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R. V. Evleth, Manager
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NERVA Rocket Operations

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DUAL MODE NERVA APPLICATIONS STUDY
LETTER PROGRESS AND STATUS REPORT NO 2
(Contract NAS8-28119)

21 January 1972

DMNAS-72-3

J. E. Boretz, Study Manager
Dual Mode NERVA Applications Study Project

Prepared For:

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TRW
SYSTEMS GROUP

1.0 INTRODUCTION

This is the second letter progress and status report for the Dual Mode NERVA Applications Study and covers activities for the period January 1 to 15, 1972.

2.0 REFERENCES

(a) TRW Progress and Status Report No. 1 (DMNAS-71-16) of 6 January 1972.

3.0 GENERAL

3.1 The Reference (a) report was prepared and submitted to NASA/MSFC

3.2 The first TRW/ANSC Technical Coordination meeting was held at TRW Systems on 11 January 1972 and reported in ANSC Report N2010:063 of 13 January 1962. Significant technical conclusions are discussed in Section 4 of this report.

4.0 PROGRESS

4.1 Mission Analysis (Task 1)

Work is continuing on compiling the data for the mission velocity requirements. The velocity increments for earth orbital transfer and to accomplish plane changes at various orbital altitudes are summarized in Tables 1 to 3. Similar data has been generated for the lunar transfer velocity requirements (Table 4). In addition, mission velocity requirements have been determined for the inner planets Mercury and Venus (Table 5) and the outer planets Mars (including Phobos and Deimos) Jupiter (including Callisto) and Saturn (Table 6). Finally, data for velocity requirements for ballistic probes (solar and inner planetary and outer planetary regions) has also been generated (Figures 1 to 7). A four

planet grand tour mission is outlined in Table 7. This is preliminary data which is being reviewed and checked, and will ultimately be used to determine payload capabilities for the various synthesized missions.

The baseline NERVA stage characteristics have been revised to reflect the latest data available and this has been summarized in Tables 8 and 9. Utilizing these characteristics data was prepared for various propulsion modules varying in size from 4 propellant modules (tandem stacking) to 28 propellant modules. Figures 8, and 10 to 14 depict this data. A typical geosychronous shuttle mission is shown in Table 10.

Figure 7 depicts initial vehicle weight and injected payload data for the 4 propellant module vehicles including leave earth gravity losses. Similar data is in preparation for the other vehicles. The gravity loss factors being used are shown in Figures 15, 16, and 17. The data presented in these figures were generated using the (modified) Rapid Analytic Trajectory Simulation (RATS) program. The burn arc for the first thrusting maneuver was controlled by terminating the thrust period on a preselected value of true anomaly of the first elliptic (generated by the first burn). Subsequent thrusting periods were initiated at a negative true anomaly with thrusting to an equal position value of true anomaly of the elliptic (or hyperbola) generated in the same thrusting period. The third thrust period was generated in the same way as the second. In this manner near symmetry about the periapsis point was maintained during the thrusting period. The thrust pointing directors were maintained in the initial velocity vector direction.

No attempt was made to optimize the length of the burn arc for multiple burn cases. Thrust periods were determined such that the second to last burn did not generate hyperbolic velocities.

The F_L factor data for the single burn, 2 burn and 3 burn cases are based on the ratio of ideal delta velocity (ΔV_{ID}) to actual delta velocity (ΔV_a), where the ΔV_{ID} was calculated assuming a fixed specific impulse and constant thrust (constant mass flow rate) for the burn period. The ordinate of the plots represents ΔV_a, and the parameter is thrust to weight ratio.

In order to interpret what the curves represent, consider the following: The actual velocity (ΔV_a) gained for a fixed thrust period, Δt, will be less than the ideal velocity (ΔV_{ID}) gained for the same period. In other words, Δt seconds of thrust are required to generate ΔV_{ID}, but the real velocity added is only ΔV_a. Now if the actual velocity is plotted versus the ratio $\Delta V_{ID}/\Delta V_a$, it can be interpreted as (mission) required velocity versus the ratio of (actual velocity input needed)/(mission required velocity). Thus an ordinate value multiplied by an obscissa value represents the velocity the system must generate to satisfy the requirements.

These gravity loss factors will be used to generate the other payload/initial vehicle weight data which will include gravity losses for the range of propellant modules under consideration. By utilizing these curves it is planned to synthesize vehicle performance curves for the significant earth orbital and lunar transfer missions. These curves will depict returned payload versus delivered payload for various number of propellant modules and will include the effects of gravity losses for certain of the configurations. It will be assumed that these vehicles will be reusable (no staging) since this appears to be the most cost effective approach for the currently envisioned earth orbital and lunar shuttle missions. In a similar fashion, this data is being compiled for inner and outer planet missions. For selected missions the payload capability for various vehicle configurations is being determined. However, in these cases both "continuous staging" and "no staging during burn but before arrival at planet" are being considered.

Payload synthesis has been initiated and several typical classes are being considered. These include:

I <u>Orbiter Class</u>

- o Side looking radar (for heavy atmosphere planets)
- o Imaging and related data rate (i.e., radar altimeter)
- o Probes

II Sample Return Class

- o Sample gatherer
- o Requires return stage
- o Consider Callisto, Deimos, Mars, Asteroids

III Manned Mars Class

- o Synthesize all required subsystems for Mars manned landing and return

IV Lunar Class

- o Initial transportation of manned lunar base
- o Lunar logistic resupply
- o Unmanned roving vehicle for surface exploration
- o Orbiting Lunar station support

V Earth Orbit Class

- o Low altitude space station energy stage
- o Low altitude to geosynchronous orbit transfer of scientific payload
- o Others

A preliminary assessment of typical payloads that could be synthesized for unmanned missions are listed in Table 11.

Data for these typical payloads is being compiled so that weight and electrical and/or thermal energy requirements can be established. As soon as this has been completed, efforts will be directed towards establishing Dual Mode NERVA applications that are payload oriented.

The mission velocity requirements for a geosynchronous mission are presented in Table 10. This mission profile is similar to the profile used by MDAC in their Phase III Study and also used in preparing the Mission Planning Handbook. Vehicle performance for various numbers of propellant modules is shown in Figure 18. This figure presents the returned payload as a function of delivered payload for vehicles consisting of 4 to 22 propellant modules. Return payload is defined as the total payload above the RNS on the descent leg of the mission, while delivered payload is the total payload on the ascent leg of the mission. The performance is shown for a constant

specific impulse of 825 sec for each burn, no gravity losses and no aftercooling propellant utilization. The effect of variations in specific impulse for each burn, gravity losses, and aftercooling will be evaluated during the next reporting period. For comparison purposes, the performance of an eight propellant module vehicle presented in MDAC Phase III Study is also shown in Figure 18. The difference in performance is due to differences in inert vehicle weight, specific impulse variations for each burn, and aftercooling propellant utilization. As can be seen, these parameters can reduce the payload capability by about 35 percent.

4.2 Dual Mode Applications Systems Evaluation (Task 2)

A meeting was held with ANSC and WANL on 11 January 1972 to review the preliminary list of Dual Mode System Applications (Table 1 of Reference a). It was agreed that the cooldown propellant reduction application (Items 8 and 9) and those applications associated with tank pressurization, boiloff conservation and engine chilldown or conditioning (Items1, 2, 3, and 11) had merit for significant weight savings and would require electric power for their implementation. It was pointed out that all the engine electrical actuators are currently designed to meet stringent reliability requirements (i.e. >R=0.9999) and hence considering an improvement in this area by using oversize actuators would be of marginal benefit to the stage. Hence, it was mutually agreed to eliminate item 4 of Table 1 of Reference (a).

The analyses for the use of the high temperature and low temperature radiators to reduce afterburning cooldown propellants is being extended at the recommendation of ANSC to include the following:

(a) A low temperature radiator specific weight of 0.5 lb/ft^2.
(b) A high temperature radiator specific weight of 3.0 lb/ft^2.
(c) Overdriving of the low temperature radiator to 660°R during the initial cooldown period.
(d) Cooldown of the reactor to values below $200KW_t$.

Analyses have been initiated to evaluate the use of propellant module boost pumps as a potential Dual Mode application. This approach would permit reduction of the maximum required tank pressure (i.e. from 30 psia to approximately 15 psia) and a potential weight saving with

respect to both gas pressurant weight ard tank structural weight. The use of the boost pump for the propellant destratification and chilldown applications is also under consideration.

As may be noted, the initial emphasis in this task has been towards engine related Dual Mode applications. This was done since considerable data was available on the engine subsystems. It is anticipated that payload oriented applications will be initiated as soon as definitive payload characteristic information becomes available from Task 1.

5.0 PLANNED ACTIVITIES FOR THE NEXT REPORTING PERIOD

5.1 The payload characteristics for the unmanned missions (subtasks 1.1-2 and 1.2-2) will be finalized. Data will then be available to permit initial screening of candidate Dual Mode systems to be conducted. This will also permit the synthesis of the required vehicle performance for these selected missions.

5.2 The payload data for the inner and outer planet missions for various vehicle configurations (subtasks 1.1-1 and 1.2-1) will be completed. Initial screening of current and new missions (subtasks 1.1-3 and 1.2-3) will be conducted.

5.3 The work pertaining to the Dual Mode applications (subtasks 2.1-1, 2.1-2 and 2.2-2) pertaining to aftercooling propellant consumption reduction will be finalized and include the ANSC recommended changes in parameters. In addition, the applications pertaining to the pressurization, chilldown, and reliquefication applications will be continued. The work pertaining to the vehicle and payload applications will continue with the preparation of additional parametric data for the alternate power systems (i.e., nuclear dynamic and nuclear static systems) for use in comparison with the Dual Mode systems. An initial assessment as to the desirable electric power range for the Dual Mode system as it applies to the unmanned payload characteristics will also be made.

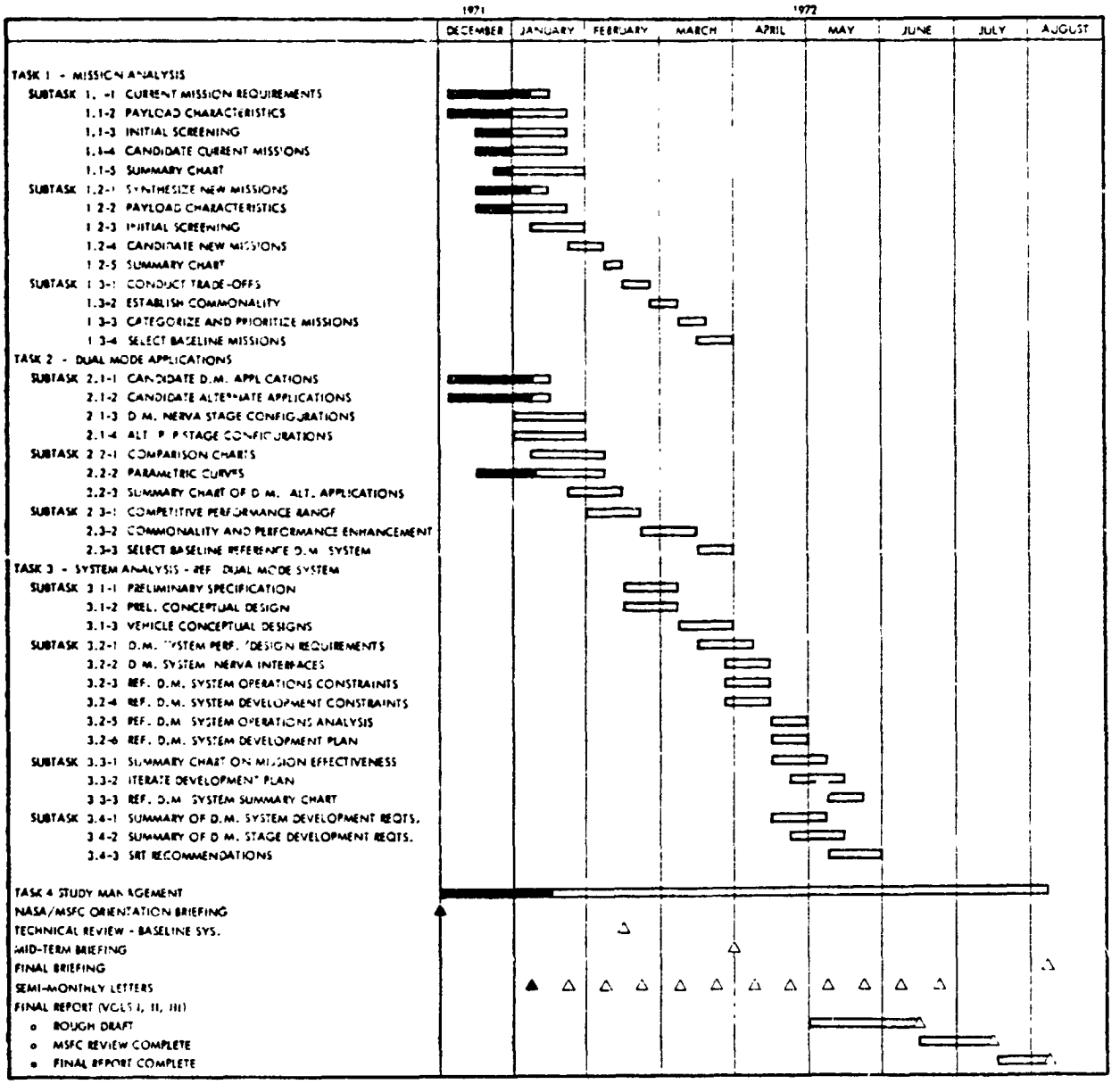
6.0 PROGRAM SCHEDULE

6.1 The status of the work completed is shown on the Task Progress Report Schedule (Attachment I). The definition of payload characteristics has been a pacing study item. However, now that the mission requirements and vehicle payload capabilities have been very nearly completed, considerable effort is being expended in this area. Hence, at this time there does not appear to be any significant problem which would preclude completion of Tasks 1 and 2 with the designated milestone date (30 March 1972).

