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Report

STUDY OF MULTIMISSION MODULAR SPACECRAFT (MMS) PROPULSION REQUIREMENTS

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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by

N. H. Fischer and A. E. Tischer

N. H. Fischer

N. H. Fischer, Study Manager

D. S. Edgecombe

Approved by: D. S. Edgecombe
Space Systems and
Applications Section

BATTELLE
Columbus Laboratories
505 King Avenue
Columbus, Ohio 43201

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1.0 INTRODUCTION

The Multimission Modular Spacecraft (MMS) is a spacecraft bus being developed for use with the Shuttle. The MMS is designed as a standard bus to be used on a wide variety of missions, many of which will require propulsion after separation from the Shuttle Orbiter to achieve the mission objectives. This study examines the propulsion requirements for MMS spacecraft. The objectives of the study are to:

(1) Determine the cost effectiveness of various propulsion technologies for Shuttle-launched MMS missions, with specific attention to the potential role of ion propulsion

(2) Find the cost effectiveness of appropriately mixing propulsion technologies for Shuttle-launched MMS missions

(3) Eliminate from possible future study those propulsion technologies and mixes thereof that are not cost effective

(4) Identify for possible future study the propulsion technolgies and mixes thereof that may be cost effective

(5) Study those propulsion technologies and mixes thereof that are cost effective.

To satisfy these objectives, it was necessary to choose a criterion for comparison for the different types of propulsion technologies. In this study the primary criterion chosen was the total propulsion related cost, including the Shuttle charges, propulsion module costs, upper stage costs, and propulsion module development. In addition to the cost comparison, other criteria such as reliability, risk, and STS compatibility are examined.

1.1 Study Approach

The study is divided into seven subtasks, as follows:

(1) MMS mission models
(2) Propulsion technology definition
(3) Trajectory/performance analysis
(4) Cost assessment
(5) Program evaluation
(6) Sensitivity analysis
(7) Conclusions and recommendations.

In the mission model subtask, estimates of MMS activity during the 1980-1991 time period are made. The study guidelines limit the consideration to geosynchronous and near-Earth orbits. The specific ground rules used, along with the mission models developed, are presented in Subsection 1.2. In addition to the projected MMS missions, some selected propulsion applications not presently included in MMS planning were examined. These special application missions are identified in Subsection 1.3.

The propulsion technology definition subtask provides the necessary technical data of the different technologies to determine what size modules are required and which technologies are applicable to each mission. The technologies considered in this study include ion engines, Earth-storable bipropellants, catalytic hydrazine, high-performance electrothermal hydrazine, solid motors and LOX/LH_2. The propulsion data defined in this subtask are presented later in Section 2.

The trajectory and performance analysis subtask determines the size of the propulsion modules needed. In this subtask, the requirements of all types of MMS missions are determined, including return and servicing missions, and also those additional missions identified in Subsection 1.3 which may or may not be MMS missions. The ground rules, discussion, and results of these analyses are shown in Section 3.

The cost assessment subtask consists of two parts: (1) providing a cost data base for the propulsion modules, etc., and (2) developing a methodology to compare different propulsion technologies. The cost data are presented in Section 4. The cost methodology discussion is included in Section 5.

The program evaluation subtask is the actual cost evaluation of the various propulsion families identified. Closely connected to this subtask is the sensitivity analysis subtask, which examines perturbations in cost data, module definition, mission models, etc. The results of both of these subtasks are discussed in Section 5. The final subtasks summarize the results and state the conclusions and recommendations of the study; these subtasks comprise Sections 6 and 7, respectively, of this report.

1.2 MMS Mission Models

Possible mission applications for MMS have been assembled to form an MMS mission model. The primary purpose for constructing the model is to evaluate the cost effectiveness of different propulsion technologies which could be used to satisfy MMS propulsion requirements. Alternative mission models are also presented so that the sensitivity to some of the key assumptions can be analyzed. The following ground rules have been established for assembly of the mission models:

(1) Shuttle missions only
(2) Earth orbital missions
(3) No small multimission spacecraft (Scout class)
(4) Emphasis on servicing missions
(5) 1980-1991 time period.

The MMS bus is being designed to be compatible with either the Shuttle or the Delta.[(1-1)*] The Solar Maximum Mission (SMM) and Landsat D/E follow-on are both to have their first launch on a Delta launch vehicle. These initial flights on the Delta vehicle will require minimal spacecraft propulsion, or none at all. Other studies[(1-2)] have examined the propulsion requirements for Landsat D/E for both the Delta- and Shuttle-launched missions. In our study, the Delta launched missions are considered to be too "near term" for inclusion, and in any case, the propulsion requirements are minimal. For Shuttle-launched missions, the propulsion requirements include orbit maneuvers between the Shuttle parking orbit and the final spacecraft orbit, attitude control, orbit maintenance, and maneuvers required for rendezvous or retrieval. In the mission definition, only the final orbits are given. The assumptions on the shuttle orbit can influence the trajectory in some cases. For example, if one of the "standard" Shuttle orbits has an inclination of 57 deg, then 57 deg is likely to be chosen for a mission which may go to an inclination between 50 and 60 deg. Previous studies have made a variety of assumptions about where the Shuttle can (or will) deliver the payload. In our study, the general guideline will be to use the Shuttle in a manner most conducive to payload sharing. Potentially, this could also impact the mission model.

The MMS bus and modules could be used in a variety of ways for many kinds of missions. However, in this study, only Earth-orbit missions are considered because these missions are expected to provide the bulk of MMS applications and, correspondingly, to define the "nominal" range of MMS

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*References, denoted by superscript numbers, are at end of section (Subsection 1.4).

propulsion requirements. In addition, the distinctive design features of MMS such as serviceability and recoverability are most applicable to Earth-orbit missions.

The basic size of the MMS bus is determined by the desire that it be usable for Delta class missions. Spacecraft in this class range in weight from 600 to 2000 kg (1320 to 4000 lb). For near-term use with Shuttle, the upper end of this range may grow to about 4500 kg (10,000 lb). For MMS, these weights would include both the spacecraft bus and the payload. Since the MMS modules are being designed for this class of mission, they end up being oversized for Scout class missions. Consequently, Scout class missions are not considered in this study.

One of the key motivations in the design of a modular spacecraft bus is the flexibility it affords in terms of on-orbit servicing. Previous studies[1-3] have compared three different modes of space operations using Shuttle. These modes can be characterized as delivery only, return, and on-orbit servicing. In all three mission modes, the desire is to have an operational satellite continually in orbit. In the delivery mode, spare satellites are kept on ground and launched when the on-orbit satellite fails (or is sufficiently degraded). In the return mode, a replacement satellite is launched, and the failed satellite is returned for subsequent ground refurbishment. In the on-orbit servicing mode, modules are brought up in the Shuttle to refurbish the spacecraft in orbit. In the different mission models based upon these three concepts, the on-orbit operational capability of a given program is held approximately constant. The baseline assumption is that low-Earth operational missions and experimental missions with a mission design life of longer than 1 year will be considered as candidates for servicing (retrieval or replacement).

The prediction of future missions is always difficult, and this is expecially so at the present time because advance plans for future missions appear to be undergoing substantial reappraisal and revision. For example, recent projections of NASA automated spacecraft missions for the early 1980's, the time when the MMS is to be introduced, indicate a lower level of activity than was indicated in earlier plans. Although these data are still "soft", they appear to be consistent with the general trend in NASA budgetary projections and associated new starts. Consequently, the approach adopted in this study was to give preference to the more recent advance mission planning data and not to use the older data, unless corroborated. Correspondingly, the principal sources of advance mission planning data used in this study are: the "National Payload Model (August 1976), the "STS Transition Planning Model" (September, 1976) and the "Battelle Outside User Model" (October, 1976).[1-4,1-5,1-6] These data sources consider missions through 1991; thus, the time period for this study begins with Shuttle initial operational capability (IOC) and extends through current planning horizons, i.e., 1980-1991.

1.2.1 MMS Mission Parameters

Several NASA missions are considerd to be prime candidates for MMS. Solar Maximum Mission (SMM), approved as a 1977 new start, would introduce the first MMS with a Delta launch in 1979.[1-7,1-8] A series of subsequent launches using the Shuttle would then continue throughout the decade of the 1980's to maintain continuous monitoring throughout the entire cycle of solar activity. The Landsat D/E combination is considered as a likely candidate for MMS. Current planning indicates the two Landsat spacecraft will alternate in-orbit and be refurbished on ground.[1-9] The first geosynchronous mission using MMS may be Stormsat.[1-10] Many of NASA's future explorer spacecraft are expected to use MMS, and there is some speculation that it may be mandatory for all explorer spacecraft (in the appropriate weight class) to be MMS.[1-11] Interest in MMS has been expressed by the Canadians, particularly regarding the servicing capabilities of the MMS. Additional spacecraft considered as potential candidates for MMS include Earth observation satellites such as TIROS O/P, Earth Survey Satellites, Earth Resources Satellites, and ITOS follow-on.

The missions in the models are identified both by name and by the SSPD code numbers, a data system developed by the Shuttle Payload Planning Working Groups at MSFC. The necessary mission parameters for this study include: flight schedules, weights, launch site, payload lengths, orbital parameters, and on-orbit velocity requirements. The MSFC payload descriptions[1-12] provided a source for some of these data, in particular the spacecraft weights and lengths. In the mission model data, the spacecratt weight and length pertain to the instrument section above the MMS bus. The MMS bus is taken to be 1.22 meters long and to have a mass of 635 kg. A mass breakdown of the MMS is given in Table 1-1 and a drawing of the components

TABLE 1-1. MMS WEIGHT STATEMENT (lb)(1-12)

Component	Baseline Configuration		Fully Redundant Configuration		
	Quantity	Total Weight	Quantity	Total Weight	Remarks
2.3.1 Communications & Data Handling		(101.0)		(131.0)	
* STADAN Transponder	2	16.0	2	16.0	
** Omni Antennas	2	4.0	2	4.0	
* Transponder Preregulator	2	4.0	2	4.0	
** RF Switches	2	1.0	2	1.0	
Command Demod/Decoder	2	8.0	2	8.0	
Format Generator, Clock, Bus Controller	2	8.0	2	8.0	
Remote Interface Unit	2	4.0	2	4.0	
Computer Interface Unit	2	6.0	2	6.0	Std. Low Cost Unit
Computer NSSC-1	1	30.0	2	60.0	Std. Low Cost Unit
Premod. Processor	2	4.0	2	4.0	
Pwr. Protect & Conditioning	1	6.0	1	6.0	
Harness & RF Cable	A/R	10.0	A/R	10.0	
2.3.2 Electrical Power Module		(266.0)		(522.0)	
Battery Charger (Part I)	1	22.0	1	22.0	
Battery Charger (Part II)	1	20.0	1	20.0	
* Battery 20 AH @ 51#	2	102.0	3	357.0	3 Batteries for Redundant Conf. are 50 AH
Signal Conditioning Assy.	1	19.5	1	19.5	
Power Disconnect & Current Assy.	1	20.0	1	20.0	
S/C Interface Connector Assy.	1	10.0	1	10.0	
Bus Protection Assembly	1	4.5	1	4.5	
Ground Charge Diode Assy.	2	6.8	2	6.8	
Remote Decoder @ 0.5#	1	0.5	2	1.0	
Remote Multiplexer @ 0.5#	1	0.5	2	1.0	
Module Harness	1	35.0	1	35.0	
Heat Sink Divider	1	6.5	1	6.5	
Misc, Brackets, Structure	A/R	12.0	A/R	12.0	
2.3.3 Attitude Control Module		(264.0)		(332.0)	
** Reference Gyro Assembly	1	40.0	2	80.0	
Bus Protection	1	8.0	1	8.0	
Magnetometer	1	5.0	2	10.0	
Interface Assembly	1	20.0	1	20.0	
Coarse Sun Sensor	8	4.0	8	4.0	
* Star Trackers	2	22.0	2	22.0	
* Reaction Wheels	3	60.0	4	80.0	
Drive Unit Electronics	1	30.0	1	30.0	

TABLE 1-1. (Continued)

Component	Quantity	Total Weight	Quantity	Total Weight	Remarks
2.3.3 Attitude Control Module (Cont.)					
** Magnetic Torquers	3	30.0	3	30.0	
Remote Multiplexer	4	2.0	8	4.0	
Remote Decoder	2	1.0	4	2.0	
Harness	1	20.0	1	20.0	
* Digital Sun Sensor	1	10.0	1	10.0	
Additional Structure	1	20.0	1	20.0	
2.3.4 Structure (Delta Launched)		(403.0)		(403.0)	
Transition Adapter	1	150.0	1	150.0	
Module Support Structure	1	73.0	1	73.0	
Module Structures	3	150.0	3	150.0	
Shuttle Launch & Retrieval Hardware	1	30.0	1	30.0	
2.3.5 Thermal Control		(62.1)		(62.1)	
Louvers & Covers (4.8 #ea. & cover)		30.0		30.0	
Blankets, 102 sq. ft.		8.2		8.2	
Paint, 3 mil		5.0		5.0	
Heaters, 25 sq. ft.		3.0		3.0	
OSR, 6 mil		12.9		12.9	
Silver-Teflon, 5 mil		3.0		3.0	
2.3.6 Electrical Integration		(73.0)		(73.0)	
Signal Conditioning & Control Module		25.0		25.0	
Wire, Cable, Connectors	A/R	45.0	A/R	45.0	
Misc. Clips, Tie Downs	A/R	3.0	A/R	3.0	
2.3.7 Vehicle Adapter (Delta 2910)		(66.0)		(66.0)	
Launch Vehicle Adapter		43.0		43.0	
Separation Mechanism		20.0		20.0	
Misc. Connectors, Harness		3.0		3.0	
TOTAL		1235.1		1589.1	

* Exists
** Mod. of existing hardware

is shown in Figure 1-1. The MSFC payload descriptions are of sufficient detail to enable a split between the instrument and bus portions of the spacecraft, although not all payloads in the MMS models developed here are based upon using MMS in the MSFC data base. For those payloads which are not indicated to be MMS, the total lengths and masses for an MMS configuration are different than for the configuration in the MSFC data base or other mission models.

The flight schedules for the different MMS models depend partially on the mission modes: deploy only, return, or on-orbit servicing. For each of these three options, a mission model is assembled which represents approximately the same on-orbit mission capability. These three mission modes have been analyzed from an overall mission cost in other studies.[1-3]

1.2.2 Deploy-Only MMS Model

In the deploy-only mission model, none of the spacecraft take advantage of the Shuttle (and MMS) capability for retrieval and on-orbit servicing. The mission model is shown in Table 1-2. To maintain the desired on-orbit capability over a period of years for scientific satellites such as SMM or Earth observation satellites such as Landsat or ITOS follow-on, a number of launches are required. This number is dependent upon the expected mission life of a satellite. The estimates of lifetimes are based upon planned lifetimes of satellites such as Landsat A/B and the atmospheric explorers.

1.2.3 Retrieval MMS Mission Model

One of the potential capabilities of the Shuttle is to retrieve payloads from orbit. The MMS is designed to be fully compatible with this mode of operation. A spacecraft could be brought back to Earth for a variety of reasons, including return for refurbishment or retrieval of experiments or data. For spacecraft being returned for refurbishment, there are two basic options in the method of operation. The first option is for a spacecraft to be returned on the same Shuttle flight as is used to launch the replacement spacecraft. The second option is that the return launch is different from subsequent launches in the series. Current plans indicate that Landsat D/E will employ the first option. This provides relatively continuous service and relieves

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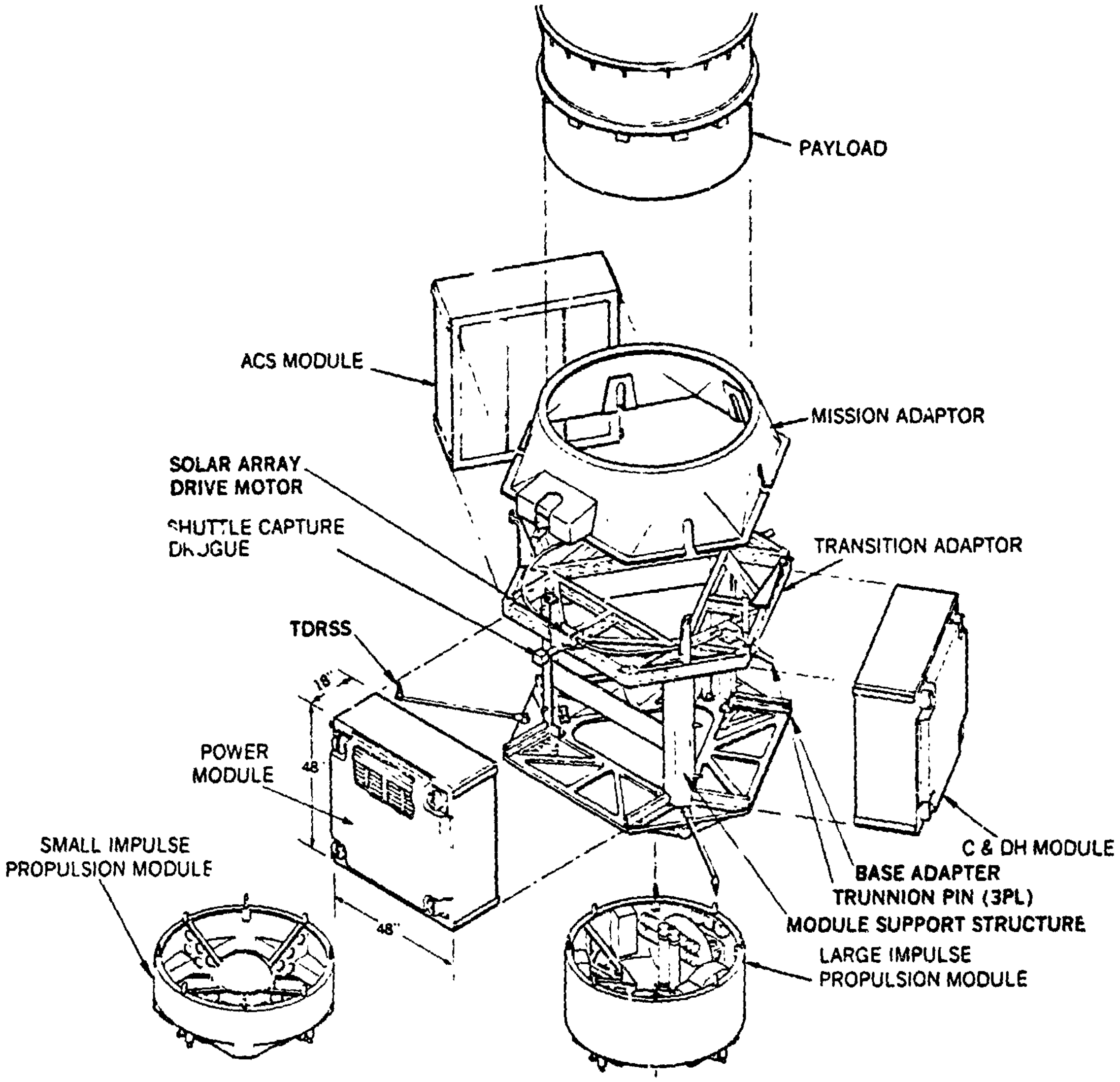

FIGURE 1-1. MMS COMPONENTS (1-12)

TABLE 1-2. DEPLOY-ONLY MMS MISSION MODEL

MISSION NAME	SSPD CODE	LAUNCH SCHEDULE 79	80	81	82	83	84	85	86	87	88	89	90	91	92	93			TOTAL	PAYLOAD PARAMETERS MASS kg	LENGTH DIA. m	MISSION LIFE	DELIVERY ORBIT APOGEE/PERIGEE km	INCL. deg.	LAUNCH SITE	ON ORBIT ΔV M/S	t_Ω
HIGH ENERGY ASTROPHYSICS																											
SMALL HIGH ENERGY OBS	HE-07A				1														1	100	0.5	1	500	28.5	ETR	-	
LARGE HIGH ENERGY OBS	HE-08A					1					1								2	8000	5.2	2	370	28.5	ETR	100	
LOW ENERGY ASTROPHYSICS	HE-27A						1												1	100	0.3	1	370	56	ETR	-	
SOLAR PHYSICS																											
SOLAR MAX MISSION	SO-03A					1		1		1		1		1					5	730	2.0	2	575	28.5	ETR	-	
ATMOSPHERIC AND SPACE PHYSICS																											
UPPER ATMOSPHERE EXPLORER	AP-01A				1							1							2	160	0.4	4	3000/300	28.5	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A													1					1	160	0.4	4	3000/300	56	ETR	600	
UPPER ATMOSPHERE EXPLORE	AP-01A					1				1									2	160	0.4	4	3000/300	90	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A						1				1								2	160	0.4	4	3000/300	10	ETR	600	
MEDIUM ALTITUDE EXPLORER	AP-02A				1								1						1	100	0.3	1	10,000/1000	28.5	ETR	-	
MEDIUM ALTITUDE EXPLORER	AP-02A								1				1						2	100	0.3	1	10,000/1000	56	ETR	-	
EARTH OBSERVATIONS																											
LANDSAT F/O D-E	EO-08A						1			1			1						3	960	2.8	3	705	98.2	WTR	25	9:00 am
TIROS O	EO-12A						1												1	1000	4.0	2	825	98.6	WTR	25	9:00 am
TIROS-P	EO-13A									1									1	1000	4.0	2	825	98.6	WTR	25	3:00 pm
STORMSAT	EO-15A						1			1			1						3	360	1.5	3-5	GEOSY		ETR	150	
EARTH SURVEY SATELLITE	EO-61A							1		1		1		1					4	500	2.0	2	700	98.2	WTR	30	11:00 am
ITOS F/O	EO-64A								1				1						3	1000	4.0	2	825	98.6	WTR	25	9 or 3
EARTH RESOURCES SATELLITE	EO-65A								1	1	1	1	1	1					6	2000	5.0	2	555	97.4	WTR	40	12 noon
FOREIGN																											
ALL WEATHER SEA MONITORING	OPN-02A							1						1					2	250	1.0	4	795	85	WTR	60	
TOTAL					3	3	5	3	3	7	4	4	5	5					42								

t_Ω = ascending node, local crossing time certain operational problems connected with payload retrieval on a shared flight. This will be discussed further in Section 3. An example of a satellite for which the second option would be preferred is the BESS (Biomedical Experimental Science Satellite), where subsequent flights would be dependent upon the results of previous flights. (Note: BESS is not included in the MMS mission model, since recent planning has eliminated it from current consideration.) Other spacecraft may have conflicting demands which would lead to preference of one option or the other. In Table 1-3, which presents the retrieval MMS mission model, the payloads which are to be returned are indicated with a circle; any additional launches for returning payloads are not indicated, nor is the option under which a payload is to be returned. The model is labeled ground refurbishment MMS mission model to coincide with terminology used in Reference (1-3).

1.2.4 Servicing MMS Mission Model

From an overall mission operation standpoint, on-orbit servicing of spacecraft has great potential for saving money.(1-3) On-orbit servicing is among the factors which influenced the inception and design of the MMS. The on-orbit servicing is potentially applicable to both operation satellites (i.e., weather satellites such as TIROS) and long-term scientific missions (e.g., Solar Maximum Mission). Current interest in servicing is limited primarily to low-altitude missions. The on-orbit servicing of spacecraft typically involves replacing one or more modules on the spacecraft, which could include replacing the propulsion module.

The MMS on-orbit service module is given in Table 1-4. In the model, only the initial satellite placement flights are shown. Since on-orbit servicing is typically based upon repair of a satellite, the frequency and time of service missions can best be described statistically. In this study, however, the assumption will be made that each mission (which provides for servicing) will be serviced once, approximately halfway through its nominal mission life. In practice, should a satellite require service early in its mission life, provision for additional servicing might be provided (i.e., additional propulsion capability) or the satellite might be returned to ground to correct a design deficiency. These impacts will not be considered in this study.

TABLE 1-3. GROUND REFURBISHMENT MMS MISSION MODEL

MISSION NAME	SSPD CODE	LAUNCH SCHEDULE 79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL	PAYLOAD PARAMETERS MASS kg	LENGTH / DIA. m	MISSION LIFE	DELIVERY ORBIT APOGEE / PERIGEE km	INCL. deg.	LAUNCH SITE	ON ORBIT ΔV m/s	t_Ω
HIGH ENERGY ASTROPHYSICS																									
SMALL HIGH ENERGY OBS	HE-07A				1												1	100	0.5	1	500	28.5	ETR	-	
LARGE HIGH ENERGY OBS	HE-08A					1					1						2	8000	5.2	2	370	28.5	ETR	100	
LOW ENERGY ASTROPHYSICS	HE-27A						1										1	100	0.3	1	370	56	ETR	-	
SOLAR PHYSICS																									
SOLAR MAX MISSION	SO-03A					1		1		1		1		1			5	730	2.0	2	575	28.5	ETR	-	
ATMOSPHERIC AND SPACE PHYSICS																									
UPPER ATMOSPHERE EXPLORER	AP-01A				1							1					2	160	0.4	4	3000 / 300	28.5	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A													1			1	160	0.4	4	3000 / 300	56	ETR	600	
UPPER ATMOSPHERE EXPLORE	AP-01A					1				1							2	160	0.4	4	3000 / 300	90	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A						1				1						2	160	0.4	4	3000 / 300	10	ETR	600	
MEDIUM ALTITUDE EXPLORER	AP-02A				1												1	100	0.3	1	10,000 / 1000	28.5	ETR	-	
MEDIUM ALTITUDE EXPLORER	AP-02A								1				1				2	100	0.3	1	10,000 / 1000	56	ETR	-	
EARTH OBSERVATIONS																									
LANDSAT F/O D-E	EO-08A						1			1			1				3	960	2.8	3	705	98.2	WTR	25	9:30 am
TIROS O	EO-12A						1										1	1000	4.0	2	825	98.6	WTR	25	9:00 am
TIROS-P	EO-13A									1							1	1000	4.0	2	825	98.6	WTR	25	3:00 pm
STORMSAT	EO-15A						1			1			1				3	360	1.5	3-5	GEOSY		ETR	150	
EARTH SURVEY SATELLITE	EO-61A							1		1		1		1			4	500	2.0	2	700	98.2	WTR	30	11:00 am
ITOS F/O	EO-64A								1		1		1				3	1000	4.0	2	825	98.6	WTR	25	9 or 3
EARTH RESOURCES SATELLITE	EO-65A								1	1	1	1	1	1			[illegible]	2000	5.0	2	555	97.4	WTR	40	12 noon
FOREIGN																									
ALL WEATHER SEA MONITORING	OPN-02A							1						1			2	250	1.0	4	795	85	WTR	60	
TOTAL					3	3	5	3	3	7	4	4	5	5			42								

t_Ω = ascending node; local crossing time

○ = RETURN MISSION

TABLE 1-4. ON-ORBIT SERVICE MODEL

MISSION NAME	SSPD CODE	LAUNCH SCHEDULE 79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL	PAYLOAD PARAMETERS MASS kg	LENGTH / DIA. m	MISSION LIFE	DELIVERY ORBIT APOGEE / PERIGEE km	INCL. deg.	LAUNCH SITE	ON ORBIT ΔV M/S	t_Ω
HIGH ENERGY ASTROPHYSICS																									
SMALL HIGH ENERGY OBS	HE-07A				1						1						1	100	0.5	1	500	28.5	ETR	-	
LARGE HIGH ENERGY OBS	HE-08A					▽1											1	8000	5.2	4	370	28.5	ETR	100	
LOW ENERGY ASTROPHYSICS	HE-27A						1										1	100	0.3	1	370	56	ETR	-	
SOLAR PHYSICS																									
SOLAR MAX MISSION	SO-03A					▽1				▽1				▽1			3	730	2.0	4	575	28.5	ETR	-	
ATMOSPHERIC AND SPACE PHYSICS																									
UPPER ATMOSPHERE EXPLORER	AP-01A				▽1							▽1					2	160	0.4	6	3000 / 300	28.5	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A													1			1	160	0.4	4	3000 / 300	56	ETR	600	
UPPER ATMOSPHERE EXPLORE	AP-01A					1				1							2	160	0.4	4	3000 / 300	90	ETR	600	
UPPER ATMOSPHERE EXPLORER	AP-01A						1				1						2	160	0.4	4	3000 / 330	10	ETR	600	
MEDIUM ALTITUDE EXPLORER	AP-02A				1								1				1	100	0.3	1	10,000 / 1000	28.5	ETR	-	
MEDIUM ALTITUDE EXPLORER	AP-02A								1				1				2	100	0.3	1	10,000 / 1000	56	ETR	-	
EARTH OBSERVATIONS																									
LANDSAT F/O D-E	EO-08A						▽1						▽1				2	960	2.8	5	705	98.2	WTR	40	9:30 am
TIROS O	EO-12A						▽1										1	1000	4.0	4	825	98.6	WTR	50	9:00 am
TIROS-P	EO-13A									▽1							1	1000	4.0	4	825	98.6	WTR	50	3:00 pm
STORMSAT	EO-15A						1			1			1				3	360	1.5	3-5	GEOSY		ETR	150	
EARTH SURVEY SATELLITE	EO-61A							▽1				▽1					2	500	2.0	4	700	98.2	WTR	60	11:00 am
ITOS F/O	EO-64A								▽1				▽1				2	1000	4.0	4	825	98.6	WTR	50	9 or [illegible]
EARTH RESOURCES SATELLITE	EO-65A								▽1		▽1		▽1				3	2000	5.0	4	555	97.4	WTR	80	12 noon
FOREIGN																									
ALL WEATHER SEA MONITORING	OPN-02A							▽1						▽1			2	250	1.0	7	795	85	WTR	100	
TOTAL					3	3	5	2	3	4	2	3	4	3			32								

t_Ω = ascending node, local crossing time

▽ : SERVICED MISSION

1.2.5 Mission Model Variations

Since all mission models are subject to uncertainty, variations of the mission model will be considered to determine what impact they have on the MMS propulsion requirements. The two main approaches to mission model variations are: (1) define a set of alternative models a priori and then determine their impact on propulsion requirements, or (2) after the propulsion requirements for the baseline models are evaluated, determine what perturbations of the model are required to basically alter the conclusions. In this study the second approach is used. The variations include mission frequency as well as specific mission parameters. The basic types of missions will remain the same; however, additional specific missions (discussed in Subsection 1.3) not in the model are analyzed separately. The discussion of the impact of mission model variations is in Section 5.

1.3 Special Mission Applications

The missions contained in the MMS mission models presented in Section 1.2 are missions which have, to varying degrees, appeared in various planning exercises. Some additional missions which could be considered for MMS, were also examined as a part of this study. The four considered are: (1) drag make-up, (2) Sun-synchronous satellite nodal change, (3) geosynchronous satellite final placement and North-South stationkeeping, and (4) geosynchronous North-South stationkeeping and return to Shuttle altitude. Each of these missions has propulsion requirements in addition to the original placement of the spacecraft. These additional requirements are potentially applications of advanced propulsion requirements such as ion propulsion or augmented electrothermal hydrazine.

1.3.1 Drag Make-Up

For satellite at altitudes near or below the Shuttle altitude (300 km), the nominal lifetime is at most a few months. If it were desirable to place a satellite at such an altitude over a longer period of time, a propulsion system would be required on the spacecraft to counteract the atmospheric drag. The nominal mission description assumed is as follows:

the altitude will be between 120 and 400 km, the inclination will be 28.5 deg, and the nominal mission life will be between 1 and 7 years. Three different launch times during the 11-year solar activity cycle are considered. These correspond to the minimum, maximum, and average drag cases. Two different spacecraft sizes are used in the analysis, a Scout class spacecraft and a Delta class spacecraft.

1.3.2 Sun-Synchronous Nodal Change

The key characteristic of a Sun-synchronous satellite is that the Earth under the satellite is always viewed with the same lighting conditions. That is, the local time* at the Earth's surface (at the equator) is constant. From an experimenter's viewpoint, the local time is a key parameter in designing his experiments. The length of shadows, amount of sunlight, and average cloud cover can be correlated to the local time. In a typical Earth observations satellite (e.g., Landsat D/E), several experimenters are involved. These experimenters are not always in agreement on what is the best choice of a local time, since each individual experimenter has different objectives. For example, in Landsat, two different local times have been under consideration, 9 a.m. and 11 a.m.

The local equatorial crossing time is determined by the longitude of the ascending node. A change of 2 hours in local viewing time corresponds to a 30-deg change in the longitude of nodes. Propulsion could be added to change from one orbit to the other for the required number of times. The propulsion requirements are reduced if sufficient time (months) can be allocated for the transfer between the viewing conditions.

Since this propulsion application is not contained in any planned mission, specific requirements cannot be defined. Assuming the satellite is to be in a Sun-synchronous circular orbit, the following basic parameters are required to define the propulsion requirements:

(1) Spacecraft mass
(2) Mission altitude
(3) Change in viewing conditions
(4) Number of changes between viewing conditions
(5) Time allocated for changing viewing conditions.

*Local time at a point on the Earth is defined as the time of day determined by the Sun's position in the sky.

The analysis of this mission is presented later in Section 5, where specific parameters are discussed. To the extent possible, the parameters are treated over wide ranges to determine under what conditions the mission is feasible and what types of propulsion systems are best.

1.3.3 Geosynchronous Satellite Placement and Stationkeeping

Geosynchronous spacecraft are initially placed in an orbit that is only approximately geostationary. From this orbit, the correct longitude is achieved, and then the orbit is trimmed to become nearer to geostationary. After the desired orbit is achieved, stationkeeping is required to maintain the orbit during its operational lifetime.

For this analysis, the spacecraft are taken to be SSUS-D or SSUS-A class spacecraft. This implies two ranges of spacecraft weights rather than two specific weights. The apogee burn is done with a solid apogee kick motor (AKM); thus, the errors after the AKM burn are due to errors in both the SSUS and the AKM. At this time, the spacecraft is nominally despun to become three-axis controlled.

Since the desired spacecraft lifetime is usually several years, there is a high reliability requirement on the stationkeeping system. Typically this implies redundant thrusters.

1.3.4 Geosynchronous Satellite Return

For a geosynchronous satellite with a high specific impulse system used for stationkeeping, satellite placement/moving, etc., it may potencially be feasible to use this system to bring back the satellite in the event of a malfunction early in the mission life. This would require additional propellants to be loaded in the spacecraft propulsion system for the return capability. The analysis of this mission possibility is not a trade-off between technologies (since the velocity requirements are too severe for a single-stage hydrazine or bipropellant system), but an analysis of the impacts of providing a return capability with an ion system.

1.4 References

(1-1) Low Cost Modular Spacecraft Description, X-700-75-140, NASA/Goddard Space Flight Center, May, 1975.

(1-2) Landsat/MMS Propulsion Module Design, Tasks 4.3 and 4.4, SD-76-SA-0095-2, NASA/Goddard Space Flight Center, September 24, 1976.

(1-3) Shuttle User Analysis (Study 2.2) Final Report, Volume III, Part 5, Aerospace Corporation, February 28, 1975.

(1-4) National Payload Model, Plan B (PM01), NASA/Marshall Space Flight Center, August, 1976.

(1-5) STS Transition Planning Model (MO), NASA Office of Space Science, September, 1976.

(1-6) Buckeye, D. L., Outside Users Payload Model, BMI-NLVP-IM-76-9, Battelle's Columbus Laboratories, Columbus, Ohio, October 8, 1976.

(1-7) Shumann, W. A., 1978 Mission Set for Study of Solar Flares, Aviation Week and Space Technology, March 25, 1974, p 52-53.

(1-8) Solar Mission will Pace Modular Spacecraft Idea, Aviation Week and Space Technology, January 19, 1976.

(1-9) NASA Considers Technology Demonstration Satellite as OFT Payload, Aerospace Daily, Vol. 80, No. 25, August 5, 1976, p 192-193.

(1-10) Stormsat Emerges as Future NASA Start, Aviation Week and Space Technology, July 12, 1976.

(1-11) NASA Favors Electrodynamics, IR Explorers, Studies Eight Others, Aerospace Daily, Vol. 80, No. 33, August 17, 1976, p 253.

(1-12) Payload Descriptions, Volume 1, Automated Payloads, NASA/Marshall Space Flight Center, July, 1975.

2.0 PROPULSION TECHNOLOGY DEFINITION

The propulsion technologies considered in this study are limited to the following six categories:

(1) Catalytic hydrazine

(2) Solid rocket motor

(3) Earth-storable bipropellant

(4) LO_2/LH_2

(5) High-performance electrothermal hydrazine

(6) Ion propulsion.

Systems employing these technologies are being considered to satisfy propulsion requirements for MMS spacecraft, which include delivery of the spacecraft to its desired orbit from Shuttle orbit, return to Shuttle orbit for retrieval/servicing, and orbit maintenance/attitude control. Not all propulsion technologies are applicable to each of the various requirements. The six propulsion technologies are presented in the following subsections; and no attempt has been made to determine which technologies are best (or even applicable) for use with MMS.

2.1 Catalytic Hydrazine

Hydrazine is the most widely used propellant in current spacecraft reaction control systems. Hydrazine is also used in the attitude control system for the Titan Transtage. Flight-proven hydrazine engines ranging in thrust from 0.4 to 2700 N (0.1 to 600 lb_f) have been developed by such U.S. firms as Hamilton Standard, Hughes, Rocket Research, Inc., and TRW. These systems are all characterized by the use of a catalyst (e.g., Shell 405) to decompose the hydrazine.

Specific impulse (I_{sp}) for catalytic hydrazine is basically dependent upon the engine inlet pressure. However, the range of I_{sp} values in operational systems is sufficiently narrow (213 to 230 sec) to allow selection of a representative specific impulse for use in sizing analyses. An I_{sp} value of 220 sec is recommended.

Like all liquid propulsion systems, hydrazine systems have expended mass fraction values less than comparable solid motors. The mass fraction is calculated by dividing the expended mass by the initial mass of the system.

Investigation of several spacecraft systems coupled with previous work in this area indicate that a mass fraction of 0.82 would be typical for monopropellant hydrazine. In the majority of cases, the propellant required is relatively insensitive to reasonable deviations from this selected value.

2.2 Solid Rocket Motors

To define representative performance parameters for solid rocket motors, an analysis was made of 23 existing or proposed motors. Table 2-1 lists pertinent parameters for these motors, which are divided into the categories of current technology and advanced technology. Propellant mass ranges from approximately 70 kg (150 lb) to a little over 1000 kg (2300 lb) for current technology motors. Advanced motors have propellant loads ranging from 500 kg (1000 lb) to 3000 kg (6000 lb). Data are included for motors manufactured by Thiokol, Chemical Systems Division of United Technologies, Aerojet, and Hercules. These motors provide a suitable base of information from which parameter values believed to be typical of motors in each technology class have been selected. The results of this analysis are discussed in the subsections that follow.

2.2.1 Specific Impulse

To determine a representative specific impulse (I_{sp}) value for use in a sizing analysis, a plot was made of motor effective specific impulse versus expended mass. The expended mass includes the propellant and any inert materials expended during the burn. The effective specific impulse is determined by dividing the motor total impulse by the expended mass. This information is displayed in Figure 2-1. The solid curves on this figure indicate the recommended relationships between specific impulse and expended mass.

For current technology solid motors, a representative effective I_{sp} value of 286 sec was selected. For the majority of the motors shown in Figure 2-1, this value is a reasonable approximation of motor I_{sp}. The points corresponding to the TE-M-442-1, SVM-3, BE-3-A, and BE-3-B motors are substantially lower than the selected value. These motors, with the exception of the SVM-3, employ nozzles with expansion ratios of approximately

TABLE 2-1. EXISTING AND PROPOSED SOLID MOTORS

Motor	Application	Total Mass, kg (lb_m)	Propellant Mass, kg (lb_m)	Propellant I_{sp}, sec	Thrust, N (lb_f)
		Current Technology			
1. TE-M-479	Radio Astronomy Explorer AKM	79 (174)	70 (153)	290.0	11,077 (2,490)
2. TE-M-521	IMP H AKM	124 (274)	112 (247)	289.9	16,014 (3,600)
3. TE-M-604	Skynet II AKM	216 (476)	198 (437)	288.4	17,767 (3,994)
4. TE-M-442-1	Burner IIA Apogee Motor	261 (576)	238 (524)	272.4	34,453 (7,745)
5. TE-M-616	CTS AKM	362 (799)	333 (734)	293.1	26,690 (6,000)
6. TE-M-364-1	Surveyor Retro	620 (1,368)	558 (1,230)	290.0	38,407 (8,634)
7. TE-M-364-3	Delta Third-Stage	717 (1,581)	653 (1,440)	290.4	43,060 (9,680)
8. TE-M-364-4	Extended Delta	1,122 (2,473)	1,039 (2,290)	285.5	68,826 (15,472)
9. TE-M-364-11	Improved Extended Delta	1,141 (2,516)	1,058 (2,333)	289.5	62,923 (14,145)
10. TE-M-364-19	FLTSATCOM AKM	913 (2,013)	845 (1,863)	287.6	55,472 (12,470)
11. TE-M-364-22	Development	1,161 (2,559)	1,077 (2,375)	291.0	--
12. FW-4	Scout Delta	301 (665)	275 (606)	284.0	23,794 (5,349)
13. FW-5	ANIK AKM	293 (647)	262 (577)	285.4	18,132 (4,076)
14. SVM-1	INTELSAT II AKM	87 (192)	74 (163)	288.3	12,927 (2,906)
15. SVM-2	INTELSAT III AKM	159 (350)	139 (306)	283.8	13,968 (3,140)
16. SVM-3	Unspecified	72 (159)	62 (137)	277.5	7,580 (1,704)
17. SVM-4A	INTELSAT IV AKM	706 (1,557)	642 (1,416)	289.7	55,516 (12,480)
18. SVM-5	SMS AKM	316 (697)	286 (631)	283.2	22,509 (5,060)
19. BE-3-A	Scout	97 (214)	87 (191)	276.0	26,246 (5,900)
20. BE-3-B	AKM	110 (243)	99 (219)	276.0	34,253 (7,700)
		Advanced Technology			
21. Star 30	High Performance AKM	489 (1,079)	463 (1,021)	295.0	24,466 (5,500)
22. Star 48	Orbit Insertion Motor	1,678 (3,700)	1,591 (3,509)	295.0	51,686 (11,619)
23. Small IUS	--	2,904 (6,403)	2,721 (6,000)	297.3	77,536 (17,430)

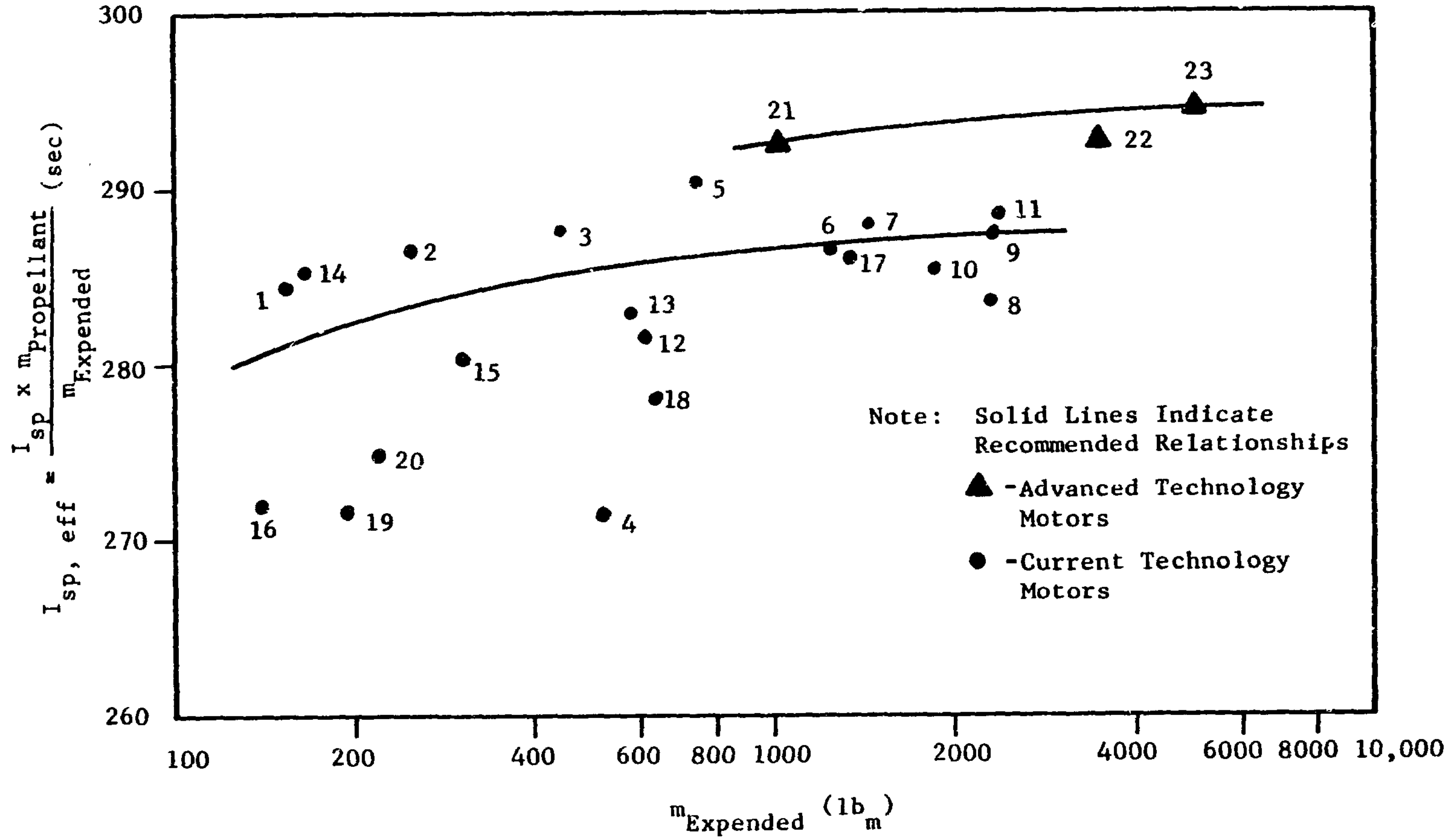

FIGURE 2-1. EFFECTIVE SPECIFIC IMPULSE OF THE SOLID ROCKET MOTORS LISTED IN TABLE 2-1 AS A FUNCTION OF EXPENDED WEIGHT

17:1 (the other motors have expansion ratios $\geq$ 30). The SVM-3 has a low aluminum content (~2%), which results in a lower I_{sp}.

For advanced technology motors, an I_{sp} value of 293 sec was selected for sizing purposes. This value is consistent with data shown in Figure 2-1. At present, a gap exists between the propellant load of the TE-M-364-22 and the small IUS motor. Thiokol is developing the Star 48 motor to bridge this gap, and it is likely that other solid motor manufacturers will follow suit. Motors which are developed in this class will likely achieve performance values comparable to the Star 48 and small IUS motors.

2.2.2 Expended Mass Fraction

The effect of the motor expended mass fraction (MF) was investigated by plotting MF versus expended mass for the motors in Table 2-1. Expended mass, as mentioned in the previous section, includes the mass of the propellant plus the mass of any expended inert materials. Inclusion of the inerts increases the MF values for solid propellant motors. The graph obtained is presented as Figure 2-2. The solid curves on this figure indicate the recommended relationships between expended mass and the MF.

To calculate the propellant required for a given mission, it is necessary to assume an initial value for the MF. Calculations indicate that the assumed mass fraction is not critical, except for missions in which the propellant mass is greater than the payload mass. The term payload, as used here, is defined as everything above the motor, i.e., the spacecraft and adapters. For the missions included in this study, such a condition could occur only for propellant loadings exceeding 500 kg (1000 lb). Motors of this size and larger have a much narrower range of MF values.

On the basis of the above considerations, an expended mass fraction of 0.925 would yield acceptable initial estimates of the propellant required for current technology motors.

For advanced technology motors, a MF of 0.954 is recommended. The IUS motor has an expended MF less than this value, but thrust vector control capability is included on this motor as currently defined. Removal of this system would likely raise the MF to a value similar to that of other motors in this class.

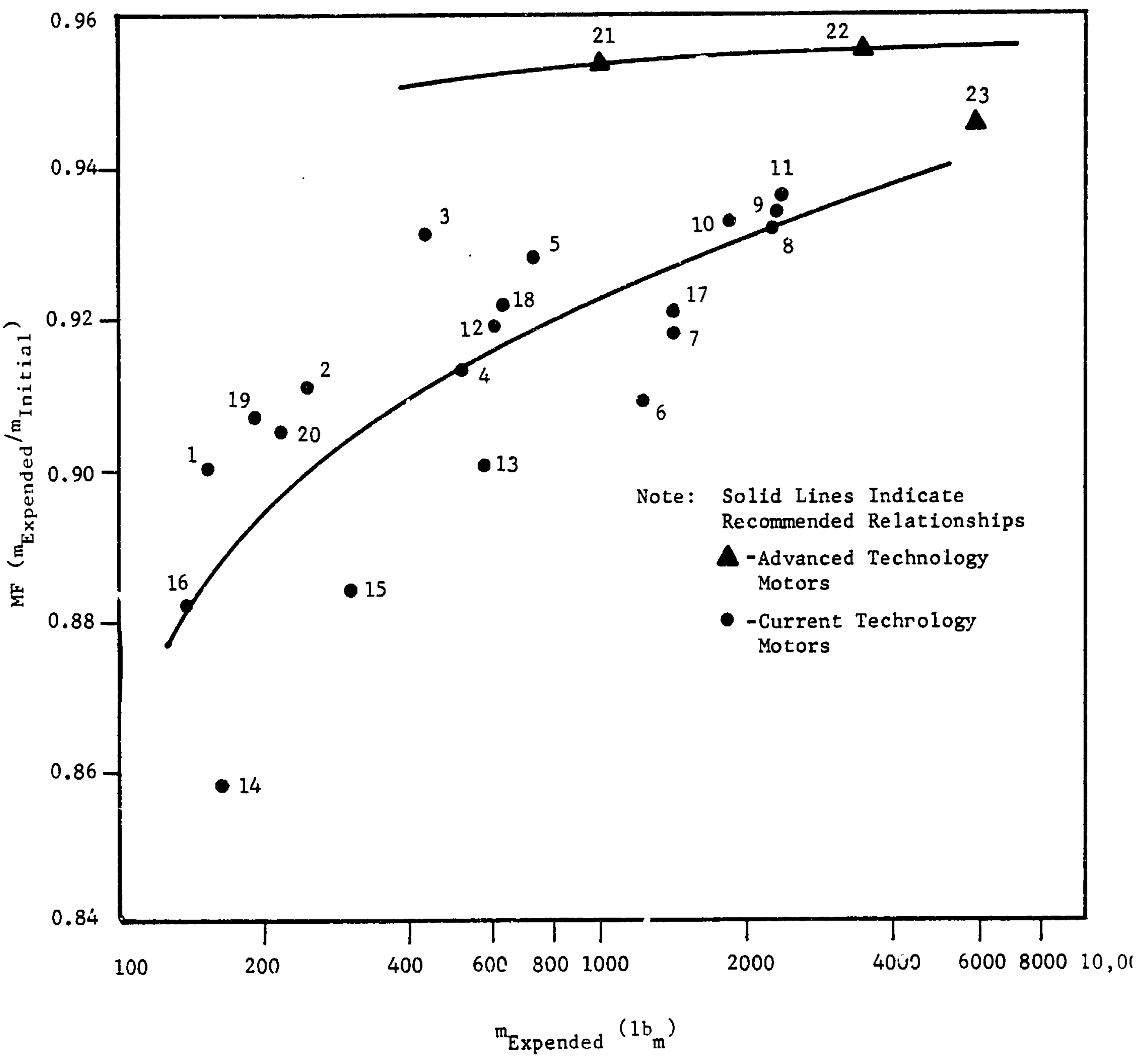

FIGURE 2-2. EXPENDED MASS FRACTION OF THE SOLID ROCKET MOTORS LISTED IN TABLE 2-1 AS A FUNCTION OF EXPENDED WEIGHT

2.2.3 Motor Thrust

Thrust values for solid rocket motors cannot be correlated well with propellant mass only (thrust level is also a strong function of chamber pressure). As shown in Table 2-1, motors which have propellant loadings up to 500 kg usually have thrust values in the area of 18,000 to 27,000 N (4000 to 6000 lb_f). Motors with a propellant mass exceeding 500 kg can be characterized by thrust levels in the range of 45,000 to 53,000 N (10,000 to 12,000 lb_f).

2.3 Earth-Storable Propulsion System

For Earth-storable systems, the state of the art is represented by systems using cold-gas pressurized N_2O_4 and hydrazine with pressure-fed ablative conduction, or radiation-cooled engines operating at chamber pressures of 70 to 140 N/cm^2 (100 to 200 psia). Spacecraft propulsion systems using this technology include TRW's Multi-Mission Bipropellant Propulsion System (MMBPS); Mariner and Viking propulsion systems designed by JPL; NASA's Apollo Service Module; Lunar Module descent and ascent systems; the Titan Transtage; and several reaction control systems. Operating characteristics of four existing Earth-storable bipropellant engines are shown in Table 2-2.

In August 1975, TRW completed its study of the "Design of Multi-Mission Chemical Propulsion Modules for Planetary Orbiters".[2-1]* Sizing estimates in this study were based on the initial assumptions that I_{sp} = 296 sec and MF = 0.82 for Earth-storable systems. These numbers are in reasonable agreement with information from other sources, including Battelle's IUS/Tug Auxiliary Stage Study.[2-2] TRW's Multi-Mission Bipropellant Propulsion Stage has a specific impulse of 295 sec and a mass fraction of 0.88.[2-3]

For missions under consideration in this study, it is recommended that initial sizing of propellant mass be based on an I_{sp} of 295 sec and a mass fraction of 0.85.

*References, denoted by superscript numbers, are in Subsection 2.8.

TABLE 2-2. CHARACTERISTICS OF EXISTING EARTH-STORABLE BIPROPELLANT ENGINES

	MMBPS	Shuttle RCS	MBB Symphonie	Mariner 71
Propellant	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/A50	N_2O_4/MMH
Thrust, N (lb_f)	391 (88)	2880 (872)	391 (88)	1317 (296)
Specific Impulse (sec)	295	290	303	287
Chamber Pressure, N/cm^2 (psi)	63 (91)	105 (152)	70 (102)	79 (115)
Nozzle Area Ratio	52:1	22:1	77:1	40:1
Engine Mass, kg (lb_m)	4.54 (10)	6.6 (14.5)	1.95 (4.3)	7.8 (17.1)

2.4 Oxygen/Hydrogen Propulsion Systems

Several contractors have conducted recent investigations of cryogenic systems for use as upper stages and propulsion modules on the Shuttle and expendable launch vehicles.

Hughes Aircraft has studied a LO_2/LH_2 stage containing approximately 1700 kg (3800 lb) of propellants.(2-4) This concept would provide both the perigee and apogee burns for a spacecraft currently launched by the Delta 3914. In addition, Hughes has investigated a cryogenic apogee kick motor (AKM) for use in satellites of the Atlas-Centaur class.(2-4)

The Aerojet Liquid Rocket Company has also conducted a preliminary cost-effectiveness study of LO_2/LH_2 kick stages.(2-5) This effort was aimed at stages to augment the performance of an Earth-storable IUS.

The Hughes and Aerojet propulsion modules are summarized in Table 2-3. Recommended oxygen/hydrogen performance parameters have been based in part on these designs, as described in the following paragraph.

Examination of the data in Table 2-3 indicates that the Hughes value for effective I_{sp} may be optimistic. An I_{sp} of 425 sec is recommended for current technology LO_2/LH_2 systems. This number corresponds to a pressure-fed engine and could be increased by the use of a pump-fed system, but such a modification would increase the stage complexity and costs, and, for small systems, might decrease the stage mass fraction. Therefore, it is considered doubtful that a pump-fed engine would be used on a stage of the size being considered in this study. For a pressure-fed propulsion module, a mass fraction of 0.75 to 0.80 is recommended for initial sizing estimates.

2.5 Non-Catalytic Hydrazine

Non-catalytic hydrazine thrusters are divided into the categories of: (1) electrothermal and (2) augmented electrothermal. Both employ heated platinum screens to initiate hydrazine decomposition. Elimination of the catalyst bed improves the thruster pulsing characteristics and also significantly increases the operational lifetime of the system. Each of these systems is intended primarily for use in spacecraft attitude control systems. They are described in greater detail in the following subsections.

TABLE 2-3. OXYGEN/HYDROGEN KICK STAGES

Concept	Total Mass, kg (lb_m)	Propellant Mass, kg (lb_m)	$I_{sp,eff}^{(*)}$ sec	$MF_{eff}^{(*)}$	Thrust, N (lb_f)
Hughes Delta-class propulsion module	2094 (4616)	1707 (3764)	437.3	0.82	2200 (500)
Hughes Atlas/Centaur AKM	807 (1780)	612 (1349)	434.9	0.76	2200 (500)
Aerojet spacecraft propulsion module	2920 (6438)	2268 (5000)	425.74	0.78	13,300 (3000)

*Effective values of I_{sp} and mass fraction include effects of propellant boil-off and loss of tank pressurant.

2.5.1 Electrothermal Hydrazine

TRW currently manufactures electrothermal thrusters which operate in a blowdown mode. Initial thrust for these engines is 0.3 N (0.06 lb_f), which decreases to about 0.09 N (0.02 lb_f). Approximately 4 to 5 watts of electrical power are required to heat the platinum screen to a nominal temperature of 540°C (1000°F). At this temperature, hydrazine decomposition is initiated, and for steady-state operation, the heater can be turned off (heat released by the reaction is sufficient to maintain operation). The non-catalytic thruster has a slightly improved I_{sp} (230 sec, as compared to 220 sec for catalytic hydrazine) and also exhibits an improved thruster pulse curve. The major benefit of a non-catalytic system is the extended lifetime which results from elimination of catalyst degradation due to contamination, crushing, and nitriding. These systems do, however, place an additional requirement on the spacecraft power supply and also increase the complexity of the attitude control system.

Hydrazine blends which are compatible with Shell 405 catalyst have freezing temperatures of approximately 4.4°C (40°F). This thermal constraint places restrictions on spacecraft operations; extended periods of exposure to deep space would have to be avoided. The use of electrothermal hydrazine thrusters allows selection of hydrazine blends with freezing points between -18°C (0°F) and -40°C (-40°F).

TRW is currently involved in an effort to scale up this technology for use in a 22 N (5 lb_f) thruster. Input power for the heater is 15 to 20 watts for this system. This effort is under contract to NASA/Goddard. MF values for electrothermal systems can be assumed to be identical to those for catalytic systems for initial sizing purposes. Therefore, recommended values are 230 sec for I_{sp} and 0.82 for MF.

2.5.2 Augmented Electrothermal Hydrazine

TRW and Avco are currently developing augmented electrothermal thrusters which will be used primarily for North-South stationkeeping on geosynchronous communications satellites. The first expected use of these systems will be on the INTELSAT V spacecraft currently under development by the Ford Aerospace and Communications Corporation. The higher I_{sp} available with this technology enables designers to reduce the amount of on-board hydrazine needed at the expense of additional power requirements and added system complexity. Operational characteristics for these systems are summarized in the following paragraphs.

The TRW system is limited to thrust levels of less than 0.44 N (0.10 lb_f), with the nominal value being approximately 0.13 N (0.03 lb_f). The I_{sp} is 320 sec with a power input of 1.2 watts per 10^{-3} N (2.2 x 10^{-4} lb_f). Chamber temperature is roughly 1930°C (3500°F). TRW considers this system as an alternative to ion engines for spacecraft attitude control.

The Avco design specifies a "blowdown" thrust from 0.13 N (0.03 lb_f) to 0.04 N (0.01 lb_f). The average I_{sp} is approximately 300 sec. A power input of 1.1 to 1.6 watts per 10^{-3} N (2.2 x 10^{-4} lb_f) is required. Chamber temperature is unavailable at the present time.

Recommended values for augmented electrothermal systems are 305 sec for I_{sp} and a power input of 1.3 watts per 10^{-3} N.

2.6 Ion Propulsion

Concept definition and analysis studies for solar electric propulsion stages (SEPS) were completed by Boeing and Rockwell in early 1975.[2-6, 2-7] These systems employed the Hughes 30-cm ion engine and have been used in this study to define typical performance parameters for similar primary propulsion modules.*

TRW is currently evaluating potential applications for ion engines in the areas of attitude control and auxiliary propulsion.[2-8] An 8-cm engine is the baseline thruster used in the TRW study. The performance parameters associated with primary and secondary ion propulsion systems are discussed in the following subsections.

2.6.1 SEPS (Primary Propulsion)

Performance parameters for the Hughes 30-cm ion engine are shown in Table 2-4 for four power levels. From these data, an I_{sp} of 3000 sec appears reasonable for initial sizing purposes. According to Boeing analyses, performance of the system is relatively insensitive to reasonable deviations from this value.[2-6]

*Data on SEP configuration modifications resulting from the recent Halleys Comet activities were not available in time for inclusion in this study. It is not believed that they would significantly alter the indicated results.

TABLE 2-4. TYPICAL PERFORMANCE DATA FOR THE HUGHES 30-CM THRUSTER FOR FOUR POWER LEVELS [2-6]

Power in Thruster Element	Full Power			73% Power			43% Power			25% Power		
	V, volts	I, amp	P, watts	V, volts	I, amp	P, watts	V, volts	I, amp	P, watts	V, volts	I, amp	P, watts
Beam	1100	2.00	2200	1100	1.45	1595	1100	0.85	935	1074	0.50	550
Total Power, watts		2625			1912			1134			692	
Electrical Efficiency, %		83.8			83.5			82.5			79.5	
Propellant Efficiency, %		94.6			90.7			79.2			63.9	
Total Efficiency, % (Meter Value)		79.3			75.7			65.3			50.8	
Effective Specific Impulse, sec		2990			2888			2556			2083	
Thrust, Newtons		0.128			0.094			0.056			0.033	

The overall electrical efficiency is of major importance in an ion propulsion system. The most important driver of this value is the thruster efficiency. Table 2-5 compares the AST* data supplied to Boeing and Rockwell. From this information, a value of 0.718 was selected for thruster efficiency. This value is shown in Table 2-6 along with the other factors contributing to the overall electric propulsion system (EPS) efficiency.

TABLE 2-5. AST DATA COMPARISON(2-6)

DATA SOURCE	EFFICIENCY * (2.0 A)	INPUT POWER W.
HRL ENGINEERING MODEL THRUSTER	0.716	2631.5
JPL SN 403 (GRID SN 638)	0.657	2765.7
THRUSTER CONTROL DOCUMENT (JPL)	0.72	2600

*CORRECTED FOR Hg++ AND BEAM DIVERGENCE

TABLE 2-6. BASELINE EPS PERFORMANCE SELECTION(2-6)

ASSEMBLY	EFFICIENCY *
THRUSTER	0.718
CABLING	0.999
POWER PROCESSOR	0.9**
EPS OVERALL	0.646

*EXTRAPOLATED TO 2.1A

**MSFC GROUNDRULE

*Advanced Systems Technology (AST) is a NASA/OAST program designed to bring electric propulsion technology to a flight-ready status.

Selection of the proper SEP power level for a given primary propulsion application is a complex process dependent upon such factors as desired payload, mission difficulty, allowable flight times and costs associated with the power system and flight operations. For this study, it is assumed that SEPS-type hardware is available. Previous studies have examined the feasibility of using SEPS hardware for low-Earth orbital missions, including servicing. The assumed power levels for these analyses were 21 kw and 15 kw. These levels were the same as assumed for SEPS in geosynchronous delivery and planetary mission analyses, and were used in the previous low-Earth orbit analysis to keep a common SEPS configuration. For the MMS, the SEP power level should be selected to maximize MMS performance and cost effectiveness. Since the SEPS thrusters and power processors assumed for this study were modularized in 3 kw units, the MMS power level will be some multiple of 3 kw. Initial estimates of the MMS power level selected a value of 6 kw for the servicing mission. Preliminary performance analyses are being performed using this assumed power level. Additional analyses may indicate that some other power level (e.g., 3 kw, 9 kw) might be more desirable. The primary trade involved is that increased power will increase performance, but at the cost of increasing the initial mass and array costs. If a change in assumed power level appears desirable, the SEPS evaluation will be made using the revised value.

The mass properties assumed for a SEPS stage are shown in Table 2-7. The numbers represent a system composed of two 30-cm thrusters, two power processing units (PPU), a switching matrix for interconnection of the thrusters and PPU's, and a 6-kw solar array (60 m^2). One item not included is the low-thrust propellant subsystem for which the mass is mission-dependent. This system includes the mercury tanks, pressurant, valves, feedlines, transducers, and a control module. Based on the previous SEPS studies, an expended mass fraction of 0.965 has been selected for the propellant subsystem.

2.6.2 Enhancement of Reliability

The long thrust periods required with SEPS may introduce the possibility of one or more components failing during the mission. To alleviate this situation, it may be desirable to carry an additional thruster and power processing unit which would be used only in the event of a failure in one of the prime components. The mass penalty would be approximately 23.7 kg (52.2 lb). A switching matrix for interconnection of the thrusters and power processing units was included in the initial SEPS mass statement and is therefore not an additional item under this approach.

TABLE 2-7. SEPS MASS SUMMARY

Item	Mass, kg	(lb)
Thrusters	15.6	(34.4)
PPU's	31.8	(70.1)
Switching matrix	5.0	(11.0)
Solar array	90.3	(199.2)
Subtotal	142.7	(314.7)
Contingency (15%)	21.4	(47.2)
Total	164.1	(361.9)

Since the additional thruster and power processing unit are not normally used, there is no need for additional solar array area. The relatively small mass penalty associated with this concept may make it an extremely attractive option in terms of the enhanced system reliability. Both Rockwell and Boeing have used similar schemes in their studies of ion propulsion systems.

2.6.3 Attitude and Velocity Control System (Secondary Propulsion)

For attitude control and stationkeeping of an Earth orbital spacecraft, NASA/Lewis is sponsoring research on an 8-cm ion engine.[2-8] Operational characteristics of this thruster are presented in Table 2-8. These data provide a reasonable estimate of system performance for use in a sizing analysis.

TABLE 2-8. OPERATING CONDITIONS FOR AN 8-CM THRUSTER(2-8)

Thrust (ideal), mN	5.1
Specific impulse, sec	2955
Total input power, w	125.4
Total efficiency, percent	58.8
Power efficiency, percent	68.9
Beam current, J_B, mA	72
Output beam power, w	86.4
Accelerator voltage, v_A, v	-300
Power/thrust, W/mN	24.6

The mass characteristics for a spacecraft attitude control system are shown in Table 2-9. The addition of approximately 20 kg (44.1 lb) of mercury to this total should provide sufficient capability to maintain an INTELSAT V class spacecraft on-orbit for 7 years. As a result of the relatively low power input (∿400w) required for this system, large dedicated solar arrays are not necessary. The electric power for the attitude control function can be obtained without design change on most advanced communications satellites. For 3 or 4 years, the power would be available from the excess in the spacecraft solar array. For the remainder of the orbital lifetime, power would be obtained from the spacecraft batteries.

2.7 Operational Considerations

Each of the propulsion technologies previously described has operational characteristics which may limit its consideration for certain missions. Extended space missions, which require multiple operations of the propulsion system would require major design modifications for most existing chemical propulsion systems. Some of the propulsion systems are more readily adapted to meet this requirement than others, but additional factors can and will influence concept selection. The areas of concern for each technology are summarized in the following paragraphs.

