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MARS LANDING MISSION MODE COMPARISON

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ABSTRACT

Four modes of landing a vehicle on Mars are compared in this document. In addition, several variations to these four basic mission modes are suggested. The most efficient mission mode for a Mars landing depends on the type of trajectories used.

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MARS LANDING MISSION MODE COMPARISON

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SUMMARY

Four modes of landing a vehicle on Mars are studied and compared in this document to determine their relative merits in terms of the total spacecraft weight required. These four modes are combinations of the techniques of aerodynamic braking and propulsive orbit insertion with hyperbolic rendezvous or direct return. Aerodynamic braking is shown to be a very effective technique for reducing the required spacecraft weight. The propulsive-orbit-insertion and direct return-to-Earth mode is shown to be very inefficient when compared to the other possible modes, regardless of the type of propulsion system used. The aerodynamic braking with hyperbolic rendezvous mode is particularly effective because it appears to be most consistent with the payload capability of the Saturn V launch vehicle.

Several variations to the four basic mission modes are suggested, and the resulting total spacecraft weights are listed. Mars lander weights in excess of 300 000 pounds are shown to be feasible if the lander is sent to Mars as a separate unmanned vehicle.

INTRODUCTION

A mode analysis, as defined for this study, is the comparative analysis of techniques for accomplishing a given mission within the prescribed energy requirements. The analysis is aimed at evaluating the performance advantages or disadvantages of the various techniques. Some degree of preliminary trajectory design work is assumed to reduce the number of parameters to be considered. Otherwise, so many possible combinations of parameters exist that the analysis becomes confused in a maze of options that often makes a comparison of mission modes meaningless. The purpose of this analysis is to determine the performance characteristics of the various mission modes, with the same basic trajectory assumptions applied to all modes.

The opposition class of trans-Mars trajectories is assumed for this study. The trade-off of mission energy requirements and of total mission duration is not a continuous function, and a large jump in energy requirements exists between the various classes of missions. The minimum-energy Mars mission is the near-1000-day conjunction class discussed in reference 1. The next mission in the energy spectrum is the Venus-flyby/Mars-landing mission which has a duration of 600 to 680 days. The use of the Venus-flyby technique to reduce mission energy requirements is discussed in

references 2 and 3. The 400- to 500-day opposition class of missions (refs. 4 to 8) represents the practical limit for reducing the mission duration for a manned Mars landing and requires a 60- to 75-percent increase in energy relative to the minimum-energy landing mission.

Numerous modes for accomplishing a manned Mars landing mission have been investigated (refs. 9 to 15), but these investigations have been carried out with considerable variation in assumptions and mission constraints. This document presents a mode analysis with the four basic modes subjected to the same assumptions. Cryogenic, hypergolic, and nuclear propulsion systems are compared as applied to mission modes which use aerodynamic braking to assist in the Mars-orbit insertion, hyperbolic rendezvous near Mars, direct retrograde maneuvers into Mars orbit, and direct return-to-Earth (that is, without hyperbolic rendezvous). The variations to the basic modes studied include crew size and a separate unmanned Mars lander launched from Earth orbit and guided into Mars orbit by a crew already orbiting the planet.

SYMBOLS

g_0	sea level acceleration because of gravity, 32.2 ft/sec^2
I_{SP}	specific impulse, sec
k_D	ratio of Mars departure ΔV to ideal engine exit velocity, $\frac{\Delta V_D}{g_0 I_{SP}}$
k_I	ratio of Mars-orbit insertion ΔV to ideal engine exit velocity, $\frac{\Delta V_I}{g_0 I_{SP}}$
W_1	total weight staged prior to Mars-orbit insertion, lb
W_2	total weight staged while in Mars orbit, lb
W_{EM}	Earth-entry module weight, lb
W_{FT}	total spacecraft propellant, lb
W_f	vehicle mass after impulse
W_i	vehicle mass before impulse
W_L	Mars lander weight, lb

W_{MM}	mission module weight, lb
W_0	total spacecraft weight, lb
W_{PL}	payload launched from Mars orbit for return-to-Earth, lb
ΔV	impulsive velocity change, ft/sec
ΔV_D	impulsive velocity change to depart from Mars orbit for return-to-Earth, ft/sec
ΔV_I	impulsive velocity change for Mars-orbit insertion, ft/sec
ϵ	ratio of Mars-entry structure weight to total entry weight
λ	ratio of propulsion module stage weight to fuel weight
λ_1	ratio of propulsion module stage weight to fuel weight, first stage
λ_2	ratio of propulsion module stage weight to fuel weight, second stage

MISSION MODES

The four basic mission modes are described in the following paragraphs and are numbered for convenience of later notation as follows.

Mission Mode Number	Description
1	Aerodynamic braking with hyperbolic rendezvous
2	Aerodynamic braking with direct return-to-Earth
3	Propulsive Mars-orbit insertion with hyperbolic rendezvous
4	Propulsive Mars-orbit insertion with direct return-to-Earth

Schematic diagrams of the spacecraft configurations illustrate the mission modes (figs. 1 to 4). These schematic diagrams do not show actual or anticipated spacecraft shapes; they are used only to illustrate the required functional modules.

Mission Mode 1 — Aerodynamic Braking with Hyperbolic Rendezvous

The spacecraft for Mars mission mode 1, which combines aerodynamic braking at Mars with a hyperbolic rendezvous, is shown schematically in figure 1. This mode requires the following two spacecrafts.

1. An orbiting spacecraft, vehicle 1
2. A flyby spacecraft, vehicle 2

The flyby spacecraft is launched to Mars so that it arrives at Mars near the end of the planned stay time of the first vehicle. The first vehicle enters Mars orbit by using aerodynamic braking and then stages a lander for surface exploration. The lander is launched from the surface to rendezvous with the orbiting spacecraft which then departs from Mars orbit for rendezvous with the flyby spacecraft and subsequent return-to-Earth.

In the orbiting spacecraft, the mission module provides crew quarters during the transplanetary phase and is jettisoned immediately prior to atmospheric entry at Mars. The lander and the rendezvous modules provide crew support until the rendezvous with the flyby vehicle. The Mars-entry structure is jettisoned immediately after exit from the Mars atmosphere, and the propulsion module provides the velocity increment required to establish the Mars orbit and to rendezvous with the flyby spacecraft.

The flyby vehicle is composed of an Earth-entry module, a mission module, and an experiments package and/or propulsion module. The propulsion module on the flyby spacecraft would permit powered-turn flybys and would reduce the velocity increment required of the orbiting vehicle. A powered turn would also eliminate passage through the asteroid belt.

Mission Mode 2 — Aerodynamic Braking with Direct Return-to-Earth

The spacecraft for Mars mission mode 2, which uses aerodynamic braking at Mars and a subsequent direct return-to-Earth, is shown schematically in figure 2. This mission mode requires only a single spacecraft assembled in and launched from Earth orbit. The sequence of events is similar to that of the orbiting vehicle in mission mode 1. However, the entire spacecraft is slowed by atmospheric braking to near orbital velocity at Mars, and the return-to-Earth is direct rather than the result of a rendezvous with a flyby spacecraft. The propulsion module provides the velocity increment to attain Mars orbit and to return to Earth. The lander is staged from orbit as described for mission mode 1. The mission module provides crew support during both the transplanetary and trans-Earth phases, and the Earth-entry module provides for the return of the crew to the surface of the Earth.

Mission Mode 3 — Propulsive Mars-Orbit Insertion with Hyperbolic Rendezvous

The spacecraft for Mars mission mode 3, which combines a propulsive insertion into Mars orbit with a hyperbolic rendezvous, is shown schematically in figure 3. The sequence of events for mission mode 3 is the same as that described for mission mode 1 except that a propulsive maneuver, instead of aerodynamic braking, is used for Mars orbit insertion. Vehicle 1 requires two propulsion modules. Propulsion module 1 provides the total velocity increment to orbit Mars, and propulsion module 2 provides the velocity increment to rendezvous with the flyby spacecraft. The lander can be staged either before or after orbital insertion. Fuel is saved if the lander is staged prior to orbit insertion, but serious operational problems are associated with the landing-site selection. The flyby spacecraft is the same as the flyby spacecraft described for mission mode 1.

Mission Mode 4 — Propulsive Mars-Orbit Insertion with Direct Return-to-Earth

The spacecraft for Mars mission mode 4, which uses a propulsive insertion into Mars orbit and a subsequent direct return-to-Earth, is shown schematically in figure 4. The sequence of events for mission mode 4 is the same as that described for mission mode 2 except that propulsive maneuvers, instead of aerodynamic braking, are used for Mars-orbit insertion. The spacecraft requires two propulsion modules; one module is used for insertion of the spacecraft into Mars orbit, and the other module is used to launch the spacecraft to Earth. The mission module and the Earth-entry module are, therefore, carried into and out of Mars orbit. The lander can be staged either before or after orbital insertion, as described for mission mode 3, with the same implications.

ANALYSIS

The total spacecraft weight is chosen as the parameter for performance comparison in the analysis, because the spacecraft weight determines the number of Earth launches required for orbit assembly. Payload size, complexity, and the number of Earth launches can be directly related to program cost in terms of new development, hardware, and operations.

