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REPORT NUMBER BMI-NLVP-TM-69- 5

on

STATUS AND TRENDS OF AUTOMATED SPACECRAFT PROPULSION AND THEIR IMPLICATIONS TO SPACE TRANSPORTATION SYSTEM PLANNING

to

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

by

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ABSTRACT

This report summarizes the results of a recent examination of current automated spacecraft propulsion technology and probable future requirements for automated spacecraft propulsion. The implications to space transportation system planning are emphasized. An attempt was made to cover all pertinent forms of spacecraft propulsion (particularly chemical and solar electric propulsion systems) for missions of potential interest to OSSA. The report is intended to serve as a general reference document for space transportation system advance planners.

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INTRODUCTION

In the past, the development of automated spacecraft propulsion systems has been primarily the responsibility of individual spacecraft project offices. This division of propulsion efforts between individual spacecraft projects has been consistent with basic OSSA operational policy and has worked reasonably well. Two foreseeable future propulsion developments may require revision of this policy. These are the possible development and use of the following systems:

(1) Relatively large chemical spacecraft propulsion systems for proposed future planetary orbiters and landers, and

(2) Solar powered electric propulsion systems for interplanetary and other missions.

To understand why these potential developments may require changes in the current policy, it is necessary to discuss the past and present status of spacecraft propulsion in some detail. This will be done in the next section. Following this, the likely nature of future large chemical and solar electric spacecraft propulsion systems, and the general nature of the problems they may create will be discussed. Finally, the impact of these systems on space transportation system advance planning will be considered.

PRESENT STATUS OF SPACECRAFT PROPULSION

Appendix A contains a detailed summary of the present status of automated spacecraft propulsion technology. Some of the essential features of these systems are summarized in this section.

Propulsion systems for automated spacecraft cover a wide range of sizes and types. Thrusts range from the micropound to the kilopound level, while the systems range from subliming solid propellant control motors to more conventional solid or liquid propellant orbit injection and midcourse motors. Figure 1 shows useful (or preferred) ranges for typical automated spacecraft propulsion systems as a function of thrust and total impulse.[1-3]* The boundaries for these regions can be only loosely defined since the requirements for specific missions must eventually be considered on an individual basis.

Spacecraft designs are usually, of necessity, weight limited, so that there is a natural desire to keep the weight of any included propulsion systems low. As the total impulse increases, and as the weight of the propulsion system becomes a significant fraction of the

* Superscript numbers refer to References shown at end of this report.

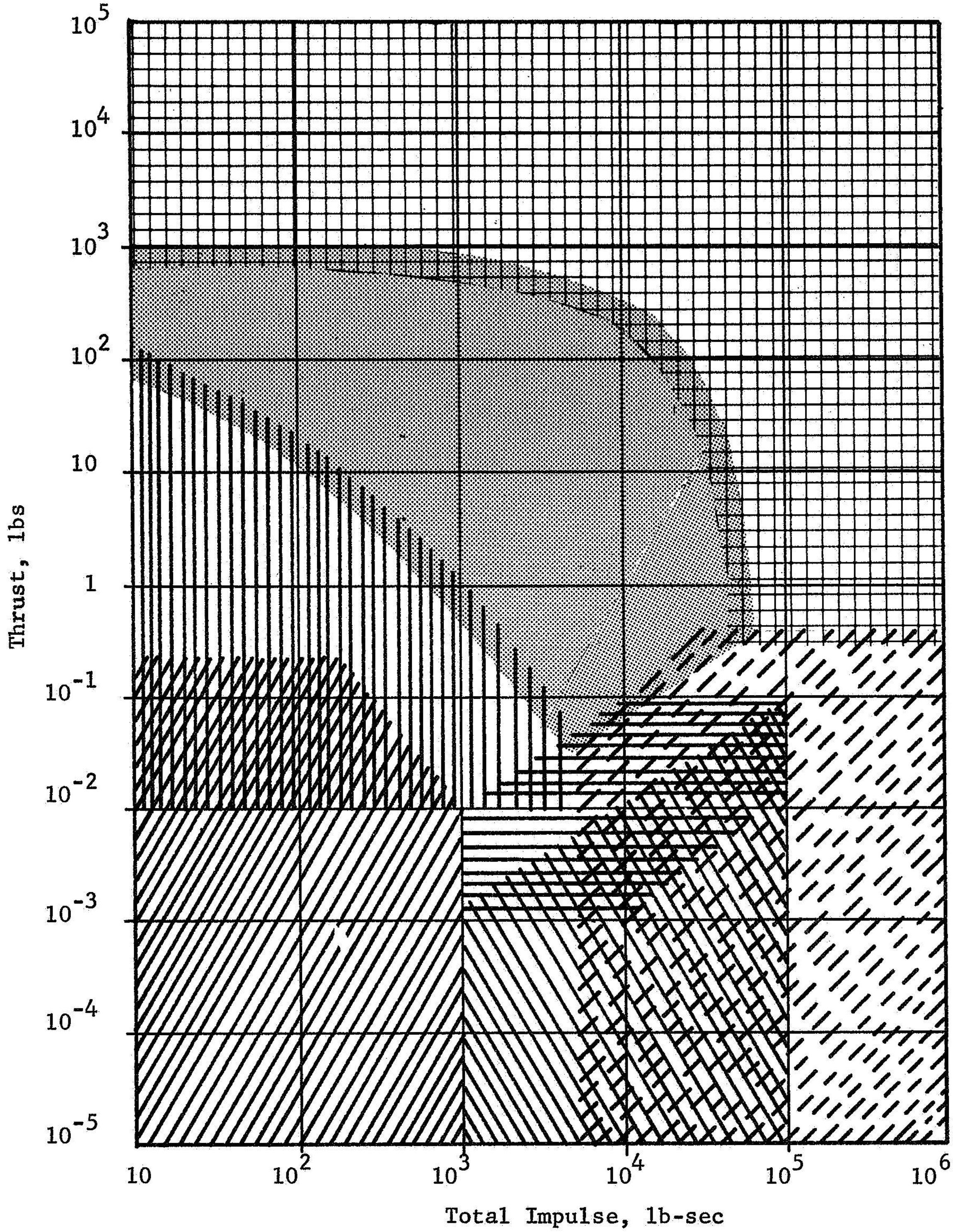


FIGURE 1. POTENTIAL OPERATING REGIMES FOR REPRESENTATIVE AUTOMATED SPACECRAFT PROPULSION SYSTEMS(1-3)

Subliming solids and vaporizing liquids

Stored gas systems

Gaseous bipropellants

Ion engines

Hybrid electro-chemical

Liquid monopropellants

Solids or liquid bipropellants

total spacecraft weight, weight saving (such as through the use of higher specific impulse propellants and systems) becomes increasingly important or even mandatory. Thus, the specific impulse (I_{SP}) of the preferred systems in Figure 1 increases from left to right. For systems providing small total impulses, the weight of the propulsion system is also small. In these instances, propellants and systems are chosen on the basis of system development cost, expected reliability, total system weight, and other factors than specific impulse. Thus, stored gas systems (e.g., high pressure gaseous N_2, $I_{SP} \sim 75$ sec) are acceptable for total impulses of 10^1 - 10^3 lb-sec for thrusts around 1 lb, while a solid or liquid bipropellant system (e.g., NTO-Aerozine 50, $I_{SP} \sim 310$ sec) is preferable when the total impulse requirement is increased to 10^5 - 10^6 lb-sec at the same thrust level.

Several undesirable features are associated with the use of higher I_{SP} propellants and systems. In most cases, higher specific impulse is accompanied by higher operating temperatures and greater complexity, which leads to reduced lifetime and reliability, and to increased cost. In some cases, such considerations can dictate that a propulsion system be operated at a lower specific impulse than the maximum obtainable. For example, the Vela nuclear detection satellite built by TRW had an orbit adjustment propulsion system using high pressure gaseous N_2 heated by electrical resistors. The system was operated at a specific impulse of 123 sec, even though it was capable of yielding higher I_{SP}. Operation at the higher temperature (I_{SP}) would have increased the chance of the exhaust jet interfering with the satellite communication system.(4)

Small systems are generally sufficiently simple that new units can be developed at relatively low cost. Under these conditions, each system can be custom designed for the individual spacecraft, with spacecraft interactions the predominant consideration. Consequently, the need for central coordination and direction in this area is minimal.

In many cases, low total impulse systems also have low thrust requirements, so that even nonexpulsive propulsion techniques may be attractive. For example, the Mariner IV spacecraft experienced small net torques about two axes due to an unequal distribution of solar pressure. It was nececessary to provide a long term attitude control system as well as a system for major maneuvers, such as for midcourse orientation. A stored gas system was used for both the midcourse maneuver orientation and attitude control. However, for attitude control about the two Sun-perturbed axes, the stored gas system was supplemented by using the solar pressure itself. Controllable solar sails provided the necessary correcting torques by varying the solar pressure force on each paddle.[5,6]

In contrast to the situation for small spacecraft propulsion systems, large systems can require major, relatively costly development efforts. These systems can be as large as a launch vehicle upper stage and may require the same or, perhaps, more advanced technology. Hence, there exists an obvious need to coordinate the development of such systems to insure their maximum utility and to avoid costly developments of single use items.

Having argued that low total impulse (or ΔV) systems can, and probably should, be designed for each individual spacecraft, and that high total impulse (or ΔV) systems should not be, it becomes necessary to decide what constitutes high and low total impulse (or ΔV) systems. Appendix B discusses the most common current and projected automated spacecraft propulsion applications and describes their associated total impulse or velocity increment requirements. The data presented there provides a useful basis for defining what constitutes large systems.

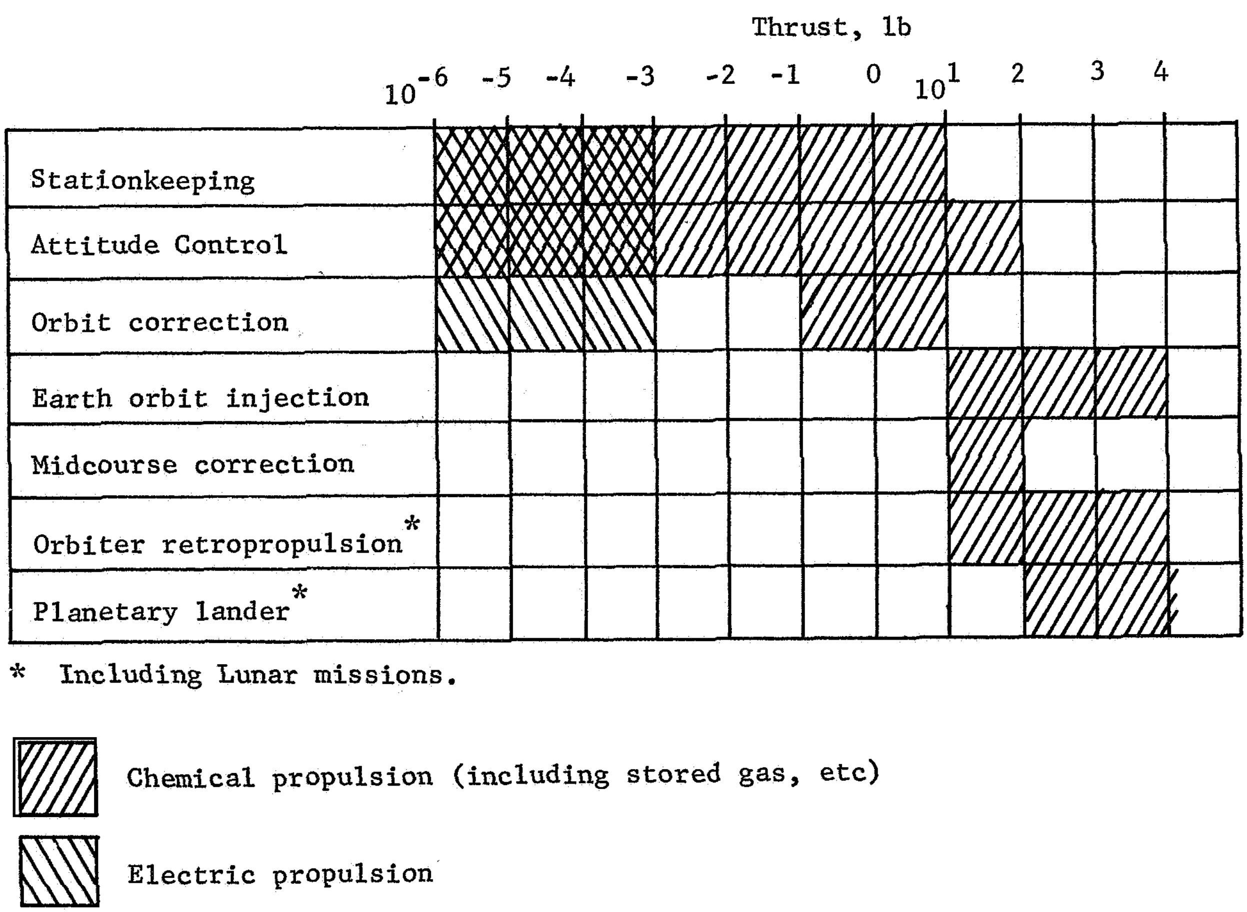
Tables 1 and 2 summarize automated spacecraft propulsion applications discussed in Appendix B and their typical ΔV (or total impulse) and thrust ranges. Of the applications shown, Earth orbit injection, orbiter retropropulsion and planetary lander systems can be considered to definitely fall in the high total impulse (high ΔV) category while north-south stationkeeping and midcourse correction systems could be classed as being on the borderline between small and large systems.

Figures 2 and 3 show the range of total impulse and thrust and weight, respectively, for some existing spacecraft propulsion systems in these categories.[5,6,8-18] Table 3 provides some details on the propulsion systems shown in Figures 2 and 3.[5,6,8-18] Appendix C presents additional details.

TABLE 1. PRESENT AUTOMATED SPACECRAFT PROPULSION APPLICATIONS AND ASSOCIATED TOTAL IMPULSE OR VELOCITY INCREMENT REQUIREMENTS

Automated Spacecraft Propulsion Application	Typical Total Impulse (lb-sec) or Velocity Increment (ft/sec) Requirements
Stationkeeping	
East-West	~ 10 ft/sec (per year)
North-South	~ 200 ft/sec (per year)
Attitude Control	~ 300-400 lb-sec
Earth Orbit Injection (Including Earth Escape Injection)	~ 100-10000 ft/sec
Orbit Correction	~ 1-100 ft/sec
Midcourse Correction	~ 50-120 ft/sec
Orbiter Retropropulsion	~ 1000-40,000 ft/sec
Lander Propulsion	~ 6000-20,000 ft/sec

TABLE 2. PRESENT AUTOMATED SPACECRAFT PROPULSION APPLICATIONS AND ASSOCIATED THRUST LEVELS[7]



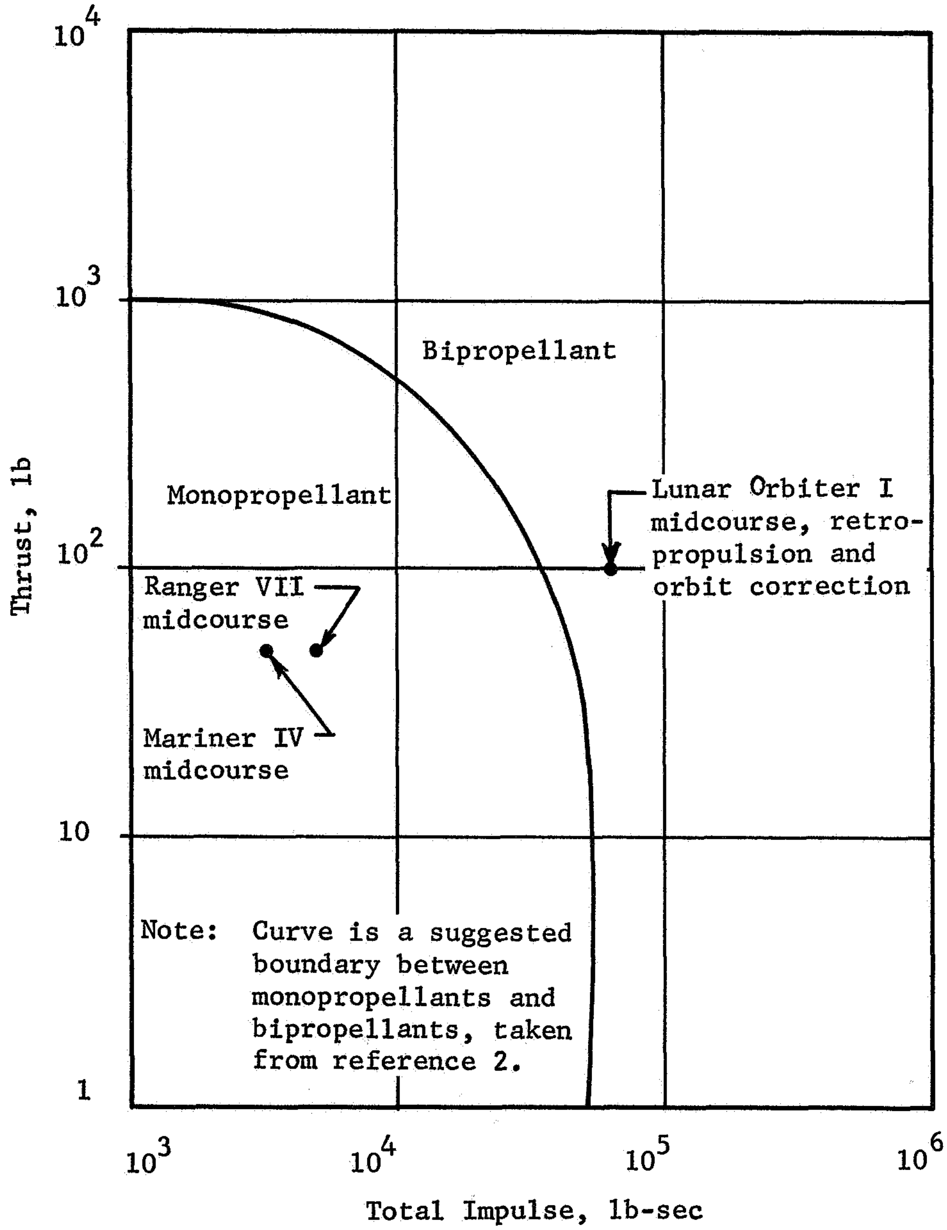


FIGURE 2. THRUST AND TOTAL IMPULSE LEVELS FOR SOME EXISTING AUTOMATED SPACECRAFT PROPULSION SYSTEMS(2,5,8-10,12-15)

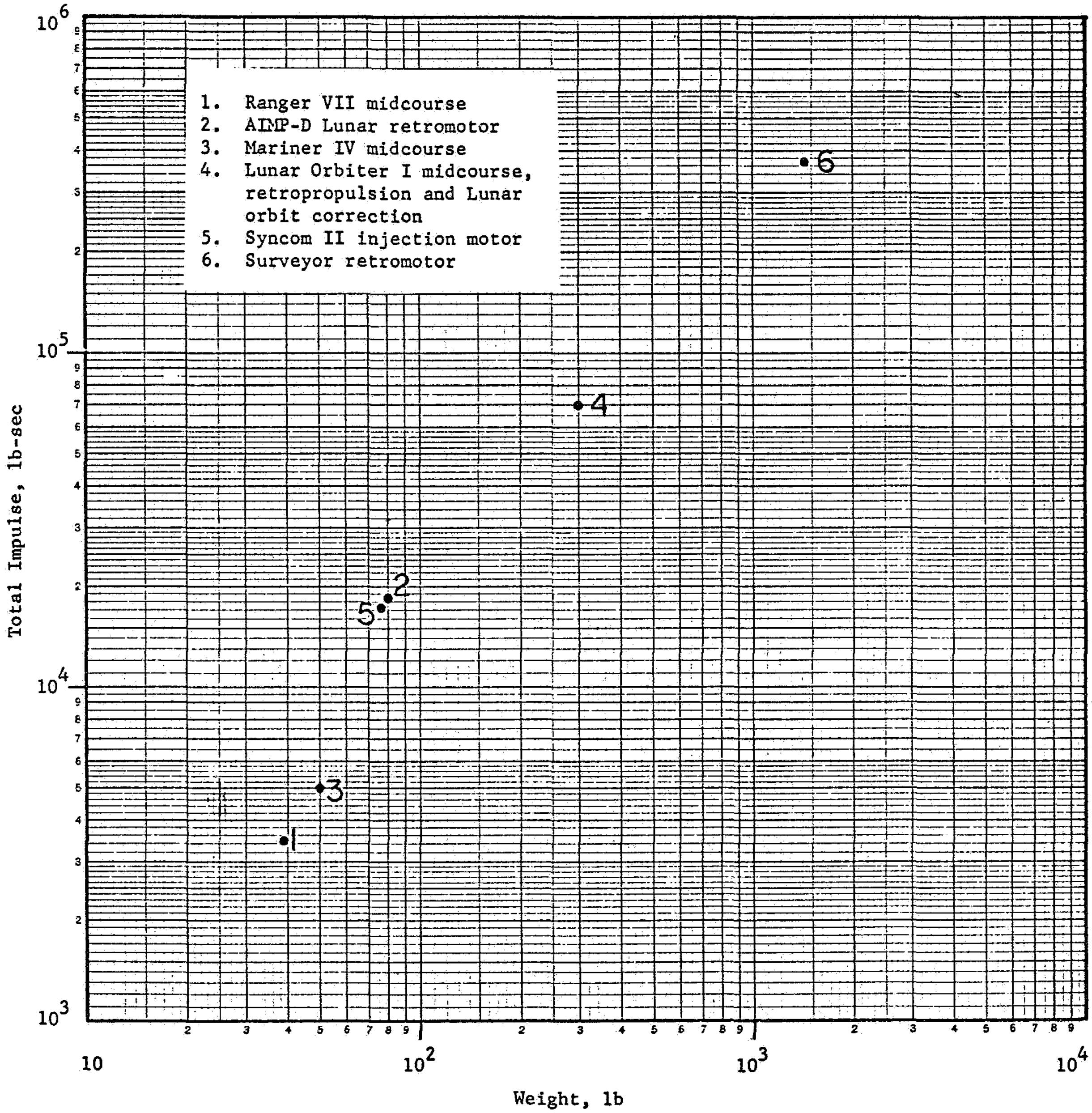


FIGURE 3. WEIGHT AND TOTAL IMPULSE LEVELS FOR SOME EXISTING AUTOMATED SPACECRAFT PROPULSION SYSTEMS(5,8-10,12-18)

TABLE 3. DETAILS OF SOME EXISTING AUTOMATED SPACECRAFT PROPULSION SYSTEMS(5,6,8-18)

Spacecraft Name	Type of Mission	Midcourse System	Attitude Control	Retromotors	Orbital Maneuver	Comments
Ranger VII	Lunar Impact	39.14 lb wt, 50 lb thrust anhydrous hydrazine monopropellant motor.	Cold-gas jets (nitrogen)	None	None	---
Mariner IV	Mars Flyby	49.9 lb wt, 49.4 lb thrust hydrazine mono-propellant motor.	Multiple system--cold-gas jets (nitrogen) and solar pressure vanes (2 axis)	None	None	Used midcourse motor derived from Mariner A development
Mariner II	Venus Flyby	31.36 lb wt hydrazine monopropellant motor	Multiple system--cold-gas jets (nitrogen) and solar pressure vanes (1 axis)	None	None	Used Ranger midcourse system, adapted for Mariner
Lunar Orbiter I & II	Lunar Orbit	One 100 lb thrust bipropellant (NTO-Aerozine 50) liquid engine for up to two midcourse corrections and lunar orbit maneuvers	Cold-gas jets (nitrogen)	Used same motor for midcourse, retro, and orbital maneuver	Used same motor for midcourse, retro, and orbital maneuver	Lunar Orbiter used a motor originally developed for use on Apollo spacecraft
Surveyor	Lunar Soft Landing	Three liquid propulsion variable thrust (30-104 lb each) engines	Cold-gas jets (nitrogen)	One large (8K to 10K lb thrust) solid retromotor (TE364) plus the same vernier motors used for midcourse correction	None	Fine control vernier landing motors were originally used for attitude control on Gemini
SYNCOM II	Synchronous Earth Orbit	None	Spin stabilized	None	A small (76 lb wt) solid propellant apogee injection motor (TE375) giving a ΔV of 4712 ft/sec, and a set of hydrogen peroxide and gaseous nitrogen jets for orbit correction and longitude positioning	---
AIMP (Anchored Interplanetary Monitoring Platform)	Lunar Orbit	None	Spin stabilized	Small (80 lb wt) solid propellant velocity motor (TE458)	None	---

Two significant general conclusions can be drawn from Figures 2 and 3, and Table 3. First, the existing systems shown are characterized by modest size and thrust levels (i.e., past automated spacecraft requirements have not led to development of large costly propulsion units). The moderate size of these systems is also reflected in the fact that advanced propellants have not been required. With one exception, the systems shown in Figures 2 and 3 use either liquid monopropellants (e.g., hydrazine) or solid propellants, both of which have modest I_{SP} values. The one exception shown is the Lunar Orbiter, which used NTO-Aerozine 50.

The second general conclusion is that there has been a significant amount of fortuitous cooperation between spacecraft propulsion groups as evidenced by the fact that several of the propulsion systems involved have been used on different spacecraft. Examples include the Ranger VII and Mariner II, the Gemini and Surveyor, and the Apollo and the Lunar Orbiter. This cooperation is fortuitous in the sense that there was no single central spacecraft development group that could plan and enforce cooperation between various spacecraft developers, yet, lacking this overall central planning, the propulsion systems that were developed proved to be usable on several spacecraft. In addition, the largest motor shown (i.e., the TE364 Surveyor retromotor) is now used in the Burner II, which has been used on Atlas and Thor for several USAF launches, and is now integrated as spin-stabilized final stage on TAT/Delta for future NASA missions.

Briefly, the current and past situation in spacecraft propulsion is characterized by the following:

(1) The limited number of propulsion systems developed as needed by the individual spacecraft groups

(2) The modest size of these units

(3) The general use of low energy (i.e., low I_{SP}) propellants

(4) The effective multiple use of a number of existing spacecraft propulsion systems including some utilization as upper launch vehicle stages.