TABLE 2-9. ATTITUDE CONTROL SYSTEM MASS[(2-8)]

Item	Unit Mass, kg	(lb)	Quantity Required	Mass, kg	(lb)
Thruster & gimbal[(a)]	3.4	(7.5)	4	13.6	(30.0)
Reservoir[(b)]	1.5	(3.3)	4	6.0	(13.2)
Power electronics unit	6.7	(14.8)	4	26.8	(59.0)
Digital interface unit	2.3	(5.1)	4	9.2	(20.3)
Controller	2.3	(5.1)	4	9.2	(20.3)
Squib valve	0.1	(0.2)	8	0.8	(1.8)
Filter	0.1	(0.2)	2	0.2	(0.4)
Propellant lines	--	--	2	(c)	--
Cables	--	--	44	1.0	(2.2)
Total dry mass				66.8	(147.2)

(a) Includes temperature sensors.
(b) Includes pressurant, fill valves, pressure sensor, temperature sensor.
(c) Less than 0.1 kg.

Solid rocket motors have two potential areas of operational concern. The first deals with the thrust levels associated with solids. It is not uncommon for spacecraft to experience accelerations of 3 to 10 g's when solid motors are used. These accelerations would preclude use of a solid motor burn while antennas, arrays, etc., were deployed. While it is possible to tailor the thrust level to provide a "softer" ride, the cost of development and qualification of such motors must be considered. Slow-burning solid propellant motors are not currently available with propellant loadings in the range under consideration. The second potential problem with solids concerns their ability to be stored for long periods of time in space. Problems are encountered with propellant outgassing due to the vacuum, and grain cracking as a result of unsymmetric heating of the case. A possible solution to the outgassing would be to seal the nozzle so as to maintain atmospheric pressure inside the motor. This would increase the complexity of the system and degrade reliability and the motor mass fraction. The heating situation could be resolved by slowly rotating the system to distribute the thermal loads, but this may not be practical from the spacecraft standpoint.

Systems involving the use of hydrazine or Earth-storable propellants in Earth orbits may encounter extended deep space exposure periods during which time the propellant could freeze. This can be countered by increased insulation of the tanks or the addition of heaters. In either case, the dry weight of the propulsion module would be increased. In the case of bipropellant systems, an attractive option would be to substitute hydrazine for the MMH normally used. This would allow the monopropellant attitude control system as well as the main propulsion engine(s) to feed from a common propellant tank. Discussions with propulsion systems contractors indicate that this modification should not be too difficult to accomplish and would not substantially alter system performance. Since this approach has never been used in flight programs, the costs associated with modifying the system may be unattractive. An opposite approach of using MMH for the attitude control function is viable only if non-catalytic monopropellant hydrazine thrusters are used. As in the previous case, this has never been attempted on flight-qualified hardware.

Cryogenic stages suffer from the inverse situation of requiring insulation or cooling to prevent excessive propellant boiloff during exposure to sunlight. The amount of insulation required in this case is significantly higher than would be required for hydrazine or Earth-storable propellants. The addition of a venting system would also be needed to allow extended use of cryogenic propellants in space.

As mentioned previously, a propulsion module employing SEPS might have to contain an additional reserve thruster to enhance system reliability. The extended thrust periods inherent with low-thrust propulsion may make this scheme necessary; however, a mass penalty would be incurred.

Propulsion systems employing bipropellants that are hypergolic (i.e., N_2O_4 and MMH) may require careful design to avoid undesirable safety characteristics in connection with Shuttle operations. This situation does not appear to be prohibitive, in view of the fact that TRW's design for the NASA Tracking and Data Relay Satellite (TDRS) features a bipropellant apogee motor. Likewise, there is substantial precedence for using hydrazine in a main propulsion role, since the majority of spacecraft using the Space

Transportation System (STS) will already contain relatively large amounts of hydrazine. The status of cryogenic propellants for use on payloads using the Shuttle is unclear at this time. There do not ap ar to be any overriding safety or operational characteristics which would preclude the use of cryogenic stages; however, there are no known spacecraft or stages currently under development for STS use that would use this propulsion technology.

2.8 References

(2-1) Design of Multi-Mission Chemical Propulsion Modules for Planetary Orbiters, 26085-6001-TUOO, TRW Systems Group, August 15, 1975.

(2-2) Wright, J. L., IUS/Tug Auxiliary Stage Study, BCL-I/TAS-FR-76-1, Battelle's Columbus Laboratories, February 20, 1976.

(2-3) TRW's Multimission Bipropellant Propulsion System, TRW Systems Group, February 1975.

(2-4) Cryogenic Stage Feasibility, Hughes Aircraft Co., 1975.

(2-5) Kick Stages for the STS-A Preliminary Cost Effectiveness Comparison, Aerojet Liquid Rocket Company, January 23, 1975.

(2-6) Concept Definition and System Analysis Study for a Solar Electric Propulsion Stage, D180-18553-1, Boeing Aerospace Company, January 1975.

(2-7) Concept Definition and Systems Analysis Study for a Solar Electric Propulsion Stage, SD74-SA-0176-2-2, Rockwell International, February 3, 1975.

(2-8) NAS3-20113 Task 1: Spacecraft Designs, TRW Systems Group, November 4, 1976.

3.0 TRAJECTORY/PERFORMANCE ANALYSIS

The trajectory analysis for MMS missions is divided into two areas, chemical propulsion and ion propulsion. In the chemical propulsion analysis (Section 3.1) the velocity additions are assumed to be impulsive; a low thrust trajectory code is used in the ion propulsion analysis (Section 3.2). The results of the trajectory analysis are used to determine appropriate stage sizes for hydrazine and bipropellant (Section 3.3).

3.1 Chemical Propulsion Trajectory Analysis

The goal of the trajectory analysis is to determine the ΔV requirements for the various satellites that may use the MMS bus. The ΔV requirements are an input to propulsion system sizing for technologies other than ion propulsion; in turn, sizing is needed to do costing. The most general mission profile is that of the satellite that is launched by the Shuttle and later serviced on-orbit. In this mode, the spacecraft supplies propulsion to go from the Shuttle to its operational orbit, then for servicing a round trip from its operational orbit to another Shuttle flight. The approach used in determining ΔV requirements in this study is first to establish what Shuttle flights are available to do servicing, then determine which satellites can be serviced from which Shuttle flights and, finally, identify the trajectories and corresponding ΔV's which best accomplish the servicing.

3.1.1 Shuttle Shared Flight Availability

There are several problem areas which must be examined to determine how real the possibility is of using a shared Shuttle flight for servicing. These include establishing (1) whether the required flight support equipment can fit in the Shuttle bay and satisfy the center of gravity requirements, (2) whether the flight support/module exchange equipment can be integrated into the Shuttle cargo in sufficient time, and (3) whether a launch window can be found which satisfies the requirements of both the existing cargo and the rendezvous requirements of the spacecraft already in orbit. The Shuttle load factor analysis is not part of this study; therefore, it will be assumed that there is space available. The second problem, whether the necessary equipment can be integrated in a single payload in the required time, is not part of this study. Nevertheless, it is necessary in some cases to estimate the required time from when it is determined servicing is required to when the Shuttle can actually do the servicing. The cargo integration time will be one of the factors that influence this time. In this study, it will be assumed that the minimal time from the decision to service to when the Shuttle can actually rendezvous with the spacecraft is 4 months. The third problem, launch window compatibility, will be discussed later.

3.1.2 ΔV Requirements for Servicing and Return Modes

The ΔV requirements analysis for satellite servicing is based on certain general assumptions which will apply regardless of the type of satellite being serviced or the launch site. Two modes of servicing are considered. In the first mode the satellite is serviced with the Shuttle by replacing failed modules. In the second mode the satellite is returned to Earth, refurbished, and later relaunched. For the on-orbit servicing mode, spacecraft propulsion is required for each of three mission phases: initially placing the satellite in orbit from the Shuttle; returning the satellite to the Shuttle when servicing is required; and replacing the satellite in orbit after servicing is complete. The total ΔV which must be supplied by the propulsion system is the sum of the ΔV's required for each of three mission legs. (The ΔV required for phasing when the satellite and Shuttle rendezvous is assumed to be negligible.) For the Earth refurbishment mode of operation, a complete mission consists of only the first two of these mission phases, placing the Satellite in orbit and returning it to the Shuttle for pickup.

In this study it is assumed that, when the satellite is initially placed in orbit, the Shuttle orbit is in the same plane as the desired satellite orbit. This minimizes the ΔV required for the initial up-leg since it can be achieved by a simple Hohmann transfer. The ΔV (km/sec) required is:(3-1)*

$$\Delta V_1 = \sqrt{\frac{\mu}{r_o}} \left[\sqrt{\frac{2(r_s/r_o)}{1 + (r_s/r_o)}} \left(1 - \frac{r_o}{r_s}\right) + \sqrt{\frac{r_o}{r_s}} - 1 \right], \qquad (3\text{-}1)$$

where

μ = Earth's gravitational constant (398,601 km^3/sec^2)
r_o = radius of Shuttle orbit (km)
r_s = radius of satellite orbit (km).

Computation of the ΔV requirements for the other two legs is more complicated and depends on the type of satellite being serviced and the inclinations of the satellite and Shuttle orbits.

The estimated Shuttle launches for the time period 1981-1991 were taken from the December 1976 Working Draft of the STS Traffic Manifest 1980-1991.(3-2) While these flights are not actual planned missions, the level of activity is representative of what the Shuttle launch activity might be in that time period. Table 3-1 shows the non-Spacelab/non-DoD launches to the four standard orbits used in the manifest.

Each of the four inclinations shown in Table 3-1 puts different constraints on the set of Shuttle flights which could be used for servicing. For 45-56 deg and 90 deg, there are not enough flights to make the assumption of a shared Shuttle launch for servicing reasonable. Thus, in determining the propulsion requirements for servicing missions with inclinations in these two ranges, a dedicated Shuttle flight will be assumed.

Many of the payloads launched to 28.5-deg inclinations are communications satellites which have launch window constraints. Therefore, the primary consideration in flight sharing at this inclination is compatibility of the launch windows of the satellite being launched and the servicing mission. Appendix A contains a discussion of launch window analysis. Briefly, this analysis shows that a convenient parameter used in evaluating launch windows for communication satellites is the longitude of ascending node of the parking orbit. Figure 3-1 shows

*References, denoted by superscript numbers, are at end of section (Subsection 3.4).

TABLE 3-1. NON-SPACELAB/NON-DOD SHUTTLE LAUNCHES

Year	Number of Launches for Indicated Inclination 28.5 Deg	45-56 Deg	90 Deg	100 Deg
1981	4	0	0	0
1982	2	2	0	0
1983	9	0	0	2
1984	8	0	0	3
1985	9	0	0	3
1986	8	0	1	3
1987	7	1	1	4
1988	11	0	1	5
1989	11	1	0	3
1990	13	0	0	5
1991	9	0	1	3

a representative launch window for a communications satellite. Overlayed on this is the longitude of nodes of a spacecraft in a 160-nmi, 28.5-deg orbit. From this figure it can be seen that the spacecraft in orbit and the communications satellite will have the same longitude of ascending node, and hence the same window during the day, about every 20 days. The windows will overlap for 2 to 3 days. For a spacecraft at a different altitude, the slope of its line of nodes will be different. Thus, for a spacecraft returning to a 160-nmi orbit from a higher orbit, there would be increased flexibility in rendezvousing with Shuttle by varying the time when the spacecraft came down to 160 nmi. This flexibility and an assumed minimal flexibility of the Shuttle launch date imply that the propulsion requirements for servicing a spacecraft in a 28.5-deg orbit will not be significantly greater than what is required for a Hohmann transfer between the orbits and minor terminal phasing requirements.

Appendix A contains launch windows for a number of specific communications satellites that will be (or may be) launched from the

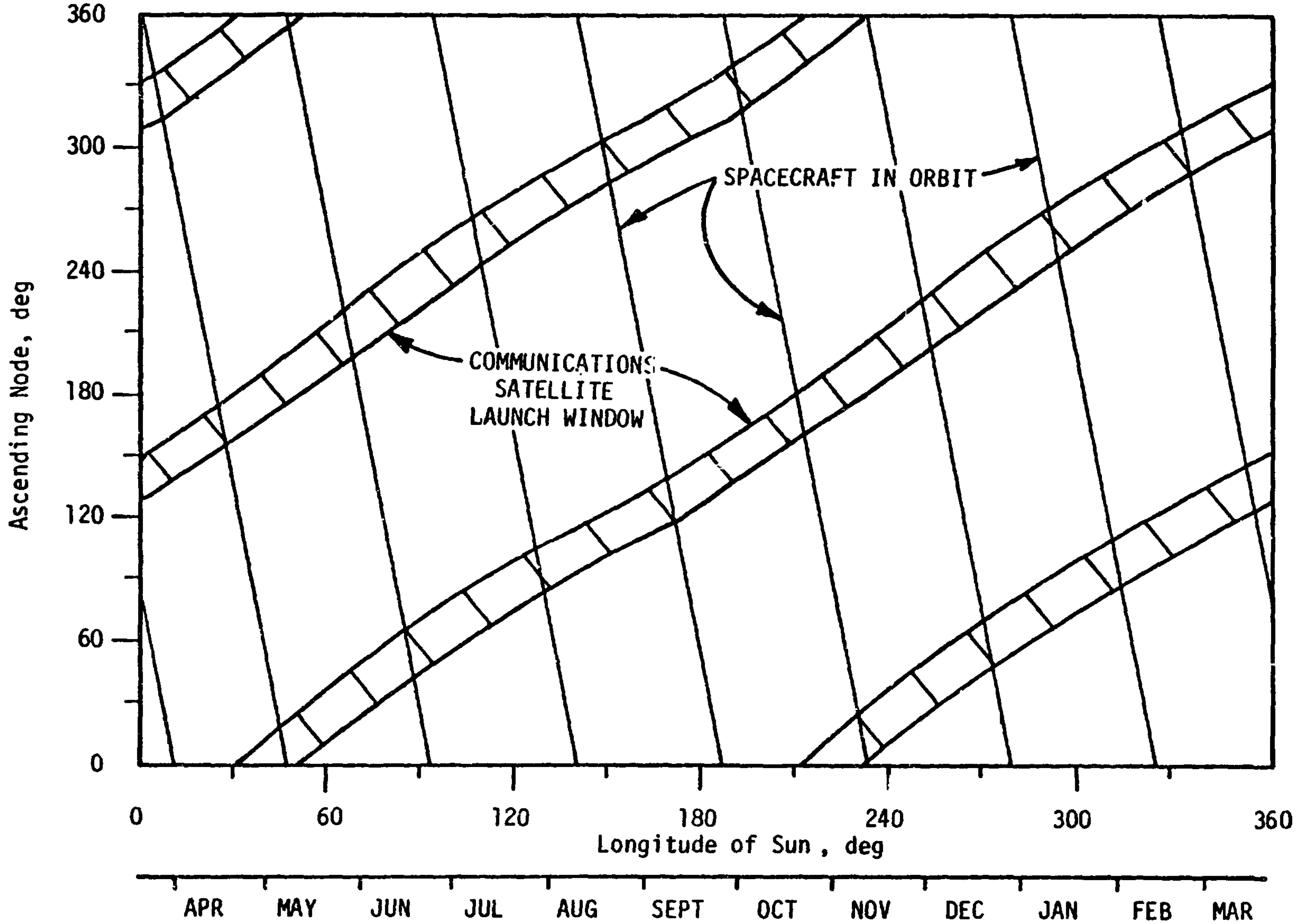

FIGURE 3-1. COMMUNICATIONS SATELLITE LAUNCH WINDOW

Shuttle. It can be seen that, if the spacecraft longitude of node lines were overlayed on these, as in Figure 3-1, the same general conclusions would be drawn.

3.1.2.1 Servicing Sun-Synchronous Missions. Servicing of satellites in 100-deg inclination orbits presents special problems. Most such satellites are in Sun-synchronous orbits; that is, they precess at the rate of 360 deg per year so that the local time at the ascending node is always the same. A full description of the analysis of servicing such missions is rather lengthy, and appears in Appendix B. The analysis starts with the assumption that satellites in 100-deg orbits are serviced by Shuttle flights in 100-deg orbits. This means that Sun-synchronous satellites are serviced by flights which launch other Sun-synchronous satellites. The analysis shows that the primary determinant of the ΔV requirement is the difference between the local times of the ascending nodes of the satellite being serviced and the satellite being launched. The larger this difference is, the more ΔV is required. Servicing a satellite with a 9 a.m. ascending node from a Shuttle flight that launches a satellite into a 9 a.m. orbit requires little more than simple Hohmann transfers since the satellite and Shuttle orbits are nearly coplanar. Servicing this same satellite from a Shuttle flight that launches a satellite into a 3 p.m. orbit will require considerably more ΔV, as explained in Appendix B.

The ascending node times for the Sun-synchronous missions included in the service-oriented MMS mission model are listed in Table 3-2.

As explained in Appendix B, the satellite is brought into an orbit with the same ascending node time as the Shuttle orbit by first placing it into an intermediate parking orbit which precesses at such a rate that when the Shuttle arrives on orbit the satellite's line of nodes is the same as the Shuttle's. If the satellite is serviced on-board the Shuttle it will then be placed into a second parking orbit which precesses back to the original satellite line of nodes. In this study, the sum of the two times spent in the parking orbits plus the time on-board

TABLE 3-2. ASCENDING NODE TIMES FOR SUN-SYNCHRONOUS MISSIONS

	Ascending Node Time for Indicated Year						
Mission	1984	1985	1986	1987	1988	1989	1990
Landsat	9 a.m.			9 a.m.			
Earth Survey Sat		11 a.m.				11 a.m.	
TIROS-O/P	9 a.m.			3 p.m.			
ITOS F/O			9 a.m.				3 p.m.
Earth Resources Sat			12 noon		12 noon		12 noon

the Shuttle is called the total service time. If the satellite is returned to Earth for refurbishment, only the down-leg parking time is included in the service time, since it is assumed that a separate flight with appropriate launch time is used to replace the refurbished satellite in orbit. In either mode of operation, the total mission velocity requirement depends on the total service time. The longer the service time, the smaller the ΔV requirement.

To determine the velocity requirements for specific missions, sets of curves were prepared showing the velocity requirements that occur for the different combinations of servicing one mission from the launch of another. (These curves appear in Appendix B.) Then a set of Shuttle launches was set up according to ascending node times. These are shown in Figure 3-2 along with the times required for servicing the Earth Survey Satellite, the Earth Resources Satellite and TIROS-O in the on-orbit servicing mode of operation. Figure 3-3 shows the same information for the ground refurbishment mode of operation. It is assumed that 4 months are required to prepare a servicing mission for launch; this establishes the minimum servicing time. If the servicing could be scheduled far enough in advance, this 4-month requirement could be eliminated. The rest of the points on the curves in Figures 3-2 and 3-3 were plotted by finding the Shuttle launch that gives the minimum servicing time for a given

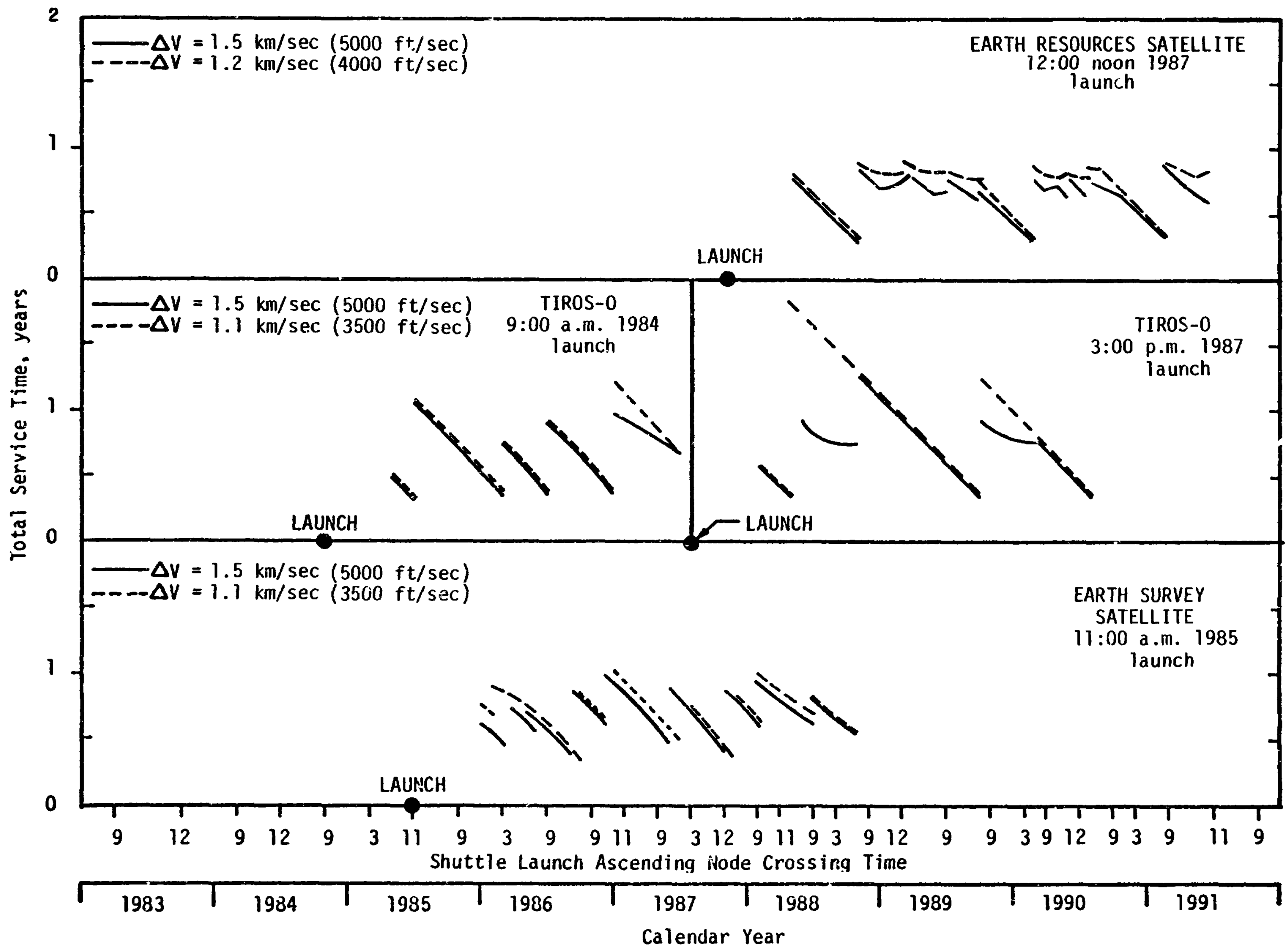

FIGURE 3-2. TOTAL SERVICING TIMES FOR VARIOUS SATELLITES WHEN SERVICING IS DONE ON-BOARD SHUTTLE

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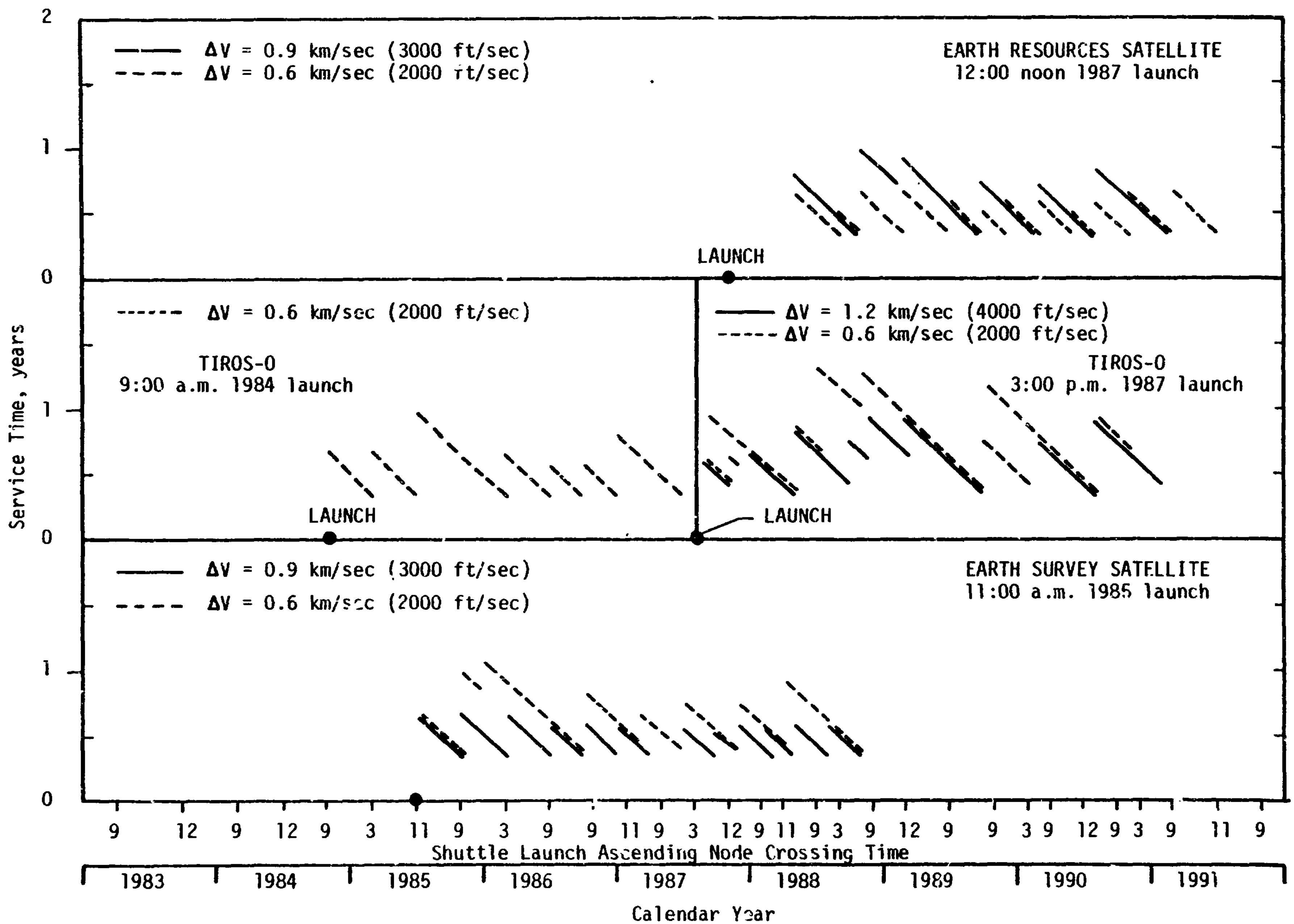


FIGURE 3-3. SERVICE TIMES FOR VARIOUS SATELLITES WHEN SERVICE IS BY GROUND REFURBISHMENT

ΔV at each point in time. For example, referring to Figure 3-2, suppose the Earth Survey Satellite fails in January 1985. Since 4 months are required to prepare a servicing mission, the next 3 p.m. launch cannot be used for servicing since it is only 2 months away. The following 9 a.m. launch can be used, and if a total ΔV of 1.5 km/sec (5000 ft/sec) is available, the total servicing time, including down- and up-legs, is 6.5 months. If the satellite failed at the end of February, the mid-1986 9 a.m. launch would be used, and the servicing time would jump to about 9 months. Following this reasoning and using the curves in Appendix B to find the servicing times, the rest of the points in Figures 3-2 and 3-3 were established. If the average servicing time is to be less than 9 months, then the velocity requirements for Sun-synchronous missions are given approximately in Tables 3-3 and 3-4.

TABLE 3-3. SUN-SYNCHRONOUS SERVICING[a] MISSION ΔVs (On-Orbit Servicing Mode)

Mission	Ascending Node Time	Velocity Requirement, m/sec (ft/sec)	
		Serviced by Shared Flight	Serviced by Dedicated Flight
Earth Survey Satellite	11 a.m.	1065 (3500)	670 (2200)
TIROS-O	9 a.m.	1065 (3500)	850 (2800)
TIROS-P	3 p.m.	1525 (5000)	850 (2800)

(a) Average servicing time approximately 9 months.

3.1.2.2 Servicing the All-Weather Microwave Satellite. Current plans for the proposed Canadian All-Weather Microwave Satellite call for a 795-km altitude, 85-deg inclination orbit. Since there will be very few Shuttle flights to orbits near this inclination, the All-Weather Microwave Satellite will have to be serviced either from a dedicated Shuttle flight or from a flight that launches a Sun-synchronous satellite. Since these flights have inclinations near 100 deg, a considerable plane change would be required, which in turn would require a large velocity

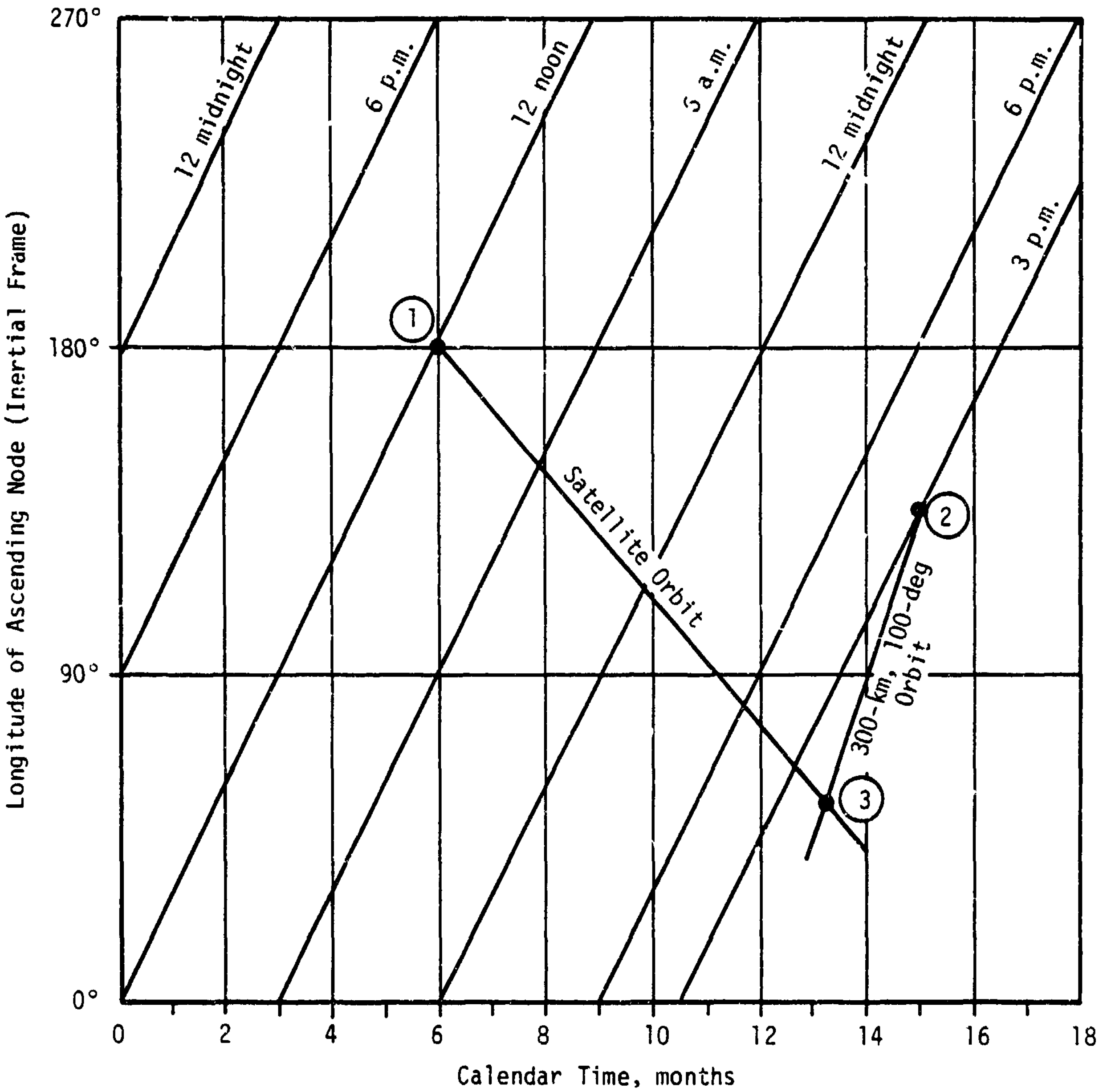
TABLE 3-4. SUN-SYNCHRONOUS SERVICING[a] MISSION ΔVs
(Ground Refurbishment Mode)

Mission	Ascending Node Time	Velocity Requirement, m/sec (ft/sec) Serviced by Shared Flight	Velocity Requirement, m/sec (ft/sec) Serviced by Dedicated Flight
Earth Survey Satellite	11 a.m.	610 (2000)	450 (1470)
TIROS-O	9 a.m.	610 (2000)	570 (1870)
TIROS-P	3 p.m.	1070 (3500)	570 (1870)

(a) Average servicing time approximately 6 to 9 months.

increment. On the down-leg of a servicing mission, i.e., when the satellite is brought down to the Shuttle for service, a plane change is also required to change the line of nodes of the satellite orbit so that it aligns with the Shuttle orbit. Since the satellite orbit is not Sun-synchronous, its line of nodes may not be required to be in any special orientation; so on the up-leg of a servicing mission, no plane change would be needed to correct the line of nodes.

It so happens that the plane change to correct the line of nodes can be accomplished without expending propellant. This is done by taking advantage of precession. By properly choosing the time at which the satellite is moved from its 795-km, 85-deg orbit into a 300-km, 100-deg orbit, the satellite can be scheduled to arrive at the proper line of nodes to rendezvous with the Shuttle. Figure 3-4 shows how this is done. The vertical axis of the graph is longitude measured in an Earth-centered inertial reference frame. The horizontal axis is time in months. The lines labeled with times of day show how the longitude of a point on the Earth's equator with a particular local time varies as the year progresses. These lines have a slope of 360 deg/year (0.9856 deg/day). The time origin is chosen arbitrarily as the vernal equinox. Suppose that the satellite fails 6 months after the vernal equinox, and at this moment is in an orbit whose ascending node has a 12 noon local time. The satellite, therefore, is at Point 1 on the graph. Suppose further that



(See text for explanation of Points 1, 2 and 3)

FIGURE 3-4. GRAPH FOR LINE-OF-NODES COMPUTATIONS FOR SERVICING ALL-WEATHER MICROWAVE SATELLITE the next Shuttle flight available for servicing is 9 months later, and that its parking orbit will have a 3 p.m. ascending node. This is represented as Point 2 on the graph. The longitude of the ascending node, Ω, of any circular orbit precesses at a rate:

$$\dot{\Omega} = -9.97\left(\frac{R_e}{R}\right)^{3.5} \cos(i), \qquad (3\text{-}2)$$

where

$\dot{\Omega}$ = precession rate (deg/day)

R_e = Earth's radius

R = orbit radius

i = orbit inclination.

Applying this equation to the All-Weather Microwave Satellite, the result is $\dot{\Omega}$ = -0.58 deg/day. This is represented in Figure 3-4 by the line labeled "satellite orbit". It shows how the longitude of the ascending node of this orbit changes with time. The same equation applied to a 300-km, 100-deg orbit yields 1.48 deg/day. This is represented in the figure by the line with the end Points 2 and 3. The maneuver used to rendezvous the satellite with the Shuttle permits the satellite to stay in its existing orbit for approximately 7-1/4 months, at which time it will be at Point 3. Then a two-burn maneuver is used to place it into a 300-km, 100-deg orbit. The burns are done at the equator, so no line-of-nodes change is produced. The satellite will then precess positively, as shown in the figure, and arrive at Point 2 in time to meet the Shuttle. Repeated application of the graph in Figure 3-4 to a variety of satellite and Shuttle ascending node times allows a picture of the general servicing requirements to be built up. The results are shown in Figure 3-5.

Figure 3-5 is similar to Figures 3-2 and 3-3, which show service times for servicing various Sun-synchronous satellites. The ascending node crossing times are shown for a set of assumed Shuttle launches which could service the satellite. The vertical axis shows servicing time in months, where servicing time is defined as the total time from satellite failure until replacement in orbit. This is equal to the time to perform the complete maneuver shown in Figure 3-5 plus a small amount of additional time to replace modules and return the satellite to its original

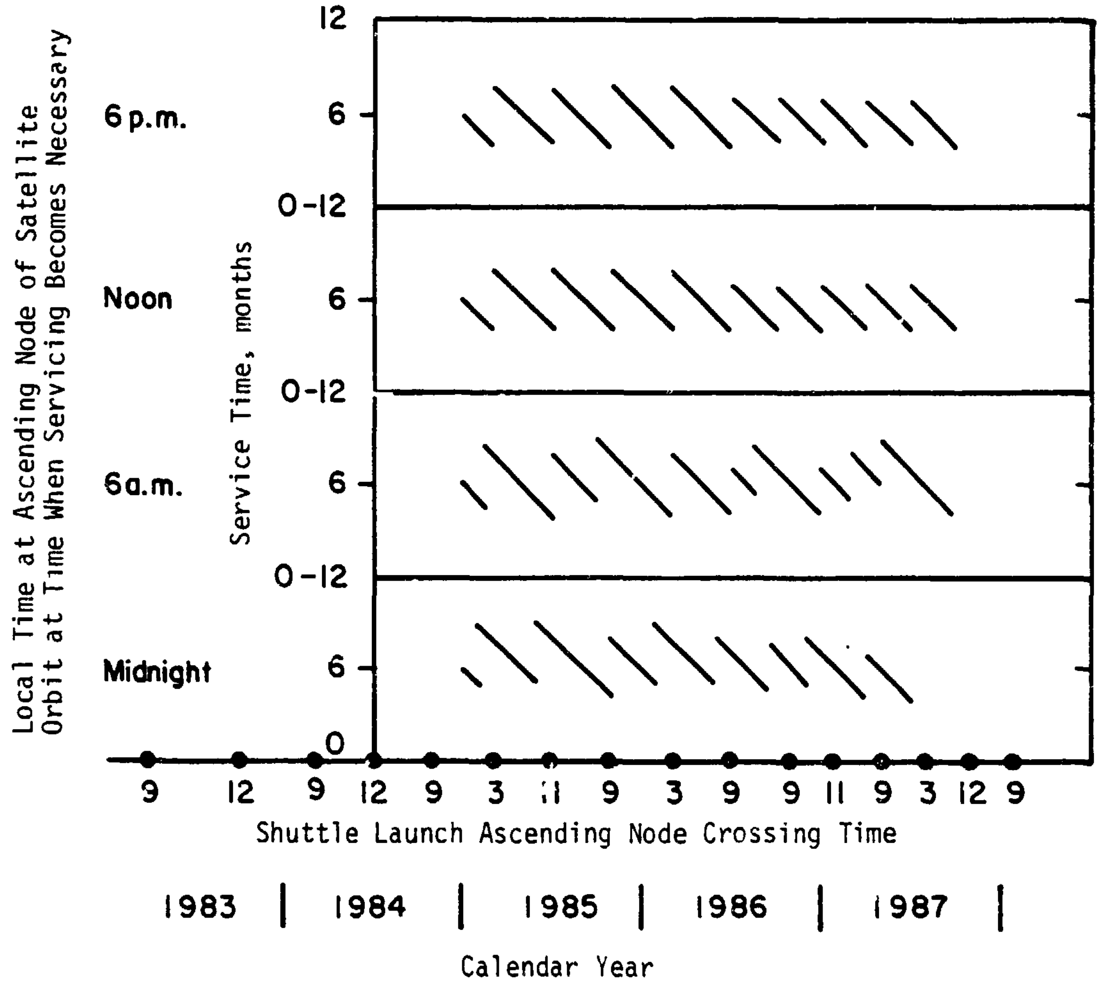


FIGURE 3-5. SERVICING TIMES FOR ALL-WEATHER MICROWAVE SATELLITE ON-BOARD SHUTTLE

orbit. Four different assumptions have been made about the ascending node crossing time of the satellite at the time failure occurs. These are 6 p.m., 12 noon, 6 a.m. and 12 midnight. It can be seen from the figure that this has little effect on the servicing time. The average time is approximately 6 months and the longest time is 9 months.

Because the line-of-nodes change can be accomplished with the help of precession, the ΔV requirements for servicing the All-Weather Microwave Satellite consist only of the velocity changes needed to change altitude and inclination. If this is done using an elliptical transfer orbit with half the plane change done at the first burn and half at the second, a 2.0-km/sec total velocity increment is required to change from a 300-km, 100-deg orbit to a 795-km, 85-deg orbit. If the initial satellite placement is done from a polar orbit (delivery ΔV = 0.71 km/sec), then the total requirement for servicing from a Sun-synchronous orbit is 4.7 km/sec and the requirement for a return to a Sun-synchronous orbit is 2.7 km/sec.

3.1.3 Total Mission Requirements

The total mission requirements include on-orbit velocity requirements in addition to the requirements to go between the Shuttle orbit and the desired spacecraft orbit. The on-orbit velocity requirements are due to attitude control, stationkeeping, drag makeup, and orbital maneuvers.

The upper atmospheric explorers typically carry 600 m/sec of propulsion to maneuver in and out of the upper atmosphere during the mission. These satellites start in elliptic orbits at the beginning of the mission and end up in approximately circular orbits at the end of the mission. Thus, it is assumed for this study that if the satellite is to return to the Shuttle, the nominal on-orbit propellant will be sufficient to enable return to the Shuttle. In a servicing mode, an entire new propulsion module would replace the old module. The Upper Atmosphere Explorer with a 10-deg inclination requires a large initial impulse to change the plane and raise apogee. This impulse is a likely candidate for a solid motor and is identified as a separate requirement from the on-orbit velocity requirements. In a similar manner, the perigee and apogee burns of the Stormsat mission are identified separately. Table 3-5 shows the total mission velocity requirements for the missions in the MMS mission model (Section 1.2).

For the Tiros-P and All-Weather Microwave missions, different options for dedicated or shared Shuttle flights are shown. These options will give some flexibility in the stage sizing.

3.2 Low-Thrust and Secondary Propulsion Analysis

The use of low thrust for primary propulsion applications requires the use of several 30-cm ion thrusters. Typically, the most promising missions for low-thrust applications are the more demanding missions. From Table 3-5, it can be seen that the missions with velocity requirements greater than 1 km/sec are some of the explorers, Stormsat, and the return and servicing of Sun-synchronous missions (and the All-Weather Microwave).

The explorer missions are not well suited for low-thrust applications. For the atmospheric explorers, which can be launched directly into the proper inclination, approximately half of the total velocity requirements are for on-orbit maneuvers. These on-orbit maneuvers involve placing the satellite in an orbit that dips into the upper portions of the atmosphere for a few revolutions and then raising the orbit to be above the atmosphere. Use of low-thrust propulsion would present significant difficulty in raising the orbit properly, when the atmospheric drag at perigee could be much greater than the thrust of the ion system. Additionally, the main purpose of the atmospheric explorer series is to measure the drag of the atmosphere, which would be more difficult with a satellite that is continuously thrusting when there is an uncertainty in the thrust. When inclinations other than those that can be achieved directly by the Shuttle are desired for an atmospheric explorer, a solid motor can be used in addition to the nominal propulsion system. As a result of the above technical problems, plus the cost differential between the hydrazine and ion systems, ion propulsion was ruled out for the atmospheric explorers.

TABLE 3-5. TOTAL MISSION VELOCITY REQUIREMENTS

Mission Name	SSPD Code	Apogee/ Perigee, km	Velocity Requirements, m/sec			
			On-Orbit ΔV	Deploy	Return	Service
Small High Energy Obs	HE-07A	500	—	131	272	358
Large High Energy Obs	HE-08A	370	100	186	463	694
Low Energy Astrophysics	HE-27A	370	—	86	—	—
Solar Maximum Mission	SO-03A	575	—	156	312	468
Upper Atmosphere Explorer	AP-01A	3000/300	600	1248	1248[a]	1248[a]
Upper Atmosphere Explorer	AP-01A[b]	3000/300	600	2648[b] + 610	—	—
Medium Altitude Explorer	AP-02A	10,000/1,000	—	1620	—	—
Landsat F/O D-E	EO-08A	705	8/yr	258	491	1125
TIROS-O	EO-12A	825	12/yr	310	635	1090
TIROS-P	EO-13A	825	12/yr	310	1095	1575
TIROS-P	EO-13A[c]	825	12/yr	310	605	900
Stormsat	EO-15A	Geosync	150	2324 + 1743	—	—
Earth Survey Sat	EO-61A	700	15/yr	255	640	1095
ITOS F/O	EO-64A	825	12/yr	310	635	1090
Earth Resources Sat	EO-65A	555	20/yr	186	420	810
All-Weather Microwave	OPN-02A	795	15/yr	767	2783	4789
All-Weather Microwave	OPN-02A[d]	795	15/yr	—	1494	2211
All-Weather Microwave	OPN-02A[c]	795	15/yr	—	1051	1326

(a) On-orbit velocity used for return to Shuttle orbit, propulsion module replaced during servicing.
(b) Spacecraft requires 10-deg inclination, first velocity requirement is a single burn.
(c) Assumes dedicated Shuttle flight for return or servicing.
(d) Assumes a polar orbit Shuttle flight for return or servicing.

Other explorer series missions, such as AP-02A, have measurement of various properties of the radiation belts as a primary goal. It is these very same belts which are damaging to solar panels, the power source for the ion thrust systems. Thus, this type of explorer mission would also not be well suited for an ion thrust propulsion system.

Previous studies[3-3] have shown that the delivery of small to medium sized satellites to geosynchronous orbit is not cost effective with ion propulsion. Thus, for missions such as Stormsat the normal use of solids appears appropriate. The only missions left are the retrieval and servicing of payloads launched from WTR. The analysis of low-thrust propulsion for these missions is presented in the next subsection (3.2.1). The remainder of this section is concerned with trade-offs and applications involving low-thrust systems.

3.2.1 Low Thrust for Sun-Synchronous Missions

The trajectory analysis using high thrust (Subsection 3.1.2) indicated the time for return to the Shuttle (and return to orbit after servicing) can be significant if the Shuttle flight used for return or servicing is launched at a different longitude of nodes (because of the requirements of another payload, or whatever). The times for a servicing mission range from 4 months to a year or more, as seen in Figures 3-2 and 3-3. The 4-month minimum time is based upon a ground rule that a service or return on a shared Shuttle flight could not be scheduled any sooner than 4 months in advance. This time would be required to integrate the necessary cradles, servicing equipment, etc., into the existing Shuttle cargo to obtain a new Shuttle cargo which satisfies Shuttle center of gravity (c.g.) constraints, etc. This assumes all the cradles and servicing equipment are existing hardware ready to be used.

The low-thrust code used for the analysis was SECKSPOT, developed for GSFC by Draper Labs.[3-4,3-5] The program was developed primarily to evaluate geosynchronous missions; however, the framework is sufficiently general to handle the appropriate constraints associated with Sun-synchronous retrieval and servicing missions. These constraints can be specified by the orbital elements (semimajor axis, eccentricity, inclination, and longitude of nodes) at the beginning and end of the trajectory. However, several problems were encountered in the analysis.

The various mission types considered can be divided into various trajectory legs. One leg which is common to both the retrieval and servicing mission is the return from the operational orbit to Shuttle orbit. The altitudes of Sun-synchronous orbits generally range from 500 to 900 km. To demonstrate a bound on times required to return to Shuttle, the 900-km altitude is chosen. The initial and final conditions are shown in Table 3-6.

TABLE 3-6. INITIAL AND FINAL CONDITIONS FOR RETURN LEG

Orbital Element	Initial Condition	Final Condition
Semimajor axis, km	7278	6674
Altitude, km	900	296
Eccentricity	0	0
Inclination, deg	99	100
Longitude of nodes, deg	-45	0.9856 T_f[a]

(a) T_f is time to rendezvous with Shuttle in days.

These conditions represent a return from a Sun-synchronous orbit with a 9 a.m. local viewing time to a Shuttle that is prepared to launch a satellite into a local noon viewing condition orbit. The final boundary condition on the longitude of nodes is expressed as a product of the number of days to return to the Shuttle orbit and the precession of the longitude of nodes in a Sun-synchronous orbit, since a constant viewing condition of a Sun-synchronous orbit corresponds to an orbit such that the longitude of nodes (measured with respect to the vernal equinox) precesses at the same rate the Earth travels around the Sun. A slight modification to the SECKSPOT code was required to handle a boundary condition depending on the final time. The spacecraft mass used is 950 kg, and the ion propulsion module consisted of two 30-cm thrusters. The mass statement for the ion propulsion module was given earlier in Table 2-7.

In obtaining converged trajectories with SECKSPOT, it is beneficial to first generate a trajectory without considering shadowing; then, using these results as initial guesses, a trajectory can be generated which includes the shadowing effects. The apogee/perigee, inclination, and longitude of nodes for the converged trajectories with and without shadow effects are shown in Figures 3-6 through 3-8. Figure 3-6 clearly shows that these trajectories cannot be realized, since the perigee in both cases becomes less than the radius of the Earth. This is a result of the formulation of the SECKSPOT code in two ways: (1) no constraint on intersecting the Earth is included and (2) the formulation is based upon a time optimal solution which was assumed to be fuel optimal. It should be noted that these restrictions do not impact the use of the program for generation of geosynchronous trajectories, which was the principal purpose of the program. Before discussing the generation of a trajectory that does not go through the Earth (or its atmosphere), some of the problems in using low thrust for these types of trajectories will be discussed.

The critical parameter that drives the altitude below the radius of the Earth and the overshoot in inclination is the constraint on meeting the longitude of nodes in minimum time. As time proceeds, the required longitude of nodes is increasing at the rate of the Earth around the Sun. Thus, the difference between the precession rate of the orbit and the Earth's rate around the Sun is a measure of the rate of achieving the final desired boundary condition. This difference is plotted in Figure 3-9 for circular orbits as contours versus altitude and inclination. From this figure it can be seen that as altitude decreases and/or inclination increases the differential drift rate increases. Thus, to satisfy the desired longitude-of-node constraint in minimum time, it is beneficial to overshoot on both inclination and altitude and then come back to the desired Shuttle orbit. However, it can also be seen from Figure 3-9 that the differential drift rate in the desired Shuttle orbit is positive (approximately 0.5 deg/ day), so that the spacecraft could proceed directly to the Shuttle orbit and coast for a prescribed time before Shuttle rendezvous.

A trajectory going directly to the Shuttle orbit was generated using SECKSPOT by letting the final value of the longitude of nodes be open. From the results of the SECKSPOT Trajectory, the final time can be calculated by:

$$0.9856\ T_f = 0.9856\ (T_B + T_C) = \Omega_{T_B} + 1.4771\ T_C\ , \qquad (3\text{-}3)$$

where

T_f = final time (days)

T_B = thrusting time (days)

T_C = final coast time (days)

Ω_{T_B} = longitude of nodes (deg) at end of thrusting phase.

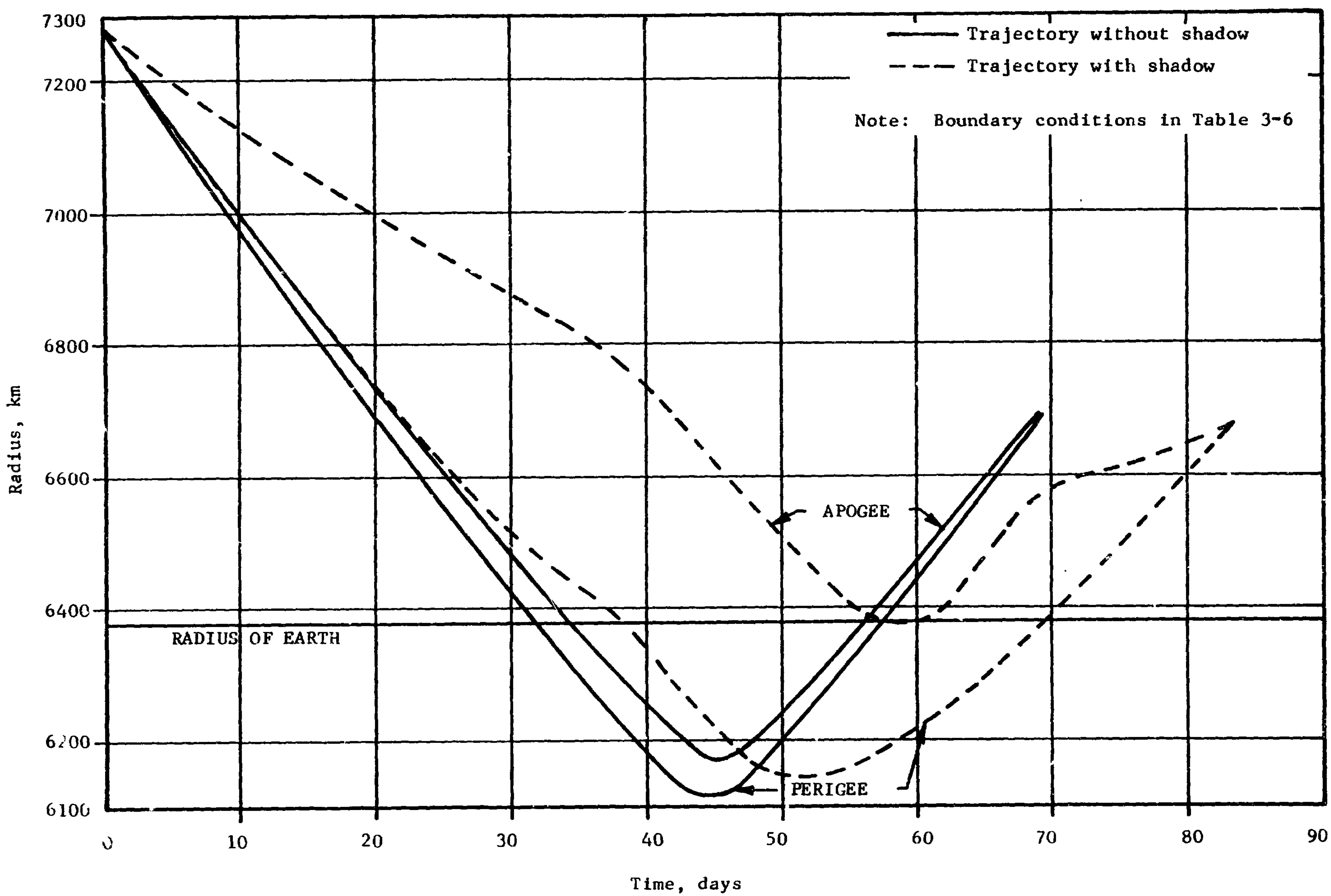


FIGURE 3-6. APOGEE/PERIGEE FOR RETURN TO SHUTTLE WITH CONTINUOUS THRUST

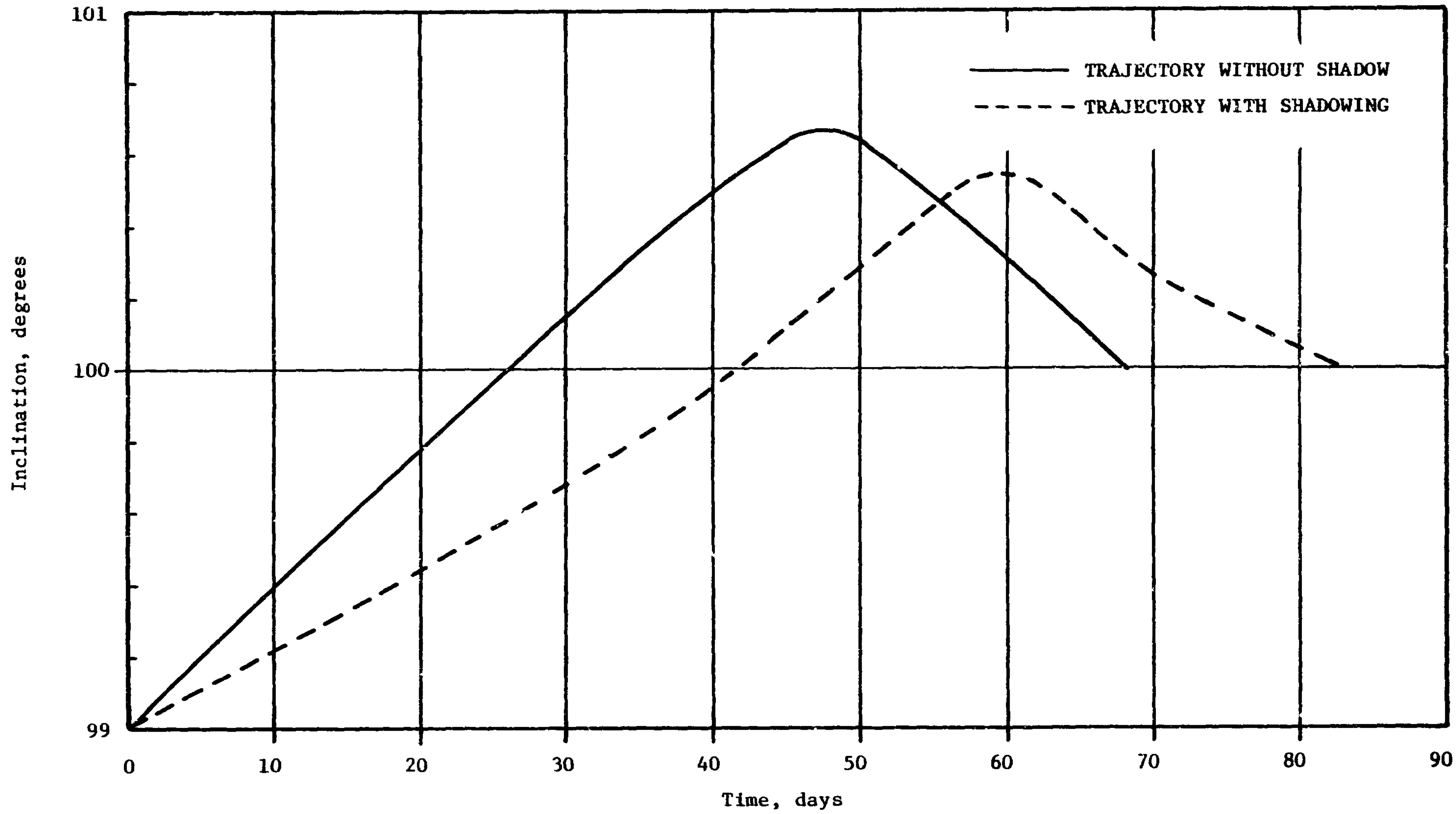


FIGURE 3-7. INCLINATION FOR RETURN TO SHUTTLE WITH CONTINUOUS THRUST

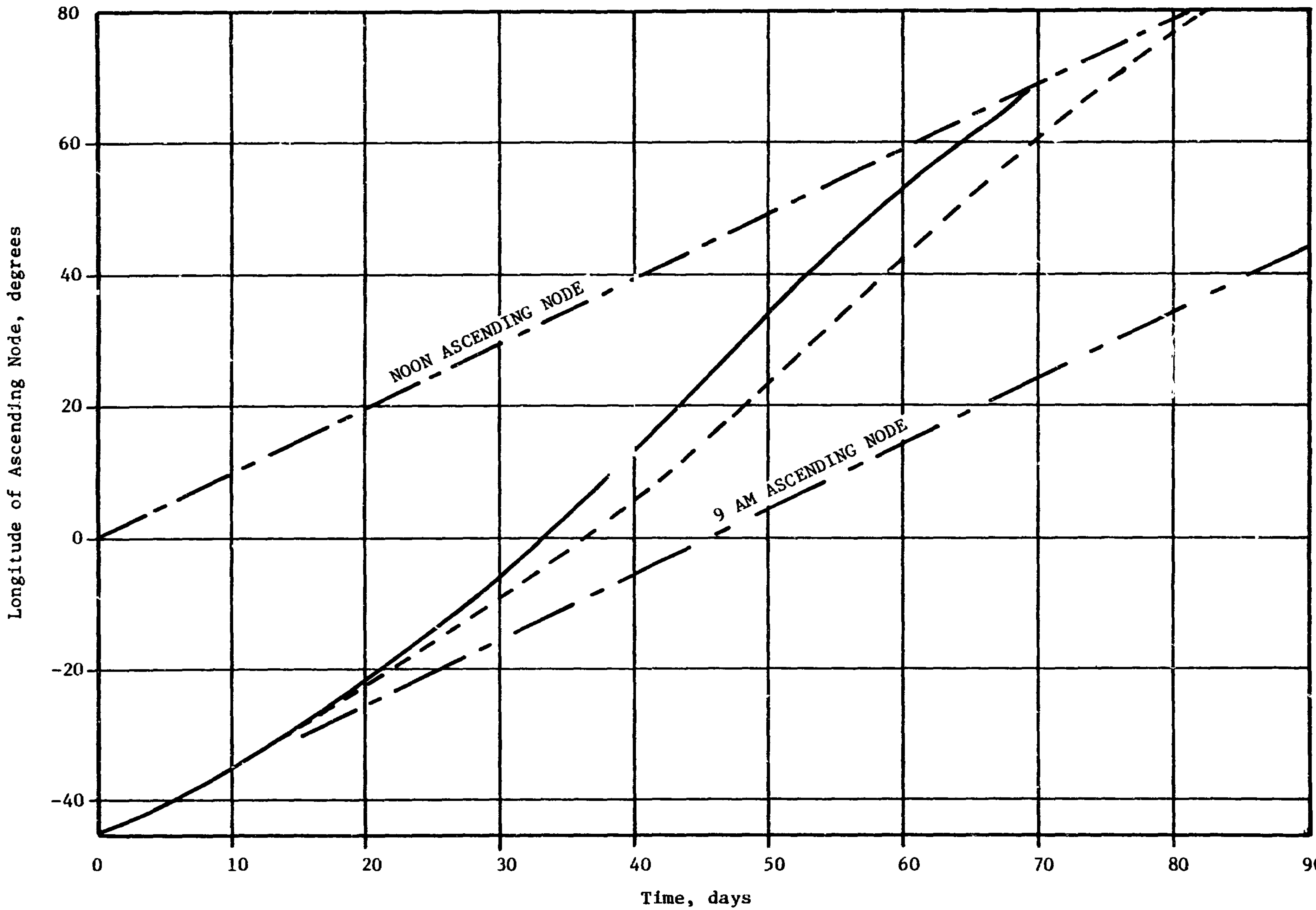


FIGURE 3-8. LONGITUDE OF NODES FOR RETURN TO SHUTTLE WITH CONTINUOUS THRUST

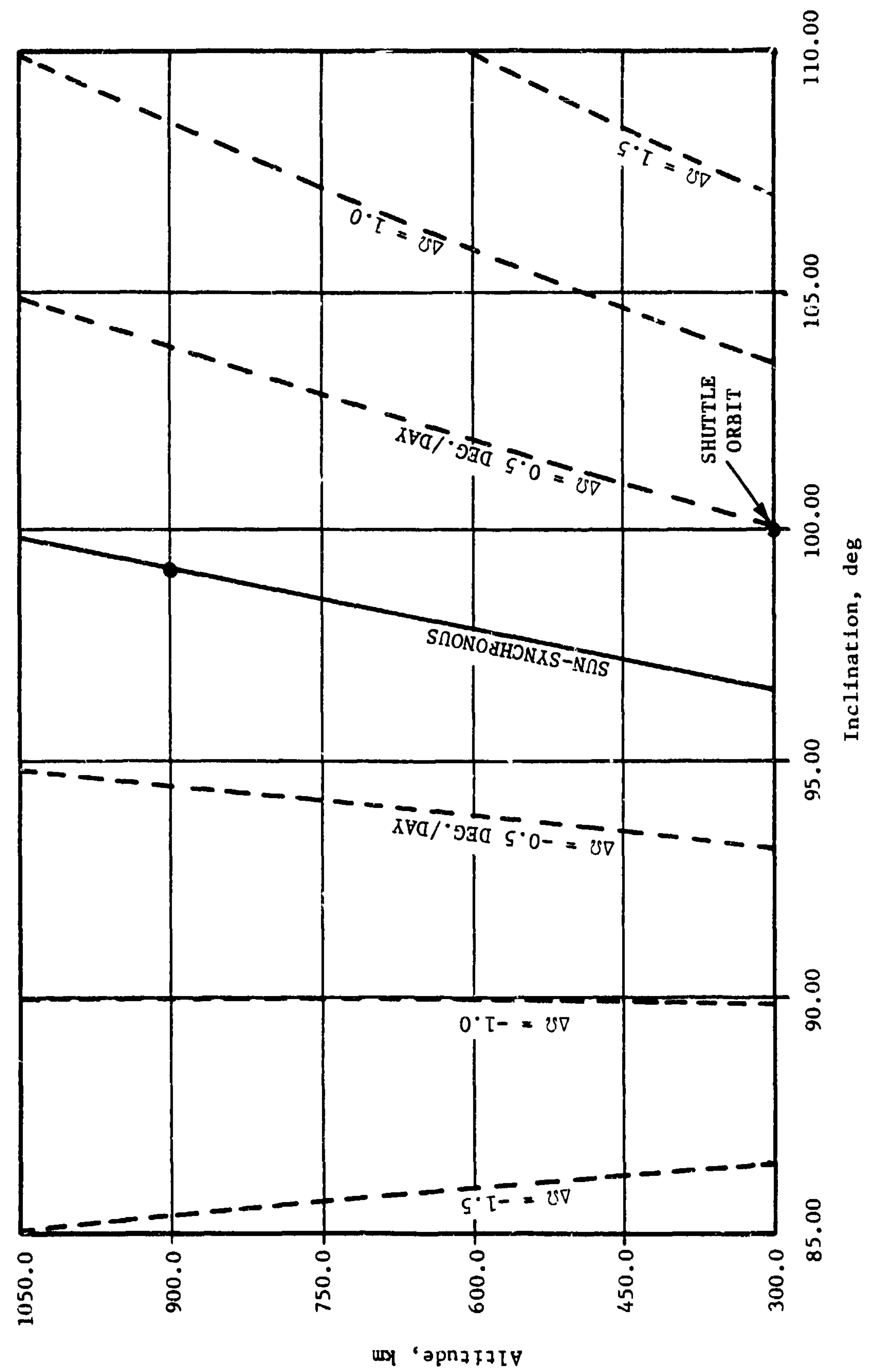


FIGURE 3-9. DIFFERENTIAL DRIFT RATE (FROM SUN-SYNCHRONOUS)

T_B and ΩT_B are obtained from the SECKSPOT trajectory. Note that T_B includes the coast times due to shadowing during the thrusting phase. The total time required for these trajectories is given in Table 3-7.

TABLE 3-7. TIME AND PROPELLANT REQUIREMENTS FOR RETURN TO SHUTTLE

Trajectory Solution	Time, days	Mercury, kg
Continuous thrust (no shadow effects)[a]	68.41	44.9
Continual thrust (with shadowing)[a]	83.22	42.2
Continual thrust followed by final coast	110.06	17.1

(a) These trajectories violate altitude constraints.

The apogee/perigee, inclination, and longitude of nodes for the low-thrust trajectory with a final coast are shown in Figures 3-10 through 3-12. These figures show no overshoot in either inclination or altitude since there isn't any requirement on the longitude of nodes. The propellant requirements (Table 3-7) indicate that the minimum time solution is not the minimum fuel solution. An operational problem with the trajectory with a final coast occurs if the Shuttle launch is delayed. The longitude of nodes will be in error by about 0.5 deg per day of launch delay. Potentially, the Shuttle Orbiter can use its Orbital Maneuver System (OMS) to correct for an error in longitude of nodes. The velocity requirements to change the longitude nodes by the Orbiter are approximately 130 m/sec per degree of node change; thus, the Orbiter could correct for a 1- or 2-day launch delay at most, depending upon how complicated the mission profile is and whether an OMS kit can be added. Since the final coast is about 73 days, the delay in schedule could be caused by a large number of reasons, including a delay in the previous launch, which might be totally unrelated to this mission.

