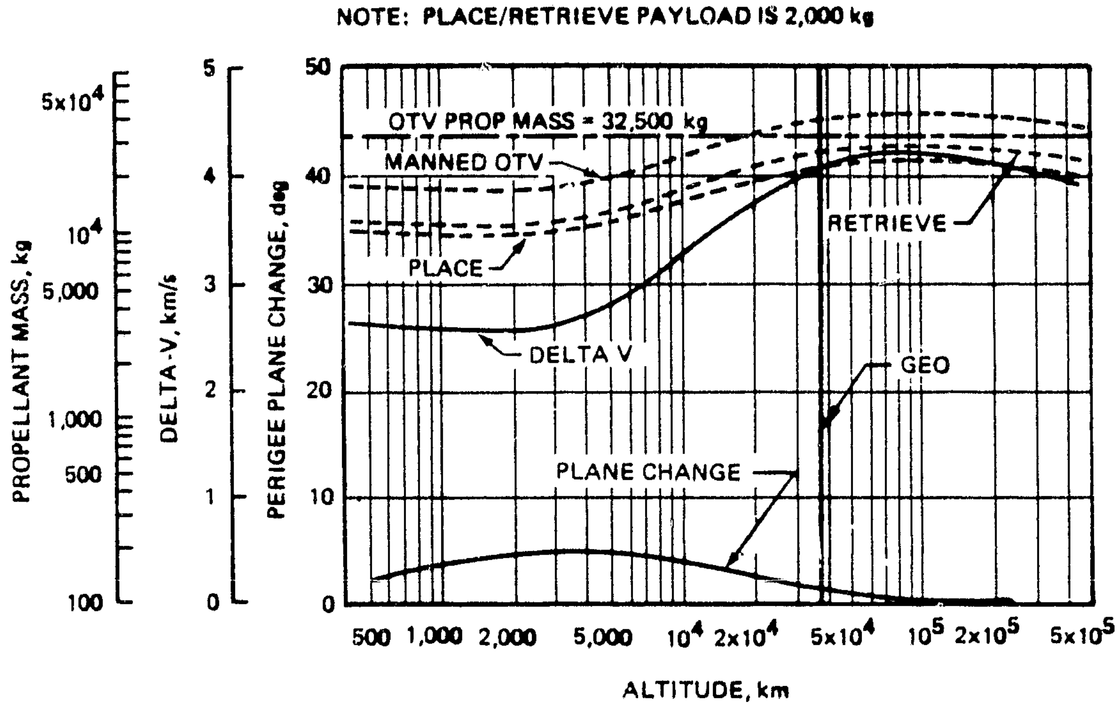

Figure 4-15. OTV Free-Flyer Access From Space Station With 20-Degree Plane Change

NOTE: PLACE/RETRIEVE PAYLOAD IS 2,000 kg

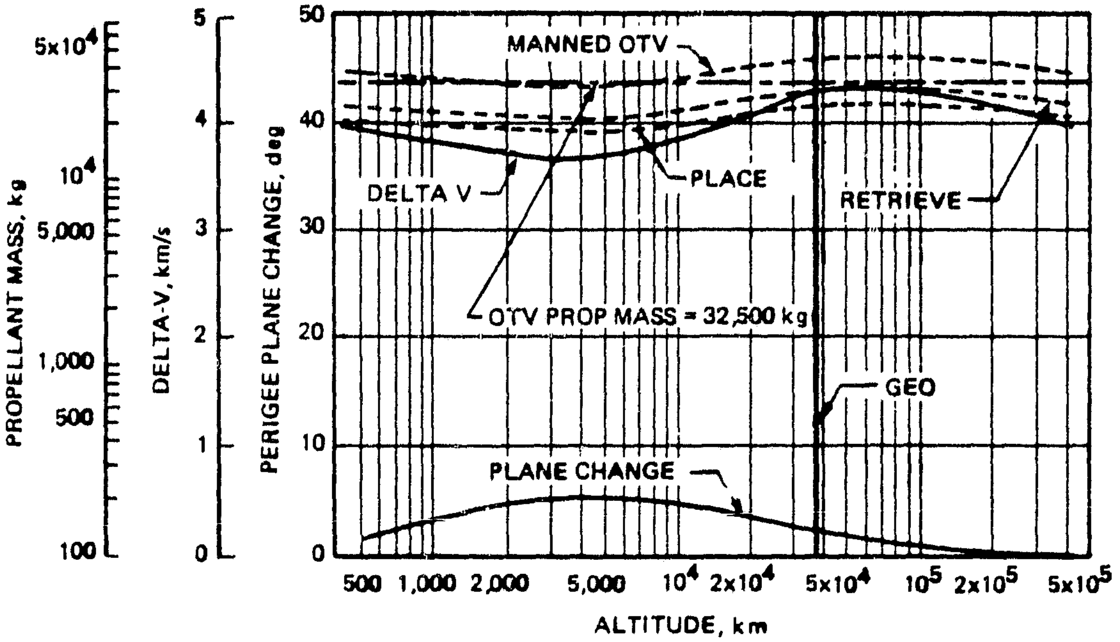

Figure 4-16. OTV Free-Flyer Access From Space Station With 30-Degree Plane Change

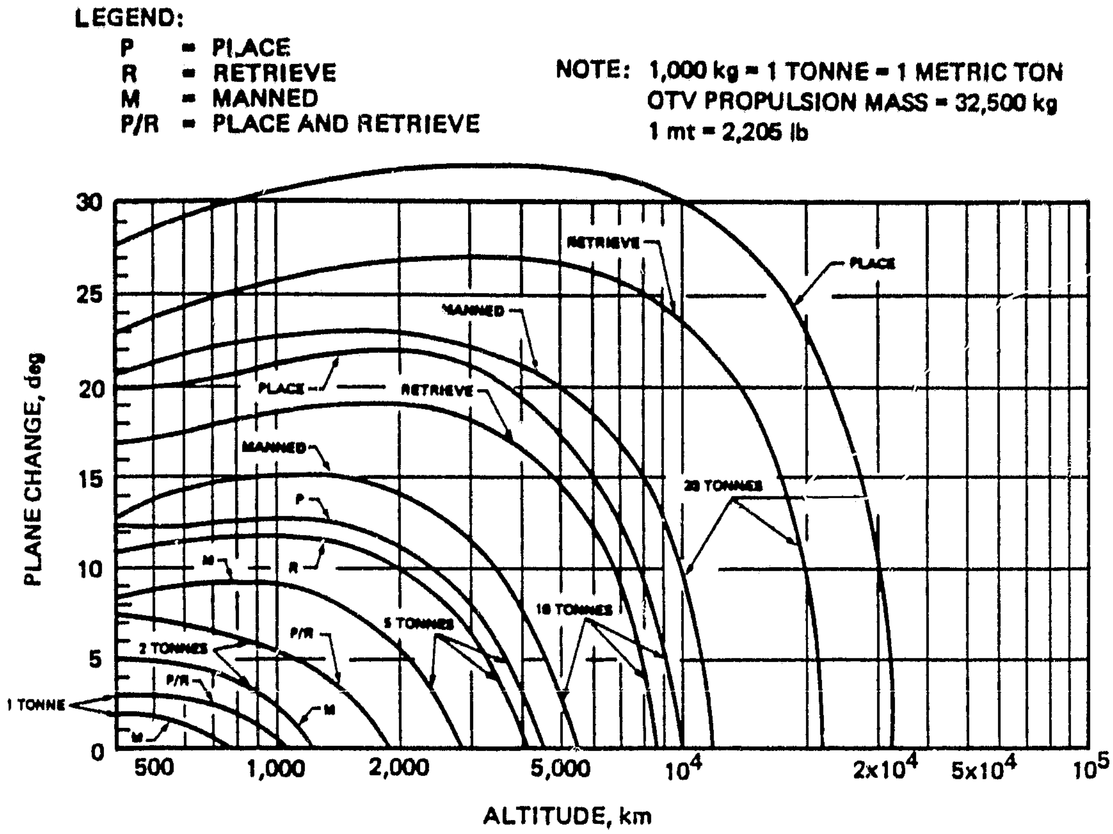

Figure 4-17. OTV Propellant Requirement Contours for Plane and Altitude Change

Propellant mass is shown here in tonnes. These calculations indicate that the OTV can transfer up to two metric tonnes for altitude increases up to 500,000m and inclination changes up to 30 deg.

4.2.3.3 OMV Servicing Capability

The OMV is currently planned for IOC in approximately 1989. The first version of the OMV will probably be an STS-based vehicle with a remote manipulation system (RMS). This configuration is expected to be adopted and used on the Space Station by approximately 1991. The primary modifications would be those required to make the vehicle capable of being space-based.

The OMV is expected to be equipped with a propulsion system that uses storable propellants for executing large delta-V maneuvers and a clean-firing, cold-gas propulsion system for free-flyer and Space Station close-proximity operations. An on-orbit refueling capability should also be provided.

Figure 4-18, which illustrates the OMV delta-V capability for the three propellants being considered, shows that the maximum delta-V for the volume-limited OMV is achieved by using the storable bipropellant combination of nitrogen tetroxide (N_2O_4) and monomethyl hydrazine (MMH).

Figure 4-19 shows the propellant requirements for an OMV using N_2O_4/MMH for various payloads. The normal fuel load shown in figures 4-19, 4-20, and 4-21 is based on normal OMV propellant tankage capacity. When high propellant levels are required, a λ' of 0.95 is used to determine additional tank and ancillary equipment weight (λ' = propellant mass/OMV mass).

If the OMV's mission is to service a free-flyer, all (or a portion) of the payload may be left at the free-flyer. Two cases are compared: one where only half the payload is delivered, which represents high propellant requirements, and one with an empty payload return.

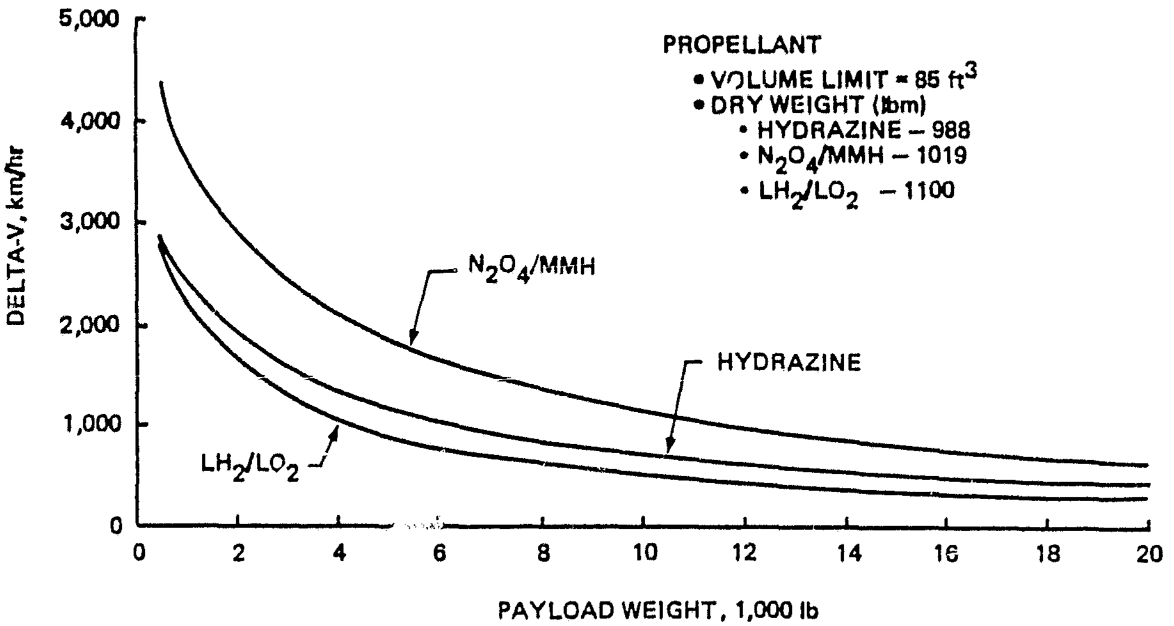

Figure 4-18. OMV Delta-V Capability

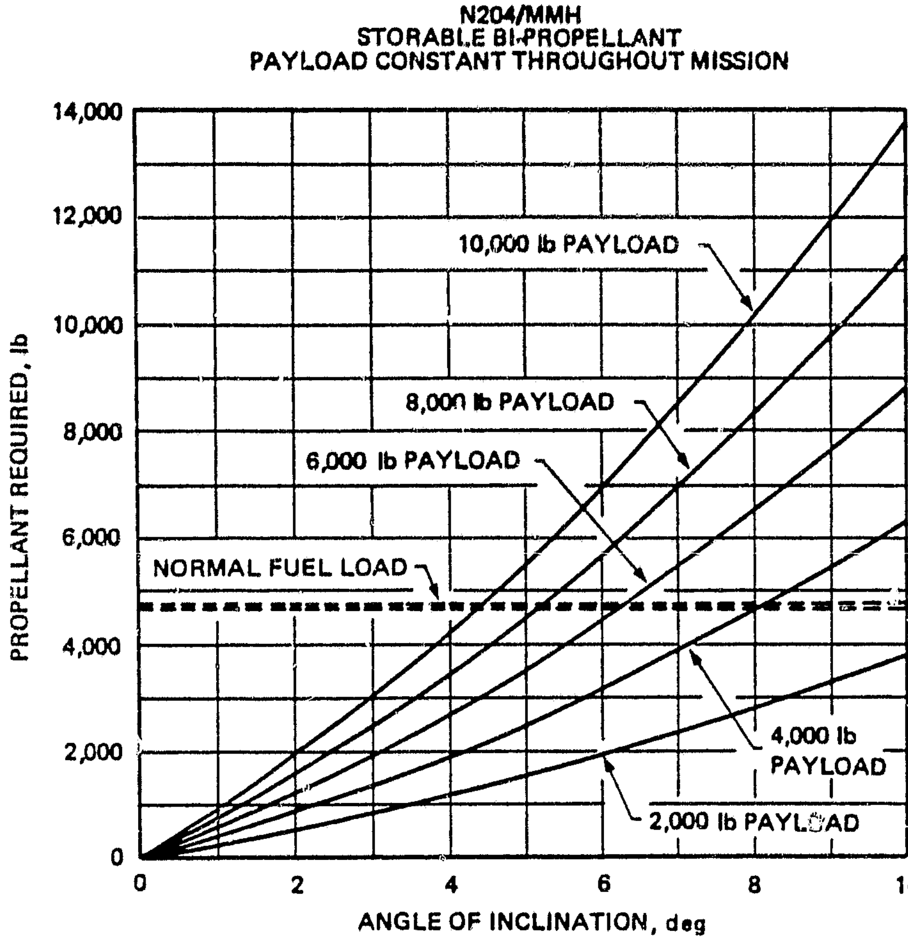

Figure 4-19. OMV Inclination Change Using N_2O_4/MMH Return With Constant Payload

Figure 4-20 shows the OMV propellant requirements for returning half a payload. In this case, the inclination angle was carried to 30 deg to illustrate the effect on propellant required for this extreme case. The figure shows that transferring 4,535g (10,000 lbm) 30 deg and returning 2,268g (5,000 lbm) requires about 48,989g (108,000 lbm) of propellant.

Figure 4-21 illustrates the propellant required for an OMV returning with no payload for up to a 30-deg inclination, using N_2O_4/MMH. A comparison of figures 4-20 and 4-21 shows that propellant requirements are cut in half if the OMV returns without a payload as opposed to half a payload.

Therefore, the bas .ine OMV (using N_2O_4/MMH as propellants), can provide an inclination change of approximately 4 to 14 deg to an orbiting payload, depending on payload size and how much of the payload is brought back on the return trip.

There is another class of missions in the immediate vicinity of the Space Station for which even the baseline OMV has excess capability. These missions include: (1) deploying, servicing, and retrieving free-flyers that are within a few kilometers of the station; (2) transferring payloads to and from a co-orbiting but undocked Orbiter; (3) supporting construction operations; and (4) retrieving hardware or rescuing personnel in the vicinity of the station. These missions require less than 100 to 200 m/s delta-V and the required propellant load is less than 10% of the inert mass plus payload mass. A separate servicing vehicle may not have to be developed if the OMV can be adapted for these functions.

4.2.3.4 OTV Servicing Capability

The OTV is capable of a high delta-V and, therefore, is not well suited for servicing free-flyers that reside in the vicinity of the Space Station. The OTV is best suited for servicing free-flyers with large differences in inclination, altitude, or a combination thereof. The OTV is also useful for other missions, such as placing satellites in GEO or on interplanetary trajectories. The relative OMV and OTV capabilities are compared in figure 4-22.

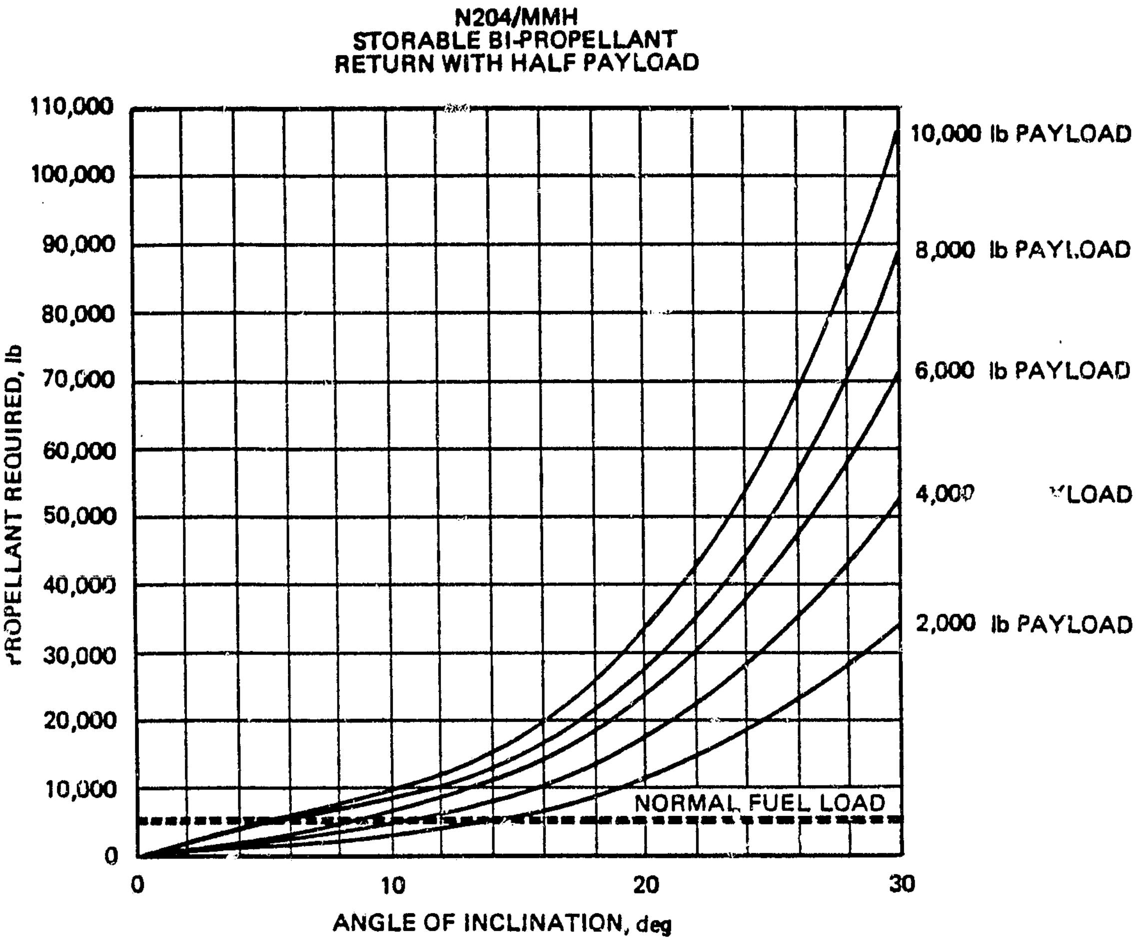

Figure 4-20. OMV Inclination Change Using N_2O_4/MMH Return With Half Payload

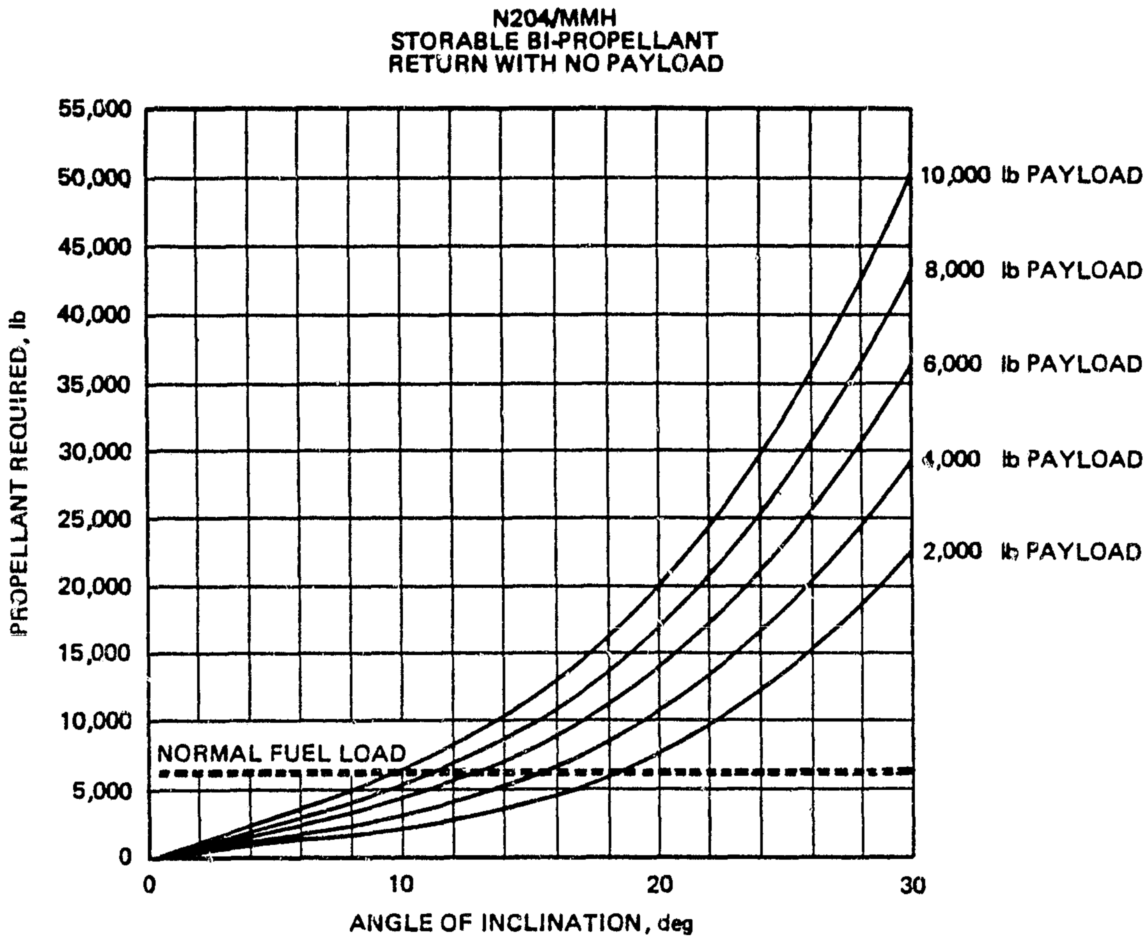

Figure 4-21. OMV Inclination Change Using N_2O_4/MMH Return With No Payload

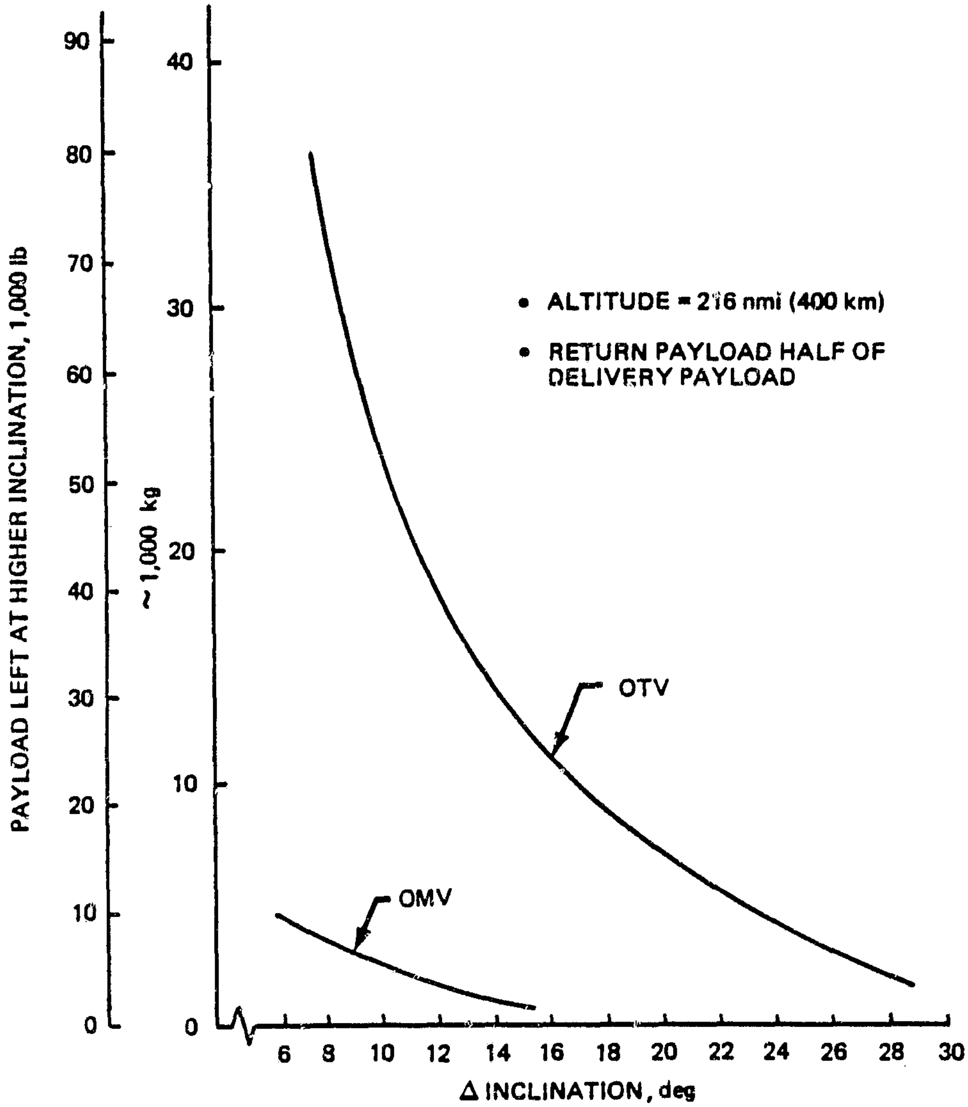

Figure 4-22. OMV and OTV Plane-Change Capability

When it becomes available, the manned, space-based OTV will greatly extend the range of access for LEO free-flyer servicing. Free-flyers in orbits of significantly different inclination and altitude than the Space Station, including GEO orbits, will then be accessible for servicing. Staging OTV service operations from the Space Station with a manned OTV will reduce the number and complexity of STS flights required. This is especially true where multiple-flight missions would otherwise be needed; space-basing makes OTV operations distinct, separating them from STS operations.

4.2.4 Servicing Operat s Missions Analysis

The mission analysis presented in this section is based on the NASA Mission Model and a software package that was developed under BAC IR&D to analyze the complex inter-relationships between Space Station operations, Space Station payloads, and the Space Transportation System. The original analysis was developed based on two studies that Boeing performed for NASA: The Space Operations Center (SOC) study for JSC and the Space Station Needs, Attributes, and Architectural Options (SSNAAO) studies done for NASA Headquarters. These studies revealed that each payload and operation needed to be analyzed both individually and as a yearly group to determine such factors as STS manifesting. The software package developed for this purpose, which is shown in figure 4-23, operates on a payload-by-payload and year-by-year basis. This software was used in this study to develop a servicing sensitivity analysis.

4.2.4.1 Mission Analysis Software Overview

A number of screening procedures were used in the mission analysis software to determine payload manifesting for the STS. In addition, numerous resource analyses were performed, including propulsion, power, manpower, and skill requirements. The program generates a highly detailed document style output comprising the recommended STS manifesting and the results of various analyses on a year-by-year basis. The software operates from three data input files: (1) payload description data, (2) traffic model, and (3) transportation-vehicle parameters.

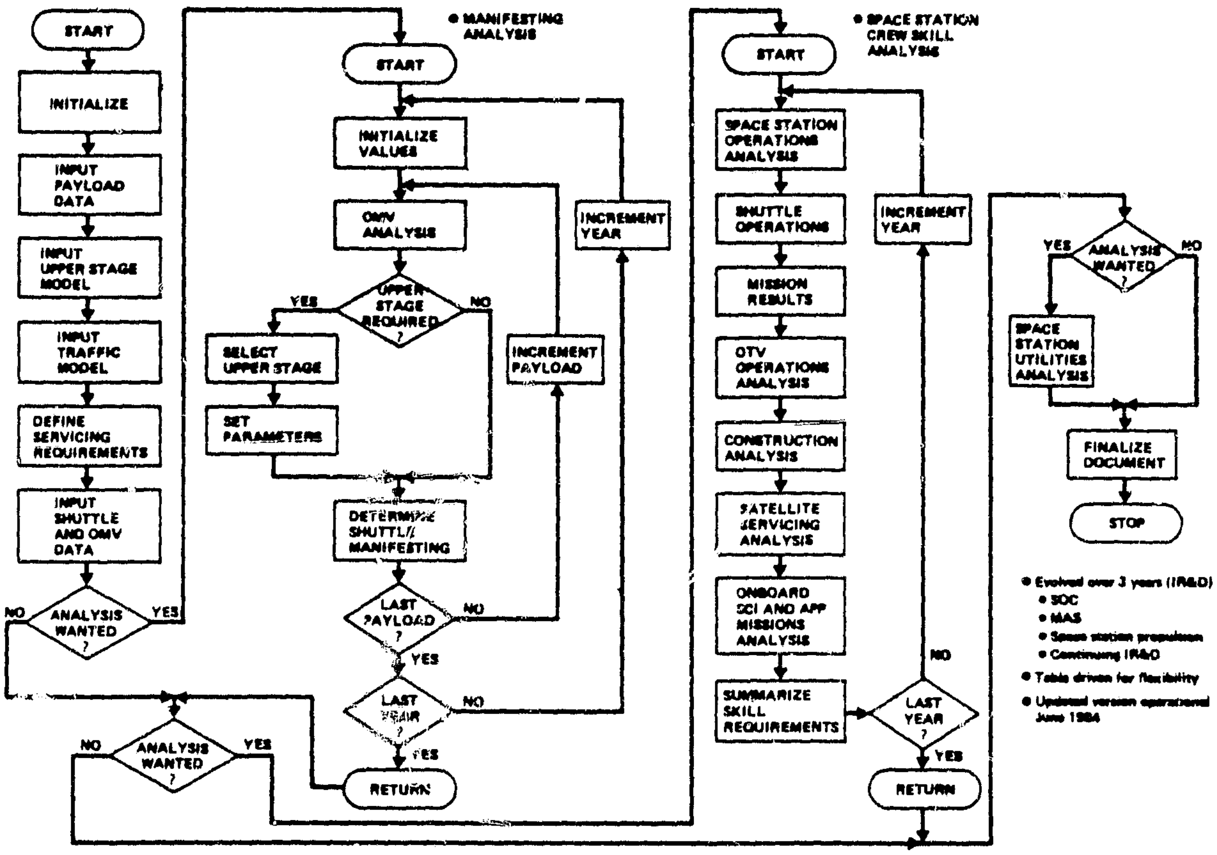

Figure 4-23. Mission Analysis Software

A manifesting of the STS is determined through a series of rules, such as priorities for pairing payloads with each other, manifesting restrictions, and physical characteristics. The output includes the following on a year-by-year basis:

a. Flight-by-flight payload manifest.
b. Number of shuttle flights.
c. Number of OMV flights.
d. Number of self-propelled free-flyer servicing operations.
e. Quantity of OMV and free-flyer propellant used.
f. Number of OTV's expended.
g. Number of OTV's reused.
h. Quantity of OTV propellant required.
i. Number of Space Station flight servicing operations.
j. Number of Space Station construction operations.
k. Total Orbiter fleet time.
l. Ideal minimum fleet size.
m. Peak power requirements.
n. Average payload power requirements.

One of the major jobs of this task was to determine the amount of propellant required to deliver and service the many payloads in the model.

4.2.4.2 Mission Model

The majority of data used in the payload parameter and traffic models came directly from the NASA mission requirements working group (MRWG) mission model. The MRWG has three discipline panels: science and applications, commercial utilization, and technology development, which are each responsible for interfacing with their respective user communities to analyze and assemble potential missions that could be supported by a Space Station system.

Some modifications were made to the mission model during the creation of the data base. Additions were made to the MRWG missionset data to fill in gaps left by data that wasn't available. The data used were taken from best estimates available to fill in the missing data points with a cutoff date of December 1983. The orbital altitude of many of the payloads from the MRWG model was standardized at 525 km to match the baseline Space Station orbit. Also, some data points thought to be incorrect were assigned more reasonable values, such as payloads that were longer than the STS payload bay. Additionally, numbers were corrected, such as the mass of the large deployable reflector (NASA code SAAX020), which was listed as having a mass of 55,000 kg, while all other available information on that particular payload gave the mass as 27,000 kg.

The manifesting role the program uses for OMV and OTV propellant resupply is as follows: the resupply tank occupies 1.5m of payload bay length and the entire diameter of 4.5m. The resupply tank filled weight is 4 tonnes with 2.5 tonnes being propellant weight.

The decision as to whether a free-flyer is serviced in-situ or returned to the station was based on the servicing time required, as specified by the MRWG. It was assumed that free-flyers requiring more than 10 hours of servicing would be returned to the station.

Because all the projected payloads have not been identified as free-flyers or station-attached, three general scenarios were used as a basis for establishing propulsion requirements: a high case that assumed identified payloads would be free-flyers; a low case that assumed all would be attached; and a nominal case that assumed payloads would be distributed between the two.

The STS, OMV, and OTV capabilities and operating parameters were set to agree with those used for other portions of this study.

The results of the mission analysis are presented in figures 4-24 through 4-30. The STS, OMV, and OTV usage for the various years reflect payload deployments as specified in the MRWG mission model.

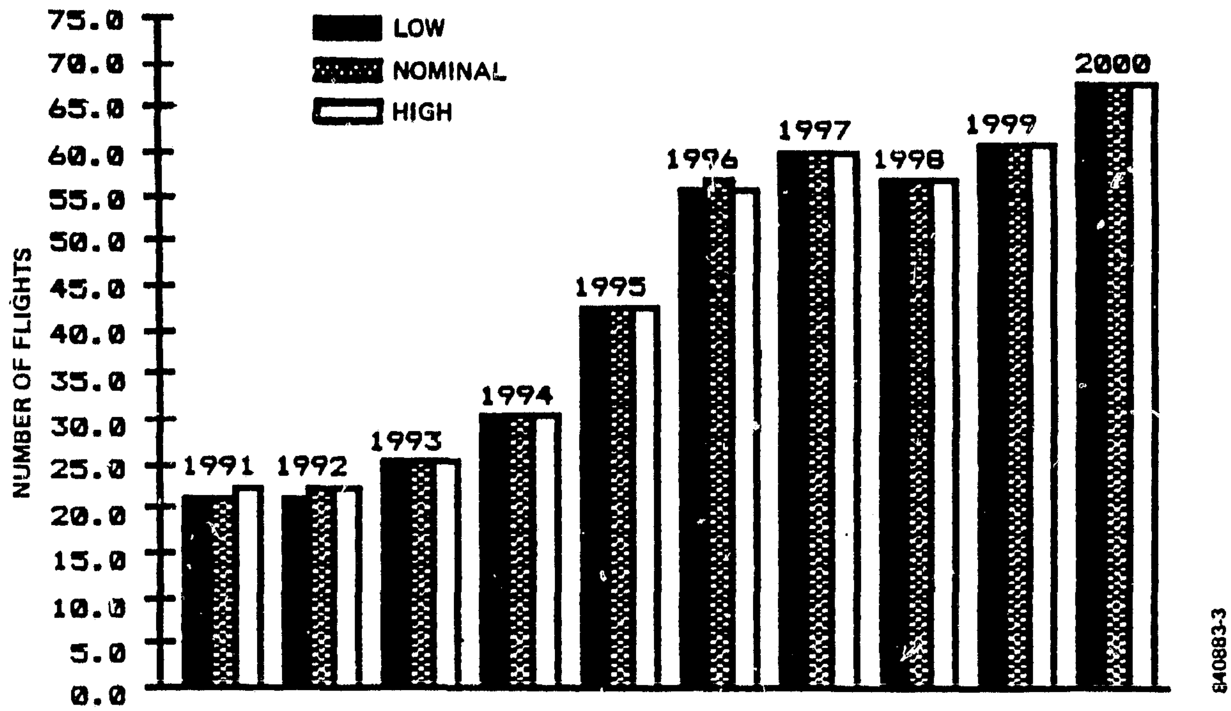

Figure 4-24. STS Flights Per Year

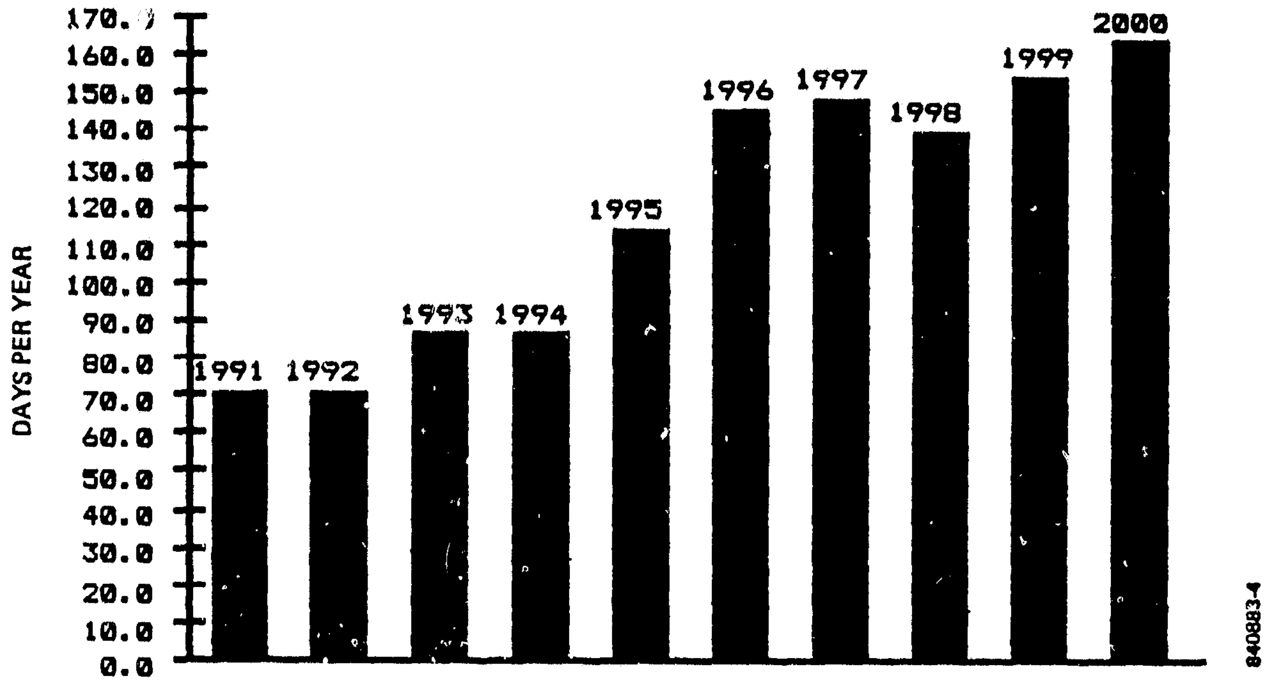

Figure 4-25. Orbiter Fleet Time On Orbit

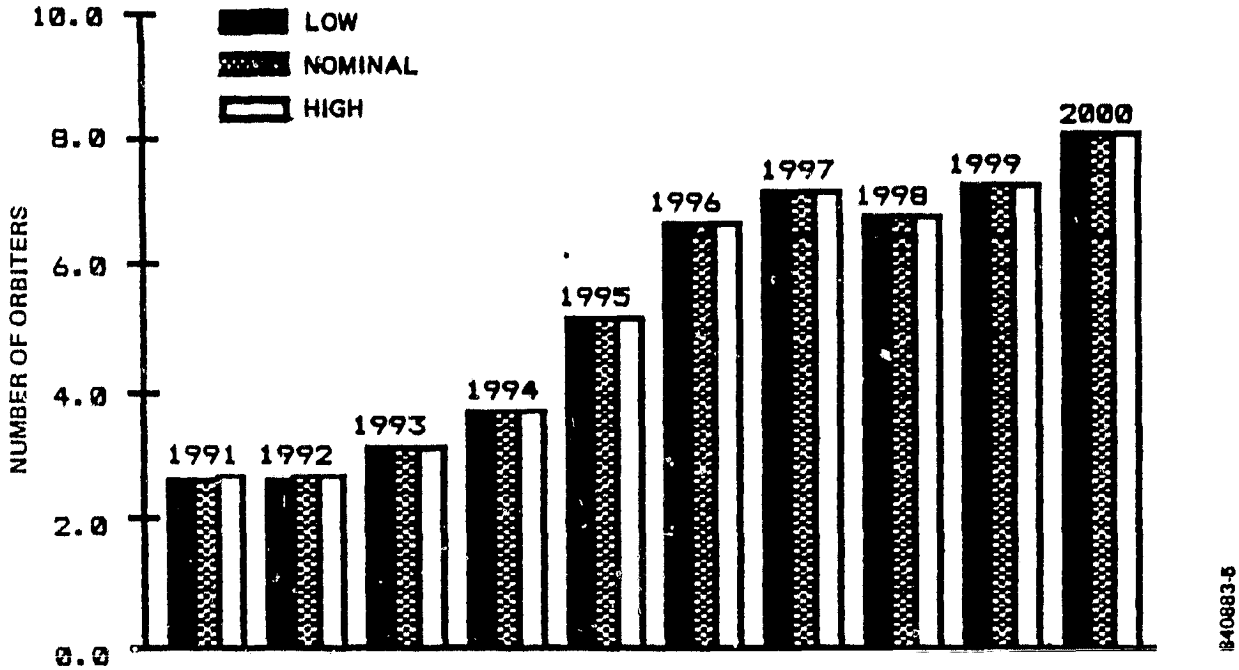

Figure 4-26. Minimum Orbiter Fleet Size

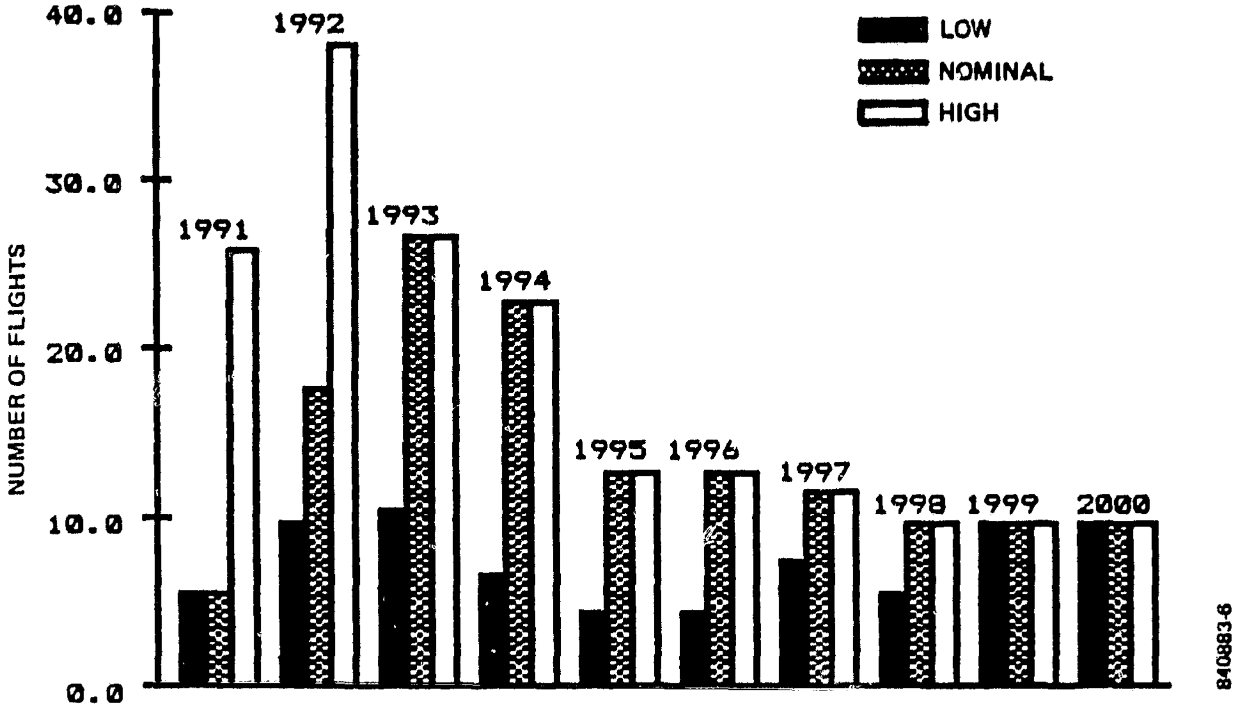

Figure 4-27. OMV Flights Per Year

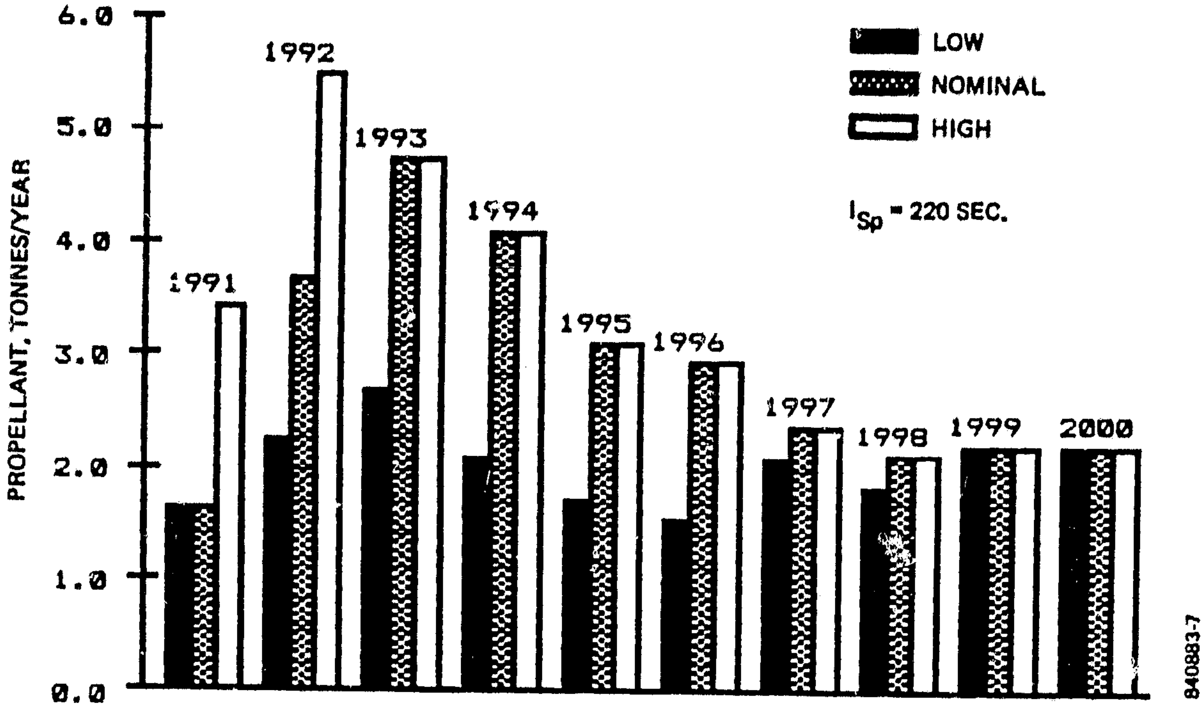

Figure 4-28. OMV Propellant Required

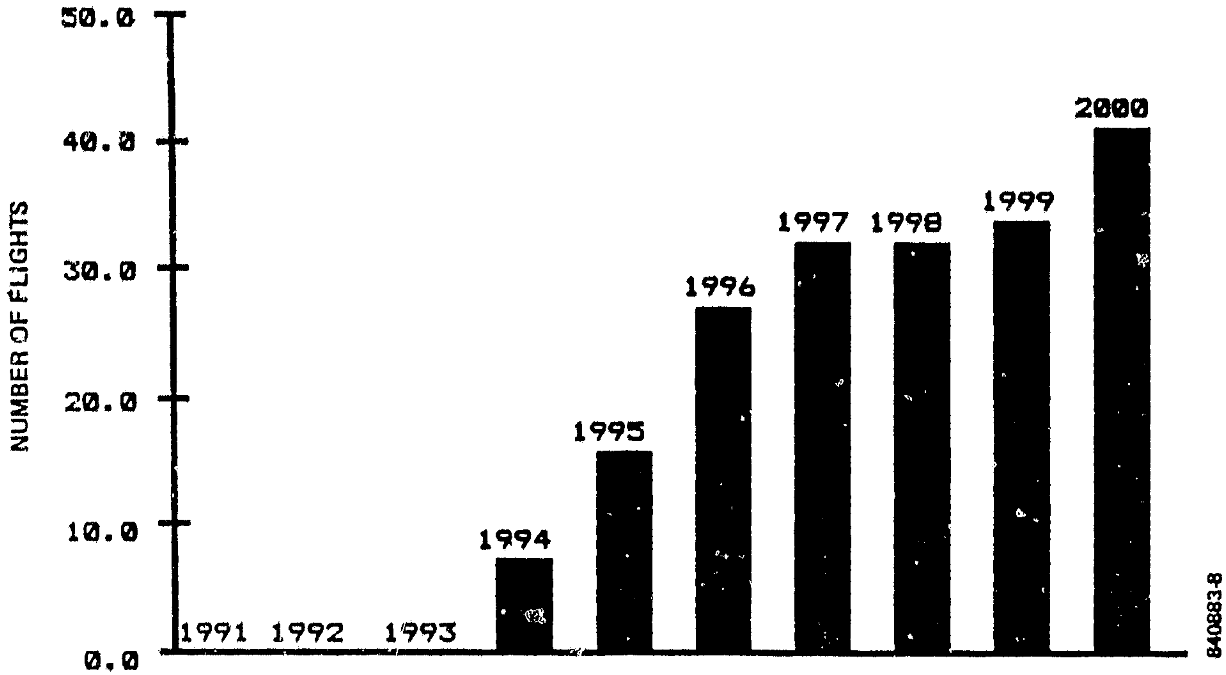

Figure 4-29. OTV Flights Per Year

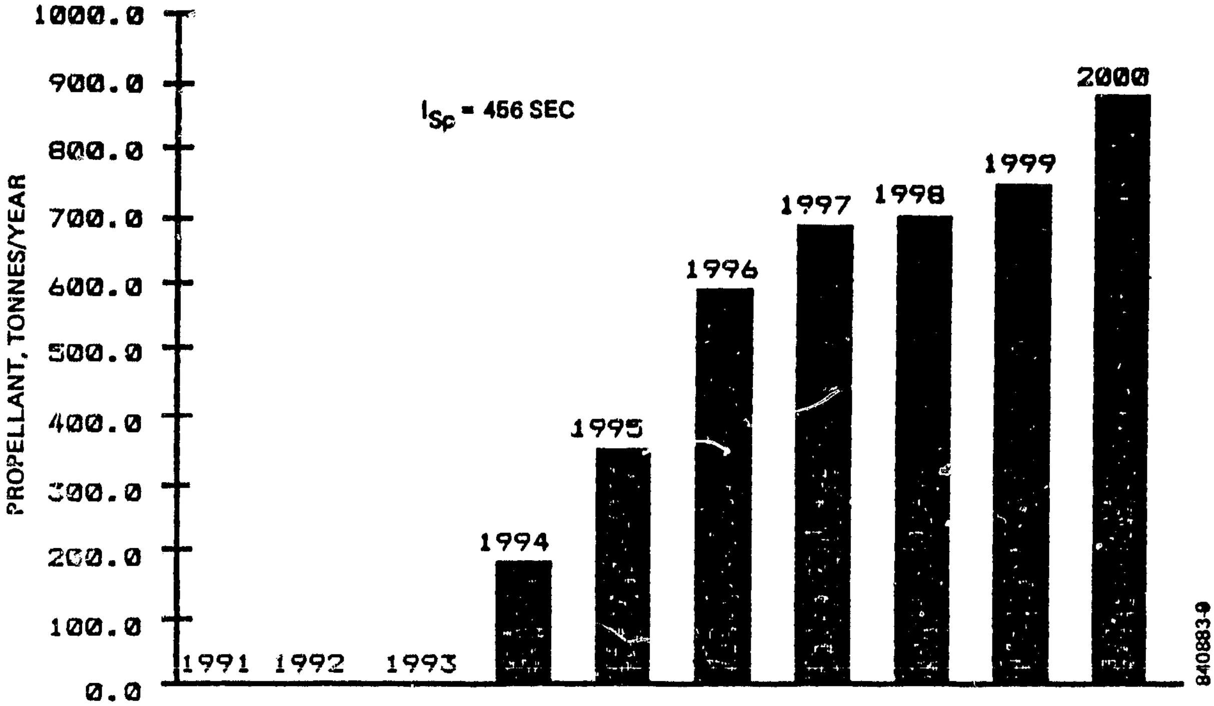

Figure 4-30. OTV Propellant Required

C-3

Figures 4-27 and 4-28 show what the projected number of OMV flights and propellant requirements are, based on the fact that all free-flyer servicing will be done by OMV's. The units of propellant are tonnes, where 1 tonne equals 1,000 kg. The OMV was assumed to have a specific impulse of 340 sec (representative of an N_2O_4/MMH propulsion system).

The OTV is used to place payloads that require a large delta-V, i.e., those requiring transfer to a much higher altitude than the station or to a significantly different inclination. OTV flights, by year, are shown in figure 4-29 and the propellant requirements are shown in figure 4-30. The OTV-specific impulse was assumed to be 456 sec for LO_2LH_2. Again, the OTV requirements shown here are driven solely by payload demands and do not necessarily relate to NASA's plans for OTV procurement.

Figures 4-31 and 4-32 show the projected number of payloads that will be deployed by year and how they will be distributed between the Space Station, free-flyers, and platforms.

4.2.5 Free-Flyer Servicing Summary

Whenever practical, all co-orbiting free-flyers should be returned to the Space Station for maintenance and resupply. Free-flyers that are too large to either be transferred by a servicing vehicle or serviced within the Space Station can be serviced in-situ. Figure 4-33 shows the LEO servicing regions for the space-based vehicles. Free-flyers that could be serviced by the Space Station fall within an area that is bounded by OTV core stage capabilities for half-range and maximum-range payload retrieval performance when limited to one STS propellant delivery flight. For example, the OTV half-range retrieval capability defines the maximum plane-change maneuver for bringing a free-flyer back to the Space Station for servicing and then returning it to its original orbit. Free-flyers beyond the OTV half-range capability can also be returned to the Space Station for servicing if needed; however, it would be more economical if they were serviced in-situ. As shown in figure 4-37, an OTV can provide in-situ servicing to a momentum

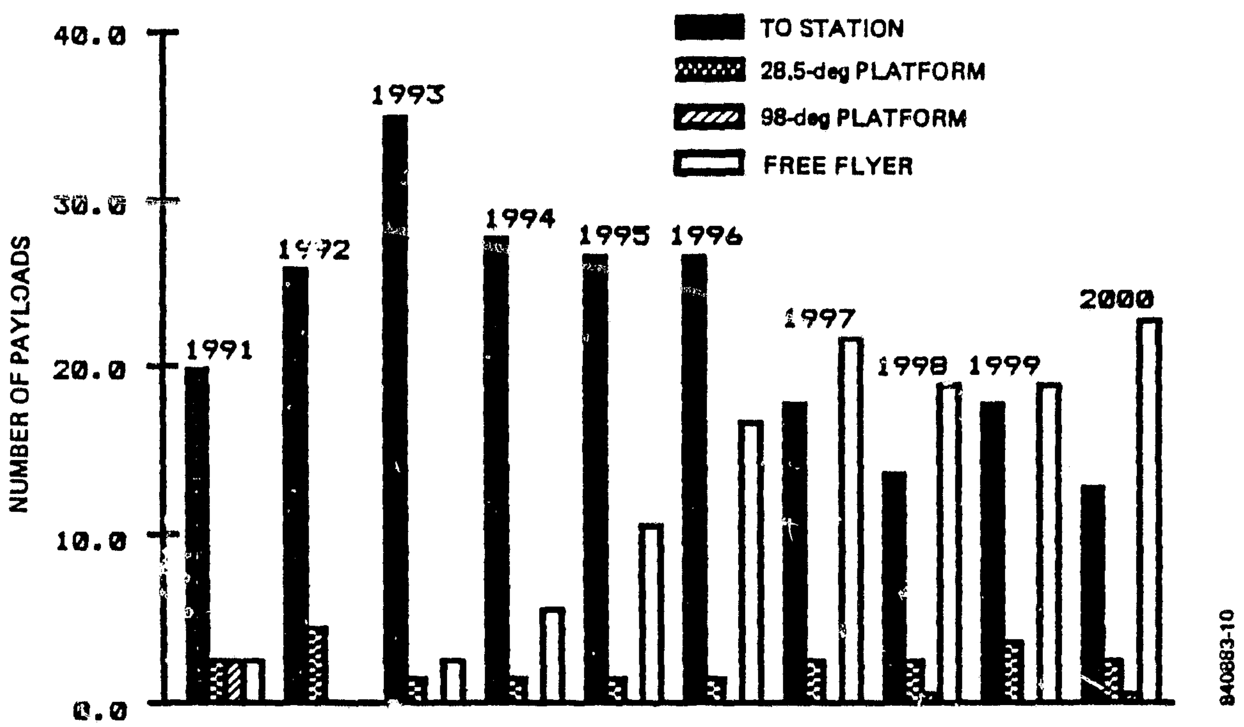

Figure 4-31. Payloads Delivered by Type Per Year

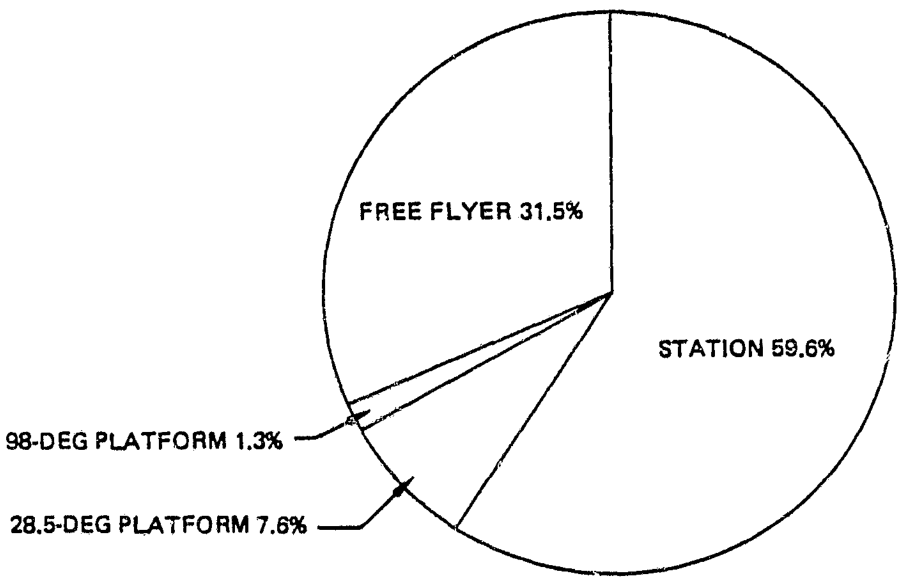

Figure 4-32. Total Payloads Delivered by Type 1991-2000

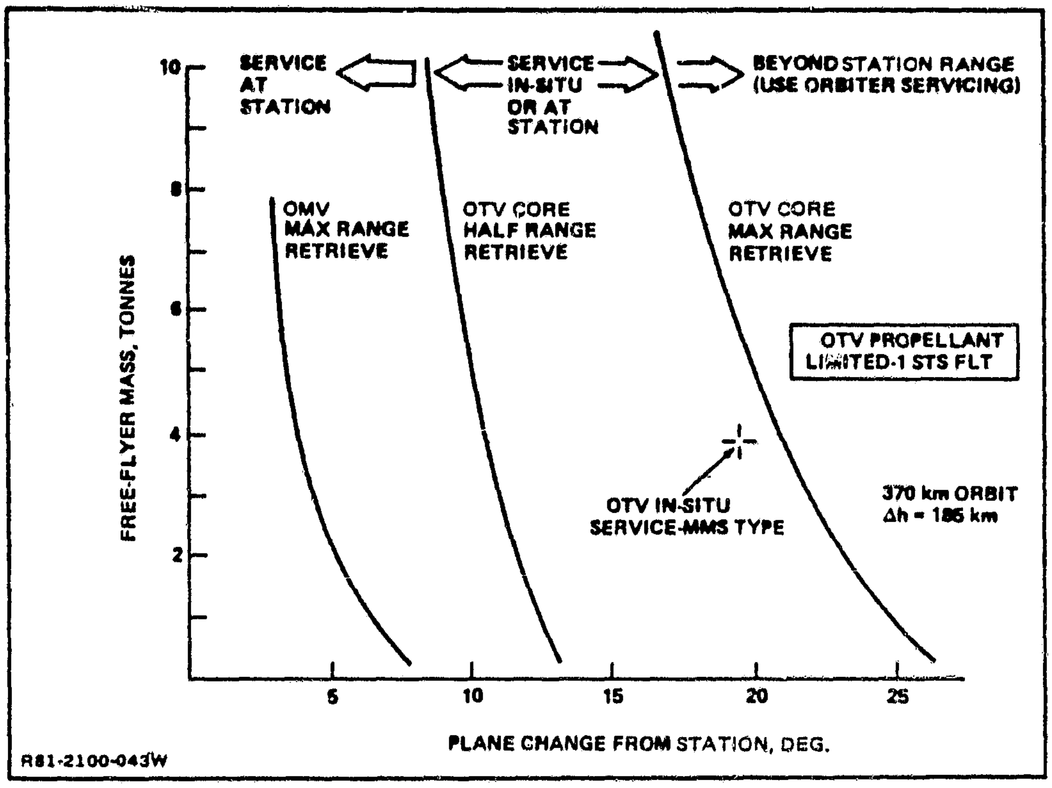

Figure 4-33. LEO Free-Flyer Servicing Regions for OMV and OTV

management system (MMS)-class free-flyer in a 185-km higher orbit that is also 20 deg out of plane with respect to the Space Station. The maximum payload retrieval angle of the OMV is also shown for comparison.

4.3 Servicing Strategy Summary

Propulsively moving the station to and from higher and lower orbits to receive payloads is less efficient than using the STS or servicing vehicles. The potential problems associated with maintaining the free-flyer constellation, and the added complexity of simultaneously accomplishing an STS launch and station deboost, led to eliminating variable-altitude servicing strategies in the remainder of this study. The strategy in which the station's orbit decays because of aerodynamic drag to effect STS rendezvous was not carried further because of the aforementioned free-flyer constellation problem and the intractability of atmospheric drag prediction. In view of the long-range effort that goes into a launch, any attempt to make full use of the STS capability and yet hit an ill-defined "window" would make an already difficult job impossible. The strategy of servicing a 90-deg station from a 28.5 deg station using a servicing vehicle is eliminated because neither the OMV or OTV is capable of servicing a station at a 90-deg inclination from another at 28.5 deg and return. Therefore, the Space Station strategy selected for examination is a high-altitude orbit accessible by the STS through direct insertion or with an STS/servicing-vehicle combination.

The next section evaluates propulsion and propellant requirements for normal and emergency operations for the Space Station and free-flyers. Station servicing strategies are compared and one is selected to serve as the baseline for the propulsion system analysis in section 6.0.

5.0 PROPULSION REQUIREMENTS

The Space Station and free-flyer propulsion requirements depend on the configuration design, deployment altitude, and the servicing strategy that is chosen. With the proper configuration treatment, attitude control can be managed non-propulsively by using CMG's and torque rods. Therefore, the primary task of the propulsion system is orbit maintenance.* Secondary tasks are emergency propulsion, CMG backup, and CMG desaturation backup. Both Space Station and free-flyer propulsion requirements are developed in this section.

5.1 Space Station Propulsion and Propellant Requirements

Propellant requirements over a given time interval are a direct function of the impulse requirements over that interval and are inversely proportional to the specific impulse of the propulsion system. This can be expressed in the following relationship:

$$\text{Propellant Mass} = \frac{\text{Mission Impulse}}{\text{Specific Impulse}}$$

Figure 5-1 illustrates the relationship between mission impulse and propellant mass for a variety of specific impulses applicable to the Space Station for a 10-year mission. The impulse imparted to the station can be for (a) orbit maintennce, (b) torque cancellation and/or (c) CMG desaturation, and (d) emergency situations.

*There is a scenario that involves raising the station's altitude by lowering the Orbiter on a tether after a servicing mission. Releasing the Orbiter then leaves the station at a higher altitude and the Orbiter at a lower altitude, which facilitates its re-entry. However, since this study evaluates only propulsive means for orbit maintenance, this approach is beyond the scope of this study.

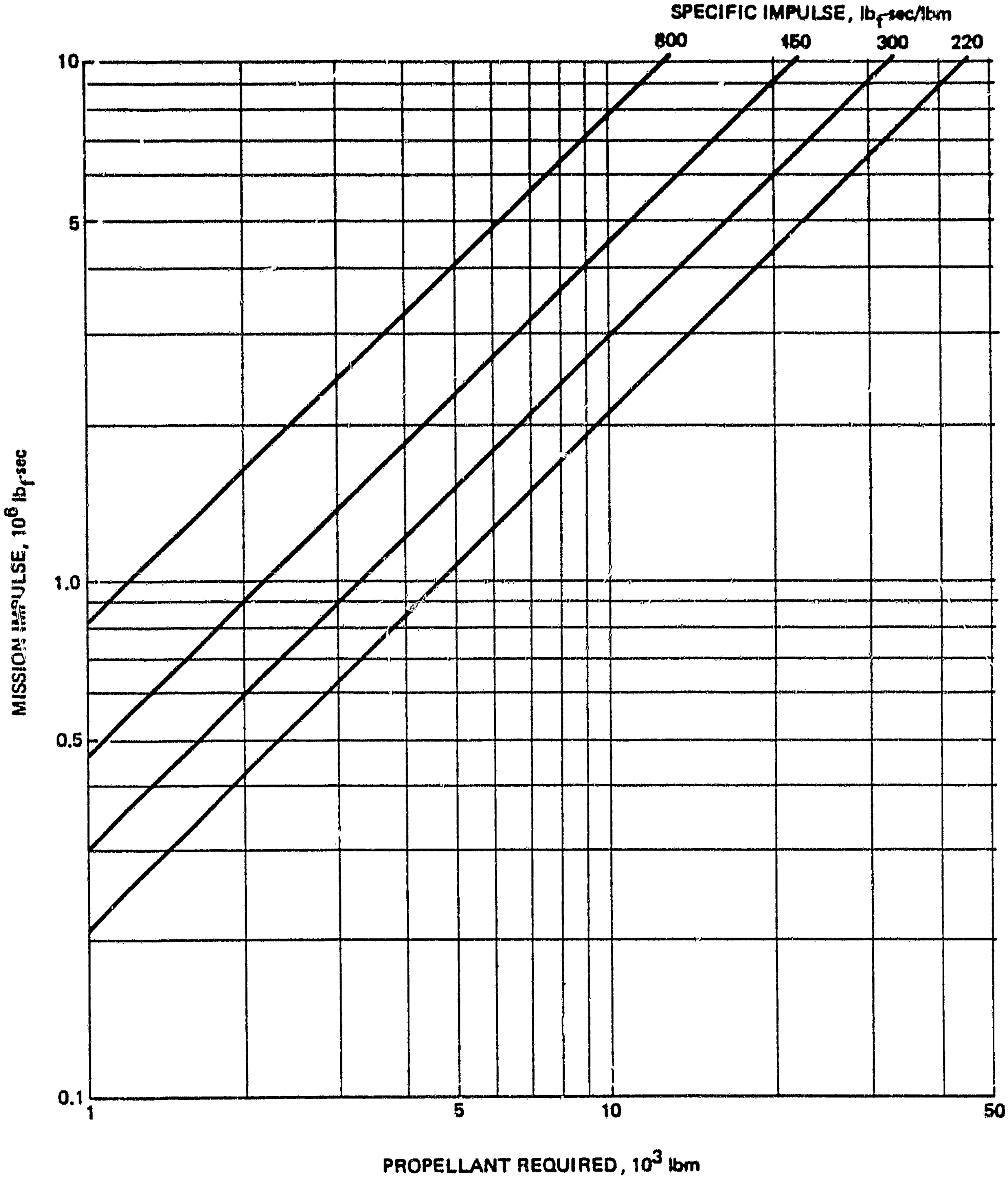

Figure 5-1. Mission Impulse Effects on Propellant Requirements

5.1.1 Orbit Maintenance Propulsion Requirements

Orbit maintenance is required to counter the aerodynamic drag experienced by the station. Otherwise, the drag will cause a gradual loss in station altitude until re-entry finally occurs. Orbit maintenance can be accomplished in a variety of ways, depending on thrust level, thrust duration, thrust frequency, and permissible orbit decay. Figure 5-2 shows various orbit maintenance scenarios based on a 525 km altitude and neutral atmosphere. In this figure, each of the three station sizes represents a specific value for total mission impulse requirements. Each thrusting scenario must satisfy mission impulse requirements, i.e., the thrust level and thrust duration product yield the required mission impulse. The lower-most curve of figure 5-2 is the thrust level required if constant thrusting, or drag balancing, is desired. Thrust requirements for the nominal atmosphere may be determined by multiplying the data from figure 5-2 by 0.221. Likewise, the increased thrust level required for 400, 450 and 500 km orbits may be determined by multiplying the data by 5.56, 2.71, and 1.39, respectively.

Depending on the scenario selected, orbit maintenance thrust levels vary from a few hundredths of a pound to about 100 lb_f (Figure 5-2). Continuous thrusting that closely approximates the drag force minimizes valve cycling but maximizes thruster burn time; i.e., in one year, the burn time would exceed 8700 hours. There are a limited number of propulsion systems that could be used for such a burn time duration at the requisite low thrust level, and are discussed in section 6. However, if a slightly higher thrust level is selected, e.g., 0.1 lb_f, then the thrust duration is 30-minutes per orbit. This diminishes the burn time from the continuous burn situation by a factor of 3 but the valves must now cycle once per orbit, or about 5840 cycles each year. Therefore, although the higher thrust level reduced burn time, valve cycling increased.

5.1.2 Orbit Maintenance Propellant Requirements

Figures 5-3(a) through (d) show what the orbit maintenance propellant requirements are for the baseline stations over a 10-year interval for a

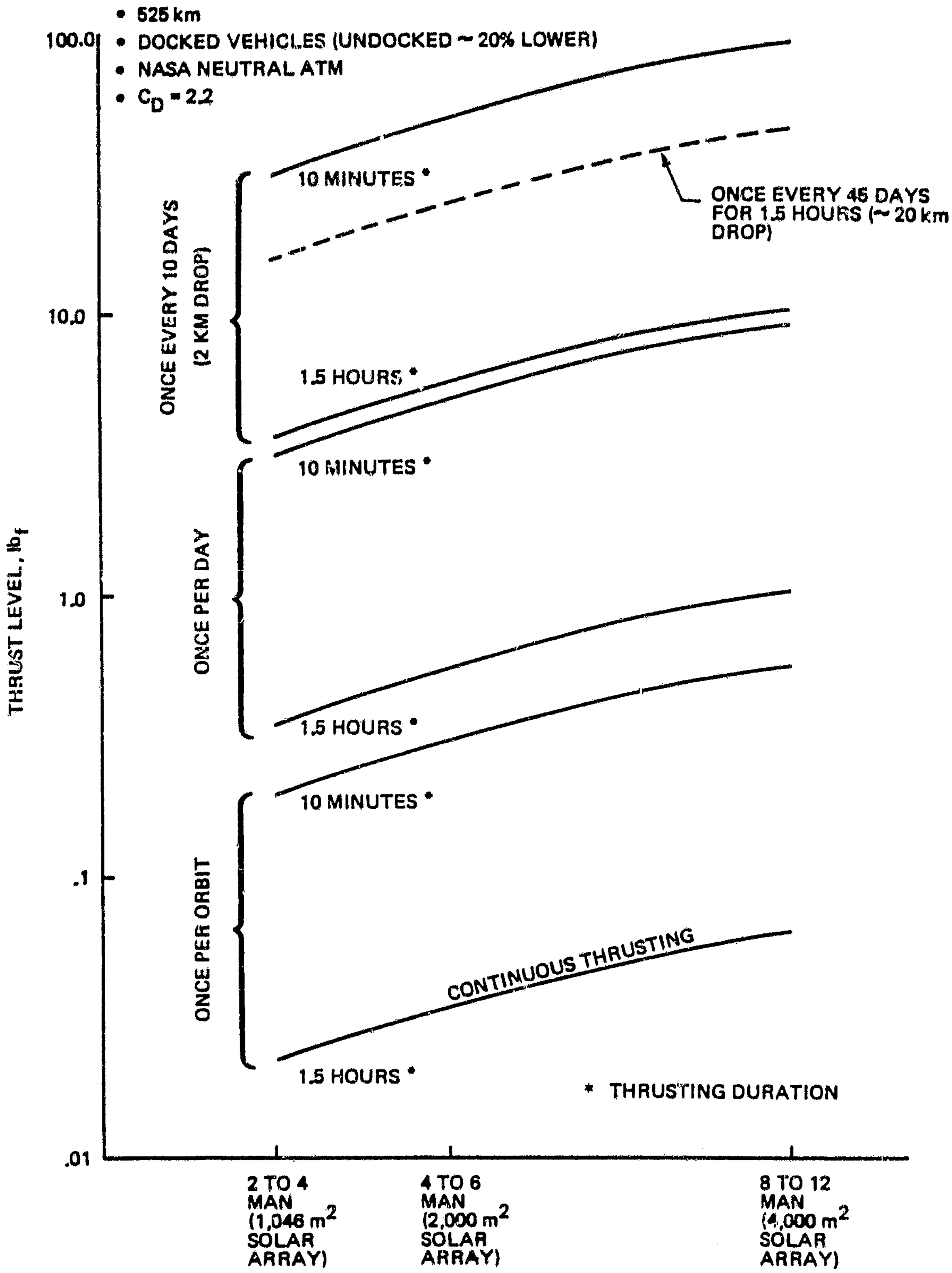

Figure 5-2. Orbit Maintenance Thrust Levels for Various Duty Cycles

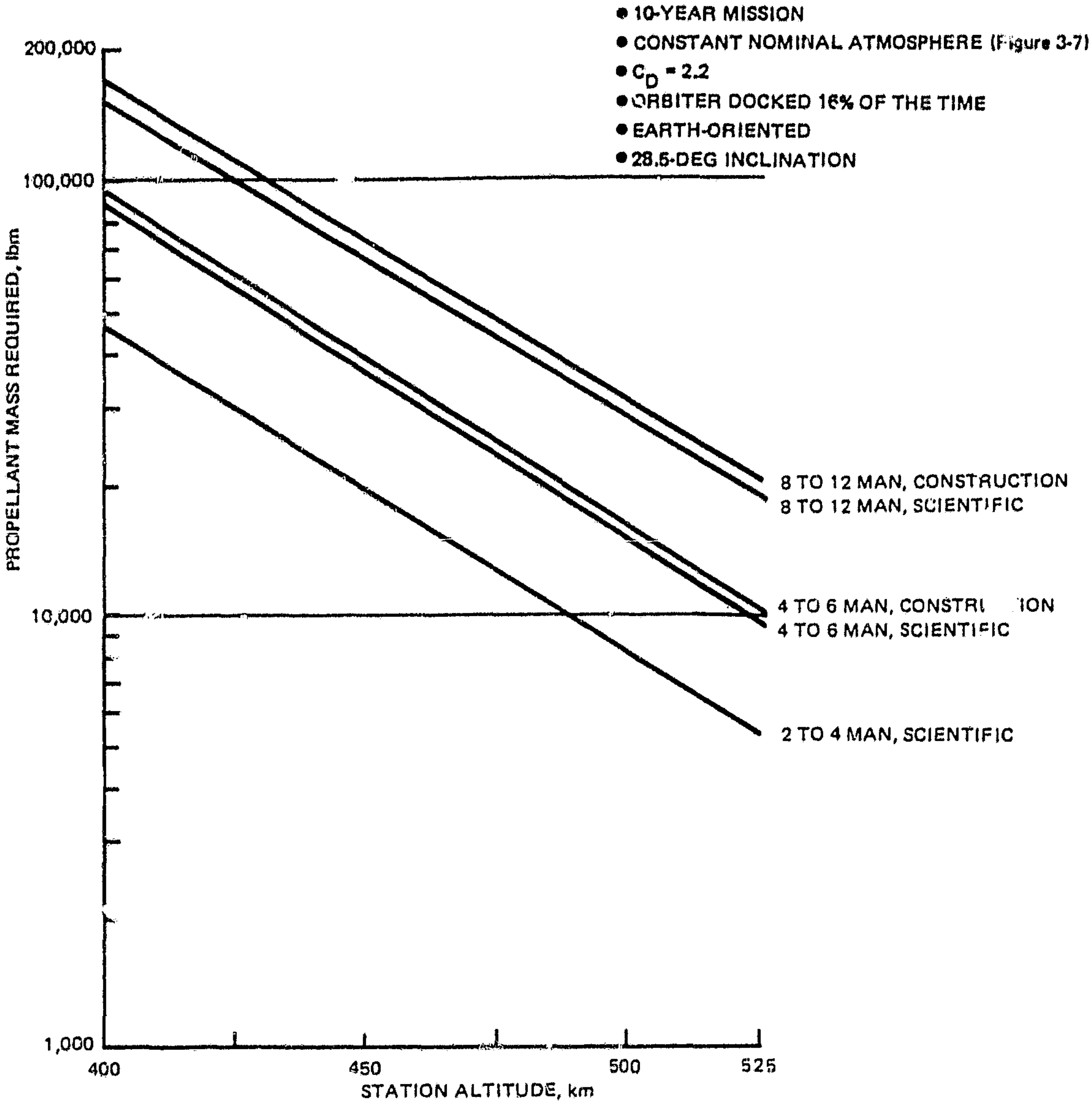

Figure 5-3(a). Propellant Requirements for the Baseline Stations With a 220-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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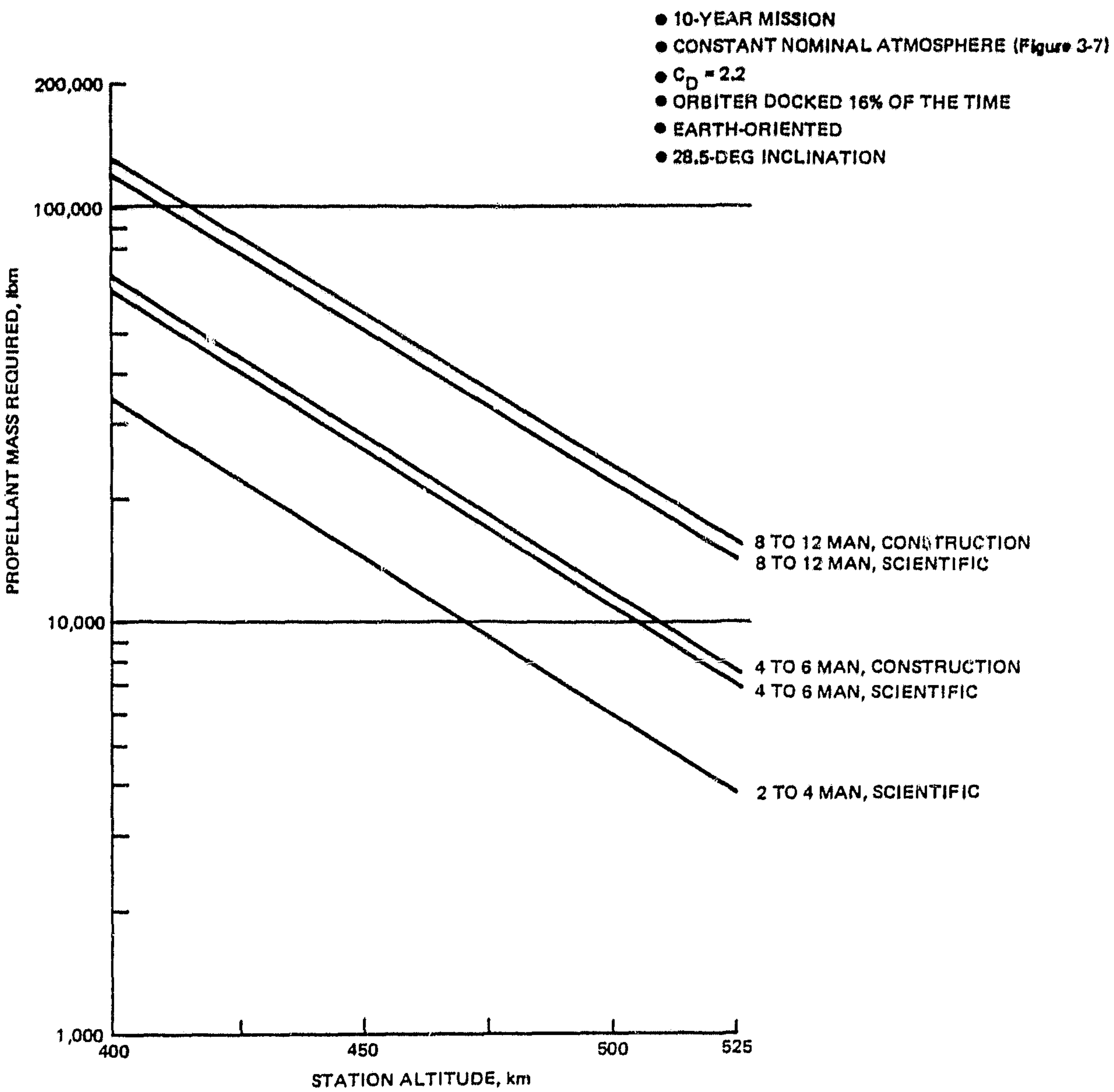

Figure 5-3(b). Propellant Requirements for the Baseline Stations With a 300-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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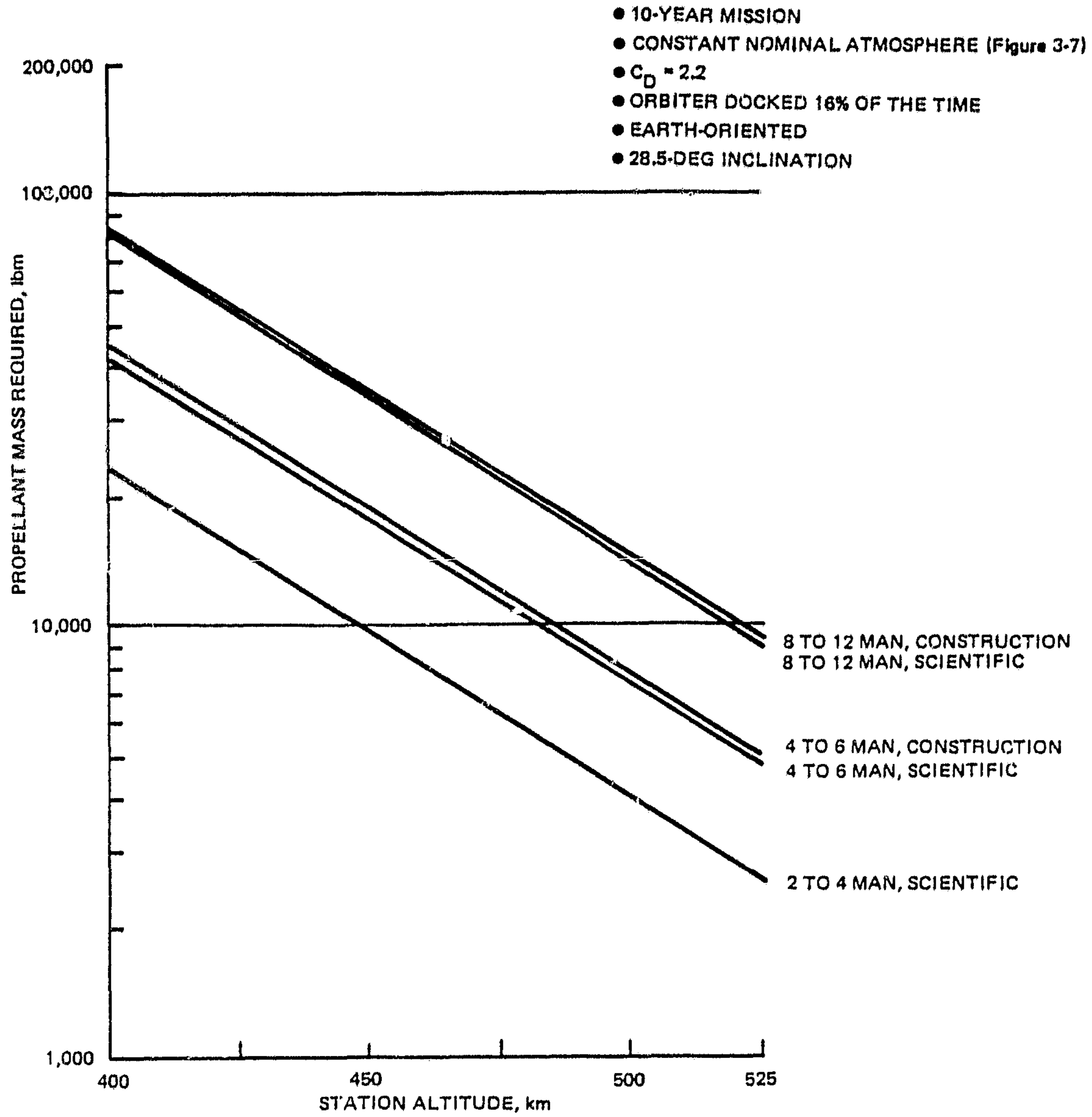

Figure 5-3(c). Propellant Requirements for the Baseline Stations With a 450-sec Specific Impulse

Altitude (km)	Multipliers to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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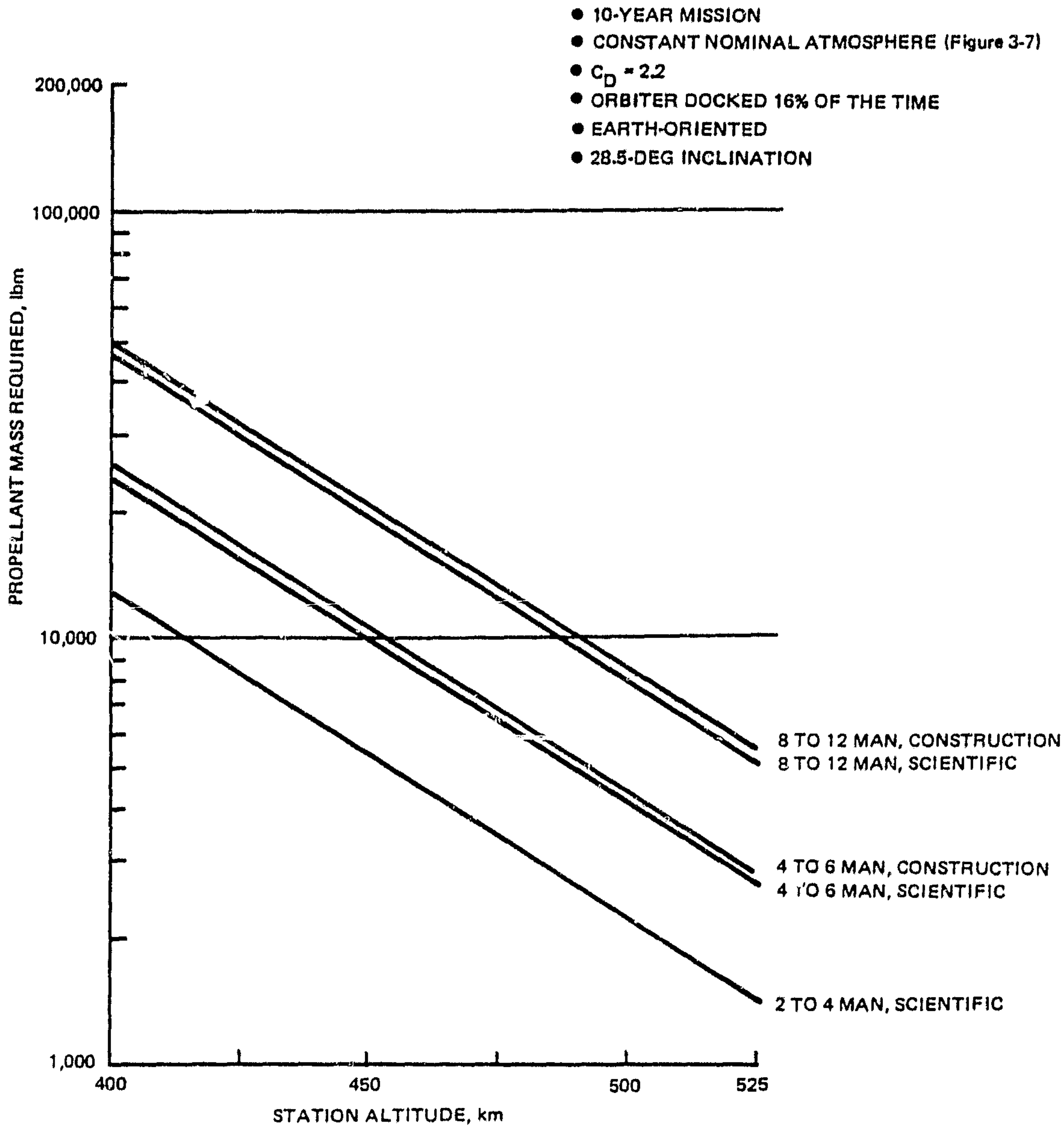

Figure 5-3(d). Propellant Requirements for the Baseline Stations With a 800-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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220, 300, 450, and 800 lb_f-sec/lbm specific impulse as a function of altitude. The figures assume constant altitude, i.e., more or less continuous orbit decay cancellation. Figures 5-4(a) through (d) show comparable data for the polar and Sun-synchronous 2- to 4- and 8- to 12-man stations. As indicated, these figures are for a nominal atmosphere; however the table at the bottom of the figures shows factors necessary for the conversions for minimal, neutral, and short-time maximum atmospheres. Based on figures 5-3(a) to (d), propellant requirements increase exponentially as altitude decreases. For example, the required propellant mass for a 4-man construction station with an I_{sp} of 450 lb_f-sec/lbm, at an altitude of 450 km is approximately 20,000 lbm. At 400 km, however, the same station with the same I_{sp} requires approximately 45,000 lbm of propellant. As figure 5-4(d) shows, the station that requires the least amount of propellant (206 lbm) is inertial, Sun-synchronous, 2- to 4-man crew, with an I_{sp} of 800 lb_f-sec/lbm, and at an altitude of 525 km. The station that requires the largest amount of propellant (168,360 lbm) is Earth-oriented, 28.5 deg inclination, 8- to 12-man crew, construction, with an I_{sp} of 220 lb_f-sec/lbm and at an altitude of 400 km (see figure 5-3(a)).