LIKELY FUTURE TRENDS IN SPACECRAFT PROPULSION

There are two possible future developments which could produce major changes in the spacecraft propulsion picture outlined in the previous section. The first of these is a possible requirement for the development of large chemical spacecraft propulsion systems. Table 4 presents a list of some possible future automated space missions requiring high total impulse spacecraft propulsion systems. These data are from the 1967 NASA OSSA Prospectus (June edition) and were supplied to a contractor for a current NASA-sponsored study of spacecraft propulsion propellant selection.[19] Table 5 presents some preliminary estimates of the required propulsion system weight for various propellant choices for each of these missions.[19] Figures 4 and 5 show the total impulse and system thrust and weight, respectively, for these propulsion systems and for the systems shown previously in Figures 2 and 3.[2,5,6,8-19]

TABLE 4. POSSIBLE FUTURE AUTOMATED SPACE MISSIONS REQUIRING HIGH TOTAL IMPULSE SPACECRAFT PROPULSION SYSTEMS(19)

Mission	Propulsion Function	Mission Year	Payload (lb)	ΔV (ft/sec)	Nominal Thrust (lb)
Mars Orbiter	Orbit Injection	1973	8143	6950	8000
Venus Orbiter	Orbit Injection	1977	7000	13500	8000
Jupiter Orbiter	Orbit Injection	1981	2000	7600	2000
Saturn Orbiter	Orbit Injection	1984	2000	6000	2000

TABLE 5. PRELIMINARY ESTIMATES OF REQUIRED SPACECRAFT PROPULSION SYSTEM WEIGHTS FOR SOME POSSIBLE FUTURE AUTOMATED SPACE MISSIONS(19)

	Propulsion Module Weight (lb)		
Mission	Earth Storable (N_2O_4/A-50)	Space Storable (Flox/CH_4)	Deep Cryogenic (F_2/H_2)
Mars Orbiter	11,720	8,600	8,150
Venus Orbiter	21,010	13,430	12,410
Jupiter Orbiter	5,840	4,430	4,750
Saturn Orbiter	4,530	3,620	4,280

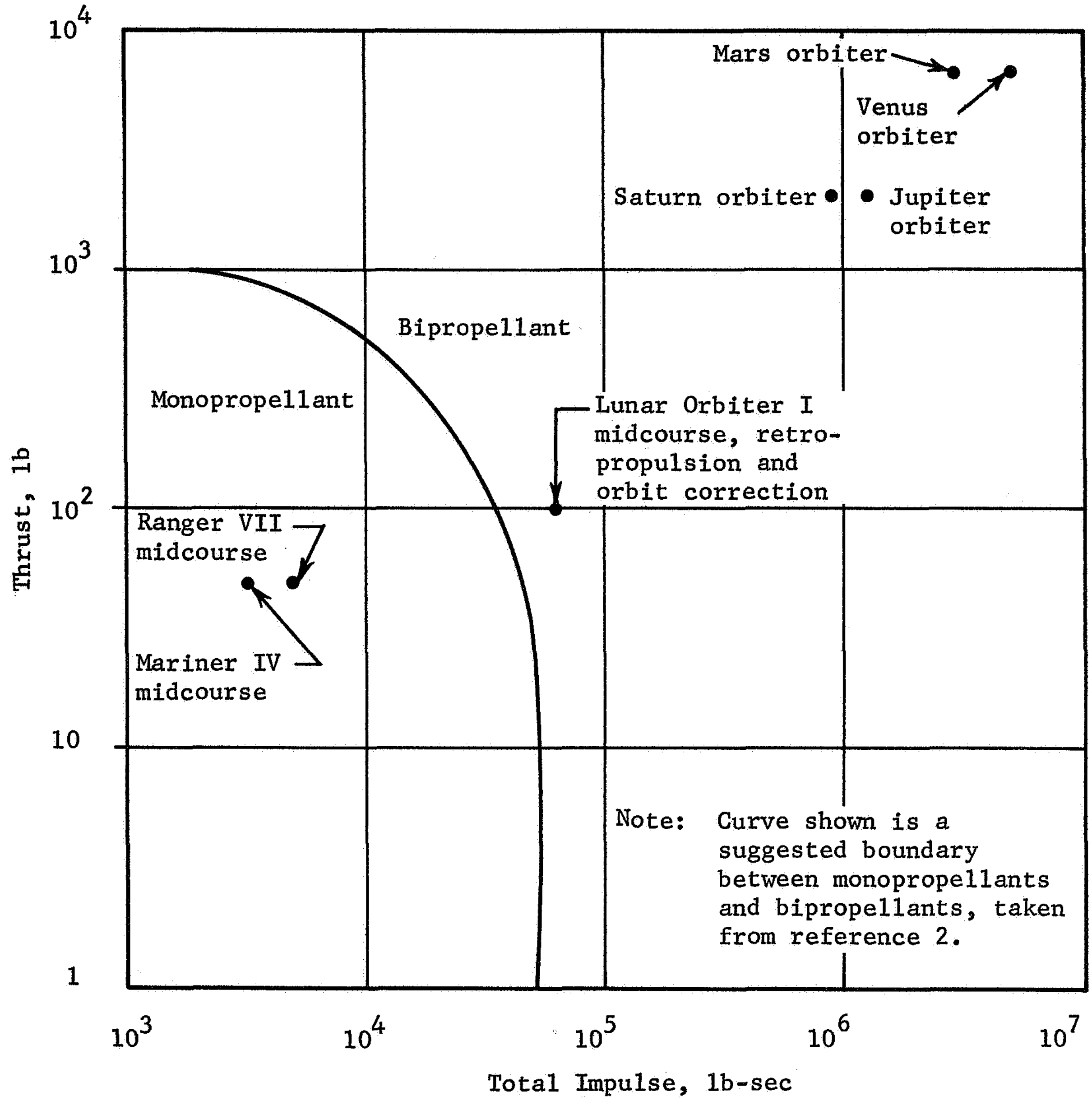


FIGURE 4. THRUST AND TOTAL IMPULSE LEVELS FOR SOME EXISTING AND POSSIBLE FUTURE AUTOMATED SPACECRAFT PROPULSION SYSTEMS[2,5,8-10,12-15,19]

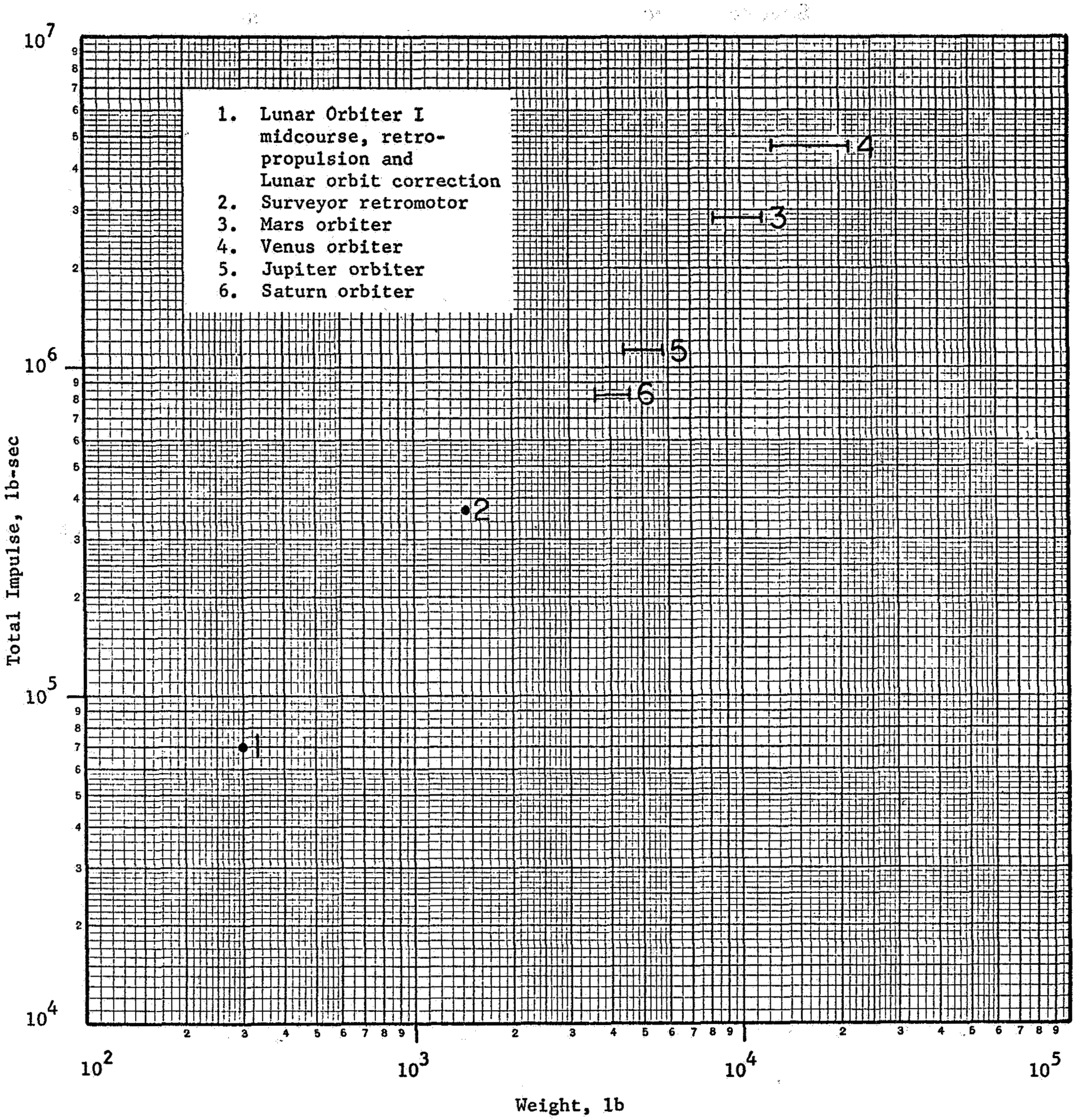


FIGURE 5. WEIGHT AND TOTAL IMPULSE LEVELS FOR SOME EXISTING AND POSSIBLE FUTURE AUTOMATED SPACECRAFT PROPULSION SYSTEMS(14,15,17,19)

Several conclusions can be drawn from these Tables and Figures. First, the propulsion systems required for missions like the Mars, Venus, Jupiter, and Saturn orbiters would be roughly an order of magnitude larger in total impulse, weight, and thrust than the largest automated spacecraft propulsion system used to date (the Surveyor retromotor). Second, if it is desired to hold the weight of these propulsion systems to reasonable levels so as to conserve usable payload capability, high energy propellants not previously used for spacecraft propulsion will have to be employed.

The second possible near future development that threatens to disrupt the development control situation outlined in the previous section is the introduction and operational use of solar powered* electric propulsion systems.[20] Although the economic implications associated with introducing these systems are far from clear at present, it is reasonable to assume that the previous argument that high specific impulse implies system complexity which, in turn, implies higher costs would hold here also. In addition, the use of these systems to provide a portion of the prime propulsion would have a far greater feedback on launch vehicle requirements than that exerted by any spacecraft propulsion system to date. Table 6 illustrates the launch vehicle and spacecraft characteristics for several possible missions for which combined solar electric--chemical propulsion systems have been proposed.[21,22]

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* Nuclear powered systems could have a similar effect in a later time period.

TABLE 6. DATA ON POSSIBLE FUTURE COMBINED CHEMICAL SOLAR-ELECTRIC PROPULSION MISSIONS

Mission	Net Payload (lb)*	Flight Time (days)	Gross Spacecraft Weight (lb)	Launch Vehicle	Launch Vehicle V_C, actual and required ballistic** (ft/sec)	Propulsion System I_{sp} (sec)	Electrical Power at Destination (kw)
Jupiter Flyby	320	500	2,672	SLV3X/Centaur	39,400 49,800	3,010	~ 1
Mercury Flyby	4045	150	13,000	Titan IIIX (1207)/Centaur	38,500 47,800	3,170	~ 60
0.1 a.u. Solar Probe	1520	639	5,200	Titan IIIX (1205)/Agena	37,500 66,600	3,500	~ 30

* Net Payload = Gross spacecraft weight minus propulsion system, propellant and associated structure.

** V_C actual is the characteristic velocity equivalent to the spacecraft's actual injection velocity.
V_C required ballistic is the characteristic velocity required to perform the mission ballistically.
The ballistic flight time would be less than the indicated flight time for the second and third missions.

Several characteristics of these systems which are of concern in space transportation system planning are immediately evident. First, the spacecraft propulsion system would provide a substantial percentage of the required energy. For the cases shown, the equivalent ΔV_C provided by the electric system (the difference between V_C required ballistic and V_C actual) ranges from ~ 10,000 to 30,000 ft/sec, or from about 20% to nearly 50% of the total propulsion energy requirement. The existence of such spacecraft propulsion systems would clearly have a strong effect on requirements for launch vehicles, particularly on the requirements for upper stages.

Second, the interface between the launch vehicle and spacecraft is no longer clearly defined. Not only does the spacecraft propulsion system make a substantial energy contribution, but it also interacts strongly with the operations of the spacecraft since it would likely be active over a substantial fraction of the spacecraft's lifetime, and there could be joint usage of power from the same source.

Third, solar electric propulsion systems have potential for reducing planetary approach velocities. Since the solar electric systems would operate throughout most of the mission, there is a substantial amount of freedom available in controlling thrust history and, therefore, trajectory shaping. For those missions involving high planetary approach velocities when performed with a ballistic trajectory (e.g., Mercury), solar electric systems can be used to reduce substantially the planetary approach velocity and, thus, reduce the terminal retro ΔV requirement.[23]

In summary, probable future spacecraft propulsion systems will likely differ substantially from present and past systems with regard to the following:

(1) Required total impulse (as well as system thrust and/or weight) could be an order of magnitude or more larger.

(2) System specific impulses (and, thus, system complexity and cost) can be expected to be higher to satisfy the higher total impulse requirements.

(3) These systems (particularly, with the solar electric, and chemical planetary orbiter and lander systems) could provide a substantial portion of the total mission propulsion energy requirements.

(4) The interaction and potential interferences of the propulsion system with other spacecraft subsystems could be greatly increased, because of the increased size and possible long operating or residence time.

(5) Large planetary approach velocities (larger orbiter propulsion ΔV's) could be substantially reduced by trajectory shaping with solar electric systems.

IMPLICATIONS TO ADVANCE PLANNING FOR SPACE TRANSPORTATION SYSTEMS

The implications of current trends in spacecraft propulsion to advance planning for space transportation systems follow in a relatively straight manner from the discussions of the previous section. The size and specific impulse of the anticipated propulsion systems imply a costly development effort. Such systems can no longer be classed as "secondary" propulsion. They will require a major development program, and should not be developed without reference to other possible spacecraft and launch vehicle propulsion requirements. It is highly desirable, for example, that the current basic and exploratory research projects supported in advanced propulsion be selected with care so as to provide a basis that can eventually be of use in developing spacecraft and launch vehicle propulsion systems of benefit to the overall NASA program effort. To know what areas will be of potential use requires that the new spacecraft propulsion technologies be examined from an overall NASA mission requirements viewpoint rather than from the narrow perspective of a single program or project's needs or desires.

As an example of this requirement for overall NASA requirements planning, consider the propulsion modules shown in Table 7. These module designs were developed during the previously mentioned NASA-sponsored study of spacecraft propulsion propellant selection.[19,24] They are for the Mars orbiter mission shown in Table 4 and are based on more detailed design studies than were used for the preliminary estimates shown in Table 5. The modules are of sufficient size and use energetic enough propellants to be potentially of interest as upper stages for launch vehicles. Thus, the possibility exists that modules of these types would be potentially adaptable for dual (orbiter retropropulsion and launch vehicle) usage*, with possible

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* The concept of using a single propulsion module for both spacecraft and launch vehicle applications is not new, however. A design using Earth storable propellants was proposed in Reference 22, for example. As noted before the TE364 Surveyor retromotor is used in Burner II and on Delta.

TABLE 7. REPRESENTATIVE MARS ORBITER PROPULSION MODULE WEIGHTS(24)

Subsystem Weight (lb)	N_2O_4/A-50 Pump Fed	Flox/CH_4 Pump Fed	OF_2/B_2H_6 Pressure Fed	F_2/H_2 Pump Fed
Structure				
Base Structure	36	178	175	72
Meteoroid Panels	120	--	--	78
Internal Structure	41	50	48	--
Tank Supports	101	110	108	69
Engine Support	--	--	--	91
Attachments, Etc.	27	27	27	27
Bulkhead Insulation	45	45	45	45
Propellant Feed Assembly				
Tanks	301	290	294	311
Valves and Filters	32	55	55	51
Insulation	18	73	52	233
Meteoroid Bumper	--	75	87	71
Pressurization System	24	59	105	36
Engine System	158	152	384	152
Contingency	90	114	138	125
Residuals				
Propellant	139	91	82	86
Vapor Weight	4	77	49	126
He-Gas	2	11	19	5
Performance Reserve	137	76	89	73
Propellants	8,260	6,485	6,591	5,587
Propulsion Module Weight	9,535	7,968	8,348	7,238
	Specific Impulse (sec)			
	335	410	414	468

overall NASA cost savings. It could be wasteful, therefore, if such dual applications were not examined so that future design and technology development decisions could be based on a wider mission requirement model.

Launch vehicle Earth escape stage and other applications (orbiters of Venus and Jupiter, Lunar cargo missions) of modified* versions of the Mars orbiters of Table 6 are presently being examined as a part of the second phase of the spacecraft propellant selection study.[26] The results of that examination were not completed at the time this report was prepared. Therefore, in response to an earlier suggestion from OART[27], a preliminary examination was made of the possibility of adapting the Mars orbiter module designs of Table 7 for use as upper stages on OSSA launch vehicles.

Table 8 shows the preliminary estimate of the characteristics of stages based on the Mars orbiter propulsion modules. For all four stage designs, the orbiter module meteoroid protection was eliminated and the insulation reduced to be compatible with launch site hold and a 2-hour coast requirement. Guidance and telemetry systems, stage power supplies and attitude control systems were added. The guidance and telemetry system weights were obtained from recent kick stage guidance studies performed at Battelle's Columbus Laboratories for the NASA Electronic Research Center under NASA Contract NAS12-550. The weights for the stage electrical and attitude control systems were based on previous kick stage design experience.[28-30]

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* Modifications include redesign to a 10-ft maximum diameter to allow use of the proposed Titan Centaur launch vehicle.

TABLE 8. STAGE DESIGN WEIGHT BASED ON MARS ORBITER PROPULSION MODULES

Stage Subsystem Weight, (lb)	N_2O_4/A-50 Pump Fed	Flox/CH_4 Pump Fed	OF_2/B_2H_6 Pressure Fed	F_2/H_2 Pump Fed
Structure				
Base Structure	36	178	175	72
Internal Structure	41	50	48	--
Tank Supports	101	110	108	69
Engine Support	--	--	--	91
Attachments	27	27	27	27
Bulkhead Insulation	45	45	45	45
Propellant Supply System				
Tanks	301	290	294	311
Valves & Filters	32	55	55	51
Insulation	--	18	13	60
Pressurization System	24	59	105	36
Engine System	158	152	384	152
Contingency	90	114	138	125
Residuals				
Propellant	139	91	82	86
Vapor Weight	4	71	49	126
He-Gas	2	11	19	5
Performance Reserve	137	76	89	73
Propellants	8,260	6,485	6,591	5,587
Guidance System				
IMU & Computer	70	70	70	70
Batteries	10	10	10	10
Wiring Harness	20	20	20	20
Telemetry, Tracking, Command & Control	20	20	20	20
Environmental Control	15	15	15	15
Stage Electrical System	50	50	50	50
Attitude Control	50	50	50	50
Stage Weight	9,632	8,067	8,457	7,151
Stage Burnout Weight	1,372	1,582	1,866	1,564
Interstage (to lower stage)	700	700	700	700
σ_p = Propellant Fraction				
	.858	.804	.779	.781
Isp (sec)				
	335	410	414	468

Figure 6 shows the performance of the stages of Table 8 when used on the Titan IIIX(1205)/Centaur. Based on these data, it appears that the concept of using modified spacecraft orbiter modules as launch vehicle stages may have some validity if the spacecraft modules use space storable or deep cryogenic propellants. As might be expected, the performance of the module-based stages is less than that of stage designs intended to be used as stages only.* The potential cost savings due to dual usage might be sufficient to overcome this reduced performance and make the system cost-effective.

On the bases of current trends it is anticipated that there will be strong interaction between launch vehicle, solar electric, and conventional spacecraft planning. The ability of a solar electric propulsion system to supply a large percentage of a total mission energy requirement will obviously have a strong effect on future needs for launch vehicle upper stages and chemical spacecraft propulsion systems. Similarly, the degree to which existing and potential upper stages and chemical spacecraft propulsion can provide an increased mission capability (even though it may be a lesser capability than might be provided by solar electric systems) will strongly influence the willingness of spacecraft designers to risk the use of a new technology such as solar electric propulsion. Thus, future planning in either area must include consideration of the effects of possible future developments in the other.

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* The Kick stage performance shown in Figure 6 is the same as that shown in Reference 22. It is based on H_2-F_2 propellants, an assumed propellant fraction of .85 and a specific impulse of 455 seconds.

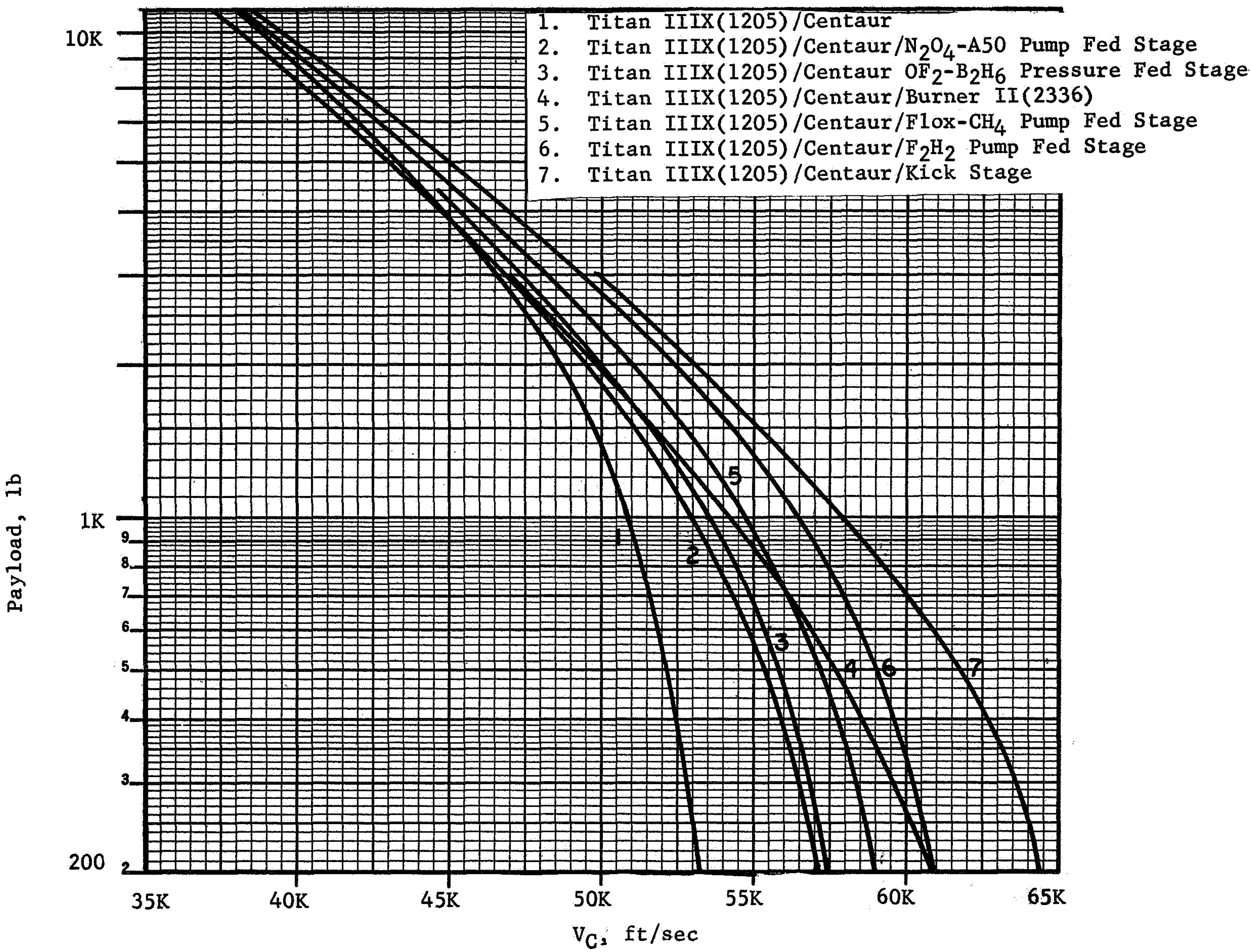

FIGURE 6. PERFORMANCE OF STAGES BASED ON ORBITER MODULES ON TITAN IIIX(1205)/CENTAUR

CONCLUSIONS

The previous sections have detailed the need and benefits of coordinated planning in the area of future launch vehicle and automated spacecraft propulsion developments. A certain amount of such coordination already exists. It is recommended that this coordination be enhanced and that, for this purpose, consideration be given to the establishment of a working group consisting of representatives of the launch vehicle planning, spacecraft planning, and propulsion technology groups within NASA.

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APPENDIX A

SUMMARY OF SPACECRAFT PROPULSION TECHNOLOGY

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SUMMARY OF SPACECRAFT PROPULSION TECHNOLOGY

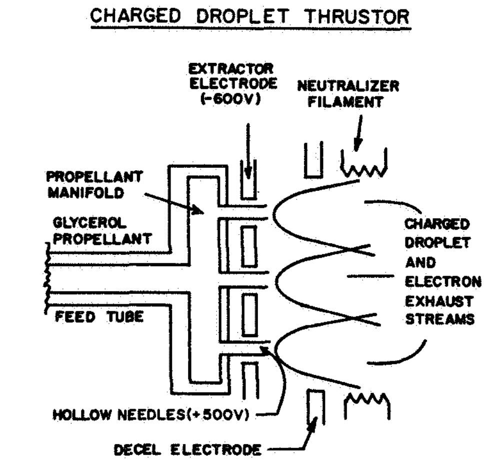
There is an almost bewildering variety of propulsion systems available or potentially available for spacecraft propulsion applications. Existing spacecraft propulsion systems already use more different energy sources and propellants than are used for launch vehicle or primary propulsion. Figure 1 of the main text suggests the range of application of some of the various spacecraft propulsion propellants.

It would be neither possible nor desirable to include detailed technical discussions of all the various spacecraft propulsion systems in this Appendix since volumes of information on this subject have already been published. Instead, an overview of the technology combined with references to more detailed sources will be offered here.

Following the procedure used in Reference A-1, a distinction will be maintained between micropropulsion (10^{-6} to 1.0 lb thrust) and what will be termed nominal propulsion (thrust$>$1.0 lb). As can be noted from Table 1 of the main text, the primary present applications of micropropulsion in automated spacecraft missions are stationkeeping, attitude control and orbit correction. The nominal systems are applicable to midcourse corrections, Earth orbit injection, orbiter retropropulsion and planetary landers. Spacecraft using solar electric primary spacecraft propulsion systems could have thrustors ranging from ~0.1 to 10 lb thrust, and could thus fall into either class.

Figure A-1 (taken from References A-1 to A-4) illustrates some micropropulsion systems. Some nominal propulsion systems are illustrated in the Figures of Appendix C.

Some of the various micro and nominal propulsion systems will now be described briefly.



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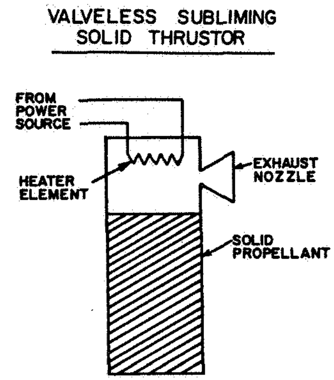

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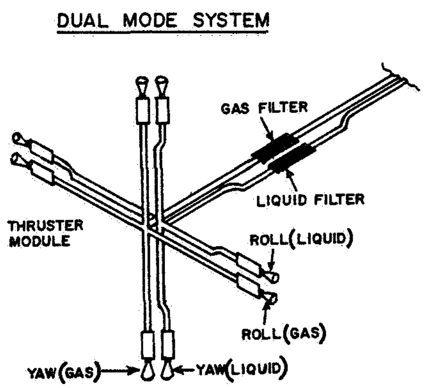

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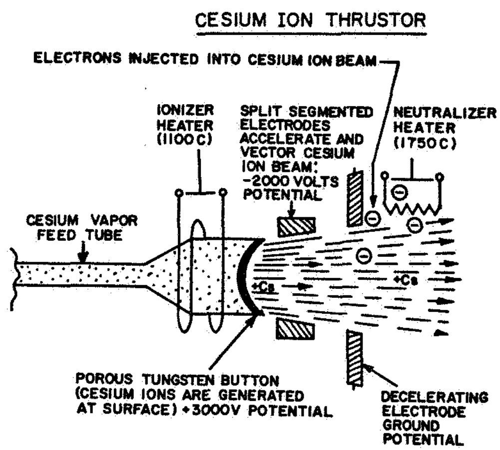

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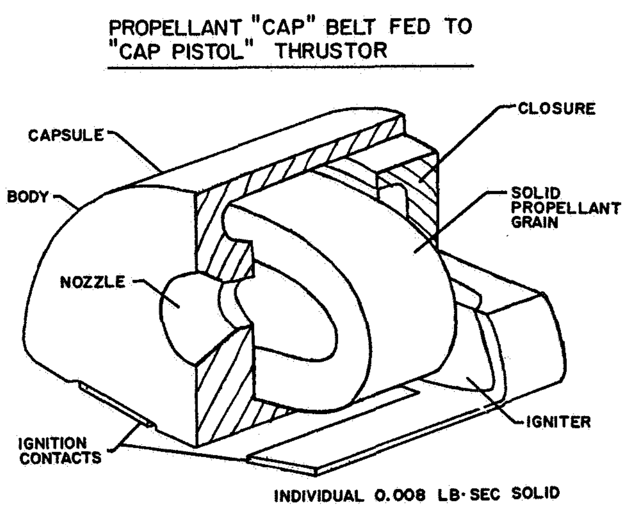

Based on Reference A-4.