The analysis of low thrust applied to these Sun-synchronous missions must also account for operational considerations. The two items of consideration are the size of the propulsion system and the length of time required by various maneuvers. For example, if the minimum time trajectory were desired on the example discussed, the trajectory generated by SECKSPOT as a minimum time trajectory is not realizable, since the

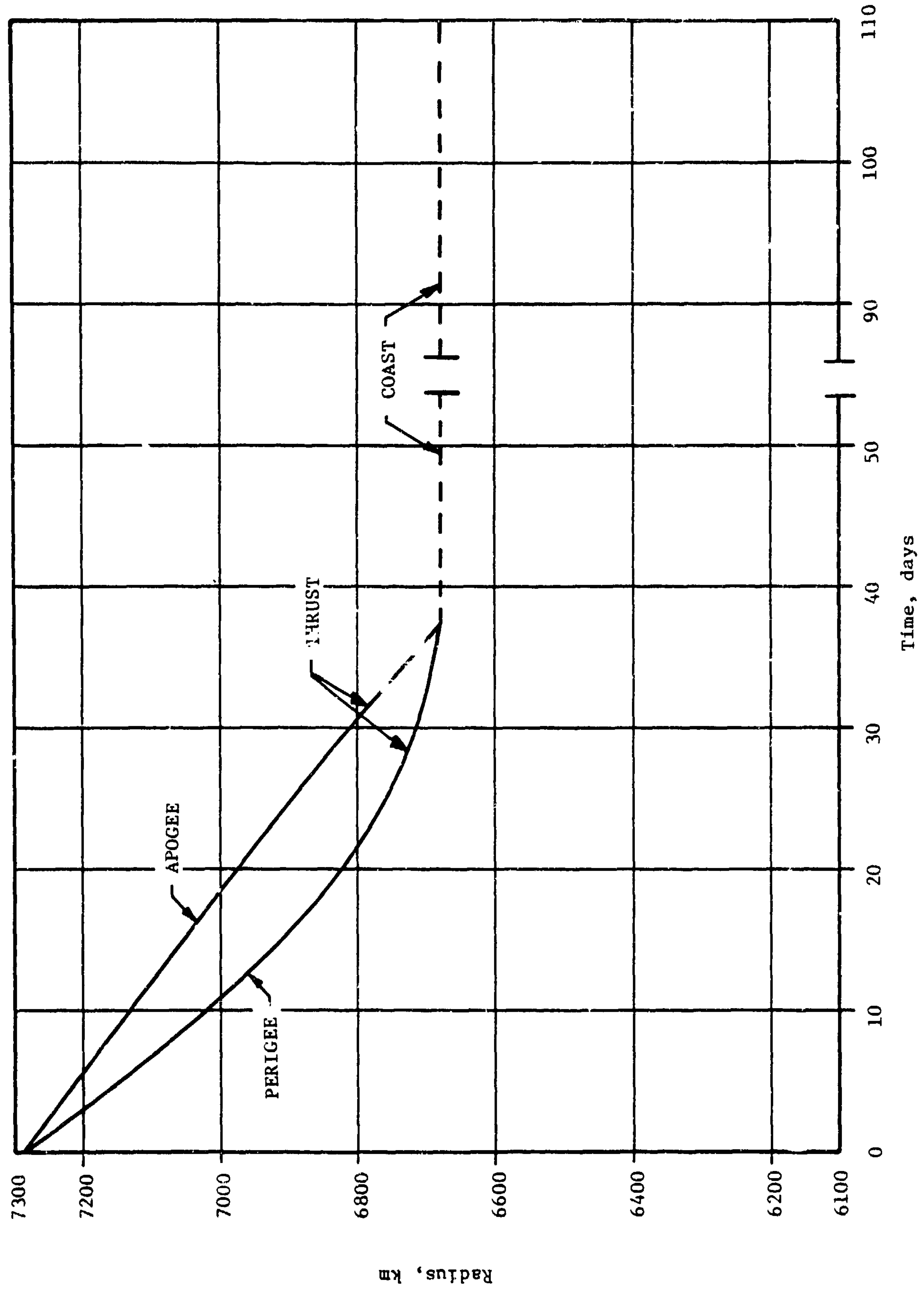


FIGURE 3-10. APOGEE/PERIGEE FOR RETURN TO SHUTTLE WITH COAST

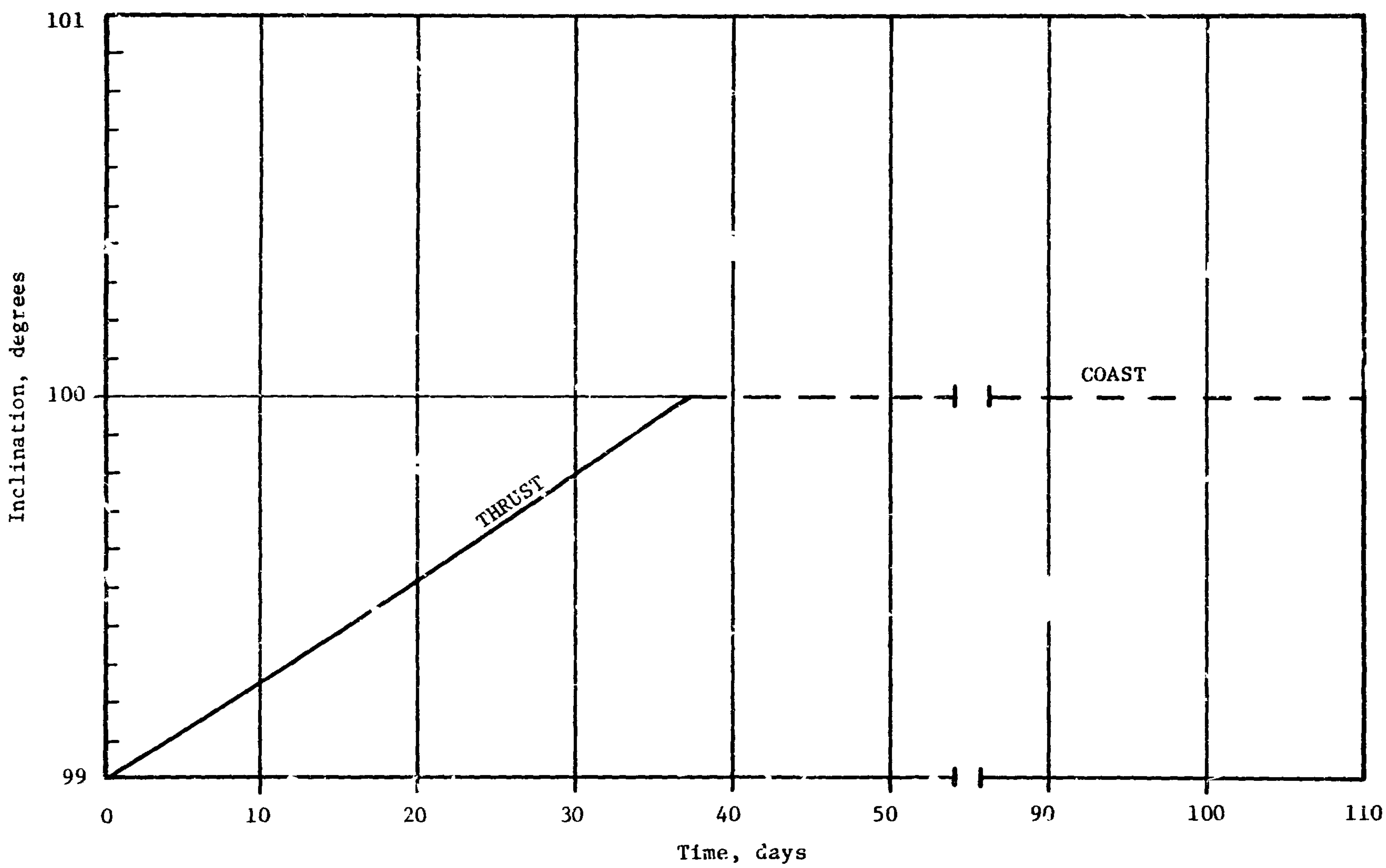


FIGURE 3-11. INCLINATION FOR RETURN TO SHUTTLE WITH COAST

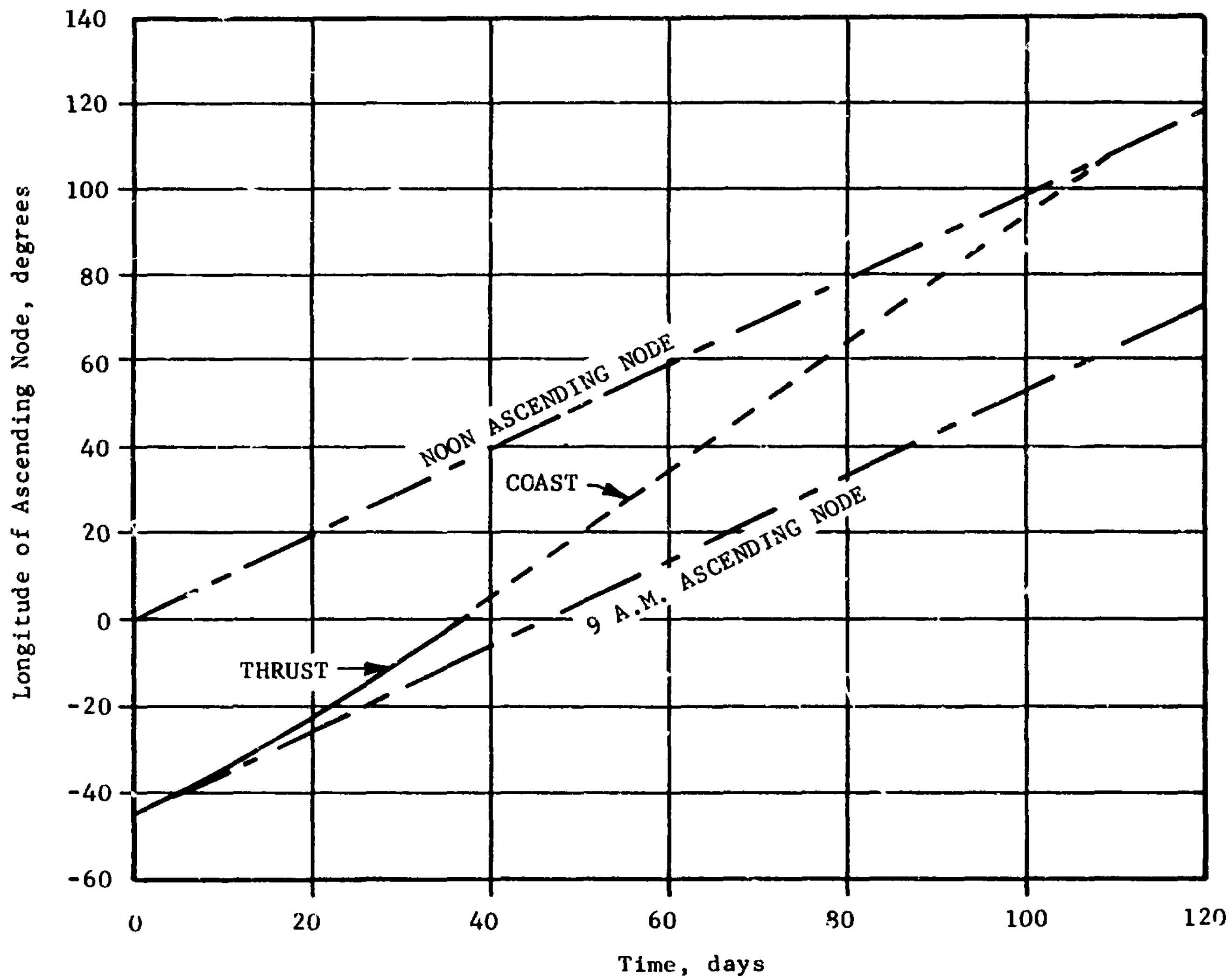


FIGURE 3-12. LONGITUDE OF NODES FOR RETURN TO SHUTTLE WITH COAST

altitude goes below the Earth's surface and the trajectory generated by SECKSPOT with no constraint on longitude of nodes is not time optimal either. The latter trajectory would have a lower time required if the inclination overshot the desired value and then came back. Generation of these types of trajectories with SECKSPOT could be done by putting in a state variable constraint, which might be an extensive modification.

In the high-thrust analysis, it was assumed that the minimum time for a return to the Orbiter (which is not prescheduled) would be 4 months; this time would be needed for Shuttle scheduling, Orbiter cargo integration and testing, etc. A similar assumption would be valid here; thus, for Sun-synchronous orbits with a 3-hr forward shift in longitude of nodes, the mission time associated with the low-thrust system is compatible with that of the high-thrust chemical systems.

Analysis of several low- and high-thrust trajectories is required in order to fully compare the time requirements of a low-thrust system versus a high-thrust chemical system. The computation of low-thrust trajectories using programs such as SECKSPOT tends to be costly, and since this study was not primarily a trajectory study, the actual number of converged trajectories was kept to a minimum.

A possible set of trajectories to evaluate would be those listed in Table 3-8. The 900-km altitude represents the most demanding requirement for Sun-synchronous missions. Additionally, the effect of spacecraft mass and the number of 30-cm ion engines used should be analyzed. From these trajectories, various retrieval and servicing missions could be patched together. For example, an ITOS follow-on is to be launched in 1986 and subsequently serviced on a Shuttle flight that will launch an Earth Resources Satellite in 1988. Realizing that the ITOS orbit is lower than the 900 km, and employing the Trajectory Identification Numbers cited in Table 3-8, a sequence of possible trajectories would be 1, 5, and 7, if the initial Shuttle flight were also launching something with chemical propulsion to the same ascending node condition. Other trajectory legs could replace 1, such as 10, if the ITOS satellite were the controlling element of the Shuttle cargo in determining the launch window constraints, or 7 if the initial launch of ITOS were on a Shuttle flight which was launching a satellite to a noon local time viewing condition. The retrieval of TIROS-P by an Earth Resources Satellite Shuttle flight could be represented by Trajectories 1 and 3. Although the various Sun-synchronous missions in the mission model would not be represented exactly, due to variations in spacecraft mass and mission altitude, bounds on mission times and propulsion system masses could be obtained.

TABLE 3-8. TRAJECTORIES REQUIRED TO EVALUATE LOW THRUST FOR SUN-SYNCHRONOUS RETURN AND SERVICING OPERATIONS

Trajectory Identification Number	Initial Conditions		Final Conditions		Local Crossing Time, Δ hr
	Inclination, deg	Altitude, km	Inclination, deg	Altitude, km	
1	100	297	99	900	0
2	100	297	98.2	500	0
3	99	900	100	297	-3
4	99	900	100	297	0
5	99	900	100	297	+3
6	99	900	100	297	+6
7	100	297	99	900	-3
8	100	297	99	900	+3
9	100	297	99	900	-6
10	100	297	99	900	Open

Generation of converged trajectories using SECKSPOT was not possible for all the cases required because of the cost of the many computer runs necessary to achieve converged trajectories using SECKSPOT and because this is an overall propulsion study, not a trajectory analysis study. However, from the limited data generated, certain basic conclusions can be obtained. The standard Shuttle orbit of 297 km (160 nmi) and 100-deg inclination has a differential drift rate of approximately +0.5 deg/day and, by definition, the Sun-synchronous orbits have a zero differential drift rate. Thus, the trajectories with a positive drift requirement (i.e., Trajectories 5, 6, and 8) have a natural drift rate which will aid in achieving the desired longitude of nodes. However, for those trajectory legs with a negative drift requirement (i.e., Trajectories 3, 7 and 9), the nominal drift of the Shuttle standard orbit is counterproductive to achieving the desired longitude of nodes. To some extent, this is also true of those trajectories with a requirement of no shift in longitude of nodes. Achievement of these trajectories requires that, for part of the total time, the trajectory lies to the left of the Sun-synchronous line in Figure 3-9. This would also be true of a trajectory with a zero shift requirement. This can be seen from the thrusting part of both Trajectory 5 (see Figure 3-12 and Table 3-8), which has about a +10-deg differential shift in longitude of nodes, and Trajectory 10, which has a 5-deg differential shift in the longitude of nodes. The data for Trajectory 10 are shown in Table 3-9. No converged, or partially converged, trajectories were obtained for any of the cases requiring a negative differential drift; however, the attempted cases tended to indicate that the mission times were comparable to those obtained using hydrazine (or bipropellant) systems.

TABLE 3-9. PLACEMENT TRAJECTORY WITHOUT NODE CONSTRAINT

Parameter	Initial Value	Final Value[a]
Time, days	0	21.70
Semimajor axis, km	6674	7276.14
Inclination, deg	100	98.98
Longitude of nodes, deg	0	26.27
Mass, kg	1170	1154.88
Eccentricity	0	0.006

(a) The desired final value of semimajor axis was 7288 km, the desired final inclination was 99 deg, and the desired final eccentricity was 0.

The following comparisons between chemical and low-thrust systems summarize our findings:

(1) The mission times on the return to Shuttle trajectories for low thrust are comparable to those of the chemical systems, and for both systems are less than 4 months (the minimum time assumed for fitting into Shuttle scheduling).

(2) The mission times to return to the desired orbit after being serviced by Shuttle when a shift in longitude of nodes has occurred are approximately the same (or slightly longer) for ion systems as for chemical systems (3 to 6 months).

(3) The mission time on the initial delivery leg is significantly longer for ion systems than for chemical systems (22 days for ion systems versus a few hours for chemical systems for a 900-km final orbit).

(4) Ion propulsion systems have less flexibility with regard to reacting to Shuttle launch delays than chemical systems.

(5) The propellant mass requirements for ion propulsion systems are significantly less than for chemical systems (15 to 60 kg per trajectory segment for ion systems versus 160 to 500 kg per trajectory segment for chemical systems).

Based upon these comparisons, the major advantage of the ion systems is the smaller propellant masses required. The traditional disadvantage of ion systems, long mission times, does not appear to be a disadvantage for the Sun-synchronous application, with the possible exception of the initial deployment. However, the operational flexibility of the ion system compared to the chemical systems in contingency situations has certain drawbacks. A potential application in the Sun-synchronous mission area which uses the best advantages of the ion system, low propellant mass requirements, is the change of on-orbit viewing conditions in addition to placement, retrieval and servicing of the satellite. This application, together with some approximation formulas for low-thrust trajectories, is presented later in Subsection 3.2.3.

3.2.2 Drag Makeup Mission

The drag makeup mission is a long lifetime mission near Shuttle altitude. It is assumed that no propulsion is required for initial satellite placement. The four systems considered for this mission are listed in Table 3-10. Two different spacecraft will be considered. The key parameters in a drag makeup analysis are the spacecraft drag coefficient (C_D), the cross-sectional area (A), and the spacecraft mass ($m_{s/c}$). The spacecraft considered (Table 3-11) are representative of a Scout class spacecraft and a Delta class spacecraft.

TABLE 3-10. PROPULSION MODULES FOR DRAG MAKEUP

System	Thrusters	I_{sp}, sec	Total Thrust, N
A	0.1-lb hydrazine thruster	220	0.445
B	0.1-lb electrothermal hydrazine thruster[a]	320	0.445
C	Two 8-cm ion thrusters	2955	0.01
D	Four 8-cm ion thrusters	2955	0.02

(a) 0.1-lb thruster or equivalent in smaller thrusters.

TABLE 3-11. REPRESENTATIVE SPACECRAFT DATA

Spacecraft Class	C_D	$m_{s/c}$, kg	$A^{(a)}$, m^2
Scout	2.2	100.	0.45
Delta	2.2	1500.	3.75

(a) Area corresponds to cross-sectional area of Scout and Delta shrouds.

The drag (F_D) on a spacecraft due to the upper portions of the atmosphere is given by:

$$F_D = \frac{1}{2} \rho V_R^2 C_D A \quad , \tag{3-4}$$

where ρ is the atmospheric density and V_R is the velocity of the spacecraft relative to the atmosphere. At orbital velocities, the velocity relative to the atmosphere is approximately equal to the orbital velocity. For a circular orbit with an altitude h, the square of the orbital velocity, V^2, is given by:

$$V^2 = \mu/(r_e + h) \quad , \tag{3-5}$$

where μ (= 398601 km^3/sec^2) is the Earth's gravitation parameter and r_e (= 6378 km) is the radius of the Earth.

The density of the atmosphere varies with many parameters including altitude, year, season of the year, time of day, latitude, etc.

The atmospheric model used in this analysis is based upon a model described in Reference (3-6). The model separates the dependencies of the many parameters considered by introducing a reference temperature, T (also called the exoatmospheric temperature), which depends upon time and the location relative to the momentary subsolar point. Then the density is given as a function of altitude and temperature:

$$\rho = \rho(h,T) \quad . \tag{3-6}$$

This relationship is empirically shown in Figure 3-13.

The reference temperature is then expressed by the following relationship:

$$T = f_r \{362 + (f_{DD} + 3.6)F\} + \delta T_a \quad , \tag{3-7}$$

where f_r is a spatial factor depending on the latitude and local time, f_{DD} is a correction factor for a semiannual variation, F is a solar flux index, and δT_a is a temperature adjustment dependent upon a geomagnetic index. When a satellite is orbiting around the Earth, the spatial factor assumes the full range of possible values. Thus, in this analysis, the average value of the spatial factor (1.13) is used. The semiannual effect is shown in Figure 3-14.

The solar flux index and temperature correction due to the geomagnetic index are random variables. They are correlated with sunspot activity. MSFC updates their 10-year forecasts on these indices periodically.(3-7) These forecasts include a nominal (50 percent) and 2σ (95 percent) estimate of the indices for several future dates. The percentage given indicates the probability that the index will be less than the given value. To illustrate the accuracy of these forecasts, both the 1968 and 1976 forecasts of the solar flux are shown in Figure 3-15. The disagreement of the two forecasts in the 1977-79 region could be considered as an error in the 1968 prediction of when the minimum activity would occur. The data used in this analysis will be based upon the 1976 forecasts.(3-7) The temperature correction factor due to the geomagnetic index is also published with the solar flux forecasts. This term, however, has a small effect on the temperature (1 to 11°K) and will not be discussed further.

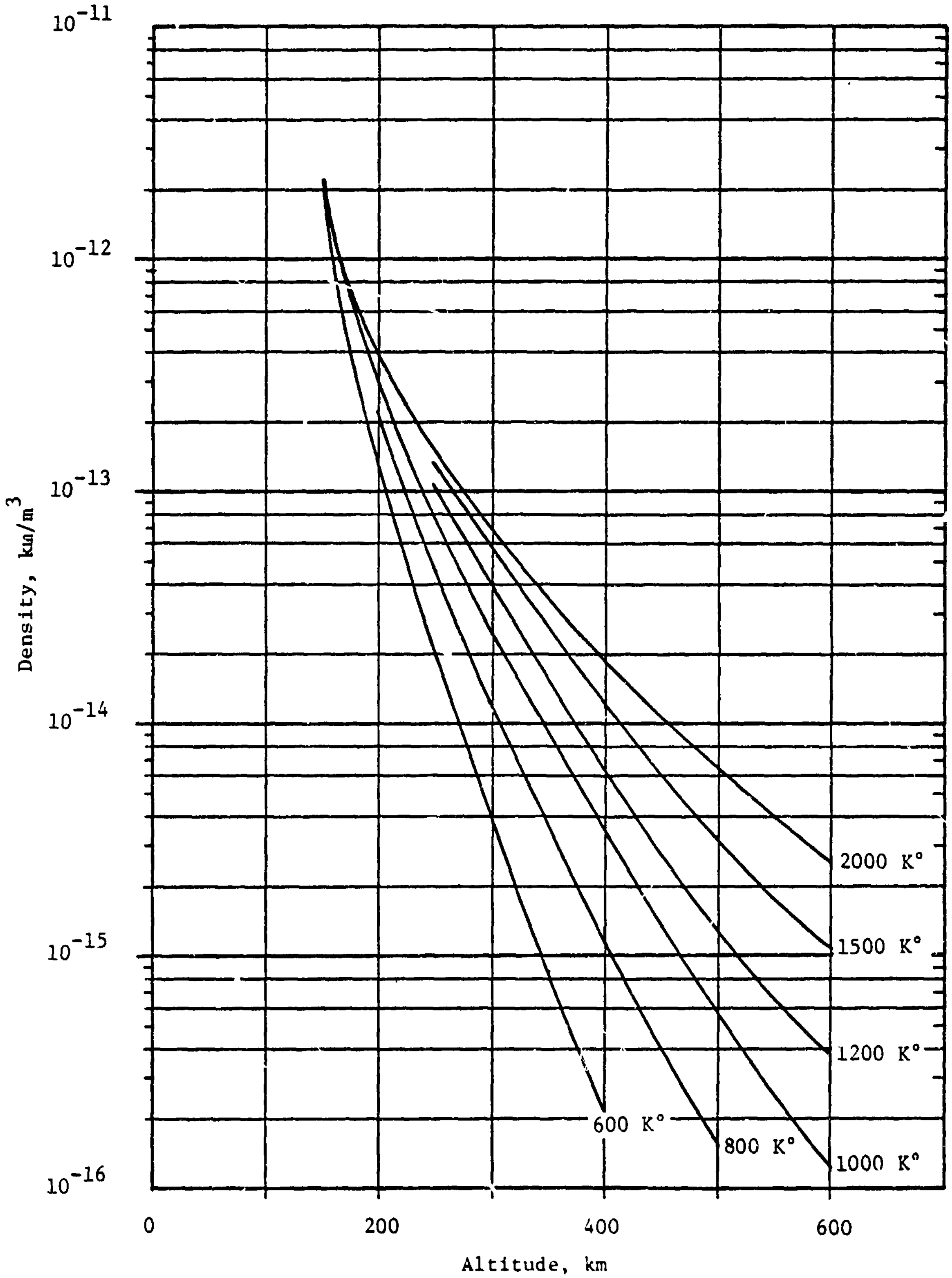


FIGURE 3-13. ATMOSPHERIC DENSITY FOR ALTITUDE AND EXOATMOSPHERIC TEMPERATURE

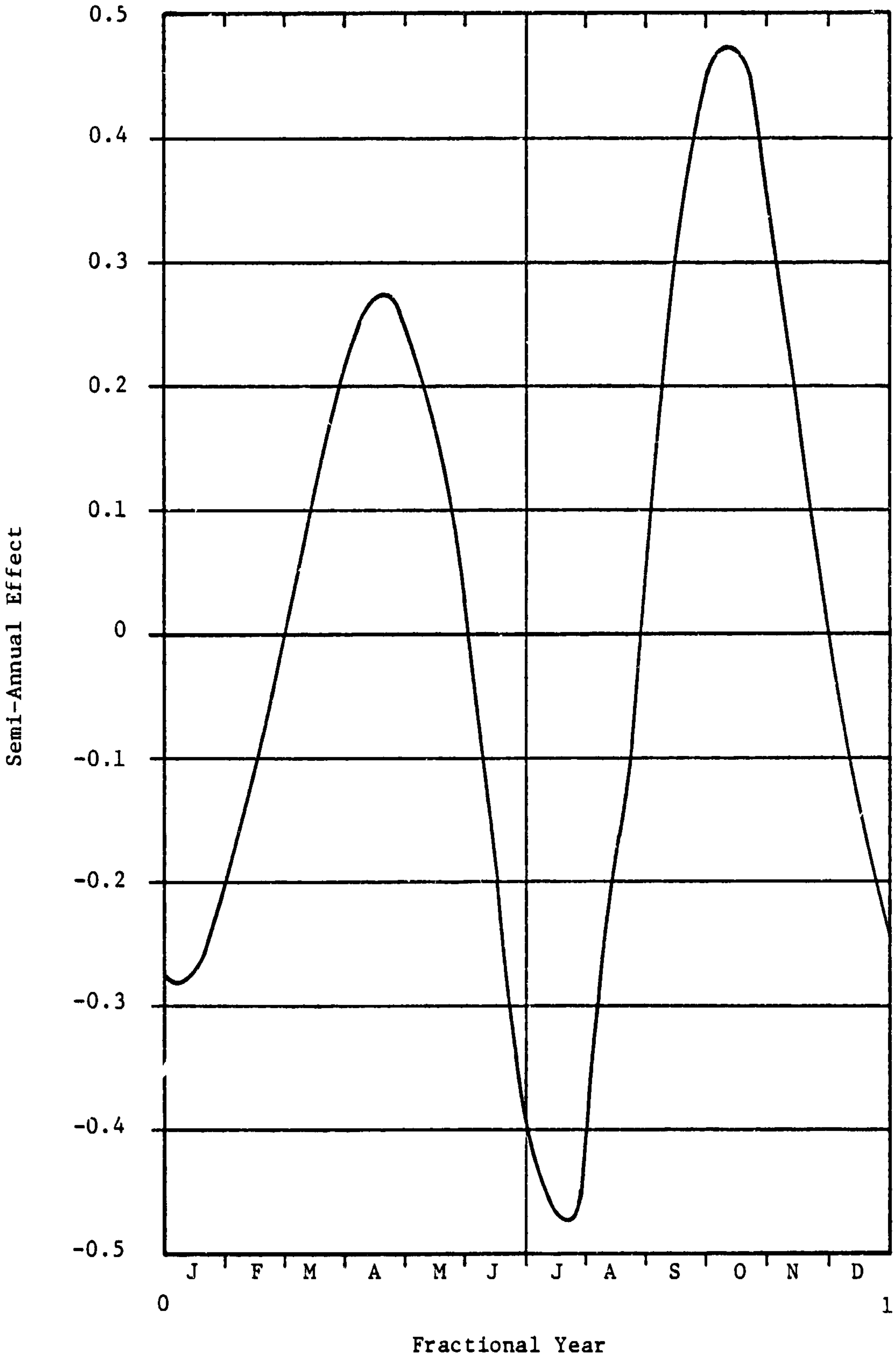


FIGURE 3-14. SEMIANNUAL TEMPERATURE CORRECTION FACTOR

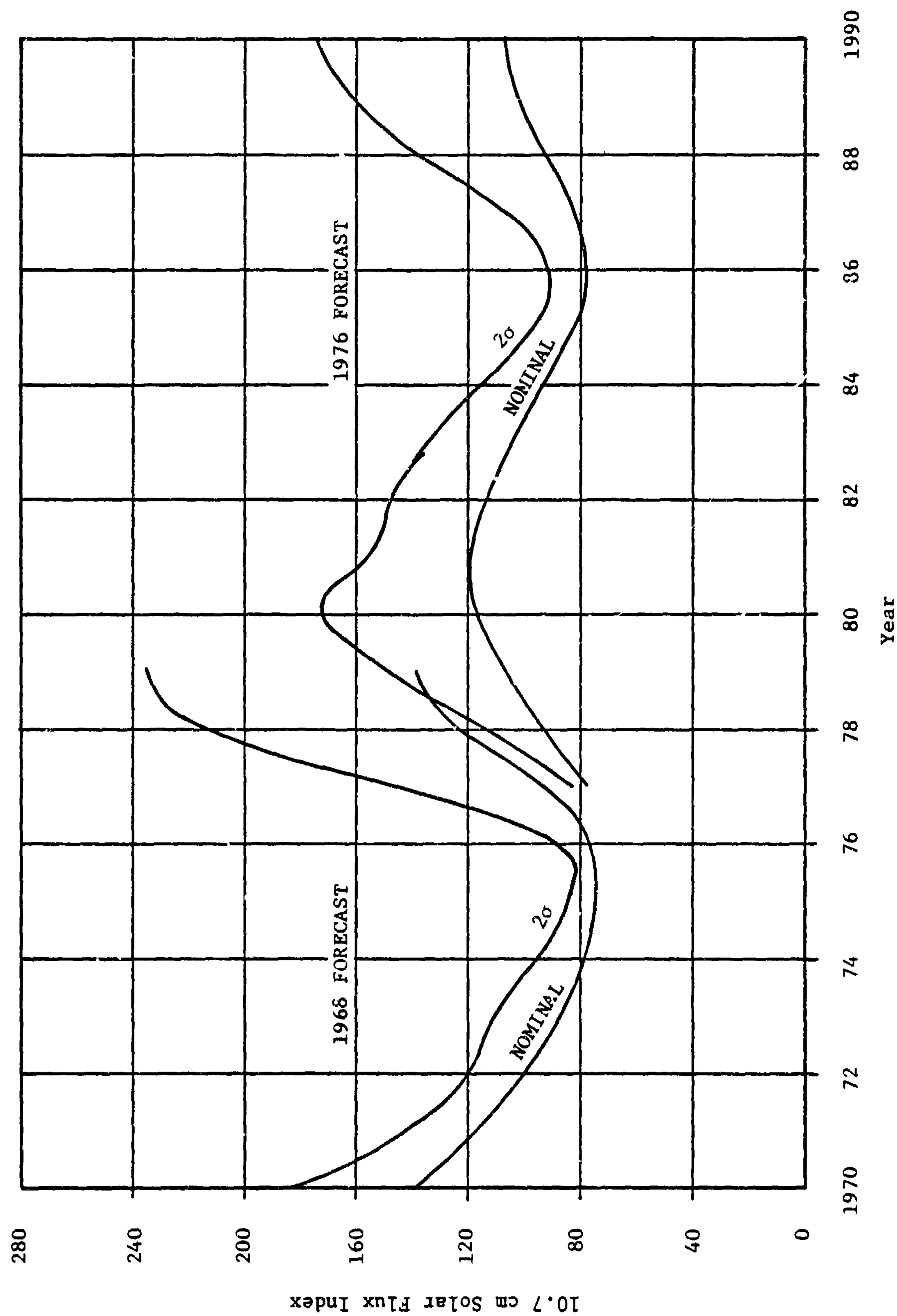


FIGURE 3-15. SOLAR FLUX FORECASTS

The density is needed to determine two key parameters of the drag makeup propulsion systems, the thrust and the propellant mass. The thrust must be sufficient to balance drag over short-term peaks so that the satellite does not decay to such an altitude that the propulsion system can never recover. Thus, the thrust requirement is based upon the density for the worst part of the semiannual cycle and the 2σ solar flux. The fuel consumption, however, is based upon average requirements; thus, the average semiannual effect and nominal flux are used. For the thrust calculation, the density is calculated using the temperature, T_m, given by:

$$T_m = 409 + 4.6\ F_{95} \quad , \tag{3-8}$$

where F_{95} is the 2σ solar flux estimate. The density is based upon the average thrust, $\bar{\rho}(h,y)$, over the year, as given by:

$$\bar{\rho}(h,y) = \int_{y}^{y+1} \rho(h,T(y))\ dy \quad , \tag{3-9}$$

where y is the year of interest, and the ref⌐rence temperature is calcu-lated using the nominal solar flux estimates in Equation (3-7). The density values used for the thrust calculations correspond to temperatures of 750 to 1350°K and the density values used for fuel stimates correspond to temperatures of 700 to 950°K (see Figure 3-13).

Each of the propulsion systems shown in Table 3-10 was considered for both the Scout and Delta class spacecraft (see Table 3-11). For each spacecraft, a given thrust level determines a minimum altitude below which the propulsion system cannot recover from a period of high drag. These altitudes are shown in Table 3-12.

TABLE 3-12. MINIMUM ALTITUDES FOR DRAG MAKEUP SYSTEMS

System[a]	Minimum Altitude, km Scout Class S/C	Minimum Altitude, km Delta Class S/C
A or B	125	155
C	200	305
D	180	270

(a) Systems are described in Table 3-10.

The minimum altitude varies with year, due to the variations in solar flux. However, these variations are only a few kilometers for the forecasts from 1977 through 1990. The altitudes shown in Table 3-12 represent the highest minimums, which occur in 1990.

The propellant mass, M_p, requirements are defined to be equal to the integral of the drag force over the lifetime of the mission divided by the specific impulse, I_{sp},

$$M_p = \frac{1}{I_{sp}} \int_{y_o}^{y_o+\ell} F_D \, dy \quad , \tag{3-10}$$

where y_o is the launch year and ℓ is the mission lifetime. This relationship is valid, since the thrust must be used to overcome the drag to maintain the orbit. The choice of the density estimates determines how conservative the design is. In this analysis, the nominal atmosphere is used to compute the propellant. Mission requirements are approximated by summing yearly requirements:

$$M_p = \frac{1}{I_{sp}} \sum_{i=y_o}^{y_o+\ell} \overline{F}_{D_i} \, (\Delta t) \quad , \tag{3-11}$$

where $\overline{F}_{D_i}$ is an average drag force over year i, and Δt is the time interval (seconds in a year). $\overline{F}_{D_i}$ is computed from Equations (3-4) and (3-9).

Mission durations of 1, 3, 5, and 7 years have been considered. To illustrate the effect of launch year on propellant mass requirements, 1-, 3-, and 7-year missions for Scout and Delta size payloads are shown in Figure 3-16. The Delta payload is shown with a catalytic hydrazine system, while the Scout payload has an augmented electrothermal hydrazine system. The variations are largest for the shorter missions, since they tend to follow the peaks and valleys of the solar activity cycle, while the longer missions average over larger portions of the solar activity cycle.

To compare different technology systems when the Shuttle is being used for transportation, it is necessary to determine the system length for Shuttle charges. While, in practice, an existing tank design

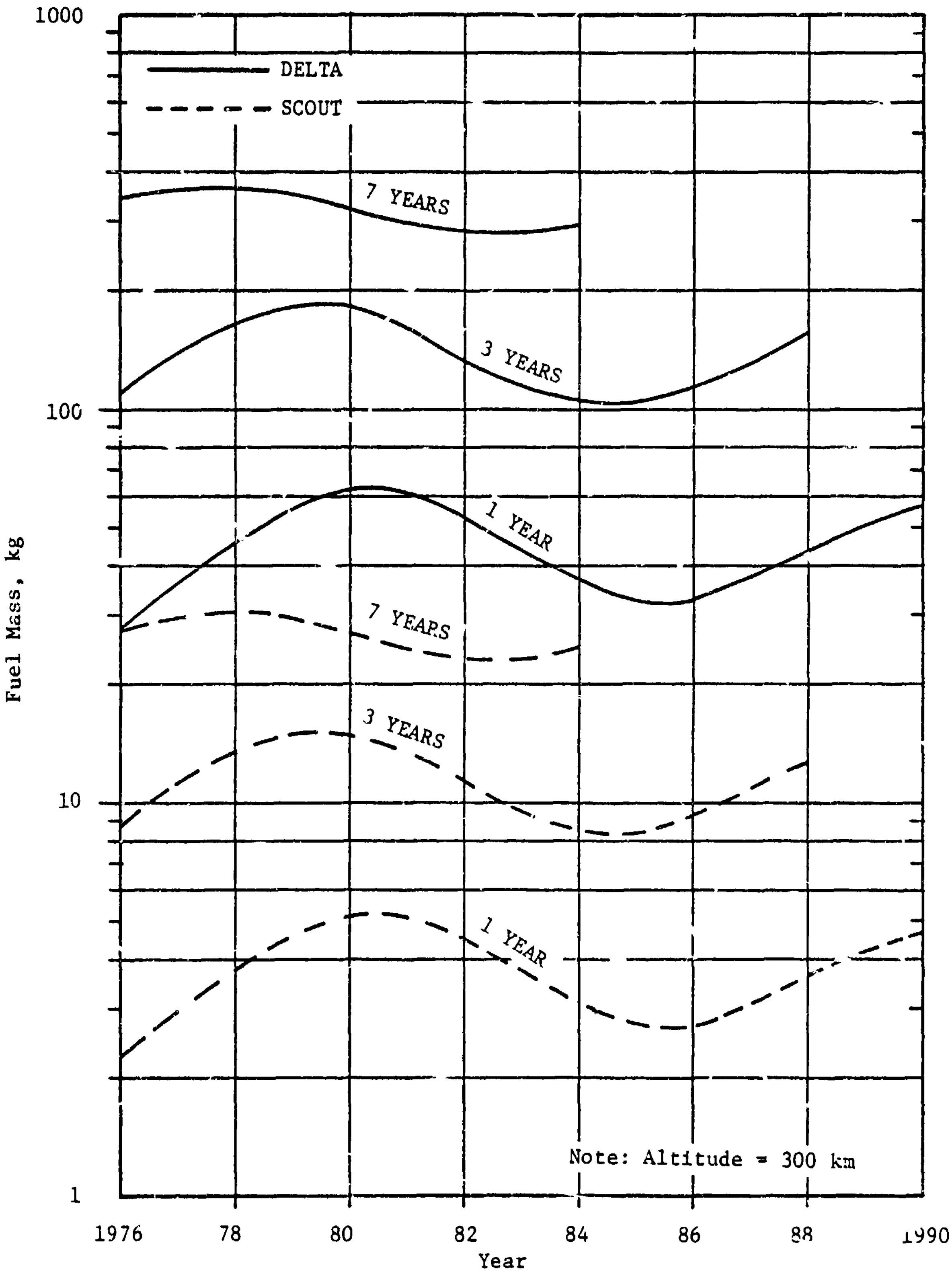


FIGURE 3-16. LAUNCH YEAR EFFECT ON FUEL MASS

would probably be used, the trade-offs here will be based upon a spherical tank of the appropriate size. An average density, ρ_h, for a hydrazine system with a 3 to 1 blowdown ratio is taken to be 670.7 kg/m^3. The length of the system, L, is defined using the length of the propellant tank, which assumes the thrusters do not add any significant length. Thus, L is given by:

$$L = 2\ (3\ M_p/4\pi\rho)^{1/3} \quad . \qquad (3\text{-}12)$$

The length of an 8-cm ion system is taken from the system description in Reference (3-8). The propellant requirements for the 8-cm ion systems do not change enough to impact the length. The system lengths for 1- and 7-year missions are shown in Figures 3-17 and 3-18 for the Scout and Delta class payloads, respectively. These curves have definite end points for the lower altitudes but not for the upper altitudes, as indicated by the arrows.

The final comparison of the systems is based upon using the cost data (Section 4) to determine not only which systems are cost effective, but under what conditions the cost-effectiveness occurs. This discussion is contained in Subsection 5.5.

3.2.3 Sun-Synchronous Nodal Change

The Landsat users are not in agreement on what the spacecraft ascending node should be. The two most likely ascending node local crossing times are 9 a.m. and 11 a.m. This corresponds to a 30-deg shift in the longitude of nodes. Since Landsat will have propulsion on board, there exists the possibility of sizing the propulsion to allow the spacecraft to change from one orbit to the other. There are three basic modes for performing this transfer: (1) a direct high-thrust transfer, (2) a transfer to an intermediate orbit that has a different precession of the longitude of nodes with chemical propulsion, a coast to achieve the desired precession, and a transfer to the desired orbit, and (3) a low-thrust maneuver.

The direct high-thrust transfer has the advantage of going directly from one operational orbit to the other, but requires a large velocity change. The required velocity, ΔV, can be computed as follows:

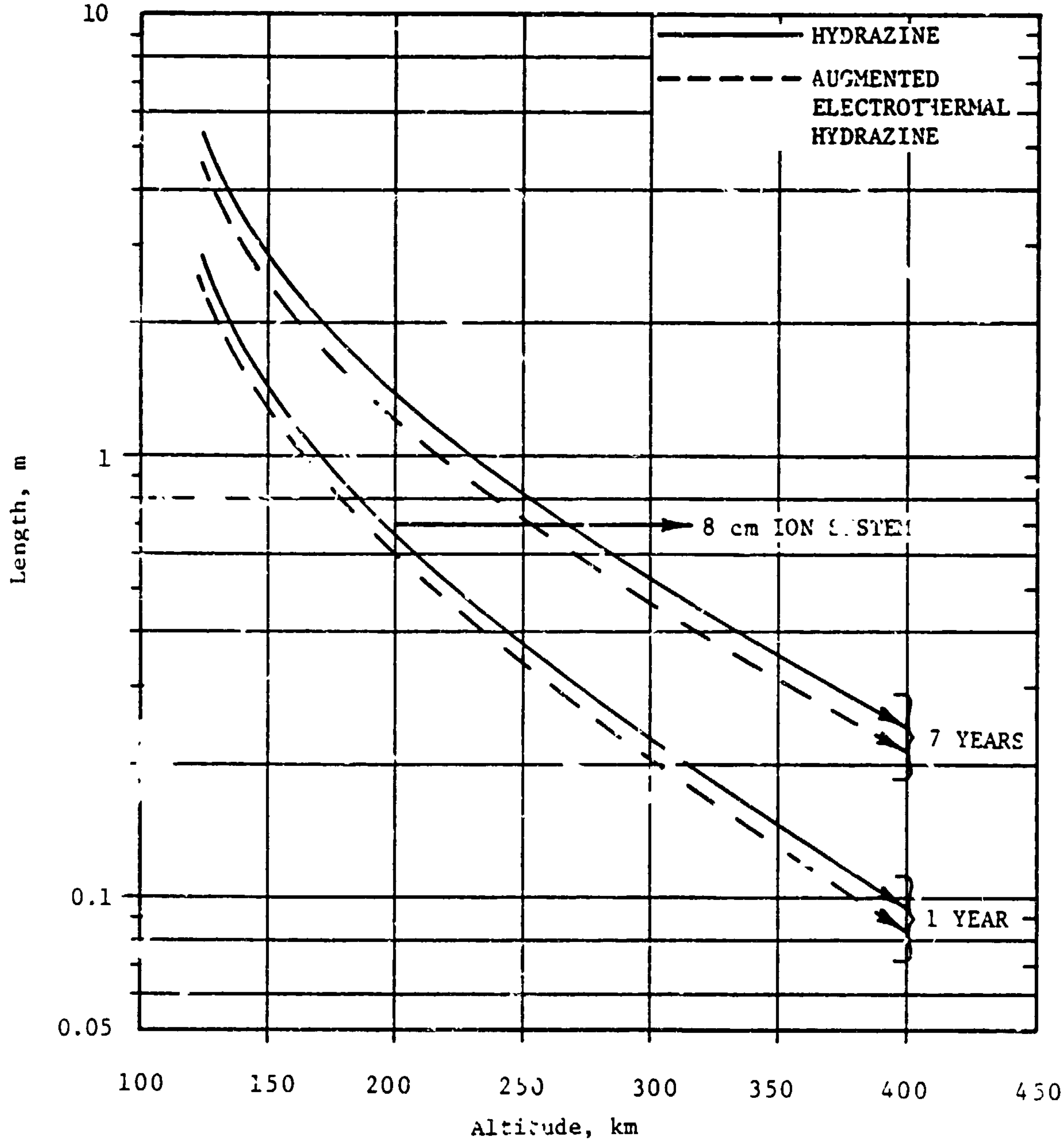


FIGURE 3-17. PROPULSION MODULE LENGTH FOR SCOUT CLASS PAYLOAD

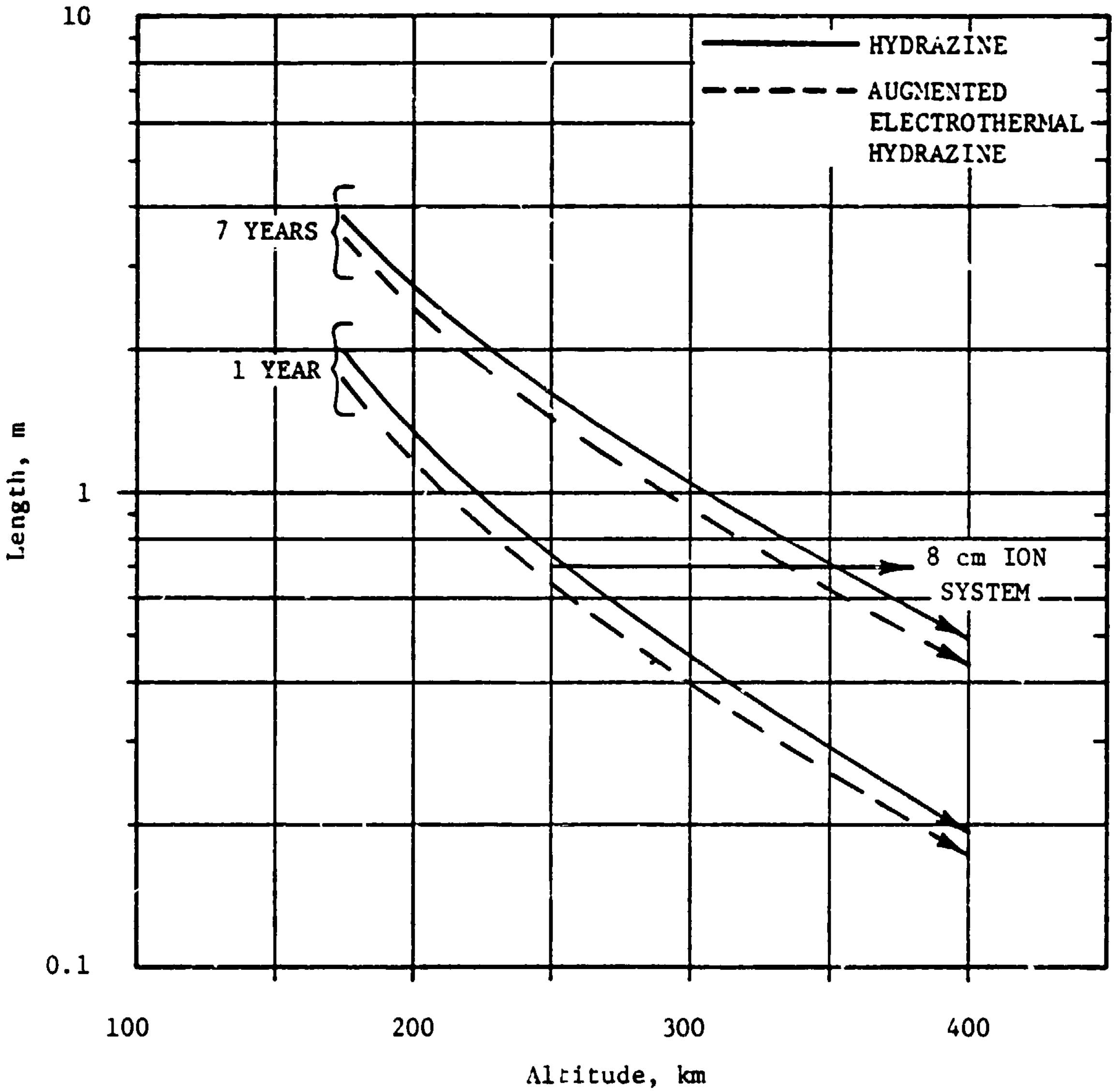


FIGURE 3-18. PROPULSION MODULE LENGTH FOR DELTA CLASS PAYLOADS

$$\cos \Delta\theta = \cos^2 i + \sin^2 i \cos \Delta\Omega \quad , \tag{3-13}$$

$$\Delta V = 2V \sin \Delta\theta/2 \quad , \tag{3-14}$$

where i is the inclination of the Landsat orbit, $\Delta\Omega$ is the change in the longitude of the ascending node, $\Delta\theta$ is a plane change, and V is the orbital velocity of the Landsat orbit. For Landsat, a 30-deg change in the longitude of nodes (a 2-hr shift in the local ascending node crossing time) requires a velocity of 3,843 m/sec (12,610 ft/sec), which is too large for spacecraft propulsion.

The second mode allows a reduction in the velocity required with the sacrifice of some operational time. In this mode, a two-impulse transfer is used to transfer to an intermediate orbit which has a different altitude and inclination from the Landsat orbit. After the coast to achieve the desired change in longitude of ascending node, a second two-impulse maneuver is used to transfer back to the Landsat orbit. The precession of the longitude of nodes was given by Equation (3-2). Using first-order approximations, the change in altitude, Δh, is related to the velocity increment, ΔV, by:

$$\Delta h/a = 0.75 \, \Delta V/V \cos \phi \quad , \tag{3-15}$$

and the change in inclination, Δi, is given by

$$\Delta i = \Delta V/V \sin \phi \quad , \tag{3-16}$$

where a is the semimajor axis, V is the orbital velocity, and ϕ is the out-of-plane angle of the two impulses. The differential drift rate, $\delta\dot{\Omega}$, is given by (considering first-order terms only):

$$\delta\dot{\Omega} = -\tan i \, \dot{\Omega} \, \Delta i - \frac{7}{2} \dot{\Omega} \, \Delta h/a \quad , \tag{3-17}$$

where $\dot{\Omega}$ is the precession rate of a Sun-synchronous orbit, 0.9856 deg/day. The desired total change in viewing conditions, $\Delta\overline{\Omega}$, is equal to the integral of the differential drift rate:

$$\Delta\overline{\Omega} = \int_0^T \delta\dot{\Omega} \, dt = \delta\dot{\Omega} \, T \quad , \tag{3-18}$$

where T is the number of days allowed for the maneuver. Combining Equations (3-15) through (3-17) and substituting the optimum choice of ϕ ($\tan^{-1} \frac{2}{7} \tan i$), the total velocity requirement for these maneuvers is given by:

$$\Delta V = \frac{2}{7} \frac{V}{\dot{\Omega}} \frac{|\Delta\Omega|}{T} \cos \left(\tan^{-1} \frac{2}{7} \tan i\right) \quad . \tag{3-19}$$

For Landsat and a 30-deg shift in the longitude of ascending node, the velocity requirement in km/sec is given by:

$$\Delta V = 28.956/T \quad . \tag{3-20}$$

The total velocity requirement then depends upon the time allowed for the changeover from one viewing condition to another and the number of changeovers to be accomplished.

When low thrust is used instead of chemical thrust, the orbit must be changed gradually, which provides a constantly changing differential drift rate in the precession of the line of nodes. Assuming $\delta\dot{\Omega}$ changes approximately linear with time, the integral in Equation (3-18) may be reevaluated and the following expression obtained to estimate the time required for the total maneuver:

$$T^2 = \frac{8}{7} \frac{V}{\bar{a}} \frac{|\Delta\Omega|}{\dot{\Omega}} \cos \left(\tan^{-1} \frac{2}{7} \tan i\right) \quad , \tag{3-21}$$

where $\bar{a}$ is the effective acceleration of the low-thrust system (expressed in km/sec/day). The effective acceleration is reduced because of shadowing and thrusting at points other than equatorial crossing. The value of $\bar{a}$ can be estimated by multiplying the maximum acceleration by 0.404. For a Landsat spacecraft of 1800 kg (including the low-thrust system of n pairs of 30-cm ion thrusters), the time to perform the shift in longitude of node is approximated by:

$$T^2 = 12.96 \text{ m/n} \quad , \tag{3-22}$$

where m is the total spacecraft mass, including propulsion system, in kg.

Representative times are shown in Table 3-13 for different combinations of total spacecraft mass and number of thrusters. The propellant requirements for a transfer using low thrust are approximately

0.196 kg/day/thruster. This value accounts for thrusters not being used during shadowing. Thus, using six thrusters on an 1800-kg spacecraft, 104 kg of propellant would be required for the transfer. Performance of the same maneuver with a hydrazine system would require approximately 300 kg of hydrazine.

TABLE 3-13. REPRESENTATIVE LOW-THRUST TRANSFER TIMES

Spacecraft Mass, kg	Low-Thrust Transfer Times for Indicated Number of 30-Cm Ion Thrusters[a], days			
	Two	Four	Six	Eight
500	80.5	56.9	46.5	40.2
1000	113.8	80.5	65.7	56.9
1500	139.4	98.6	80.5	69.7
1800	152.7	108.0	88.2	76.4

(a) Transfer times are those required to achieve a 2-hour shift in the local time of the ascending node.

Additionally, the other propulsion requirements should also be taken into consideration. Consider a Sun-synchronous spacecraft with the following requirements:

(1) Spacecraft mass of 1800 kg.

(2) Altitude of 705 km.

(3) Spacecraft has capability to return to Shuttle for servicing.

(4) Spacecraft can shift between 9 a.m. and 11 a.m. viewing conditions three times (3 months allocated for each transfer).

(5) Spacecraft has nominal altitude control and orbit adjustment capabilities.

The total velocity requirements using chemical propulsion would be approximately 2110 m/sec [Table 3-5 and Equation (3-20)]. Assuming an I_{sp} of 220 sec and an expended mass fraction of 0.82 for a catalytic hydrazine system (Subsection 2.1), this would require a propulsion module with a mass of approximately 4260 kg. The total mass of an ion system with six 30-cm thrusters and sufficient propellant to perform the required maneuvers would be approximately 1000 kg. Although a hydrazine propulsion module of over 4000 kg would be feasible (by clustering tanks and designing new tanks), it would be larger than most spacecraft designers would like to consider. Using an ion system, the total mass would be about one-fourth as large, and the density comparison between mercury and hydrazine (about 13.5 to 1) implies the propellant volume for hydrazine would be about 100 times as large as the propellant volume for mercury.

Considering Shuttle charge formulas, the ion system could be cost effective over hydrazine (assuming this mission does not bear the development cost for the ion system). The advantages of going to the ion system would be greater flexibility for system growth, both in terms of spacecraft mass and mission complexity. An approximate cost trade is done in Subsection 5.6, and the overall merits of the different approaches are discussed in Section 6.

3.2.4 Geosynchronous North-South Stationkeeping

The transportation cost is not necessarily the primary concern in comparing propulsion systems for stationkeeping. A key parameter, which will be used for comparison, is net spacecraft mass in orbit. For communication satellites, the net mass in orbit translates into communication capability which, in turn, yields revenue. Thus, the trade-offs for this propulsion application will be in terms of net spacecraft mass in orbit, not transportation cost.

Both spin-stabilized and three-axis-stabilized geosynchronous spacecraft have been built in the past. Table 3-14 lists several representative spacecraft, their use, and the size and number of thrusters on board. The tendency is for future spacecraft to be three-axis stabilized as opposed to spin stabilized. Intelsat V is three-axis as opposed to the spin stabilized Intelsat IVA; RCA's new Satcom (which required development of the Delta 3914) is three-axis stabilized, the European satellites (e.g., Symphonie) are three axis, and Telsat has gone to three-axis stabilization. The Hughes spacecraft, however, appear to favor spin stabilization.

TABLE 3-14. THRUSTERS ON SELECTED GEOSYNCHRONOUS SPACECRAFT

Spacecraft	Thruster Size	Number of Thrusters	Use
INTELSAT IVA[3-9] Hughes Aircraft Co. 1745 lb_m(*) Spin Stabilized	5 lb_F	6 (2 radial, 2 axial, and 2 spin)	Spacecraft spinup, spin trim, active nutation damping, precession for AKM firing, orbital station acquisition, stationkeeping, and attitude control.
MARISAT[3-10] Hughes Aircraft Co. 720.5 lb_m(*) Spin Stabilized	5 lb_F	2	Oriented to fire parallel to spin axis. Augments apogee motor thrust for orbital injection in steady-state mode. Used for attitude control in the pulsed mode.
	1 lb_F	2	Radial thrusters used for East-West stationkeeping in the pulsed mode.
			All four thrusters are biased in the spinup direction for adjustments in spin rate.
Palapa[3-11] Hughes Aircraft Co. 647 lb_m(*) Spin Stabilized	1 lb_F	4	Two of the thrusters are oriented to fire parallel to the spin axis and are used for inclination control in the steady-state mode and for altitude control in the pulsed mode. The remaining two thrusters are oriented through the spacecraft c.g. and are fired in the pulsed mode only for period and eccentricity control.
Satcom F-1[3-12] RCA Astro-Electronics 1027.1 lb_m(*) Three-axis Stabilized	0.1 lb_F	12	Thrusters are placed 4 per axis and are fired in various combinations to generate torques or to provide velocity changes with respect to all three axes. They provide velocity increments for stationkeeping; serve as momentum management devices; and serve as a backup to on-board magnetics for attitude control.

TABLE 3-14. (CONTINUED)

Spacecraft	Thruster Size	Number of Thrusters	Use
GOES(3-13) Aeronutronic Ford 630 lb_m(*) Spin Stabilized	5 lb_F	4 (2 axial and 2 radial)	Prior to apogee boost motor separation, these thrusters provide thrust vectors for active nutation damping and attitude control. After ABM separation, they are used for East-West velocity control, coarse attitude control, and spacecraft spin rate adjustments. East-West stationkeeping is accomplished by pulsing one of the radial thrusters in phase with the spin rate. North-South stationkeeping is accomplished by steady firing of one of the axial thrusters.
	0.5 lb_F	2	Oriented in radial direction. These thrusters are used for precision attitude control.
CS (Japanese Communications Satellite)(3-13) Aeronutronic Ford 792.4 lb_m(*) Spin Stabilized	5 lb_F	4	Two radial and two axial thrusters provide redundant capability for East-West and North-South stationkeeping, precession of spacecraft spin axis, and spin rate adjustments. The canted axial thrusters, when fired in a continuous mode, are used for orbital inclination changes.

TABLE 3-14. (CONTINUED)

Spacecraft	Thruster Size	Number of Thrusters	Use
INTELSAT V(3-13) Aeronutronic Ford 4100 lb_m(**) Three-axis Stabilized	5 lb_F	2	Active nutation damping and precession (reorientation).
	0.6 lb_F	10	Spinup (contingency), despin, East-West stationkeeping, pitch and yaw control, and backup on North-South stationkeeping.
	0.1 lb_F	4	Roll control.
	0.03 lb_F	4	Electrothermal thrusters (improved I_{sp}) for North-South stationkeeping.
Symphonie (ESA)(3-14) Consortium Industrial Franco-Allemand pour le Satellite Symphonie 507.1 lb_m(*) Three-axis Stabilized	2.92 lb_F	7	Bipropellant (N_2O_4/MMH) thrusters used to translate spacecraft to final geosynchronous longitude and for maintenance of orbit altitude.

(*) On-Station Mass (Initial).
(**) Launch Mass.

Two general observations can be made from the information in Table 3-14: (1) the spin-stabilized satellites have four to six thrusters, while the three-axis-stabilized satellites have between 7 and 20 thrusters, and (2) the thrust levels used on the three-axis-stabilized spacecraft for stationkeeping are lower than the thrust levels for the spin-stabilized satellites. The first observation is connected with the various redundancy schemes employed for the different types of spacecraft. The second observation is related to the required thrust levels for a spin-stabilized spacecraft. Since the trade-offs in this analysis are between hydrazine and lower thrust systems such as electrothermal hydrazine or the 8-cm ion engine, the spin-stabilized spacecraft will not be considered. In the area of redundancy, it will be assumed that the on-orbit operations (North-South stationkeeping, etc.) require a backup thruster system, but none is required for the initial station acquisition. When electrothermal thrusters are utilized, catalytic hydrazine thrusters can be used as a backup and can be fueled from the same propellant tanks. However, when an ion system is used, backup ion thrusters are required.

In analyzing the propulsion requirements for North-South stationkeeping of a geosynchronous spacecraft, it is necessary to also consider the requirements of establishing the orbit after apogee kick motor (AKM) burn. When spinning solid motors are used for the perigee and apogee burns on a transfer from a low-altitude parking orbit to a geosynchronous equatorial orbit, a correction is needed to remove the errors introduced by the solid motors. The velocity correction requirements needed to overcome perigee kick motor (PKM) induced errors are estimated by the method developed in Reference (3-15).

The PKM errors are represented as a percent 3σ magnitude error, η, and a 3σ pointing error, θ. The normalized apogee error, $\delta r_a/r_a$, due to PKM errors is given by:

$$\delta r_a/r_a = 14.65\ (1-0.761 \cos \Delta I_p)\ \eta - 11.15 \sin \Delta I_p \theta \quad , \qquad (3\text{-}23)$$

where ΔI_p is the plane change done by the PKM. The transfer inclination error, δI_T, is given by:

$$\delta I_T = 0.761 \sin \Delta I_p\ \eta - (1-0.761 \cos \Delta I_p)\theta \quad . \qquad (3\text{-}24)$$

The correction velocity required to remove PKM errors, δV_{pkm}, is:

$$\delta V_{pkm} = V_f \left\{ \left| 0.402 \frac{\delta r_a}{r_a} - 0.420 \; \delta I_T \right| + \left| 0.25 \frac{\delta r_a}{r_a} \right| \right\} \quad , \tag{3-25}$$

where V_f is the final orbital velocity (3.08 km/sec). The PKM errors are described statistically, and care must be used in evaluating Equation (3-25). A conservative approach is to add the terms in the two separate absolute values, replace the inclination error and apogee error with the expressions from Equations (3-23) and (3-24) and then RSS (root-sum-square) the resulting independent errors to obtain the velocity correction, as given by:

$$\delta V^2_{pkm} = V_f^2 \left\{ \left(9.55 \; (1-0.761 \cos \Delta I_p) + 0.32 \sin \Delta I_p\right)^2 \eta^2 + \left(0.42 \; (1-0.761 \cos \Delta I_p) - 7.27 \sin \Delta I_p\right)^2 \theta^2 \right\} . \tag{3-26}$$

The errors due to the apogee kick motor are directly proportional to the velocity increment provided by the apogee kick motor.

In addition to the correction of the errors from the PKM and the AKM burns, a velocity correction is required to achieve geosynchronous orbit due to the apogee bias i the nominal transfer. The apogee bias is designed into the trajectory for several reasons, one of which is to have an initial drift in longitude to achieve the desired station location. The velocity, ΔV_{AC} to correct for the apogee bias is given by:

$$\Delta V_{AC} = V_f \left\{ \sqrt{\frac{2(1+\Delta r_a/r_a)}{2+\Delta r_a/r_a}} - 1 \right\} \quad , \tag{3-27}$$

where Δr_a is the apogee bias. A typical apogee bias is 1500 km, which requires 27 m/sec to correct. These values will be assumed in this analysis.

The North-South stationkeeping requirements are taken to be 50 m/sec for each year of operation. Compared to the North-South requirements, the East-West requirements are minimal. Thus, the masses for the thrusters will be considered, but the propellant masses will not be.

The technologies under consideration are catalytic hydrazine, electrothermal hydrazine, and 8-cm ion engines. Since Symphonie uses bipropellants, they will be considered also. Both single-technology and multiple-technology systems are considered. The mass properties, number of thrusters and other characteristics necessary to define the systems are listed in Table 3-15.

TABLE 3-15. GEOSYNCHRONOUS SPACECRAFT PROPULSION SYSTEMS

System	Dry Mass, kg	No. of Thrusters	Thrust per Thruster, N	I_{sp}, sec	Tankage Factor[(a)]
A: Hydrazine	37.1	8	0.445	220	0.176
B: Bipropellant	38.2	8	13.0	295	0.176
C: Augmented electrothermal hydrazine	33.1	8	0.13	320	0.176
D: 8-cm ion	80.5	8	0.005	2955	--
E: Hydrazine +	35.1	4	0.445	220	0.176
augmented electrothermal hydrazine[(b)]	31.1	4	0.13	320	0.176
F: Hydrazine +	35.1	4	0.445	220	0.176
8-cm ion[(b)]	80.5	8	0.005	2955	--

(a) The tankage factor represents that part of the system proportional to the propellant required.
(b) In the combined systems, the hydrazine is used for initial station acquisition, and the ion system for North-South stationkeeping.

Table 3-16 shows the net spacecraft mass using the various auxiliary propulsion systems. These masses correspond to the maximum capability of the SSUS-D and SSUS-A. The spacecraft is assumed to have an integral apogee kick motor which is not jettisoned before use of the auxiliary propulsion module. For SSUS-D class spacecraft, the dry weights of the propulsion systems are sufficiently large that the dual systems are not competitive. The low thrust of the 8-cm ion system (0.005 N) could result in an unacceptable time to achieve orbit. The propellant requirements are about 1.15 kg, depending on the errors introduced by the PKM and AKM with a flow rate of 0.0149 kg/day; this results in a time to achieve orbit of 77 days plus the time needed for drifting. Thus, the augmented electrothermal hydrazine system (which has a thrust level about 25 times as large as the 8-cm ion system) is attractive for the SSUS-D class payloads.