All fuel weight computations for this study are based on impulsive velocity changes, which are computed by using the "ideal rocket equation."

$$\Delta V = g_0 I_{SP} \log_e \frac{W_i}{W_f} \quad (1)$$

where ΔV = impulsive velocity change

$$g_0 = 32.2 \text{ ft/sec}^2$$

I_{SP} = specific impulse

W_i = initial mass

W_f = final mass

The equation for total spacecraft weight for mission mode 1 is

$$W_0 = \frac{(1 + \lambda_1)W_{FT} + W_L + W_{PL}}{(1 - \epsilon)} + W_{MM} \quad (2)$$

where W_{FT} is the total fuel required. The equation for W_{FT} is

$$W_{FT} = \frac{W_{PL} (e^{k_D} - 1) + (W_L + W_{PL}) (1 - e^{-k_I})}{(1 + \lambda_1) (e^{-k_I}) - \lambda_1 e^{k_D}} \quad (3)$$

where $k_D = \frac{\Delta V_D}{g_0 I_{SP}}$ and $k_I = \frac{\Delta V_I}{g_0 I_{SP}}$. The total spacecraft weight and fuel requirements for mission mode 2 can be determined from equations (2) and (3) by including the mission module weight W_{MM} in the return-to-Earth payload W_{PL} .

The advantage of the hyperbolic rendezvous technique is apparent in equation (3) because of the terms involving W_{PL} . The total fuel required increases rapidly as W_{PL} increases. The advantages of aerodynamic braking are obvious in the terms involving ΔV_I . Without aerodynamic braking, the required velocity increments approach the theoretical limit for a single stage vehicle; and, thus, a two-stage propulsion module is much more efficient.

The equation for total spacecraft weight for mission modes 3 and 4 is

$$W_o = W_1 + \frac{W_2}{e^{-k_I} (1 + \lambda_1) - \lambda_1} + \frac{W_{PL}}{\left[e^{-k_I} (1 + \lambda_1) - \lambda_1 \right] \left[e^{-k_D} (1 + \lambda_2) - \lambda_2 \right]} \quad (4)$$

The advantage of the hyperbolic rendezvous technique is obvious in equation (4) since it results in a reduced W_{PL} term. However, another option to be considered in equation (4) is that the Mars lander module can be staged prior to orbit insertion; thus, the weight of the lander is included in W_1 . If the lander is not staged, the second term in equation (4) becomes very large because e^{-k_I} is very small. Equation (4) also indicates that payload weight carried to hyperbolic injection is very expensive in terms of W_o .

RESULTS

The relative merits of the four modes of operation previously described can only be determined if the modes are compared by using the same basic criteria for computing spacecraft weight. Table I lists the values of the parameters selected in optimistic, expected, and pessimistic categories. The nominal specific impulse of the propulsion module is listed as 380 seconds. A hypergolic system was selected for the propulsion module because it appears that such a system will have to be developed for the Mars lander. The system will thus be available to reduce development time and cost. A cryogenic chemical system ($I_{SP} = 420$ seconds) and nuclear system ($I_{SP} = 820$ seconds) are also possibilities for the propulsion module. A brief comparison of these systems is also included.

The spacecraft module weights used in this study are based on the results of a series of spacecraft design studies conducted at the Manned Spacecraft Center during 1966. The characteristics of the nuclear propulsion system are based on information in reference 3. References 4, 5, and 6 are useful in obtaining a spacecraft velocity budget.

Parametric Mission Mode Analysis

The following paragraphs present a parametric analysis of the four basic mission modes based on the use of a hypergolic propulsion module.

Mission mode 1. - Mission mode 1 uses aerodynamic braking to attain Mars orbit; and, therefore, the velocity increment required for orbit insertion can be made relatively small. Figure 5 shows the effect of the orbit insertion ΔV on total spacecraft weight when the insertion maneuver is preceded by atmospheric braking. The required

ΔV decreases as the apoapsis and periapsis altitude (resulting from the braking maneuvers) increase. A ΔV of 150 ft/sec is sufficient to attain a circular 100-n. mi. orbit if, after braking, the periapsis altitude is 0 n. mi. and the corresponding apoapsis altitude is 100 n. mi. The total spacecraft weight increases at approximately 24 pounds per ft/sec for this ΔV . A nominal ΔV of 150 ft/sec is assumed for the remainder of this study.

The payload required for rendezvous with the flyby spacecraft is a sensitive parameter because of the large ΔV required for the launch from orbit. A small rendezvous module (15 000 pounds) is used to reduce the total spacecraft weight. This payload was assumed to be a four-man Earth-entry module, which would be required earlier in the mission for abort while in Earth vicinity. Figure 6 shows the effect of this payload on the total spacecraft weight. The total spacecraft weight increases approximately 5-1/3 pounds for each pound of rendezvous payload.

The effect of small variations in specific impulse is shown in figure 7. The variation of total spacecraft weight with specific impulse is negligible for hyperbolic launch velocities of 10 000 ft/sec or less and is very large at 20 000 ft/sec (approximately 1800 lb/sec).

The increase in total spacecraft weight caused by the addition of an entry structure for aerodynamic braking to Mars orbit is difficult to assess without getting involved in a spacecraft configuration study. Therefore, a parameter ϵ , defined as the ratio of entry structure weight to total entry weight, is assumed and is shown in figure 8 as a function of total spacecraft weight and ΔV for hyperbolic injection. It is assumed that the mission module is staged prior to atmospheric entry. If the mission module is not staged prior to entry, the total spacecraft weight is increased by a factor of approximately ϵW_{MM} .

The ratio of the propulsion module tank and engine weight to fuel weight λ is plotted in figure 9 as a function of total spacecraft weight and hyperbolic injection velocity. This is a sensitive parameter when ΔV_D approaches the theoretical limit for a one-stage vehicle. For a moderate ΔV (for example, $\Delta V_D = 15\ 000$ ft/sec), the total spacecraft weight variation with a 10-percent change in λ is 14 000 pounds.

The total spacecraft weight variation for mission mode 1 is summarized in figure 10 for the optimistic, expected, and pessimistic values of the design parameter (table I). The expected total spacecraft weight for a ΔV_D of 15 000 ft/sec is 250 000 pounds and is well within the payload capability of the present Saturn V booster for attaining a low Earth orbit. By using pessimistic values and a ΔV_D equal to 15 000 ft/sec, the total spacecraft weight is only 340 000 pounds. If we assume that each spacecraft has a 300 000-pound total spacecraft weight and multiple Saturn IVB stages for transplanetary injection, then mission mode 1 can be accomplished with eight Saturn V launches.

Mission mode 2. - Mission mode 2 uses aerodynamic braking to attain Mars orbit as described for mission mode 1, but the return-to-Earth is accomplished without rendezvous with a flyby spacecraft. The return-to-Earth payload for mission mode 2 must,

therefore, include a mission module as well as an Earth-entry module. Figure 11 is a plot of the total spacecraft weight as a function of the trans-Earth payload and hyperbolic launch velocity. A 150-ft/sec ΔV for orbit insertion is assumed here as was assumed for mission mode 1. The expected value for the return-to-Earth payload is 73 000 pounds for an eight-man crew and 55 000 pounds for a four-man crew. The total spacecraft weight for $\Delta V_D = 15\ 000$ ft/sec is 450 000 pounds and 550 000 pounds for four-man and eight-man crews, respectively. The total spacecraft weight increases by approximately 5-1/2 pounds for each pound of trans-Earth payload.

The effect of variations in the specific impulse of the propulsion module is shown in figure 12. The total spacecraft weight decreases by 2800 pounds for each second of increase in I_{SP} at $\Delta V_D = 15\ 000$ ft/sec. The reason for the increased sensitivity, as compared to mission mode 1, is that a much greater amount of fuel is being consumed.

The total spacecraft weight is also more sensitive to the ratio of tank and engine weight to fuel weight as is shown in figure 13. A 10-percent increase in λ will result in an increase of 15 000 pounds in total spacecraft weight for $\Delta V_D = 15\ 000$ ft/sec.

The ratio of entry structure weight to total entry weight is plotted in figure 14 as a function of total spacecraft weight. A 10-percent increase in ϵ will result in an increase in total spacecraft weight of 800 pounds (for $\Delta V_D = 15\ 000$ ft/sec) and a nominal ϵ of 0.15.

The total spacecraft weight variation for mission mode 2 is summarized in figure 15. The expected total spacecraft weight for $\Delta V_D = 15\ 000$ ft/sec is 550 000 pounds. When compared to mission mode 1, mission mode 2 shows a significant increase in the sensitivity of total spacecraft weight to variations in the design parameters. The optimistic to pessimistic variation of spacecraft weight at $\Delta V_D = 15\ 000$ ft/sec is 130 000 pounds for mission mode 1 and 270 000 pounds for mission mode 2.

Mission mode 3. - Mission mode 3 does not use aerodynamic braking to attain Mars orbit as did mission modes 1 and 2. However, hyperbolic rendezvous is used to reduce the payload inserted into Mars orbit. Two propulsion modules are used instead of one as in mission modes 1 and 2, because of the increased ΔV requirements and the obvious advantages of staging.

Since Mars-orbit insertion is accomplished propulsively, it is important to reduce the spacecraft weight as much as possible at this point. One possibility is to stage the lander for a hyperbolic entry prior to orbit insertion. The effect of staging the lander either before or after orbit insertion is shown in figure 16 (assuming a ΔV of 15 000 ft/sec for orbit insertion). The total spacecraft weight is increased by 420 000 pounds if the lander is staged after orbit insertion (that is, the total spacecraft weight is doubled if the lander is not staged before orbit insertion). This weight savings cannot be completely realized since the lander weight must increase to accommodate the higher Mars-entry velocity when the lander is staged prior to Mars-orbit insertion.

The total spacecraft weight variation for mission mode 3 is summarized in figures 17 to 20. This summary assumes that the lander is staged prior to orbit insertion. The expected value of total spacecraft weight for $\Delta V_D = 15\ 000$ ft/sec and $\Delta V_I = 15\ 000$ ft/sec is 480 000 pounds, which is slightly less than for mission mode 2. However, the difference between the optimistic and pessimistic values for spacecraft weight is 400 000 pounds, which indicates that mission mode 3 is much more sensitive to the design parameters than either mission mode 1 or 2.