FIGURE A-1. SCHEMATICS OF REPRESENTATIVE MICROTHRUST SPACECRAFT PROPULSION SYSTEMS

SELECTED SYSTEM DESCRIPTIONS

Subliming Solids

This system has many variations (valved or valveless, mono or bipropellant), one of which is illustrated in Figure A-1. The basic principle involved in all the variations is the use of a solid propellant which sublimes to form a low molecular weight vapor. The vapor pressure provides the chamber pressurization and the vapor is used as a propellant. The rate of sublimation (and, thus, the thrust) is controlled by the thermal input to the solid propellant. Typical thrust levels are $\sim 1.0 \times 10^{-5}$ lb, with a vacuum specific impulse of 60-70 sec, a system total impulse to weight* ratio of 60 lb-sec/lb, and a typical total impulse of 500 lb-sec.[A-1,A-5]**

Vaporizing Liquid

The basic principle for this system is the same as for the subliming solid. Again, a liquid with a low molecular weight vapor is required. Because of liquid containment requirements, a separate storage tank and thrust chamber are required. This extra weight is offset by the lower (compared to solid propellants) molecular weights of the two most commonly used propellants (NH_3 and H_2O). Thrust levels are again $\sim 10^{-5}$ lb, at a vacuum I_{SP} of up to 100 sec, a system total impulse to weight ratio of ~80 lb-sec/lb, and a typical total impulse of 5,000-10,000 lb-sec.[A-1]

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* System weight does not include raw power supply (~10 kilowatts/lb of thrust.

** Superscript numbers refer to References shown at end of this Appendix.

Cap Pistol

This system employs a belt fed set of "caps" which are actually miniature motors, complete with propellant, igniter, and exhaust nozzle. A single cap is shown in Figure A-1. These caps are fed into a thrustor housing where they are fired electrically to produce small impulse bits (typically around .08 lb-sec/bit).[A-4] The specific impulse is in the 190-210 sec range, with a typical average thrust of ~.5 lb. Typical total impulse to weight ratios are 100 lb-sec/lb at higher total impulse levels (1,000-3,000 lb-sec). For smaller total impulses, the ratio may drop to 1-10 lb-sec/lb.

Stored Gas

Three types of propulsion systems using stored gas propellants have been studied. In all these systems, the propellants are stored under high pressure (~300 psia). In cold gas systems, the propellant is exhausted directly through a nozzle to produce the thrust. In a hot gas system, the gas is heated electrically before being exhausted through the nozzle. In bipropellant gaseous systems, two gaseous propellants are used which are burned in a combustion chamber prior to exhausting. Table A-1 lists the characteristics of stored gas systems.

TABLE A-1. PERFORMANCE CHARACTERISTICS OF STORED GAS PROPELLANT SYSTEMS[A-1,A-5]

System Type	Typical Propellant(s)	Typical Specific Impulse (sec)	Typical Thrust Level (lb)	Typical Total Impulse to Weight Ratio (lb-sec/lb)	Typical Total Impulse (lb-sec)
Cold Gas	N_2	60-80	1.0	25	500
Hot Gas*	N_2H_4	95-150	1.0	25	500
	NH_3	100-250	.1-.5	150	5000
Gaseous Bipropellant	OF_2/CH_4 F_2/C_2H_4 O_2/H_2	350-400	.1-.5	240	50000

* This system also requires a power input of ~20 kw/lb of thrust.

Liquid Monopropellant

This system uses the decomposition of the propellant (augmented by a catalyst) to heat the propellant. Examples of such systems are discussed in Appendix C. The predominant propellant is hydrazine (N_2H_4), although hydrogen peroxide (H_2O_2) was used in earlier applications. Typical thrust levels range from 0.1 lb (altitude control) to 50 lb (midcourse motors). Hydrazine lander propulsion systems with thrust levels of 400 to 1000 lb thrust are currently being studied.[A-6] Typical specific impulses for H_2O_2 are 160-165 sec, and 235-245 sec for hydrazine. Typical total impulse to weight ratios are 120 lb-sec/lb for H_2O_2 and 180 lb-sec/lb for hydrazine. Typical total impulse levels are 10,000 lb-sec for hydrozen peroxide and 10,000-60,000 lb-sec for hydrazine.

Earth Storable Liquid Bipropellant

The term Earth storable propellants refers to those propellants that remain liquid at Earth surface temperatures (See Table A-2). The most commonly used Earth storables are nitrogen tetroxide (N_2O_4 or NTO) and Aerozine 50, a mixture of 50% hydrazine and 50% unsymmetrical dimethyl-hydrazine (UDMH).*

Typical specific impulses for Earth storables are 310-335 sec. (See Table A-3).[A-10] Typical propulsion module weights for Earth storable propellants are shown in Figure A-2.[A-11] The module weights are presented as a function of the payload (spacecraft weight), the required ΔV, and the module specific impulse. As an example of the use of Figure A-2, consider the 2 x 20 planetary radii orbit of Venus Shown in Table B-3. The required

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* Other common Earth storables are N_2O_4-UDMH and inhibited red fuming nitric acrid (IRFNA)-UDMH.

TABLE A-2. NOMINAL PROPELLANT LIQUID STATE TEMPERATURE RANGES(A-7-A-9)

Propellant Type	Oxidizer	Fuel	Freezing Point, °R	Boiling Point, °R (at 200 psia)
Monopropellant (hydrazine)	--	--	495	890
Earth Storable	IRFNA		396	610
	N_2O_4		472	650
		MMH	395	660
		UDMH	389	606*
		A-50	479	619
Space Storable	OF_2		89	316
	O_2		98	224
	F_2		96	215
	Flox(70-30)		98	218
		CH_4	163	282
		B_2H_6	184	447
		NH_3	352	431
Deep Cryogenic	O_2		99	224
	F_2		96	215
		H_2	25	60

* At 14.7 psia.

NOTE: Temperatures are rounded to nearest degree.

TABLE A-3. TYPICAL SPECIFIC IMPULSE VALUES FOR LIQUID BIPROPELLANTS (A-7-A-10)

Propellant Type	Oxidizer	Fuel	Theoretical Vacuum I_{SP} (sec)*
Earth Storable	N_2O_4	N_2H_4	342
		A-50	339
	Nitric Acid	UDMH	320
Space Storable	OF_2	CH_4	405
		B_2H_6	426
	Flox	CH_4	418
	F_2	B_2H_6	431
		NH_3	417
Deep Cryogenic	F_2	H_2	473
	O_2	H_2	454

* P_C = 100 psia, ϵ = 40, shifting equilibrium.

NOTE: Delivered I_{SP}'s fall below the theoretical values quoted in this table. Typical working values would be

		Pressure Fed		Pump Fed
N_2O_4/A-50	-	300-310	-	310-320
OF_2/B_2H_6	-	390-410	-	--
Flox/CH_4	-	--	-	400-410
O_2/H_2	-	--	-	440-445
F_2/H_2	-	--	-	455-460

All of these values are for near-term technology. By the mid 1980's, values such as 470 and 490 seconds for H_2/O_2 and H_2/F_2, respectively, may be attainable at chamber pressures greater than 100 psia and expansion ratios greater than 40.

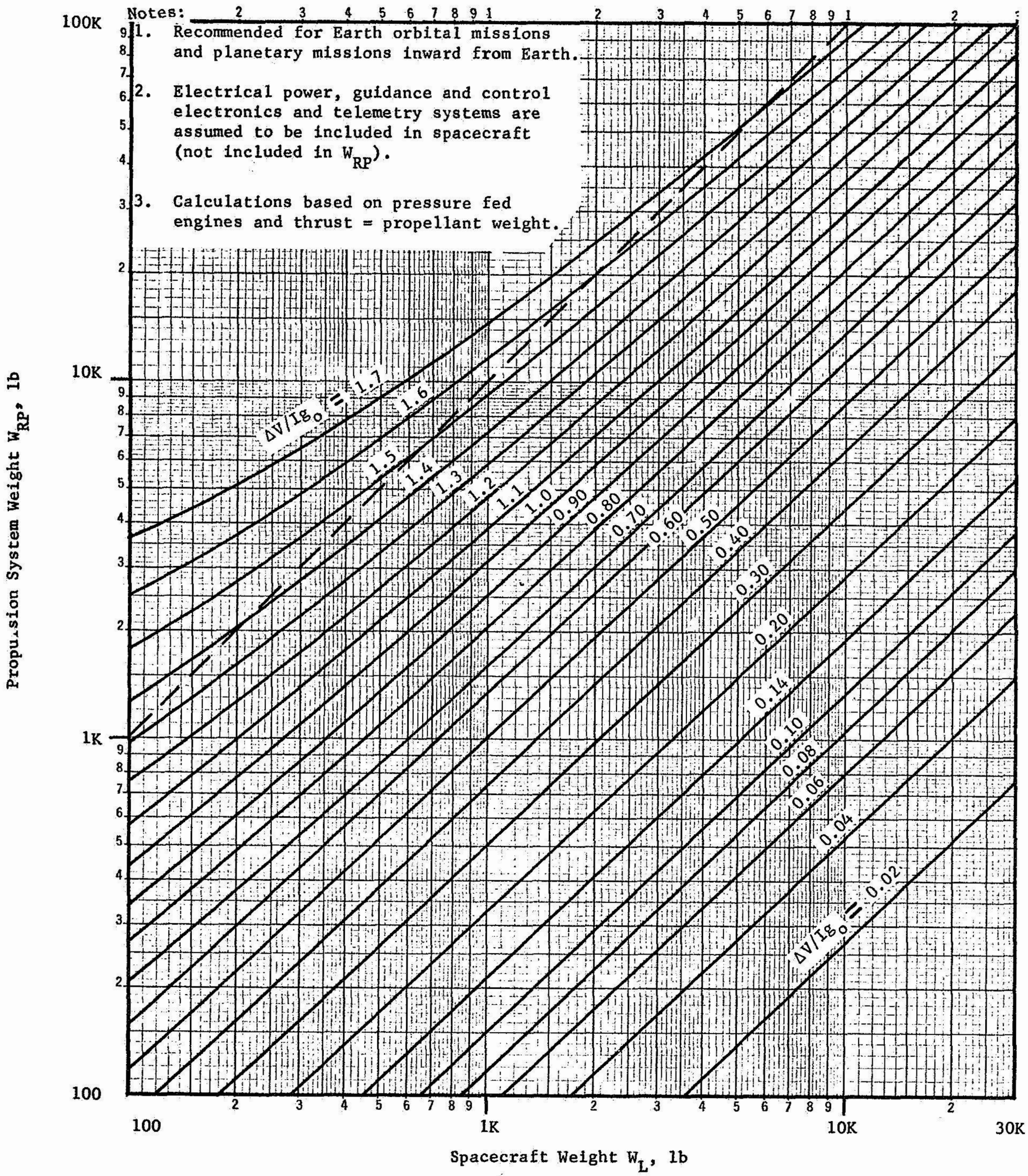

FIGURE A-2. SPACECRAFT PROPULSION SYSTEM WEIGHT USING EARTH STORABLE PROPELLANTS(A-11)

ΔV for this mission is 5,400 ft/sec. If the required payload in Venus orbit were 1000 lb and a 310 sec specific impulse, NTO-Aerozine 50 module were used, the orbiter propulsion module weight could be calculated as follows. The value of $\Delta V/Ig_o$ is

$$\frac{(5,400 \text{ ft/sec})}{(310 \text{ sec})(32.2 \text{ ft/sec})} = .525$$

From Figure A-2, for a 1000 lb spacecraft, this mission would require a propulsion module slightly in excess of 1100 lb.

It should be noted that the data of Figure B-2 are based on the module being used for Earth orbital or inner planetary missions. Missions to the outer planets would require increased module insulation and thermal control (resulting in heavier modules than indicated by the Figure) to prevent propellant freezing.

Space Storable Liquid Bipropellant

Space storable, as used here, applies to propellants that remain liquid between roughly 100 to 400°R* (See Table A-2). The interest in these propellants results from the facts that their performance is nearly that of the deep cryogenics while their bulk densities are nearly as high as the Earth storables, and that required storage temperatures can be provided passively in near-Earth space.

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* i.e., mild cryogenics.

This is a current intensive NASA OART program for developing the technology of space storable propellants.[A-12] The program is concentrating on two space storable combinations--OF_2/diborane (B_2H_6) and Flox (fluorine and oxygen mixture)/methane (CH_4). The OF_2/diborane work is being conducted primarily through JPL, while the Flox/methane work is concentrated at the Lewis Research Center.

Typical specific impulses for space storables are 400-430 sec. Typical propulsion module weights for space storables are shown in Figure A-3.[A-11] This Figure is analogous to Figure A-2, and is used in the same manner as Figure A-2 was in the example of the previous discussion.

The data of Figure A-3 are based on the module being used for outer planetary missions (i.e., Mars and beyond). Earth orbital or inner planetary missions would require increased insulation and thermal control (and thus increased module weights) to prevent excessive boiloff.

Deep-Cryogenic Liquid Bipropellants

Deep cryogenic propellants, as used here, refer to those propellant combinations employing liquid hydrogen. The two most common combinations are hydrogen-oxygen and hydrogen-fluorine.

Deep cryogenics were not previously regarded as good candidates for spacecraft propulsion applications due to the difficulty of storing the liquid hydrogen. Recent advances in superinsulation and other storage methods have changed this outlook. With improved insulation capability, the inherently high performance of deep cryogenics (I_{SP}'s of 440-470 secs) has tended to overcome the hydrogen storage problems and make the deep

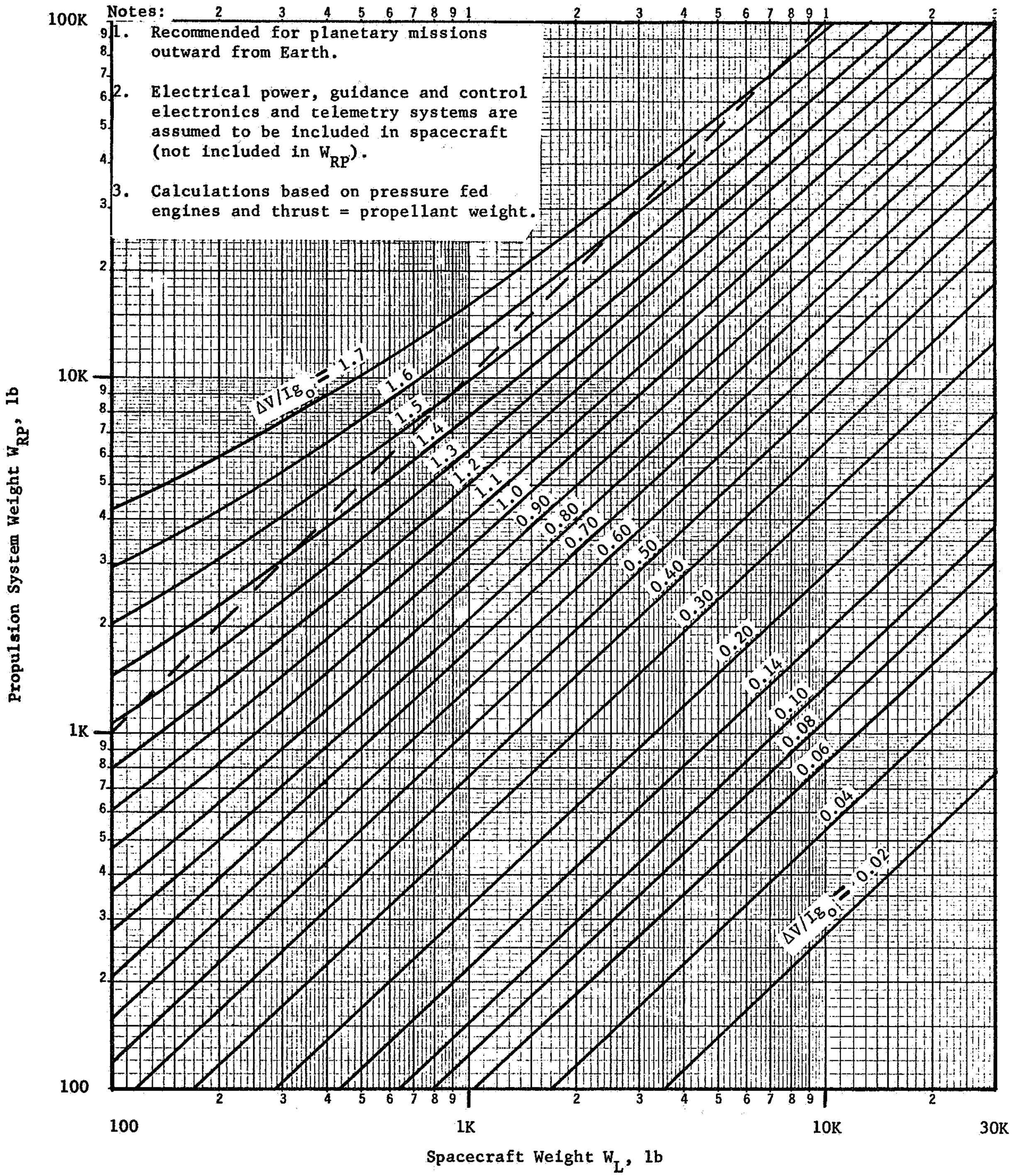

FIGURE A-3. SPACECRAFT PROPULSION SYSTEM WEIGHT USING SPACE STORABLE PROPELLANTS(A-11) cryogenics, particularly hydrogen/fluorine, more competitive. For example, in Reference A-13, the initial design studies showed that hydrogen/fluorine resulted in the lowest module weights for the Mars orbiter missions.

Figure A-4 shows typical propulsion module weights for hydrogen/fluorine propellants. The Figure is analogous to Figure A-2, and is to be used in the same manner as Figure A-2 was in the example given in the Earth storable propellants section. The data of Figure A-4 are based on the module being used for outer planetary missions (Mars and beyond).

Solid Propellant

Solid propellant systems are of interest for spacecraft propulsion applications because of their high propellant densities and good storability. These advantages are counterbalanced by the generally low specific impulses available and by problems encountered in applications requiring precise or repeated energy management.

There is a wide variety of solid propellant combinations available. The general I_{SP} range of conventional propellants is 260-290 sec although there is a series of high energy propellants being developed with I_{SP}'s up to 320 sec. Typical solid propellant module weights are shown in Figure A-5. The Figure is analogous to Figure A-2 and is used in the same manner as Figure A-2 was in the example shown in the Earth storable propellant section. These modules can be used for outer and inner planetary missions, although they are more commonly restricted to Earth orbital missions, particularly apogee kick motors.

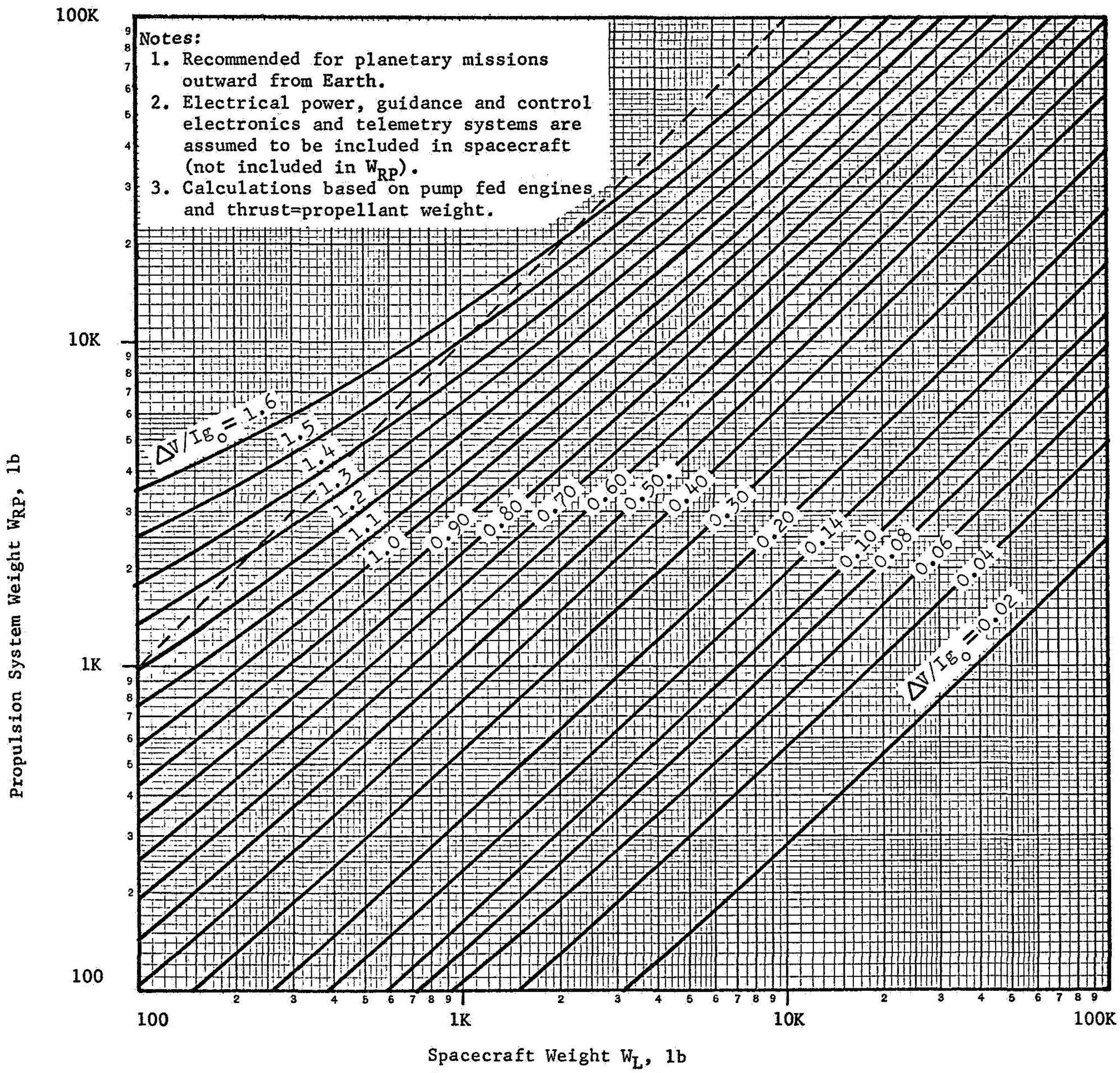

FIGURE A-4. SPACECRAFT PROPULSION SYSTEM WEIGHT USING CRYOGENIC PROPELLANTS (H_2F_2;DENSITY = 33 LB/FT^3)

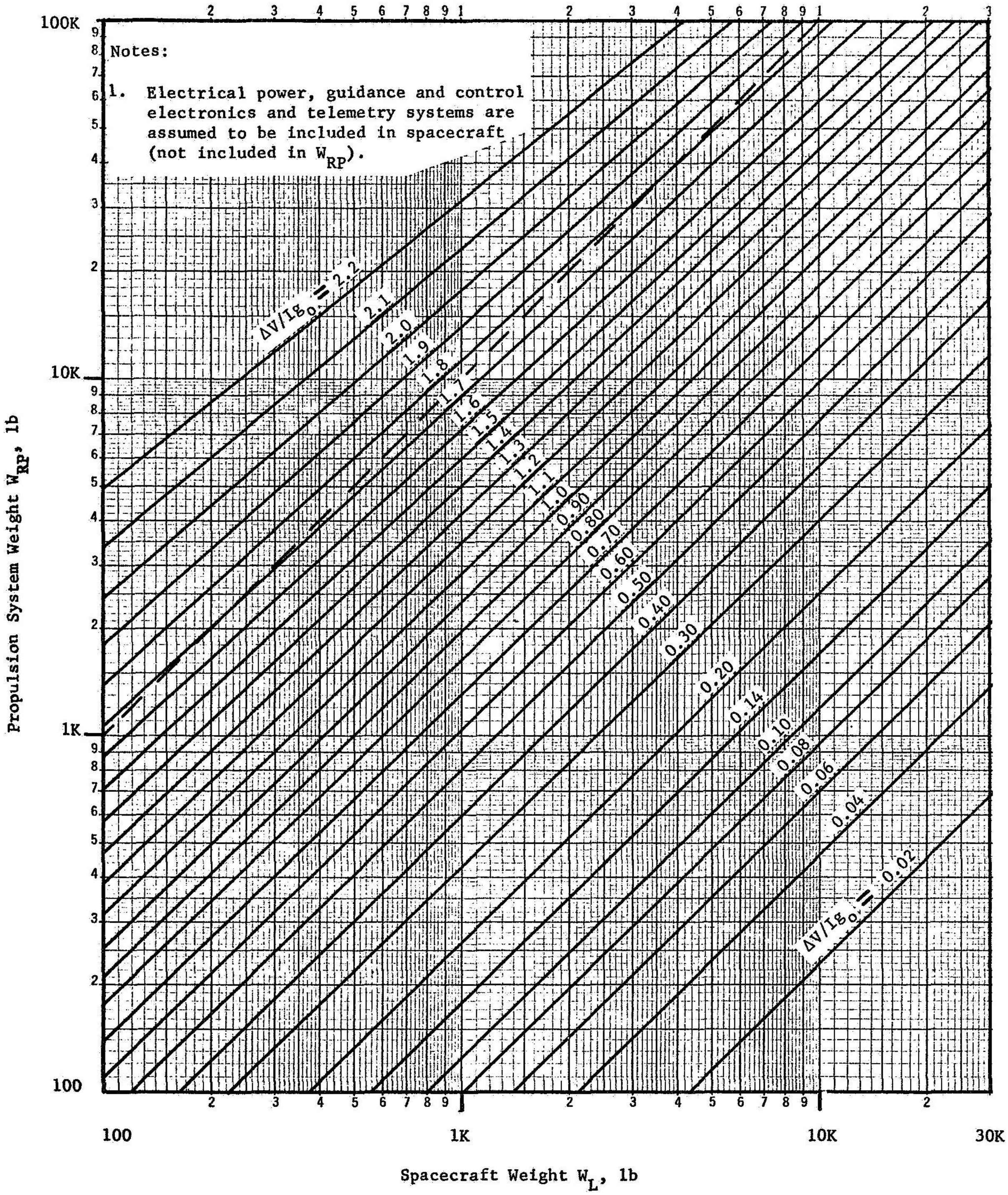

FIGURE A-5. SPACECRAFT PROPULSION SYSTEM WEIGHT USING SOLID PROPELLANTS(A-11)

The data in Figure A-4 are generalized and refer to properties of modules that could be developed. In the case of solid propellant motors, a series of motors have already been developed and used. Table A-4 (taken from Reference A-14) gives the properties (thrust, total impulse, weight, etc.) of some of these motors.

Hybrid Propellants

Hybrid propulsion systems use a thrust chamber in which a solid and a liquid propellant are burned. The most common combination is a liquid oxidizer (e.g., Flox, N_2O_4, N_2H_4) and a solid fuel (e.g., HFX). Although very high specific impulses are theoretically attainable through the use of high energy oxidizers and fuels (e.g., 498 sec for H_2O_2/BeH_2), most hybrid studies to date have concentrated on less energetic combinations (I_{SP}'s from 280-380 sec), emphasizing instead the operational advantages of such systems (e.g., restartability, precise impulse control, throttling capability). The usual practice for hybrid systems is to cast the solid fuel in the combustion chamber, and store the liquid oxidizer in a separate tank. The liquid is then injected into the combustion chamber as required. Because of the greater complexity of hybrids compared to solid propellant motors, the hybrids generally have poorer propellant fractions.* Table A-5 lists typical propellant fractions for propulsion systems using some form of solid propellants.

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* Propellant fraction = propellant weight/propulsion system weight.