5.2 Attitude Control Propulsion and Propellant Requirements

Attitude control of the Space Station is a complex issue and may involve the use of momentum management devices (MMD), torque rods, and thrusters. A complete treatment of this subject is beyond the scope of the current study, but a few general observations can be made to permit an assessment of attitude control impact on propulsion requirements.

The general assumptions made in this assessment are:

a. Since a cost analysis is not included in this study, the measure of "worth" for a given approach is conservation of propellant and, to a lesser extent, safety, reliability, and maintainability.

b. The station configuration (e.g., with balanced solar arrays and Orbiter docked in either a neutral gravity or gravity-gradient-stable location and orientation) is designed to produce minimal

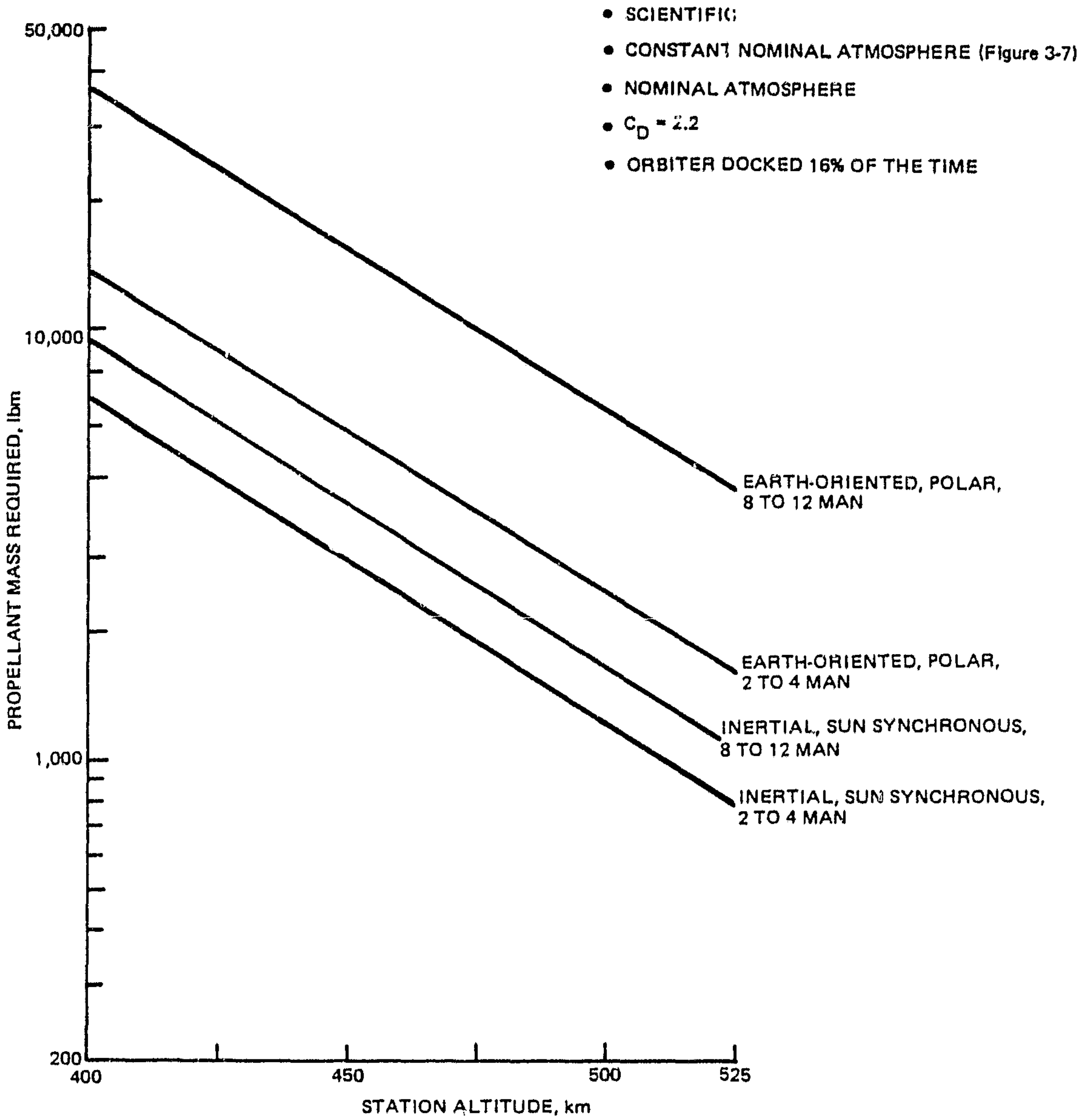

Figure 5-4(a). Propellant Requirements for the Polar and Sun-Synchronous Stations With a 220-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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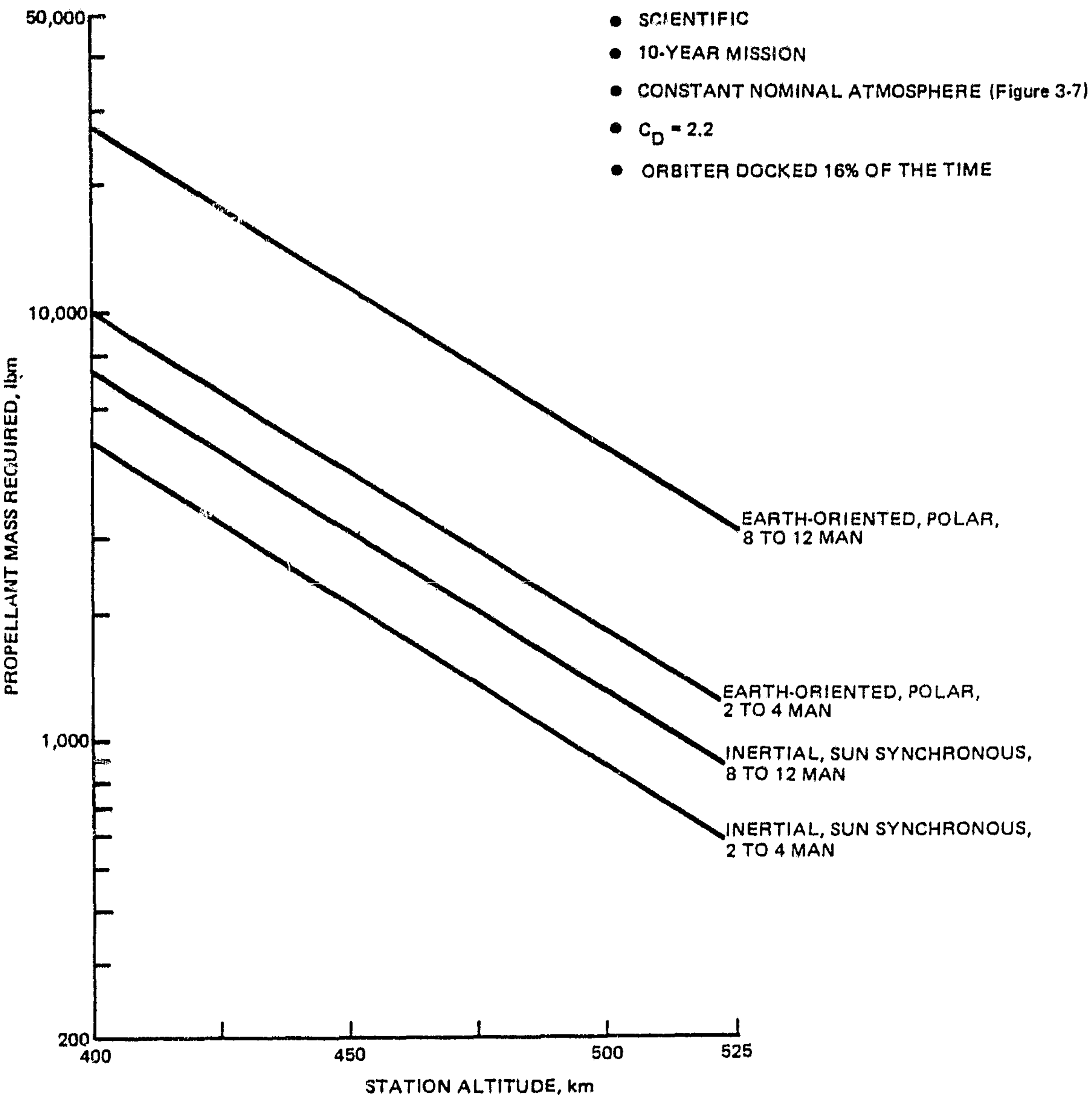

Figure 5-4(b). Propellant Requirements for the Polar and Sun-Synchronous Stations With a 300-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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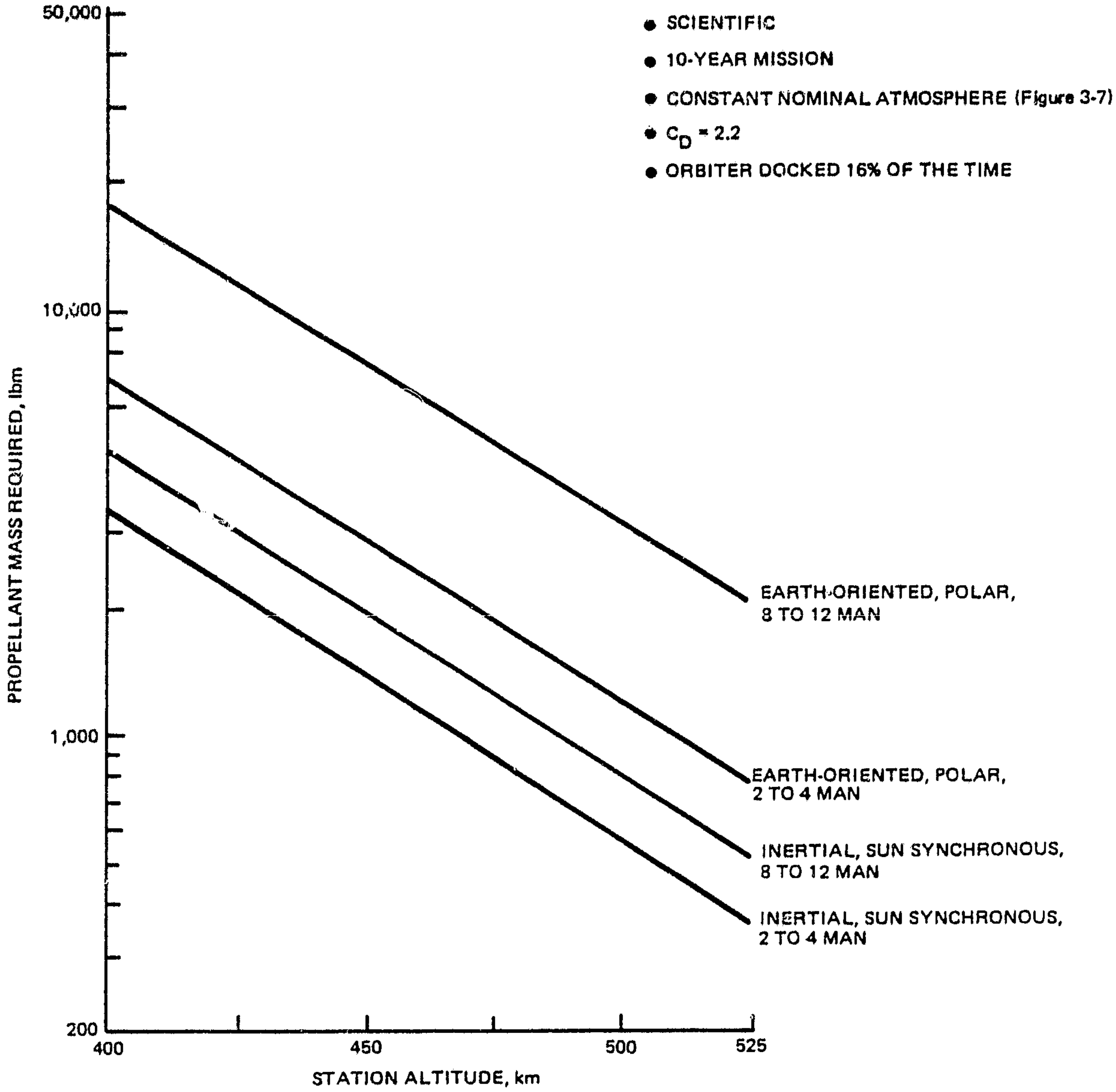

Figure 5-4(c). Propellant Requirements for the Polar and Sun-Synchronous Stations With a 450-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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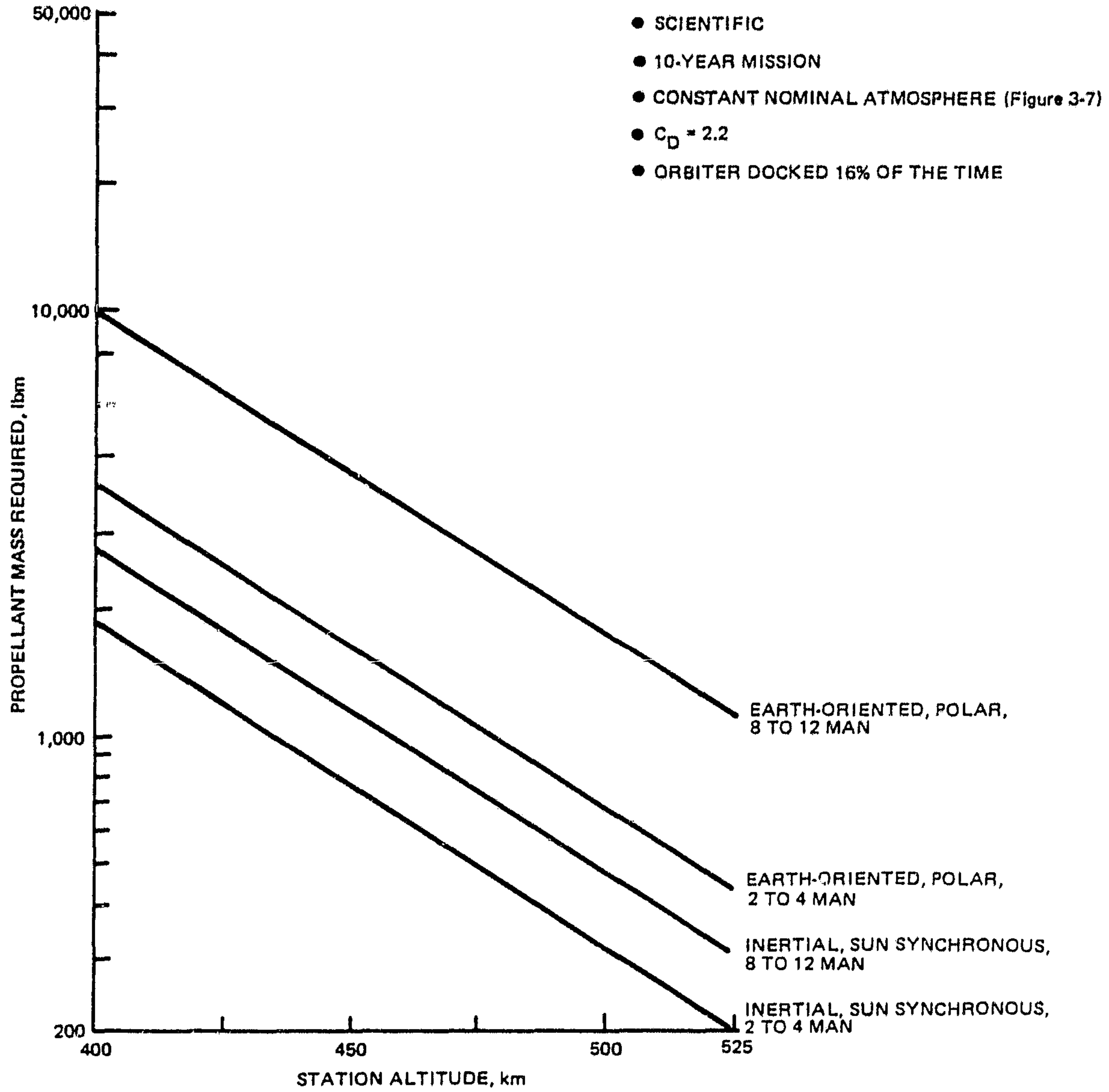

Figure 5-4(d). Propellant Requirements for the Polar and Sun-Synchronous Stations With a 800-sec Specific Impulse

Altitude (km)	Multiplier to convert nominal atmosphere to:		
	Minimum	Neutral	Short-time max
400	.2500	2.857	7.143
450	.1843	3.250	9.165
500	.1600	4.000	10.474
525	.1584	4.525	16.425

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aero and gravity-gradient torques and momenta which must be overcome to maintain the desired Earth-oriented attitude.

c. The attitude control system must minimize the impact on onboard experiments due to acceleration, contamination, and deviation from the desired orientation.

As section 3.0 illustrated, the major attitude control problems the station will encounter are due to aerodynamic and gravity-gradient torques. Several methods for countering these effects are discussed in this section.

Torque cancellation can be accomplished in four ways:

(1) The orbit maintenance thrust can be offset from the center-of-mass to produce a torque in the opposite direction of the aerodynamic and/or gravity-gradient torque. This approach, by necessity, requires that the thrusters provide an almost constant countering torque, which implies a very low, almost constant, orbit maintenance thrust. Figure 5-5 illustrates this concept. The disparity between CP and CM will vary, depending on experiment deployment, docked vehicles, the depletion of expendables, crew and hardware relocation, and other factors. If thrusters are used to simultaneously counter drag and torques, their moment arms (from the CM) must lie outside the station CM envelope. Referring to figure 5-5, if thruster 1 is fired, there will be a pitch down torque, a yaw right torque, and a thrust along the direction of motion. The yaw moment arm is 'a' and the pitch moment arm is 'b'. If only a yaw torque is desired, thrusters 1 and 4 must be fired simultaneously (assuming c = b and 1 and 4 have the same thrust level). It is apparent that an infinite number of pitch and yaw torque combinations can be achieved if duty cycles are also varied. Roll torques, on the other hand, cannot be overcome by using orbit maintenance thrust but require a couple.

(2) Providing a propulsive couple. This approach is commonly used by attitude control systems and can create a roll, pitch, or yaw torque by providing equal thrusts in opposite directions separated by a given

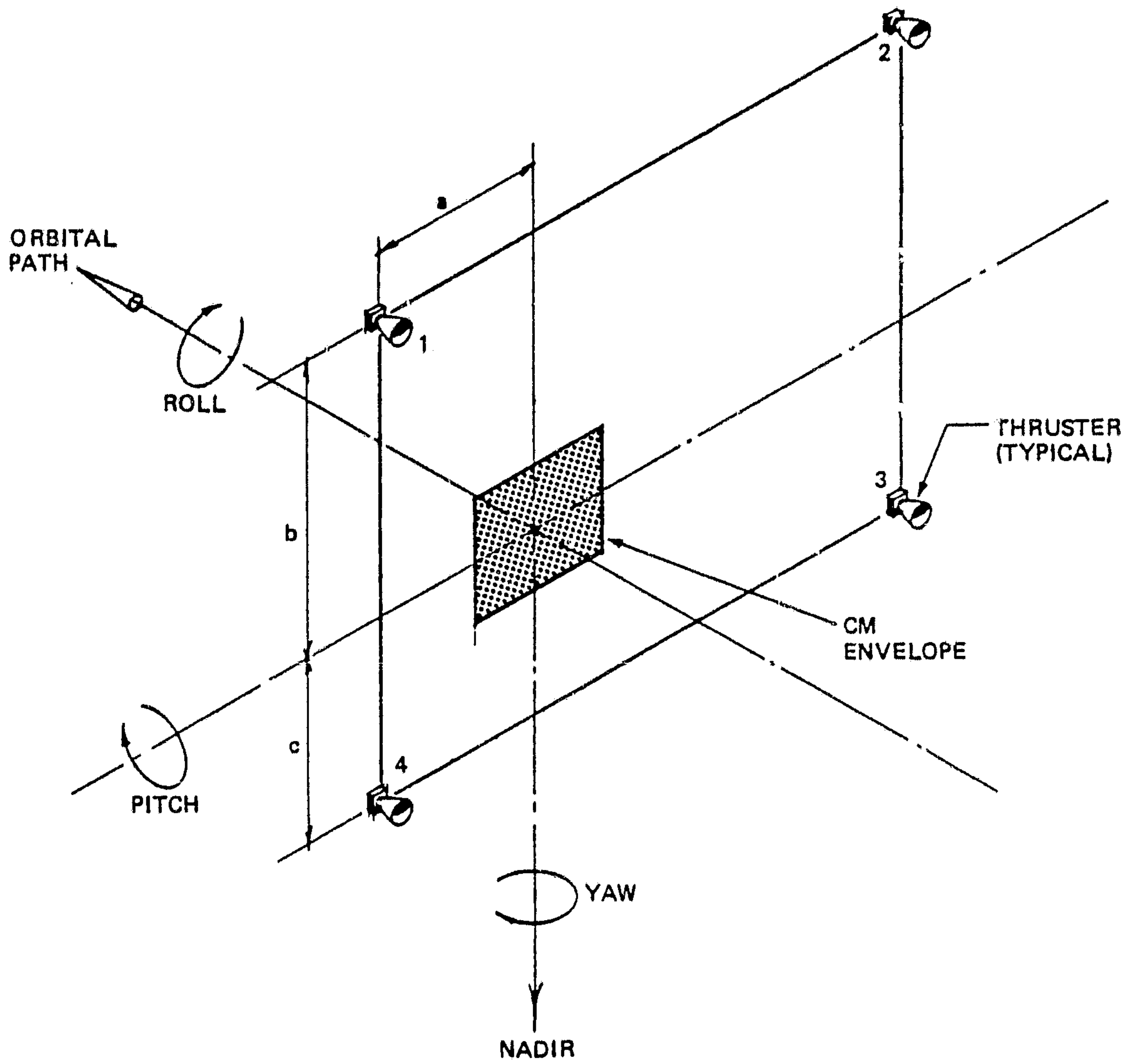

Figure 5-5. Offset Thruster Depiction

distance. Countering torques in this manner requires that the thrust times the moment arm be equal to the aerodynamic or gravity-gradient torques (i.e., low thrust or numerous short pulses). This approach requires more propellant than the first method proposed for torque cancellation because the couple cannot simultaneously overcome drag forces.

(3) Torque rods can be used to provide a countering torque magnetically. Torque rods are electromagnets that are designed to impose a torque on a spacecraft by interacting with the Earth's magnetic field. The largest torque rod produced, used on the Space Telescope, weighs 96 lbm, is 98-in. long, and can deliver a maximum torque of 0.078 N-m at 20% linearity in a 0.2G field, which is typical for the selected station altitudes. The size of the torque rods produced to date has been limited to that of the manufacturer's (Ithaco Inc.) curing oven, but, theoretically, there is no limit to the maximum torque rod size. RCA Astroelectronics has used magnetic torquing for more than 20 years for attitude and momentum control on over 65 spacecraft.* Figure 5-6 shows a typical torquing coil application. The power requirement for the largest torque rod, at saturation, is 56W. The station will require on the order of 10 torque rods for attitude control depending on actual design and pointing requirements. Based on the foregoing, it appears that station attitude control requirements could be met by using torque rods and expending relatively little electrical power. Figure 5-7 provides design information on the torque rods now being manufactured by Ithaco, Inc. that may be used on the station.

(4) Control moment gyros (CMG's) can also be employed to counter torques. CMG's are momentum management devices that are essentially a flywheel that can be either accelerated or decelerated to create a torque about its spin axis. CMG's are particularly useful for countering cyclic torques, where the direction of torque on the spacecraft is reversed

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*Schmidt, G. E., Jr. and Muhlfelder, L., "The Application of Magnetic Torquing to Spacecraft Attitude Control," Paper AAS 81-002, Feb. 1981.

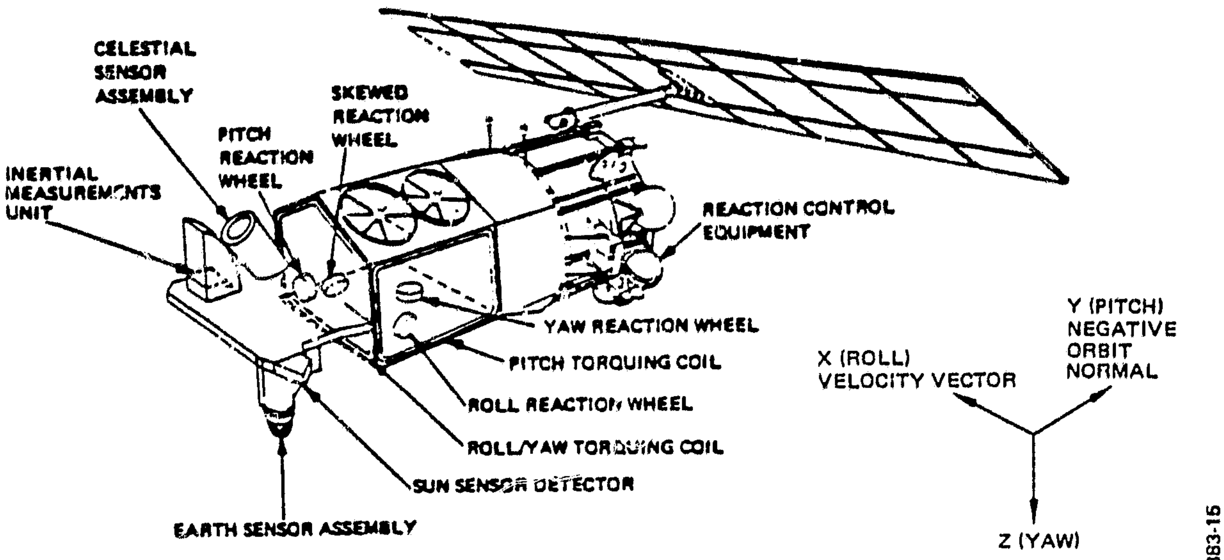
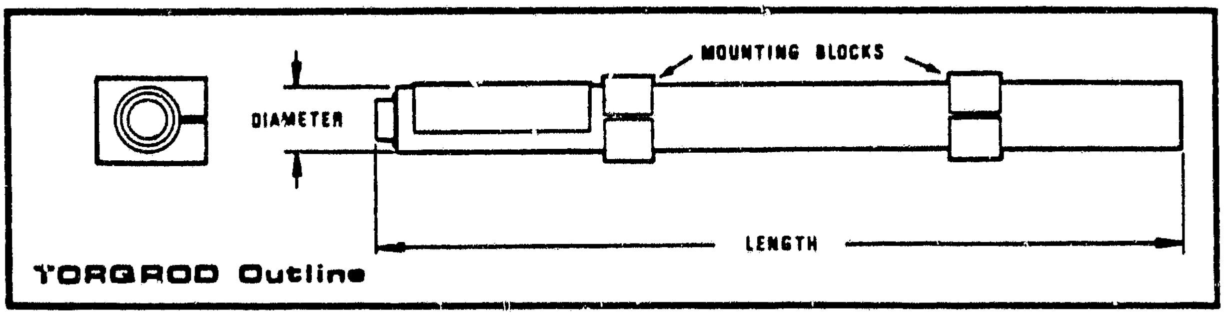


Figure 5-6. Spacecraft Configuration With Torquing Coil Application

(Source: G. E. Schmidt, Jr. and L. Muhlfelder, "The Application of Magnetic Torquing to Spacecraft Attitude Control," AAS 81-002, February 1981.)



DIPOLE MOMENT [Am²]		SIZE [cm]		WEIGHT [kg]	POWER * [W]	
Linearity				Includes mounting blocks	Linearity	
1%	20%	length	diameter		1%	20%
10	15	40	1.8	0.4	0.6	1.0
15	20	45	1.8	0.5	0.6	1.5
20	30	49	1.9	0.6	0.7	1.7
30	50	56	2.1	0.9	0.7	1.8
60	85	64	2.6	1.7	0.8	2.0
100	150	72	3.6	2.8	1.1	2.7
150	250	84	3.8	3.2	1.3	3.5
250	350	104	4.3	6.2	1.8	4.4
350	500	115	4.7	8.3	2.1	5.0
500	700	130	5.0	11.1	2.3	5.5
1,250	1,750	200	5.3	18.5	3.3	7.6
2,900	4,000	250	7.6	49.9	6.0	16.0

1 Ampere meter² = 1000 p–cm

*When a single winding is used, power doubles.

Figure 5-7. Typical Torque Rod Sizing Information (Courtesy Ithaco, Inc.)

for part of the orbit. CMG's used solely for cyclic torques must be sized to absorb the momentum generated during a portion of the cycle, which will then be given up during the following, opposing, portion of the cycle. Torques that tend toward one inertial direction are called secular torques and will eventually saturate a CMG. After becoming saturated, the CMG must be desaturated by reacting against a torque in the opposite direction, which is created by using torque rods or a propulsion system. The CMG's used as a baseline in this study are an advanced version of the CMG's used on Skylab and have a momentum storage capacity of 4000 N-m-s.

Some combination of these strategies will undoubtedly be employed in the Space Station design. CMG's and torque rods will probably be the primary system, and propulsion will be a backup system.

Figure 5-8 shows the 90-day and 10-year propellant requirements for CMG roll-torque desaturation for the baseline Space Station in a 525 km orbit. The figure covers the I_{sp} range of interest (220 to 800), assuming a 10m thruster moment arm. For reference, the 90-day requirements for the cantilevered array configuration are also shown, which exceed those for the balanced-array configuration for 10-years. The 10-year propellant requirement for the cantilevered array is off the scale. Since the orbit altitude has little effect on gravity-gradient forces, the data are applicable to a station at any altitude identified in this study.

Figure 5-9 illustrates the propellant requirements to overcome secular aerodynamic torques for a 1m CP-CM moment arm assuming a 10m thruster moment arm. (A 1m CP-CM offset appears to be an attainable goal for design purposes.) Both 90-day requirements for a neutral atmosphere and 10-year requirements for a nominal atmosphere are shown for a variety of I_{sp} values. The data shown were calculated for a 525-km orbit but can be determined for 400, 450, and 500-km orbits by multiplying the propellant requirements from figure 5-9 by 5.56, 2.71, and 1.39, respectively. The propellant requirements for secular aerodynamic torque cancellation, therefore, are low compared to orbit maintenance requirements. Also, since CMG desaturation can be accomplished by using orbit maintenance thrusters

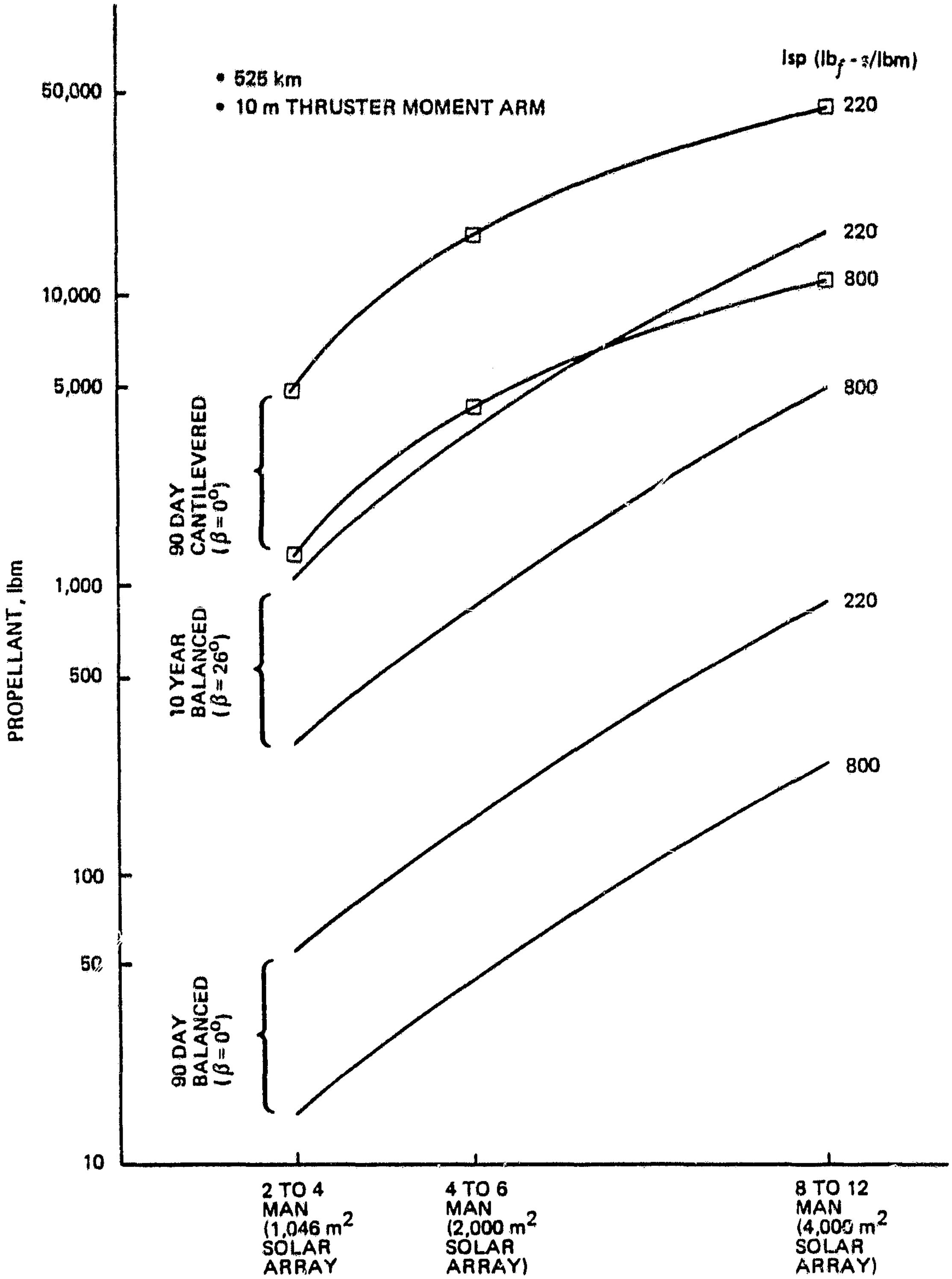


Figure 5-8. Gravity Gradient Roll Torque Desaturation Requirements

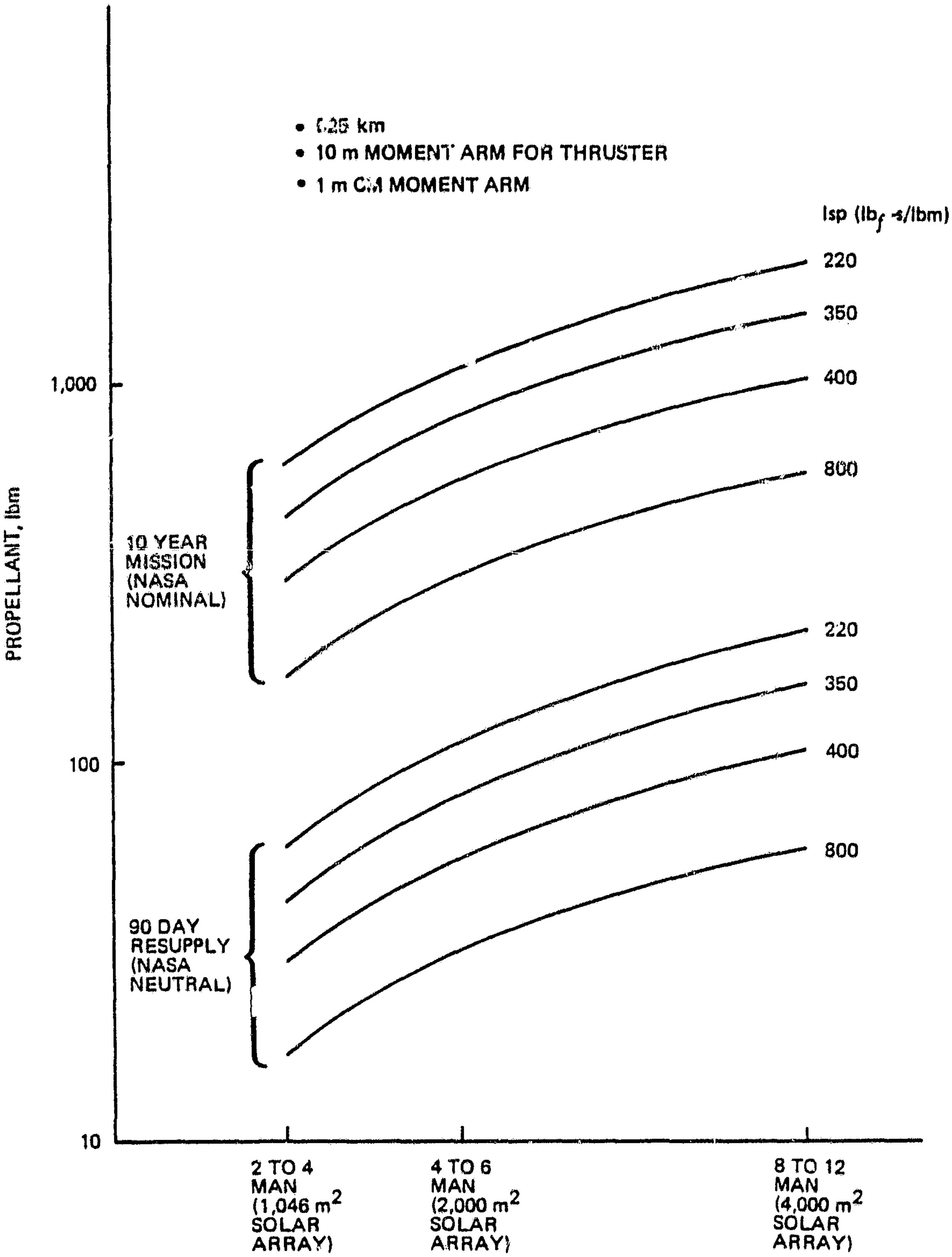


Figure 5-9. Aerodynamic Torque Desaturation Propellant Requirements

offset from the center-of-mass, it does not require additional propellant. Aerodynamic torques are always either in the pitch or yaw direction.

Gravity-gradient roll torques are shown in figure 5-10 for both the balanced and cantilevered array configurations and for both 90-day and 10-year durations. Specific impulse values of 220 and 800 lb_f-sec/lbm are shown. Gravity- gradient produces only roll torques for the balanced array and predominately roll torques for the cantilevered array. Propellant usage assumed a 10m moment arm for the thrusters. Since the gravity-gradient varies only slightly over the altitude range studied, the data in figure 5-10 are valid for all altitudes.

Figure 5-11(a) illustrates the mission impulse and figure 5-11(b) the combined gravity-gradient and orbit maintenance propellant requirements for a 10-year mission at 525 km for the baseline stations with balanced arrays. The data can be converted for the lower altitudes as in the previous figure. It should be noted that the propellant mass doesn't exceed 1000 lbm per 90-day servicing mission for the lowest I_{sp} propulsion system.

In summary, attitude control may well be accomplished non-propulsively. However, the propellant requirements for attitude control are modest if a propulsion system is used.

5.3 Propulsion and Propellant Requirements for Retained Servicing Strategies

Three Space Station servicing strategies were selected for further evaluation from those described in section 4.1. The propulsion requirements associated with these strategies are given below and a single strategy is selected.

5.3.1 Fixed High Altitude, Servicing via Direct Insertion

The propellant requirements associated with the fixed high altitude depend on the altitude chosen. The range suggested in this study is between 450 and 525 km. The upper limit is imposed by the Van Allen Radiation belt and

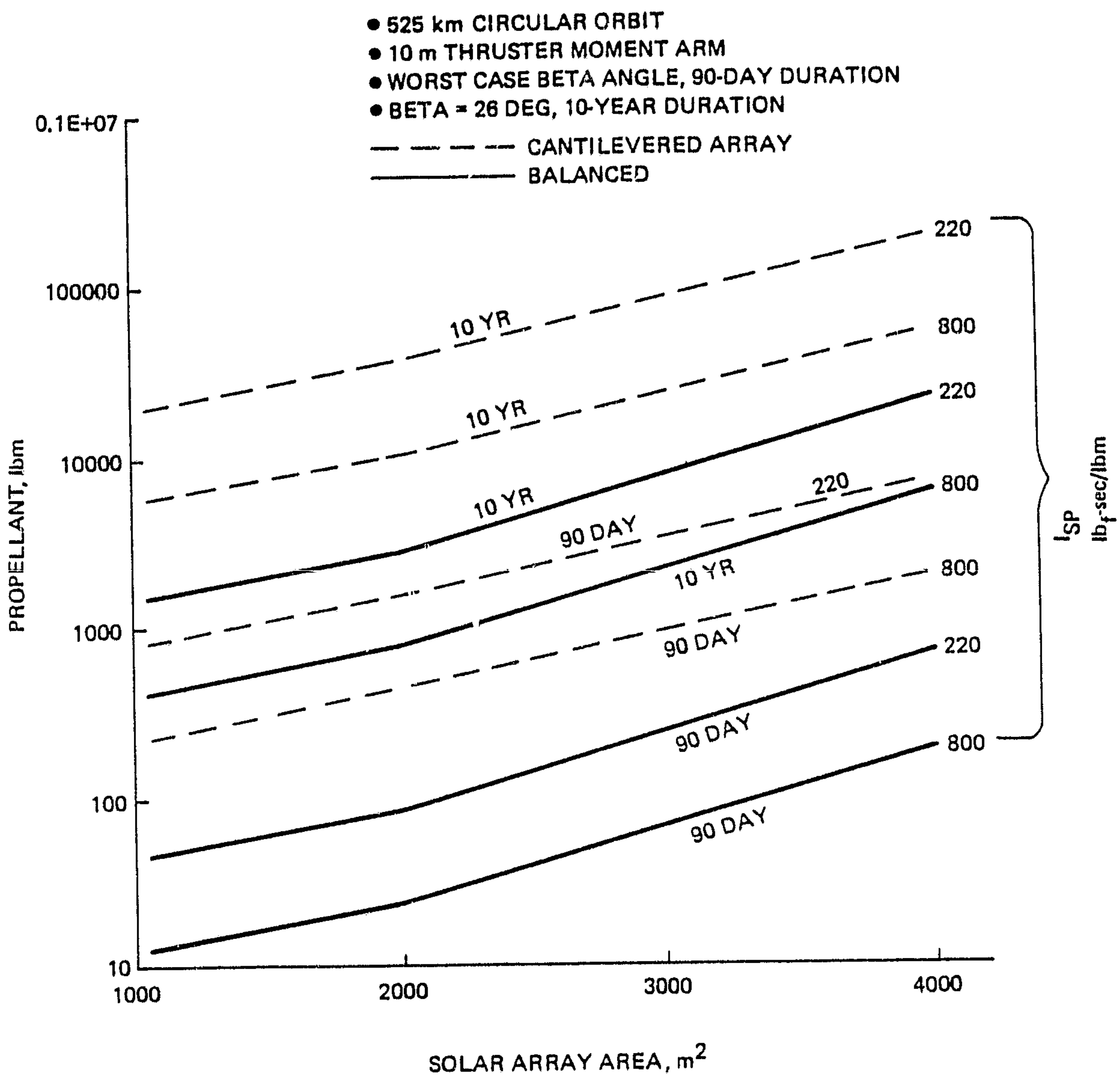


Figure 5-10. Propellant Requirements Due to Gravity Gradient Roll Torque (Baseline Aspect Ratios)

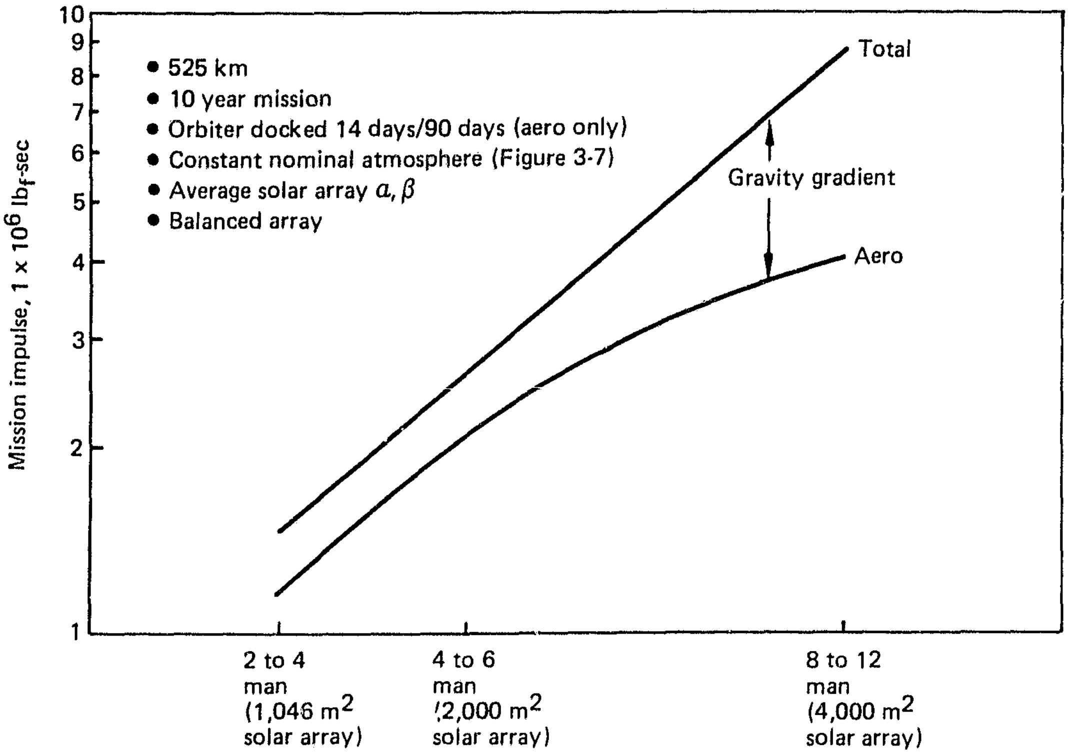


Figure 5-11(a). Mission Impulse

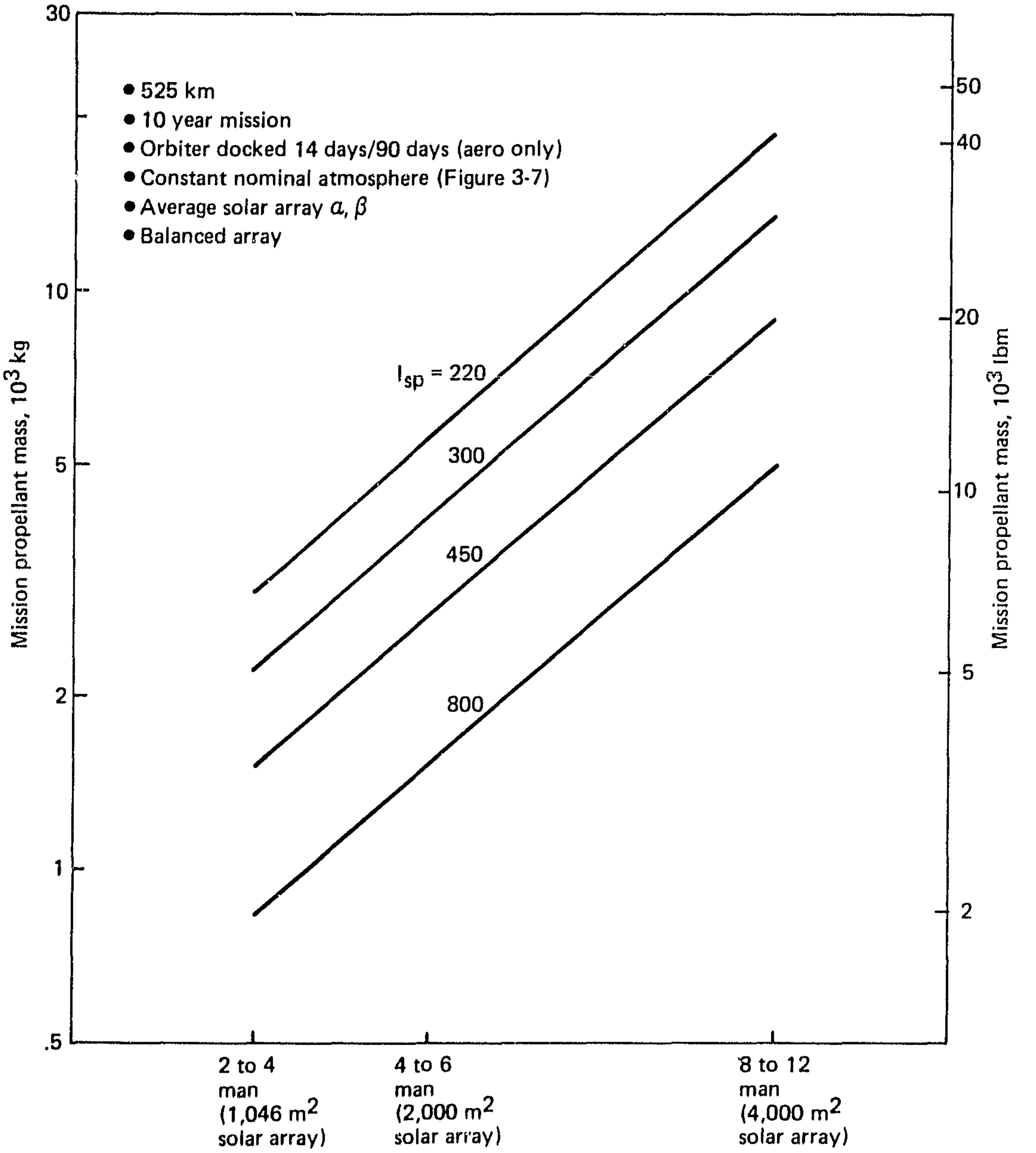


Figure 5-11(b). Mission Propellant Requirements Resulting From Aerodynamic Drag and Gravity Gradient Roll Torque the lower limit is based on the 90-day safe altitude requirement. This section shows propellant requirements for both the 450 and 525 km altitudes. It should be mentioned that a Space Station could operate within the Van Allen belt, but personnel and equipment would have to be shielded from the radiation, which was not a design criterion included in this study. The OMV and OTV will be designed to withstand short periods within the belt while transporting spacecraft between high (GEO) and low Earth orbits.

Table 5-1 presents the 90-day propellant requirements for orbit maintenance at 525 km in a neutral atmosphere for the baseline station configurations with an Orbiter docked for 14 out of the 90 days. The orbit maintenance propellant requirement for 10 years can be obtained by multiplying the data from table 5-1 by 8.96 (which is the number of 90 day periods in 10 years multiplied by the nominal-to-neutral atmosphere density ratio at 525 km). Therefore, the 10-year propellant requirement for a construction station and an I_{sp} of 220 lbf-sec/lbm is 19,130 lbm. Since there are 40 servicing missions over this 10-year interval if a servicing mission is conducted every 90 days, the average propellant resupply requirement is 480 lbm.

Table 5-2 shows the propellant requirements for orbit maintenance at 450 km, again in a neutral atmosphere and for the same baseline Space Stations. To obtain the propellant requirements for 10 years in a nominal atmosphere, multiply the data by 12.48, which is the number of 90-day periods in 10 years muliplied by the nominal-to-neutral atmosphere density ratio at 450 km. For example, the 10-year requirement for the 8- to 12-man construction station for an I_{sp} of 220 lb_f-sec/lbm is 72,159 lbm. The average propellant requirement per servicing mission is 1780 lbm. Therefore, the decrease in propellant mass between the 450 and 525 km orbits is 1300 lbm each servicing mission. If a propulsion system with a high I_{sp} is used, the amount of propellant would be reduced for both altitudes.

Based on the foregoing analysis, the fixed high altitude servicing option is a satisfactory approach. Considering the reduction of payload and the orbit maintenance propellant saved, there is a 700 lbm penalty every 90 days for operating at 525 versus 450 km for an I_{sp} of 220 lb_f-sec/lbm.

Table 5-1. 90 - Day Propellant Requirement for a Fixed High Altitude Space Station (525 km)

STATION DESCRIPTION	SPECIFIC IMPULSE (lb_f - sec/lb_m) 220	300	450	800
	PROPELLANT REQUIRED (lb_m)			
2 TO 4 MAN SCIENTIFIC	584	428	286	161
4 TO 6 MAN SCIENTIFIC	1064	780	520	293
4 TO 6 MAN CONSTRUCTION	1146	840	560	315
8 TO 12 MAN SCIENTIFIC	2054	1506	1004	565
8 TO 12 MAN CONSTRUCTION	2135	1566	1044	587

Table 5-2. 90 - Day Propellant Requirements for a Fixed High Altitude Space Station (450km)

STATION DESCRIPTION	SPECIFIC IMPULSE (lb_f - sec/lb_m) 220	300	450	800
	PROPELLANT REQUIRED (lb_m)			
2 TO 4 MAN SCIENTIFIC	1582	1159	775	436
4 TO 6 MAN SCIENTIFIC	2882	2113	1408	794
4 TO 6 MAN CONTRUCTION	3104	2275	1517	853
8 TO 12 MAN SCIENTIFIC	5563	4079	2719	1530
8 TO 12 MAN CONTRUCTION	5782	4241	2828	1590

(BOTH TABLES ARE BASED ON A NASA NEUTRAL ATMOSPHERE)

5.3.2 Variable High Altitude

In this servicing strategy, the Space Station remains in high orbit until it must rendezvous with the Orbiter. At that time, it descends to the 450 km lower limit of the high-altitude range. Figure 5-12 shows the propellant requirements associated with this servicing strategy for a variety of propulsion systems. For example, based on a station mass of 100,000 kg, approximately 4,000 kg of hydrazine would be needed for the 75 km round trip from 525 to 450 km. Although the amount of propellant can be reduced with higher specific impulse values, the variable high orbit requires substantially more propellant than a fixed high orbit. This places additional limitations on Orbiter cargo space for other mission requirements. Therefore, due to the high propellant requirements and other problems associated with a variable orbit which were discussed in section 4 (i.e., station/free-flyer nodal regression and experiment disruption), this strategy has been eliminated.

5.3.3 Fixed High Altitude, Remote Servicing

This variation of the high altitude servicing strategy uses a servicing vehicle (OMV) to transfer payloads from the Orbiter at a lower altitude to the Space Station at a high altitude. The Space Station remains in a fixed high orbit at 525 km. The STS is launched using nominal insertion to 300 km, where it would be met by the OMV. The OMV then transfers payloads from the Orbiter to the station.

The advantages of this strategy are linked to the tradeoffs between STS deliverable payload and OMV propellant requirements. The Orbiter can carry 4,000 kg more payload to 300 km than to 450 km, and 6,000 kg more than to 525 km (using direct insertion for the latter two). The Orbiter payload capacity to 300 km is 33,000 kg (77,175 lbm), according to figure 2-15.

The OMV retrieval and delivery capabilities are shown in figures 5-13(a) and 5-13(b). The OMV has a 27,200 kg (60,000 lbm) capacity to 525 km, which is 5800 kg less than what the Orbiter can deliver to 300 km (33,000 kg - 27,000 kg). However, if 1300 kg of the excess 6000 kg can be

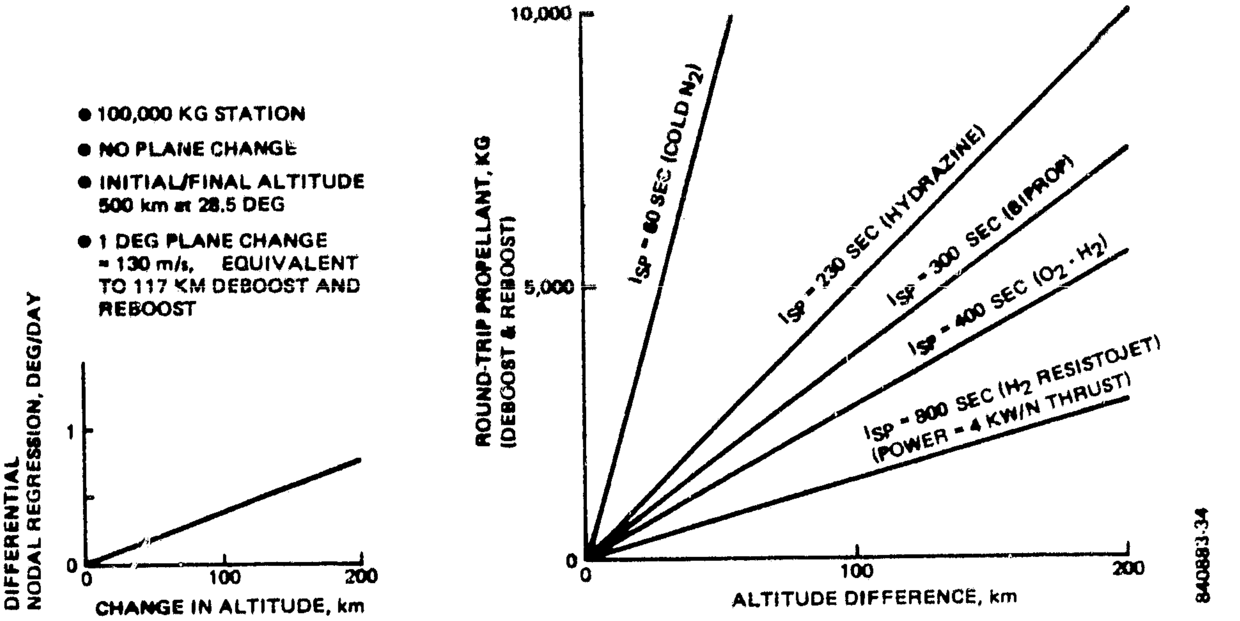


Figure 5-12. Propellant Consumption for the Variable High Altitude Servicing Strategy

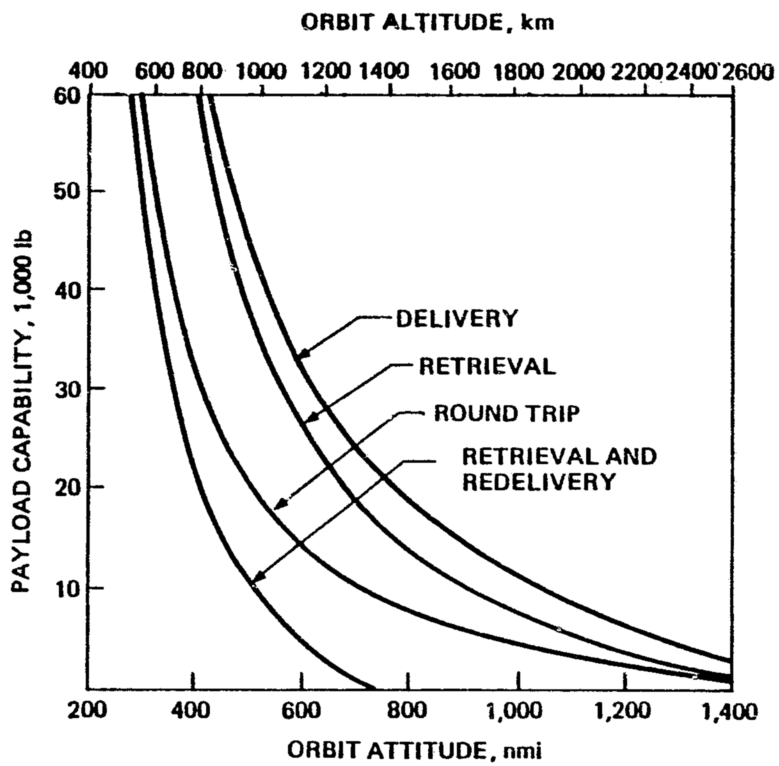


Figure 5-13(a). OMV Performance Capability, Coplanar

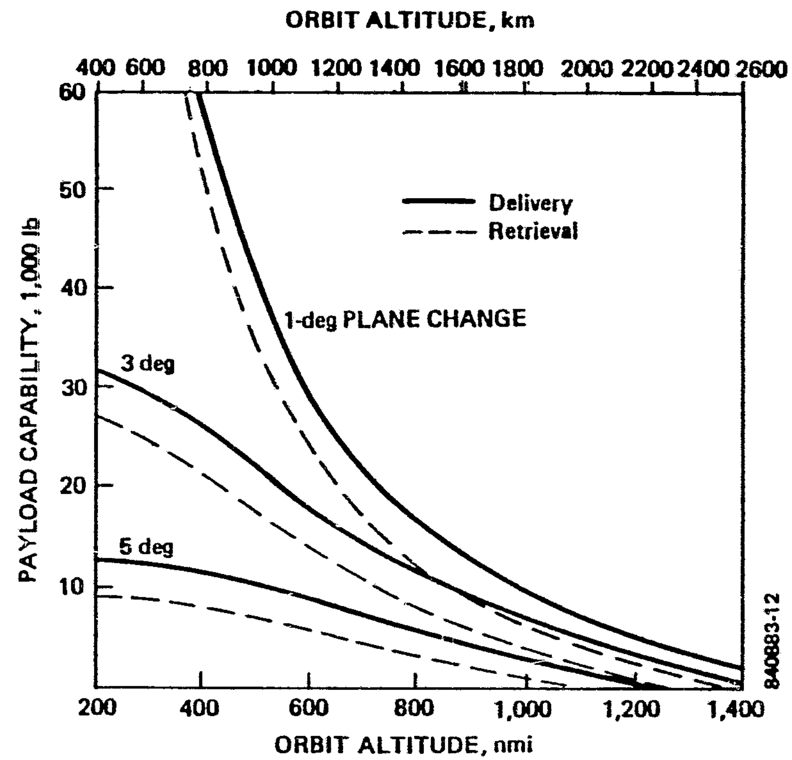


Figure 5-13(b). OMV Performance Capability with Plane Change propellant for refueling the OMV, it could transport the remaining payload (31,700 kg) to the station, based on the retrieval curve of figure 5-12.

Therefore, the OMV flyup method can deliver 4500 kg more payload than direct insertion can to a 525 km orbit. However, the 1300 kg of propellant required by the OMV for a second trip to the Space Station is lost payload. Hence, the actual increase in payload capacity using the OMV to transfer payloads from the Orbiter at 300 km to the station at 525 km is 3200 kg.

The negative aspects of this servicing strategy are that the docking, payload transfer, and refueling operations are more complex than in other servicing modes. It should be emphasized that the OMV is an unmanned, remotely operated, vehicle, which requires even more precise planning and operations.

5.4 Selected Servicing Strategy

Based on the foregoing discussion, it is apparent that Space Station servicing is facilitated by locating the station at a fixed altitude. Since most payloads are volume, rather than mass, limited, the higher altitude may have little effect on the payloads delivered by direct insertion. The difference in propellant requirements for orbit maintenance at either end of the 450 to 525 km high altitude range are significant, but are not expected to affect payloads or servicing. The 450 to 525-km altitude is also based on the effects of atomic oxygen and airglow. As mentioned previously, these effects diminish with altitude and atmospheric density. Therefore, the recommended servicing strategy is by direct insertion to a station operating at a fixed altitude between 450 and 525 km.

5.5 Emergency or Critical Propulsion Requirements

In addition to normal operating conditions, there are several critical situations that require a much higher thrust level than for orbit maintenance: docking, collision avoidance, rescue operations, and safe end-of-life station disposal.

5.5.1 Orbiter Docking Disturbance

The worst case docking disturbance will occur when the maximum weight Orbiter, 230,000 lbm, docks with the station at the peak differential velocity. The following assumptions are made to facilitate an assessment of the effect of docking on the propulsion system:

- The maximum closing velocity between the station and the Orbiter is 0.5 ft/sec;
- A "hard" or rigid connection is made one second after initial contact of the Orbiter with the docking port;
- The velocity imparted by the Orbiter as a result of docking must be reduced to zero in one minute;
- After the rigid connection is made, the Orbiter/station combination is a completely rigid unit;
- The Orbiter's relative velocity vector (0.5 ft/sec) passes through both the Orbiter and station CM's.

The velocity imparted to the station as a result of docking with the Orbiter is readily found from conservation of momentum principles. The mass of the various stations and the docking delta-V's are shown in Table 5-3. Based on these velocity changes, the accelerations and forces experienced during docking are as shown in Table 5-3. Similarily, the thrust required to arrest this delta-V within one minute is 60 lb_f (for all station sizes).

Table 5-3. Docking Disturbances

Station		Docking		
Crew Size	Mass (lbm)	Delta-V (ft/sec)	Acceleration(g)	Force (lb_f)
2 to 4	134,434	0.316	0.0098	1320
4 to 6	207,368	0.263	0.0082	1695
8 to 12	313,240	0.212	0.0066	2064

The station configurations contemplated in this study are expected to be highly flexible as opposed to the rigid structure assumed in the foregoing analysis. Although a structural dynamic analysis is beyond the scope of this study, the results above are expected to be affected significantly by dynamic effects.

5.5.2 Collision Avoidance

A collision with space debris, a runaway servicing vehicle, satellite, or Orbiter could have disastrous consequences and must be avoided. To achieve a reasonable safety margin for collision avoidance, the Space Station must have a high thrust, high performance, and highly reliable emergency propulsion system. This study assumes that the Space Station will require a maximum 1000m translation (movement in any direction perpendicular to the velocity vector) as rapidly as possible to avoid a potential collision. The minimum time required to effect this translation is dependent on the allowable acceleration that can be tolerated by the station. The propulsion system required for these maneuvers must be capable of providing the thrust levels shown in table 5-4 for the accelerations listed. Equivalent thrust levels required for the 4- to 6- and 8- to 12-man stations are found by multipling by 1.54 and 2.33, respectively.

Table 5-4. Collision Avoidance
(for a 2- to 4-man station)

Maximum Allowable Acceleration (g)	Thrust (lb_f)	Minimum time to Translate 1000m* (sec)
0.003	403	245
0.006	807	173
0.009	1210	141

*Constant thrusting.

In discussing avoidance maneuvers, it is convenient to describe the Space Station's "original position" as the orbital position it would be in at any time if no maneuvers had taken place. Return to the "original position" means to recover the initial state vector and therefore to resume formation flying with other vehicles that have not maneuvered (some free-flyers have no propulsion).

Once a maneuver impulse has been delivered, distance from the original position will continue to increase. The foregoing table shows the time needed to translate 1000m for the acceleration shown. The station must thrust an additional two times to return to its original position: once to establish a state vector that returns to the original position and a second to correct the state vector to the original one when it is intercepted. These latter maneuvers do not require high thrust since time is no longer a primary constraint. It will be important to consider the presence of co-orbiting free-flyers in collision avoidance maneuver planning; this is a real-time operational need that cannot be addressed in this study.

5.5.3 Rescue Operations

All potential rescue operations could be performed by either an OMV, OTV, soft or hard suited MMU, personal rescue system, or an Orbiter (if it is docked at the time). Safety specifications expected to be in force will not permit any exterior activity to be performed without one or more of these present. Therefore, due to this safety requirement and the high propellant expenditure that would be needed to perform a rescue operation with the station itself, the Space Station propulsion design requirements do not address this usage.

5.5.4 End-of-Life Station Disposal

At the conclusion of the useful life of the Space Station, it must be disposed of in a safe manner. Two likely ways for accomplishing this are: (1) by implementing a controlled atmospheric re-entry within an acceptable debris footprint, or (2) by boosting it into a high enough orbit to preclude re-entry for many years without orbit maintenance. Both strategies have advantages and disadvantages. The high-orbit disposal is the least demanding insofar as the propulsion system is concerned because an increase in altitude can be attained with any station thrust level if it is applied for a sufficiently long burn time. An OMV or OTV could also be used to place the station in high orbit. The main disadvantages of a high-orbit are that (1) it will add to the number of items in Earth orbit that must be tracked continuously, and (2) eventually, in perhaps 100 years, the orbit will decay and the station will have to be reboosted.

Controlled re-entry, on the other hand, requires a specific acceleration. For example, controlled re-entry of a 250,000-lbm station from 500 km into a footprint 100 nmi long and 10 nmi wide would entail (1) changing the orbit to an ellipsoid with the apogee at 500 km and perigee at 107 km, and (2) firing a 500-lb_f thruster in retrograde for 10 minutes at the apogee altitude. If the allowable footprint can be enlarged, the required thrust level can be reduced and the duration of firing extended. The OMV or OTV could be used to put the station on a re-entry trajectory and could then return to the Orbiter or a parking orbit. An OMV can produce approximately 880 lb_f thrust, which is enough to provide the necessary delta-V for a controlled re-entry disposal.

5.5.5 Emergency Propulsion Requirements Summary and Conclusions

Comprehensive emergency propulsion requirements have not been defined, but there are two situations in which the Space Station will require a high thrust, high performance propulsion system: during docking and for collision avoidance. The following thrust levels would be necessary for a 2- to 4-man station:

Emergency	Thrust Level (lb_f)
Docking Disturbances	60 to 1320
Collision Avoidance	100 to 1300

The upper thrust limit of 1320 lb_f is based on providing an acceleration equal to that imparted to the station during Orbiter docking. These forces are 1695 and 2064 lb_f for the 4- to 6- and 8- to 12-man stations, respectively, as shown in table 5-3. These emergency, or critical, situations identify the need for a minimum thrust of at least 100 lb_f. The maximum allowable thrust level depends on the structural characteristics of the station, which is beyond the scope of this study.

A detailed analysis is required to determine the thrust levels and propellant requirements for various rescue operations. The Space Station, however, is not used directly for rescue and will only be affected by the propellant storage requirements for the various vehicles that may be used for rescue operations.

A reboost to a higher-orbit could be accomplished with any thrust level given sufficient time. Re-entry requires a thrust level of at least 500 lb_f and a burn time of approximately 10 min to ensure that all debris falls in a footprint 100 nmi long by 10 nmi wide.

5.6 Emergency Propellant Requirements

This section defines the propellant requirements associated with the critical propulsion maneuvers described in the previous section.

5.6.1 Docking Disturbances

The Orbiter will dock at least once, and at most three times during any 90-day period. Each time the Orbiter docks, it is assumed that the maximum docking conditions discussed in section 5.5.1 will exist. Figure 5-14 shows the propellant requirements for different specific impulses under maximum docking conditions. As the figure shows, the propellant requirements for docking disturbances are small. Thus, since I_{sp} and reaction time are not critical for docking disturbances, a high-performance propulsion system is not required.

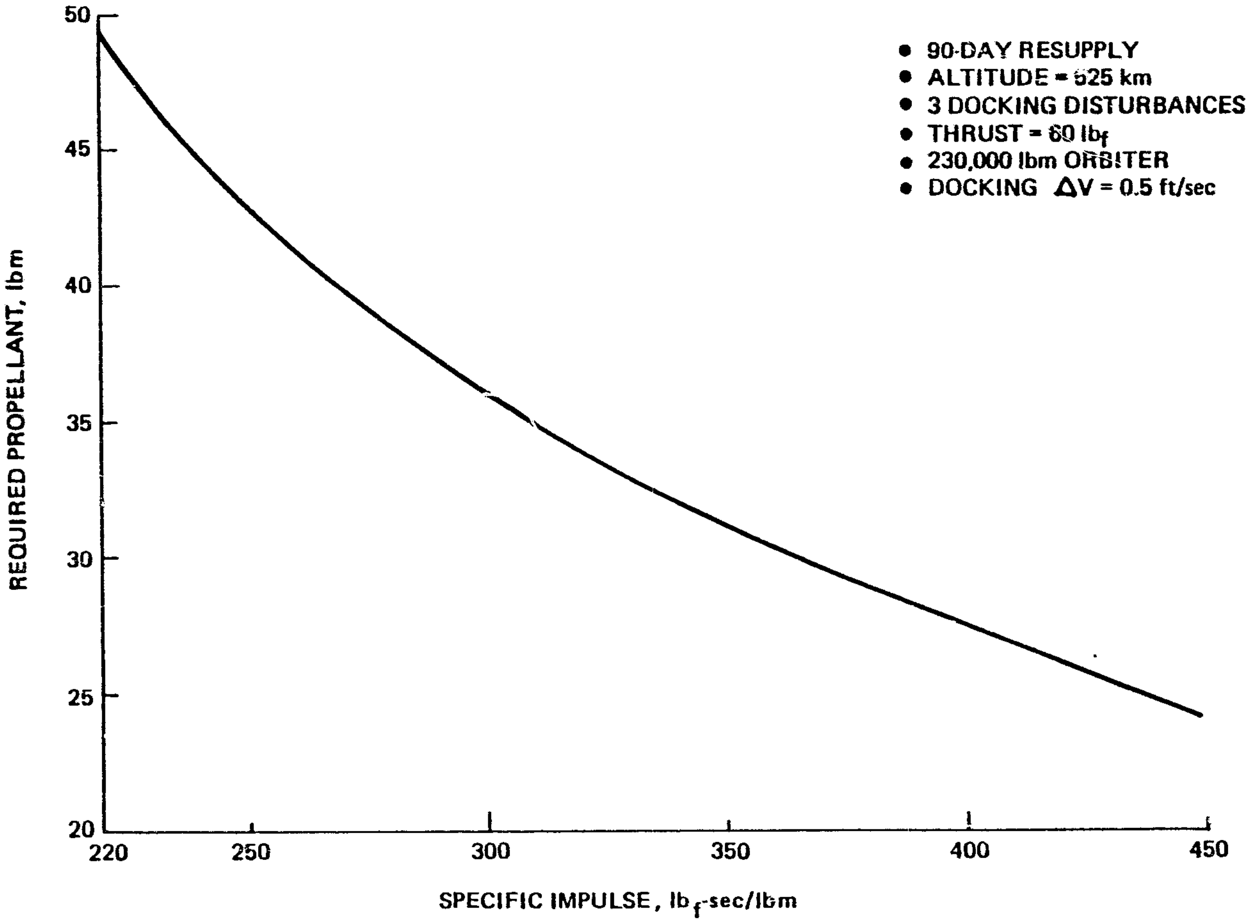


Figure 5-14. Docking Disturbance Propellant Requirements

5.6.2 Collision Avoidance

As stated in section 5.5.2, collision avoidance of large space debris is critical. An SOC study performed by Boeing showed that there is virtually no chance of the Space Station being struck over the 10-year life of the station by natural debris of significant size. Natural debris of micrometeorite size are impossible to avoid, however, so the Space Station will need a protective shield. For man-made debris, however, the SOC study shows that the probability of the station being struck over a 10-year period is unity. Therefore, for any 90-day period, a maximum of one collision avoidance maneuver will be assumed and the Space Station should have enough propellant onboard to perform one maneuver at all times.

Figure 5-15 shows the propellant requirements to perform a collision avoidance maneuver as described in section 5.5.2. Since we have assumed that only a short time is available to avoid a collision, it is important to have both a high-performance and a highly reliable system.

5.6.3 Rescue Operations

There are potential rescue operations which will require significant propellant usage, but not necessarily by the Space Station systems. The OMV, OTV or Orbiter could be used for rescue operations. An in-depth analysis of rescue operations is beyond the scope of this study.

5.7 Free-Flyer Propulsion and Propellant Requirements

The propulsion requirements for free-flyers differ from those for the Space Station primarily in terms of magnitude. Each free-flyer requires orbit maintenance and attitude control but, due to its smaller size, lower total impulse is required. Each free-flyer must have sufficient tankage to store the propellant used between servicing missions, but the free-flyers are not restricted to the station's 90-day servicing interval. Most of the free-flyers are serviced by the OMV or OTV, but some may be able to rendezvous autonomously with the station for servicing. Free-flyer servicing is discussed in detail in section 4.2.

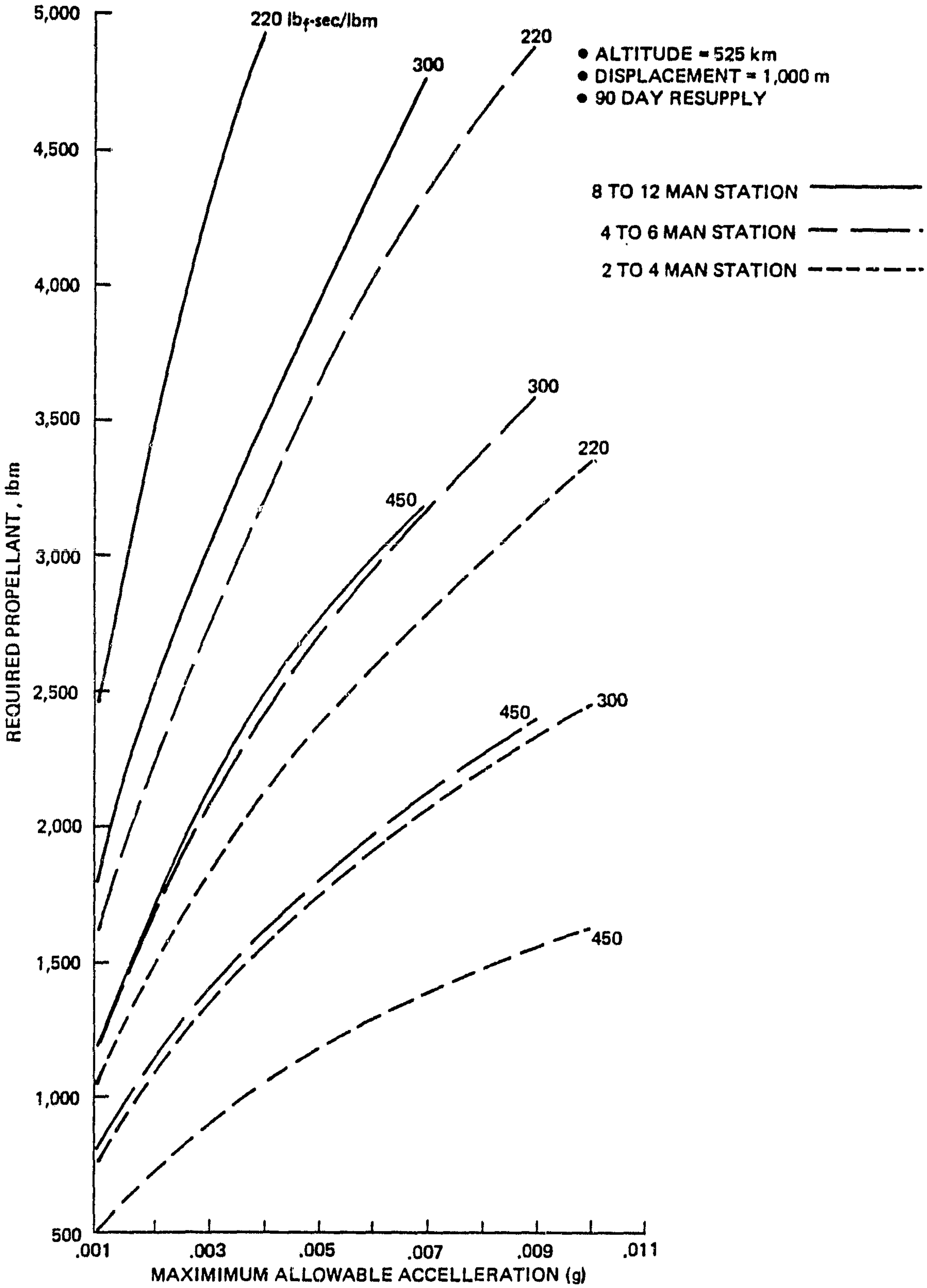


Figure 5-15. Collision Avoidance Propellant Requirements

The free-flyer propulsion requirements discussed in this section pertain to both free-flyers that have been defined in this study and those defined by NASA.

5.7.1 Free-Flyers Defined in This Study

The free-flyers developed as a part of the current contract are described in section 2.3.8. These free-flyers are listed in table 5-5 with their mass, average projected area along the direction of flight, and the area-to-mass ratio.

Table 5-5. Free-Flyer Properties

Free-Flyer	Mass (kg)	Area* (m^2)	A/M (m^2/kg)
Propellant Farm			
Full	128,675	113.5	0.0009
Empty	6,466	113.5	0.0176
SASP			
12.5 kW	7,528	267	0.0355
25 kW	11,250	533	0.0474
STPGM	4,610	518	0.1124

*Average area for the entire orbit; Sun-oriented vehicles.

5.7.1.1 Free-Flyer Orbit Decay

Figures 5-16 and 5-17 estimate free-flyer orbit lifetimes in nominal and neutral atmospheres, respectively, for the area-to-mass range applicable to the free-flyers. If the propulsion system fails, the propellant farm lifetime can vary dramatically, depending on the propellant load. Table 5-6 illustrates the lifetime expected for each of the free-flyers at 400, 450, 500, and 525 km for a neutral atmosphere.

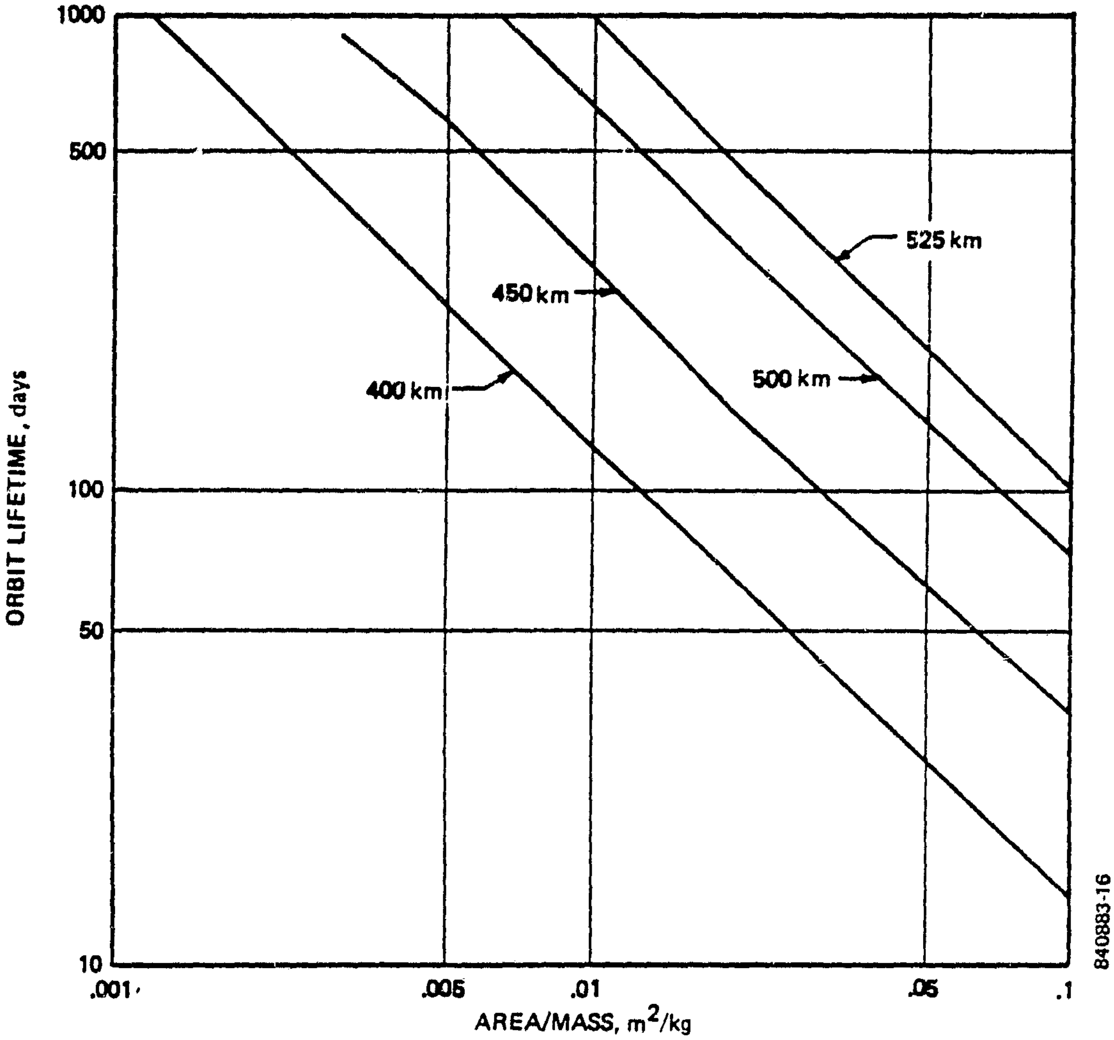


Figure 5-16. Free Flyer Orbit Lifetime, Nominal Atmosphere, F10.7 = 158.7, AP = 12.

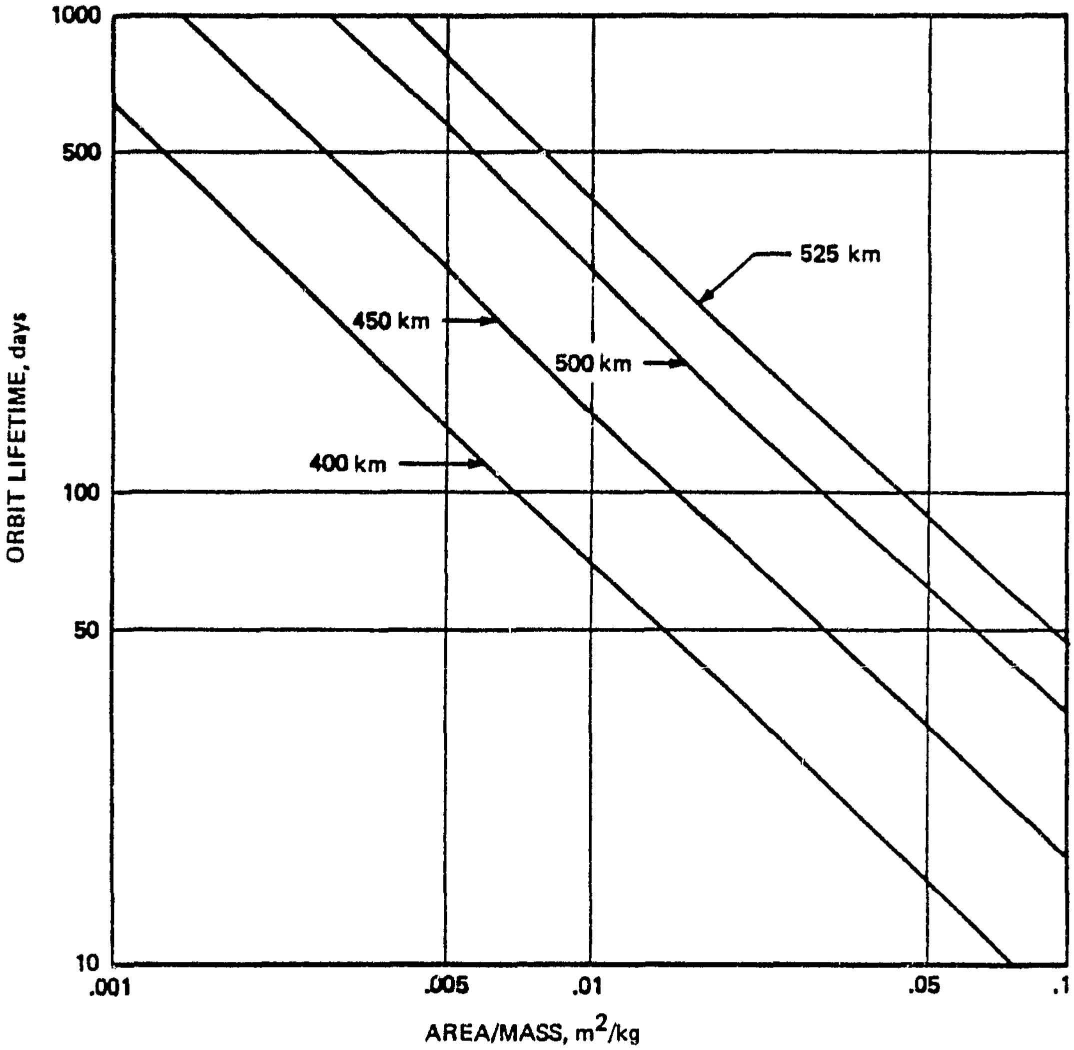


Figure 5-17. Free-Flyer Orbit Lifetime, Neutral Atmosphere, F10.7 = 230., AP = 20.3

Table 5-6. Free-Flyer Lifetimes Following Propulsion System Loss for Various Initial Altitudes (Neutral Atmosphere)

Free-Flyer	Area-to-Mass Ratio (m^2/kg)	Orbit Lifetime After Propulsion Loss (Days) Starting Orbit Altitude (km) 400	450	500	525
Propellant Farm					
Full	0.0009	700	>1000	>1000	>1000
Empty	0.0176	43	90	170	240
SASP					
(12.5 kW)	0.0355	21	44	85	120
(25 kW)	0.0474	16	34	67	90
STPGM	0.1124	5	16	30	44

These free-flyers are expected to be in close proximity to the station (within line-of-sight) so that they can be serviced by the OMV (either in situ or brought back to the station). Free-flyer servicing is not considered to be bound by the 90-day safe-orbit criterion used for the station. However, since a propulsion system failure can occur at any time without warning, OMV and other required resources may be in use and unavailable for immediate servicing operations. This study assumes that line-of-sight contact must be maintained for the propellant farm and SASP free-flyers. The STPGM is closer to the station because it is attached by a tether. This tether remains flexible because the STPGM onboard propulsion system provides thrust to retain the slackness. However, if there is a loss of propulsive ability, the tether will become taught quickly. Table 5-7 shows the resulting tension levels in a neutral atmosphere for the highest drag orientation.

Table 5-7. STPGM Tether Tension Due to Propulsion Loss

Altitude (km)	Max. Tension (lb_f)
400	0.111
450	0.054
500	0.027
525	0.020

These forces will not adversely affect the tether, but they could introduce significant torques in both the station and the STPGM attachment points. This study assumes that the center-of-mass alignment will remain stable and that the attitude control will be maintained non-propulsively. Therefore, servicing can be accomplished within an acceptable period of time if there is a propulsion system loss.

The Long Term Earth Satellite Orbital Prediction computer program (LTESOP) was used to determine the length of time that line-of-sight contact (direct communication) could be maintained between the Space Station and its co-orbiting free-flyers if the propulsion systems failed on the latter. The results are shown in table 5-8.

Table 5-8. Free-Flyer/Space Station Communication Duration with Propulsion System Failure

	Altitude/Days			
Free-Flyer	400	450	500	525
Propellant Farm (empty)	6	9	13	16
SASP (12.5 kW)	5	7	10	12
SASP (25 kW)	4	6	8	10

If the propellant farm is full of fuel, it could remain in the line-of-sight for at least 300 days. However, as table 5-8 shows, at the lower altitudes direct communication between the station and all other free-flyers would be lost in less than 10 days. There would be no immediate danger of re-entry, but the period during which the OMV would have to effect a servicing operation would be reduced at the lower altitudes.