6.2 The cumulative manhours for this study are shown on Attachment II.

DUAL MODE NERVA APPLICATIONS STUDY

TASK PROGRESS REPORT



DUAL MODE NERVA APPLICATION STUDY CUMULATIVE MANHOURS

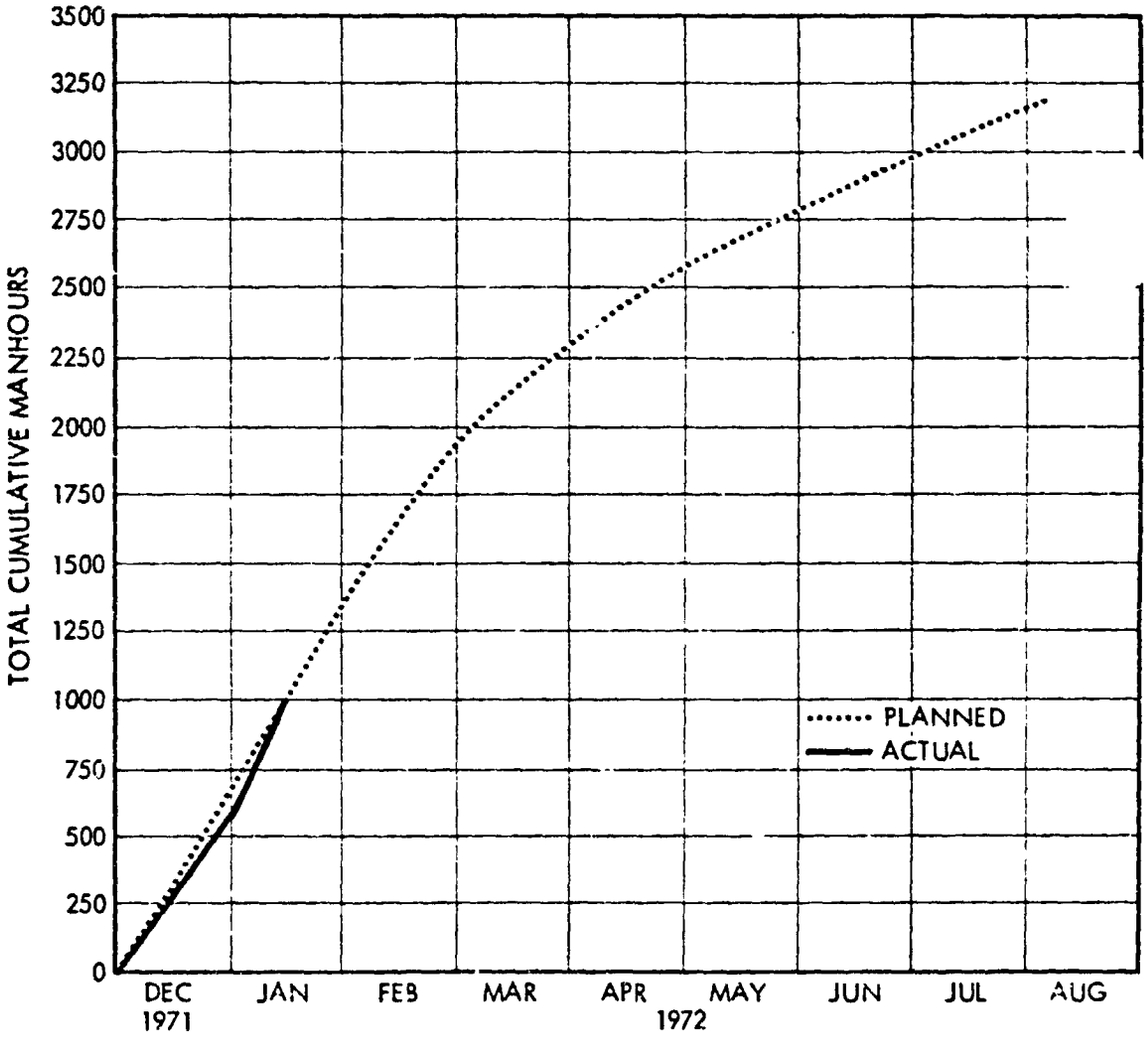


Table 1. Earth Orbital Transfer Velocity Requirements

FINAL ORBIT ALTITUDE AT APOGEE, h_a (nmi)	ΔV_1 SINGLE HOHMANN TRANSFER to h_a (FPS)	ΔV_2 PLANE CHANGE CIRCULARIZE, i = 0° (FPS)	ΔV_2 PLANE CHANGE CIRCULARIZE, i = 15° (FPS)	ΔV_2 CIRCULARIZE i = 31.5° (FPS)	ΔV_2 PLANE CHANGE CIRCULARIZE, i = 45° (FPS)	ΔV_2 PLANE CHANGE CIRCULARIZE, i = 75° (FPS)	ΔV_2 PLANE CHANGE CIRCULARIZE, i = 90° (FPS)
300	70.	13495.	7135.	70.	5840.	18420.	24290.
1000	1115.	12160.	6490.	1045.	5350.	16565.	21825.
10000	6315.	7325.	5440.	4505.	5150.	9080.	11335.
19324 (Synchronous)	7795.	6200.	5200.	4755.	5060.	7225.	8600.
60000	9380.	4420.	4110.	3985.	4070.	4765.	5270.
100000	9750.	3730.	3560.	3490.	3535.	3925.	4220.

ASCENT PROFILE

1) Initial Parking orbit, 260 n mi circular, i = 31.5°.
2) Single Hohmann transfer from 260 n mi parking orbit to h_a (ΔV_1).
3) Combined single impulse plane change and circularization (ΔV_2).

NOTE:
The velocity requirements indicated in the above table assume controlled perigee, i.e., inject into transfer ellipse at nodal crossing (ω = 0° or 180°).

Table 2. Earth Orbital Transfer Velocity Requirements

FINAL ORBIT ALTITUDE AT APOGEE, h_a (nmi)	ΔV_1 SINGLE HOHMANN TRANSFER to h_a (FPS)	ΔV_2 PLANE CHANGE $i = 0°$ (FPS)	ΔV_2 PLANE CHANGE $i = 15°$ (FPS)	ΔV_3 CIRCULARIZE $i = 31.5°$ (FPS)	ΔV_2 PLANE CHANGE $i = 45°$ (FPS)	ΔV_2 PLANE CHANGE $i = 75°$ (FPS)	ΔV_2 PLANE CHANGE $i = 90°$ (FPS)
300	70	13,550	7,160	70	5,870	18,495	24,390
1,000	1,115	13,060	6,905	1,065	5,655	17,830	23,510
10,000	6,315	12,470	6,595	4,505	5,405	17,030	22,460
19,324 (Synchronous)	7,795	12,765	6,747	4,755	5,525	17,425	22,927
60,000	9,380	13,230	6,995	3,985	5,730	18,060	23,810
100,000	9,750	13,355	7,060	3,490	5,785	18,235	24,045

Ascent Profile:

1) Initial parking orbit, 260 nmi circular, $i = 31.5°$.
2) Single Hohmann transfer from 260 nmi parking orbit to h_a (ΔV_1).
3) Inclination change at ascending node ($\omega = 270°$) (ΔV_2).
4) Circularize at transfer orbit apogee (ΔV_3).

Table 3. Earth Orbital Transfer Velocity Requirements

FINAL ORBIT ALTITUDE AT APOGEE, h_a (nmi)	ΔV_1 SINGLE HOHMANN TRANSFER to h_a (FPS)	ΔV_2 CIRCULARIZE $i = 31.5°$ (FPS)	ΔV_3 PLANE CHANGE $i = 0°$ (FPS)	ΔV_3 PLANE CHANGE $i = 15°$ (FPS)	ΔV_3 PLANE CHANGE $i = 45°$ (FPS)	ΔV_3 PLANE CHANGE $i = 75°$ (FPS)	ΔV_3 PLANE CHANGE $i = 90°$ (FPS)
300	70	70	13,515	7,145	5,850	18,445	24,325
1,000	1,115	1,065	12,400	6,555	5,370	16,930	22,325
10,000	6,315	4,505	7,130	3,765	3,090	9,730	12,830
19,324 (Synchronous)	7,795	4,755	5,474	2,895	2,370	7,474	9,858
60,000	9,380	3,985	3,280	1,735	1,420	4,487	5.905
100,000	9,750	3,490	2,570	1,360	1,115	3,510	4,625

Ascent Profile:

1) Initial parking orbit, 260 nmi circular, $i = 31.5°$.
2) Single Hohmann transfer from 260 nmi parking orbit to h_a (ΔV_1).
3) Circularize at transfer orbit apogee (ΔV_2).
4) Inclination change at nodal crossing in circular orbit (ΔV_3).

Table 4. Lunar Transfer Velocity Requirements

Transfer Time (Hr)	Lunar Position	Mission Characteristic Velocity (FPS)	Lunar Transfer ΔV (FPS)	Equivalent Hyperbolic Excess Velocity At Moon (FPS)	Circularize Lunar Orbit ΔV (FPS)			Lunar Orbit Plane Change ΔV (FPS)					
								15° Plane Change			30° Plane Change		
					60 nmi	100 nmi	938 nmi	60 nmi	100 nmi	938 nmi	60 nmi	100 nmi	938 nmi
50	Apogee	35,420	10,395	5,875	4,300	4,275	4,240						
	Perigee	35,300	10,275	4,925	3,760	3,740	3,515						
60	Apogee	35,300	10,275	4,625	3,610	3,590	3,290						
	Perigee	35,180	10,155	3,890	3,245	3,225	2,900						
72	Apogee	35,175	10,150	3,685	3,100	3,075	2,775	1,395	1,365	1,020	2,765	2,710	2,014
	Perigee	35,080	10,055	3,260	2,945	2,920	2,555						
90	Apogee	35,130	10,105	3,100	2,870	2,850	2,485						
	Perigee	35,060	10,035	2,970	2,800	2,775	2,425						
108	Apogee	35,110	10,085	2,910	2,765	2,735	[illegible],400						
	Perigee	35,050	10,025	2,900	2,760	2,730	2,395						

Ascent Profile

1. Inject into Lunar Transfer Trajectory from 260 nmi circular orbit.
2. Circularize at closest approach to moon.
3. Plane change in circular orbit