TABLE A-4. PROPERTIES OF SOME EXISTING SOLID PROPELLANT MOTORS

Motor	Case Diameter (in.)	Propellant	Total Impulse (lbf-sec)	Average Thrust (lbf)	Maximum Pressure (psi)	Expansion Ratio	Total Weight (lbs)	Application
TE-M385	12.8	TP-L-3014A	14,000	2,150	865	23.0	69.2	Gemini R/A
TE-M-345	13.5	TP-G-3129	18,200	838	607	30.0	81.4	Titan II
TE-M-345-11	13.5	TP-G-3129	17,652	841	607	22.8	79.2	Hitchhiker
TE-M-375	13.5	TP-G-3129	17,300	776	565	60.0	76.0	SynCom I
TE-M-427	13.5	TP-H-3109	17,875	1,066	620	48.2	77.8	--
TE-M-444	13.5	TP-H-3062	21,200	1,370	965	56.0	88.3	--
TE-M-458	13.5	TP-G-3129	18,780	841	560	43.3	80.2	AIMP
TE-M-456-2	15.0	TP-L-3098	24,500	5,000	580	8.0	107.5	Trailblazer
TE-M-479	17.4	TP-H-3062	44,500	2,290	840	58.4	175.4	RAE
TE-M-184-3	25.06	TP-H-3034	129,800	7,600	585	13.8	514.9	Cygnus
TE-M-442	26.0	TP-H-3114	140,000	6,420	785	18.7	602.7	--
TE-M-364	37.0	TP-IL-3062	347,130	8,376	570	53.0	1,315.3	Surveyor
TE-M-364-1	37.0	TP-H-3062	357,000	8,930	561	53.0	1,338.6	Surveyor
TE-M-364-2	37.0	TP-H-3062	402,000	9,248	600	53.2	1,532.6	Burner II
TE-M-364-3	37.0	TP-H-3062	415,440	8,945	613	53.0	1,580.1	Improved Delta
TE-M-364-5	37.0	TP-H-3062	377,000	8,950	576	53.2	1,435.5	Surveyor
TE-M-186-2	40.1	TP-H-3034	443,000	16,800	727	12.8	2,159.8	Cygnus
TE-M-521	17.4	TP-H-3062	71,500	3,850	850	57.9	273.3	IDCS

TABLE A-5. PROPELLANT FRACTIONS FOR PROPULSION SYSTEMS USING SOLID PROPELLANTS(A-15)

Propulsion System Type	Typical Range of Propellant Fractions*
All Solid	.86-.88 at W_P = 200 lb**
	.88-.90 at W_P = 1000 lb
	.90-.92 at W_P = 10000 lb
Hybrid	.81-.85 at W_P = 200 lb
	.84-.87 at W_P = 1000 lb
	.87-.89 at W_P = 10000 lb
Liquid Augmented Solid	.83-.86 at W_P = 200 lb
	.86-.88 at W_P = 1000 lb
	.89-.90 at W_P = 10000 lb

* Propellant fraction = Weight of Propellant/Propulsion System Weight
** W_P = Propellant Weight

Liquid Augmented Solids

Liquid augmented solid propellant propulsion systems are a result of an attempt to retain the desirable features of both hybrid and solid propellant systems. In a liquid augmented system, a small amount of liquid propellant (usually an oxidizer) is injected into the combustion chamber of a solid propellant to serve as a control agent. The solid propellant is designed so that combustion cannot be sustained without liquid augmentation. Thus, the liquid injection can be used to control the restart, total impulse, and throttling of the solid propellant motor. The liquid propellant requires a separate tankage system. However, the total amount of liquid propellant is usually small (a few percent of the solid propellant weight), so that propellant fractions nearly equal to those of solid propellant motors are possible (See Table A-5). Specific impulses naturally tend to be nearly those of solid propellants (although the use of high energy liquid oxidizers such as fluorine can result in I_{SP}'s around 330-340 sec).

Electro-Chemical

The primary current example of an electrochemical system is the electrolysis rocket. In its most common form (called the "water rocket"), electrical energy is used to reduce water to gaseous hydrogen and oxygen which are subsequently ignited by a spark and burned in a conventional thrust chamber when desired. The system has high performance (I_{SP}'s of 350-475 sec) and good propellant bulk density, but does require an electrical power supply (~7 watt hours per lb-sec of impulse). For this reason, the electrolysis rocket is not competitive with conventional chemical propulsion systems at high thrusts and large total impulses. The system is probably most applicable where total impulses up to 100 lb-sec and thrusts in the 0.1-3.0 lb range are required.

Solar Electric Propulsion

Solar electric propulsion systems use arrays of silicon solar cells to generate electrical power which is used to operate a thrustor. There are a great variety of thrustors that have been or are being studied. Basically, all of these attempt to use the electrical energy to heat a propellant or to generate electric and magnetic fields which can accelerate a conductive or charged propellant.

A wide range of specific impulses are available, ranging from 100-800 sec for resistojets, 500-1500 sec for thermal arc jets, 1000-4000 for plasma engines, 2000-10,000 or greater for ion engines. Systems can be designed with thrusts ranging from a few micropounds to several pounds. System specific weights vary with system size. For those systems intended for primary spacecraft propulsion (e.g., systems like those listed in Table 6 of the main text), the power generation and conversion weights dominate the system weights, and specific weights of the order of 10^3 lb per pound of thrust are typical. For smaller systems, specific weights tend to be higher (2×10^5 to 4×10^5 lb per pound of thrust). The power system may be shared with other systems, so that power consumptions of 10-25 kw/lb of thrust are typical for small systems.

Since some of the major differences between possible solar electric propulsion systems stem from differences in the basic nature of the thrustors used, a short description of the major thrustor types will be given. The most highly developed thrustor system is the ion engine, one of which is shown in Figure A-1. Although collective phenomena do exist and influence the performance of the ion engine, the basic principle is the generation of individual ions and their acceleration by an electric field. A variation on this system is the charged droplet thrustor (also shown in Figure A-1). Here the accelerated particles are charged liquid droplets rather than ions. The droplets are produced by electrostatically spraying a liquid (usually glycerol).

Plasma engines are based on the mechanism of a bulk or body force (the Lorentz or $\bar{J} \times \bar{B}$ force) distributed over the working fluid. The propellant is an easily ionized gas with a low molecular weight (e.g., argon). MPD (magneto-plasma-dynamic) engines are the most complicated and least understood of all the electric thrustors. Here, an electric arc is used to heat and ionize the working fluid (normally, hydrogen or ammonia). The fluid is then accelerated by a combination of pressure and magnetic forces.

In the thermal arc jet, an electric arc is used to heat the propellant (also normally H_2 or NH_2) which is then accelerated through a conventional nozzle. The resistojet is similar, the arc being replaced by electrically heated resistors or thermal storage elements.

General Discussion

There are two general sources of more detailed information on spacecraft propulsion technology that are highly recommended. First, for micropropulsion, Reference A-1 is a survey article published 3 years ago that dicusses most of the significant micropropulsion concepts and has a lengthy list of further references. For larger chemical (liquid monopropellant, liquid bipropellant, solid propellant, and hybrid) spacecraft propulsion systems, Reference A-15 is a five-volume summary published in 1968. These volumes, prepared under NASA OART sponsorship, are the most complete and authoritative surveys in the field.

In addition to these surveys, information on some current NASA research on large chemical spacecraft propulsion systems has been collected.(A-16) Table A-6 summarizes this work by technical areas.

A few comments concerning requirements for the introduction of new spacec .ft propulsion systems needs to be made. There are, at this time, a family of existing, flight proven propulsion systems to perform most of the tasks listed in Table 1 of the main text. In general, these systems can be characterized as relatively simple, highly reliable and of low-to-medium performance. Higher performance alternatives to these systems have been suggested or studied. In general, however, spacecraft designers have been reluctant to incorporate such systems in their missions. Their attitude has been that they are not willing to risk reducing the reliability of their spacecraft by the introduction of a new propulsion subsystem unless a significant overall mission advantage could be demonstrated for the new subsystem. The term "overall mission advantage" is the key here. Development and use of new spacecraft propulsion techniques have sometimes been advocated on the basis of propulsion subsystem performance calculations alone, when what is actually required is to relate subsystem performance to overall mission effectiveness.

TABLE A-6. SUMMARY OF SOME CURRENT NASA SPONSORED RESEARCH ON LARGE CHEMICAL SPACECRAFT PROPULSION

Type of Study	Research Organization	NASA Agency	Latest Known Status
Chemical Propulsion - Systems and Applications			
Examination of the Potential Problems Associated with Impingement of High Altitude Exhaust Plumes on Adjacent Surfaces.	Cornell Aeronautical	MSFC	Measurement techniques being examined.
Theoretical Studies of Exhaust Plumes.	General Applied Science Laboratory, (Westbury, L. I.)	MSFC	Computer codes developed.
Wind Tunnel Study of Rocket Exhaust Plumes Using Laser Doppler Measurements.	In House	MSFC	Measurement method verified.
Analysis of Mission Capabilities and Requirements for Vehicles Using Chemical Propulsion. (Spacecraft Propulsion Using Earth Storable, Space Storable, Cryogenic and Solid Propellants are to be studied.)	In House	LeRC	Not available.
Construction of a Spacecraft Propulsion Research Facility to Provide a Simulated Space Environment for Ground Test of Spacecraft Propulsion Systems.	In House	LeRC	Not available.
Study of Pump Fed Low Thrust (100 to 10K lbf) Rocket Engines.	--	MSFC	Contractor not selected.
Design and Evaluation of Propellant Gaging Systems for Small Modules.	--	MSFC	New work.
Study of the Performance of Various Thrustor Systems at Thrust Levels from 10^{-6} to 5 lbsf, For Altitude Control and Stationkeeping.	In House	LeRC	Several thrustors tested.
Experimental Study of Heterogenous Combustion (Slush Hydrogen-Oxygen, Lox-Solids).	In House	LeRC	In progress.
Experimental Study of Throat Inserts for Ablative Chambers for Earth and Space Storable Propellants.	In House	LeRC	Report issued on Earth storables, other work near completion.
Development of Criteria of Merit for Upper Stage/Spacecraft Propulsion Systems (This study also considers electric propulsion systems).	In House	OART-MAD	In progress.
Study of Applicability of Earth Storable, Space Storable, Cryogenic and Solid Propellants to Manned and Automated Planetary Lander and Return Stages.	In House	OART-MAD	In progress.
Propellant Selection for Spacecraft Systems.	Lockheed Missiles and Space Co.	OART(Hdq) (NASw 1664)	Completed
Liquid Monopropellants			
Development of large (400-1000 lb) hydrazine motors for lander applications.	TRW	LVPP/JPL	New work.
Study of Feasibility of High Voltage Ignition of Hydrazine for Rapid Thermal Conditioning of Shell 405 Catalyst Beds.	In House	GSFC	Concept has been demonstrated for cold gas plenum feeds. Demonstration with hydrazine engine is underway.
Study of Monolithic Catalytic Beds for Hydrazine Reactors.	--	JPL	Pending
Monopropellant Torroidal Tank Bellows.	In House	JPL	Active

TABLE A-6. SUMMARY OF SOME CURRENT NASA SPONSORED RESEARCH ON LARGE CHEMICAL SPACECRAFT PROPULSION

Type of Study	Research Organization	NASA Agency	Latest Known Status
Liquid Bipropellants (Earth Storable)			
Experimental Study of Feasibility of Allowing Earth Storable Propellants to Freeze During Non-Propulsive Periods.	In House	MSFC	New work.
Development of Combustion Chamber Liners to Alleviate Combustion Instability in Earth Storable Propellant Motors.	Pratt & Whitney	MSC	Analytical designs completed, experimental studies proceeding.
Study of High Chamber Pressure (up to 4000 PSIA) Combustion of Earth Storable Propellants.	Purdue University	LeRC	In progress.
Development of a Flight weight NTO-MMH 1500-2000 lb Thrust Throttable Engine for Planetary Landers.	In House	JPL	New work.
Development of Earth Storable Liquid Propulsion Components.	In House	JPL	Active
Liquid Bipropellants (Space Storable)			
Experimental Investigation of OF_2 Gels.	Thiokol RMD	LeRC	Report received.
Design and Fabrication of a Small OF_2/B_2H_6 Transportable Propellant Supply Module.	In House	JPL	Not available.
Chamber Technology for Space Storable Propellants.	Rocketdyne	JPL	In progress.
Testing of Ablative Thrust Chamber Liners for Flox.	In House	LeRC	In progress.
Study of Polymeric Expulsion Bladders for OF_2-B_2H_6.	Penisular Chem Research (Gainesville, Florida)	JPL(NAS7-668)	In progress.
Study of Flox-LPG Combustion Instabilities.	In House	LeRC	Hardware designed.
Study of Clogging in OF_2- B_2H_6 Flow.	TRW	JPL(NAS7-549)	In progress.
Chemical Kinetics of OF_2-B_2H_6 Reactions.	In House	JPL	Experimental efforts in progress.
Experimental Study of Flox-LPG Vacuum Ignition.	In House	LeRC	In hardware acquisition phase.
Experimental Study of Ignition and Combustion of Space Storable Propellants.	In House	JPL	Equipment used in earlier NTO-Hydrazine studies is being adapted for this work.
Systems Analysis of Gelled Space Storable Propellants.	Aerojet	JPL(NAS7-473)	Active
Development of a small (1000 lb thrust) Flight Weight OF_2-B_2H_6 Engine.	In House	JPL	Component evaluations in progress.
Development of a Moderate Sized (5-8000 lb thrust) Flox-Methane Breadboard Engine.	Not selected	LeRC	RFP due late 69.

TABLE A-6. SUMMARY OF SOME CURRENT NASA SPONSORED RESEARCH ON LARGE CHEMICAL SPACECRAFT PROPULSION

Type of Study	Research Organization	NASA Agency	Latest Known Status
Liquid Bipropellants (Space Storable) (Continued)			
Low P_c Nozzle Performance With Space Storables.	Rocketdyne	LeRC	Final Report.
Space Storable Regenerative Cooling Study.	Rocketdyne	LeRC	In progress.
Space Storable Regenerative Cooling Study.	Pratt & Whitney	LeRC	In progress.
Space Storable Engine Characterization Study.	Rocketdyne	LeRC	Almost complete.
Space Storable Engine Characterization Study.	Pratt and Whitney	LeRC	Almost complete.
B_2H_6 Storage and Transport.	Callery Chemical	OAST(Hdq) (NASw 1838)	Active
OF_2 Production Methods.	Air Products	OART(Hdq) (NASw 1827)	Active
Slush Methane Characterization.	National Bureau of Standards	OART(Hdq) (NASw 1827)	Active
Foam Stabilized Boundary Injection.	TRW	JPL(NAS7-681)	Active
Design Criteria for Space Storable chambers.	Rocketdyne	JPL(NAS7-304)	Active
Pressurized Gas Systems Perturbations.	IITRI	JPL	Active
Hybrid Pressurization System.	NORAIR	JPL	Active
Space Storable Propellant Acquisition.	Martin	JPL(NAS7-754)	Active
Improvement of Expulsion Bladders for Space Storable Propellants.	TRW	JPL(NAS7-682)	Active
Feed System Interaction Analyses.	TRW	JPL	Pending
Heat Tube Technology for Rocket Engines.	Aerojet	JPL(NAS7-697)	Active
Evaluation of Rocket Engine Materials for Space Storable Propellants.	Aerojet	JPL	Pending
Flox/B_2H_6 Boundary Reactions.	Aerojet	JPL(NAS7-659)	Active
Properties of High Conductivity Alloy for Bladders.	In House	JPL	Active
Advance F_2 Bladders.	Astropower	JPL(NAS7-603)	Active
Refractory Metal Composite Structures.	Astropower	JPL	RFP pending.
Insulation for Rocket Engines.	Rocketdyne	JPL(NAS7-474)	Active
Advanced Pyrolitic Chamber Materials.	Marquardt	JPL(NAS7-555)	Active
Advanced Bladders.	TRW	JPL(NAS7-446)	Active
Advanced Nitroso Rubber Bladders.	Thiokol RMD	JPL(NAS7-451)	Active
Physically Crosslinked Rubber.	Stanford Research Institute	JPL(NAS7-523)	Active
Permeation of Liquids and Gases Thru Teflon.	TRW	JPL(NAS7-505)	Active
Advanced Bladders.	NARMCO	JPL	Active
Propellant Materials Compatibility (OF_2/B_2H_6).	In House	JPL	Active (Tests initiated)
Advanced Spacecraft Valves.	TRW	JPL(NAS7-436)	Active
Space Storable Bipropellant Shutoff Valves.	Aerojet	JPL(NAS7-773)	Active
Distributed Energy Release in Space Storable Propellants.	Aerojet	JPL	RFP Pending
OF_2/B_2H_6 Vacuum Performance.	Rocketdyne	JPL(NAS7-741)	Active

TABLE A-6. SUMMARY OF SOME CURRENT NASA SPONSORED RESEARCH ON LARGE CHEMICAL SPACECRAFT PROPULSION

Type of Study	Research Organization	NASA Agency	Latest Known Status
Liquid Bipropellants (Space Storable) Continued)			
Liquid Phase Reactions in Hypergolic Propellants.	Rocketdyne	JPL(NAS7-739)	Active
Injector Spray Separation Phenomena.	Rocketdyne	JPL(NAS7-720)	Active
Injector Design Variables vs Drop Size Distribution.	Rocketdyne	JPL(NAS7-726)	Active
Wave Phenomena in Hybergolic Combustion.	Dynamic Sciences	JPL(NAS7-476)	Active
Flox/B_2H_6 Vacuum Ignition.	Thiokol RMD	JPL(NAS7-660)	Completed
Analytical Study of Control Devices for High Amplitude Combustion.	MAGI	JPL(NAS7-752)	Active
Combustion and Ignition of Space Storable Propellants.	Dynamic Sciences	JPL(NAS7-438)	Active
Rocket Plumes in Counterflows.	MITHRAS	JPL(NAS7-516)	Active
Transonic Flow in Converging-Diverging Nozzles.	G.V.R. Rao	JPL(NAS7-635)	Active
Space Storable Nozzle Performance.	Rocketdyne	OART(Hdq) (NASw-1224)	Completed
Space Storable Propellant Performance.	Rocketdyne	LeRC(NAS3-11190)	Completed
Space Storable Regenerative Cooling Study.	Pratt & Whitney	LeRC(NAS3-12029)	Active
Flourinated Oxidizer Valve.	McDonnell/Douglas and Aerojet	LeRC(NAS3-12029)	Active
Small Rotating and Positive Displacement Pumps.	Rocketdyne	LeRC(NAS3-12022)	Active
Space Storable Engine Characterization.	Pratt & Whitney	LeRC(NAS3-12010)	Completed
Space Storable Propellant Performance.	TRW	LeRC(NAS3-11200)	Completed
Space Storable Cooling Investigation.	Rocketdyne	LeRC(NAS3-11191)	Completed
Thermal Barrier for Space Engines.	Aerojet	LeRC(NAS3-7985)	Active
LPG Pressurization Gas Absorption.	In House	LeRC(NAS3-7985)	Active
Fluorine Oxidizer Vent Valve.	--	LeRC(NAS3-7985)	Pending
Space Storable Engine Characterization.	Rocketdyne	LeRC(NAS3-12024)	Completed
Spacecraft Propulsion Module Using Space Storable Propellants.	In House	JPL	Active
Advanced Combustion Device Development.	In House	JPL	Active
Advanced Injectors for Space Storable Propellants	Aerojet	JPL(NAS7-713)	Completed
Liquid Bipropellant (Deep Cryogenic)			
Theoretical and Experimental Study of Reliquification of Cryogenic Propellants for Increased Storability.	Air Products	MSFC	Hardware designs have been formulated, subcomponent tests have been conducted.
Theoretical and Experimental Study of Slush Hydrogen for Lunar, Earth Orbital and Planetary Probe Missions.	Lockheed Aircraft Corporation	MSFC	Mission feasibility study complete, demonstration effort underway.
Experimental Study of Performance of Liquid Hydrogen gels.	Technidyne (Westchester, Pennsylvania)	LeRC	Final report received.
Experimental and Theoretical Study of Techniques of Eliminating Cryogenic Propellant Stratification During Long Term Storage.	In House	MSFC	Potential systems assesed, analytical methods of analysis developed, small scale experiments conducted.

TABLE A-6. SUMMARY OF SOME CURRENT NASA SPONSORED RESEARCH ON LARGE CHEMICAL SPACECRAFT PROPULSION

Type of Study	Research Organization	NASA Agency	Latest Known Status
Liquid Bipropellant (Deep Cryogenic) (Continued)			
Development of Instrumentation for Handling Slush Hydrogen.	ORTEC (Oak Ridge, Tennessee)	MSFC	Complete.
Development of Instrumentation for Handling Slush Hydrogen.	Engineering Physics Company (Rockville, Maryland)	MSFC	Probably complete.
Development of RF Propellant Gaging System for Cryogens under Zero g's.	Bendix	MSC	In progress.
Study of Feasibility of Resonance Tube Ignition of H_2/O_2 in Space Environment.	In House	LeRC	Preliminary work indicates feasibility (TMX 1460). Work on design parameters required.
H_2/F_2 Rocket Engine Research	Pratt & Whitney	LeRC(NAS3-7991)	Completed
Flight Weight LF_2 Feed System	McDonnell/Douglas	LeRC(NAS3-11195)	Completed
Main Tank Injection Pressorization	McDonnell/Douglas	LeRC(NAS3-7963)	Completed
Active Heat Exchanger	Lockheed Missiles and Space Co.	LeRC(NAS3-12033)	Active
LH_2 Mixing Unit	Air Research	LeRC(NAS3-11268)	Completed
Solid and Hybrid Propellants			
Study of High Energy Solid Propellants Based on Nitronium Perchlorate for use in Spacecraft.	In House	JPL	Remote operations cell being constructed.
Study of Methods of Improving the Thermal Stability of Nitronium Perchlorate.	Not selected	JPL	Unsolicited proposal being evaluated, procurement pending.
Development of a Hybrid Propulsion System with 8:1 Throttling and an I_{SP} of at Least 365 sec.	Lockheed Redlands	LeRC	In progress.
Study of High Energy Solid Propellants Using Light Metal Hydrides for Spacecraft Applications.	In House	JPL	Facility being constructed.
Design Review of ATS-F Apogee Motor.	In House	JPL	Not available.
Study of Feasibility of Sterilizing Solid Propellant Motors.	NOTS	JPL	Feasibility indicated, demonstration proceeding.
Study of the Factors Influencing the Stability of Ammonium Perchlorate .	In House	JPL	In progress.
Stress Analysis of Solid Propellant Grains Subjected to Sterilization.	In House	JPL	New work.

There is, at present, an increasing conviction among some spacecraft propulsion system developers that working relations can be found and used which would allow propulsion subsystem requirements and design decisions to be related to overall mission effectiveness. In this respect, the aircraft industry has, in several instances (e.g., on the 747 program), been able to derive guidelines for subsystem designers which allow these designers to relate specific design choices to overall vehicle cost-effectiveness. A methodology for deriving similar guidelines for spacecraft propulsion systems was proposed in Reference A-15. Although the approach is not yet demonstrated, it is of sufficient importance to warrant considerable future examination.

Some comments regarding the subject of propellant toxicity need to be made at this point. Most of the propellants shown in Table A-2 are toxic (specifically, N_2O_4, OF_2, F_2, Flox, B_2H_6, IRFNA, MMH, UDMH and NH_3). Because of the high toxicity of fluorine-based propellants and B_2H_6, special handling procedures and perhaps special launch restrictions will be required during the use of these propellants.

The technical problem of propellant handling appears to be manageable. N_2O_4 has been launched from the ETR in large quantities in the Titan. The more toxic propellants such as OF_2, F_2 and B_2H_6 also appear to be technically manageable through the use of remote loading and no-vent, on-site storage methods.(A-17)

The greatest unknown in this area is, thus, the possible launch restrictions that may be placed on such vehicles by the range safety requirements. There are, at present, a number of studies being conducted in this area. The possible restrictions depend upon the toxicity of the propellant. A large amount of work has been conducted on LF_2 toxicity. Recent studies indicate that the traditional allowable levels in parts per million might be conservative by about a factor of 10, and that there are both total dosage and dose rate limits with fluorine.[A-18] The general feeling is that fluorine toxicity is becoming fairly well understood. The same does not hold true for some of the other propellants such as OF_2 and B_2H_6. OF_2 is particularly troublesome since it is much harder to detect than F_2.

Once the toxicity is established, it is necessary to relate this information to estimates of potential hazards to flight and civilian personnel from accidental spills or vents. This requires the formulation of models of likely meteorological conditions and propellant reactions with the surrounding environment. Much work remains to be done in this area.

In summary, the question of possible launch constraints imposed by the use of toxic spacecraft propellants is still unsettled. Although there are some proposed studies (e.g., the upcoming JPL-KSFC joint study of the use of an OF_2-B_2H_6 module at ETR)[A-19] which should help, much work remains to be accomplished before this question is satisfactorily answered.

A final word needs to be added on a requirement unique to planetary orbiters and landers. Current NASA specifications call for planetary missions to keep the probability of contaminating the planet with Earth organisms less than 10^{-3}.[A-20] A typical past procedure was to sterilize the entire spacecraft prior to launch by exposure to a 135°C temperature dry heat environment. Such treatments place the spacecraft propulsion system under stresses not normally encountered during operation* and require consideration during the initial design period.

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* There are indications that some of the early failures on the Ranger series may have been induced by the sterilization process.[A-20]

APPENDIX A

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APPENDIX A

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APPENDIX B

SUMMARY OF SPACECRAFT PROPULSION APPLICATIONS

APPENDIX B

SUMMARY OF SPACECRAFT PROPULSION APPLICATIONS

Table 1 of the main body of this report summarizes common automated spacecraft propulsion applications and their associated total impulse or velocity increment requirements. This appendix contains a more detailed discussion of these requirements. The applications will be discussed in the order shown in Table 1.

Stationkeeping

Stationkeeping refers to the maintenance of the spacecraft in some desired Earth orbit. A common case is the maintenance of a geostationary longitudinal position for synchronous (equatorial or other) spacecraft.

A synchronous spacecraft inevitably has some drift rate relative to the desired geostationary longitude. This drift results from a variety of perturbative forces and from inability to establish the precise synchronous altitude and velocity. Limitations in ability to measure spacecraft velocity precisely and in ability to deliver precise velocity corrections, and the continuous presence of small Lunar, Solar, and noncentral geopotential disturbing forces insure that these drifts will persist.

Normal operating procedures for synchronous spacecraft (for example, the Syncom system[(B-1)*]) are to accept the existence of drifts and minimize (or in some instances, utilize) their effect. During the initial positioning of the spacecraft, restrictions on the initial injection longitude resulting from launch vehicle constraints (such as limited coast capability) can be overcome by placing the spacecraft in

*Superscript numbers refer to References shown at end of this Appendix.

a near synchronous orbit at a longitude other than the desired and allowing the spacecraft to drift to the desired longitude. Once the desired longitude is reached, the drift velocity is reduced as much as possible. Subsequent drift is usually small and is allowed to persist until the spacecraft has wandered several nautical miles from the desired position. A new velocity correction is then applied to reverse the direction of the drift. The net result is that the spacecraft slowly wanders in the near vicinity of the desired geostationary position.

The required ΔV capability for this east-west stationkeeping depends upon a number of factors such as the required spacecraft lifetime, the accuracy of the tracking equipment, and the accuracy and reproducibility of the stationkeeping propulsion system corrections. Experience with the Syncom system indicates a nominal value of 10 ft/sec per year.[B-1]

In addition to east-west stationkeeping, there may be a requirement for north-south adjustments. In this case, the orbital element being adjusted is the inclination, whereas in the east-west case the period (semimajor axis) is adjusted. The required velocity increments in the north-south case can be shown to be proportional to Δi, the incremental change in the inclination, while the ΔV's for the east-west case can be shown to be proportional to $\Delta a/a$ (a = semimajor axis). In general, the required corrections are such that Δi term is numerically much larger than the $\Delta a/a$ term. As a result, the total ΔV per year for north-south stationkeeping is considerably higher. For the Syncom spacecraft, the requirement was estimated at ~175 ft/sec per year.

Attitude Control

After injection by the launch vehicle, a spacecraft would normally be expected to have some residual angular rotational rates. In addition, the spacecraft in cruise or operational condition would be expected to experience perturbing torques due to imbalance in the solar pressure, micrometeroid impact, etc. As a result, the spacecraft, if left to itself, would tumble. For certain functions, such as antenna pointing and solar panel orientation, a continuously varying attitude is unacceptable, and some sort of attitude control is required. This control may be nonpropulsive, such as with gravity gradient, spin, or solar paddle stabilization. Where such measures are not possible or not adequate, propulsive attitude control may be required.

The procedure adopted for propulsive attitude control is similar to that used for stationkeeping. The angular orientation is allowed to drift within the specified angular limits (normally a few tenths of a degree) about the nominal pointing direction. This direction may be fixed in space, such as the direction to a reference star, such as Canopus, or varying, such as the direction from the spacecraft to Earth or Sun. Once any of the angular limits is exceeded, a corrective torque is applied to reverse the angular drift rate. Thus, the spacecraft undergoes an oscillatory motion within the angular deadband.