5.7.1.2 Free-Flyer Orbit Maintenance

Propellant requirements for orbit maintenance will vary for each free-flyer, based on drag, specific impulse, and servicing interval, as shown in the following equation:

$$\frac{\text{Drag x time}}{I_{sp}} = \text{Propellant Requirement}$$

The equation indicates the advantages of using a higher impulse or more frequent servicing interval than every 90 days.

The drag forces experienced by the various free-flyers in a neutral atmosphere are shown for different altitudes in table 5-9. The propellant farm will experience little drag due to its low power requirements and subsequent small solar arrays. Figure 5-18 shows the propellant requirements for each free-flyer studied based on a high 220-sec I_{sp} and 90-day servicing interval in a neutral atmosphere.

Table 5-9. Free-Flyer Drag Forces

	Altitude/Drag (lb_f)			
Free-Flyer	400	450	500	525
Propellant Farm	0.0111	0.0054	0.0028	0.0023
SASP				
12.5 kW	0.0305	0.0149	0.0076	0.0055
25 kW	0.0606	0.0295	0.0152	0.0109
STGPM	0.0588	0.0286	0.0147	0.0106

The propellant farm will probably use either the OMV or OTV propellants. Although there may be frequent OMV and OTV dockings with the propellant farm to obtain fuel, resupply docking with the Orbiter may be no more frequent than once every 90 days. Therefore, the propellant farm will have a docked vehicle attached only a small percent of the time, thereby minimizing drag and orbit maintenance propellant requirements. Any orbit reboosting, or attitude control that is needed during docking could be provided by a servicing vehicle or, perhaps, by the Orbiter.

Propellant requirements for SASP are high due to the drag imposed by its large solar arrays. The relatively high power production and low contamination requirements for SASP make it an excellent candidate for water electrolysis. The relatively high I_{sp} obtainable from an H_2/O_2

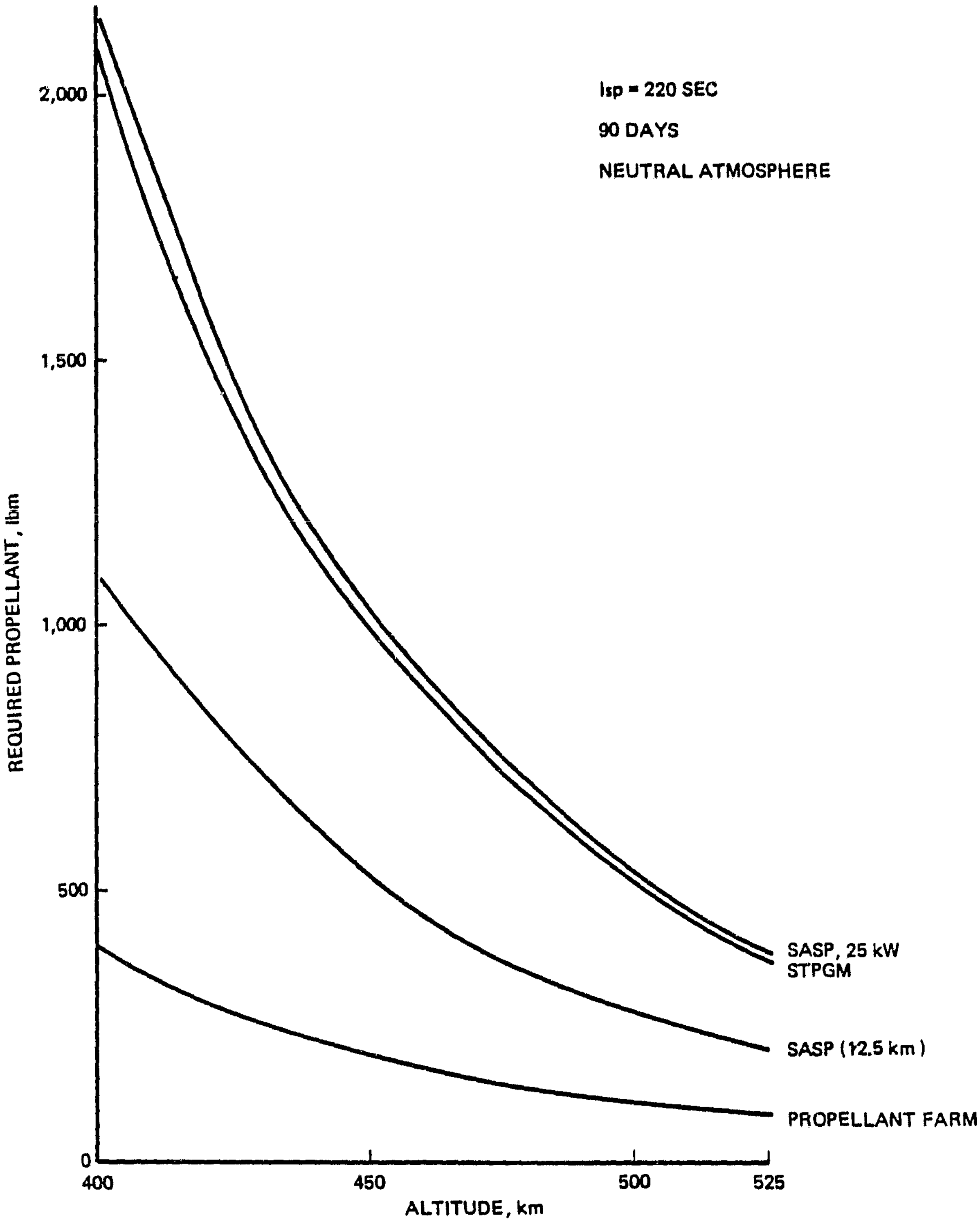


Figure 5-18. Free-Flyer Propellant Requirements

propulsion system would reduce the propellant mass requirements. A higher servicing frequency for the STPGM, made possible by its proximity to the station, would reduce propellant storage requirements. However, using an H_2 resistojet (I_{sp} about 600) would pose an oxygen disposal problem. Ultimately, the STPGM's propulsion system will depend on that chosen for the Space Station because their thrusting strategies (frequency, duration, accelerations) will have to be compatible to keep the tether slack.

In summary, the free-flyers will have different orbit maintenance and propellant requirements, depending on their configuration and drag effects, usage, and function relative to the Space Station. Whereas higher thrust levels or more frequent servicing intervals may decrease propellant storage mass, mission requirements such as low-disturbance experiments on the SASP may mitigate these factors.

5.7.1.3 Free-Flyer Attitude Control

Free-flyers will employ the same methods as the Space Station for achieving attitude control: by using CMG's and torque rods, with a propulsion system backup in case one of these elements fails. Thrusters on the free-flyers must be capable of producing a moment about all three axes and propellant reserves must be adequate to maintain attitude control until repairs to the primary system(s) can be made. Since all systems will have redundant elements, it is unlikely that there will be a complete failure of one system, but a contingency allowance should be incorporated into service planning. In fact, if the thruster location envelope encompasses the center-of-mass envelope to create a sufficient moment arm, all torque cancellation and CMG desaturation functions could be satisfied concurrently with orbit maintenance thrusting. (See section 6.0 for a complete discussion.) This capability is considered to be a design feature of the free-flyers included in this study.

5.7.1.4 Free-Flyer Collision Avoidance

Collision avoidance has not been used as a criterion for sizing free-flyer propulsion system elements in this study because it is assumed that orbit maintenance capabilities will be adequate to laterally displace or rotate the free-flyer to avoid a potential collision. Collision maneuvers are considered to be a rarely occurring phenomenon that could be accomplished with the existing propulsion system.

5.7.1.5 Safe End-of-Life Disposal

Since free-flyers will be close to the Space Station, it is unlikely that they will have to be disposed of due to hardware failure or absolescence: they can be readily serviced like the Space Station, they could be boosted into a high orbit by their own propulsion systems, or they could re-enter the atmosphere via the OMV. Re-entry is the most probable option because they represent a smaller salvage value and pose less danger upon re-entry due to their size. Again, disposal is not considered a factor in propulsion requirements.

5.7.1.6 Free-Flyer Docking Disturbances

The free-flyers will assume a passive role when being docked with an Orbiter, OTV or OMV, all of which will have sufficient attitude control capabilities to dock with other spacecraft. The attitude control system therefore will not encompass docking disturbance propulsion requirements.

5.7.2 NASA-Defined Free-Flyers

The free-flyers NASA defined for the 1991 to 2000 time period were discussed previously and were listed in table 2-1. That table also shows the number of free-flyers that are expected to be deployed, their altitudes, inclinations, and servicing frequencies. With few exceptions, these free-flyers will be deployed at altitudes of 500 km and above. Therefore, they will require relatively small propulsion systems. Most will require only propellant for attitude control, and the Space Telescope will not require a propulsion system at all. Many of these free-flyers will also not need to be serviced after deployment. Information concerning mass and aerodynamic properties of these free-flyers was not available for this study; therefore, specific propellant requirements could not be established. However, the free-flyers discussed in section 5.7.1 are representative of most types of free-flyers that will be serviced by the Space Station.

OMV and OTV propellant requirements for deploying and servicing the free-flyers defined in this study are given in section 4.0. By comparison, the propulsion requirements for NASA-defined free-flyers are insignificant.

5.8 Propulsion Requirements Conclusions

Our analysis of propulsion requirements yields the following conclusions:

We found that placing the Space Station orbital altitude between 450 and 525 km offers relief from high propellant consumption due to aerodynamic drag. Shuttle direct insertion flight operations can reach this altitude range with very modest performance penalties. Sizing of the propulsion system, especially with regard to thrust capability and reserve propellant, should also consider the lowest altitudes at which the Space Station might operate, even in unusual or emergency circumstances. This leads to requirements for thrust capabilities of several Newtons, depending on solar array size and selected minimum altitude.

The simplest resupply operations are obtained when the Space Station remains in a fixed high-altitude orbit, i.e., in the range noted above, with frequent orbit adjustments.

For normal operations at these altitudes, thrust levels below the acceleration sensitivity limits of most Space Station missions are sufficient to maintain the orbit. Orbit maintenance can be achieved by many practical combinations of thrust level and duration. The lower limit is on the order of a tenth of a Newton, and the upper limit is set by station acceleration limits. Structural analyses were not a part of this study, but the upper limit appears to exceed 1000 Newtons based on the range of Space Station loads expected from other sources such as docking.

Space Station propellant mass requirements are moderate, a few kg per day, when altitude is set within the 450 to 525 km range. The degree to which resupply operations are affected by propellant consumption depends on other resupply requirements, outside the scope of this study. Ensuring that the logistics module normal landing mass is less than the shuttle's 14,515 kg limit appears to be a more difficult problem than that of exceeding the shuttle's lift capability. High specific impulse should not be the predominant factor in selecting a propulsion technology(very low specific impulse such as delivered by a cold gas system will present resupply problems). The most important issues are reliability, safety, system synergism, maintainability, and cost. These factors are all discussed in the next section of this report.

Many alternatives are available for attitude control. These include propulsion, momentum storage and management devices, magnetic torquers, and momentum management device desaturation through off-nominal attitude bias (causing desaturating gravity-gradient torques). In the altitude range cited above, desaturation by propulsion requires about the same daily impulse as orbit makeup for the NASA neutral atmosphere. Suitable location of thrusters permits altitude makeup and much of the desaturation to be accomplished by the same impulse. Desaturation intervals will be short, a few hours to days, for typical configurations.

Docking to the Space Station by the shuttle will create disturbances best corrected by propulsive attitude control. Moderate thrust levels 80 to 180 N (20 to 40 lbf) for one minute, are sufficient to counteract the worst disturbances.

Collision avoidance thrust and propellant requirements could be significant, depending on the avoidance scenario.

The next section describes and evaluates the propulsion systems that have been investigated in this study and recommends those that could be applicable to the Space Station, servicing vehicles, and free-flyers.

6.0 PROPULSION SYSTEM ANALYSIS

This section identifies and discusses propulsion systems that can satisfy the Space Station propulsion requirements defined in section 5.0. The propulsion systems evaluated in detail were selected based on factors found by this study to be most critical. Discussions on current state-of-the-art and advances expected to be available for the IOC station will constitute the majority of this section. Also addressed are issues that cannot be quantified; for example, safety and developmental risk. Several propulsion system combinations are identified that could satisfy the requirements set forth in this study. Although a cost analysis is beyond the scope of this study, cost will be a significant driver in propulsion system selection. Hence, for comparison, systems requiring three levels of DDT&E expenditure are defined: the first system utilizes only SOA components and is expected to have the lowest DDT&E and initial cost. The second system requires a modest level of technological development, and therefore, higher DDT&E costs. The third system requires significant technological development and, although the DDT&E costs may be high, the lifecycle costs may be lower than for the other systems. Finally, a brief discussion on free flyer propulsion systems and their interaction with the Space Station is presented.

6.1 Factors that Affect Propulsion System Selection

The Space Station propulsion subsystem will interact directly or indirectly with virtually every other subsystem on the Space Station, as well as with the other space vehicle systems that interact with the station. Figure 6-1 depicts these relationships, many of which involve the mutual interaction of several systems or system functions.

Additionally, safety is involved in all system considerations. Six evaluation criteria were used in selecting (or eliminating) the candidate

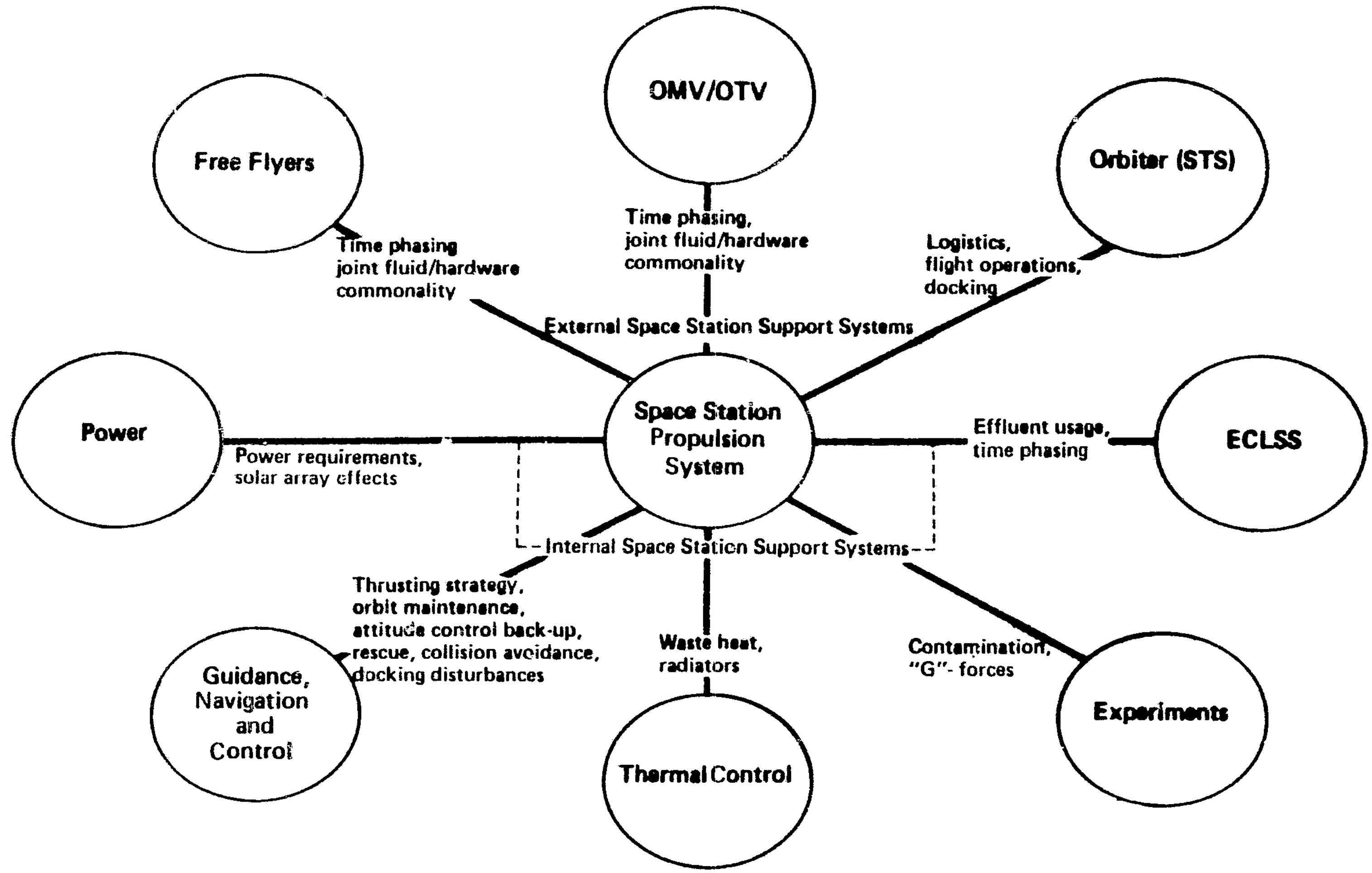


Figure 6-1. Space Station Propulsion System Interactions

propulsion systems and the corresponding propellants. These six criteria are used as the determining parameters due to their direct impact on propellant and propulsion system selection. They are: (1) thrusting strategy; (2) volume and mass limitations; (3) safety and contamination; (4) power; (5) time phasing; and (6) synergism. Volume and mass limitations are included in the resupply requirements and synergism is considered for all of the functions included in potential joint hardware/fluids commonality.

6.1.1 Thrusting Strategy

Thrusting strategy includes consideration of (1) thrust level, (2) thrust frequency, and (3) thrust duration, all of which vary depending on the functional requirements of the Space Station. A number of station maneuvers requiring the propulsion system and their associated thrust levels are shown below.

Low Thrust (0.01 to 10 lbf)	Moderate Thrust (10 to 100 lbf)	High Thrust (>100 lbf)
Drag Cancellation	Reboost	Reboost
Precise Maneuvers	Docking Cancellation	Collision Avoidance
EVA Safety	Desaturation	Disposal (controlled
Torque Cancellations	Disposal (to high orbit)	re-entry or to high orbit)
Desaturation	Collision Avoidance	Docking Cancellation
		Desaturation

This table shows that a range of thrust levels can be used for different Space Station operations. Obviously, the ranges shown could be narrower or wider depending on specific requirements. Collision avoidance (see sections 5.5.2 and 5.6.2) can require thrust levels from low to high depending on the time interval that the station has to perform the maneuver and the distance to be moved. The Space Station RFP (Sept. 1985), for instance, requires only that the station be translated 10 Km in 24 hours; placing virtually no demands on the propulsion system at any thrust level. The arrangement of the above table results from those examples discussed in section 5.0.

Orbit maintenance is the primary function of the Space Station propulsion system. Secondary functions are attitude control system and momentum management device desaturation backup. During low solar cycle years and configuration imbalance, for operational or other reasons, attitude control might be the primary function. A near-continuous thrusting strategy requires a low thrust system which counteracts drag forces and maintains a nearly constant orbit altitude. This strategy also facilitates groundtracking, Orbiter docking, free-flyer servicing, and a low-g station environment. However, a low-thrust strategy creates a high duty cycle for the thrusters thereby potentially decreasing life by increasing the valve cycling, number of ignitions (if required), stress on the combustor materials, etc. Continuous operation creates the lowest possible rate of contaminant production from thrusters.[1] However, all of the gases considered (i.e., CO_2, H_2, NH_4, CH_4) exceeded the column density specifications, as pointed out in the Space Station work package for RFP, by a factor of 1000, except CO_2, which only exceeded it by a factor of 60. Hence, it is apparent that all thrusters will exceed the contamination criterion for column density during operation.

An infrequent-thrusting strategy can result in significant orbit altitude dispersions, depending on the time-lapse between firings and current vehicle drag, necessitating a higher thrust system (see Figure 5-2). This strategy reduces burn time and valve cycling but complicates groundtracking operations, Orbiter docking, free-flyer servicing, and potentially disrupts sensitive onboard experiments due to acceleration. Figure 5-2 shows how orbit maintenance thrust levels change for a variety of duty cycles. Each thrusting frequency and duration corresponds to a particular thrust level for each station size. Each thrusting strategy is capable of satisfying the orbit maintenance requirements but each also has its advantages and disadvantages. Therefore, it is appropriate that an analysis be conducted on thrusting strategies to determine the benefits each strategy can provide.

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1 Ruggeri, R. T.; "The Contamination Effects of Continuous Thrust for Space Station Reboost," Boeing Aerospace Co., Memo 2-1681-5RTR-003, August 1984.

The thrust levels required to counter torques and/or desaturate MMD's depend on the moment arm and thrusting duration. As section 3.2.2 showed, the aerodynamic torques are a function of the drag forces multiplied by the CP-CM moment arm. Gravity-gradient torques are the result of more complicated configuration factors also discussed in section 3.3.2. Orbit maintenance thrusters with thrust levels of 0.1 lb_f or lower (approximate station drag), may be used to counter torques directly depending on whether or not the moment arm-thrust product is as large as the torques in question. (See the discussion on torque cancellation in section 5.2.)

The roll component of gravity-gradient torque cannot be countered by the orbit maintenance thrusters because the torque occurs around the axis along the flight path. These roll torques, which vary from 0.01 to 1.0 N-m, require roll-oriented thrusters if they are to be countered propulsively. Since the roll momentum buildup is quite small compared to the pitch momentum buildup (per radian, balanced array), torque rods rather than roll-unique thrusters can be used to counter the roll torque.

Orbiter docking disturbance cancellation is expected to require a thrust of approximately 60. Since docking maneuvers may occur 15 to 25 times per year, it may be cost-effective to select a thrusting strategy to accomplish both the orbit maintenance and docking functions with one set of thrusters. The collision avoidance scenario developed in section 5.5.2 required thrusters with an aggregate force of 300 to 500 lb_f, depending on station size. End-of-life station disposal via re-entry could also require a high (500 lb_f) thrust level. Both the collision avoidance and disposal scenarios need to be evaluated more thoroughly to see if a separate, dedicated propulsion system is needed.

If a propulsion system with a thrust level greater than approximately 25 lb_f is selected for station orbit maintenance, there may also be a need for a low-thrust system (<10 lb_f), utilizing cold gas thrusters. This low-thrust system could be used for precise station translations during vehicle docking and when a large hot plume cannot be tolerated, such as during vehicle docking or EVA. Selective deactivation of thrusters could alleviate some instances of hot plume impingment on sensitive areas but could not eliminate this problem completely during all phases of the Space Station mission. Conversely, if a low-thrust orbit maintenance propulsion system is chosen that has continuous or near-continuous thrusting, there must also be a higher thrust system for docking cancellation and high delta-V maneuvers.

Therefore, the Space Station could require two, and perhaps three, levels of thrust depending on collision avoidance and end-of-life disposal requirements. A discussion of propulsion combinations to meet high/low thrust combinations can be found in section 6.6.

Thus, as a first evaluation criterion, the Space Station propulsion system must be capable of producing thrust in one or more of the above ranges.

6.1.2 Volume and Mass Limitations

Propellant volume and mass are significant factors in propulsion system selection because most, if not all, STS payloads will be volume and/or weight limited. STS volume is currently limited to 42,390 ft^3 and the maximum allowable weight is about 65,000 lbm,* although, as of this writing no payload has exceeded approximately 45,000 lbm (see section 2.2.1 for STS capability.)

Propellant volume required for a given interval (e.g., the 90-day period) is a function of the density and specific impulse of the various propellants and the impulse requirements. For example, an LO_2/LH_2 combination requires a much larger volume than a N_2O_4/MMH combination because of the extremely low density of H_2 (4.43 lb/ft^3) even though LO_2/LH_2 has an I_{sp} of approximately 450 lb_f-sec/lbm and N_2O_4/MMH has an I_{sp} of approximately 300 lb_f-sec/lbm. Propellant mass is a function specific

*The Air Force is investigating the use of an unmanned launch vehicle (ULV). This ULV would have a payload capability of 143,620 lbm and an envelope of 25 ft diameter by 90 ft in length, or a total volume of approximately 176,625 ft^3. This is approximately four times the STS volume capability. These capabilities were not considered during this study in comparing the various propulsion systems.

impulse for a given total impulse requirement. However, a low density propellant, such as H_2, can require a storage tank that is considerably heavier than a more dense fuel (depending on the pressure which affects tank thickness). Figure 2-15 shows STS delivery capability for various payload weights.

The second evaluation criterion for a Space Station propulsion system is that the propellant requirements must meet the volume and mass constraints for the 90-day STS resupply interval.

6.1.3 Safety and Contamination

Safety and contamination are non-quantitative subjects that must be acknowledged in choosing a propulsion system. Safety is, of course, paramount and the requirements that must be addressed in the selection and design of a Space Station propulsion system include (1) plume effects, which include the temperature, contamination, and corrosiveness of the plume, (2) propellant corrosiveness as it relates to the ability to store the propellants for long periods, (3) explosiveness, i.e., whether a propellant is stable or unstable, (4) flammability, (5) toxicity, and (6) electrical hazards. One or more of these categories is often the deciding factor in choosing or discarding propulsion/propellant system combinations.

Contamination criteria were provided in the Space Station RFP and are waived during thrusting periods (per the RFP). Other requirements addressing cumulative effects of condensing exhaust species have not been defined so quantitative conclusions cannot be made as to how much or what type of contaminants are acceptable. Qualitative judgements are made, which instead, compare the various propellants and their exhaust products. Other actions that may be taken in the overall control of contamination include:

- o Establish a contamination control plan early in the Space Station development program to control fluids allowed to escape or be directed overboard, material selection, manufacturing practices, and handling and cleaning procedures.

- o Bake Space Station components in a thermal vacuum.
- o Cover high outgassing materials with low outgassers.
- o Cover station components with an environmental shelter prior to installation in the Orbiter and cover individual sensitive surfaces during ground and flight operations.
- o Avoid direct impingement of thruster exhaust on sensitive surfaces through hardware design or flight operations.
- o Avoid direct or reflected line-of-sight from contamination sources to sensitive surfaces.

Thruster exhaust, of any type, is a contaminant for some surfaces and/or experiments, but the effects of some contaminants are more serious than others. Free carbon, for example, is an unacceptable contaminant because it can accumulate on the solar arrays and cause electrical short circuits and resultant electrical power reductions. Additional effects of contamination are the degradation of thermal control coatings, optical surfaces, and sensors.

The third evaluation criterion used in selecting a propulsion system is that it meet the safety and contamination requirements stated above. These requirements are discussed in more detail in section 6.5.

6.1.4 Electrical Power

Electrical power is a valuable resource on the Space Station. Sections 2.2.4 and 2.3.5 describe, in detail, the BOL, EOL, and average power levels available for different station sizes, altitudes, and inclinations. Any significant propulsion system power requirement (i.e., beyond valves, pumps, and instrumentation) would impact solar array sizing. Hence, a fourth evaluation criterion for selecting a propulsion system is that its power requirements be minimal.

6.1.5 Time Phasing

The integration of various systems with the Space Station, and the time sequence when the integration occurs, can have a significant affect on the rationale for selecting a particular propulsion system. Figure 6-2 illustrates a time-phasing scenario assumed for illustrative purposes in this study. Other scenarios need to be investigated, particularly as the definition of the Space Station program develops. As different systems are introduced during the life of the station, synergistic opportunities are created. For example, the OMV, which may use N_2O_4/MMH for its propellant, is expected to be available during initial Space Station operations. The OTV, which is expected to use LH_2 and LO_2 for its propellant, is assumed to be available in 1995. Hence, by 1995 the problems associated with resupplying the station with large quantities of these cryogenics will need to be resolved for the above assumption. However, prior to 1995, if LH_2/LO_2 is used on the station, most of the same problems must be resolved, but the relatively small quantities involved makes their selection less practical. On the other hand, it will be difficult to make a transition from a propulsion system tailored for the IOC station to a system tailored for a mature station that services an OTV and a large number of free-flyers. Hence, providing the lowest DDT&E expenditure may result in a much higher life-cycle cost.

As stated previously, a cost analysis is beyond the scope of this study but will be required before the final propulsion system selection is made. The fifth evaluation criterion for selecting a propulsion system is that it be capable of being phased into or out of the station or have potential to be combined with another propulsion (thruster) system.

6.1.6 Synergistic Opportunities

The Space Station propulsion system may benefit from enhancing the synergistic opportunities that exist with other systems (i.e., OMV, OTV, free-flyers, ECLSS, and Orbiter). For example, from the Space Station ECLSS, as conceived in this study, CO_2 gaseous effluent is available and

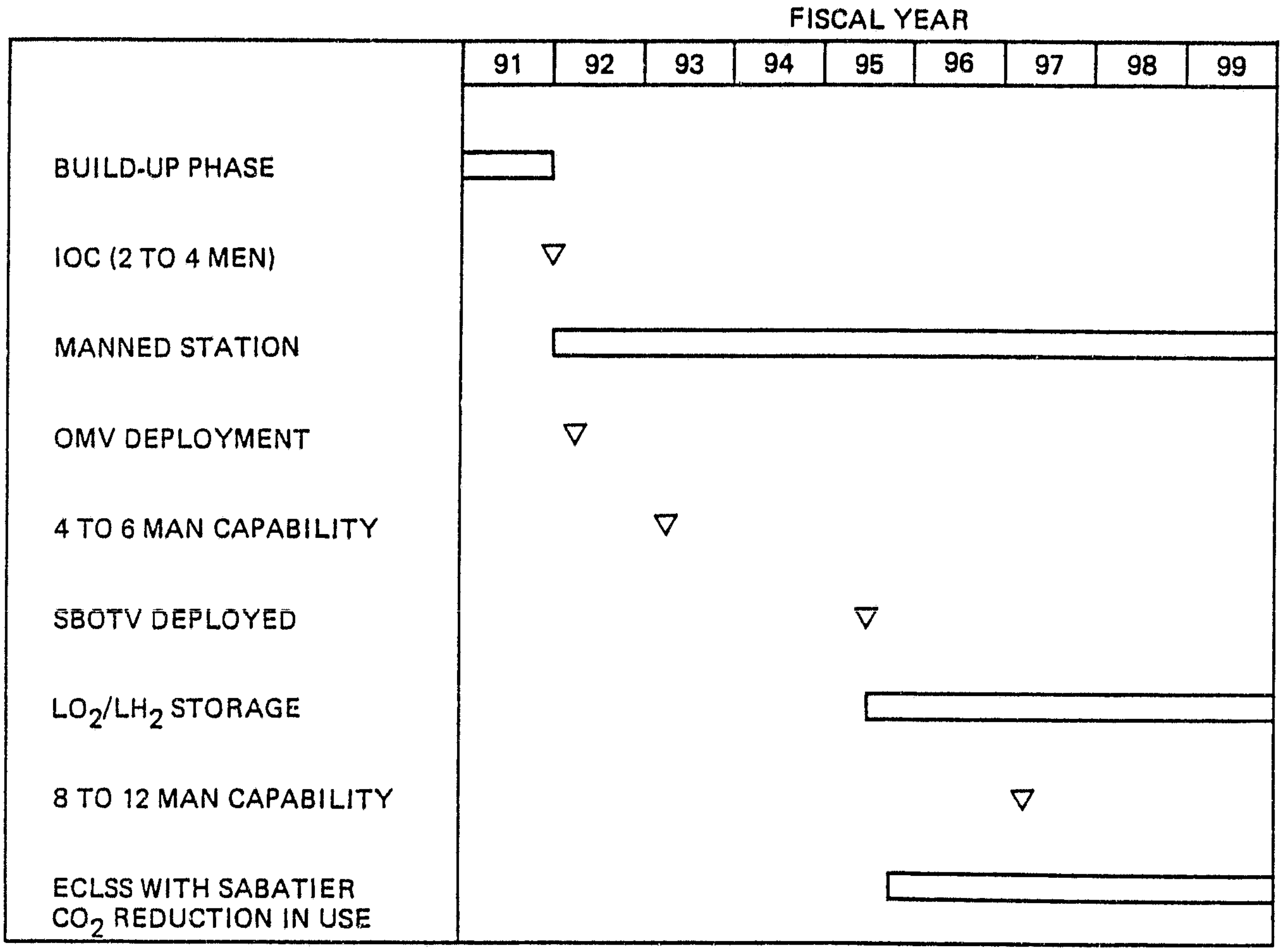

Figure 6-2. Assumed Scenario for Space Station Operational Capability, 1991 Through 1999

may be utilized by the propulsion system as "free" propellant. Other fluids that are used for different subsystems on the Space Station, are N_2 for station atmosphere and N_2 and helium for sensor cooling. Also expected to be available are LO_2/LH_2 from ET/Orbiter scavenging and OTV storage; N_2O_4/MMH from OMS scavenging; and O_2/H_2 from water electrolysis.

The following five subsections describe potential synergistic opportunities. Taking advantage of synergism can reduce Space Station fluid resupply and also reduce, or possibly eliminate, the return of unwanted fluids back to Earth.

6.1.6.1 ECLSS Effluent

The following illustrates potential synergistic opportunities between the propulsion and ECLS systems consistent with this study. The baseline ECLSS developed by NASA's Concept Development Group (CDG), is depicted schematically in figure 6-3. It is expected to produce 17.9 lbm/day of CO_2 for an 8-man crew which, due to a restriction against overboard CO_2 venting, is to be liquified and returned to Earth via the Orbiter. Over the 10-year lifetime of the station, this amounts to 65,335 lbm of CO_2. There are a number of problems associated with this method of CO_2 disposal. The amount of station power required to liquify the CO_2 is significant, there are handling problems, and the liquefication system adds weight and operating concerns. The CO_2 cannot be vented overboard because contamination limits would be violated. CO_2 vented through a nozzle yields 67 sec I_{sp} (assuming the CO_2 is heated to at least 300^oF to avoid solidification within the nozzle). Figure 5-6 shows that the 8- to 12-man station has a 7×10^6 lb_f-sec total impulse requirement for the 10-year time period. Therefore, it is seen that if CO_2 were to be used as a propellant, the requirement for CO_2 is 104,500 lbm or about 1.5 times that available. A CO_2 resistojet, however, can increase the I_{sp} to as high as 200 lb_f-sec/lbm for which case the requirement diminishes to 35,000 lbm; well below that available. Similar conclusions are obtained for the 2- to 4- and 4- to 6-man stations from figure 5-11 when a proportionate reduction in CO_2 production is assumed. Thus, two problems can be simultaneously

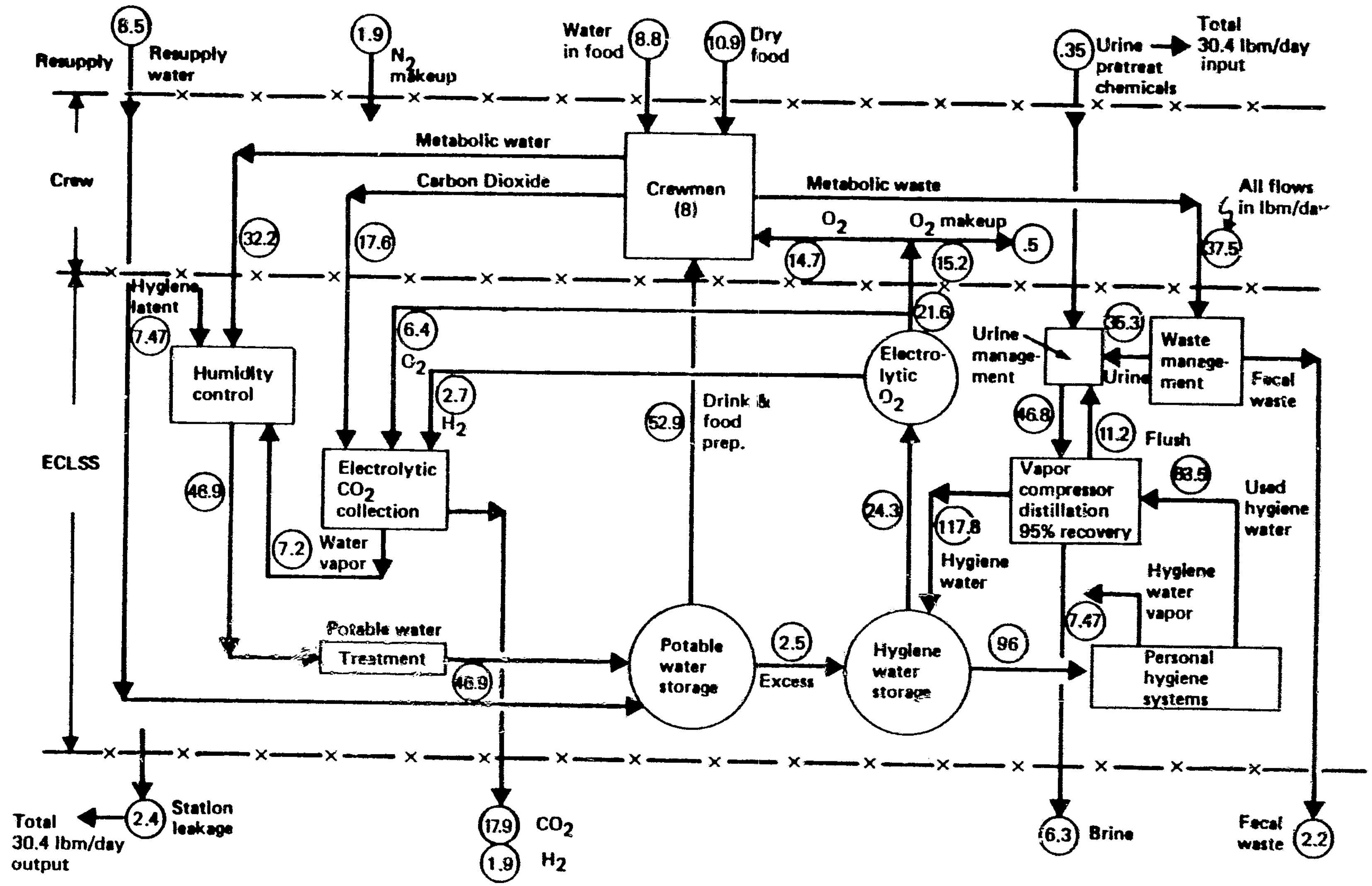

Figure 6-3. Open CO_2 Loop ECLSS Material Balance

resolved by using the CO_2 for propellant: i.e., problems associated with disposal can be eliminated and the total impulse requirements can be satisfied.

The ECLSS is expected to become more "closed" during the life of the station. The introduction of a Sabatier CO_2 reduction system is expected to change the 8-man crew effluent to about 10.6 lbm/day of a CH_4/CO_2 mixture[1] (see figure B-9 of Appendix B). If this mixture can be used with a resistojet and get an I_{sp} of at least 178 lb_f-sec/lbm, there would still be sufficient propellant to satisfy the above requirements.

6.1.6.2 LH_2 Boil-Off

It is currently envisioned that an LO_2/LH_2-propelled space-based OTV will be introduced in about 1995. If this occurs, LH_2 and LO_2 will have to be stored at the station. Boiloff from the LH_2 can be used to cool the LO_2 tank penetrations and effectively eliminate O_2 boiloff. This H_2 boiloff could vary from 13 to 113 lbm/day, depending on the OTV usage rate.[1] The 10-year totals would be 47,450 lbm to 412,450 lbm, respectively. Using the 10-year total mission impulse of 7×10^6 lb_f-sec from figure 5-2 and a cold GH_2 I_{sp} of 283 lb_f-sec/lbm, 24,750 lbm of H_2 is required, which is well below even the low estimate for the H_2 available. If an H_2 resistojet is used and an I_{sp} of 600 lb_f-sec/lbm is assumed, the requirement diminishes to 11,700 lbm for 10 years. Appendix D expands on the use of GO_2/GH_2 and discusses some of the associated problems.

6.1.6.3 External Tank and Orbiter Scavenging

There are also sources external to the Space Station that may have some synergistic possibilities. Scavenging propellants can reduce the quantity and cost of otherwise resupplying propellants. There are two propellant combinations that may be scavenged from the STS. The first is LO_2/LH_2 from

1 Donovan, R., CDG Propulsion Study Issues, Informal correspondence dated October 20, 1983.

the external tank (ET) and the Orbiter. The scavenging of propellants from the ET and the Orbiter has been the subject of considerable investigation.[2] Although a total of 5236 lbm of propellant could be transfered from these sources to the station every 90 days, there are some potential complexities associated with scavenging cryogenic propellants in zero-gravity. This subject is discussed in more detail in Appendix D.

The second propellant combination available is N_2O_4/MMH from the OMS tanks. The OMS propellant quantity onboard the STS is dictated by a worst-case scenario; i.e., if a SSME fails during launch, there is additional fuel (above that normally required) available for the abort maneuvers. If the engines function properly, however, there will be an excess of N_2O_4 and MMH within the OMS which can be used for station and/or free-flyer propellant. Based on the 7×10^6 lb_f-sec 10-year total impulse requirement, used in conjunction with the 340 lb_f-sec/lbm I_{sp} attainable from these propellants, a mass requirement of 20,600 lbm or, 4 STS flights per year, a total of 515 lbm per STS flight would be required. A Rockwell study has shown that an average of 9.075 lbm of N_2O_4/MMH may be recovered per mission[3], almost half the requirement for an entire year.

6.1.6.4 Water Electrolysis

Water electrolysis is a potential source of O_2 for ECLSS and also of O_2/H_2 for propulsion purposes. Propulsively, in order to produce the 11,687 lbm of H_2 required for an H_2 resistojet over a 10 year interval, 105,311 lbm of water must be processed (yielding 93,613 lbm of O_2; enough for a 12-man crew) or 2602 lbm every 90 days (the nominal resupply inverval). The energy required to electrolyze 2602 lbm of water is about 6376 kWh, for an average load of about 3 kW over the 90-day resupply interval.

2 Anon, "Space Operations Center Shuttle Interaction Study Extension Final Review," Rockwell International, Doc. PD82-1A, Contract NAS9-16153, February 1982.
3 "STS Propellant Scavenging Systems Study", Contract NAS9-16994.

If additional water is resupplied for electrolysis and a Sabatier CO_2 reduction system is not used, it may be possible to utilize a GO_2/GH_2 propulsion system with a small increase in energy consumption. An example of such a system is presented in Appendix B.

6.1.6.5 Free-Flyers

One of the functions of the Space Station is to act as a "service station" for a number of free-flyers. This means, in addition to performing maintenance duties on the free-flyers, the station will also be responsible for resuppling propellant. These free-flyers are expected to employ N_2O_4/MMH, hydrazine, nitrogen, carbon dioxide, or hydrogen cold gas thrusters or resistojets. Thus, one or more of these propellants must be stored onboard the station in quantities large enough to satisfy the free-flyers requirements. Additionally, commonality between the free-flyers and station, when possible, will reduce overall program costs.

The sixth and final evaluation criterion as defined in this study, is that synergism with other systems and fluids operating on or in conjunction with the Space Station be considered in the propulsion system selection.

6.1.7 Summary

The six evaluation criteria, as developed in the preceding text, that will be used to select potential Space Station propulsion systems are: 1) it must be able to develop thrust in one or more of three identified thrust ranges; 2) its propellant must meet the volume and mass constraints for the 90-day STS resupply interval; 3) it must meet specific safety and contamination requirements; 4) its power needs must be minimal; 5) it must be capable of being phased into or out of the station or have potential to be combined with another propulsion (thruster) system; and 6) that synergism with other systems and fluids operating on or in conjunction with the Space Station be fully considered.

6.2 Candidate Propulsion Systems

Based on the evaluation criteria defined in section 6.1, a number of propulsion systems were evaluated. The following section will introduce the candidate propulsion systems and select those systems and propellants to be evaluated and discussed in greater detail based on the foregoing factors. The three major categories are ion, monopropellant, and bipropellant systems. Some mono and bipropellants will not be further considered simply because their characteristics are similar to another, more readily available, propellant. Figures 6-4 and 6-5 show both mono and bipropellant propulsion system schematics as examples of possible systems. An arcjet system (not shown) would be similar to that of a resistojet system.

6.2.1 Ion Systems

Ion thrusters produce thrust by electrostatic acceleration of ions extracted from an electron bombardment ionization chamber. The propellants commonly used in ion thrusters are mercury, xenon, and argon. Some of the currently conceived ion propulsion systems include a nuclear ion, thermophotovoltaic ion, and solar thermionic power source. The first type is eliminated to avoid the nuclear complications. The latter two types require a power processing unit to enable the ion thrusters to make use of the raw solar power. However, the ion thrusters require that the solar power be extensively conditioned (up to 12 power supplies). For example, a 50-cm argon ion thruster would require approximately 18 kW of processed power to produce 0.1 lb_f of thrust and a specific impulse of 6600 lb_f-sec/lbm[1] (see Figure E-3). This power requirement would result in a large sizing impact on the solar arrays. In addition, all three propellants have contaminating exhaust plumes and are not synergistic with other systems. Hence for the purposes of this study, ion sytems have been eliminated from further consideration for use on the Space Station.

1. Boeing, "Advanced Propulsion Systems Concepts for Orbital Transfer" Final Volume II; D180-26680-1, 1981.

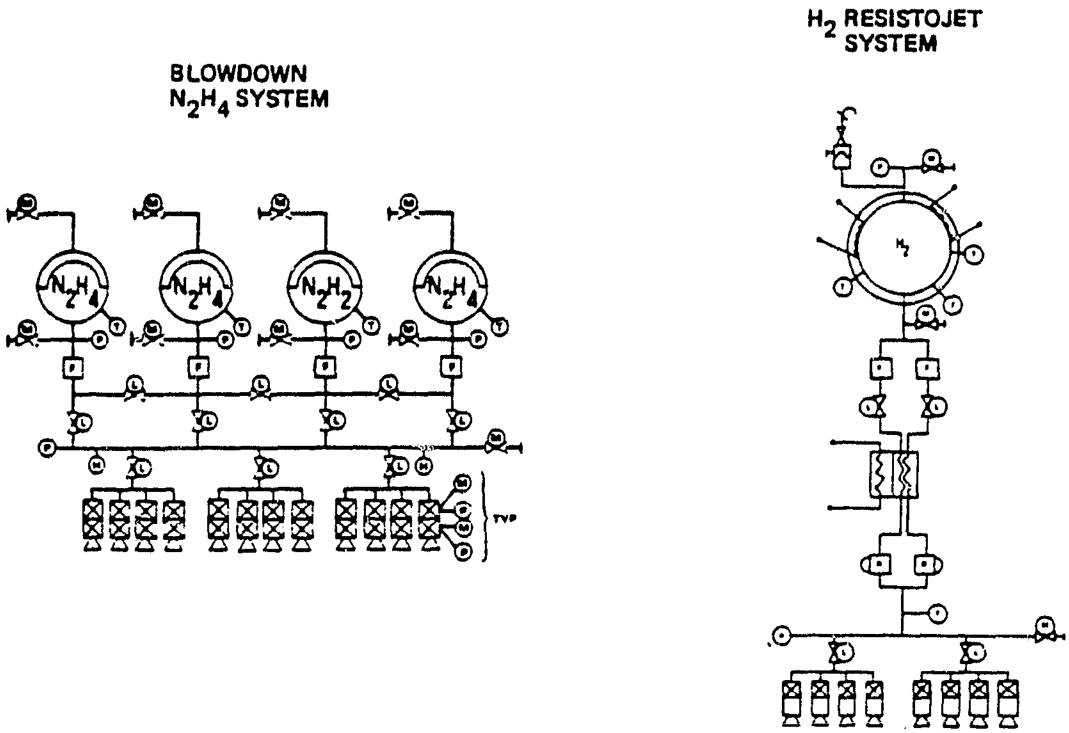

Figure 6-4. Monopropellant Propulsion System Schematics

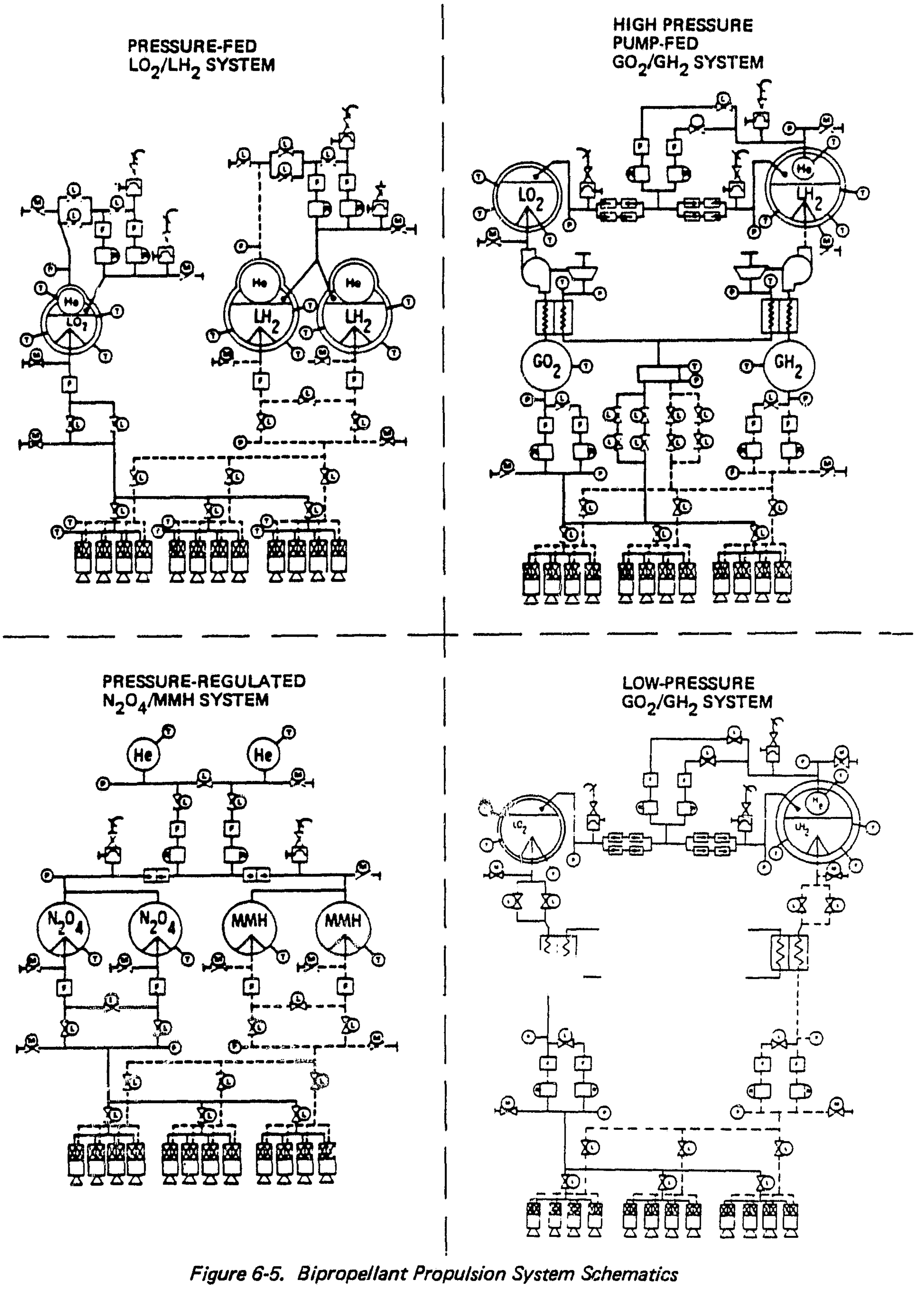

Figure 6-5. Bipropellant Propulsion System Schematics

6.2.2 Monopropellant Systems

The monopropellant systems considered are cold gas, hydrazine (both augmented and unaugmented), resistojets, and arcjets.

6.2.2.1 Cold Gases

A propulsive force can be obtained by allowing a gas to expand through a plenum-nozzle into the vacuum of the Space Station environment. This type of "cold" gas thruster represents a mature state-of-the-art and has a large variety of space-qualified, off-the-shelf hardware. Such thrusters are relatively simple, reliable, cost effective, and easy to develop, virtually any gas can be used, and they deliver very small, precise, and repeatable impulse bits. The most significant disadvantage of a cold gas system is a low specific impulse and the resulting requirement for a large amount of propellant. The primary life-limiting factor associated with a cold gas system is valve cycling, which is directly related to the thrusting strategy used.

Potential Space Station cold gas propellants and their respective specific impulses are shown in table 6-1. The data are based on a chamber temperature of 200^{O}F and chamber pressure of 10 psia. Specific impulse can be improved slightly if the gases are "warmed" as with waste heat. All of these gases will be used in other station subsystems or as an effluent from the ECLSS, and could, therefore, be available to use in cold or warm gas thrusters.

Table 6-1. Potential Cold Gas Propellants

Gas	Specific Impulse* I_{sp} (lb_f-sec/lbm)
H_2	283
H_2O (steam)	110
CH_4	130
NH_3	106
CO_2**	67
N_2	78

*P_c = 10 psia, T_c = 660°R

**CO_2 must be heated to approximately 300°F to prevent solidification within the nozzle.

H_2 is produced from water electrolysis or boiloff from an LH_2 source. H_2O is available from non-recycled waste water, which may or may not be practical to use as a propellant.

As much as 2 lbm per day of CO_2 is generated by each crew member which will be available in varying percentages depending on the degree of ECLSS closure. Methane (CH_4) and ammonia (NH_3) are available from ECLSS with high closure but, of course, the amount of all gases is diminished. N_2 is required for the station atmosphere and sensor cooling. Oxygen is not considered because of priority for its use for crew respiration.

The gases selected for further analysis in this study are therefore H_2, CO_2 and N_2 because they are readily available and represent the full range of specific impulses obtainable from cold gases.

6.2.2.2 Liquid Monopropellants

Liquid monopropellant engines produce thrust by catalytically decomposing a propellant within a catalyst bed. The released energy then causes an increase in temperature which produces a high flow velocity through the nozzle. There are many compounds, or mixtures of compounds, that have been suggested for use as monopropellants. Most have been discarded due to instability and handling requirements. The monopropellants that have been most frequently mentioned are hydrazine, hydrogen peroxide (95% and 98%), hydrazine-hydrainium nitrate, ethylene oxide, nitromethane and tetranitromethane. Only the hydrogen peroxide pair and hydrazine have been used extensively. Hydrogen peroxide tends to deteriorate at about 5% per year and, when the temperature exceeds 350^{O}F, complete decomposition can become spontaneous. Hydrazine, on the other hand, remains stable and can be safely heated in excess of 500^{O}F when in its pure state. Although hydrazine has a higher performance (220 lb_f-sec/lbm) then does either 95% hydrogen peroxide (170 lb_f-sec/lbm) or 98% hydrogen peroxide (180 lb_f-sec/lbm), they are all comparable when measured in a per volume basis (15,000 lb_f-sec/ft^3). When evaluated against the six stipulations defined previously, all three propellants are again comparable except that hydrazine is safer, synergistic with more free-flyers, and hydrazine thrusters exist in any thrust range required by the station. Therefore, for purposes of this study, only hydrazine will be considered for further analysis.

There are two major concerns associated with using a hydazine thruster. The first is its relatively high freezing point (35^{O}F) and the second is the catalyst life. A high freezing temperature could create problems on the Space Station, that were not a concern in previous, smaller spacecraft due to remote, widely separated thruster locations. Also, the catalyst tends to lose particles during the life of a hydrazine thruster. This can create voids in the catalyst bed large enough to collect propellant which then decomposes suddenly causing large chamber pressures and fluctuations. These concerns are addressed in greater detail in section 6.3.1 and possible solutions are proposed in section 6.4.1.

6.2.2.3 Resistojets

Resistojets are strong candidates because of proven technology and potential for high I_{sp}. The equation below shows that the specific impulse of a propellant increases proportionally with the square root of the chamber temperature for a given weight flow, W, thrust coefficient, C_F, and mass, M.

$$I_{sp} = \frac{C_F}{W} \left(\frac{T_C}{M}\right)^{\frac{1}{2}}$$

This temperature effect on specific impulse is the main principle behind a resistojet. The propellant gas undergoes significant heating by using electrical resistance heaters as it flows into the chamber.

The performance of a resistojet varies with both the propellant thermodynamics and the level of energy input. Many resistojet thrusters have been developed and tested during the past 15 years. The most commonly used propellants have been hydrogen, ammonia, hydrazine, and carbon dioxide, all of which will be discussed in further detail in sections 6.3.2 and 6.4.2.

6.2.2.4 Arcjets

A thermal arcjet converts electrical energy to thermal energy by transfering heat from an arc discharge to a propellant; thermal energy is converted to kinetic energy by gas expansion through a nozzle. The arcjet thruster can obtain a specific impulse range of 80 to 1500 lb_f-sec/lbm, depending on the propellant and operating conditions. Some of the propellants that can be used are hydrogen, ammonia, nitrogen, carbon dioxide, helium, argon, lithium hydride, and hydrazine. By their nature, they are high power consumers, and therefore were not carried forward in this study.

There are essentially no current state-of-the-art arcjets because most of the developmental work occurred between 1960-1965, and little work has been done on arcjets since 1966. This may be due to a shift in advanced propulsion research from electrothermal to electrostatic, or electromagnetic forces, which has led to such devices as the electron bombardment thruster (Ion) and the magnetoplasmadynamic (MPD) thruster. However, recent years have seen the resurgence of arcjet development in aerospace industry research and development labs. Hence, even though arcjets are not considered in this study, as a possible propulsion system, their SOA and projected SOA development are discussed in Appendix F.

6.2.3 Bipropellant Systems

A bipropellant propulsion system is similar to a monopropellant system except that it utilizes combustion of an oxidizer/fuel combination to raise chamber temperature thereby obtaining high exhaust velocity. However, although a bipropellant yields higher performance, it is also a more complex system. Many propellant combinations have been used since the development of bipropellant systems, most of which have been discarded in favor of the more energetic or easily handled types. Boron, aluminum, and other additives have been studied in an effort to improve performance. Figure 6-6 lists the bipropellants that were initially considered for the Space Station. From this initial list, a number of the propellant combinations were eliminated for the reasons discussed in the following paragraphs.

All fluorine combinations such as oxygen difluoride and liquid hydrogen (OF_2/LH_2), liquid fluorine and hydrazine (LF_2/N_2H_4), etc., were eliminated due to their extreme toxicity and corrosiveness. Though fluorine does provide the most energetic reactions, its extreme toxicity and corrosiveness negate its use on the Space Station for the forseeable future. Propellants that incorporate exotic additivies like B_2H_6, LC_3H_8, or B_5H_9 have been eliminated because there is limited information on their operating characteristics, especially for small thrusters. Other combinations deleted due to a lack of data for small thrusters were O_2/CH_4 and O_2/RP-1.

Table 6-2 lists the bipropellants that are analyzed in greater detail and shows the usually used mixture ratio, vacuum specific impulse, and the bulk density. Additional information on these propellants is presented in table 6-3. There are a number of drawbacks concerning the use of UDMH. First, there is currently very limited production capability and it is expensive to produce. Also, when combined with N_2O_4, its performance is similar to the N_2O_4/MMH combination, but the exhaust products have more

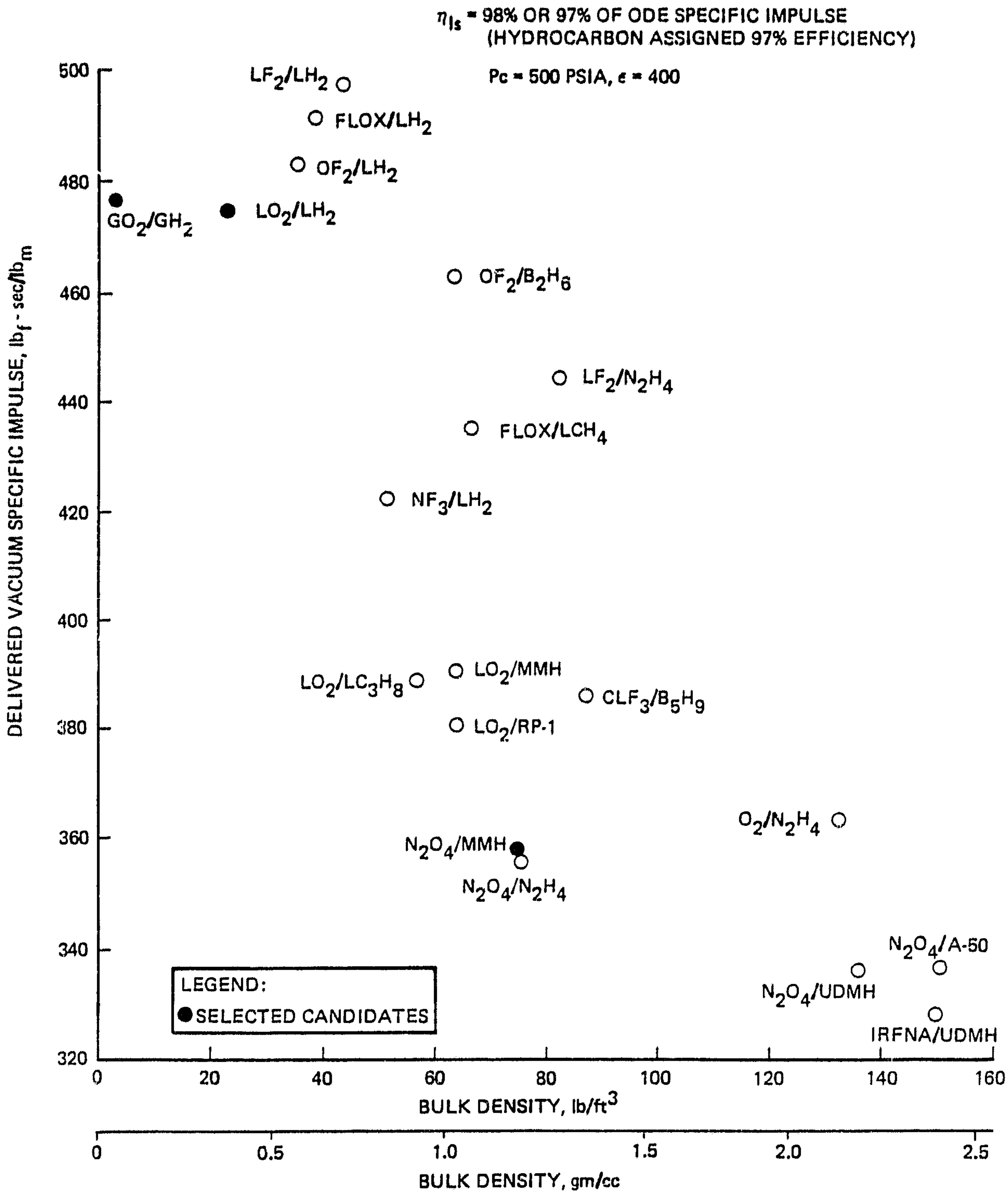

Figure 6-6. Comparison of Specific Impulse and Density for Cadidate Propellants

Table 6-2. Bipropellant Options

OXIDIZER	FUEL	MIXTURE RATIO (O/F)*	SPECIFIC IMPULSE, sec $\epsilon = 40$, $P_c = 300$ psia	BULK DENSITY (gm/cc)
LO_2	LH_2	4.5	456	0.31
N_2O_4	N_2H_4	1.2	342	1.2
N_2O_4	A-50	2.0	340	1.19
N_2O_4	MMH	2.2	340	1.19
N_2O_4	UDMH	2.6	337	1.17
IRFNA+	UDMH	3.0	325	1.25
O_2	N_2H_4	1.0	368	1.07
GO_2	GH_2	4.7	457	0.025

* BY WEIGHT

\+ INHIBITED RED FUMING NITRIC ACID

Table 6-3 Propellant Characteristics

PROPELLANT	HEAT OF COMBUSTION		DENSITY (lb/ft^3)		TEMPERATURE (°F)	
	(BTU/lb)	(BTU/ft^3)	GAS AT 0°C, 1 atm	LIQUID AT B.P.	BOILING AT 1 atm	FREEZING AT 1 atm
OXYGEN			0.0892	71.24	-297.4	-361.0
HYDROGEN	51,500	222,000	0.0056	4.43	-422.97	-434.4
HYDRAZINE (N_2H_4)	8,345	525,000		62.90	236.3	34.7
NITROGEN TETROXIDE (N_2O_4)				90.23	70.1	11.8
A-50*	N/A	N/A		56.38	158.0	19.0
MMH (CH_3NHNH_2)	12,180	665,760		54.66	192.5	-62.5
UDMH ($(CH_3)_2N_2H_2$)	14,190	696,445		49.08	146.0	-71.0
IRFNA				96.64	77.0	-63.4

*A-50 IS COMPOSED OF 50% HYDRAZINE AND 50% UDMH. THE HEAT OF COMBUSTION IS EXPECTED TO BE APPROXIMATELY 600,000 BTU/ft^3

contaminants. Therefore, the N_2O_4/MMH combination is carried and the N_2O_4/UDMH combination is dropped. The N_2O_4/A-50 combination was also eliminated because A-50 is 50% UDMH. The IRFNA/UDMH combination is discarded because of the UDMH component and poor performance. The performance characteristics of N_2O_4/N_2H_4 and N_2O_4/MMH are similar, but the N_2O_4/N_2H_4 combination has been known to cause combustion instabilities under certain conditions. All three propellants will be available at the station since many free-flyers will use hydrazine and the OMV and Orbiter will use N_2O_4/MMH. In view of the foregoing, and since there is considerably more information available for engines using MMH, the N_2O_4/N_2H_4 combination is dropped in favor of N_2O_4/MMH. Therefore, the remaining bipropellants are N_2O_4/MMH and O_2/H_2.

The main reasons for choosing N_2O_4/MMH and O_2/H_2 are (1) they span the range of specific impulses for the bipropellants of interest; (2) they represent a fairly large range of bulk densities for bipropellants; (3) N_2O_4/MMH has proven technology for the thrust ranges desired; (4) O_2/H_2 thrusters are currently being tested and qualified for the desired thrust ranges; (5) they are both synergistic with other systems on, or in conjunction with the Space Station; and (6) they are relatively safe to handle.

Of the two propellant combinations chosen, only oxygen and hydrogen are considered in both their gaseous and liquid states. Although both these and the supercritical states are viewed as possible for storage purposes, only GO_2/GH_2 thrusters will be considered. This is done in recognition of the current state of development of small, liquid oxygen/liquid hydrogen thrusters and the complexity of a cryogenic distribution system on-board the Space Station. For these reasons only GO_2/GH_2 and N_2O_4/MMH thrusters will be considered as potential candidates for a bipropellant propulsion system.

6.3 State-of-the-Art for Retained Propulsion Options

This section assesses current state-of-the-art technologies that are applicable to Space Station propulsion requirements. "State-of-the-art technologies" include: (1) systems and propellants that have been used in space propulsion applications or have been tested under similar conditions; and (2) technologies that have been shown analytically to have such a capability and do not rely on largely unproven concepts or hardware elements.

Thruster concepts and propellants that have been retained from the initial screening and could satisfy either high and/or low thrust requirements include: (1) conventional monopropellant thrusters that use hydrazine or cold or warm nitrogen, hydrogen, or carbon dioxide gas; (2) heated gas thrusters (resistojets) operating on hydrogen, carbon dioxide, ammonia, or hydrazine; and (3) bipropellant thrusters that use either nitrogen tetroxide and monomethyl hydrazine or gaseous oxygen and gaseous hydrogen. The following subjects of interest are addressed:

1) Performance characteristics (thrust, I_{sp}, pulsing capability) of each hardware/propellant combination;

2) Physical characteristics (size and weight) of the hardware required for various performance levels; including requirements (tank size, weight, pressure, temperature, exhaust composition) imposed by each propellant;

3) Penalties imposed upon each concept by adding throttling, gimballing, and/or orbital replacement capabilities.

The information presented comes from technical publications, industry and BAC analyses. Some of the technologies discussed are applicable across a wide range of thrust levels. As was discussed in section 6.1.1 on thrusting strategies, various thrust levels may exist for different propulsion requirements. Hence, depending on how each individual maneuver is handled (i.e., to minimize propellant consumption) a potential thrust band or range exists from 0.01 to 500 lb_f. Since one system will, most likely, not be optimal in all situations, a breakdown of the thrust band in which each system performs most efficiently is defined. Table 6-4 shows a breakdown of the thrust band and thrust region of emphasis for each system.

6.3.1 Conventional Monopropellant Thrusters

Two types of conventional monopropellant thrusters are of interest. The first and simplest of these systems allows gaseous propellant to expand through a nozzle. In the second type, a catalytic or thermally-induced reaction produces hot gas from a liquid propellant. This system is more complicated because it adds a reactor, but it generally yields a higher performance.

6.3.1.1 Performance Characteristics

RRC has tested a nozzle of 30.8:1 area ratio using gaseous N_2 propellant at inlet pressures ranging from 100 to 235 psia. For gas temperatures varying from $-100^{o}F$ to $150^{o}F$, measured vacuum specific impulse ranged from 55 to 79 lb_f-sec/lbm. The range of measured thrust levels was 0.8 to 2.1 lb_f.

No test data were located for CO_2 monopropellant thrusters. An analysis was performed in the projected capability assessment to evaluate the theoretical performance of CO_2 expanding through a nozzle. A possible complication is that CO_2 would freeze as it expands, thereby limiting performance. Two possible solutions to this problem are to have heater elements along the nozzle or to preheat the propellant.

Hydrazine thrusters can also produce thrust levels applicable to the Space Station. Numerous engines in general use lie in the 0.1- to 1-lb_f and 5-lb_f thrust classes, and essentially any engine desired can be built. Steady-state firing produces I_{sp} values from 220 to 230 lb_f-sec/lbm, depending on the size. Duty cycles as low as isolated single 10-msec pulses have been demonstrated and employed, with a 0.1-lb_f thruster yielding impulse bits of 0.002 lb_f-sec and an I_{sp} of 110 lb_f-sec/lbm in this mode. A typical example of a hydrazine thruster using a catalyst bed for dissociation is shown in figure 6-7. This is a nominal 5-lb_f thruster that has been qualified for 45,500 lb_f-sec total impulse, 500,000 pulses,

Table 6-4. Defined Thrust Ranges for Various Systems

	Thrust band (lb_f)	Thrust region emphasized (lb_f)
Chemical bipropellants		
N_2O_4/MMH	0.01 to 100	5.00 to 100
O_2/H_2	0.01 to 500	5.00 to 100
Monopropellants		
N_2H_4	0.01 to 100	0.10 to 100
N_2 (Blowdown)	0.01 to 10	0.05 to 5
CO_2 (Blowdown)	0.01 to 10	0.05 to 5
H_2 (Blowdown)	0.01 to 10	0.05 to 5
Resistojets		
N_2H_4, H_2, NH_3, CO_2 — isp 120 to 800 depending on species, power level	0.05 to 1.0	0.05 to 1.0

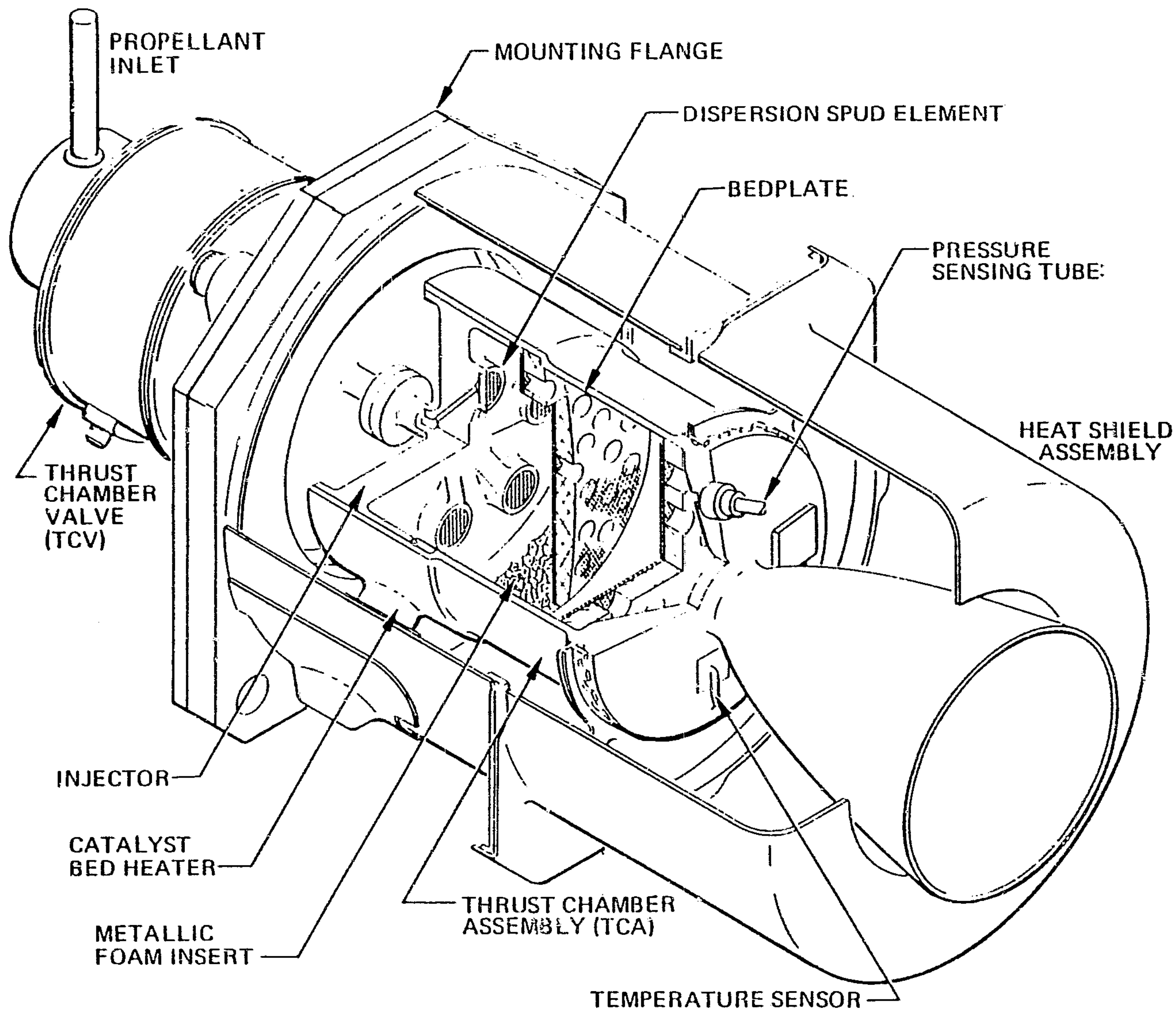

Figure 6-7. Hydrazine Thruster (P-95 MR-50 5-lb$_f$)

and a thrust range of 2 to 8.9 lb_f. Hydrazine thrusters have been used on Intelsat V, GPS, Voyager, Viking, and other spacecraft and is planned for use in the Gamma Ray Observatory (1989).

6.3.1.2 Hardware Physical Characteristics

Conventional monopropellant thrusters are among the smallest and lightest available. Table 6-5 shows the weights and envelope sizes for many typical hydrazine engines. The data presented in this table refer to specific RRC, Hamilton Standard, and Bell Aerospace engines, and are presented as typical values. Information in this table was taken from envelope drawings and product data sheets. Gaseous monopropellant thrusters, which would not require reactors, would be somewhat smaller and lighter.

6.3.1.3 Propellant Quantities and Tank Sizing

All of the three monopropellants being considered are storable as liquids. This is generally preferable as a storage mode, even though pressurization or refrigeration would be required to liquify CO_2 or N_2. Table 6-6 shows the quantities of these propellants that would be required for a 90-day period (assuming the Orbiter is docked for 14 of those days). The study assumes a NASA neutral atmosphere at an altitude of 525 km and an Earth-oriented, 28.5-deg inclination for a 2- to 4-man station.

The CO_2 I_{sp} value in table 6-6 is a theoretical calculation and assumes the CO_2 is heated to 300^{o}F prior to expansion. The N_2 value is midrange for observed I_{sp} values in RRC nozzle tests. N_2 is assumed to be cryogenically stored, and the mass of any refrigeration elements that might be required is not included.

6.3.1.4 Exhaust Constituents

Both CO_2 and N_2 are chemically stable during expansion, although the CO_2 becomes solid below -200^{o}F. Hydrazine exhaust plumes consist mainly of H_2, N_2, and NH_3, proportions of which vary with the extent of NH_3 dissociation.

Table 6-5. Hydrazine Engine Sizes and Weights (Including Valves)

THRUST, lb_f	MANUFACTURER	LENGTH, in.*	DIAMETER, in.*	WEIGHT, lb
0.1	ROCKET RESEARCH	5.78	1.35	0.53
0.2	BELL AEROSPACE	6.12	–	0.86
0.2	HAMILTON STANDARD	5.5	0.9	0.63
0.2	ROCKET REASEARCH	5.78	1.35	0.73
0.5	HAMILTON STANDARD	7.4	1.6	0.68
0.5	ROCKET REASEARCH	8.6	1.4	0.78
1.0	HAMILTON STANDARD	5.48	0.93	0.68
1.0	ROCKET RESEARCH	6.6	1.4	0.73
5.0	HAMILTON STANDARD	8.79	1.45	1.13
5.0	ROCKET RESEARCH	5.78	1.3	1.08
8.0	ROCKET RESEARCH	8.80	1.36	1.56
12.0	HAMILTON STANDARD	6.58	1.89	1.44
30.0	HAMILTON STANDARD	9.60	2.42	2.19
40.0	HAMILTON STANDARD	11.50	3.30	2.17
40.0	ROCKET RESEARCH	7.38	2.94	1.96
100.0	ROCKET RESEARCH	15.5	4.40	4.18
125.0	HAMILTON STANDARD	16.6	5.83	3.78

*MAXIMUM ENVELOPE

Table 6-6. Monopropellant Requirements for a 90-Day Period

Parameter	Propellant		
	CO_2	N_2	N_2H_4
I_{sp}, lbf-sec/lbm	55	67	230
Total required, lbm*	4671	3834	1117
Volume, ft^3**	116	84	18
Tank material	Ti 6AL-4V	AL 2219	Ti 6AL-4V
Tank weight (est), lbm	910	72	40
Tank pressure, psia	1000	20	20
Tank temperature, °F	75	-320	75

*Includes 50% contingency for attitude control backup, desaturation, docking disturbances, etc.

**Includes 10% ullage for liquids only

Table 6-7 shows a breakdown of the exhaust constituents for a 0.1 lb_f Hamilton-standard hydrazine thruster.[1] This exhaust sample was obtained while firing the engine for a 60-sec steady state run at an inlet pressure of 195 psia, with an unknown propellant composition. It has been shown in previous experiments that the water survives its passage through the catalyst bed, so no attempt was made to determine the amount of water in the gas sample. In some cases, methane was detected in various amounts, which was caused by the breakdown of the aniline within the thruster catalyst bed. As much as 1.6%, by mass, of unreacted hydrazine at a pulse-width of 1 sec. was detected in the exhaust plume. It is expected that a greater amount of unreacted hydrazine will appear with a duty cycle of 100 msec.

6.3.1.5 Throttling and Installation Penalties

There are many circumstances that exist, in which the throttling of an engine is required. Acceleration, deceleration, and precision maneuvering are but a few. The actual throttling process can be performed in three different ways. These are flow control, pulsing, and gimballing; each of which can be performed in different ways.

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1 Baerwald, R. K., and Dassamaneck, R. S., JPL, "Monopropellant Thruster Exhaust Plume Contamination Measurements," AFRPL-TR-77-44.

Table 6-7. Exhaust Constituents for a Representative Hydrazine Thruster

CONSTITUENT	MASS IN SAMPLE (grams)	MASS (%)	CONDENSATION TEMPERATURE AT 10^{-4} Pa MOLSINK* PRESSURE (°K)
H_2	2.8×10^{-2}	8.10	4
N_2	2.3×10^{-1}	66.43	26
NH_3**	8.37×10^{-2}	24.20	101
H_2O	+	0.71	159
N_2H_4	40×10^{-6}	0.01	165
ANILINE	+	0.55	190

* MOLECULAR SINK VACUUM CHAMBER

** PERCENT AMMONIA DISSOCIATION EQUALS 65.5% AND IS CALCULATED USING THE MOLE FRACTION RATIO OF H_2/NH_3 TO ACCOUNT FOR ANY DISSOLVED N_2 PRESSURANT IN THE PROPELLANT

\+ ASSUMED TO BE THE SAME AS FOUND IN THE PROPELLANT

Monopropellant thrusters can be throttled either by regulating feed pressure or by cycling propellant flow control valves. Either action causes a loss in efficiency as heat losses become larger fractions of total energies at lower thrust levels. The efficiency loss varies with thruster size and duty cycle. Specific impulse for N_2H_4 ranges from around 230 sec steady-state to as low as 100 sec for a single 10-msec pulse. Also, the lower impulse bits tend to increase the percentage of unreacted hydrazine in the exhaust plume as discussed in section 6.3.1.4. Cycling the valves also require a power input and may affect valve life. Penalties associated with gimballing or orbital replacement capabilities include the weight of hardware required to implement these capabilities and cosine losses associated with gimballing. Flexible feed lines, gimbals, actuators, disconnect fittings, and other parts should not affect the behavior of monopropellant thrusters.

6.3.2 Augmented-Gas Thrusters (Resistojets)

Resistojets differ from conventional monopropellant thrusters by adding energy to the working fluid after its introduction or decomposition and prior to expulsion. In resistojets produced or tested to date, the augmentation energy has been input via an electrically-powered heat exchanger.

6.3.2.1 Performance Characteristics

Several resistojet thrusters have been developed and tested. RRC and TRW have delivered flight units that operate on hydrazine, and AVCO has used hydazine in test units. Marquardt Corporation and TRW have tested resistojets using NH_3 and H_2. NASA (LeRC) has also tested a hydrogen resistojet. RRC has analyzed the performance of its Augmented Catalytic Thruster (ACT), which is designed to use hydrazine, NH_3, H_2, or CO_2. Table 6-8 summarizes the thrust and specific impulse ranges exhibited by the various thruster/propellant combinations tested or analyzed to date. Additional performance data will be discussed in detail in section 6.5.

Table 6-8. Observed or Analyzed Resistojet Thrust and I_{sp} Ranges

	PROPELLANT:	N_2H_4	NH_3	H_2	CO_2
AVCO	F, mlb_f	1.5 - 17	—	—	—
	I_{sp}, sec	120 - 235	—	—	—
ERNO	F, mlb_f	11.2 - 45	—	—	—
	I_{sp}, sec	300	—	—	—
MARQUARDT	F, mlb_f	—	8.8 - 17	9.3 - 14.5	—
	I_{sp}, sec	—	120 - 320	230 -590	—
NASA	F, mlb_f	—	—	600 - 1000	—
	I_{sp}, sec	—	—	580 - 710	—
RRC	F, mlb_f	30 - 200	24 - 68*	43 - 117*	17 - 44*
	I_{sp}, sec	290 - 305	208 - 360*	340 - 690*	165 - 220*
TRW	F, mlb_f	41 - 79	45 - 49	31 - 33	—
	I_{sp}, sec	292 - 315	234 - 256	510 - 550	—

*PREDICTED BY ANALYSIS; ONGOING NASA FUNDED EFFORT AT RRC WILL TEST NH_3 AND H_2 PROPELLANTS

6.3.2.2 Hardware Physical Properties

Resistojets have specific power requirements that are on the order of several watts per millipound thrust. Limitations on available electrical power have previously confined thrust levels to below 1-lb_f. Table 6-9 presents weights and envelope sizes for representative resistojets of the radiating-wire, coiled-tube, and vortex-chamber types.

Resistojet heat exchangers have been built in three general configurations. Two of these are shown in figures 6-8 and 6-9. Radiating-wire heat exchangers use refractory-metal structures heated by filaments, generally of tungsten. Coiled-tube thrusters use thin-walled gas transfer tubes as resistive heaters. The TRW HiPEHT passes the gas directly over the heater element in a vortex chamber.

The suitability of these designs for pulse-mode operation decreases in the order as listed in table 6-8. Due to their relatively large volumes, none of these devices generate a crisp pulse shape; they are generally unsuitable for use where pulse centroids are closely constrained. The thermal inertia of the heat-exchanger structure buffers it against temperature excursions, so a radiating-wire resistojet can be off-pulsed (propellant flow interruped); the RRC ACT has operated at duty cycles as low as 3 per cent, with off-times up to 7.7 sec.

A lack of thermal inertia can have serious consequences. Coiled-tube thrusters experience rapid rises in tube temperature when flow is interrupted, which may shorten or end tube life. The vortex-chamber resistojet's heating element operates at a high power density, and can be damaged in a very short time period. This time period is determined by the amount of power in use, the thermal-transient time constant of the heater element, and factors such as coil geometry and emittance of the element. For a given design it is necessary to ensure that if the flow is interrupted, element temperatures will not rise to unacceptable levels. If other heat-rejection mechanisms do not provide such assurance, electronic means must be provided to monitor temperature and to shut off the current flow if tolerances are exceeded. The TRW HiPEHT has a controller that incorporates

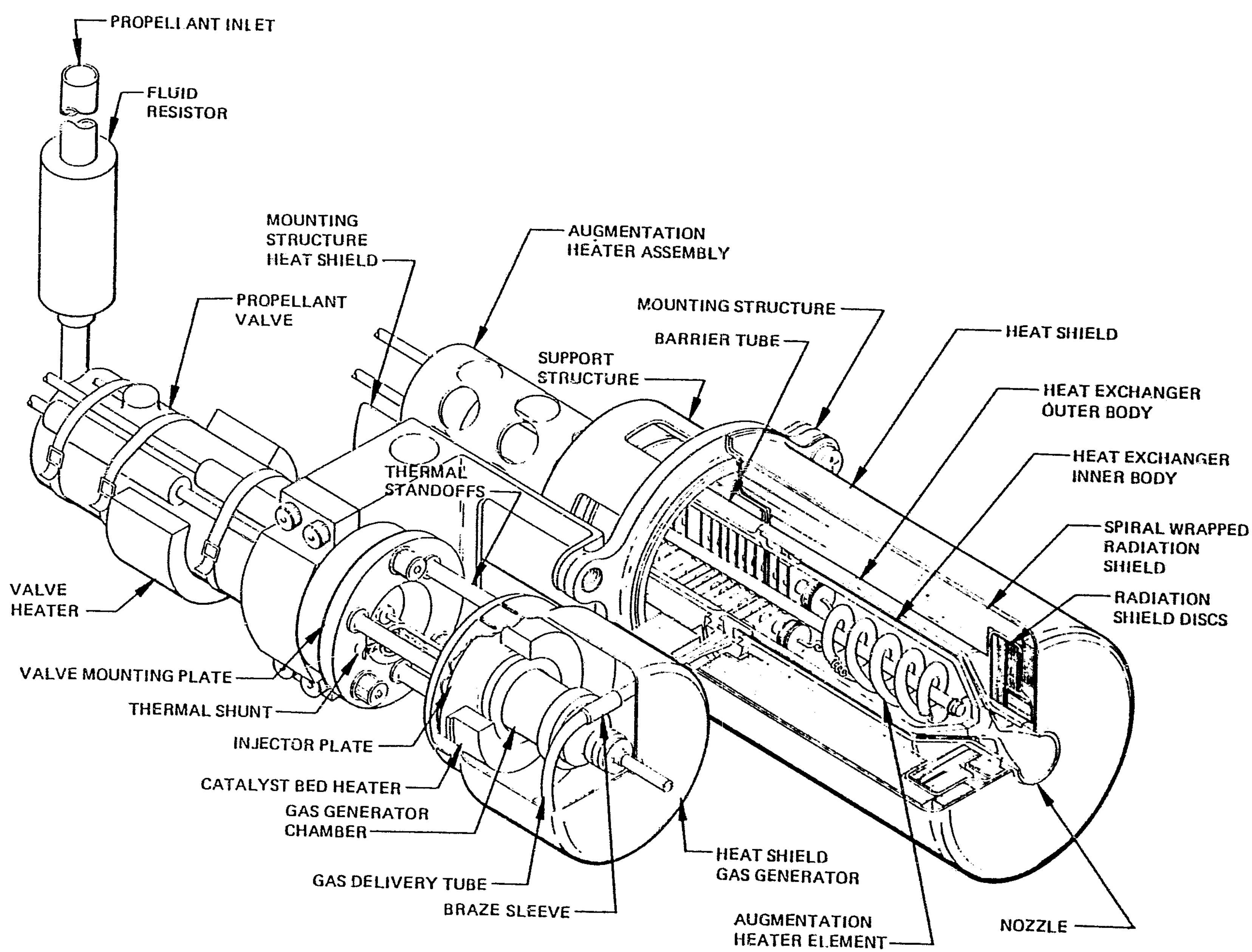

Figure 6-8. Augmented Catalytic Thruster

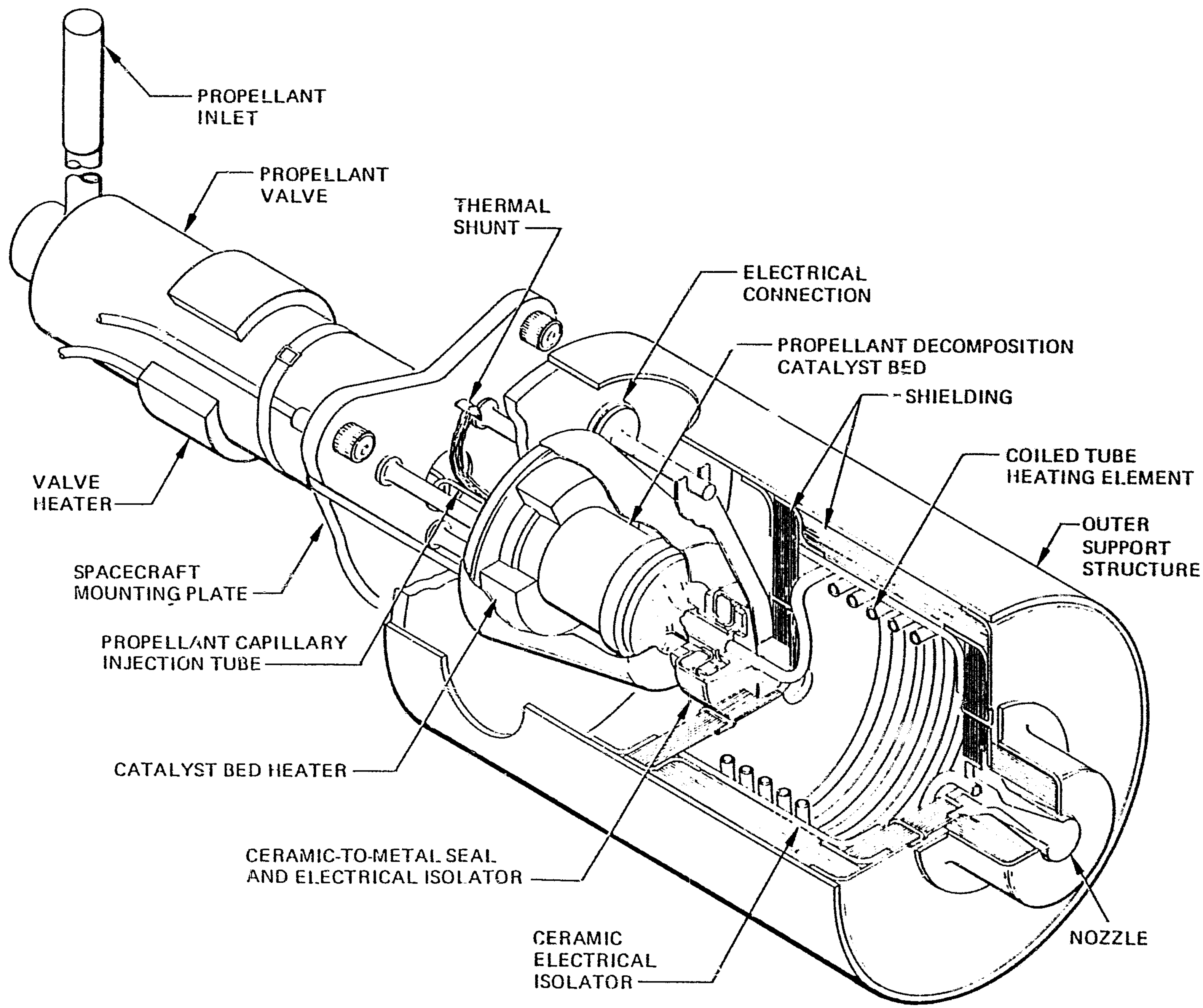

Figure 6-9 Coiled Tube Augmented Catalytic Thruster

Table 6-9. Resistojet Sizes and Weights

DEVICE	THRUST, mlbf*	ENVELOPE SIZE, in	WEIGHT, lb
RRC ACT	30 - 200	7.65 x 3.48 x 2.1	1.8
RRC CTAT	25 - 95	8.44L x 2.47 dia	1.3 (EST)
TRW HIPEHT	35 - 115	5.40 x 1.75 x 1.78	0.8
SCALED - UP ACT	450	9.96 x 4.88 x 3.13	3.5 (EST)

*OPERATING ON N_2H_4

Table 6-10. Resistojet Propellant and Storage Tank Requirements for a 90-day Period

PROPELLANT	N_2H_4	NH_3	H_2	CO_2
SPECIFIC IMPULSE, sec*	300	240	500	200
TOTAL REQUIRED, lbm **	856	1070	514	1284
VOLUME, ft^{3+}	15	32	130	32
TANK MATERIAL	Ti 6AL-4V	Ti 6AL-4V	AL 2219	Ti 6AL-4V
TANK WT, lbm^{++}	36	88	96	250
TANK PRESSURE, psia	20	150	20	1000
TANK TEMPERATURE, °F	75	75	-425	75

* FOR A MIXTURE RATIO OF 1.6:1

**INCLUDES 50% CONTINGENCY (I.E. ATTITUDE CONTROL BACKUP, DOCKING DISTURBANCES, ETC....)

+ INCLUDES 10 % ULLAGE FOR LIQUID PROPELLANTS ONLY

++ TANK WEIGHT ASSEMBLY = $K_{T+A}\, K_{L/D}\, 1.5\, (\rho/F_{T_u})(V)$ (UFS x P max oper) a monitor for this purpose.