Table 5. Inner Planet Mission ΔV Requirements

PLANET	MISSION TYPE	ENCOUNTER TYPE	LAUNCH DATE/ ENCOUNTER DATE	FLIGHT TIME (DAYS)	INJECTION ΔV (FPS)	PLANET RETRO ΔV (FPS)	PLANET ORBIT (nmi)	ESCAPE VELOCITY FROM PLANET SURFACE (FPS)
MERCURY	DIRECT	FLYBY	1977-1996	70-140	16,000-28,900	-	-	-
MERCURY	VENUS SWING-BY	FLYBY	1981-1996	120-350	13,100-16,200	-	-	-
MERCURY	DIRECT	FLYBY	AUG 1985	98	17,245	-	-	-
MERCURY	VENUS SWING-BY	FLYBY	APR 1980/ SEP 1980	254	11,615	-	-	-
MERCURY	DIRECT	ORBITER	FEB 1982	105	19,525	27,775 24,750	1350(CIRC) 135x74,250	13,800 13,800
MERCURY	VENUS SWING-BY	ORBITER	DEC 1981/ JUN 1982	348	16,355	29,560	540(CIRC)	13,800
MERCURY	VENUS SWING-BY	ORBITER	JAN 1982/ JUL 1982	304	13,585	28,280	540(CIRC)	13,800
MERCURY	VENUS SWING-BY	ORBITER	APR 1980/ SEP 1980	254	11,615	42,355	540(CIRC)	13,800
VENUS	DIRECT	FLYBY	1977-1990	120-130 (Type I)	11,400-11,900	-	-	-
VENUS	DIRECT	FLYBY	1977-1990	155-170 (Type II)	11,200-11,800	-	-	-
VENUS	DIRECT	FLYBY	NOV 1981	170	11,475	-	-	-
VENUS	DIRECT	ORBITER	APR 1983	170	11,200	9,320 3,150	3350(CIRC) 3350x67,000	34,100 34,100

Table 3. Outer Planet Velocity Requirements

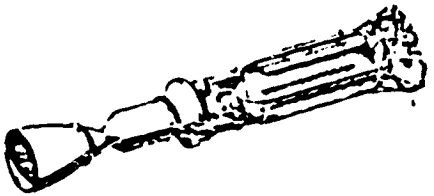
PLANET	MISSION TYPE	ENCOUNTER TYPE	LAUNCH DATE/ ENCOUNTER DATE	FLIGHT TIME (DAYS)	INJECTION ΔV (FPS)	PLANET RETRO ΔV (FPS)	PLANET ORBIT (nmi)	ESCAPE VELOCITY FROM PLANET SURFACE (FPS)
MARS	DIRECT	FLYBY	1977-1990	190-220 (Type I)	11,500-13,600	-	-	-
MARS	DIRECT	FLYBY	1977-1990	250-265 (Type II)	11,800-12,600	-	-	-
MARS	DIRECT	ORBITER	MAR 1986	220	11,525	3,795	180x19,670	-
MARS	DIRECT	PHOBOS ORBIT	MAR 1986	220	11,525	7,440	4985(CIRC)	
MARS	DIRECT	DEIMOS ORBIT	MAR 1986	220	11,525	7,670	12,384(CIRC)	-
MARS	DIRECT	ORBITER	DEC 1979	265	11,825	3,225	180x19,670	
MARS	RETURN	ORBITER	DEC 1981	250	12,120	13,095 6,975	260(CIRC) 260x10,000	
JUPITER	DIRECT	FLYBY	1977-1990	400-600	21,700-32,600	-	-	
JUPITER	DIRECT	ORBITER	JUN 1986	600	21,975	54,965 42,435	3,850(CIRC) 38,500(CIRC)	
JUPITER	DIRECT	CALLISTO ORBIT	JUN 1986	600	21,975	21,205	1.03×10^6(CIRC)	
JUPITER	DIRECT	FLYBY	MAY 1985	600	21,725	-	-	
SATURN	DIRECT	FLYBY	1977-1990	500-1625	23,600-56,500	-	-	
SATURN	DIRECT	ORBITER	APR 1986	1625	23,780	34,650 26,980 17,840 17,450	3,260(CIRC) 32,600(CIRC) 326,000(CIRC) 1,630,000(CIRC)	
SATURN	JUPITER SWINGBY	ORBITER	DEC 1980/ JUL 1986	2040	24,280	10,695	342,000(CIRC)	
URANUS	DIRECT	ORBITER	MAY 1984	2350	28,675	26,780 21,020	12,700(CIRC) 127,000(CIRC)	
URANUS	JUPITER SWINGBY	ORBITER	DEC 1980/ OCT 1985	1755	27,725	41,340 28,870	25,400(CIRC) 25,400x330,000	

Table 7. Four Planet Grand Tour Mission

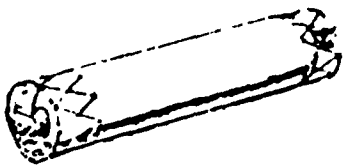
DEPART EARTH	NOV 1979	DEC 1980
ARRIVE NEPTUNE	JAN 1993	DEC 1991
TOTAL TIME OF FLIGHT	4116 DAYS	4015 DAYS
JUPITER SWING-BY DATE	MAY 1981	MAR 1982
JUPITER PASSAGE DISTANCE	97.5 RADII	58.5 RADII
SATURN SWING-BY DATE	AUG 1983	MAR 1984
SATURN PASSAGE DISTANCE	4.4 RADII	2.25 RADII
URANUS SWING-BY DATE	OCT 1988	MAR 1988
URANUS PASSAGE DISTANCE	17.0 RADII	9.6 RADII
EARTH DEPARTURE ΔV FROM 260 N.M. CIRCULAR	24075 FT/SEC	27375 FT/SEC

Table 8. Baseline Stage Characteristics

PROPULSION MODULE



PROPELLANT MODULE



COMMAND AND CONTROL MODULE

SUBSYSTEM	MODULE WEIGHTS (LB)		
	PROPULSION	PROPELLANT	COMMAND AND CONTROL
STRUCTURE	1300	3560	690
METEOROID/THERMAL	710	1240	120
DOCKING/CLUSTERING	80	370	280
MAIN PROPULSION - STAGE	23,300	510	540
NERVA	23,300	--	--
AUXILIARY PROPULSION (WET)	160	--	950
ASTRIONICS	345	115	1905
CONTINGENCY	200	280	200
SUBTOTAL	26,095	6075	4685
PROPELLANT			
RCS	--	--	1250
RESIDUAL	440	465	--
USABLE	10,410	36,035	--
SUBTOTAL	10,850	36,500	1250
TOTAL	36,945	42,575	5935

(Revision 1)

Table 9. RNS Stage Weight Characteristics

NO. OF PROPELLANT MODULES	INITIAL VEHICLE WEIGHT LESS PAYLOAD LBS	INITIAL USABLE PROPELLANT WEIGHT LBS	BURNOUT VEHICLE WEIGHT LESS PAYLOAD LBS
1	85,455	46,445	39,010
2	128,030	82,480	45,550
4	213,180	154,550	58,630
6	298,330	226,620	71,710
8	383,480	298,690	84,790
10	468,630	370,760	97,870
16	724,080	586,970	137,110
22	979,530	803,180	176,350
28	1,234,980	1,019,390	215,590

Table 10. Geosynchronous Shuttle Mission Velocity Requirements (MDAC Mission Profile)

Maneuver	Ideal Velocity (fps)
Leave 260 nmi orbit (Inject into 260 by 19,324 nmi transfer ellipse - orbital plane change 2.4°)	7885
Midcourse correction(s)	50
Circularize transfer orbit at 19,324 and orbital plane change 29.1°	6014
Deorbit from geosynchronous orbit (Inject into 260 by 19,324 transfer ellipse - orbital plane change 29.1°)	6014
Midcourse correction(s)	50
Circularize transfer at 260 nmi - orbital plane change 2.4°	7885
Total	27,898

Table 11. Potential Unmanned Payloads

Planet	Payload Components
Gas giant planets (Jupiter, Saturn, Uranus, Neptune	Side looking radar, imaging, probes
Venus and Mercury	Imaging
Moons of Jupiter	Side looking radar, imaging, probes and sample return
Mars	Imaging and sample return
Moons of Mars (Phobos, Deimos)	Imaging and sample return
Comets and asteroids	Sample return and imaging