Mission mode 4. - Mission mode 4 uses a propulsive orbital insertion and a direct return-to-Earth. It is even more important for mission mode 4 than for mission mode 3 that the lander be staged prior to Mars-orbit insertion, as shown in figure 21. The total spacecraft weight is increased by 500 000 pounds if the lander is not staged before orbit insertion. The total spacecraft weight is well over 1 500 000 pounds for a ΔV_D of 15 000 ft/sec even if the lander is staged before orbit insertion.

The total spacecraft weight variation is summarized in figure 22 (assuming a ΔV_I of 12 000 ft/sec). Even with this very low ΔV for orbit insertion, the expected spacecraft weight for $\Delta V_D = 15\ 000$ ft/sec is 1 250 000 pounds with a variation of 750 000 pounds between optimistic and pessimistic values of the design parameters. A much higher specific impulse is required for mission mode 4 to be realistic.

Comparison of the Basic Mission Modes

The comparison of the basic mission modes, which use the hypergolic, cryogenic, and nuclear propulsion systems, is shown in figure 23. The lander is staged after achieving Mars orbit in each mission mode.

The total spacecraft weight in figure 23 is the sum of all vehicle weights for those modes requiring more than one spacecraft even though they are launched to Mars as separate vehicles. Thus, the spacecraft weight scale is the total weight required to be launched to Mars.

Table II lists the expected characteristics of the propulsion systems, and table III lists the assumed velocity budget. The velocity budget is somewhat arbitrary; and, although a different budget would alter the spacecraft weights, it would not greatly affect the relative comparisons. The engine weight is included in λ for the hypergolic and cryogenic systems, but this weight is listed separately for the nuclear engine. A 10 000-pound reactor shield is included in the nuclear system. The specific impulse variation (optimistic to pessimistic) for the cryogenic system is ± 5 seconds and for the nuclear systems, ± 20 seconds. The nuclear engine weight and the engine radiation shield weight are each assumed to vary by ± 1000 pounds from the expected value. All other spacecraft design parameters are as shown in table I.

The second vehicle weight for mission modes 1 and 3 (that is, weight for the flyby spacecraft) is 180 000 pounds, which allows 18 000 pounds for the Earth-entry module, 55 000 pounds for the mission module, and 107 000 pounds for the experiments package and/or propulsion module. Figure 23 indicates that mission modes 3 and 4 are not

competitive with mission modes 1 and 2 unless a nuclear propulsion system is used. Mission mode 1 is particularly effective because the first and second vehicles can each be launched into Earth orbit by a single Saturn V launch vehicle, since the weight of the largest vehicle is approximately 270 000 pounds; the Earth-orbit assembly is thus simplified. The sensitivity of the various modes to uncertainties in the design parameters is indicated by the length of each bar in figure 23. Mission mode 1 is the least sensitive, and the sensitivity increases through mission mode 4.

Variations of the Four Basic Mission Modes

Many possible variations to the four basic mission modes exist. These variations include the propulsion system used, the crew size, and the number of vehicles used. A complete analysis of all possible variations is unnecessary and beyond the intended scope of this study, but an assessment of some of the more meaningful variations is included. The variations selected for analysis illustrate the effect of systems requirements and mission profiles on the spacecraft module weights and the number of launches required.

The Mars lander could be launched to Mars as a separate, unmanned vehicle and guided to Mars orbit by using aerodynamic braking by a manned vehicle already in Mars orbit. The manned vehicle would then rendezvous with the lander in Mars orbit and continue the mission. The advantage of this operation is in the increased lander weight capability. The lander could weigh as much as 300 000 pounds and could be launched to Mars by using a propulsion stage assembled to the lander. This assumed that the Saturn V can deliver 306 000 pounds to a low Earth orbit. The increased lander weight could be used to increase the Mars surface exploration capability. This variation will be applied to mission modes 2 and 4 and referred to as mission modes 2a and 4b.

The reduction of crew size is an obvious possibility; the spacecraft weights for four-man and eight-man crews are computed for mission modes 1, 2a, 4, and 4b.

Tables IV and V list the lander weight, total spacecraft weight, total weight in Earth orbit, and the number of Saturn V launches required. The weights in table IV are based on an eight-man crew, and the weights in table V are based on a four-man crew. The assumed spacecraft velocity budget is shown in table III.

The total weight in Earth orbit and the number of launches required are based on the following assumptions.

1. A 306 000-pound payload capability in a 100-n. mi. circular Earth orbit
2. A characteristic velocity increment of 16 000 ft/sec required for transplanetary injection
3. A nuclear transplanetary injection stage used only if the Mars spacecraft uses nuclear propulsion — the specific impulse (I_{SP}) of the nuclear injection stage assumed as 820 seconds, the engine weight as 35 000 pounds, λ as 0.15, and the total weight as 306 000 pounds

Tables IV and V indicate that a nuclear propulsion system will reduce the number of Saturn V launches required for the Mars landing mission. For example, mission mode 4b with a four-man crew and nuclear propulsion is capable of placing a 300 000-pound lander on the surface of Mars by using only six Saturn V launches. However, a hypergolic mode 1 and a cryogenic mode 2a are equivalent to the nuclear modes 4 and 4b, except that mission modes 1 and 2a require two additional Saturn V launches. The Mars landing mission can be accomplished without a nuclear propulsion system, but at the cost of two to three additional Saturn V launches depending on the mission mode selected.

CONCLUSIONS

The most efficient mission mode for the Mars landing depends on the type of trajectories used, and the conclusions derived from this study apply only to the opposition class of missions with flight times of 400 to 500 days. The conclusions are further qualified by the fact that the study is strictly a performance analysis and does not include a concurrent assessment of spacecraft design feasibility. Therefore, the following is a list of "qualified" conclusions.

1. Aerodynamic braking is equal to or better than nuclear propulsion for either direct return-to-Earth or hyperbolic rendezvous missions.
2. Propulsive orbital insertion with direct return-to-Earth requires the largest total spacecraft weight and is the most sensitive to uncertainties in spacecraft design parameters when compared to the other mission modes studied.
3. Aerodynamic braking into Mars orbit with hyperbolic rendezvous for direct return-to-Earth is the mission mode most consistent with the current Saturn V launch-vehicle payload and can be accomplished without the development of nuclear Earth-orbit launch stages.
4. The use of nuclear Earth-orbit launch stages reduces the number of launches required and, thus, either simplifies Earth-orbit operations or permits a larger spacecraft weight in the Earth-orbit operations for a given number of launches.
5. Sending the lander to Mars in an unmanned configuration as a separate vehicle could provide lander weight capability in excess of 300 000 pounds and thus provide for greater surface exploration capability.

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TABLE I. - PRELIMINARY DESIGN PARAMETERS FOR MARS LANDING MISSION

Parameter	Optimistic	Expected	Pessimistic
Ratio of propulsion module stage weight to fuel weight	0.10	0.12	0.14
Ratio of Mars-entry structure weight to total entry weight10	.15	.20
Propulsion module specific impulse, sec	385	380	375
Mars lander weight, lb	100 000	110 000	120 000
Mission module weight (four-man crew), lb	30 000	40 000	50 000
Mission module weight (eight-man crew), lb	45 000	55 000	65 000
Payload required for interplanetary rendezvous, lb	11 250	15 000	18 750
Earth-entry module, lb	15 000	18 000	21 000

**TABLE II. - COMPARISON OF PROPULSION SYSTEMS
FOR MARS LANDING MISSION MODE**

Propulsion	Specific impulse, sec	λ	Engine weight, lb
Hypergolic	380	0.12	--
Cryogenic	420	.15	--
Nuclear	820	.15	15 000

**TABLE III. - COMPARISON OF VELOCITY BUDGET
FOR MARS LANDING MISSION MODE**

Maneuver	With aerodynamic braking, ft/sec	Without aerodynamic braking, ft/sec
Earth-to-Mars midcourse	500	500
Deboost-to-Mars orbit	500	15 000
Return-to-Earth	15 000	15 000
Mars-to-Earth midcourse	500	500

TABLE IV. - COMPARISON OF MARS LANDING MISSION MODES (EIGHT-MAN CREW)

Mode	Propulsion	Lander weight, lb	Earth-orbit weight, lb		Spacecraft weight, lb		Number of launches	
			Spacecraft 1	Spacecraft 2	Spacecraft 1	Spacecraft 2	Spacecraft 1	Spacecraft 2
1	Hypergolic	110×10^3	1200×10^3	1220×10^3	284×10^3	306×10^3	4	4
1	Nuclear	110	918	918	302	306	3	3
2a	Cryogenic	300	1540	1220	416	306	5	4
2a	Nuclear	300	880	918	266	306	3	3
4	Nuclear	110	1600	--	680	--	6	--
4b	Nuclear	300	1370	918	451	306	4	3

TABLE V. - COMPARISON OF MARS LANDING MISSION MODES (FOUR-MAN CREW)

Mode	Propulsion	Lander weight, lb	Earth-orbit weight, lb		Spacecraft weight, lb		Number of launches	
			Spacecraft 1	Spacecraft 2	Spacecraft 1	Spacecraft 2	Spacecraft 1	Spacecraft 2
2a	Cryogenic	300×10^3	1220×10^3	1220×10^3	314×10^3	306×10^3	4	4
2a	Nuclear	300	526	918	220	306	2	3
4	Nuclear	110	1530	--	602	--	5	-
4b	Nuclear	300	985	918	373	306	3	3