The amount of corrective torque required depends upon such considerations as the specified angular limits, the mass moments of inertia of the spacecraft, the amount of perturbing torque (which is influenced by such factors as the net distribution of the spacecraft surfaces and their reflective and radiative properties), and the number of required attitude maneuvers or reference acquisitions. Table B-1 shows some typical attitude control gas requirements for past spacecraft missions.

Earth Orbit Injection

This term refers to applications where the spacecraft carries a propulsion unit to provide the ΔV to place the spacecraft in its final Earth orbit. Examples of this application can be found with many operational spacecraft including Syncom I and II, and Explorer I.

The ΔV requirements for final injection depend, of course, upon the final and transfer orbit. There are so many ways of attaining a final orbit that no completely general rules can be stated. However, some common cases occur with sufficient frequency to warrant quoting. For synchronous orbit out of ETR, the ΔV for circularization at the apogee of the transfer orbit is ~4800 ft/sec if the transfer orbit is for a 100 n. mi. parking orbit. The normal ΔV for combined circularization and plane change at the apogee of the transfer orbit for synchronous equatorial missions out of the ETR is approximately 6000 ft/sec.

Figures B-1 and B-2, taken from References B-5 and B-6, provide a means of obtaining ΔV's for certain common cases of interest.* Figure B-1 gives the ΔV required to establish a desired final orbit when the injection takes place at the apogee of the transfer ellipse from 100 n. mi. For example, if the desired final orbit is 400 x 1000 n. mi., the

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* Data are based on an assumption of impulsive velocity change, and should be used only for cases where the vehicle thrust to weight ratio is sufficiently large (say > 0.1) that this is a reasonable assumption.

TABLE B-1. TYPICAL ATTITUDE CONTROL GAS REQUIREMENTS FOR PAST SPACECRAFT(B-2 to B-4)

Spacecraft	Control Gas Use Rate* (lb/day)	Total Gas Carried* (lb)	Estimated Total Impulse Capability* (lb-sec)
Mariner IV	$3\text{-}4 \times 10^{-3}$	5.25	394
Ranger VII	5×10^{-2}	4.24	318
Surveyor I	2×10^{-1}	4.49	337

* Total for all gas jets.

desired procedure would be to inject the spacecraft into a 100 x 1000 n. mi. orbit (transferring to the final apogee first results in lower total velocity requirements). Then, using the 1000 n. mi. line on the lower scale, the ΔV to raise the final apsis altitude (in this case the perigee) to 400 n. mi. is approximately 500 ft/sec.

Figure B-1 can also be used to find the ΔV's for injecting the spacecraft into orbit from a 100 n. mi. parking orbit. In this case, the apogee altitude of the intial transfer orbit is 100 n. mi., and the left hand edge of the figure only is used. For example, the ΔV to transfer from a 100 n. mi. parking orbit to a 100 x 1000 n. mi. final orbit is found by using the 100 n. mi. line on the lower scale and finding the point where the final apsis altitude (in this case the final apogee equals 1000 n. mi.). The ΔV is then 1400 ft/sec.

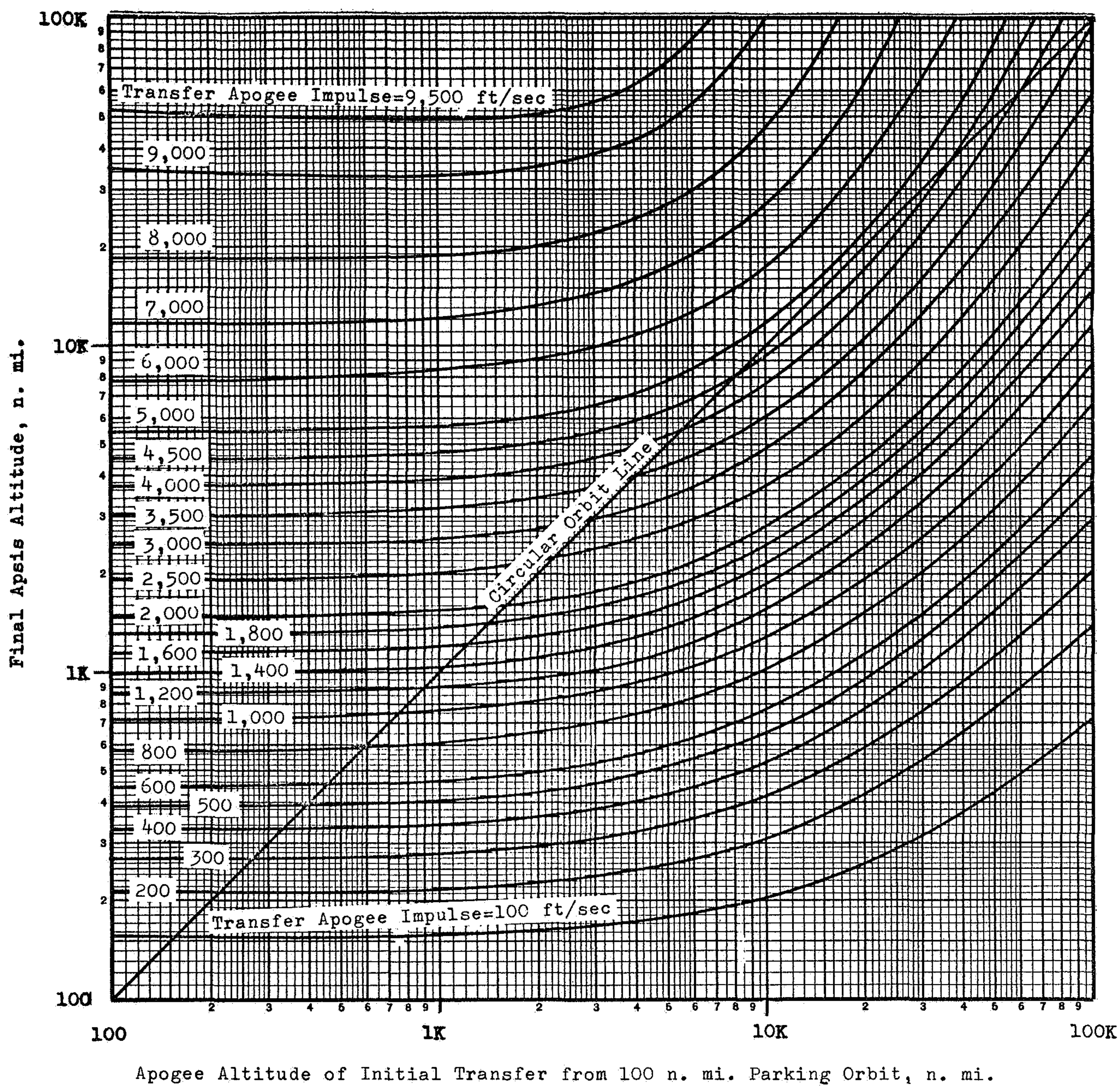

FIGURE B-1. TRANSFER APOGEE IMPULSE REQUIREMENTS AFTER TRANSFER FROM 100 N. MI. ORBIT(B-5)

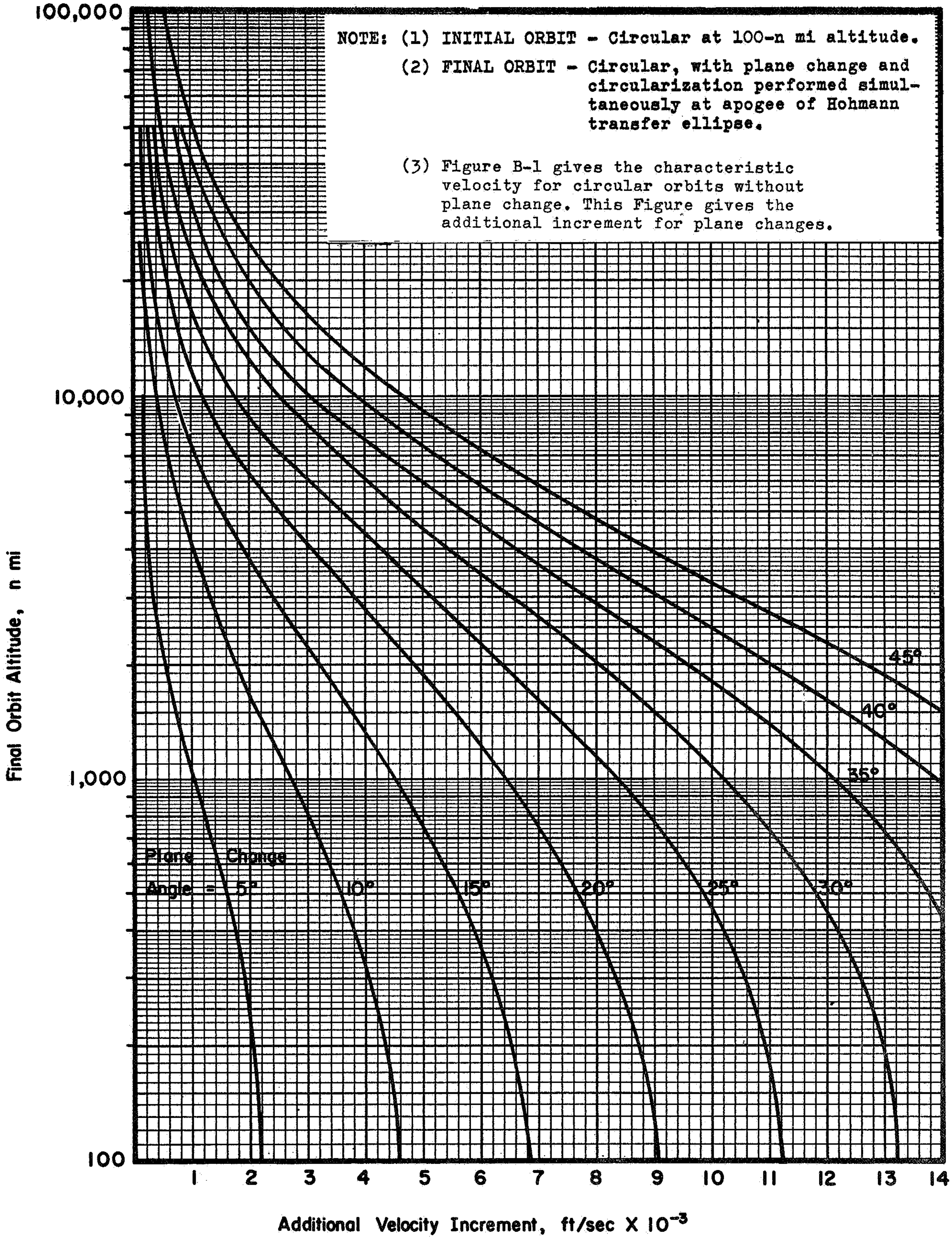

FIGURE B-2. ADDITIONAL VELOCITY INCREMENT REQUIRED FOR ORBITAL PLANE CHANGE FOR CIRCULAR ORBITS (B-6)

Figure B-2 gives the required additional ΔV for plane changes in establishing circular orbits.* These data are based on the assumption that the spacecraft is either already in a circular orbit at 100 n. mi. altitude (in which case B-2 gives the required spacecraft injection motor ΔV) or that the plane change occurs simultaneously with the circularization (in which case the required ΔV is the sum of those given by Figures B-1 and B-2). For example, if the spacecraft were in a 100 n. mi. circular orbit of 30° inclination, then a ΔV of 1320 ft/sec would be required to place it in a 100 n. mi. circular orbit of 0 or 60° inclination.

As a further example, consider a spacecraft in a 100 x 600 n. mi. orbit of 30° inclination. If it is desired to inject the spacecraft into a 600 n. mi. circular orbit of 0° inclination, the required spacecraft ΔV can be found as follows. From Figure B-1, the velocity increment for circularization of a 100 x 600 n. mi. orbit at the apogee is ~800 ft/sec. From Figure B-2, the required additional velocity increment for a 30° plane change at 600 n. mi. is 1180 ft/sec. Therefore, the required spacecraft ΔV is 1980 ft/sec.

Orbit Correction

Orbit correction refers to applications where the spacecraft is in a generally satisfactory orbit that must be modified somewhat to conduct the required mission. Examples can be found from past spacecraft missions. The Syncom II injection motor placed the spacecraft into an orbit that was nearly synchronous but with an undesirable drift rate.

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* Reference B-5 contains other data for more generalized orbital maneuvers.

To correct this, a ΔV of 109.8 ft/sec was applied to establish the desired drift rate. The propulsion system on the Lunar Orbiter initially placed it in a lunar orbit with a 200 km perilune. Prior to initiating the photographic mission, the perilune was reduced to 40 km with a ΔV of 40.2 m/sec. Seven days later, this orbit was further adjusted by the application of a 5.4 m/sec ΔV.[B-7,8] As illustrated in these examples, most orbit corrections involve small ΔV's (100 ft/sec or less).

Two concepts involving repeated orbit corrections have been proposed in recent years. One is the so-called "yo-yo" spacecraft which involves multiple orbit transfers to perform investigations over a range of orbits with various perigees and apogees. The other is a low-perigee spacecraft where the spacecraft lifetime is extended by applying repeated corrective ΔV's to cancel the rate of apogee decay due to atmospheric drag. The required ΔV per orbit for these applications would also be low. Total ΔV requirements over an extended time period could be fairly high, however.

It should be noted that, in general, orbit transfers require fairly high thrust levels. Orbit corrections of the type discussed here could reasonably use low thrust propulsion units operated either intermittently or--as in the last example--continuously.

Midcourse Correction

One or more small midcourse velocity corrections are normally required to reduce target arrival errors introduced by guidance system errors and off-nominal launch vehicle propulsion system performance on planetary and lunar missions. The magnitude of the required ΔV is a function of both the nominal mission trajectory (which determines the manner in which initial errors are propagated into terminal errors) and the launch vehicle guidance and propulsion system performance.[B-9] Because of the random nature of the variations in the performance of these systems, the ΔV requirements can only be given statistically (i.e., in terms of an expected mean value and variance). Statistically, there is a chance that the ΔV requirement may be large or quite small. For this reason, midcourse propulsion systems must be designed under different groundrules than most other propulsion systems. The minimum deliverable impulse (or ΔV increment) is a matter of concern, and is generally designed to be relatively small to increase the probability of satisfying a small ΔV requirement. For example, the Mariner IV midcourse motor could provide a minimum ΔV of ~1.5 ft/sec. Similarly, an excess of propellant over that required to provide the expected mean correction is carried to increase the probability of satisfying a large ΔV requirement. This point is illustrated in Table B-2, which gives midcourse ΔV's for some past missions which may be considered typical.

TABLE B-2. TYPICAL MIDCOURSE PROPULSION SYSTEM REQUIREMENTS[B-2,3,4,7,8,10,11]

Mission	Midcourse ΔV (m/sec)	Total ΔV Capability (m/sec)	Time of Correction	Total Mission Time
Ranger VII	29.89	~60	~17.5 hours	~68.5 hours
Mariner II	~31.1	~60	~62.5 hours	~109 days
Mariner IV	16.70	~125	~7 days	~228 days
Lunar Orbiter I	37.8	--*	~28.5 hours	~92 hours
Surveyor I	~20.3	--*	~16 hours	~63 hours
Surveyor IV	10.1	--*	~38 hours	~62 hours

* Midcourse propulsion system was not a separate entity.

Orbiter Retropropulsion

The required ΔV for lunar and planetary orbiter retropropulsion depends upon the relative approach velocity at the planet (or Moon) and the periapse and eccentricity of the final orbit. Figures B-3 through B-11 allow the calculation of the required ΔV for planetary and lunar orbiters. The ΔV's can be calculated as follows: Figures B-3 through B-9 give the approach velocity relative to the target body. For Mercury, only the minimum approach velocity for each year (there are an average of about three opportunities per year) are shown since the required retro ΔV's become excessive for the other opportunities. For Mars, Venus, Jupiter, Saturn and Uranus, the approach velocities corresponding to the minimum launch velocity and launches at the extremes of a 30 day opportunity width are shown for various years and flight times. For Neptune, the approach velocities are given as a function of flight time alone since they vary little with opportunity during the next two decades. For lunar missions, equivalent approach velocity at the Moon is shown in Figure B-9 as a function of trip time for the Moon at perigee and apogee. In general, the approach velocities will lie between these curves.

Knowing the approach velocity, the required retro ΔV can be found once the desired orbit periapse and apoapse are chosen.* Figure B-10 gives the escape velocity at periapse as a function of periapse radius for the planets and the Moon. The ratio of the approach velocity to the escape velocity is then computed and used to enter Figures B-11 or B-12 to find the ratio of the required retro ΔV to the escape velocity at periapse. Since the escape velocity at periapse was previously determined, the retro ΔV is thus known.

* For the data given here, the lunar apoapse should not exceed 22 lunar radii.

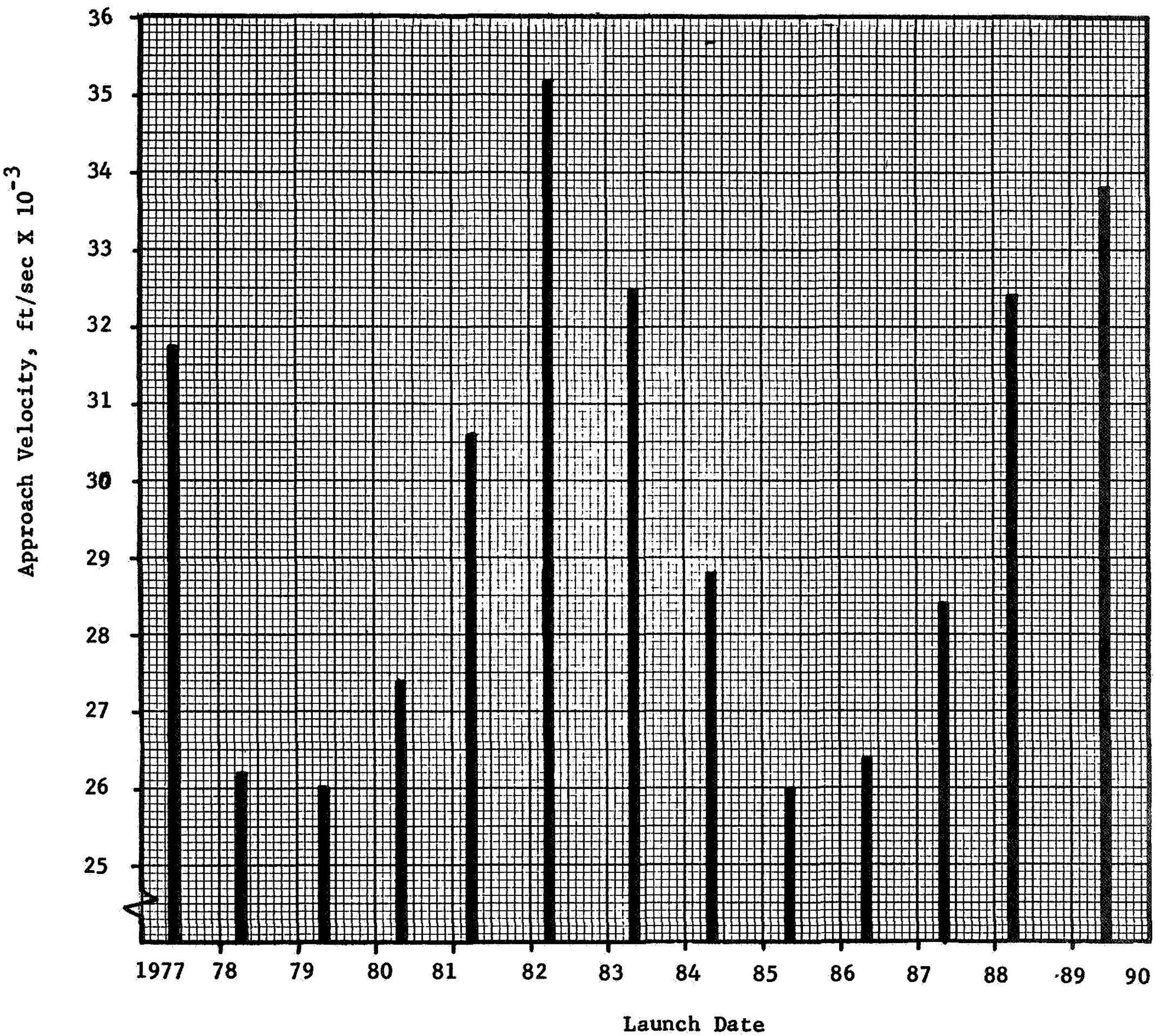

FIGURE B-3. MINIMUM APPROACH VELOCITIES FOR DIRECT MERCURY FLIGHTS

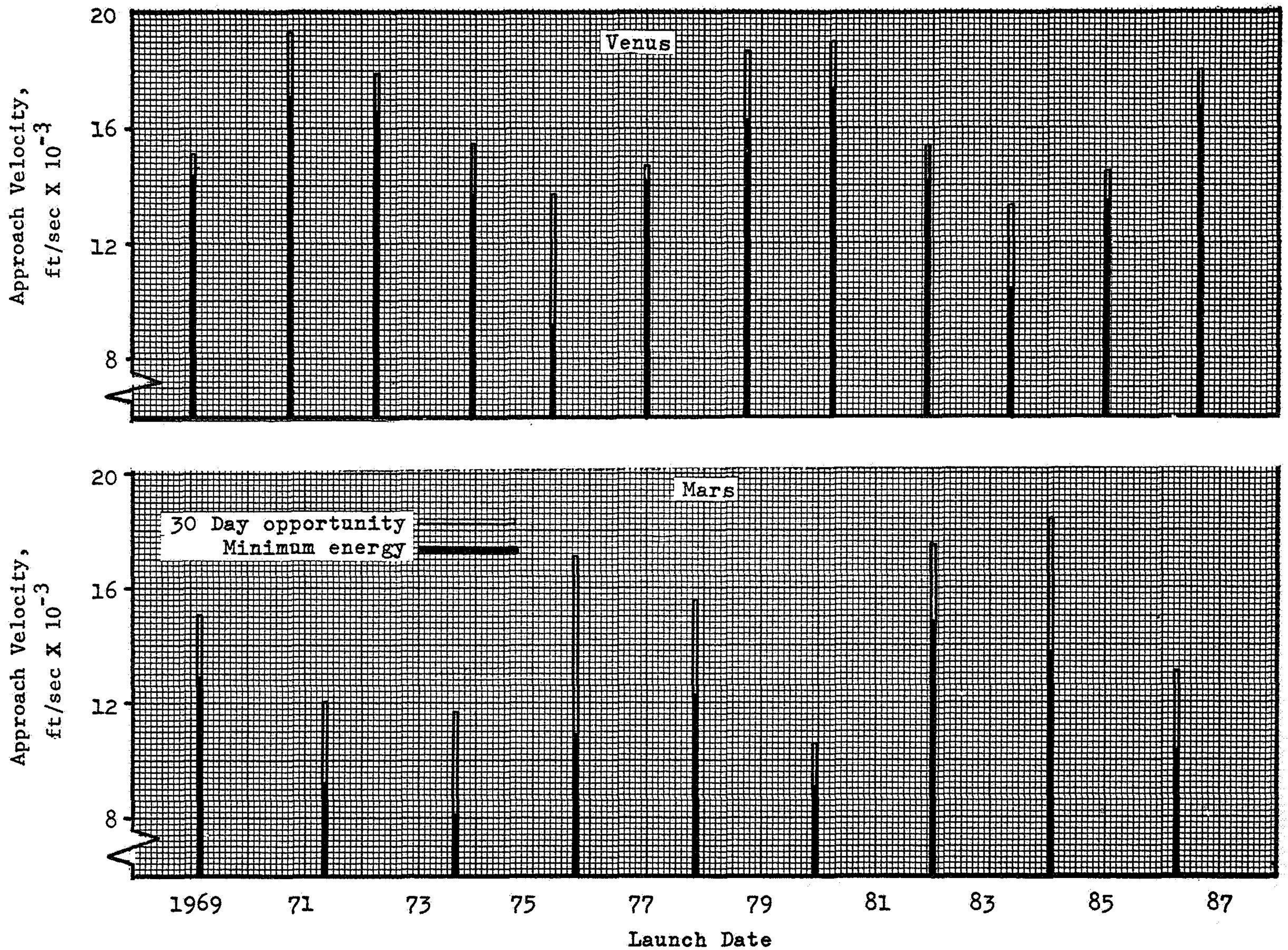

FIGURE B-4. APPROACH VELOCITIES FOR MARS AND VENUS

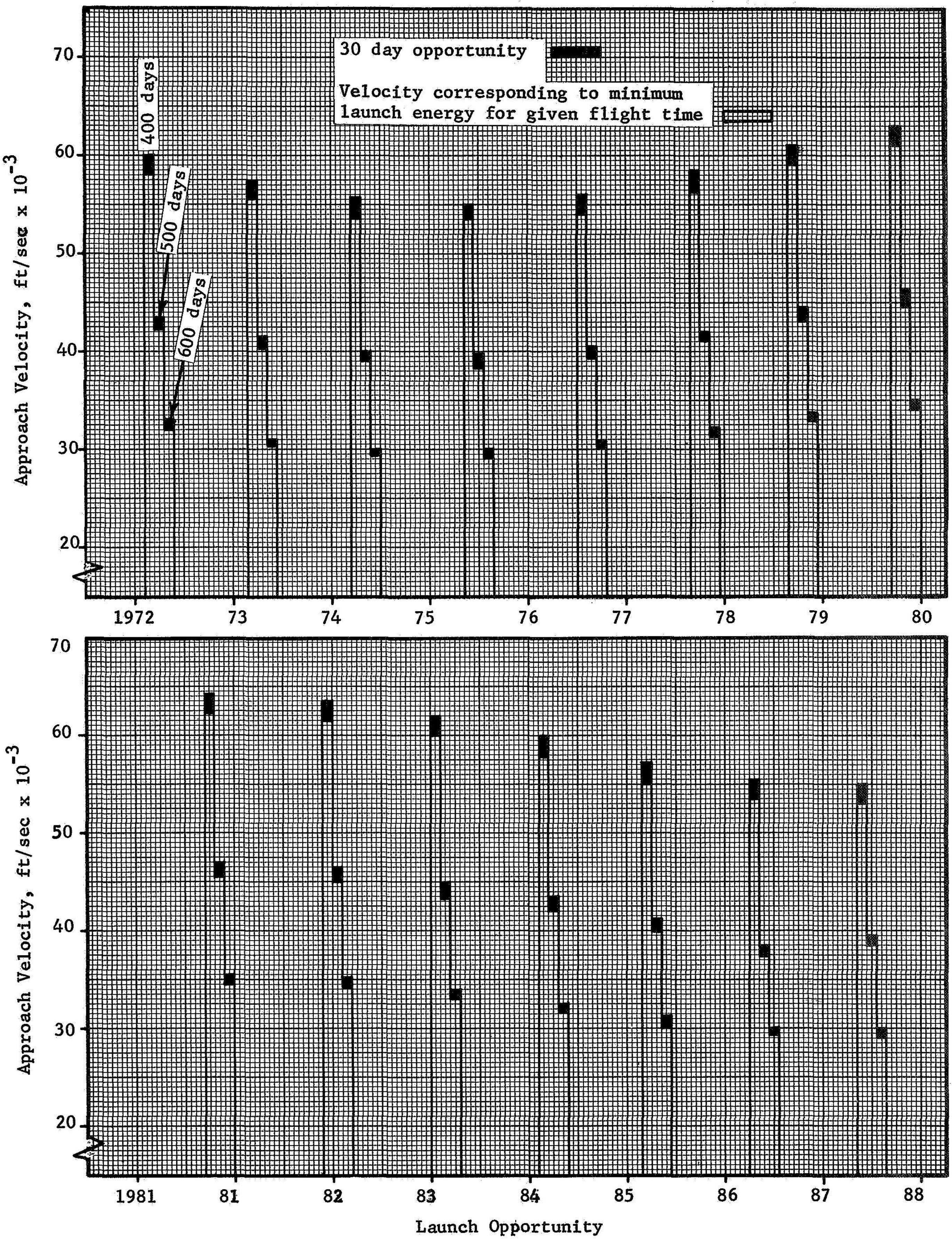

FIGURE B-5. APPROACH VELOCITY FOR JUPITER (400, 500, and 600 DAY FLIGHT TIMES)(B-5)