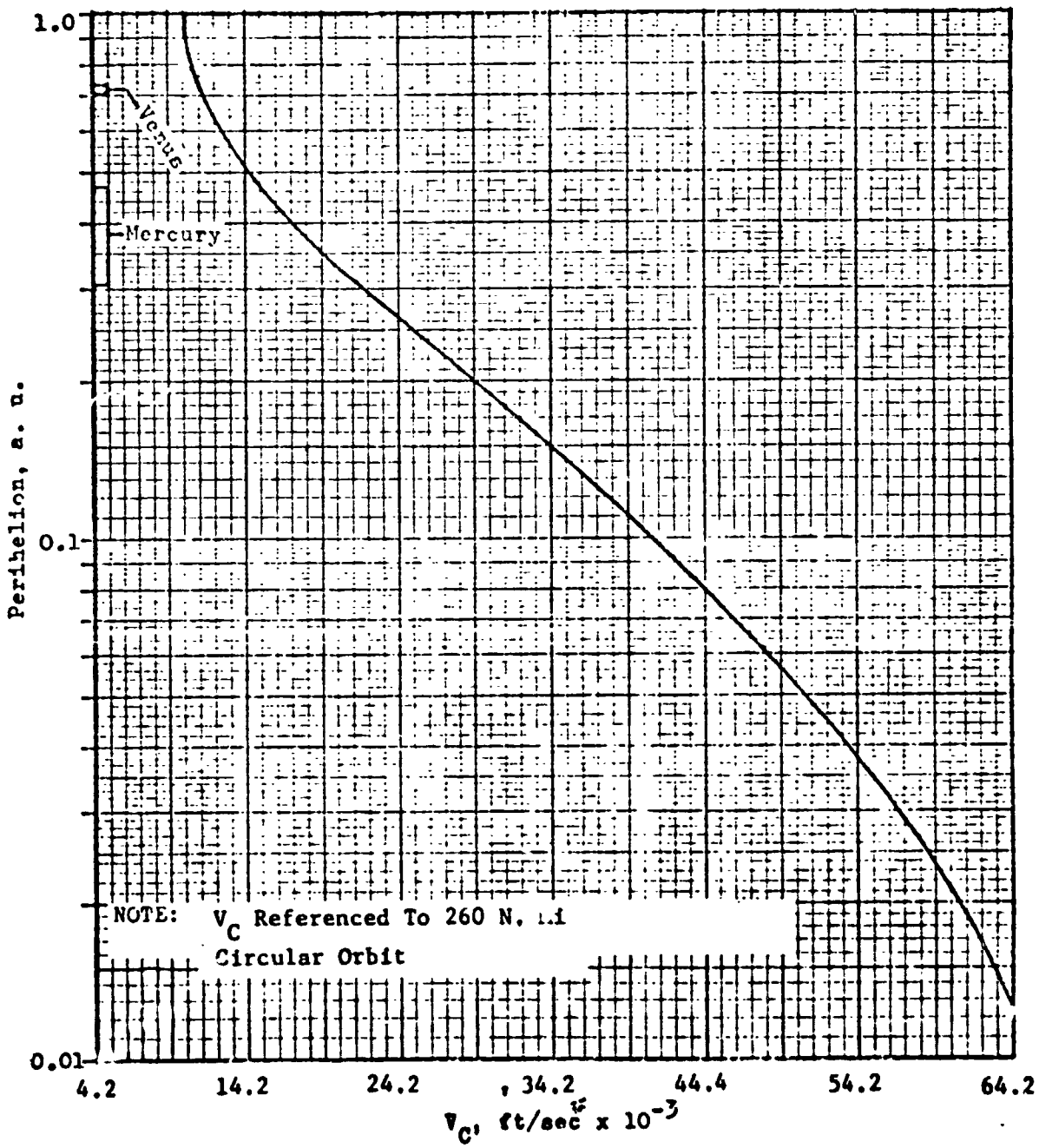


Figure 1. Velocity Required for Ballistic Solar and Inner Planetary Probes

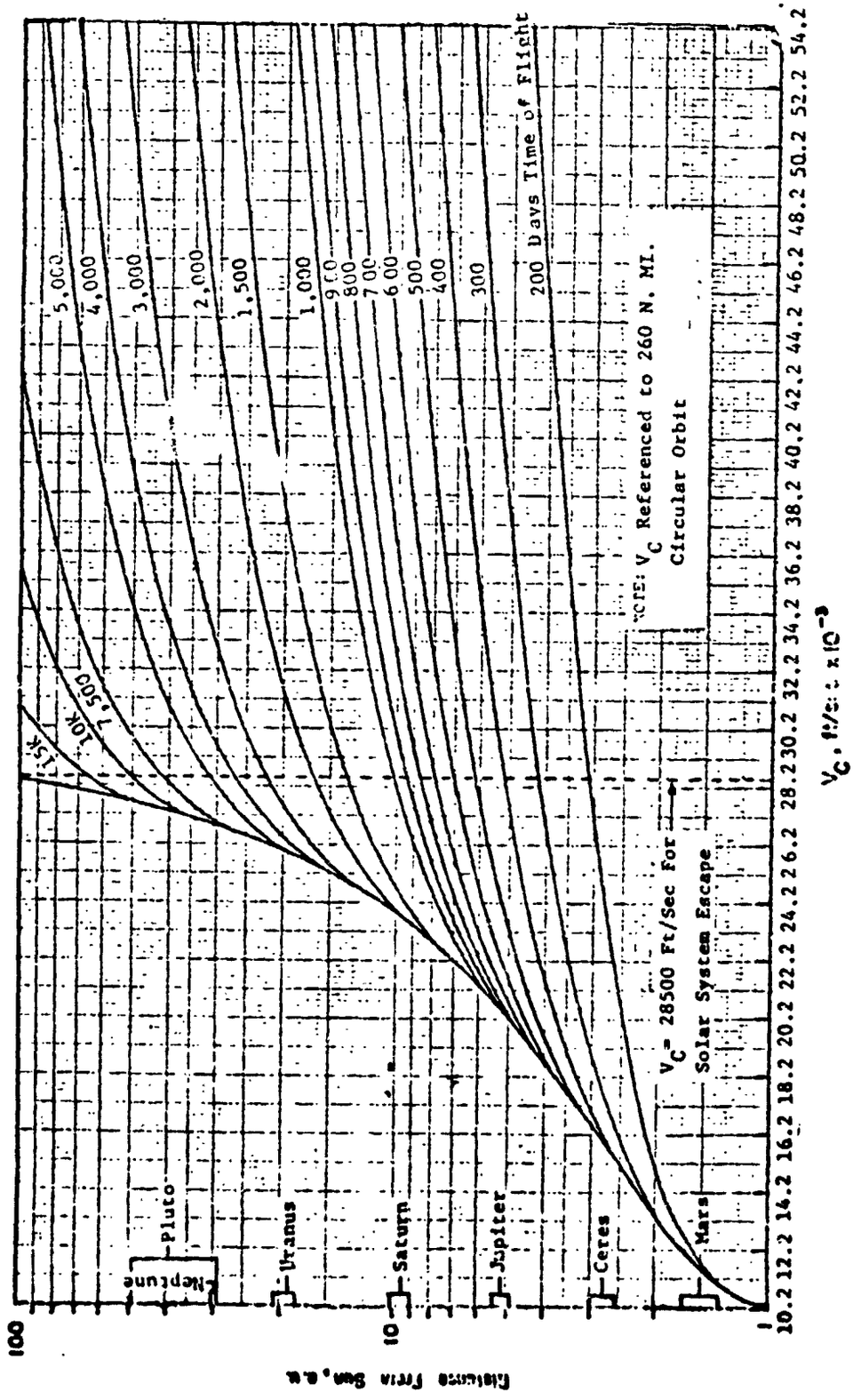


Figure 2. Velocity Required for Ballistic Probes to Outer Planetary Regions

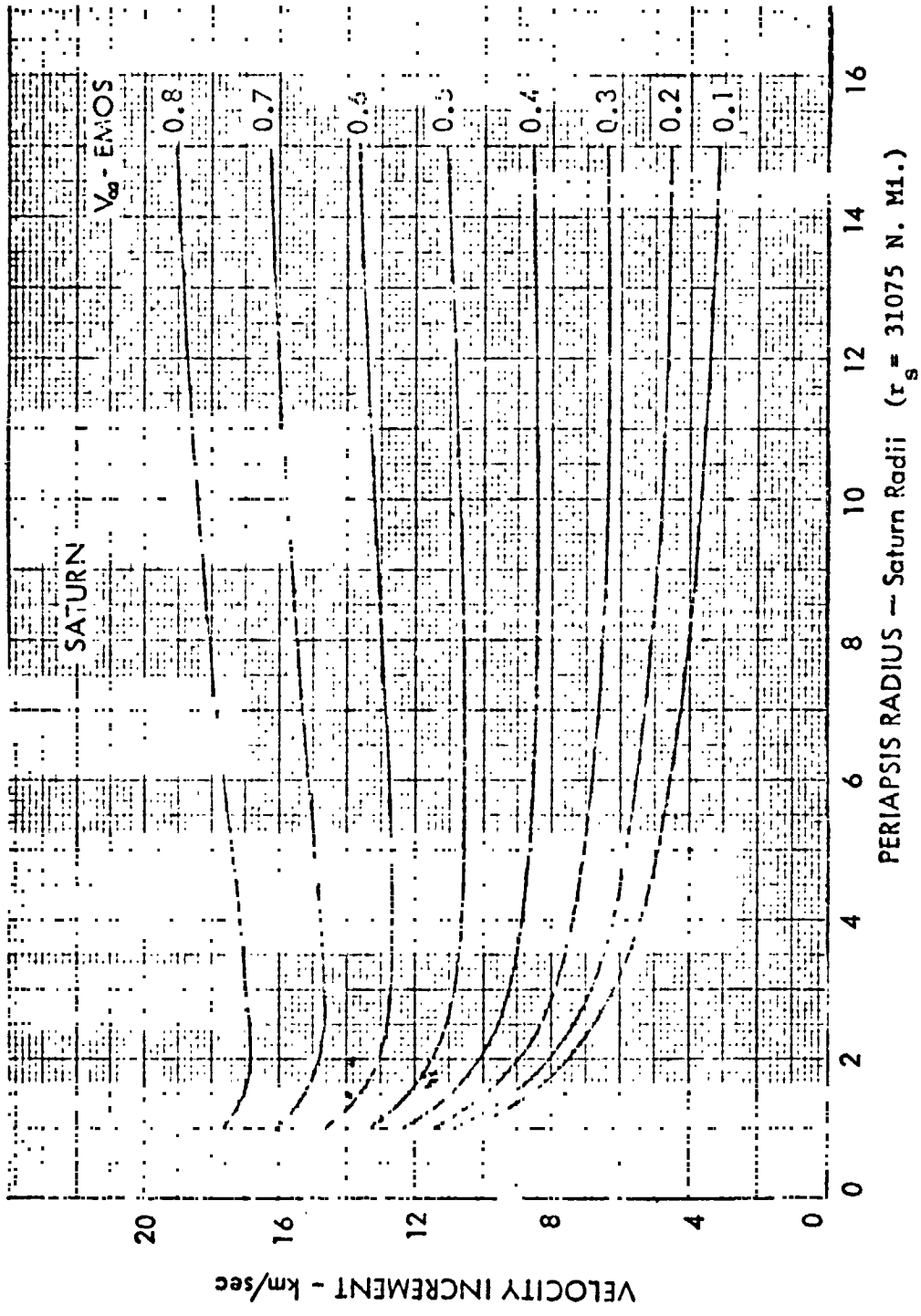


Figure 4. Velocity Increments Required to Enter Circular Capture Orbits-Saturn

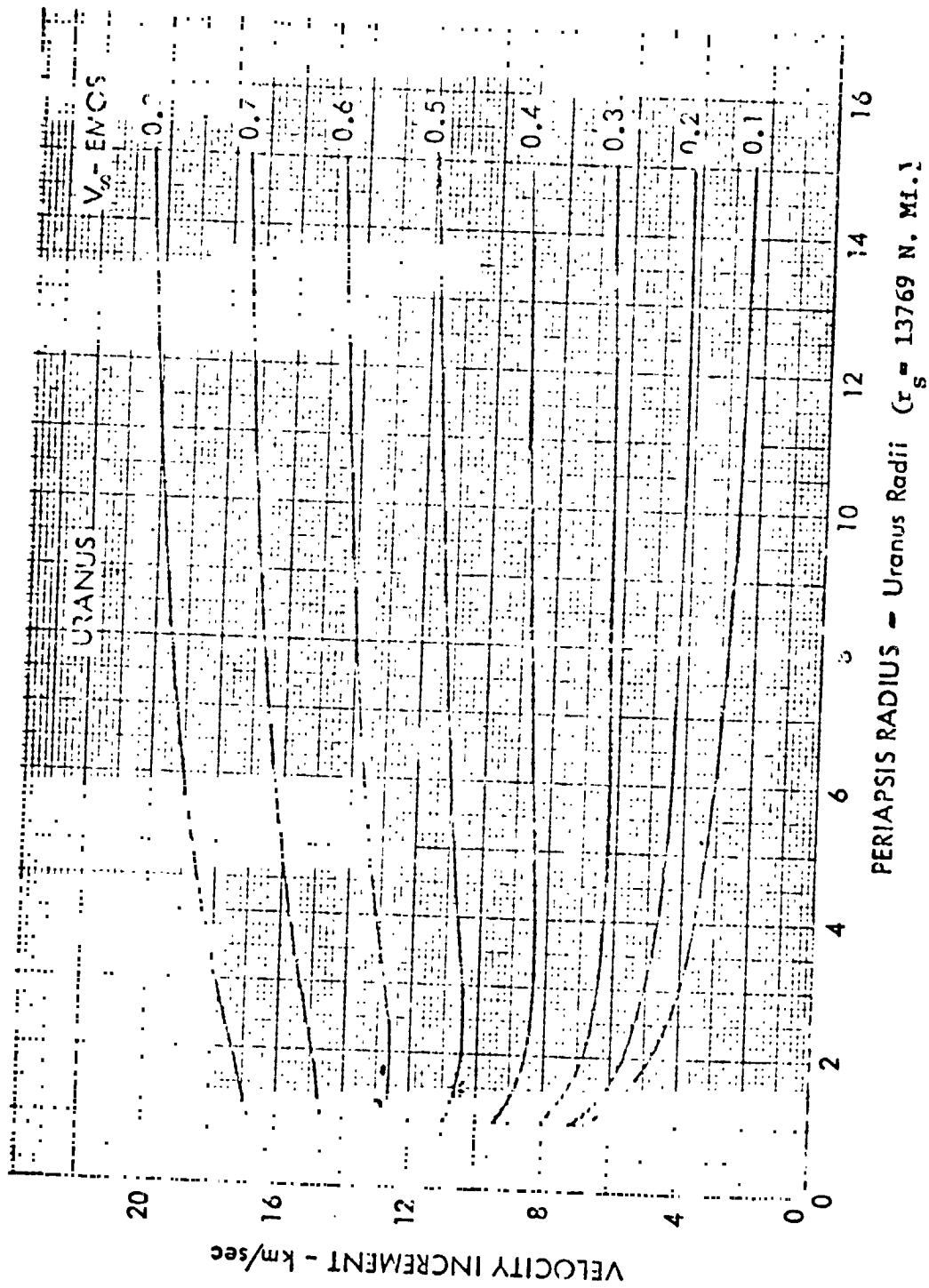