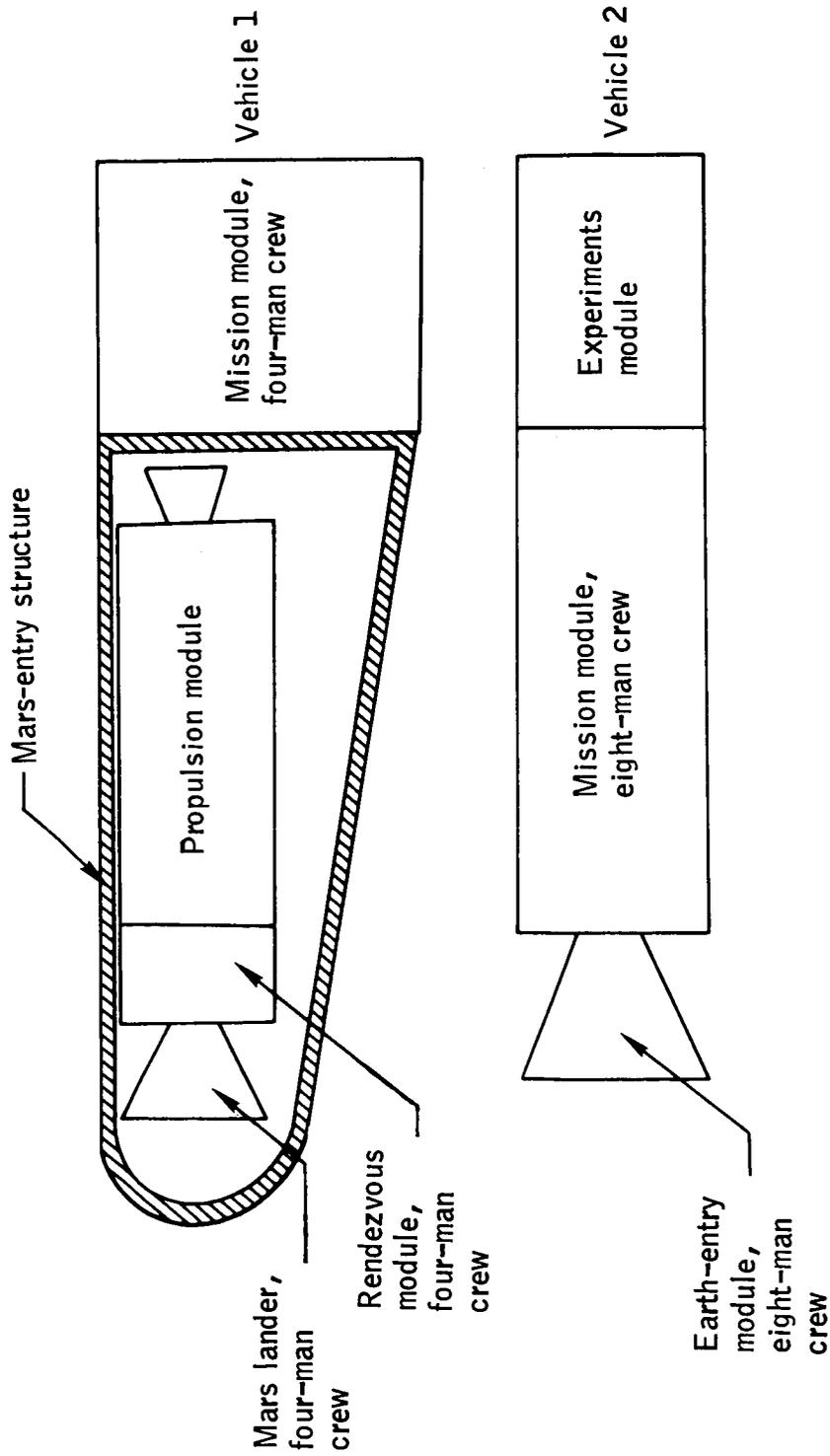


Figure 1. - Aerodynamic braking with interplanetary rendezvous for Earth return (Mars landing mission, mode 1).

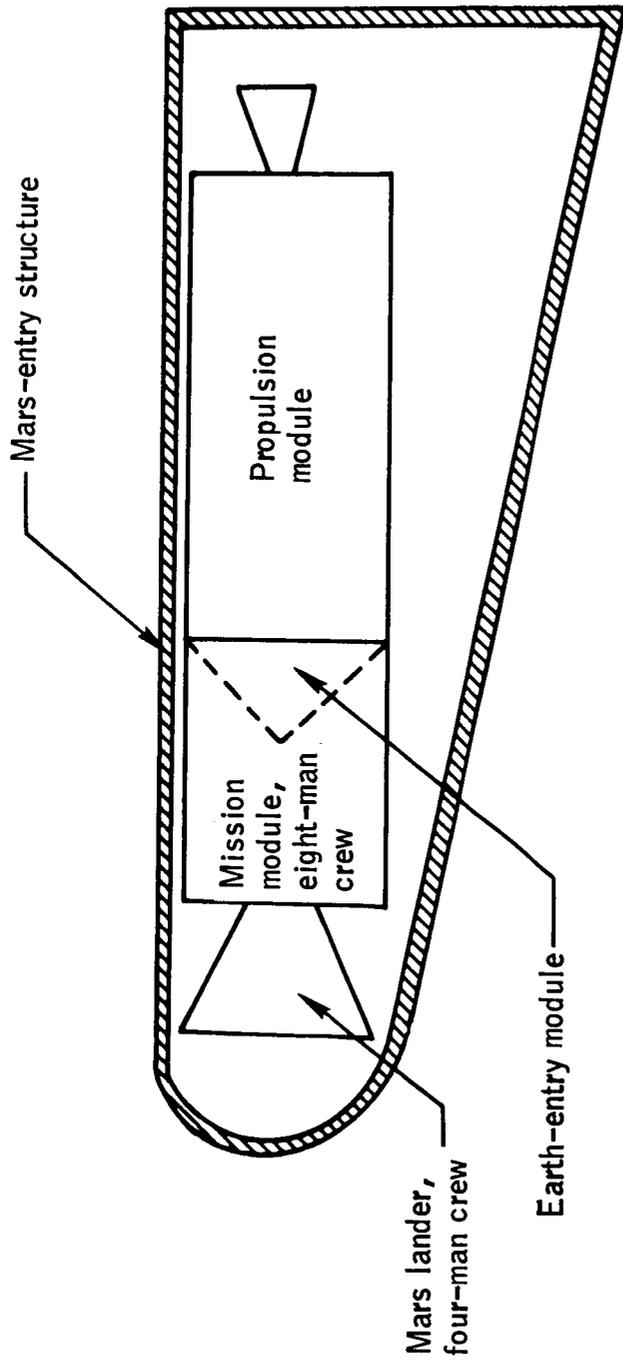


Figure 2. - Aerodynamic braking with direct return-to-Earth (Mars landing mission, mode 2).

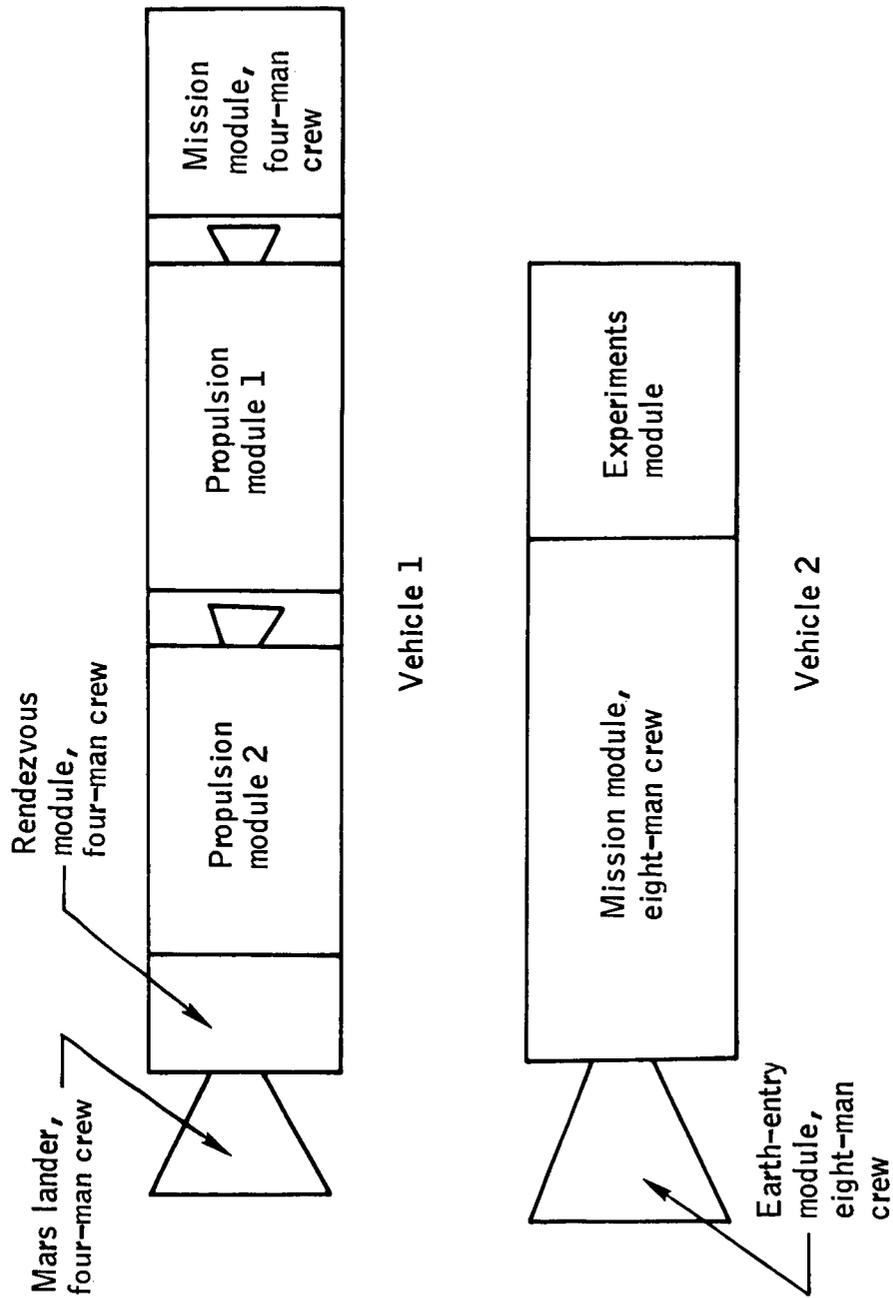


Figure 3. - Deboost to Mars orbit with interplanetary rendezvous for Earth return (Mars landing mission, mode 3).