6.3.2.3 Propellant Quantities and Tank Sizing

Of the resistojet propellants studied, hydrogen poses the most challenging storage problem due to its low density, high leak/boiloff rate, and possible refrigeration requirements. Hydrazine problems include toxicity and its high freezing temperature ($35^{O}F$). A constraint associated with using carbon dioxide is its relatively low temperature at which it begins to dissociate (2000^{O}F). Table 6-10 shows the propellant quantities required for a 90-day period (assuming the Orbiter is docked for 14 of those days). The study assumes a NASA neutral atmosphere at an altitude of 525 km and an Earth-oriented, 28.5-deg inclination, 2- to 4-man station. The table represents resistojet I_{sp} levels, and the associated storage system parameters.

6.3.2.4 Exhaust Constituents

The behavior of N_2H_4 exhaust has been discussed in the section on monopropellant thrusters. NH_3 may dissociate, producing N_2 and H_2. The constituents of H_2 and CO_2 exhausts will be discussed in section 6.4.

6.3.2.5 Potential Throttling and Installation Penalties

As noted in the section on performance characteristics, some types of resistojets are not effective as pulse-mode devices. Their low thrust, however, balances the need for very short pulse durations. The fact that these thrusters have very low thrust levels may obviate the need for short pulse widths on vehicles as large as the Space Station. Beyond these restrictions, the comments made with respect to throttling and installation of conventional monopropellant engines apply to resistojets. Throttling can be accomplished by regulating propellant feed pressure and heater power. The only penalties incurred in a gimballed or removable installation would be the weight of the additional hardware and the cosine losses associated with gimballing.

6.3.3 N_2O_4/MMH Bipropellant Thrusters

Engines employing N_2O_4/MMH propellants have been used for years and most 0.1 to 150 lb_f bipropellant thrusters have used N_2O_4/MMH. Perhaps the most visible N_2O_4/MMH thruster application is on the Space Shuttle, which uses six 25-lb_f thrusters for orbit adjustment and attitude control. The four major manufacturer's that provide N_2O_4/MMH thrusters for the desired thrust range are Aerojet, Bell Aerospace, Marquardt, and Rocketdyne.

6.3.3.1 Performance Characteristics

N_2O_4/MMH thrusters have demonstrated specific impulse values in the 160 to 320 lb_f-sec/lbm range depending on whether the engine is designed to run steady-state or in a pulse mode. The pulsing engines normally perform in the 160 to 200 lb_f-sec/lbm range. The lower performance is caused by engines not running long enough to reach steady-state temperatures.

For steady-state thrusters, a typical range is 240 to 320 lb_f-sec/lbm depending on the thrust level, chamber pressure, expansion ratio, and mixture ratio. Figure 6-10 shows a performance spread for thrusters in the range from 0.5 to 600 lb_f. As the figure shows, the higher thrust engines perform better than the lower thrust engines (i.e., below the 10-lbf range). Also, the normal nozzles have a higher performance than the short or scarfed nozzles for the same thrust range. Figures 6-11 through 6-14 show examples of I_{sp} performance for 5 and 100 lb_f thrusters as functions of mixture ratio and thrust level. The data for these figures was generated by The Marquardt Co.[1] Table 6-11 shows performance variations of thrusters in the 0.5 lb_f to 325 lb_f thrust range.

6.3.3.2 Hardware Physical Properties

Table 6-12 shows the weights and envelope sizes of thrusters in the 0.1 through 150 lb_f range that are produced for the four major companies. The

1 Marquardt Co., "Propulsion Systems and Engines for Satellites," A-82-7-1990.

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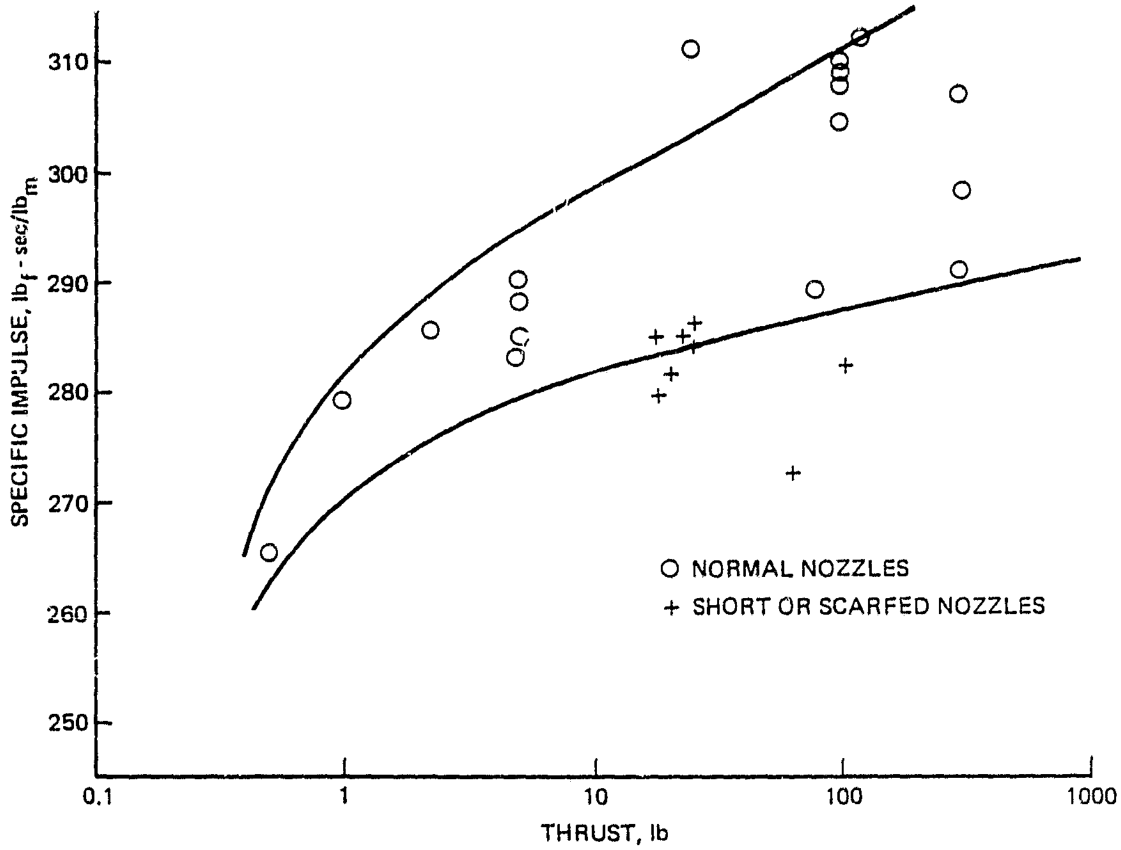

Figure 6-10. Specific Impulse Trending from Existing N_2O_4/MMH Thruster Data

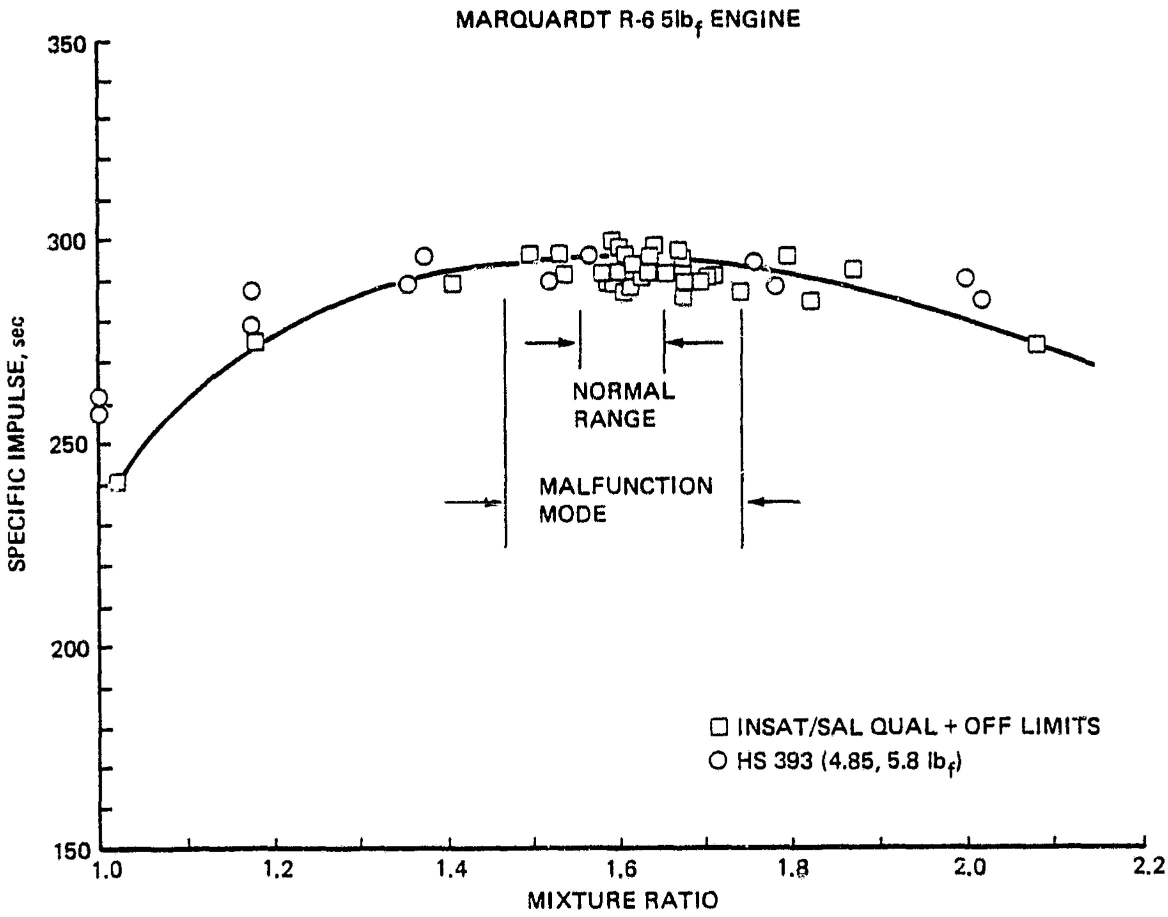

Figure 6-11 Performance of a N_2O_4/MMH 5-lb_f Engine for Varying Mixture Ratios

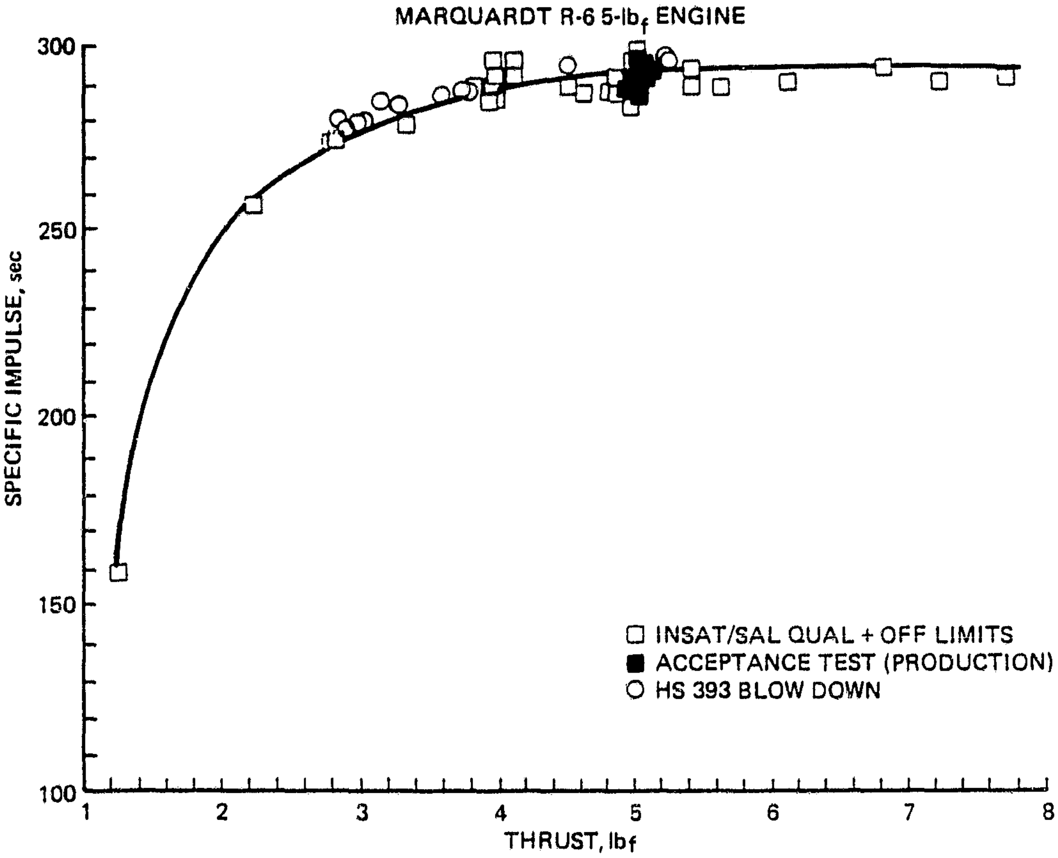

Figure 6-12. Performance of an N_2O_4/MMH 5-lb$_f$ Engine for Varying Thrust Levels

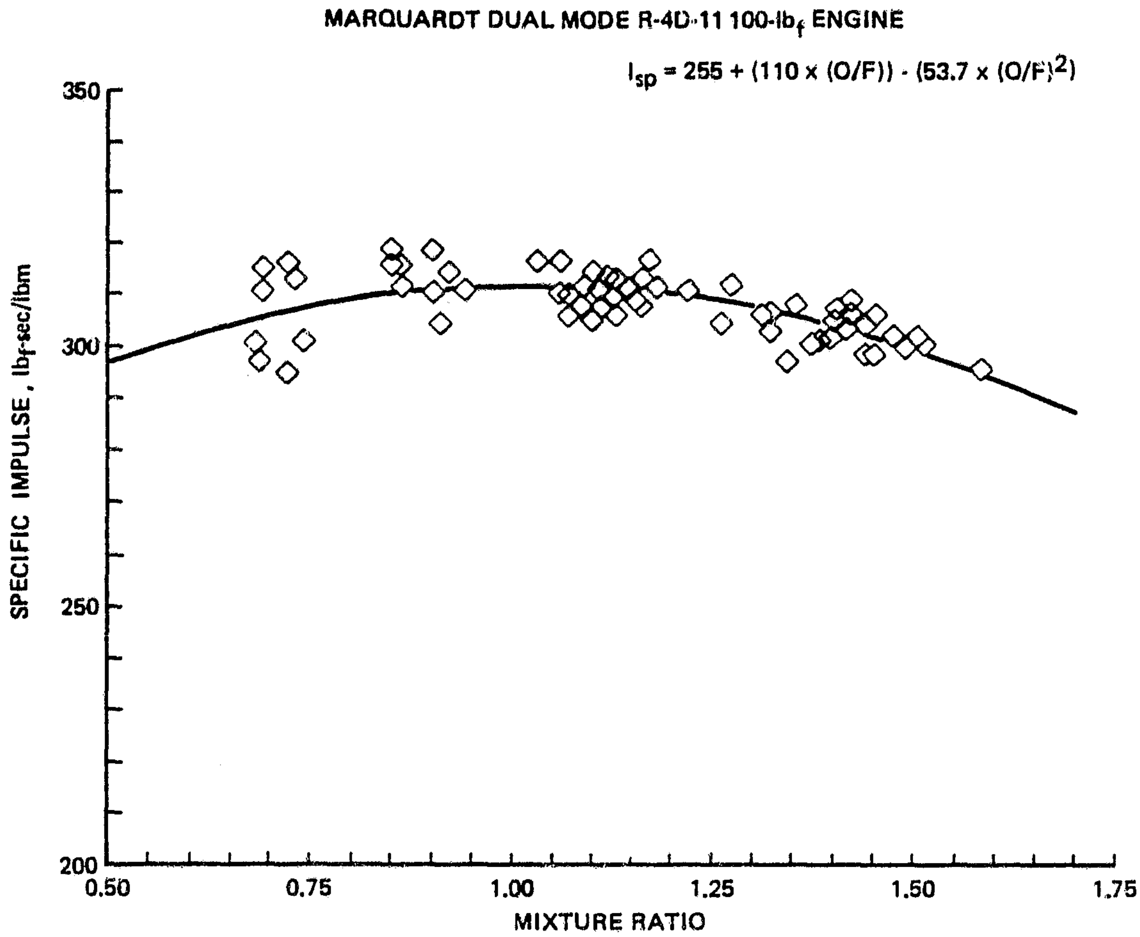

Figure 6-13. Performance of an N_2O_4/MMH 100-lb_f Engine for Varying Mixture Ratios

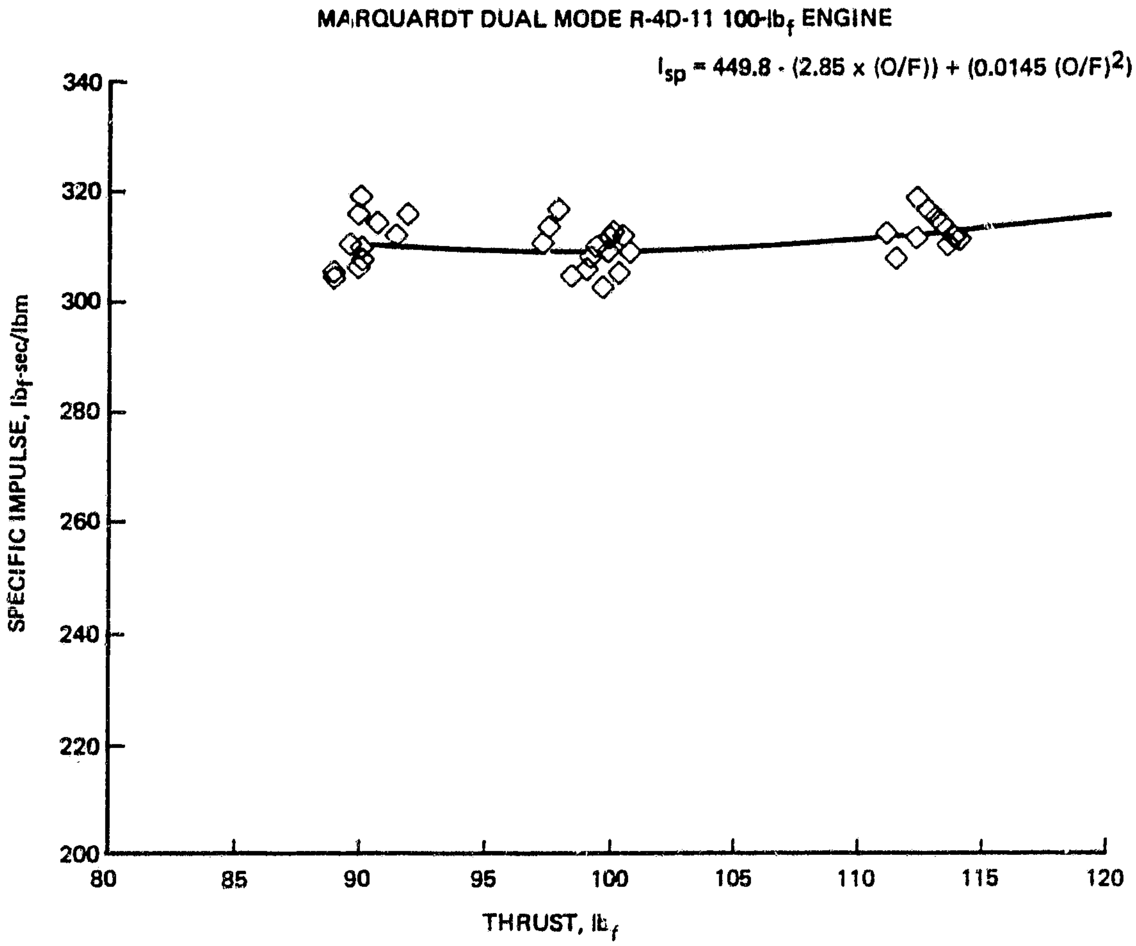

Figure 6-14. Performance of an N_2O_4/MMH 100-lb$_f$ Engine for Varying Thrust Levels

Table 6-11. N_2/O_4/MMH Engine Performance

Thrust, lb_f	Manufacturer	Specific impulse, sec*	Minimum impulse bit, lb_f-sec
0.5	Aerojet	275	0.002
1.0	Marquardt	280	0.003
1.0	Rocketdyne	300	———
2.2	Marquardt	285	0.007
5	Aerojet	280	0.090
5	Bell Aerospace	286	———
5	Marquardt	289	0.013
5	Rocketdyne	200	0.025
14	Aerojet	290	0.210
15	Aerojet	285	———
18	Bell Aerospace	280	———
18	Rocketdyne	285	———
20	Aerojet	282	———
23	Bell Aerospace	200	———
25	Bell Aerospace	272	———
25	Marquardt	290	0.200
25	Rocketdyne	300	0.200
75	Aerojet	272	———
80	Rocketdyne	289	0.530
100	Aerojet	311	5.0
100	Bell Aerospace	310	———
100	Marquardt	310	———
100	Rocketdyne	305	0.610
110	Marquardt	312	0.600

*Steady - State

Table 6-12. N_2O_4/MMH Engine Sizes and Weights (Including Valves)

Thrust, lb_f	Manufacturer	Length, in	Diameter, in	Weight, lb
0.5	Aerojet	4.090	1.04	0.60
1.0	Marquardt	7.000	3.20	1.20
2.2	Marquardt	9.036	1.50	1.38
5	Aerojet	7.310	2.10	1.10
5	Bell Aerospace	10.700	----	1.90
5	Marquardt	9.905	2.17	1.48
5	Rocketdyne	6.700	2.60	1.37
14	Aerojet	9.000	2.78	3.00
15	Aerojet	10.000	4.00	2.30
18	Bell Aerospace	6.500	----	2.40
23	Bell Aerospace	8.000	----	2.70
25	Bell Aerospace	9.000	----	2.70
25	Marquardt	13.300	5.50	4.50
25	Rocketdyne	13.400	5.40	2.44
80	Rocketdyne	17.200	7.90	5.00
100	Aerojet	19.900	8.73	5.67
100	Bell Aerospace	21.000	----	6.00
100	Rocketdyne	15.500	6.90	5.12
110	Marquardt	21.815	11.00	8.30

values presented are nominal. Actual values may vary depending on the design and constraints required by a particular system. The information gathered in this table was obtained from the individual manufacturers at the request of BAC.

An example of a recently developed engine is the 1.0 lb_f thruster by Marquardt. This engine has demonstrated a thrust range between 0.56 and 1.35 lb_f at 100 and 350 psia inlet pressures, respectively. It has attractive I_{sp}, life, and impulse bit characteristics.

6.3.3.3 Propellant Quantities and Tank Sizing

N_2O_4 and MMH are a storable propellant combination. At room temperature (68^{0}F), N_2O_4 has a density of 89.899 lbm/ft^3 and MMH has a density of 54.814 lbm/ft^3. N_2O_4 is an oxidizer that is only mildly corrosive when pure. When it is moist or allowed to mix with water, it becomes a strong acid. It is hypergolic with many fuels (i.e., MMH) and can also cause spontaneous ignition with many common materials such as paper and leather. N_2O_4 fumes are a reddish-brown and are extremely toxic. Due to its high vapor pressure (111 psia at 160^{0}F), it must be kept in relatively heavy tanks. It is compatible with aluminum, stainless steel, nickel alloy, and teflon. Because of the small range between its freezing point (11^{0}F) and boiling point (70^{0}F), N_2O_4 should be stored at a minimum pressure of 30 psia to prevent it from changing to vapor.

MMH is highly toxic and is spontaneously ignitable with N_2O_4. It is a stable, storable fuel, with a freezing point of -63^{0}F and a boiling point of 187^{0}F. It has a low vapor pressure, 8.8 psia, at 160^{0}F and is compatible with aluminum, 304.307 stainless steel, Teflon, Kel-F and polyethylene.

Table 6-13 shows the quantities of N_2O_4 and MMH that would satisfy drag make-up requirements for a 90-day period (assuming the Orbiter is docked for 14 of those days). The drag data assume a NASA neutral atmosphere, an altitude of 525 km, and an Earth-oriented 28.5 deg inclination, for a 2- to 4-man station.

Table 6-13. N_2O_4/MMH Bipropellant Requirements for a 90-Day Period

Propellant	N_2O_4/MMH
Specific impulse, sec	290
Total required, lb_m*	545/341
Volume, ft^3**	6.9/6.9
Tank material	Ti 6AL-4V
Tank weight, lb_m	21/21
Tank pressure, psia	50/50
Tank temperature, $^{\circ}F$	$68^{\circ}/68^{\circ}$

*Includes 50% contingency (i.e., attitude control backup, docking disturbances, etc.)
**Includes 10% ullage for liquid propellants only.

6.3.3.4 Exhaust Constituents

To examine the exhaust constituents and to understand what the key parameters governing them are, the engine operation and design must be examined. A bipropellant engine system has two distinguishable subsystems: the propellant feed module and the engine valve assembly (figure 6-15).[1] Figure 6-15, also shows a simplified breakdown of a nozzle plume flow. The engine can be operated either under steady-state conditions or in a pulsing mode. Unburned fuel and oxidizer are produced during start-up and shut-down operations. During start-up, the remaining fuel in the manifold is expelled slightly before the oxidizer starts flowing. After the thrusters shut down, the oxidizer remaining in the manifold between the valve and the injector face vaporizes and flows out of the thruster. Thus, the shorter pulse times tend to produce increasing amounts of unburned fuel and oxidizer.[2]

Figure 6-16 shows an example of the total propellant droplet outflow from an Aerojet 5 lb_f AJ10-181-2 engine. The fuel droplets are dominant during pulse buildup, while the oxidizer droplets dominate when the pulse is extinguished. Figure 6-17 shows the total propellant droplet outflow for the same engine after the engine is shut down. The results shown in figure 6-17 were obtained by means of laser mie light scattering.[2] Two other characteristics that affect the exhaust constituency are manifold dribble volume and variable mixture ratio. The volume of the manifold between the valves and the injector face influence the impact of thruster pulsing on contamination production. The larger the volume, commonly referred to as dribble volume, the larger the amount of unburned oxidizer and fuel production. Also, since the oxidizer-to-fuel ratio varies across the radius of the combustion chamber, the chamber wall area becomes fuel rich. This depends, however, on the design of the injectors and their spray

1 Chirivella, Jose E., Ergo-Tech and Furstenau, Ronald P., AFRPL, "Verification of CONTAM II using Bipropellant Engine Data", JANNAF 23-25, March 1982.

2 JANNAF Handbook, "Rocket Exhaust Plume Technology," CPIA Publication 263, June 1983.

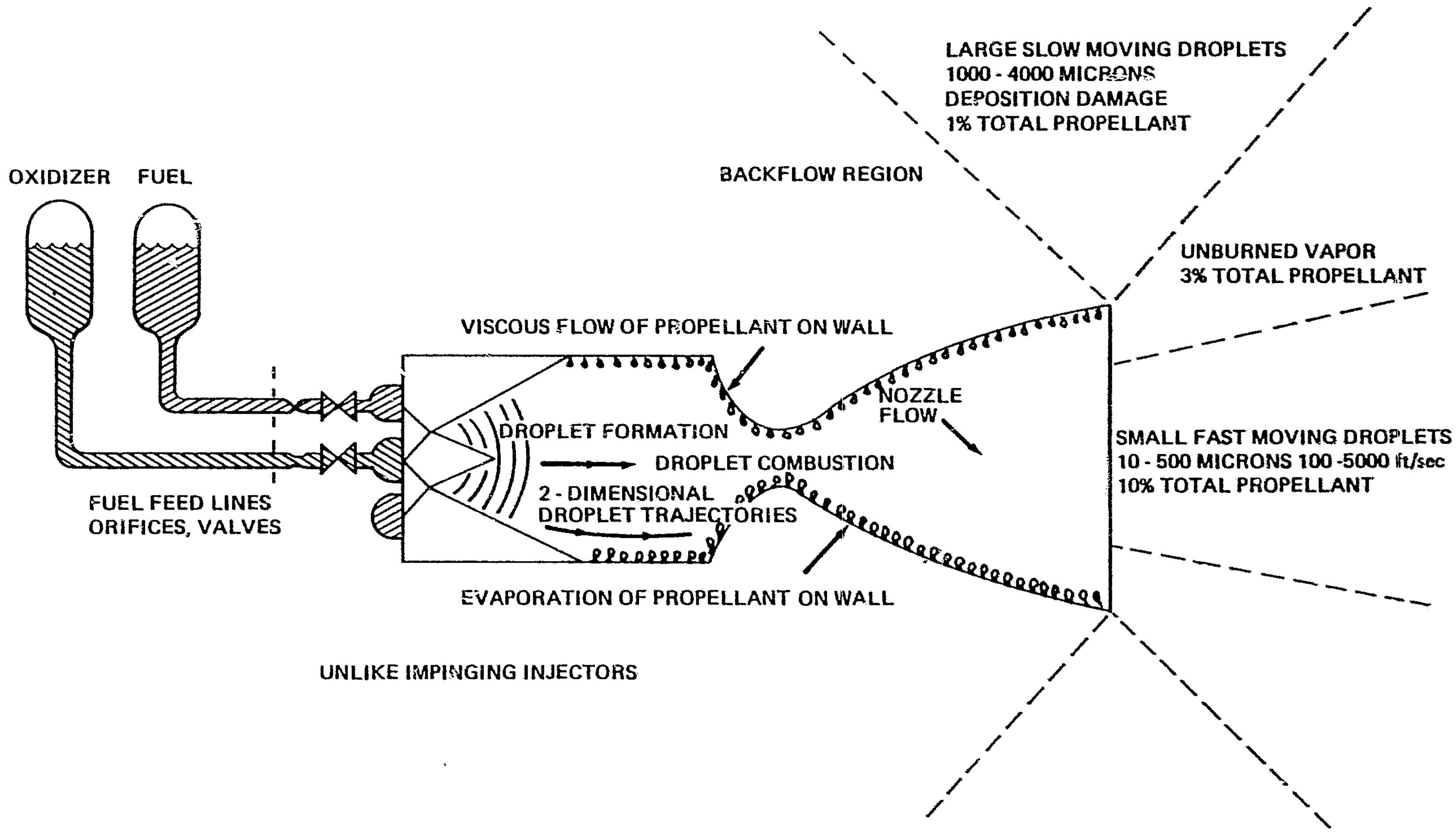

Figure 6-15. Bipropellant Engine System Contamination Production and Transport

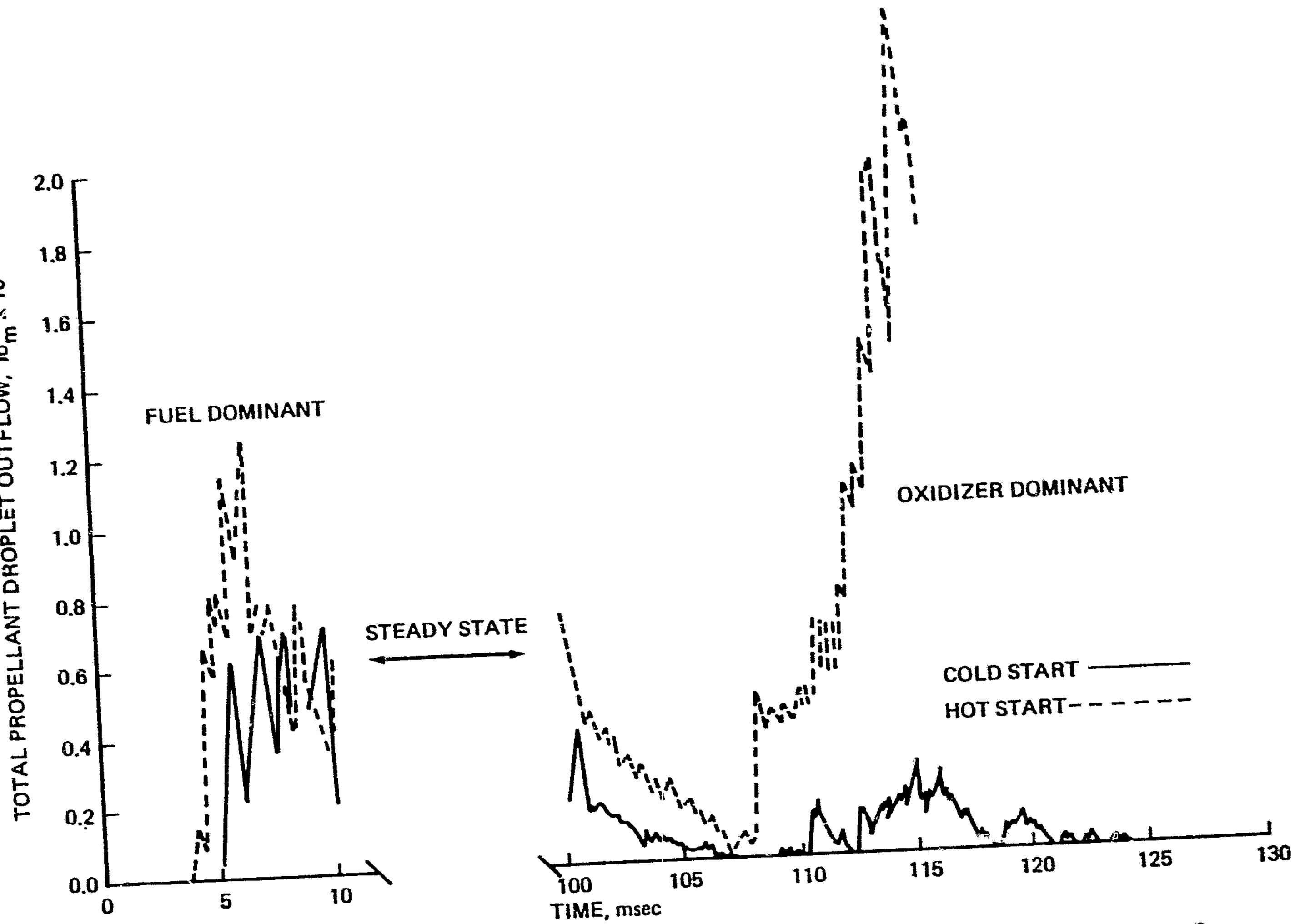

Figure 6-16. Total Propellant Droplet Outflow Versus Time for the AJ10-181-2 Engine for Cold and Hot Start

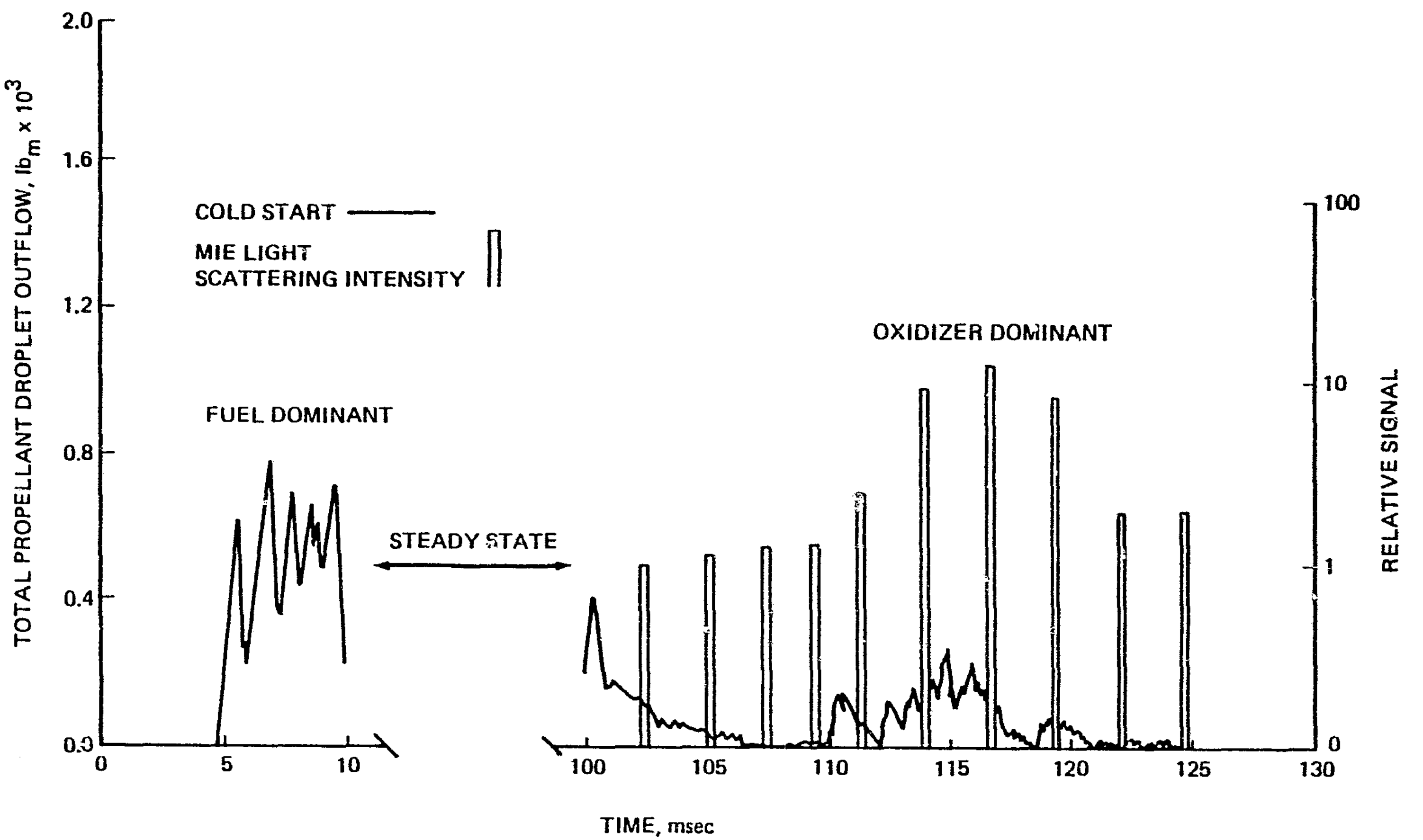

Figure 6-17. Total Propellant Droplet Outflow for the AJ10-181-2 Engine for Startup and Shutdown

patterns. Table 6-14 shows a breakdown of N_2O_4/MMH exhaust products for steady-state operations of the Marquardt R-4D 110 lb_f and R-1E 25 lb_f rocket engines.[3]

6.3.3.5 Potential Throttling and Installation Penalties

Bipropellant thrusters can be throttled by either regulating the feed pressure or by using a pulse mode. Both cases cause a loss in efficiency. Figures 6-18(a) and 6-18(b) show an example of the effects of varying the feed pressure or impulse bit on thrust level. Both figures represent performance data for Marquardt's model R-2 1 lbf rocket engine.[4] Figure 6-19 also shows how performance drops as the impulse bit becomes smaller.

6.3.4 O_2/H_2 Bipropellant Thrusters

Qualified O_2/H_2 engines with thrust levels of 500 lb_f or less do not exist. However, experimental work has been conducted to evaluate low thrust and low chamber-pressure thruster technology. All of the work completed to date has been either theoretical or developmental. Thrust levels as low as 0.1 lb_f have been attained. Ignitor-injector operations which have been demonstrated for pressures as low as 30 psia have been established. Because O_2/H_2 is not a hypergolic propellant combination, an ignitor is required. For Space Station applications, the ignitor must be reliable for thousands of firings. In tests conducted at the Jet Propulsion Laboratory, a commercially available sparkplug was modified for use with a "breadboard" exciter and produced favorable results. The major manufacturers that have conducted theoretical and developmental work on GO_2/GH_2 thrusters in the sub-500 lb_f class are Aerojet, Bell Aerospace, Marquardt, and Rocketdyne.

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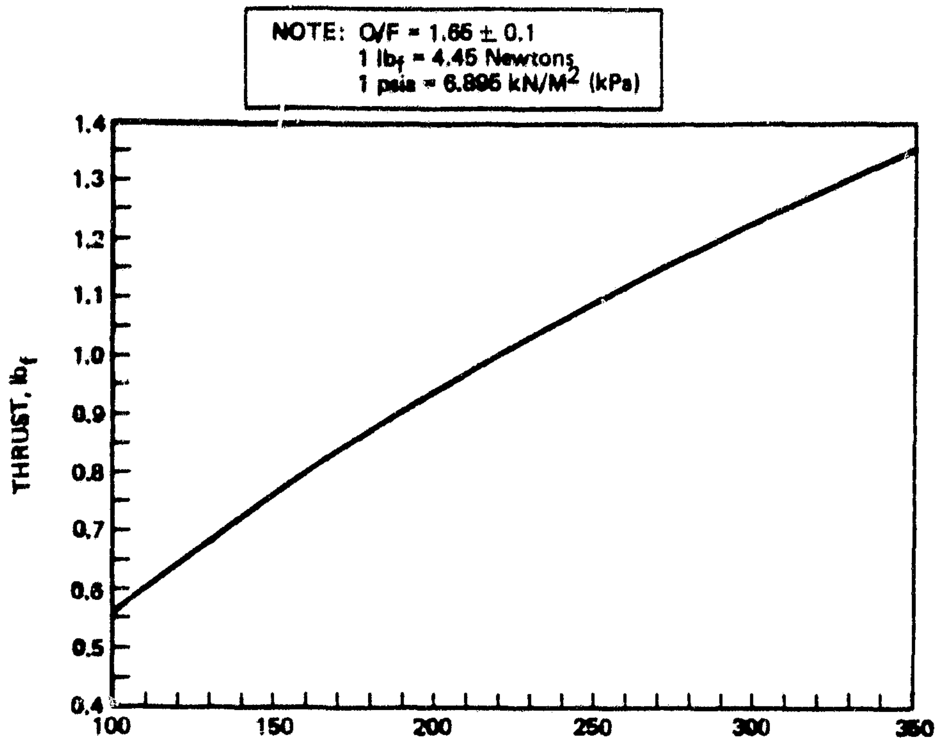
3 "Payload Accomodations Handbook," JSC 07700 Vol. XIV, Change No. 36 Aug. 2, 1981.

4 Marquardt Co., "Propulsion Systems and Engines for Satellites," 1982, and Garrison, P. W. (JPL), Rosenburg, S. D. and Judd, D. C. (Aerojet), "Integrated Space Station Propulsion Systems," JANNAF 7-9, Feb. 1984.

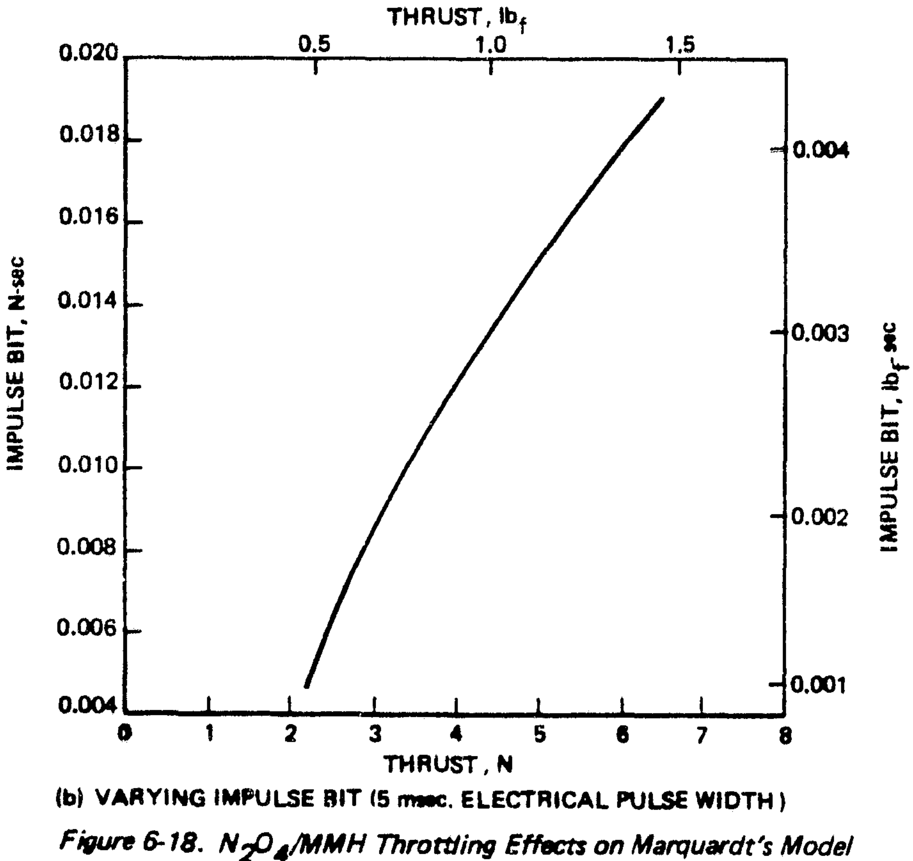
Table 6-14. N_2O_4/MMH Exhaust Products (Mole Fractions) and Partially Reacted Contaminants

Steady State Operations	RCS (110 lbf)	VRCS (25 lbf)
Completely Reacted Products		
H_2O	0.339	0.333
N_2	0.309	0.312
H_2	0.163	0.181
CO	0.129	0.129
CO_2	0.042	0.042
O_2	0.002	Traces
NO	0.001	——
Free Radicals		
H	0.012	0.003
OH	0.006	Traces
C	Traces	Traces

Pulse Mode Trace Products (Solid or Liquid)	
MMH	$N_2H_3CH_3$
MMH - Nitrate	$N_2H_2\,CH_3\,NO_3$; $N_2H\,CH_3\,(NO_3)_2$
Nitrate Nitric Acid	HNO_3
Ammonium Nitrate	$NH_4\,NO_3$



(a) VARYING INLET PRESSURE



(b) VARYING IMPULSE BIT (5 msec. ELECTRICAL PULSE WIDTH)

Figure 6-18. N_2O_4/MMH Throttling Effects on Marquardt's Model R-2 1-lb_f Rocket Engine

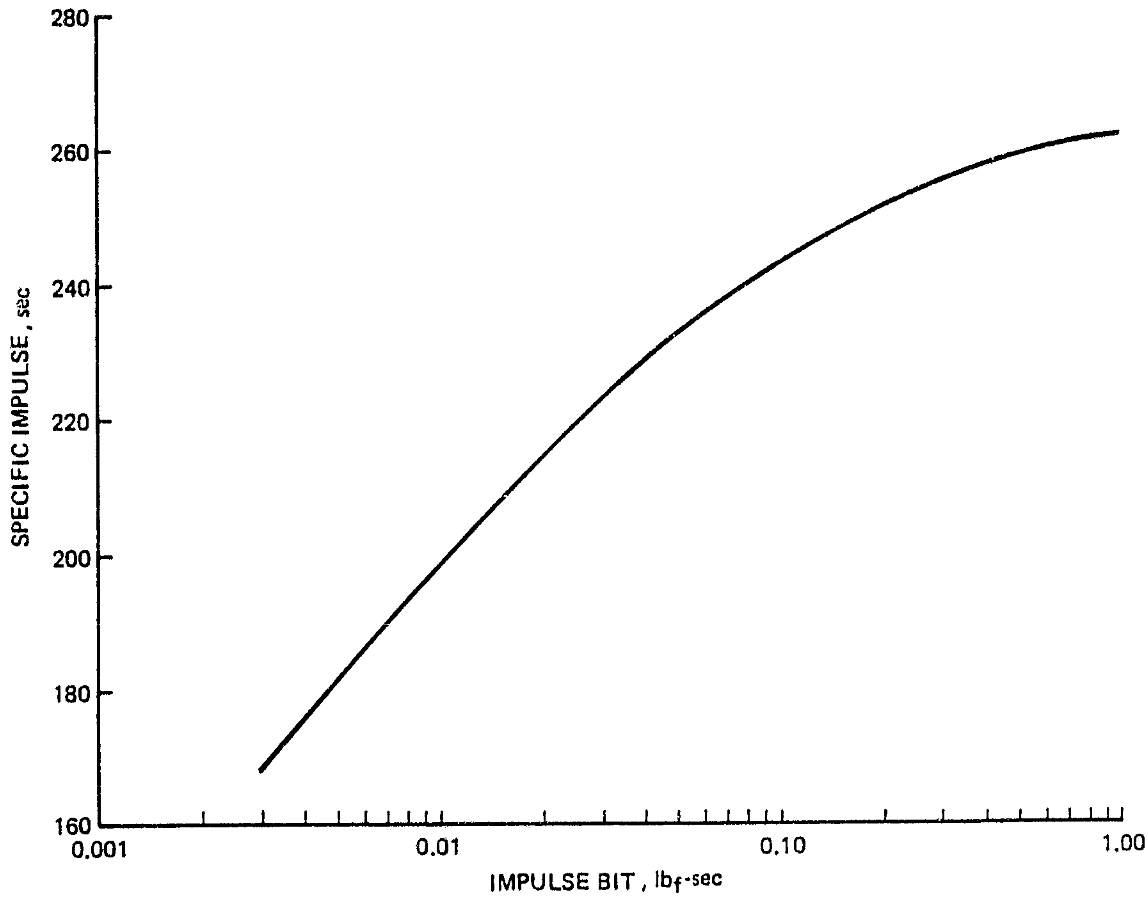


Figure 6-19. Varying Impulse Bits (Marquardt R-2 1-lb_f Rocket Engine)

6.3.4.1 Performance Characteristics

BAC has generated theoretical specific-impulse data using a one-dimensional equilibrium (ODE) rocket motor code. Figures 6-20 through 6-24 illustrate the ideal I_{sp} for various mixture ratios, chamber pressures, and expansion ratios for an inlet temperature of $525^{o}R$. Performance data available for GO_2/GH_2 thrusters in the size and range of interest have indicated relatively large cooling losses.[1,2] I_{sp} efficiency factors (I_{sp} actual/I_{sp} ideal) were calculated based on these data and are approximately 75% for thrusters in the 0.1 lb_f to 25 lb_f range, and 80% for thrusters greater than 25 lb_f but less than 150 lb_f. These efficiency factors are considered to be conservative. Table 6-15 shows performance data for various thrust levels and manufacturers. Figure 6-25 shows actual performance data of a Marquardt 5 lb_f GO_2/GH_2 engine for thrust levels ranging from 2.5 to 7.1 lb_f thrust.

Comparing figure 6-25 with figure 6-20, it can be seen that for a mixture ratio of 8:1 and a chamber pressure of 100 psia, the actual performance of the 5 lb_f engine is approximately 80% of the ideal performance.

6.3.4.2 Hardware Physical Characteristics

Table 6-16 shows the weights and envelope sizes of the four major manufacturers for thrusters in the range of 0.1 through 500 lb_f. The weight and envelope data presented are nominal, actual values may differ depending on the design and constraints required by a particular system. The information gathered in this table was provided by the individual manufacturers at the request of BAC.

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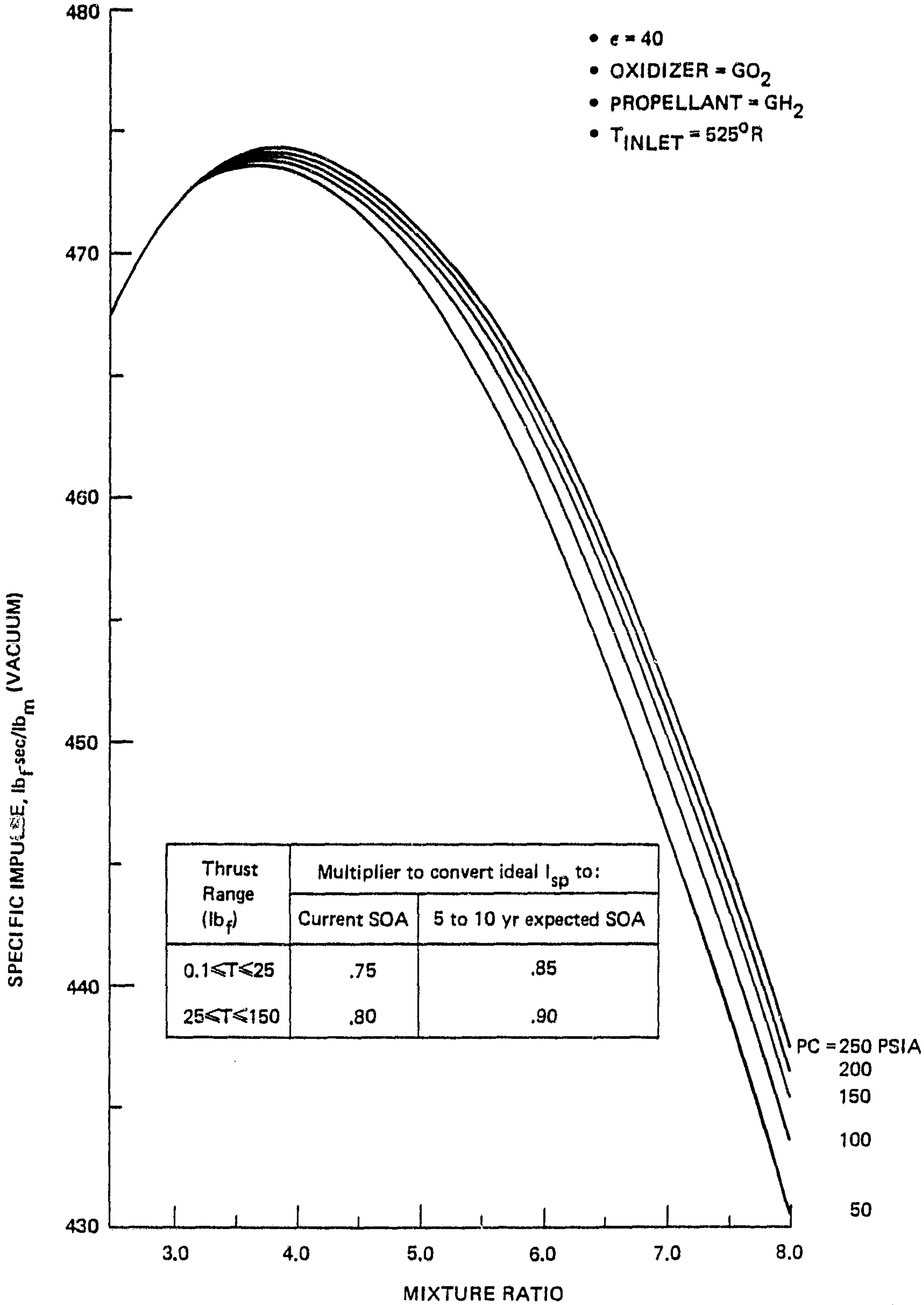
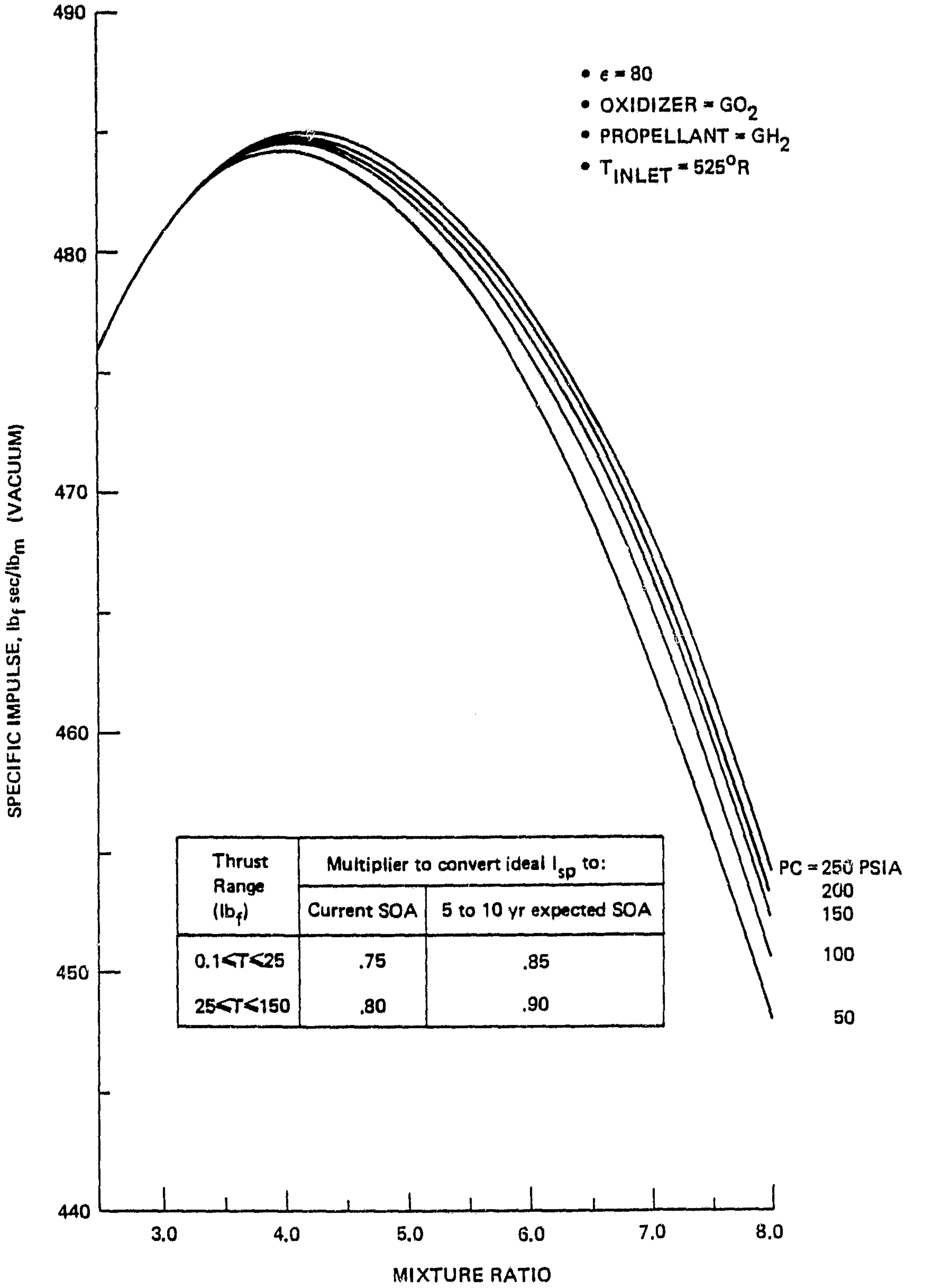
1 Appel, M. A., JPL, Schoenman, L., Berkman, D. K., Aerojet; "Oxygen/Hydrogen Thrusters for the Space Station Auxiliary Propulsion Systems."

2 Stechman, C., Campbell, J., Hudson, T. E., "The Future...Liquid Bi-propellant Rocket Engines/Systems for Satellites and Spacecraft," AIAA-82-1194, June 1982.



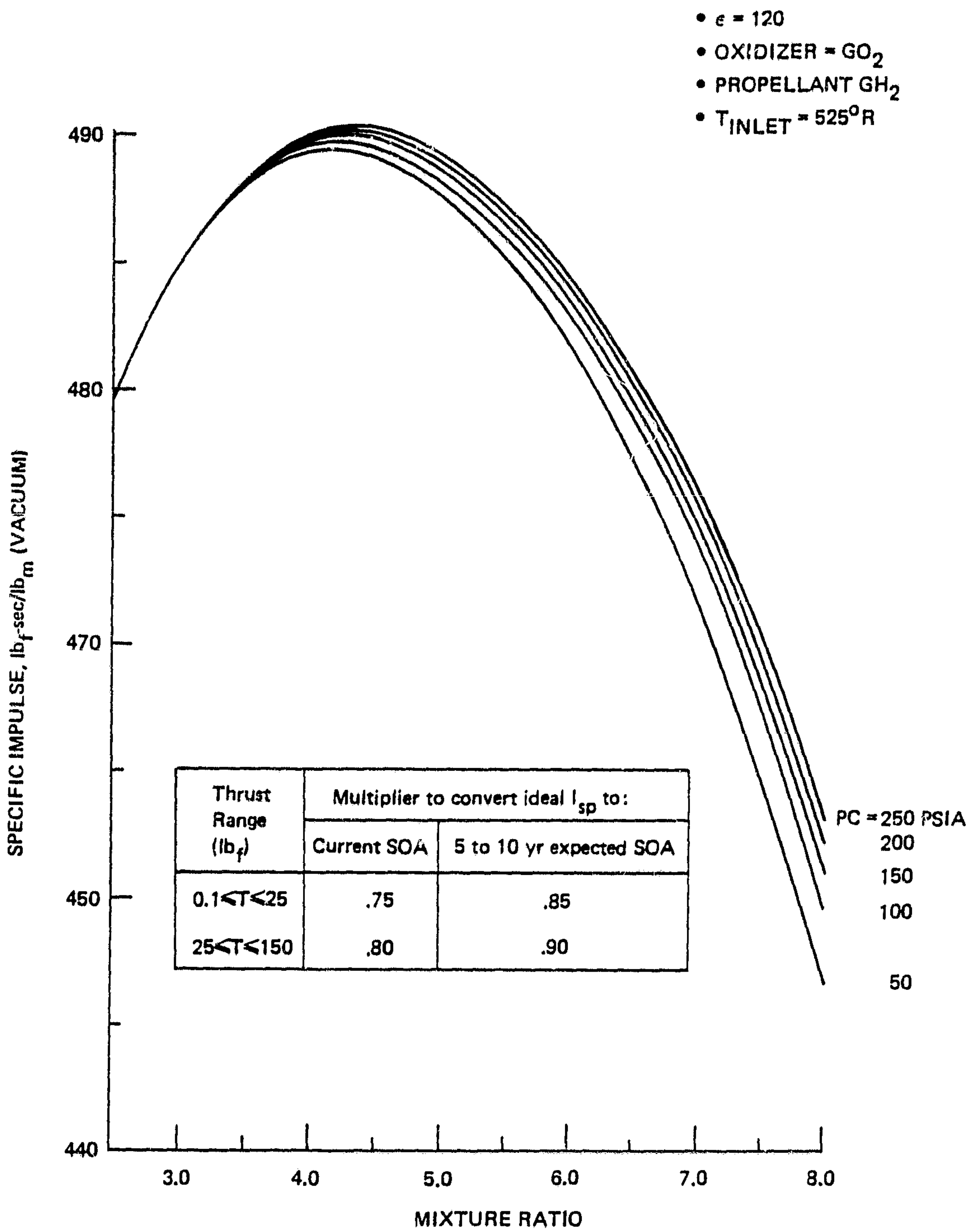
Thrust Range (lb_f)	Multiplier to convert ideal I_{sp} to:	
	Current SOA	5 to 10 yr expected SOA
$0.1 \leq T \leq 25$	.75	.85
$25 \leq T \leq 150$	.80	.90

Figure 6-20. Ideal Specific Impulse Performance of GO_2/GH_2 for an Expansion Ratio of 40



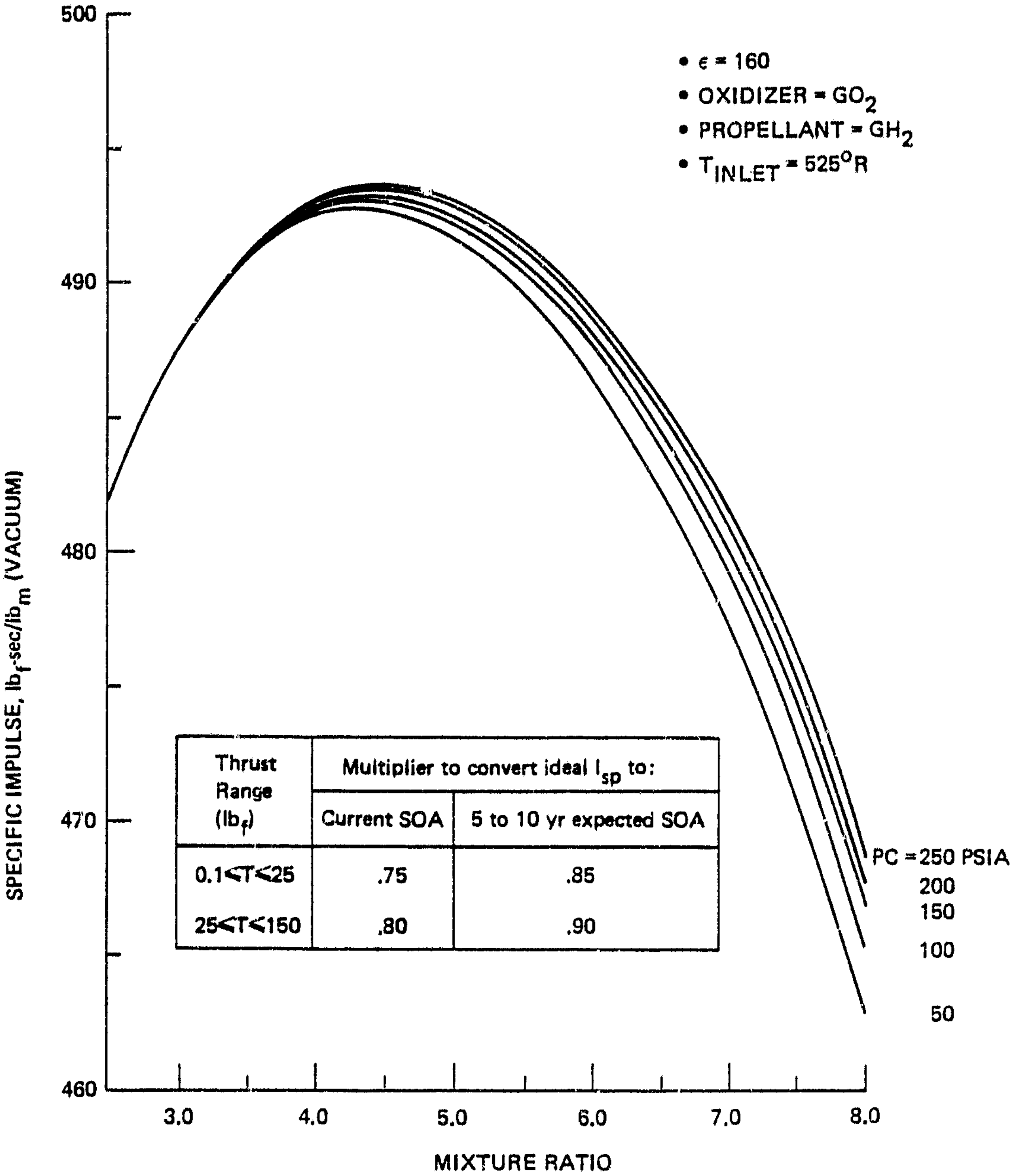
Thrust Range (lb_f)	Multiplier to convert ideal I_{sp} to:	
	Current SOA	5 to 10 yr expected SOA
$0.1 < T < 25$	.75	.85
$25 < T < 150$	.80	.90

Figure 6-21. Ideal Specific Impulse Performance of GO_2/GH_2 for an Expansion Ratio of 80



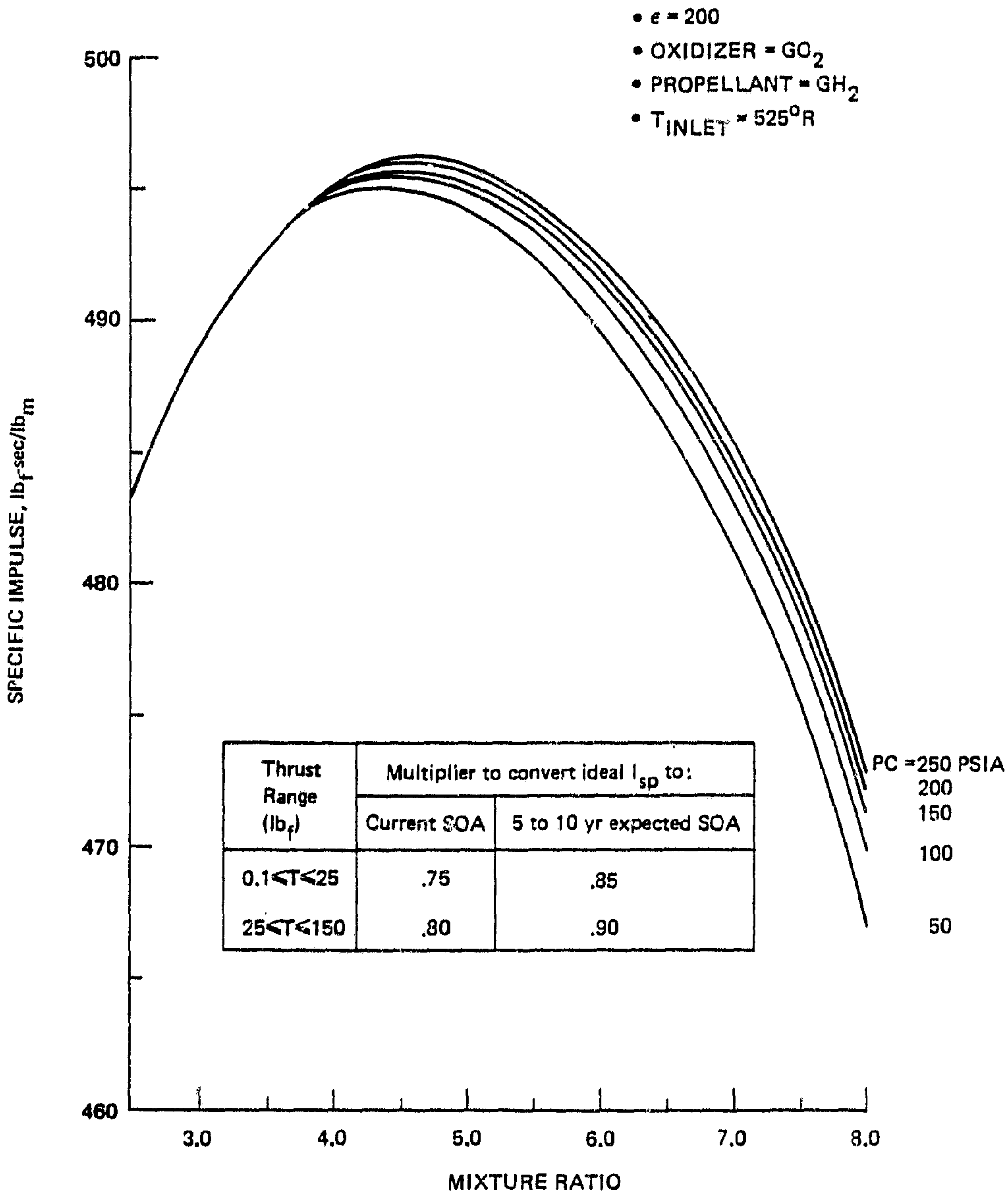
Thrust Range (lb_f)	Multiplier to convert ideal I_{sp} to:	
	Current SOA	5 to 10 yr expected SOA
$0.1 < T < 25$	.75	.85
$25 < T < 150$	.80	.90

Figure 6-22. Ideal Specific Impulse Performance of GO_2/GH_2 for an Expansion Ratio of 120



Thrust Range (lb_f)	Multiplier to convert ideal I_{sp} to:	
	Current SOA	5 to 10 yr expected SOA
$0.1 \leq T \leq 25$	.75	.85
$25 \leq T \leq 150$	.80	.90

Figure 6-23. Ideal Specific Impulse Performance of GO_2/GH_2 for an Expansion Ratio of 160



Thrust Range (lb_f)	Multiplier to convert ideal I_{sp} to:	
	Current SOA	5 to 10 yr expected SOA
0.1≤T≤25	.75	.85
25≤T≤150	.80	.90

Figure 6-24. Ideal Specific Impulse Performance of GO_2/GH_2 for an Expansion Ratio of 200

Table 6-15. O_2/H_2 Engine Performance

Thrust, lb_f*	Manufacturer	Specific impulse, sec	Type
0.1	Marquardt	290	Gas
0.5	JPL	418	Gas
5.0	Marquardt	355	Gas
25	Aerojet	400	Liquid
25	Rocketdyne	390	Liquid
50	Aerojet	450	Gas
50	Bell Aerospace	425	Gas
500	Aerojet	465	Liquid
500	Rocketdyne	465	Liquid

* A study issued by NASA/LeRC to develop a 25 or 100 lb_f gaseous O_2/H_2 thruster system is currently underway

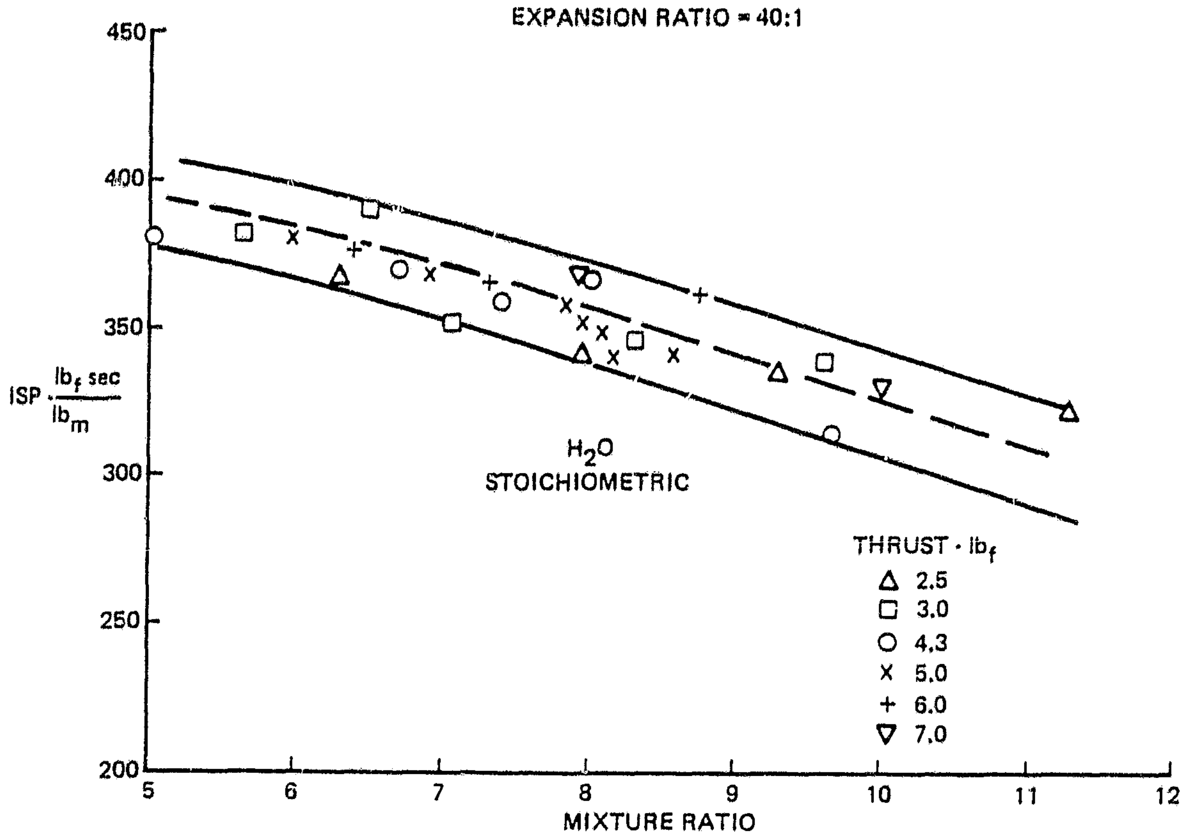


Figure 6-25. Performance of the Marquardt 5-lb_f GO_2/GH_2 Engine

Table 6-16. O_2/H_2 Engine Sizes and Weights (Including Valves)

Thrust, lb_f	Manufacturer	Length, in	Diameter, in	Weight, lb
0.1	Marquardt	N9.5	——	——
0.5	JPL	——	——	——
5.0	Marquardt	12.5	1.8	——
25*	Rocketdyne	16.2	5.4	7.5
50	Aerojet	N9.5	2.5	3.0** 6.0+
50	Bell Aerospace	18.7	8.0	——
100*	Rocketdyne	26.0	10.8	12.6

* As proposed for the NASA/LeRC Space Station On-board Propulsion Study

** Regeneratively cooled

\+ Rhenium chamber

6.3.4.3 Propellant Quantities and Tank Sizing

O_2 and H_2 can be considered as storable gases. At room temperature (68^oF) and atmospheric pressure, hydrogen has a density of 0.0052 lb/ft^3 and oxygen has a density of 0.08295 lb/ft^3. Table 6-17 shows the quantities of O_2 and H_2 that would satisfy drag makeup requirements for a 90-day period (assuming the Orbiter is docked for 14 of those days). The drag data assume a NASA neutral atmosphere, an altitude of 525 km, a Earth- oriented 28.5-deg inclination, 2- to 4-man station.

Storage of O_2/H_2 for long intervals can be complicated and difficult. However, there are two techniques for storing O_2/H_2 that may be used: supercritically, as cryogens, and as water using electrolysis conversion to GO_2/GH_2. Supercritical cryogenic and high-pressure storage of fluids ensures single-phase vapor delivery under all gravity conditions, and also permits a large quantity of fluids to be stored with minimum system volume and weight. Therefore, supercritical cryogenic storage is often used for large-capacity and high-consumption applications that require minimum storage volume and weight.

Supercritical storage of fluids is illustrated by the pressure-enthalpy diagram of figure 6-26. The initial full-tank condition, indicated by point 1, is a mixture of saturated liquid and vapor at atmospheric pressure. After fill, heating results in pressurization at constant density. During this process (1-2), the liquid expands until it fills the entire container and becomes a supercritical fluid.

Once the desired supercritical pressure is reached at point 3, fluid delivery can be initiated. Constant pressure operation, as indicated by path 3-4, is achieved by the simultaneous addition of heat to the storage volume and fluid withdrawal. As long as supercritical pressures are maintained, the stored mass remains a homogeneous, single-phase fluid.

As figure 6-26 shows, the fluid temperature rises during operation (indicated by increasing enthalpy). When the temperature of the fluid in the vessel becomes significantly higher than the critical temperature, it

Table 6-17. Gaseous O_2/H_2 Propellant and Storage Tank Requirements for a 90-Day Period
(2 to 4 man station)

Propellant	O_2/H_2
Specific impulse, $\frac{lb_f\text{-}sec}{lbm}$*	400
Total required, lbm**	320/80
Volume, ft^3	143/572
Tank material	AL 2219
Tank weight, lbm+	[illegible]/2400
Tank pressure, psia	400/400
Tank temperature, °F	75/75

* For a mixture ratio of 4:1

** Includes 25% contingency (i.e., attitude control backup, docking disturbances, etc.)

\+ Tank weight assembly = $K_{T+A}\ K_{L/D}\ 1.5\ (\rho/F_{T_{u^*}})(V)$ (UFS x P max oper)

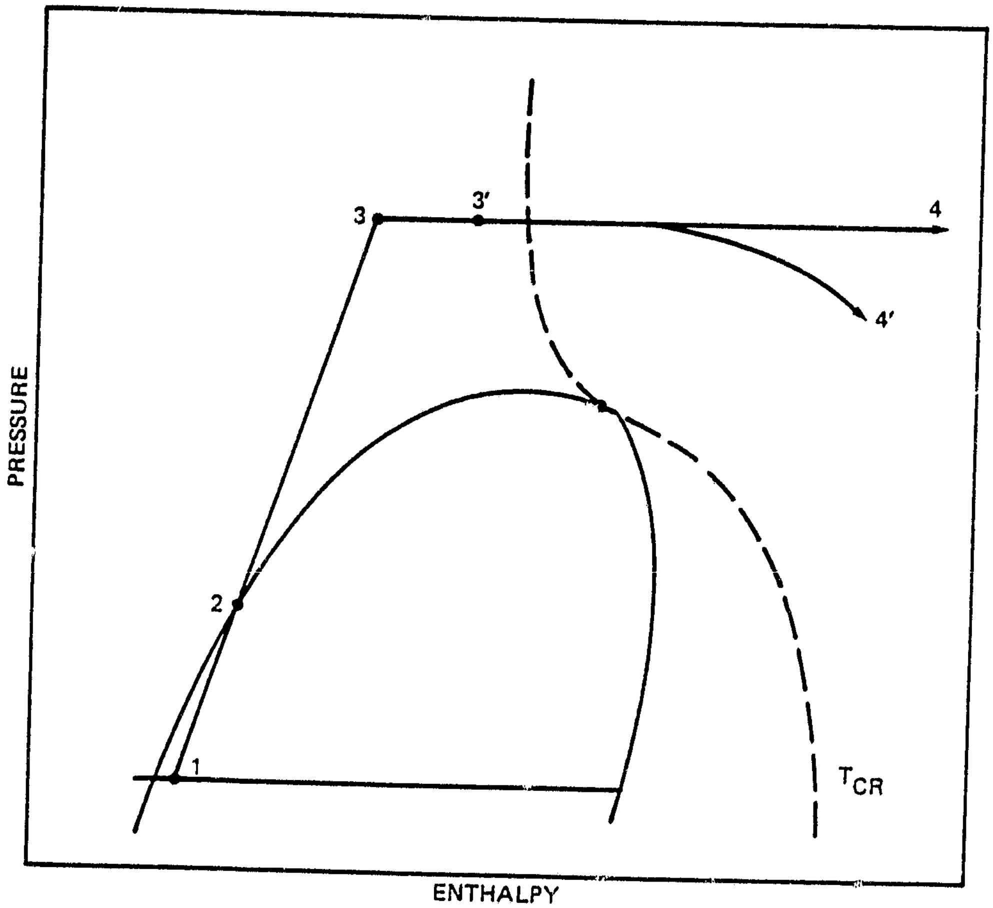


Figure 6-26. Pressure-Enthalpy Diagram, Thermally Pressurized Supercritical Storage

is not necessary to control heat input to maintain a single phase. The ambient heat-leak into the tank in most cases is adequate to maintain a somewhat isothermal path to point 4', while tank pressure is free to decay. Nearly full utilization of the stored fluid is still realized since the difference in residual densities at points 4 and 4' is of secondary importance to the total original charge.

Using supercritical storage for H_2 and O_2 eliminates the acquisition, venting, and gaging problems associated with storing these propellants in liquid form. It would be relatively easy to convert supercritical to subcritical H_2/O_2 storage when these concerns are resolved as a result of the CFMF and other NASA programs.

Electrolyzing water to provide a source of hydrogen and oxygen for propulsion presents a number of interesting features. The advantages and disadvantages are summarized below:

Advantages

- Ground and launch handling convenience
- Safety until electrolysis occurs
- Less expensive storage and delivery systems
- High density (water requires 40% of the volume required by the same weight of LH_2/LO_2)
- Ready availability from the thermal control system
- Fuel source for fuel cells

Disadvantages

- Requires electrical power
- Storage of GH_2 and GO_2 after electrolysis (low density)
- Water has high freezing temperature

A preliminary assessment has been made of the requirements for a water electrolysis system based on an 8- to 12-man scientific station at 525 km, with a 420-kW (BOL) array. For a 90-day period, this station requires 533 kg (1174 lbm) of propellant. The total impulse required is the product of the propellant mass and the specific impulse (I_{sp} = 400 lb_f-sec/lbm) and equals 469,688 lb_f-sec for 90 days. This equals an average daily impulse requirement of 5219 lb_f-sec. If the electrolyzing unit and the H_2/O_2 thrusters are used during the sunlit hours only, then this daily impulse requirement corresponds to approximately 0.6 lb_f of continual thrust for 16 hours each day.

The Marquardt Company has developed prototypes of low thrust H_2/O_2 rockets that have an I_{sp} of about 380 lb_f-sec/lbm. The mass flow rate is the thrust (0.6 lb_f) divided by the I_{sp} (380), or 0.0016 lbm/sec. This results in 92 lbm propellant expended each 16-hour day. The system would require about 8300 lbm water each 90-day resupply cycle.

Marquardt's electrolysis unit weighs 29 lbm and could possibly be reduced to 20 lbm. This works out to a ratio of about 10 lbm of system for each lbm of propellant produced in a day. Therefore, this configuration requires an electrolysis unit of about 900 lbm. Marquardt's system electrolyzes continuously, stores the gases, and thrusts occasionally, so continuous thrusting (16 hours/day) would negate propellant storage requirements and reduce system weight.

6.3.4.4 Exhaust Constituents

The exhaust of an O_2/H_2 engine mainly consists of H_2O and either H_2 and/or O_2, depending on the mixture ratio. Table 6-18 shows a breakdown of O_2/H_2 exhaust products generated by a one-dimensional equilibrium rocket motor program. This theoretical performance data assumes equilibrium composition during expansion. This particular example is for a mixture ratio of 8:1, expansion ratio of 40:1, chamber pressure of 50 psia, and an exit temperature of 3847.6^{o}R. As the table shows, the major product in the exhaust is H_2O (90%), with H_2(4.5%), O_2(2.2%) and OH(2.0%) making up the bulk of the remainder. This type of breakdown is characteristic only of a stoichiometric mixture ratio.

As the mixture ratio drops below stoichiometric, the exhaust product tends to be approximately 70% H_2 and 30% H_2O with all other products being trace.

Table 6-18. O_2/H_2 Exhaust Products (Mole Fractions) and Partially Reacted Contaminants

Theoretical rocket performance equilibrium composition during expansion for a stoichiometric (8:1) mixture ratio

Completely reacted products		Mole fraction
H_2O		.91510
H_2		.04537
O_2		.02221
—	Free radicals	
H		.00813
HO_2		Trace
H_2O_2		Trace
OH		.02018
O		.00261
—	Trace Solid or liquid	
H_2O (S)		Less than .50000 E - 05
H_2O (L)		Less than .50000 E - 05

As the mixture ratio rises above stoichiometric, the exhaust product tends to be approximately 70% O_2 and 30% H_2O, with all other products being trace. Only at and close to a stoichiometric mixture ratio is there any real concern of contamination from the formations of free radicals like H, O, and OH, which are attracted to other substances (i.e., Space Station) and can cause decomposition.

6.3.4.5 Potential Throttling and Installation Penalties

Any level of average thrust can be attained from O_2/H_2 thrusters by throttling through feed pressure regulation or by using a pulse mode. Both result in an efficiency loss. An example is Aerojet's 50-lb_f thruster, which is currently in the demonstration stage. This engine is modifiable for a thrust range of 2.5 to 50 lb_f. A film-cooled rhenium chamber is used for the 2.5 to 25-lb_f thrust level and a regeneratively cooled chamber is used for the 26 through 50 lb_f thrust range. The chamber pressure is varied from 30 to 500 psia and the mixture ratio is varied from 2.2 to 4.0:1 to achieve the variable thrust. Therefore, the specific impulse varies from 400 to 450 lb_f-sec/lbm.[1]

6.4 Projected Thruster Capability Assessment

This section summarizes the results of the projected thruster capability assessment. The principal focus of this task is on resistojet and arcjet systems, as called out in the Statement of Work. Because of the relative importance and potential applicability of mono and bipropellant engines to the Space Station, performance and lifetime projections for cold gas and catalytic, and bipropellant thrusters are also examined. In many cases, state-of-the-art projections cannot be made without assuming significant technology development programs beyond those currently underway.

Factors that limit capabilities of current hardware and the likelihood of extending these limits with additional development efforts are noted.

1 Uhrhammer, Tom, Aerojet, Telecon 8/2/84, Bipropellant Thruster Data.

These factors differ among the various thruster types, so the report is divided into major sections that discuss each thruster technology.

6.4.1 Monopropellant Thrusters

Current monopropellant thrusters have two basic features. The first is that, in order to run at low chamber pressures, the thruster size must increase. Low chamber pressures can be created by utilizing by-products from on-board processes and storing these gases with a safe but low-pressure containment scheme. The second feature is lifetime of the thruster system. Propellant valve cycling life and catalyst bed life for N_2H_4 systems are the primary concern. Both of these limitations are discussed below.

6.4.1.1 Low Pressure Operation

The basic nozzle thruster equation is written as:

$$F = P_c \times A_t \times C_f$$

where:

F = thrust (lb_f)
A_t = area of the throat (in^2)
C_f = thrust or nozzle coefficient
P_c = chamber pressure (lb_f/in^2)

This equation illustrates that, for a given thrust level, there is an inverse correlation between chamber pressure and hardware size because C_f (thrust coefficient) values generally lie in a narrow range. Propellant quantity considerations aside, a cold-gas thruster designed for "large" (e.g. 50 lb_f) thrust levels would be relatively large and heavy unless the gas could be supplied under pressure of at least a few (Earth) atmospheres. However, most gases available for these thrusters are attained at low pressures from on-board processes. To illustrate the effects of low pressures on thruster sizing consider a CO_2 thruster designed for 50 lb_f at a chamber pressure of 10 psia. It would have a C_f value of approximately

1.66, a throat area of 3.01 in^2, a throat diameter of nearly 2 inches, and a nozzle exit diameter of almost 14 inches (for a 50:1 expansion ratio). Hence, the use of cold-gas thrusters is, therefore, practical only for thrust levels at the lower end of the 0 to 100 lb_f thrust range.