TABLE 3-16. NET GEOSYNCHRONOUS SPACECRAFT MASS VERSUS AUXILIARY PROPULSION MODULES

System[a]	Mass of Indicated Element When SSUS-D is Used as a PKM, kg			Mass of Indicated Element When SSUS-A is Used as a PKM, kg		
	Spacecraft	Propulsion System 1[b]	Propulsion System 2[b]	Spacecraft	Propulsion System 1[b]	Propulsion System 2[b]
A: Hydrazine	384	146	--	740	237	--
B: Bipropellant	408	121	--	786	191	--
C: Electrothermal hydrazine	420	110	--	803	175	--
D: 8-cm ion	442	88	--	883	94	--
E: Hydrazine + electrothermal	381	53	95	760	68	149
F: Hydrazine + 8-cm ion	390	53	87	817	68	92

(a) System characteristics defined in Table 3-10.
(b) Propulsion system masses include dry weight and propellants.

For the larger SSUS-A spacecraft, the dual systems become more competitive. The largest spacecraft mass is achieved with a single 8-cm ion system; however, this approach is not attractive because of initial placement times of up to 6 months. The next best systems are the hydrazine/8-cm ion combination and the augmented electrothermal hydrazine systems. The dual system looks attractive compared to the electrothermal for two reasons: (1) 14 kg of additional spacecraft mass and (2) smaller propellant requirements for additional on-orbit capability.

For payloads less than the maximum capability of each stage, the relative masses of the spacecraft for each of the different systems are shown versus total SSUS load for the SSUS-D and SSUS-A in Figures 3-19 and 3-20, respectively.

3.2.5 Geosynchronous Satellite Return

Consider the case of a geosynchronous spacecraft which has an ion propulsion system for stationkeeping and other propulsion needs. By allocating additional mass to the amount of propellant on board, it is feasible that the stationkeeping system could have the capability of returning to low altitude for retrieval by Shuttle in the event the spacecraft malfunctions. To estimate the amount of propellant required, the following approximations are developed.

In developing an approximation for a low-thrust trajectory, it is desirable to consider variations in orbital elements which change slowly. Starting with Lagrange's planetary equations for rates of change of semimajor axis and inclination:

$$\frac{da}{dt} = \frac{2a^2}{\sqrt{\mu p}} \left\{ F_r \, e \sin \theta + F_t \, (1 + e \cos \theta) \right\} \quad , \tag{3-28}$$

$$\frac{di}{dt} = \frac{r \, F_n}{\sqrt{\mu p}} \cos u \quad , \tag{3-29}$$

where p is the semilatus rectum, e is the eccentricity of the orbit, F_r, F_t and F_n are the radial, transverse, and normal components of acceleration, θ is the true anomaly, and u is the argument of latitude. At geosynchronous

C-2

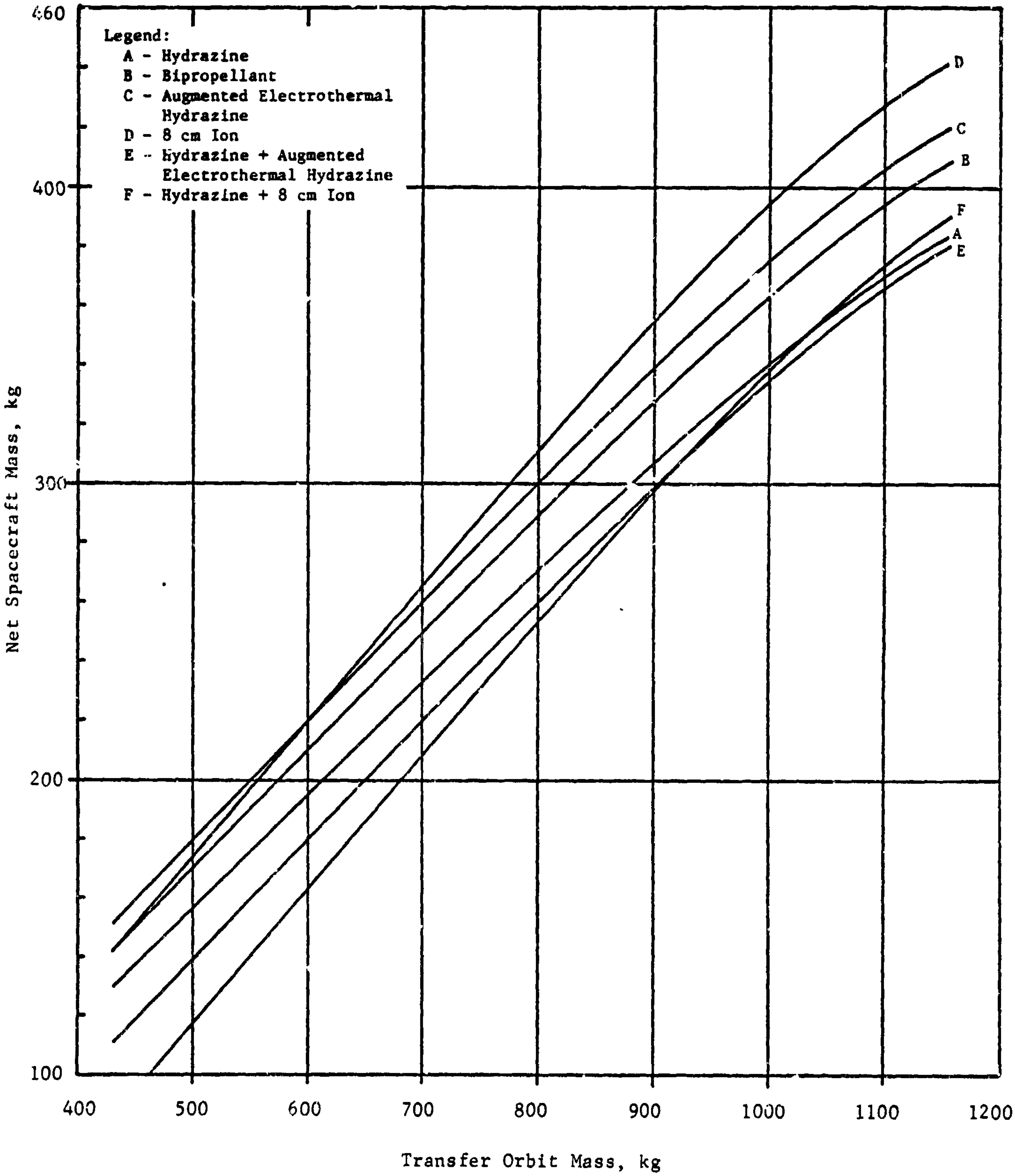


FIGURE 3-19. SPACECRAFT MASS ON SSUS-D

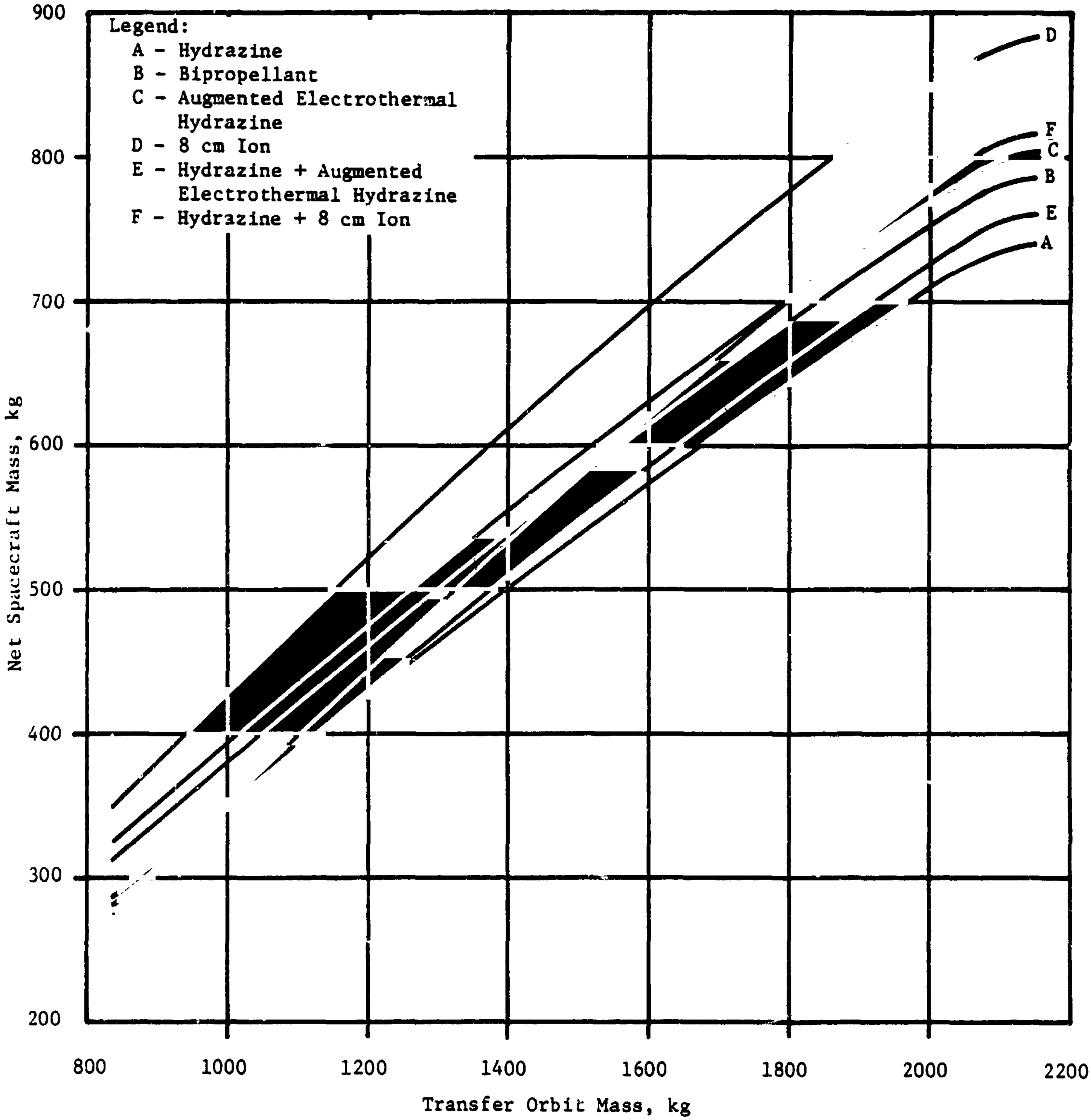


FIGURE 3-20. SPACECRAFT MASS ON SSUS-A

orbit the eccentricity is zero, and it will be assumed to remain zero. The components of acceleration will be taken as:

$$f_r = 0,\ F_t = \frac{T \cos \phi}{m_o - \dot{m}t}\ ,\ F_n = \frac{-T \sin \phi}{m_o - \dot{m}t} \operatorname{sgn}(\cos u) \quad , \tag{3-30}$$

where ϕ is an angle which represents the split of the thrust between altitude change and inclination change, m_o is the initial mass, $\dot{m}$ is the mass flow rate, and sgn is the sign function. The formulation is being developed for raising the orbit and reducing the inclination, but the final results will also apply to the return case. Substituting the components of acceleration into Lagrange's equations and letting the eccentricity be zero (which implies the semilatus rectum is equal to the semimajor axis), gives the following:

$$\frac{da}{dt} = \frac{2a^{3/2}}{\sqrt{\mu}} \ \frac{T \cos \phi}{m_o - \dot{m}t} \quad , \tag{3-31}$$

$$\frac{di}{dt} = - \sqrt{\frac{a}{\mu}} \ |\cos u| \ \frac{T \sin \phi}{m_o - \dot{m}t} \quad . \tag{3-32}$$

Separating variables in Equation (3-31) gives:

$$\frac{da}{a^{3/2}} = \frac{T \cos \phi}{m_o - \dot{m}t} \cdot \frac{2dt}{\sqrt{\mu}} \quad , \tag{3-33}$$

and integrating holding ϕ constant gives:

$$\frac{1}{\sqrt{a_f}} - \frac{1}{\sqrt{a_o}} = \frac{T \cos \phi}{\dot{m} \sqrt{\mu}} \log \left(1 - \dot{m}t_f / m_o\right) \quad , \tag{3-34}$$

where a_f and a_o are the final and initial values of semimajor axis and t_f is the final time. Typically, the initial and final altitudes are known, as well as the system parameters T, $\dot{m}$ and m_o; thus, the final time could be determined if the angle ϕ were known. In preparation for integrating Equation (3-32), the $\sqrt{a/\mu}$ as a function of time is given as:

$$\sqrt{\frac{a}{\mu}} = \left\{ \frac{T \cos \phi}{\dot{m}} \log \left(1 - \frac{\dot{m}t}{m_o}\right) + \sqrt{\frac{\mu}{a_o}} \right\}^{-1} \quad . \tag{3-35}$$

Substituting this result into Equation (3-32) gives:

$$\frac{di}{dt} = -\left\{\frac{T\cos\phi}{\dot{m}}\log\left(1-\frac{\dot{m}t}{m_o}\right)+\sqrt{\frac{\mu}{a_o}}\right\}^{-1}\left|\cos u\right|\frac{T\sin\phi}{m_o-\dot{m}t} \quad . \tag{3-36}$$

This equation can be integrated in closed form if the $|\cos u|$ could be represented by a constant, $1/K$. The average value of $|\cos u|$ is $2/\pi$, which would correspond to changing the inclination all around the orbit. The more optimal strategy would be to do the inclination change at the nodes only where $|\cos u|$ is 1. The actual choice of the constant will be discussed with the evaluation of the other constants. Letting $|\cos u|$ be $1/K$ and $x = \log(1-\dot{m}t/m_o)$, we have, by integration:

$$\frac{K\dot{m}\Delta i}{T\sin\phi} = \int\frac{dx}{\frac{T}{\dot{m}}\cos\phi\, x+\sqrt{\frac{\mu}{a_o}}} \quad , \tag{3-37}$$

$$= \frac{\dot{m}}{T\cos\phi}\log\left(\frac{T\cos\phi}{\dot{m}}x+\sqrt{\frac{\mu}{a_o}}\right)\Bigg|_{x=0}^{x=\log\left(1-\frac{\dot{m}t_f}{m_o}\right)} \quad . \tag{3-38}$$

To simplify Equation (3-38) and use terminology consistent with low-thrust systems, the following relationships are used:

$$T = \dot{m}c,\ p_J = \frac{1}{2}\dot{m}c^2,\ v = \sqrt{\frac{\mu}{a}} \quad , \tag{3-39}$$

where c is the jet velocity, p_J is the jet power, and v is the equivalent circular orbit velocity; additionally, at $t = t_f$, from Equation (3-35):

$$\frac{T}{\dot{m}}\cos\phi\log(1-\dot{m}t_f/m_o)+\sqrt{\frac{\mu}{a_o}} = v_f \quad . \tag{3-40}$$

Thus, the angle ϕ can be determined from:

$$\tan\phi = \frac{K\,\Delta i}{\log(v_f/v_o)} \quad . \tag{3-41}$$

For the upbound leg, Δi is negative, v_f is less than v_o, and ϕ is between 0 and 90 deg; for the down leg, Δi is positive, v_f is greater than v_o, and ϕ is between 180 and 270 deg. However, in both cases the same equations are valid. Solving Equation (3-34) for t_f and substituting the relationships in Equation (3-39) gives:

$$t_f = \frac{m_o c^2}{2p_J}\left(1 - e^{\frac{v_f - v_o}{c \cdot \cos\phi}}\right) \approx \frac{m_o c}{2p_J}\frac{(v_o - v_f)}{\cos\phi} \quad . \qquad (3\text{-}42)$$

These last two equations provide a method for estimating performance to and from geosynchronous orbit with a low-thrust system once a value of K is chosen.

Several assumptions have been made in the development of these approximations. These have been examined by comparing the results of these approximations with data generated by MSFC. The key assumptions are:

(1) The eccentricity remains zero.

(2) The rate of change of semimajor axis and inclination are approximately proportional (i.e., ϕ is constant).

(3) The radiation belts are not considered.

(4) K is chosen as the average of the two extremes (K = 1.2854).

(5) The transfers are between circular orbits.

Due to Assumption (3), data were checked only for cases completely above the radiation belts. The results and the various assumptions were found to hold reasonably well; the eccentricity remained small, holding ϕ constant is a valid assumption, and the estimates of the transfer times agreed within a few percent.

The following method has been developed to extend the procedure to trajectories which traverse the radiation belts. A radiation flux model and solar cell damage model were obtained from MSFC.[3-16] The major effect of the radiation is to alter the thrust. Thus, Equations (3-31) and (3-32) can be numerically integrated, with the thrust being evaluated from the integrated flux and the radiation damage model. The choice of ϕ is obtained from Equation (3-41). By replacing $|\cos u|$ with a constant factor, the numerical integration did not have to be done at steps commensurate with the orbital motion, but rather several days per step. Although the trajectories from low Earth orbit to geosynchronous do not remain circular, the final time estimates agreed well with data from MSFC.[3-16] The obvious advantage of this procedure is that it enables data and trade-offs of various parameters to be obtained without requiring the lengthy computer runs needed for converged trajectories from programs such as SECKSPOT or MOLTOP. Those programs, however, are required to evaluate how accurate the approximations are.

Other authors(3-17,3-18) have considered different approximations which do not directly give the transfer times for the cases considered here.

Consider the case of a spacecraft on a SSUS-A with an ion system used for stationkeeping. If sufficient mass is allocated for propellant so that the spacecraft could return to Shuttle orbit immediately after going on-station, a contingency would be provided in the event of a spacecraft failure. Using the approximation developed above, it is impractical to consider the 8-cm ion system for the return, since the return trip time would be in excess of 10 years.

Thus, the following system is proposed: two 30-cm thrusters and six 8-cm thrusters. The 30-cm thrusters provide the thrust for initial station placement and the capability of returning to Shuttle altitude in the event of spacecraft failure. The six 8-cm thrusters combine with the 30-cm thrusters to give complete redundancy in North-South stationkeeping, East-West stationkeeping, and altitude control. A dry mass statement of this system is shown in Table 3-17. This mass statement assumes that different power processing units (PPU) are required to power the 8-cm and 30-cm thrusters. Further, it is assumed that, at most, two would need to be fired at any one time. Additionally, 3.5 percent of the propellant mass is allocated for propellant tanks, etc.

TABLE 3-17. COMBINED 8-CM/30-CM ION SYSTEM

Item	Unit Mass, kg	Number Required	Mass, kg
30-cm thrusters	7.8	2	15.6
8-cm thrusters	3.4	6	20.4
PPU (for 30-cm thrusters)	25	3	75
PPU (for 8-cm thrusters)	10	3	30
Switching matrix	5	1	5
Solar array	90	1	90
Cables, propellant lines, contingency			20
Total mass			256

The total mass of a SSUS-A class spacecraft after AKM burn and satellite placement is a maximum of 944 kg. For this initial mass, the propellant required to return to the Shuttle is estimated to be 165.4 kg. Subtracting the system dry mass, propellant and tanks leaves a net spacecraft mass of 517 kg. The impact on net spacecraft mass can be seen in Table 3-18, where the spacecraft net masses are calculated with consistent assumptions without return capability.

TABLE 3-18. NET GEOSYNCHRONOUS SPACECRAFT MASS WITH/ WITHOUT RETURN CAPABILITY

Spacecraft Propulsion[a]	Return to Shuttle	PKM	Net Spacecraft Mass, kg
Hydrazine	No	SSUS-D	384
Augmented electro-thermal hydrazine	No	SSUS-D	420
8 + 30-cm ion	Yes	SSUS-A	517
Hydrazine	No	SSUS-A	740
Hydrazine + 8-cm ion	No	SSUS-A	817

(a) Spacecraft propulsion for non-returning spacecraft taken from Table 3-16.

The net spacecraft mass for a return capability falls between the maximum SSUS-D and SSUS-A capabilities. Comparison of the net spacecraft mass with the SSUS-A spacecraft mass shows the spacecraft with a return capability has a net mass one-fourth less than the spacecraft using hydrazine for stationkeeping and one-third less than the spacecraft using an ion system for stationkeeping. Comparing the net mass of 517 kg to existing spacecraft shows it is larger than all SSUS-D or Delta class spacecraft, but somewhat less than the Atlas/Centaur class spacecraft. The Intelsat IVA does not use full Atlas/Centaur capability; but its net mass, using the definitions of net mass used here, is about 620 kg.

Another concern is that the spacecraft is using propellants for stationkeeping. This reduces the capability for returning to the Shuttle altitude. The nominal propellant use for stationkeeping is about 1.6 kg/ year. This results in higher retrieval orbit. The trade-off based upon years from launch is shown in Figure 3-21. If it is desired to be able to return to 300 km for a period of a few years, it would be necessary to add 1.6 kg of propellant per year to the initial propellant capability.

3.3 MMS Module Sizing

In sizing propulsion modules for the MMS missions many options are available. In this section, a set of modules based upon hydrazine technology and a set of modules based upon use of bipropellants are proposed. The ground rules for sizing the modules will be examined in the sensitivity analysis to determine their impact on the overall cost.

The hydrazine modules are based on using the SPS-I and SPS-II designs contained in Rockwell's Landsat analysis(3-19) and modifications of SPS-II using multiple Viking tanks clustered to maintain the length of the SPS-II system. Additionally, two missions, Upper Atmospheric Explorer (10-deg inclination) and Stormsat, were assigned to solids due to the large impulses required by these missions.

There are several uncertainties connected with the MMS payload positioning and retention system which potentially affect module configurations. Current information shows the retention system as a 3.3-m pallet which is mounted in the Shuttle cargo bay. It is unclear as to whether or not the full length of the pallet must be carried, regardless of payload length. Uncertainty in the details of how the payload is attached within the retention system also leads to speculation concerning the ability to reduce total payload length by shortening the propulsion.

The actual module and cradle design is not within the scope of this study; thus, it was assumed that the MMS could be operated from a cradle similar to that used for SSUS. This configuration does not add any substantial length to that already occupied by the payload/propulsion system. Furthermore, the uncertainty with respect to the attack points affects tank arrangements for multiple tank configurations. To alleviate this area of concern, all of the designs are structured to permit access to the three corners of the MMS bus from the aft end.

Use of the 5-lb MMS thrusters for these spacecraft results in a low ratio of thrust to weight. Studies by Rockwell(3-19) indicate that this

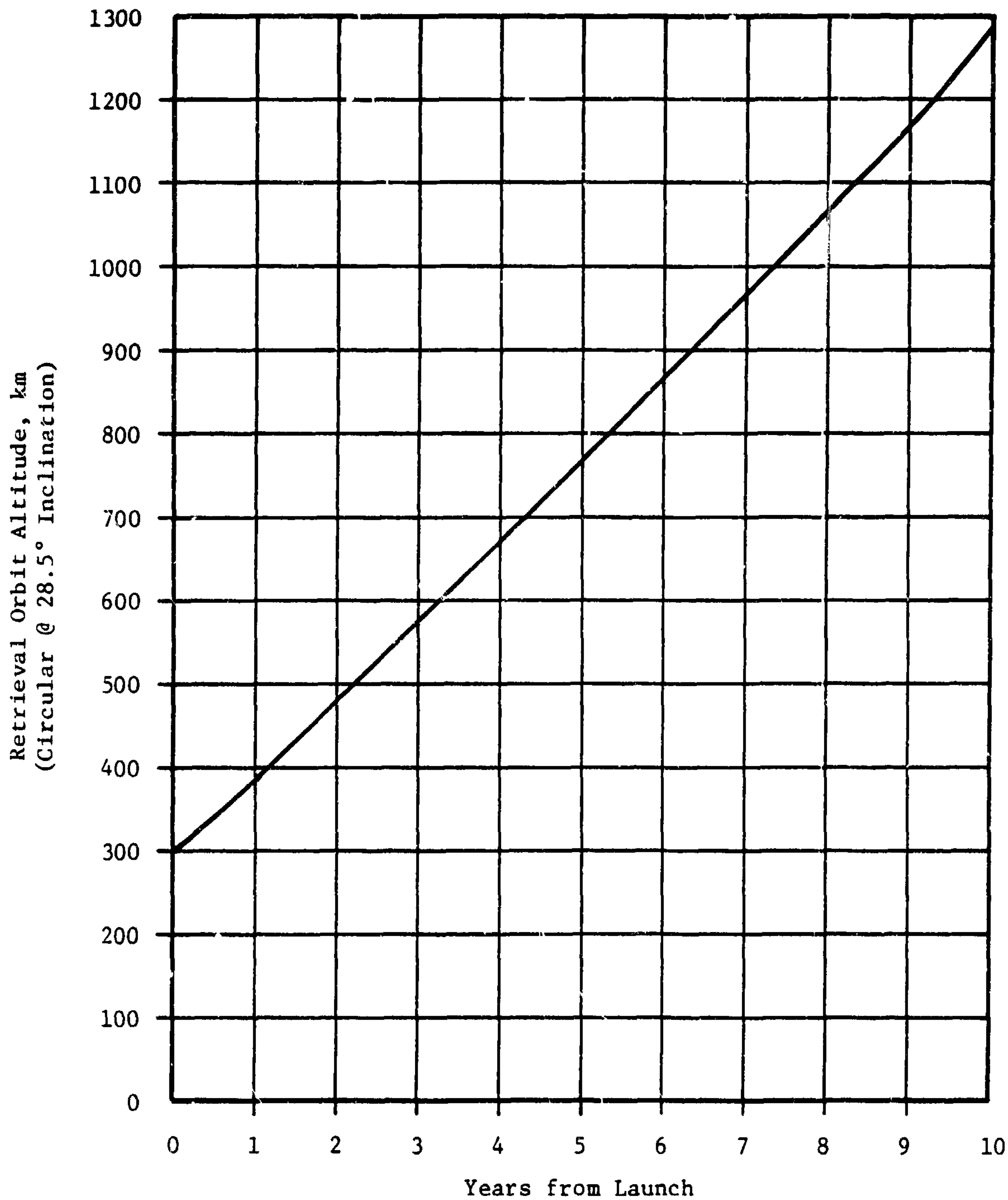


FIGURE 3-21. SEP CAPABILITY TO RETURN SPACECRAFT TO RETRIEVAL ORBIT

increases the velocity requirements by 4 to 6 percent. Thus, to allow sufficient reserves (5 percent), the velocity requirements detailed earlier in Table 3-5 were multiplied by 1.1 for hydrazine and 1.05 for solids or bipropellants. The module weights for the missions shown in Table 3-5 are given in Table 3-19. In addition to module weights and sizes, Shuttle load factors have been calculated. In all cases, the payloads were length critical, that is, the Shuttle charge would be based upon the length factor.

The bipropellant modules are based upon using TRW's multimission module(3-20) for the missions it could handle and a cylindrical tank with common bulkhead design for the larger missions. The TRW module has four propellant tanks; however, for some of the missions with small propulsion requirements, a two-tank version will suffice. Nitrogen tetroxide and MMH have density values which would produce an offset in the propulsion module lateral c.g. when using this two-tank derivative. This situation could be used to compensate for misalignment in the lateral c.g. location for the combined MMS bus plus payload. If balance problems persist, ballast could be added. Further studies should investigate any complications involved with this configuration or any similar design. A reserve of 5 percent was assumed for bipropellant systems; the module sizes for the missions are shown in Table 3-20. As in the hydrazine case, the Shuttle length factor is critical.

The hydrazine and bipropellant propulsion systems used in this analysis are shown in Figures 3-22 through 3-29.

TABLE 3-19. MONOPROPELLANT HYDRAZINE SYSTEMS

Mission	Shuttle RET/SER[a]	Mission Type	ΔV[b], m/sec	Mass of Indicated Component, kg: S/C + Bus	Dry	Hydrazine	Upper State	Total	Mass Load Factor[c]	Length of Indicated Component, m: S/C	Bus	Hydrazine	Upper Stage	Total	Length Load Factor[d]
HE-08A	–	Deploy	156	8635	206	663	0	9504	0.43	5.2	1.3	1.5	0	8.0	0.58
	SH	Return	156	8635	206	663	0	9504	0.43	5.2	1.3	1.5	0	8.0	0.58
	SH	Service	312	8635	360	1400	0	10,395	0.47	5.2	1.3	1.5	0	8.0	[illegible]
HE-07A	–	Deploy	126	735	69	48	0	852	0.04	0.3	1.3	0.5	0	2.1	0.15
HE-27A	–	Deploy	46	735	69	17	0	821	0.04	0.3	1.3	0.5	0	2.1	0.15
SO-03A	–	Deploy	172	1635	132	147	0	1914	0.09	2.0	1.3	1.5	0	4.8	0.35
	SH	Return	343	1635	132	304	0	2071	0.09	2.0	1.3	1.5	0	4.8	0.35
	SH	Service	515	1635	132	476	0	43	0.10	2.0	1.3	1.5	0	4.8	0.35
AP-01A (100)	–	(e)	2780 (perigee)	795	132	338	2593[f]	3858	0.17	0.3	1.3	1.5	1.9	5.0	0.36
	–	Deploy	671 (apogee)	–	–	–	–	–	–	–	–	–	–	–	–
(28.50)	SH	(g)	1373	795	206	891	0	1892	0.09	0.3	1.3	1.5	0	3.1	0.23
(560)	DED	(g)	1373	795	206	891	0	1892	0.10	0.3	1.3	1.5	0	3.1	0.23
(900)	DED	(g)	1373	795	206	891	0	1892	0.14	0.3	1.3	1.5	0	3.1	0.23
AP-02A (28.50)	–	Deploy	1782	735	283	1307	0	2325	0.11	0.3	1.3	1.5	0	3.1	0.23
(560)	–	Deploy	1782	735	283	1307	0	2325	0.12	0.3	1.3	1.5	0	3.1	0.23
EO-08A	–	Deploy	284	1595	132	243	0	1970	0.18	2.0	1.3	1.5	0	4.8	0.35
	SH	Return	540	1595	132	491	0	2218	0.20	2.0	1.3	1.5	0	4.8	0.35
	SH	Service	1238	1595	360	1515	0	3470	0.32	2.0	1.3	1.5	0	4.8	0.35
EO-12A	–	Deploy	341	1635	132	303	0	2070	0.19	4.0	1.3	1.5	0	6.8	0.50
	SH	Return	698	1635	206	703	0	2544	0.23	4.0	1.3	1.5	0	6.8	0.50
	SH	Service	1200	1635	360	1484	0	3479	0.32	4.0	1.3	1.5	0	6.8	0.50
EO-13A	–	Deploy	341	1635	132	303	0	2070	0.19	4.0	1.3	1.5	0	6.8	0.50
	SH(SS)	Return	1204	1635	360	1491	0	3486	0.32	4.0	1.3	1.5	0	6.8	0.50
	SH(SS)	Service	1732	1635	512	2645	0	4792	0.44	4.0	1.3	1.5	0	6.8	0.50
	–	Deploy	341	1635	132	303	0	2070	0.19	4.0	1.3	1.5	0	6.8	0.50
	DED	Return	666	1635	206	666	0	2507	0.23	4.0	1.3	1.5	0	6.8	0.50
	DED	Service	990	1635	283	1117	0	3035	0.28	4.0	1.3	1.5	0	6.8	0.50
EO-15A	–	(e)	2440 (perigee)	995	0	0	5301[h]	7463	0.34	1.5	1.3	0	4.5	7.3	0.5[illegible]
	–	(e)	1830 (apogee)	–	–	–	1167[i]	–	–	–	–	–	–	–	–
EO-61A	–	Deploy	280	1135	132	176	0	1443	0.13	4.0	1.3	1.5	0	6.8	0.50
	SH	Return	704	1135	132	489	0	1756	0.16	4.0	1.3	1.5	0	6.8	0.50
	SH	Service	1204	1135	283	1060	0	2478	0.23	4.0	1.3	1.5	0	6.8	0.50
EO-64A	–	Deploy	341	1635	132	303	0	2070	0.19	4.0	1.3	1.5	0		0.50
	SH	Return	69[illegible]	1635	206	703	0	2544	0.23	4.0	1.3	1.5	0		0.50
	SH	Service	1200	1635	360	1484	0	3479	0.32	4.0	1.3	1.5	0		0.50
EO-65A	–	Deploy	205	2635	132	276	0	3043	0.28	4.0	1.3	1.5	0	6.8	0.50
	SH	Return	462	2635	206	678	0	3519	0.32	4.0	1.3	1.5	0	6.8	0.50
	SH	Service	891	2635	360	1531	0	4526	0.42	4.0	1.3	1.5	0	6.8	0.50
OPN-02A	SH(P)	Deploy	844	885	132	487	0	1504	0.11	1.0	1.3	1.5	0	3.8	0.28
	DED	Return	1156	885	206	773	0	1864	0.14	1.0	1.3	1.5	0	3.8	0.28
	DED	Service	1459	885	283	1129	0	2297	0.17	1.0	1.3	1.5	0	3.8	0.28

(a) RET = return; SER = service; SH = shared; DED = dedicated; SH(SS) = shared sun-syncronous, and SH(P) = shared polar.
(b) ΔV is defined as 1.10 times mission velocity requirement.
(c) Load factor = (cargo mass/Shuttle maximum mass) x 1.33.
(d) Load factor = (cargo length/Shuttle maximum length) x 1.33.
(e) ΔV for solid motors is defined as 1.05 times mission requirement.
(f) Offloaded version of small IUS motor (338 kg of propellant were removed).
(g) ΔV assumed constant for deploy, return, and service missions.
(h) Stretched version of small IUS motor with 4200 kg of propellant (including 66-kg adapter).
(i) TE-M-364-4 motor (mass includes 45-kg adapter).

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TABLE 3-20. BIPROPELLANT PROPULSION SYSTEMS

Mission	Shuttle RET/SER[a]	Mission Type	ΔV[b], m/sec	Mass of Indicated Component, kg: S/C + Bus	Dry	Bipro-pellant	Upper Stage	Total	Mass Load Factor[c]	Length of Indicated Component, m: S/C	Bus	Bipro-pellant	Upper Stage	Total	Length Load Factor[d]
HE-08A	–	Deploy	149	8635	86	461	0	9182	0.42	5.2	1.3	1.2	0	7.7	0.56
	SH	Return	149	8035	86	461	0	9182	0.42	5.2	1.3	1.2	0	7.7	0.56
	SH	Service	298	8635	265	966	0	9866	0.45	5.2	1.3	1.7	0	8.2	0.60
HE-07A	–	Deploy	121	735	71	35	0	841	0.02	0.3	1.3	1.2	0	2.8	0.20
HE-27A	–	Deploy	44	735	71	12	0	818	0.04	0.3	1.3	1.2	0	2.8	0.20
SO-03A	–	Deploy	164	1635	71	100	0	1806	0.08	2.0	1.3	1.2	0	4.5	0.33
	SH	Return	328	1635	71	206	0	1911	0.09	2.0	1.3	1.2	0	4.5	0.33
	SH	Service	491	1635	86	318	0	2039	0.09	2.0	1.3	1.2	0	4.5	0.33
AP-01A (100)	–	Deploy	2780 (perigee)	795	0	0	2352[e]	3432	0.16	0.3	1.3	1.2	1.9	4.7	0.34
	–	Deploy	640 (apogee)	–	71	214	0	–	–	–	–	–	–	–	–
(28.50)	SH	(f)	1310	795	86	505	0	1386	0.06	0.3	1.3	1.2	0	2.8	0.20
(560)	DED	(f)	1310	795	86	505	0	1386	0.07	0.3	1.3	1.2	0	2.8	0.20
(900)	DED	(f)	1310	795	86	505	0	1386	0.10	0.3	1.3	1.2	0	2.8	0.20
AP-02A (28.50)	–	Deploy	1701	735	265	800	0	1800	0.08	0.3	1.3	1.7	0	3.3	0.24
(560)	–	Deploy	1701	735	265	800	0	1800	0.09	0.3	1.3	1.7	0	3.3	0.24
EO-08A	–	Deploy	271	1595	71	164	0	1830	0.17	2.0	1.3	1.2	0	4.5	0.33
	SH	Return	516	1595	86	328	0	2009	0.18	2.0	1.3	1.2	0	4.5	0.33
	SH	Service	1181	1595	265	938	0	2798	0.26	2.0	1.3	1.7	0	5.0	0.36
EO-12A	–	Deploy	326	1635	71	203	0	1909	0.18	4.0	1.3	1.2	0	6.5	0.47
	SH	Return	667	1635	86	446	0	2167	0.20	4.0	1.3	1.2	0	6.5	0.47
	SH	Service	1144	1635	265	922	0	2822	0.26	4.0	1.3	1.7	0	7.0	0.51
EO-13A	–	Deploy	326	1635	71	203	0	1909	0.18	4.0	1.3	1.2	0	6.5	0.47
	SH(SS)	Return	1150	1635	265	927	0	2827	0.28	4.0	1.3	1.7	0	7.0	0.51
	SH(SS)	Service	1654	1635	265	1465	0	3365	0.31	4.0	1.3	1.7	0	7.0	0.51
	–	Deploy	326	1635	71	203	0	1909	0.18	4.0	1.3	1.2	0	6.5	0.47
	DED	Return	635	1635	86	422	0	2143	0.20	4.0	1.3	1.2	0	6.5	0.47
	DED	Service	946	1635	265	734	0	2634	0.24	4.0	1.3	–	0	6.5	0.47
EO-15A	–	Deploy	2440 (perigee)	995	0	0	5375[g]	7747	0.35	1.5	1.3	1.7	2.8	7.3	0.53
	–	Deploy	1830 (apogee)	–	265	1112	0	–	–	–	–	–	–	–	–
EO-61A	–	Deploy	268	1135	71	117	0	1323	0.12	4.0	1.3	1.2	0	6.5	0.47
	SH	Return	672	1135	86	319	0	1540	0.14	4.0	1.3	1.2	0	6.5	0.47
	SH	Service	1150	1135	86	596	0	1817	0.17	4.0	1.3	1.2	0	6.5	0.47
EO-64A	–	Deploy	326	1635	71	203	0	1909	0.18	4.0	1.3	1.2	0	6.5	0.47
	SH	Return	667	1635	86	446	0	2167	0.20	4.0	1.3	1.2	0	6.5	0.47
	SH	Service	1144	1635	265	922	0	2822	0.26	4.0	1.3	1.7	0	7.0	0.51
EO-65A	–	Deploy	195	2635	71	189	0	2895	0.27	4.0	1.3	1.2	0	6.5	0.47
	SH	Return	441	2635	86	448	0	3169	0.29	4.0	1.3	1.2	0	6.5	0.47
	SH	Service	850	2635	265	990	0	3890	0.36	4.0	1.3	1.7	0	7.0	0.51
OPN-02A	SH(P)	Deploy	805	885	86	312	0	1283	0.09	1.0	1.3	1.2	0	3.5	0.26
	DED	Return	1104	885	86	451	0	1422	0.11	1.0	1.3	1.2	0	3.5	0.26
	DED	Service	1392	885	265	711	0	1861	0.14	1.0	1.3	1.7	0	4.0	0.29

(a) RET = return; SER = service; SH = shared; DED = dedicated; SH(SS) = shared sun-synchronous, and SH(P) = shared polar.
(b) ΔV is defined as 1.05 times mission velocity requirement.
(c) Load factor = (cargo mass/Shuttle maximum mass) x 1.33.
(d) Load factor = (cargo length/Shuttle maximum length) x 1.33.
(e) Small IUS motor with 600-kg offload (includes 47-kg adapter).
(f) ΔV assumed constant for deploy, service, and return missions.
(g) Stretched version of small IUS motor with 4450 kg of propellant (mass includes 70-kg adapter).

Diameter = 1.2 m
Length = 0.5 m

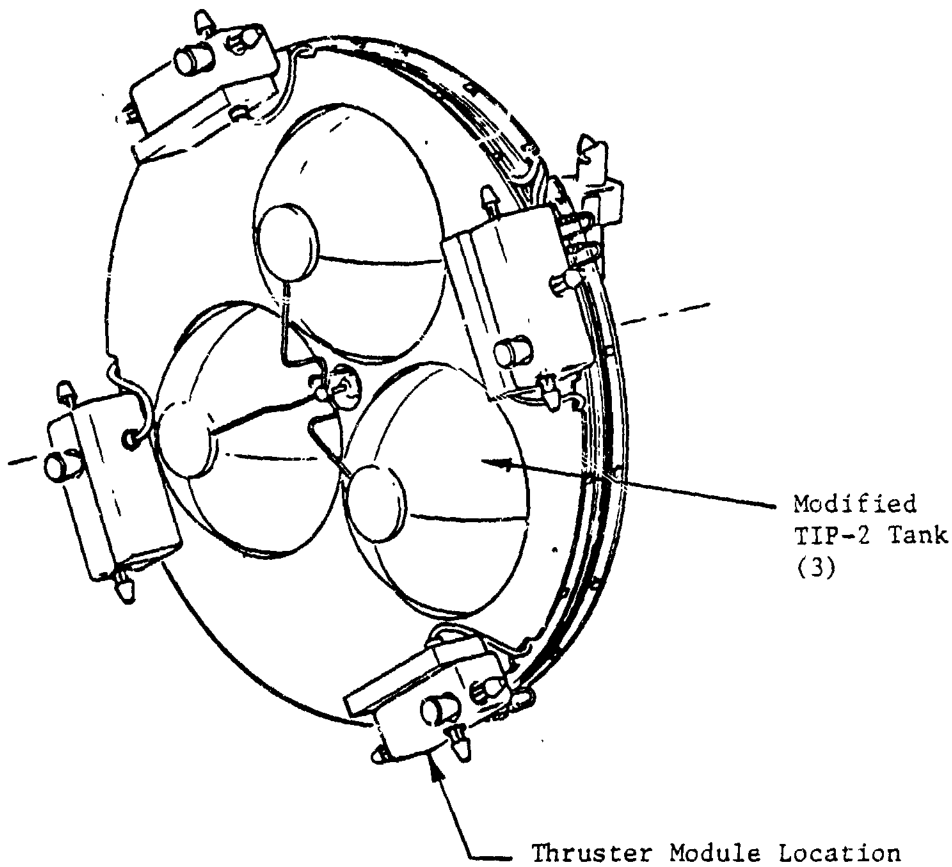

FIGURE 3-22. ROCKWELL SPS-I DESIGN (3-19)

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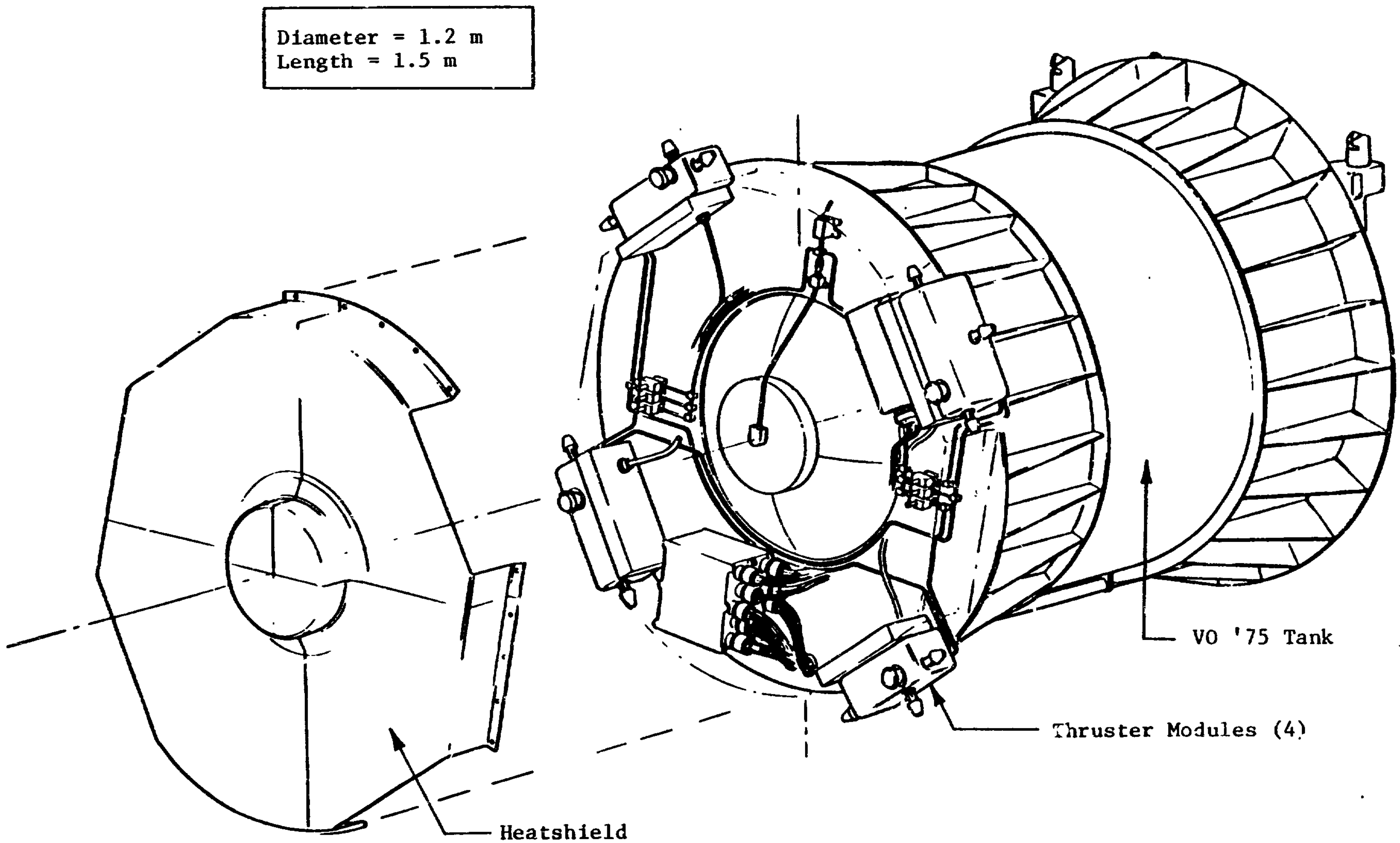

FIGURE 3-23. ROCKWELL SPS-II HYDRAZINE SYSTEM (3-19)

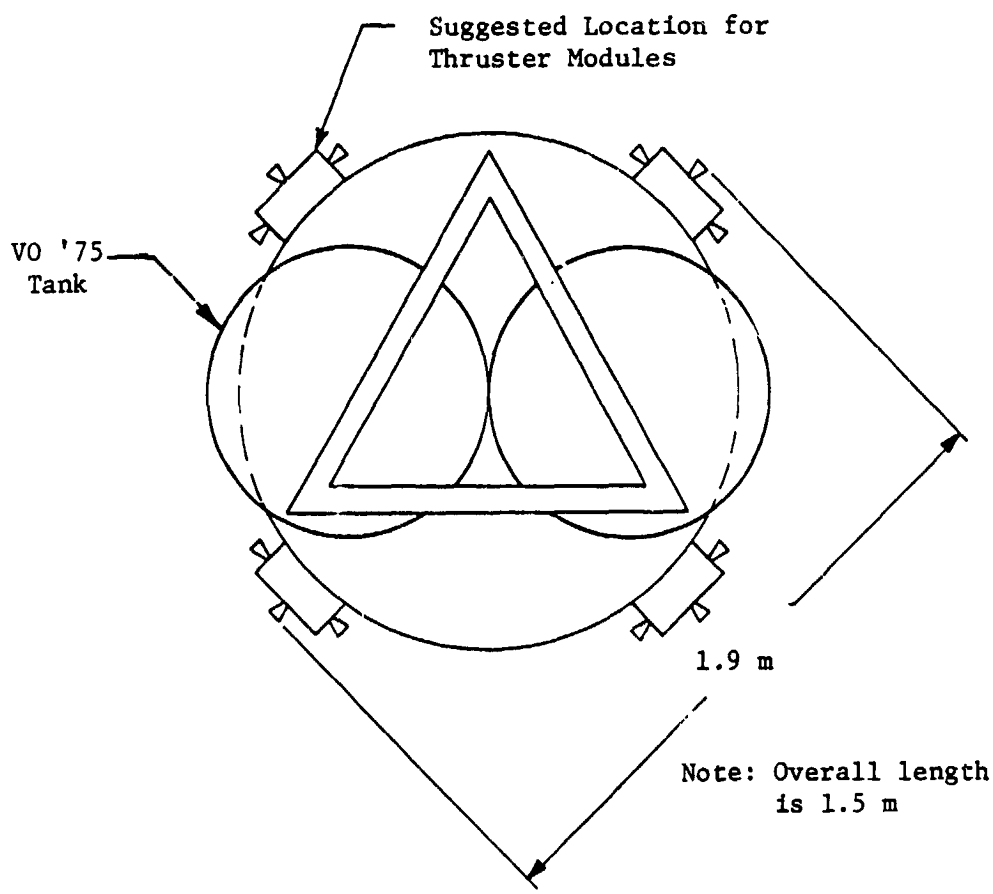

FIGURE 3-24. TWO-TANK MODIFIED SPS-II HYDRAZINE DESIGN (TOP VIEW)

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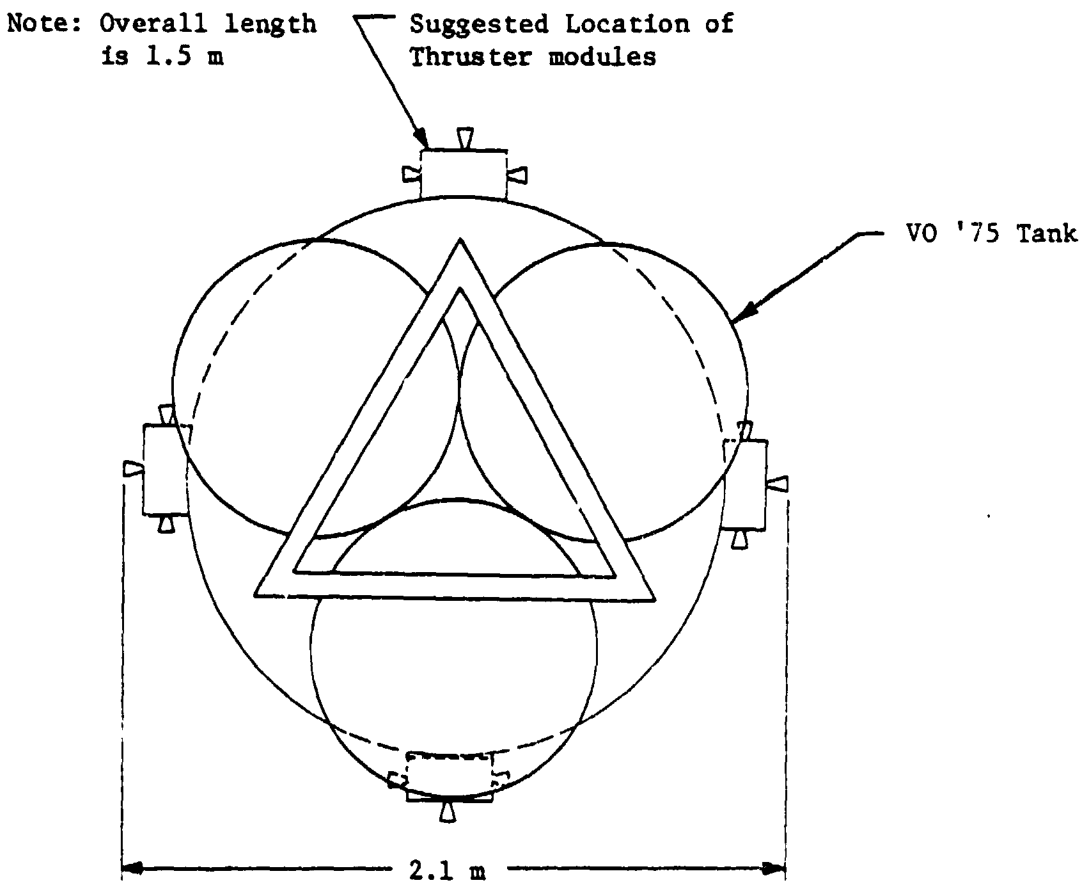
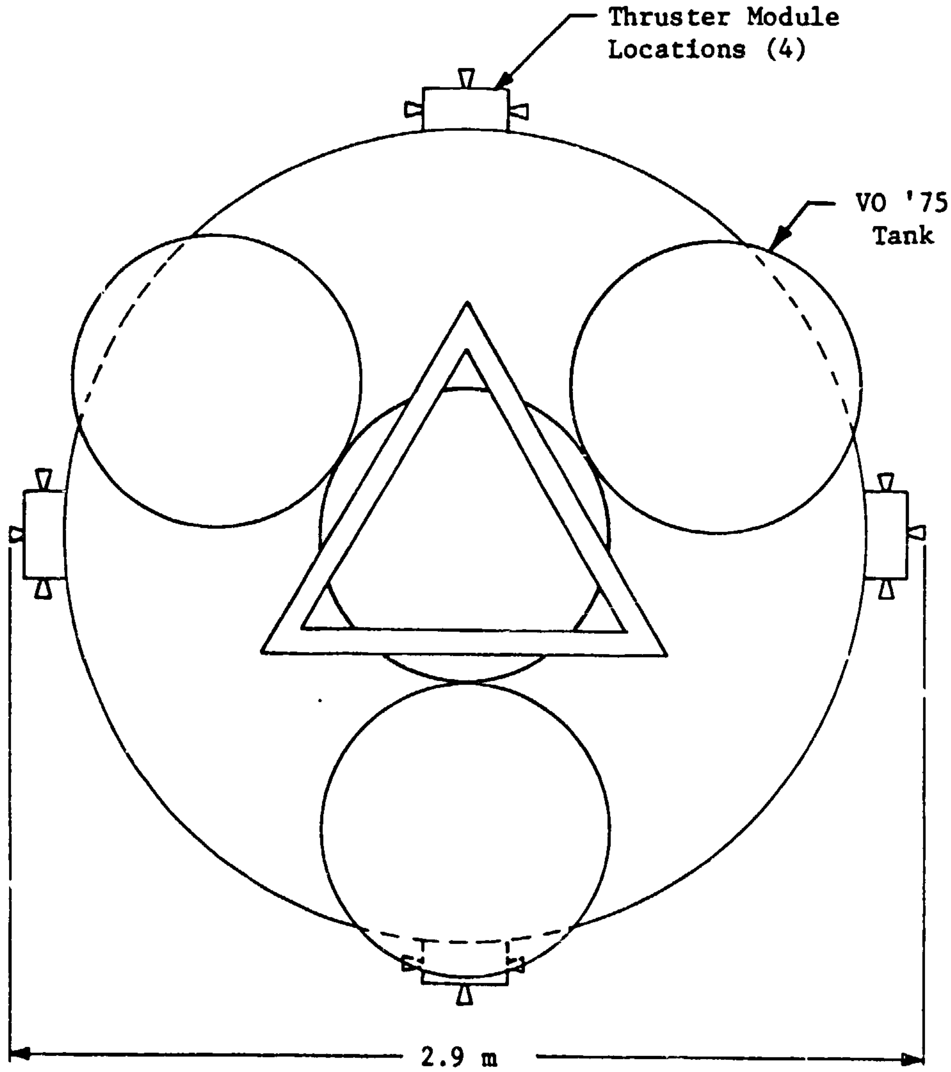
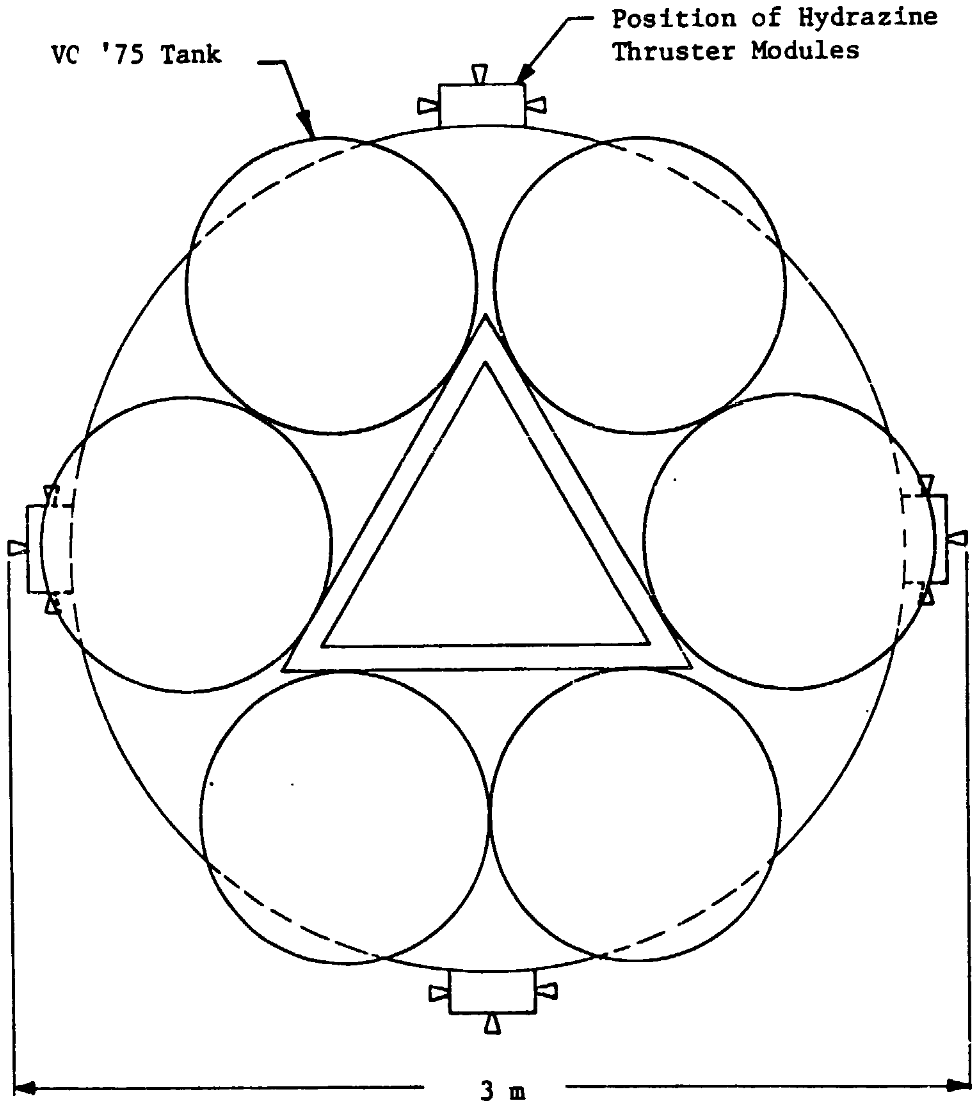


FIGURE 3-25. THREE-TANK SPS-II DERIVATIVE MODULE
(TOP VIEW)



NOTE: Total length is 1.5 m

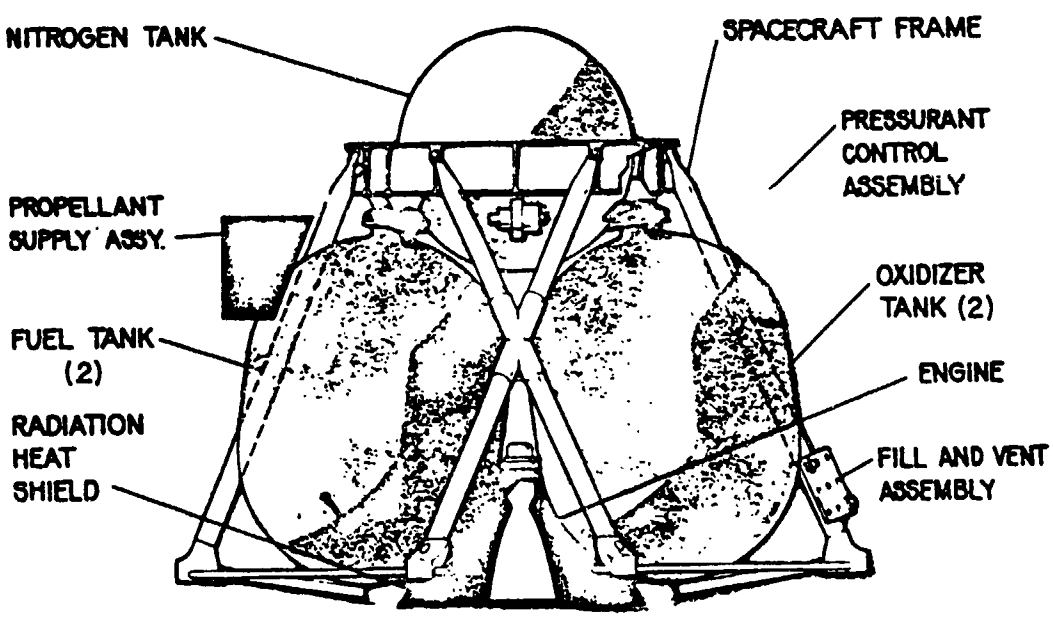
FIGURE 3-26. MODIFIED SPS-II SYSTEM WITH FOUR VIKING ORBITER 1975 TANKS (TOP VIEW)



NOTE: Overall length is 1.5 m

FIGURE 3-27. SIX-TANK MODIFIED SPS-II PROPULSION SYSTEM (TOP VIEW)

Diameter = 1.5 m
Length = 1.2 m



Note: Two tank version is derived by removing 1 oxidizer and 1 fuel tank.

FIGURE 3-28. TRW MULTIMISSION BIPROPELLANT PROPULSION SYSTEM (3-20)

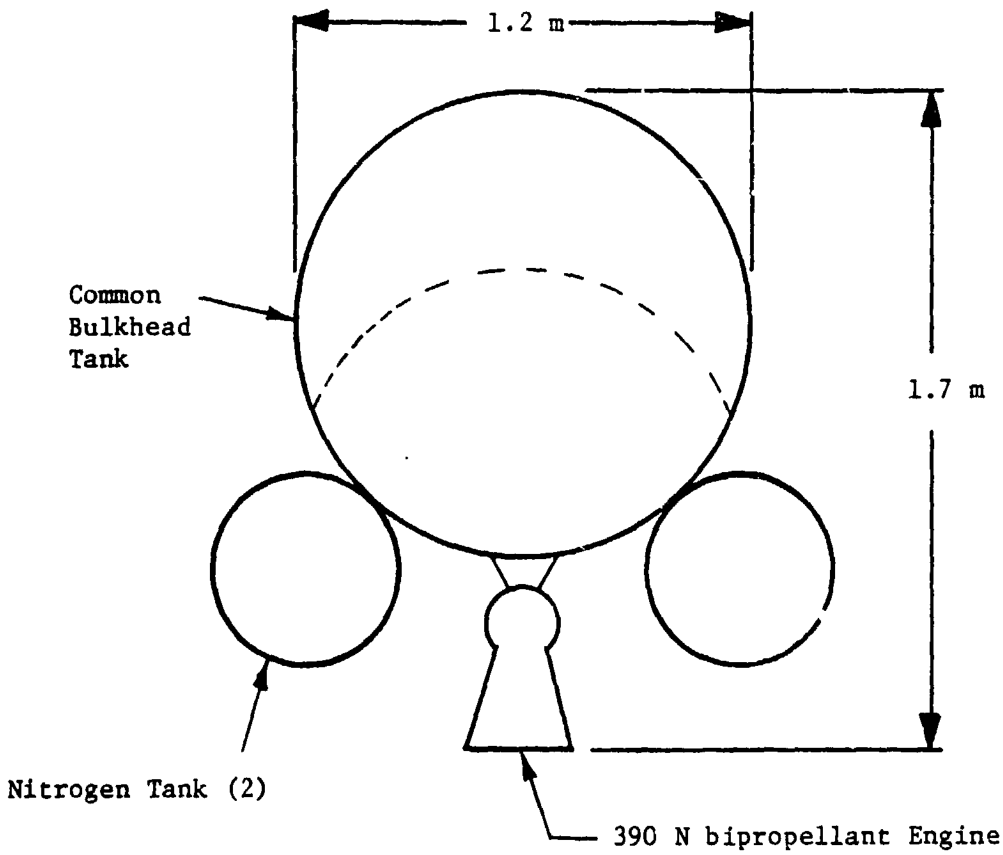


FIGURE 3-29. LARGE BIPROPELLANT SYSTEM CONTAINING APPROXIMATELY 1500 KILOGRAMS OF PROPELLANTS

3.4 References

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4.0 COST ASSESSMENTS FOR MMS PROPULSION REQUIREMENTS

4.1 Introduction

This section of the report documents and derives where necessary the development and recurring unit costs used in analyzing the controllable program transportation costs. The greatest attention is focused on monopropellant (N_2H_4) and bipropellant (N_2O_4/N_2H_4) module technologies since these are available without extensive developments and are applicable to the missions under consideration. Solar electric propulsion (SEP) costs are also discussed in some detail since this is considered to be the most promising future technology to meet long-range propulsion requirements. We consider SEP in both a primary propulsion role and for secondary propulsion (stationkeeping, attitude control, drag makeup) in conjunction with primary chemical propulsion.

Chemical propulsion technologies, and especially monopropellant hydrazine, are currently in use and are planned for the initial, expendable vehicle use of the MMS. The government program costs for the initial monopropellant hydrazine module are thus fixed and not subject to control, in that they cannot be selected or rejected, as is the situation for potential future technologies such as SEP. In addition, the government support of the hydrazine modules is part of the support for the MMS bus program and not readily separable from that program. The government support for bipropellant modules, approximately equivalent to that for monopropellant modules, can be expected to involve only a few additional people. Accordingly, this report considers only the hardware and space transportation costs for both of these storable chemical propulsion modules.

Solar electric propulsion technology, however, is not yet operationally available and is expected to cost significantly more both in terms of hardware and support. The significant difference in the cost implications between the two programs is handled by using the SEP hardware costs and contractor estimates of the SEP support costs parametrically in the analysis of the benefits achieved for SEP applications for primary propulsion. This will underburden SEP applications in relation to chemical propulsion when only hardware costs are considered, and will overburden

SEP when both hardware and support are considered. This method thus provides upper and lower bounds on the SEP module in comparison with chemical propulsion.

Electrothermal hydrazine thrusters for secondary propulsion are also considered. Because relatively little experience with this technology is available, the cost implications are not well known. The thruster assemblies themselves are not expected to be the significant cost in using this technology. The significant cost, rather, is expected to come from the provision of electrical power. Since the payload competes with electrical requirements for propulsion, the benefits of potentially lower propulsion weight (and cost) must be traded against higher costs and weights for the solar arrays (and batteries) to judge the net benefit for this technology.

Transportation costs used in this study for the Shuttle, as well as identified Shuttle services, are derived from the latest available documentation. The costs for solid rocket motor (SRM) propulsion for the cases where it is applicable are taken from the latest available documentation. While this documentation does not reflect formal NASA estimates, the costs given are comparable with historical costs for SRM stages in unmanned applications.

Hydrogen/oxygen propulsion modules are not considered in this study both for the technical reason that the cryogenic propellants would evaporate during extended missions and for cost reasons: no cases were identified where the benefit of reducing Shuttle charges through the lower weight and size of H_2/O_2 propulsion would justify the high development cost and significantly higher recurring costs in relation to a storable propellant module of the same capability.