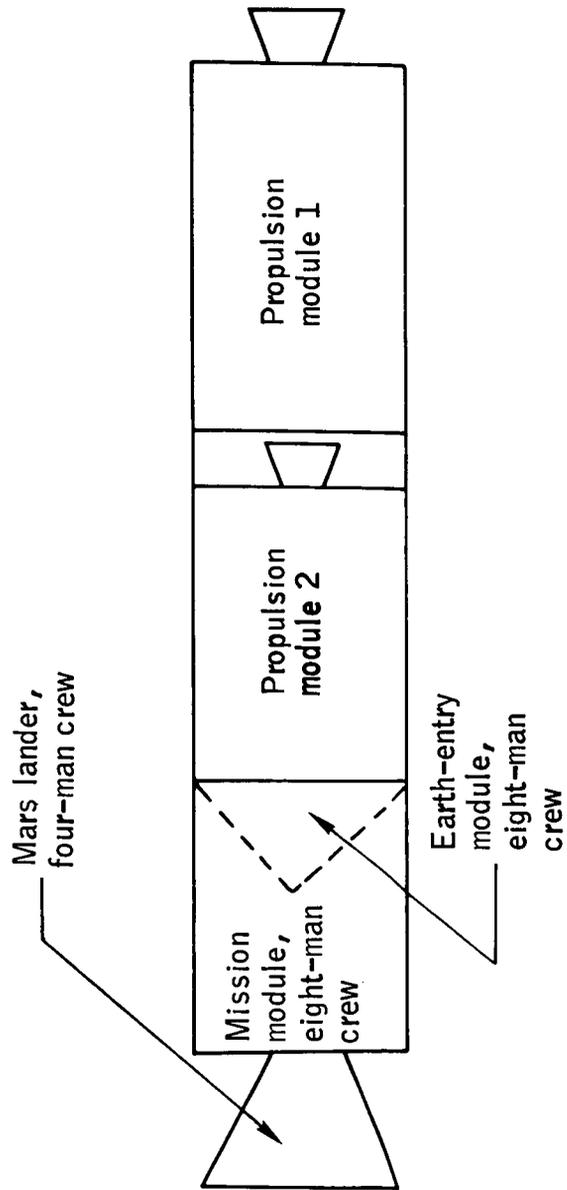


Figure 4. - Deboost to Mars orbit with direct return-to-Earth
(Mars landing mission, mode 4).

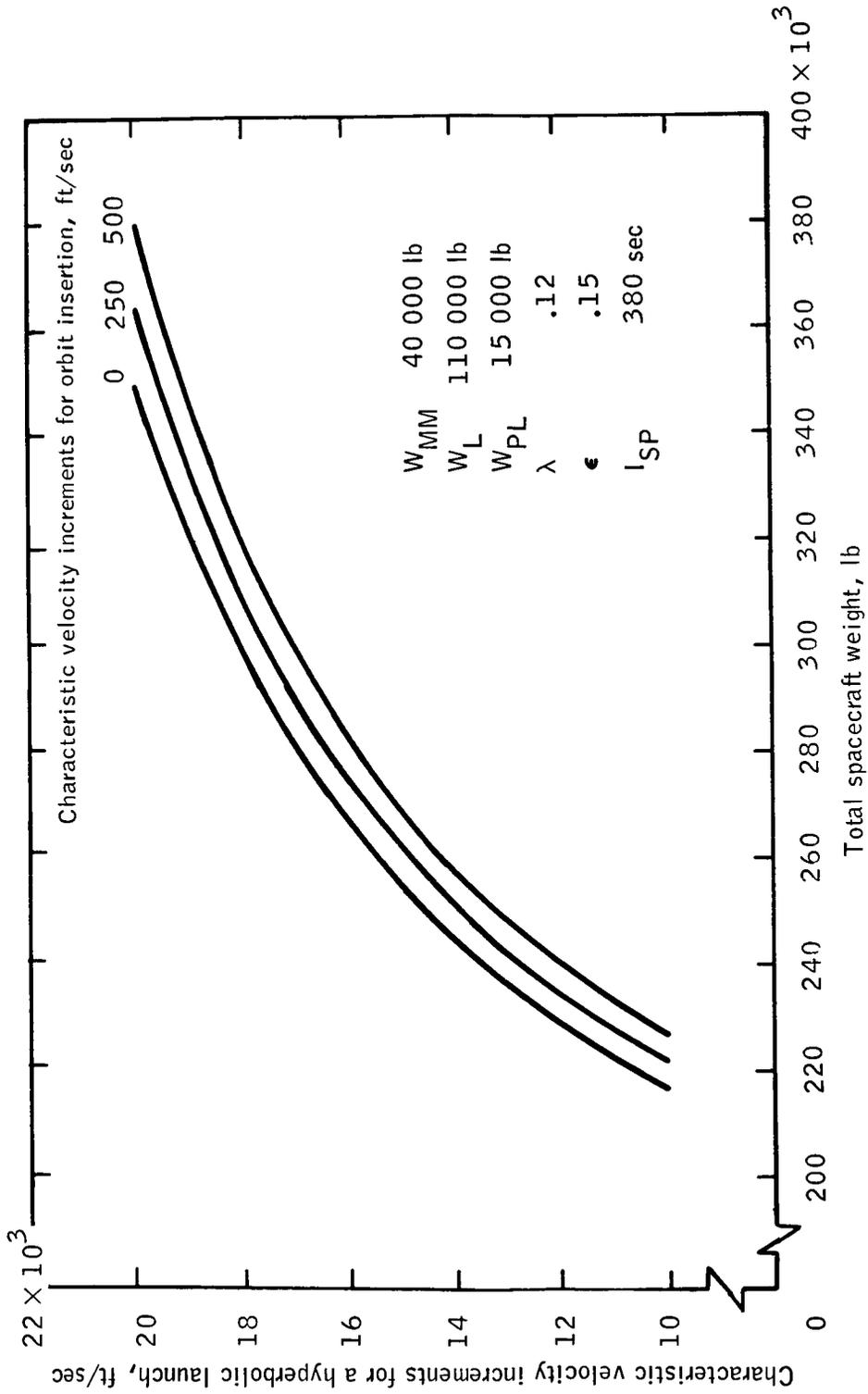


Figure 5. - Total spacecraft weight versus velocity requirements (Mars landing mission, mode 1).

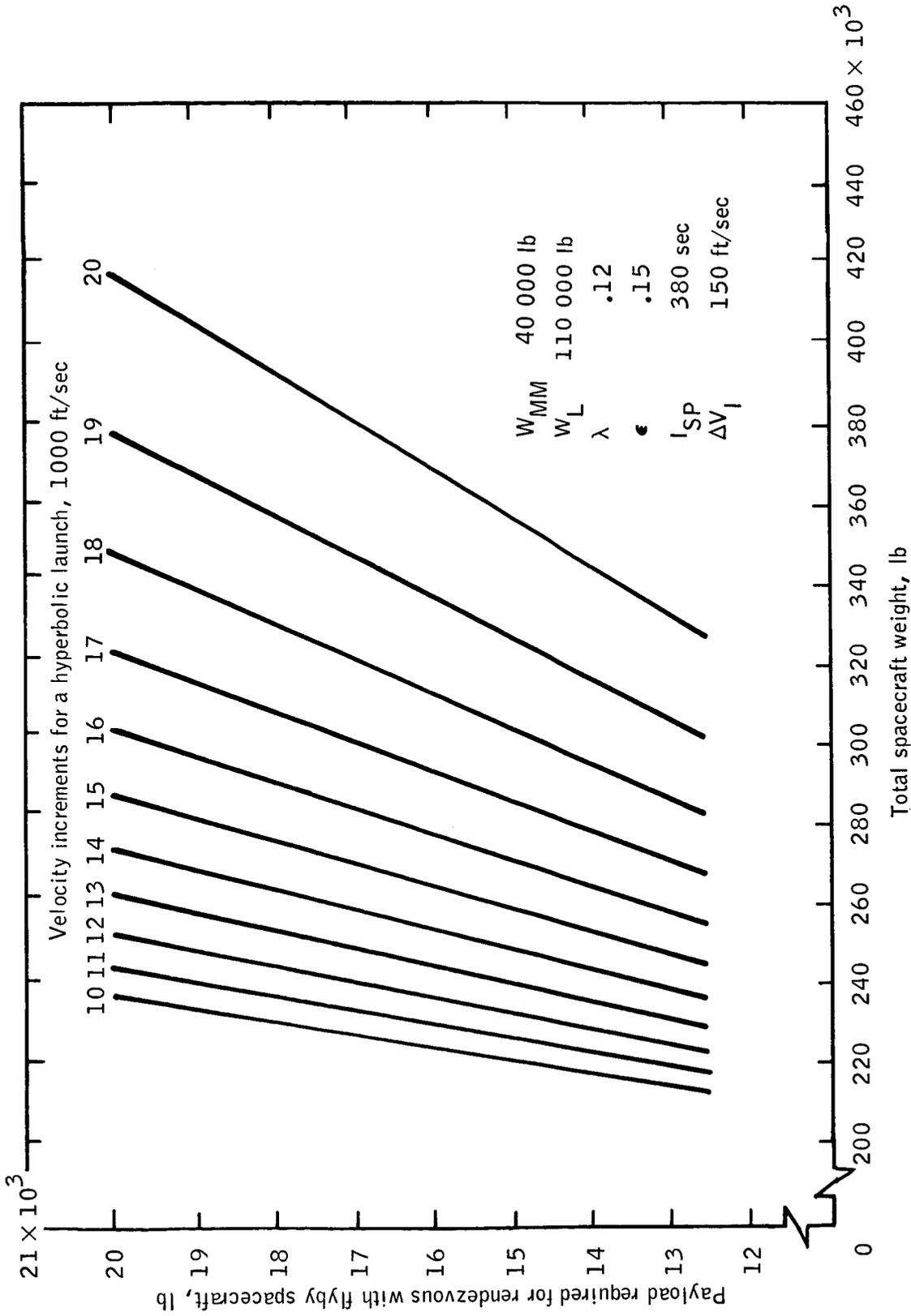


Figure 6. - Total spacecraft-weight versus rendezvous payload (Mars landing mission, mode 1).