The use of very low chamber pressures also entails performance losses associated with the adverse effect of low Reynolds numbers on thrust coefficient. Research performed at NASA-LeRC showed that there is a correlation between throat Reynolds numbers and thrust coefficients using various nozzle geometries and H_2 at temperatures ranging from 70°F to 3540°F.[1] RRC has used this correlation to estimate theoretical thrust coefficient and I_{sp} values for CH_4, CO_2, H_2, N_2 and NH_3 at a chamber temperature of 200°F and pressures ranging from 0.1 to 10 psia. The accuracy of the correlation for gases other than hydrogen has not been verified, so the results should be viewed as relative rather than absolute. However, some degradation of both C_f and I_{sp} at P_c values below about 5 psia can be seen in figures 6-27 and 6-28. This effect would need to be considered if a low-pressure propulsion system is otherwise desirable.

6.4.1.2 Thruster Life

To understand technology life limits, a somewhat arbitrary 1000-hr lifetime has been used as a design requirement for a Space Station thruster. For a cold-gas thruster, the only component prone to a disabling failure over this lifetime would be the propellant valve. The service life of such valves are generally described in terms of operating cycles. The correlation between valve cycles and firing time depends on the nature of the firing duty cycle. A typical 5-lb_f thruster propellant valve is qualified to 50,000 cycles, which would support 1000 hours of firing if the average pulse length is at least 72 seconds. Numerous valves have demonstrated

1 Spisz, E. W., Brinich, P. F., and Jack, J. R., "Thrust Coefficients of Low-Thrust Nozzles," NASA TND-3056, Lewis Research Center, 1965.

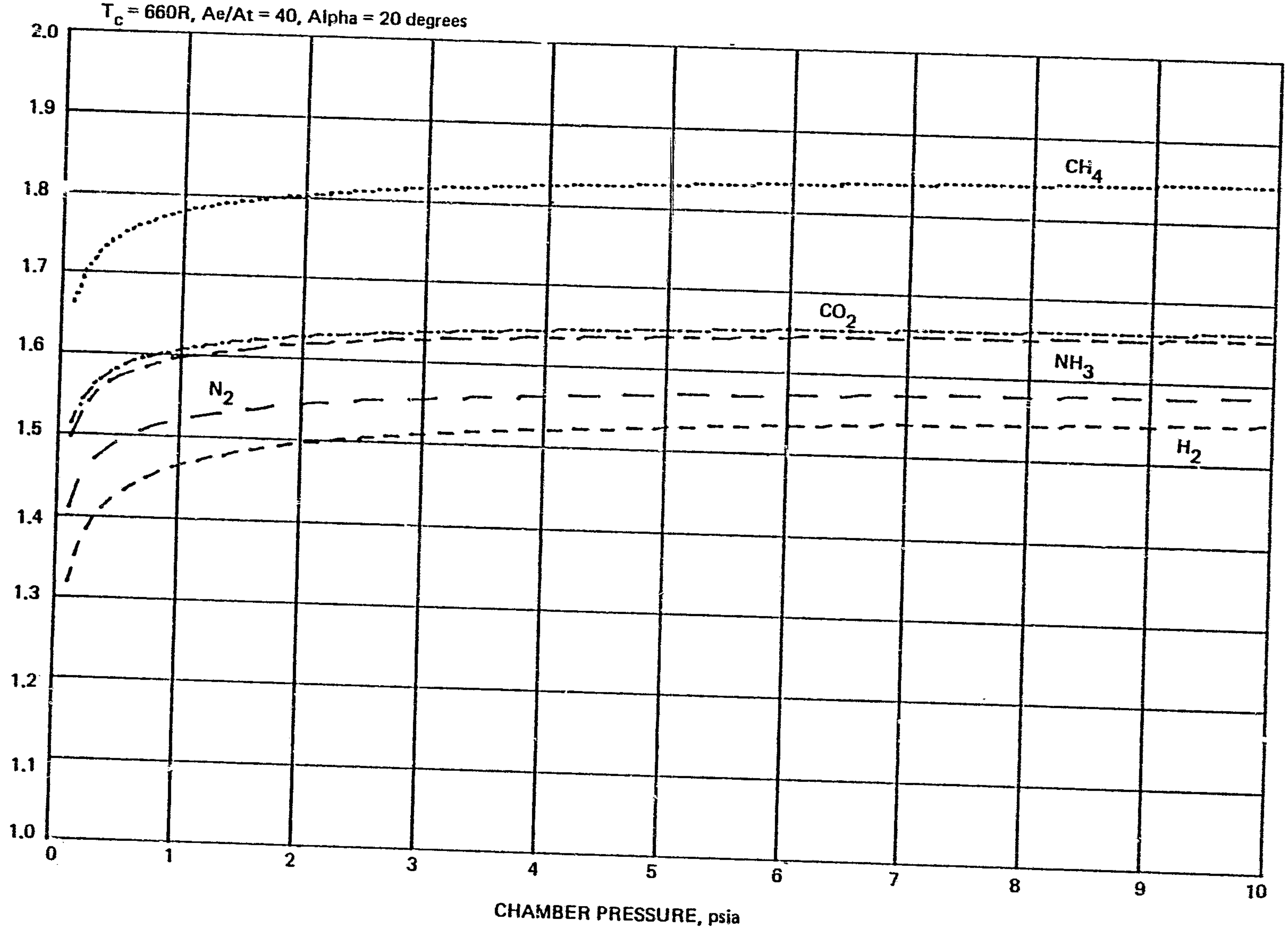


Figure 6-27. Theoretical Thrust Coefficients of Various Monopropellants

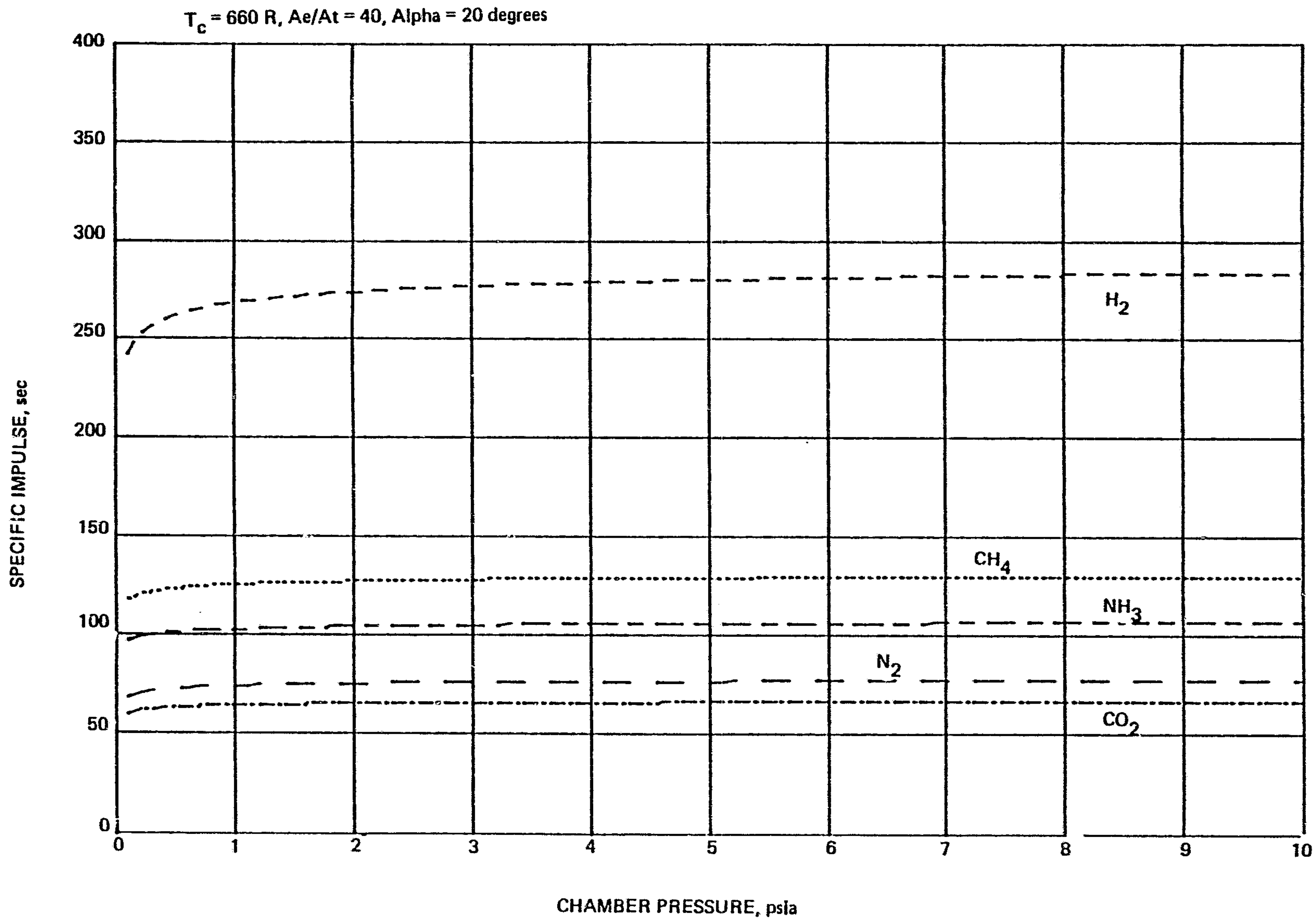


Figure 6-28. Theoretical Performance Values of Various Monopropellants

Table 6-19. Life-Related Capabilities Demonstrated in Various Hydrazine Engine Tests

Tests	Total Impulse (lbf-sec)	Total Throughput (lbm)	Total On Time (hr)	Duty Cycles	Feed Pressures (psia)	Propellant Temp. (°F)	Flow Rate Range (lbm/hr)	Total Valve Cycles
CTAT GG 500-hr IRAD	95,000	454	500	1-hr ss firing, cool to 200°F	250—50	Valve temp 140—175		500
CTAT Phase III Life	80,000	300	430	1-hr ss, 10 min. cool	450—90		1.2—0.4	480
RRC/RCA ACT 1st Dev Unit	39,000	147	224	SS, cool to 400°F	300—100	70	0.9—0.4	305
ACT 1st qual unit	43,000	146	222	SS, cool to 400°F	300—100	70	0.9—0.4	208
ACT 2nd qual unit	76,000 30,640 ss 27,900 pulse 15,000 other	252 105 pulsing	472 472	SS 1.5 sec on/0.5 off 1 sec on/1 off 0.5 on/0.5 off	300—100	70—120	0.9—0.4	533,000
Improved GG IRAD		430	275	SS 0.5 sec on/0.5 off 0.6 sec on/0.3 off	300—100	120 valve at 200	1.3—0.6	800,000
INTELSAT V 0.6 lbf	58,500	260		Most ss and 95%				420,000
RRC/AFRPL 5-lbf Long-Life	800,000 103,650	3,529 475	40 10	1-hr ss misc. pulses	230—100	40—140	80	40 528,038

CTAT = Coiled-Tube augmented thruster

GG = Gas generator

IRAD = Independent research and development at RRC

ACT = Augmented catalytic thruster

lives well beyond 50,000 cycles (Table 6-19), and low-thrust engines on large space structures would probably fire for relatively long intervals. Therefore, the 1000 hour requirement could probably be met by using current valve technology.

Catalytic engines operate at substantially higher temperatures than cold-gas units and, consequently, require materials that can extend service at elevated thermal and pressure loads. Two degradation mechanisms also affect catalytic thruster life. Thermal and mechanical stresses cause catalyst particles to gradually breakup and disintegrate. The resulting voids collect propellant that decompose intermittently, producing chamber-pressure oscillations and increases in the rate of catalyst attrition. Moreover, heat is conducted from the reactor into the tube that transfers propellant from the valve. This heat can cause propellant to vaporize in the tube, allowing non-volatile residues to collect in the tube and possibly crack it due to localized thermal stresses. The former problem is most evident in engines subjected to high-rate firing cycles; apparently, the thermal shock and pressure transients of startup are major contributors to catalyst attrition. The problem of propellant-tube degradation is generally confined to small engines (1.0 lb_f) with small propellant tubes. RRC has experimented with various methods for extending the lifetime of catalytic hydrazine engines. The problem of catalyst attrition was the subject of an AFRPL-sponsored effort in the late 1970's. As part of this effort, RRC developed an advanced 5.0-lb_f thruster that utilized a radial flow catalyst bed design with a new catalyst-bed retention device. The retention technique employs a torsional spring to compensate for catalyst attrition and differential expansion effects during long-life testing. As catalyst losses occur, the band is tightened by a torsional spring so that voids are less likely to form. The design concept for this long-life engine is shown in figure 6-29 and in isometric form in figure 6-30. This design approach has successfully demonstrated more than 900,000 pulses, over 770 cold starts, and 800,000 lb_f-sec total impulse (see table 6-20). An advanced technology 25.0 lb_f engine/gas generator uses a torsional spring combined with a radial flow catalyst bed design concept identical to the 5.0 lb_f long-life thruster. Other designs have used a spring-driven penetrating injector or any of several similar concepts. In a satellite maneuvering mission life simulation test at AFRPL, the RRC 5 lbf long-life engine completed approximately 1 million pulses.

The materials from which hydrazine engines have been built include corrosion-resistant steels and various alloys of nickel and cobalt, such as Haynes 25, Inconel, and Hastelloy. These materials are suitable for engines with 1000-hr lives, although greater thicknesses than are currently employed might be required. The chemical activity of Shell 405 catalyst is sufficient for 1000 hours of firing in long burn times, but may not be adequate for high-rate pulsing.

Efforts at extending the life of the small-engine injector have concentrated on reducing the flow of heat to the propellant-transfer tube, usually by moving the reaction zone toward the reactor's exit end. As seen in table 6-19, various thrusters in the 0.1 lb_f class have been run for almost 500 hours (CTAT Phase III, ACT QU2, Improved GG).

In summary, it appears that N_2H_4 engines could be built with existing materials for 1000-hr lives but have yet to be demonstrated for steady-state firing or long pulses at thrust levels above 1 lb_f. To go below this thrust level would require the development of small-engine injectors. To operate in high or even moderate-rate pulsing duty cycles at any thrust level requires additional work on catalyst-bed design, and might call for a catalyst other than Shell 405, or the use of some other (e.g., thermal) method of initiating propellant decomposition.

6.4.2 Resistojets

Resistojet performance is a function of specific augmentation power and propellant thermodynamics. For a given propellant and flow-rate, I_{sp} may be limited by the available electrical power. If excess power is available, the performance limit is determined by the temperature capabilities of the resistojet's materials.

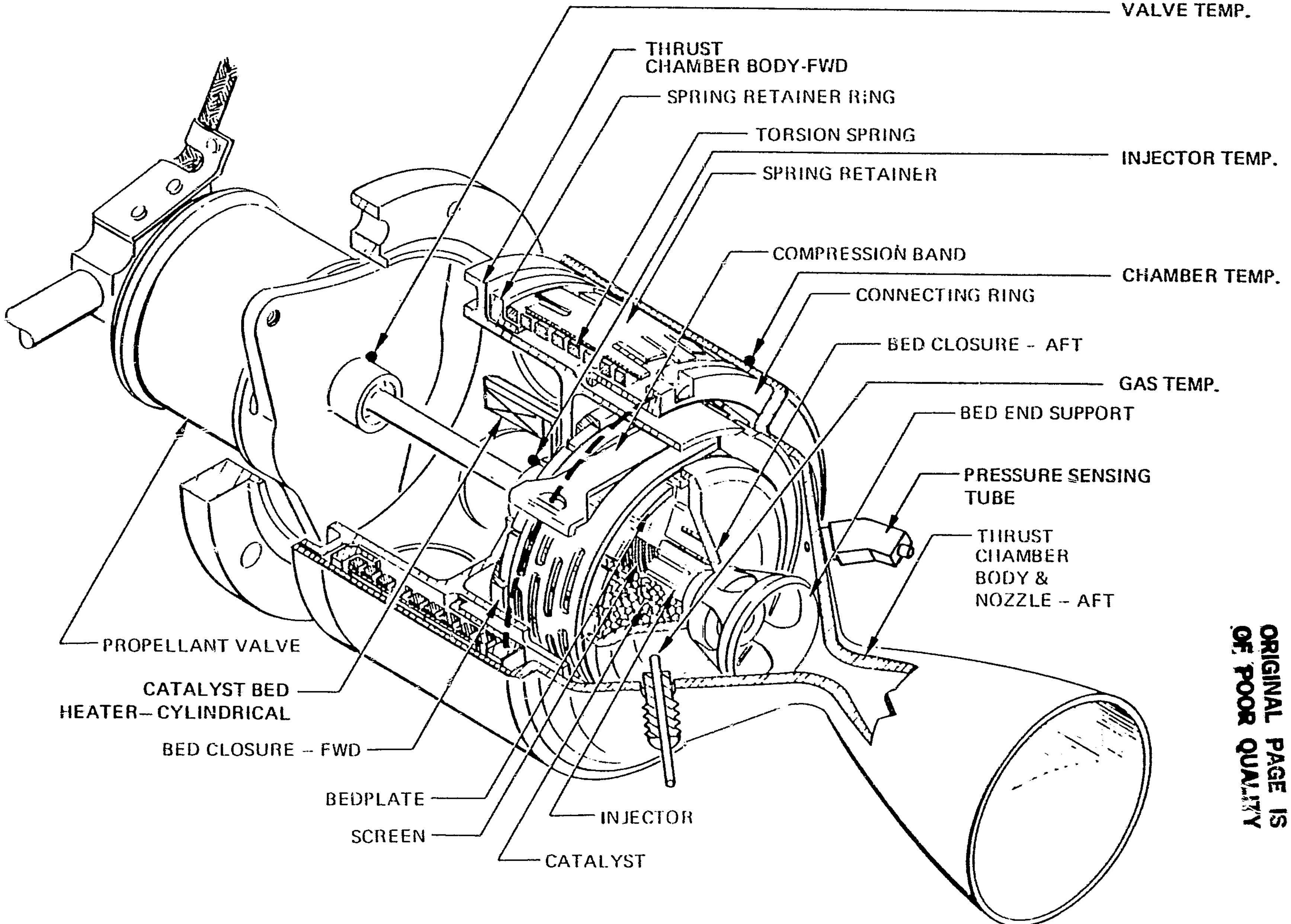


Figure 6-29. Long-Life Rocket Engine Assembly

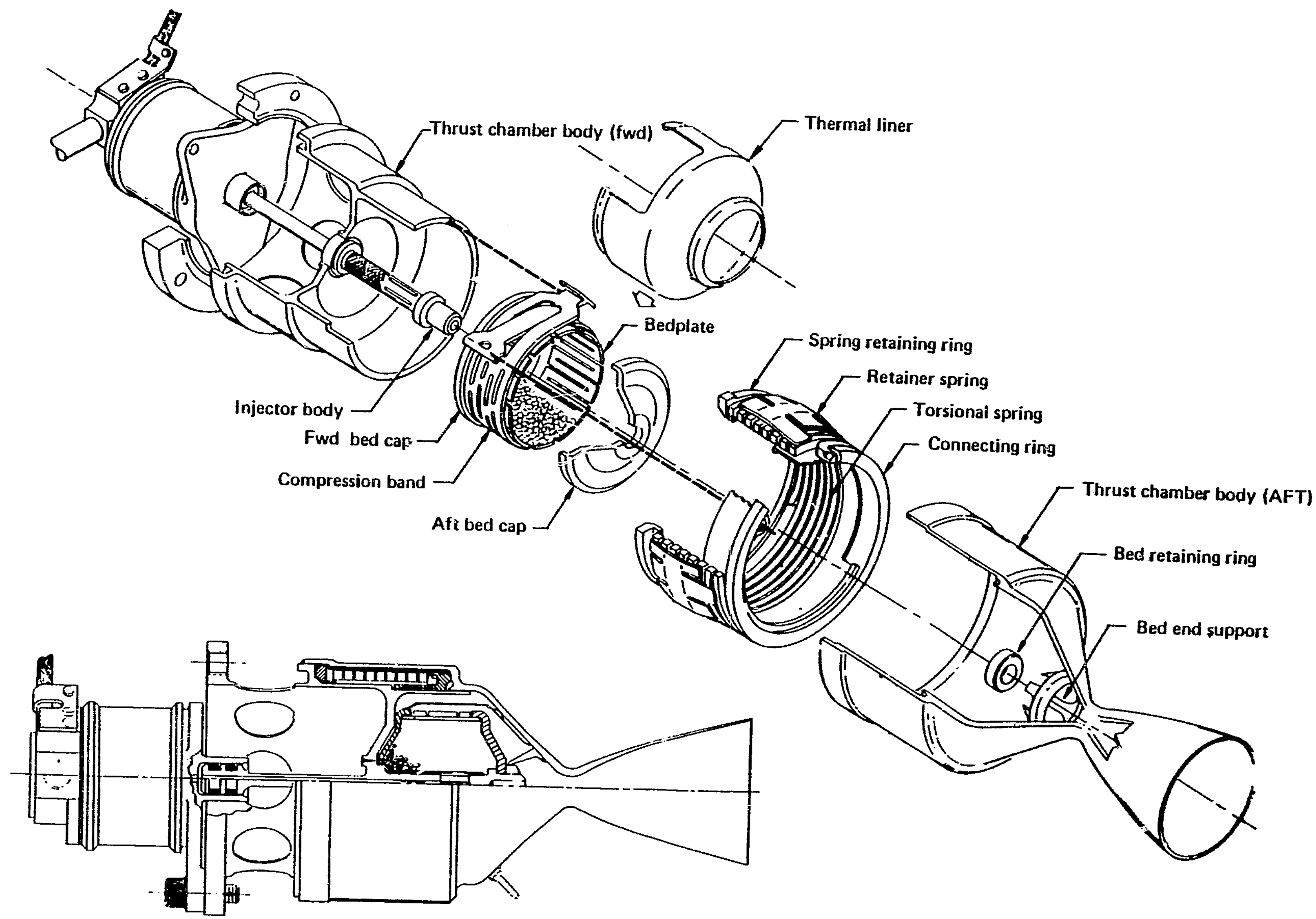


Figure 6-30. Isometric of 5-lb_f Radial Flow Long-Life Monopropellant Hydrazine Engine

Table 6-20. Summary of MR-50 5.0-lbf engine testing

Parameters	P-95 Blk I qualification (prototype)	SMS qualification	MJS T/VA (prototype)	Viking RCS qualification (prototype)	P-95 Blk III qualification	P-95 Blk III development		RRC extended steady-state tests	AFRPL long life engine technology program
						phase I	Phase II		
Throughput, propellant, lbm	128	100	33	70	215	227	148	210	3,529
Total impulse lbf-sec	27,600	21,086	7,000	14,000	44,600	45,500	29,500	46,000	806,274
Pulses (T_c initial > 125° F)	198,500	21,452	15,800	10,200 (20,000)	470,700	537,000	360,000	—	528,038*
Cold start pulses (T_c initial < 90°F)	50	52	—	—	105	39	24	—	400 cold starts with 40°F propellant and 40°F bed**
Steady-state (single burn), sec	300 (1,200)	300	30	1,210 (1,500)	300	100	300	3,600	3,600
Steady-state (cumulative), sec	2,100	3,826	225	1,705	3,100	1,000	2,000	10,800	144,000
Total cumulative burn time, sec	6,100	5,539	—	2,860	13,400	12,000	9,200	10,800	144,000
Documentation, RRC rpt no	70-R-217 R1 (70-R-211)	72-R-297	75-R-485	73-R-397 (73-R-349)	74-R-458	74-R-439		70-R-227	AFRPL-TR-79-68

*All pulses conducted at worst case thermal duty cycle conditions, in excess of 900,000 pulses tested at AFRPL

**AFRPL subsequently accumulated in excess of 300,000 lbf-sec and 774 cold starts on a single engine

6.4.2.1 Performance Enhancement

Current Space Station propulsion requirements are not expected to demand high performance resistojets. However, future development that may limit propellant availability and also the special requirements of free-flyers make a study of resistojet performance enhancement of value.

Figures 6-31, 6-32, and 6-33 present the theoretical performance available from CO_2, H_2, N_2H_4 and NH_3 as functions of the specific power input to the gas. These calculations were made using the NASA Thermodynamics Chemical Equilibirum Computer Code.[1] Figures 6-34, 6-35 and 6-36 show performance expected from the RRC ACT unit using NH_3, H_2, and CO_2. Table 6-21 shows the values of I_{sp} and specific power corresponding to these propellants plus N_2H_4, all at 4500^oF. These numbers represent upper-bound estimates of resistojet performance for the near future. The most suitable materials available for constructing the hot sections of a resistojet are molybdenum,

Table 6-21. Resistojet Specific Impulse and Specific Power Levels for Various Working Fluids at 4500^oF

Propellant*	Specific Impulse	Specific Power(W/mlb)
CO_2	250	7.6
H_2	960	18.9
NH_3	480	13.5
N_2H_4	407	6.4

*Chemical equilibrium was assumed for CO_2, H_2, and NH_3; 90% NH_3 dissociation was used for N_2H_4.

1 Gordon, Sanford, et al., "Computer Program for Calculations of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations," NASA N78-17724, March, 1976.

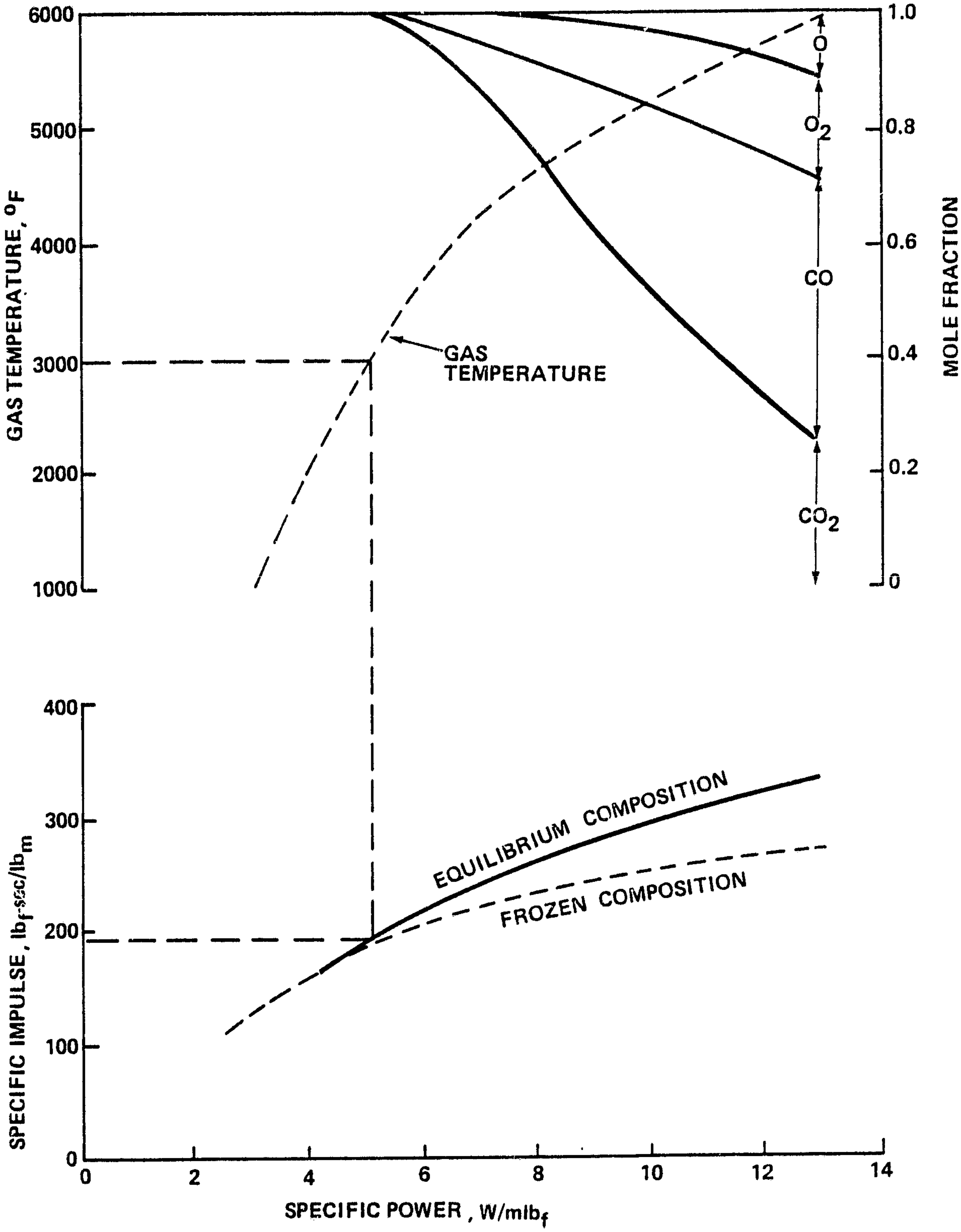


Figure 6-31. Theoretical Performance of CO_2 Resistojet

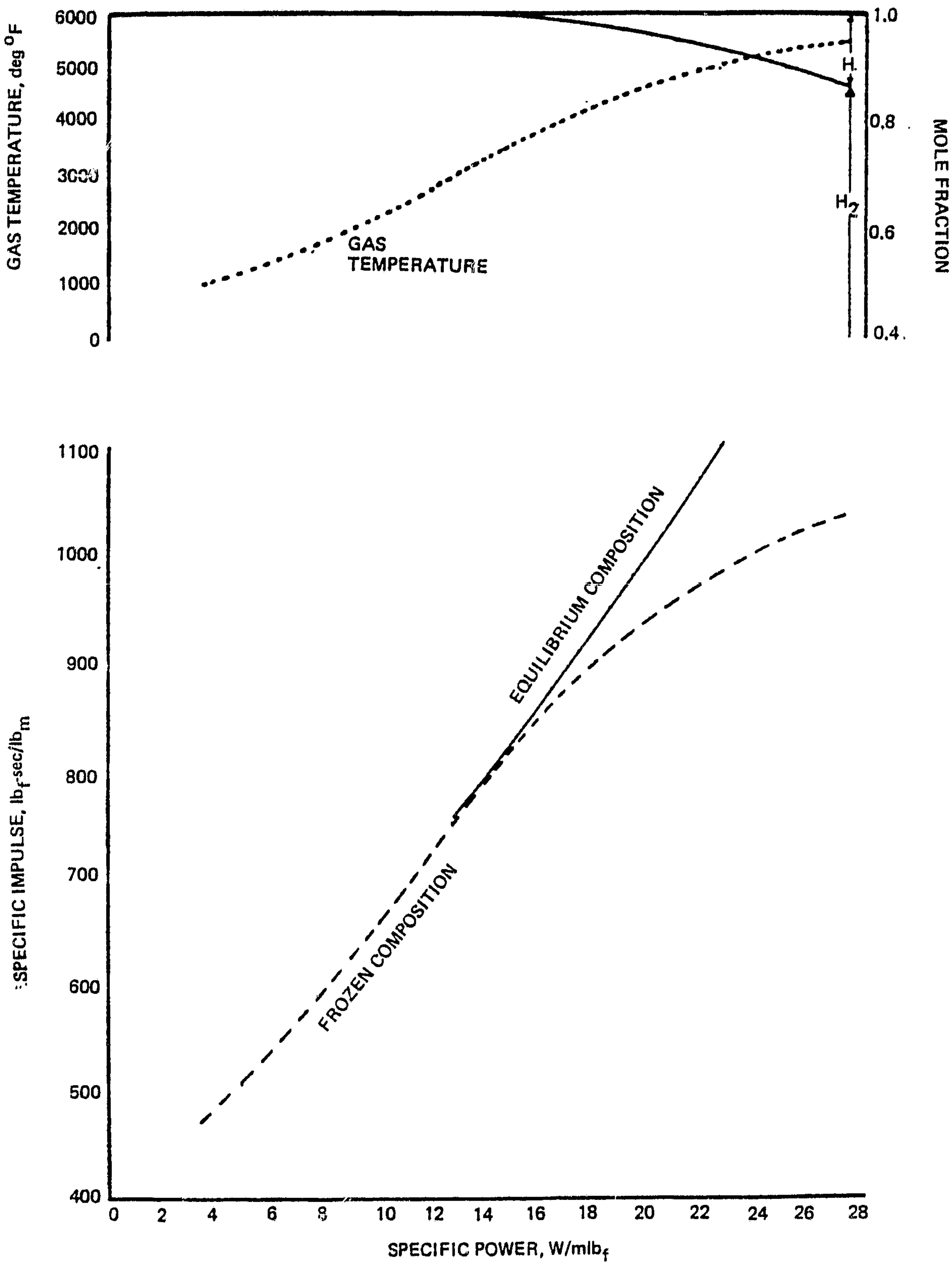


Figure 6-32. Theoretical Performance of H_2

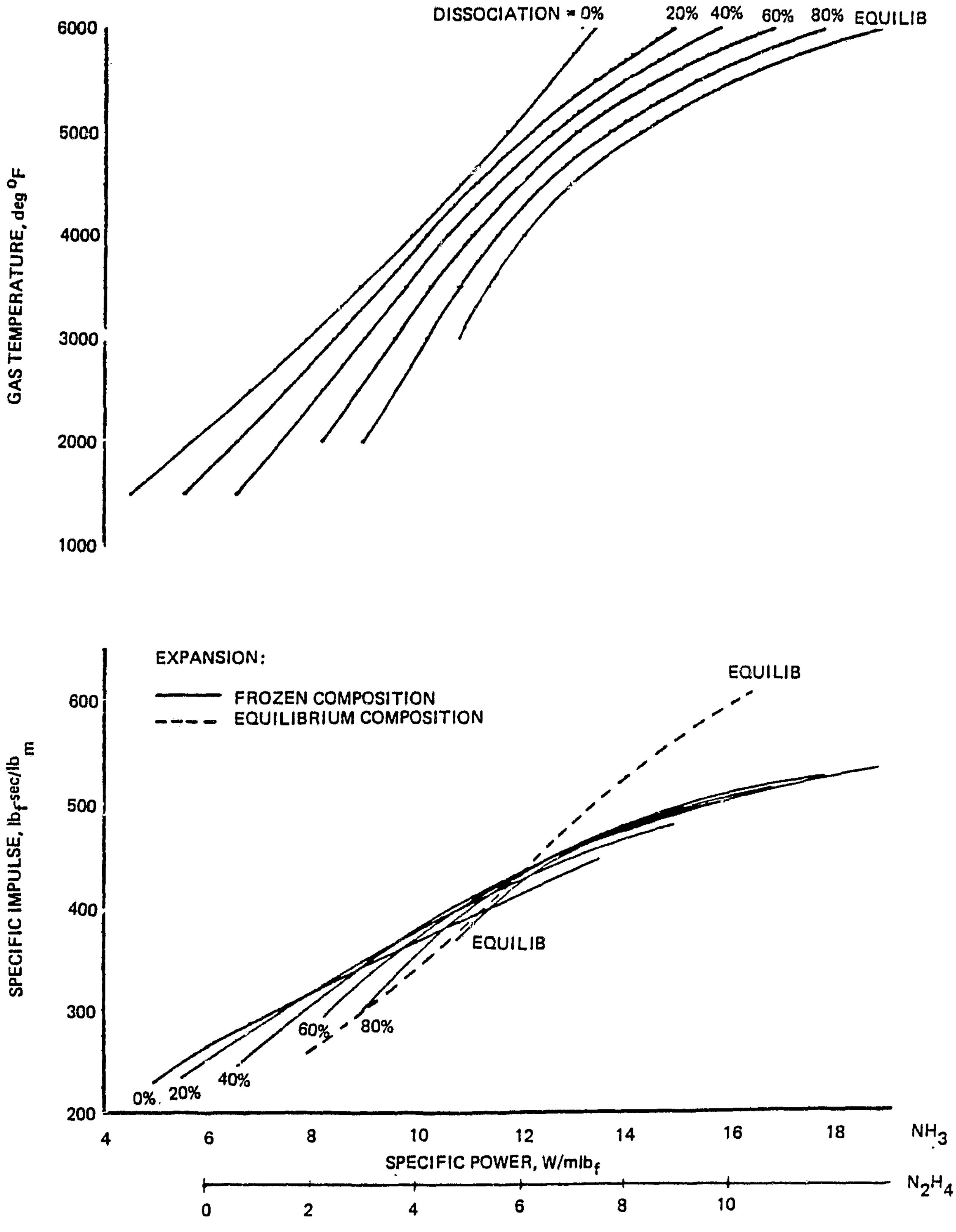


Figure 6-33. Theoretical Performance of NH_3 and N_2H_4

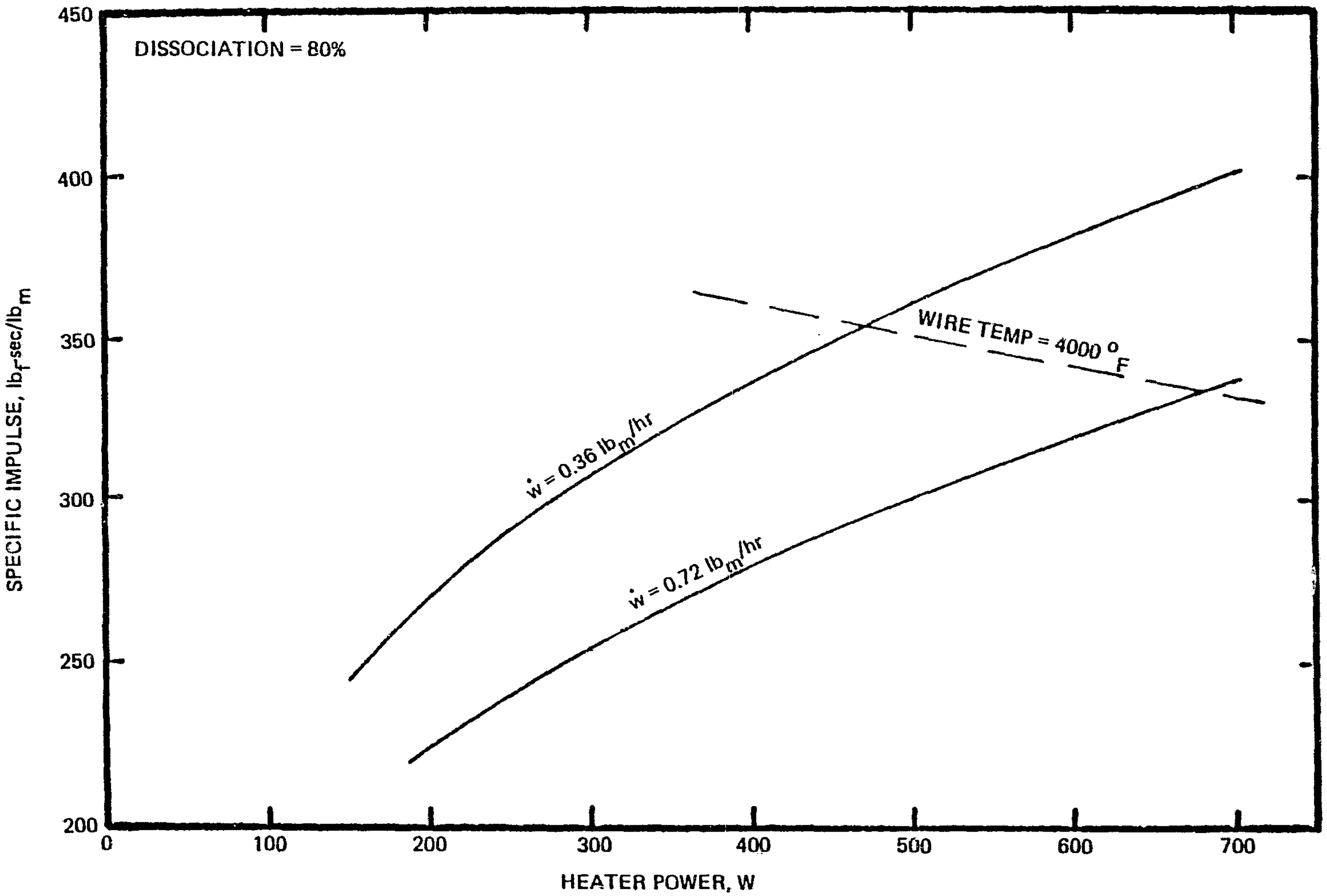


Figure 6-34. Gaseous NH_3 Predicted Specific Impulse Using RRC ACT

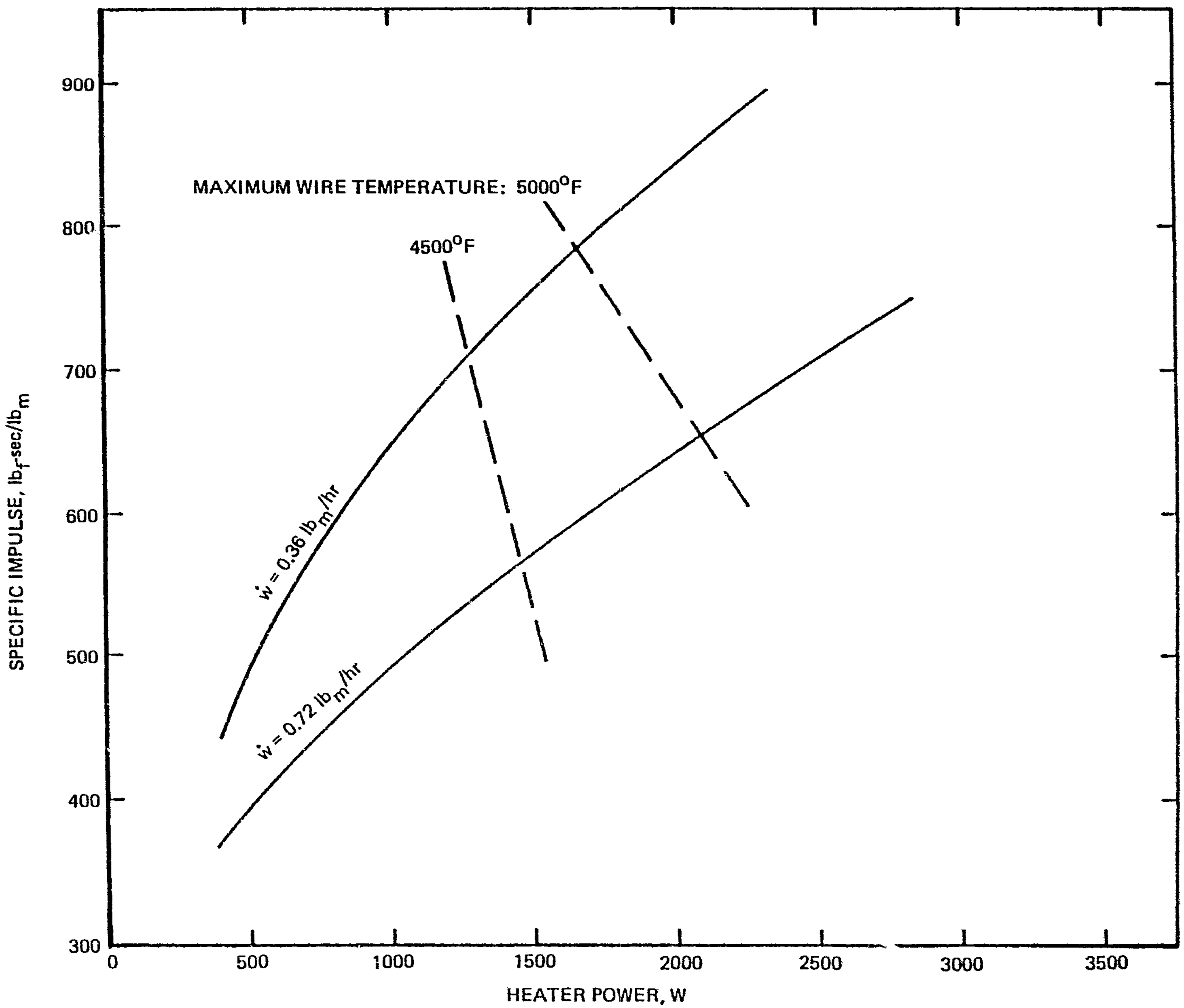


Figure 6-35. Gaseous H_2 Predicted Specific Impulse Using RRC ACT

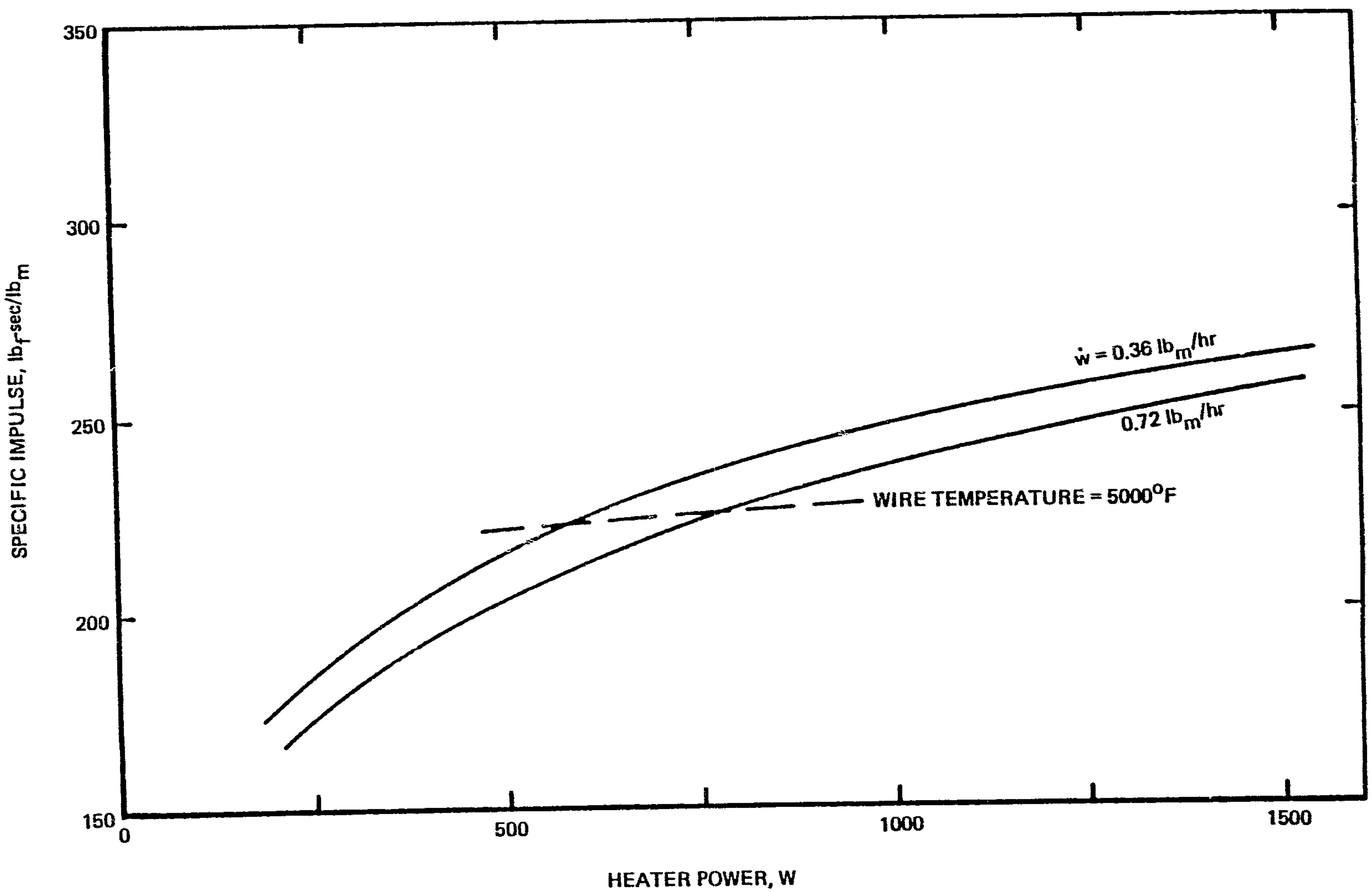


Figure 6-36. Gaseous CO_2 Predicted Specific Impulse Using RRC ACT

rhenium, tungsten, and various alloys thereof. For long-term operations, $4500^{\circ}F$ is a reasonable estimate of the maximum gas exit temperature obtainable with such materials. Platinum has also been considered, but it has a melting point of only $3215^{\circ}F$.

The specific power values in Table 6-21 suggest that a resistojet developing 1.0-lbf thrust would require an electrical power input ranging from 6.4 kW for N_2H_4 to 18.9 kW for H_2, even at a heat-exchanger efficiency of 100%. It is therefore unlikely that high thrust (1.0 lb_f) resistojets would be suitable for the Space Station.

RRC has attempted to estimate the gross characteristics of various types of resistojets designed to operate at thrust levels of 1, 2, and 5N (225, 450, and 1125 mlb_f). These three designs are shown in Figures 6-37, 6-38 and 6-39. As shown here, each of these designs is fed from the catalyst bed shown at the bottom of each figure. Figure 6-37 shows the ACT (Augmented Catalytic Thruster), which is currently in use on the RCA SATCOM G and H satellites. This device has a classic heat exchanger in which propellant runs through a plenum outside a heater element coil. The propellant does not contact the heater element directly. Figure 6-38 illustrates the CTAT (Coiled Tube Augmented Thruster). A hollow coil, which serves as the resistive element and as the propellant passage, forms the heat exchanger. This design is examined in a RRC/AFRPL study that was recently completed. The ETT (Electrothermal Thruster) is the TRW design in use on the Intelsat V spacecraft. In this thruster, a vortex chamber with a heater coil serves as the heat exchanger. Gas passes directly over the coil and is mixed by vortex action.

The results of the scaling study are shown in table 6-22. The figures shown are for N_2H_4, but may be assumed to represent scaling characteristics for any resistojet propellant. The table shows that as thrust level increases there are slight improvements in efficiency and specific impulse, and large increases in thrust/weight ratio, since heat exchanger size increases only slightly. The efficiency of these devices seems to be limited to around 91% because of inherent thermal loss. The 1-N CTAT thruster has lower efficiency than the rest because of the conflicting

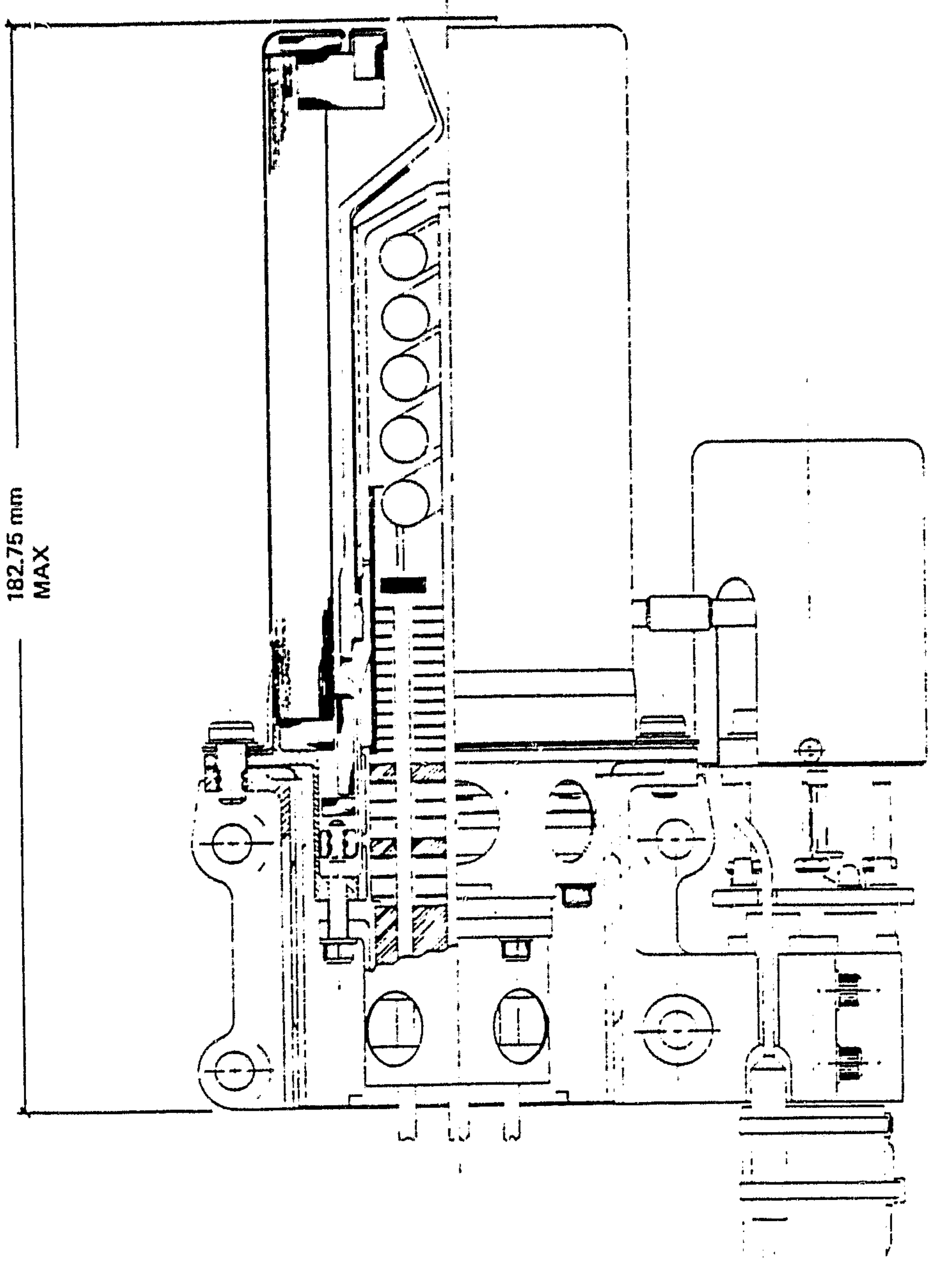


Figure 6-37. ACT .45-lb$_f$ (2 N) Thruster

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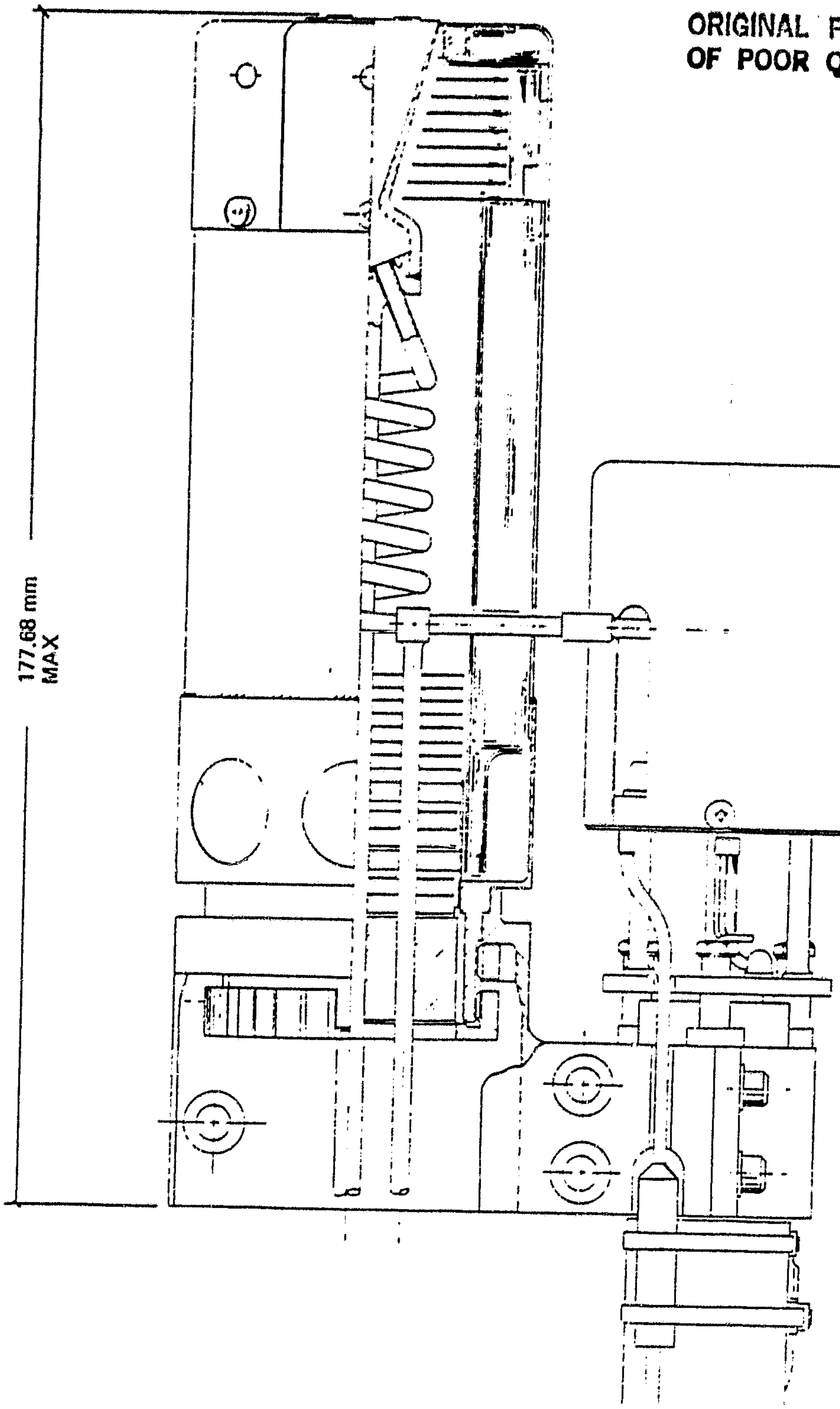

Figure 6-38. CTAT .45-lb$_f$ (2 N) Thruster

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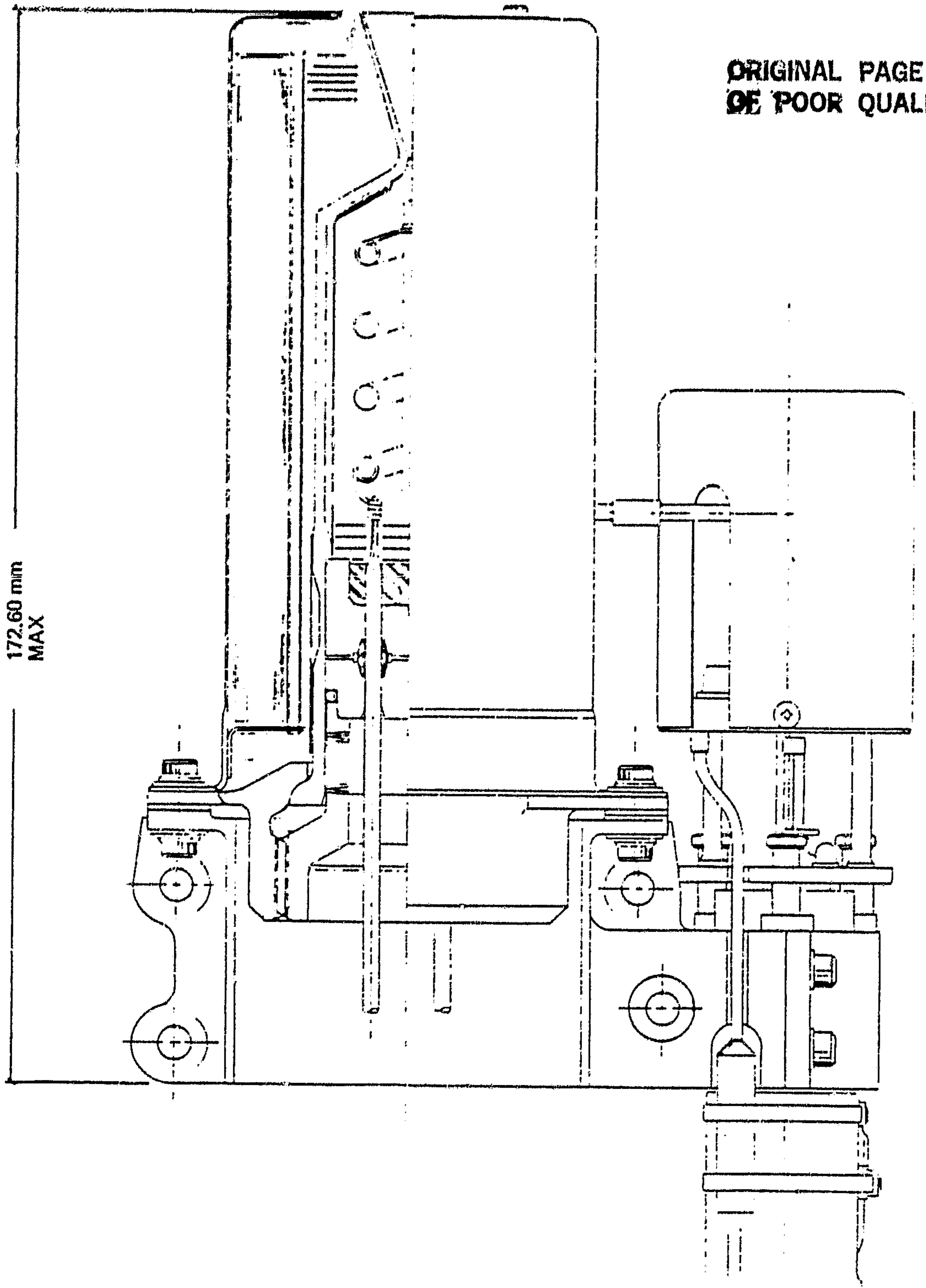

Figure 6-39. ETT .45-lb$_f$ (2 N) Thruster

Table 6-22. Representative Design Parameters for Three Resistojet Configuration (1,2 and 5 N N_2H_4 Thrusters)

Thruster	P, Watts	Electrical Efficiency, %	Specific impulse, sec	Heat exchanger Length, in	Heat exchanger Diameter, in
ACT, 1 N	805	88	300	1.46	0.73
ACT, 2 N	2040	91	300	2.08	1.04
ACT, 5 N	3750	91	300	2.71	1.36
CTAT, 1 N	1060	76	300	3.41	1.72
CTAT, 2 N	2115	89	300	2.51	1.42
CTAT, 5 N	4050	89	300	3.09	1.64
ETT, 1 N	830	88	300	1.28	0.84
ETT, 2 N	2070	90	300	1.77	1.08
ETT, 5 N	3865	91	300	2.68	1.54

ACT = Radiative-wire augmented catalytic thruster (RCC)

CTAT = Coiled-tube augmented catalytic thruster (RRC)

ETT = Vortex-chamber augmented thermal engine (TRW) requirements on the tube to serve as a pressure vessel and a resistive element. The tube is oversized with regard to the heat transfer surface, which causes a greater radiative thermal loss.

6.4.2.2 Thruster Life

In general, a resistojet is less reliable than a conventional engine of similar output because of the additional failure modes associated with the augmentation heater. However, it is possible to make augmentation heaters that are capable of running for 1000 hr or more. Table 6-23 summarizes the results of a test program conducted by the Marquardt Corp. in which several resistojets were run on H_2 and NH_3 for as long as 8,000 hours.[1] Although one unit developed a leak and another short-circuited, these problems apparently occurred beyond the 1000-hr point. Almost all of the resistojets built to date have had less than 1 lb_f of thrust. At this thrust level it would be more difficult to make a hydrazine decomposition reactor last 1000 hr than to make an augmentation heater do so, largely because of degradation of the propellant tube.

Rhenium or moly-rhenium generally are used in the heater structure for high performance and long life. However, since rhenium has a high affinity for oxygen, CO_2 would not be the propellant of choice in these thrusters because, at the temperatures exploiting the rhenium benefits, CO_2 will dissociate into CO, O, O_2, and C. The temperature limit to prevent CO_2 dissociation is under 3000^oF (see figure 6-31). Performance will therefore be limited to around 190 lb_f-sec/lbm I_{sp} and could, therefore, use refractory metals. Platinum or certain ceramics would allow somewhat higher performance and yet not be sensitive to oxidation from the dissociation products. Operation at the lower perforance levels necessary for the Space Station will enhance resistojet life.

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1 Yoshida, R. Y., Halback, C. R., Page, R. J., Short, R. A., and Hill, C. S., "Resistojet Thruster Life Tests and High Vacuum Performance," NASA CR-66970, 1970.

Table 6-23. Nominal Values for Resistojet Thruster Life Tests

	Thruster Serial No.						
	S-1	S-2	B-2	S-3	S-4	S-5	S-6
Propellant	H_2	H_2	H_2	NH_3	NH_3	NH_3	NH_3
Specific impulse, sec (measured)	550	560**	560	300	300	300	300
Specific impulse, sec (estimated for space)	660 300*	670	670	320	320	320	320
Test duration, hours	7,858	1,426	6,023	8,048	8,152	8,052	8,134
Electric power, watts	258	220	222	131	189	145	145
Supply pressure, psia	50	50	50	44	44	44	44
Thrust, millipound	11.7	10.0	10.5	9.2	11.9	10.6	10.5
Mass flow, lb/sec x 10^5	2.13	1.79	1.86	3.06	3.97	3.53	3.50
Average max. gas temperature, °K	2,150	2,150	2,150	2,100	2,100	2,100	2,100
Average max. engine temperature, °K	2,200†	2,200†	2,200†	2,200†	2,200†	2,200†	2,200†

LEGEND:

*Developed leak in outer heating element, test continued for temperature data.

**Shorted

†Estimated from data. During course of the test, the thrusters rose to temperatures of 2,400°K for periods of weeks because of drift of facility controls (regulators, relays, etc.)

6.4.3 Bipropellant Thrusters

There are three basic limitations affecting current SOA bipropellant thrusters: thruster performance at low chamber pressures, thruster life, and development of quick disconnect component interfaces for easy and rapid component changeout. These limitations are discussed in the following sections.

6.4.3.1 Low-Pressure Performance

There are many reasons for operating bipropellent thrusters at a low chamber pressure. One reason is to avoid using a compressor on board the station which would increase the propulsion system weight and power consumption and decrease its reliability. Another reason for using a low chamber pressure is to minimize tank weight. For liquid propellants, as the tank pressure decreases, the tank weight also decreases. It is assumed for gaseous propellants (i.e., GO_2/GH_2) that for long-term (90-day) storage, they will be kept in another state (i.e., H_2O or LO_2/LH_2). Thus, only small accumulators would be needed and a small increase in volume for the lower pressure would not be significant. A third reason for using a low chamber pressure is to minimize safety hazards. High-pressure storage tanks create more of a safety hazard than low-pressure storage tanks. Using a lower chamber pressure may seem ideal, but there are tradeoffs.

As noted in section 6.4.1.1, there is an inverse correlation between the chamber pressure and thruster size (i.e., as the chamber pressure decreases, the thruster size increases). Figures 6-40 and 6-41 show how lower pressures effect the thrust level for both 110 lb_f and 1 lb_f thrusters, and figures 6-42 and 6-43 show the effects of lower chamber pressures on specific impulse performance. At the lower chamber pressures, both thrusters perform at approximately 55% of their normal thrust levels.

6.4.3.2 Thruster Life

There are four factors that affect thruster life: (1) chamber temperature, (2) propellant throughput, (3) the number of start-stop cycles, and (4)

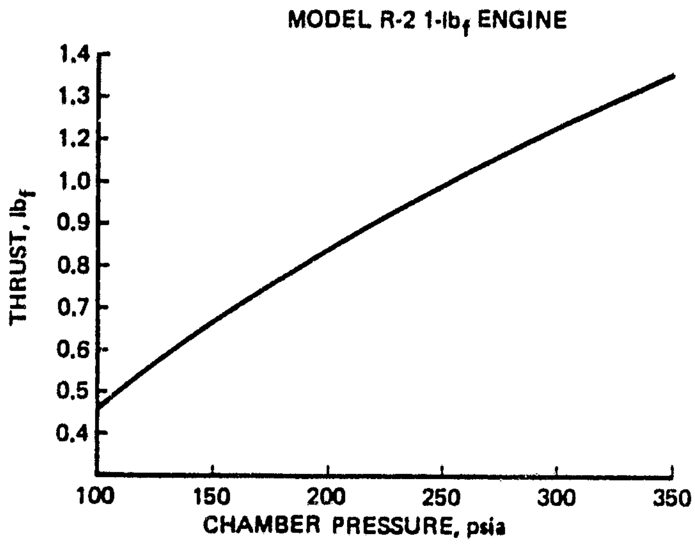


Figure 6-40. Thrust Performance of a 1-lb$_f$ Engine for Varying Chamber Pressures

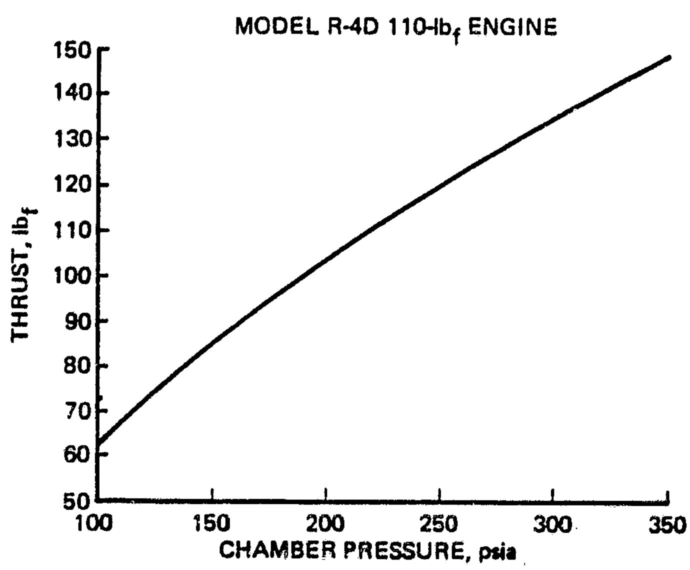


Figure 6-41. Thrust Performance of a 110 lb$_f$ Engine for Varying Chamber Pressures

NOTE: O/F = 1.65 ± 0.1
1 lb$_f$ = 4.45 NEWTON
1 psia = 6.895 kN/m^2 (kPa)

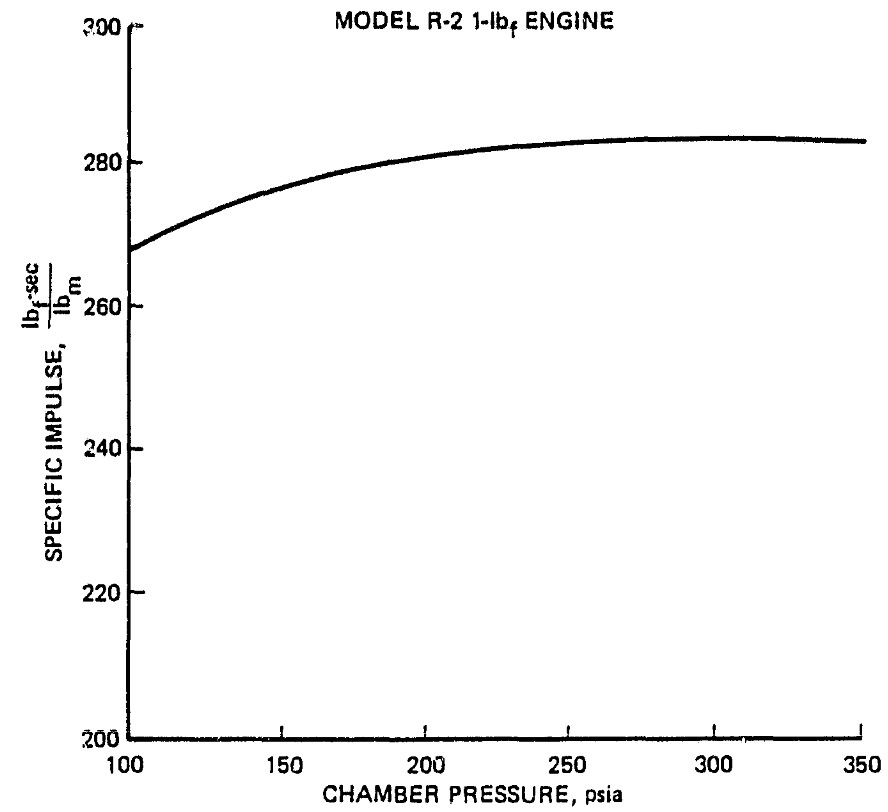


Figure 6-42. Performance of a 1-lb$_f$ Thruster for Varying Chamber Pressures

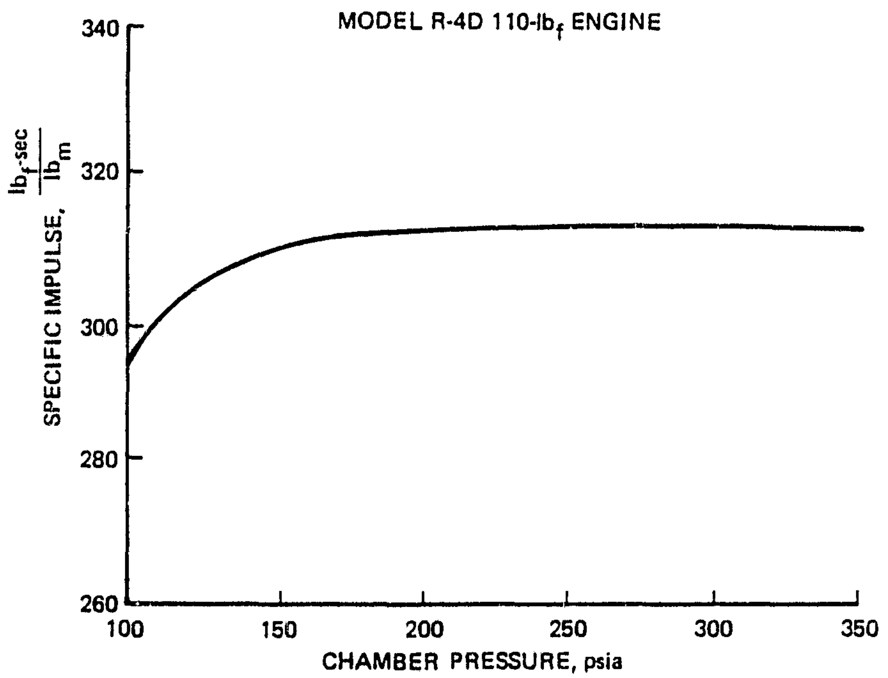


Figure 6-43. Performance of a 110 lb$_f$ Thruster for Varying Chamber Pressures

exhaust constituents.

Chamber temperature is the most prominant factor affecting thruster life. The high combustion temperatures (4000° to $6000^{\circ}F$) and the high heat transfer rates from the hot gas to the chamber walls (0.5 to 50 BTU/in^{2}-sec) can cause the thrust chamber materials to weaken and eventually fail. The most common types of cooling methods are regenerative and film cooling. Both use propellant to cool the chambers, but regenerative cooling is usually more efficient because of the manner in which it uses the heat from the thrusters to warm the fuel before it enters the chamber.

The Jet Propulsion Laboratory, in conjunction with Aerojet Tech Systems, performed a series of experiments comparing a rhenium thrust chamber and a regeneratively cooled thrust chamber to determine low-thrust, long-life oxygen/hydrogen thruster capabilities.[1] Rhenium possesses good qualities in that it has a high melting temperature ($5760^{\circ}F$) and good strength at high temperatures. However, because rhenium readily oxidizes, a film of hydrogen is required as an oxidation barrier. Figures 6-44, 6-45, 6-46, and 6-47 show thermocouple locations and their corresponding chamber pressures for the rhenium and regeneratively cooled thrusters, respectively. Due to the lower pressure and flow rate, the regeneratively cooled thruster lacked sufficient cooling. Therefore, the run times were shortened by the high temperatures of the head-end seal (T_{C1}). Overall, the rhenium thruster shows promise for use on the Space Station. Further testing is still required on both thrusters for: (1) compatibility with other propellants, (2) increased temperatures, (3) long burn times, and (4) improved cooling efficiency.

Chamber temperature can also be lowered by reducing the mixture ratio. For example, at a chamber pressure of 50 psia and an expansion ratio of 40:1, gaseous oxygen and hydrogen optimize at approximately 3.7:1 for an ideal

1 Appel, M. A., JPL; Schoenman, L. and Berkman, D. K., Aerojet Tech. Systems Co., "Oxygen/Hydrogen Thrusters for the Space Station Auxiliary Propulsion system," NASA Contract.

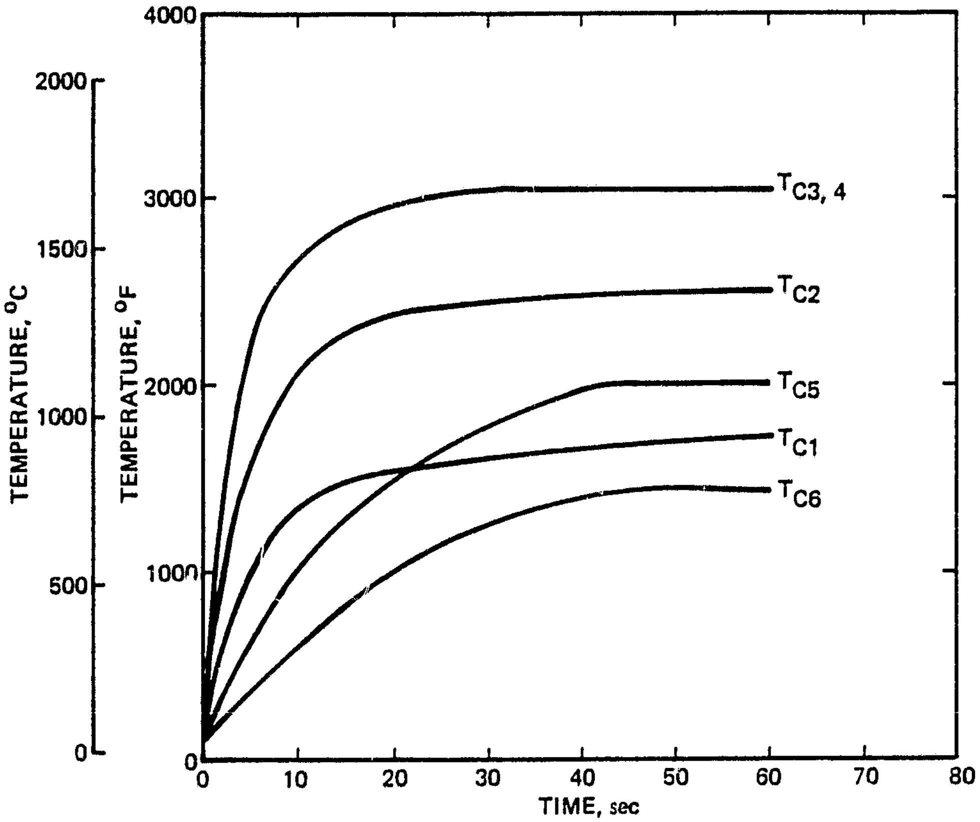


Figure 6-44. Rhenium Chamber Temperatures

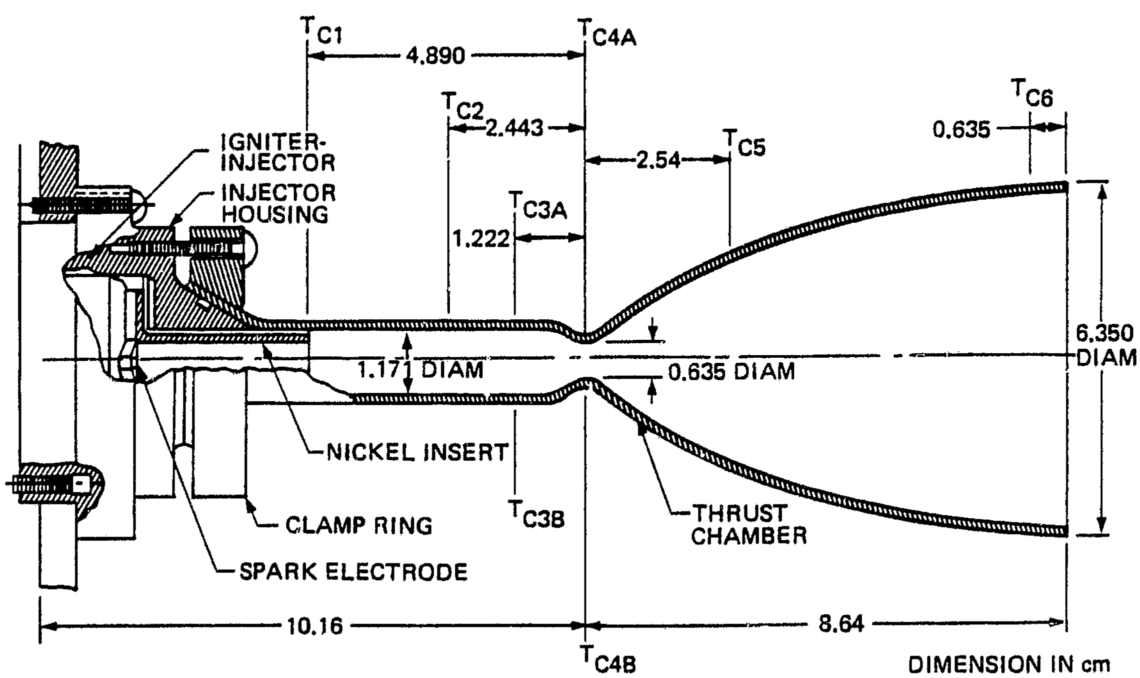


Figure 6-45. Cross Section of Rhenium Thrust Chamber Showing Test Thermocouple Locations

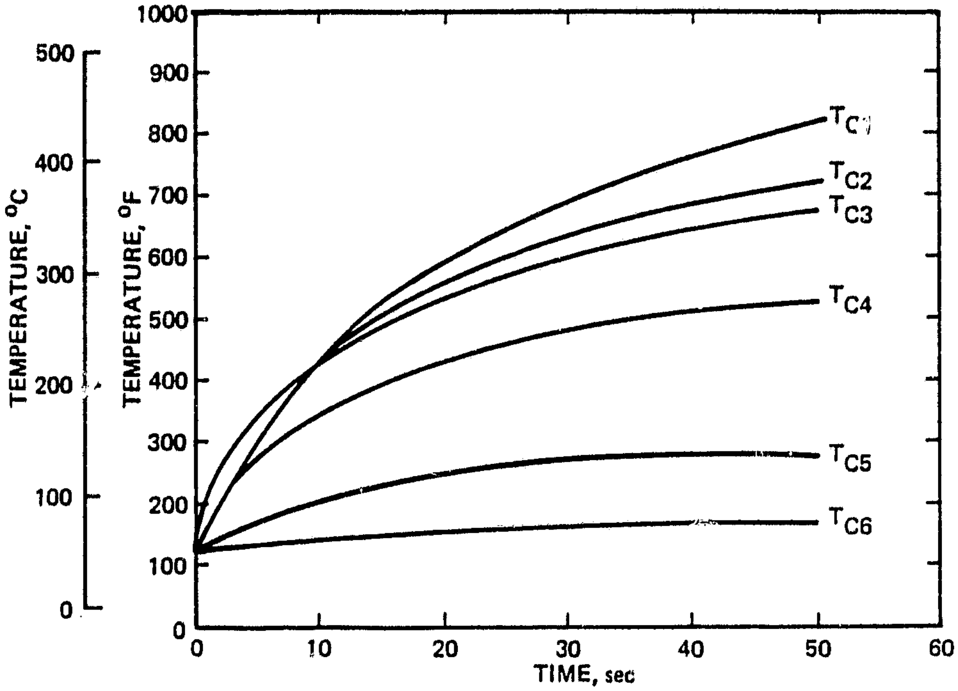


Figure 6-46. Regenerative Chamber Temperatures

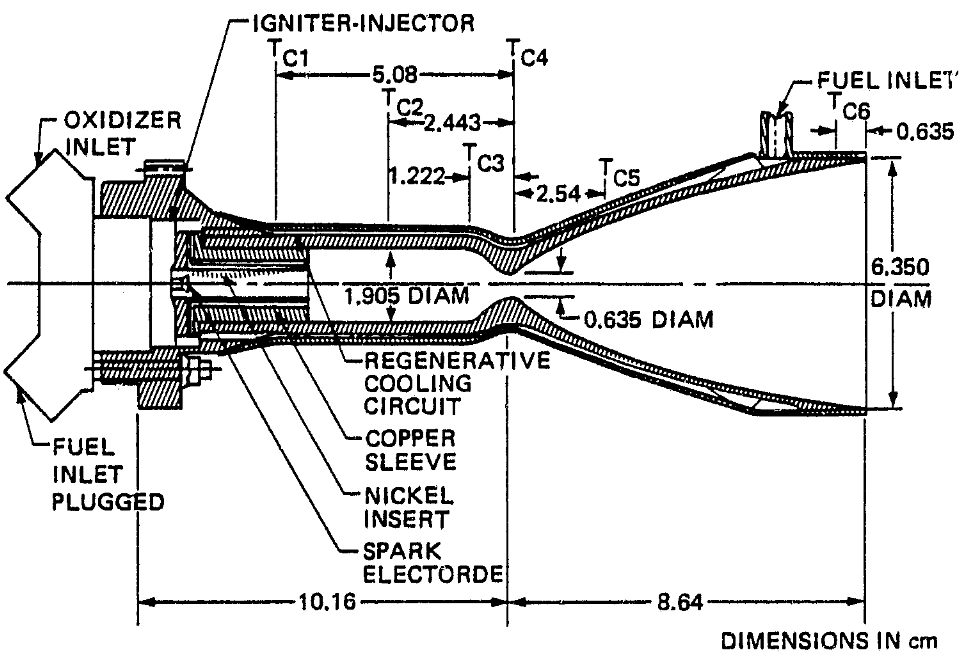


Figure 6-47. Cross Section of Regeneratively Cooled Chamber Showing Test Thermocouple Locations

specific impulse of approximately 473 lb_f-sec/lbm (see figure 6-20). At this mixture ratio, the ideal chamber temperature is approximately $5200^{\circ}R$ (see figure 6-48). By lowering the mixture ratio to 2.5:1, the ideal specific impulse drops only about one percent, but the ideal temperature of the chamber gas drops to approximately $4280^{\circ}R$. This could reduce the actual chamber wall temperature below $2000^{\circ}R$, well below the limit of many currently used thruster materials, and thereby increasing thruster life tremendously.

Propellant throughput is the total amount of propellant that flows through an engine in a given time and also affects thruster life. The more propellant that flows through the engine, the greater the wear on the material. However, this varies with the type of propellant (i.e., its corrosiveness), the temperature, the flowrate, the cross-sectional area, and the pressure (gases mainly).

A third factor that affects thruster life is the number of start-stop cycles a thruster must endure. As discussed earlier in the section on thrusting strategy, the number of start-stop cycles an engine must perform depend on the thrust level chosen, thrust duration, and duty cycle. Currently, bipropellant thrusters in the 0.1 lb_f to 150 lb_f thrust range, are capable of approximately 350,000 starts. Most of the manufacturers agree that projected capabilities will enable an unlimited number of start-stop cycles.

A fourth factor that can affect the life of a thruster is the exhaust constituents. For example, running an O_2/H_2 engine close to stoichiometric (8:1) will produce approximately 30% O_2, 2% O, 2% H, and 2% OH. All four of these exhaust constituents can cause corrosion and limit the life of the thruster.

6.4.3.3 Quick Disconnect Development

It is essential to develop quick disconnect component interfaces for easy and rapid component changeout, if an orbit changeout by a suited astronaut is required. The Johnson Space Center has issued an RFP for the design,

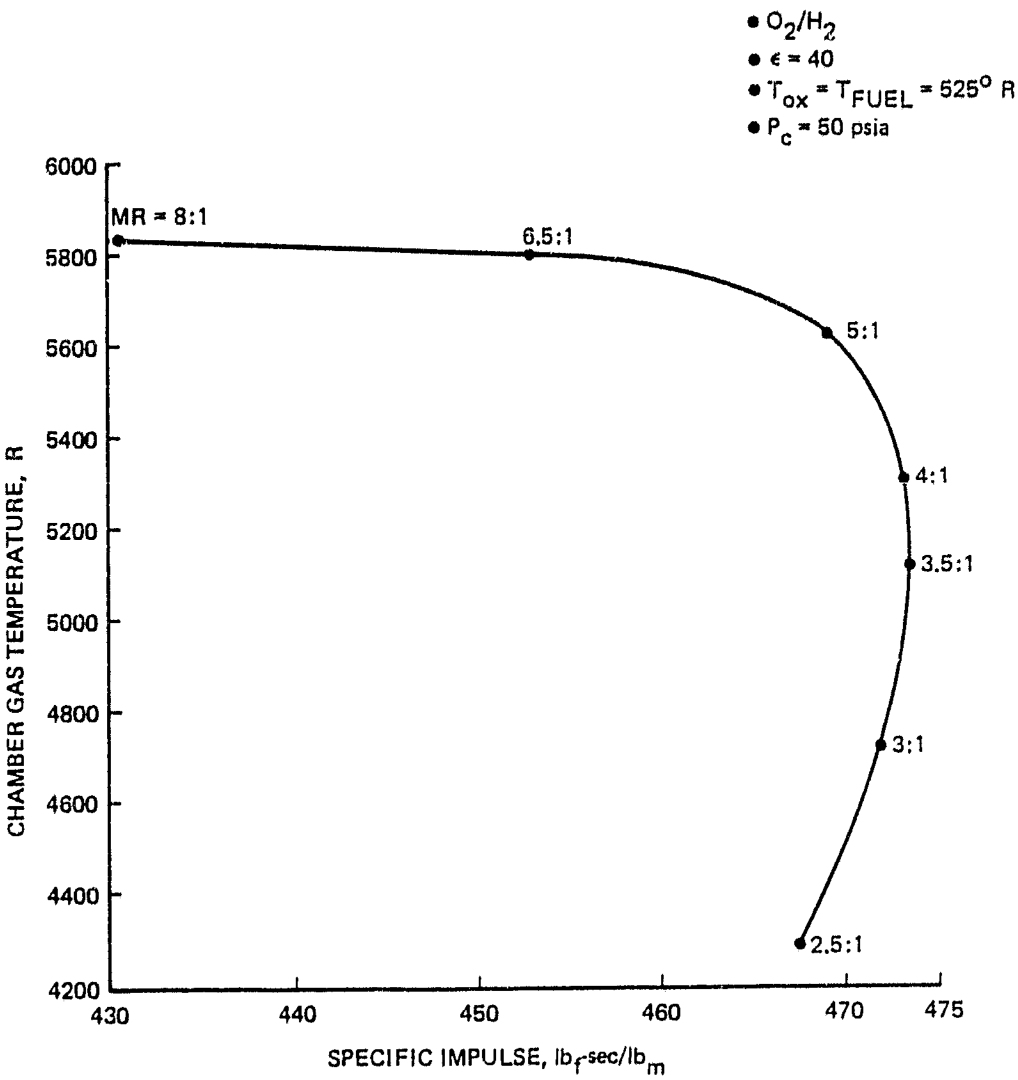


Figure 6-48. Ideal Chamber Gas Temperature for Varying Mixture Ratios

and fabrication of couplings for manual operations in space. It was initially intended to entail only monopropellant hydrazine, but it may be expanded to include other fluids and temperatures, such as O_2/H_2.