4.2 Monopropellant and Bipropellant Module Cost Estimates

The hardware and support cost structure used for this study is summarized in Table 4-1. This is a generalized structure which is modified to reflect differences in technology and terminology specific to that technology.

4.2.1 Monopropellant Module

The cost estimates for the monopropellant hydrazine module are extracted from a Rockwell International report: Landsat/MMS Propulsion Module Design[(4-1)*], and are increased to reflect fees (7.5%) and inflation from 1976.7 to 1977.5 (5%). The fee rate of 7.5% represents a typical negotiated fee for aerospace contracts of moderate risk; the inflation

*References, denoted by superscript numbers, are in Subsection 4.7.

adjustment of 5% is the estimated change in the Consumer Price Index (CPI) during the year between the studies. All estimates in this report are adjusted to 1977 (June) dollars based on the CPI. The CPI reflects inflation in the economy overall rather than specifically in the manufacturing sector. Our analysis indicates that aerospace, as a labor-sensitive industry, also reflects inflation in the same manner as the CPI.(4-2)

TABLE 4-1. HARDWARE AND SUPPORT COST STRUCTURE

1. Hardware
    (a) Development of Hardware (including Qualification Test Vehicle)

        Structure
        Thermal Control
        Propulsion - Main Thrusters
                     Attitude Thrusters
                     Tanks
                     Other

        Electrical and Electronics
        Integration and Assembly

    (b) Contractor Program

        Aerospace Ground Equipment (AGE)
        Simulators
        Identification of Launch Tasks
        Design/Manufacturing Verification Tests
        Systems Engineering
        Project Management
        Fee at 7.5%

    (c) Recurring Unit Costs for a and b Above.

2. Government Support (Uncosted except for SEP Module)

    Software
    Systems Engineering
    Shuttle Adaption
    Develop Procedures for Launch Operations
    New AGE and Other New GFE
    Module/Bus/Spacecraft Design Verification and Integration
    Launch Operations
    Mission Support
    NASA Program Management

The development and recurring costs presented in Table 4-2 are for a Shuttle-launched module of 1000 lb propellant weight using four 5-lb catalytic thrusters for primary propulsion and twelve 0.2-lb catalytic thrusters for auxiliary propulsion. The development costs include those of a qualification test module (QTM). The estimate is directly applicable to referenced designs and covers the hardware manufacturer's costs and fee only. Government support costs outside the manufacturer's plant are not included. The hardware costs generally reflect an existing, flight-qualified component and assume no concurrent production. The hardware cost tolerance cited in the Rockwell report is ± 15% in 1976 dollars.[4-1] A telephone conversation with Mr. W. Cooper, one of the authors of the Rockwell report, confirmed that the cost estimates are dependent upon the continuing availability of the flight-qualified components selected, or their equivalents.

The four-thruster design was selected to avoid the complexities and additional costs associated with a gimbaled single thruster.

TABLE 4-2. MONOPROPELLANT HYDRAZINE MODULE HARDWARE CONTRACTOR COSTS[a] ($, Millions, 1976)

	Maximum[b] of SPS I and II	
	Non-Recurring[c]	Recurring
Hardware		
Structure	$0.246M	$0.050M
Thermal Control	0.025	0.008
Propulsion		
- Four 5-Lb Thrusters	0.058	0.043
- Twelve 0.2-Lb Thrusters	0.170	0.160
- Tanks	0.090	0.060
- Other	0.433	0.227
Electrical and Electronics	0.136	0.113
Integration and Assembly	0.096	0.010
	1.254	0.671
Contractor Program		
Aerospace Ground Equipment	0.054	0.001
Simulators	0.011	0.002
Identification of Launch Tasks	0.022	-
Design/Manufacturing Verification Tests	0.136	0.030
System Engineering	0.222	0.081
Project Management	0.265	0.070
Fee at 7.5%	0.147	0.064
	$2.111M	$0.919M
Adjust to 1977.5 @ 5%	$2.215M	$0.965M

(a) Based on Rockwell Landsat/MMS Module, Reference (4-1).
(b) The maximum is taken to fully reflect non-recurring costs.
(c) Non-recurring costs include a Qualification Test Module (QTM).

4.2.2 Bipropellant Module

The bipropellant module cost estimate given in Table 4-3 is for a module approximately equivalent in performance to the monopropellant module of the previous section. The bipropellants selected are N_2O_4/N_2H_4 (hydrazine) rather than N_2O_4/MMH (monomethyl hydrazine) so that the main propellant tank can feed hydrazine to both the main engine (22 or 23-lb thrust) and the auxiliary thrusters, as in the case of the monopropellant module. This configuration is believed to have advantages for long-term propellant management as well as lower component costs over separate tanks for the auxiliary thrusters. The use of hydrazine rather than MMH also provides a slightly higher specific impulse and avoids the rapid degradation of typical catalysts by methylated hydrazines. (See Subsection 2.7 for the discussion of the technical reasons and problems associated with this choice.)

The cost estimates of Table 4-3 are based on data from Lewis Research Center (LeRC) for the propulsion system.[(4-5)] Other systems and contractor program costs are based on adjusted Rockwell Landsat/MMS costs. The adjustment is by a factor of 1.2 applied to the monopropellant subsystems except for the structure. This factor was determined from the relative costs of equivalent monopropellant and bipropellant auxiliary propulsion systems in Reference (4-4). This factor adjusts the cost impact of the relative complexity of bipropellant in relation to monopropellant technology.

4.2.3 Cost Effects of Propellant Weight Variation for Chemical Propellant Modules

The cost effect of propellant weight variation for the propulsion modules is estimated in two different ways; the number of propellant tanks can be increased or decreased or the size of the main tanks can be changed. Within some ranges of requirements of the mission model used for later analysis, it appears that changing the number of tanks is the least-cost method, while for some missions the use of a large number of tanks is not feasible. This report considers both methods.

TABLE 4-3. BIPROPELLANT N_2O_4/HYDRAZINE MODULE HARDWARE CONTRACTOR COSTS[a]
($, Millions, 1976)

	Non-Recurring[b]	Recurring
Hardware - Pressurized System, one 22-lb engine		
Structure[c]	$0.246M	$0.050M
Thermal Control[c]	0.030	0.010
Propulsion		
Fill and Drain	0.119	0.017
Pressurization System	0.555	0.137
Propellant Control	0.705	0.126
Propellant Feed System	0.658	0.152
Propellant Vent System	0.055	0.003
Thruster Assembly	0.615	0.052
Attitude Control Thrusters[c]	0.204	0.192
Attitude Control Valves, Latches Control[c]	0.090	0.090
Instrumentation	0.020	0.002
Integration and Assembly[c]	0.115	0.012
	3.412	0.843
Contractor Program[c]		
Aerospace Ground Equipment	0.065	0.001
Simulators	0.013	0.002
Identification of Launch Tasks	0.026	--
Design/Manufacturing Verification Tests	0.163	0.036
System Engineering	0.266	0.097
Project Management	0.318	0.084
Fee at 7.5%	0.364	0.088
	$4.629M	$1.151M
Adjust to 1977.5@ 5%	$4.86M	$1.20M

(a) Based on Rockwell Landsat/MMS Module[4-1] and Lewis Research Center data[4-3], with complexity adjustment based on Reference (4-4).
(b) Includes the cost of a Qualification Test Module (QTM).
(c) Items costed from Rockwell Landsat/MMS study with adjustment for change from monopropellant to bipropellant.

Both in discussions with the authors of Reference (4-1) and from other sources, we concluded that reasonable changes in the structural designs of a module to accommodate different numbers of tanks or the size of one tank is not a major cost item in either development or recurring cost. The major cost comes in the tanks, valves and their control mechanisms. Accordingly, costs for modules using multiple tanks are parameterized by the number of tanks using tanks and related costs, both for recurring and non-recurring costs. These estimates are shown later in the cost estimate summary (Section 4.6).

Only one design using a specialized single large tank is required for the mission model. For this design, cost estimating relationships developed in Subsection 4.2.4 were used for the tank; for other items, costs developed in this section were used. This larger bipropellant module with a propellant weight of 1522 kg (3350 lb) contained in a single tank with double-diaphram separating bulkhead is used as a baseline for analysis. An alternative employing two separate tanks is also used. The double-diaphram single tank design is estimated at $6.5M non-recurring and $2.0M recurring, while the two-tank design is estimated at $6.7M non-recurring and $2.1M per recurring unit. The cost impact of the new tanks is determined in the following section. The costs just given also reflect necessary revisions in the propellant management devices.

4.2.4 Propellant Tank Costs

In this study, a variety of module sizes are considered to meet requirements of the mission model and other forecasts of desired capabilities. Most of these requirements can be met with the same thruster combinations used on the baseline configuration, but require multiple propellant tanks or tanks of different sizes. The structure, unless under very severe weight constraints, is considered to have a much lower impact on costs than tanks and lines. In most of the modules considered, the non-recurring hardware costs have been determined under the assumption that existing hardware is adopted to the module. This results in multiple tank designs which have relatively high transportation costs from a less efficient loading in the Shuttle bay. For one bipropellant module, a single spherical tank with a twin bulkhead forming two hemispherical tanks is considered to determine whether the additional module costs can be offset by reduced Shuttle charges.

To judge the cost impact of this special design as well as the cost implications of going to specialized tank designs, the Precision Sheet Metal Division of Fansteel Corporation, an aerospace tank manufacturer, was contacted for estimates on tanks in the range from 227 to 2268 kg (500 to 5000 lb) of hydrazine propellant. The estimates for the specific sizes are given in Table 4-4 and the development and unit prices are given in Figure 4-1.

TABLE 4-4. DEVELOPMENT AND RECURRING COSTS FOR SPHERICAL PROPELLANT TANKS[a]

Propellant Weight[b]		Approximate Volume[c]		Diameter		Approximate Tank Wt.		ROM 1977 Costs[d], thousands of dollars	
Kg	Lb	M^3	$In.^3$	Cm	In.	Kg	Lb	Development	Unit
227	500	0.25	15,000	77.7	30.6	9	20	200	30
454	1000	0.49	30,000	97.8	38.5	18	40	290	47
680	1500	0.74	45,000	112.3	44.2	27	60	362	62
907	2000	0.98	60,000	123.2	48.5	36	80	428	67
1361	3000	1.48	90,000	141.2	55.6	54	120	546	99
1588	3500[e]	1.72	105,000	148.6	58.5	63	140	600	110
2268	5000	2.46	150,000	167.6	66.0	91	200	750	140

(a) The costs quoted do not represent a formal bid or estimate.
(b) Hydrazine propellant, nominal - includes allowance for internal propellant management devices.
(c) Includes 5% allowance for internal propellant management devices (bladder type).
(d) 1977 dollars in thousands; ROM = Rough Order of Magnitude = ± 15%.
(e) Special spherical tank with twin bulkhead for bipropellants.

From Figure 4-1, it is noted that the twin bulkhead tank of 1588 kg (3500 lb) nominal propellant weight and 1515 kg (3340 lb) net propellant weight has a lower development cost than might be expected for a single chamber tank of the same nominal propellant capacity. The recurring cost, however, is about the same as for a single chamber tank. From the relatively slow growth in tank costs as a function of propellant weight, it appears that tank costs, as a relatively small fraction of total design costs, need not be a barrier to design optimization.

The large tank module costs used in this report contain the estimated $600,000 development cost for the special tank and an additional allowance for propellant management devices. Other designs are estimated on the basis of existing components.

4.3 Solar Electric Propulsion Module

The cost estimates for a solar electric propulsion module (SEPM) are derived from a solar electric propulsion stage (SEPS) study by Boeing.[4-5] The cost estimates in this study are comparable to those in a similar study by Rockwell International[4-6] and are available to us in more detail than for the RI study. The estimates developed are also compared with electric

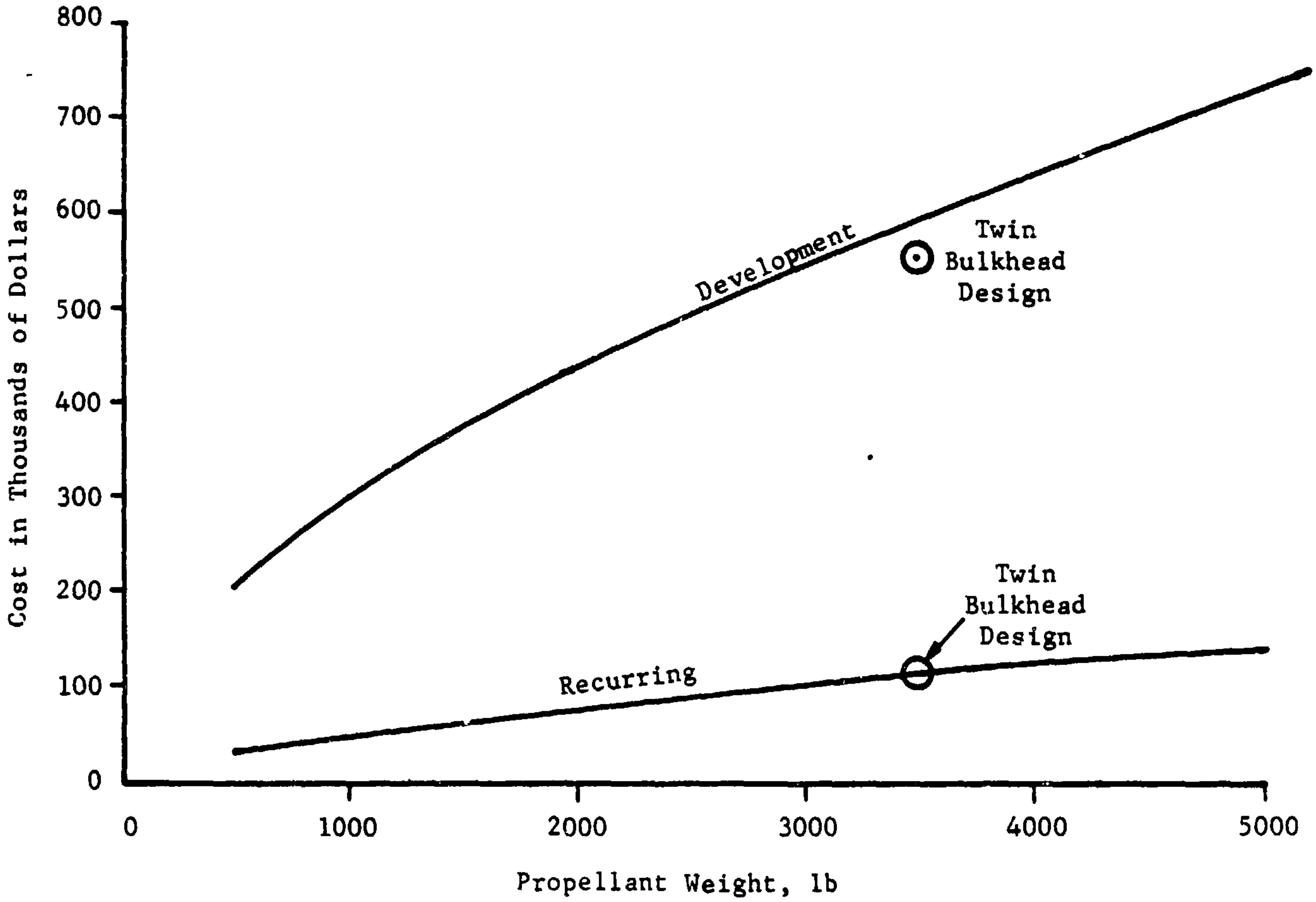

FIGURE 4-1. TANK DEVELOPMENT AND RECURRING COSTS AS A FUNCTION OF PROPELLANT WEIGHT

propulsion costs provided by LeRC[4-3] which do not include some elements such as structures. While the two sources use significantly different approaches, the results are shown to be compatible. The estimates derived from the Boeing data are used in subsequent analyses since they provide both hardware and program costs. The hardware development and recurring costs are used to provide a lower bound on the cost impact of SEP technology in comparison to chemical propulsion. The upper bound is then provided by the estimate of total program costs.

The solar electric propulsion module used as a baseline for costing purposes has three 30-cm ion engines and a solar array with an initial power level of 6.5 kw. The Boeing SEPS has ten 30-cm thrusters and an initial array power of 25 kw. The hardware related costs are scaled on power and number of thrusters. This scaling assumes that the SEPM will be designed and procured very quickly after a SEPS stage has been procured. Production scaling is based on a run of six modules, and will proceed at a rate which minimizes costs. The Boeing program estimates and the overall SEPM scaling are presented with the specific scale factors used in Table 4-5. The scaling of the stage to the module, consisting of predominantly hardware costs, is presented in a similar manner in Table 4-6. These estimates are in 1975 dollars and do not include either contractor fees or NASA program costs. Adjustments for these factors are shown in Table 4-7, where inflation from 1975.0 to 1977.5 is estimated at 16%, the fees are 7.5%, and NASA program costs associated with development and use of the stage or module are estimated at 15%. This estimate of NASA programmatic costs does not include any payload specific costs and reflects only support in the use of the stage or module as a propulsion system.

The recurring cost data, however, were provided in terms of estimates of the first unit cost of the propulsion hardware and in terms of the twentieth unit cost under the assumption of an 80% learning curve. The estimate for the recurring unit cost for a run of six as derived from the Boeing data falls between the two LeRC estimates. The Boeing data are used as being representative of the relatively short production runs which can be forecast at the present time and with present technology and costs. A detailed comparison between the two estimates is given in Table 4-8.

TABLE 4-5. SOLAR ELECTRIC PROPULSION STAGE AND MODULE HARDWARE COST ESTIMATES ($, Millions, 1975)

ELEMENT	PLANETARY STAGE DDT&E [a]	PLANETARY STAGE AVG. REC. [a]	DDT&E SCALING [b]	SEPM DDT&E	RECURRING SCALING [b]	SEPM RECURRING
Structure & Mech	2.9	0.42	0.26	0.75	0.75	0.315
Electrical Propulsion	3.64	3.01	0.30	0.95	0.30	0.948
Communications	4.65	0.78	0	--	--	--
Computer & Data Handling [c]	2.17	0.36	0.50	1.0	--	--
Guidance, Nav., Control	8.90	0.78	0.50	4.45	0.50	0.390
Reaction Control	0.49	0.15	1.00	0.49	1.00	0.150
Solar Array	5.0	5.05	0.26	1.30	0.26	1.310
Power Control & Dist.	1.4	0.24	1.00	1.40	1.00	0.240
Thermal Control	1.0	0.21	1.00	1.00	1.00	0.210
Adapters	0.64	0.27	1.00	0.64	0.50	0.140
Assembly & Checkout	0.41	1.72	1.00	0.41	0.30	0.570
	31.0	12.99		12.30		4.273
		ADJUSTMENTS:				
		Fees (7.5%)		13.320		4.593
		Inflation (16%)		15.450		5.328

(a) Source: Reference (4-5).
(b) Hardware scaling factors determined from number of thrusters and power requirements.
(c) MMS computer is assumed.

TABLE 4-6. SCALING OF THE SOLAR ELECTRIC PROPULSION STAGE TO THE MODULE PROGRAM COST ESTIMATES ($, Millions, 1975)

Element	Planetary DDT&E [a]	Planetary Recurring [a]	DDT&E Scaling [b]	SEPM DDT&E	Recurring Scaling [b]	SEPM Recurring
Project Management (6%)	$4.7	1.13	1.00	1.54	1.0	0.402
System Engineering & Integ.	8.5	0.45	0.26	2.21	0.5	0.225
Stage/Module	31.0	12.99	calc.	12.39	calc.	4.273
GSE	24.8	0.80	0.26	6.45	0.5	0.400
System Test Hardware	8.3	--	0.26	2.16	--	--
System Test Ops	0.8	0.02	0.26	0.21	1.0	0.020
Logistics	--	1.42	--	--	0.3	0.473
Software	4.7	0.28	Boeing	1.10	1.0	0.28
Facilities	--	--	--	--	--	--
Launch Ops.	--	2.03	--	--	0.3	0.676
Flight Ops.	--	0.90	--	--	0.5	0.450
Refurbishment	--	--	--	--	--	--
	$82.8	20.00		27.23		7.199
Adjustments						
1975.0 to 1977.5 (16%)	96.0	23.2		31.6		8.3
Fee @ 7.5%	103.3	24.9		34.0		9.0
NASA Program (15%)	118.7	28.7		39.0		10.3

(a) Source: Reference (4-5).
(b) Scaling for SEP hardware items is determined from number of thrusters and power requirements.

TABLE 4-7. SOLAR ELECTRIC POWER MODULE PROGRAM COST COMPARISONS

	Batteile Estimates[a]	
	DDT&E SEPM	Recurring Unit Run of Six
Contractor Costs, $M 1975.0	27.2	7.2
Fee at 7.5%	29.2	7.6
Inflation, 1975.0 to 1977.5 (16%)	34.0	9.0
NASA Program (15%)	39.0	10.3
	LeRC Recurring Unit Estimates[b], $M 1977	
	Single Unit	Twentieth Unit
Three 30-cm Thrusters (∿6.5 kw)	7.92	3.03
Four 8-cm Auxiliary Thruster Units (400-600 w total power)	2.44	0.928

(a) Derived from Reference (4-5), and based on three 30-cm ion thrusters and 6.5 kw (initial) solar power array.

(b) Source: Reference (4-3). Note: LeRC DDT&E estimates were not made.

TABLE 4-8. DETAILED SEPM RECURRING COST ESTIMATE COMPARISON[a]

Element	Recurring Unit Estimates: Battelle[b], Six Units, \$M 1975		LeRC[c], \$M 1977: 1st-3rd Unit	LeRC[c], \$M 1977: 19th-21st Unit
Project Management (6%)	0.402			
System Engineering and Integ	0.225			
Module	--			
Structure and Mech	0.315			
Control	0.390			
Reaction Control (RCS)	0.150			
Solar Array	1.310		3.3	1.26
Power Control and Dist	0.240		1.23	0.48
Thermal Control	0.210	and other items	0.42	0.15
Adapters	0.140			
Assembly and Checkout	0.570			
Electrical Propulsion	0.948			
Propellant Supply, Dist			2.4	0.915
Thrusters			0.561	0.213
GSE	0.400			
System Test Ops	0.020			
Logistics	0.473			
Software	0.280			
Launch Ops	0.676			
Flight Ops	0.450			
	7.199		7.92	3.030

(a) Based on three 30-cm ion thrusters and 6.5 kw (initial) solar power array.
(b) Scaled from Reference (4-5).
(c) Source: Reference (4-3).

4.4 Secondary Propulsion Cost Estimates

4.4.1 Solar Electric Secondary Propulsion

In addition to potential use as primary propulsion, SEP has a potential for secondary propulsion in applications for drag makeup in low Earth orbits, for stationkeeping at geosynchronous altitude, and for some attitude control applications.(4-7) Accordingly, this study considers a propulsion module which uses two or four 8-cm ion thrusters for these applications. It is assumed that the millipound ion thrusters in conjunction with the momentum wheel attitude control provided by the MMS bus will provide sufficient attitude stabilization, and no hydrazine propulsion system will be required. The power requirements of 400 to 600 watts can reasonably be met from the MMS arrays, but the auxiliary propulsion is then in competition with the payload for electrical power. The estimate of Table 4-8 accordingly reflects alternative assumptions about the provision of additional solar power on an incremental or marginal basis.

While estimates of the recurring costs for the 8-cm thrusters and associated hardware are available, no data on the development and operational test costs of this electric propulsion application could be found. Since it is unlikely that this technology would be used on the MMS unless the hardware production capability and experience information were available from other programs, the development costs were not pursued further, and, accordingly, only the recurring estimates are provided. Under the circumstances of prior development and power available from existing solar panel designs, it is also likely that the program costs would not change significantly. The recurring cost estimate of Table 4-9 is based on the hydrazine module cost data(4-1) of Table 4-2 and the estimates from the LeRC data package(4-3).

The potential impact of providing additional solar electric power to yield the additional 600 watts so that the power for the payload can remain at the nominal level is estimated from the Boeing report.(4-5) As part of their SEPS costing effort, solar arrays were investigated in detail; the results are summarized on pages 107 and 108 of Reference (4-5). The major cost of the 25-kw array came from the cost of purchasing 276,000 solar cells at $7.15 each and 276,000 cover glasses at $4. each. Thus, for two 12.5-kw arrays, the solar cells cost $1.937M and the cover glasses cost $1.104M in 1975 dollars. The recurring unit cost of the two array wings was estimated at \$5.05M, or \$0.51M less than the first unit cost based on a production run of six units. Boeing presents these costs as being lower than those obtained from Lockheed at that time.

TABLE 4-9. RECURRING COST ESTIMATE FOR SOLAR ELECTRIC AUXILIARY PROPULSION (\$, Millions, 1977.5)

Element	Two 8-Cm Thrusters 1st Unit	Two 8-Cm Thrusters 20th Unit	Four 8-Cm Thrusters 1st Unit	Four 8-Cm Thrusters 20th Unit
Hardware[a]	0.181	0.181	0.181	0.181
Contractor Program[a]	0.248	0.248	0.248	0.248
Power Processor[b]	0.750	0.286	1.500	0.572
Thrusters[b]	0.210	0.080	0.420	0.160
Controllers[b]	0.200	0.076	0.400	0.152
Propellants[b]	0.060	0.022	0.120	0.044
	\$1.649M	\$0.893M	\$2.869M	\$1.357M

(a) Structure, thermal control, interface, integration and assembly estimated from Reference (4-1), as stated in Table 4-2.
(b) Source: Reference (4-3).

From the large number of cells and glass covers required, it is assumed that the production efficiencies accrue to assembly costs rather than to the cells and glasses. Accordingly, the array costs are scaled linearly with the total cost of the array rather than assuming a learning curve. At some time in the future, it is very likely that solar cell unit costs will decline significantly, in the manner of solid-state electronic components. This report does not attempt to forecast this future time, and thus electric propulsion is costed on the basis of current knowledge.

The recurring unit cost of \$5.05M for 25 kw results in an estimate of \$202 (\$1975) per watt[4-5] in an incremental cost for solar power for the secondary propulsion where additional solar power is required. This is adjusted for inflation to 1977.5 by 16%, since a major cost will still be manpower, at \$234 per watt. Thus the incremental cost of providing an additional 600 watts to existing solar array design is estimated at \$140,600 on a recurring basis.

4.4.2 Electrothermal Secondary Propulsion

A 1974 TRW report[4-8] provides considerable technical data on thrusters which use electrothermal decomposition of hydrazine to provide a significantly higher specific impulse than can be achieved by catalytic decomposition. This technology is viewed as being potentially advantageous for long missions in that it can reduce the final requirements for secondary propellants.

The power requirements in this application are of the same order of magnitude (600 watts) as for secondary ion propulsion. A potential advantage of electrothermal over ion propulsion is that this power requirement may not need to be continuous or near-continuous as in the case of ion propulsion. The potential disadvantage of this technology is that it will compete with the payload for electrical power, as does ion propulsion.

Accordingly, the cost impact of electrothermal hydrazine secondary propulsion is not expected to come from the thrusters or propulsion equipment since the propulsion components are expected to cost about the same, on a recurring cost basis, as catalytic propulsion components. The major cost impact of this technology is expected to come from the cost of supplying the electrical power in competition with the payload. Hence, no specific cost is attached to this choice and the cost impact is judged on the cost of providing power through solar arrays.

The cost of providing an incremental 600 watts is estimated from Section 4.4.1 at $234 per watt[4-5], or $140,600 on a recurring basis.

4.5 Shuttle and Shuttle Upper Stage Charges

The controllable transportation costs for the MMS program include Shuttle and Shuttle-related charges as specified by NASA. These are predominantly recurring operations costs associated with each flight and do not include amortization and overhead costs, which are charged to commercial users of the Space Transportation System (STS). The charges for the Shuttle are taken from the STS Users Handbook[4-9] and NMI 8610[4-10], and reflect announced NASA policy for the transportation charges. Charges for STS services such as extended mission time are taken from the Users Handbook and other sources. These other charges represent estimates, and are considered more likely to change than the Shuttle transportation charge. Charges for the

Spinning Solid Upper Stages (SSUS) and the Interim Upper Stage (IUS) are based on estimates which reflect the launch costs as well as hardware and hardware amortization charges. At the present time it appears that the SSUS will be provided as packages by two different contractors, one for the SSUS-D (Delta equivalent) and another for the SSUS-A (Atlas/Centaur equivalent). The IUS will be provided by the U.S. Air Force under an interagency agreement. Current agreements provide for reimbursements of launch costs as well as hardware.

The Shuttle transportation charge algorithm is shown in Figure 4-2, and is based on the larger of the payload's weight or length in the Shuttle Orbiter bay. The charge to NASA in 1975 dollars is $16 million for a launch to the standard Shuttle orbit of 160 nmi. This is adjusted for inflation to 1977.5 to be $18.5 million. The curve of Figure 4-2 is then used to determine the fraction of this charge attributable to the MMS payload. Additional charges for STS services used in later analyses are[4-11]:

$300,000 for a service mission
$100,000 for a return mission.

The charges for the SSUS-A and SSUS-D are not formal NASA or contractor estimates but are taken from our previous effort on a different task under contract to NASA[4-12]. The charges, as adjusted to reflect inflation to 1977.5, are $1.12M for the SSUS-D and $1.46M for the SSUS-A.

The cost for the IUS, provided informally by SAMSO as a planning estimate, was $4.6M in 1978 dollars for the hardware and $1.0M for operations for a two-stage vehicle. This is reduced to a total of $5.2M for 1977 under the assumption of 5% inflation. This is considered equivalent to an estimate of $4.8M made recently[4-13] in dollars of unstated year.

SHARED PRICE = C_f x DEDICATED PRICE

$$\text{LOAD FACTOR} = \left\{ \begin{array}{c} \dfrac{\text{PAYLOAD WEIGHT, LB}}{\text{SHUTTLE CAPABILITY}} \\[2ex] \dfrac{\text{PAYLOAD LENGTH, FT}}{60} \end{array} \right\} \text{WHICHEVER IS GREATER}$$

SHUTTLE CAPABILITY	
INCLINATION IN DEGREES	WEIGHT IN THOUSANDS OF POUNDS
28.5	65
56	57
90	37
104	30

NASA DEDICATED PRICE
$18.5M , 1977.5)
Source: NMI 8610

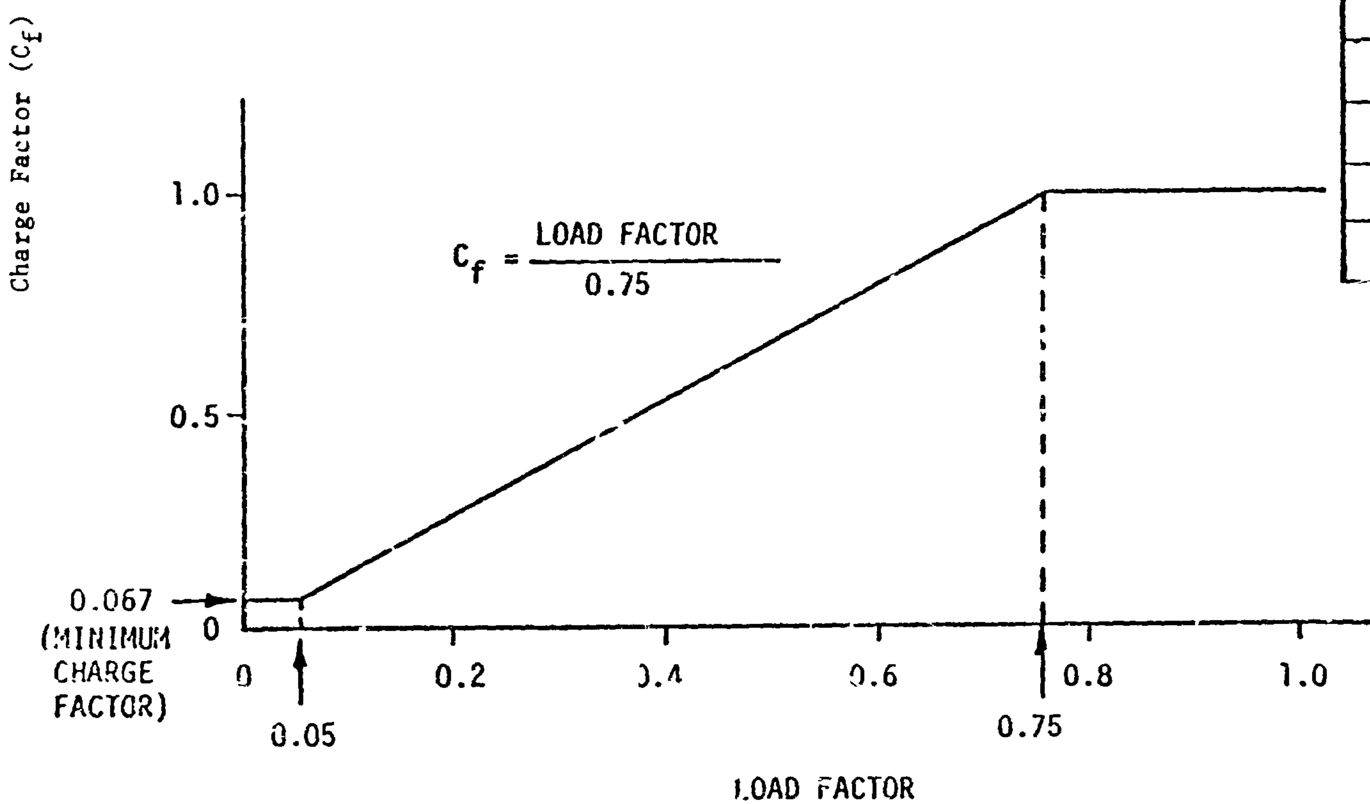

FIGURE 4-2. DETERMINATION OF SHUTTLE PRICE FOR 160 NMI

4.6 Summary of Cost Estimates

The cost estimates used in the analysis are summarized in Table 4-10.

TABLE 4-10. SUMMARY OF COST ESTIMATES USED IN ANALYSIS

System	Costs, millions of 1977.5 dollars Non-Recurring	Recurring
Monopropellant Hydrazine Module		
One tank	2.215	0.965
Two tanks	2.314	1.075
Three tanks	2.413	1.186
Four tanks	2.512	1.296
Six tanks	2.710	1.518
SPS-I	2.160	0.886
Bipropellant (N_2H_4/N_2O_4) Module		
Two tanks	4.86[a]	1.200
Four tanks	4.86[a]	1.335
Large twin tank	6.5	2.000
Solar Electric Propulsion Module		
Hardware only	15.45	5.300
Total program costs	39.0	10.300
Solar Electric Auxiliary Propulsion Module		
Two 8-cm thrusters	--	1.649
Four 8-cm thrusters	--	2.869
Electrothermal Secondary Propulsion on Primary Chemical Propulsion Module (increment for additional power of 600 watts)	--	0.140
Shuttle Charges		
Dedicated Shuttle Flight	--	18.500
Additional Charges		
Service mission	--	0.300
Return mission	--	0.100
SSUS-D	--	1.120
SSUS-A	--	1.460
IUS (2-stage including operations)	--	5.300

(a) May be reduced if pursued as a joint development. Our estimate is a total of $6.5M.

4.7 References

(4-1) Etheridge, F., Cooper, W., and Mansfield, Jr., Landsat/MMS Propulsion Module Design: Appendix - Cost Analyses, Rockwell International Space Division, Report No. SD 76-SA-0095-3, September 24, 1976.

(4-2) Cork, T. R., Deflation Factors for Economic Analyses, BMI-NLVP-ICM-75-20, Battelle's Columbus Laboratories, April 20, 1975.

(4-3) LeRC Memorandum, Subject: Data Package for MMS Study, From: 6132/Head, Large Thruster Section (D. C. Byers), March 22, 1977.

(4-4) Nichols, J. F., Study of Liquid Oxygen/Liquid Hydrogen Auxiliary Propulsion Systems for the Space Tug, Final Report, Technology Programs, Space Systems and Applications, Space Division, Rockwell International Corporation, June 15, 1975.

(4-5) Concept Definition and System Analysis Study for a Solar Electric Propulsion Stage, Volume V, Cost Data; Boeing Aerospace Company, January 1975.

(4-6) Concept Definition and System Analysis Study for a Solar Electric Propulsion Stage, Rockwell International Space Division, February 3, 1975.

(4-7) NAS 3-20113 - Task 1: Spacecraft Design, November 4, 1976, Viewgraph Presentation by TRW Defense and Space Systems Group.

(4-8) Kuenzly, J. D., Study of Monopropellants for Electrothermal Thrusters, TRW Systems Group, June 1974.

(4-9) Space Transportation System Users Handbook, NASA Headquarters, June 1977.

(4-10) NASA Management Instruction 8610, Subject: Reimbursement for Shuttle Services Provided to Non-U.S. Government Users, January 1977.

(4-11) (Preliminary) STS Reimbursement Guide, NASA/Johnson Space Center, June 1977.

(4-12) Earhart, R. W., Recurring Cost Estimates for Spin-Stabilized Shuttle Upper Stages, BMI-NLVP-ICM-76-28, July 30, 1976.

(4-13) Parker, R.N., DDDR&E, quoted in Aerospace Daily, April 25, 1977, page 317.

5.0 PROPULSION SYSTEM TRADE-OFFS

Initial sizing of hydrazine and bipropellant propulsion systems for MMS is described in Section 3.3. These estimates were based on the use of existing and/or proposed hardware such as the Rockwell SPS-I and SPS-II hydrazine systems, modified SPS-II modules with clustered Viking Orbiter 197_ tanks, and the TRW Multimission Bipropellant Propulsion System.(5-1)* This approach was considered desirable for achieving a reasonable commonality of system components to reduce overall program costs.

A review of the costing analysis shows that, in view of the announced NASA policy for determination of STS transportation charges, all of the missions included would be charged based on the load factor associated with payload length. This indicates that a cost reduction might be derived from the development of unique tanks for the MMS propulsion systems which would result in reduced overall length. In light of the transportation charges involved, this concept was deemed worthy of further study, as had been mentioned in the Rockwell Landsat/MMS Propulsion Module Design Study.(5-2)

This section discusses a preliminary analysis of reconfiguring the MMS propulsion systems to reduce overall length. Included are estimates of the effects of these design changes on system development costs and on transportation charges. Both hydrazine and bipropellant modules have been included to adequately determine the most appropriate design for the MMS propulsion system(s). This discussion is followed by a brief assessment of the cost effectiveness of the new propulsion modules, and includes a comparison of the results based on both discounted and undiscounted costs.

The section is concluded with a cost trade-off/analysis for each of the additional mission concepts discussed in Subsection 3.2.2 through 3.2.5.

5.1 Baseline Costs for Hydrazine-Bipropellant Systems

Program costs for deploy-only, ground refurbishment, and on-orbit servicing mission models have been compiled for both the initial hydrazine and initial bipropellant designs. Included in the program costs are

*References, denoted by superscript numbers, are at the end of section (Subsection 5.7).

engineering development (nonrecurring cost) of the propulsion modules, Shuttle transportation charges, recurring cost of the propulsion module, and recurring cost of the MMS bus. Table 5-1 summarizes the costs assumed and indicates the distribution of these costs with respect to the launch year (LY) of a mission. The information for each mission model was processed using the Battelle-developed NASA Interactive Planning System (NIPS). Table 5-2 is a sample output from the NIPS program accompanied by descriptive remarks to clarify the displayed information.

TABLE 5-1. COSTS AND COST DISTRIBUTION FOR COMPONENTS OF PROGRAM TOTALS

Item or Service	Cost	Distribution
STS transportation charges	Load factor[a] X \$18.0M + additional services (if any)	LY-2 (20%) LY-1 (34%) LY (46%)
MMS bus	\$4.2M	LY-2 (50%) LY-1 (50%)
Propulsion modules	(b)	LY-1 (100%)
Engineering development	(c)	YFU-3[d] (50%) YFU-2 (50%)

(a) Larger of two load factors associated with mass and length.
(b) Recurring cost for individual propulsion modules is shown in Table 4-10.
(c) Engineering development (nonrecurring) costs are shown in Table 4-10.
(d) YFU = year of first use.

In generating total program costs, the data have been summed on a yearly basis for each funding type (i.e., ED, PM, etc.) to facilitate identification of funding spikes which result from the various mission models.

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TABLE 5-2. SAMPLE COST DATA OUTPUT

RH
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

			79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
COST FOR ENTIRE MODEL	0 RH	ED	3.4	5.8	2 4					.					.			11 6
		PM	.		3.1	3 1	4.1	3 0	3 3	6 5	4 2	4 1	4.5	5 2				41 1
		L/V	.	2.2	7 9	20 1	32 6	40 5	48 3	50 4	65 8	63.8	68 9	43 4	33.8	16.4	14.3	488 4
		BUS	.	6.3	12 6	16 8	16 8	12 6	21.0	23 1	16 8	18 9	21 0	10.5	.	.	.	176 4
																		717.5
Costs associated with specific categories of mission model	1 HERH	ED		.		.		.	.			.	.				.	
		PM	.		9	1 1	9	.	.		1 1					.	.	3 9
		L/V	.	.5	3.0	5 3	6 6	2 7	2 0	2 1	3 5	5.7	1 4	2 0	.			34 8
		BUS	.	2.1	4 2	4 2	2 1		.	2 1	2 1		.					16 8
	1 SOLAR MAX MISSION	ED	.	.				.										
		PM	.	.		1 0		1 0		1 0		1.0	.	1 0		.		4 8
		L/V	.	.	1 3	2.1	5 0	3 6	7 0	3 6	7 0	3.6	7 0	3 6	5 7	1 4	2 0	52 3
		BUS	.	.	2 1	2 1	2 1	2.1	2.1	2 1	2 1	2.1	2 1	2 1		.	.	21 0
	1 APRH	ED	.	.				.				.				.	.	
		PM	.	.	2 3	1 1	1.0		1.2	1 1	1 0	1 1	1 2	1.1	.	.		10 9
		L/V	.	1 7	3 6	7 5	9 5	12 3	10 7	4 8	8 0	11.8	13.4	4 8	3 9	.		92 7
		BUS	.	4.2	6.3	4 2	2 1	2.1	4 2	4 2	4 2	4 8	4 2	2 1	.	.	.	42 0
	1 EORH	ED	.	.	.	.						.			.	.	.	
		PM	.	.	.		2 2	1.0	2 1	4 5	2 1	2.0	3.3	2 0			.	19 4
		L/V	.	.	.	5.1	10.5	20 1	22 7	33 5	38 0	32 7	38 1	31 4	18.2	8.8	3.9	261 4
		BUS	.	.	.	6 3	8.4	6.3	14.7	14 7	8 4	12.6	12.6	4 8	.		.	88 2
	1 ALL WEATR & SEA M	ED	.	.	.							.					.	
		PM	.	.			.	1 1				.		1 1	.		.	2 1
		L/V	.		.	.	1 0	1 7	6 0	6 2	8 4	.	1.0	1 7	6 0	6 2	8.4	46.7
		BUS	.	.			2.1	2 1				.	8 1	2 1	.	.	.	8 4
	1 RHPROP	ED	3.4	5.8	2.4		.	.	.	.	.		.	.		.		11.6
		PM	.	.	.	.	.	.	.	.		.	.	.		.	.	.
		L/V	.	.	.	.	.	.	.	.		.	.	. .				.
		BUS	.	.	.	.	.	.	.	.	.						.	.

Abbreviations: DB - Deploy model/bipropellants
DH - Deploy model/hydrazine
RB - Return model/bipropellants
RH - Return model/hydrazine
SB - Service model/bipropellants
SH - Service model/biprdpellants
LV - STS transportation charges

AP - Astrophysics mission category
HE - High energy explorer category
EO - Earth observation category
PROP - Propulsion modules
ED - Engineering development cost
PM - Recurring costs of propulsion module
BUS - Recurring costs of MMS bus

Example: EODH stands for Earth observation satellites using hydrazine for deployment only

5.1.1 Deploy-Only Mission Model

Program costs for the deploy-only mission model using both hydrazine and bipropellant propulsion systems are shown in Tables 5-3 and 5-4. Missions included in this analysis were presented earlier in Table 1-1. Four hydrazine systems are required to meet the propulsion requirements of this model. They include the SPS-I, SPS-II, and two- and three-tank SPS-II derivative systems. To perform all of the missions using bipropellants would require a two-tank version of the TRW Multimission Bipropellant Propulsion System (MBPS), the standard four-tank MBPS, and a new bipropellant system with a propellant capacity of 1500 kg.

Comparison of the propulsion module engineering development costs for these technologies indicates that hydrazine is less expensive at \$9.1M than the bipropellants, which have a development cost of \$13.0M. This advantage of hydrazine is intensified when propulsion module costs (recurring costs) are taken into consideration. Hydrazine systems would cost approximately \$39.5M, while bipropellants would require an expenditure of \$56.4M. MMS bus costs are constant for the two propulsion technologies since they are dependant only upon the mission model (i.e., number of flights) under consideration.

The use of bipropellants results in lower transportation charges due to the reduced overall length. STS charges for all of the missions are \$286.7M when bipropellants are used and \$300.2M for hydrazine propulsion. Summation of all four funding types results in a total program cost of \$525.2M for hydrazine and \$532.5M for bipropellant systems. For the deploy-only model, hydrazine appears to be the most cost-effective propulsion alternative. Although the cost margin between the systems is not dramatic, the reduced program cost coupled with the reduced system complexity and more favorable safety characteristics (i.e., bipropellants such as N_2O_4 and N_2H_4 are hypergolic) favor the selection of hydrazine for MMS propulsion applications.

It should be noted at this point that careful comparison of the costs associated with specific categories of the mission model may show instances in which bipropellants appear to be lower in cost than hydrazine. A case in point is the Solar Maximum Mission, which has a total cost of \$56.7M for bipropellants and \$57.3M for hydrazine. When viewed on a per flight basis, this difference amounts to approximately \$120,000. A cost

TABLE 5-3. PROGRAM COSTS FOR DEPLOY MISSION MODEL USING HYDRAZINE MODULES

DH

******* FOR INTERNAL PLANNING USE ONLY *******

RATE DOLLARS IN MILLIONS

		79	80	81	82	83	8<	85	86	87	88	89	90	91	92	93	TOTAL
0 DH	ED	3 4	4 6	1 1													9 1
	PM			3 1	3 1	4 0	2 9	3.1	6 1	4.0	4 0	4 3	4 9				39 5
	L/V		2 2	7 9	19 3	25 7	27 7	27.7	35 7	42 7	33 8	32 9	29 2	15 4			300 2
	BUS		6 3	12 6	16 8	16 8	12 6	21 0	23 1	16 8	18.9	21 0	10 5				176 4
																	525.2
1 HEDH	ED																
	PM			9	1 1	9				1 1							3 9
	L/V		5	3 0	5 3	5 7	1 2		2 1	3 5	4 8						26 3
	BUS		2 1	4 2	4 2	2 1			2 1	2 1							16 8
1 SOLAR MAX MISS	ED																
	PM				1 0		1 0		1 0		1.0		1 0				4 8
	L/V			1 3	2 1	4 2	2 1	4 2	2 1	4 2	2 1	4 2	2 1	2 9			31 5
	BUS			2.1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1				21 0
1 APDH	ED																
	PM			2 3	1 1	1 0		1 2	1 1	1 0	1 1	1 2	1 1				10 9
	L/V		1.7	3 6	6 7	4 4	4 2	2 2	4.8	5 2	5 6	4 1	3 3	1 9			47 6
	BUS		4.2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4 2	4 2	2 1				42 0
1 EODH	ED																
	PM					2 1	1 0	1 9	4 1	1 9	1 9	3 1	1 9				18 0
	L/V				5 1	10 5	18 4	19 0	26 7	29 8	21 3	23 6	22 0	8 3			184 8
	BUS				6 3	8 4	6 3	14 7	14 7	8 4	12 6	12 6	4 2				88 2
1 ALL WEATR & SEA M	ED																
	PM						1 0						1 0				1 9
	L/V					1 0	1 7	2 3				1 0	1 7	2 3			10 1
	BUS					2 1	2 1					2 1	2 1				8 4
1 DHPROP	ED	3 4	4 6	1 1													9 1
	PM																
	L/V																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

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TABLE 5-4. PROGRAM COSTS FOR DEPLOY MISSION MODEL
USING BIPROPELLANT SYSTEMS

DB
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 DE	ED	6 5	6 5														13 0
	PM			4 5	3 9	6 8	3 7	4 4	9 3	4 9	4 9	7 6	6 3				56 4
	L/V		2 3	7 8	19 0	24 8	26 9	26 2	33 9	40 3	32 1	31 1	27 8	14 3			286 7
	BUS		6 3	12 6	16 8	16 8	12 6	21 0	23 1	16 8	18 9	21 0	10 5				176 4
																	532.5
1 HEDS	ED																
	PM			1 2	1 3	1 2				1 3							5 1
	L/V		7	3 2	5 8	5 9	1 7		2 0	3 4	4 6						27 4
	BUS		2 1	4 2	4 2	2 1			2 1	2 1							16 8
1 SOLAR MAX MISSION	ED																
	PM				1 2		1 2		1 2		1 2		1 2				6 0
	L/V			1 2	2 0	3 9	2 0	3 9	2 0	3 9	2 0	3 9	2 0	2 7			29 7
	BUS			2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1				21 0
1 APDE	ED																
	PM			3 3	1 3	1 2		2 0	1 3	1 2	1 3	2 0	1 3				15 1
	L/V		1 6	3 4	6 2	4 0	4 0	2 2	4 6	4 7	5 2	3 8	3 2	1 7			44 7
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4 2	4 2	2 1				42 0
1 EODE	ED																
	PM					4 4	1 2	2 4	6 8	2 4	2 4	5 6	2 4				27 6
	L/V				4 9	10 1	17 6	18 0	25 3	28.3	20 2	22 4	21 0	7 8			175 6
	BUS				6 3	8 4	6.3	14 7	14 7	8 4	12 6	12 6	4 2				88 2
1 ALL WEATR & SEA M	ED																
	PM						1 3						1 3				2 7
	L/V					9	1 6	2 2				9	1 6	2 2			9 4
	BUS					2 1	2 1					2 1	2 1				8 4
1 BIPROP	ED	6 5	6 5														13 0
	PM																
	L/V																
	BUS																

NOTE: For explanation of abbreviations
refer to Table 5-2.

differential of this magnitude could be the result of a 0.1-m uncertainty in overall length. As a result of the rounding off of component lengths and load factors, it is not clear whether either system has a cost benefit over the other. A more accurate assessment might be that, if the cost difference on a per flight basis is small (low enough that roundoff is a probable explanation), then cost should not be used as the sole selection criterion between hydrazine and bipropellants.

5.1.2 Ground Refurbishment Mission Model

Cost information for the ground refurbishment mission model is shown in Tables 5-5 and 5-6. These data cover the missions described earlier in Table 1-2. Hydrazine systems to perform these missions include the SPS-I; SPS-II; and two-, three-, and four-tank modified SPS-II modules. The bipropellant stages, as described in Subsection 5.1.1, will satisfy all propulsion requirements connected with this model.

The trends in program cost discussed in the previous subsection on the deploy-only mission model are also evident for the ground refurbishment case. Hydrazine shows a slight edge over bipropellants in terms of engineering development costs and propulsion system recurring cost. Bipropellants have lower STS transportation charges, but total costs for all four funding types would indicate a cost advantage in favor of hydrazine. The overall program cost of $717.5M for hydrazine and $724.5M for bipropellants should realistically be viewed as roughly equivalent, in light of the previous discussion of roundoff error in component lengths and load factors.

Since neither propulsion technology shows a definite cost advantage, a decision based on factors such as system complexity and safety considerations would likely result in selection of hydrazine to fulfill the propulsion needs of the ground refurbishment mission model.

5.1.3 On-Orbit Servicing Mission Model

Costs associated with the on-orbit servicing mission model are displayed in Tables 5-7 and 5-8. Total program cost for the hydrazine systems is $512.4M. The use of bipropellant propulsion systems to perform the same missions results in a total cost of $530.3M. For this mission model, the cost differential between these two propulsion technologies

TABLE 5-5. COSTS FOR GROUND REFURBISHMENT MODEL EMPLOYING HYDRAZINE PROPULSION

RH
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 RH	ED	3.4	5.8	2 4													11 6
	PM			3.1	3 1	4.1	3.0	3.3	6 5	4 2	4 1	4.5	5 2				41 1
	L/U		2.2	7 9	20 1	32 6	40 5	48 3	50 4	65 8	53.8	58 9	43.4	33 8	16 4	14 3	488 4
	BUS		6 3	12 6	16 8	[illegible]	12.6	21.0	23 1	16 8	18.9	21 0	10 5				176 4
																	717.5
1 HERH	ED																
	PM			.9	1 1	9				1 1							3 9
	L/U		.5	3 0	5 3	6 6	2 7	2 0	2 1	3 5	5.7	1 4	2 0				34 8
	BUS		2.1	4 2	4 2	2 1			2 1	2 1							16 8
1 SOLAR MAX MISSION	ED																
	PM				1 0		1 0		1 0		1.0		1 0				4 8
	L/U			1 3	2.1	5.0	3 6	7 0	3 6	7 0	3.6	7 0	3 6	5 7	1 4	2 0	52 8
	BUS			2 1	2 1	2 1	2.1	2.1	2 1	2 1	2 1	2 1	2 1				21 0
1 APRH	ED																
	PM			2 3	1 1	1.0		1.2	1 1	1 0	1 1	1 2	1 1				10 9
	L/U		1 7	3 6	7 5	9 5	12 3	10 7	4 8	8 9	11.8	13.4	4 8	3 9			92 7
	BUS		4 2	6.3	4 2	2 1	2.1	4 2	4 2	4 2	4 2	4.2	2 1				42 0
1 EORH	ED																
	PM					2 2	1.0	2 1	4 5	2 1	2.0	3.3	2 0				19 4
	L/U				5.1	10.5	20 1	22 7	33 8	38 0	32.7	38 1	31 4	18 2	8.8	3.9	261 4
	BUS				6 3	8.4	6 3	14.7	14 7	8 4	12.6	12.6	4 2				88 2
1 ALL WEATR & SEA M	ED																
	PM						1 1						1 1				2 1
	L/U					1 0	1 7	6 0	6 2	8 4		1.0	1 7	6 0	6 8	8.4	46 7
	BUS					2 1	2 1					2 1	2 1				8 4
1 RHPROP	ED	3 4	5 6	2.4													11.6
	PM																
	L/U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

TABLE 5-6. PROGRAM COSTS FOR PERFORMING GROUND REFURBISHMENT MODEL WITH BIPROPELLANTS

PB
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 RE	ED	6 5	6 5														13 0
	PM			4 5	3 9	7 1	3 9	4 7	10 5	5 2	5 2	8 0	6 5				59 5
	L/U		2 3	7 8	13 9	31 6	39 7	47 0	48 9	63 8	52 0	57 1	42 1	32 7	16 4	14 3	475 6
	BUS		6 3	12 6	16 8	16 8	12 6	21 0	23 1	16 8	15 9	21 0	10 5				176 4
																	724.5
1 HERE	ED																
	PM			1 2	1 3	1 2				1 3							5 1
	L/U		7	3 2	5 8	6 7	3 1	2 0	2 0	3 4	5 5	1 4	2 0				35 9
	BUS		2 1	4 2	4 2	2 1			2 1	2 1							16 8
1 SOLAR MAX MISSION	ED																
	PM				1 2		1 2		1 2		1 2		1 2				6 0
	L/U			1 2	2 0	4 8	3 5	6 7	3 5	6 7	3 5	6 7	3 5	5 5	1 4	2 0	51 0
	BUS			2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1	2 1				21 0
1 APRE	ED																
	PM			3 3	1 3	1 2		2 0	1 3	1 2	1 3	2 0	1 3				15 1
	L/U		1 6	3 4	7 1	9 1	12 2	10 6	4 6	8 4	11 5	13 1	4 7	3 6			89 8
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4 2	4 2	2 1				42 0
1 EOPE	ED																
	PM					4 7	1 3	2 7	8 0	2 7	2 7	6 0	2 7				30 7
	L/U				4.9	10 1	19 3	21 9	32 6	36 9	31 6	34 9	30 4	17 8	8 8	3 9	253 0
	BUS				6 3	8 4	6 3	14 7	14 7	8 4	12 6	12 6	4 2				88 2
1 ALL WEATR & SEA M	ED																
	PM						1 3						1 3				2 7
	L/U					9	1 6	5 8	6 2	8.4		9	1 6	5 8	6 2	8.4	46.0
	BUS					2.1	2.1					2 1	2 1				8 4
1 BIPROP	ED	6 5	6 5														13 0
	PM																
	L U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

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TABLE 5-7. ON-ORBIT SERVICING MISSION MODEL COSTS FOR HYDRAZINE PROPULSION

SH
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 SH	ED	3.4	5 8	2 4			1 4	1 4									14 3
	PM			3 1	3 3	4 6	2.4	3 8	3 8	2 3	2.3	5.3	3.2				34 0
	L/V		2.2	7 9	20.1	27 8	32 4	31 7	37 2	37 2	27 7	29 7	30 8	20 4	14 0	10.6	329 7
	BUS		6.3	12 6	16 8	14 7	10 5	14 7	12 6	8 4	14.7	16 8	6.3				134 4
																	512.4
1 HESH	ED																
	PM			9	1 3	9											3 1
	L/V		.5	3 0	5 3	6 6	2 8	2.1									20 3
	BUS		2 1	4.2	4 2	2 1											12 6
1 SOLAR MAX MISSION	ED																
	PM				1 0				1 0				1 0				2 9
	L/V			1 3	2 1	3 8	1 5	3 3	2 1	3 8	1.5	3.3	2 1	3 8	1 5	2 1	32 3
	BUS			2 1	2 1			2 1	2 1			2 1	2 1				12 6
1 APSH	ED																
	PM			2 3	1 1	1 0		1 2	1 1	1 0	1.1	1.2	1.1				10 9
	L/V		1.7	3 6	7.6	5 9	6.2	2 2	4 8	5 2	5 6	5.0	4 8	4 0			56 5
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4.2	4 2	4 2	4 2	2 1				42 0
1 EOSH	ED																
	PM					2 8	1 2	2 6	1 7	1 3	1.2	4.1					14 8
	L/V				5 1	10 5	20 2	18 0	24 0	19 7	20 6	20.4	22 2	6.6	6.8		173 5
	BUS				6.3	8.4	6 3	8 4	6.3	4 2	10.5	8 4					58 8
1 ALL WEATR & SEA M	ED																
	PM						1 2						1 2				2 4
	L/V					1 0	1 7	6 0	6 3	8 5		1.0	1 7	6 0	6 3	8 5	47.1
	BUS					2 1	2 1					2 1	2 1				8 4
1 SHPROP	ED	3 4	5 8	2 4			1 4	1 4									14 3
	PM																
	L/V																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

TABLE 5-8. COSTS FOR BIPROPELLANT SYSTEMS TO PERFORM ON-ORBIT SERVICING MISSION MODEL

SB
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 SB	ED	6 5	6 5														13 0
	PM			4 5	4 7	8 4	3 3	6 0	6 7	3 2	2 7	10 0	4 7				54 1
	L U		2 3	8 0	20 4	27 8	32 8	31 5	37 2	36 6	27 4	29 3	30 9	20 0	14 0	10 6	322 2
	BUS		6 3	12 6	16 8	14 7	10 5	14 7	12 6	8 4	14 7	16 8	6 3				134 4
																	530.3
1 HESE	ED																
	PM			1 2	2 0	1 2											4 4
	L/U		7	3 4	6 0	7 1	3 2	2 1									22 5
	BUS		2 1	4 2	4 2	2 1											12 6
1 SOLAR MAX MISSION	ED																
	PM				1 3				1 3				1 3				4 0
	L/U			1 2	2 0	3 6	1 5	3 2	2 0	3 6	1 5	3 2	2 0	3 6	1 5	2 1	31 2
	BUS			2 1	2 1			2 1	2 1			2 1	2 1				12 6
1 APSB	ED																
	PM			3 3	1 3	1 2		2 0	1 3	1 2	1 3	2 0	1 3				15 1
	L/U		1 6	3 4	7 1	5 5	6 1	2 2	4 6	4 7	5 2	4 7	4 7	3 7			53 6
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4 2	4 2	2 1				42 0
1 EOSE	ED																
	PM					6 0	1 3	4 0	4 0	2 0	1 3	8 0					26 7
	L/U				5 2	10 5	20 3	18 0	24 3	19 7	20 6	20 3	22 4	6 6	6 2		174 1
	BUS				6 3	8 4	6 3	8 4	6 3	4 2	10 5	8 4					58 8
1 ALL WEATR & SEA M	ED																
	PM						2 0						2 0				4 0
	L/U					1 0	1 8	6 1	6 3	8 5		1.0	1 8	6 1	6 3	8.5	47 4
	BUS					2 1	2 1					2.1	2 1				8 4
1 EIPROP	ED	6 5	6 5														13 0
	PM																
	L U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

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cannot be dismissed as the result of length uncertainties, as was done for the previous models discussed. In this case, the cost advantage of hydrazine has been enhanced due to STS transportation charges for the bipropellants, which are nearly identical to those for hydrazine. Bipropellants had previously tended to offset their higher development and recurring costs through reduced launch charges.

Since the STS transportation costs dominate in those cases analyzed, a brief investigation was conducted to determine what caused the bipropellant advantage in this area to essentially disappear. The results indicate that the cause of this effect is the hypothesized bipropellant design containing 1500 kg of propellants. This design was based on a single spherical tank with common bulkhead and an axially mounted engine which produces 391 N of thrust. These design assumptions produce a stage which is longer than might be desired. Since this system is used for 18 of 34 servicing missions, multiple tanks and/or the use of several smaller thrusters located off-axis would have resulted in total program costs much closer to those for hydrazine. Since this configuration does not currently exist, it is likely that it would be designed in a more efficient fashion. It is, therefore, difficult to justify either system solely on the basis of cost. Any decision involving non-cost considerations would probably result in the selection of hydrazine, as discussed in the preceding subsections.

Comparison of the engineering development costs associated with this model reveals that, in this area, hydrazine is more expensive. This situation is a result of the larger number of hydrazine configurations (six, as compared to three for bipropellants) needed to satisfy the full range of propulsion requirements. The SPS-II module, which is based on a single Viking Orbiter 1975 tank, is used for only five of the 32 missions in this scenario. Of the remaining 27 spacecraft, 25 would require two-, three-, four-, or six-tank versions of the SPS-II system. This observation opens the possibility of further decreasing cost by development of a new tank which is not only shorter in length but has a more optimal capacity to reduce the number of versions required.

5.2 Reconfigured Propulsion Modules

Preliminary analysis of new tank designs was undertaken to determine the effects on overall program cost. The following subsections describe the resulting hydrazine and bipropellant tank configurations and also summarize the recurring and nonrecurring costs of the reconfigured propulsion systems.

5.2.1 Design of New Hydrazine Modules

Primary emphasis in the redesign of the hydrazine tank(s) was placed on reducing the length of this component. Previous experience indicated that it might also be advantageous to minimize the number of tank and propulsion module designs that would require development. As a first step in this analysis, the hydrazine requirements shown earlier in Table 3-19 were reviewed. Only one case was found that needed a propellant loading in excess of 1550 kg. This lone instance was the on-orbit servicing mission for EO-13A using a shared STS flight for rendezvous and refurbishment The propellant requirement of 2645 kg for this mission can be reduced to 1117 kg if a dedicated Shuttle flight is used to service the spacecraft. Since only one flight is involved, it is unlikely that the mission planners would fund development of a unique propulsion system. The efforts of this analysis have, therefore, assumed that the much larger system need not be considered.

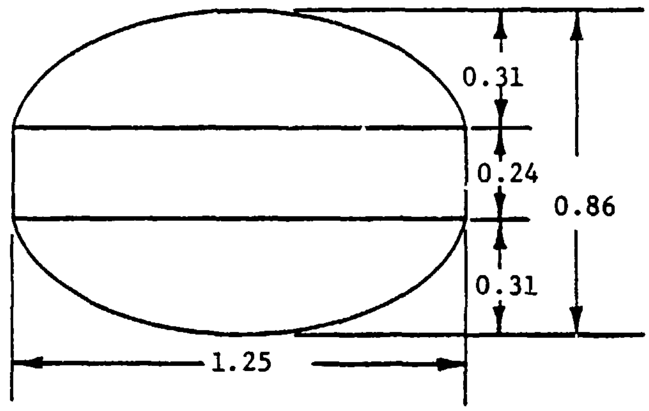
The initial design iteration was based on a maximum propellant capacity of 1590 kg, to allow a reasonable margin in the event that the requirements shown in Table 3-19 would increase for the new system. Calculations also assumed that a mission planner would not routinely operate this system at less than 40 percent of capacity, since this would necessitate paying for a relatively large excess capability. A smaller system with a propellant load of approximately 640 kg would be used for the lower range of mission requirements.

A convenient gap exists in the results of the previous sizing effort between 491 kg and 663 kg of propellant. This opens the possibility of designing a single tank of approximately 530 kg capacity, which can be used alone to satisfy the propulsion needs of the lower energy missions or used in a three-tank cluster to fulfill the propulsion requirements of the more demanding missions such as return or on-orbit servicing. Such a scheme would limit the number of new modules to two and reduce the total propulsion development costs. Two mission categories, namely HE-07A and HE-27A, would call for substantially less propulsion than is available from either of these systems. Since these missions can be performed with the SPS-I module, which is being developed for MMS use in conjunction with expendable launch vehicles, it would appear reasonable to continue the use of this system for these categories.

A propellant capacity of 530 kg translates into a tank volume of about 0.80 m^3. This value was calculated by using an effective hydrazine density of 665 kg/m^3, which corresponds to a pressure blowdown ratio of 3:1. To reduce overall length, the decision was made to use oblate spheroid tanks with a diameter-to-height ratio of 2:1. Cylindrical sections can be added, if necessary, to obtain sufficient volume, with a reduction in height of 17 percent over a sphere of equal di meter and volume. The diameter-to-height ratio was selected for pressure containment purposes and appears to be consistent with current tank and solid motor designs.

As mentioned in Subsection 3.3, there are a number of current uncertainties connected with the MMS cradle and its retention system. Rather than spend an undue amount of time dwelling on this topic, it was decided that the propulsion system tankage should not hinder access from the rear to the three corners of the MMS bus. It was also considered appropriate that the overall diameter of the new three-tank module would not exceed the 2.74-m diameter of the four-tank, modified SPS-II system that it was designed to replace. These restraints led to the selection of 1.25 m as the diameter of the new tank.

Evaluation of all the tankage parameters just discussed led to the configuration shown in Figure 5-1. This design has a length of 0.86 m, which is approximately half the length of the Viking Orbiter 1975 tank. The use of four 22-N thrusters arranged around the perimeter of this tank results in a module length identical to that of the tank. An estimated mass summary for the one- and three-tank systems is shown in Table 5-9. These numbers were derived from Reference (5-2). The integrated modules are shown in Figures 5-2 and 5-3.