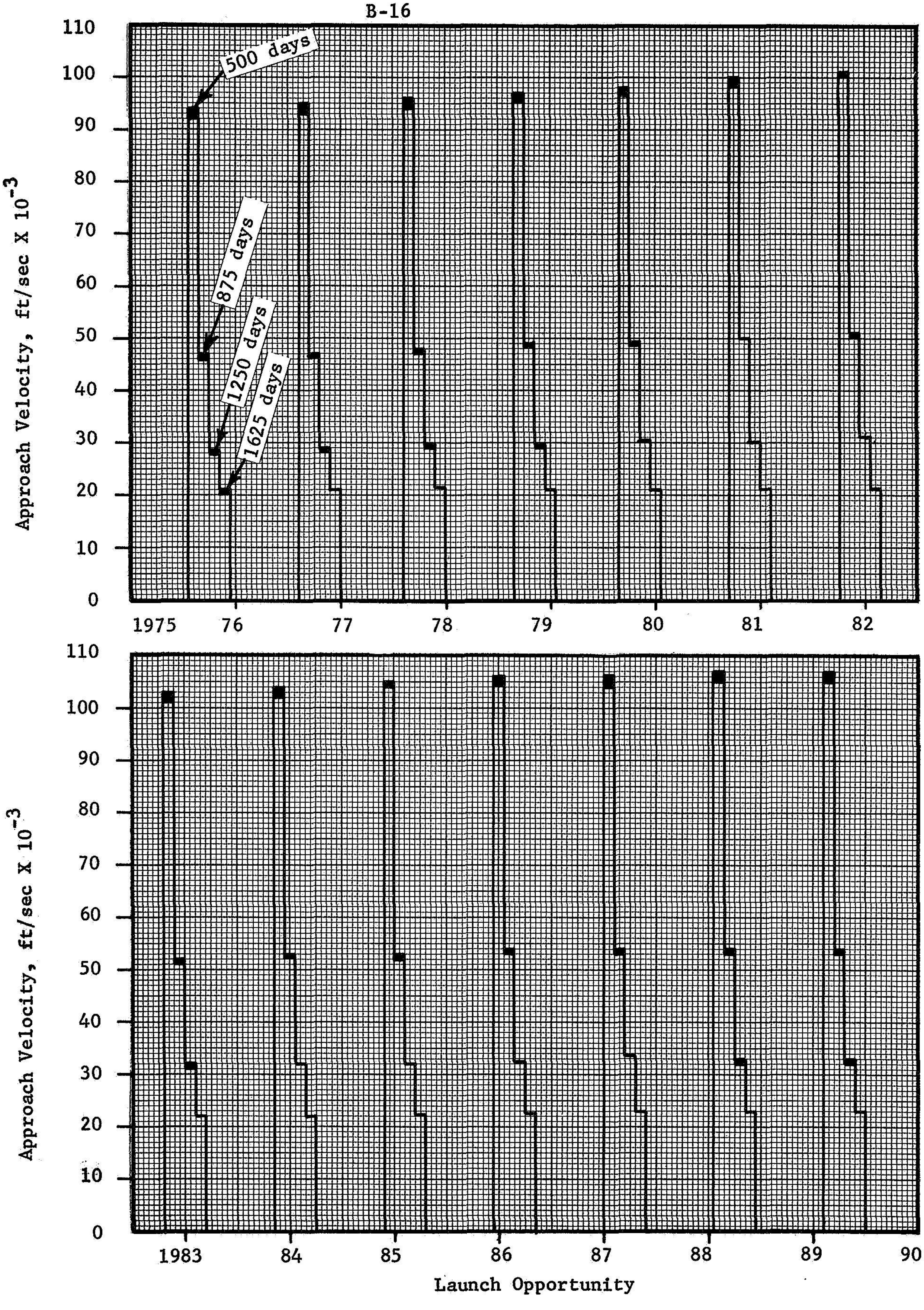

FIGURE B-6. APPROACH VELOCITY FOR SATURN (500, 875, 1250, AND 1625 DAY FLIGHT TIMES)

30 Day opportunity (when not shown, increment is negligible)

Velocity corresponding to minimum launch energy for given flight time

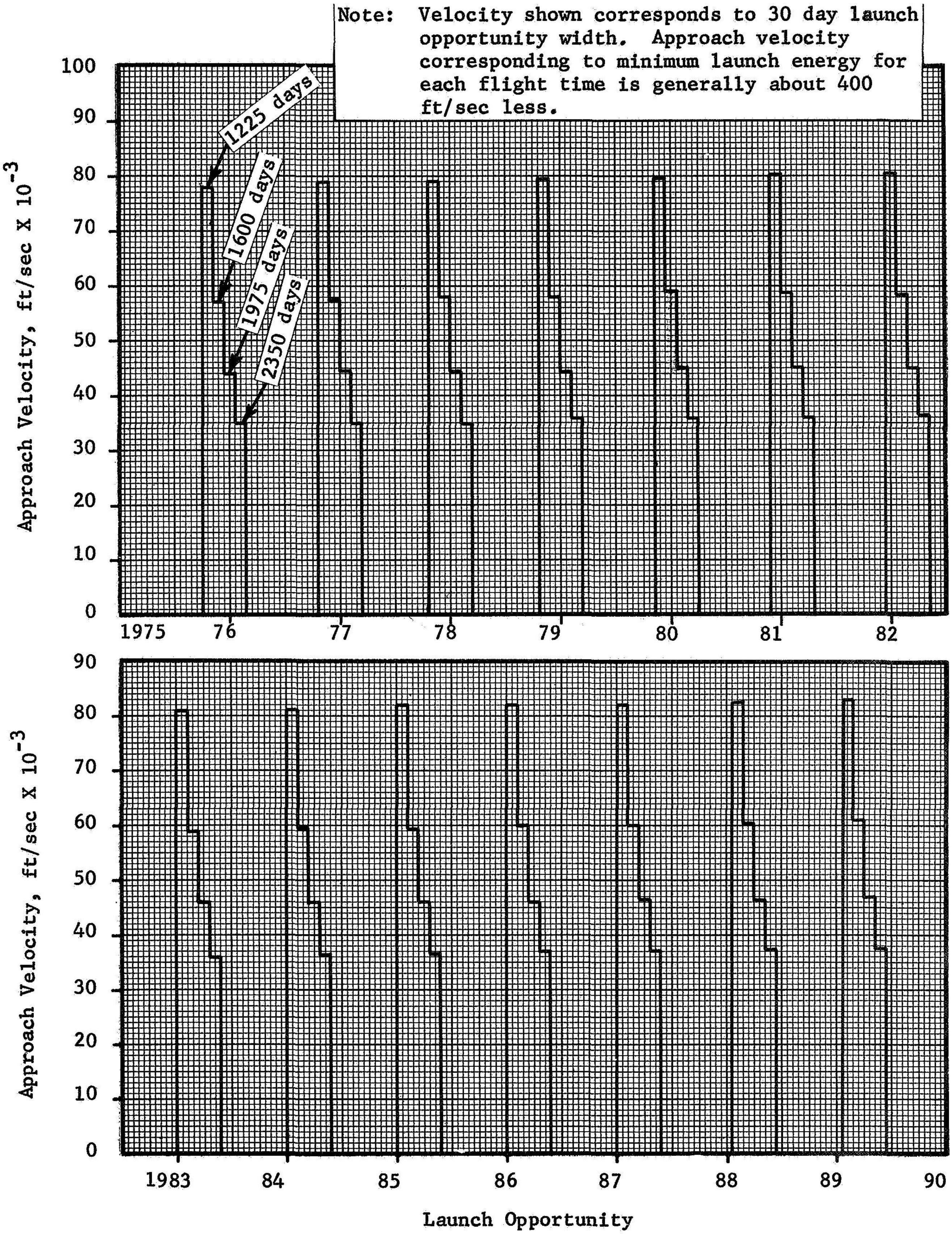

FIGURE B-7. APPROACH VELOCITY FOR URANUS (1225, 1600, 1975, and 2350 DAY FLIGHT TIMES) (B-5)

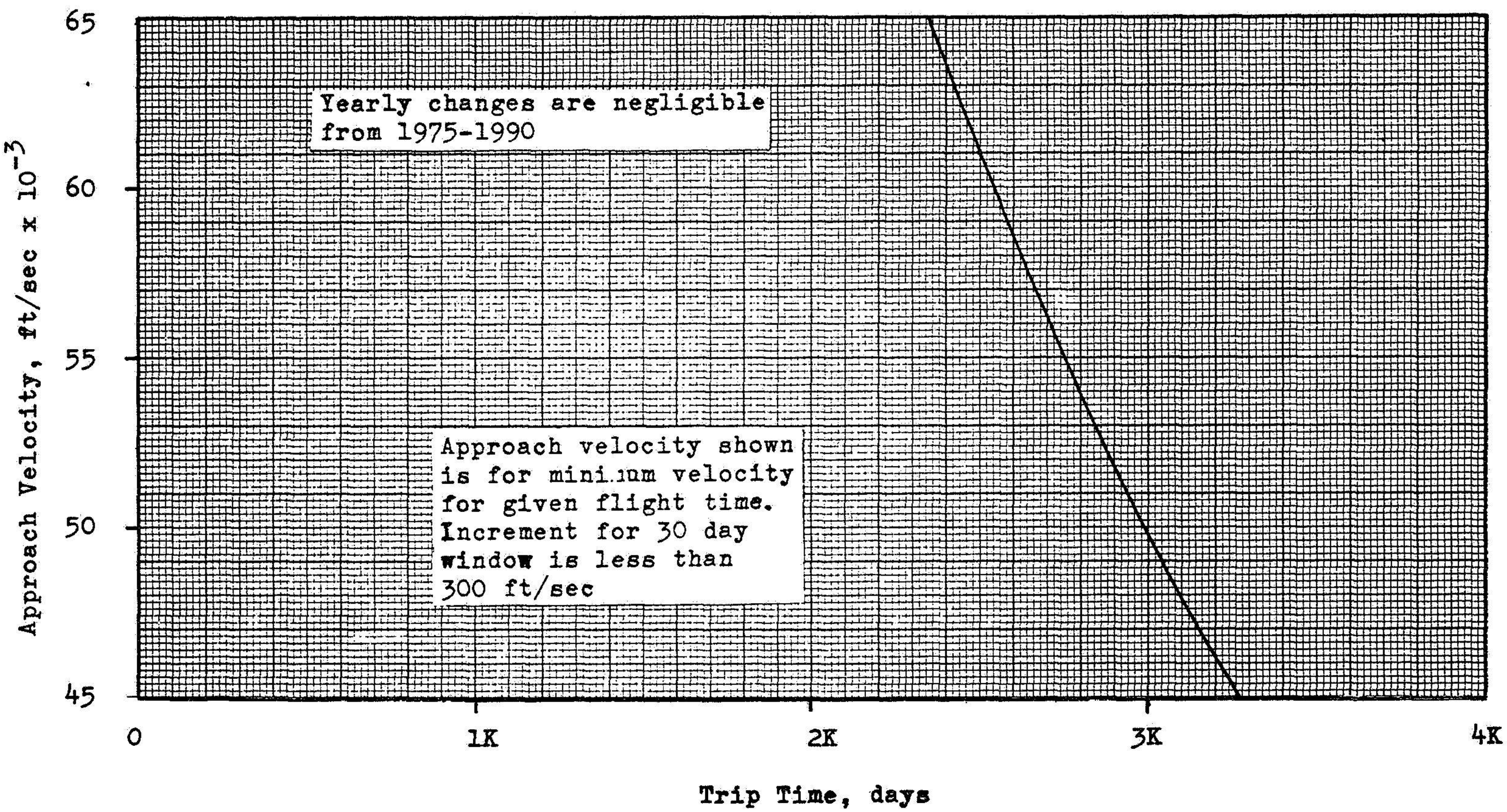

FIGURE B-8. APPROACH VELOCITY FOR NEPTUNE(B-5)

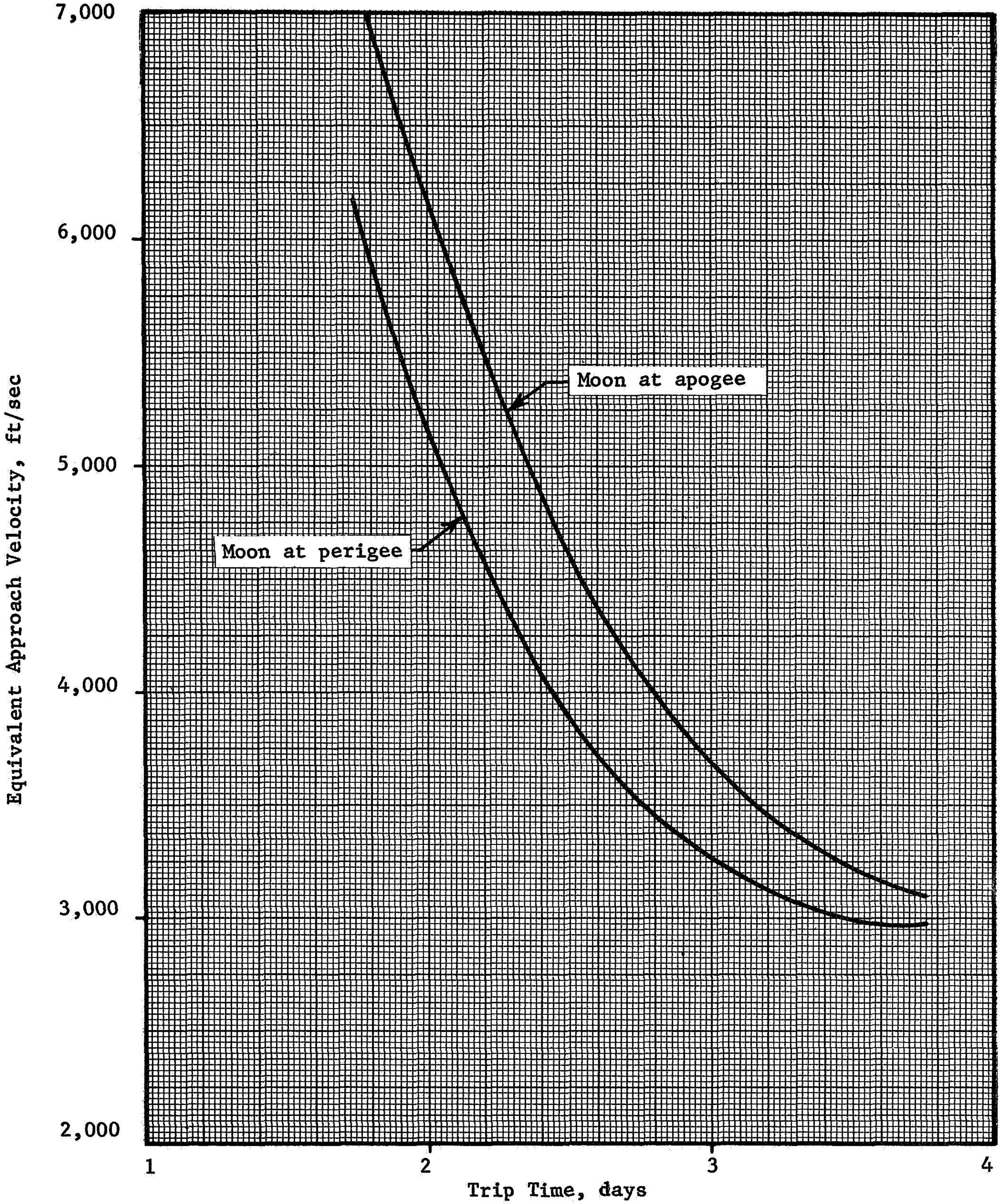

FIGURE B-9. APPROACH VELOCITIES FOR LUNAR MISSIONS (B-5)

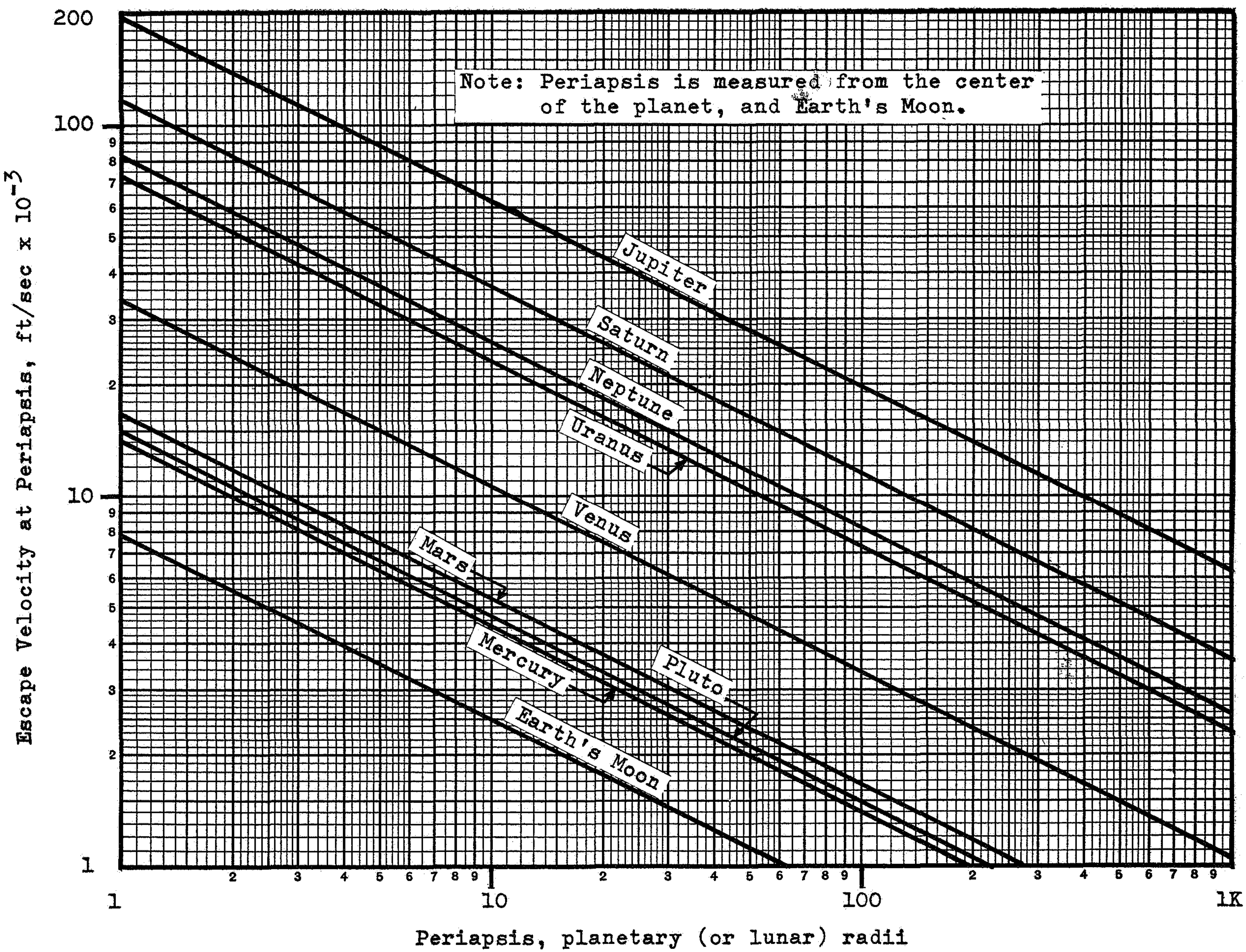

FIGURE B-10. ESCAPE VELOCITIES FOR THE PLANETS AND EARTH'S MOON (B-5)

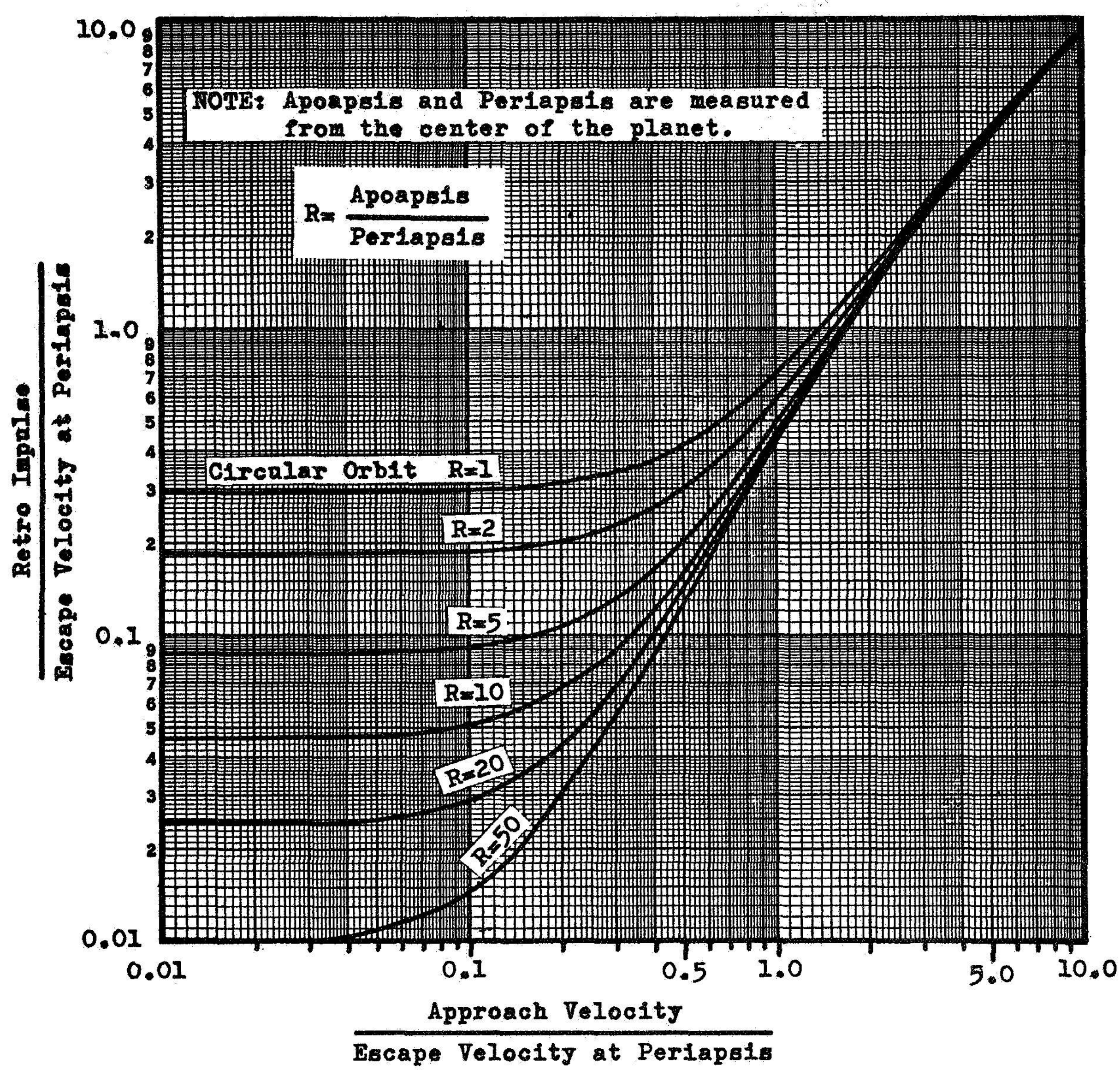

FIGURE B-11. RETRO IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS (B-5)

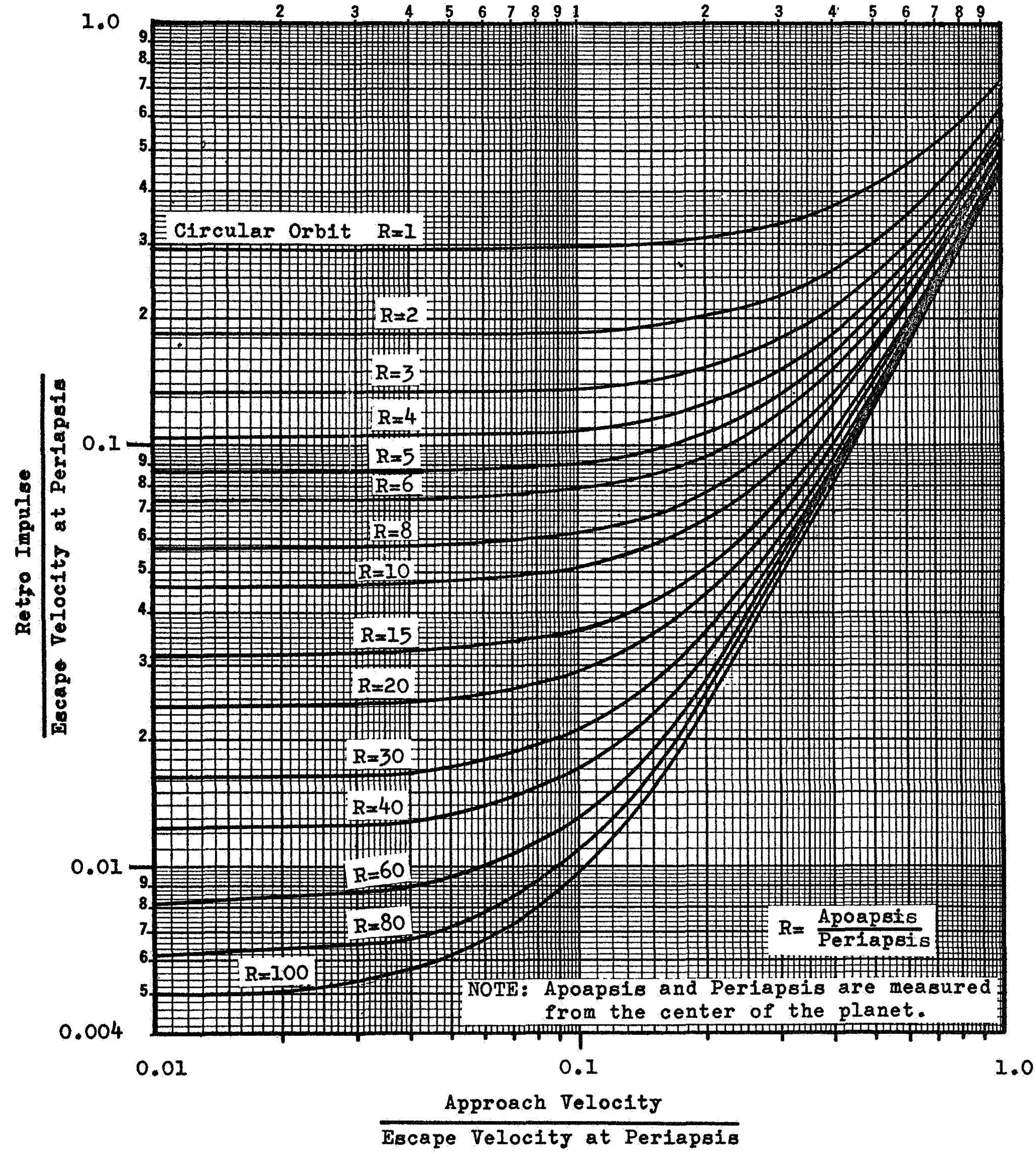

FIGURE B-12. RETRO IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS (B-5)

(An expanded scale of Figure B-11)

For example, for a 1975 Venus orbiter, the approach velocity, from Figure B-4, is 14,700 ft/sec for a 30 day opportunity width. Let the orbit of interest be circular at 2 planetary radii (altitude = 1 radius), from Figure B-10, for a periapse of 2.0, the escape velocity for Venus is seen to be 24,000 ft/sec. Thus, the ratio of approach velocity to escape velocity at periapse is 14,700/24,000 = .613. Using this value to enter Figure B-12, the value of the ratio of the retro impulse to escape velocity at periapsis for the circular orbit curve is seen to be .47. Thus, the required retro ΔV is given by (.47)(24,000) = 11,300 ft/sec. Table B-3 presents some additional ΔV's for other planets and the Moon. As can be seen, the retro ΔV's for close or loose orbiters of Mercury and Neptune and close orbiters of Jupiter, Saturn and Uranus are large. It should also be noted that the choice of orbit has only a small effect for the smaller planets such as Mercury or Mars, but has a significant effect for the major planets of Jupiter, Saturn, and Uranus. In any case where the retro ΔV is 20,000 ft/sec or greater, a multistage retropropulsion system would probably be required.