Figure 5. Velocity Increments Required to Enter Circular Capture Orbits-Uranus

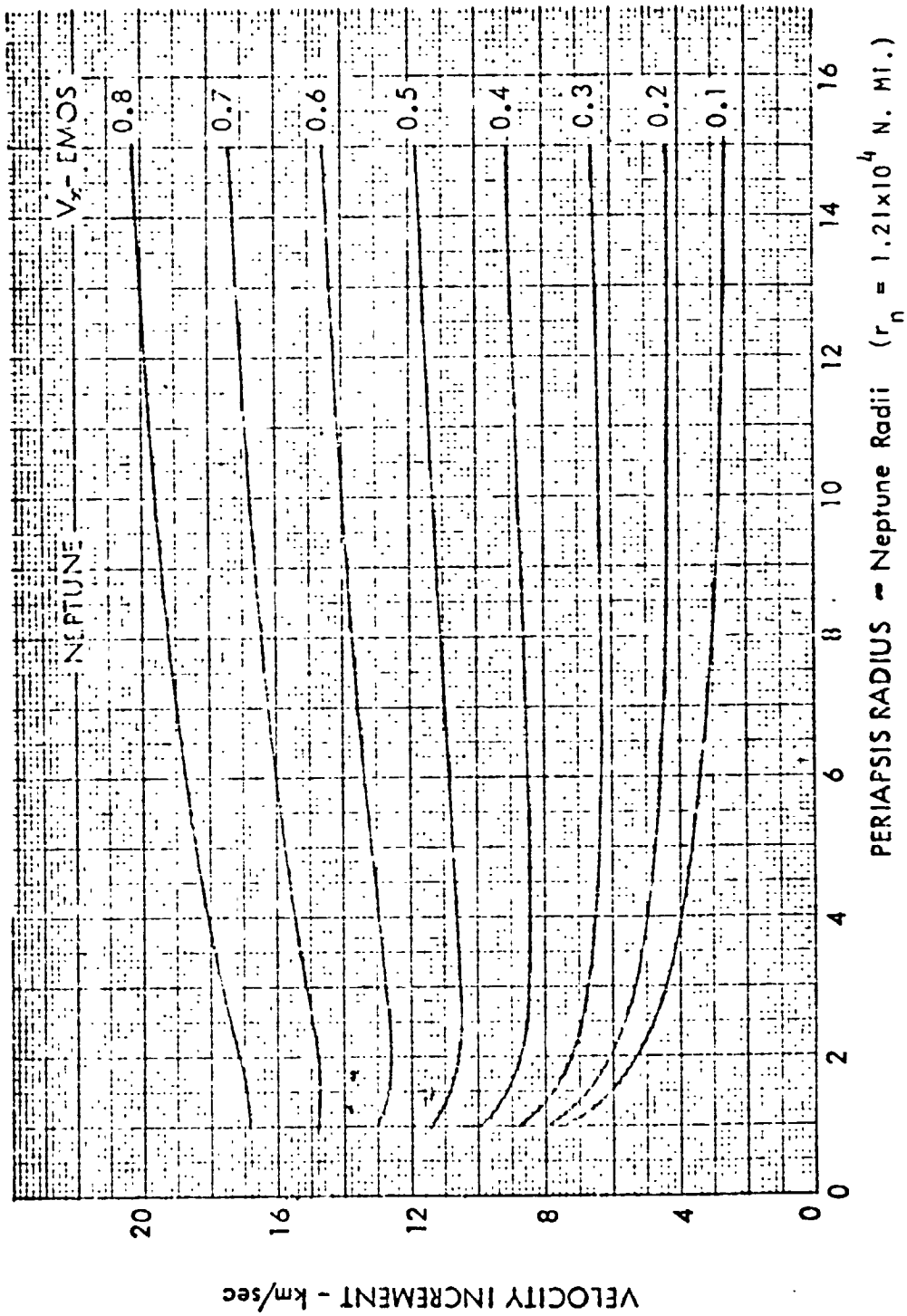


Figure 6. Velocity Increments Required to Enter Circular Capture Orbits-Neptune

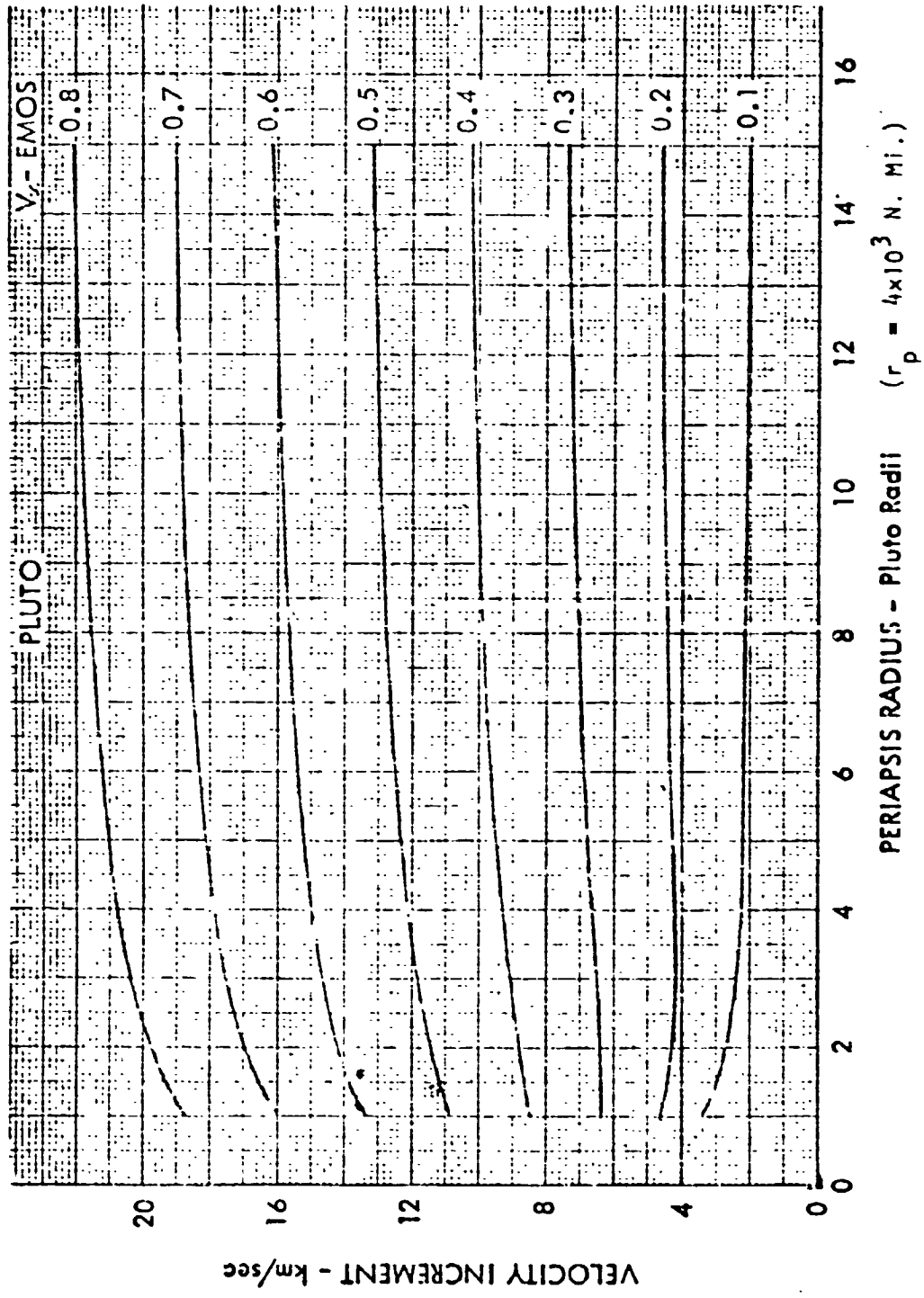


Figure 7. Velocity Increments Required to Enter Circular Capture Orbits-Pluto

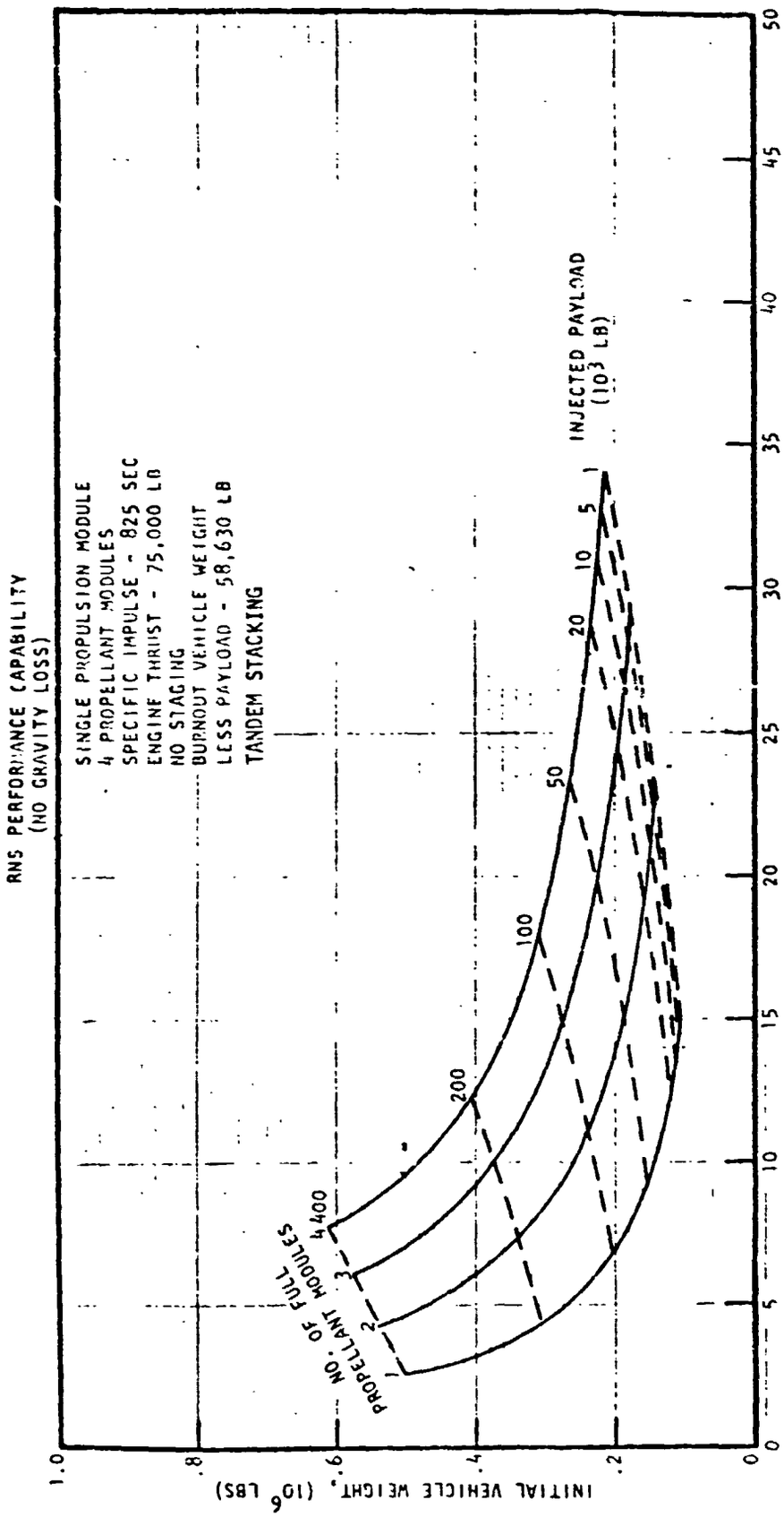


Figure 8. Total Vehicle Characteristic Velocity Increment, ΔV_c - (1000 FPS)