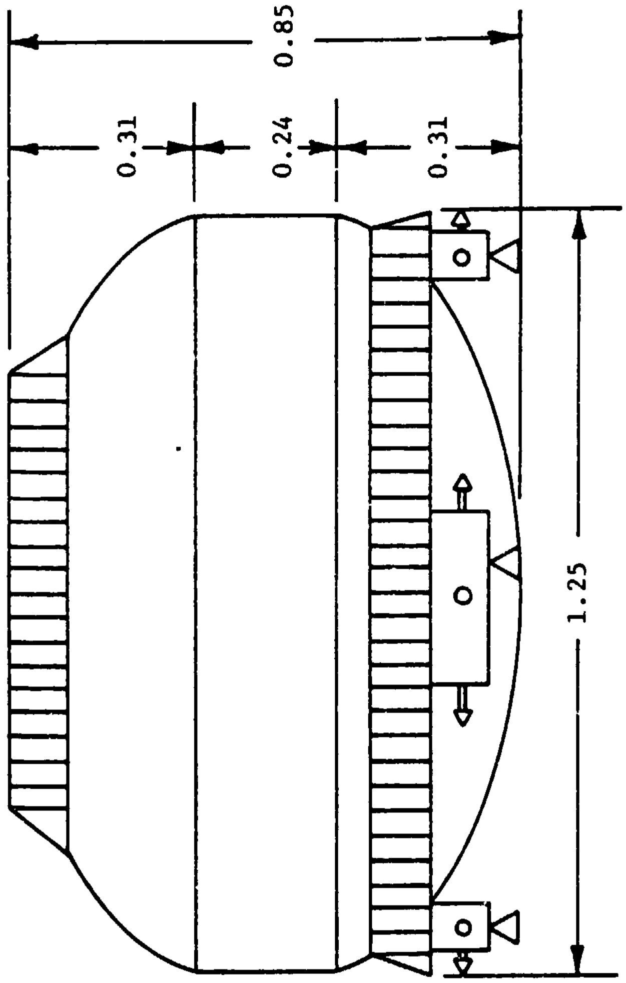


NOTE: All dimensions in meters

FIGURE 5-1. NEW HYDRAZINE TANK

TABLE 5-9. HYDRAZINE PROPULSION MODULE MASS STATEMENT

Component	One-Tank System	Three-Tank System
Tank	79	237
Structure	19	22
Thrusters	8	8
Valves, plumbing, etc.	10	12
Electrical and electronics	29	29
Total	145	308



NOTE: All dimensions in meters

FIGURE 5-2. ONE-TANK HYDRAZINE MODULE

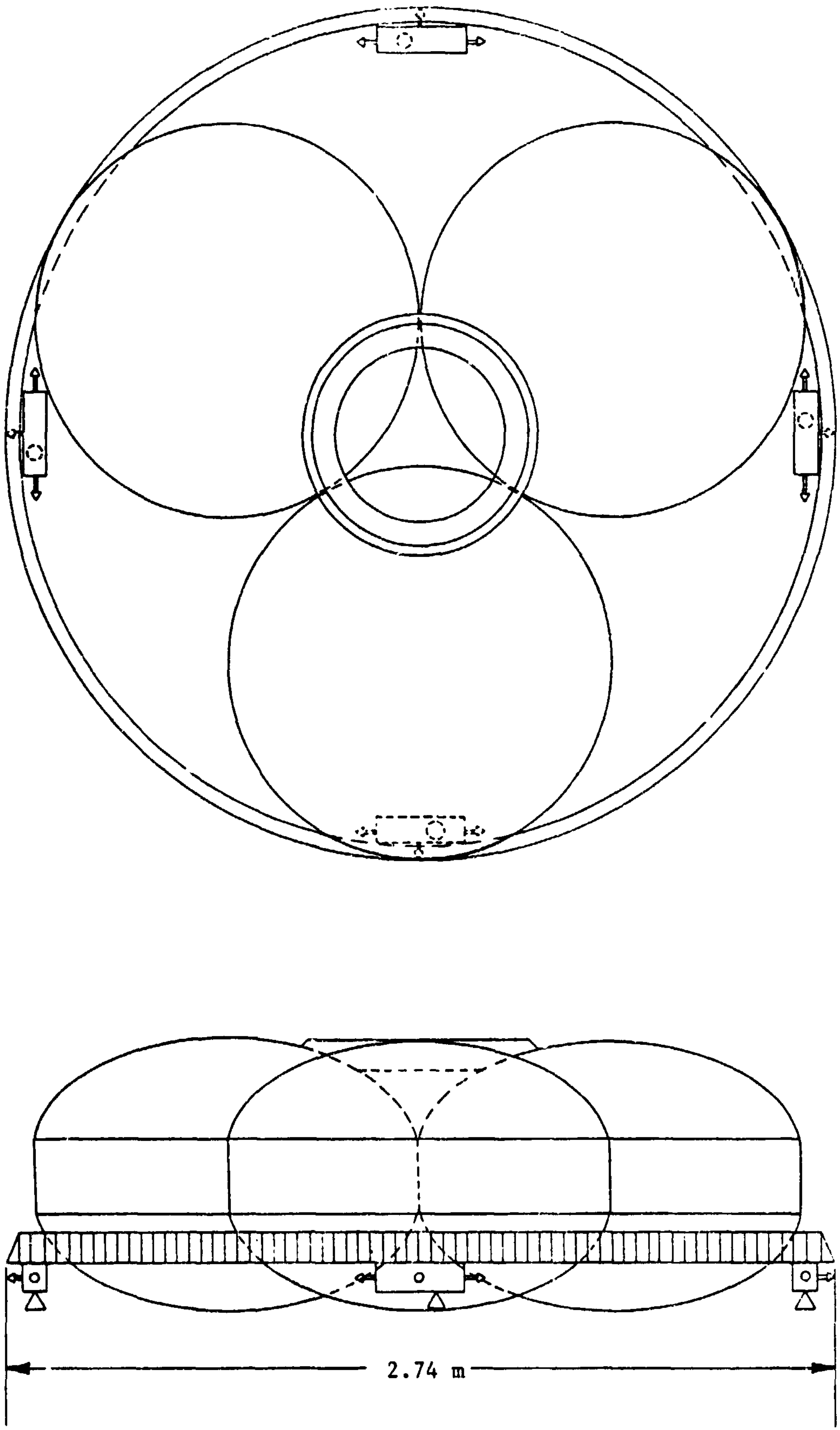


FIGURE 5-3. THREE-TANK HYDRAZINE SYSTEM

A sizing analysis was conducted using the SPS-I and the two new modules. The results, presented in Table 5-10, show the length load factor to dominate, in general. However, the mass and length load factors are now much closer, which is a desirable trend in view of the STS pricing policy. Redesign of the larger hydrazine systems produced one example (i.e., the servicing mission of EO-08A) in which Shuttle transportation charges would be determined by mass and not by length.

Costs associated with the new monopropellant hydrazine modules were estimated using the relationships developed in Section 4. Recurring cost was estimated to be $0.9,M for the one-tank module and $1.20M for the three-tank propulsion system. Engineering development costs associated with the one- and three-tank configurations are, respectively, $2.515M and $2.713M.

5.2.2 New Bipropellant Tank Design

Efforts toward redesigning the bipropellant tankage and modules were directed at defining two systems using the philosophy previously discussed for hydrazine. Review of the initial bipropellant sizing study indicated a maximum capacity requirement of 1200 kg. Assuming the same 40 percent minimum load already discussed, a module of this size could be used for propellant loadings down to 480 kg. As in the case of hydrazine, mission requirements were such that a gap existed between this value and approximately 328 kg of bipropellants. It was decided to investigate a new tank of 400-kg capacity with the higher energy missions, utilizing a three-tank arrangement to achieve the needed maximum loading. Rather than develop an extremely small bipropellant system to handle the HE-07A and HE-27A missions, it has been assumed that these missions would continue to use the SPS-I module.

At this point, it was necessary to evaluate the advantages and disadvantages of completely separate tanks for the oxidizer and fuel, as opposed to a common bulkhead design. The total separation of these propellants might appear to be favorable from a safety standpoint. Since nitrogen tetroxide (NTO) and monomethyl-hydrazine (MMH) are hypergolic, the mounting of two distinct tanks even a small distance from one another serves to reduce the possibility of simultaneous rupture and ignition. However, due to density differences between NTO and MMH, the use of two distinct tanks may result in an undesirable lateral center-of-gravity (c.g.) position. Any c.g. problems could be countered by the use of four tanks, with the two oxidizer tanks

TABLE 5-10. RECONFIGURED MONOPROPELLANT HYDRAZINE SYSTEMS

Mission	Shuttle RET/SER(a)	Mission Type	ΔV(b), m/s	Mass of Indicated Component, kg: S/C + Bus	Dry	Hydrazine	Upper Stage	Total	Mass Load Factor(c)	Length of Indicated Component, m: S/C	Bus	Hydrazine	Upper Stage	Total	Length Load Factor(d)
HE-08A	–	Deploy	156	8635	308	671	0	9614	0.43	5.2	1.3	0.86	0	7.36	0.54
	SH	Return	156	8635	308	671	0	9614	0.43	5.2	1.3	0.88	0	7.36	0.54
	SH	Service	312	8635	308	1391	0	10334	0.47	5.2	1.3	0.86	0	7.36	0.54
HE-07A	–	Deploy	126	735	69	48	0	852	0.04	0.3	1.3	0.5	0	2.1	0.15
HE-27A	–	Deploy	46	735	69	17	0	821	0.04	0.3	1.3	0.5	0	2.1	0.15
SO-03A	–	Deploy	172	1635	145	148	0	1928	0.09	2.0	1.3	0.56	0	4.16	0.30
	SH	Return	343	1635	145	307	0	2087	0.09	2.0	1.3	0.86	0	4.16	0.30
	SH	Service	515	1635	145	480	0	2260	0.10	2.0	1.3	0.86	0	4.16	0.30
AP-01A (10)	–	Deploy	2780(e) (perigee)	795	145	343	2593(f)	3876	0.18	0.3	1.3	0.86	1.9	4.36	0.32
	–	–	671(e) (apogee)	–	–	–	–	–	–	–	–	–	–	–	–
(28.5)	SH	(g)	1373	795	308	981	0	2084	0.09	0.3	1.3	0.86	0	2.46	0.18
(56)	DED	(g)	1373	795	308	981	0	2084	0.11	0.3	1.3	0.86	0	2.46	0.18
(90)	DED	(g)	1373	795	308	981	0	2084	0.17	0.3	1.3	0.86	0	2.46	0.18
AP-02A (28.5)	–	Deploy	1782	735	308	1339	0	2382	0.11	0.3	1.3	0.86	0	2.46	0.18
(56)	–	Deploy	1782	735	308	1339	0	2382	0.12	0.3	1.3	0.86	0	2.46	0.18
EO-08A	–	Deploy	284	1595	145	245	0	1985	0.19	2.0	1.3	0.86	0	4.16	0.30
	SH	Return	540	1595	145	495	0	2235	0.22	2.0	1.3	0.86	0	4.16	0.30
	SH	Service	1238	1595	308	1475	0	3378	0.33	2.0	1.3	0.86	0	4.16	0.30
EO-12A	–	Deploy	341	1635	145	305	0	2085	0.20	4.0	1.3	0.86	0	6.16	0.45
	SH	Return	698	1635	308	742	0	2685	0.26	4.0	1.3	0.86	0	6.16	0.45
	SH	Service	1200	1635	308	1446	0	3389	0.33	4.0	1.3	0.86	0	6.16	0.45
EP-13A	–	Deploy	341	1635	145	305	0	2085	0.20	4.0	1.3	0.86	0	6.16	0.45
	SH(SS)	Return	1204	1635	308	1452	0	3395	0.33	4.0	1.3	0.86	0	6.16	0.45
	DED	Service	990	1635	308	1131	0	3074	0.30	4.0	1.3	0.86	0	6.16	0.45
EO-15A	–	Deploy	2440(e) (perigee)	995	0	0	5301(h)	7463	0.34	1.5	1.3	0	4.5	7.3	0.53
	–	–	1830(e) (apogee)	–	–	–	1167(i)	–	–	–	–	–	–	–	–
EO-61A	–	Deploy	280	1135	145	177	0	1457	0.14	4.0	1.3	0.86	0	6.16	0.45
	SH	Return	704	1135	145	494	0	1774	0.17	4.0	1.3	0.88	0	6.16	0.45
	SH	Service	1204	1135	308	1078	0	2521	0.25	4.0	1.3	0.86	0	6.16	0.45
EO-64A	–	Deploy	341	1635	145	305	0	2085	0.20	4.0	1.3	0.86	0	6.16	0.45
	SH	Return	698	1635	308	742	0	2685	0.26	4.0	1.3	0.86	0	6.16	0.45
	SH	Service	1200	1635	308	1446	0	3389	0.33	4.0	1.3	0.86	0	6.16	0.45
EO-65A	–	Deploy	206	2635	145	277	0	3057	0.30	4.0	1.3	0.86	0	6.16	0.45
	SH	Return	462	2635	308	703	0	3646	0.36	4.0	1.3	0.86	0	6.16	0.45
	SH	Service	891	2635	308	1505	0	4448	0.44	4.0	1.3	0.86	0	6.16	0.45
OPN-02A	SH(P)	Deploy	344	885	145	493	0	1523	0.12	1.0	1.3	0.86	0	3.16	0.23
	DED	Return	1156	885	308	846	0	2039	0.16	1.0	1.3	0.86	0	3.16	0.23
	DED	Service	1459	885	308	1153	0	2346	0.19	1.0	1.3	0.86	0	3.16	0.23

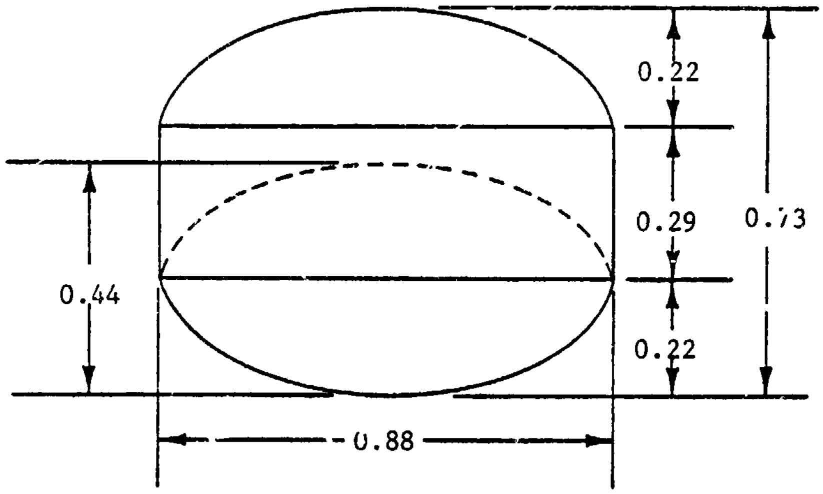
(a) RET = return; SER = service; SH = shared; DED = dedicated; SH(SS) = shared sun-synchronous, and SH(P) = shared polar.
(b) ΔV is defined as 1.10 times mission velocity requirement.
(c) Load factor = (cargo mass/Shuttle maximum mass) x 1.33.
(d) Load factor = (cargo length/Shuttle maximum length) x 1.33.
(e) ΔV for solid motors is defined as 1.05 times mission requirement.
(f) Offloaded version of small IUS motor (338 kg of propellant were removed).
(g) ΔV assumed constant for deploy, return, and service missions.
(h) Stretched version of small IUS motor with 4200 kg of propellant (including 66-kg adapter).
(i) TE-M-364-4 motor (mass includes 45-kg adapter).

mounted opposite one another and the two fuel tanks mounted opposite each other. Large numbers of tanks, however, can lead to reduced system reliability due to the increased number of components subject to failure. From the standpoints of c.g. control and reliability, a common bulkhead tank would appear to be advantageous. In such a configuration, the oxidizer and fuel would be mounted axially, one over the other, thus negating the density variation and reducing system complexity by minimizing the number of tanks involved. Any safety reservations that may result from this design can be partially alleviated by using a double-walled bulkhead to separate the propellants. TRW is currently designing such a common-bulkhead tank for use with their TDRS liquid apogee motor.(5-3) A decision, therefore, was made to base the analysis on a design similar to TRW's.

The volume required to contain 400 kg of NTO/MMH was calculated using a bulk density of 1120 kg/m^3 for these propellants. The resulting volume was approximately 0.36 m^3. For the size of thruster under consideration in this study, it is typical to use an oxidizer-to-fuel ratio of about 1:6. This results in equal volumes of NTO and MMH. Calculation of an oblate spheroid tank with a volume of 0.18 m^3 and a diameter-to-height ratio as discussed in the previous subsection yields a diameter of 0.88 m and a height of 0.44 m. Addition of an identical volume to contain the fuel results in the tank configuration shown in Figure 5-4. This tank, when used with four 22-N bipropellant thrusters, produces a propulsion module which is approximately 0.73 m in length. Redesign has produced a 38 percent reduction in overall system length when compared to the TRW Multimission Bipropellant Propulsion System.

Mass statements for the new bipropellant systems are shown in Table 5-11. These data are based on the component masses of the TRW bipropellant system.(5-1) Estimates of the module masses include external pressurant tank(s) and control assembly. The use of external pressure supplies can lead to complications for extended missions if even small leaks occur in the system. Multiple pressurant feed systems with explosively operated connects and disconnects have been added to these systems to seal off this subassembly between uses.

The complete bipropellant modules are shown in Figures 5-5 and 5-6. The results of a sizing analysis using these designs are presented in Table 5-12. These data indicate again that STS transportation charges



NOTE: All dimensions in meters

FIGURE 5-4. NEW BIPROPELLANT TANK

TABLE 5-11. REDESIGNED BIPROPELLANT SYSTEM MASS

Component	One-Tank System	Three-Tank System
Tank	36	108
Pressurant tank(s) - Full	35	70
Pressurant control assy. (2)	12	12
Propellant supply assy.	3	4
Fill and vent assy.	1	1
Structure	22	22
Thrusters (4)	2	2
Total	111	219

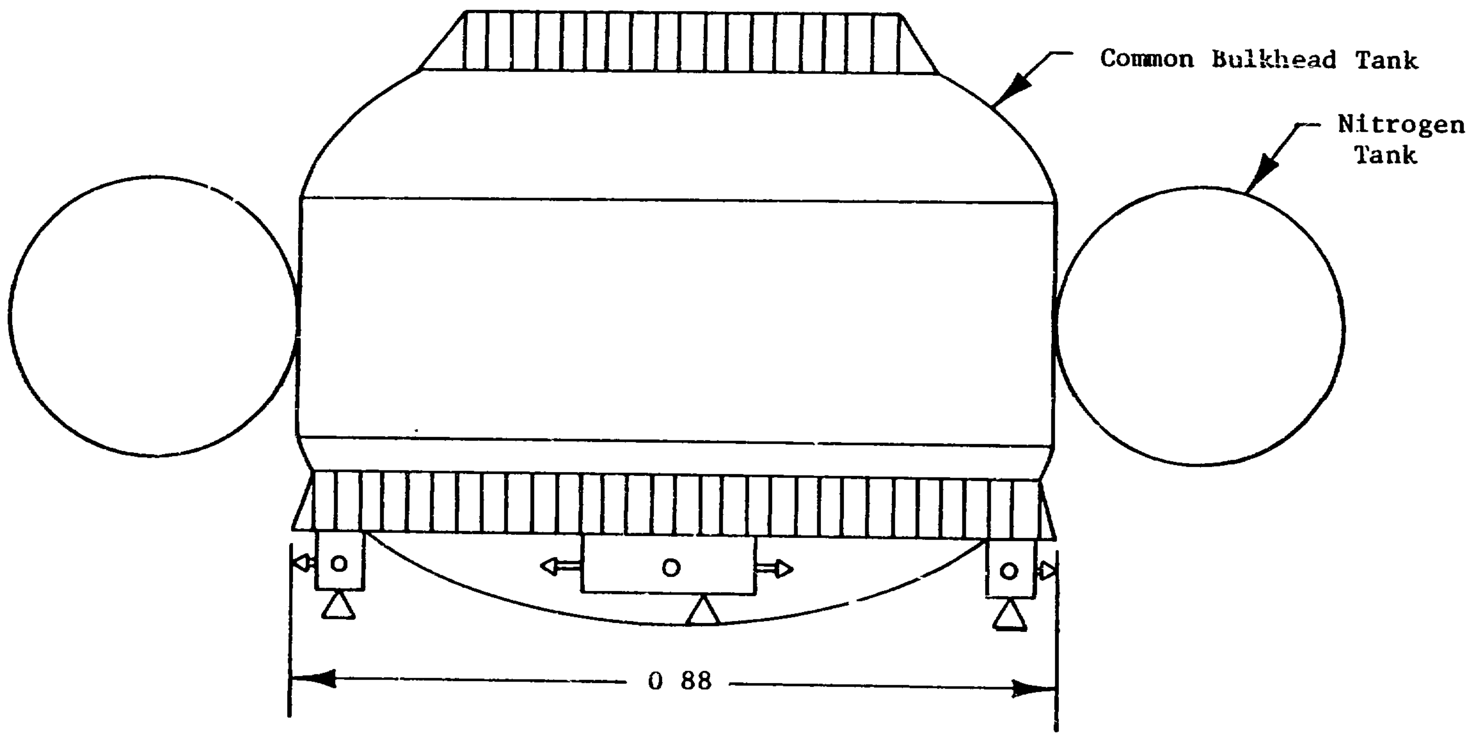


FIGURE 5-5. ONE-TANK NEW BIPROPELLANT SYSTEM

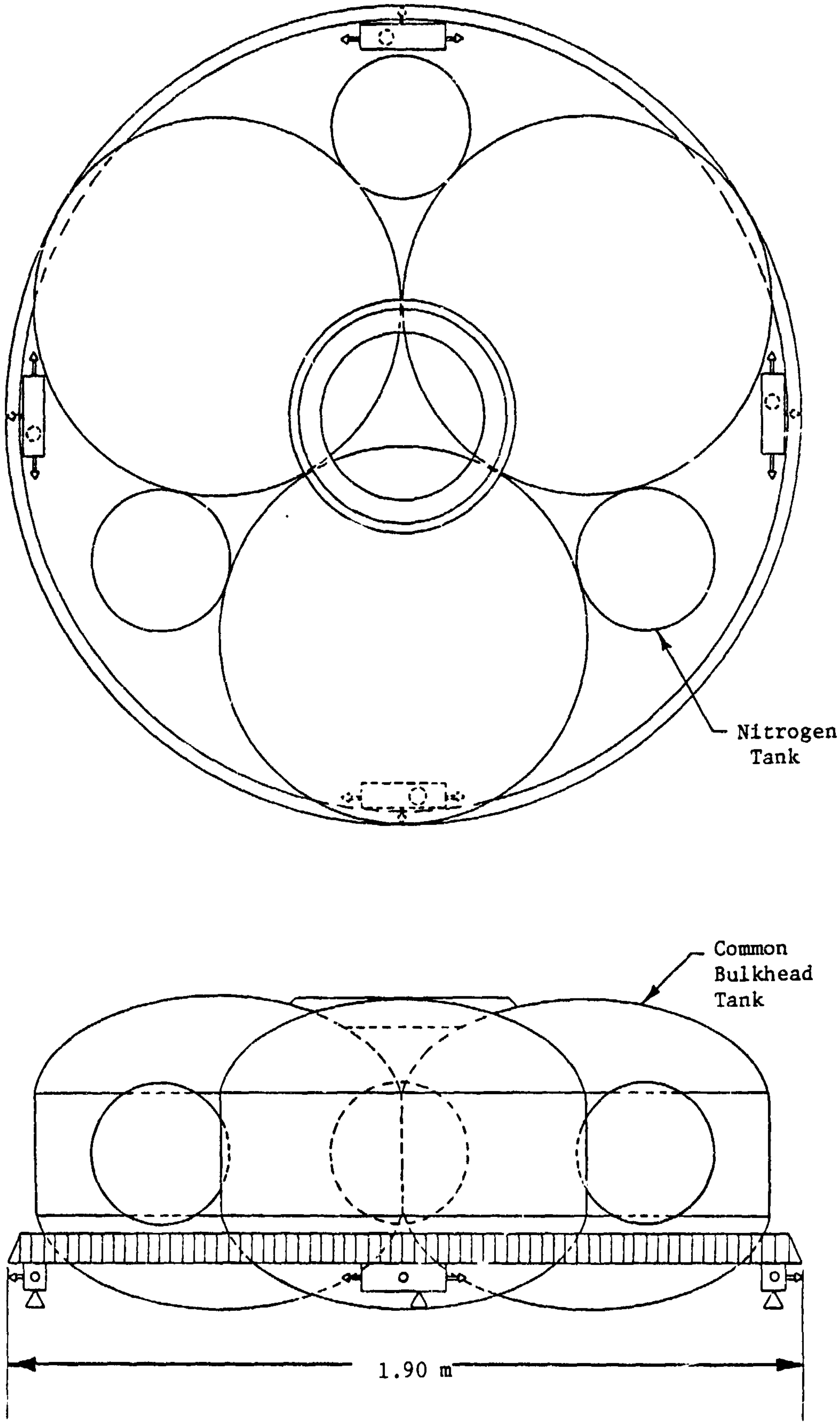


FIGURE 5-6. NEW THREE-TANK BIPROPELLANT MODULE

TABLE 5-12. RECONFIGURED BIPROPELLANT PROPULSION SYSTEMS

Mission		Shuttle RET/SER[a]	Mission Type	ΔV[b], m/s	Mass of Indicated Component, kg: S/C+Bus	Div	Bipropellant	Upper Stage	Total	Mass Load Factor[c]	Length of Indicated Component, m: S/C	Bus	Bipro-pellant	Upper Stage	Total	Length Load Factor[d]
HE-08A		–	Deploy	149	8635	219	485	0	9339	0.42	5.2	1.3	0.74	0	7.24	0.53
		SH	Return	149	8635	219	485	0	9339	0.42	5.2	1.3	0.74	0	7.24	0.53
		SH	Service	298	8635	219	996	0	9850	0.45	5.2	1.3	0.74	0	7.24	0.53
SO-03A		–	Deploy	164	1635	111	106	0	1852	0.08	2.0	1.3	0.74	0	4.04	0.29
		SH	Return	328	1635	111	217	0	1963	0.09	2.0	1.3	0.74	0	4.04	0.29
		SH	Service	491	1635	111	335	0	2081	0.09	2.0	1.3	0.74	0	4.04	0.29
AP-01A	(10)	–	Deploy	2780 (perigee)	795	111	233	2352[e]	3491	0.16	0.3	1.3	0.74	1.9	4.24	0.31
			–	640 (apogee)	–	–	–	–	–	–	–	–	–	–	–	–
	(28.5)	SH	(f)	1310	795	219	606	0	1620	0.07	0.3	1.3	0.74	0	2.34	0.17
	(56)	DED	(f)	1310	5	219	606	0	1620	0.06	0.3	1.3	0.74	0	2.34	0.17
	(90)	DED	(f)	1310	795	219	606	0	1620	0.13	0.3	1.3	0.74	0	2.34	0.17
AP-02A	(28.5)	–	Deploy	1701	735	219	799	0	1753	0.08	0.3	1.3	0.74	0	2.34	0.17
	(56)	–	Deploy	1701	735	219	799	0	1753	0.09	0.3	1.3	0.74	0	2.34	0.17
EO-08A		–	Deploy	271	1595	111	174	0	1880	0.18	2.0	1.3	0.74	0	4.04	0.29
		SH	Return	516	1595	111	346	0	2052	0.20	2.0	1.3	0.74	0	4.04	0.29
		SH	Service	1181	1595	219	954	0	2768	0.27	2.0	1.3	0.74	0	4.04	0.29
EO-12A		–	Deploy	326	1635	111	216	0	1962	0.19	4.0	1.3	0.74	0	6.04	0.44
		SH	Return	667	1635	219	500	0	2354	0.23	4.0	1.3	0.74	0	6.04	0.44
		SH	Service	1144	1635	219	938	0	2792	0.27	4.0	1.3	0.74	0	6.04	0.44
EO-13A		–	Deploy	326	1635	111	216	0	1962	0.19	4.0	1.3	0.74	0	6.04	0.44
		SH(SS)	Return	1150	1635	219	944	0	2798	0.27	4.0	1.3	0.74	0	6.04	0.44
		DED	Service	945	1635	219	746	0	2600	0.25	4.0	1.3	0.74	0	6.04	0.44
EO-15A		–	Deploy	2440 (perigee)	995	219	1123	5375[g]	.337	0.11	1.5	1.3	0.74	2.8	6.34	0.46
		–	–	1830 (apogee)	–	–	–	–	–	–	–	–	–	–	–	–
EO-61A		–	Deploy	268	1135	111	125	0	1371	0.13	4.0	1.3	0.74	0	6.04	0.44
		SH	Return	672	1135	111	339	0	1585	0.16	4.0	1.3	0.74	0	6.04	0.44
		SH	Service	1150	1135	219	689	0	2043	0.20	4.0	1.3	0.74	0	6.04	0.44
EO-64A		–	Deploy	326	1635	111	216	0	1962	0.19	4.0	1.3	0.74	0	6.04	0.44
		SH	Return	667	1635	219	500	0	2354	0.23	4.0	1.3	0.74	0	6.04	0.44
		SH	Service	1144	1635	219	938	0	2792	0.27	4.0	1.3	0.74	0	6.04	0.44
EO-65A		–	Deploy	195	2635	111	198	0	2944	0.29	4.0	1.3	0.74	0	6.04	0.14
		SH	Return	441	2635	219	488	0	3342	0.33	4.0	1.3	0.74	0	6.04	0.44
			Service	830	2635	219	1014	0	3868	0.38	4.0	1.3	0.74	0	6.04	0.44
OPN-02A		SH(P)	Deploy	305	885	111	332	0	1328	0.13	1.0	1.3	0.74	0	3.04	0.22
		DED	Return	1104	885	219	535	0	1639	0.16	1.0	1.3	0.74	0	3.04	0.22
		DED	Service	1392	885	219	713	0	1817	0.14	1.0	1.3	0.74	0	3.04	0.22

(a) RET = return; SER = service; SH = shared; DED = dedicated; SH(SS) = shared sun-synchronous, and SH(P) = shared polar.
(b) ΔV is defined as 1.05 times mission velocity requirement.
(c) Load factor = (cargo mass/Shuttle maximum mass) x 1.33.
(d) Load factor = (cargo length/Shuttle maximum length) x 1.33.
(e) Small IUS motor with 600-kg offload (includes 47-kg adapter).
(f) ΔV assumed constant for deploy, service, and return missions.
(g) Stretched version of small IUS motor with 4450 kg of propellant (mass includes 70-kg adapter).

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would be based on the length load factor. As in the case of the new hydrazine systems, the difference between the two load factors has been decreased. Due to the fixed spacecraft dimensions, it is unclear at this point as to whether equal load factors for length and mass can be achieved no matter how short the propulsion module is. Further efforts to reduce propulsion system length must be carefully weighed to determine their cost effectiveness in view of the STS pricing policy.

Costs for the reconfigured bipropellant systems were estimated using the information of Section 4. Recurring costs of $1.2M for the one-tank system and $1.8M for the three-tank system appear reasonable. Nonrecurring or engineering development costs for the one-tank system would be about $5.5M, with the three-tank module costing approximately $6.5M.

5.3 Program Costs for New Propulsion Systems

Total program costs associated with each propulsion technology have been calculated for the deploy-only, ground refurbishment, and on-orbit servicing mission models using the reconfigured propulsion modules. The data are presented in the same format as used in Subsection 5.1, to facilitate comparison.

5.3.1 Deploy-Only Costs

Program costs for the 42-mission deploy-only model are shown in Tables 5-13 and 5-14 for the new hydrazine and bipropellant systems, respectively. Comparison of this information does not show a dramatic cost advantage for either technology. The margin in favor of hydrazine has been increased somewhat by the reduction in development charges, which results from decreasing the number of required configurations. From the results shown here, there appears to be no justification for selecting bipropellants to satisfy the needs of this model. The sum for all four funding types is $491.8M for hydrazine and $505.4M for the bipropellants.

TABLE 5-13. DEPLOY MISSION MODEL COSTS USING NEW HYDRAZINE SYSTEMS

DHN
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 DHN	ED	2.4	3.7	1.3													7.4
	PM			3.3	3.4	4.0	2.9	3.1	6.2	4.1	4.1	4.3	5.1				40.4
	L/U		1.8	6.8	17.1	23.2	25.1	24.4	32.0	38.4	30.4	29.2	26.0	13.3			267.6
	BUS		6.3	12.6	16.8	16.8	12.6	21.0	23.1	16.8	18.9	21.0	10.5				176.4
																	491.8
1 HEDHN	ED																
	PM			.9	1.2	.9				1.2							4.2
	L/U		.5	2.9	5.1	5.4	.2		1.9	3.3	4.5						24.8
	BUS		2.1	4.2	4.2	2.1			2.1	2.1							16.8
1 SOLAR MAX MISS	ED																
	PM				1.0		1.0		1.0		1.0		1.0				4.8
	L/U			1.1	1.8	3.6	1.8	3.6	1.8	3.6	1.8	3.6	1.8	2.5			27.0
	BUS			2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1				21.0
1 APDHN	ED																
	PM			2.4	1.2	1.0		1.2	1.2	1.0	1.2	1.2	1.2				11.5
	L/U		1.3	2.9	5.4	3.7	3.8	1.7	3.9	4.4	4.7	3.2	2.6	1.5			38.9
	BUS		4.2	6.3	4.2	2.1	2.1	4.2	4.2	4.2	4.2	4.2	2.1				42.0
1 EODHN	ED																
	PM					2.1	1.0	1.9	4.1	1.9	1.9	3.[illegible]	1.9				18.0
	L/U				4.8	9.7	16.9	17.2	24.[illegible]	27.1	19.3	21.5	20.2	7.6			168.6
	BUS				6.3	8.4	6.3	14.7	14.7	8.4	18.6	18.6	4.2				88.2
1 ALL WEATR & SEA M	ED																
	PM						1.0						1.0				1.9
	L/U					.8	1.4	1.9				.8	1.4	1.9			8.3
	BUS					2.1	2.1					2.1	2.1				8.4
1 DHNPROP	ED	2.4	3.7	1.3													7.4
	PM																
	L/U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

TABLE 5-14. COSTS FOR DEPLOY MODEL USING NEW BIPROPELLANT MODULES

DBN
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
● DBN	ED	4 3	7 1	2.8													14.2
	PM			4.5	4 8	6.3	3.6	4.2	9 6	5.4	5.4	7.2	6.6				57 6
	L/U		1 8	6 6	16.4	22.2	23.9	23.5	30 8	36.9	29.4	28.0	24 8	12.9			257.2
	BUS		6.3	12 [illegible]	16.8	16 8	12 6	21.0	23.1	16.8	18.9	21.0	10 5				176 4
																	505.4
1 HEDBN	ED																
	PM			.9	1.8	.9				1 8							5.4
	L/U		.5	2.8	5 0	5 3	1.2		1 9	3.2	4.4						24 5
	BUS		2.1	4 2	4 2	2.1			2 1	2.1							16 8
1 SOLAR MAX MISSION	ED																
	PM				1 2		1 2		1 8		1.2		1.2				6 0
	L/U			1.0	1 8	3.4	1.8	3.4	1 8	3 4	1 8	3 4	1 8	2 4			26 1
	BUS			2 1	2 1	2 1	2 1	2.1	2 1	2.1	2.1	2 1	2.1				21 0
1 APDBN	ED																
	PM			3 6	1 8	1 2		1.8	1 8	1 2	1.8	1.8	1 8				16 8
	L/U		1.2	2 7	5 1	3 6	3.5	1 7	3.7	4 2	4.6	3.1	2.4	1 4			37 1
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 4	4.2	4.2	2 1				42 0
1 EODBN	ED																
	PM					4 2	1 2	2 4	6 6	2 4	2 4	3 4	2 4				27 0
	L/U				4 4	9 1	16 1	16 6	23 4	26 0	19.7	20 7	.9 2	7 3			161 5
	BUS				6 3	8 4	6 3	14 7	14 7	8 4	1[illegible].8	18 6	4 2				88 2
1 ALL WEATR & SEA M	ED																
	PM						1.2						1.2				2 4
	L/U					.8	1.3	1.8				.8	1 3	1.8			7.9
	BUS					2 1	2.1					2.1	2 1				8 4
1 NBIPROP	ED	4 3	7 1	2.8													14.2
	PM																
	L/U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

5.3.2 Costs for Ground Refurbishment Model

Cost information for the ground refurbishment mission model using both the new hydrazine and bipropellant modules is shown in Tables 5-15 and 5-16, respectively. As in the case of the deploy-only model, the total cost differential between hydrazine at $683.0M and the bipropellants at $701.4M has increased. This results from the proportionately larger decrease in transportation charges for hydrazine and the reduced hydrazine engineering development cost, as mentioned earlier. The emergence of the favorable cost position for hydrazine, coupled with its reduced complexity and more acceptable safety characteristics, would likely result in selection of hydrazine for this mission model.

5.3.3 On-Orbit Servicing Cost with New Propulsion Systems

The costs pertaining to the on-orbit servicing model for the new hydrazine and bipropellant modules show little in the way of trends not previously discussed. As shown in Tables 5-17 and 5-18, inclusion of all four funding types results in a total price of $496.3M for the reconfigured hydrazine case and $512.7M for the bipropellants. As discussed in the preceding subsections, there does not appear to be any need to reverse our earlier selection of hydrazine for use with this mission model.

5.4 Analysis of Transportaton Costs with Discounting

Space Transportation System (Shuttle) costs are the driving factors representing 80 percent or more of total transportation costs. Thus, reasonable developments which can reduce Shuttle charges can be justified. One of these is an oblate (short) tank, considered in the previous subsection. The transportation costs of the mission model for each of the technologies and scenarios are given in rank order in Table 5-19, together with their discounted costs. The module development costs are also shown; the development costs do not have any significant effect on the total transportation costs in comparison with other factors such as the operational scenarios and Shuttle charges.

The discounting procedure used for the results of Table 5-19 permits consideration of the time value of funds invested in the alternative methods of achieving an equivalent space transportation capability. In

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TABLE 5-15. COSTS FOR GROUND REFURBISHMENT MODEL EMPLOYING NEW HYDRAZINE PROPULSION

RMN
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 RMN	ED	2.4	3.7	1.3													7.4
	PM			3.3	3.4	4.2	3.1	3.6	6.7	4.6	4.3	4.7	5.5				43.4
	L/V		1.8	6.8	17.9	30.0	37.8	45.0	46.7	61.5	50.3	55.1	40.2	31.7	16.4	14.3	455.8
	BUS		6.3	12.6	16.8	16.8	12.6	21.0	23.1	16.8	18.9	21.0	10.5				176.4
																	683.0
1 HERMN	ED																
	PM			.9	1.2	.9				1.2							4.2
	L/V		.5	2.9	5.1	6.2	2.7	2.0	1.9	3.3	5.3	1.4	2.0				33.4
	BUS		2.1	4.2	4.2	2.1			2.1	2.1							16.8
1 SOLAR MAX MISSION	ED																
	PM				1.0		1.0		1.0		1.0		1.0				4.8
	L/V			1.1	1.5	4.4	3.3	6.4	3.3	6.4	3.3	6.4	3.3	5.3	1.4	2.0	48.3
	BUS			2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1				21.0
1 APRMN	ED																
	PM			2.4	1.2	1.0		1.2	1.2	1.0	1.2	1.2	1.2				11.5
	L/V		1.3	2.9	6.2	8.8	11.8	10.8	3.9	8.0	11.0	12.6	4.0	3.4			84.1
	BUS		4.2	6.3	4.2	2.1	2.1	4.2	4.2	4.2	4.2	4.2	2.1				42.0
1 EORMN	ED																
	PM					2.4	1.0	2.4	4.5	2.4	2.2	3.6	2.2				20.5
	L/V				4.8	9.7	18.8	21.0	31.4	38.4	30.8	34.0	29.5	17.4	8.8	3.9	245.2
	BUS				6.3	8.4	6.3	14.7	14.7	8.4	12.6	12.6	4.2				88.2
1 ALL WEATR & SEA M	ED																
	PM						1.2						1.2				2.4
	L/V					.8	1.4	5.6	6.2	8.4		.8	1.4	5.6	6.2	8.4	44.9
	BUS					2.1	2.1					2.1	2.1				8.4
1 DMMPROP	ED	2.4	3.7	1.3													7.4
	PM																
	L/V																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

TABLE 5-16. GROUND REFURBISHMENT COSTS FOR MISSIONS USING NEW BIPROPELLANT SYSTEMS

RBN
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 RBN	ED	4.3	7.1	2.8													14.2
	PM			4.5	4.8	6.9	4.2	5.4	10.8	6.6	6.0	8.4	7.8				65.4
	L/V		1.8	6.6	17.2	29.0	36.7	44.1	45.5	60.1	49.4	54.0	39.0	31.3	16.4	14.3	445.4
	BUS		6.3	12.6	16.8	16.8	12.6	21.0	23.1	16.8	18.9	21.0	10.5				176.4
																	701.4
1 HERBN	ED																
	PM			.9	1.8	.9				1.8							5.4
	L/V		.5	2.8	5.0	6.2	2.7	2.0	1.9	3.2	5.2	1.4	2.0				33.0
	BUS		2.1	4.3	4.2	2.1			2.1	2.1							16.8
1 SOLAR MAX MISSION	ED																
	PM				1.2		1.2		1.2		1.2		1.2				6.9
	L/V			1.0	1.8	4.3	3.2	6.3	3.2	6.3	3.2	6.3	3.2	5.2	1.4	2.0	47.4
	BUS			2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1	2.1				21.0
1 APRBN	ED																
	PM			3.6	1.8	1.2		1.8	1.8	1.2	1.8	1.8	1.8				16.8
	L/V		1.2	2.7	6.0	8.7	11.7	10.1	3.7	7.8	10.8	12.3	3.9	3.4			82.3
	BUS		4.2	6.3	4.2	2.1	2.1	4.2	4.2	4.2	4.2	4.2	2.1				42.0
1 EORBN	ED																
	PM					4.8	1.2	3.6	7.8	3.6	3.0	6.6	3.0				33.6
	L/V				4.4	9.1	17.8	20.4	30.4	34.3	30.[illegible]	33.1	28.6	17.3	8.8	3.9	238.2
	BUS				6.3	8.4	6.3	14.7	14.7	8.4	12.6	12.6	4.8				88.2
1 ALL WEATR & SEA M	ED																
	PM						1.8						1.8				3.6
	L/V					.8	1.3	5.5	6.2	8.4		.8	1.3	5.5	6.2	8.[illegible]	44.5
	BUS					2.1	2.1					2.1	2.1				8.4
1 NBIPROP	ED	4.3	7.1	2.8													14.2
	PM																
	L/V																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

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TABLE 5-17. ON-ORBIT SERVICING MODEL USING NEW MONOPROPELLANT HYDRAZINE

SHN
******* FOR INTERNAL PLANNING USE ONLY *******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 SHN	ED	2 4	3 7	1 3													7 4
	PM			3 3	3 4	4 4	2 4	3 6	3.6	[illegible]	2 4	5 0	3.4				33 5
	L/U		1 8	6 8	18 1	25 6	30.3	29.4	34.7	37 9	30.5	33 8	28 5	19.1	14 0	10.6	321 0
	BUS		6 3	12 6	16 8	14 7	10 5	14 7	12 8	8 [illegible]	14.7	16 8	6 3				134 4
																	496.3
1 HESHN	ED																
	PM			9	1 2	9											3 0
	L/U		5	2 9	5 1	6 3	2 8	2 1									19 6
	BUS		2 1	4 2	4 2	2 1											12 6
1 SOLAR MAX MISSION	ED																
	PM				1 0				1 0				1 0				2 9
	L/U			1 1	1 8	3 4	1 5	3 1	1 8	3 4	1 5	3 1	1 8	3 4	1 5	2 1	29 6
	BUS			2 1	2 1			2 1	2 1			2 1	2 1				12 6
1 APSHN	ED																
	PM			2 4	1 2	1 0		1 2	1 2	1 0	1 8	1 2	1 2				11 5
	L/U		1 3	2 9	6 3	5 2	5 7	1 7	3 9	4 4	4 7	4 1	4 1	3 5			47 9
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4 2	4 2	2 1				42 0
1 EOSHN	ED																
	PM					2 6	1 2	2 4	1 4	1 8	1 8	3 8					13 7
	L/U				4 9	9 9	10 0	16 8	22 7	21 6	24 3	25 7	21 8	6 6	6 8		178 7
	BUS				6 3	8 4	6 3	8 4	6 3	4 2	10 5	8 4					58 8
1 ALL WEATR & SEA M	ED																
	PM						1 2						1 2				2 4
	L/U					8	1.4	5.6	6 3	8.5		8	1 4	5 6	6 3	8 5	45.3
	BUS					2 1	2 1					2 1	2 1				8 4
1 DHNPROP	ED	2 4	3 7	1.3													7 4
	PM																
	L/U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

TABLE 5-18. NE'' BIPROPELLANT COSTS FOR ON-ORBIT SERVICING

SBN
******* FOR INTERNAL PLANNING USE ONLY*******
RATE DOLLARS IN MILLIONS

		79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
0 SBN	ED	4 3	7 1	2 8													14 2
	PM			4 5	4 8	7.5	3.6	5.4	6.6	3.0	3.6	9.0	4.8				52 8
	L/U		1 8	6 6	17 2	24.5	29.0	28.7	33.8	36.8	29 7	32.6	27 2	18.9	14 0	10.6	311.3
	BUS		6.3	12 6	16 8	14 7	10 5	14 7	12 6	8.4	14.7	16 8	6 3				134 4
																	512.7
1 HESBN	ED																
	PM			9	1 8	9											3 6
	L/U		5	2 8	5 0	6 2	2.8	2.1									19 4
	BUS		2.1	4.2	4 2	2 1											12 6
1 SOLAR MAX MISSION	ED																
	PM				1.2				1 2				1.2				3 6
	L/U			1 0	1.8	3.3	1 5	3.1	1 8	3 3	1.5	3 1	1 8	3 3	1.5	2 1	29 0
	BUS			2 1	2 1			2 1	2 1			2.1	2.1				12 6
1 APSBN	ED																
	PM			3 6	1 8	1 2		1 8	1 8	1 2	1.8	1.8	1 8				16 8
	L/U		1 2	2 7	6 0	5 1	5.6	1 7	3 7	4 2	4.6	4 0	4 0	3 5			46 1
	BUS		4 2	6 3	4 2	2 1	2 1	4 2	4 2	4 2	4.2	4 2	2 1				42 0
1 EOSBN	ED																
	PM					5 4	1 8	3 6	3 6	1 8	1.8	7 8					25 2
	L/U				4 4	9 1	17 8	16 3	22 0	20 9	23 8	24 8	20 1	8 6	6 2		171 9
	BUS				6 3	8 4	6 3	8 4	6 3	4 2	10.5	8 4					58 3
1 ALL WEATR & SEA M	ED																
	PM						1 8						1 8				3 6
	L/U					8	1.3	5.5	6 3	8.5		8	1 3	5 5	6 3	8 5	44 9
	BUS					2 1	2 1					2 1	2 1				8 4
1 NBIPROP	ED	4 3	7 1	2.8													14 2
	PM																
	L/U																
	BUS																

NOTE: For explanation of abbreviations refer to Table 5-2.

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TABLE 5-19. RANKING OF ALTERNATIVE PROGRAMS BY TRANSPORTATION COSTS AND TRANSPORTATION COSTS DISCOUNTED AT 10 PERCENT

	$, Millions, 1977					
	Undiscounted Costs			Discounted Costs		
Alternative Programs	Development	Total	Rank	Development	Total	Rank
Deployment Scenarios						
Short Deployed Modules-Hydrazine	7.4	324.4	1	6.8	167.8	1
Short Deployed Modules-Bipropellant	14.2	329.0	2	13.1	178.0	2
Deployed Modules-Hydrazine	9.1	348.8	3	8.5	185.0	3
Deployed Modules-Bipropellant	13.0	356.1	4	12.4	192.6	4
Servicing Scenarios						
Short Serviced Modules-Hydrazine	7.4	361.9	1	6.8	186.0	1
Serviced Modules-Hydrazine	14.3	378.0	2	12.3	198.2	3
Short Serviced Modules-Bipropellant	14.2	378.3	3	13.1	197.8	2
Serviced Modules-Bipropellant	13.0	395.9	4	12.4	209.2	4
Refurbishment Scenarios						
Short Refurbished Modules-Hydrazine	7.4	506.6	1	6.8	251.7	1
Short Refurbished Modules-Bipropellant	14.2	525.0	2	13.1	264.2	2
Refurbished Modules-Hydrazine	11.6	541.1	3	10.7	271.3	3
Refurbished Modules-Bipropellant	13.0	548.1	4	12.4	276.9	4

evaluating programs on the basis of discounted costs, it is assumed that funds not expended on the programs under consideration can produce benefits by being expended elsewhere. Thus, if two alternative methods have different funding profiles over their life, but the same (undiscounted) total costs, the comparison of their discounted costs indicates that the program with the lower discounted costs (other things being equal) should be selected. The funds not required during an early phase of the project with higher discounted costs are available for use elsewhere.

The formula used here is:

$$\text{Discounted Total Costs} = \sum_{j=0} \frac{c_j}{(1 + i)^j} ,$$

where c_j is annual program transportation costs (exclusive of the MMS bus) for a given year, as in Table 5-19, and i is the discount rate. The first year of costs (1979) is $j = 0$ and the last year (1993) is $j = n = 14$. The discount rate used here, 10 percent ($i = 0.1$), is a standard used in government analyses of this type, as given in DOD Instruction 7041.3.[5-4] The exact value of the discount rate is somewhat arbitrary; the object is to illuminate the effect of the time value of money in assessing projects in relation to a return on that investment available elsewhere. The effect of a discounting analysis is to favor projects which require major expenditures in the distant future over projects which require major near-term expenditures.

As Table 5-19 indicates, the alternatives examined retain generally the same ranking under discounting as they had without discounting. The exceptions occur in the middle of the rankings and where the undiscounted costs are also very close. The major conclusion drawn is that the rankings are correct when the time value of money is considered. The cause determining this conclusion is the high level (80 to 85 percent) of Shuttle charges that comprise the total transportation cost. These are distributed evenly across the time span considered and have a heavy weight in determining both discounted and undiscounted costs.

In summary, the propulsion module development costs have little effect on total mission model transportation costs, and the relatively high STS use charges indicate that reasonable amounts of development funding spent to reduce these charges can be justified. The development and use scenarios are correctly ranked by their undiscounted transportation costs when the time value of money at 10 percent per year is considered.

5.5 Cost Effectiveness of New Propulsion Designs

Analysis of hydrazine and bipropellant propulsion modules has indicated that hydrazine is the likely selection to fulfill the propulsion needs of the MMS. Discussion of the cost effectiveness of redesigning the propulsion modules is limited to the hydrazine case, but the methods and conclusions are also applicable to bipropellant modules if additional performance is later found to be required.

In terms of total undiscounted program costs, redesign of the hydrazine propulsion systems would result in an approximate savings of $33.4M for the deploy-only mission model. This figure translates into a reduction of 6.4 percent in total program costs. Similar values are $34.5M (4.8 percent) for the ground refurbishment model and 16.1M (3.1 percent) for the on-orbit servicing missions. Our analysis assumes that spacecraft and MMS bus lengths are fixed, thus yielding undiscounted launch costs for these two components of $227.2M for the deploy-only and ground refurbishment models and $160M for the on-orbit servicing missions. This approach dictates that potential savings must be derived from reducing propulsion module length. Redesign of the propulsion systems results in a 45 percent reduction in launch charges for the propulsion modules. Other savings, not analyzed in this report, are also available from appropriate designs of the MMS and payload which save transportation costs without unduly increasing total costs. Comparison of the discounted total costs yields total cost savings of $18.1M for deploy only, $19.6M for ground refurbishment, and $12.2M for on orbit servicing.

The new designs would be viewed as cost effective when the discounted savings exceed the discounted incremental development by a sufficient amount to cover inherent uncertainties in cost information. (About 20 to 30 percent might be used, as the available estimate of cost accuracies is ± 15 percent.) According to this criterion, the reconfigured (hydrazine) systems would appear to be a slightly better alternative than modifications of current designs. The picture is clouded somewhat, because the transportation charges did not decrease as much as originally expected. Review of the spacecraft dimensions used for this study shows that the propulsion module length for the majority of cases is a relatively small factor in overall payload length. In view of this, it is easier to see why reducing the propulsion system length does not result in a more substantial reduction in transportation charges. The spacecraft dimensions roughly coincide with what might be expected for missions compatible with both the current expendable launch vehicles and the Shuttle. Selection of either the SPS-II derivatives or the new systems purely on the basis of overall program costs as shown here may be misleading.

Use of the reconfigured monopropellant modules would result in an STS charge reduction of 9 percent for the EO-64A mission, while use of this same system results in a 21 percent decrease in transportation charges for AP-01A (28.5 deg). Since most of the spacecraft included in our models do not currently exist, it must be assumed that designers of these payloads will attempt to minimize length, unless this goal adversely affects spacecraft costs. The MMS bus length is a significant factor in the total length occupied within the cargo bay. This component is currently being designed, and few changes to its present configuration are expected. Reductions in spacecraft length would be derived by shortening the experiment package, which rides on top of the MMS bus. Evolution of the designs included in this study into configurations in which reduced propulsion module length could play a major role is not difficult to envision.

Furthermore, it is possible that reconfiguring the propulsion system to reduce length, and thus transportation costs, may attract users with severe expenditure constraints who might otherwise be priced out of the MMS market.

5.6 Cost Trade-offs on Special Propulsion Applications

Several special propulsion concepts were examined from a technical standpoint in Section 3.2. To determine the benefits of each of the concepts, it is advantageous to do a cost trade-off. For the variety of concepts considered, the cost trade-offs must be done differently for each application.

5.6.1 Drag Makeup Cost Trade

In the drag makeup analysis, the net mission remained fixed in terms of spacecraft weight, size, and lifetime. The differences were only in the propulsion system, which would be used for drag makeup. The cost comparison is, therefore, based upon the recurring costs of the propulsion modules and the marginal transportation cost. Any potential development costs are not included, since no estimates are made as to the number of drag makeup satellites or other potential uses of a given size of module over which these development costs might be spread. However, in practice, a single mission may be required to bear all development costs.

The four systems considered are a hydrazine system, augmented electrothermal hydrazine system, and two different 8-cm ion systems. The costs for these systems are taken from Section 4, and summarized here in Table 5-20. The costs used in the trade-off are based upon the twentieth unit costs, not the first unit cost. The impact of using the first unit cost will be discussed later.

TABLE 5-20. AUXILIARY PROPULSION HARDWARE COSTS[a]

System	Base Unit Cost, $M	Additional Costs, $M
Hydrazine[b]	0.540	0.105/1000 lb propellant[c]
Augmented electrothermal hydrazine[b]	0.540	0.105/1000 lb propellant[c] + 0.122 for power[d]
Two 8-cm ion engines[e]	0.893[f]	0.094 for power[d] + 0.756 if first unit
Four 8-cm ion engines[e]	1.357[f]	0.140 for power[d] + 1.512 if first unit

(a) Recurring costs only; 1977 dollars.
(b) No primary propulsion (i.e., no 5-lb or larger thrusters, etc.).
(c) First 1000 lb of propellant included in base cost.
(d) Assumes power cannot be obtained from spacecraft power.
(e) Spacecraft attitude control by momentum wheels.
(f) Twentieth unit costs, based upon data from LeRC.[5-5]

These drag makeup missions are near Shuttle altitude. The delivery mode in this analysis is assumed to be by the Shuttle only. If an additional stage is required, the impact on this stage will not be considered, since it is likely that a single propulsion system would be used for both propulsion requirements (drag makeup and satellite placement). Note that, for those altitudes less than the standard Shuttle orbit, the drag itself could be used to achieve the desired final orbit.

The only transportation charge is the Shuttle charge formula, which will be based upon the marginal increase in length of the spacecraft. The spacecraft is assumed to be mounted horizontally in the Shuttle bay, and the interface between the propulsion module for drag makeup and the spacecraft is taken to be a well-defined plane. In practice this may not be the case, but the actual spacecraft design is not a part of this study. The Shuttle charge for the spacecraft is assumed to be based upon the length factor, and it is also assumed that the addition of the propulsion does not alter this. Thus, the marginal Shuttle charge is based upon the length of the propulsion module. The base Shuttle charge for a dedicated flight by a NASA user is taken to be $18.5M in 1977 dollars, which corresponds to $1.349M per meter of Shuttle bay length used. Figure 5-7 shows the module cost for a Scout class payload for 3- and 5-year missions. The actual cost data shown are for a 1981 launch. Since the cost data are based upon the module size, which is a function of the launch year, the cost estimates are dependent upon launch year. The variation in costs for different years does not significantly change the trade-off between systems as to which is more cost effective. The maximum variation in costs is 9 percent for a 3-year mission and 5 percent for a 7-year mission.

For Scout class payloads, the augmented electrothermal hydrazine is the most cost effective for altitudes less than 180 km. At these altitudes, the thrust levels of the ion systems are insufficient to balance drag during the high solar activity times and the hydrazine mass requirements are large enough that the higher I_{sp} is beneficial. The 8-cm ion systems are most cost effective on the 7-year missions between 180 and 220 km. At these altitudes, the ion systems have sufficient thrust and the propellant requirements are still large enough (on the hydrazine systems) that the much higher I_{sp} of the ion systems can save on Shuttle length

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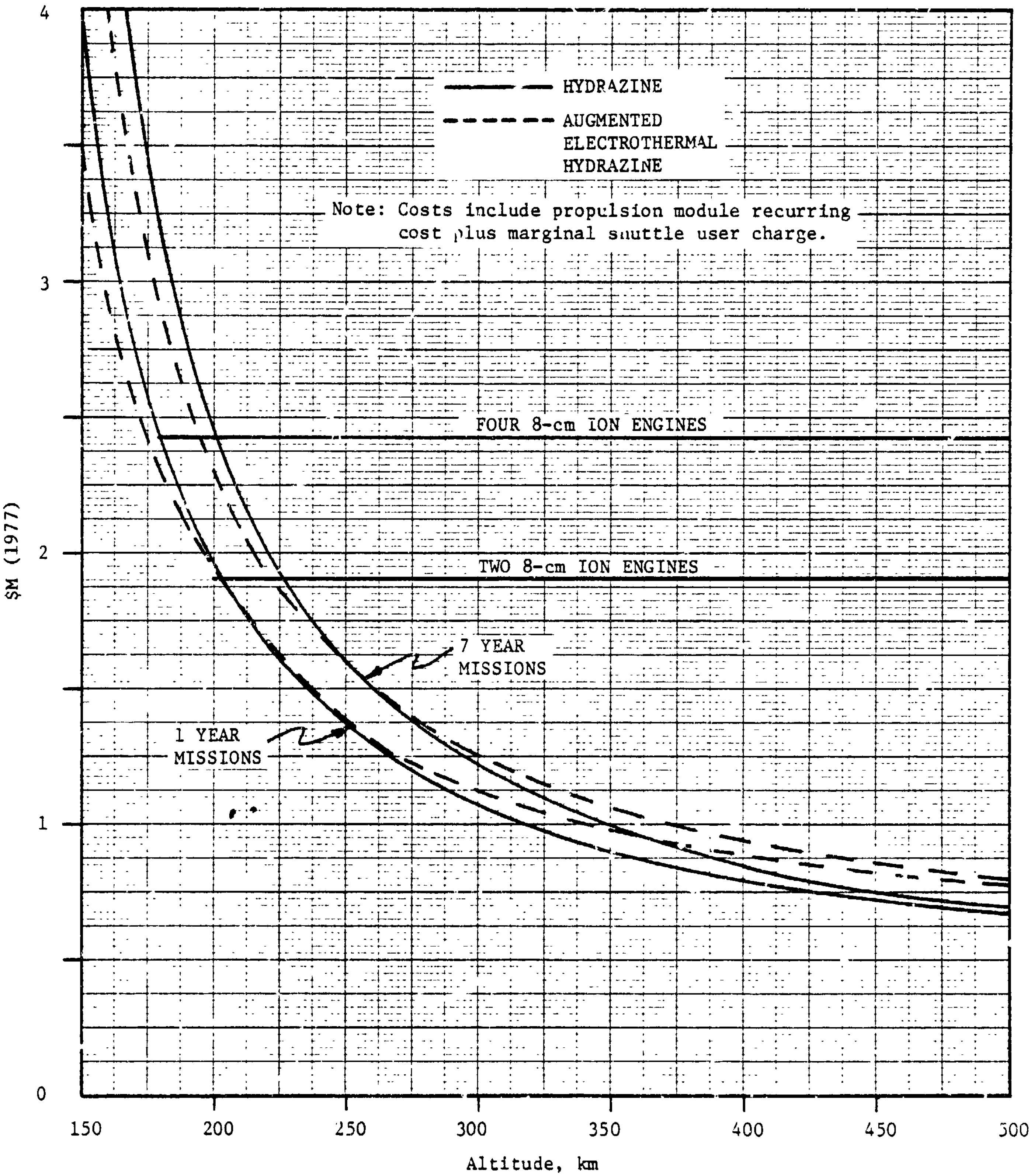

FIGURE 5-7. DRAG MAKEUP SYSTEMS COST FOR SCOUT CLASS PAYLOADS

charges to overcome the ion systems' higher unit cost. The propellant requirements for the ion systems are not significant, thus the ion systems costs are independent of altitude and duration. For shorter missions at these altitudes the hydrazine systems become more cost effective. At altitudes above 250 km, the propellant requirements are minimal and the catalytic hydrazine's smaller recurring cost makes it the most cost-effective system. The cost differences between the two hydrazine systems for altitudes above 200 km are generally less than $0.1M; however for long missions at low altitudes (7 years at 150 km) the difference becomes as large as $0.8M. At the minimum operational altitudes of the ion systems, the savings over hydrazine can be as great as $0.6M.

For Delta class spacecraft, Figure 5-8, the hydrazine systems are more cost effective than the ion systems at all altitudes. This is due to the high cross-sectional area of the spacecraft, which requires more thrust to balance drag. The crossover point between the catalytic hydrazine and the augmented electrothermal hydrazine is about 325 km. Below this altitude, the higher specific impulse of the electrothermal system results in lower propellant requirements, which yield lower transportation costs and lower overall system cost. Above these altitudes, the unit cost of the catalytic system dominates the transportation cost, with the result that the catalytic system is the most cost effective.

The one factor which could potentially impact the trade-offs is the potential of integrating the propulsion module into the spacecraft. The hydrazine modules are dominated (at altitudes less than 350 km) by the propellant tank. Thus, integrating the propulsion module into the spacecraft amounts to integrating the spacecraft around the propellant tank. For the ion systems, however, there may be more options, since they are composed of smaller components.

5.6.2 Sun-Synchronous Satellite Orbit Change Cost Trade-off

There are two basic cost trade-offs to examine for this type of mission. One trade-off is between chemical and ion propulsion and the other is involved in mission operations and the number of operational satellites. Performance of the second trade-off requires spacecraft costs, mission operations costs, and benefits received from the satellite. This is beyond the scope of this study.

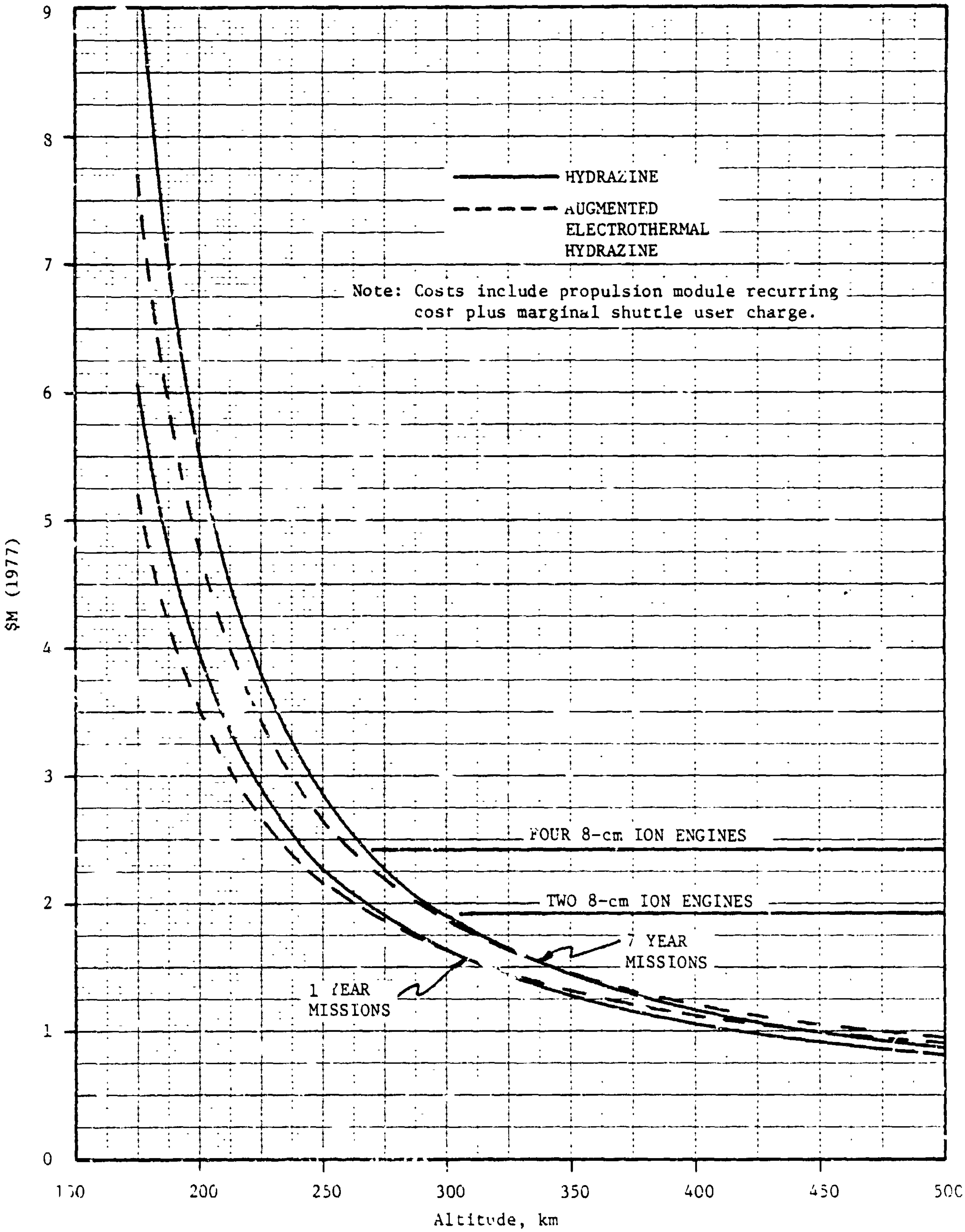

FIGURE 5-8. DRAG MAKEUP SYSTEMS COST FOR DELTA CLASS PAYLOADS

The trade-off between chemical and ion propulsion is based upon the following assumptions:

(1) A need exists for a satellite to alternate between different Sun viewing conditions.

(2) The satellite has the capability of returning to the Shuttle for servicing.

(3) A transfer time of 3 months between viewing conditions is acceptable.

(4) A single propulsion system is used for all main propulsion requirements.

(5) Three transfers between the different viewing conditions are required.

(6) Total mission life is at least 4 years.

(7) The spacecraft and orbit are based upon the Landsat D/E mission.

From the assumptions above, the velocity requirements can be estimated for a chemical propulsion system. From Table 3-5 and Equation (3-20), the total velocity is taken to be approximately 2200 m/sec. A cluster of seven Viking tanks on a Landsat sized spacecraft would provide only about 2000 m/sec. This configuration would be approximately 3 m in diameter and have about the largest number of tanks that could be considered for a single layer. The next step is to consider a cluster arrangement where, at some point, staging occurs. For this arrangement of tanks the propulsion module would double in length, but the mass increase would result in a Shuttle load factor of 0.56 based upon mass. The recurring cost of such a propulsion module would be approximately $2.2M. Thus, the recurring module cost plus the Shuttle launch charge would be approximately $12.7M.