6.4.4 Discussion

The technology improvements required to enable long-life, efficient, and reliable thrusters for the Space Station can realistically be obtained in the next five to ten years. High-energy thrusters, such as arcjets and ion thrusters, are not currently applicable to the Space Station system. However, conventional monopropellant thrusters, bipropellant thrusters, and resistojets with funded development can be available and qualified by 1991 to satisfy the requirements discussed in section 5.0.

6.5 Safety and Other Issues

Propellants and thruster candidates have also been compared to the extent possible on the basis of complexity, safety, maintainability, interface requirements, throttleability, and development risks. Some of these issues overlap and must ultimately be addressed at a systems level to be evaluated. In each case, only those propellants that pose potential problems are discussed. In many cases, it is possible to do so only qualitatively.

6.5.1 Complexity

An efficient way to define complexity in this study is by the use of an example. Consider the two systems: cold-gas and bipropellant. The cold-gas system is a simpler or less complex system than the bipropellant system. The cold-gas system operates by expending a propellant through a nozzle (as discussed in section 6.2.2.1). Since it is normally a monopropellant system requiring little or no heat addition, it does not require a lot of wiring for heaters, pumps, igniters, etc. and it does not require a cooling system for the chamber, throat, or nozzle. The bipropellant system, on the other hand, requires more tanks, valves, gauges, wiring, and unless hypergolic, it also uses an igniter. Depending on the type of propellant used, the system can become even more complicated (i.e., cryogenic propellants). Hence, what is meant by complexity actually includes all of the qualitative and quantitative conditions together. To accurately determine how complex each propulsion system is requires an in-depth trade study and an agreed-upon ranking system. In general, a system becomes more complex as the number of parts and interfaces increases, with a corresponding increase in the potential for failure and higher costs. This study assumes that thruster hardware increases in complexity as follows: cold-gas, warm-gas, catalytic, resistojet, arcjet, bipropellants. Propellant system complexity also varies, increasing as follows: storable liquids, pressurized liquids, and cryogenic liquids. Table 6-24 is a simplistic attempt to rank each system in order of complexity.

6.5.2 Safety

The safety issues include four categories: propellant toxicity, plume effects, electrical hazards, and propellant flammability.

6.5.2.1 Propellant Toxicity

Several criteria are used to characterize the toxicity of gases. One of these is the threshold limit value, which is an allowable exposure level for an 8-hr day/40-hr week. Another is the short-term exposure limit, which prescribes allowable exposure levels up to 15 minutes. Still higher exposure levels are permissible for healthy populations under medical supervision; these levels are established by the emergency exposure level.

Since the propellants in Table 6-25 must be handled in a hazardous-materials processing facility at the launch site, turnaround operations will be costly and complex. Special safety procedures will also have to be used on-orbit whenever EVA operations involving propellant systems are conducted. Emergency procedures for handling on-orbit propellant spills that cause suit contamination will have to be developed.

Table 6-24. Relative Complexity of Hardware/Propellant Combinations (Higher Number = More Complex)

System/propellant	Subsystem complexity			
	Propellant	Engine	Electrical	Total
Monopropellant/CO_2	2	1	1	4
Monopropellant/N_2	3 (for large quantities)	1	1	5
Monopropellant/N_2H_4	1	2	1	4
Resistojet/CO_2	2	3	2	7
Resistojet/N_2H_4	1	4	2 (ACT)	7
Resistojet/H_2	3	3	2	8
Resistojet/NH_3	1	3	2	6
Arcjet/N_2H_4	1	5	3	9
Arcjet/NH_3	1	4	3	8
Arcjet/H_2	3	4	3	10
Bipropellant/MMH-NTO	3	5	1	9
Bipropellant/GH_2-GO_2	4	5	2	11
Bipropellant/LH_2-LO_2	5	6	2	13

Table 6-25 shows the values of these limits for CO_2, NH_3, and N_2H_4. Propellants N_2 and H_2, not shown in Table 6-25, are not toxic but are asphyxiants.

Table 6-25. Allowable Exposure Levels for CO_2, NH_3, and N_2H_4

Propellant	Threshold limit value, ppm (8-hr)	Short-term exposure limit, ppm (15-min)	Emergency exposure level, ppm			Ranking (3=worst)
			10-min	15-min	1-hr	
CO_2	5,000	15,500				1
NH_3	25	35	500	300	300	2
N_2H_4	0.1	0.1	30	20	10	3

6.5.2.2 Plume Effects

Exhaust plumes in gaseous or particulate form can affect the Space Station by heating impinged surfaces, leaving sedimentation or abrasions, or by absorbing, reflecting, or emitting radiation; any or all of which can interfere with EVA. EVA exclusion areas may have to be declared during thrusting, depending on the thrust level and location of the thrusters. This, in turn, would place a limit on permissible station imbalance since sufficient imbalance requires desaturation thrusting at more frequent intervals than typical EVA periods (4 to 6 hours).

Different propulsion systems will affect the station in various ways depending on the type of propellant(s), its temperature, and the force with which it is expelled. Most monopropellant exhausts are transparent but CO_2 plumes may freeze and form visible dry-ice flakes. Pressures and temperatures decrease rapidly with distance from the nozzle exit, so a person performing EVA in front of a firing thruster might not suffer damage to his suit. However, the astronaut could receive a force input equal to the engine's thrust.

6.5.2.3 Electrical Hazards

Of the propulsion systems narrowed to in this study for use on the Space Station, only resistojets pose any significant electrical hazards. The voltage and power levels at which the resistojets might operate at are a function of the desired thrust level and performance (I_{sp}). Since resistojets are in the less than one pound force category, their primary function might be for back up attitude control or damping of small distubances, thus reducing the need for high performance and thus large energy requirements.

6.5.2.4 Propellant Flammability

Explosion hazards exist for almost all liquid and gas bipropellant combinations. Some bipropellants, such as hypergolic combinations like N_2O_4/MMH, tend to be more hazardous than others. However, improvements in handling techniques have reduced this hazard to a reasonable safety level. Handling conditions vary with the surrounding environment. Ground handling is usually considered more hazardous because of the air which is a readily available oxidizer. The propellants that present a flammability hazard in air are hydrogen, hydrazine, and ammonia. Table 6-26 presents the flammability limits for these vapors.

Flammability in space is less likely to occur, since the fuel and oxidizers are in separate tanks and would require an igniter in most cases, even if some flammable mixture did occur. In those cases where hypergolic fuels are used, special precautions must be taken to ensure zero leakage from any tanks, pipes, valves, regulators, etc. that might be used.

Table 6-26 Flammability of NH_3, N_2H_4, and H_2 in Air (Volume, Percent)

Propellant	Lower Limit	Upper Limit	Ranking (1=best)
NH_3	15	28	1
H_2	4.0	75	2
N_2H_4	4.7	100	3

Source: U.S. Bureau of Mines Bulletin 503

6.5.3 Maintainability

Mission success and cost minimization require that system downtime be minimized. This will require an in-depth assessment of the resupply techniques used (i.e., tools, replacement parts, propellant), system safety, system integration, and overall reliability. An example of a propulsion system which could simplify maintenance is a removable thruster module as shown in figure 6-49. An arrangement of this type allows tanks, thrusters, igniters, and valves to be easily changed-out with minimum effort and down-time.

6.5.4 Interfaces

The propulsion system has a functional interface with the guidance, navigation, and control system. The latter will generate thrusting commands for attitude control, CMG desaturation, orbit makeup, and maneuvering. Figure 6-1 showed a schematic of the various sytems that will interface with the propulsion sytem. All of the devices being considered have a mechanical, electrical, and thermal interface to the vehicle, and definitions of these interfaces are usually included in propulsion system specifications.

Physical interfaces can be complex. As figure 6-50 shows, distributing propellant from the resupply module to the thrusters involves removeable disconnects and may involve intervening modules, depending on the selected configuration. Physical connection to the electrical power system and the

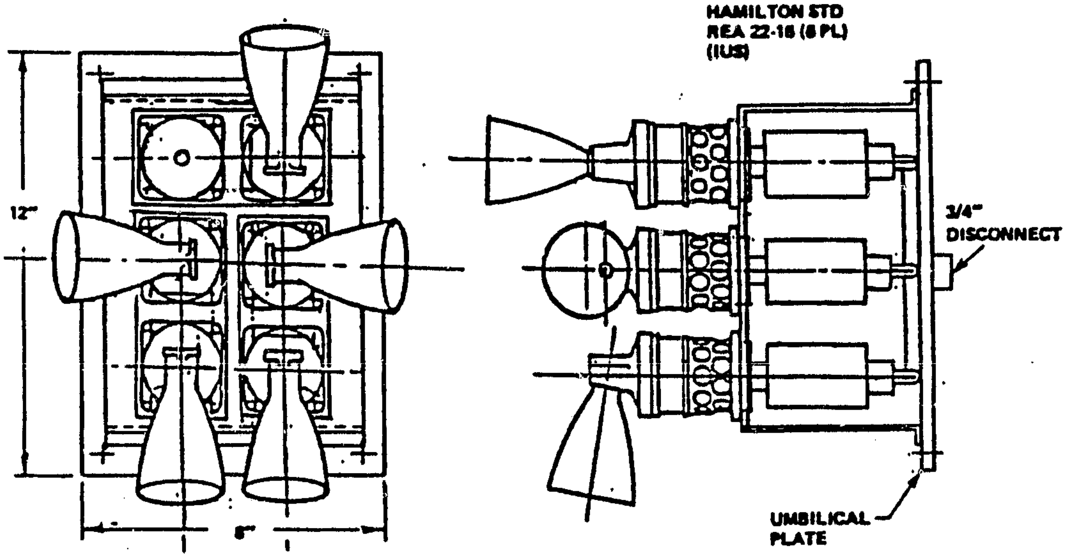


Figure 6-49. Removable Thruster Module

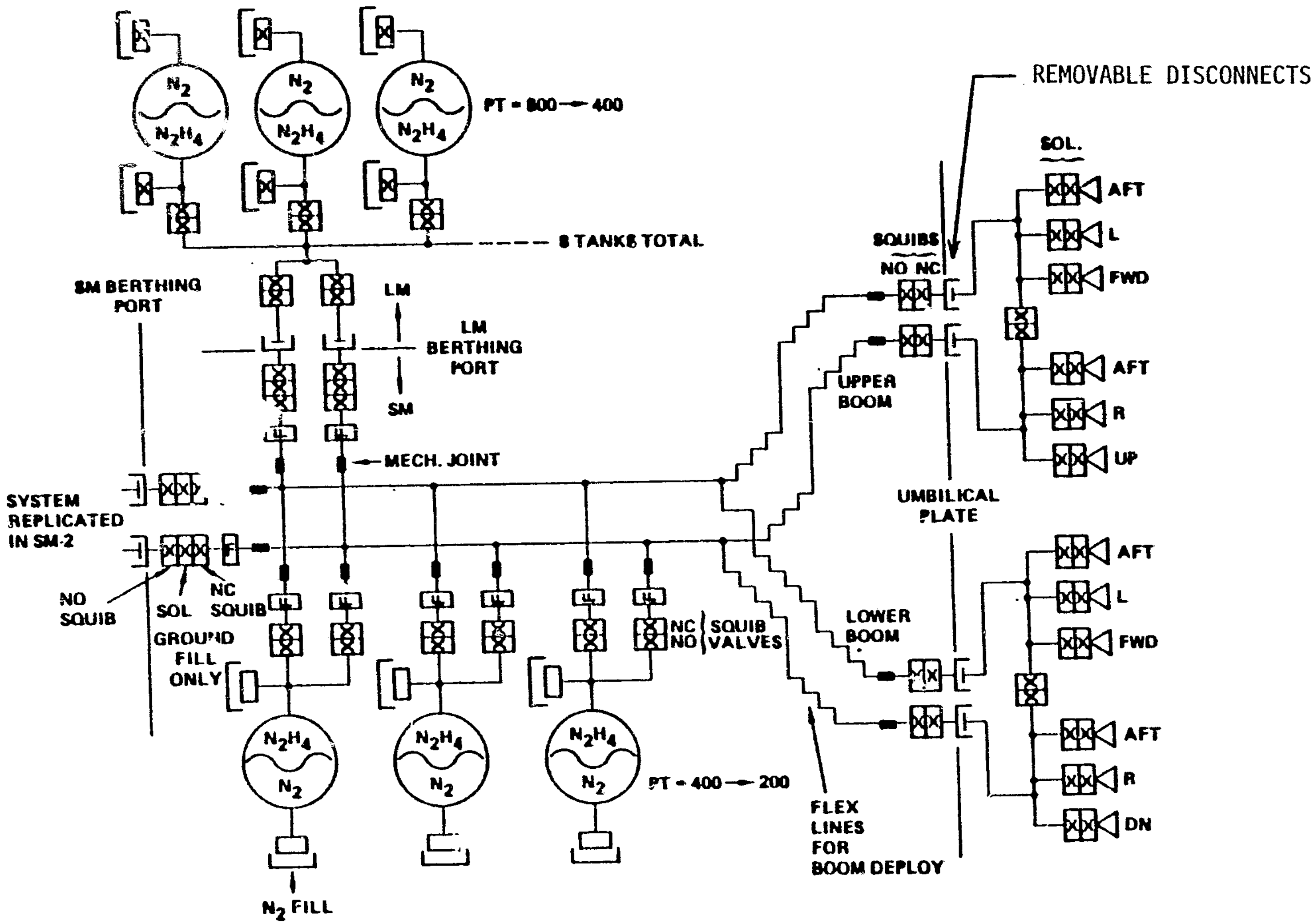


Figure 6-50. Propulsion System Schematic with Removable Disconnects

data management system are necessary in all cases. Hydrazine requires thermal control (heaters, blankets), and probably active heating of distribution lines.

Electrical interfaces vary, depending on the type of thruster. Conventional engines use electrically powered valves and heaters, with total power consumption ranging into the tens of watts. Resistojets currently use .5 to 2.5 kW; arcjets have used up to 30 kW, although 1.5 to 5 kW thrusters are probably adequate for the Space Station. The voltages and currents have been previously discussed.

The thermal interfaces also vary. Non-augmented thrusters operate at the lowest interface temperatures and impose heat loads in the tens of watts on the vehicle. Resistojets can lose up to 50 percent of their electrical input as waste heat, with half of this (i.e., 25 percent of the total) being conducted to the vehicle. As scale increases, so does thermal efficiency; the average heat load on the vehicle imposed by a resistojet would be about 8 percent of the augmentation heater power. The heat is transmitted mainly by radiation from the thruster's outer heat shields.

6.5.5 Throttleability

Small-thruster throttling is normally accomplished by controlling the propellant inlet flow rate through the use of a pintle valve. Before microprocessors were available, it was a difficult costly process to control these valves because small differences from pintle-to-pintle caused significant differences in flow area for a given pintle position. Additionally, the flow rate was not a linear function of pintle position. Now, however, the use of a microprocessor requires only that a measurement of flow rate as a function of pintle position be determined. The microprocessor is then provided with this information and, when a certain flow rate is desired, the pintle is commanded to move to the position that provides that flow rate. Virtually any single-valued curve can be accommodated.

Cold-gas and catalytic monopropellant thrusters are duty-cycle limited only by the operating characteristics of their propellant valves. The engines themselves are capable of any duty cycle, from a single pulse to steady firing. These thrusters can be throttled to operate over a wide range and, in fact, some throttling often improves catalyst performance and life. However, certain high-rate, variable-length pulse sequences can accelerate catalyst attrition.

Radiating-wire resistojets can be off-pulsed and the RRC augmented catalyic thruster has been run at cycles as low as 3 percent. The resulting high structural temperatures cause specific impulse to increase, but may shorten thruster life. Other types of resistojets--the coiled-tube and vortex-chamber types--are essentially limited to steady-state firing. The same restriction probably applies to arcjets.

N_2O_4/MMH bipropellant thrusters are duty-cycle limited by valve characteristics. These thrusters are capable of virtually any duty cycle and most are designed for steady-state operation. These thrusters can be throttled to operate over a wide thrust range.

H_2/O_2 bipropellant thrusters are duty-cycle limited primarily by valve and ignitor life and operating characteristics. Throttling over a wide range (e.g., 2.5 to 50 lb_f) has been demonstrated using prototype thrusters.

6.5.6 Development Risk

The monopropellant thrusters available can produce thrust levels in the range of interest (0.1 to 100 lb_f). As noted previously, there is no signficant degradation mechanism for a cold-gas thruster, so virtually any proposed mission of any duration could be performed with this technology. Catalytic engines are limited to about 500 hours of firing by the mechanisms already mentioned. This has been adequate, so development past this point has not been aggressively pursued. The catalyst generally used (Shell 405) is chemically capable of several times this figure, and life extension to 1000 hours is primarily a matter of refining current designs to improve their bed voiding and/or transport tube occlusion characteristics. The associated risk is relatively low.

Bipropellant thrusters using N_2O_4/MMH are available from 1.0 lb_f and up and have been in use for years. Considerable developmental work and testing has been conducted on N_2O_4/MMH thrusters with thrust as low as 0.5 lb_f. Very little risk exists for N_2O_4/MMH thruster development of any size down to this level.

Bipropellant thrusters utilizing hydrogen and oxygen can be categorized by propellant state (i.e., liquid and gas). Requirements for O_2/H_2 thrusters below 500 lb_f have not existed until recently and, as a result, there are no space-qualified thrusters available in this range. Work is planned on a NASA-LeRC contract on 25 and/or 100 lb_f LH_2/LO_2 thrusters. Considerable developmental work has been conducted with GH_2/GO_2 for thrusters in the 5 to 50 lb_f thrust range and down to 0.1 lb_f. There is moderate developmental risk for thrusters in the 20 lb_f range, the biggest concern being the ignitor spark life.

6.6 Propulsion Combinations to Meet High/Low Thrust Requirements

The propulsion system combinations selected for discussion in this section were chosen for their ability to satisfy the requirements established in the previous sections and are divided into three levels of DDT&E expenditure. These various propulsion system combinations are merely examples of ways to meet high/low thrust requirements and should not be construed as a recommendation for a particular propulsion system. The three levels of expenditure are: (1) low DDT&E, with little or no synergism; (2) moderate DDT&E, with increased synergism and growth progression; and (3) high DDT&E, with maximum synergism and growth progression.

High/low thrust combinations are suggested based on the premise that a satisfactory throttleable system that can meet all thrusting requirements will not be available for Space Station use. "High" thrust is considered to be 10 lb_f and above and low thrust is anything below 10 lb_f.

6.6.1 Low DDT&E

Based on the propulsion requirements set down in the previous sections, there is only one system that minimizes the front-end DDT&E costs. The hydrazine system is state-of-the-art, hydrazine thrusters are available in virtually any size, and they are space qualified. However, depending on the particular duty cycle desired, which effects the thrust level, thrusting frequency, and thrusting duration, the long-term or life-cycle cost could become large. The chosen duty cycle will affect the life of the catalyst bed and the amount of contamination emitted into the surrounding Space Station environment. Also, the hydrazine system will only be synergistic if it is also used on some of the free-flyers. If the duty cycle is relatively high, the hydrazine thrusters may have to be changed out rather frequently thereby increasing the life-cycle cost of the system. It should be noted however, that if a long-life catalyst bed system can be qualified, as discussed in section 6.4.1.2, the life-cycle costs of the hydrazine system can be greatly reduced, making it more attractive to use hydrazine thrusters.

To satisfy the low thrust requirements, a reliable system is a cold gas, blowdown GN_2 system. In addition to being reliable, this system is state-of-the-art, qualified, simple, clean, and has a cool exhaust with low contamination potential. Nitrogen also is available onboard for use in the ECLSS. The only concern regarding this system is the quantity of propellant required in view of the low specific impulse attainable.

6.6.2 Moderate DDT&E

There are three possible systems that stand out as viable choices for a high/low thrust combination requiring moderate DDT&E costs. They are (1) a N_2O_4/MMH high-thrust system coupled with a CO_2 and/or H_2 resistojet low-thrust system; (2) a GO_2/ GH_2 high-thrust system (using supercritical storage) coupled with a CO_2 and/or H_2 resistojet low thrust system; and (3) a GO_2/GH_2 high-thrust system (using water electrolysis) coupled with a CO_2 and/or H_2 resistojet low-thrust system.

6.6.2.1 N_2O_4/MMH Thrusters with CO_2 and/or H_2 Resistojets

An N_2O_4/MMH system is viable because the thrusters are state-of- the-art, qualified, and available in almost any size desired to meet the required thrust range. Any bipropellant system is, however, more complex and initially more expensive than a hydrazine system. However, over the life of the Space Station, N_2O_4/MMH may prove to be less expensive than hydrazine because of longer thruster life and synergism with the OMV and the Orbiter OMS enabling the N_2O_4/MMH system to use scavenged propellant from the OMS tanks and share the storage tanks required for the OMV. Scavenging propellant from the OMS tanks, however, would require an acquisition and transfer system which would add additional costs. A major disadvantage of using N_2O_4/MMH is the contamination from the exhaust plume emitted into the surrounding environment, especially during engine startup and shutdown.

Low thrust requirements can be satisfied using effluent from other station systems which otherwise must be stored and returned to Earth via the Orbiter. Effluent that has promise as propellant are CO_2 resulting from crew metabolism and H_2 from the water electrolysis unit of the ECLSS (see Figure 6-3) used to produce O_2 for crew respiration. Table 6-27 illustrates the annual CO_2 production for the various station sizes and manning levels and the associated I_{sp} requirements. It is seen that, at the lower manning levels, the required I_{sp} can be attained by exhausting CO_2 through a nozzle with only slight warming since 300^oF CO_2 produces 67 lb_f-sec/lbm (see Table 6-1). At the higher manning levels, an excess of CO_2 exists for the baseline ECLSS. As shown in Figure 6-2, in 1995 an advanced ECLSS with a Sabatier CO_2 reduction unit is expected to be incorporated which will diminish the effluent by 38% and from all CO_2 to a CO_2/CH_4 mixture. An advanced ECLSS schematic is shown in Figure B-9 of Appendix B. With the advanced ECLSS at the lower manning levels, the I_{sp} requirement becomes approximately 115 lb^f-sec/lbm while, at the higher manning levels, it is not necessary to add power to the CO_2 since 67 lb_f-sec/lbm is adequate. Figure 6-31 illustrates the I_{sp}, dissociation, and gas temperature as a function of specific power for a CO_2 resistojet.

Station size (number men)	Total impulse* (lbf-sec)	Annual production (lbm)		Specific impulse (lbf-sec/lbm)		Specific power (kW/lbf)	
		CO_2	H_2	CO_2	H_2	CO_2	H_2
2 to 4	115,148	1624 to 3249	189 to 379	71 to 35	609 to 304	–	8.8 to < 0.5
4 to 6	225,962	3249 to 4873	379 to 568	70 to 46	596 to 398	–	8.1 to < 1.0
8 to 12	420,992	6497 to 9746	758 to 1136	65 to 43	555 to 371	–	6.5 to < 0.8

* Nominal atmosphere at 525 km; aero only, assumes other requirements resolved non-propulsively; construction version used for 4 to 6 and 8 to 12 man stations

Table 6-27. Annual CO_2, H_2, and Associated Resistojet Specific Impulse and Power Requirements

Advantages associated with using CO_2 as a propellant are to minimize or eliminate the expence of returning the CO_2 to the Earth via the Orbiter and the associated storage and handling concerns. The primary disadvantage is the contamination from CO_2 in the vicinity of the station.

The ECLSS electrolysis unit produces oxygen for crew respiration and, as a by-product, hydrogen gas. As shown in Figure 6-3, the ECLSS provides 16.6 lbm of O_2 per day for an 8 man crew or 2.1 lbm per day per crew member. The corresponding H_2 production is 0.26 lbm per day per crew member or, annually, 94.7 lbm per crew member. Table 6-27 lists the annual H_2 production for the various station sizes (and the crew ranges for each), the I_{sp}, and specific power requirements to provide the annual total impulse. Note that, for the maximum manning levels, very little power will be required to produce thrust equivalent to the station drag of 0.005, 0.008, and 0.014 lb_f for the three station sizes.

Advantages to using an H_2 resistojet for orbit maintenance are: (1) the elimination of the problems and expense of storing excess H_2; (2) the elimination of the need to transport the excess H_2 back to the Earth via the Orbiter; and (3) the H_2 has the lowest contamination potential of all propellants considered. The disadvantages foreseen are that: (1) required resistojet lifetimes are not currently available (see section 6.4.2.2 on resistojet life), and (2) the H_2 resistojet exhaust exceeds H_2 column density limitations as stated in the Space Station RFP.

6.6.2.2 O_2/H_2 Thrusters with CO_2 and/or H_2 Resistojets

An O_2/H_2 thrust system is also considered in the moderate DDT&E range because the thrusters are not qualified. The advantages of using an O_2/H_2 propulsion system for the IOC station are that it (1) could be designed to facilitate growth; (2) would be synergistic with other systems; (3) has less contaminating exhaust products than other systems (at a low mixture ratio most of the exhaust plume is H_2O and H_2); and (4) is competitive with hydrazine in terms of life-cycle costs. The disadvantages are (1) higher initial DDT&E costs; (2) the system is unproven for thrust levels below 500 lb_f; (3) the production of O_2/H_2 by electrolysis requires energy; (4) the storage, acquisition, gaging and transference become problematic if O_2/H_2 is stored cryogenically due to zero-gravity effects; and (5) it is difficult to make the system modular because of the large and heavy tanks required to compensate for the rapid boil-off of hydrogen.

It may be beneficial to use water electrolysis for propellant production on the IOC station, later adopt a supercritical storage system, and then evolve into cryo storage when the OTV is integrated into the system in 1995. Additional trade studies are required to develop the optimal transition scenario.

The low thrust system could use the same CO_2 and/or H_2 resistojet system that was discussed previously.

6.6.3 High DDT&E

When the OTV is introduced in 1995, O_2/H_2 may be obtained as boiloff from the OTV storage supply. It would require an extremely high DDT&E expenditure to store LO_2/LH_2 onboard the station simply to use the boiloff prior to OTV introduction. This too, requires further trade studies and cannot be completely ruled out as yet. NASA has a number of studies underway that are addressing this issue.

At some point in time, the ECLSS is expected to evolve into a more closed system which would enable much of the oxygen from the CO_2 to be recovered. When this happens, the amount of ECLSS effluent will decrease dramatically. If a Sabatier reduction system is used (see figure B-9, Appendix B), the effluent will be reduced by approximately 40%. The resulting CO_2/CH_4 mixture may then be used in the existing CO_2 resistojets to satisfy low-thrust requirements. However, this possibility requires further investigation.

An arcjet system is not included as a viable alternative for any of these DDT&E levels because the technology is far from mature, the development cost of such a system would be high, and its power requirements are large.

6.6.4 Summary

In summary, there are many different ways to meet the high/low thrust requirements of the Space Station and this report suggests several possibilities. To date, no individual system can be designed to meet a thrust range of 0.01 lb_f to 100 lb_f, which indicates that two systems may be required.

6.7 Free-Flyer Propulsion Systems

The free-flyers that will co-orbit with and require support from the Space Station differ widely in their purposes. Many co-orbiting free-flyers contain experiments that must avoid contamination, micro-g's, pointing excursions, electromagnetic interference, or other potentially disruptive situations. Therefore, the propulsion system chosen for them will have to be responsive to these requirements. Some of the free-flyers may serve as propellant storage facilities or power generation stations and will have less stringent requirements than the Space Station.

In analyzing the factors that influence free-flyer propulsion requirements and comparing them to those for the station, a generality may be drawn: the free-flyer servicing interval is not dictated by the STS 90 day turnaround period. Free-flyers in proximity to the station can have more frequent servicing. Indeed, the propellant farm will be serviced each STS servicing mission. The STPGM will be no further than a kilometer from the station (see figure 2-39) and hence can be readily serviced. The servicing frequencies for the NASA-defined free-flyers (see section 2.1.3) vary from none to 6 times per year, as shown in table 2-1.

The orbit maintenance requirement is generally less for free-flyers. An exception to this is the STPGM because, when this free-flyer is in use, the station has no solar arrays (see section 5.7.1.2). None of the free-flyers will have an ECLSS onboard so CO_2, N_2, and CH_4 will not be available synergistically.

The following subsections discuss a variety of potential free-flyer types.

6.7.1 Propellant Farm

A free-flying propellant farm would store LO_2, LH_2, hydrazine and/or N_2O_4/MMH so there would be several readily available propellant options. The propellant farm is free from micro-g, stringent pointing, and most contamination concerns. However, as with most free-flyers, the propellant farm will not have synergistic propellant opportunities. The use of either N_2O_4/MMH or hydrazine for the propellant farm would entail the lowest development risk. Sections 2.3.8.1 and 2.3.8.2 discuss the propellant farm concept in detail.

6.7.2 Slack-Tethered Power Generation Module (STPGM)

A STPGM has a unique orbit maintenance requirement: it must keep the tether slack and yet maintain a safe distance from the station. This will require at least one thrusting interval per orbit, depending on the altitude and tether requirements. Because of its proximity to the station, the STPGM could be readily serviced by the OMV or perhaps an MM, and could use the ECLSS effluent since the station itself would have low orbit-maintenance requirement since it has no solar arrays. Because of the proximity of power generation equipment, the use of water electrolysis on the STPGM module to produce hydrogen and oxygen for H_2/O_2, a cold H_2 and/or O_2 thruster, or an H_2 resistojet propulsion system. An H_2 resistojet system would require disposing of the O_2. Although water electolysis is a more complex system than a hydrazine system, it may also be used if a reliable small H_2/O_2 or H_2 resistojet can be developed. Perhaps the O_2 could be returned to the station for respiration when water is brought to the STPGM. Water electrolysis is discussed in more detail in Appendix B. Alternately, the water (perhaps even unpotable waste water from the station) could be converted to steam for the thrusters. A low development risk approach for the STPGM would be to use hydrazine. However, the impact of the hydrazine exhaust, in view of the proximity of the STPGM to the station, may make another choice attractive depending on contamination requirements. The aforementioned high duty-cycle requirements would also mean frequent hydrazine thruster change-out which, because of the proximity, may be possible to accomplish during an EVA using the MMU. The STPGM is discussed in detail in section 2.3.8.4.

6.7.3 Science and Applications Space Platform (SASP)

The SASP is used as an example of an experiment platform and will probably have stringent micro-g, contamination, and pointing requirements. SASP also would have a large area-to-mass ratio and thus would have relatively high orbit-maintenance requirements. Because the SASP also has a large power supply, water electrolysis could be used with either small H_2/O_2 thrusters or H_2 resistojets. Section 2.3.8.3 discusses SASP in detail.

6.7.4 NASA-Defined Free-Flyers

Based on the variety of propulsion options available, the varying free-flyer propulsion requirements can be met for any thrust level or reasonable contamination limit. However, the most reliable propulsion system should be used for some of the NASA-defined free-flyers because of the infrequent or non-existant servicing intervals (see table 2-1). Some of these free-flyers will be at geosynchronous altitudes, which will require almost no propulsion for orbit maintenance. Others have very low duty-cycle requirements and will require little operating time annually. Therefore, the propulsion requirements for most of these NASA-defined free-flyers could be satisfied with hydrazine, augmented hydrazine, or N_2O_4/MMH thrusters.

7.0 CONCLUSIONS AND RECOMMENDATIONS FOR FURTHER WORK

The objectives for this NASA-LeRC-sponsored study were to parametrically define Space Station and free-flyer propulsion requirements. In order to accomplish this study, the effort was divided into four tasks. The first three tasks examined the factors that drive propulsion requirements while the fourth examined the propulsion systems capable of meeting those requirements. The conclusions arrived at concerning configurations, servicing strategies, and operating options that minimize propulsion system total impulse requirements and those for the propulsion systems that satisfy the requirements are summarized in the following sections. An additional section provides recommendations for additional work that should be accomplished to clarify certain issues or examine areas beyond the scope of this study.

7.1 Configuration

A number of Space Station sizes and free-flyer configurations were defined and analyzed in order to examine propoulsion requirements. It was found that these vehicles should be designed to minimize aerodynamic torques (CP-CM offset), gravity-gradient torques, and cross-products of inertia for all loading and docked configurations. When secular aerodynamic torques are unavoidable, the effects can often be countered non-propulsively by introducing an intentional gravity-gradient torque or bias in the opposite direction.

The primary contributors to the torques and momenta are the solar arrrays or other large masses, such as docked vehicles, located far from the center-of-mass. The core modules have a lesser effect on these parameters and thus should be sized and arranged only for operational considerations. A planar core arrangement for the station modules with the plane aligned with the velocity vector for minimum drag area was used in this study. The thermal radiators can have a significant effect, primarily aerodynamic, depending on their orientation. A concept was developed as a part of this study which allowed the radiator to be constantly aligned with the velocity vector and yet follow the beta angle for a minimum Sun view factor without requiring a "slip ring" fluid connection.

Cantilevered and balanced solar array configurations were analyzed with the conclusion that a balanced solar array must be used because of the high gravity-gradient torques and cross-products of inertia resulting from the cantilevered approach. This was probably the single-most significant configurational finding during the study. Additional work was done to determine the optimum balanced solar array aspect ratio from the view-point of minimizing gravity-gradient torques and inertia cross products.

Configurations for propellant farms (tethered and un-tethered) and a slack-tethered power generation module free-flyers were defined and these as well as NASA SASP configuration were analyzed. A limited capability Orbital Maneuvering Vehicle (OMV) and a greater capability Orbit Transfer Vehicle (OTV) were also defined and capabilities assessed. These vehicles are quite similar to the common perception for the OMV and OTV.

7.2 Operational Concerns

These are concerns related to altitude, thrust level, attitude control, and servicing of the Space Station and free-flyers.

7.2.1 Altitude

The Van Allen radiation belt precludes station operation at or above 550 km without shielding to protect personnel and electronic equipment. An analysis of station minimum operating altitude to assure a 90-day orbit decay life-time without orbit maintenance resulted in an altitude of 450 km. Obviously, the lower limit may be reduced for modifications to decrease the area-to-mass ratio. The required analysis was beyond the scope of this study but the effects can be readily determined from the parametric analysis performed on lifetime vs. area-to-mass ratio (e.g., see figure 3-14). The altitude range selected for this study was 450 to 525 km.

Factors which require additional study that may drive the altitude toward the higher limit of this range are the atomic oxygen erosion and airflow concerns that are thought to be related and diminish as the atmospheric density decreases.

7.2.2 Thrust Level

Orbit maintenance, or reboost, can be accomplished in a variety of ways. The variables that may be manipulated to achieve a virtually infinite number of possible combinations are thrust level, thrust duration, and thrusting frequency. Thus, the extremes are continuous thrusting to balance drag to a once per 90-day short duration reboost requiring several hundred pounds of thrust. Continuous or very frequent thrusting requires a thrust level of less than 0.1 lb_f with thruster operation for 2160 hrs for each 90-day interval as compared to 1 to 2 hrs every 90-day for the highest thrust approach. This has an impact on viable thruster candidates and/or required techology development.

Other considerations that place requirements on the thrust level are docking disturbances, collision avoidance, rescue, and end-of-life disposal. Collision avoidance and rescue are so requirement-dependent that virtually any level can be dictated, depending on the assumptions made. Docking disturbance cancellation would seem to require a thrust level of 20 to 40 lb_f. End-of-life disposal includes both atmospheric re-entry or high altitude deployment. The former can best be accomplished via an OMV while the latter can be done with the station's normal system over a long period.

7.2.3 Attitude Control and Desaturation

Attitude control was assumed to be accomplished primarily with momentum exchange devices with propulsive attitude control strictly a back-up function with relaxed attitude requirements. Desaturation may be accomplished using torque rods and/or simultaneously with reboost by thrusting on a moment arm from the center-of-mass such that the force times moment arm times duration yields the required desaturation. This places additional constraints on thruster placement, thrust level, thrust frequency, and thrust duration.

7.2.4 Servicing Strategy

A variety of servicing strategies were evaluated. These included: station orbit decay for Orbiter rendezvous; station fly-down for Orbiter rendezvous; OMV fly-up for payload transfer; fixed station low altitude; and fixed station high altitude. It was concluded that servicing can best be accomplished through Orbiter direct insertion to rendezvous with the station at a fixed altitude between 450 and 525 km.

Free-flyer servicing is usually dictated by the free-flyer location with respect to the station, its size, and whether or not the free-flyer has a propulsion system. Those free-flyers without a propulsion system will be serviced by the OMV or OTV, while those with propulsion may return to the station. The half-range capacity of each will determine whether it is better to service in situ or to return the free-flyer to the station.

7.3 Propellant Mass Considerations

A parametric analysis of propellant mass to satisfy the station total impulse requirements was performed for the practical range of thruster specific impulse values. Using a balanced solar array, the minimum station altitude, Orbiter docked 16% of the time, the largest station considered, and the lowest specific impulse, about 4 tonnes of propellant per year are required for an average of about 2200 lbm every 90 days. In contrast, based on the mission traffic analysis conducted and the requirements for payload as provided by NASA, about 900 tonnes per year of LH_2/LO_2 will be required by the year 2000 for the OTV. Based upon this finding, the station propellant resupply concerns can be expected to pale in comparison to those for the OTV propellant.

Propellant storage requirements can obviously be reduced by using higher I_{sp} advanced-technology thrusters. If waste products such as CO_2 effluent from the life-support system or boiloff of hydrogen and oxygen from OTV propellant storage tanks are used, propellant resupply requirements will also be reduced. Employing these options would incur higher development costs than using an off-the-shelf system. The selection of an off-the-shelf system would not, of course, result in zero development costs for thrusters. The long-life required for thrusters on Space Stations, assuming propulsive desaturation, will necessitate a significant increase in thruster life or the development of a thruster replaceble by a suited astronaut.

In the absence of a detailed cost/benefit analysis, and a more refined operational scenario and configuration design, it is difficult to make logical propulsion-system selections. The payoff for advanced propulsion systems must offset development costs within a reasonable time period. The biggest payoff for reducing propulsion requirements for the station at this stage of development is in the area of configuration design and operation. It is apparent, however, that the biggest payoff for advanced propulsion technology is in the area of servicing-vehicle propulsion systems in view of the propellant masses involved.

7.4 Propulsion System Candidates

Propulsion system options have been identified for the Space Station and the free-flyers. Cost is expected to be an overwhelming driver in making the final propulsion system selection. However, minimizing cost can be considered in two ways; minimum start-up or DDT&E cost or minimum life-cycle cost. In the absence of a cost analysis and direction as to which should be minimized, three variants have been studied which progress from estimated minimum to maximum DDT&E cost but also from maximum to minimum life cycle cost (the cycle cost is based strictly on a subjective judgement). Free-flyer propulsion selection was based primarily on propellant availability or commonality with the station or servicing vehicles but may be driven by user requirements for contamination or acceleration.

A wide variety of mono and bipropellant thrusters were examined for state-of-the-art, projected capability up to and during the life of the station, and non-quantitative but important factors such as safety, maintainability, and risk.

Attractive Space Station propulsion systems are summarized in Table 7-1. Free-flyer propulsion system recommendations are summarized in Table 7-2.

7.5 Propulsion Insights Into Other Configurations

The type of configuration analyzed in this study is only one type of a wide range of possibilities. Several of these other possibilities considered in other studies were shown in section 2.0.

The "delta" configuration (figure 2-4) has the advantage of high rigidity but the vehicle and solar arrays are "locked" together such that the entire station must be Sun- or Earth-oriented; the latter resulting in a large area penalty for the solar array. This approach eliminates the need for a slip ring for power distribution. The propulsion system required for the delta configuration would depend on the orientation. An Earth-oriented delta configuration with the velocity vector parallel to the three sides of the "delta" would present a small area to the flow field and hence have low drag and minimal orbit maintenance requirements. In the Sun-oriented mode, the aerodynamic and gravity-gradient torques are expected to be high.

The "Big-T" configuation (figure 2-5) is an Earth-oriented vehicle and, since the two major planes comprising the station are aligned with the velocity vector, the drag and concomitant orbit maintenance requirements will be small. This configuration is also quite rigid and hence is expected to have few dynamic problems arising from thruster firing. The solar array for the Big-T will have to be about twice that for a Sun-oriented array. Any tilting to achieve a better solar viewing angle must consider gravity-gradient torques in the design.

A gravity-gradient stabilized or "power tower" configuration is shown in figure 2-6. This configuration differs from that for this study primarily in the extension from the mid-point between the arrays toward the Earth's surface at the bottom of which are the manned modules. The orbit maintenance requirements for this configuration will be similar to those for

Table 7-1. Space Station Propulsion Systems

DDT&E cost	Features	Attractive systems	
		High thrust	Low thrust
Low	SOA, little growth	10 to 75 lbf hydrazine	$\leq$1 lbf GN_2
Moderate	Synergism, growth	50 to 100 lbf, N_2O_4/MMH from OMV fuel storage or O_2/H_2 from electrolysis	$\leq$1 lbf CO_2 from ECLSS and/or H_2 from electrolysis
High	Synergism, lowest contamination, growth, several sources	Up to 500 lbf, O_2/H_2 boiloff from OTV fuel storage	$\leq$1 lbf H_2 from boiloff or electrolysis

Table 7-2. Free-Flyer Propulsion Systems

Free-flyer	Features	Attractive systems	
		Low risk	Higher risk
Propellant farm	Few contamination and no micro-g concerns	Hydrazine or N_2O_4/MMH based on commonality	O_2/H_2 from boiloff
STPGM	No micro-g concerns	Hydrazine	O_2/H_2 from electrolysis
SASP	High environmental concerns	Hydrazine	H_2 resistojet
NASA-defined	Wide variety	Most will use hydrazine	Augmented hydrazine

the configuration of the current study. The gravity-gradient forces, while tending to stabilize the vehicle, result in a pointing penalty due to Earth oblateness, Earth-Sun-Moon triaxiality, and gravitational anomalies. The power tower configuration is the NASA baseline for the current Space Station RFP.

An Earth-oriented Sun-synchronous station orbiting over the terminator (figure 2-7) offers unique configuration possibilities because of the ability to have a solar array always aligned with the velocity vector yet also always perpendicular to the Sun's rays with no shadowed periods. The orbit maintenance requirements of such a station are very low and the array size can be minimal since there will be no energy storage requirements. This orbit would be ideal for a solar observatory. This station, as depicted in figure 2-7, is gravity-gradient stabilized and will have minimal aerodynamic torques.

7.6 Recommendations for Future Work

It was not the purpose of this study to recommend any one particular propulsion system, but to analyze various systems and assess their ability to satisfy the requirements set down in tasks 1 through 3. An attempt was made to estimate projected capabilities of the various propulsion systems over the next 20 years. This projection proved to be a difficult and somewhat nebulous task, since technology is advancing at such a rapid pace and because these advances are so closely coupled with yet-to-be-made decisions on where resources should be focused. During the analysis of different propulsion systems that are, or could be, applicable to the Space Station many questions were raised which were beyond the scope of this study to analyze. Therefore, analyses and/or testing of the following type are recommended to make a sound and justifiable conclusion as to which propulsion system could "best" satisfy all of the requirements of the Space Station propulsion system.

7.6.1 Life-Cycle-Cost Impact for Alternate Propulsion Systems

A life-cycle-cost analysis should be conducted for the implementation of the propulsion system concepts that are not now ready for station or free-flyer use but could be with some DDT&E expenditure. It is probable that one or more of the synergistic approaches can result in a considerable savings over the life of the station.

7.6.2 Structural Dynamic Impact on Thrusting Strategy

This study assumed the station structure to be rigid when, in fact, the configurations proposed in this study, and most others, are extremely flexible. The dynamics of the station structure resulting from excitation by the thrusters is expected to have a significant impact on the thrusting strategy.

7.6.3 ECLSS - Propulsion System Synergism

A preliminary analysis of the possibilities has been conducted as a part of the current study. A comprehensive analysis is recommended to determine if the apparent benefits really exist.

7.6.4 Solar Concentrators/Solar-Brayton Power Generation Effects

The use of such a system would have a significant effect on the station mass properties and drag. These effects will impact the propulsion system and ,since there is a possibility of the use of such systems, an analysis should be conducted.

7.6.5 Hydrazine Thruster Life Extension

Hydrazine thrusters are considered state-of-the-art technology. However, the primary drawback to their use on the Space Station relates to the limit life of the catalyst bed. Studies and testing should be conducted to determine the benefits of such ideas as spring-loading the catalyst bed to keep the bed packed and preclude voids.

7.6.6 Subcritical vs Supercritical vs Water Electrolysis vs ET Scavenging as an H_2/O_2 Source

A comprehensive trade study should be conducted to evaluate the potential sources of hydrazine and oxygen. The study should consider DDT&E and life cycle costs, risk, and synergism.

7.6.7 Simultaneous Station Reboost and Orbiter Deboost

The concept of lowering the Orbiter toward the Earth on a tether to simultaneously cause it to be deboosted and the station to be raised to a higher orbit deserves a detailed study. Problems concerning tether disposition are foreseen but may be shown to be solveable.

APPENDIX A

MASS PROPERTIES, TORQUE AND ANGULAR MOMENTUM REQUIREMENTS

A. Introduction

Angular momentum requirements for environmental torque compensation will depend upon the dynamics of area and mass properties. For the purposes of this analysis only aero and gravity torques will be considered. This approach is justified for space station since drag and gravity torques at 370-500 km altitude are expected to exceed solar pressure torque by at least three orders of magnitude. In the articles below, the angular momentum requirements for aerodynamic and gravity-gradient torque compensation will be computed. The space station is assumed to be oriented such that the Z body axis is along the local vertical. In order to ensure normal incidence with the Sun line, the solar panels must rotate relative to body axes with orbital frequency ω_0. The analysis will consider two solar panel mounting configurations and three power levels for orbit altitudes in the range of 370 to 500 km. The coordinate system and panel mounting geometry is shown in Figure A-1. Solar panel mass properties and dimensions for typical Solar Electric Power Satellite (SEPS) type arrays are given in Table A-1.

Table A-1. Solar Panel Areas and Mass Properties (Two Panels)

Power (kw)	Area (m)	Mass (kg)
110	1046	2613
210	2000	5004
420	4000	9994

Power Density = 105 W/m2
Mass Density = 2.5 kg/m2

The variation of dynamic pressure with orbit altitude is given in Table A-2 for both nominal and "NASA Neutral" atmospheres. The nominal atmosphere is prevalent and the NASA Neutral accounts for long-term increases in solar activity. Dynamic pressure in N-m is defined as Q=1/2 ($\rho\mu/r$) where ρ is the density in kg/m^3, μ is the Earth's gravitational constant (3.99×10^{14} m^3 /sec^2) and r is the orbit radius in meters. Orbits are assumed to be circular with 28 degrees equatorial inclination, which gives a worst case β angle of 52 degrees. The nominal value of orbit angular rate was computed for h = 500 km to be ω_0 = .001 r/sec. The variation of ω_0 over the altitude range is about 3%.

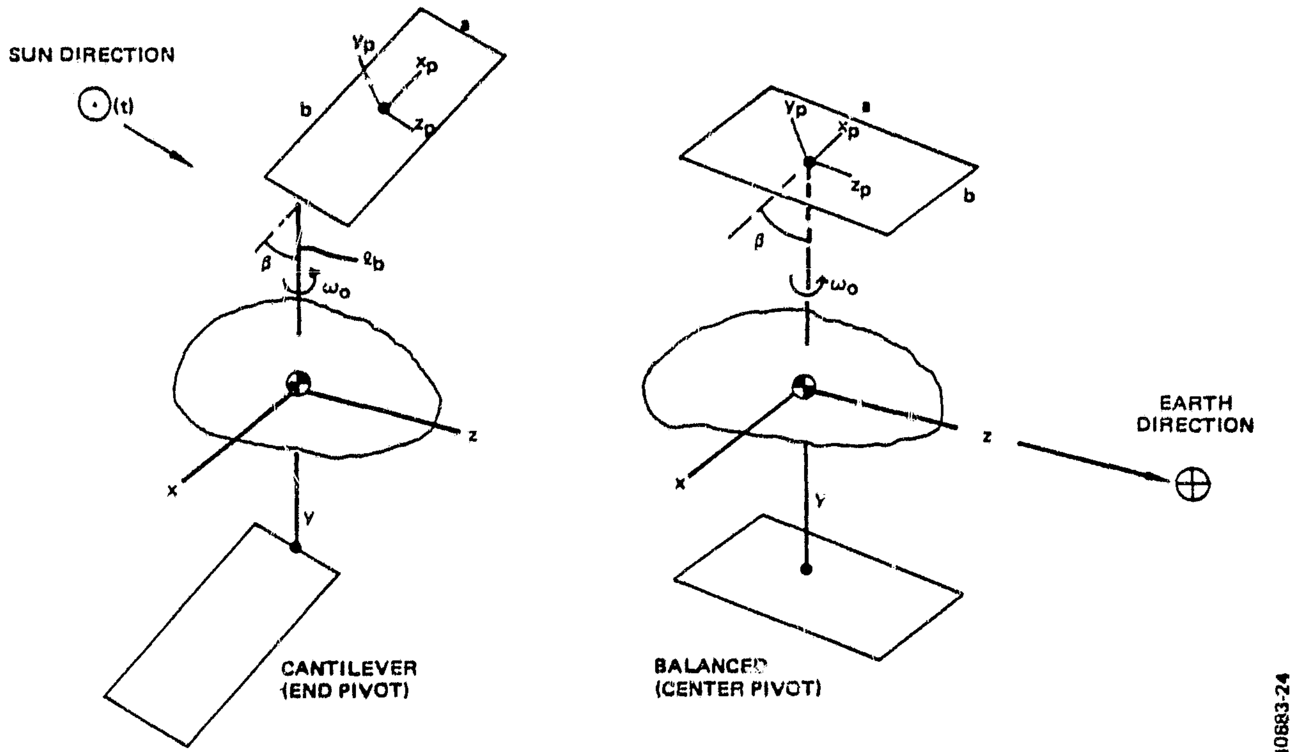


Figure A-1. Solar Panel Mounting Configurations

Table A-2. Variation of Dynamic Pressure with Orbit Altitude

h		Q(N/m2)	
km	nmi	Nominal	NASA Neutral
370	200	1.48 x 10-4	3.55 x 10-4
435	235	4.10 x 10-5	1.46 x 10-4
500	270	1.45 x 10-5	5.80 x 10-5

B. Angular Momentum State Equations

To begin this analysis it is necessary to derive the Euler torque equation in the body axis frame as illustrated in figure A-1.

The total system angular momentum is the sum of the angular momenta of the vehicle plus the control system, which is assumed to be a momentum transfer device. This is expressed in equation 1.

$$H = I\omega + H_c \tag{1}$$

Where:

- H = Total angular momentum (body frame)
- H_c = Control system angular momentum (body frame)
- I = Inertia tensor (body frame)
- ω = Angular velocity vector (body frame)

The system torque is defined as:

$$T = H \tag{2}$$

Substituting equation 1 into equation 2 yields the Euler torque equation in the body frame coordinate system.

$$T = I\omega + I\omega + \omega X I\omega + H_c + \omega X H_c \tag{3}$$

For small perturbations of the body axis system relative to the local vertical, the total angular velocity ω can be expressed as:

$$\omega = \dot{\eta} + q_o \times \eta + q_o \tag{4}$$

In this formulation the nominal local vertical body axis system is assumed to rotate with a constant angular velocity q_0 along y. Roll, pitch and yaw perturbations of the body axis system relative to a local vertical orientation are specified by the components of η Vectors η, q_0, and H_c, and are defined,

$$q_0 = \left[0, -\omega_0, 0\right]^T$$

$$H_c = \left[H_\phi, H_\Theta, H_\psi\right]^T \tag{5}$$

$$\eta = \left[\phi, \theta, \psi\right]^T$$

The quantities η and η are the state variables of the vehicle which are being controlled. However, for the purpose of estimating angular momentum for constant and slowly varying external torques the contribution of state perturbations η and η to body rate ω are negligible.

The assumptions of Equation 5 combined with Equation 4 yield:

$$\omega = q_o = \left[o, -\omega_o, o\right]^T \tag{6}$$

Substituting equation 6 into equation 3 and noting that ω is constant yields:

$$T = I\dot{q}_o + q_o \times Iq_o + \dot{H}_c + q_o \times H_c \tag{7}$$

A skew symmetric matrix, Q_o, can be formed which represents the cross product operation, i.e. q_o X ( ).

Rewriting equation 7 yields

$$T = I\dot{q}_o + Q_o I q_o + \dot{H}_c + Q_o H_c \tag{8}$$

where:

$$Q_o = \begin{bmatrix} o & o & -\omega_o \\ o & o & o \\ \omega_o & o & o \end{bmatrix}$$

The derivative of the inertia Tensor, I, in equation 8 can be expressed as a function of itself. If R is defined to be the coordinate rotation matrix from inertial to body frame then:

$$I = R\, I_i\, R^T \tag{9}$$

where:
I_i = Inertia Tensor (inertial frame).

Taking the derivative of equation 9 and noting that I_i is contant yields:

$$\dot{I} = \dot{R}\, I_i R^T + R\, I_i\, \dot{R}^T \tag{10}$$

For the rotational coordinate transformation matrix R,

$$\dot{R} = \Omega R \tag{11}$$

where:

Ω = skew symmetric matrix representing the cross product of the angular velocity.

Substituting equation 11 into equation 10 yields

$$\dot{I} = \Omega R\, I_i R^T + R\, I_i R^T \Omega^T = \Omega I + I \Omega^T \tag{12}$$

Substituting equation 12 into equation 8 and noting that for this analysis $\Omega = -Q_o$ and that $Q_o\, q_o = q_o \times q_o = 0$ yields:

$$T = \dot{H}_c + Q_o H_c \tag{13}$$

Setting the control torque, T_O, equal to the sum of the external torques yields:

$$T_o = T_a + T_g \tag{14}$$

where:

T_a = Aerodrag torque (body frame)
T_g = Gravity gradient torque (body frame)
(In this analysis only torques due to aerodrag and gravity are considered. See paragraph A for rationale).

Equation 14 can now be substituted into equation 13. Rearranging terms yields the control system angular momentum state equation.

$$H_c = -Q_o H_c + T \tag{15}$$

Equation 15 can be expanded to show that the dynamics of H_c separate to give two uncoupled sets of equations representing the out-of-plane (pitch) and inplane (roll/yaw) components.

$$H_{c\theta} = T_\theta$$

$$\begin{bmatrix} H_{c\phi} \\ H_{c\psi} \end{bmatrix} = \begin{bmatrix} o & \omega_o \\ -\omega_o & o \end{bmatrix} \begin{bmatrix} H_{c\phi} \\ H_{c\psi} \end{bmatrix} + \begin{bmatrix} T_\phi \\ T_\psi \end{bmatrix} \tag{16}$$

C. Panel Products of Inertia in Rotating Coordinates

The geometry required to calculate the solar panel products of inertia is depicted in figure 2. The calculations will involve computing the inertia tensor for the panels about the panel center of mass. This coordinate system (figure A-2) is represented by the subscript 'p'. The inertia tensor in the panel coordinate frame, I_p, is calculated. The tensor is rotated $(90-\beta)$ degrees about the Z_p axis to accommodate Sun tracking. A second rotation about the X-body axis accounts for the orbital rotation.

Rotating the panel centered inertia tensor yields the inertia tensor in the translated body frame.

$$I' = R_\theta \; R_\beta \; I_p \; R_\beta^T \; R_\theta^T \tag{17}$$

Where:

I' = *Inertia Tensor (translated body)*

I_p = *Inertia Tensor (panel centered)*

= $diag(I_{xp}, I_{yp}, I_{zp})$

R_β = *Rotation Matrix about Z_p-axis $(90 - \beta)$ degrees*

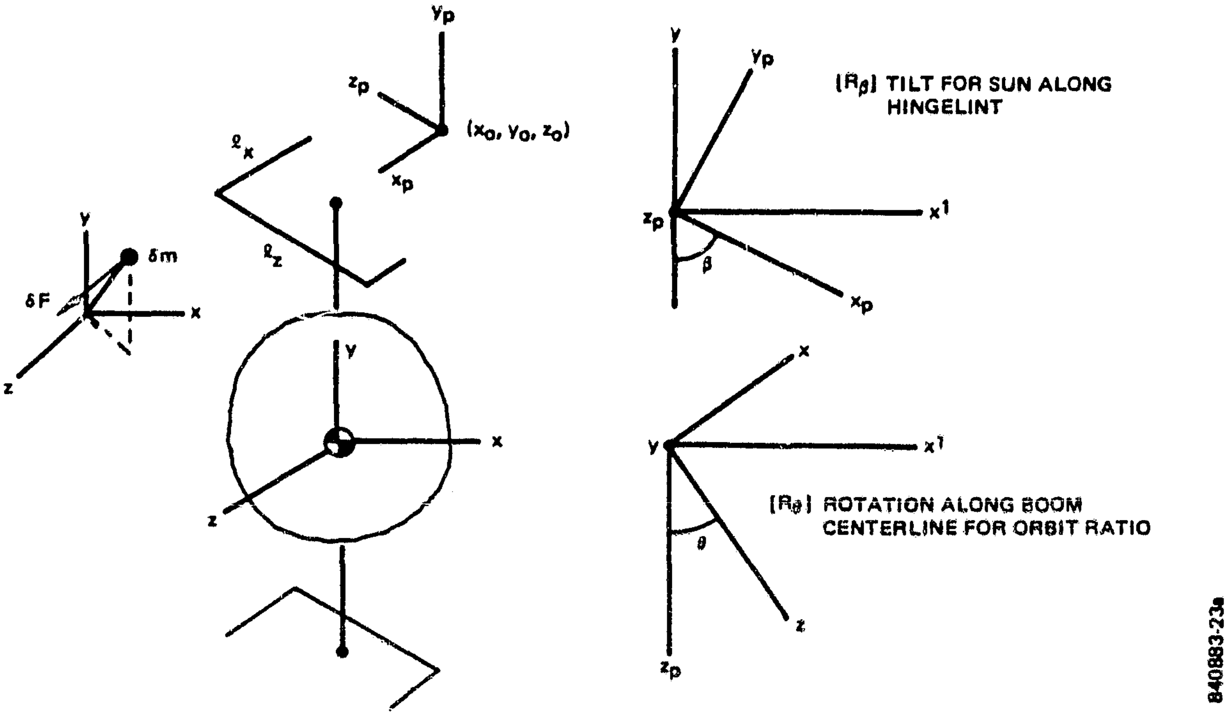


Figure A-2. Solar Panel Geometry for Computing Products of Inertia

$$R_\beta = \begin{bmatrix} \sin\beta & \cos\beta & o \\ -\cos\beta & \sin\beta & o \\ o & o & 1 \end{bmatrix}$$

R_θ = *Rotation Matrix about* Y_p *axis*

$$R_\theta = \begin{bmatrix} \cos\theta & 0 & -\sin\theta \\ o & 1 & o \\ \sin\theta & o & \cos\theta \end{bmatrix}$$

Extracting the products of inertia from I' gives

$$I'_{xy} = \left(\frac{I_{yp} - I_{xp}}{2}\right) \sin 2\beta \cos\theta$$

$$I'_{yz} = \left(\frac{I_{yp} - I_{xp}}{2}\right) \sin 2\beta \sin\theta \qquad (17a)$$

$$I'_{xz} = \left(\frac{I_{yp}\cos^2\beta + I_{xp}\,Sin^2\beta - I_{zp}}{2}\right) \sin 2\theta$$

The products of inertia are derived from (17a) by noting that for a flat rectangular plate (refer figure A-2).

$$I_{xp} = \frac{m}{12}\,\ell_z^2$$

$$I_{yp} = \frac{m}{12}\left(\ell_x^2 + \ell_z^2\right) \qquad (18)$$

$$I_{zp} = \frac{m}{12}\,\ell_x^2$$

where: m = panel mass.

Substitution of equation 18 into equation 17a yields:

$$I'_{xy} = \frac{1}{2}\,\frac{m}{12}\,\ell_x^2 \sin 2\beta \cos\theta$$

$$\Gamma_{yz} = \frac{1}{2}\,\frac{m}{12}\,\ell_x^2 \sin 2\beta \sin\theta \tag{19}$$

$$\Gamma_{xz} = \frac{1}{2}\,\frac{m}{12}\left(\ell_z^2 - \ell_x^2 \sin 2\beta\right)\sin 2\theta$$

Now that the cross products of inertia have been found for the translated body frame, the parallel axis theorem can be used to compute the cross products in the core body frame. The parallel axis theorem yields:

$$I_{xy} = \Gamma_{xy} + m\,x_o\,y_o$$

$$I_{yz} = \Gamma_{yz} + m\,y_o\,z_o \tag{20}$$

$$I_{xz} = \Gamma_{xz} + m\,x_o\,z_o$$

For balanced arrays the pivot is located at the panel center of mass where $x_0, z_0 = 0$. Since $\ell_X = b$, $\ell_Z = a$ for balanced arrays (Figure A-1) the resulting products for each panel are:

$$I_{xy} = \frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta \cos\theta$$

$$I_{yz} = \frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta \sin\theta \tag{21}$$

$$I_{xz} = \frac{1}{2}\,\frac{m}{12}\left(a^2 - b^2 \sin^2\beta\right)\sin 2\theta$$

For cantilevered arrays the pivot is located at the end of the panel where x_0, z_0 are finite and opposite in sign and $\ell_X = b$, $\ell_Z = a$ (Figure A-1). The quantities (x_0, y_0, z_0) are given by:

$$x_o = \frac{b}{2}\,\mathrm{Sin}\,\beta \cos\theta$$

$$y_o = \ell_b + \frac{b}{2}\cos\beta \tag{22}$$

$$z_o = \frac{b}{2}\sin\beta \sin\theta$$

The resulting primed products of inertia for each panel from equation 19 are:

$$I'_{xy} = \frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta \cos\theta$$

$$I'_{yz} = \frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta \sin\theta \qquad (23)$$

$$I'_{xz} = \frac{1}{2}\,\frac{m}{12}\left(a^2 - b^2 \sin^2\beta\right) \sin 2\theta$$

Substituting Equations 22 and 23 into Equation 20 and reducing terms yields the cantilevered cross products of inertia in body frame per panel.

$$I_{xy} = m\frac{b}{2}\left[\frac{b}{3}\sin 2\beta + \ell_b \sin\beta\right]\cos\theta$$

$$I_{yz} = m\frac{b}{2}\left[\frac{b}{3}\sin 2\beta + \ell_b \sin\beta\right]\sin\theta \qquad (24)$$

$$I_{xz} = \frac{1}{2}\frac{m}{12}\left[a^2 + 2b^2 \sin^2\beta\right]\sin 2\theta$$

Equations 21 and 24 are summarized in Table A-3.

Table A-3. Panel Inertia Amplitudes (body frame)

	Balanced	*Cantilevered*
I^*_{xy}	$\frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta$	$\frac{1}{2}\,\frac{m}{12}\left(4b^2 \sin 2\beta + 12b\,\ell_b \sin\beta\right)$
I^*_{yz}	$\frac{1}{2}\,\frac{m}{12}\,b^2 \sin 2\beta$	$\frac{1}{2}\,\frac{m}{12}\left(4b^2 \sin 2\beta + 12b\,\ell_b \sin\beta\right)$
I^*_{xz}	$\frac{1}{2}\,\frac{m}{12}\left(a^2 - b^2 \sin^2\beta\right)$	$\frac{1}{2}\,\frac{m}{12}\left(a^2 + 2b^2 \sin^2\beta\right)$

D. Aerodynamic Torque

The calculation of body axis torques due to aerodynamic effects assumes negligible lift in fre molecular flow so that all the forces can be attributed to drag. Since the vehicle X-body asix is colinear with the velocity vector and since drag is the only significant force, the force vector, F, can be written as follows:

$$F = F_a \,|sin\,\theta|\, [0, 1, 1]^T$$

where

Fa	$= 2\ Q(h)\ S_O\ (\beta)\ C_D$
Q(h)	= Dynamic Pressure (see Table A-2)
$S_O(\beta)$	= Projected Solar Panel Area (one panel)
C_D	= Drag Coefficient

The torque can then be written as:

$$T = r x F = F_a \,|\, sin\,\theta \,|\, [o, r_z, r_y]$$

where

$$r \;=\; \text{cp/cg offset} \qquad (25)$$

$$= [r_x, r_y, r_z]^T$$

It is seen that the pressure induced roll axis torques are negligible. The magnitude of the torque is shown in figure A-3 along with the geometry required to perform the torque calculations.

For free molecular flow the drag coefficient C_D is very nearly constant with value $C_D = 2.0$. The area S_O is the projected frontal area of a single solar panel. This area is a function of the sun angle beta. Assuming $\beta = 0$ deg results in maximum projected solar panel area.

Since the value of r is unknown, it seems reasonable to specify the aero torque per unit cp/cg offset. Accordingly the amplitude of the normalized torque computed from equation 25 and using the data in tables A-1 and A-2 are presented in Figure A-11 for β=0 deg per unit cp/cg offset. It is noted that r as well as S_O is a function of time and changes as the panels rotate. However, for estimating purposes it is reasonable to assume that r(t) is dominated by the solar panels with only negligible contribution from the station core when the panels are edge on to the flow. It is also noted that for panels of the same dimensions and tilt, zero torques are the same for both cantilevered and balanced array configurations.

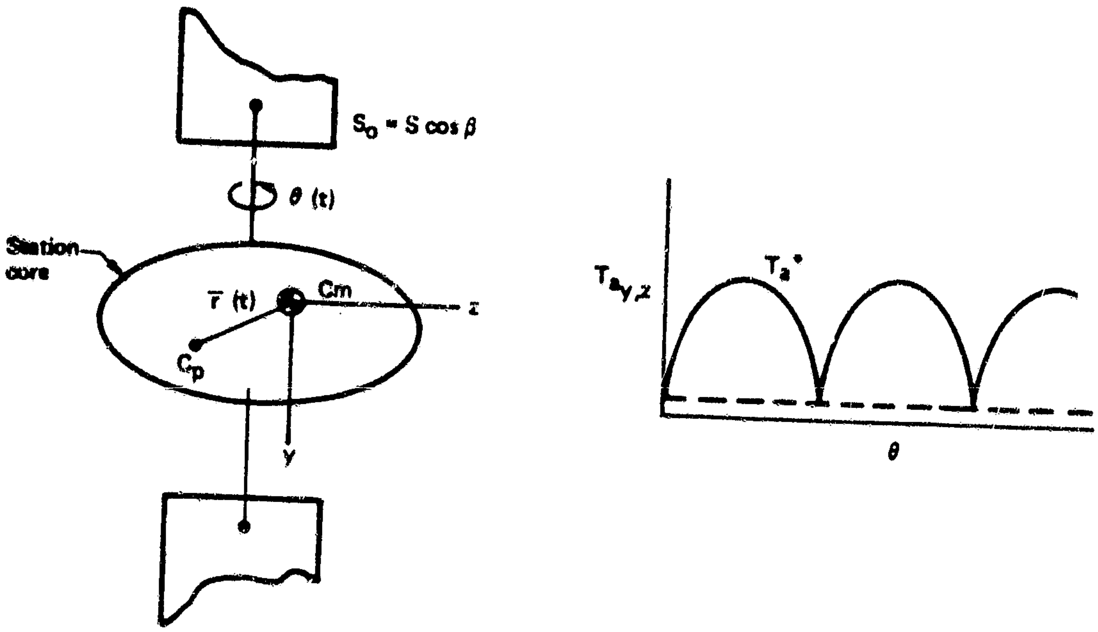


Figure A-3. Geometry for Computing Aero Torques

E. Gravity Gradient Torque

The gravity gradient torque is given by:

$$T_g = G_z + T_{gi} \tag{26}$$

Where the first term is the state dependent contribution, the second term is independent of vehicle state and dependent only on products of inertia.

For this analysis we can assume that the state dependent contribution is negligible compared to the second term, since the rate of change of the state is assumed small (see Equation 5).

Neglecting the first term and using a standard gravity torque equation for the second term yields:

$$T_g = 3\,\frac{\mu}{R^3}\,e X I e \tag{27}$$

where:

e = unit vector pointing along the local vertical in body frame coordinates

$$e = \begin{bmatrix} 0 & 0 & 1 \end{bmatrix}^T \qquad \textit{Note: For a circular orbit } \frac{3\mu}{R^3} = 3\omega_o^2$$

Expanding Equation 27 yields:

$$T_g = 3\omega_o^2 \begin{bmatrix} I_{yz} \\ -I_{xz} \\ 0 \end{bmatrix} \tag{28}$$

Substituting the relations for inertia computed in Equations 21 and 24 into. equation 28 yields:

$$T'_g = 3\omega_o^2 \begin{bmatrix} I^*_{yz}\sin\theta \\ -I^*_{xz}\sin 2\theta \\ 0 \end{bmatrix} ;\ \theta = \omega_o t \tag{29}$$

where: I_{yz}^* and I_{xz}^* are the amplitudes of the products of inertia (see Table A-3).

The resulting gravity gradient torques from equation 29 are plotted as a function of array area and aspect ratio for both configurations in figures A-4 through A-7. (See section G for a discussion of aspect ratio.)

F. Angular Momentum Computation

The angular momentum resulting from the applied aero and gravity torques is computed by integrating equations (16). Solving for the pitch component of angular momentum in Equation 16 involves a straight forward integration.

$$H_{c\theta} = \int_{t_o}^{t_1} T_\theta \, dt \tag{30}$$

Laplace transforms were used to solve for the roll and yaw angular momentum components of equation 16. From equation 16:

$$\dot{H}_c = \Omega H_c + T \tag{31}$$

where:

$$H_c = \left[H_{c\phi}, H_{c\psi} \right]^T$$

$$\Omega = \begin{bmatrix} 0 & \omega_o \\ -\omega_o & 0 \end{bmatrix}$$

$$T = \left[T_\phi, T_\psi \right]^T$$

Transforming equation 31 to the S-plane yields:

$$\begin{aligned} sH_c(s) &= \Omega H_c(s) + T(s) \\ (sI - \Omega) H_c(s) &= T(s) \\ H_c(s) &= (sI - \Omega)^{-1} T(s) \end{aligned} \tag{32}$$

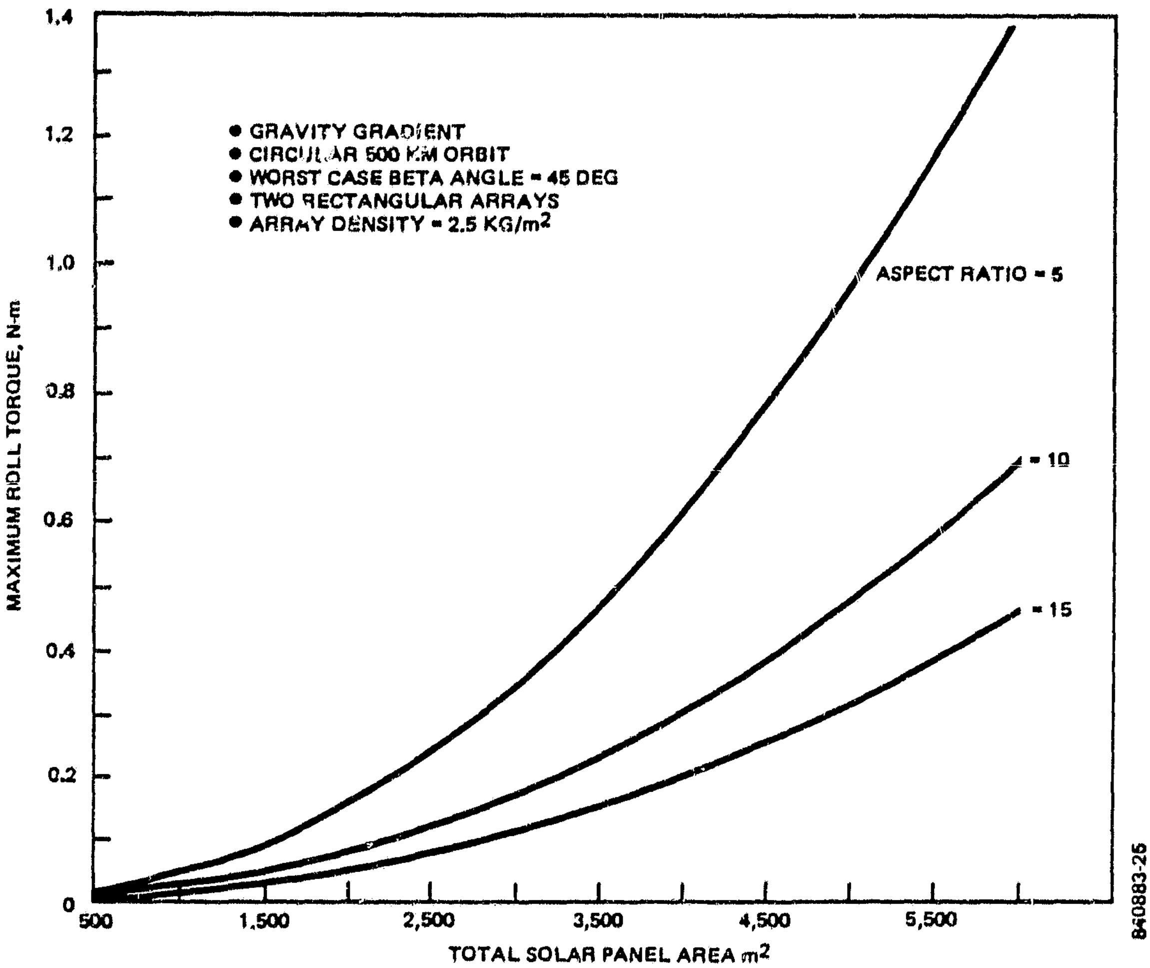


Figure A-4. Maximum Roll Torque for Balanced Array

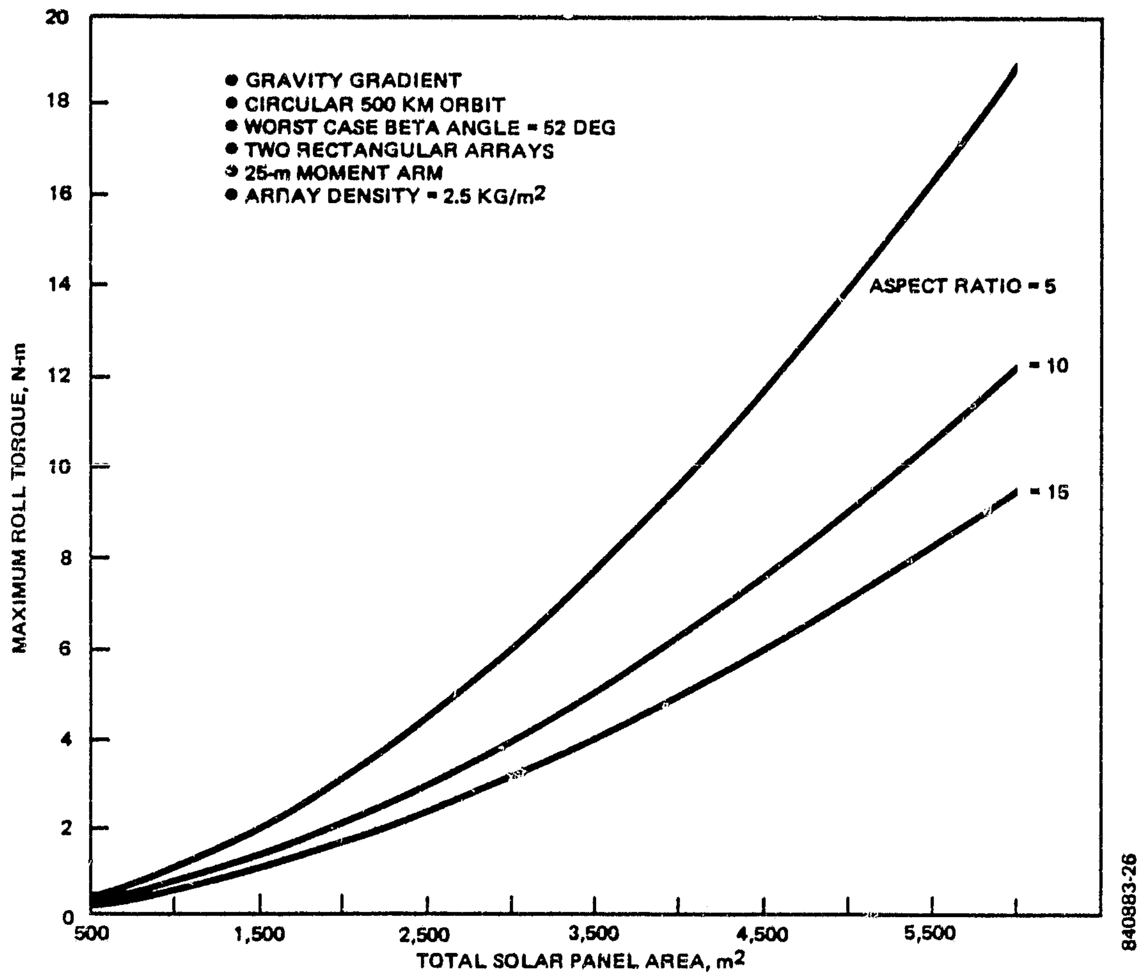


Figure A-5. Maximum Roll Torque Cantilevered Array

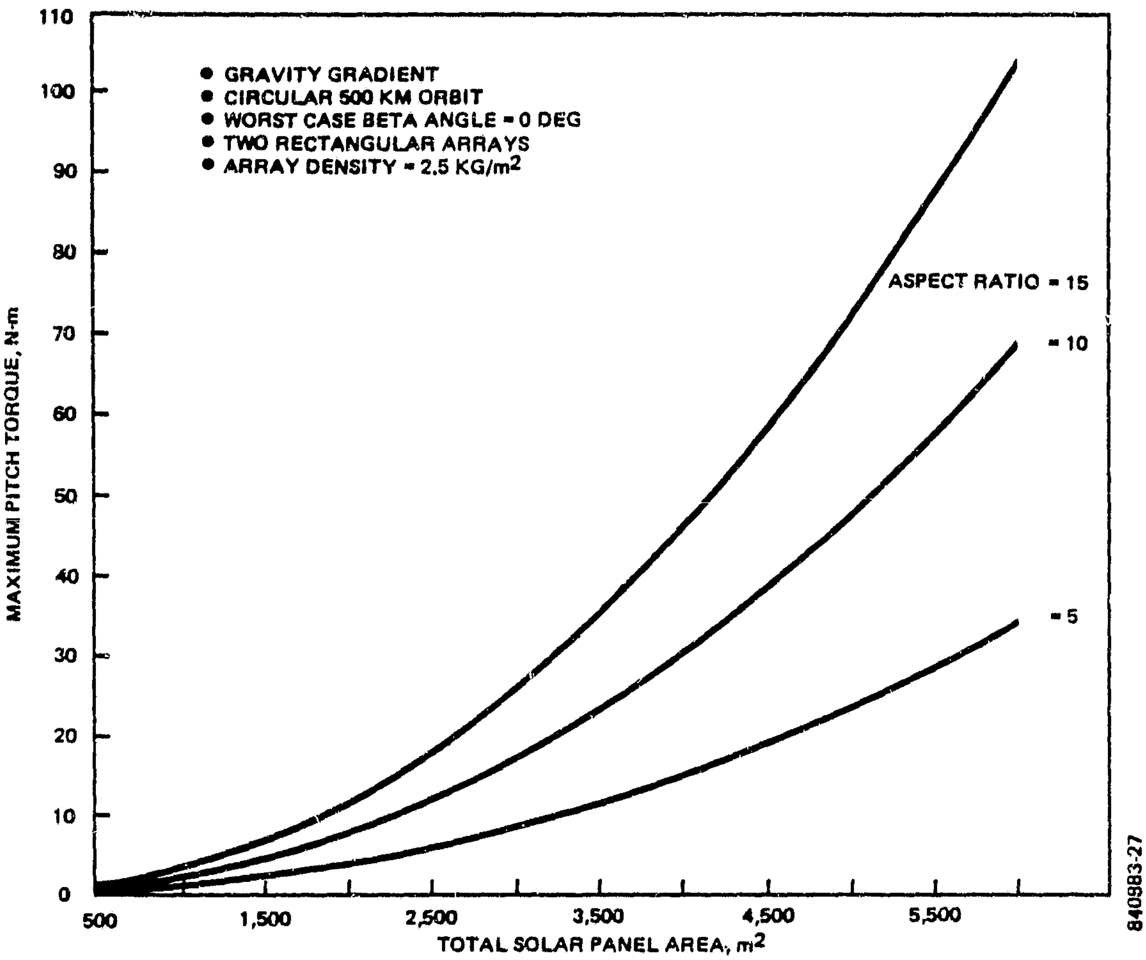


Figure A-6. Maximum Pitch Torque for Balanced Array

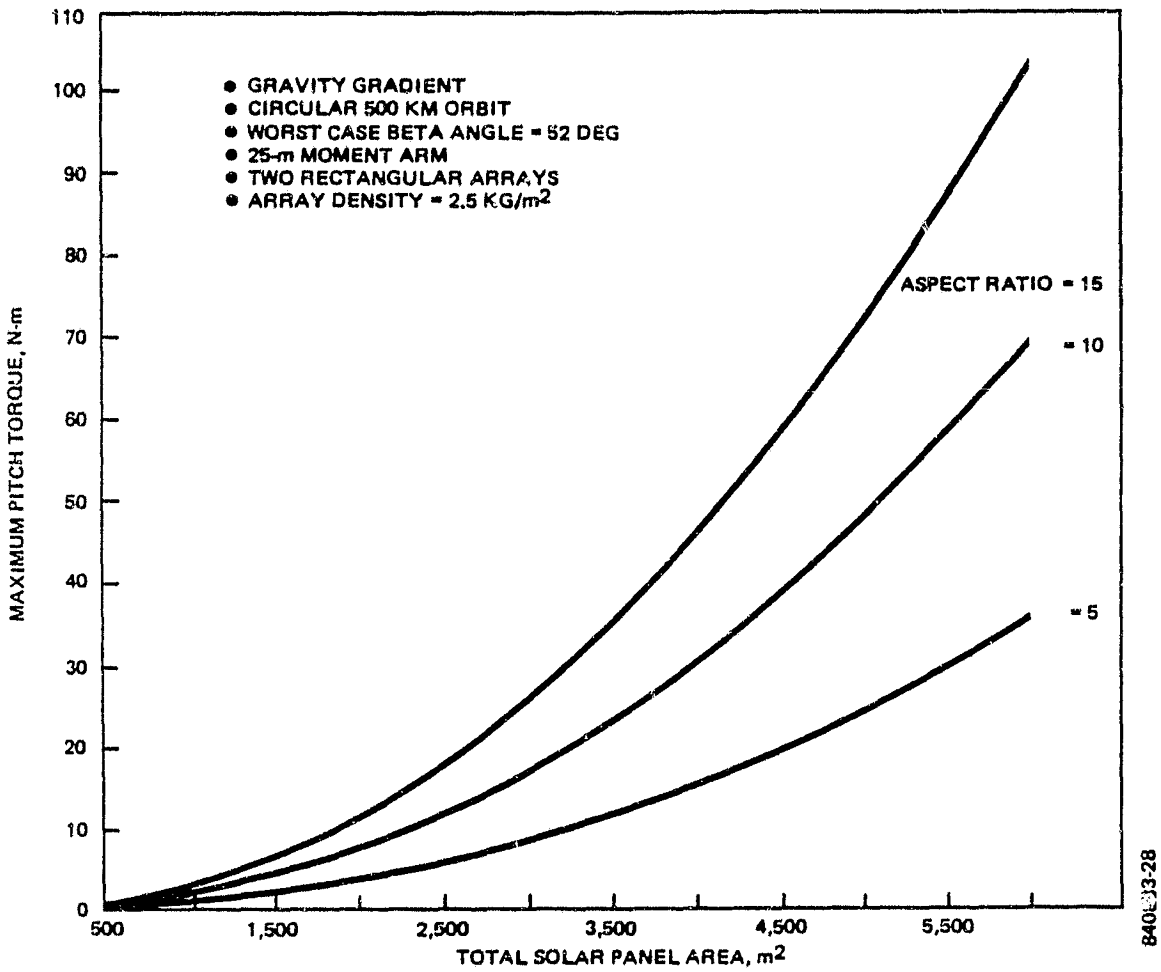


Figure A-7. Maximum Pitch Torque for Cantilevered Array

where:

I = identity matrix = diag (1, 1, 1)

Expanding the matrix $(sI-\Omega)^{-1}$ in equation 32 yields:

$$H_c(s) = \frac{\begin{bmatrix} s & w_o \\ -w_o & s \end{bmatrix} T(s)}{s^2 + w_o^2} \tag{33}$$

1. Aerodynamic Drag

Pitch Axis. Substitution of the aerodynamic torque derived in Equation 25 into the pitch angular momentum expressed in Equation 30 yields:

$$H_{C\theta} = \frac{2Q(h)S_oC_Dr_z}{\omega_o} \int_o^{2\pi} |sin\theta| \, d\theta \tag{34}$$

note: $\theta = \omega_o t$

Integrating Equation 34 and dividing by r_z yields the angular momentum requirement in the pitch axis per orbit per unit lever arm of cg/cp offset.

$$H_{C\theta} = \frac{8\,Q\,(h)\,S_o\,C_D}{\omega_o} \tag{35}$$

The pitch angular momentum expressed in Equations 34 and 35 is secular producing a linear divergence with time. It is plotted in figure A-12 for $\beta = 0$ degrees.

Roll-Yaw Plane. From equation 25 define

$$T = T_a \begin{bmatrix} o \\ |\,sin\theta\,| \end{bmatrix} \tag{36}$$

where

$$T_a = 2\,Q\,(h)\,S_o\,C_D\,r_y$$

The angular momentum equation derived in Equation 33 can now be written as:

$$H(s) = \frac{T_a \begin{bmatrix} s & \omega_o \\ -\omega_o & s \end{bmatrix}}{s^2 + \omega_o^{\,2}} \begin{bmatrix} o \\ \pounds(|sin\,\theta|) \end{bmatrix} \tag{37}$$

where $\pounds(*)$ is the Laplace operator

Expanding Equation 37 and taking inverse Laplace transforms yields:

$$H(s) = T_a \begin{bmatrix} \pounds(\sin \omega_o t)\ \pounds(|\sin \omega_o t|) \\ \pounds(\cos \omega_o t)\ \pounds(|\sin \omega_o t|) \end{bmatrix} \tag{38}$$

The following DuHamel relation will be used to find the inverse Laplace transform of Equation 38

$$\pounds(f(t))\pounds(g(t)) = \pounds\left\{ \int_0^t f(t-\tau)g(\tau)d\tau \right\} = \pounds\left\{ f(t)*g(t) \right\} \quad (convolution\ integral) \tag{39}$$

The inverse Laplace transform of Equation 38 is using Equation 39

$$H(t) = T_a \begin{bmatrix} \int_0^t \sin \omega_o (t-\tau)|\sin \omega_o \tau|\, d\tau \\ \int_0^t \cos \omega_o (t-\tau)|\sin \omega_o \tau|\, d\tau \end{bmatrix} \tag{40}$$

H(t) can be evaluated in two regions of interest

$$0 \le t < \frac{\pi}{\omega_o}$$

$$\frac{\pi}{\omega_o} \le t < \frac{2\pi}{\omega_o}$$

From equation 40 using trigonometric substitutions and evaluating the integral

$$H(t) = \frac{T_a}{2\,\omega_o} \begin{bmatrix} \sin\theta - \theta\cos\theta \\ \theta \sin\theta \end{bmatrix} \quad ;\ 0 \le \theta < \pi \tag{41}$$

$$H(t) = H\left(\frac{\pi}{\omega_o}\right) + \frac{T_a}{2\,\omega_o} \begin{bmatrix} \sin\theta + (\pi - \theta)\cos\theta \\ (\theta - \pi)\sin\theta \end{bmatrix} \quad ;\ \pi \le \theta < 2\pi \tag{42}$$

Substituting $\theta = \pi$ into Equation 41 yields, $H(\pi/\omega_o)$ which allows Equation 42 to be rewritten as:

$$H(\theta) = \frac{T_a}{2\,\omega_o} \begin{bmatrix} \pi + \sin\theta + (\pi - \theta)\cos\theta \\ (\theta - \pi)\sin\theta \end{bmatrix} \quad ;\ \pi \le \theta < 2\,\pi \tag{43}$$

Analysis of Equations 41 and 43 indicate that the aerodynamic angular momentum is cyclic in the roll-yaw plane. Also the maximum momentum generated in both roll and yaw axes per unit arm per orbit is

$$H_{max} = \frac{\pi Q(h) S_o C_D}{\omega_o} \tag{44}$$

This is plotted in Figure A-13 for β=0 deg.

2. Gravity Gradient

Pitch Axis. Substituting the pitch gravity gradient torque derived in Equation 29 into the pitch angular momentum of Equation 30 yields:

$$H_{c\theta} -3\,\omega_o^2 \int_o^t I^*_{xz} \sin 2\omega_o \tau d\tau \tag{45}$$

where:

$H_{c\theta}$ = pitch angular momentum due to gravity gradient torques

Integration of (45) gives:

$$H_{c\theta} = \frac{-3\omega_0}{2} I^*_{xz} (\cos 2\theta - 1) \tag{46}$$

Roll-Yaw Plane. From equation 29 define T and its equivalent transform, T(s).

$$T = 3\omega_o^2 \begin{bmatrix} I^*_{yz} \sin\theta \\ O \end{bmatrix} \tag{47}$$

$$T(s) = 3\omega_o^2 \begin{bmatrix} I^*_{yz} \left(\dfrac{\omega_o}{s^2 + \omega_o^2} \right) \\ O \end{bmatrix} \tag{48}$$

Substituting Equation (48) into Equation (33) yields:

$$H(s) = \frac{3\omega_o^3 I^*_{yz}}{\left(s^2 + \omega_o^2\right)^2} \begin{bmatrix} s & \omega_o \\ -\omega_o & s \end{bmatrix} \begin{bmatrix} 1 \\ 0 \end{bmatrix} \tag{49}$$

Expanding Equation (49) yields:

$$H_{c\phi}(s) = 3\omega_o^3 I^*_{yz}\left(\frac{s}{\left(s^2+\omega_o^2\right)^2}\right) \tag{50}$$

$$H_{c\psi}(s) = -3\omega_o^4 I^*_{yz}\left(\frac{1}{\left(s^2+\omega_o^2\right)^2}\right)$$

Taking the inverse Laplace transforms yields:

$$H_{c\phi} = \frac{3}{2}\,\omega_o I^*_{yz}\,\theta\sin\theta \tag{51}$$

$$H_{c\psi} = \frac{-3}{2}\,\omega_o I^*_{yz}\,(\sin\theta - \theta\cos\theta)$$

Inspecting Equations 46 and 51 it is observed that both cyclic and secular momenta occur. The maximum momenta per orbit are summarized for each access in Table A -4 below.