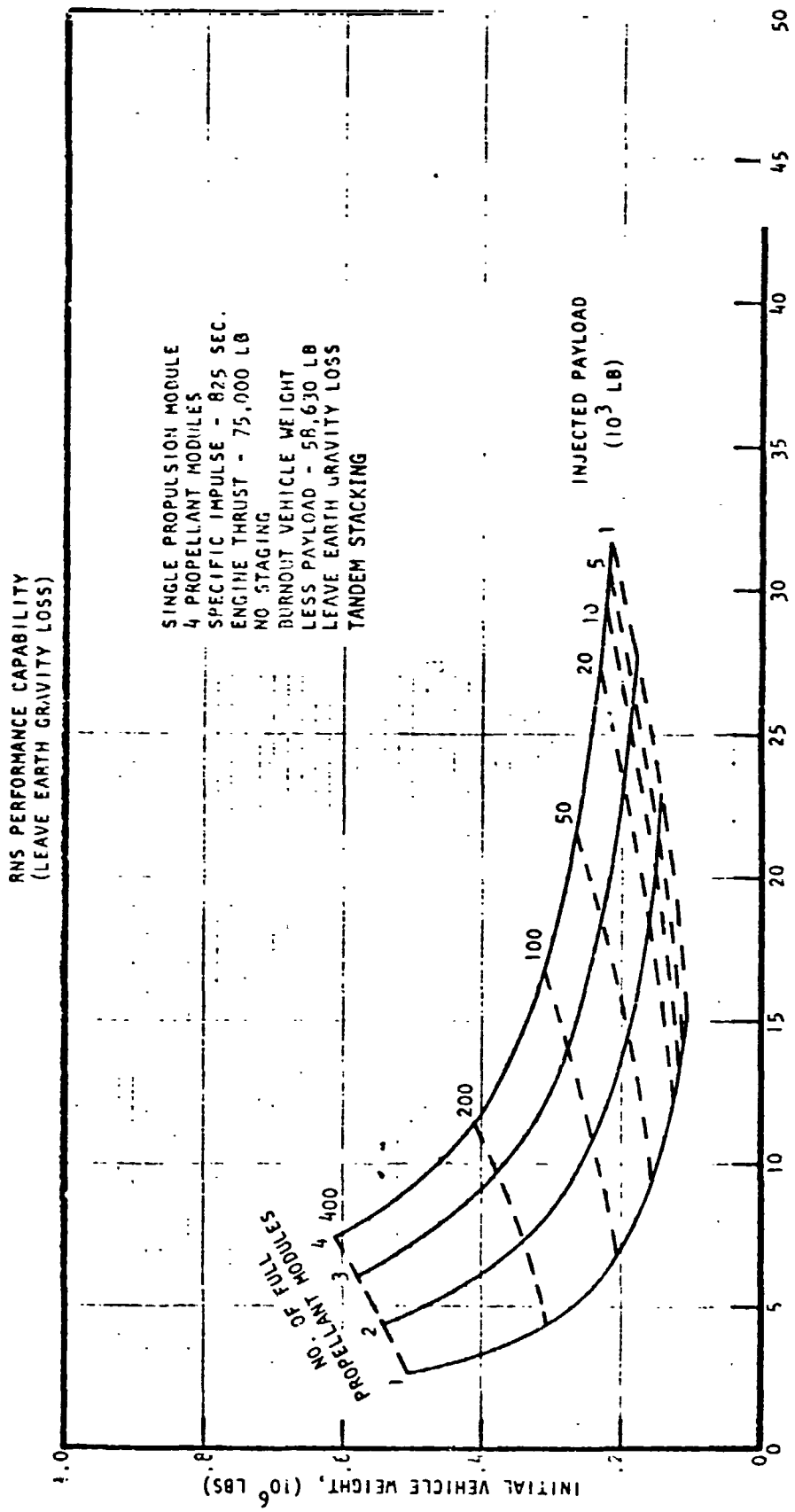


Figure 9. Impulsive Leave Earth Velocity Increment, ΔV_1 - (1000 FPS)

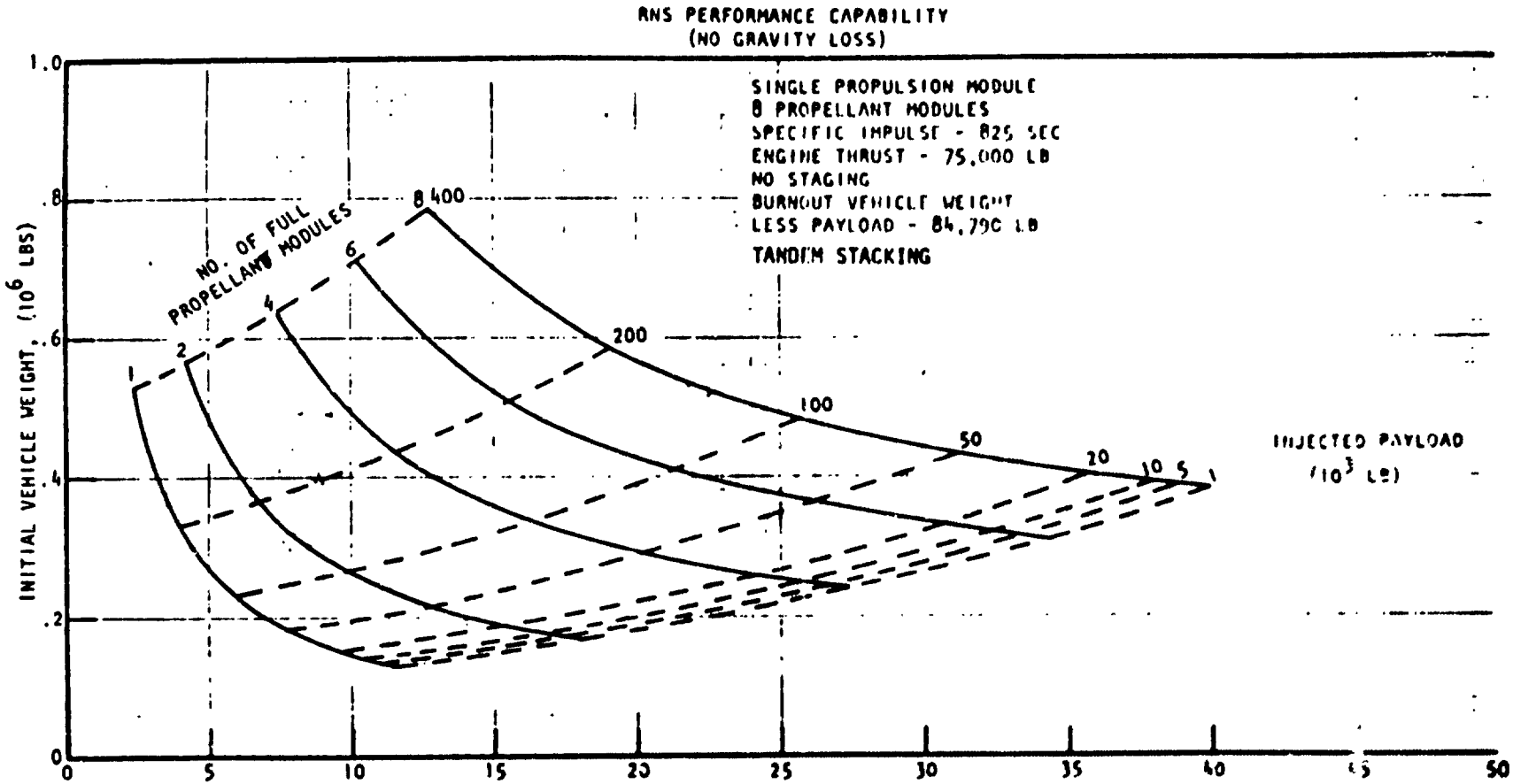


Figure 10. Total Vehicle Characteristic Velocity Increment, ΔV_c - (1000 FPS)

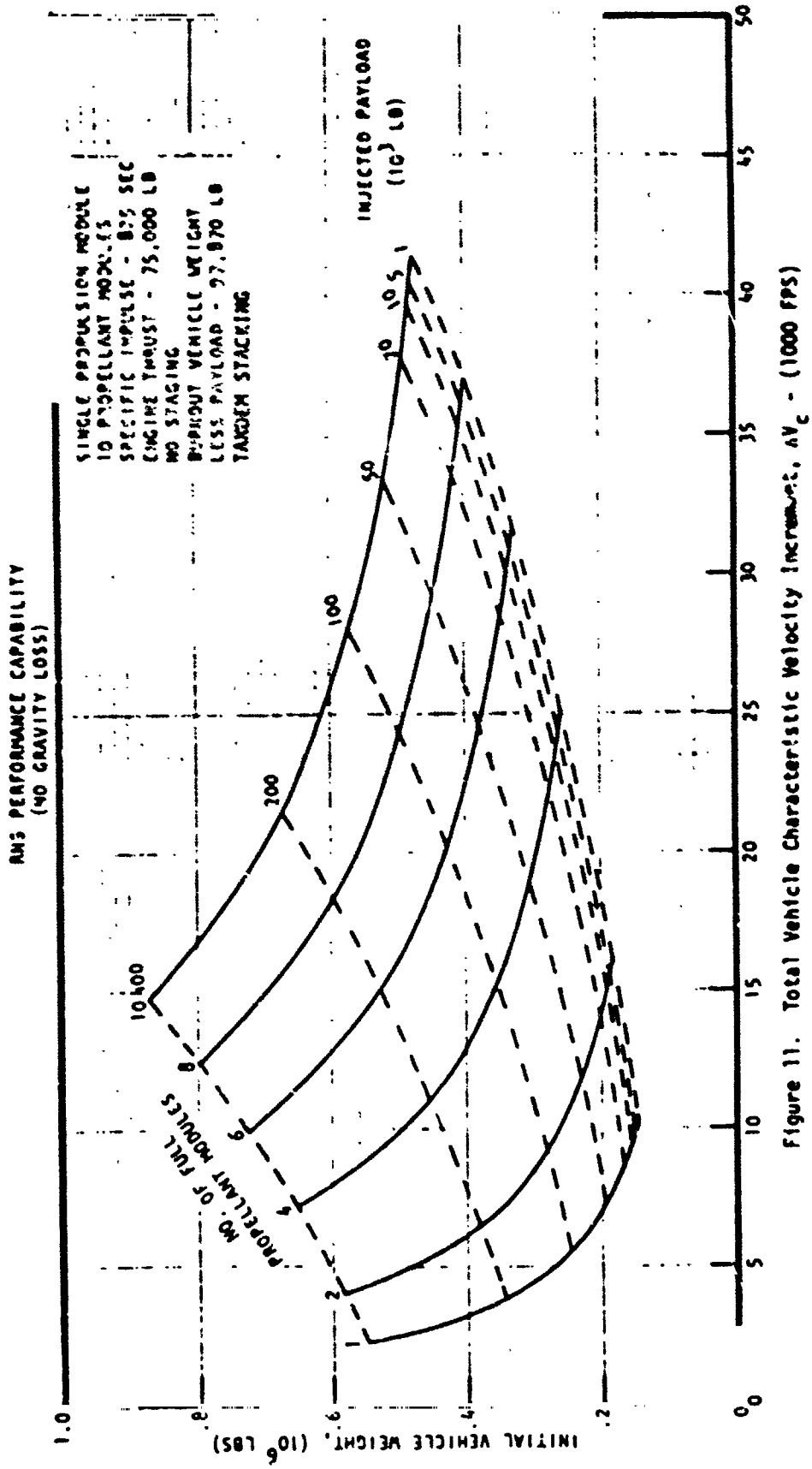


Figure 11. Total Vehicle Characteristic Velocity Increment, ΔV_c - (1000 FPS)

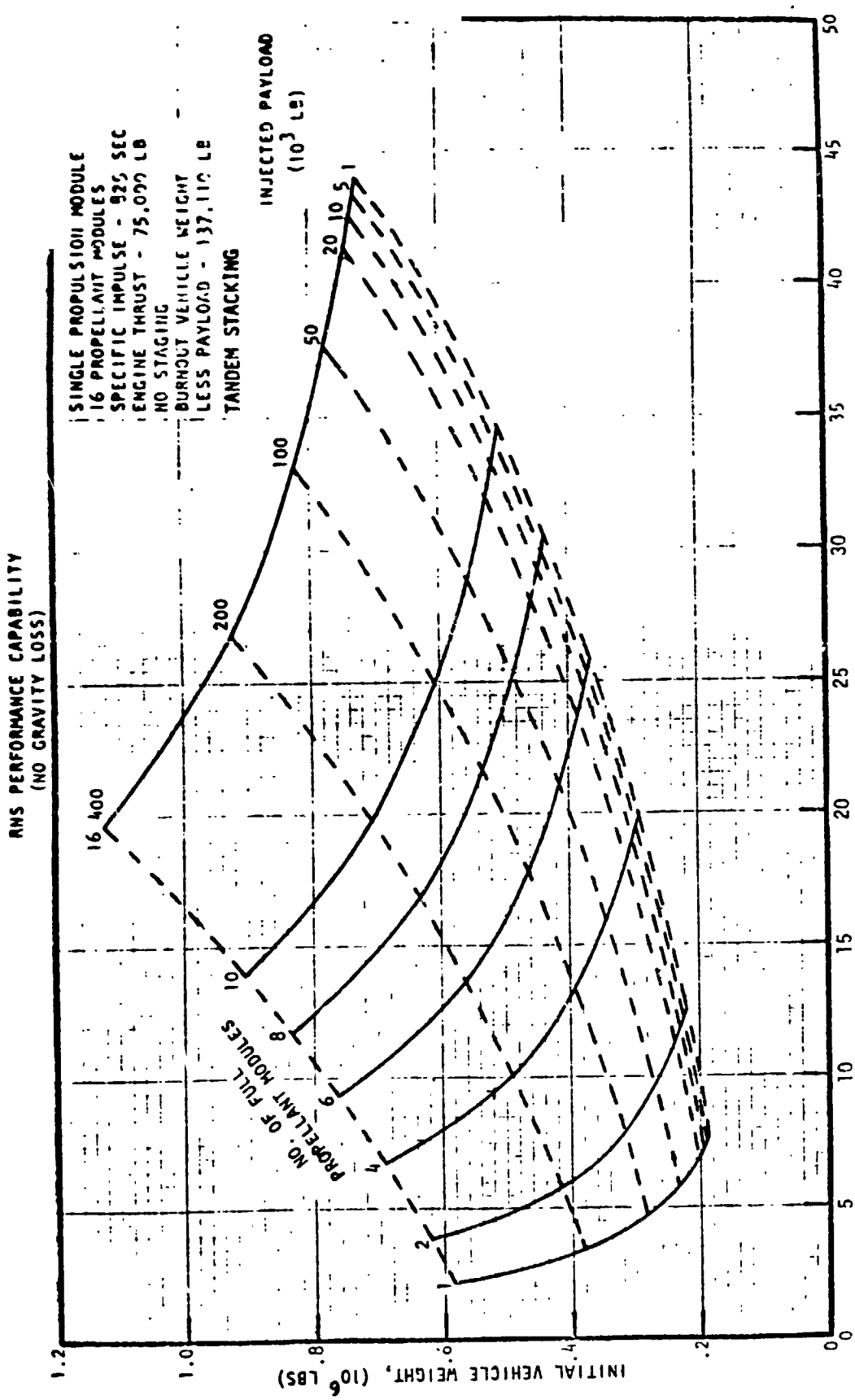


Figure 12. Total Vehicle Characteristics Velocity Increment, ΔV_c - (1000 FPS)