Using an ion system with six 30-cm thrusters, the Shuttle charge could be reduced to approximately $6.5M, but the module recurring cost would be about $12.5M*, which gives a total cost of $19M. Thus, even with a savings of almost $4M in the Shuttle launch charge, the ion system is still more costly than the hydrazine system.

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*Cost estimate scaled from data in Tables 4-5 and 4-6.

Several factors not considered in this analysis are the Shuttle charges for the servicing flight, any nonrecurring propulsion module hardware cost, and mission operations costs. These factors would favor the hydrazine module over the ion system.

Other options are possible which could potentially be more cost effective than the hydrazine option with multiple Viking tanks. Among these would be the design of a new hydrazine tank, the replacement of the entire propulsion module during servicing (which may then be the reason for servicing), or going to a bipropellant module.

To estimate the cost for the bipropellant option, consider a module composed of three of the large bipropellant tanks used in the baseline costing analysis. By clustering the tanks, the length would not increase and the mass factor would then dominate and determine the Shuttle charge of $9.2M. The recurring cost of the bipropellant module would be about the same as that of the hydrazine combination, since fewer tanks are involved and staging is not required. Thus, the bipropellant option costs potentially less than hydrazine by about $1.2M, the difference in the Shuttle charges.

The cost comparison above gives sufficient information to determine that, even for this mission with a velocity requirement of over 2 km/sec, the chemical systems are less expensive than the ion systems, due to the high recurring cost of the ion systems. For this mission, the higher I_{sp} of the bipropellant gives a small cost advantage to bipropellants. However, when the module development costs are considered, and they depend on the overall activity of missions, the cost advantages of the bipropellants could disappear.

5.6.3 Auxiliary Propulsion Costs for Geosynchronous North-South Stationkeeping

For geosynchronous communications satellites, the net mass in orbit is an extremely important parameter, since additional on-orbit mass implies additional communication channel capability, which yields additional revenue. These additional revenues are potentially much larger than the costs associated with the various propulsion systems (and the additional transponders) if the demand is available. This analysis is not within the scope of this study. Table 5-21 shows the marginal increases in net spacecraft mass using alternate stationkeeping propulsion systems compared to using hydrazine and cost estimates of the different propulsion systems. The cost estimates are for the recurring cost of auxiliary propulsion systems, and do not include development or integration costs.

Based on the ratio of incremental dollars to incremental mass, the bipropellant system is best for both SSUS-D and SSUS-A class spacecraft, with the augmented electrothermal hydrazine next, giving larger net mass increases for a larger incremental cost. The combined hydrazine and 8-cm ion systems provide additional mass over and above what is possible using the augmented electrothermal system on SSUS-A sized spacecraft.

TABLE 5-21. GEOSYNCHRONOUS SPACECRAFT MASS INCREASES AND ASSOCIATED COSTS

System[a]	Recurring Cost, $M (1977)	Spacecraft Net Incremental Mass Using Indicated PKM[b], kg	
		SSUS-D	SSUS-A
A: Hydrazine (baseline)	0.45	0	0
B: Bipropellant	0.55	24	46
C: Augmented electrothermal hydrazine	0.70	36	63
D: 8-cm ion[c]	1.50	58	143
E: Catalytic + augmented electrothermal hydrazine	0.75	-3	20
F: Hydrazine + 8-cm ion	1.90	6	77

(a) For system definitions, see Table 3-15.
(b) Incremental masses derived from Table 3-16.
(c) 8-cm ion system can require long station acquisition time after AKM burn.

The maximum increase in spacecraft net mass is achieved using the 8-cm ion system for both initial station acquisition and North-South stationkeeping. This, however, can result in long times (several months) for initial station acquisition due to nominal drifts and correction of AKM and PKM errors. Use of a liquid AKM (with a commanded shutdown) may be a potential alternative to reduce the velocity requirements for the initial station acquisition system.

5.6.4 Geosynchronous Satellite Return Costs

Identification of a need for returning a spacecraft from geosynchronous orbit presents the major difficulty when this concept is considered. One possibility that can be envisioned would be to return a sapcecraft from geosynchronous orbit if there is potentially a cost savings in the building of satellites with lower reliabilities and then returning them for refurbishment if they fail. The savings in initial spacecraft costs would have to be substantial, however, since the costs to return a payload, refurbish it, and then relaunch it are fairly large. The refurbishment cost estimate is not within the scope of this study; nevertheless, an approximate estimate of the transportation costs for the return and relaunch is about $21M.

5.7 References

(5-1) TRW's Multimission Bipropellant Propulsion System, TRW Systems Group, February 1975.

(5-2) Landsat/MMS Propulsion Module Design Study, Task 4.4 Final Report, Rockwell International Space Division, September 24, 1976.

(5-3) TDRS Design Data, Attachment 2, prepared for Battelle under Contract No. M-4639(6540), TRW, 22 November 1976.

(5-4) Economic Analysis and Program Evaluation for Resource Management, DOD Instruction 7041.3, Office of the Assistant Secretary of Defense (Comptroller), The Pentagon, October 18, 1972.

(5-5) LeRC Memorandum, Subject: Data Package for MMS Study, From: 6132/Head, Large Thruster Section (D.C. Byers), March 22, 1977.

6.0 SUMMARY

Various propulsion concepts have been evaluated in this study as candidates for main propulsion on MMS missions. The major criterion for the comparison is the transportation cost; however, other aspects such as operational flexibility, safety, etc., have been examined. The results of these analyses are summarized, by technology, in Subsection 6.1. Additionally, several technologies were analyzed to determine the best approaches for satisfying propulsion requirements on four special case missions. The applicability of each of these technologies to additional propulsion tasks is discussed in Subsection 6.2.

In the course of this study, several conclusions were reached which are not related directly to any propulsion technology. Among these are: (1) uncertainty in Shuttle operations in general and WTR Shuttle operations in particular causes uncertainty in spacecraft propulsion requirements, and potentially could alter the intended operations, (2) the large cost of a dedicated flight could be sufficient to justify a new propulsion module for a single mission, and (3) the payload characteristics (length, mass, operational procedures, etc.) are likely to evolve to best take advantage of Shuttle capability. These factors along with a discussion as to how they affect the spacecraft propulsion requirements, are presented in Subsection 6.3.

6.1 MMS Main Propulsion

A number of conclusions were reached for each technology under consideration for the MMS primary propulsion application. In at least two instances, preliminary analyses indicated unfavorable operational and/or cost factors which eliminated the respective technology from further evaluation. The reasoning behind the selection philosophy is examined here in detail, with the subsections arranged in order of increasing usefulness to MMS.

6.1.1 Oxygen/Hydrogen

The cryogenic propellants (oxygen/hydrogen) have the highest performance of any of the chemical propulsion technologies considered. The larger I_{sp} connected with this oxidizer/fuel combination results in a reduction of the quantity of propellants required to perform a given mission.

This characteristic has, in the past, dictated the use of oxygen/hydrogen for Earth-escape missions such as interplanetary trajectory injection, and for energy-intensive missions such as delivery of large paylods to Earth orbit (i.e., the Shuttle), expendable launch vehicle delivery of spacecraft to geosynchronous transfer orbit. In most cases, these missions would have been difficult, if not impossible, to achieve with lower energy propellants.

To a large extent, the performance of the cryogenics, as discussed in Section 2, requires the sacrifice of operational simplicity. Current oxygen/hydrogen engines, such as the Pratt & Whitney RL-10 which is used on the Centaur stage, exhibit complicated starting procedures involving "chilldown" of the turbopumps to eliminate cavitation. An ignition system would be required, since oxygen and hydrogen are not a hypergolic combination. The presence of liquid hydrogen would necessitate the addition of a substantial amount of insulation if the system is to be operated for extended periods of time in the deep space environment. Venting of the propellant tanks might also be required to prevent excessive pressure buildup on orbit.

Designers of payloads destined to fly in the Shuttle era will most likely react to the decreased dependence on launch vehicle performance by stressing the use of less sophisticated but more reliable components to increase spacecraft life. Also inherent in STS operations is the potential for recovery of the spacecraft with subsequent on-orbit servicing or ground refurbishment. Therefore, to the mission planner, the propulsion system would play a vital role, not only in terms of delivery to the initial orbital station but also because it must perform reliably in later maneuvers to ensure completion of all mission objectives. A cryogenic stage, with its tank insulation, vent system, engine "chilldown" cycle, and ignition system, would be unattractive in this operational environment since all of the above factors tend to reduce system performance by increasing the dry weight and/or degrade the system reliability due to increased complexity.

For the missions analyzed in this study, the propulsion requirements are relatively small and are well within the capability of other chemical propulsion technologies. Lacking an obvious driver which would require the performance associated with the cryogenic propellants, it is unlikely that mission planners would be willing to tolerate the added constraints connected with oxygen/hydrogen propulsion.

The MMS mission concept appears to favor an engine thrust in the range of 22 to 445 N (5 to 100 lb_F). A value in this thrust range would permit operation of the primary propulsion system while spacecraft appendages are deployed and allows three-axis stabilization using small auxiliary thrusters. Since there are no flight-qualified oxygen/hydrogen engines of this size, an expensive development program would be needed. The large nonrecurring investment, when coupled with the higher recurring cost of a cryogenic stage, would further reduce the probability that oxygen/hydrogen would be competitive for use with MMS. Thus, cryogenic propellants were eliminated from further investigation as a result of the items discussed in this subsection.

6.1.2 Solid Rocket Motors

Solid propellant motors display performance parameters which typically fall between the values associated with monopropellant hydrazine and the Earth-storable bipropellants. Solids have been used extensively in the past, primarily for thrust augmentation of the first stage of expendable launch vehicles, upper stages, and spacecraft apogee kick motors. Two factors tend to favor solids for these applications, even though the I_{sp} is approximately in the middle of the chemical propulsion system range. The first is an expended mass fraction that is substantially larger than can be achieved with liquid propellant stages. An increase in performance results since less nonimpulsive mass is being carried. A second, and perhaps the most important, factor in favor of solids is their low engineering development and recurring cost. In the past 2 years, this characteristic has led to the selection of solid rocket motors for use on the Interim Upper Stage (IUS) and the Spinning Solid Upper Stage (SSUS), which are under development for use with the STS.

The missions under consideration in this study in general require multiple burns of moderate magnitude. Deploy-only missions originating from the Shuttle parking orbit will, for the most part, require two burns to achieve the desired orbital parameters. The number of burns is further increased when the ground refurbishment and on-orbit servicing missions are considered; these missions need at least four and six burns, respectively. Unlike a liquid stage, which can be stopped by closing the propellant valves, thrust termination of a solid propellant motor can be accomplished only by introducing a chemical agent to extinguish the burning propellant or by releasing the chamber pressure which is required to sustain combustion.

Several solid rocket motor contractors have investigated two-burn motors, but none have been tested on actual space flights. A major disadvantage associated with solids for the MMS application is the lack of flexibility inherent with currently operational motors having only a one-burn capability. In view of this constraint, a large number of solids (as many as six or more for a servicing flight) would have to be carried, with a resulting decrease in propulsion system reliability due solely to the number of components that must function properly. Even if dual-burn solids could be developed within reasonable cost and time restraints, the operational flexibility of these motors would be questionable, since the quench and reignition systems must be designed to function over a narrow range of consumed propellant values.

Solid propellant motors cannot be offloaded to any propellant value, as can usually be done with a liquid stage. The ability to offload a solid is fixed at the time of igniter design since this component is required to ensure sufficient impingement of hot gases on the propellant surface to cause ignition. The igniter system must also be able to generate enough pressure to establish and maintain the combustion process. Offloads in excess of 25 percent are usually difficult to achieve without major modifications to the motor. A motor request for a non-standard offload would necessitate at least one test firing to verify the motor characteristics at this propellant loading, with a resulting increase in cost to the perspective user. It is possible to use energy management to remove the need for an exact propellant value, but this, in many cases, implies making an unwanted plane change at perigee which must be compensated for during the apogee burn. Also inherent in the procedure for wasting excess energy is the need for a very precise method of determining spacecraft attitude.

Most solid rocket motors are not qualified for extended operations in space. Several items limit their storability, including outgassing of volatile propellant constituents and propellant cracking due to unsymmetrical heating of the case. The outgassing problem can be solved by installing a nozzle closure to maintain the inside of the motor at atmospheric pressure. This solution, however, introduces a new component which can cause failure of the propulsion system. Propellant cracking can be eliminated by slow rotation of the motor to evenly distribute the thermal loads. Most MMS-type payloads, though, have indicated a preference for three-axis stabilization, which is incompatible with the mode of operation just discussed.

Thrust levels associated with solid propellant motors are high compared to other propulsion technologies. It is not uncommon for spacecraft operating in conjunction with solid motors to experience accelerations of between 5 and 13 g. Thrust values of this magnitude are unacceptable if the mission planner wishes to have spacecraft appendages such as solar arrays deployed during operation of the primary propulsion system. Three-axis stabilization of the combined spacecraft/solid motor would be difficult with the small monopropellant hydrazine thrusters normally used for attitude control because of the large torques created by the high thrust level of the solid. Spin stabilization could be used to counteract this effect, but as previously mentioned, this does not appear to be a viable alternative for the MMS application. Another solution to the control problem would be the installation of a thrust vector control system on the solid rocket motor. Such a system is quite effective on motors of the size used for the IUS, but would dramatically degrade the performance of the size of motor needed for most MMS missions due to the added inert mass.

Probably the single most damaging result for the potential use of solids pertains to packaging the system for Shuttle launch. As the result of announced STS pricing policy[(6-1)*], it is desirable to reduce the overall length of the payload, which translates into reduction of propulsion module length, since spacecraft and MMS bus lengths are likely to be fixed quantities. The large number of solids required for refurbishment and servicing missions would necessitate clustering to reduce stage length. The high thrust levels previously mentioned, coupled with the moment arms of clustered motors, would make three-axis control virtually impossible. The only other alternative would be to mount the solids in tandem. Since Shuttle launch charges dominate program costs, tandem mounting would eliminate rapidly any cost advantage solids might have in terms of development and recurring cost.

The net effect of all of the items discussed in this subsection was to eliminate solid rocket motors as a possible propulsion module that could satisfy the majority of MMS propulsion applications.

*References, denoted by superscript numbers, are at the end of the section. (Subsection 6.4).

6.1.3 Ion Propulsion

Ion propulsion exhibits the highest performance of any of the propulsion technologies considered in this study. With a specific impulse equal to or exceeding 3000 sec, the ion drive system has an I_{sp} higher than the chemical propulsion systems by approximately an order of magnitude. A propulsive stage employing ion technology would thus require the least amount of propellant. Achievement of the I_{sp} values mentioned above requires substantial amounts of electrical power (at 3 Kw per 30-cm thruster), and the typical 30-cm ion thruster is capable of thrust levels of only about 0.13 N (0.03 lb_F). The low thrust inherent with ion drive systems increases flight times for most missions and necessitates large dedicated solar arrays to power the propulsion subsystem. The use of a combination of Hughes 30-cm ion thrusters was investigated for the MMS primary propulsion application.

Typically, the most promising missions for low-thrust applications are the more demanding missions with velocity requirements in excess of 1 km/sec. These missions include some of the explorer series, Stormsat, and the return and servicing of Sun-synchronous missions.

The requirements of the atmospheric explorers are not well suited for low-thrust applications. Approximately half of the total velocity requirements are for on-orbit maneuvers to periodically raise the spacecraft orbit above the Earth's atmosphere. Low-thrust propulsion would have difficulty raising the orbit properly when the atmospheric drag at perigee could be much greater than the thrust of the ion system. Additionally, the primary purpose of the atmospheric explorer series is to measure the drag of the atmosphere. This goal would be hampered by a satellite propulsion system that must thrust continuously, with uncertainties in determining the actual thrust. Several of the explorer missions need orbital inclinations not directly achievable by the Shuttle. For missions such as these, which involve plane changes in excess of several degrees, it is unlikely that the spacecraft would use ion propulsion due to the excessive flight times which result. Other explorer series missions have as a primary goal the measurement of various properties associated with the radiation belts. It is these very same belts that critically degrade the solar panels providing power for the ion thrust system. Low-thrust propulsion, in view of the technical problems just discussed, was deemed insufficient for the explorer series of missions.

As outlined in the opening pages of Subsection 3.2, previous studies have shown that the delivery of small to medium-size satellites to geosynchronous orbit is not cost effective with ion propulsion. Thus, for deploy-only geosynchronous missions such as Stormsat, the normal use of solids appears appropriate. The only mission categories left are the retrieval and servicing of the Sun-synchronous payloads launched from WTR.

On the basis of the analysis conducted (see Subsection 3.2.1), it appears that the following comparisons can be made between chemical and ion propulsion systems for Sun-synchronous missions:

(1) Low-thrust mission times on the return to Shuttle trajectories are comparable to those of the chemical systems.

(2) Return to the operational orbit after servicing by the Shuttle when a shift in longitude of nodes has occurred is about the same (or slightly longer) for ion systems as for chemical stages.

(3) Mission time on the initial delivery leg is significantly longer for ion propulsion.

(4) Ion systems have less flexibility in reacting to Shuttle launch delays.

(5) Propellant mass requirements are significantly less for ion propulsion than for chemical systems.

The preceding five items show that ion propulsion is technically competitive with chemical systems such as hydrazine and Earth-storable bipropellants. Consideration of the recurring costs for the ion systems eliminates this technology from active consideration. The recurring cost associated with an ion module employing two 30-cm ion engines has been estimated at $10.2M in 1977.5 dollars (see Section 4). This figure can be compared with the total charges of 3.5M for hardware plus launch for a six-tank modified SPS-II design (the largest hydrazine module required). Even if the ion module had zero length, the total cost would be approximately three times that of hydrazine. Since the ion systems lack technical superiority to chemical propulsion for the retrieval and servicing of Sun-synchronous missions, there appears to be no viable means to recover the additional cost connected with these systems. Thus, it is concluded that propulsion modules containing 30-cm ion engines would not be cost effective for the MMS primary propulsion application.

6.1.4 Hydrazine/Bipropellants

Monopropellant hydrazine and Earth-storable bipropellants (i.e., N_2O_4 and MMH) emerged very early in this study as the most operationally feasible and cost-effective alternatives for the MMS propulsion system(s). Hydrazine has been selected for the SPS-I module being developed in conjunction with the expendable launch vehicle delivery of the MMS-based Solar Maximum and Landsat missions. Rockwell has also conducted a preliminary design analysis of a SPS-II propulsion system which could be used for Shuttle-launched Landsat missions.[(6-2)] With the elimination of the other technologies, the primary thrust of this study was concerned with analysis of the propulsion requirements for a much broader mission model (than only SMM and Landsat) to determine whether hydrazine or the higher performance bipropellants would be most appropriate in an expanded mission environment.

Initial sizing estimates of hydrazine and bipropellant propulsion systems were based on the use of existing and/or proposed hardware. Included were the Rockwell SPS-I and SPS-II hydrazine systems, modified SPS-II modules with clustered Viking Orbiter 1975 tanks, and the TRW Multimission Bipropellant Propulsion System (see Subsection 3.3). It was assumed at this point that the use of existing components would result in lower overall program costs.

Total program costs were computed for deploy-only, ground refurbishment, and on-orbit servicing mission models, as shown earlier in Tables 1-1 through 1-3. Cost components included were propulsion module engineering costs (nonrecurring), recurring propulsion system costs, STS transportation charges, and MMS bus costs. Table 6-1 summarizes the total costs that resulted for each propulsion technology. From the information in this table, it can be seen that neither propulsion technology demonstrates a clear cost advantage. In fact, the costs for these systems should be viewed as roughly equivalent in light of uncertainties in component lengths and possible roundoff errors in the calculations.

A decision based on non-cost considerations will likely result in the selection of hydrazine, as a result of its reduced system complexity and more favorable safety characteristics (N_2O_4 and MMH are hypergolic).

TABLE 6-1. COMPARISON OF HYDRAZINE AND BIPROPELLANT PROGRAM COSTS (a)

Technology	Total Program Cost, M$ Deploy Only	Ground Refurbishment	Servicing
Hydrazine	525.2	717.5	512.4
Bipropellants	532.5	724.5	530.3

(a) See Subsection 5.1 for cost breakdown.

During the course of the above investigation, it was determined that Shuttle transportation charges dominate the total program costs for the mission models studied. The current STS pricing policy further dictates that all of the payloads would be charged for a launch based on the load factor associated with total payload length. These observations indicate the possibility of decreasing program costs through reduction of the propulsion module length. A preliminary analysis was undertaken to redesign the hydrazine and bipropellant propulsion modules (Subsection 5.2), while noting the effect on overall program cost.

Calculations showed that the propulsion requirements of all three mission models could be met by careful design of a single hydrazine and a single, common-bulkhead bipropellant tank. These tanks would be used alone for the lower energy missions and would be grouped in three-tank clusters to fulfill the propulsion requirements of the more demanding missions. It was assumed throughout this analysis that the SPS-I module would be used for the HE-07A and HE-27A missions, which have extremely small propulsion requirements. For the reconfigured hydrazine modules, a propellant capacity of 530 kg was selected. Corresponding propellant load for the bipropellant system was 400 kg. By using oblate spheroid tanks with a diameter-to-height ratio of 2:1, it was possible to achieve the necessary volumes while substantially reducing propulsion system length. Due to diameter restraints connected with operational aspects of the MMS retention system, cylindrical sections were required in both the hydrazine and the bipropellant tanks. The completed hydrazine tank design has a diameter of 1.25 m and an overall height of 0.86 m. Measurements of the Earth-storable bipropellant tank include a 0.88-m diameter and a total height of 0.73 m. A common-bulkhead design was selected for the bipropellant configuration to alleviate c.g. problems which can result from separated propellant tanks (see Figures 5-1 and 5-4).

Table 6-2 summarizes the total costs associated with the reconfigured propulsion systems by mission model. On the basis of these data, it can be seen that hydrazine displays a slightly greater cost advantage due to its lower development cost. A brief analysis of the cost figures with discounting was made to identify any possible reversals in the apparent cost rankings. Results of this investigation also led to the conclusion that hydrazine is the lower cost propulsion alternative. Coupled with the previously mentioned operational and safety factors in favor of hydrazine, there appears to be no requirement for bipropellants as an MMS propulsion module.

TABLE 6-2. COMPARISON OF NEW HYDRAZINE AND BIPROPELLANT COSTS[a]

	Total Program Cost, M$		
Technology	Deploy Only	Ground Refurbishment	Servicing
Hydrazine	491.8	683.0	496.3
Bipropellants	505.4	701.4	512.7

(a) See Subsection 5.3 for cost breakdown.

The one remaining topic pertains to the cost effectiveness of redesigning the hydrazine propulsion systems. Comparison of Tables 6-1 and 6-2 shows that the reconfigured systems result in savings of 5.4 percent, 4.8 percent, and 3.1 percent, respectively, for the deploy-only, ground refurbishment, and on-orbit servicing mission models. The assumption that spacecraft and MMS bus lengths are fixed forces all potential savings to be derived from reduction of the propulsion module length. Redesign of the hydrazine systems results in a 45 percent reduction in the STS transportation charges associated with the propulsion module. It is also likely that payload designers will react to the Shuttle pricing policy by repackaging the spacecraft components mounted on top of the MMS bus to further reduce length unless this goal adversely impacts spacecraft development costs. In this environment, the redesigned hydrazine systems would play a more dramatic role in reducing total program costs. In view of the above discussion, reconfiguration of the hydrazine systems by developing a new tank is considered a cost-effective alternative.

6.2 Special Case Mission Propulsion Requirements

The four propulsion applications other than the normal requirements of the MMS missions discussed in Subsection 6.1 are (1) drag makeup requirements, (2) Sun-synchronous orbit change, (3) geosynchronous satellite stationkeeping, and (4) contingency return of a geosynchronous spacecraft. Brief descriptions of these missions are as follows:

(1) Drag makeup mission: a Scout to Delta-size payload launched from the Shuttle, requiring a long lifetime at an altitude of 125 to 400 km. No propulsion is required to establish the initial orbit, although drag may be used to achieve the desired altitude for the lower altitudes considered.

(2) Sun-synchronous orbit change: a Sun-synchronous spacecraft views the Earth at the same Sun lighting conditions each day. To view the Earth at one lighting condition for a period of time and then change the viewing conditions requires a change in the longitude of ascending node. This can be done with a large single impulse or by a transfer to an intermediate drift orbit where the change in precession rate will achieve the desired change in viewing conditions. A low-thrust system could also be used to vary the precession rate continuously and achieve the desired change in viewing conditions. These orbit change requirements are added to the nominal delivery and return/servicing requirements.

(3) Geosynchronous satellite stationkeeping: a SSUS-D or SSUS-A class spacecraft has several propulsion requirements after the apogee kick burn. These include initial station

acquisition, North-South stationkeeping, East-West stationkeeping, and attitude control.

(4) Contingency return of a geosynchronous spacecraft: a geosynchronous spacecraft with a stationkeeping system which could return the spacecraft to Shuttle orbit on a contingency basis.

These four missions cover a wide range of thrust levels, propellant requirements, etc. Not all the technologies considered in this study were applicable to each of these special case propulsion applications. Table 6-3 shows which technologies were applicable for each of these missions and which technologies have advantages for each mission as defined. Although solids and LOX/LH_2 are technologies included in the study, they were not applicable to any of these four propulsion applications. The primary reasons are the requirement of multiple thrustings of unknown duration and the long-term operational requirements. The four remaining technologies are considered in the next four subsections.

TABLE 6-3. TECHNOLOGIES APPLICABLE FOR SPECIAL CASE PROPULSION MISSIONS[a]

Technology	Drag Makeup	Sun-Synchronous Orbit Change	Geosynchronous Stationkeeping	Geosynchronous Return
Solids	NA	NA	NA	NA
Catalytic hydrazine	A	A	B	NA
Electrothermal hydrazine	A	NA	A	NA
Bipropellants	NA	A	A	NA
LO_2/LH_2	NA	NA	NA	NA
Ion	A	C	A	B

(a) NA - not applicable/not considered.
A - has definite advantages over other technologies.
B - baseline case or current usage.
C - considered in trade-off.

The missions considered here for additional propulsion requirements are not directly related to any mission projections. Thus, the need for or advantages of a given technology on a specific mission does not necessarily imply that technology should be developed. For additional details on the trajectory analysis/system sizing see Subsection 3.2, and for details on the cost trades see Subsection 5.6.

6.2.1 Ion Propulsion

In this study two ion thruster sizes were considered: the 8-cm and 30-cm thrusters. Although both of these thrusters were considered to operate at approximately 3000 sec I_{sp}, their thrust levels and other characteristics are so different that they will be discussed separately for the special case missions. The results here do not imply a role as main MMS propulsion.

The thrust levels of the 8-cm ion thruster are so low, even compared to those of the 30-cm thruster, that the application of this thruster is limited to the drag makeup and geosynchronous stationkeeping missions. In the drag makeup mission there are specific altitudes at which the 8-cm ion engine is less costly than the other systems studied. Two specific spacecraft were considered: a Scout class spacecraft and a Delta class spacecraft. The most significant spacecraft parameter is the cross-sectional area: 0.45 m^2 for Scout and 3.75 m^2 for Delta. For the Scout-size spacecraft the drag is relatively low; this, coupled with the altitudes at which the Scout system operates (between approximately 180 and 250 km for a 7-year mission), leads to costs which are less than those for other systems considered. The sensitivity with respect to launch year is minimal. For shorter missions, the altitudes at which the 8-cm ion system is less costly than the other systems are confined to the lower end of the 180 to 250-km range. The largest cost savings of the ion systems over the hydrazine systems occurs at the lowest altitudes at which the ion systems can be used. The savings can be as large as \$0.7M at these lower altitudes; however, at these altitudes there is also the greatest risk that an unexpectedly high solar activity level could produce a drag which is too large for the propulsion system to make up, thus causing the spacecraft to reenter prematurely. For the larger Delta-size spacecraft, the ion system is always more costly than the hydrazine systems, although the costs are close to those of the hydrazine system at altitudes near 300 km. The range where the ion system is less costly than the hydrazine system becomes smaller and shifts to higher altitudes as the cross-sectional area grows from a Scout-size spacecraft to a Delta-size spacecraft.

There are some potential drawbacks to the ion system for this application which must be mentioned. The ion system requires sunlight for thrusting, or a set of batteries which can be recharged. If the batteries are not provided, the system may still be able to provide sufficient thrust on average, but all the thrusting would occur over one-half the trajectory, which could result in an elliptic orbit. The other option of adding batteries would increase the cost. Another difficulty occurs in the planning process. In the early phases of mission definition, the ion system could be the most cost-effective option, but as the mission evolves in size, altitude, etc., the ion system may no longer be the best choice.

The second application for the 8-cm ion system is the geosynchronous stationkeeping mission. The mission requirements also include the initial station placement. The 8-cm ion system was considered both as a system to perform all the required tasks or as a subsystem in conjunction with hydrazine. Although the 8-cm ion system used for all the mission requirements give the largest spacecraft net mass increases for both SSUS-D and SSUS-A class spacecraft, this option is not considered as a strong candidate for 8-cm thrusters since the initial spacecraft station acquisition times would be too lengthy. The use of the 8-cm system combined with the hydrazine system on the SSUS-A class spacecraft gives the next best increase in spacecraft net mass. The increase over hydrazine alone is about 77 kg. The competition for this combined system on the SSUS-A class spacecraft comes primarly from the augmented electrothermal hydrazine, which provides only 14 kg less than the hydrazine/ion system. On the SSUS-D class payloads, the combined ion/hydrazine systems offer no advantage because of the higher dry weight of a dual system compared to any of the single systems.

Potentially, the most promising combination is to replace the solid apogee kick motor with a bipropellant system that performs the apogee kick burn and the initial station acquisition and use an 8-cm ion system for the North-South stationkeeping, attitude control, etc. The bipropellant system would be similar to a solid apogee kick motor, but its capability to shut down and restart would result in smaller errors after the apogee burn and the ability to perform the initial station acquisition in times comparable to existing systems. The ion system could then use its high specific impulse for those tasks where the low thrust levels are not a hindrance. The potential net mass increase of this system versus a solid AKM and hydrazine system is about the same as that of the solid AKM and ion system. Thus, the net mass increase of this combination over a solid AKM and catalytic hydrazine is about 60 kg for a SSUS-D class spacecraft and 140 kg for a SSUS-A class spacecraft. However, these large gains are costly, with recurring costs increasing by approximately $2.0M.

The 30-cm ion engine was considered on two of the auxiliary propulsion missions: the Sun-synchronous orbit change and the geosynchronous return mission. For the Sun-synchronous mission, its performance was equivalent to that of the chemical systems (hydrazine or bipropellants) including transfer times, but it could not compete on a cost basis. For the geosynchronous return mission, only one propulsion system was found to be feasible. This system is a combination of 8-cm and 30-cm ion engines. The spacecraft net mass, using a SSUS-A as the PKM after deducting the dry mass of the propulsion system and sufficient propellant mass for a contingency return, is 517 kg. This net mass lies between those of the SSUS-D (384 kg) and the SSUS-A (740 kg) when a hydrazine system is used for stationkeeping. The major drawback to this application is the jusification of a need or cost-saving reason for returning from geosynchronous orbit.

Summarizing, for the ion systems on the special case propulsion applications: the 8-cm ion engine was found to be attractive for small spacecraft which require drag makeup at selected altitudes and also for the geosynchronous spacecraft for the stationkeeping role, while the 30-cm engine was found to be unsuitable on any mission for which there is a real need.

6.2.2 Electrothermal Hydrazine

There are two types of electrothermal hydrazine technologies. The first is where a small power level is used to replace the need of a catalyst. The performance of this system is approximately the same as using catlytic hydrazine, the advantage being a longer thruster life. The second type is where additional power is used which improves the specific impulse by 80 to 100 sec. This second type was considered for these special case propulsion missions.

For the drag makeup missions, the augmented electrothermal hydrazine was less costly than the other systems considered for the low altitudes for both sizes of spacecraft (Scout or Delta). There are some potential problems in using the augmented electrothermal hydrazine for this application. Since power is required for operation, this would require solar arrays, and unless batteries are provided, the system would be able to thrust only on one side, which could lead to the orbit becoming elliptical. A potential technology problem is related to the thrust levels that can be achieved. For the altitudes at which the electrothermal system has an advantage it would be desirable if a thrust level of 0.4 N could be achieved. The potential savings for the augmented electrothermal hydrazine system over the catalytic hydrazine system are typically in the $0.2M to $0.6M range.

The other mission application for the augmented electrothermal hydrazine is the geosynchronous stationkeeping application. Use of the electrothermal hydrazine instead of the catalytic hydrazine gives an increase in net mass of 36 kg on a SSUS-D size spacecraft and 53 kg on a SSUS-A size spacecraft. Excluding the ion systems, these are the largest net increases of any of the systems studied. The incremental cost over hydrazine is about $0.25M, which is significantly less than the $1M plus increment required for using an ion system instead of hydrazine. The electrothermal hydrazine system is the best option considered for the SSUS-D class spacecraft, whereas on the larger SSUS-A class spacecraft there is a choice between several systems depending on how much mass increase is needed and what additional cost the spacecraft owner can afford.

6.2.3 Catalytic Hydrazine

Catalytic hydrazine was considered on all special case missions with the exception of the geosynchronous return mission. One of the primary advantages of hydrazine is the large amount of experience that exists with using hydrazine for spacecraft propulsion. For the drag makeup mission, the catalytic hydrazine system is the least costly of the systems considered for higher altitudes (above 250 km). Additionally, the system offers some potential operational flexibility over the other systems which have power requirements that may restrict operations to sunlit parts of the orbit. Since the cost advantage of the other systems is not very large at many of the lower altitudes considered, the operational flexibility and the experience with an existing technology should encourage the use of hydrazine for this application.

For the Sun-synchronous orbit change mission, hydrazine is less costly than ion systems. However, the use of bipropellants for this application should be considered. If this mission became an existing mission, the decision between hydrazine and bipropellants would have to consider the number of flights and development costs as well as the recurring costs and Shuttle charges.

Most current geosynchronous communication satellites currently use catalytic hydrazine. Although it provides less net spacecraft mass which can be used for communication equipment than other systems potentially can provide, it has two important advantages over the other systems: (1) it is the least expensive system and (2) it is designed into current production spacecraft. Therefore, as long as the communications industry can continue to use modifications of existing spacecraft without exceeding the mass capabilities of the available solids (SSUS-D and SSUS-A), there will be a role for hydrazine.

The use of catalytic hydrazine for these special missions can best be summarized by noting that the advantages of using catalytic hydrazine on geosynchronous spacecraft (low cost and past experience) apply to spacecraft propulsion in general. Thus, unless there is a need for an increase in performance (e.g., more net mass on a geosynchronous spacecraft) catalytic hydrazine should continue to be used.

6.2.4 Bipropellants

Earth-storable bipropellant systems (i.e., N_2O_4 and MMH) were considered for the Sun-synchronous orbit change mission and the geosynchronous stationkeeping mission. For the Sun-synchronous orbit change mission, a module such as that shown in Figure 5-6 could be used. As discussed under hydrazine, the decision between bipropellants and hydrazine must consider the number of flights, development costs, and other potential mission applications to determine which system is more cost effective.

The other application of bipropellants considered was with the geosynchronous spacecraft. When used as a stationkeeping system in place of hydrazine, a modest increase in net spacecraft mass is available for a small increment in recurring cost. The net mass increase is 24 kg for a SSUS-D spacecraft and 36 kg for a SSUS-A spacecraft for an additional cost of approximately $0.1M. If this increase in net mass is sufficient to satisfy some current requirement, the redesign cost associated with changing propulsion systems might be justified. It is likely that the direct change to a system with greater net mass increase which provides growth potential would be more cost effective in the long run.

If a bipropellant system is used, however, as a replacement for the solid AKM on a SSUS-A spacecraft, it allows for significant growth in net mass by replacing the stationkeeping system with an 8-cm ion system. This combination gives a potential net spacecraft mass increase of 140 kg over the current solid AKM/hydrazine combination. The use of the bipropellant apogee motor for the initial station acquisition is the key to achieving the full potential of the 8-cm ion system. Although the propulsion system recurring costs may increase by $2.5M to $3.0M, the 19 percent net mass increase may be justified when the Shuttle/SSUS-A charges (about $12M) and spacecraft costs are considered.

6.3 General Observations

In the process of performing the subtasks required for this study, several general observations were made that do not necessarily relate to any given technology. These observations are discussed in this subsection. The material is organized according to the order of the subtasks in the study plan (Subsection 1.1), and does not follow in any order of importance.

6.3.1 MMS Mission Model Observations

Prediction of future missions is continually plagued by uncertainties and change. The NASA missions are highly dependent upon the NASA budget, new starts, money needed for Shuttle development, etc. Since all of these factors are continually undergoing change, the projections of the future missions are also continually changing. At some point in this study, as in any study of this nature, the mission model must be fixed. Thus, by its nature, the mission model will be out of date at the end of the study.

The changes which take place in a mission model are divided into the following three types:

(1) Changes in flight schedules, which add or delete launches

(2) Addition of new types of missions that were not originally considered

(3) Changes in specific mission parameters such as payload mass, orbit requirements, or on-orbit velocity requirements.

In order to maintain the validity of this study, the following steps were taken to evaluate the impact of these types of changes in the mission model.

In the cost comparison of different technologies for a given mission model, the costs were calculated and compared on a per mission basis as well as totaled and compared for the entire model. Thus, it was possible to determine whether the technology that was cheaper for all missions together was less expensive on a mission by mission basis. In the cases examined in this study, the technology that was less expensive for the mission model as a whole, was no more expensive than the competing technology on a mission by mission basis (subject to cost uncertainties caused by Shuttle charge due to length uncertainties, module recurring cost estimates, etc.). From this, it can be concluded that different flight rates and/or dropping of missions from the model would not alter the conclusion as to which technology was less costly.

The impact of new missions is a difficult area to evaluate. A few special case missions for which it was felt that a different technology than hydrazine might be more cost effective were analyzed. The results of these analyses give conditions for which technologies other than hydrazine are the most cost effective or offer worthwhile advantages. While conditions have been determined which indicate that technologies other than hydrazine have a role in spacecraft propulsion, the list of potential missions treated as special cases can never be a completely exhaustive tabulation. Thus, even if no mission has been found which demonstrates the need for a particular technology, it cannot be concluded that there aren't any missions for which that particular technology would be least costly.

For any particular mission, the mission parameters (i.e, spacecraft mass, apogee, perigee, inclination, mission lifetime, etc.) usually undergo an evolution between the initial planning phases and the final mission definition. As an example, consider Landsat D/E. The first Landsat spacecraft were at an altitude of approximately 900 km. In 1975 a preliminary design of the SPS-II was published.[6-3] As the spacecraft experiments became finalized, the spacecraft mass grew. Then consideration was given to a lower altitude, 705 km, so that the next step in the design of SPS-II[6-2] indicated that the module was ideally sized for Landsat. The observation to be made is that, in practice, missions evolve in such a way that the transportation system will have sufficient capability to perform the missions. Thus, although it is desirable to be able to design modules which will not be altered by the changes in the missions, it is not necessary to consider in detail small mission changes since the missions evolve so as to conform to existing transportation capability. The exceptions to this are the major programs such as Viking or Apollo where the necessary transportation is part of the program.

6.3.2 Trajectory Analysis Observations

The trajectory analyses of MMS missions in the 1980's must consider the operational characteristics of the Shuttle. Since Shuttle IOC will not occur until 1980 at ETR and 1982 at WTR, e Shuttle operations are still in a planning stage. Two aspects that are among the prime motivations for building the Shuttle are payload sharing and retrieval/servicing. These features have not, to any large extent, been available with the expendable launch vehicles. To encourage payload sharing, a shared-flight Shuttle charge policy has evolved. The costs for retrieval/servicing can only be estimated at this time. As a result, cost policy has a strong influence on how to use the Shuttle for the users' best advantage. These policies influence the trajectory analysis. Thus, the following guideline was used in the trajectory analysis: make maximum use of payload sharing.

This presents several complications in the analysis. Currently, there is little experience in payload sharing where two (or more) payloads are put into orbit on a single launch and each payload is considered a prime payload. Shuttle operations planning is attempting to simplify the difficultie finding payloads to share a flight by defining "standard orbits". These standard orbits are currently defined as 297 km (160 nmi) altitude circular orbits at one of four standard inclinations. Although the four inclinations are not necessarily firmly chosen, they are relatively fixed. One key parameter, however, is still free to be chosen. This parameter is time of day for the launch. Due to a wide variety of constraints, different types of spacecraft require different launch times. Thus, in practice, for an MMS payload to share a flight with a non-MMS payload(s) it is necessary for these payloads to have compatible launch times. An examination of selected cases of multiple payloads being launched on the same Shuttle flight was made in this study; although some potential difficulties were indicated, the launch of multiple payloads on a single flight does not impact the trajectory significantly after orbit has been achieved (if each payload's launch constraints have not been severely violated).

The situation is significantly more involved for cases where a mission comprises launch of one or more payloads and retrieval/servicing of another. Although the Shuttle charges for retrieval and servicing are not well defined, indications are that the charges will be minimal if the retrieval or servicing can be done conveniently. There are two key words which need to be interpreted: minimal and conveniently. In this study the following interpretation was used: a retrieval/servicing could be done on a shared flight with minimal cost (significantly less than the dedicated flight price) if the spacecraft is in the Shuttle orbit. Thus, the trajectory analysis was done to determine the propulsion requirements necessary to return to a standard Shuttle orbit where the launch time is determined by the payload being launched. If this became too difficult, then the next alternative assumed was a dedicated flight for retrieval/servicing.

For ETR operations the restrictions are not very severe. It was found that the retrieval or servicing of a payload on a flight whose launch window is determined by the constraints of a typical communication satellite could be done with no more propulsion required than if the spacecraft in orbit chose the time of day for the launch and returned to the standard Shuttle orbit if the launch occurred during the proper 3 to 4 days over a range of approximately 20 days. These findings, together with current estimates of flight rates at ETR, indicate that although there are problems to be solved, retrieval/servicing is feasible from a propulsion standpoint.

The situation at WTR is significantly different. The first difficulty is that the number of flights available for sharing is projected to range between 3 and 5 per year from 1983 through 1991. Additionally, since these flights are launching Sun-synchronous spacecraft, and the spacecraft which are to be retrieved/serviced are also Sun-synchronous spacecraft, the longitude of nodes which is established by launch time must agree for the two spacecraft. The difficulty is that different spacecraft require different viewing conditions (which implies different longitude of nodes). This has definite implications on the propulsion requirements on-board the spacecraft to enable return to the Shuttle. The details are discussed in Subsection 3.1. Since the propulsion requirements are based upon current estimates of Shuttle operations at WTR, flight rates at WTR, and Sun-synchronous mission requirements, these propulsion requirements could change over the next few years. It should be noted that the Sun-synchronous mission requirements for retrieval/servicing are the most demanding of all MMS missions (except geosynchronous) in terms of total velocity increments. Should these become more severe, there could be a strong case for using bipropellants. If the requirements become too severe, mission planners may be driven from the idea of retrieval/servicing. At any rate, as discussed previously, mission planners tend to modify missions to use available propulsion.

One final observation will be made on the sensitivity of performance requirements to Shuttle operations. At this point in the development of Shuttle, the operational characteristics of the Shuttle are still evolving. For example, the 297 km (160 nmi) standard orbit may be replaced by a 278 km (150 nmi) standard orbit. While this would have an impact on the propellant requirements of the propulsion module for a given set of spacecraft, the mission definitions are sufficiently flexible at this time that the mission planners could adjust to minor changes in Shuttle performance.

6.3.3 Program Evaluation Observations

In the process of evaluating certain families of propulsion modules for a set of missions, there were cases where a particular mission could either be done with a dedicated Shuttle flight or with the development of a larger propulsion module. This typically occurred for a servicing mission. On the surface it appears to be a comparison between development of a larger module, the recurring cost of the larger module, and a shared servicing flight versus the recurring cost of a previously developed module and a dedicated servicing flight. Based on the costs in Section 4, the development of the new module is the lower cost approach. However, there are other considerations.

The development of a propulsion module would have to begin before all the mission requirements are finalized. The propulsion requirements for Sun-synchronous missions are somewhat random in nature in that they are based upon expected servicing times. The mission planner may decide he can accept a longer servicing time or he may decide that servicing will be done on an a priori determined Shuttle flight. Both of these options can lower the propulsion requirements so that an existing module could be used. Thus, the ground rule was established that a propulsion module would not be developed for a single flight.

6.4 References

(6-1) NASA Management Instruction 8610, Subject: Reimbursement for Shuttle Services Provided to Non-U.S. Government Users, January 1977.

(6-2) Landsat/MMS Propulsion Module Design Study, Task 4.4 Final Report, Rockwell International Space Division, September 24, 1976.

(6-3) Low Cost Modular Spacecraft Description, X-700-75-140, NASA Goddard Space Flight Center, May 1975.

7.0 CONCLUSIONS AND RECOMMENDATIONS

Based upon the analyses of the MMS missions performed in this study, the following conclusions are made on the applicability of the various propulsion technologies as a main propulsion module for MMS:

(1) LO_2/LH_2 does not have the long-term capability required for retrieval/servicing missions

(2) Solid rocket motors lack flexibility required of MMS missions

(3) Ion propulsion systems recurring costs are too large for ion systems to be competitive

(4) Earth storable bipropellants are feasible as an MMS propulsion module, with costs only slightly larger than hydrazine modules

(5) The lowest overall transportation cost is achieved with a family of hydrazine modules made up by clustering different numbers of a single tank design chosen to minimize Shuttle length.

From the analyses of four special case missions which are not necessarily MMS missions, the following conclusions are made:

(1) For main propulsion on a Sun-synchronous mission with greatly increased propulsion requirements, hydrazine or Earth storable bipropellants remain the possible choices with no role for an ion module

(2) Using a 30-cm ion system, a geosynchronous spacecraft on a SSUS-A can have a contingency return capability with a net mass about halfway between SSUS-D and SSUS-A capability

(3) The augmented electrothermal hydrazine and 8-cm ion engines have potential cost-saving applications on drag makeup satellites and can result in a net mass increase on geosynchronous spacecraft when used for North-South stationkeeping

(4) To achieve the maximum benefit from the 8-cm ion engine on a geosynchronous spacecraft it is necessary to use a bipropellant AKM instead of a solid AKM.

Based upon the analyses performed in this study and the conclusions obtained, the following recommendations are made:

(1) Consideration should be given to developing a new hydrazine tank that minimizes the length in the Shuttle bay and can be clustered to perform all the MMS missions which require more liquid propulsion than available with SPS-I. The single tank should contain slightly more hydrazine than required for Landsat D/E. The potential savings on Landsat alone could recover the development cost.

(2) The augmented electrothermal hydrazine and the 8-cm ion engines have potential applications (geosynchronous spacecraft stationkeeping and drag makeup); thus, research in these technologies should continue.

(3) With the continuing evolution of Shuttle operational planning and the impact the WTR operations potentially have on MMS propulsion module requirements, a study should be done on Sun-synchronous missions; the study should consider both Shuttle operations and how missions may evolve to make best use of the Shuttle so that the propulsion requirements may be further defined.

(4) Bipropellants should not be eliminated from further consideration since increases in WTR mission requirements may result in bipropellants being more cost effective than hydrazine; additionally, for SSUS-A class geosynchronous spacecraft, it is necessary to replace the solid AKM with a bipropellant to achieve the maximum benefits available from using the 8-cm ion engine for stationkeeping.

APPENDIX A

COMMUNICATIONS SATELLITE LAUNCH WINDOWS

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COMMUNICATIONS SATELLITE LAUNCH WINDOWS

This appendix describes work done under a different contract: NASW-3001, Development of Civil Spacecraft Requirements for Spinning Solid Upper Stages. This work was done for Marshall Space Flight Center and the final report was dated February 28, 1977. The results of this work are applicable to the analysis of shared flight servicing. They are reproduced here for convenience.

Launch Window Analysis

Several constraints resulting from spacecraft design, orbit geometry requirements, and spacecraft operational procedures determine acceptable time periods during which a spacecraft can be released from the Orbiter cargo bay for transfer orbit injection. If more than one payload is to be released on the same Shuttle flight, the launch windows of these spacecraft must be compatible.

In this analysis, INTELSAT V, TDRS (TRW), COMSTAR FOLLOW-ON, AEROMARISAT, RCA FOLLOW-ON, FOREIGN COMMUNICATIONS, GOES, AND TDRS (GE) were considered for potential multiple payload Shuttle flights.

The constraints considered by spacecraft manufacturers in deriving launch windows, and the launch windows for the spacecraft of this study are discussed in the following paragraphs.

Types of Constraints

Launch window constraints express the design and operational requirements of spacecraft from the SSUS perigee burn at transfer orbit injection through the final geosynchronous orbit. Occasionally some spacecraft may have additional requirements prior to the perigee burn after separation and/or while the spacecraft is still in the Shuttle cargo.

One of the most significant operational requirements of a spacecraft is the location and duration of sunlight. Various horizon and sun sensors require the Sun to be in specific regions for proper attitude determination. Solar panels must have a reasonable incidence of sunlight to maintain a power supply. Thermal constraints will often restrict or prohibit the direct exposure of some parts of a spacecraft to sunlight. To specify these requirements, acceptable values or times are given for the solar aspect angle, the Earth-spacecraft-Sun angle, the orbit normal Sun angle, and the occurrence and duration of eclipses during the transfer orbit.

The solar aspect angle is the angle between the spacecraft spin axis and the Sun, as shown in Figure A-1. This angle is usually most important at transfer orbit injection and/or at the transfer orbit apogee. Figure A-2 illustrates the Earth-spacecraft-Sun angle, and Figure A-3 shows the orbit normal Sun angle.

Two operational philosophies exist regarding the final inclination of the synchronous orbit. One type of spacecraft goes to a zero-degree inclination and maintains that inclination, while the other type has a small positive inclination and allows the inclination and the ascending node location to wander due to the influences of the sun and the moon. The latter type generally requires that the synchronous orbit inclination remain less than some upper value for the spacecraft lifetime. This requirement can be satisfied by an appropriate initial ascending node longitude. For this reason, some spacecraft require specific ranges on the initial synchronous orbit ascending node longitude. Figure A-4 illustrates how TDRS (TRW) derived a 255 to 360-deg ascending node longitude requirement to maintain an inclination less than 7 deg for 10 years.

Some spacecraft also have ground tracking station requirements for data transmission. These effects have not been included in this evaluation.

Table A-1 lists the specific constraint requirements for each spacecraft from the appropriate manufacturer requirements documents.

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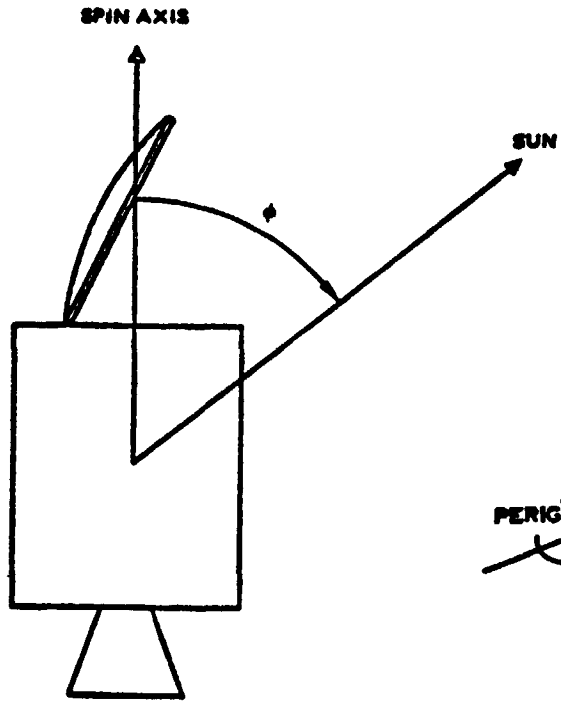

FIGURE A-1. SOLAR ASPECT ANGLE

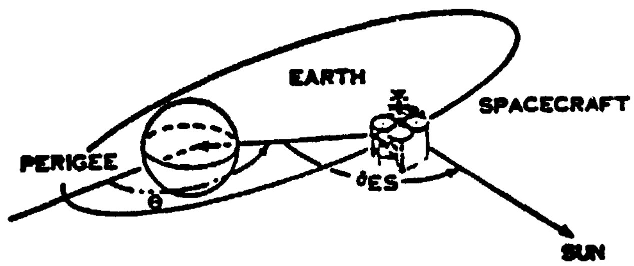

FIGURE A-2. EARTH-SPACECRAFT SUN ANGLE

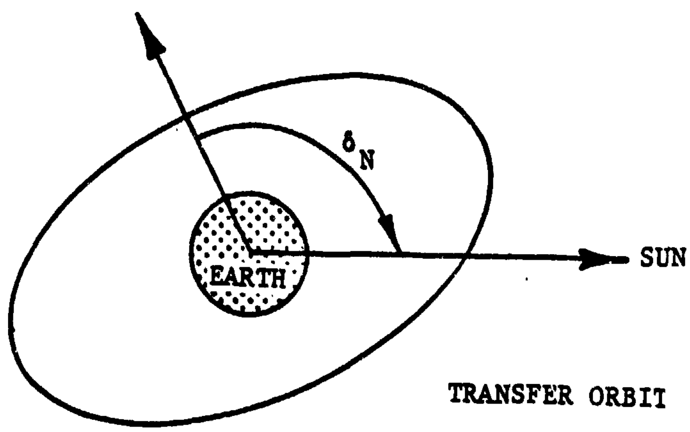

FIGURE A-3. ORBIT NORMAL SUN ANGLE

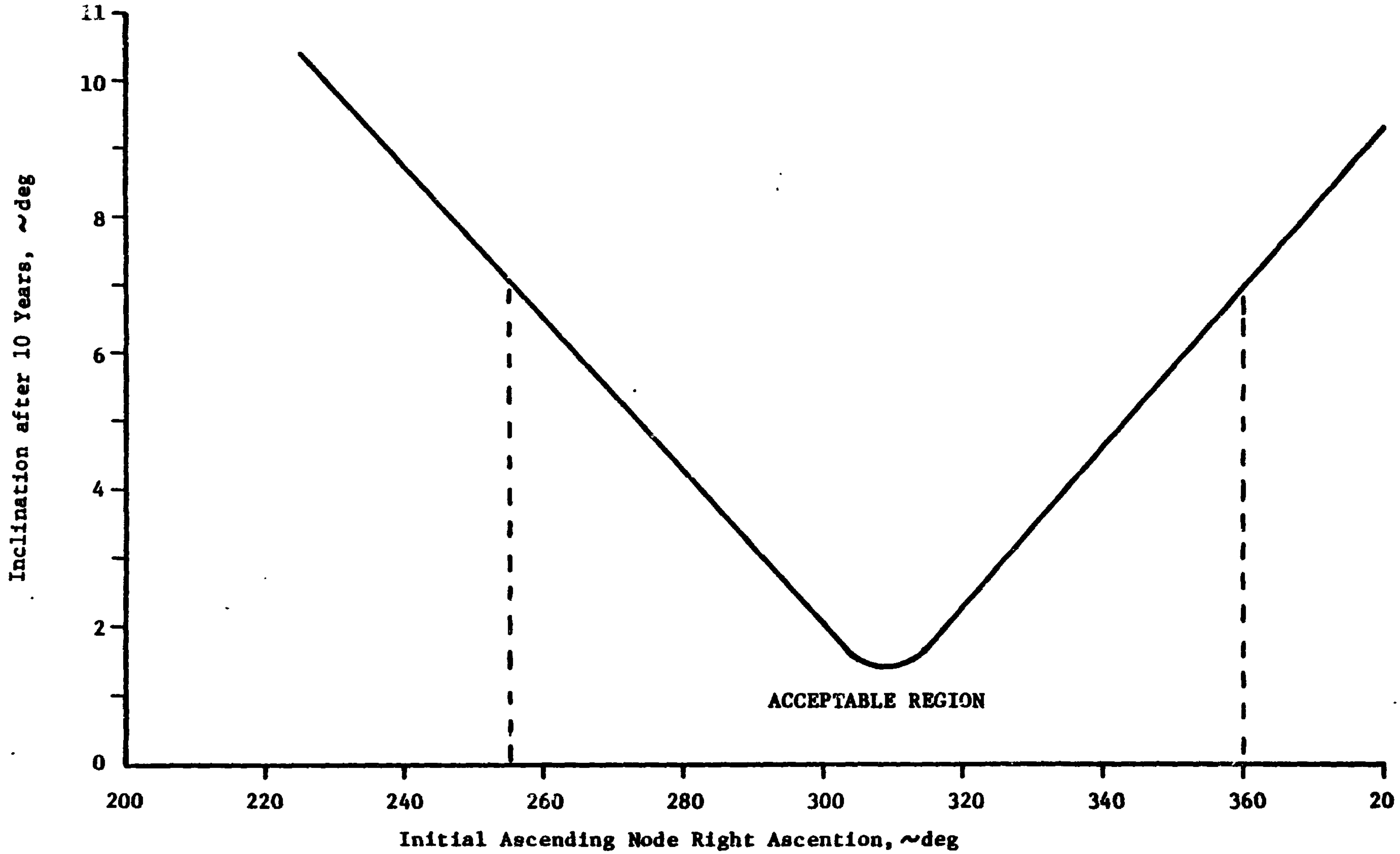

FIGURE A-4. TDRS (TRW) ORBIT INCLINATION AFTER 10 YEARS; INITIAL INCLINATION = 7 DEG

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TABLE A-1. LAUNCH WINDOW CONSTRAINTS

DATA ITEM	INTELSAT V	TDRS (TRW)	COMSTAR FOLLOW-ON	AEROMARISAT	RCA FOLLOW-ON	FOREIGN COMMUNICATIONS	GOES	TDRS (GE)
Transfer Orbit								
Apogee Altitude (nm)	20014	19324	19324	19702	19324	19324	19933	19324
Perigee Altitude (nm)	160	160	160	160	160	160	160	160
Inclination (deg)	26.6	26.6	28.5	27.44	26.0	28.5	23.8	27.0
Launch Window Constraints								
Earth-SC-Sun Angle δ_{ES}	None	None	None	None	$10 \le \delta_{ES} \le 170$ at apogee	None	None	$30 \le \delta_{ES} \le 150$ for true anomalies 160 to 180°
PKM Solar Aspect Angle, δ_P	None	None	$65 \le \delta_P \le 115$	$55 \le \delta_P \le 125$	$55 \le \delta_P \le 125$	$65 \le \delta_P \le 104$	$60 \le \delta_P \le 120$	$40 \le \delta_P \le 140$
AKM Solar Aspect Angle, δ_A	$70 \le \delta_A \le 110$	$130 \le \delta_A \le 180$	$65 \le \delta_A \le 115$	$55 \le \delta_A \le 125$	$70 \le \delta_A \le 120$	$65 \le \delta_A \le 104$	$60 \le \delta_A \le 120$	$40 \le \delta_A \le 140$
Eclipse Duration (hr)	<.50	No eclipse 15 minutes after perigee thru apogee	<1.17	<.583 and no eclipse during 70-290° true anomaly	<.75 Apogee is on the Sun-lit side of the Earth	<1.17	<.50	None
Synchronous Orbit Ascending Node Longitude, Ω	None	$255 \le \Omega \le 360$	None	$204 \le \Omega \le 384$	None	None	None	$250 \le \Omega \le 380$
Orbit Normal Sun Angle, δ_N	None	None	None	None	$55 \le \delta_N \le 125$	None	None	None

Note: Shuttle parking orbit is 160 nm circular and 28.5° inclination.

Spacecraft Launch Windows

A launch window represents the time periods during which a spacecraft could be released from the Shuttle cargo bay to proceed with a geosynchronous transfer orbit injection with all constraints being satisfied. Launch windows are generally larger on the Shuttle than on an ELV since transfer orbit injection opportunities occur at both the descending and the ascending nodes of the parking orbit.

Figures A-5 through A-12 are the launch windows for each spacecraft in Table A-1. Each window has been derived by entering the appropriate transfer orbit parameters and constraint values from Table A-1 into the Interactive Graphics Orbit Selection (IGOS) computer program developed at Battelle. The shaded regions indicate the periods during which the spacecraft could be released from the Shuttle. The descending node injection opportunities are shaded with lines, while the dotted regions represent the additional opportunities available at an ascending node injection which are not acceptable for the descending node. An ascending node injection could also occur at some of the descending node opportunities.

Examination of Figures A-5 through A-12 indicates:

- INTELSAT V, COMSTAR FOLLOW-ON, RCA FOLLOW-ON, and GOES each have two launch opportunities a day all year.
- AEROMARISAT has two launch opportunities a day, except for January 1 to February 10, May 22 to August 15, and November 15 to December 31, when there is only one opportunity a day.
- TDRS (GE) has two launch opportunities a day all year, except for March 7 to April 9, June 1 to June 15, September 8 to October 7, and November 26 to December 15, when there is only one launch opportunity a day.
- TDRS (TRW) has one launch opportunity a day all year.

Note that the most restricted launch windows are those for spacecraft with a synchronous orbit ascending node requirement.

The implications of these results for shared flight servicing are discussed in the main body of this report under trajectory analysis.