Table A-4. Maximum Angular Momenta per Orbit

	Cyclic	Secular
Roll	None	$3\pi\omega_0 I^*_{yz}$
Pitch	$3\omega_0 I^*_{xz}$	None
Yaw	$3/2\ \omega_0 I^*_{yz}$	$3\pi\omega_0 I^*_{yz}$

G. Aspect Ratio

A comparison of the torques and momentum between the two configurations can be enhanced by considering the aspect ratio of the solar panels. The aspect ratio describes the rectangular solar panel geometry. It is the ratio of the length of the solar panel in the X_p-axis divided by the length in the Zp-axis (see figure A-1), or aspect ratio = a/b

The torques and moments defined in equation 29 and Table A-4 are all proportional to the cross products of inertia listed in Table A-3. Hence, reducing a cross product will result in the reduction of the corresonding torque or momentum. For constant β and solar array area, the cross products are dependent upon the dimensions of the array or the aspect ratio. For example, from Table A-4 the secular moments are dependent upon the I_{yz} cross product of inertia. Table A-3 indicates that for both balanced and cantilevered configurations I_{yz} is proportional to the dimension "b", thus increasing the aspect ratio (decreasing "b") yields a lower I_{yz} cross product and lower secular momenta. Figures A-4 through A-10 show the relationship between aspect ratio, torque and momenta. The figures are more fully discussed in the next section.

H. Conclusions

Figures A-4 and A-5 show the maximum roll torque per orbit as a function of area for the balanced and cantilevered configurations, respectively. The three curves on each plot represent a different aspect ratio. A comparison of the two figures clearly shows the advantages of the balanced array versus the cantilevered. There is at least an order of magnitude difference in roll torques between the two configurations

Figures A-6 and A-7 display the maximum pitch torques in a similar manner as figures A-4 and A-5. Note that whereas the pitch torques are proportional to the aspect ratio, the roll torques are inversely proportional to aspect ratio. Also note that there is little difference between the maximum pitch torques for the two configurations. The pitch torques are much larger for the range of aspect ratio shown than the roll torques.

Figures A-8, A-9, and A-10 are plots of the secular and cyclic momenta (see Table A-4) as a function of aspect ratio. Each figure represents a different solar array area (see Table A-1).

The momenta are expressed as the number of control moment gyros (CMG) needed to store the momenta per orbit (4000 N-m-s per CMG).

An examination of figures A-4 through A-10 suggests certain strategies in dealing with the gravity gradient effects. For example, selecting the balanced instead of the cantilevered configuration seems the obvious choice. The pitch torque of the balanced array for an aspect ratio of 10 and an array area of 4,000 m^2 gives from figure A-4 a maximum roll torque of .3 n-m. Torques of this magnitude can be constantly countered by magnetic torque rods (MTR) of the type used on skylab. If this approach is taken then the torque used in equation 47 is reduced to zero completely eliminating all but the cyclic pitch momentum in Table A-4. The number of control moment gyros (CMG's) needed to store the pitch cyclic momentum are from figure A-10, about 7 CMG's.

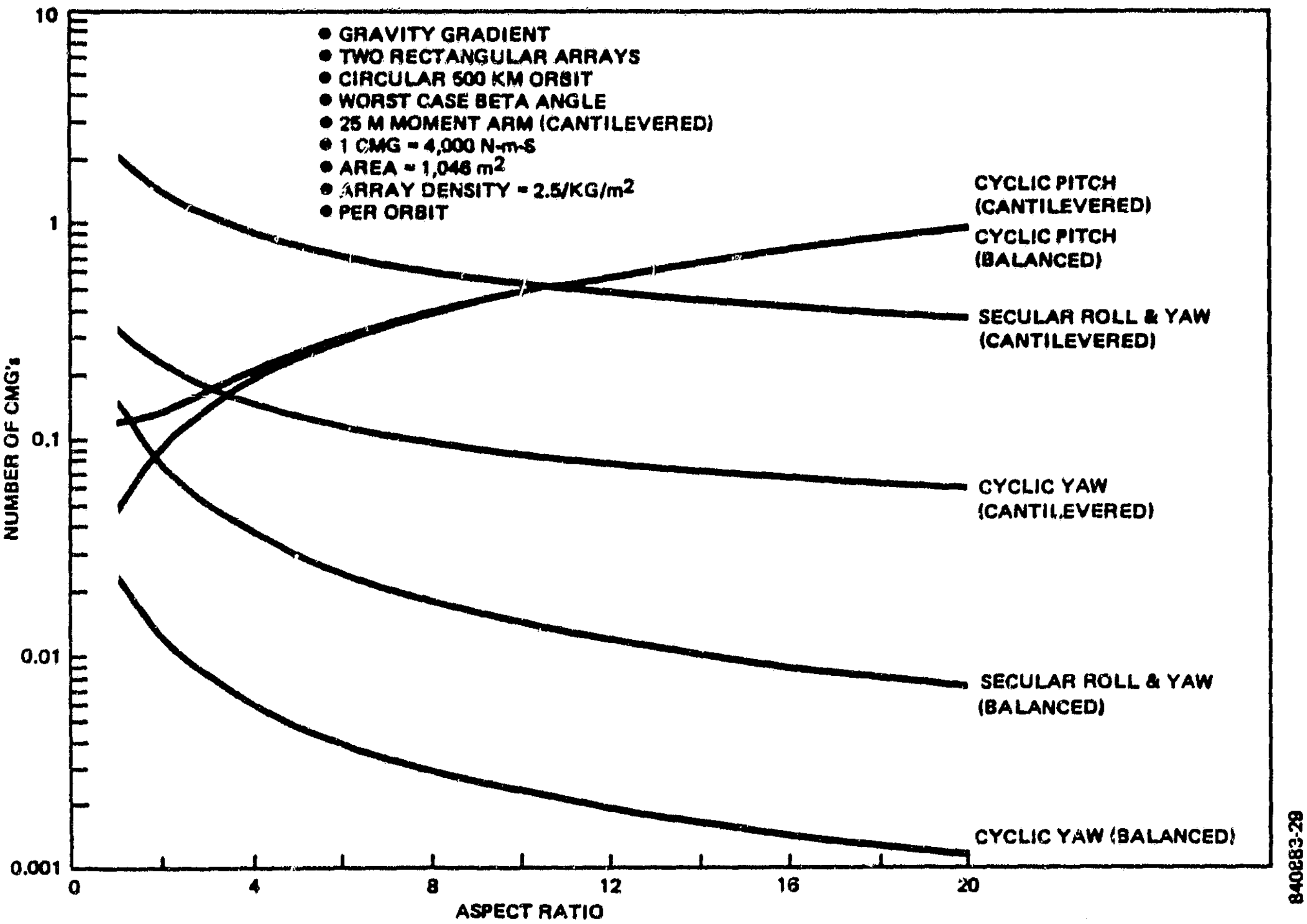


Figure A-8. Momentum per Orbit for 1046-m^2 Array

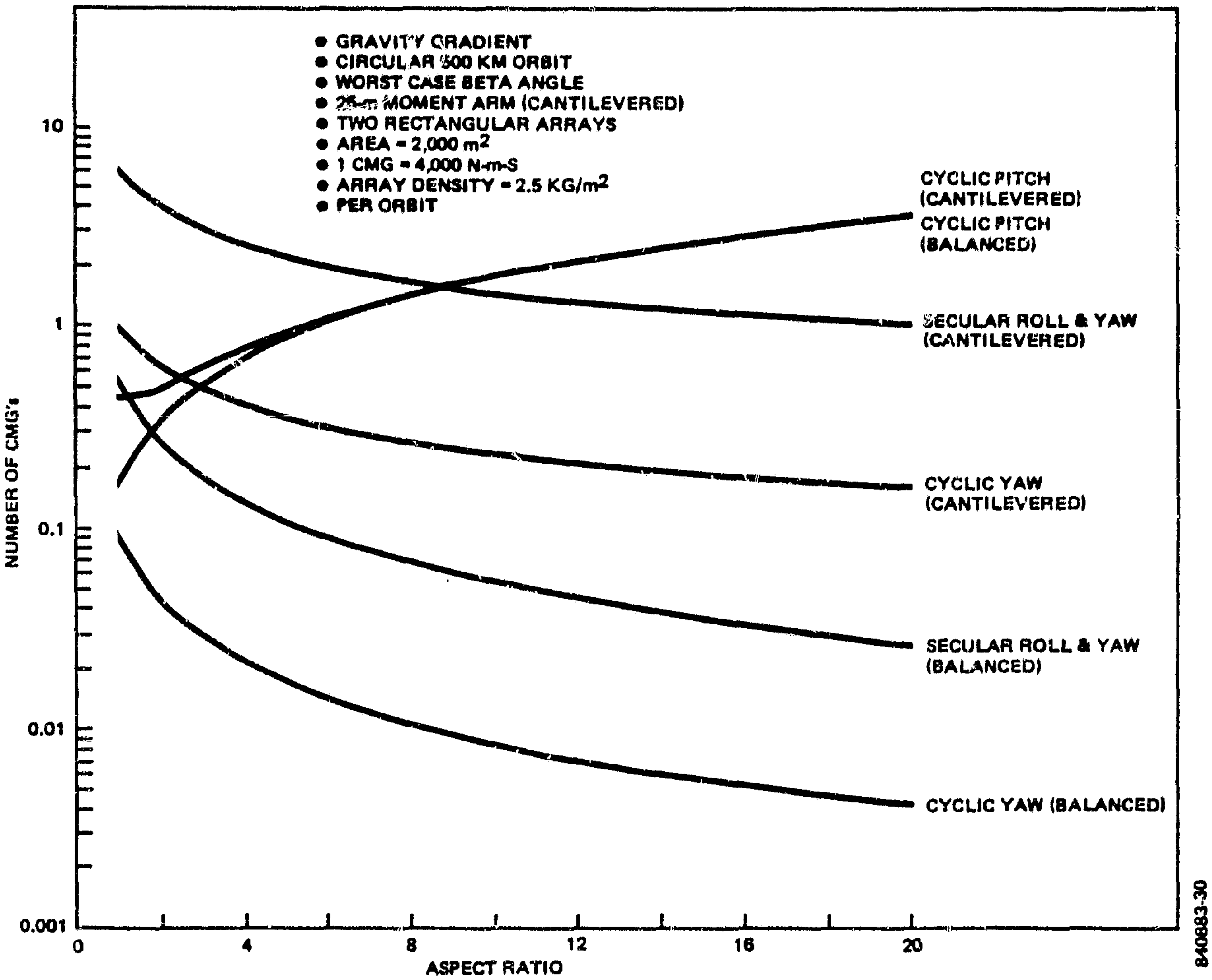


Figure A-9. Momentum per Orbit for 2,000 m² Array

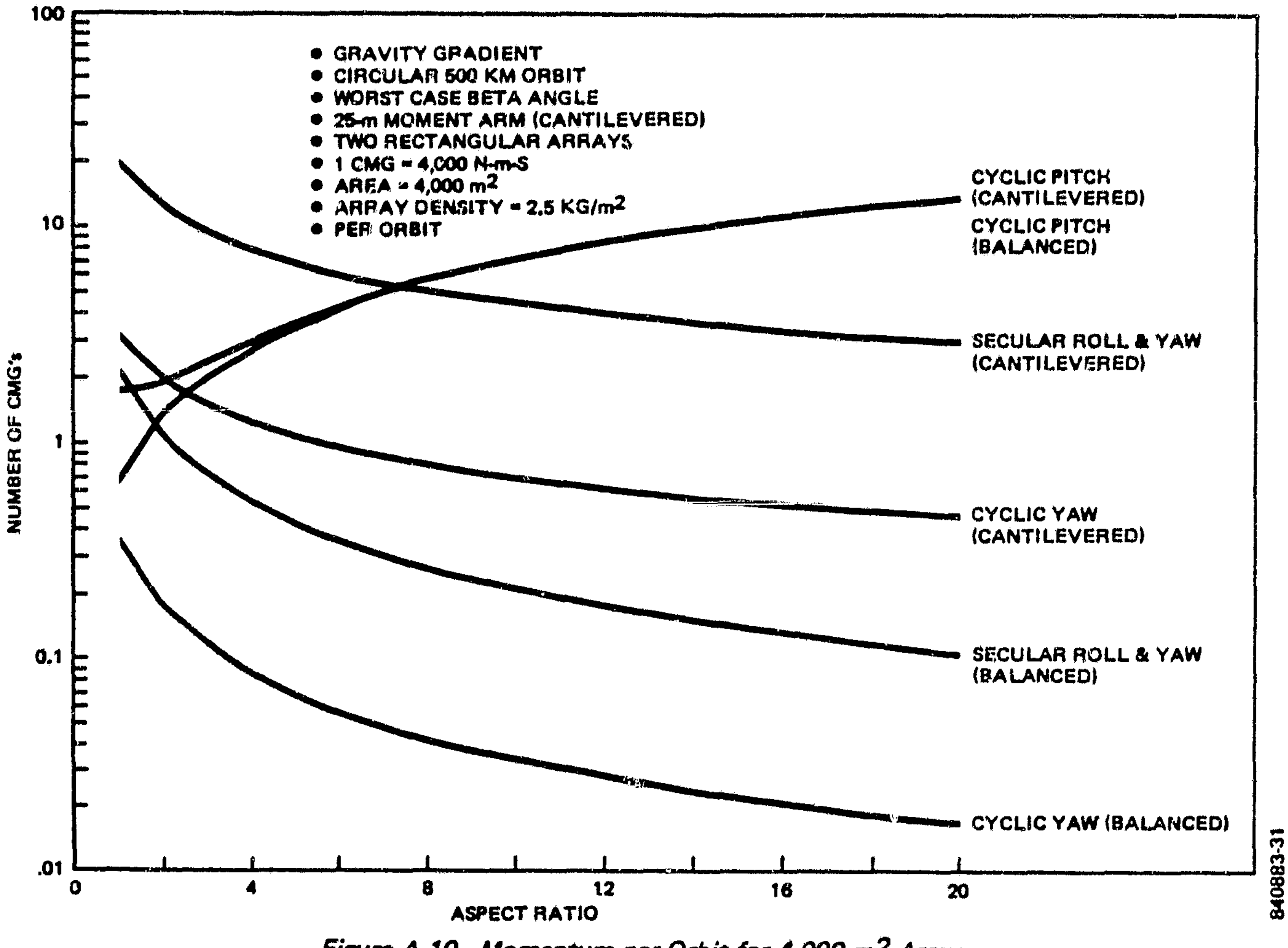


Figure A-10. Momentum per Orbit for 4,000 m^2 Array

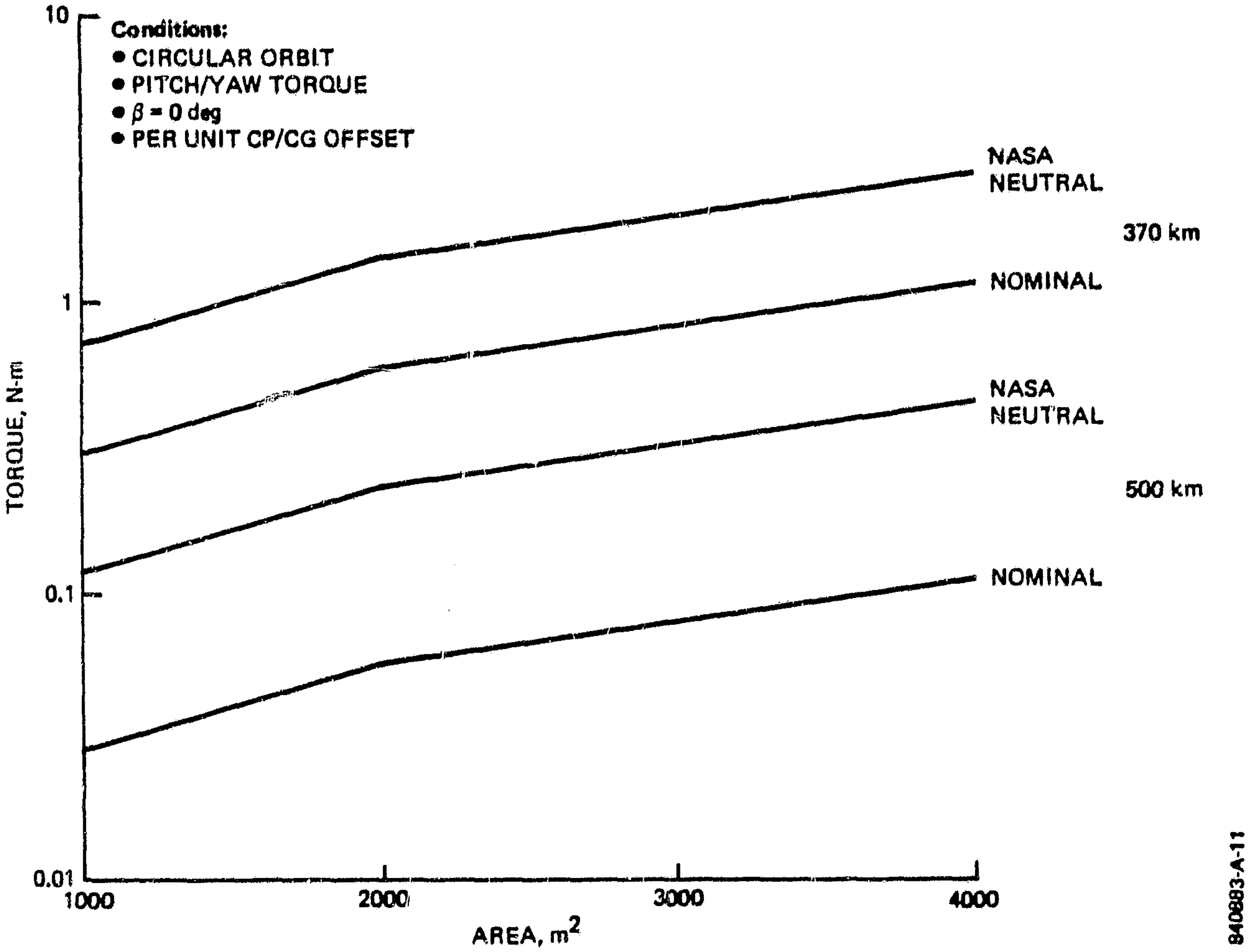


Figure A-11. Maximum Aerodynamic Torque per Orbit

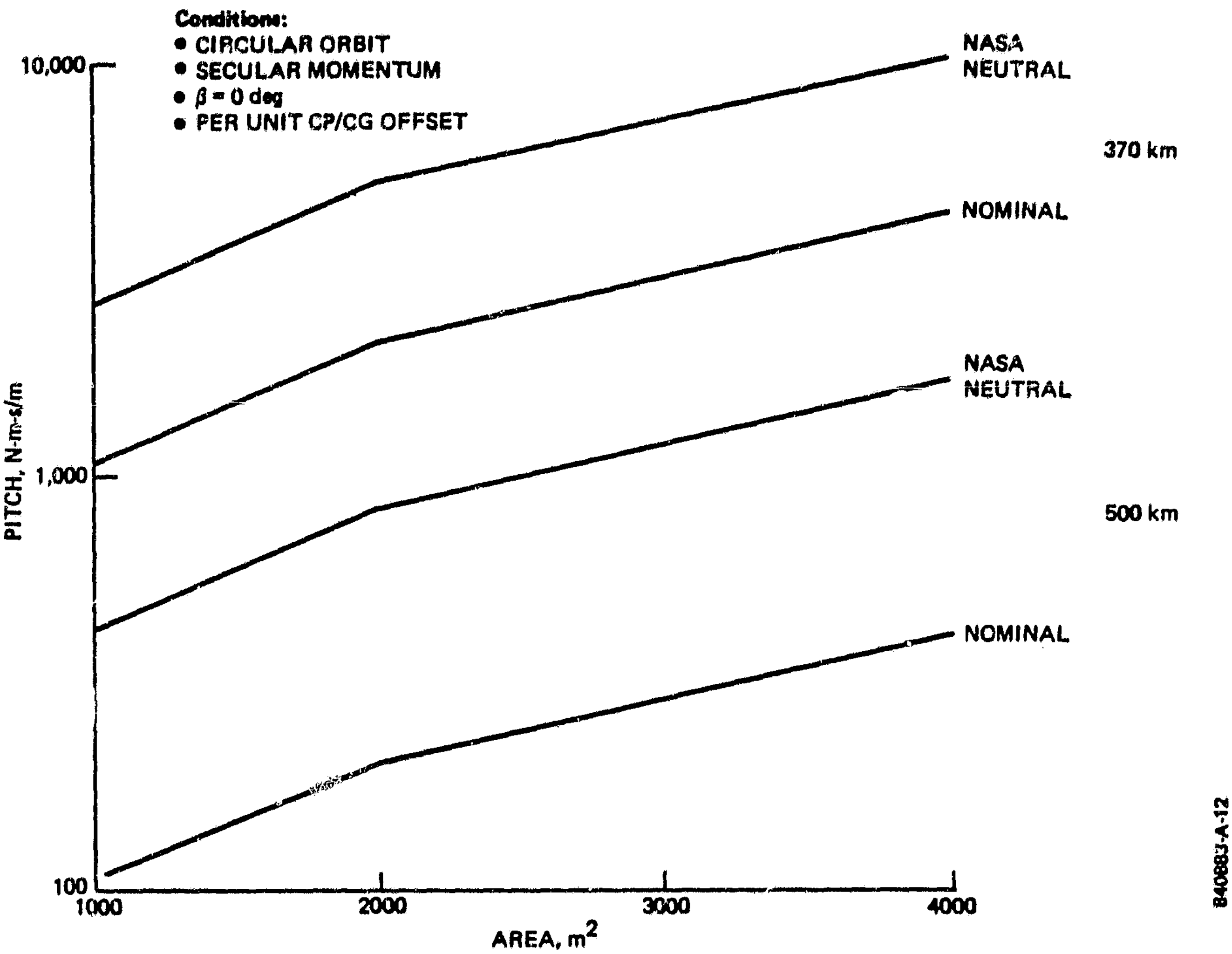


Figure A-12. Maximum Aerodynamic Pitch Angular Momentum per Orbit

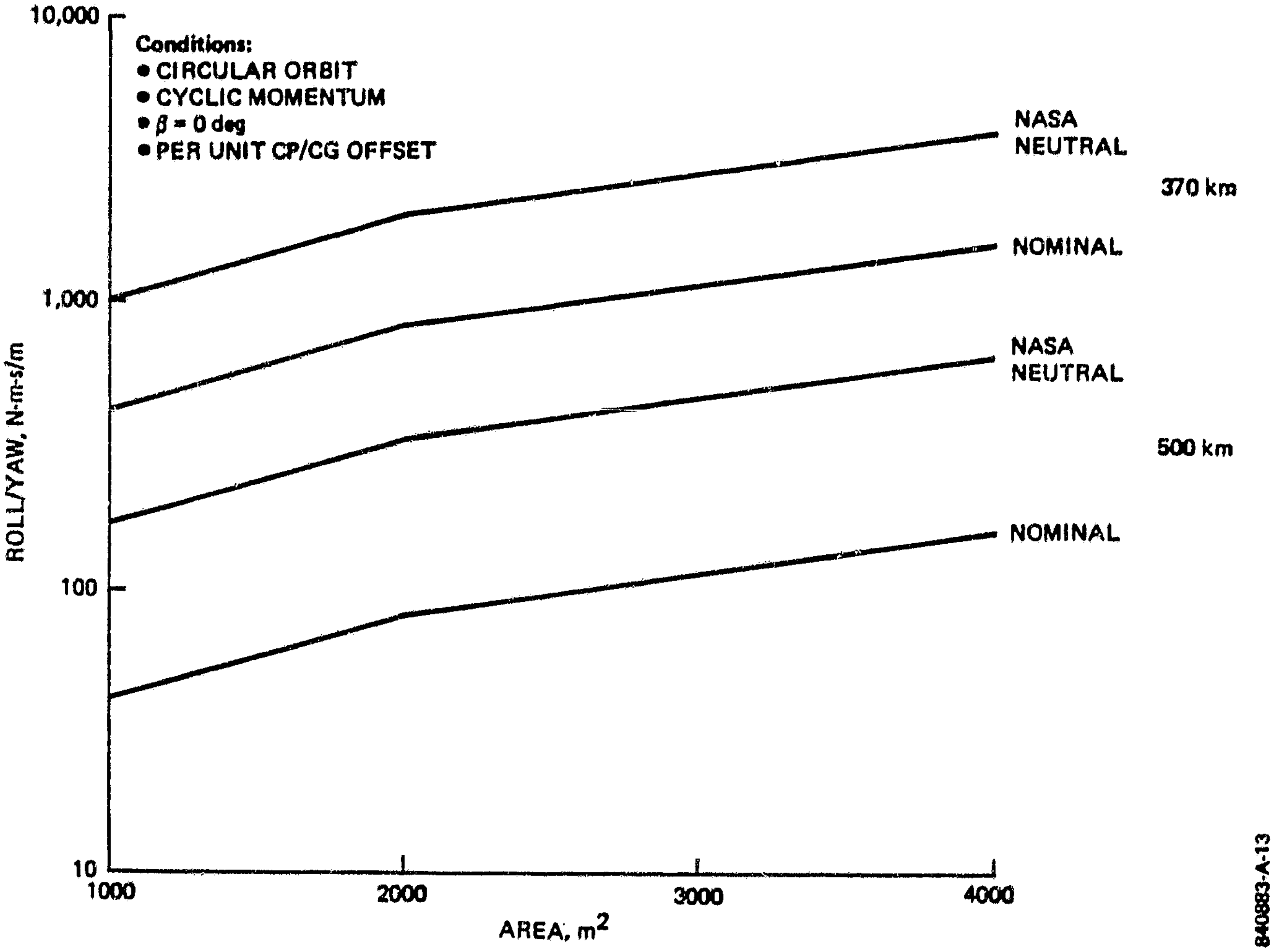


Figure A-13. Maximum Aerodynamic Roll/Yaw Angular Momentum per Orbit

Appendix B

WATER ELECTROLYSIS PROPULSION SYSTEM ANALYSIS

B.1 Introduction

Electrolyzing water to produce oxygen and hydrogen for propulsion purposes has been analysed by several investigators (i.e., Marquardt and General Electric; Aerojet; JPL; etc.). These analyses were either preliminary in nature or have concentrated on particular aspects of the system. The analysis presented here examines the use of water electrolysis in more detail as it operates as a part of the Space Station propulsion system. The purpose of this analysis is to determine the combination of variables that maximize system performance and minimize energy consumption, while satisfying the propulsion requirements discussed in sections 5.0 and 6.0. There are many factors that affect a water electrolysis propulsion system. The most prominent ones are: energy consumption; system and engine specific impulse; storage and accumulator tank sizes and weights; impact on ECLS; and finally, water resupply. Another key factor, system cost, is not addressed in this study.

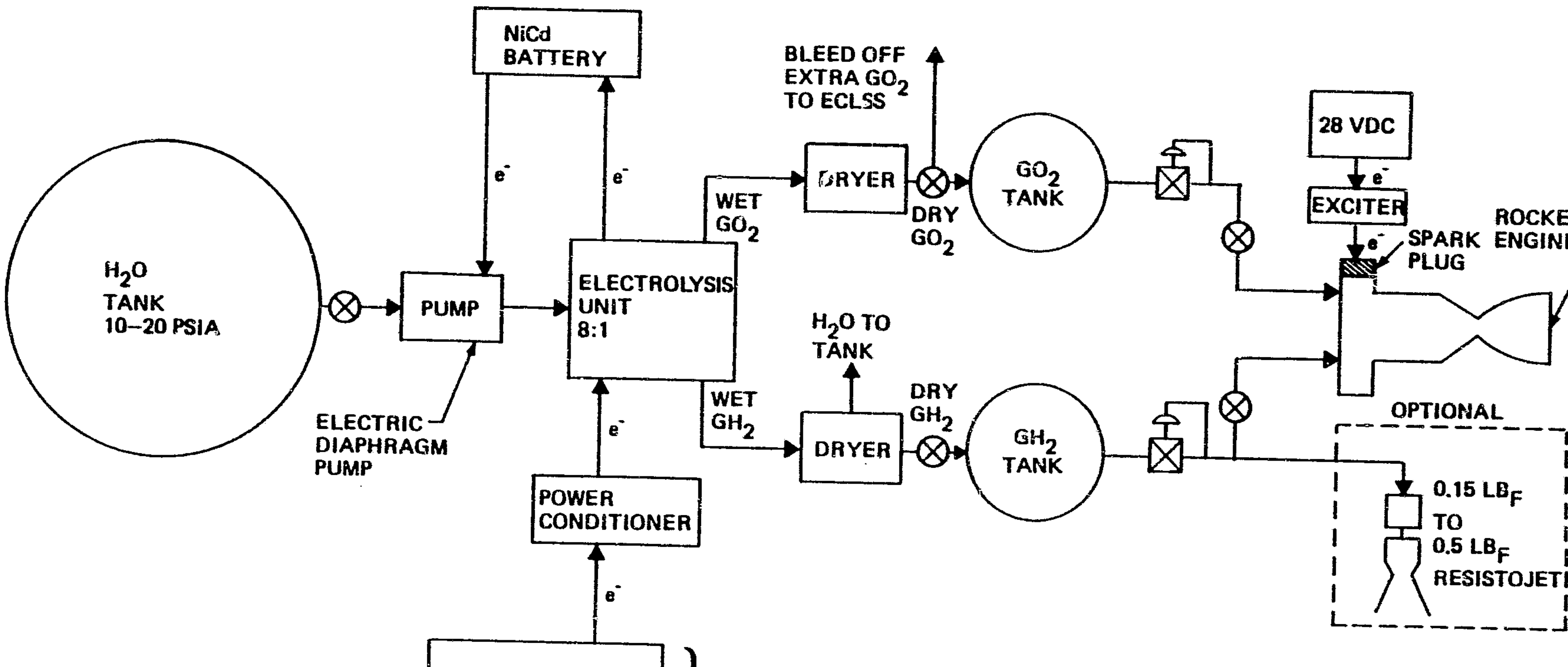
This analysis assumes the following Space Station operating parameters:

- o Altitude = 525 km
- o Neutral Atmosphere
- o Duty Cycle = One Thrust/10 Days
- o Thrust Duration per Duty Cycle = 600 sec
- o C_D = 2.2
- o Resupply Period = 90 days

B.2 Water Electrolysis Propulsion System

There are several types of water electrolysis propulsion systems. Figure B-1 shows a simplified schematic of the baseline system used in this analysis. Other systems include using a helium bottle to force the water

SIMPLIFED SCHEMATIC



CREW SIZE	2 TO 4	4 TO 6	8 TO 12
AREA (M^2):	1046	2000	4000
AVERAGE POWER (KW): B.O.L.	110	210	420

Figure B-1. Water Electrolysis Propulsion System—Simplified Schematic into the electrolysis unit and a nickel-cadmium battery (which is charged by the solar panels) to power the valves; also a more advanced concept would utilize oxygen/hydrogen fuel cells instead of batteries for energy storage.

The system shown in figure B-1 has its energy supplied by the solar panels and then sent through a power conditioner, which in turn supplies the regulated amount of current required by the electrolysis unit for its particular needs. These needs are determined by the Space Station operating parameters, which for this analysis, have already been assumed. The mixture ratio that the engine runs at also has a large impact on the electrolysis requirements. Therefore, comparisons are made between the various driving factors based on mixture ratio. After energy is sent to the electrolysis unit(s), some of the energy goes to recharge the battery, which is used to run the electric diaphragm pump and various valves. Water is then pumped into the electrolysis unit and separated into wet GO_2 and GH_2. Figures B-2 and B-3 show schematics of the electrolysis process and of the water electrolysis cell (ref. B-1). The wet GO_2 and GH_2 means that the propellants are saturated with water vapor. In the case of the GO_2, it contains about 85% water vapor, while the GH_2 contains 95% water vapor. Thus, the next process is for the wet GO_2 and GH_2 to go through dryers, after which the dry GO_2 and GH_2 are stored in accumulator tanks at the temperatures and pressures at which they were produced. Prior to storing the GO_2 in the accumulator tanks, the excess GO_2, which depends on the mixture ratio, is bled off, possibly to the ECLSS for crew respiration.

The following sections will analyze each of the dominant driving factors, how they are affected by mixture ratio, and how they in turn affect the system as a whole.

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1 R. C. Stechman, Jr., and J. G. Campbell; Marquardt, "Water Electrolysis Satellite Propulsion System," AFRPL-TR-72-132, January 1973.

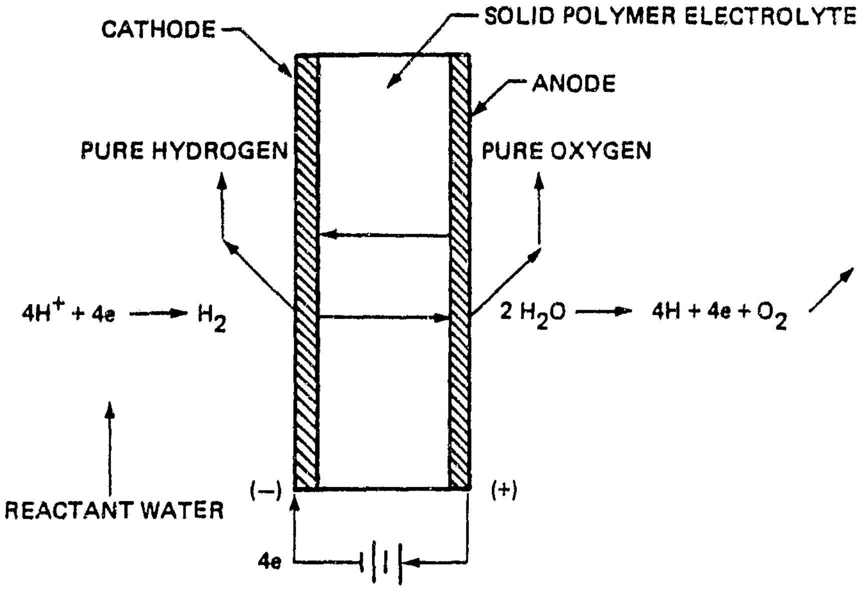


Figure B-2. Electrolysis Process

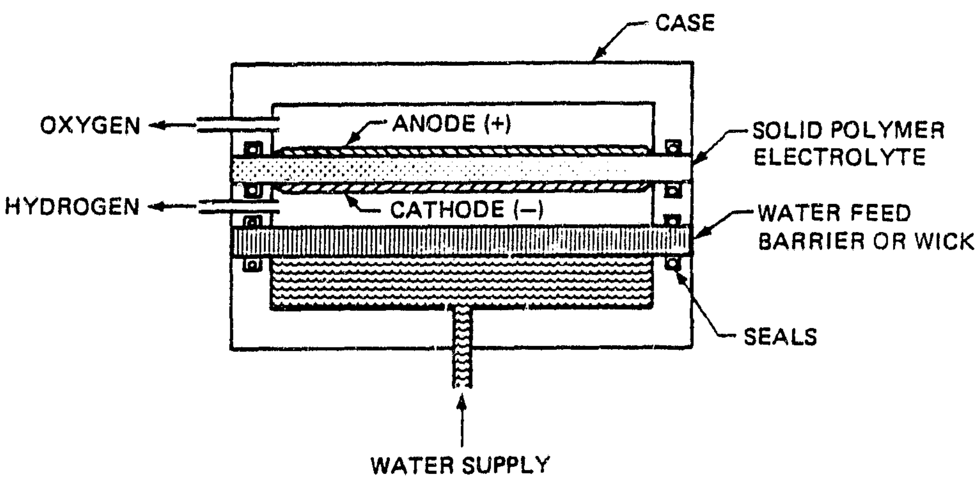


Figure B-3. Water Electrolysis Cell Schematic

B.3 Engine Specific Impulse Performance

The current SOA engine I_{sp} performance of GO_2/GH_2 thrusters is shown in section 6.3.5.1. Very little data were available on small GO_2/GH_2 thrusters. Thus, a conversion table was developed (figure 6-24 through 6-28) showing approximated SOA and projected SOA percentages, based on available data. These approximations have since been shown to be fairly accurate, although slightly conservative. The system analysis that follows is based on these approximations.

B.4 System Specific Impulse Performance

The system specific impulse is defined here as being the I_{sp} that is calculated from the flow rate and mixture ratio (stoichiometric) prior to the GO_2 being bled off (see the schematic in lower right corner of figure B-4). Hence, if the engines run at an 8:1 mixture ratio, then all of the GO_2 produced during electrolysis will be required for combustion and thus the engine and system I_{sp} will be the same. However, if the mixture ratio is anything less than stoichiometric, then the system I_{sp} will drop according to how low the mixture ratio becomes. For example, at a mixture ratio of 2.5:1, only 31% of the GO_2 produced is used for combustion, thus the figure shows the relationship between the engine and system I_{sp}.

B.5 Energy Requirements

The energy requirements for a water electrolysis system appear to have only a small effect on the overall size increase of the solar arrays. The effect of energy requirements on solar array size varies with each station and diminishes with station size. The increase in solar array size ranges from 1.0% to 2.5% for mixture ratios of 3:1 to 2.5:1 respectively. Figure B-5 shows how the energy requirements increase exponentially as the mixture ratio decreases. The minimum energy expenditure occurrs at a stoichiometric mixture. The average BOL power available for each station is shown on figure B-1 and was used in determining percentage increases of the solar

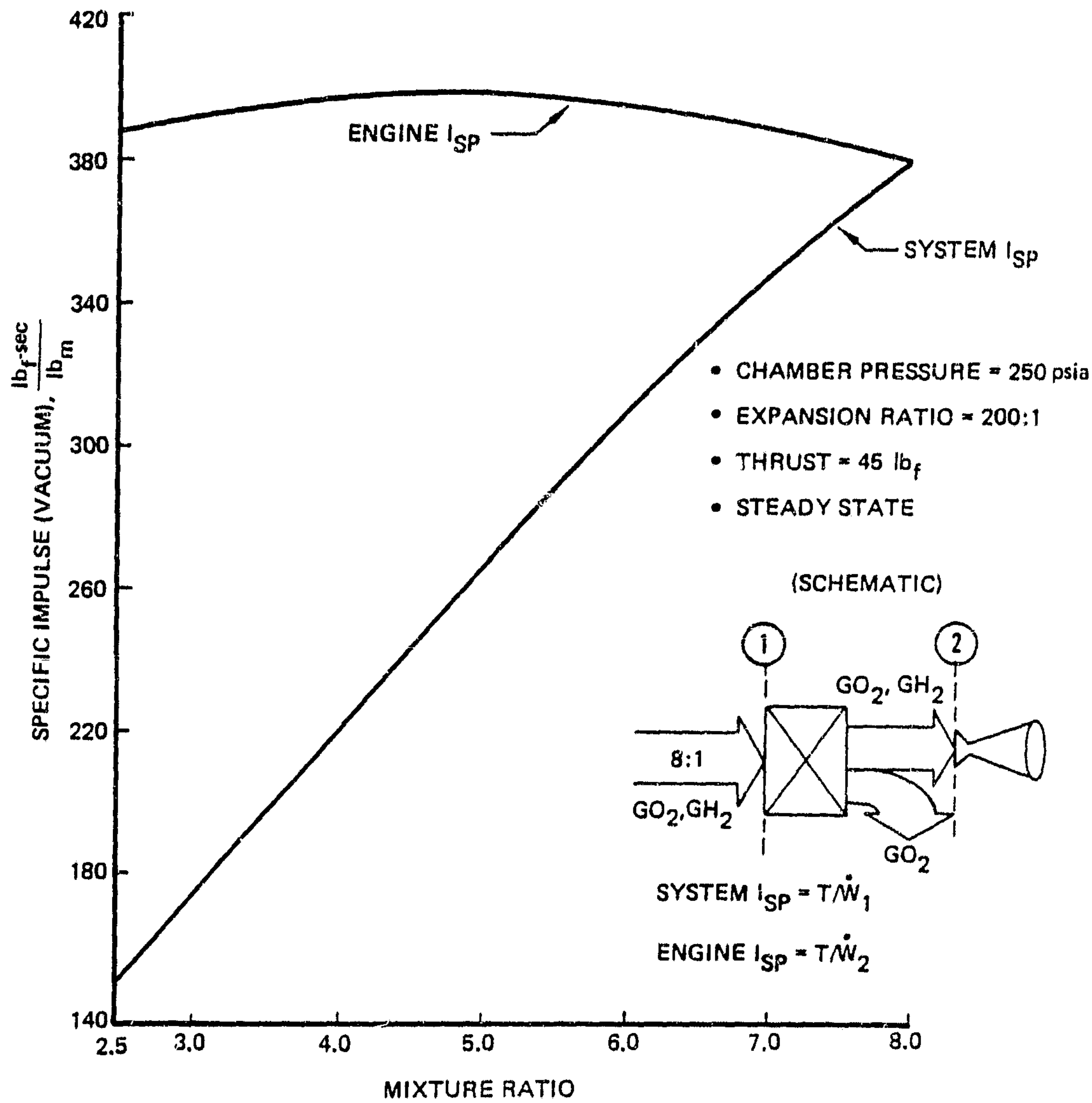


Figure B-4. System Specific Impulse Performance

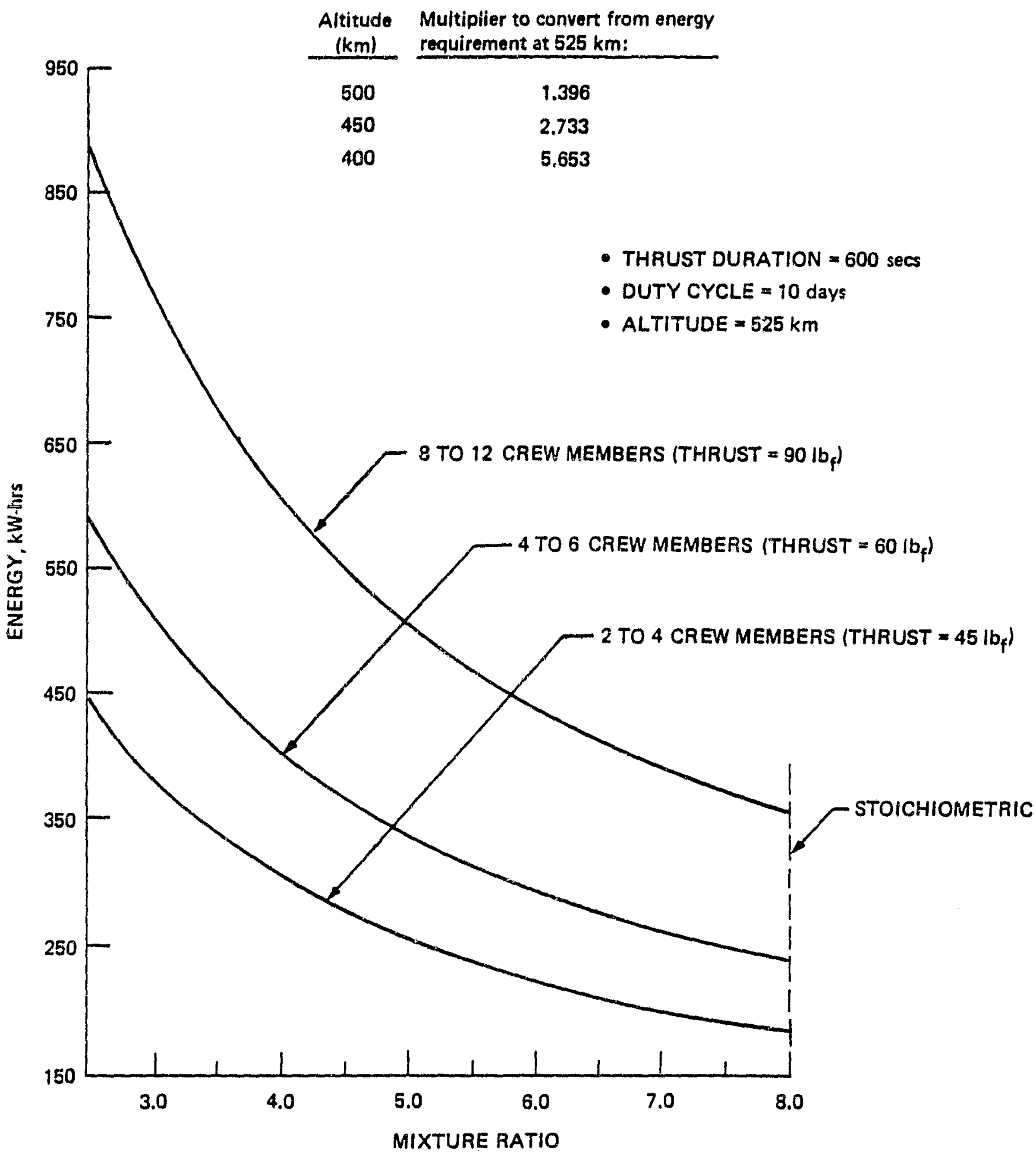


Figure B-5. Energy Requirements for Water Electrolysis Propulsion System

Table B-1. Electrolysis Cell Characteristics

Design parameter	Design data
Number of cells	6
High H_2O rate (maximum)	2.3 lb H_2O/day
Low H_2O rate (adjustable)	0.1 to 0.6 lb H_2O/day
Cell stack characteristics	
@70°F mean temperature	
Stack current	22 amps
Stack voltage	10.7 VDC
Stack input power @2.3 lb H_2O/day	235 W
Stack heat loss	44 W
Current density	100 amps/ft^2

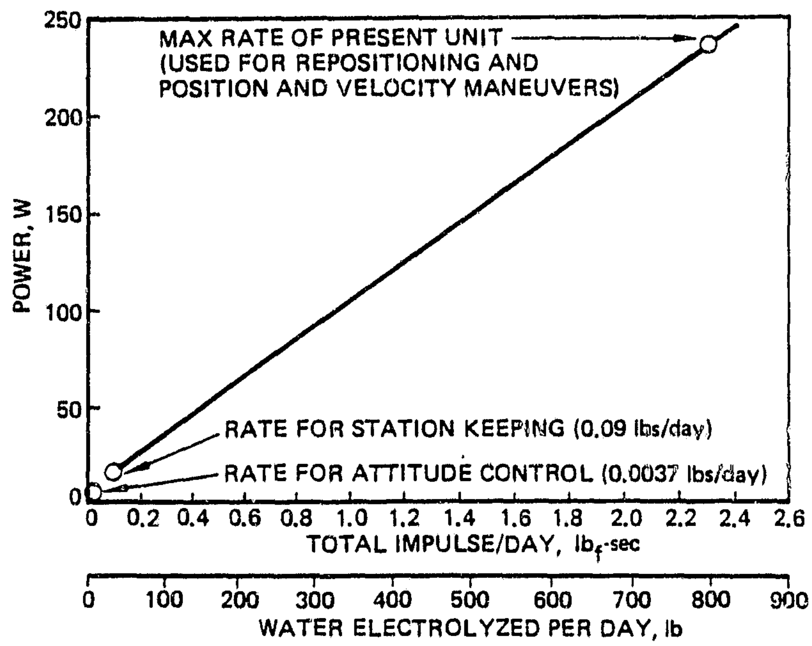


Figure B-6. Power Requirements for the General Electric 6-Cell Electrolysis Unit array. The current area of the solar arrays is also shown. In determining the energy requirements, it was assumed that the electrolysis units run at maximum output and the arrays are sunlit approximately 2/3 of the time. Using the electrolysis units produced by General Electric (ref. B-1), the power requirements are 235 watts at 2.3 lbm-H_2O/day for each 6 cell electrolysis unit. Table B-1 and figure B-6 show the electrolysis cell characteristics and power requirements re- spectively.

B.6 Accumulator Tank Sizing

In analyzing the effects of mixture ratio on accumulator tank sizes, it was assumed the tanks were to be spherical and made from 2219 aluminum. The equation used in obtaining the tank weights is as follows:

$$W = K_{T\ A}\ K_{L/D}\ 1.5\ (\rho_{2219}/F_{TU})(UFS)(P_{max.oper.})\ \frac{m\ t}{\rho_i - \rho_f}$$

where:

$K_{T\ A}$	=	1.10; conversion from theoretical to actual.
$K_{L/D}$	=	1.0; length over diameter factor; spherical tanks.
1.5	=	constant for 50% contingency factor.
ρ_{2219}	=	176.256 lb/ft^3; density of 2219 aluminum.
$P_{max.oper.}$	=	400 psia, maximum operating tank pressure.
F_{TU}	=	63,000 psia; ultimate strength at room temperature.
UFS	=	2.0; ultimate factor of safety
m	=	$f(m_R, I_{sp}, T)$ flow rate to the engine; lbm.
t	=	600 sec; thrust duration per duty cycle.
ρ_i	=	$f(T_i, P_i)$; initial density of gas in tank; lbm/ft^3.
ρ_f	=	$f(T_f, P_f)$; final density of gas in tank; lbm/ft^3.

In this analysis, composite propellant tanks were not considered due to the low operating pressure of the tanks.

It was assumed that a compressor was not desirable, as discussed in section 6.4.3.1, because it would increase the system weight and power consumption, and decrease reliability. Therefore, the maximum operating tank pressure was limited by the maximum operating pressure of the electrolysis unit (ref. B-1). To avoid having excessively large tanks, the pressure in the tank is assumed to be for a blowdown mode with pressure ranging from 500 to 75 psia. In keeping with the available pressure when the tank pressure is at a minimum, the operating chamber pressure was assumed to regulate at 50 psia. Figure B-7 shows how strongly GH_2 dictates the total weight and size of the accumulator tanks. An 8:1 mixture ratio minimizes the accumulator tanks size and weight.

B.7 Water Resupply and Storage

Probably the most attractive aspect of using a water electrolysis propulsion system is the safety associated with resupplying and storing the water. Figure B-8 shows the amount of water that is required to be electrolyzed over 90 days for the three station sizes at various mixture ratios. The amount of water required for electrolysis is proportional to both the system I_{sp} and the engine I_{sp}, however the former has a stronger impact. Therefore, since the system I_{sp} optimizes at an 8:1 mixture ratio (figure B-4), the amount of water resupply also optimizes at 8:1, thus requiring the least amount of water.

It is possible to reduce the amount of water required for resupply even further by utilizing the excess water produced from ECLSS. Figure B-9 shows a schematic of the ECLSS system using a Sabatier system. Depending on the number of people on board, the station produces an excess of from 0.475 lbm/day of water for a crew of 2 men to 2.85 lbm/day for for a crew of 12 men, for a total of 42.75 lbm to 256.5 lbm, respectively, over a 90 day period. Depending on the mixture ratio selected and the number of people on board the station, between 3 and 20% of the water required for electrolysis can be supplied from excess ECLSS water (see figure B-10). If the Sabatier system is not used, then the amount of water available for resupply from the ECLSS is 58.2 lbm per 8 crew members. Thus, the percentage of water is between 81 and 100% respectively, again depending on the mixture ratio and number of crew members on board the station. For the 12-crewmember station, there is more than 6 times as much water available

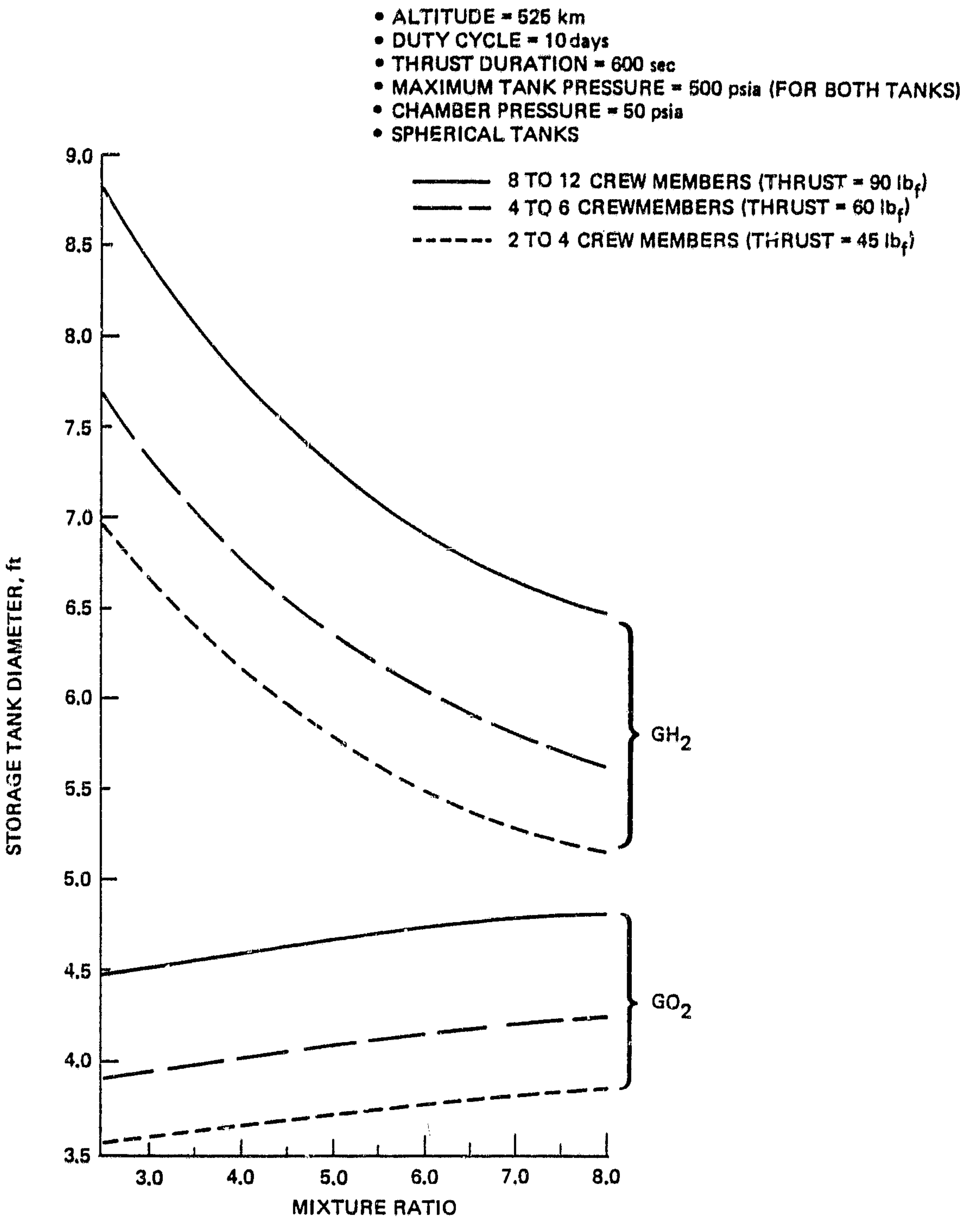


Figure B-7. Accumulator Tank Sizing

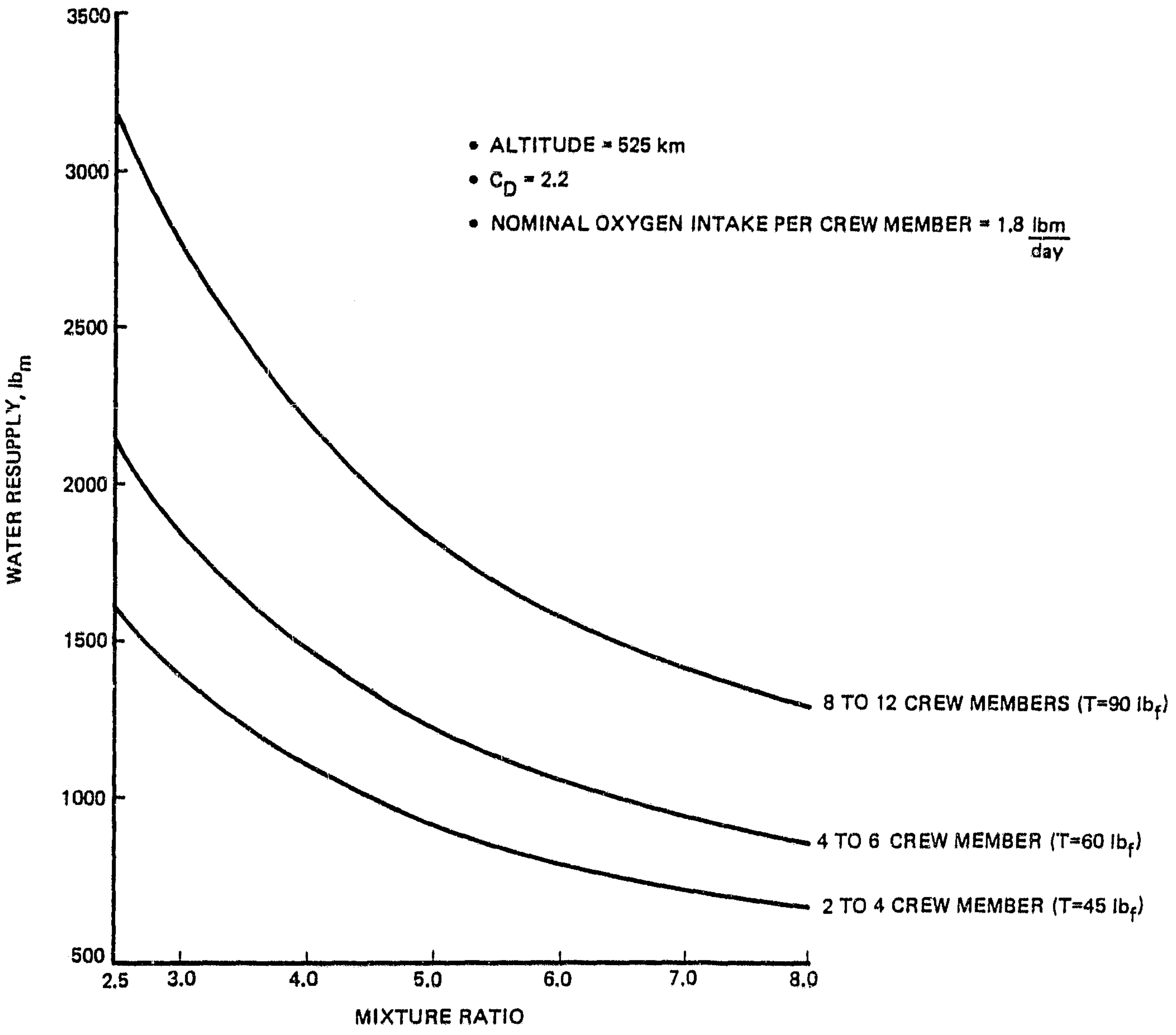


Figure B-8. Water Resupply for 90 Days

Based on 8 crew members

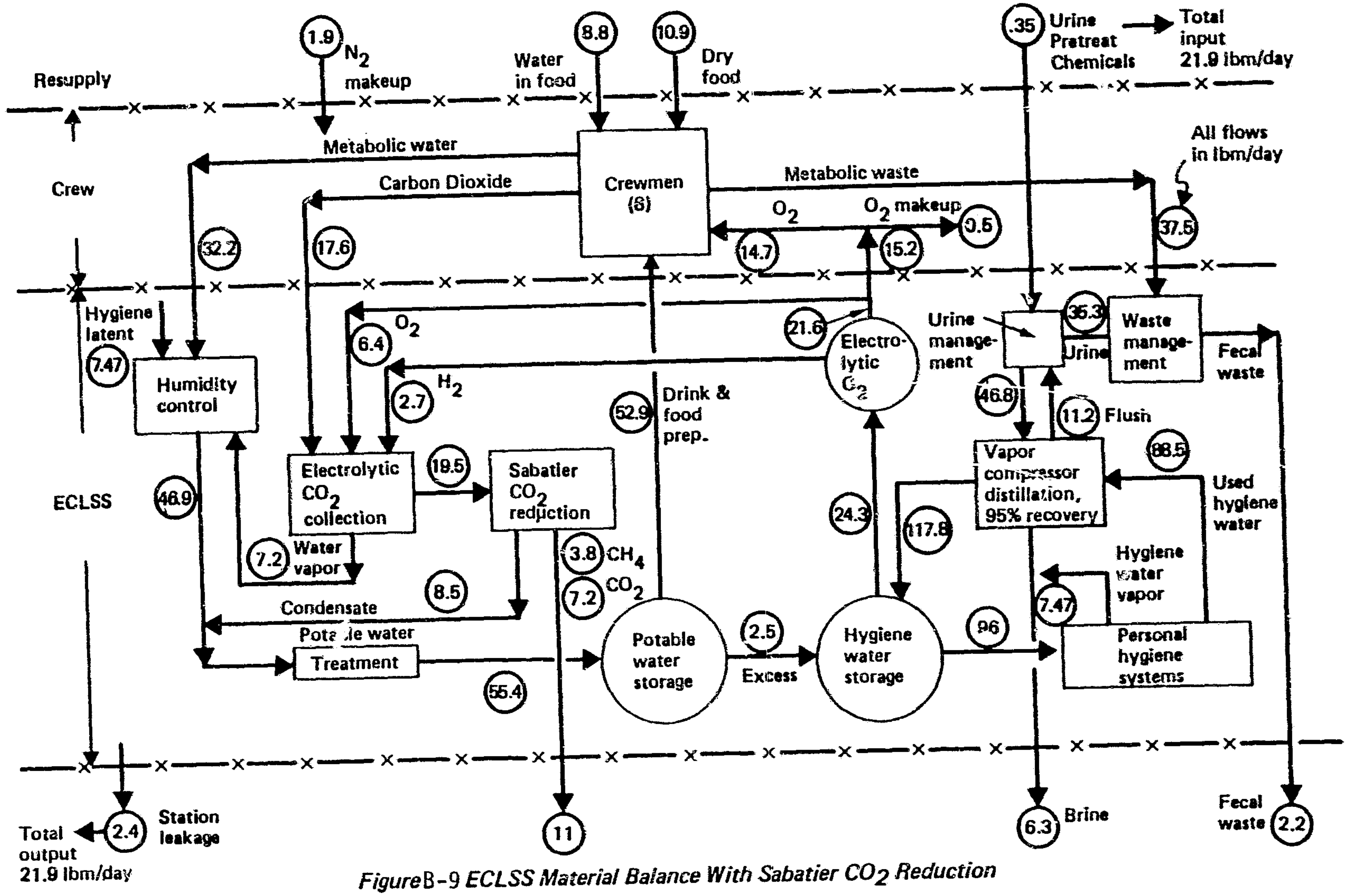


Figure B-9 ECLSS Material Balance With Sabatier CO_2 Reduction

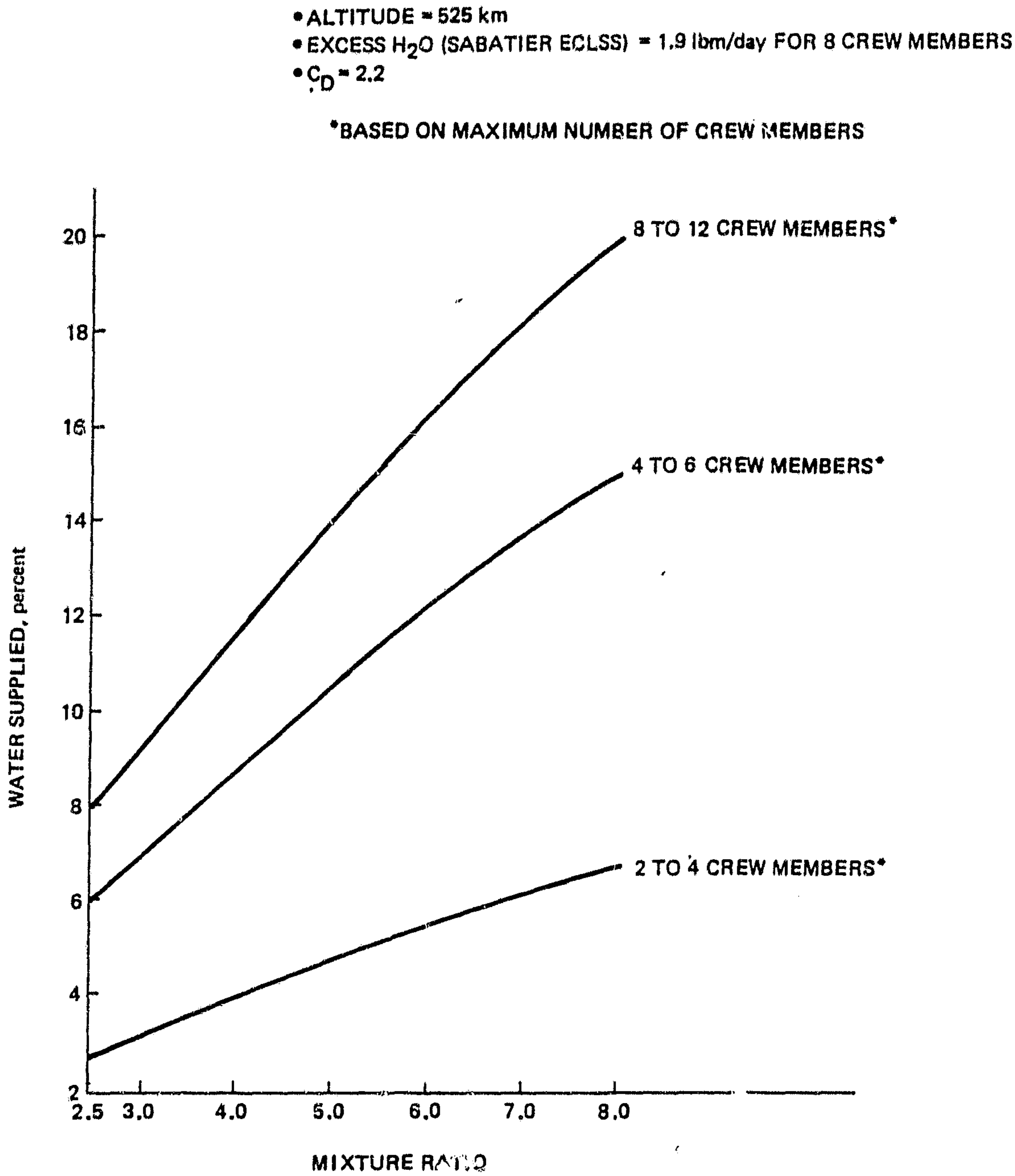


Figure B-10. Percentage of Water Supplied from ECLSS Excess then required. A trade would need to be performed to determine the cost savings of eliminating the need for resupplying water compared to not using the Sabatier system.

It should be noted that the percentages mentioned above will probably vary depending on the type of electrolysis system used. For example, separation using a solid polymer electrolyte (as is used in this appendix) might have a lower percentage of water than it can utilize, since this requires highly purified water to prevent contamination of the electrode catalyst and/or cell electrolyte. Whereas, using an alkaline electrode, static feed electrolysis system does not require purification of the water (assuming it is not sludge), since the water to be electrolyzed is separated from the electrodes and cell electrolyte by a gas compartment containing only water vapor and hydrogen. Hence, a significant part of the cost of using electrolysis, which is incurred during the purification of the water, is eliminated.

B.8 Use of Excess GO_2 for Crew Respiration

Figure B-11 shows the number of crew members whose O_2 requirements can be satisfied as a function of mixture ratio. It is seen that a large portion, if not all, of the oxygen requirements for crew respiration can be satisfied depending on the mixture ratio selected. This figure shows that, as the mixture ratio decreases, the number of people that can be supported increases. For example, on the 2- to 4-crewmember station, a mixture ratio of 4.7:1 can support two crewmembers, whereas a mixture ratio of 2.85:1 is can support four crewmembers. As the station size increases, the required mixture ratio needed to satisfy all of the crew respiration needs decreases. Whether this represents a desirable integration of subsystem operations, would have to be determined by a trade study.

B.9 Overall Performance

In addition to the factors addressed in sections B.3 through B.8, there are two additional factors that affect performance of a water electrolysis system; contamination and thruster life. Both have been addressed in sections 6.3.4.4 and 6.4.3.2, respectively. In section 6.3.4.4, it was shown that at mixture ratios 8:1 or greater, free radicals are formed and excess O_2 in the exhaust. In section 6.4.3.2, it was shown that by running thrusters at a low mixture ratio (2.5:1), the chamber temperature decreases from that of an 8:1 mixture ratio by approximately $1500^{o}R$. Both can have a significant effect on the Space Station environment and the life of the thrusters.

It is, therefore, important to determine which factors are going to be more critical to the overall performance of the system. Since it has been shown that the propellant requirements are not critical, both the system and engine specific impulse are considered secondary requirements. Also, the accumulator and resupply tanks are considered secondary, due to the relatively small propellant requirements over a 90 day period. The use of excess GO_2 for crew support is also considered secondary, since ECLSS will not be dependent on any system and is only an additional feature of water electrolysis. This limits the main factors affecting performance to energy requirements, contamination, and thruster life. Energy requirements have been shown to be fairly small even for low mixture ratios suich as 2.5:1. Therefore, contamination and thruster life are considered the two primary factors affecting overall performance.

It can be concluded that the most efficient mixture ratio to operate a water electrolysis system is somewhere in the range of 2.0:1 to 3.5:1. This is only a first general approximation and is considered preliminary. A detailed trade analysis is required on the factors that have been discussed and others, to determine a more precise range over which to operate a water electrolysis propulsion system.

B.10 Summary

The use of water electrolysis as a source for propellant proves to be a promising alternative. Many of the problems that were initially associated with water electrolysis have been addressed and shown to be unsubstantial. Power consumption was initially considered a significant factor in determining the potential of a water electrolysis-based propulsion system.

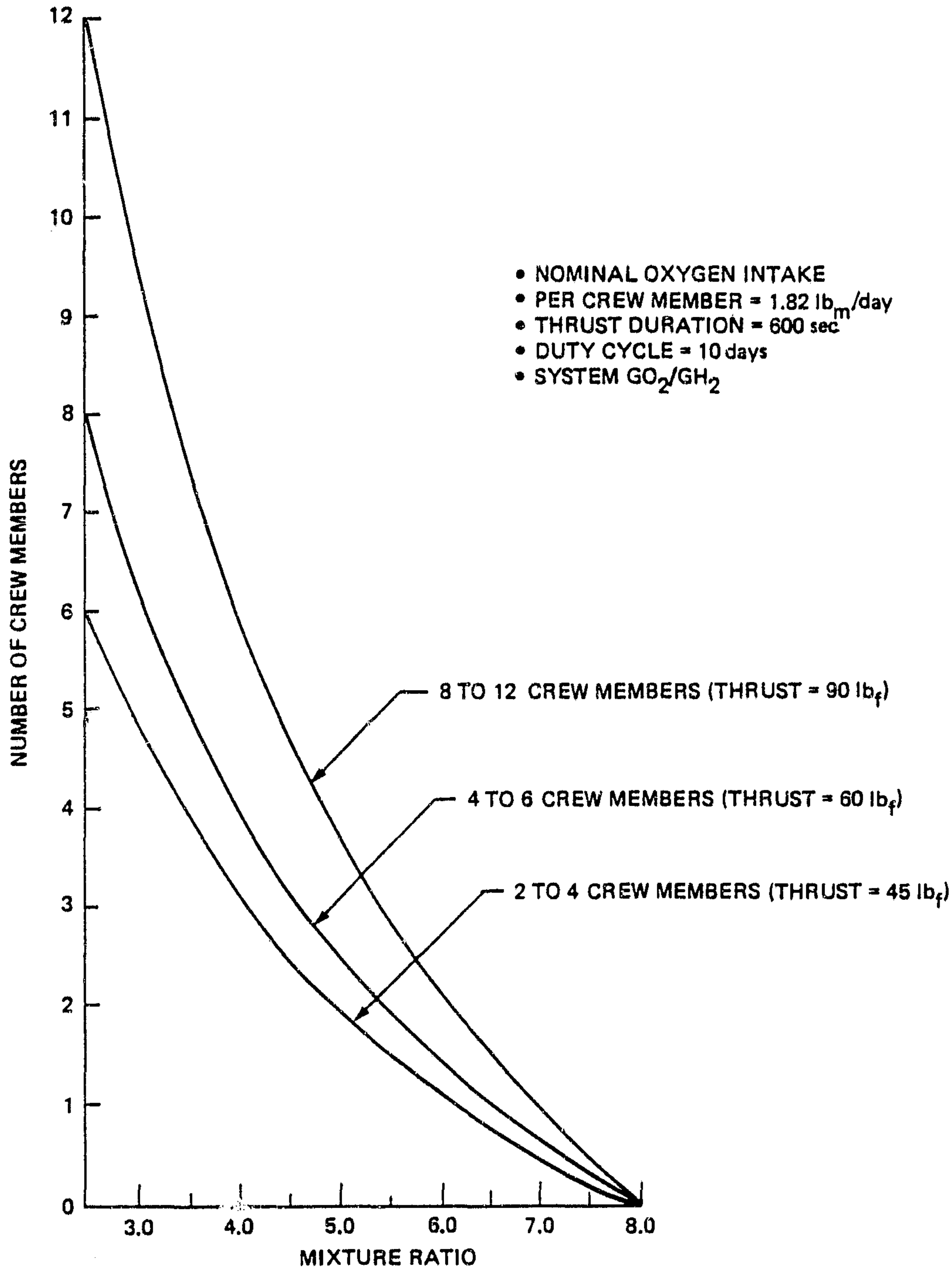


Figure B-11. ECLSS Usage of Excess Oxygen from Water Electrolysis Propulsion System

However, it was shown that the impact of a water electrolysis system on power requirements is small compared to other systems. Water electrolysis may be used in conjunction with regenerative fuel cells and the ECLSS, making its use in conjunction with the propulsion system even more viable. Water electrolysis unit performance data were obtained from a combined General Electric - Marquardt study[1] that was performed in 1973.

A concern associated with water electrolysis is the water vapor removal procedure requiring desiccant units for both GO_2 and GH_2. These units, capable of absorbing 20% of their weight in water vapor, will require frequent replacement.

1. R. C. Steehman, Jr., J. G. Campbell, "Water Electrolysis Satellite Propulsion System", Marquardt Co. AFRPL-TR-72-132, January 1973.

Appendix C

SPACE STATION HYDRAZINE PROPULSION SYSTEM RELIABILITY COMPARISON

C.1 Introduction

A reliability analysis was conducted to estimate the comparative reliability of a 27 lb_f hydrazine reboost system and a 0.1 lb_f augmented catalytic thruster (ACT) reboost system during a ten-year mission. The reliability prediction was determined by using reliability logic diagrams, mathematical models, and calculations progressing from the detailed listing of generic failure rates, the modification factors used to account for different environmental stresses and operating conditions, applicable time/cycle data, and the step by step use of this data in the reliability mathematical models.

C.2 System Configuration

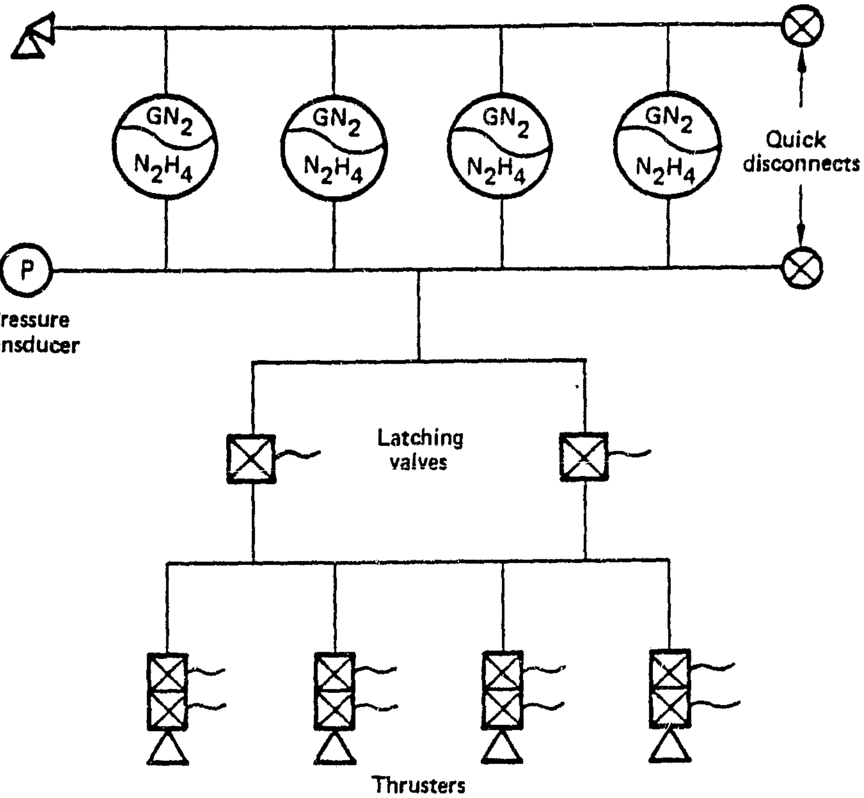
The propulsion system configuration studied is shown in figure C-1. The two systems are identical except for the following differences:

Case 1 - Propulsion system uses four 27 lb_f hydrazine thrusters.
Case 2 - Propulsion system uses four 0.1 lb_f ACT hydrazine thrusters.

C.3 Mission Profile

During the 10-year (87,600 hour) mission, the mission profile assumed (reference 27) for Case 1, 25 thrust cycles and a 17 hour-per-engine burn time with a total impulse of 6.5×10^6 lb_f-sec. The mission profile for Case 2 included 120 thrust cycles and a 4200 hour-per-engine burn time with a total impulse of 6.1×10^6 lb_f-sec.

For analysis purposes, the mission was divided into four phases. Phase A is the launch phase, which was assumed to be 0.2 hours in duration. Phase B is the time in orbit when the propulsion system is non-operating; Phase C is the propulsion system operating time, and Phase D covers the operating



4 27-lb_f* N_2H_4 thrusters

ACT units identical except add power leads

Figure C-1. N_2H_4 Propulsion System Configuration cycles while in orbit; Phase D is used to calculate part failures when failure rates are given in terms of failures per cycle.

The thruster operating time was also the power-on time for the thrust chamber valves and gas generator valves. The number of thruster pulses will be the number of series valve on/off cycles. The propellant latch valves will be cycled less frequently. Ten operating cycles were assumed for the latch valves. The propellant subsystem was treated as operating for the full duration in orbit.

C.4 Failure Rate Data Source

The failure-rate data used for this analysis have been selected from available industry data that seemed representative for a typical mission. The failure rates selected and their data source are presented in table C-1. Failure rate data sources include SAMSO-specified failure rates for some of the current aerospace programs (e.g, IUS, Fleet SATCOM, etc.); the Avco Reliability Engineering Data Series document "Failure Rates;" RRC Reliability Bulletin RB-1, RRC Failure Rate Data; and NPRD-2, "Non-electronic Parts Reliability Data."

The generic failure rates given in table C-1 were modified by an environmental factor (KE) to account for the varying stress levels during different mission environments, and an application factor (KA) to adjust for operating and non-operating (or quiescent) stresses. An environmental factor of 800 was selected for the launch period to reflect the high stress environment of the launch phase.

The selection of the KA value is important to this analysis because of the long term, nonoperating time accumulated in orbit. Since the valve failure rate can be a major factor in determining the overall system reliability, it is reasonable to derive the KA from available industry valve failure rate data. The DOD Reliability Analysis Center document NPRD-1, Non-Electronic Parts Reliability Data, gives a dormant failure rate of 0.009×10^{-6} failures per hour (f/hr) for a solenoid valve. Using the time-dependent generic failure rate of 2.27×10^{-6} f/hr selection as

Table C-1. Failure Rate Selection

Hardware Item	Failure Rate $\times 10^{-6}$ (Flight)	Data Source Remarks
1.1, 1.4 Manual/fill/vent/drain valve	0.07 f/hr	SAMSO gives a failure rate of 0.07 x 10^{-6} failures per hour.
1.2, 1.5 Fill valve cap seal	0.07 f/hr	For this analysis, the redundant seal provided by the fill/vent/drain valve caps was assumed to have the same failure rate as the valve.
1.3 Propellant tank	0.12 f/hr	SAMSO gives a failure rate of 0.120 x 10^{-6} f/hr for a propellant tank.
1.6 Lines & fittings	0.001 f/hr	RRC RB-1 failure rate for lines and fittings (SAMSO gives a failure rate of 0.0006 x 10^{-6} per inch of weld of tubing).
Solenoid valve (Latch valve & thrust chamber valve)	2.27 f/hr and 0.01 f/cy	SAMSO gives a failure rate of 0.35 x 10^{-6} f/cy for a solenoid valve without accounting for a time-dependent failure rate. The AVCO failure rate tables give a mean failure rate for a solenoid valve of 11.0 x 10^{-6} f/hr without accounting for a cycle-dependent failure rate. For this analysis, a combined failure rate that includes both time-dependent and cycle-dependent failure rates was used. RRC RB-1 gives a combined failure rate of 2.27 x 10^{-6} f/hr for the time-dependent failure rate plus 0.01 x 10^{-6} f/cy for cycle-dependent failures. The most probable valve failure mode is internal leakage, which accounts for approximately 84.9% of valve failures and corresponds to a failure rate for the fail open/leak mode of 1.92723 x 10^{-6} failures per hour and 0.00849 x 10^{-6} failures per cycle. The failure rate excluding the fail open/leak mode is then 0.34277 x 10^{-6} failures per hour and 0.00151 x 10^{-6} failures per cycle.
3.1, 4.1 Excluding fail open/leakage mode only (i.e. fail closed)	0.34277 f/hr and 0.00151 f/cy	
2.1, 2.2 Fail open/leak mode only	1.92723 f/hr and 0.00849 f/cy	
4.2 4.3 Thrust chamber assy or GG chamber assy	0.055 f/hr	SAMSO gives a failure rate of 0.055 x 10^{-6} failures per cycle for a thruster.

Table C-1. Failure Rate Selection — Continued

	Hardware Item	Failure Rate $\times 10^{-6}$ (Flight)	Data Source Remarks
4.3	Augmentation heater assembly	2.5 f/hr	SAMSO gives a failure rate of 0.01×10^{-6} f/cy for a heaters. AVCO tables give a mean failure rate of 0.02×10^{-6} f/hr for heater elements with an upper limit of 0.02×10^{-6} failures per hour. MIL-HDBK-217C gives a part failure rate during space orbit of 5.0×10^{-6} f/hr for an electron vacuum tube power rectifier after 3 years of field use. A mature augmentation heater design should achieve a failure rate lower than the power rectivier since it does not have the cathode poisoning wearout that is included in the power rectifier failure rate. A failure rate at the mid-point between a SAMSO heater failure rate and the power rectifier tube was selected as the failure rate for the augmentation heater assembly.

described in table C-1, the quiescent failure rate modification factor is estimated as follows:

$$KA = \frac{t(\text{dormant})}{G} = \frac{0.0009 \times 10^{-6}}{2.27 \times 10^{-6}} = 0.004$$

The failure rate modification factors used are as follows:

K_E = 800 during the launch period
K_E = 1.0 during orbit
K_A = 1 for operating parts
K_A = 0.004 for nonoperating parts

C.5 Logic Models and Calculations

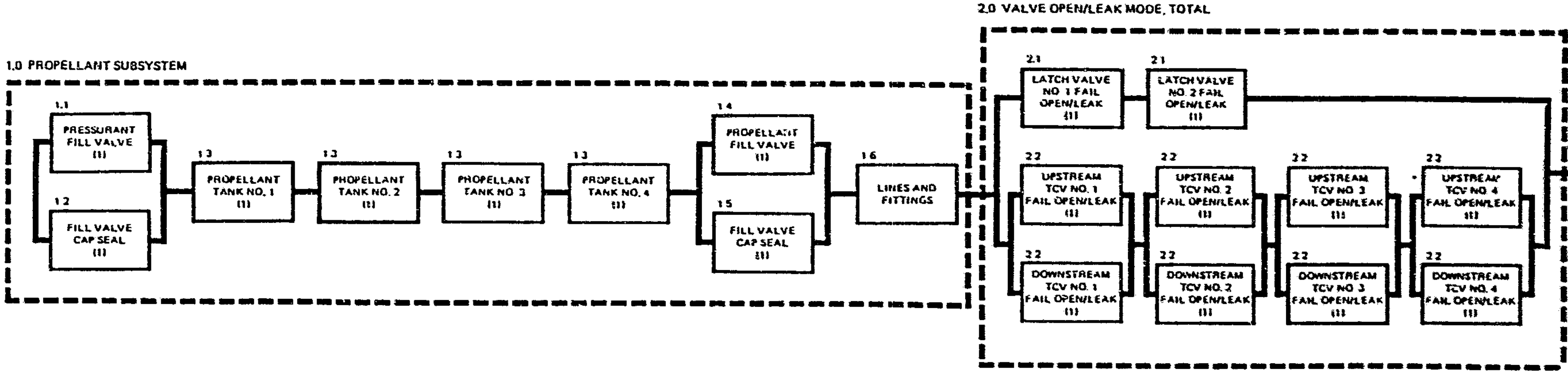
The hardware parts have been functionally grouped for the reliability prediction as shown in the reliability block diagram, figures C-2 and C-3. The reliability block diagram represents a systematic arrangement of functions that must be performed for successful completion of the mission. Redundant functions are shown in parallel and nonredundant functions are shown in series. In general, where more than one part has been grouped within a block, the success of the parts is required for successful operation of the block.

The mathematical equation for the reliability of the thruster has been derived from the block diagram and is shown in figures C-2 and C-3. The resulting reliability prediction provides the following comparison of propulsion system reliabilities:

Case 1 Propulsion System with 4-27 lb_f thrusters, R = 0.9576
Case 2 Propulsion System with 4-0.1 lb_f ACTs, R = 0.9048

For many missions, neither of these system reliability values would be acceptable; a more highly redundant system configuration would be required.

A. BLOCK DIAGRAM



B. MATHEMATICAL MODEL

$$R_{1.0} = \left[R_{1.1} + (1 - R_{1.2})\right] R^4_{1.3} \left[R_{1.4} + (1 - R_{1.4})\right] R_{1.16}$$

$$R_{2.0} = R^2_{2.1} + (1 - R^2_{2.1})\left[2R_{2.2} - R^2_{2.2}\right]^4$$

$$R_{3.0} = 2R_{3.1} - R^2_{3.1}$$

$$R_{0.4} = R^2_{4.1} R_{4.2}$$

$$R_{0.1} = R_{1.0} R_{2.0} R_{3.0} R^4_{4.0} R_{5.0}$$

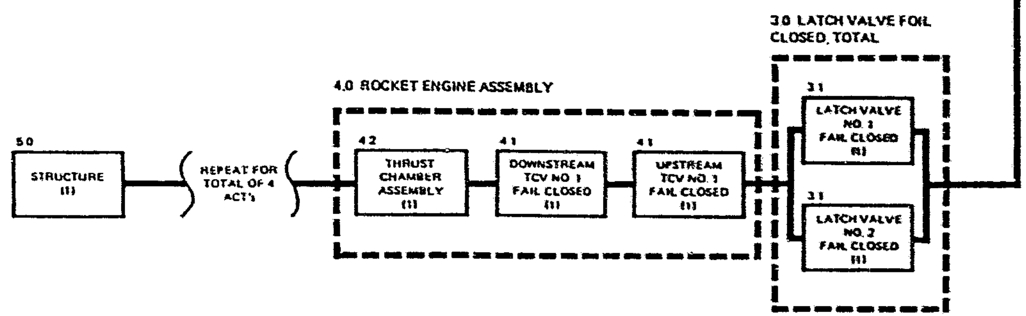


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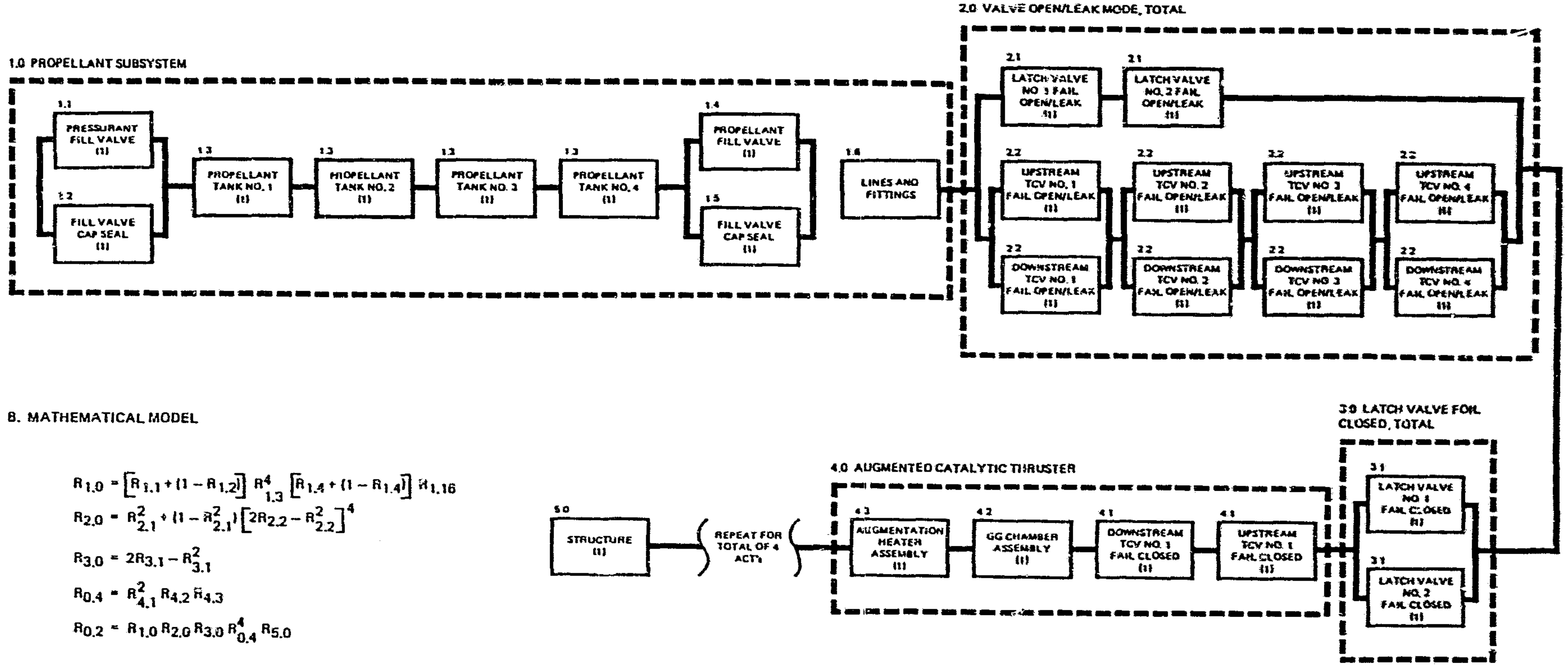
Figure C-2. Block Diagram of an N_2H_4 Monopropellant System

A. BLOCK DIAGRAM



B. MATHEMATICAL MODEL

$$R_{1,0} = \left[R_{1,1} + (1 - R_{1,2})\right] R^4_{1,3} \left[R_{1,4} + (1 - R_{1,4})\right] R_{1,16}$$

$$R_{2,0} = R^2_{2,1} + (1 - R^2_{2,1})\left[2R_{2,2} - R^2_{2,2}\right]^4$$

$$R_{3,0} = 2R_{3,1} - R^2_{3,1}$$

$$R_{0,4} = R^2_{4,1} R_{4,2} R_{4,3}$$

$$R_{0,2} = R_{1,0} R_{2,0} R_{3,0} R^4_{0,4} R_{5,0}$$

Figure C-3. Block Diagram of an N_2H_4 Augmented Catalytic Thruster

Since Space Station thrusters could be replaced on-orbit, the required level of reliability might be lower than for an unmanned mission. Whatever the required level, more reliability would have to be designed into the system configuration for resistojets than for conventional monopropellant engines. Table C-2 summarizes the subsystem reliability. An observation from table C-2 is that the propellant subsystem, which includes tanks and lines is the least reliable subsystem. A large amount of redundancy in tanks and lines must be considered in the Space Station configuration.