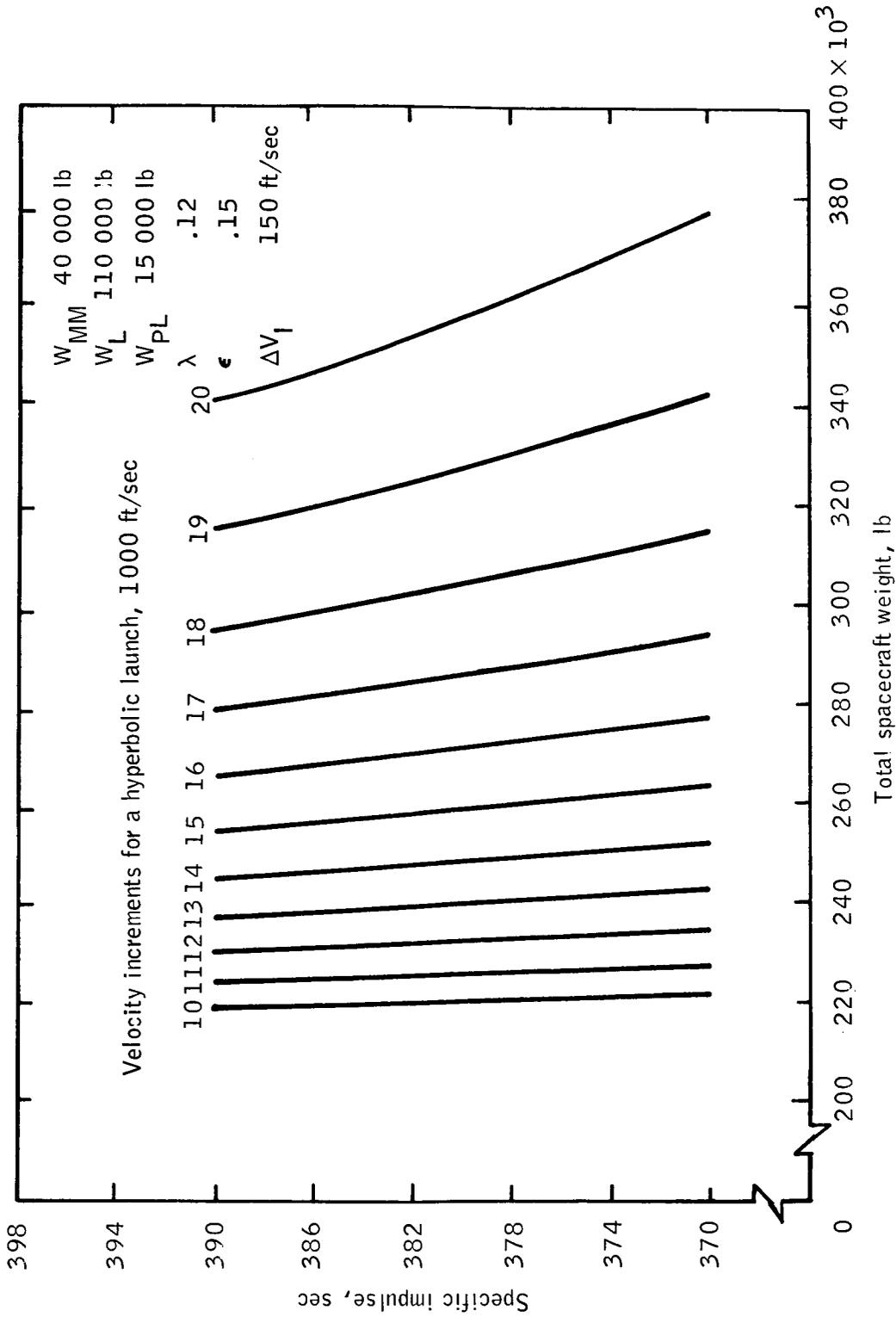


Figure 7. - Total spacecraft weight versus specific impulse (Mars landing mission, mode 1).

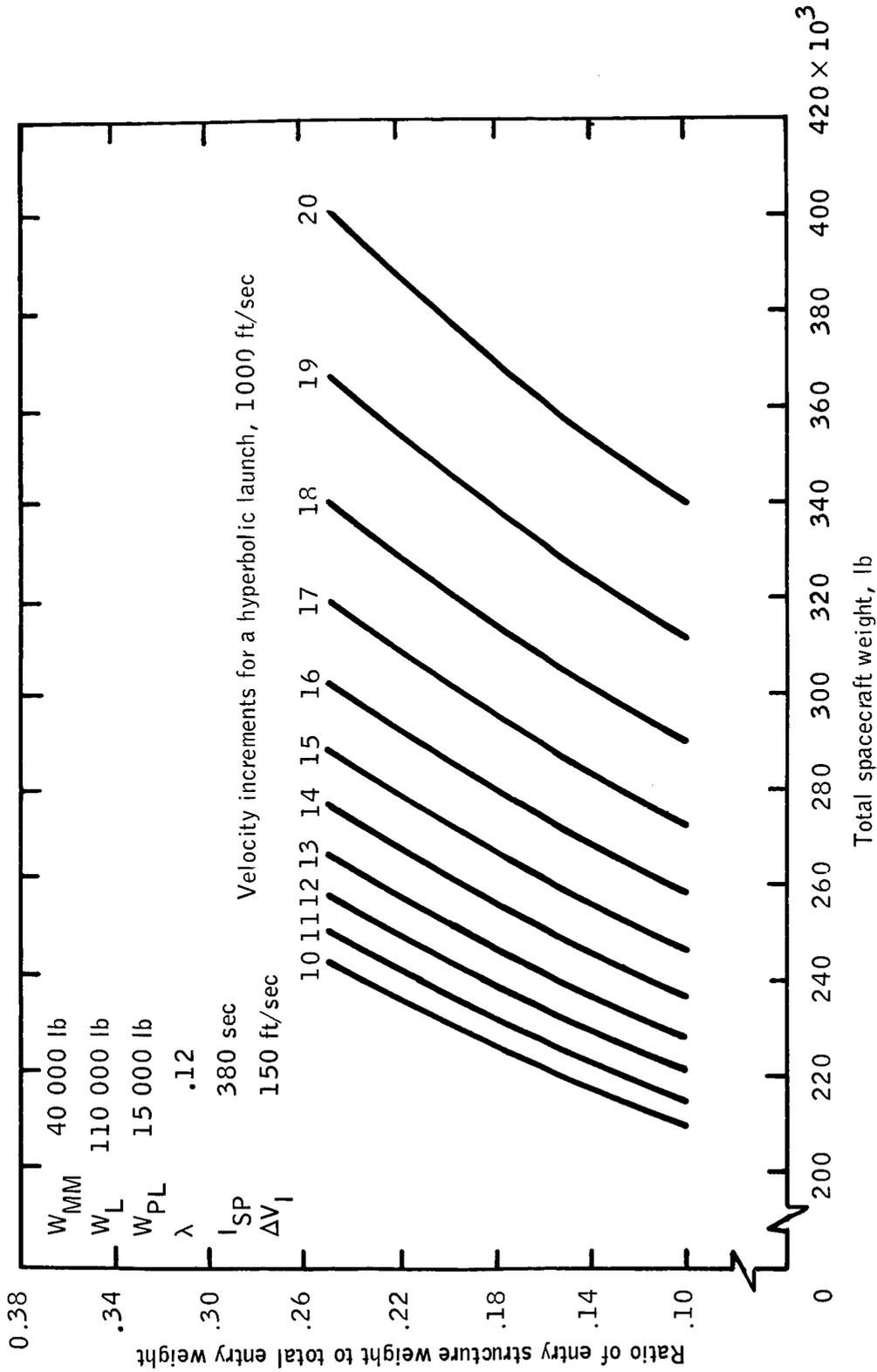


Figure 8. - Total spacecraft weight versus ratio of entry structure weight to total entry weight (Mars landing mission, mode 1).