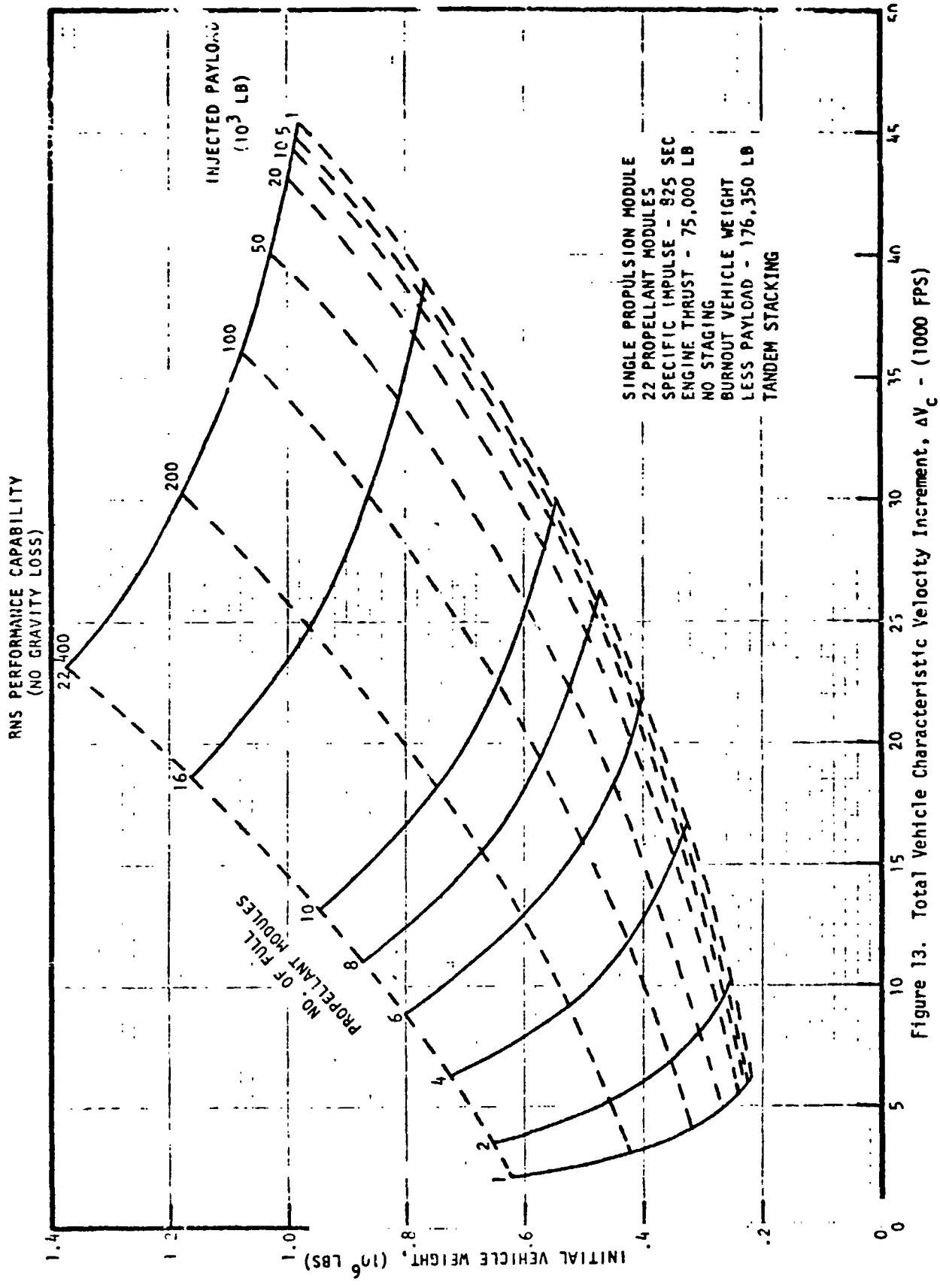


Figure 13. Total Vehicle Characteristic Velocity Increment, ΔV_c - (1000 FPS)

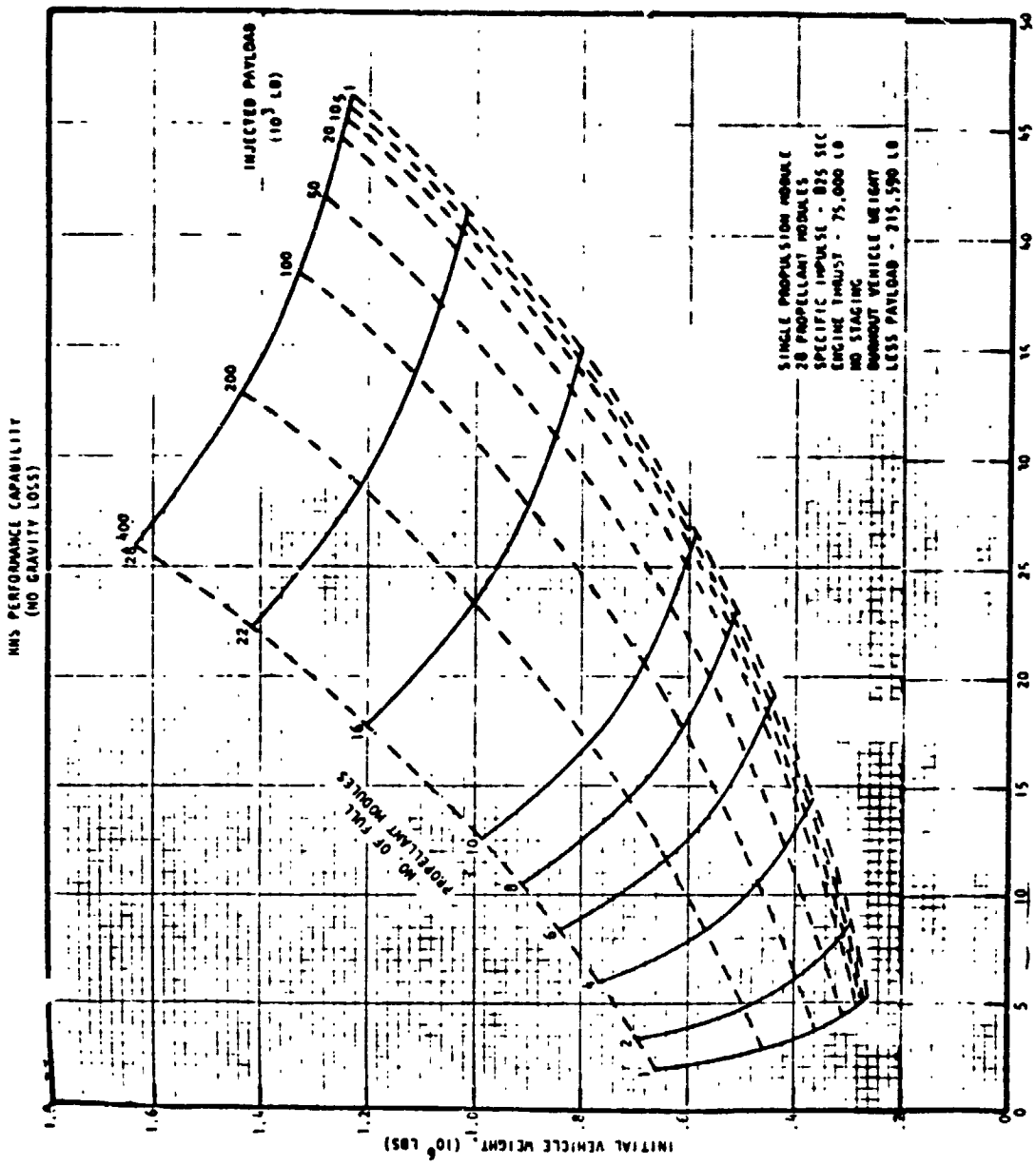


Figure 14. Total Vehicle Characteristic Velocity Increment, ΔV_c - (1000 FPS)

SINGLE BURN GRAVITY LOSS FACTOR FOR EARTH DEPARTURE
FROM A 260 NMI CIRCULAR ORBIT

SPECIFIC IMPULSE - 825 SEC

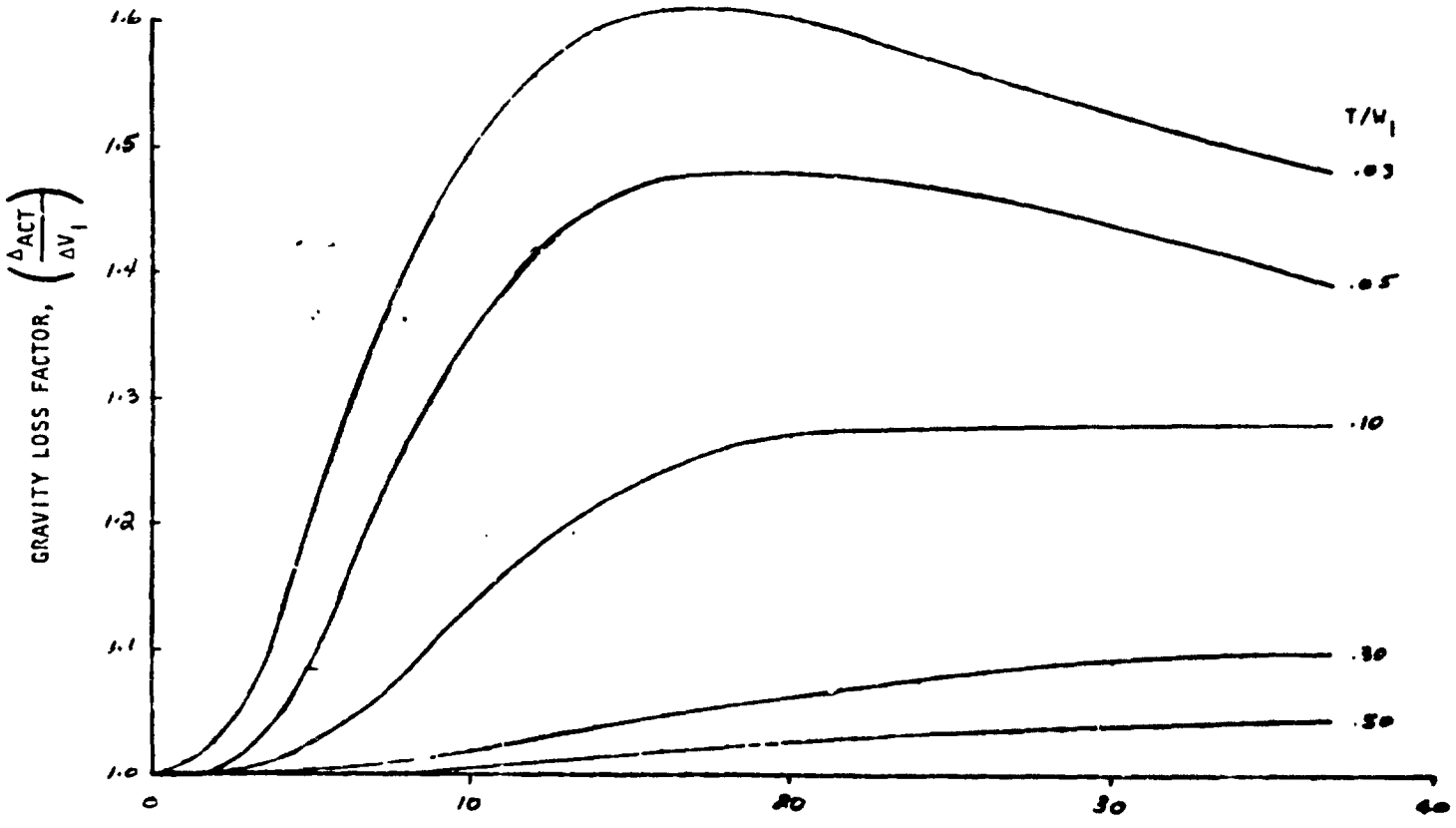


Figure 15. Delta Velocity Leave Earth, (1000 FPS)