Planetary Landers

An estimate of the ΔV requirements for direct planetary or lunar soft landers (final relative velocity ~0) can be obtained from Figures B-3 through B-10. The required velocity increment is given by the expression

$$\Delta V_{Lander} = \sqrt{V_{esc}^2 + V_{apr}^2}$$

TABLE B-3. TYPICAL RETRO ΔV's FOR LUNAR AND PLANETARY ORBITERS

Planet	Opportunity	Periapse (in planetary radii)	Apoapse (in planetary radii)	Flight Time (days)	Required Retro ΔV (ft/sec)
Mercury	1979	2.0	2.0	100	20,600
Mercury	1979	2.0	20.0	100	19,100
Venus	1975	2.0	2.0	150	11,300
Venus	1975	2.0	20.0	150	5,400
Mars	1979	2.0	2.0	240	7,700
Mars	1979	2.0	20.0	240	5,000
Jupiter	1975	2.0	2.0	600	42,900
Jupiter	1975	2.0	20.0	600	9,500
Saturn	1976	2.0	2.0	1,625	26,600
Saturn	1976	2.0	20.0	1,625	6,600
Uranus	1977	2.0	2.0	2,350	26,000
Uranus	1977	2.0	20.0	2,350	13,300
Neptune	any	2.0	2.0	3,000	36,600
Neptune	any	2.0	20.0	3,000	22,700
Moon	Moon at Perigee	2.0	2.0	3	2,460
Moon	Moon at Perigee	2.0	20.0	3	1,175

where

ΔV_{lander}	is the required ΔV, assuming that it is applied impulsively at landing,
V_{esc}	is the escape velocity at the surface of the planet, and
V_{apr}	is the approach velocity relative to the planet.*

For any given planet V_{esc} at the surface can be found by reading the extreme left hand values of the curves in Figure B-10. The approach velocity at the planet can be found from Figures B-3 through B-9.

As an example, consider a direct Venus lander in 1973. For Venus the escape velocity at the surface is, from Figure B-10, 34,000 ft/sec. From Figure B-4, the approach velocity for the 1973 opportunity is 11,700 ft/sec for a 30-day opportunity. Thus, the required ΔV for a direct lander is

$$\Delta V = \sqrt{(34{,}000)^2 + (11{,}700)^2}$$
$$= 36{,}000 \text{ ft/sec}$$

For a direct lunar lander with a 3-day flight time, the approach velocity for the Moon at perigee (from Figure B-9) is 3,275 ft/sec. The escape velocity at the lunar surface (from Figure B-10) is 7,900 ft/sec. Therefore, the estimated ΔV for the lander is

$$\Delta V = \sqrt{3{,}275^2 + 7{,}900^2}$$
$$= 8{,}560 \text{ ft/sec.}$$

The velocity increment required for an indirect lander--i.e., a lander module ejected from a spacecraft orbiting the target body--is not as straightforward. The required ΔV depends upon such factors as the initial orbital radius, the amount of planetary atmosphere, the

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* Or, more conventionally, V_∞.

aerodynamic design of the lander, etc. An estimate of the upper and lower bounds of the required ΔV is possible, however. Figure B-13 contains two such estimates. The upper curve, labeled "all propulsion", is based on the assumption that the planetary atmosphere is negligible and that virtually all of the retro ΔV would have to be provided by a lander propulsion system. The assumed flight path consists of a small initial ΔV to place the lander in a Hohmann transfer between the initial planetary orbit (assumed circular) and the planetary surface, and a large ΔV at the planetary surface to cancel the lander velocity and allow a soft landing. Planetary rotation has been neglected. The second curve, labeled "aerodynamic", is based on the assumption that the planetary atmosphere is sufficient to allow aerodynamic deceleration (e.g., drag plus parachute) of the lander. In this case, the ΔV given is the increment required to transfer from the initial orbit (again assumed circular) to a transfer orbit resulting in planetary atmosphere entry. The required ΔV is assumed to be roughly the same as the initial ΔV of the all propulsive case.

For most landers of likely interest, the actual ΔV will probably lie somewhere between these two limits. The curves shown should bound the problem, however.

As an example of their use, consider the two circular orbits of Venus and Mercury listed in Table B-3. For Mercury, whatever atmosphere may be present will not likely be of much use for lander deceleration. Therefore, the upper curve of Figure B-13 is used here. The circular orbit shown in Table B-3 has a radius of 2 planetary radii. Therefore, the ratio of the initial orbital radius to the planetary radius (which is

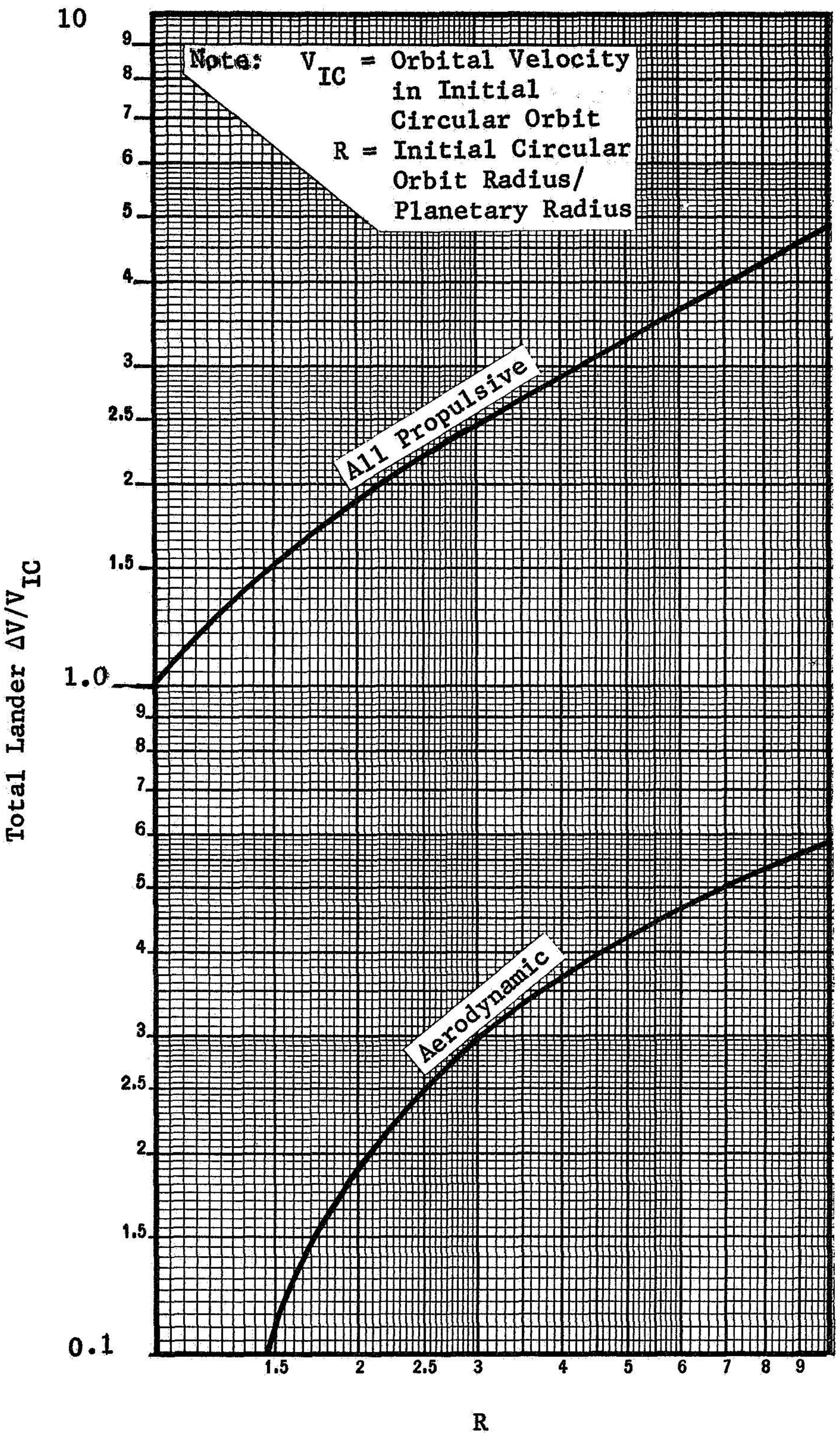

FIGURE B-13. ESTIMATED UPPER AND LOWER BOUNDS FOR INDIRECT PLANETARY LANDER ΔV REQUIREMENTS the quantity R shown in Figure B-13 is 2.0. From Figure B-13, for R =2.0, the all propulsion case gives a $\Delta V/V_{IC}$ of 1.9. V_{IC} is the velocity in the initial circular orbit. This can be found from Figure B-10. For Mercury, the escape velocity at 2 planetary radii is ~9800 ft/sec. The circular orbital velocity at any radius can be found by dividing the escape velocity at that radius by $\sqrt{2}$. Thus, $V_{IC} = 9800/\sqrt{2} = 6940$ ft/sec. Thus, the estimated total lander ΔV for the Mercury lander is

$$\Delta V_{Lander} = (\Delta V/V_{IC}) \cdot V_{IC}$$

$$= (1.9)(6940) = 13{,}180 \text{ ft/sec.}$$

For Venus, the planetary atmosphere probably allows aerodynamic deceleration--therefore, the lower curve will be used here. For an orbital radius of 2 planetary radii (R = 2.0), $\Delta V_{Lander}/V_{IC} = .185$. For Venus, the escape velocity at 2 planetary radii is (from Figure B-10) 24,000 ft/sec. Therefore, $V_{IC} = 24{,}000/\sqrt{2} = 17{,}050$. Thus, $\Delta V_{Lander} =$ $(.185)(17{,}050) = 3{,}150$ ft/sec.

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(B-5) "Launch Vehicle Estimating Factors", Launch Vehicle and Propulsion Programs, Office of Space Science and Applications, National Aeronautics and Space Administration, January, 1969.

(B-6) "Launch Vehicle Estimating Factors for Generating OSSA Prospectus 1968", Launch Vehicle and Propulsion Programs, Office of Space Science and Applications, National Aeronautics and Space Administration, December, 1967.

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(B-10) Haglund, H. A., "Surveyor IV Mission Report", JPL Technical Report No. 32-1210, January, 1968.

(B-11) Wheelock, H. J., et al, "Mariner Mission to Venus", McGraw-Hill, 1963.

(B-12) Ericke, K. A., "Space Flight-Dynamics" D. Van Nostrand, 1962.

APPENDIX C

<u>DETAILS OF SOME EXISTING SPACECRAFT PROPULSION SYSTEMS</u>

APPENDIX C

DETAILS OF SOME EXISTING SPACECRAFT PROPULSION SYSTEMS

This appendix illustrates some of the spacecraft propulsion system applications discussed in the previous appendix with examples from operational spacecraft. Systems are discussed which perform the following roles: east-west stationkeeping, attitude control, orbit correction, Earth orbit injection, midcourse correction, orbiter retropropulsion and direct planetary (lunar) lander.

East-West Stationkeeping

Figure C-1 (taken from Reference C-1)* shows a cutaway view of a Syncom spacecraft. The manner in which the Syncom was maintained in its geostationary position is discussed in Appendix B. The required ΔV's were provided by the lateral or axial jets shown in the cutaway. The jets used N_2 or H_2O_2 as propellants.

Attitude Control

Several different attitude control systems were used in the spacecraft shown in Figures C-1 through C-4 (taken from References C-1 through C-4). The Syncom spacecraft was spin stabilized, and used the axially mounted jets in a pulsed mode to orient the spin axis. The Mariner IV (Figure C-2) used a three-axis nitrogen gas jet stabilization system augmented by solar paddle torques about the pitch and yaw axes. The Lunar Orbiter also used a three-axis nitrogen gas jet system (not shown).

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*References are listed at the end of this Appendix.

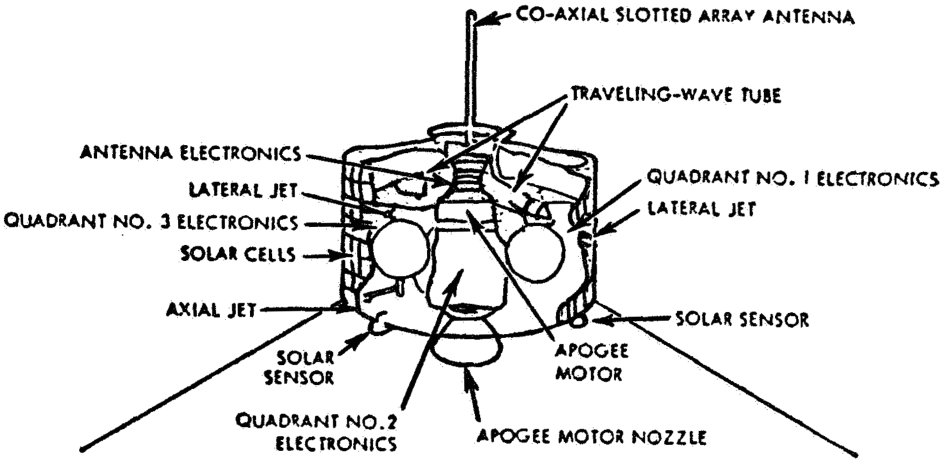


FIGURE C-1. CUTAWAY VIEW OF SYNCOM SPACECRAFT (Drawing based on information in Reference C-1.)

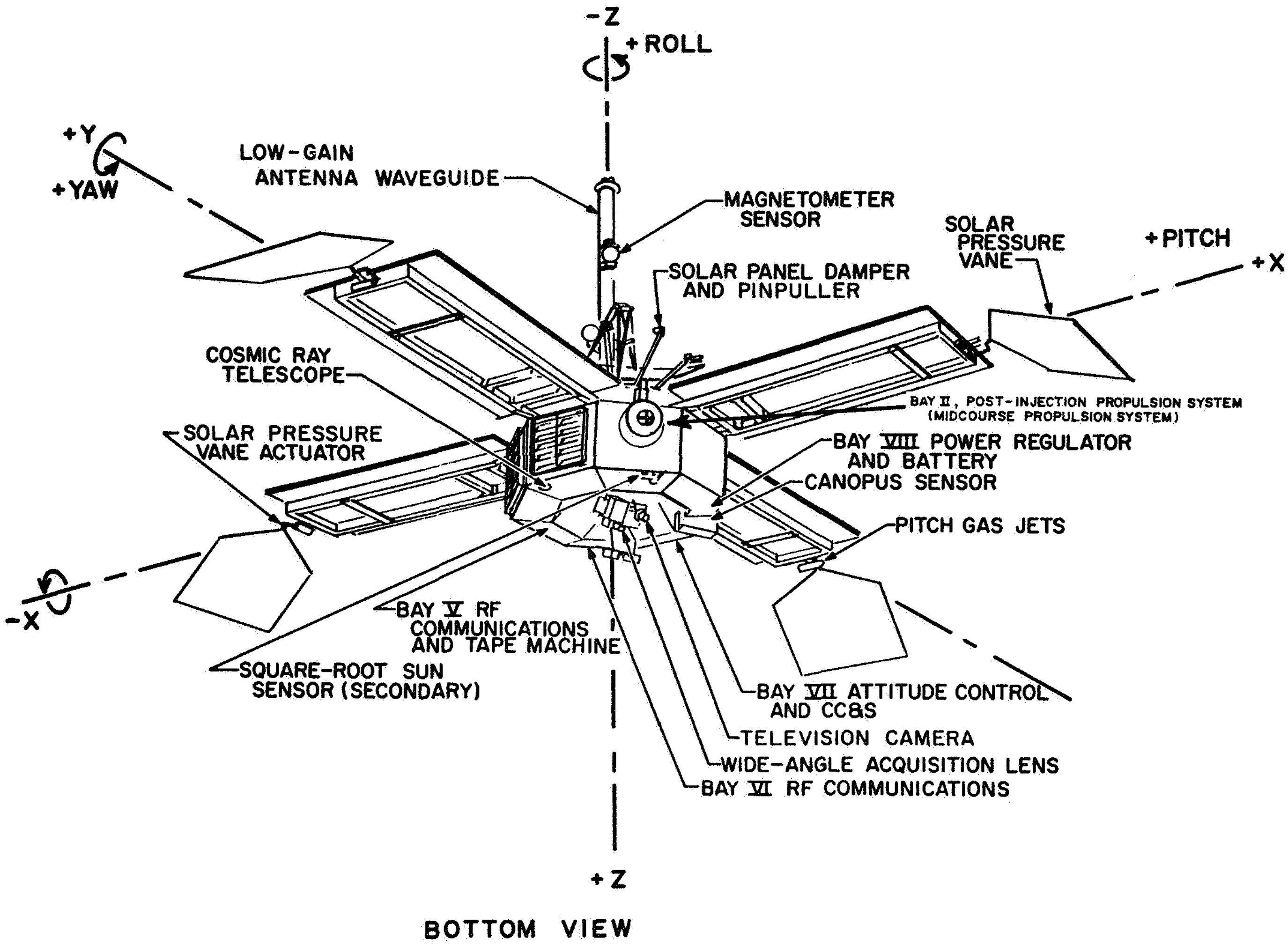
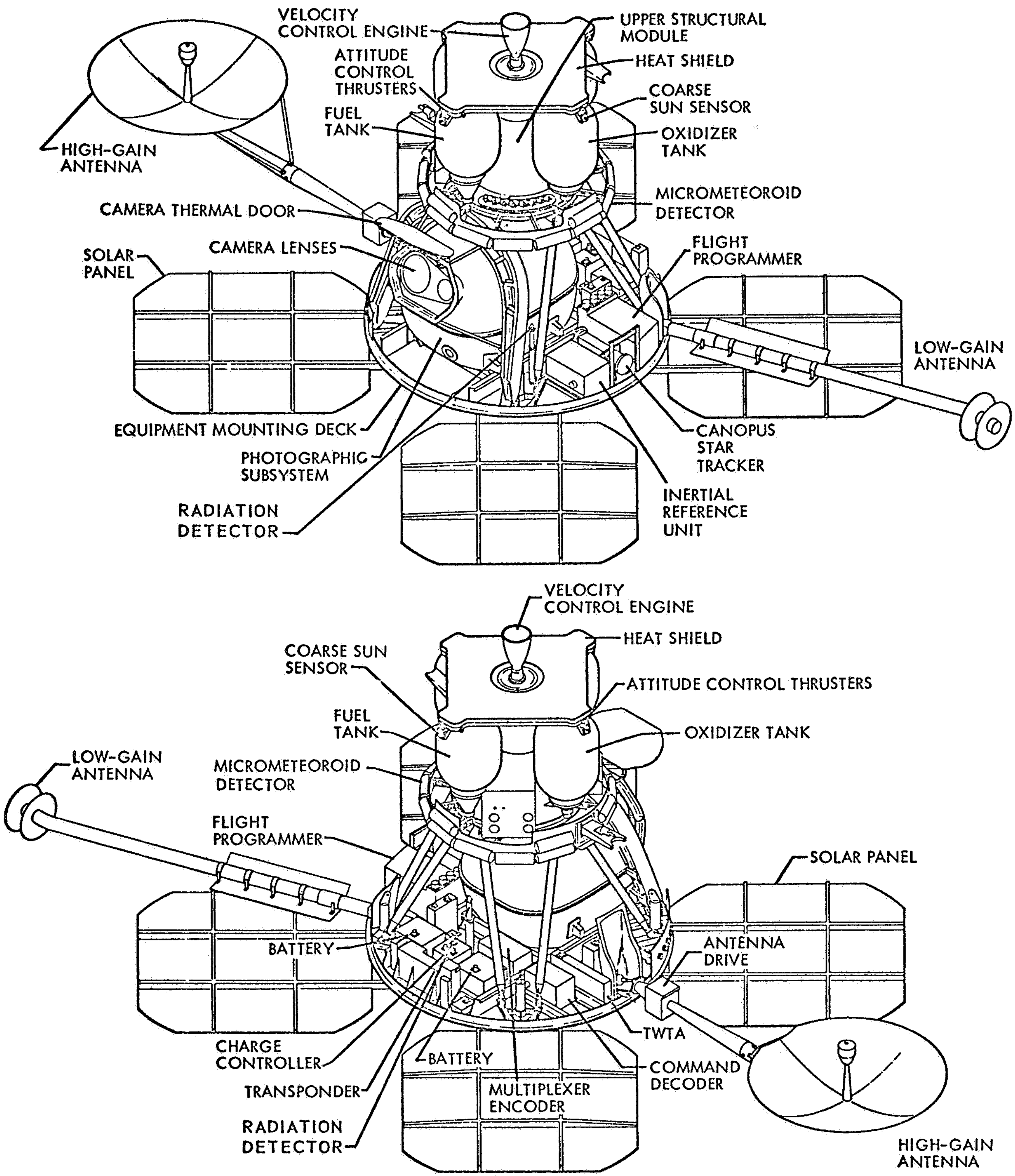


FIGURE C-2. MARINER IV SPACECRAFT CONFIGURATION$^{(C-2)}$



NOTE: SHOWN WITH THERMAL BARRIER REMOVED

FIGURE C-3. LUNAR ORBITER SPACECRAFT[(C-3)]

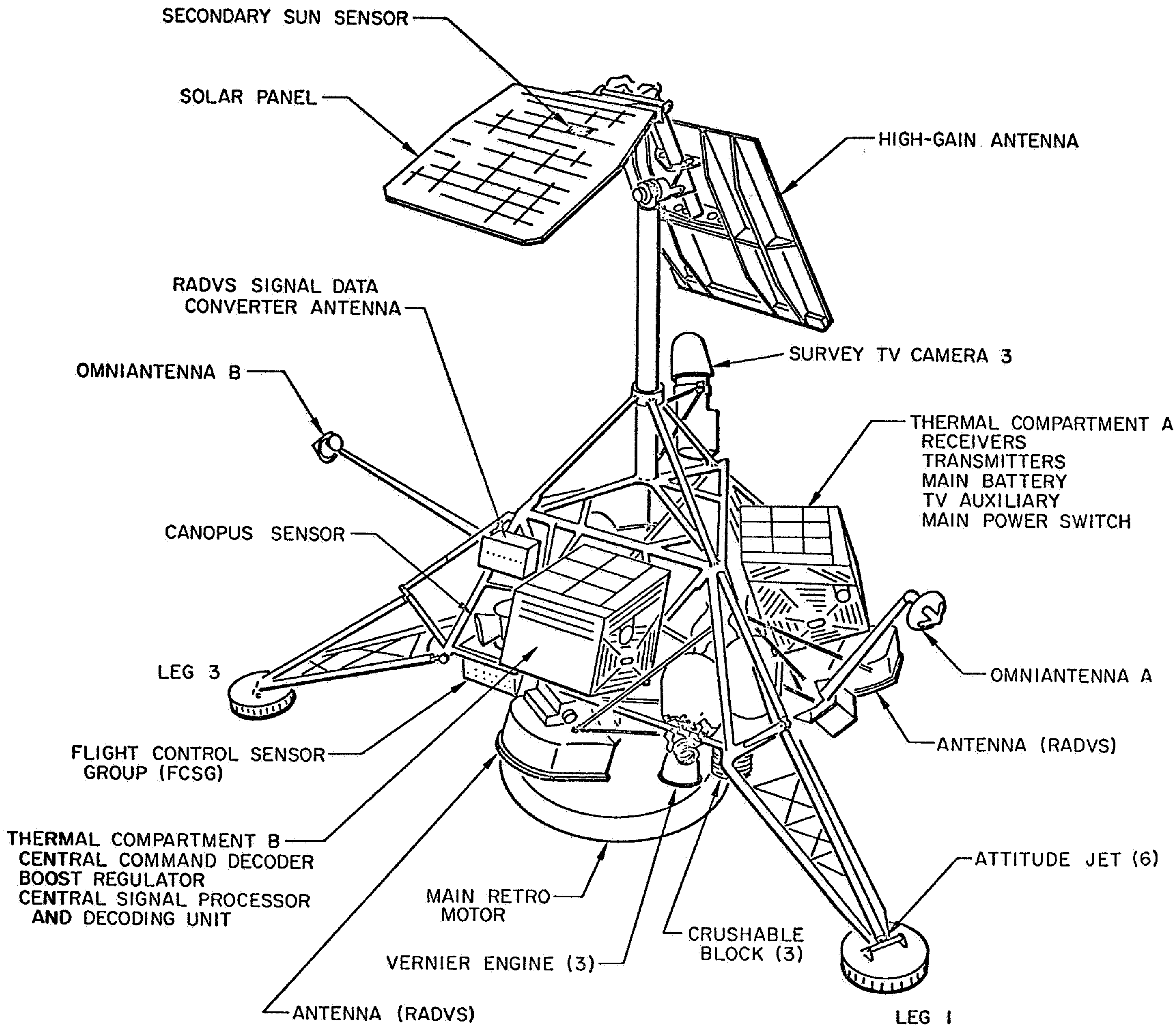


FIGURE C-4. SURVEYOR I SPACECRAFT[(C-4)]

The Surveyor used a three axis cold gas (nitrogen) jet stabilization system during coasts and midcourse firing, and used the terminal descent vernier system (shown in Figure C-4, taken from Reference C-4) for attitude control during the main retropropulsion firing and during final descent. The vernier system used nitrogen tetroxide (N_2O_4) with 10% nitric oxide (NO) added to lower the freezing point and monomethyl hydrazine monohydrate ($MMH \cdot H_2O$) as the propellants.

Earth Orbit Injection

The cutaway drawing of the Syncom spacecraft shown in Figure C-1 shows the TE-375 solid propellant apogee motor used to inject the Syncom into synchronous orbit. The motor delivered a ΔV of 4712 ft/sec, 30 ft/sec above the nominal value.* This placed the spacecraft in an orbit with an eastward drift rate of 7.03 degrees/orbit.

Orbit Correction or Transfers

The Lunar Orbiter I was commanded to make a lunar orbit transfer 11 days after launch to reduce the perilune from 200 km to 40 km, and an orbit trim 15 days after launch. Both maneuvers were made using the bipropellant (N_2O_4-Aerozine 50) velocity control engine shown in Figure C-3 (taken from Reference C-3). This engine was also fired 79 days after launch to cause a lunar impact of the spacecraft.

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* The Syncom was launched on Delta, which does not use a 100 n. mi. parking orbit. Therefore, the 4800 ft/sec nominal ΔV from Appendix B does not hold in this case.

The Syncom spacecraft was also commanded to make an initial orbit correction. The axial control jets were fired to change the drift rate from 7.03 degrees/orbit eastward to 4.58 degrees/orbit westward.

Midcourse Correction

Figure C-5 (taken from Reference C-5) shows the general configuration and mounting of the Ranger VII monopropellant (hydrazine) midcourse motor. The motor is typical of those used on the Ranger and Mariner series.

Figure C-6 (taken from Reference C-6) shows a schematic of the Mariner A midcourse propulsion systems.* The major system components are high pressure gas storage tanks (nitrogen at 3000 psia), a pressure regulator which maintains the propellant tank at 310 psia, the propellant (N_2H_4) tank, the rocket motor, the N_2O_4 initiating oxidizer supply and the valving system. The Mariner A motor contained a bed of 3/16 inch diameter spherical particle catalyst which accelerated the propellant decomposition. The catalytic reaction was not spontaneous and required that a 15 cc slug of N_2O_4 be injected into the engine at startup. This provided a 1-second bipropellant combustion period during which the catalyst bed was heated and catalytic decomposition was initiated.** The valving system used a number of explosively operated valves, which is typical for operations in which minimum long term leakage is desired.

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* The system was developed for the Mariner A spacecraft, which was cancelled due to Centaur development difficulties. A modified version was subsequently used on the Mariner IV and V spacecraft. Figure C-2 shows its location on Mariner IV.

** Future monopropellant midcourse motors will probably use a spontaneous catalytic reaction, eliminating the oxidizer start.