Table C-2. System/Subsystem Reliability Comparison of a Conventional and a Resistojet Hydrazine Propulsion System (4 tanks, 2 latch valves, 4 thrusters)

Subsystem	27-lbf Conventional	0.1-lbf ACT
1. Propellant	.9586678	.9586678
2. Valve leak/open mode	.9999999	.9999999
3. Latch valve fail closed	.9999999	.9999999
4. Engine failure	.9997465	.9856529
5. Total system	.9576922	.9048200

Appendix D

GO_2/GH_2 CONVERSION ANALYSIS

D.1 Introduction

The candidate H_2/O_2 sources to be discussed are: (1) stored liquid at the station which, through heat addition, is transformed from LO_2/LH_2 into GO_2/GH_2; (2) boiloff from the OTV storage facility; and (3) scavenging the External Tank and Orbiter for LO_2/LH_2.

There are major LO_2/LH_2 resupply and storage concerns for the Space Station as listed below:

- venting in zero gravity
- boiloff
- propellant acquisition and transfer in zero gravity
- Orbiter modifications to carry cryogenics
- complexity of the system
- safety

The resupply and storage issues will be addressed by the Cryogenic Fluid Management Facility (CFMF), which is currently being developed by the Martin-Marietta Co. for NASA/LeRC, and is scheduled to be tested onboard the Space Shuttle in 1987-88. Assuming that these and other concerns will be resolved and since the OTV will probably use LO_2/LH_2 from a storage source onboard the Space Station, it is then the purpose of the following sections to examine several candidate GO_2/GH_2 sources.

D.2 LO_2/LH_2 to GO_2/GH_2 Using a Gas Generator, Turbopump Assembly, and Heat Exchanger

The first most detailed example is of a propulsion system that utilizes a gas generator, turbopump assembly, and heat exchanger to convert LO_2/LH_2 to

GO_2/GH_2.

It must be noted that the following weights and performance values used in this scenario are the result of an analysis performed by Aerojet[1], with some modifications by BAC as part of the current study.

Figure D-1 shows a simplified schematic of one possible system configuration. The major components of this system consist of:

o LO_2 Resupply Tank	o LH_2 Resupply Tank
o GO_2 Accumulator	o GH_2 Accumulator
o Gas Generator	o Turbopump Assembly
o Heat Exchanger	o GO_2/GH_2 Thrusters
o Valves, Regulators, Transducers, etc.	
o H_2 Resistojets*	

The gas generator (G.G.) weighs 10 lbm and consumes 18.6% of the total nominal engine flow rate (8.27% of LO_2 and 10.33% of LH_2). The G.G. operates at a mixture ratio of 0.8:1, a chamber pressure of 80 psia, and produces GH_2 and superheated steam at a temperature of 1500°R. The turbopump assembly (T.P.A.) weighs 3.0 lbm and requires 1.1 horsepower to pump the propellants from 100 to 500 psia. The heat exchanger (H.E.) weighs 11.3 lbm and raises the temperature of the propellants to 530°R. The combined weight of the valving, regulators, and transducers is 47.5 lbm.

The operational procedure of this system is to store LO_2 and LH_2 in separate tanks. Then, by bleeding of 18.6% of the total nominal engine

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1. Appel, M.A., JPL; Schoenman, L., Berkman, D. K., Aerojet; "Oxygen/Hydrogen Thrusters for the Space Station Auxiliary Propulsion System"

* The H_2 resistojets could be used either as an alternative to GO_2/GH_2 thrusters for orbit maintenance and/or used to satisfy low thrust requirements (i.e., attitude control backup).

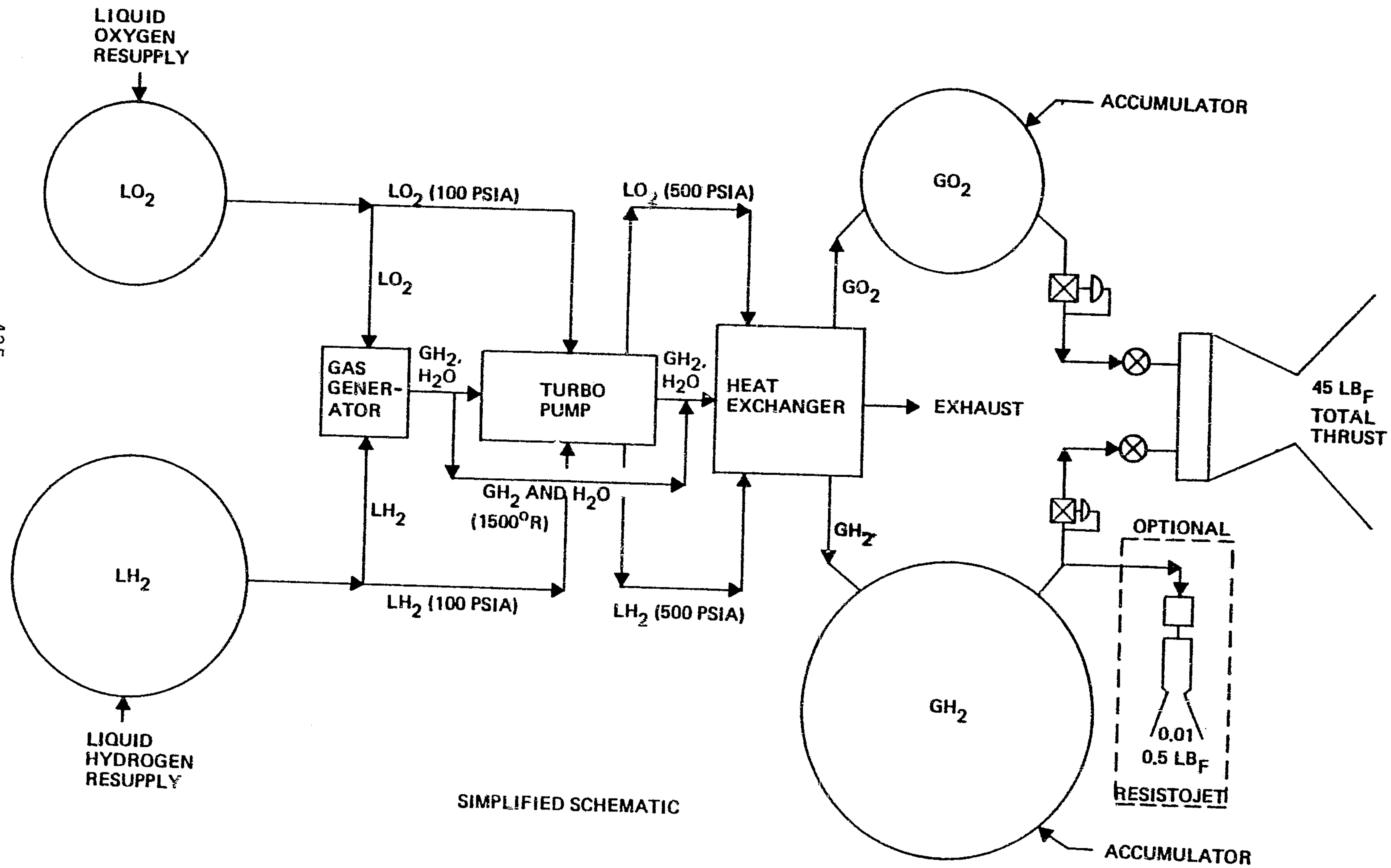

Figure D-1. GO_2/GH_2 Propulsion System With LO_2/LH_2 Resupply and Storage flowrate, the G.G. produces H_2/O_2 combustion products at $1500^{\circ}R$. Two percent of the G.G. gas flow is sent into the T.P.A. and is used to spin two small turbines which drive the LO_2 and LH_2 pumps. Within the T.P.A., the propellants are pumped from 100 to 500 psia and then passed into the H.E. The other 16.6% of the G.G. gas flow bypasses the T.P.A. and enters the H.E. directly. The exhaust from the T.P.A. also is used in the H.E., however, the temperature of the GH_2 after leaving the T.P.A. is only $1200^{\circ}R$. As the propellant is run through the heat exchanger, its temperature is raised to $530^{\circ}R$. The exhaust from the H.E. is then dumped overboard or possibly used as propellant. The propellants are then stored in accumulator tanks.

In order to size this system, the following assumptions have been used:

- o neutral atmosphere
- o altitude = 525 km
- o $C_D = 2.2$
- o propellant resupply once every 90 days
- o 2- to 4-man station
- o total thrust = 45 lb_f
- o thrust duration = 600 secs per thrust
- o duty cycle = one thrust every 10 days

It is apparent that, in sizing this system, the accumulator tanks, storage tanks, and propellant requirements are functions of mixture ratio.

Figure 6-28 shows the ideal specific impulse for a chamber pressure of 50 psia and an expansion ratio of 200:1. This I_{sp} was used to determine the flowrate for the propellant requirements shown in figure D-2. The total amount of propellant required by the entire sytem, which includes the G.G., T.P.A., H.E., and thrusters is shown in figure D-2. The increase in the amount of LO_2 required as thruster mixture ratio increases is fairly linear due to the combination of a low mixture ratio (0.8:1) required for the G.G. and the large quantity of propellant (18.6%) that is required to run the T.P.A. and H.E. Figure D-3 shows the amount of propellant consumed in order to run the G.G., T.P.A., and H.E. The effects of thruster mixture

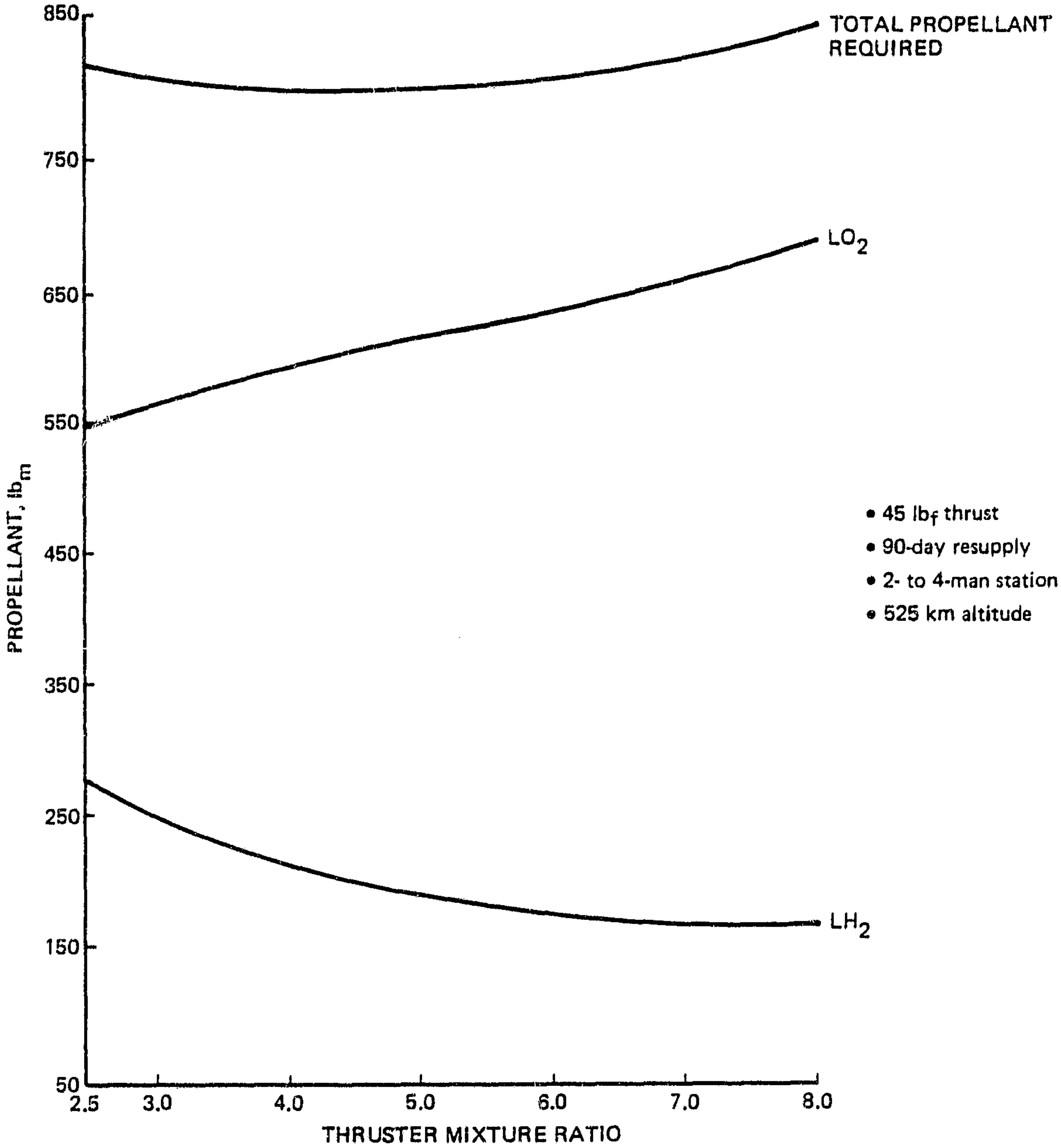

Figure D-2. Required Propellant for O_2/H_2 System

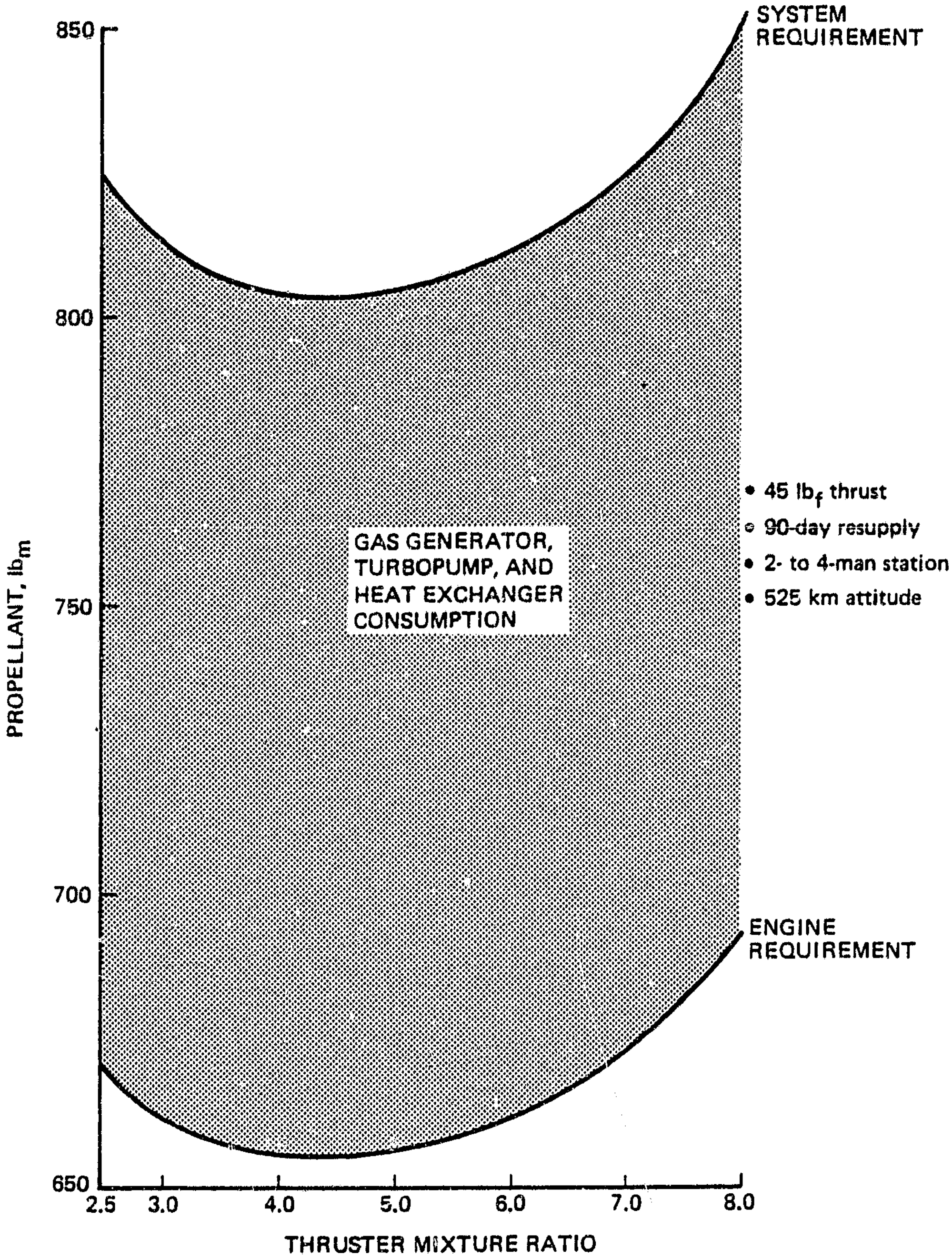

Figure D-3. Gas Generator, Turbopump, and Heat Exchanger Propellant Consumption ratio on storage and accumulator tank sizing can be seen in figure D-4. All of the tanks include a 50% propellant volume contingency and the storage tanks include a 10% ullage.

This analysis assumes the baseline altitude and thrusting strategy developed during the study. Obviously, a lower altitude would increase the propellant and storage tank requirements will be larger; also, if a different thrusting strategy is chosen, then the accumulator tanks may be larger or smaller.

D.3 H_2/O_2 Propellant from OTV Storage Facility Boiloff

As discussed in section 6.1.5, it is envisioned that a LH_2/LO_2-propelled OTV will be based at the Space Station by 1995. Therefore, LO_2/LH_2 will require storage onboard the Space Station. If the LH_2 and LO_2 are both allowed to boiloff freely, their vapor can be stored in accumulator tanks and then used in GO_2/GH_2 thrusters for propulsive requirements. The amount of vapor required depends on the mixture ratio, thrusting strategy, altitude, and station size. Using the assumptions from the previous analysis (D.2), the propellant requirements are as shown in figure D-3. Concern associated with this system is the pressure of the propellants. If a thermodynamic vent is used, which enables the tank insulation penetrations to be cooled by the propellant vapor, then the pressure of the vented gas is less than 1 psia. This gas would then have to be pumped up to a reasonable pressure so that it could be stored in accumulators and then used in the thrusters. This requires a compressor and thus energy. A storage tank thermal analysis is required to determine the amount of propellant boiloff, which is beyond the scope of this study.

D.4 H_2/O_2 Propellant from ET and Orbiter Scavenging

Scavenging propellants from the external tanks (ET) and the Orbiter has received a lot of attention and probably even more skepticism. Though there are many problems associated with this idea, it may nevertheless be a viable alternative.

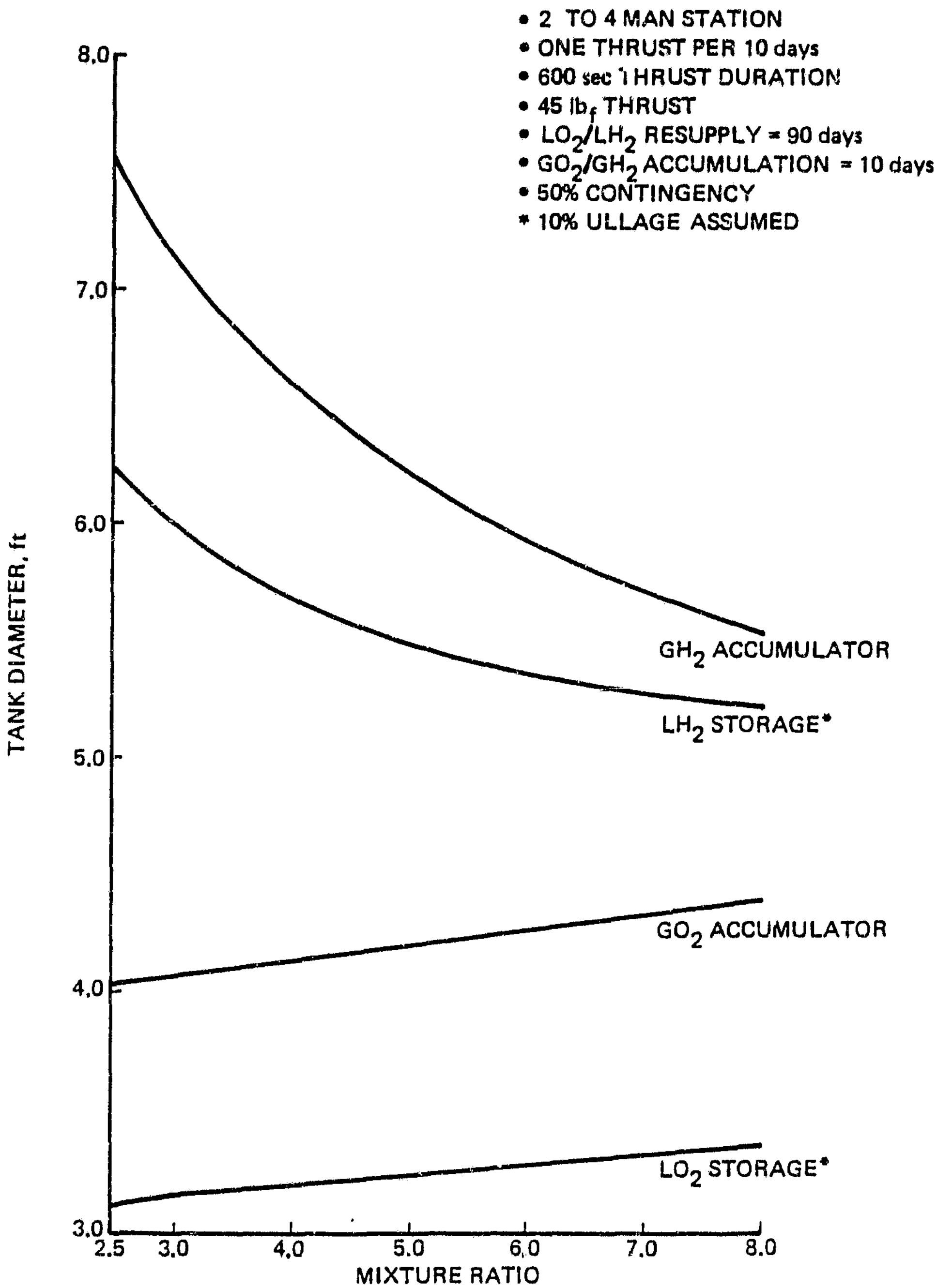

Figure D-4. Storage and Accumulator Tank Sizing

Table D-1 shows the propellant quantities transferable after main engine cutoff of the Orbiter. A total of 3363 lbm of LO_2 and 1873 lbm of LH_2 can be transferred assuming that the ET ullage pressure remains at 20.5 psia and 33 psia for the LO_2 and LH_2 tanks, respectively, and that propellant settling is accomplished by spinning the Orbiter/ET combination at a rate of 0.5 deg/sec. This spin rate is within limits currently established for the STS.

Propellant obtained from the ET and Orbiter must be converted into GO_2/GH_2. This can be accomplished actively using turbopumps and heat exchangers or allowing propellant to boil off. Major problems associated with obtaining propellant through the ET and Orbiter scavenging are cost, complexity, and safety. Even though the propellant itself may be "free", the procedure in which it is obtained is quite expensive. For example, liquid acquisition equipment must be developed and will probably be located to the extent possible on the Space Station to minimize STS weight. The technology to accomplish the acquisition must be developed and will drive up the cost of such a system. Therefore, even though scavenging may seem at first glance to be a cheap way to obtain fuel, in practice it will be expensive.

D.5 Supercritical H_2/O_2

Storage of H_2/O_2 as a supercritical fluid avoids the acquisition, venting, and gaging problems associated with subcritical storage. This subject was discussed in some detail in section 6.3.4.3. Conversion of the supercritical fluid to a gas requires only that sufficient heat be added to maintain the temperature above the critical value during any subsequent pressure drop.

D.6 Water Electrolysis

GO_2/GH_2 can be obtained directly by electrolyzing water. This approach was discussed in some detail in Appendix B. The advantages from both the utility and safety standpoints associated with storage as water as opposed to LO_2/LH_2 until needed are apparent (and may be found in Appendix B).

Table D-1. Post-Main Engine Cutoff Reserve Propellant Transfer

Item	LO_2 (lbm)	LH_2 (lbm)	Total (lbm)
Orbiter reserves	3304	249	3553
ET reserves	1413	2374	3787
Total Reserves	4717	2623	7340
Orbiter trapped	1200	100	1300
ET trapped	154	650	804
Total trapped	1354	750	2104
Orbiter transferrable	2104	149	2253
ET transferrable	1259	1724	2983
Total transferrable	3363	1873	5236

D.7 Summary

A number of candidate GO_2/GH_2 propellant sources for the Space Station propulsion system has been discussed. The relative value of each of the candidates depends on technology advancements and the necessity for the sources for other purposes. Existing technology favors either water electrolysis or supercritical storage. The use of LO_2/LH_2 from any of the sources discussed requires advancing the state-of-the-art for venting, acquisition, and gaging in a low-g environment.

APPENDIX E

ION SYSTEMS

E.1 Introduction

Ion thrusters are characterized by a specific impulse that is commonly an order of magnitude higher than chemical systems (and, in fact, is theoretically limited only by relativistic effects). These thrusters also have high power requirements, on the order of 250 kW per lb_f of thrust. An extensive ion thruster review is presented in Boeing Document D180-26680-2, "Advanced Propulsion Systems Concepts for Orbital Transfer."[1] The following is a summary of the major features of this type of system.

The essential subsystems of this concept are: (1) an array of photovoltaic solar cells, or other electrical energy source, sized to provide the required power; (2) a power processing unit (PPU) that converts source power into the power forms required to start and operate the thruster, and (3) an ion thruster that electrostatically accelerates propellant ions to produce thrust. This source-processor-thruster relationship is illustrated in figure E-1.

The salient feature of this concept is that the specific impulse can be prescribed by the system designer since it is determined by the design voltage used to accelerate the ions. Hence, the designer can optimize I_{sp} to minimize cost or maximize payload. One single-thruster design, a 30-cm mercury thruster, has demonstrated I_{sp} values ranging from 1000 sec to 5000 sec. Specific impulse must be prespecified to optimize some mission parameters, such as recurring costs. This is accomplished by specifying the acceleration voltage corresponding to the required I_{sp}. (In practice,

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1 Boeing Aerospace Company, "Advanced Propulsion Systems Concept for Orbital Transfer," D180-26680-2.

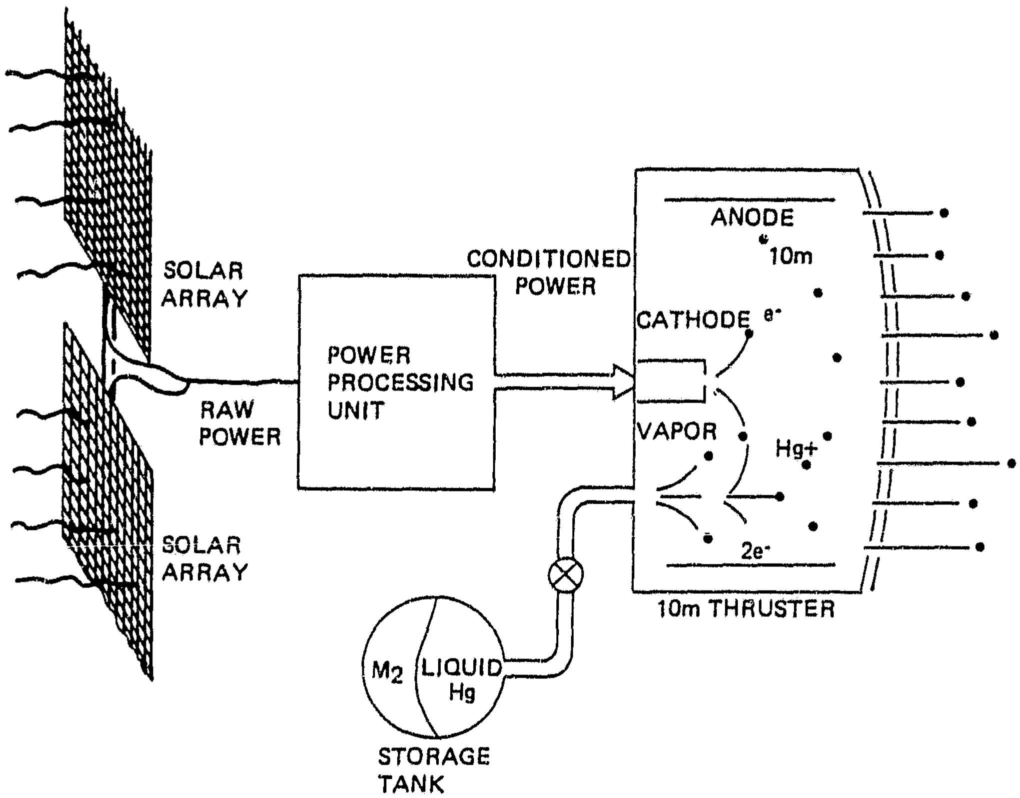

Figure E.1. Solar Electrical Propulsion System Schematic this is done by designing the PPU to develop the required accelerating voltage.) Since, for constant beam current, the required propulsion power tends to be a function of the square of I_{sp}, the maximum achievable I_{sp} is almost never selected. Instead, some comparatively low I_{sp} is usually best, generally in the range of 2000 to 6000 sec, depending on the mission, subsystem masses and efficiencies, and the propellant selected.

An ion thruster is illustrated in figure E-2. The hollow cathode produces discharge electrons. When these electrons enter the ionization (discharge) chamber, they produce ions by bombarding atoms in the diluted propellant gas. Doubly charged ions are also produced. Approximately a ten percent fraction of all ions recombine on the chamber walls, which contributes a loss of thruster input power. Those ions that enter the holes in the optics (screen grid and accelerator grid) are electrostatically accelerated to produce thrust. Neutral atoms may also pass through these holes but they do not produce thrust. Except for a slight divergence, the acceleration of ions is without loss and their velocities are proportional to the square root of the accelerating voltage. Ions that recombine on the screen grid structure have sufficient energy to erode its upstream face. This process generally limits the thruster lifetime.

The propellants commonly proposed for ion thrusters are mercury, xenon, and argon. Mercury thrusters have received considerable attention in the literature but were eliminated from this study because of contamination from the mercury ion plume. Argon and xenon thrusters produce the least contaminating exhaust plume of the propulsion systems examined. Figure E-3 illustrates the thrust attainable and power requirements for an argon thruster. Note also that efficiency increases with specific impulse.

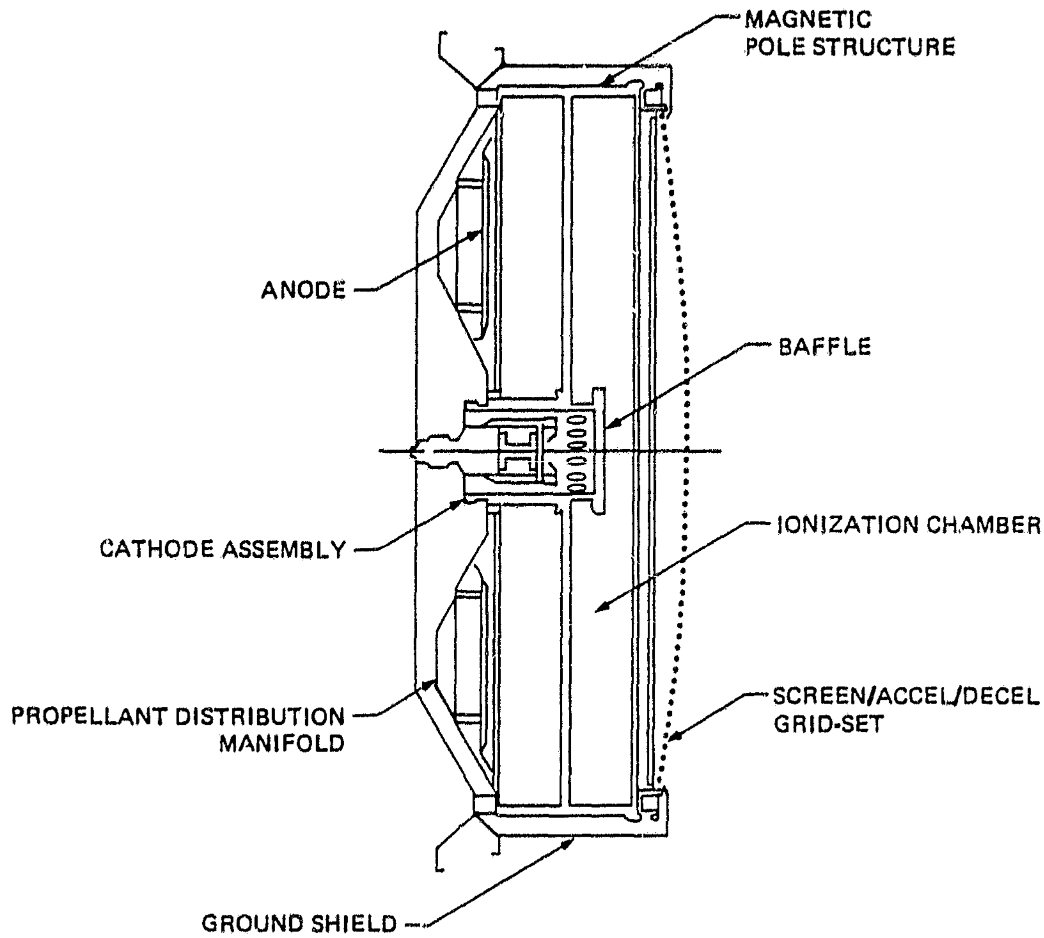

Figure E.2. Ion Thruster

Ion thrusters are also characterized by low thrust (below 0.1 lb_f) which has been demonstrated by thrusters using mercury. Due to this low thrust level, these thrusters must operate all or most of the time for orbit maintenance. Table E-1 shows demonstrated approximate lifetimes of various ion thrusters.

Table E-1 Ion Thruster Demonstrated Lifetimes

Thruster	Lifetime (hr)
30 cm	10,000
(Grid)	(>30,000)
8 cm	15,000
IAPS	8,000
5 cm	9,700

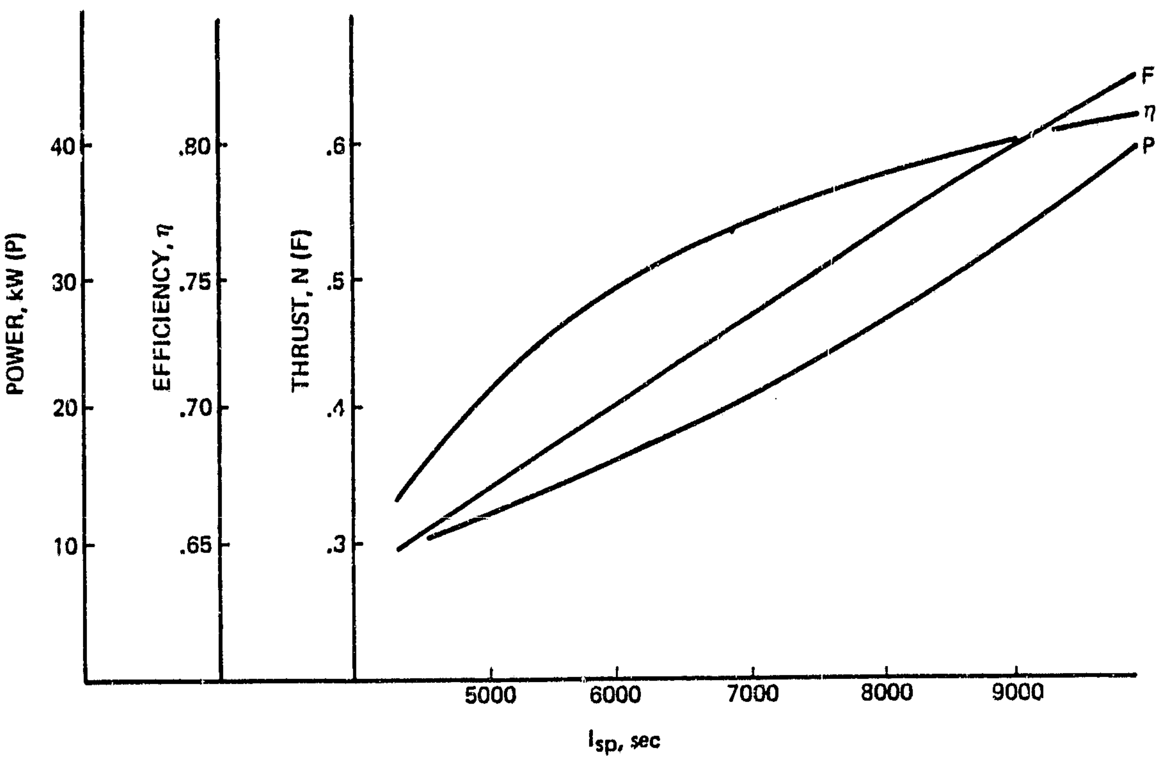

Figure E.3. 50-cm Argon Ion Thruster Performance

APPENDIX F

ARCJET THRUSTERS

F.1 Introduction

As stated before, most of the work that has been developed on arcjets occurred in the early half of the 1960's. Avco and Plasmadyne are two of the leading corporations that have done research work on arcjets. Some general comments about the design, operation, failure modes, and performance limitations, are discussed below.[1]

F.2 Performance Characteristics

Most of the experimental work done with arcjets has been at a 30-kW power level and has used hydrogen as the working fluid. The experience base at low powers (<5 kW) is minimal and the work completed has indicated problems. Ammonia has also received some life-testing at 30 kW.

Tables F-1 through F-3 illustrate the performance regimes of arcjets using various propellants at different operating conditions. Table F-1 shows performance data for a 1-kW arcjet that was tested for short durations with a variety of propellants.[2] By ratioing the molecular weights and specific impulse performance of N_2 and CO_2, the data given in table F-1 indicates that a 250-300 lb_f-sec/lbm specific impulse can be obtained from CO_2 arcjets. The lifetime of arcjets using CO_2 may be limited due to the formation of free oxygen and the potential for oxidation. This potential failure mode is analyzed in section 6.4. The incomplete data in table F-1 for argon and lithium hydride was due to incomplete measurements caused

1 Wallner, L. E., Czilea, J., "Arc-Jet Thruster for Space Propulsion," NASA TN D-2868, 1968.

2 Shepard, C. E., and Watson, V. R., "Performance of a Constructed-Arc Discharge in a Supersonic Nozzle," Paper No. 63-380, AIAA, 1963.

Table F-1. 1 kW Arcjet Performance

Propellant	Input Power, W	Weight Flow rate, lbm/sec	Voltage, V	Current, A	Thrust, lb_f	Specific impulse, lb_f-s/lbm	Efficiency, percent	Test Time, hr	Life Expectancy, hr
Ammonia	920	1.10×10^5	59	13	0.006	550	8	2	1
Nitrogen	550	2.75	40	13	.009	350	13	2	1
Helium	370	1.00	30	12	.005	550	17	2	1
Argon	130–300	2.00–5.70	18–20	9.5–15	——	80	3	Short	Short
Lithium	700	———	70	10	——	400–800	8–12	Short	Short

*Some of the input power was lost to resistance in the circuit. Therefore, $V_{arc} \times A_{arc}$ may not equal the input power.

Table F-2. Operating Envelopes for Ammonia and Hydrogen at 30 kW

Mass flow rate, g/sec (lbm/sec)	Current, A	Voltage, V	Thrust, lb_f	Specific impulse, lb_f-s/lbm	Efficiency, percent
		Hydrogen			
0.244 (.00054)	186	161	.54	1008	41.2
.230 (.00051)	191	158	.53	1042	39.7
.218 (.00048)	192	155	.51	1063	39.7
.204 (.00045)	200	152	.49	1096	38.7
.190 (.00042)	293	148	.47	1131	38.8
.177 (.00039)	210	145	.46	1174	38.7
.164 (.00036)	210	143	.44	1232	39.8
.150 (.00033)	222	138	.42	1280	38.5
		Ammonia			
0.50 (.00110)	250	122	.66	597	28.1
.45 (.00100)	255	118	.63	640	29.4
.40 (.00090)	260	116	.61	692	30.6
.38 (.00084)	260	115	.60	715	31.3
.36 (.00080)	262	115	.59	742	31.8
.34 (.00075)	265	114	.58	780	33.0
.32 (.00070)	270	112	.57	819	34.1
.30 (.00066)	270	112	.57	866	35.8
.28 (.00062)	270	111	.56	910	38.8
.26 (.00057)	275	110	[illegible]	970	38.8
.24 (.000[illegible]3)	288	106		1012	38.6

Table F-3. H_2 Arcjet Performance Summary

Power, kW	Specific impulse, $lb_f\text{-}s/lb_m$	Thrust, lb_f	Efficiency, percent	Test time	Type of test	Comments
1	1100	.01	35	25 hr	Continuous	Average over first 12 hr; decayed rapidly afterwards
2	935	.03	30	150 hr	Continuous	
30	1010	.55	41	720 hr	Interrupted	Terminated voluntarily; thruster in excellent condition
30	1010	.74	54	500 hr	Continuous	
30	1520	.46	44	250 hr	Continuous	
30*	1020	.51	38	250 hr	Interrupted	Performance was not verified throughout test

*Alternating current.

from extremely low thrust levels and lifetimes. Due to the overall variances and difficulties in running a 1-kW arcjet for long periods of time, the results shown in table F-1 cannot be considered conclusive.

Table F-2 shows the operating conditions for a 30-kW arcjet using hydrogen or ammonia.[3] Generally, the impulse for ammonia is not much lower than hydrogen at the same thrust level (i.e., efficiencies are similar), and I_{sp}'s of 900-1000 lb_f-sec/lbm for ammonia are achievable. Table F-2 also shows the strong dependence of performance on the operating conditions selected. Efficiency, in this table, is defined as the ratio of electric power to thrust power.

Table F-3 summarizes the results of the best performance achieved using hydrogen as a working fluid for various operating conditions. It is clear from this table that arcjets were at a fairly advanced stage in the early 1960's, at least for 30-kW thrusters, and that space-qualified thrusters at this power level were not far off. Performance at lower power levels was less established, but development problems at the 2-kW level appeared tractable.

F.3 Hardware Physical Characteristics

Fundamentally, a thermal arcjet design must provide two insulated conductors for the arc electrodes and a gas passage from inlet to exhaust. Because most arcjet designs are of the constricted-arc type, the discussion is based on a typical radiation-cooled design, which is shown in figure F-2. The cathode is a tungsten rod with a conical tip. A tungsten anode, which serves as the nozzle, is placed slightly downstream of the cathode. The constrictor is formed by a narrow horizontal portion of the nozzle. Surrounding the cathode is a boron nitride insulator. The outer body is made from molybdenum or tungsten. The back of the assembly is generally boron nitride, again for insulation purposes. Current is conducted into

3 John, R. R., Thirty-Kilowatt Plasmajet Rocket-Engine Development, Rept. No. RAD SR-62-182, Avco. Corp., Sept. 11, 1962.

the cathode mounting from the power supply. Propellant is led into the plenum by a passage through the upstream portion of the device and entered the arc cathode attachment region tangentially. Some regenerative cooling passages in the nozzle are also used in some designs as part of the gas inlet.

The basis for certain aspects of the design are clear. For a given input power, a constrictor provides a larger arc than would be possible in the unconstricted case, because the former has a lower current. The cathode is the cone-tipped rod, and the anode is a massive nozzle that provides a sink for the high heat load it must sustain. Furthermore, the cone tip provides a high field region for thermionic emission of electrons. The flow enters the constrictor nearly tangentially in order to stabilize the discharge by creating a vortex that forces the cooler, heavier gas to the outer regions. It is questionable whether this vortex is really needed because the data shows no significant change with the vortex, at least in engines with 30 kW of power or more.[1,2] In a smaller size engine (1 to 3 kW), the vortex has some effect, but phenomena in this device cannot be isolated and the role of the vortex cannot be correctly assessed.

Most of the engine failures are the result of (1) nozzle cracks from thermal shocks, (2) arc anode attachment off the design point due to power fluctuations, or (3) leakage of critical engine joints. These effects are often traced to deterioration by heat loads; therefore, the basic design must be considered from a heat-transfer aspect. This comes largely from experience and the specific design, but two general rules can be set down. The hottest part of an engine is the downstream end of the nozzle.[1,2] To provide sufficient cooling by radiation, adequate surface area must be provided. Also, interfaces at high heat flux regions cause detrimental hot

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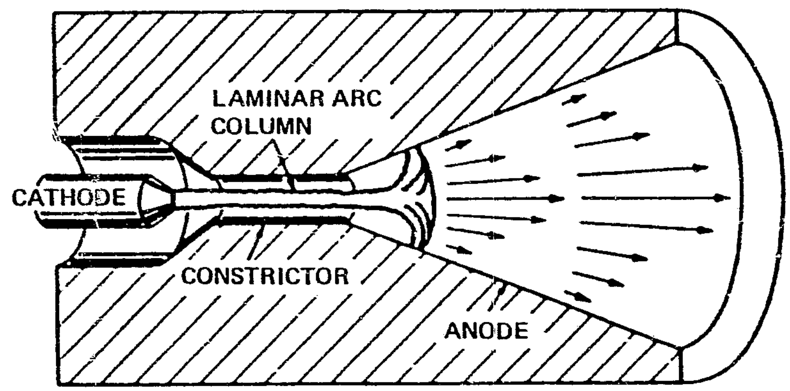
1 Cann, G. L., Moore, R. A., Buhler, R. D., and Marlotte, G. L., Thermal Arc Jet Research. Rept. No. ASD-TDR-63-632, Electro-Optical Systems, Inc., Aug. 15, 1963.

2 Anon., Thirty-Kilowatt Plasamajet Rocket-Engine Development. Third Year Development Program, QPR-2, Rept. No. RAD SR-63-244, Avco Corp., Dec. 1963.



GAS IS HEATED PRIMARILY BY JOULE HEATING IN ARC COLUMN

$$\sigma E = \quad \sigma E^2 = \mathrm{DIV}\ Q$$

"COLD" FLOW AWAY FROM ARC ALLOWS WALL TEMERATURE TO REMAIN CONSTRAINED WITH GAS IN ARC AT ELEVATAED TEMPERATURE ($> 30{,}000^{\circ}K$)

ARC STABILITY CRITICAL TO PERFORMANCE AND LIFETIME

Figure F-1. Arcjet Fundamentals

spots. For this reason, the joint of the anode and the outer body are as far upstream from the nozzle end and the constrictor as possible.

F.4 Propellant Quantities and Tank Sizing

For the purpose of comparison, hydrogen, ammonia, and nitrogen are considered to be storable as liquids. Table F-4 shows the quantities of H_2, NH_3, and N_2, for a 1-kW arcjet, which would satisfy drag makeup requirements for a 90-day period (assuming the Orbiter is docked for 14 of those days). These drag data assume a NASA-neutral atmosphere at an altitude of 525 km, an Earth-oriented 28.5 deg inclination, and 2- to 4-man station.

F.5 Exhaust Constituents

No plume diagnostics have been performed by any companies or government agencies. However, a few general theoretical statements can be made on this subject. In comparing the exhaust plume of an arcjet with a resistojet, it is assumed that they are similar for similar propellants. Inside the thruster of the arcjet there is probably greater dissociation due to the larger amount of energy input. Complete combination probably occurs slightly downstream of the thrust. Therefore, it is assumed that the arcjet plume is neutral. The only other variation in the exhaust plume from that of a resistojet, is that the exhaust temperature is a little higher, again due to the greater amount of energy input.

F.6 Potential Throttling and Installation Penalties

Unlike other thrusters, arcjets are not easily throttled. To throttle an arcjet, either the power, mass flow rate, or both must be varied. The

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1 John R., Thirty-Kilowatt Plasmajet Rocket Engine Development. Rept. No. RAD-SR-61-182, Avco Corp., Nov. 29, 1961.

2 McCaughey, O. J., Development of a Plasmajet rocket Engine for Attitude and Orbit Control. Rept. No. FR-122-651, Plasmadyne Corp., June 1964.

Table F-4. Arcjet Propellant Requirements for a 90-Day Period

Propellant	H_2	NH_3	N_2
Specific impulse, sec*	1100	550	350
Total required, lbm**	117	234	367
Volume, ft^{2+}	30	7	8
Tank material	AL 2219	Ti 64V-4V	AL 2219
Tank weight, lbm ++	22	19	7
Tank pressure, psia	20	150	20
Tank temperature, F	-425	75	-320

* 1 kW arcjet performance

**Includes 50% contingency (i.e., attitude control backup, docking disturbances, etc.)

\+ Includes 10% ullage for liquid propellants only

++ Tank weight assembly = $K_{T+A}\ K_{L/D}\ 1.5\ (\rho/F_{Tu})\ (v)\ (UFS \times P_{max\ oper})$

throttling is more difficult than for conventional thrusters because the inter-related relationship of power and mass flow for a given electrode gap, the electric potential between two electrodes, is very sensitive. Exactly how sensitive is still unknown. Although it may be easy to vary the mass flow rate, it is still not clear what the impact is of varying say a 2-kW arcjet to 1 kW or 5 kW. It is assumed that throttling can have a significant impact on thruster life and performance. The only penalties incurred in a gimballed or removable installation would most likely be the weight of the additional hardware.

F.7 Projected Arcjet Capability Assessment

Since most of the developmental work on arcjets occurred from 1960 to 1965, state-of-the-art projections are necessarily based on previous literature, theoretical concerns, and simple gas dynamic performance models. Table F-3 in the previous section showed the SOA status of H_2 arcjets. In this section, the theoretical models are compared to tests that were done in the 1960's (primarily on H_2). A simple gas dynamic model that included assumptions about arc temperature was used to project the performance of thrusters of 50-mlb_f to 500-mlb_f thrust. Nozzle efficiency losses and throat diameters were also estimated.

F.8 Arc/Gas Heating Models

There are currently two proposed models for arcjet heating processes: the core-flow theory,[1,2] and the Stine-Watson or volumetric-heat-addition theory.[3] Each theory seems to be applicable depending on the regime of

1 John, R. R., et al, "Theoretical and Experimental Invesgitation of Arc Plasma-Generation Technology." Report No. ASD-TDR-62-729, Pts. I-II, Vols. 1-2, Avco Corp., 1964.

2 John, R. R., "Thirty-Kilowatt Plasmajet Rocket Engine Development." Report No. RAD TR-64-6, Avco Corp., July 15, 1964.

3 Stine, Howard A., and Watson, Velvin R., "The Theoretical Enthalpy Distribution of Air in Steady Flow Along the Axis of a Direct-Current Electric Arc," NASA TN D-1331, 1962.

operation. Both of these models are described briefly in this section. The core-flow model assumes a positive column, small in area relative to the throat area, which :: essentially an extremely hot hydrogen "wire." Figures F-2 and F-3 illustrate the mechanisms of heat transfer and the flow regions for the core-flow model. The basic energy equation, after the axial conduction and convection terms are neglected, is reduced to:

$$\sigma(T)\, E^2 = P_{rad}\,(T) - \frac{1}{r}\,\frac{d}{r}\,\frac{(rds)}{dr}$$

where:

$\sigma(T)$ = electrical conductivity

E = electric field strength

P_{rad} = radiated power in the arc

r = radial position

s = entropy

The core-flow theory is weakened by the omission of the axial enthalpy flux, but it accurately predicts an increase in the gas power and thrust density toward the center of the flow. The theory also accurately predicts the temperature variation with nozzle distance, as shown in Figure F-4.[1]

In the Stine-Watson model, the arc is assumed to fill a relatively large part of the constrictor. Essentially all the gas passes through the arc, as shown in Figure F-5. The energy equation, including the axial convection term on enthalpy gradient, is:

$$\rho V_Z\, \frac{h}{Z} = \sigma E^2 + \frac{\delta^2 S}{\delta_r 2} + \frac{1}{r}\,\frac{\delta S}{\delta r} + \frac{\delta^2 S}{\delta_Z 2}$$

1 John, R. R., "Thirty-Kilowatt Plasmajet Rocket Engine Development." Report No. RAD-SR-61-182, Avco Corp., November 29, 1961.

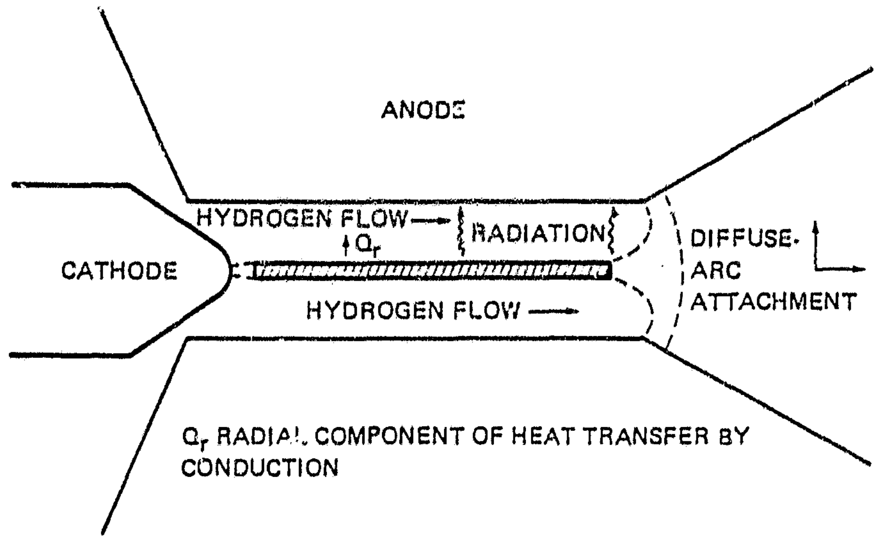


Figure F-2. Heat Transfer Mechanism in the Core Flow Model

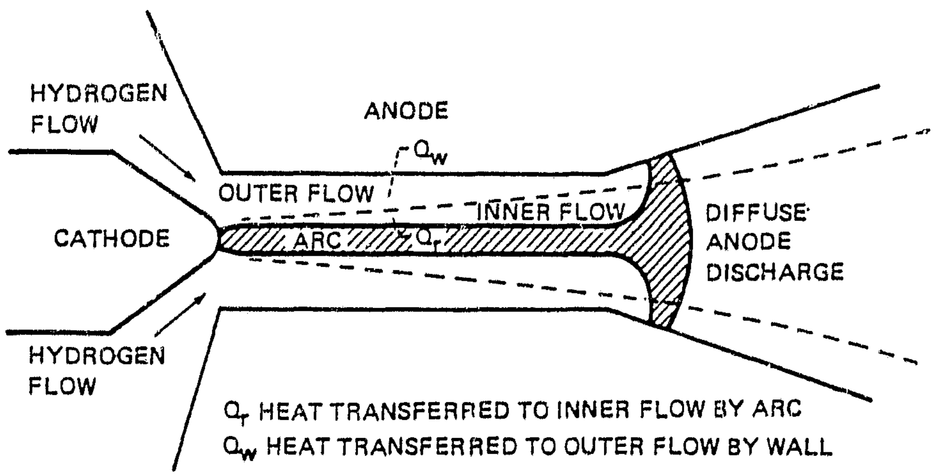


Figure F-3. Flow and Energy Regions in the Core Flow Model

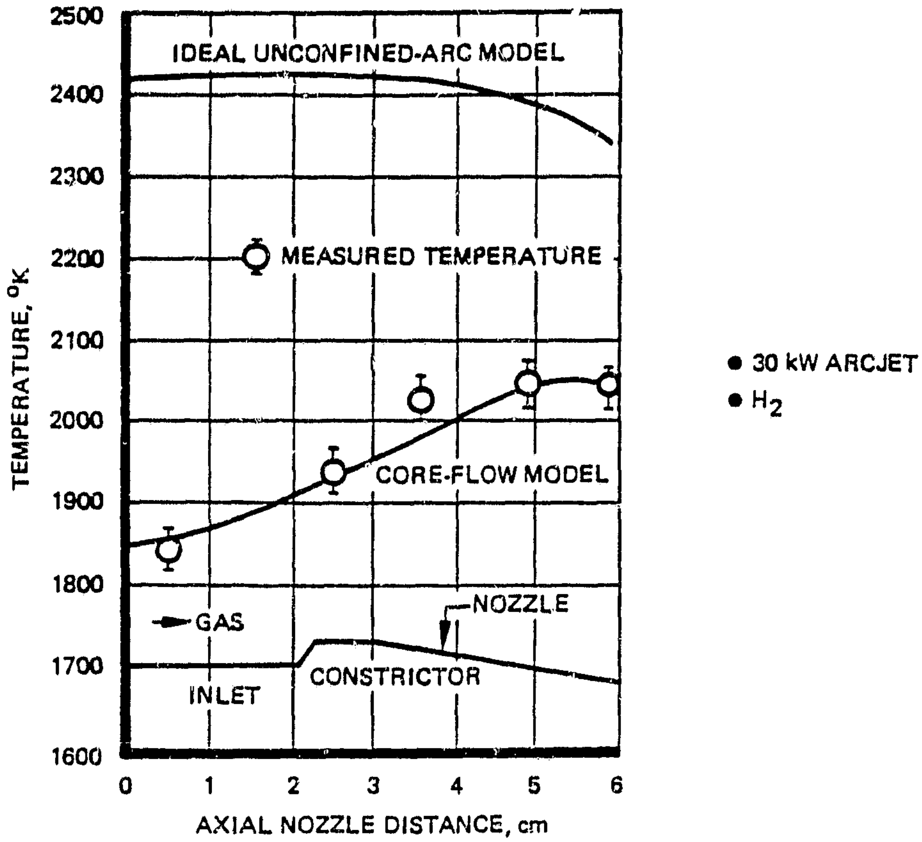
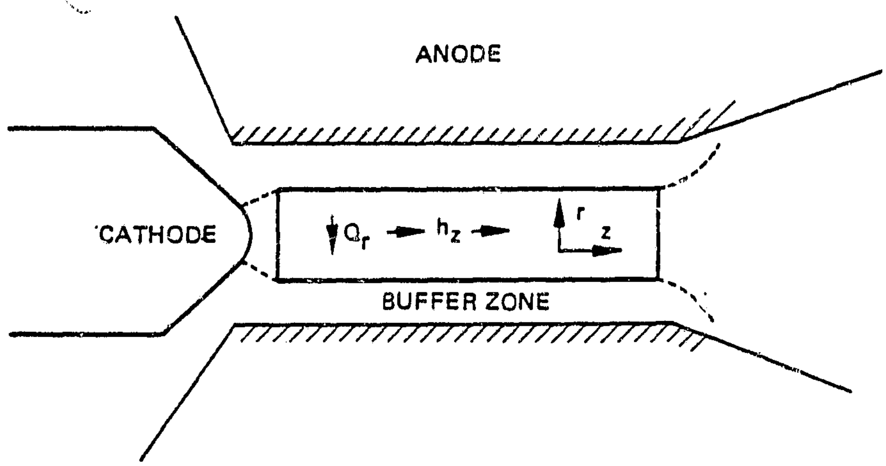


Figure F-4. Temperature Variation Predicted by the Core Flow Model



h_z AXIAL COMPONENT OF HEAT TRANSFER BY CONVECTION

Q_r RADIAL COMPONENT OF HEAT TRANSFER BY CONDUCTION

Figure F-5. Heat Transfer Mechanisms in the Stine-Watson Theory

The axial conduction term ($\delta^2 S/\delta z^2$) can be neglected because of its small contribution.

The comparative propulsive data that has been generated indicate good arc efficiency and enthalpy variation for N_2 and H_2 arcjets, as shown in figures F-6 and F-7.[2] Because the Stine-Watson model demonstrated accurate performance calculations, this large arc region was used in the performance extrapolation that follows.

F.9 Performance Extrapolation

Using a simplified gas dynamic model, the nozzle effic[illegible]ency, thrust level, specific impulse, and throat diameter were predi[illegible]. The theoretical performance data for cold-gas and hydrazine thrusters (presented here for arcjets) were generated using a computer program developed for this purpose. The program is based on the gas dynamics research of Shapiro[1] and on measurements made by NASA/LeRC of the effect of low Reynolds numbers on nozzle thrust coefficients.

The calculations include chamber gas conditions (temperature, pressure, specific-heat ratio, and molecular weight), nozzle area ratio and divergence half-angle, and desired thrust level. Other parameters required are calculated by the program. An iterative technique based on Newton's method is used to solve the following expression for exit Mach number:

$$\frac{Ae}{At} = \frac{1}{Me}\frac{2}{\gamma+1}\left(1 + \frac{\gamma-1}{2} Me^2\right)^{2\,\frac{\gamma+1}{(\gamma-1)}}$$

where Ae is the area at exit of the nozzle, At is the throat area, Me is the exit Mach number, and γ is the specific heat ratio.

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1 Shapiro, A. H., The Dynamics and Thermodynamics of Compressible Fluid Flow, Chapter 4, Ronald Press, 1953.

2 Shepard, C. E., and Watson, V. R., "Performance of a Constricted-Arc Discharge in a Supersonic Nozzle," Paper No. 63-380, AIAA, 1963.

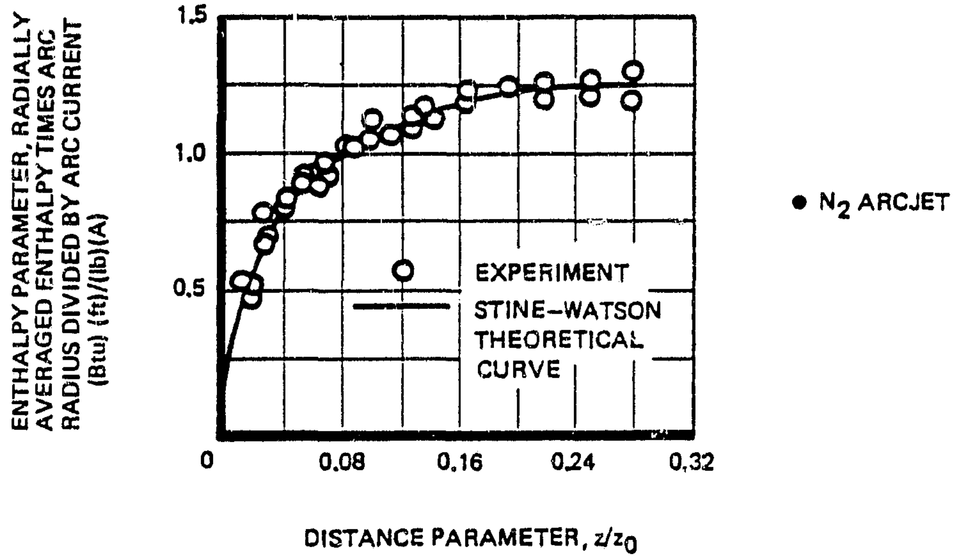

Figure F-6. Enthalpy Variation Comparison for the Stine-Watson Model

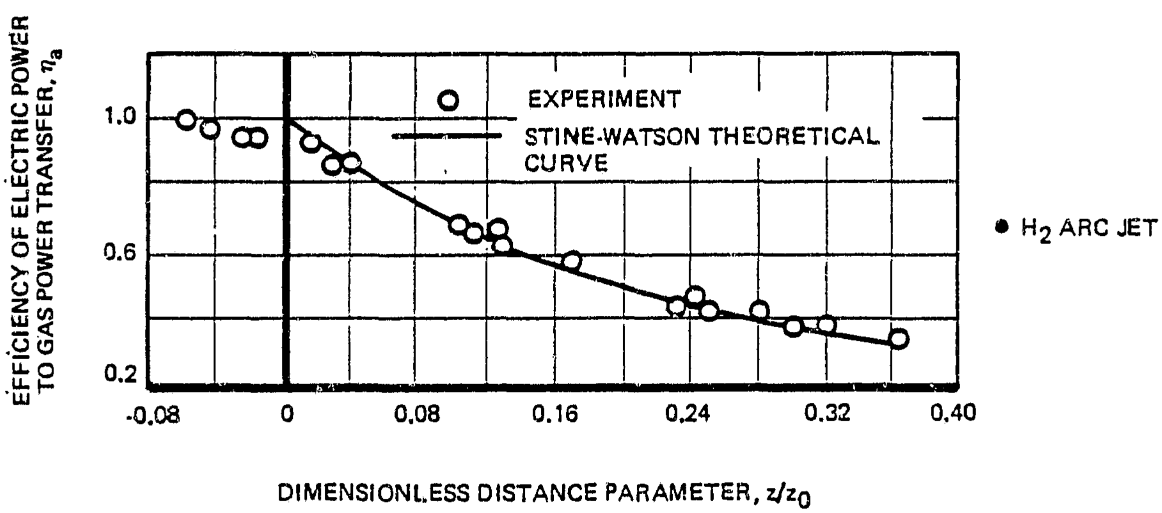

Figure F-7. Arc Efficiency Variation Comparison with the Stine-Watson Model

Isentropic flow relationships are then used to establish exit pressure, and the ideal-flow thrust coefficient is calculated:

$$C_{f1} = \frac{2}{\lambda \Gamma \Upsilon - 1} \quad \frac{(Xe)^{\frac{1}{2}}}{1+Xe} + \frac{Ae}{At} \quad \frac{Pe}{Pc}$$

where Pe and Pc are the exit and chamber pressures, respectively, the divergence factor, Υ, is 0.5 ($1+\cos\alpha$), where α is the nozzle half-angle, and

$$Xe = \left(\frac{Pc}{Pe}\right)^{\frac{\Upsilon-1}{\Upsilon}} - 1$$

A second iterative routine is now entered which uses the NASA-LeRC correlation to estimate corrected thrust coefficient:

$$C_f = C_{f1} - \frac{17.6\, e^{(0.0032\ Ae/At)}}{Re^{\frac{1}{2}}}$$

Since the thrust, I_{sp}, flow rate, Reynolds number, and Cf are interrelated, it is necessary to iterate until all of these converge at the desired chamber conditions and thrust level. The nozzle configuration and thruster performance have then been specified. Figure F-8 illustrates the foregoing process in block diagram form.

In addition to the assumption of the Stine-Watson large-arc model, it is assumed that the temperature of the plasma in the arc region is between 10,000°K and 30,000°K.[1,2] For our analysis, the arc was assumed to be a column of plasma at an average temperature of 20,000°K (35,540°F).

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1 John, R. R., et al, "Theoretical and Experimental Investigation of Arc Plasma-Generation Technology," Report No. ASD-TR-62-729, Pts I-II, Vols. 1-2, Avco Corp., 1964.

2 Stine, Howard A., and Watson, Velvin R., "The Theoretical Enthalphy Distribution of air in Steady Flow Along the Axis of a Direct-Current Electric Arc," NASA TN D-1331, 1962.

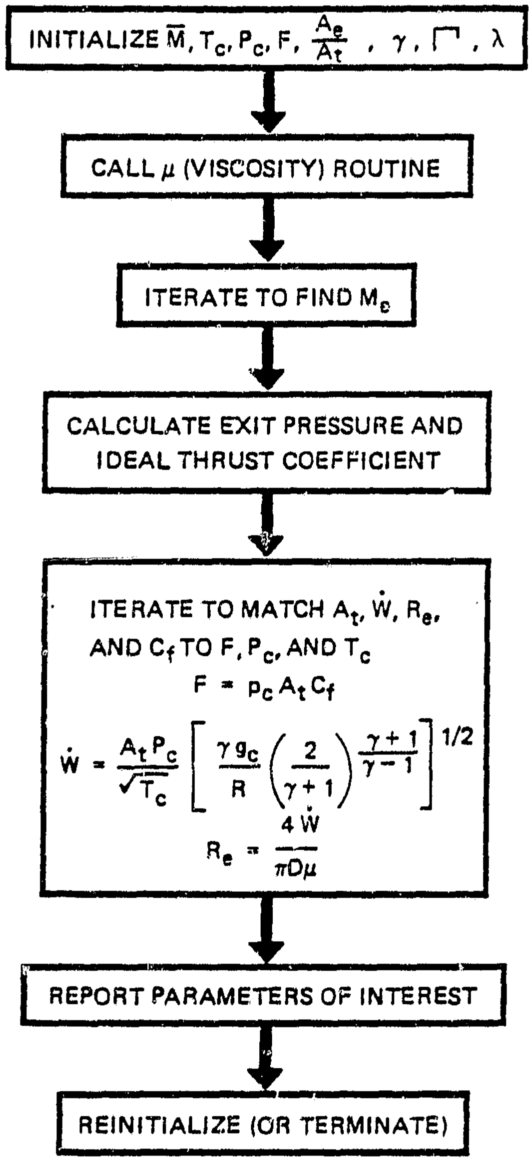

Figure F-8. Block Diagram of Nozzle Performance Program

Performance was calculated by assuming that the gas flow in the region next to the wall was held to a maximum of 4000°F. This temperature is lower than the expected wall temperature because the wall is heated by radiation from the arc. Two flows then exist: a hot (35,540°F) arc column inner flow, and a cold (4000°F) outer flow. Strictly speaking, some flow mixing will occur, but the effects of this have been neglected.

The average gas temperature was the independent variable in this exercise. To provide a given average temperature, the arc diameter/throat diameter ratio could then be calculated. Figure F-9 shows this relationship. Large arc diameters are anticipated using this model if performance significantly above the SOA monopropellant thrusters is desired.

Using a range of average temperatures between 3000°F and 25000°F, nozzle coefficients were calculated for each propellant as a function of chamber pressure. The results are shown in figure F-10. In contrast to the cold-flow Cf's shown in figure 6-29, the nozzle efficiencies for the arcjet at even moderate pressures can be very low. Elevated temperature and low mass flow yield very low Reynolds numbers and low nozzle efficiencies. A 50-mlb_f thrust level was chosen from power considerations. This thrust requires 2-3 kW of power when H_2 is used as the propellant.

Specific impulse values were calculated over the range of gas temperatures and pressures. Figures F-11 and F-12 show the results for H_2 and N_2H_4. Figure F-11 can be correlated with the existing data from table F-3. The 2-kW thruster operated at 30 mlb_f and achieved a 935-sec I_{sp}. To have these performance values, the model indicates that a very high average temperature (around 20,000°F) and a relatively high chamber pressure (3-10 atm) were required. As figure F-9 shows, this average temperature indicates a substantial arc diameter/throat diameter ratio. This implies a high radiative heat transfer from the arc to the wall with these large arc diameters, which may indicate a substantial heat transfer problem. N_2H_4 performance in this operating regime is around 500-550 sec I_{sp}, as shown in figure F-12.

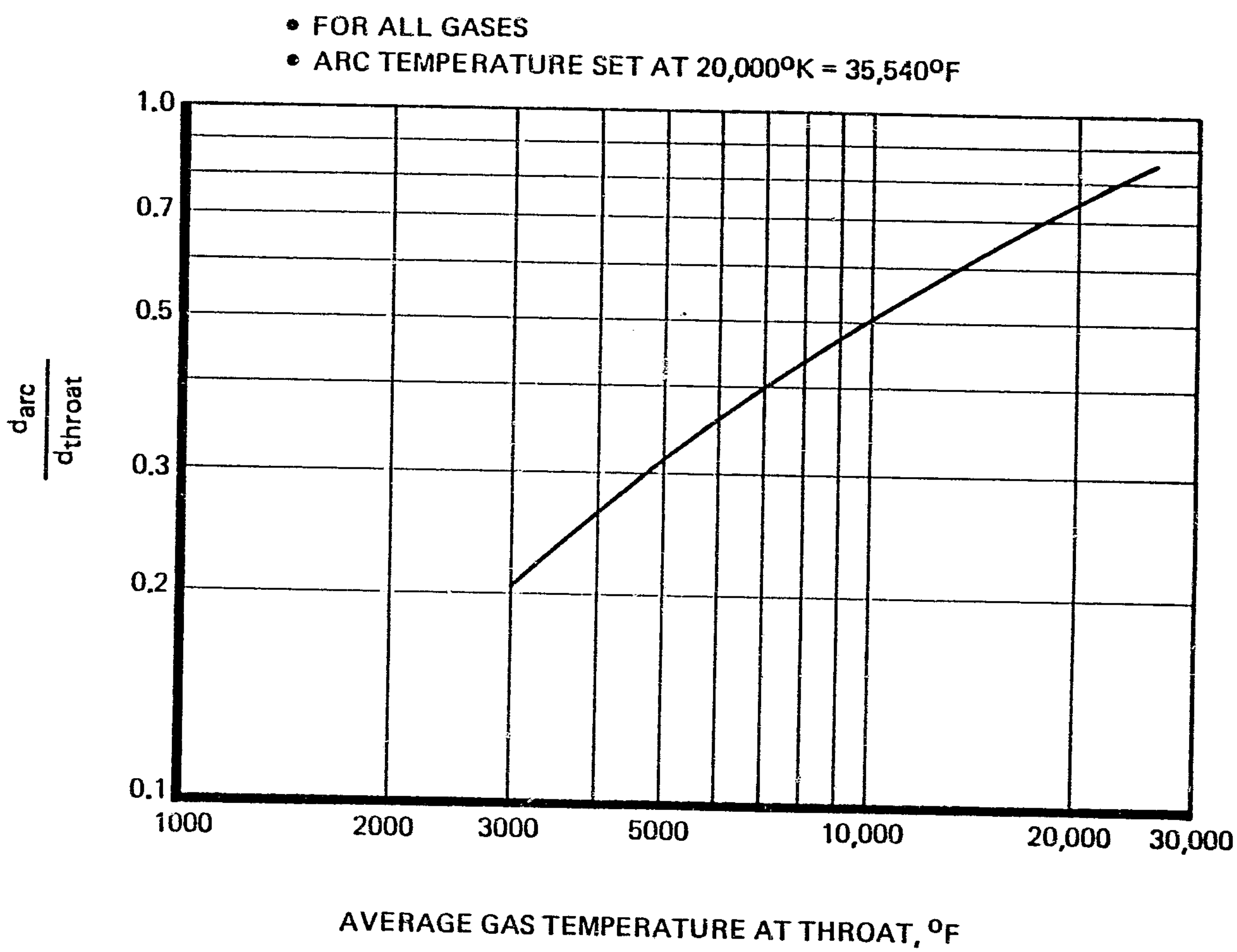

Figure F-9. Arc Diameter/Throat Diameter

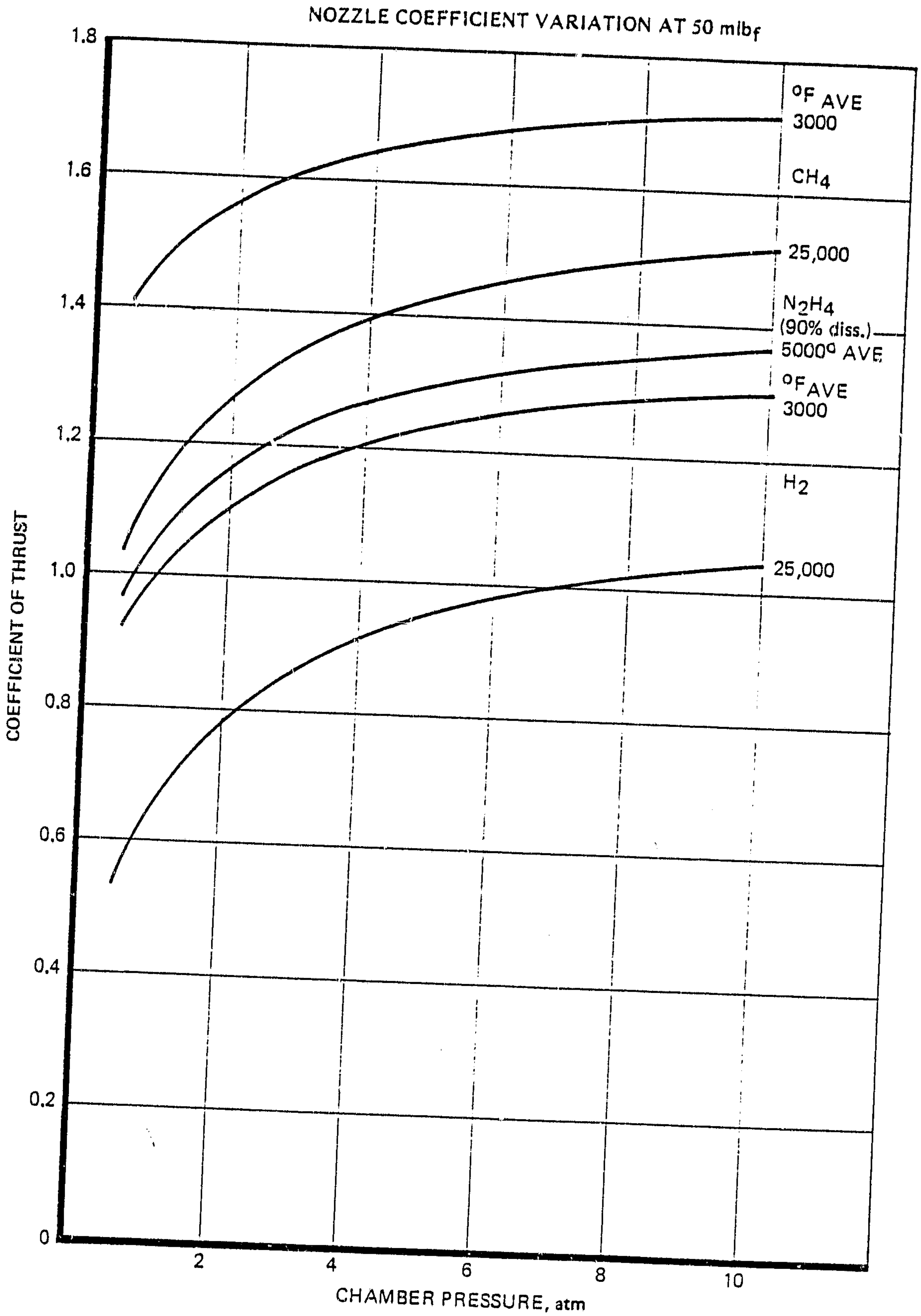

Figure F-10. Nozzle Coefficient Variation at 50 mlb$_f$

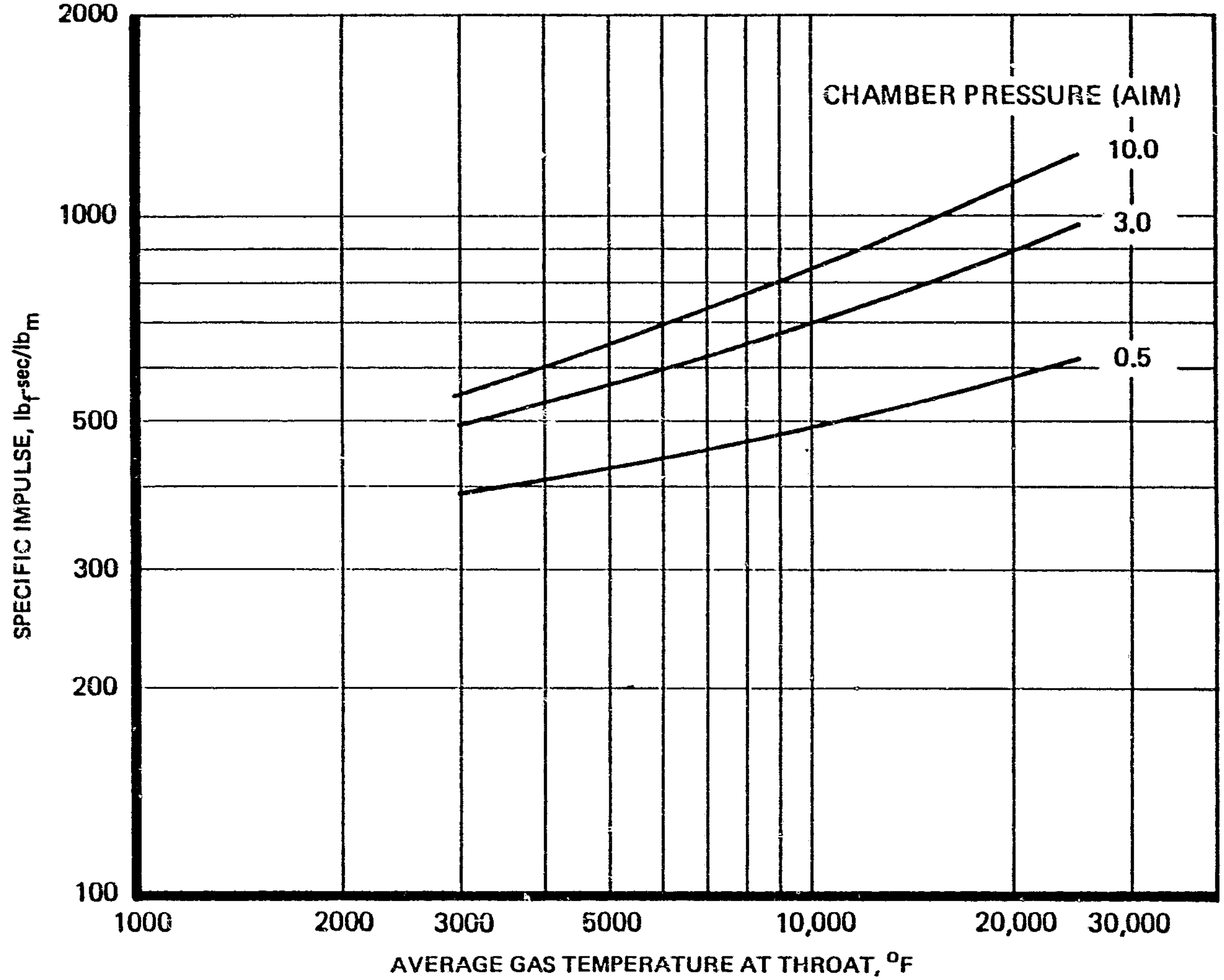

Figure F-11. Thruster Performance of Hydogen at 50 mlb$_f$

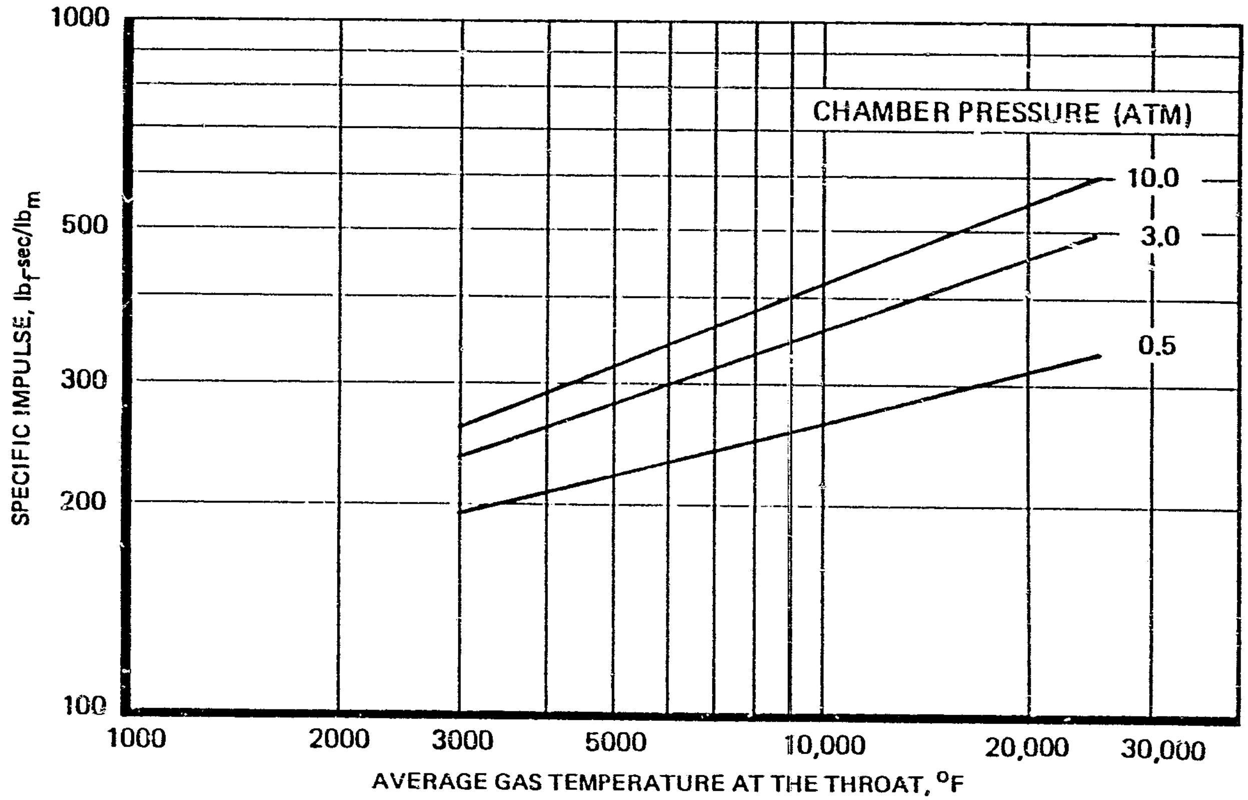

Figure F-12. Thruster Performance of Hydrazine at 50 mlb_f

As indicated earlier, performance improves with thrust level. A range of thrusts and average temperatures were used to calculate the expected I_{sp} for N_2H_4 and H_2 as illustrated in figure F-13. One interesting correlation is that performance of the RRC ACT resistojet at 100-lb_f is around 290 sec I_{sp}. The average gas temperature in this device is around 4000^{o}F. Figure F-13 predicts an I_{sp} of around 270 sec I_{sp}. Therefore, this analysis may underestimate performance by 10 to 20%.

The power levels required to yield a given thrust level are a function of other parameters such as frozen flow losses and other energy losses from the thruster. An analysis of those factors is beyond the scope of the study, however.

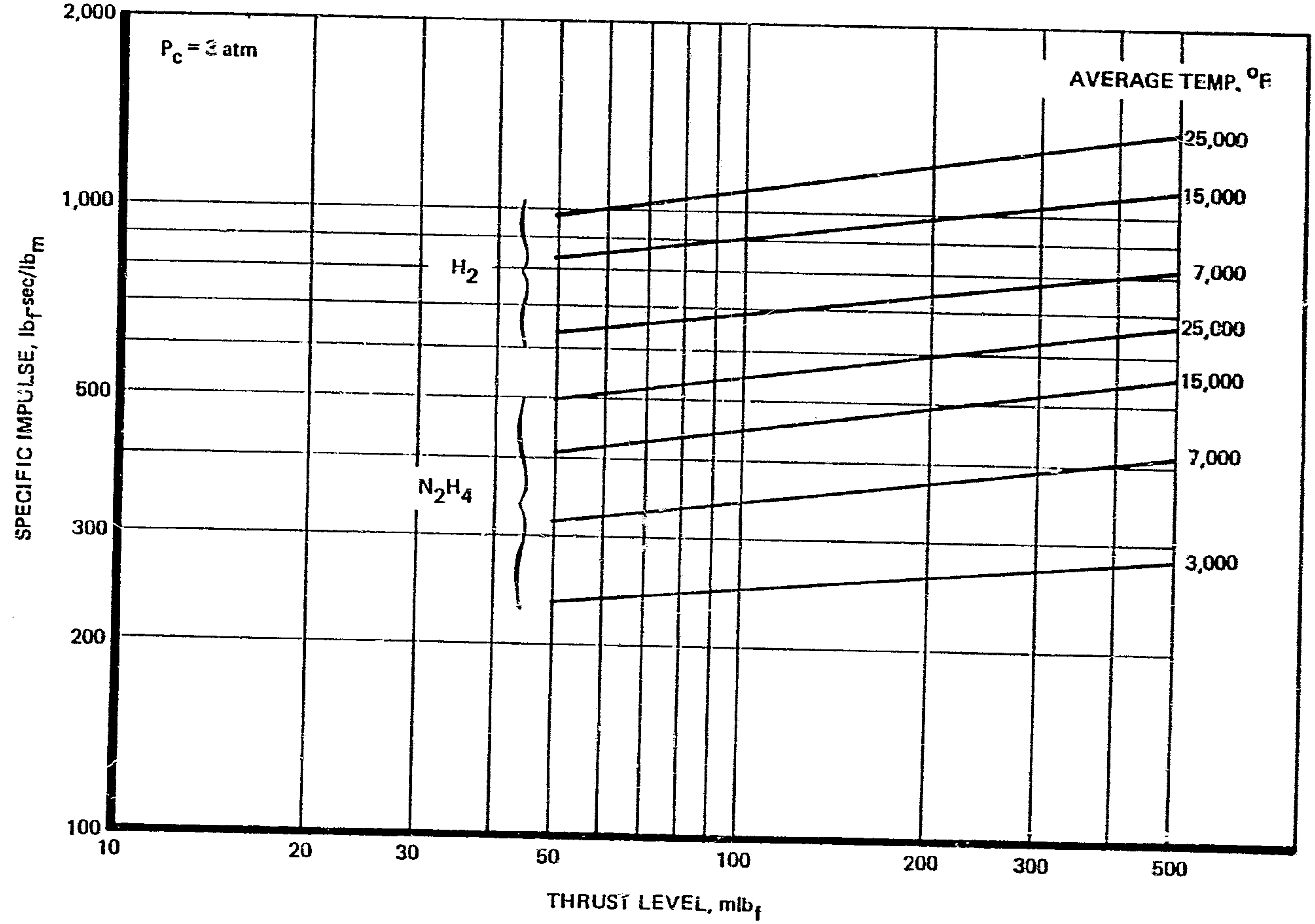

Figure F-13. Thruster Performance of Hydrogen and Hydrazine for Various Thrust Levels

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15. Supplementary Notes

Final report. Project Managers, M.E. Valgora, Advanced Programs and Planning Office and R.M. Donovan, Power System Engineering Division, NASA Lewis Research Center, Cleveland, Ohio 44135.

16. Abstract

The primary objective of the work described herein was to define propulsion system requirements to support Low Earth Orbit (LEO) manned Space Station development and evolution over a wide range of potential capabilities and for a variety of STS servicing and Space Station operating strategies. The term Space Station and the overall Space Station configuration refers, for the purpose of this report, to a group of potential LEO spacecraft that support the overall Space Station mission. The group consisted of the central Space Station at 28.5-deg or 90-deg inclinations, unmanned free-flying spacecraft that are both tethered and untethered, a short-range servicing vehicle, and a longer range servicing vehicle capable of GEO payload transfers. The time phasing for preferred propulsion technology approaches is also investigated, as well as the high-leverage, state-of-the-art advancements needed, and the qualitative and quantitative benefits of these advancements on STS/Space Station operations. The time frame of propulsion technologies applicable to this study is the early 1990's to approximately the year 2000.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
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