TWO BURN GRAVITY LOSS FACTOR FOR EARTH DEPARTURE
FROM A 260 NMI CIRCULAR ORBIT

SPECIFIC IMPULSE - 825 SEC

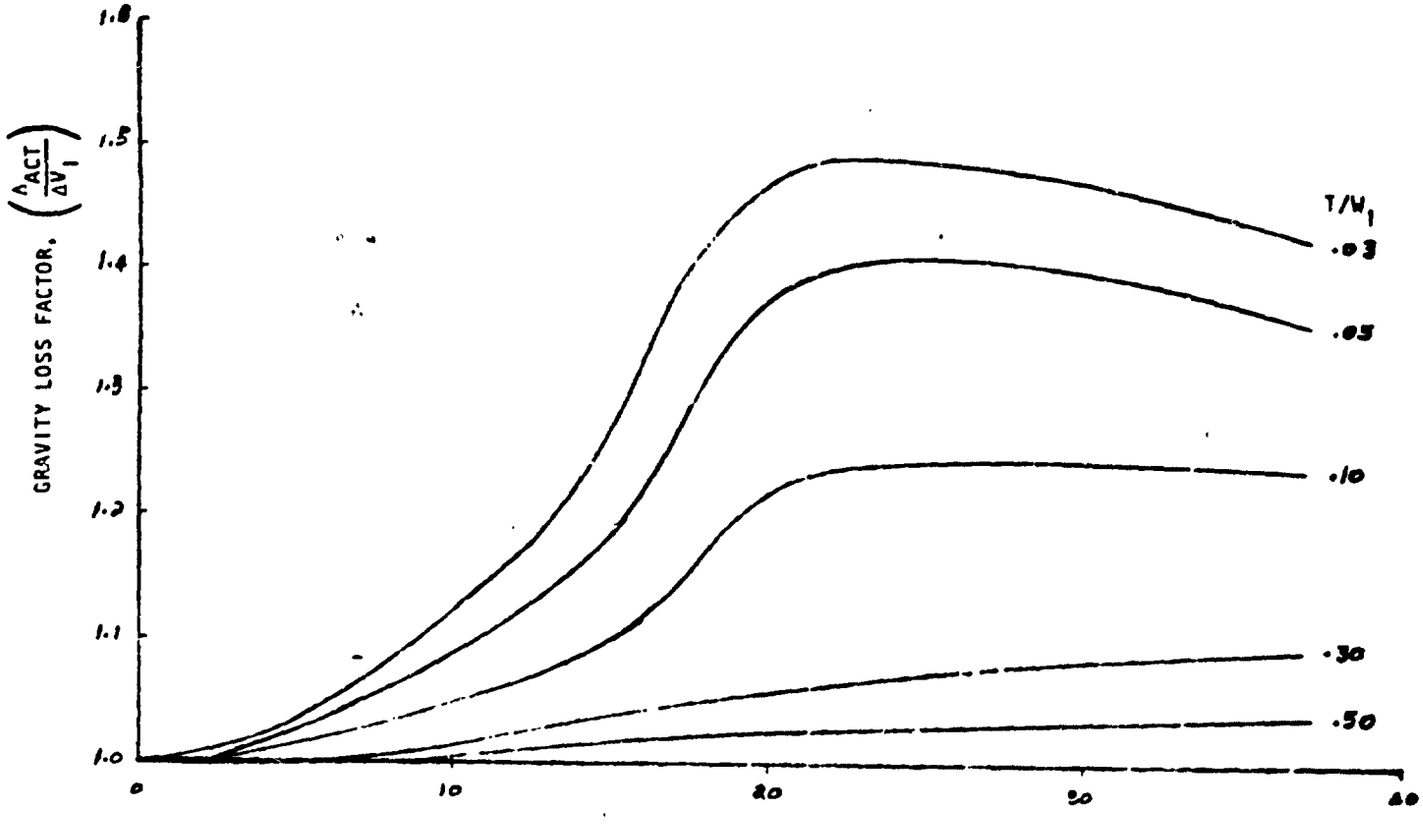


Figure 16. Delta Velocity Leave Earth, (1000 FPS)

THREE BURN GRAVITY LOSS FACTOR FOR EARTH DEPARTURE
FROM A 260 NMI CIRCULAR ORBIT

SPECIFIC IMPULSE - 825 SEC

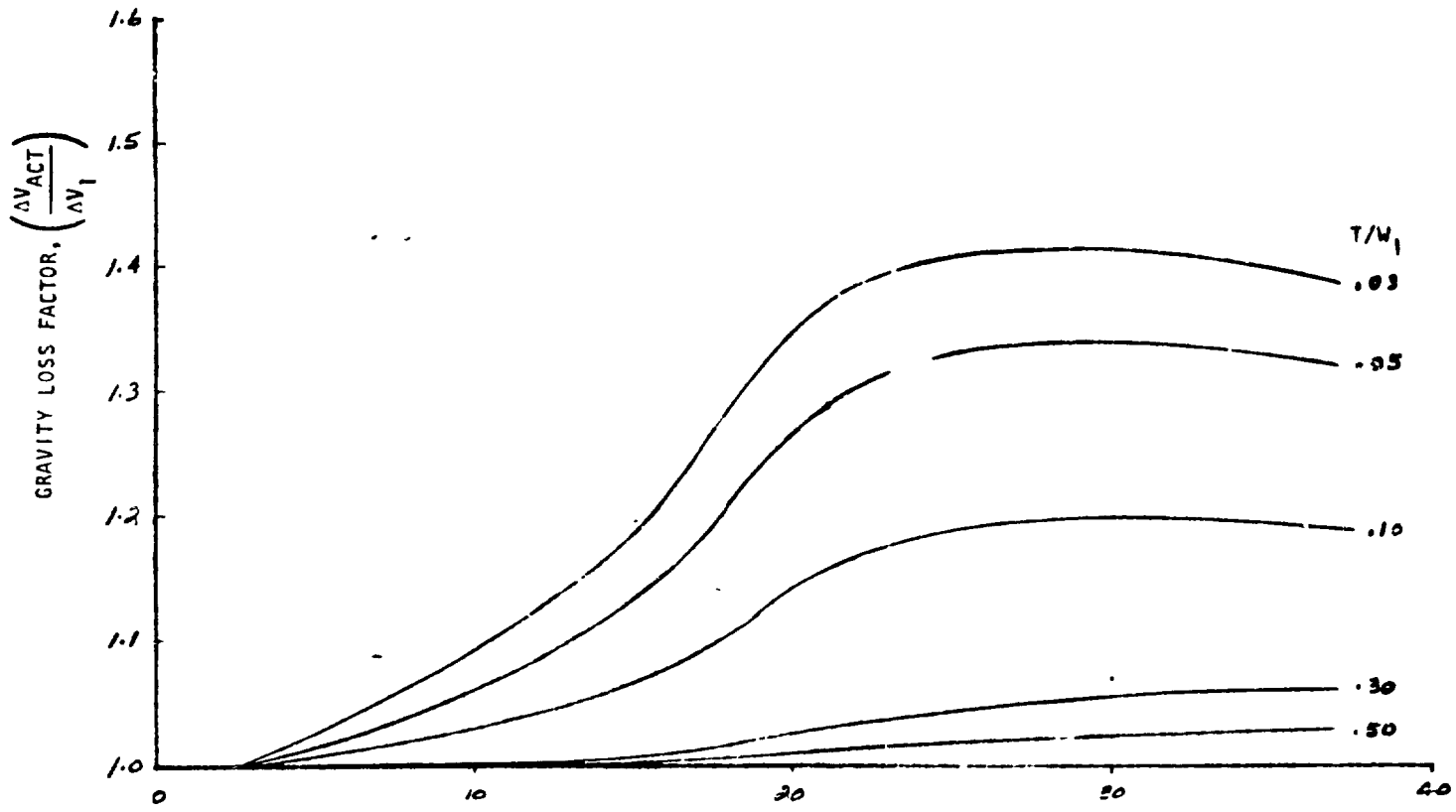


Figure 17. Delta Velocity Leave Earth, (1000 FPS)

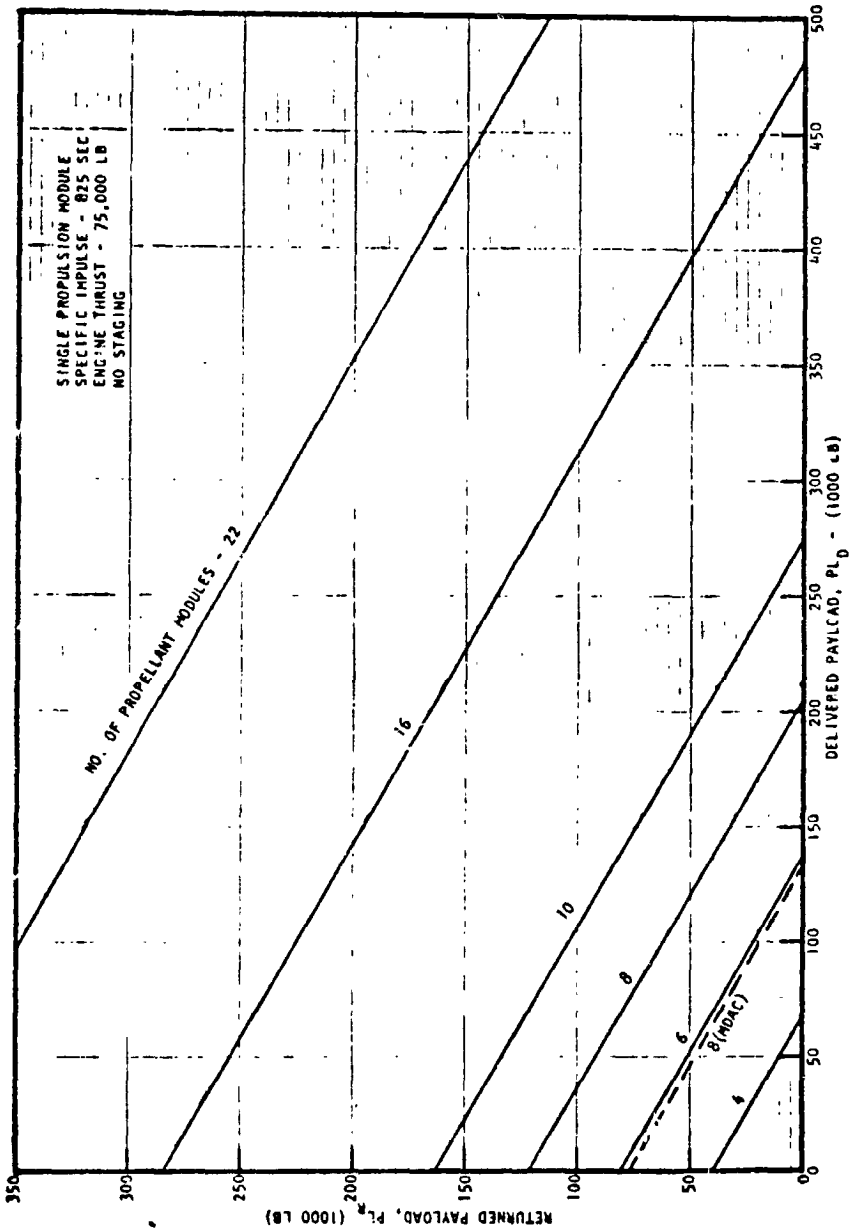


Figure 18. RNS Performance Geosynchronous Shuttle Mission (MDAC Mission Profile)