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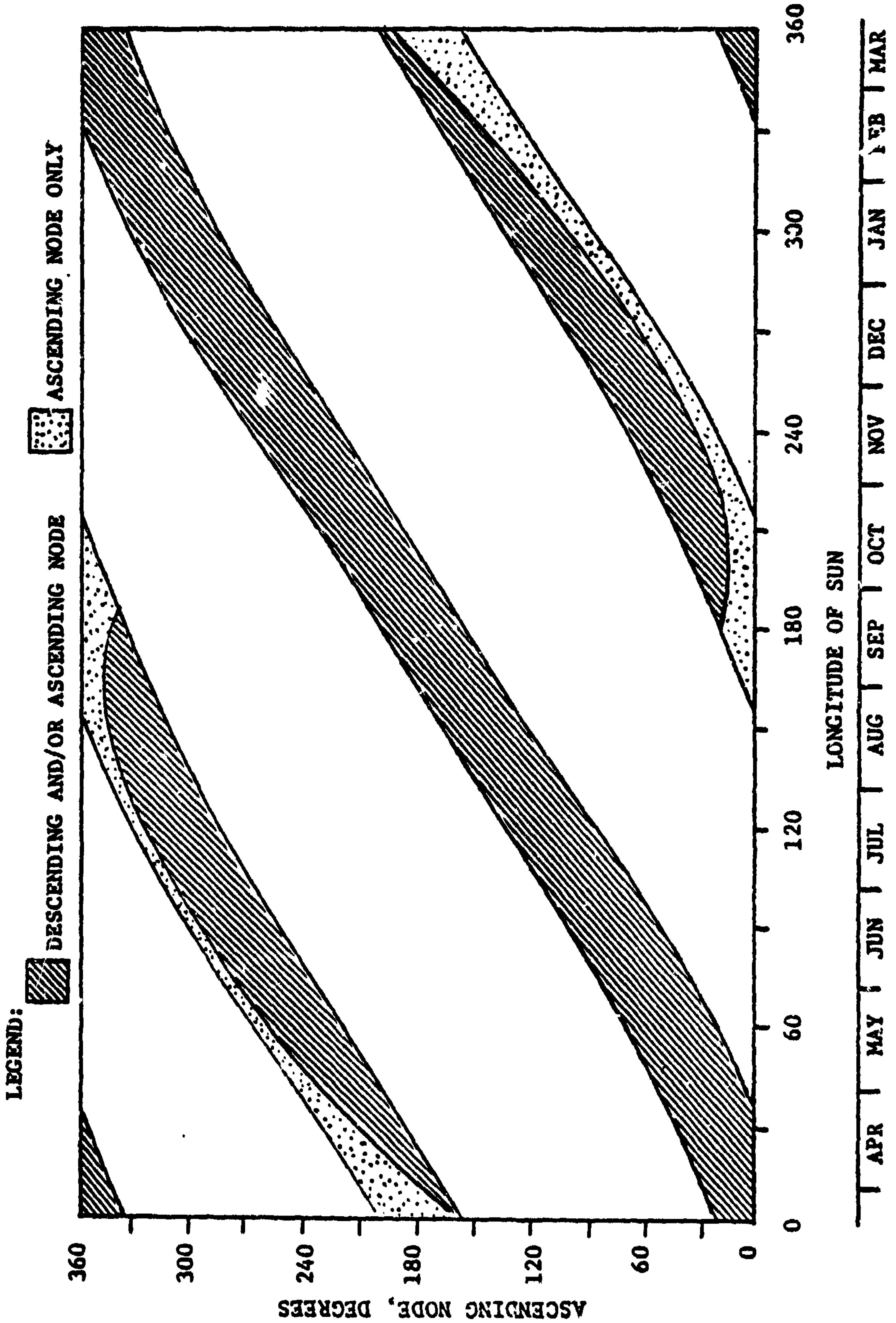

FIGURE A-5. INTELSAT V LAUNCH WINDOW

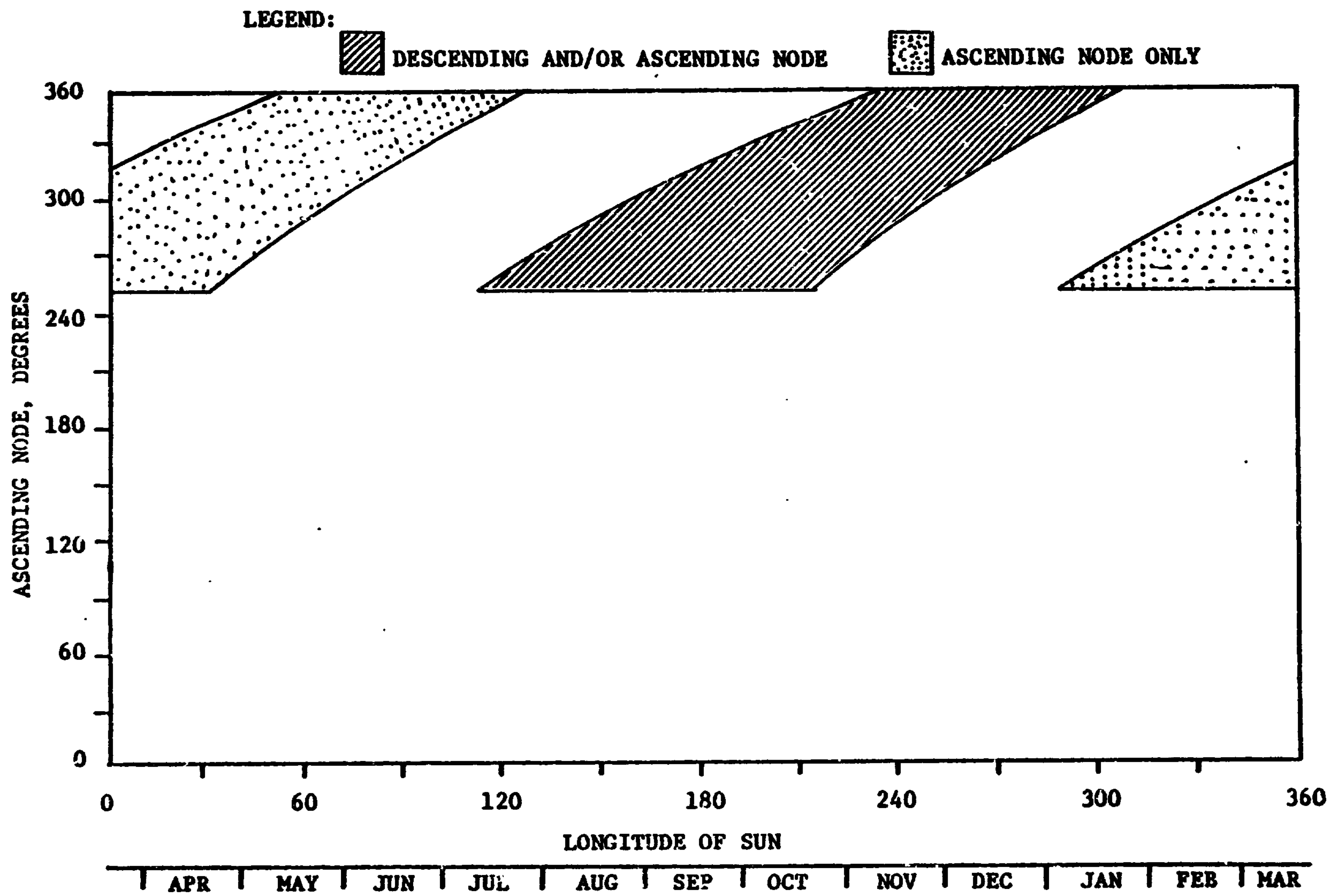

FIGURE A-6. TDRS (TRW) LAUNCH WINDOW

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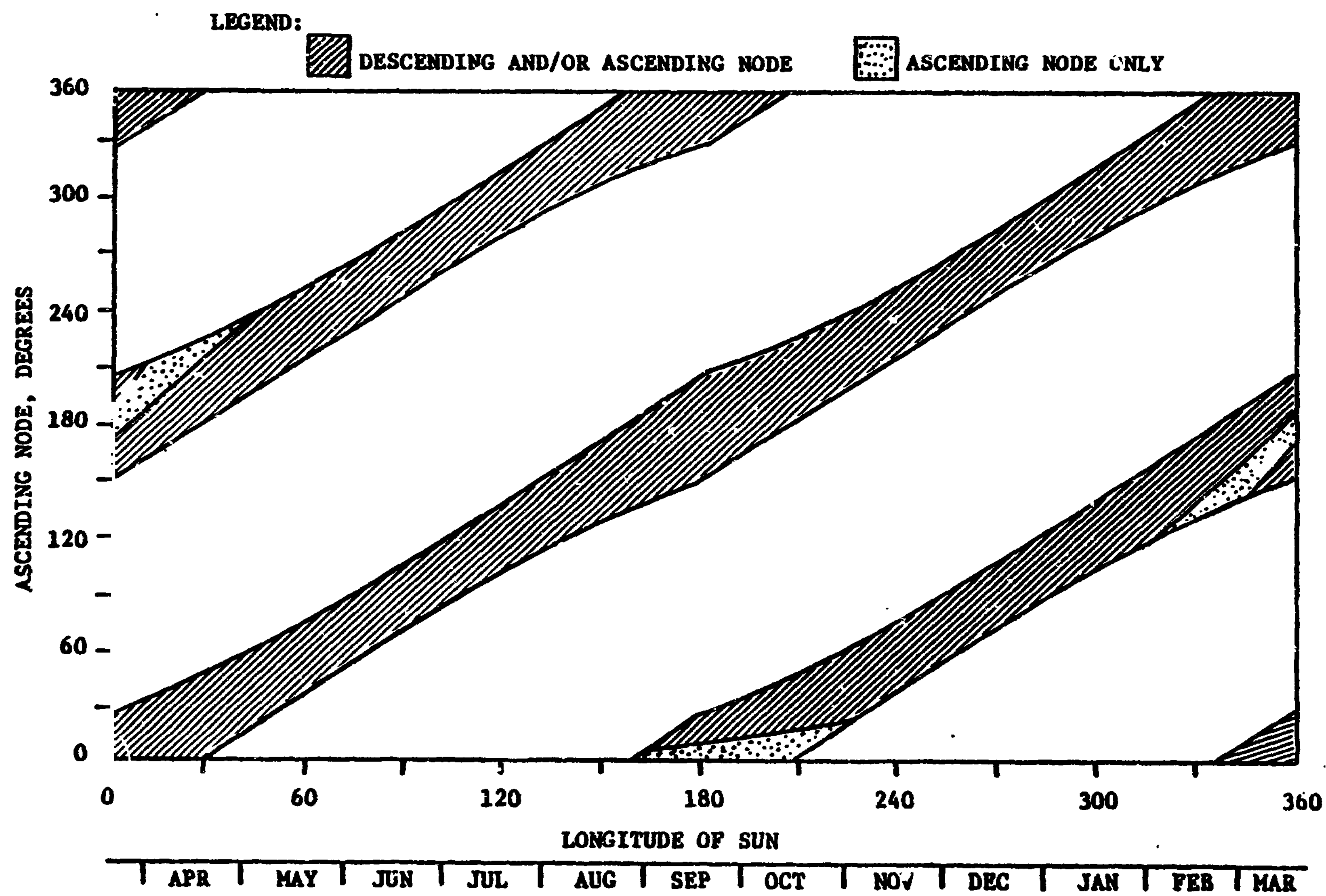

FIGURE A-7. COMSTAR FOLLOW-ON LAUNCH WINDOW

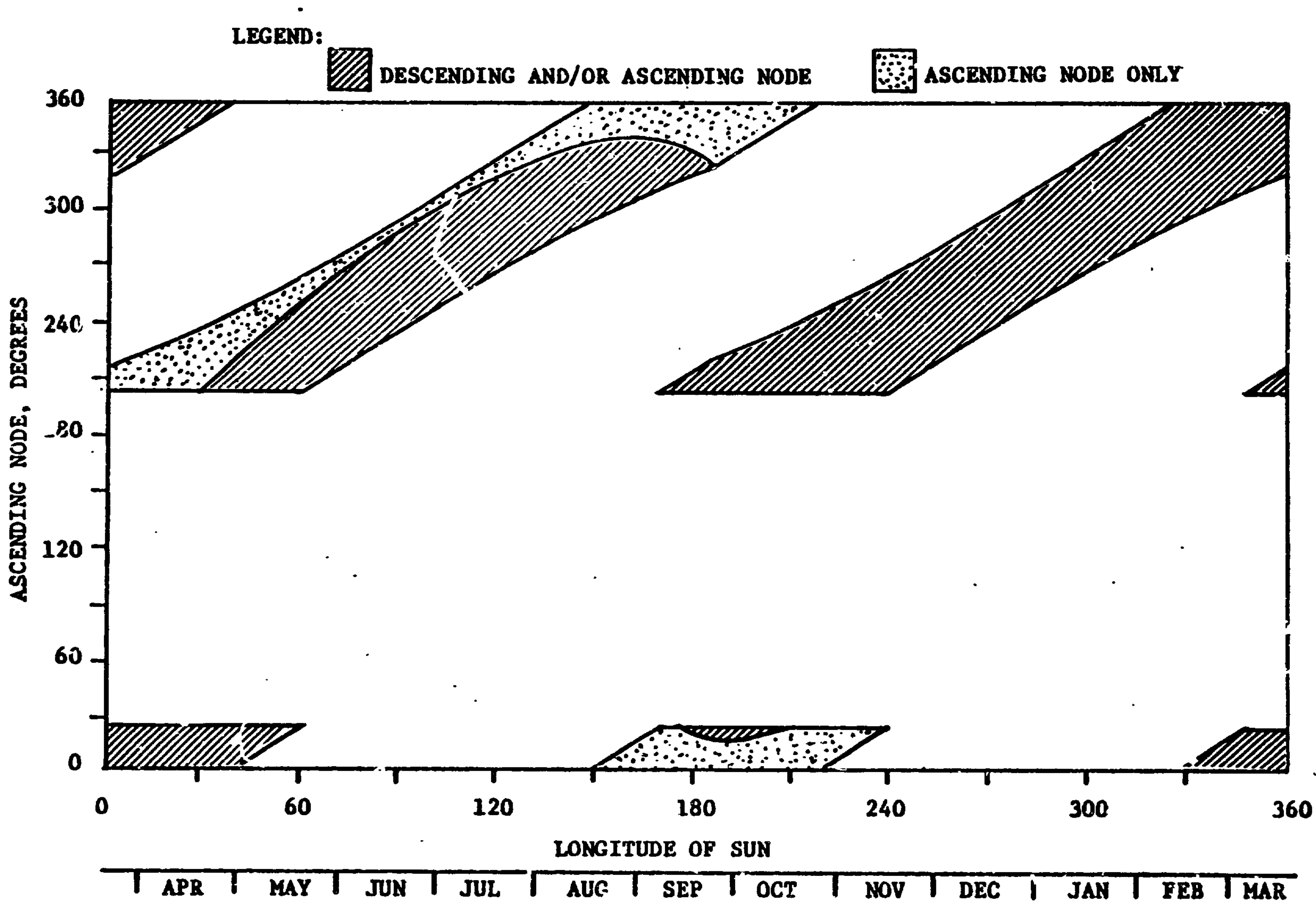

FIGURE A-8. AEROMARISAT LAUNCH WINDOW

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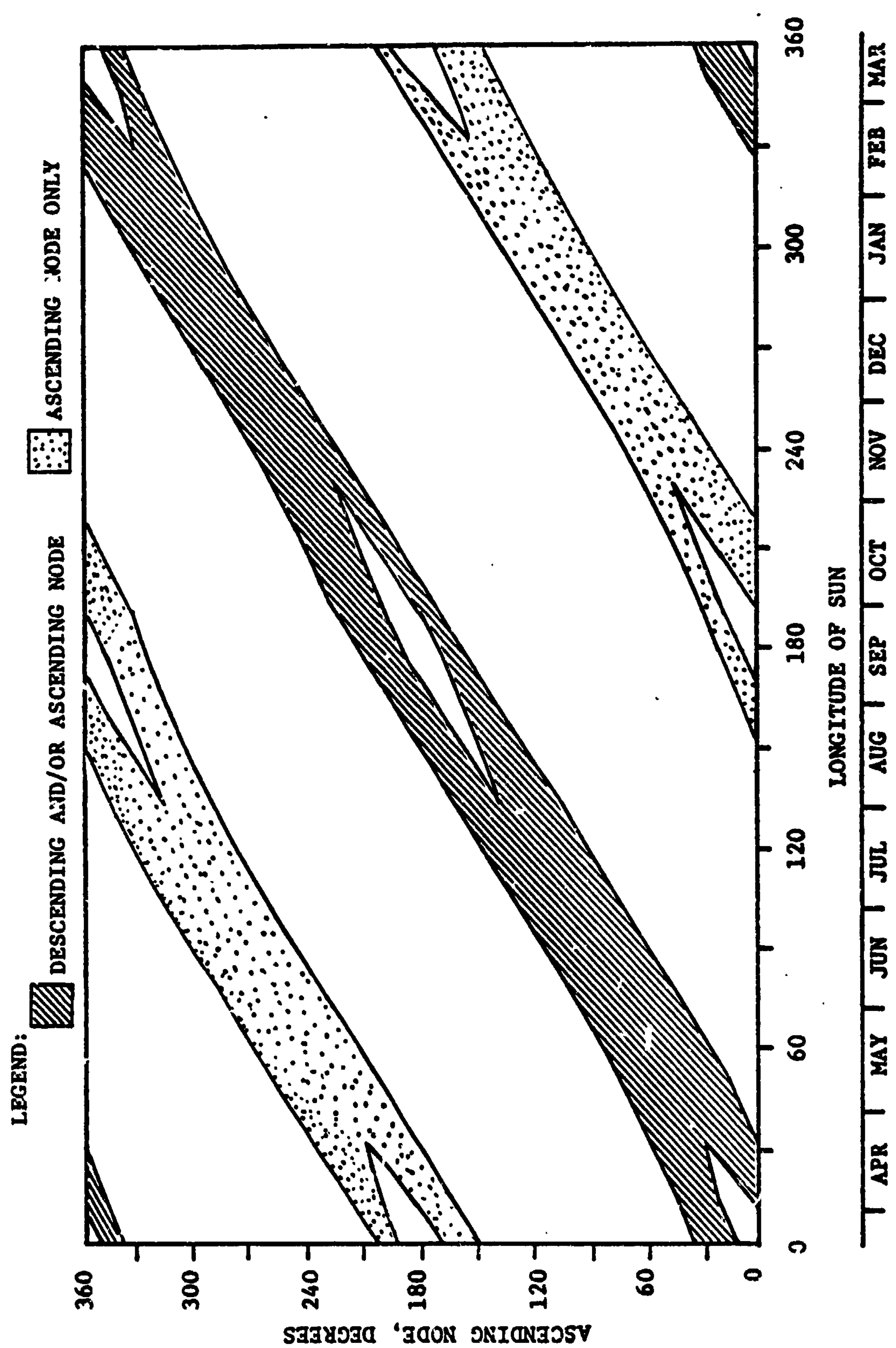

FIGURE A-9. RCA FOLLOW-ON LAUNCH WINDOW

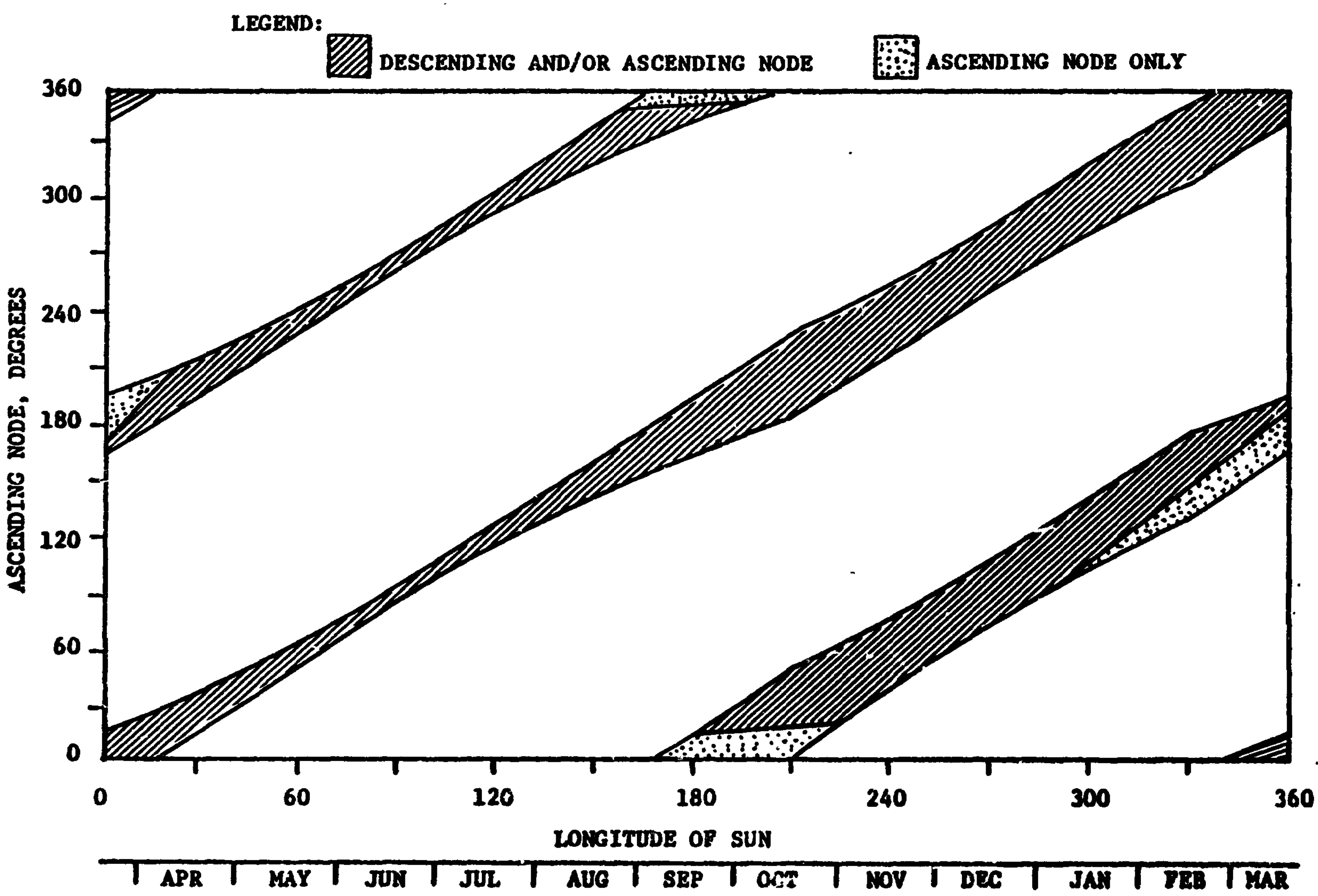

FIGURE A-10. FOREIGN COMMUNICATIONS LAUNCH WINDOW

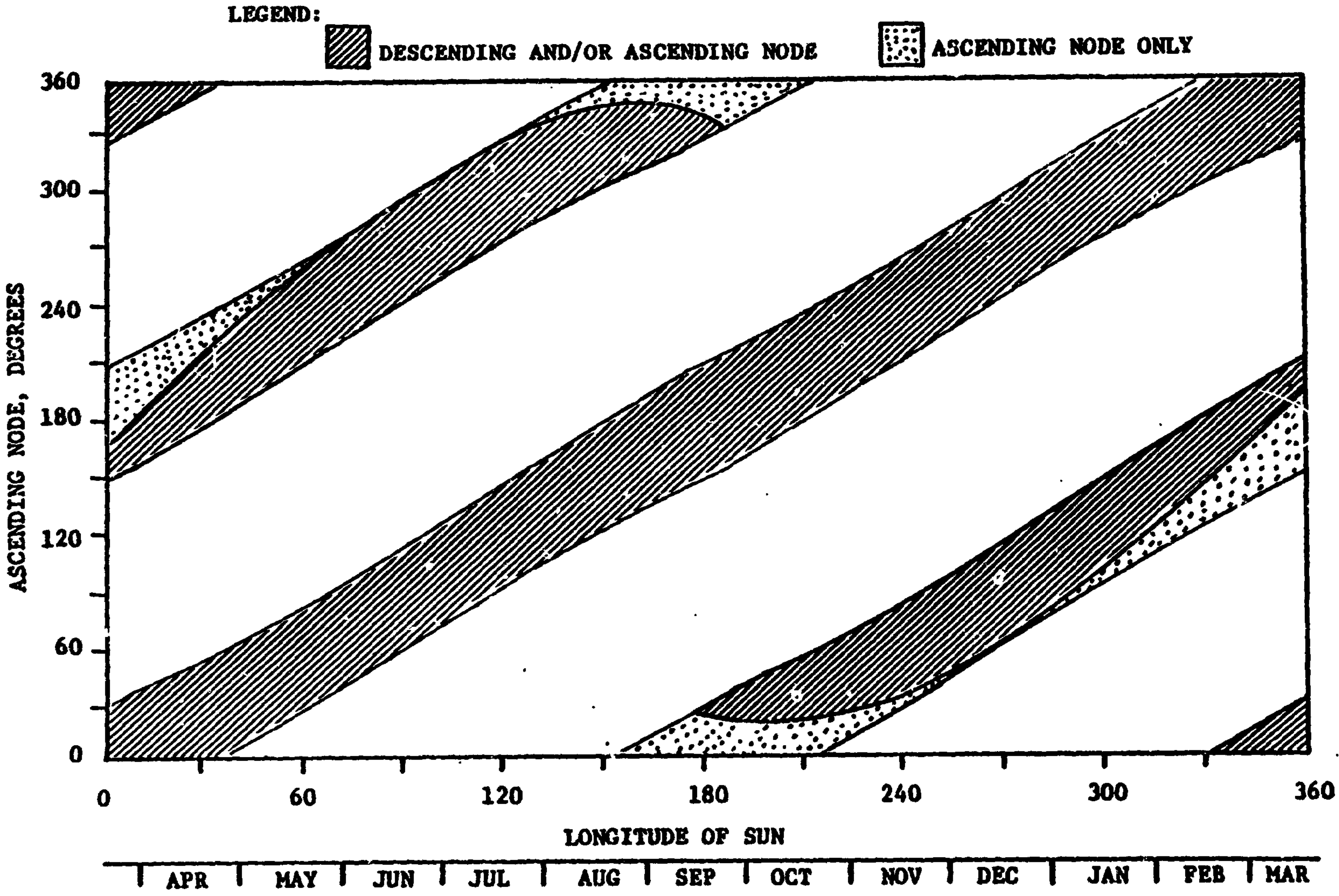

FIGURE A-11. GOES LAUNCH WINDOW

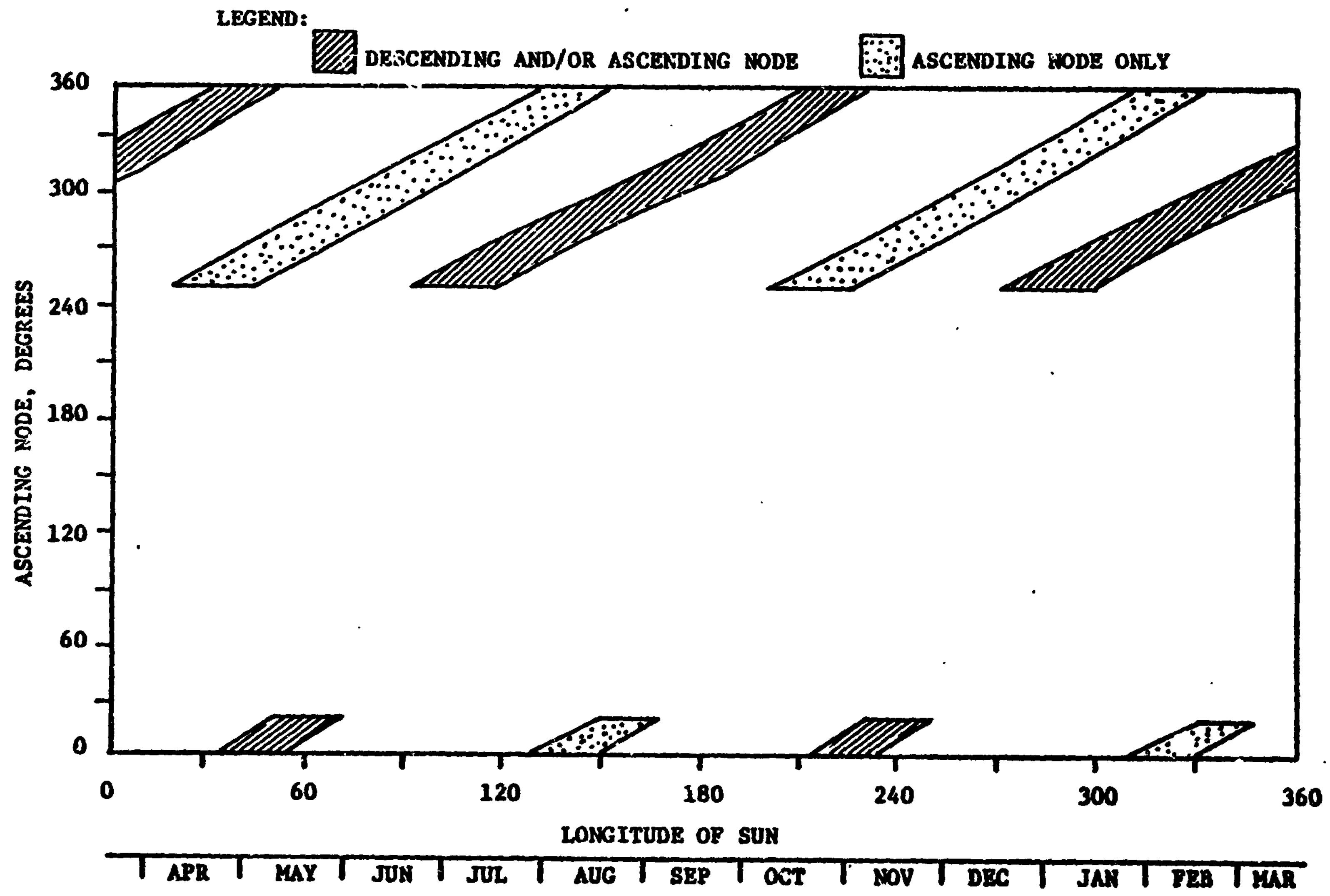

FIGURE A-12. GE TDRS LAUNCH WINDOW

APPENDIX B

SERVICING SUN-SYNCHRONOUS SATELLITES

APPENDIX B

SERVICING SUN-SYNCHRONOUS SATELLITES

Figure B-1 shows the geometry of the problem of servicing a Sun-synchronous satellite. The satellite is in an orbit with radius r_s and inclination i_s. The longitude of its ascending node is Ω_s. The corresponding elements for the Shuttle orbit are r_o, i_o, and Ω_o. Since the Shuttle mission that does the servicing likely is not dedicated to this task but has other objectives as well, its orbital elements will, in general, all be different from those of the satellite being serviced. The problem is futher complicated by the fact that both orbits precess; that is, Ω_s and Ω_o are functions of time. The precession rates are given, to a good approximation, by:

$$\dot{\Omega}_s = -9.97\left(\frac{R_e}{r_s}\right)^{3.5} \cos\,(i_s)\ \text{deg/day} \qquad , \qquad \text{(B-2)}$$

$$\dot{\Omega}_o = -9.97\left(\frac{R_e}{r_o}\right)^{3.5} \cos\,(i_o)\ \text{deg/day} \qquad , \qquad \text{(B-2)}$$

where R_e is the radius of the Earth.

A Sun-synchronous orbit has the property that the local time at the point on the Earth's surface directly under the ascending node is the same on every pass. Put another way, the angle between the plane of the orbit and the Sun's rays is constant. Figure B-2 shows the geometry of this situation. In an Earth-centered non-rotating coordinate frame, the Sun appears to rotate around the Earth at a rate ω_s = 360 deg/year = 0.9856 deg/day. Regardless of where the Sun is, the time at the point on the Earth closest to the Sun is always noon. Midnight is at the point furthest away; 6 a.m. and 6 p.m. are at the points midway between noon and midnight. If a satellite always makes its upward crossing of the equator at 9 a.m. (Sun-synchronous orbit with local time at ascending node 9 a.m.), then its orbital plane always is tilted 45 deg from the Sun's rays (measured in the ecliptic plane). For this to be so, the orbital plane must rotate

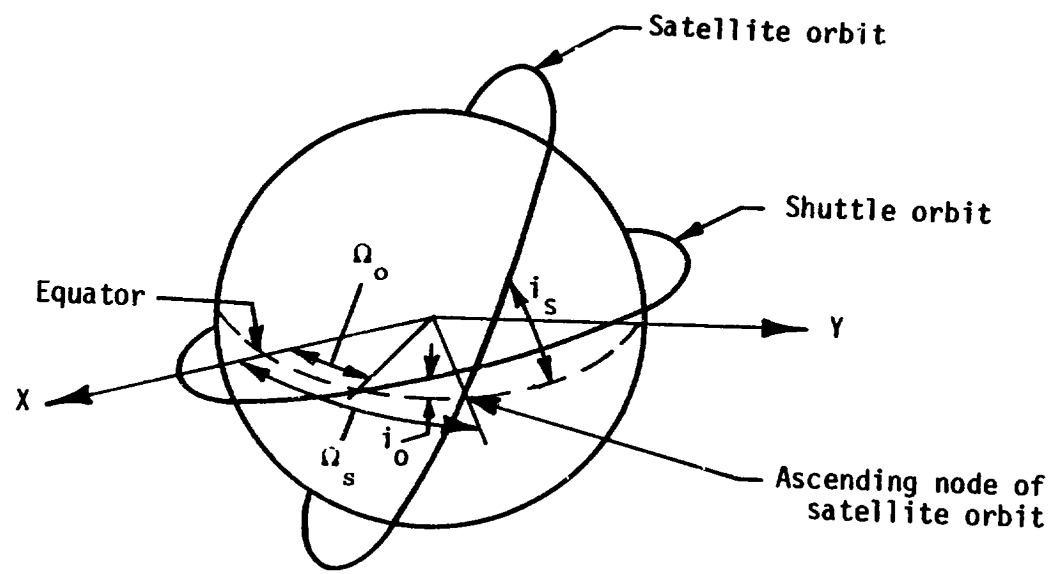

FIGURE B-1. ORBIT GEOMETRY FOR SERVICING SUN-SYNCHRONOUS SATELLITES

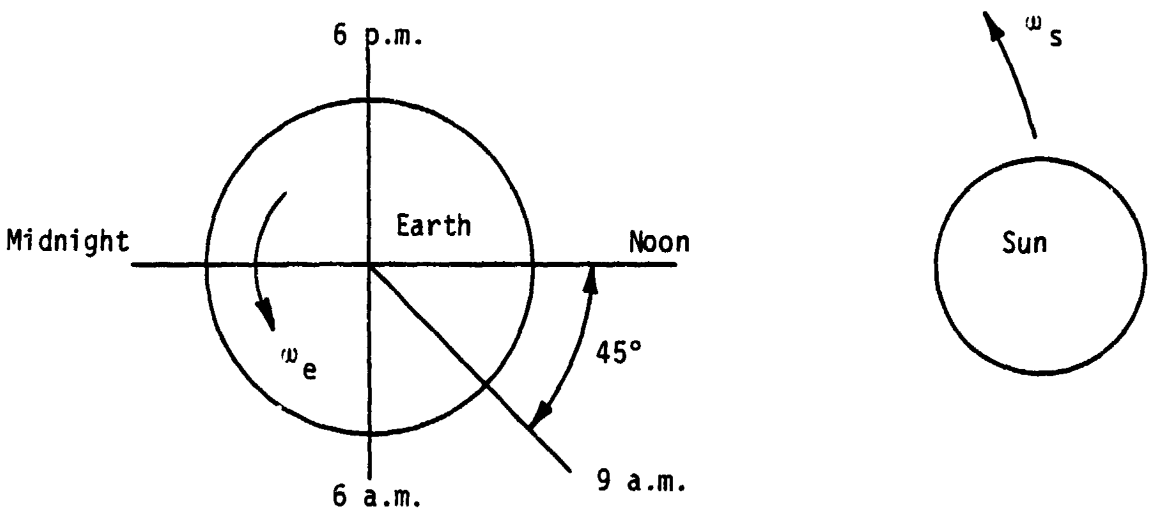

FIGURE B-2. GEOMETRY OF LOCAL TIMES OF ASCENDING NODES about the Earth at the same rate as the Sun. Therefore, all Sun-synchronous orbits have the property that:

$$\omega_s = -9.97\left(\frac{R_e}{a}\right)^{3.5} \cos(i) \quad , \qquad \text{(B-3)}$$

where a and i are the orbit's semimajor axis and inclination, respectively. Most Sun-synchronous satellites have altitudes between about 650 km and 900 km (roughly 350 to 500 nmi). From Equation (B-3) it follows that their inclinations are around 98 to 99 deg.

To service a satellite in a 650-km-altitude, 98-deg-inclination orbit with a 9 a.m. ascending node, it would be desirable to choose a Shuttle orbit with a 98-deg inclination and a 9 a.m. ascending node; then only an altitude change from 650 km down to a nominal 300-km Shuttle orbit would be required and the spacecraft ΔV would be fairly small. There will be many shuttle missions with parking orbits having inclinations near 98 deg, but there will be relatively fewer with ascending nodes at 9 a.m. or any other particular time. Therefore, in most cases, a plane change will be required to correct the line of nodes. It is out of the question to do this plan change purely by impulsive burns, since the ΔV required would be very large. For example, to go from a 900-km, 99-deg satellite orbit to a 300-km, 98-deg Shuttle orbit would require a ΔV of 0.32 km/sec to correct the altitude alone and 0.13 km/sec to correct the inclination alone. (The total ΔV would be somewhat less than the sum of these two if both elements are corrected simultaneously.) However, if the Shuttle were in an orbit with a noon ascending node while the satellite had a 9 a.m. ascending node, a 45-deg plan change would be required to change the line of nodes. This would require a ΔV of 5.7 km/sec, more than 10 times the amount required to correct altitude and inclination. Therefore, the line of nodes must be corrected by taking advantage of the precession phenomenon.

Sometime before the servicing Shuttle flight, the satellite is placed into an orbit which has the proper precession rate, $\dot{\Omega}$, so that its longitude of ascending node, Ω, will drift around to the proper point by the time the Shuttle arrives on orbit. To be specific, suppose, as before, that the satellite is in an orbit with a 9 a.m. ascending node and that when the servicing Shuttle arrives on orbit it will have a 12 noon ascending node. Referring to Figure B-2, it can be seen that the line of nodes can be corrected during a time interval of T days if the satellite is placed into an orbit which drifts 45 deg further forward in T days than a Sun-synchronous orbit would drift. In other words, the satellite must be put into an orbit whose precession rate, $\dot{\Omega}$, satisfies the equation:

$$\dot{\Omega}T = \omega_s T + 45^\circ \quad . \tag{B-4}$$

When the shuttle arrives on orbit, the satellite will then have the proper line of nodes, and only the inclination and altitudes will need to be changed in order to place the satellite into the shuttle orbit.

If it is assumed that this intermediate parking orbit is circular and that the parking time T, is fixed, then the parking orbit has an optimum altitude and inclination which minimize the total ΔV needed to place the satellite into the parking orbit and then transfer from this orbit to the Shuttle orbit. Actually, a fixed parking time implies a fixed precession rate, which means that altitude and inclination are related by an equation such as Equation (B-1) or Equation (B-2). Therefore, the optimum circular parking orbit is determined by a single parameter, either altitude or inclination. In other words, the optimum orbit can be found by a search on one parameter.

To recap the foregoing discussion, bringing a Sun-synchronous satellite down to a 300-km Shuttle parking orbit will, in general, require that the satellite orbit's altitude, inclination and longitude of ascending node all be changed to match those of the Shuttle orbit. The only practical way to achieve the node change is to place the satellite in an intermediate parking orbit whose node will precess to the proper location by the time the Shuttle arrives on orbit. There will be an optimum altitude for the parking orbit such that the total ΔV required to bring the satellite down is minimized.

If the satellite is serviced onboard the Shuttle, then, after the servicing is completed, it must be replaced in its original orbit.

The technique for doing this is the same as for bringing it down. The satellite is placed in a parking orbit which precesses to the desired line of nodes. It is then transferred to the final orbit.

In this study, a computer code has been written to find the optimum parking orbit for a given parking time. For a series of different parking times for both the down leg (bringing the satellite down for servicing) and the up leg (returning the satellite to its original orbit), the code computes the total ΔV required. The total includes the ΔV required to initially place the satellite in orbit. This initial ΔV is calculated assuming a dedicated Shuttle flight; i.e., the Shuttle parking orbit has the same inclination and line of nodes as the final satellite orbit.

A series of curves was prepared showing the total ΔV as a function of the parking times for servicing a variety of different Sun-synchronous satellites from Shuttle flights which launch other Sun-synchronous satellites. The reason for servicing Sun-synchronous satellites from flights launching other Sun-synchronous satellites is that, as mentioned earlier, Sun-synchronous satellites all tend to have inclinations near 100 deg, so a minimum plane change is required if one such satellite is serviced from a flight that launches another. Figure B-3 is a typical set of such curves. The total ΔV is plotted as a function of the parking time on the up leg, T_{up}, with the parking time on the down leg, T_{down}, as a parameter. Figures B-4 through B-8 are similar curves for on-orbit servicing of different missions from various launches. Figures B-9 and B-10 are corresponding curves for the ground refurbishment mode of servicing. Here, there is no up leg, because this is considered to be part of the next mission.

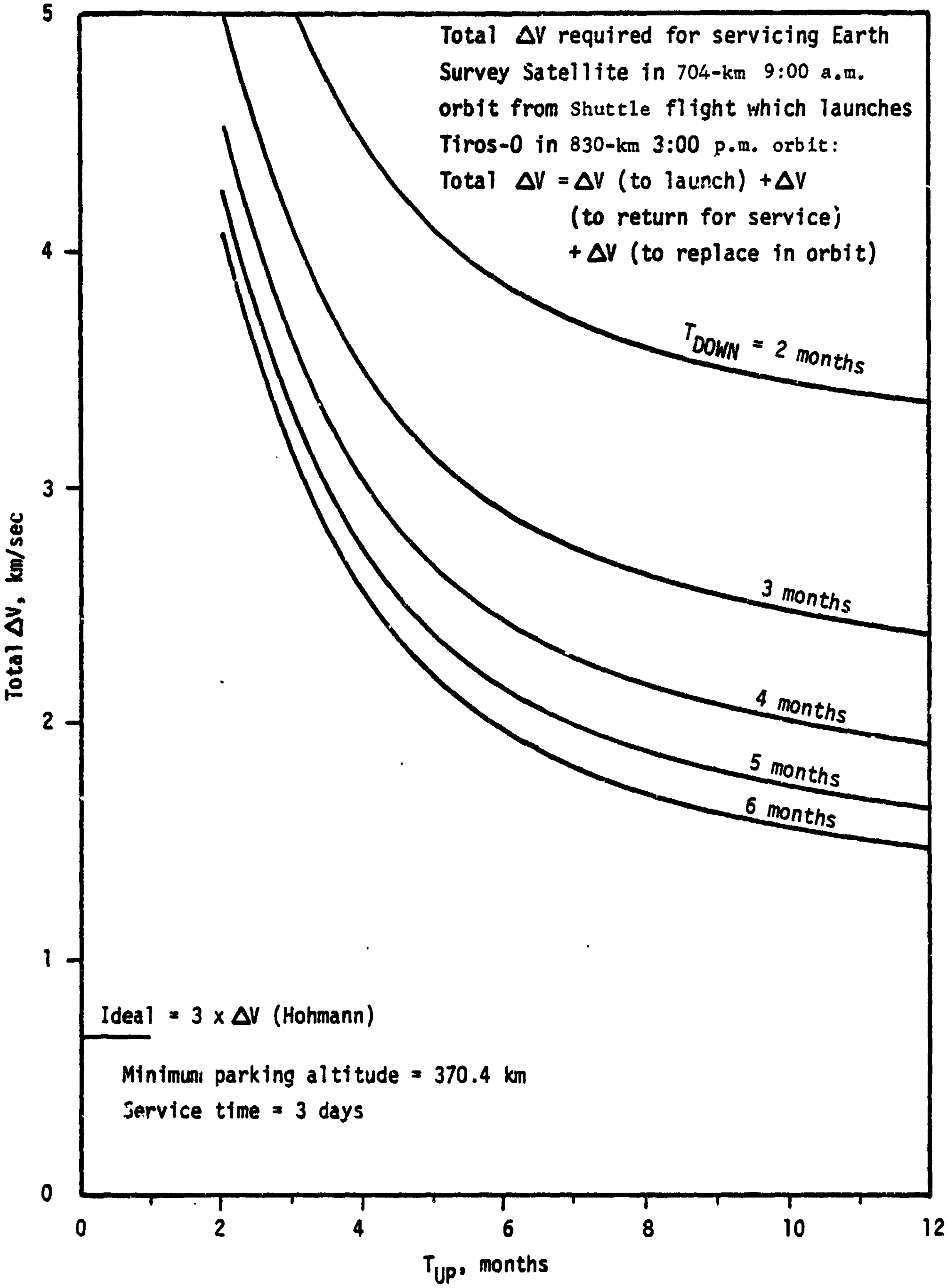

FIGURE B-3. VELOCITY REQUIREMENTS FOR SERVICING EARTH SURVEY SATELLITE IN 9 A.M. ORBIT FROM SHUTTLE LAUNCH OF TIROS-O IN 3 P.M. ORBIT

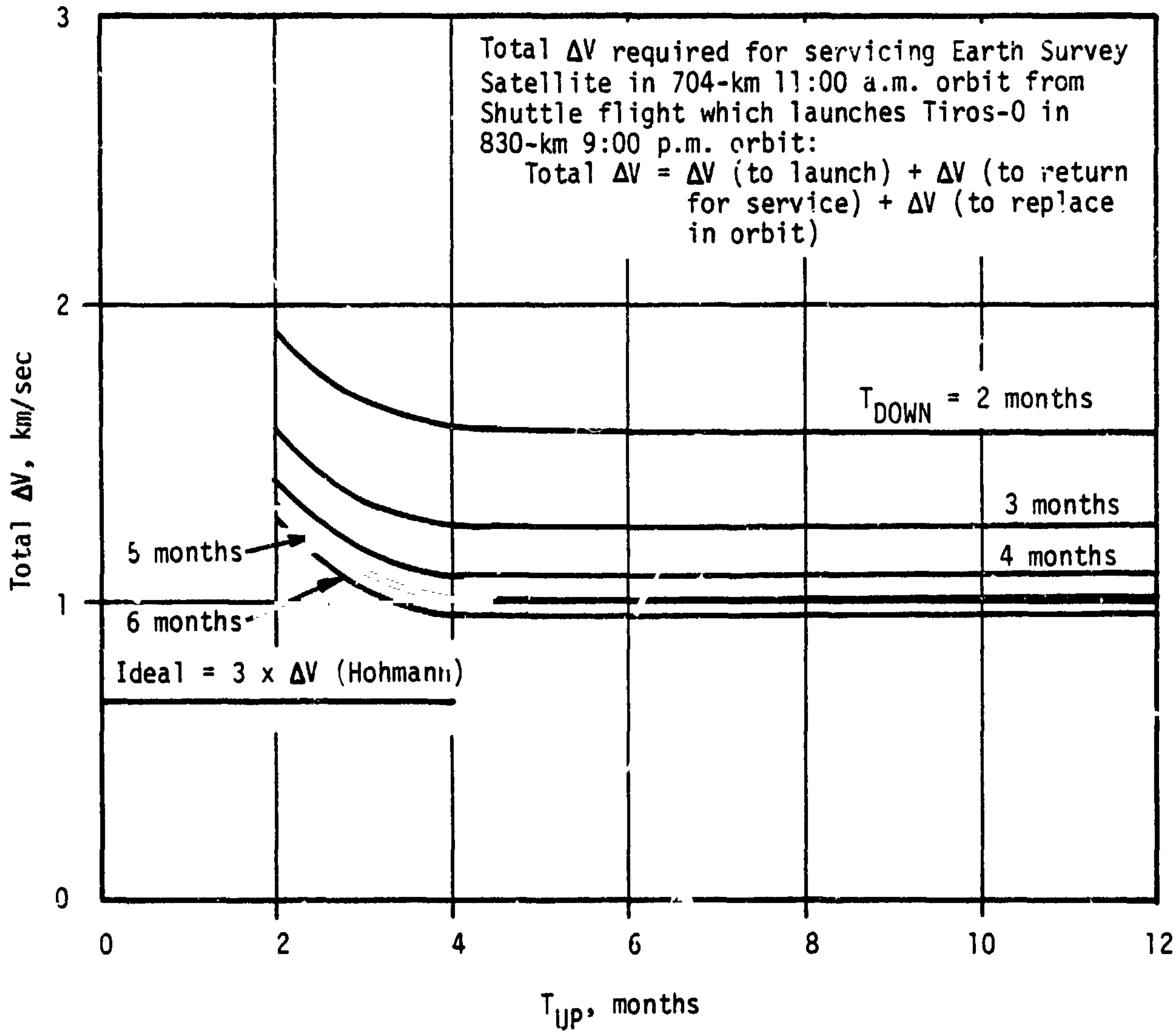

FIGURE B-4. VELOCITY REQUIREMENTS FOR SERVICING EARTH SURVEY SATELLITE IN 11 A.M. ORBIT FROM SHUTTLE LAUNCH OF TIROS-O IN 9 A.M. ORBIT

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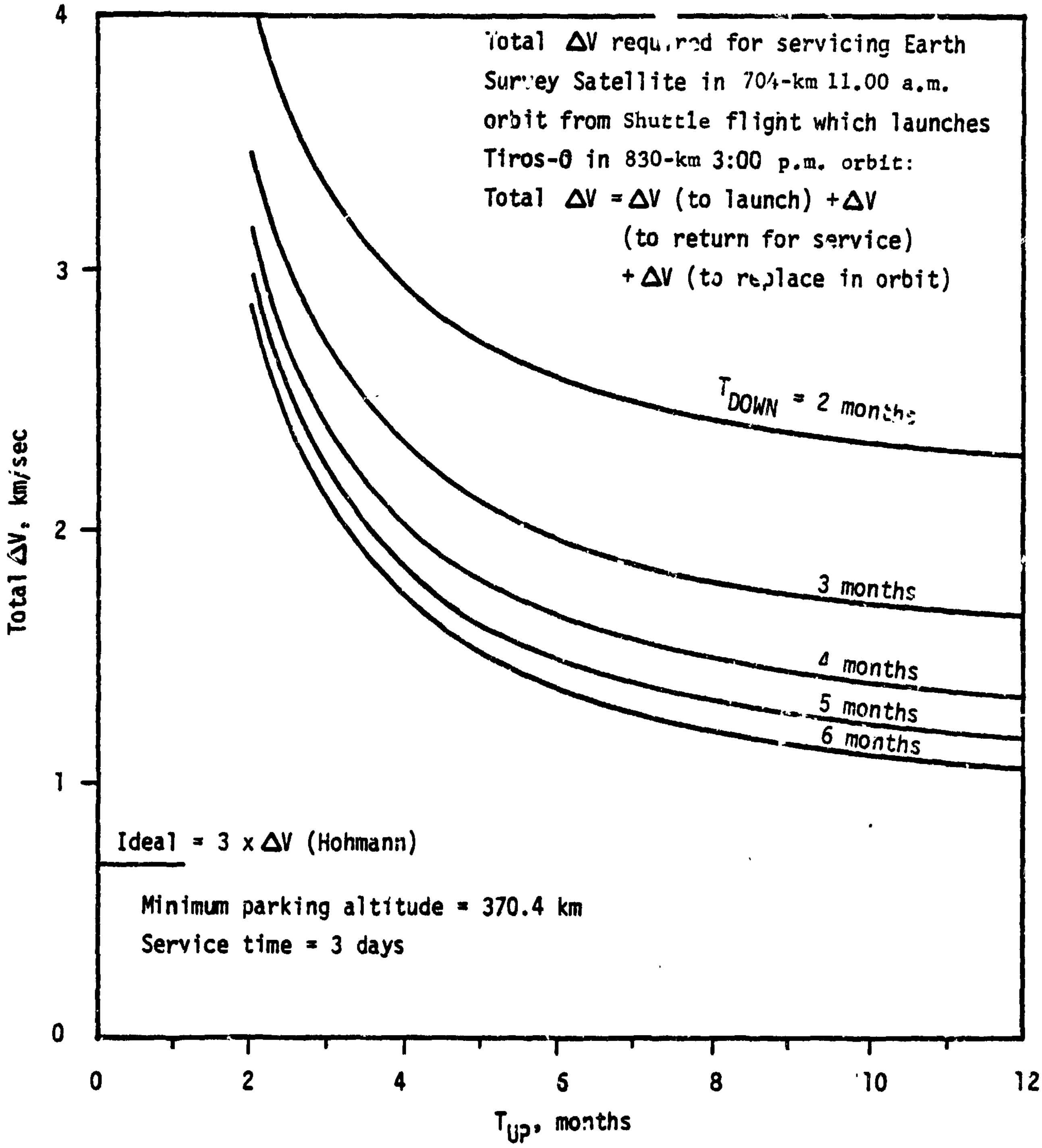

FIGURE B-5. VELOCITY REQUIREMENTS FOR SERVICING EARTH SURVEY SATELLITE IN 11 A.M. ORBIT FROM SHUTTLE LAUNCH OF TIROS-O IN 3 P.M. ORBITS

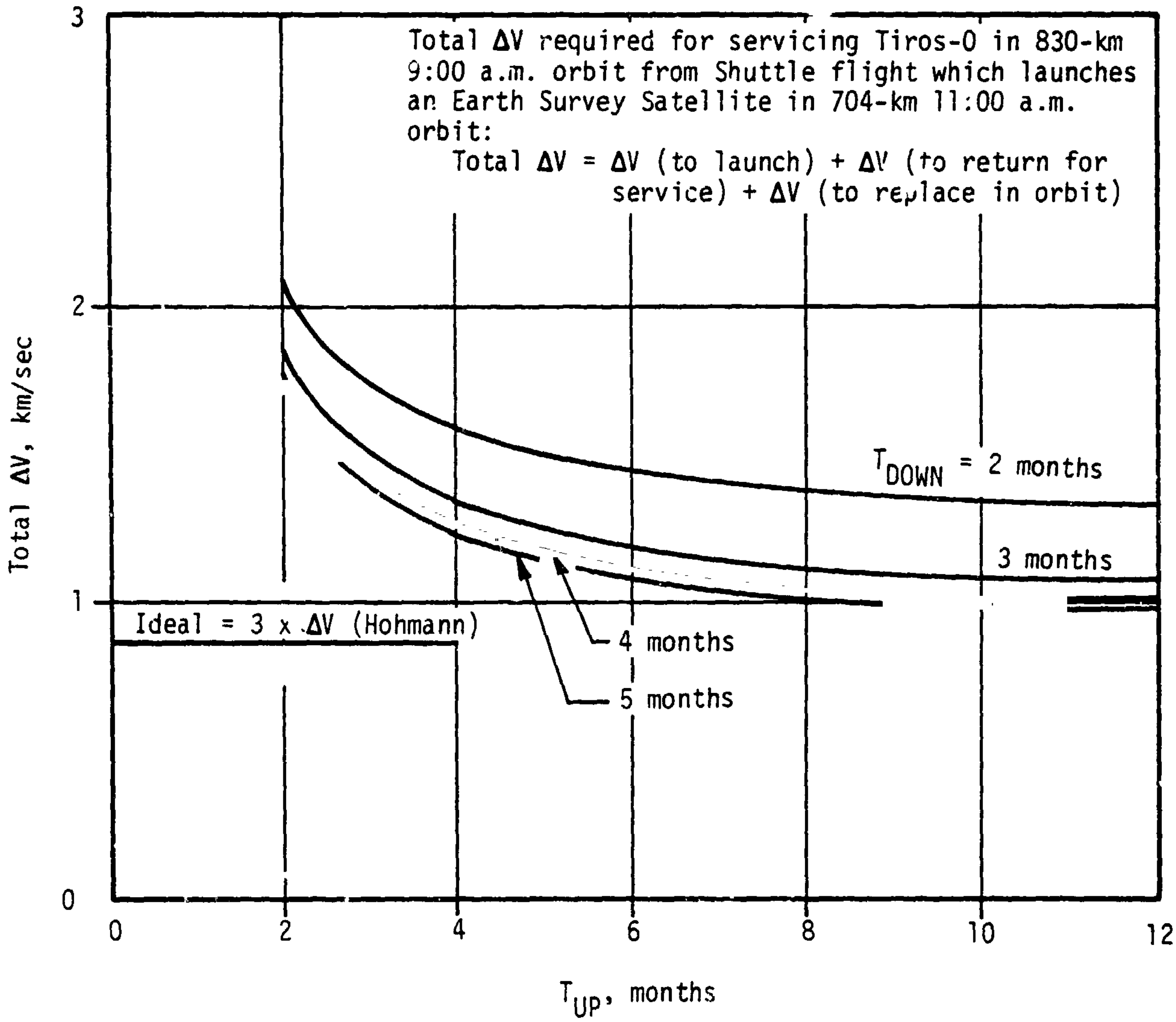

FIGURE B-6. VELOCITY REQUIREMENTS FOR SERVICING TIROS-O IN 9 A.M. ORBIT FROM SHUTTLE LAUNCH OF EARTH SURVEY SATELLITE IN 11 A.M. ORBIT

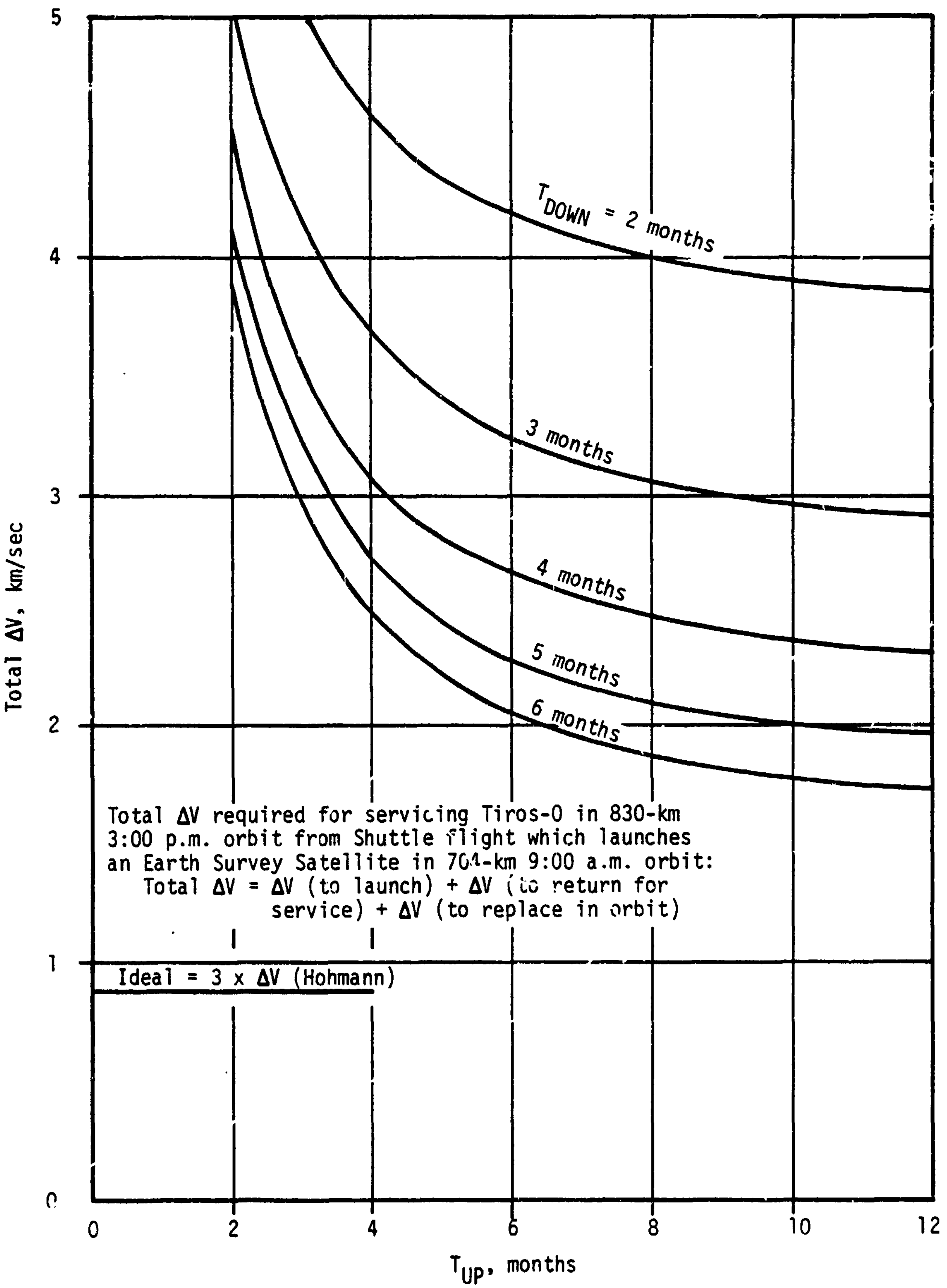

FIGURE B-7. VELOCITY REQUIREMENTS FOR SERVICING TIROS-O IN 3 P.M. ORBIT FROM SHUTTLE LAUNCH OF EARTH SURVEY SATELLITE IN 9 A.M. ORBIT

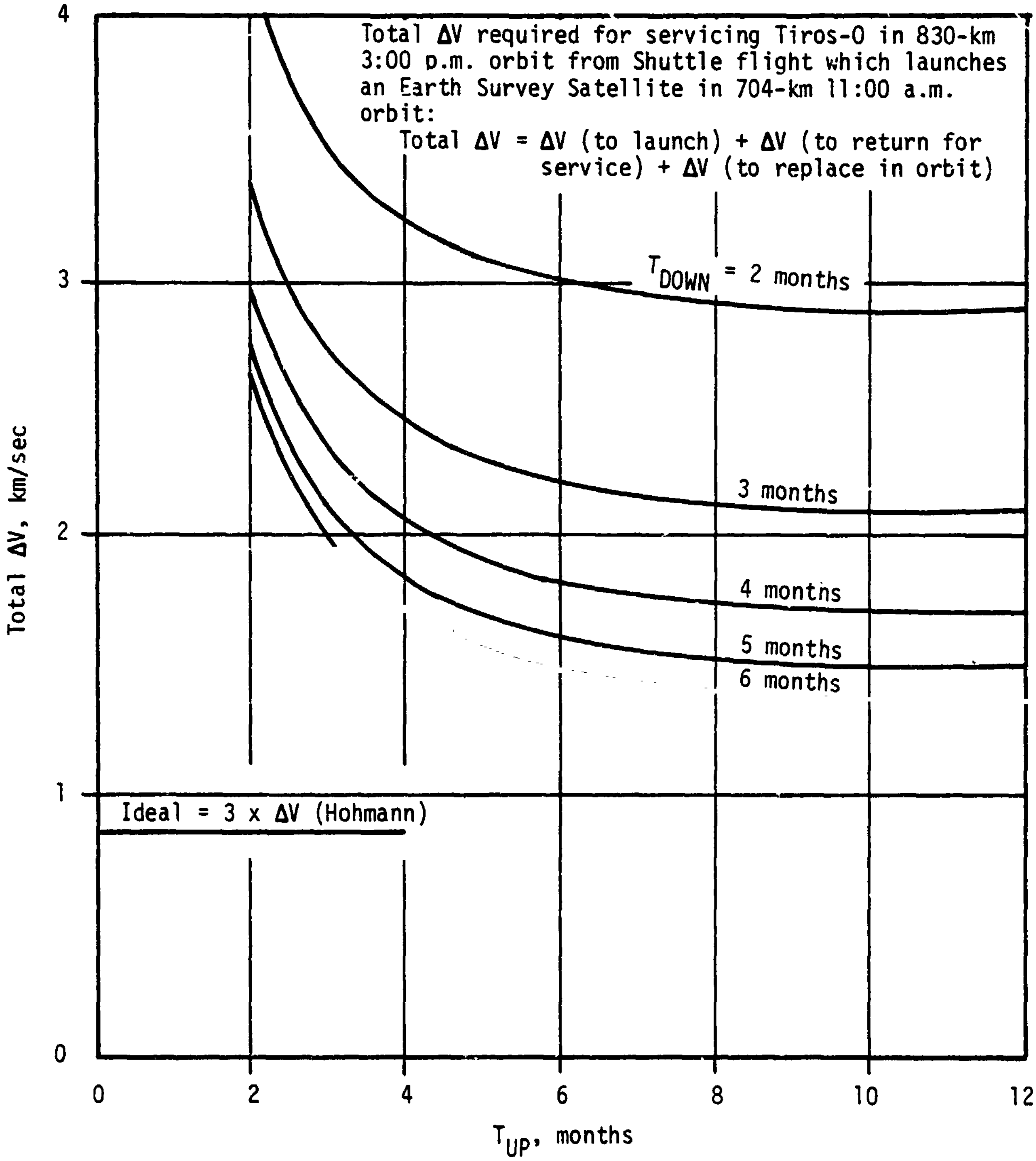

FIGURE B-8. VELOCITY REQUIREMENTS FOR SERVICING TIROS-O IN 3 P.M. ORBIT FROM SHUTTLE LAUNCH OF EARTH SURVEY SATELLITE IN 11 A.M. ORBIT

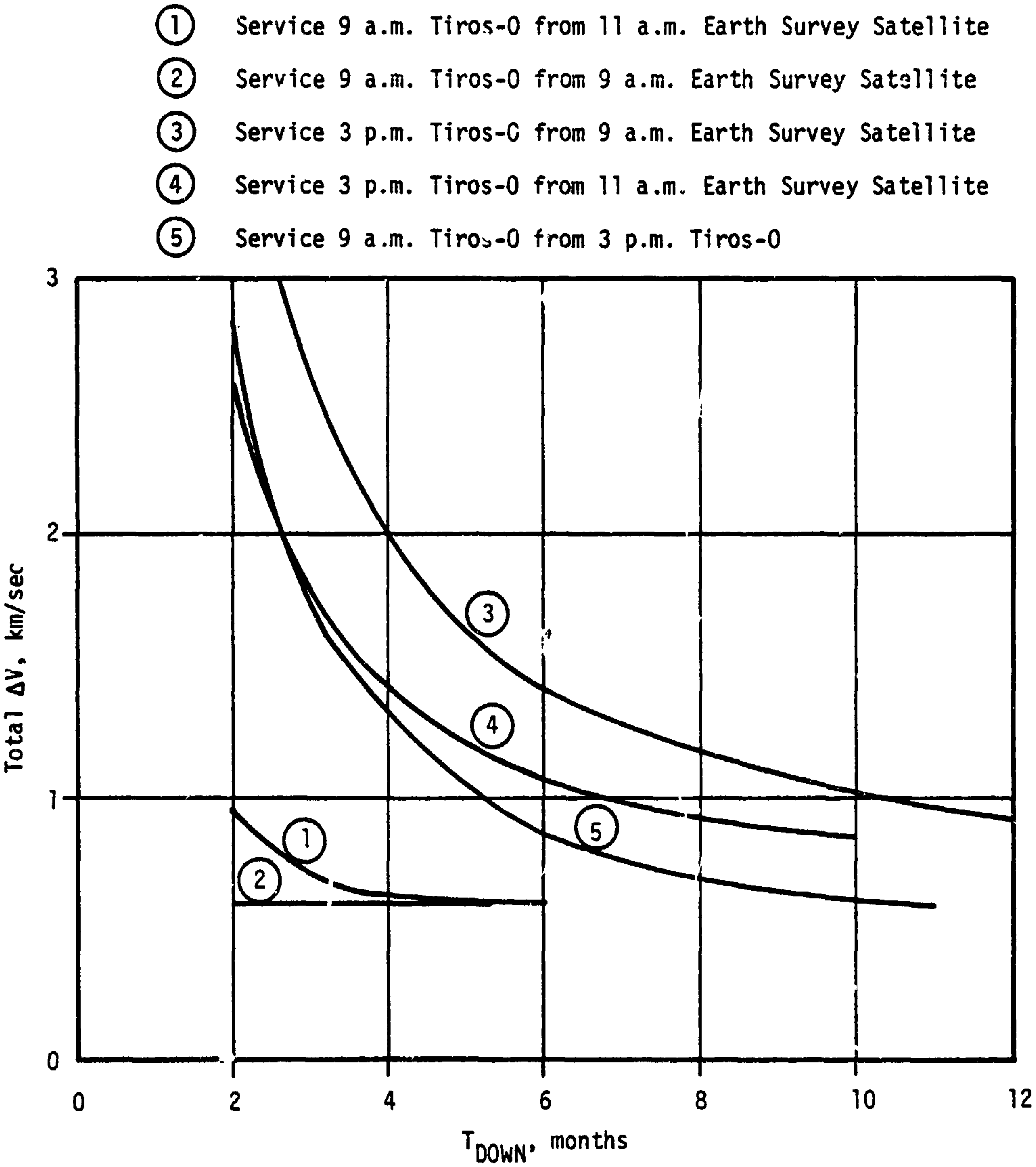

FIGURE B-9. VELOCITY REQUIREMENTS FOR GROUND REFURBISHMENT MODE OF SERVICING TIROS-O FROM EARTH SURVEY SATELLITE OR ANOTHER TIROS-O

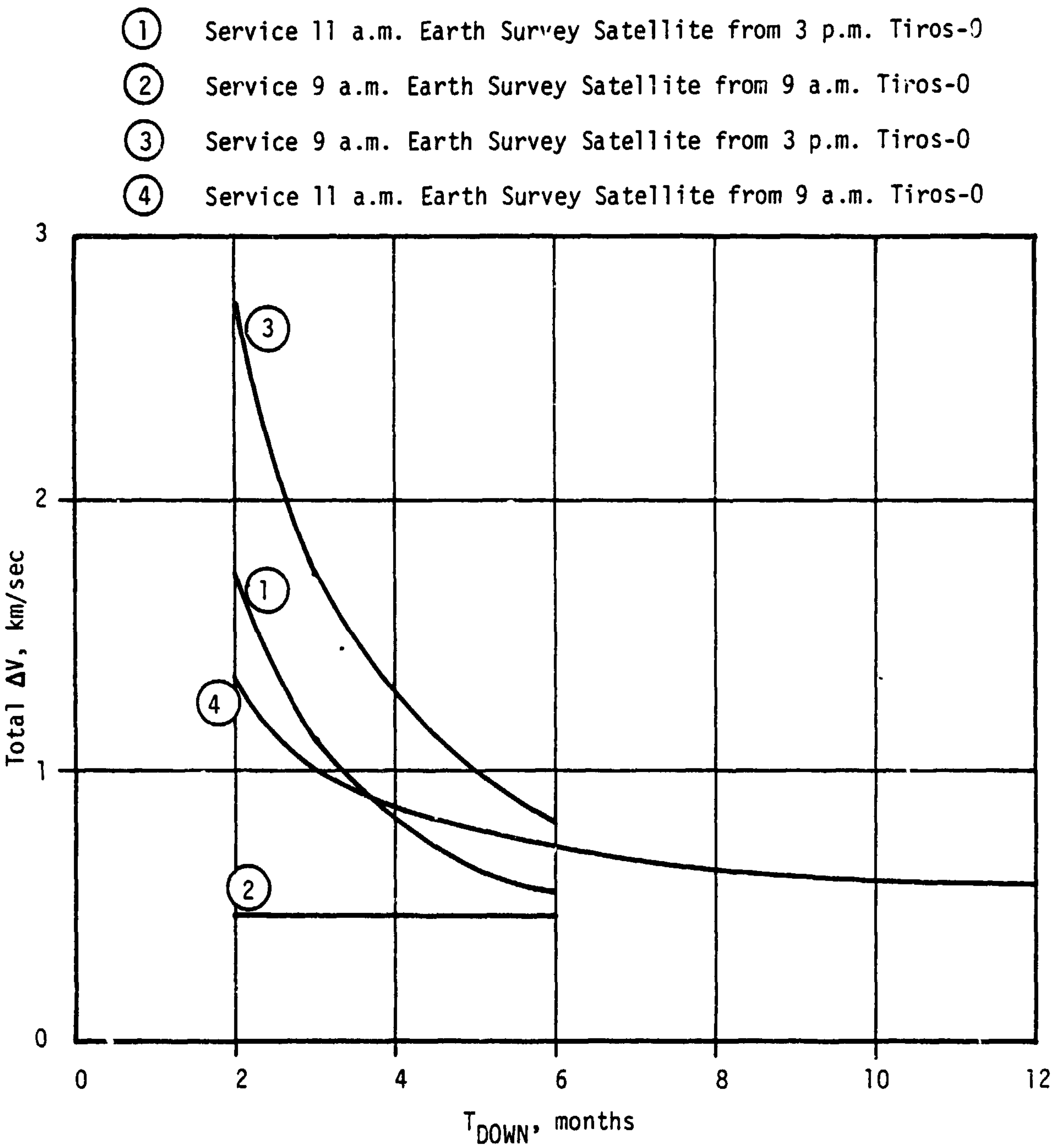

FIGURE B-10. VELOCITY REQUIREMENTS FOR GROUND REFURBISHMENT MODE OF SERVICING EARTH SURVEY SATELLITE FROM TIROS-0