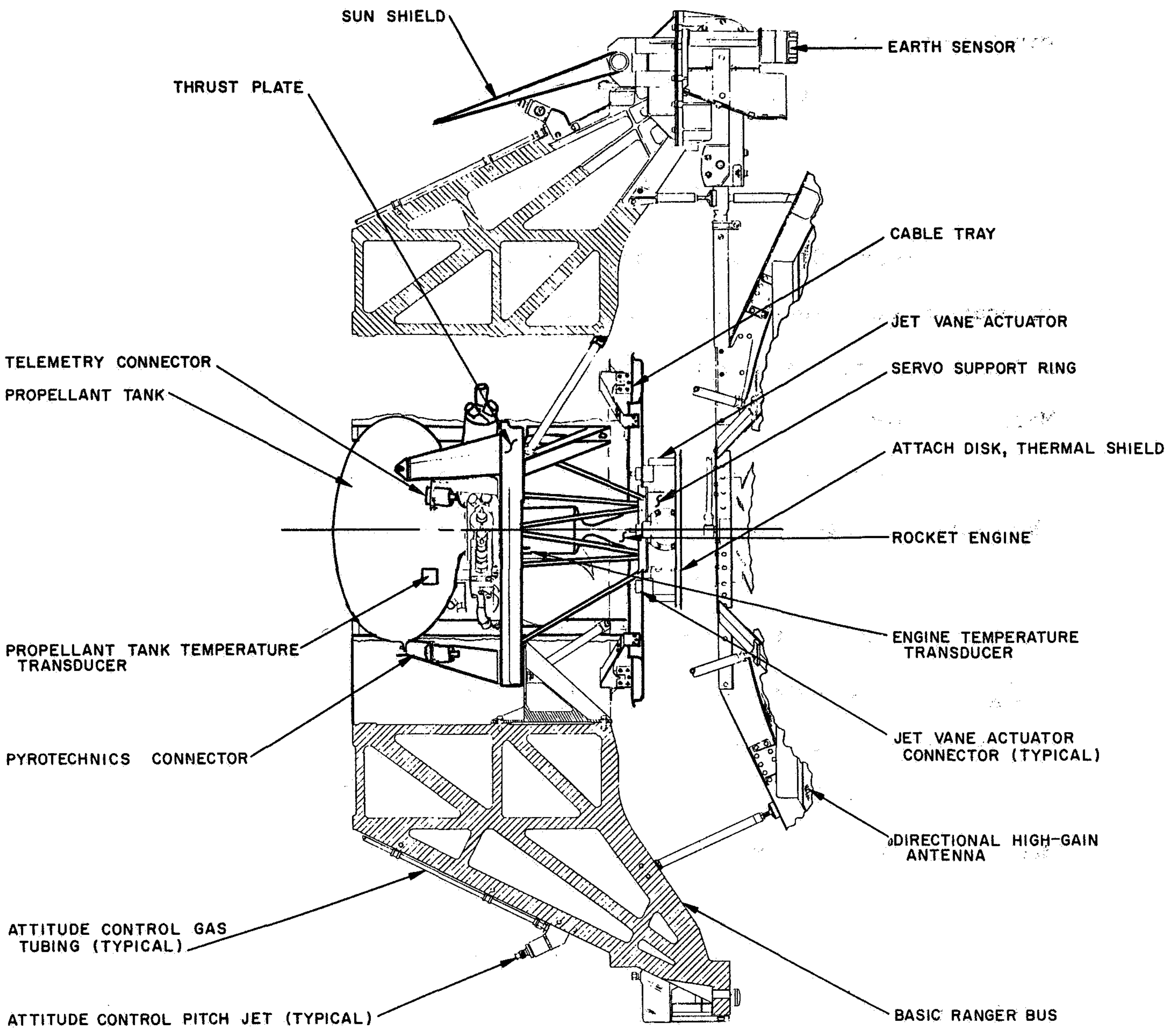
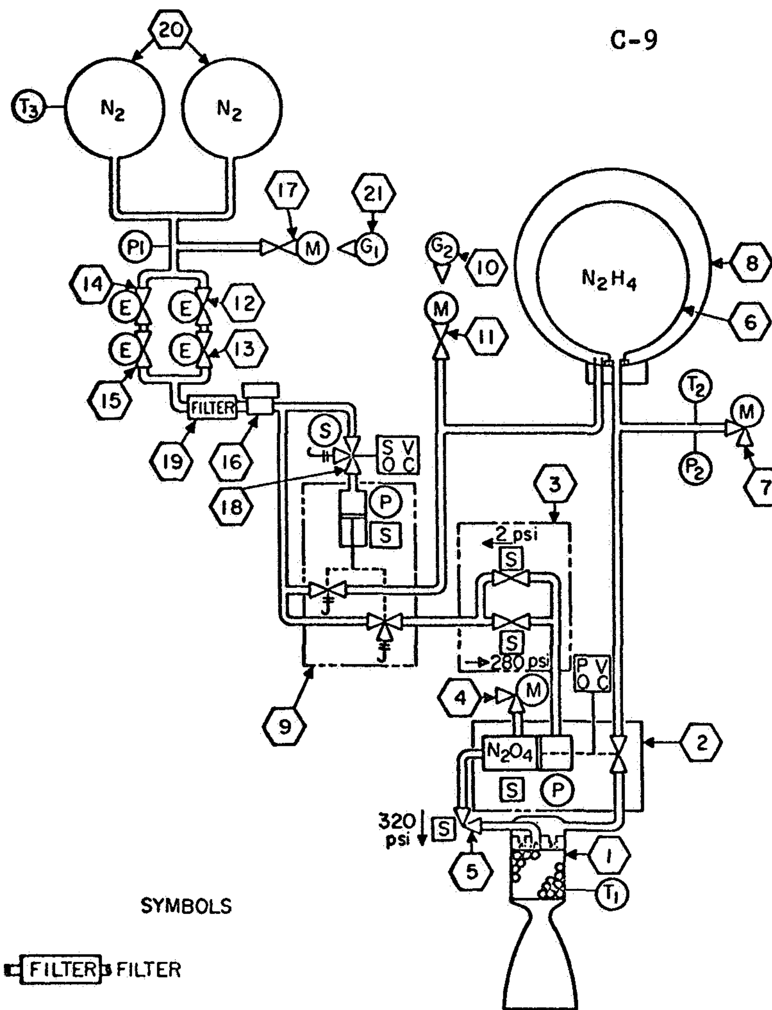


FIGURE C-5. RANGER VII MIDCOURSE PROPULSION SYSTEM[(C-5)]



COMPONENTS

1 ROCKET ENGINE
2 PROPELLANT VALVE, INCLUDING OXIDIZER RESERVOIR AND PUMP
3 NITROGEN DELAY VALVE
4 OXIDIZER FILL VALVE
5 OXIDIZER BACK PRESSURE VALVE
6 PROPELLANT BLADDER
7 PROPELLANT FILL VALVE
8 PROPELLANT TANK
9 PILOT OPERATED CONTROL VALVE
10 PROPELLANT TANK VISUAL PRESSURE GAGE, 0-300 psi
11 PROPELLANT TANK NITROGEN PREPRESSURIZATION VALVE
12 NITROGEN START VALVE A, NORMALLY CLOSED
13 NITROGEN SHUTOFF VALVE A, NORMALLY OPEN
14 NITROGEN START VALVE B, NORMALLY CLOSED
15 NITROGEN SHUTOFF VALVE B, NORMALLY OPEN
16 NITROGEN PRESSURE REGULATOR
17 NITROGEN TANK FILL VALVE
18 SOLENOID PILOT VALVE
19 NITROGEN FILTER
20 NITROGEN TANK
21 NITROGEN TANK VISUAL PRESSURE GAGE, 0-4000 psi

SYMBOLS

FILTER FILTER
PRESET REGULATOR
10 COMPONENT NUMBERS
P2 INSTRUMENTATION NUMBERS
M TWO-WAY VALVE, MANUALLY OPERATED
G1 VISUAL PLUG-IN PRESSURE GAGE
S BACK PRESSURE SPRING RETURN VALVE, PRESSURE OPERATED
TWO-WAY DOUBLE SEAL VALVE, INTERNALLY VENTED CAVITY WHEN CLOSED
TWO-WAY VENTED VALVE, PNEUMATICALLY OPERATED
E TWO-WAY VALVE, EXPLOSIVELY OPERATED
M TWO-WAY ANGLE VALVE, MANUALLY OPERATED
S TWO-WAY VENTED VALVE, SOLENOID OPERATED
P S TWO-WAY SPRING RETURN VALVE, PNEUMATICALLY OPERATED

INSTRUMENTATION

PRESSURE TRANSDUCERS
P1 NITROGEN TANK
P2 PROPELLANT TANK

PRESSURE GAGES (VISUAL)
G1 NITROGEN TANK
G2 PROPELLANT TANK

TEMPERATURE TRANSDUCERS
T1 ROCKET ENGINE
T2 PROPELLANT
T3 NITROGEN TANK

VALVE POSITION INDICATORS
P V / O C PROPELLANT VALVE OPEN AND CLOSE MICROSWITCH
S V / O C SOLENOID VALVE OPEN AND CLOSE SIGNAL GENERATOR

FIGURE C-6. SCHEMATIC OF MARINER A MIDCOURSE AND APPROACH-CORRECTION PROPULSION SYSTEM(C-6)

Volume restrictions on the Mariner A limited the motor nozzle to a 44:1 expansion ratio, which resulted in a vacuum Isp of 235 seconds. Use of a 100:1 expansion ratio would add approximately 10 seconds to the specific impulse. As is typical for propulsion systems where factors other than performance are significant (in this case, storability, reliability, and impulse reducibility considerations dominate) the propellant fraction (weight of N_2H_4/total engine weight) is low (λ_p = .503).

The Lunar Orbiter and Surveyor both used their liquid bipropellant motors for the midcourse correction. The schematic for the Lunar Orbiter motor is shown in Figure C-7 (taken from Reference C-3). The arrangement of the Surveyor liquid bipropellant system (the vernier system) is shown in Figure C-8 (taken from Reference C-4).

Orbiter Retropropulsion

Figure C-9 illustrates the TE-458 solid propellant retromotor of the AIMP-D (taken from Reference C-6). For this spacecraft, a loose Lunar orbit was desired, with the exact final orbital elements not a critical consideration to operation of the spacecraft. Therefore, it was possible to use a solid propellant retromotor, with the ignition command given from the ground and the motor allowed to thrust until the propellant was depleted.

For the Lunar Orbiter (see Figure C-3), control of the retro ΔV and, thus, the initial orbital characteristics was more critical because of the photographic mission. Furthermore, multiple uses of the motor were required for orbit changes and trim. Therefore, a

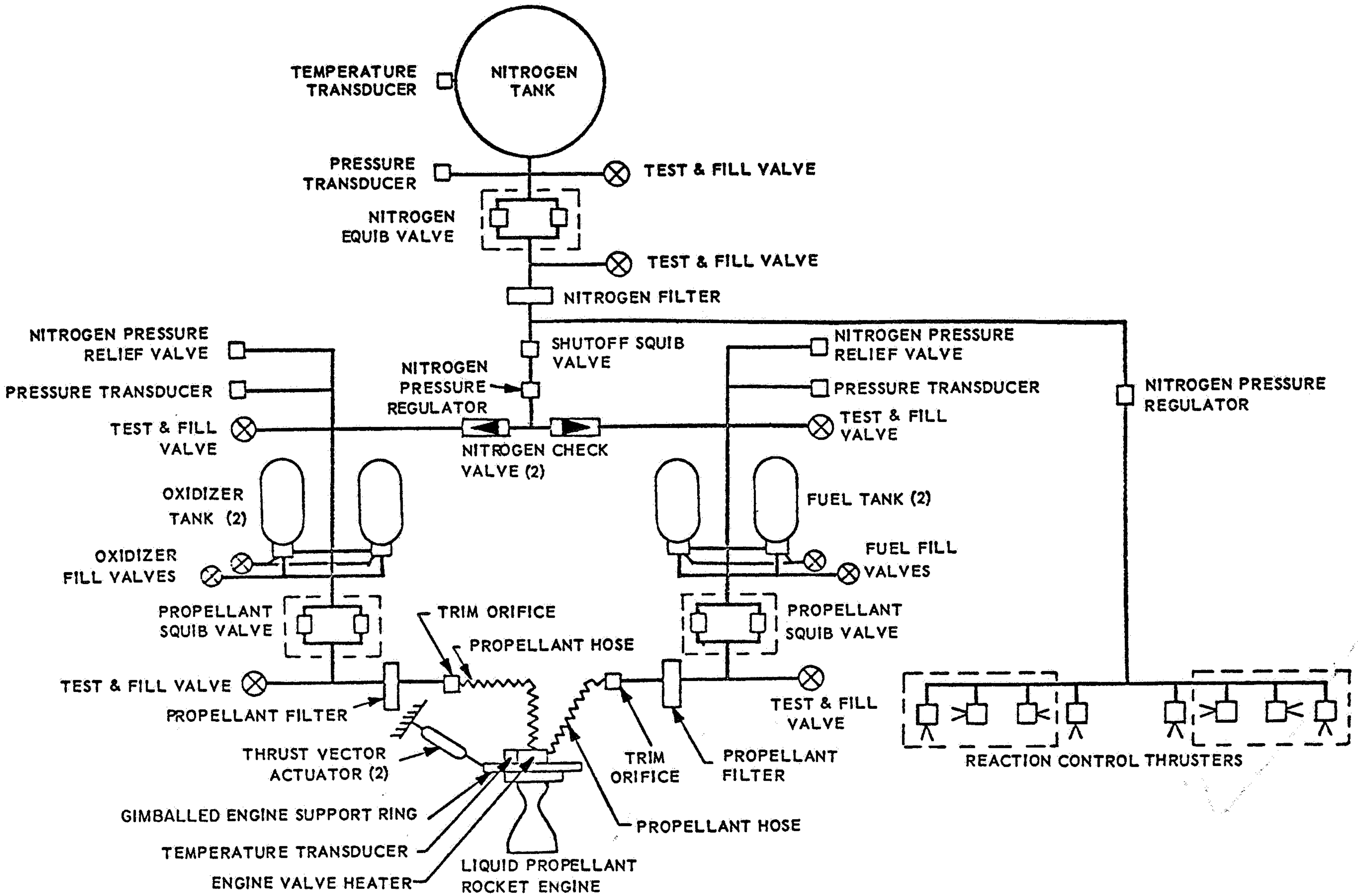


FIGURE C-7. LUNAR ORBITER VELOCITY AND REACTION CONTROL SUBSYSTEM(C-3)

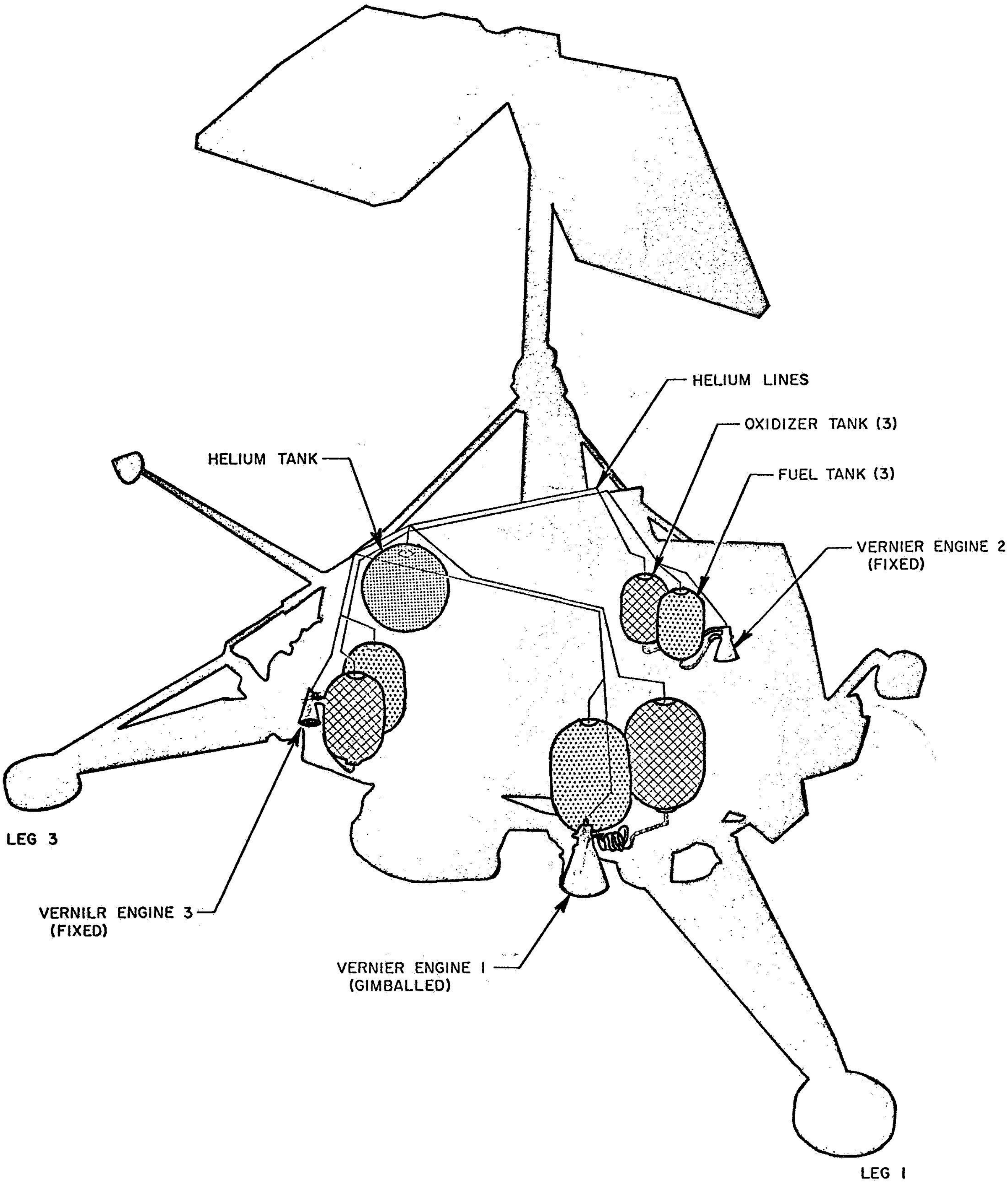


FIGURE C-8. SURVEYOR I VERNIER PROPULSION SYSTEM(C-4)

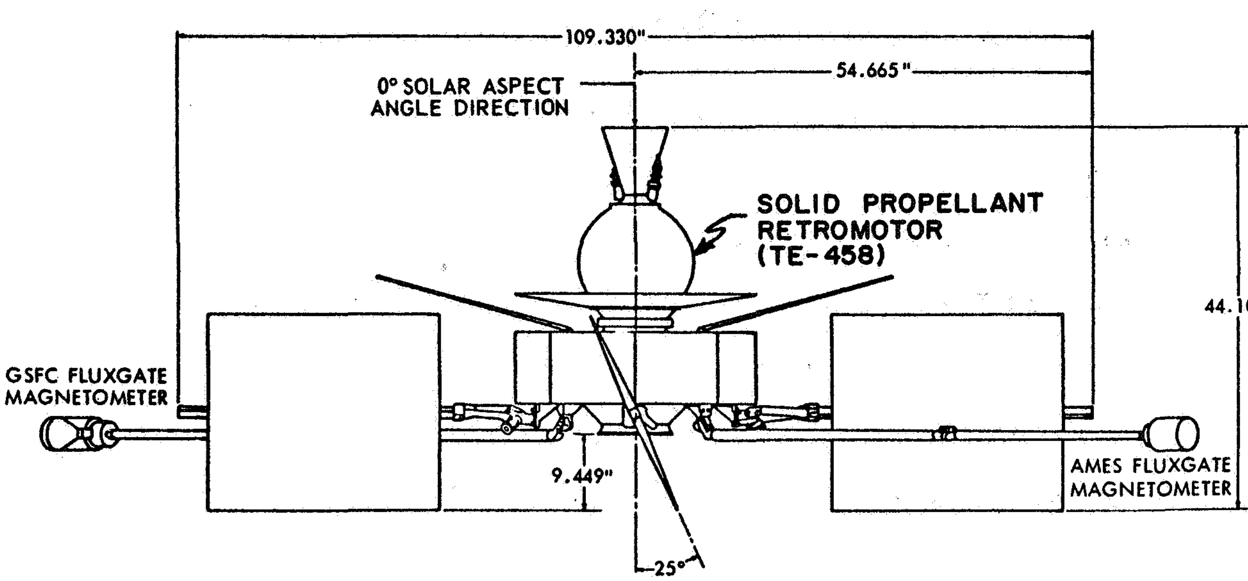


FIGURE C-9. AIMP-D SPACECRAFT SIDE VIEW(C-6) liquid bipropellant motor was used. The precision available from this type of system is illustrated by the fact that the delivered initial retro ΔV differed from the desired by only .35 m/sec (790.0 m/sec desired, 789.65 m/sec delivered).

Planetary Direct Lander

A number of Surveyor spacecraft have been soft landed on the Moon. The descent control propulsion is provided by two subsystems. The main retro ΔV is provided by the solid propellant motor (TE364) shown in Figure C-10 (taken from Reference C-4). Terminal descent control is provided by the vernier propulsion system shown in Figures C-4 and C-8. The nominal sequence of events during terminal descent are shown in Figure C-11 (taken from Reference C-4). Altitude and velocity information is provided by a multibeamed doppler radar system. During the terminal descent the attitude reference is provided by an inertial reference system. The velocity vector is maintained parallel to the flight path so that the spacecraft executes a gravity term in descending to the Lunar surface.

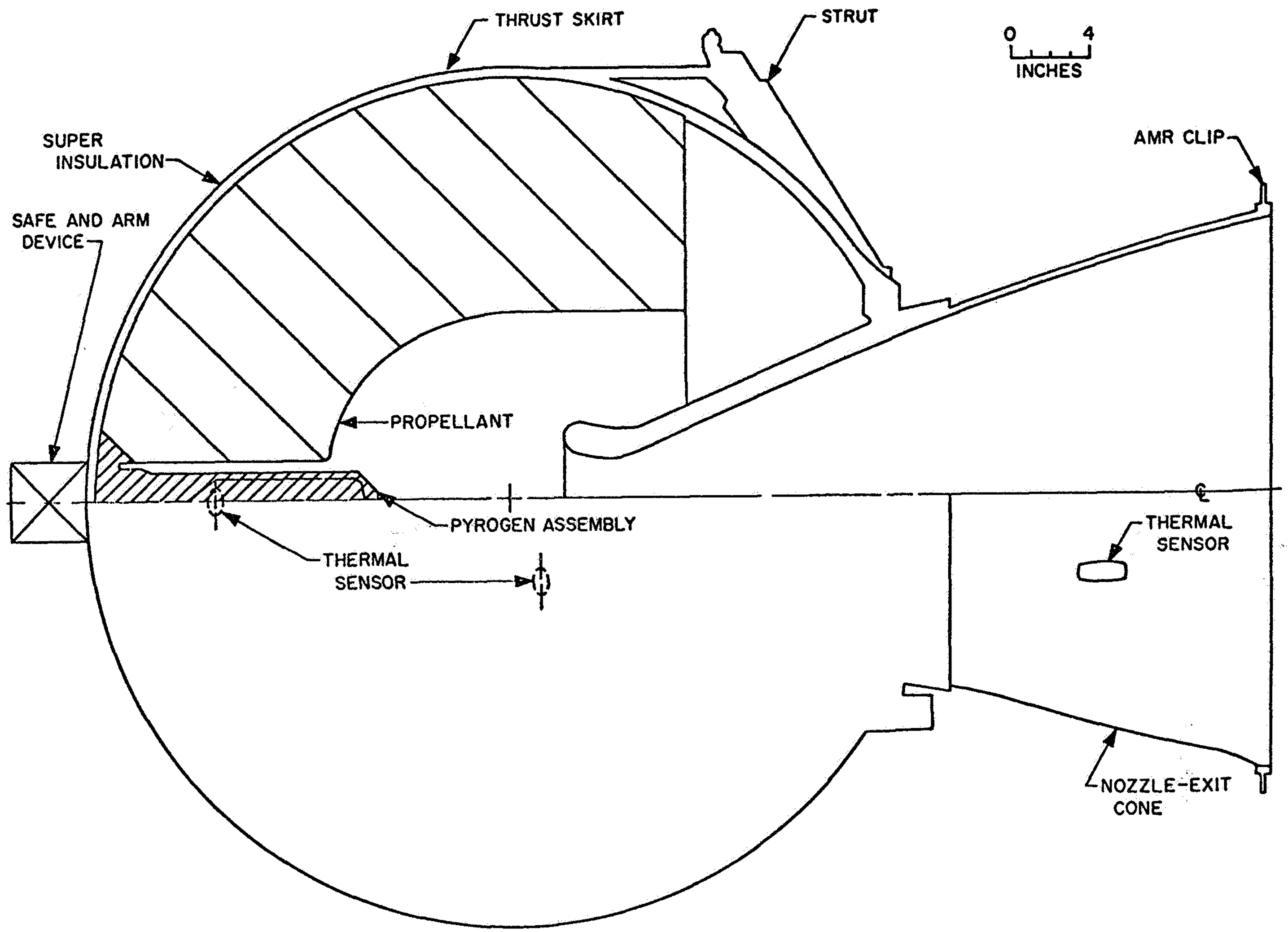


FIGURE C-10. SURVEYOR I MAIN RETROMOTOR (TE-364)(C-4)

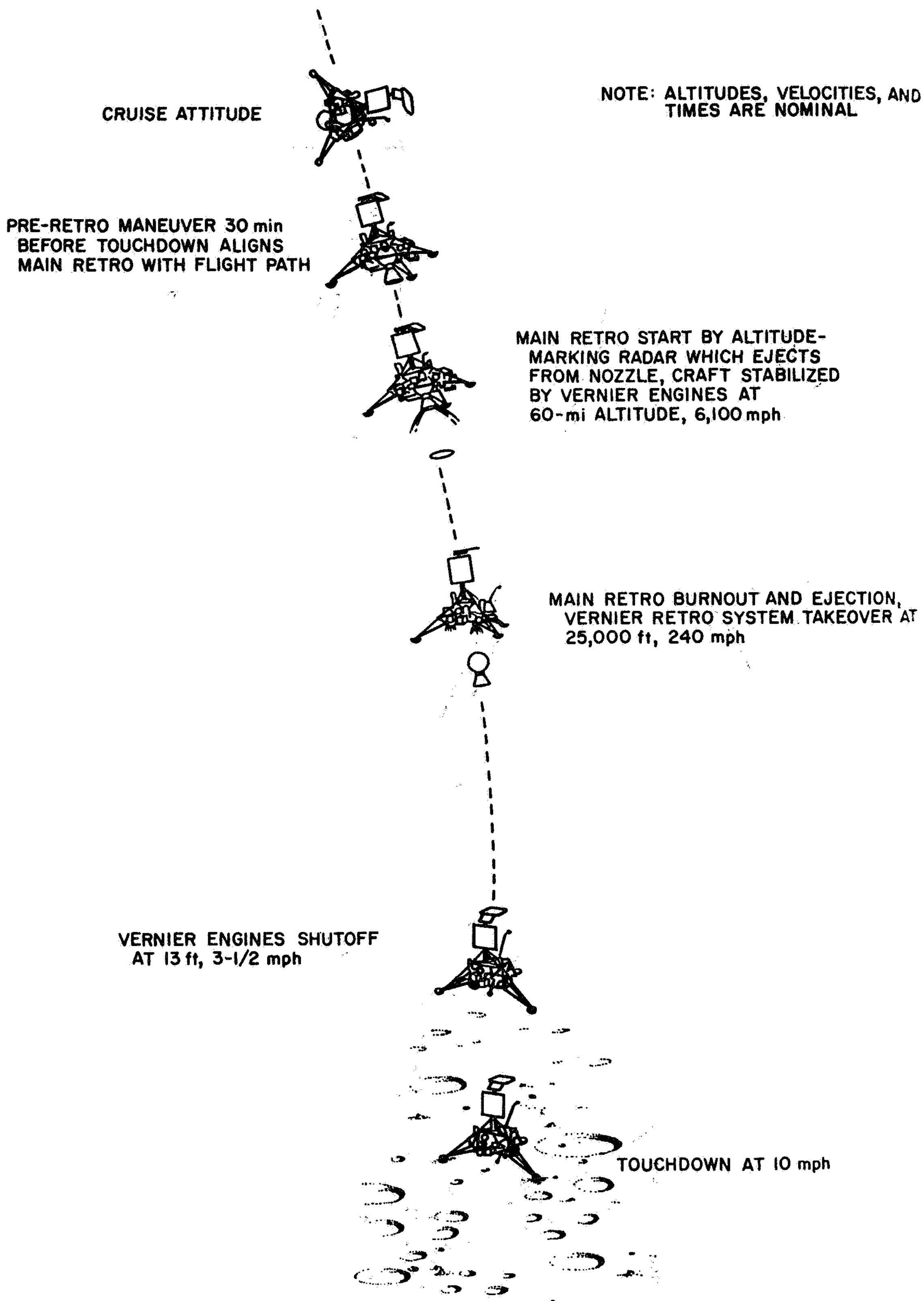


FIGURE C-11. SURVEYOR I TERMINAL DESCENT RETROPROPULSION NOMINAL EVENT SEQUENCE(C-4)

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