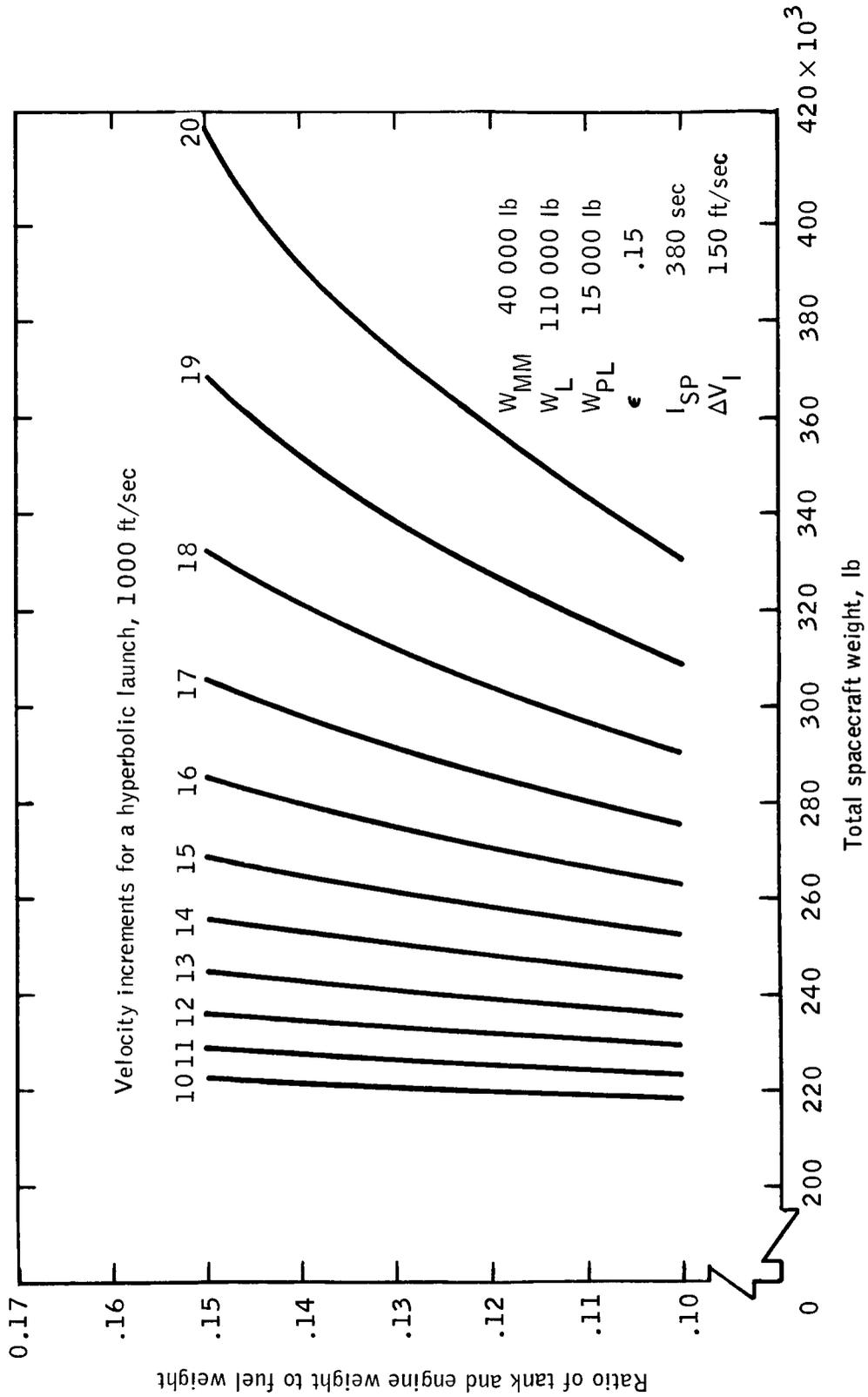


Figure 9.- Total spacecraft weight versus ratio of tank and engine weight to fuel weight (Mars landing mission, mode 1).

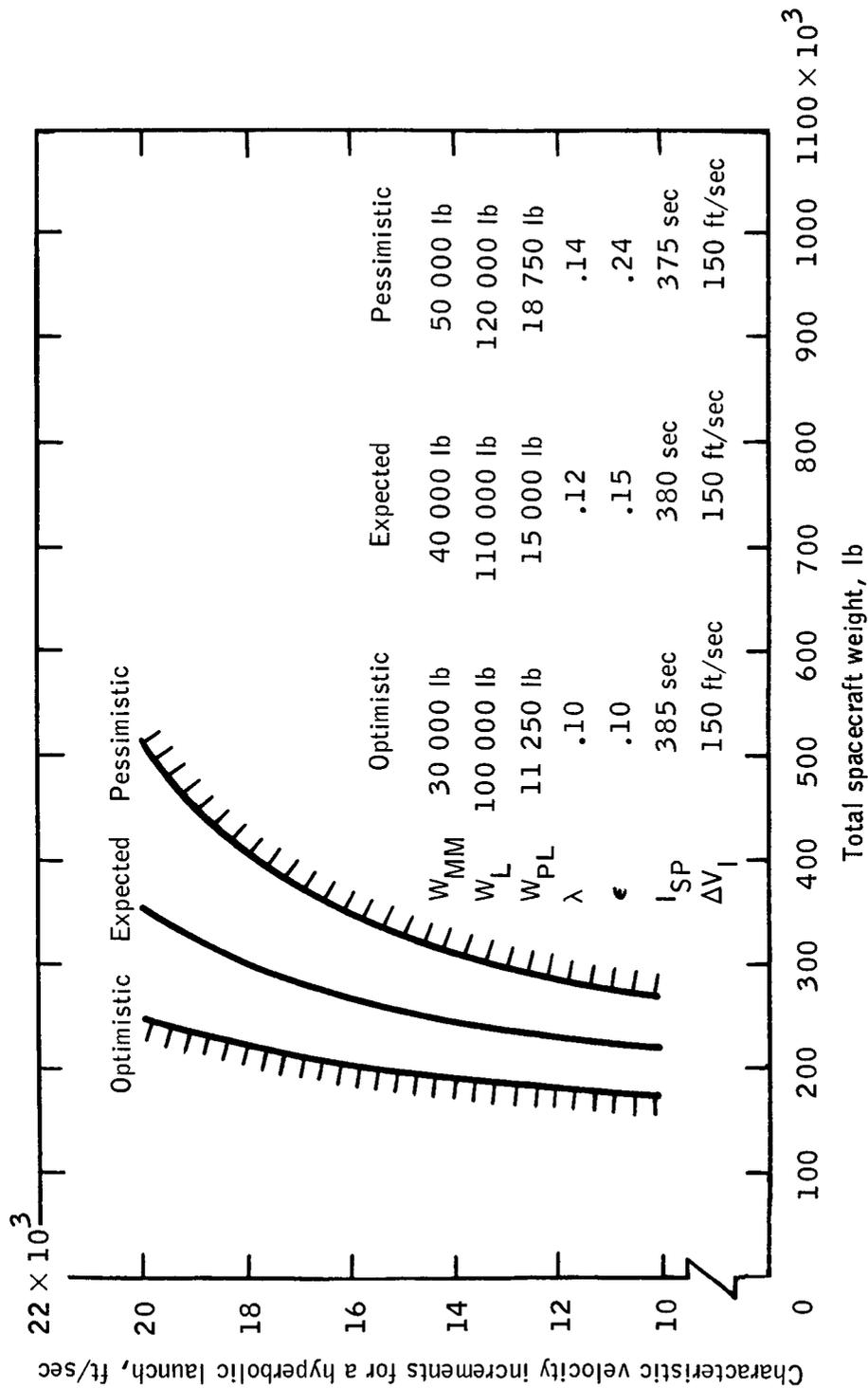


Figure 10. - Design range for total spacecraft weight (Mars landing mission, mode 1).

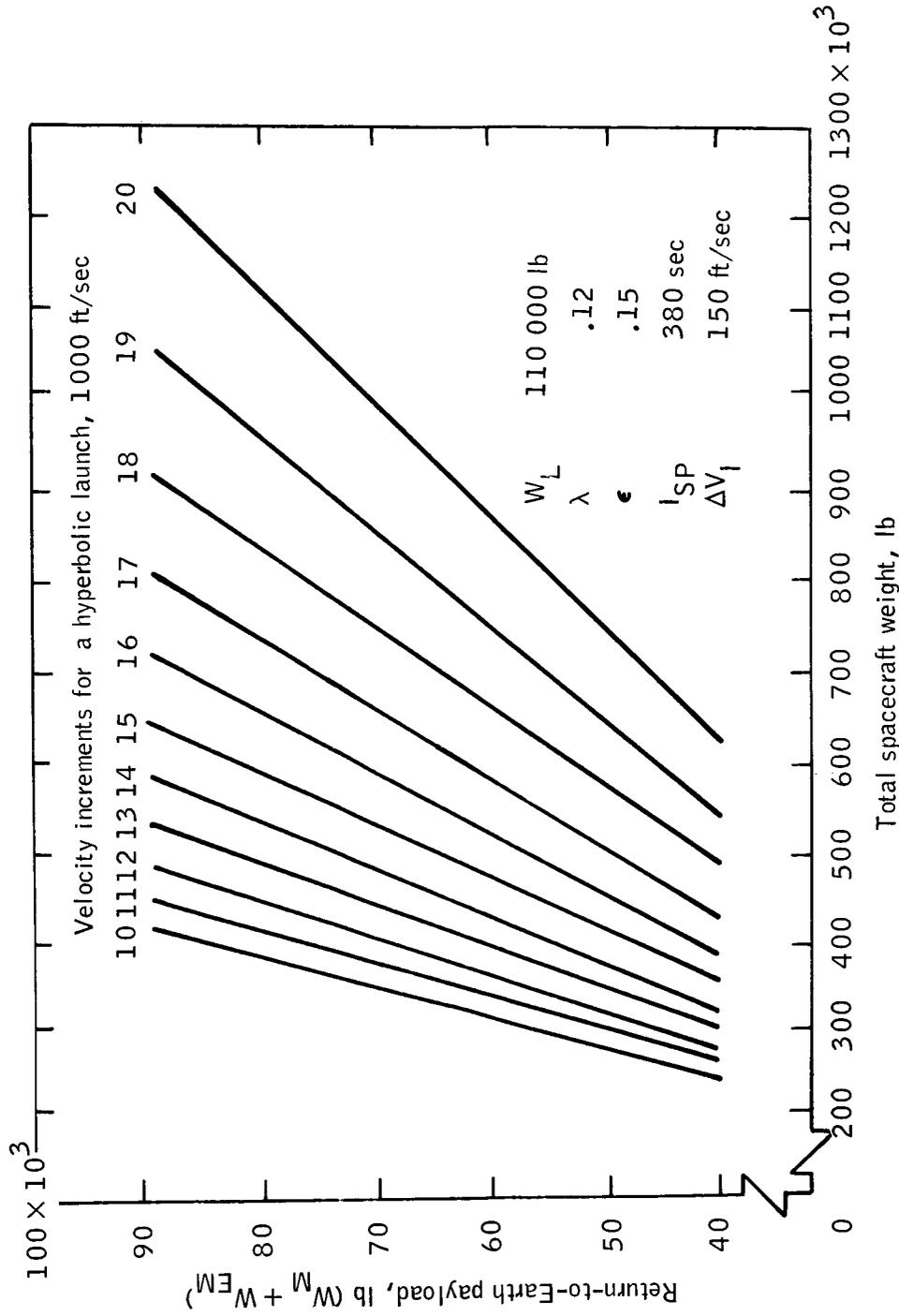


Figure 11.- Total spacecraft weight versus return-to-Earth payload (Mars landing mission, mode 2).

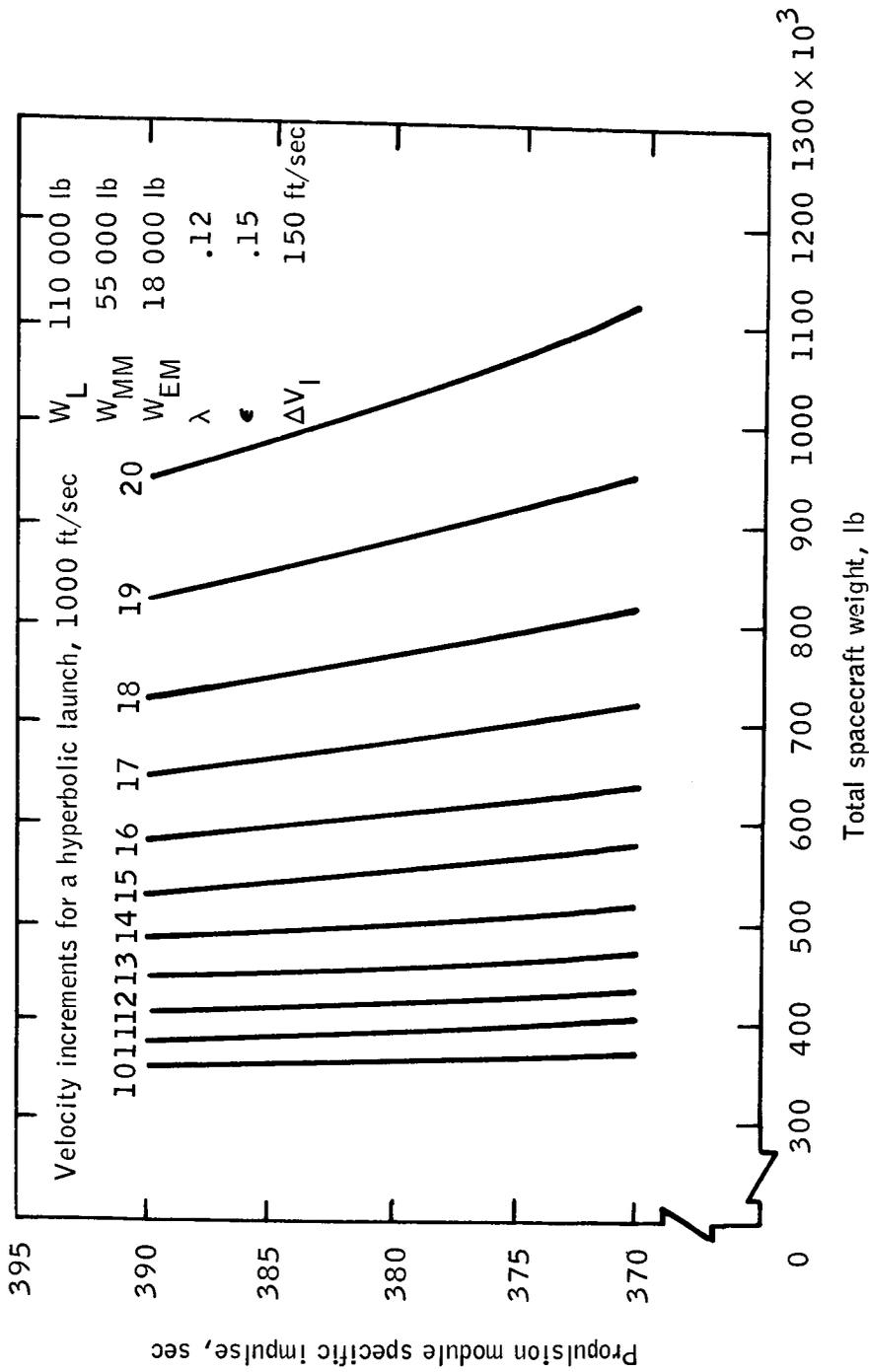


Figure 12. - Total spacecraft weight versus specific impulse (Mars landing mission, mode 2).

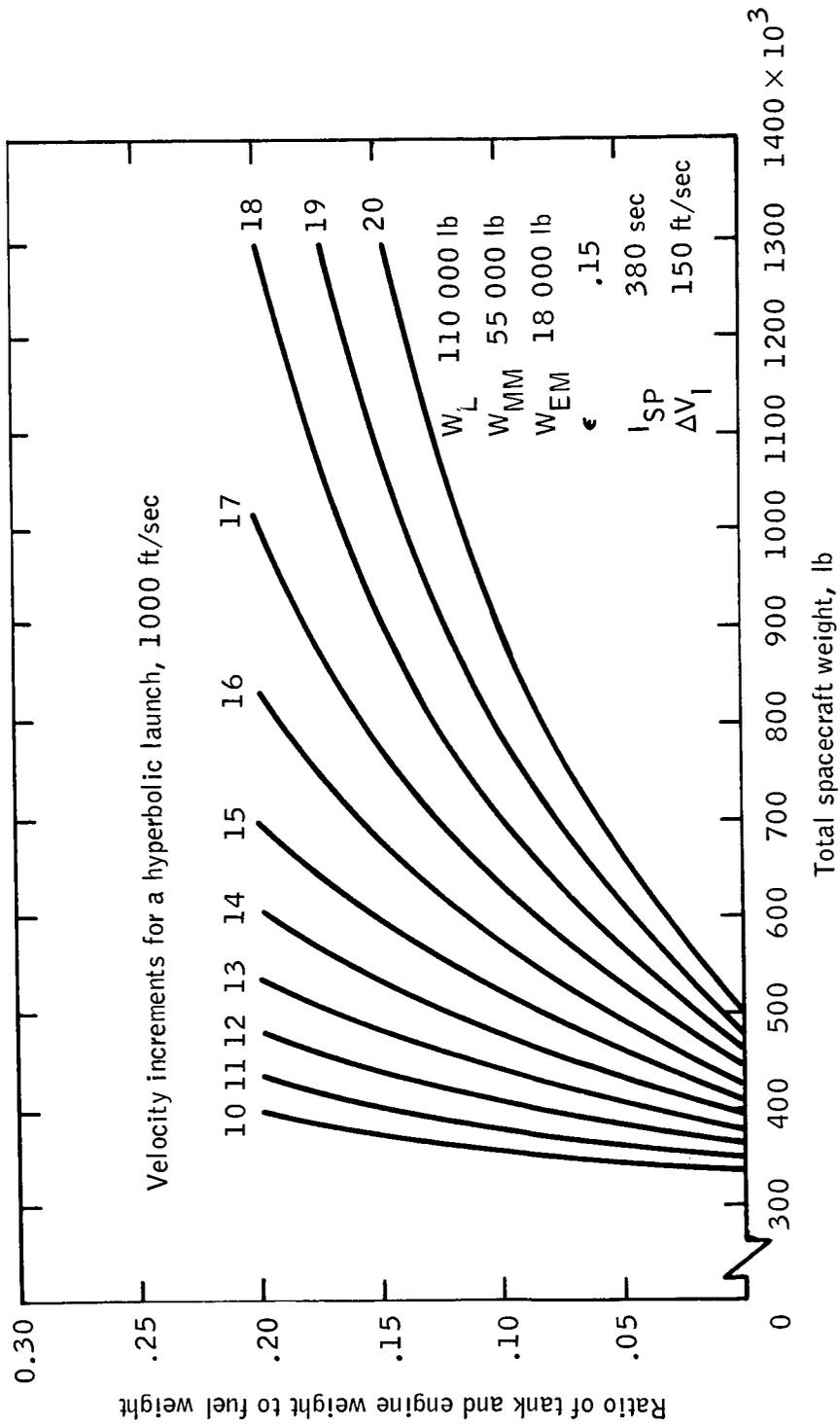


Figure 13. - Total spacecraft weight versus ratio of tank and engine weight to fuel weight (Mars landing mission, mode 2).

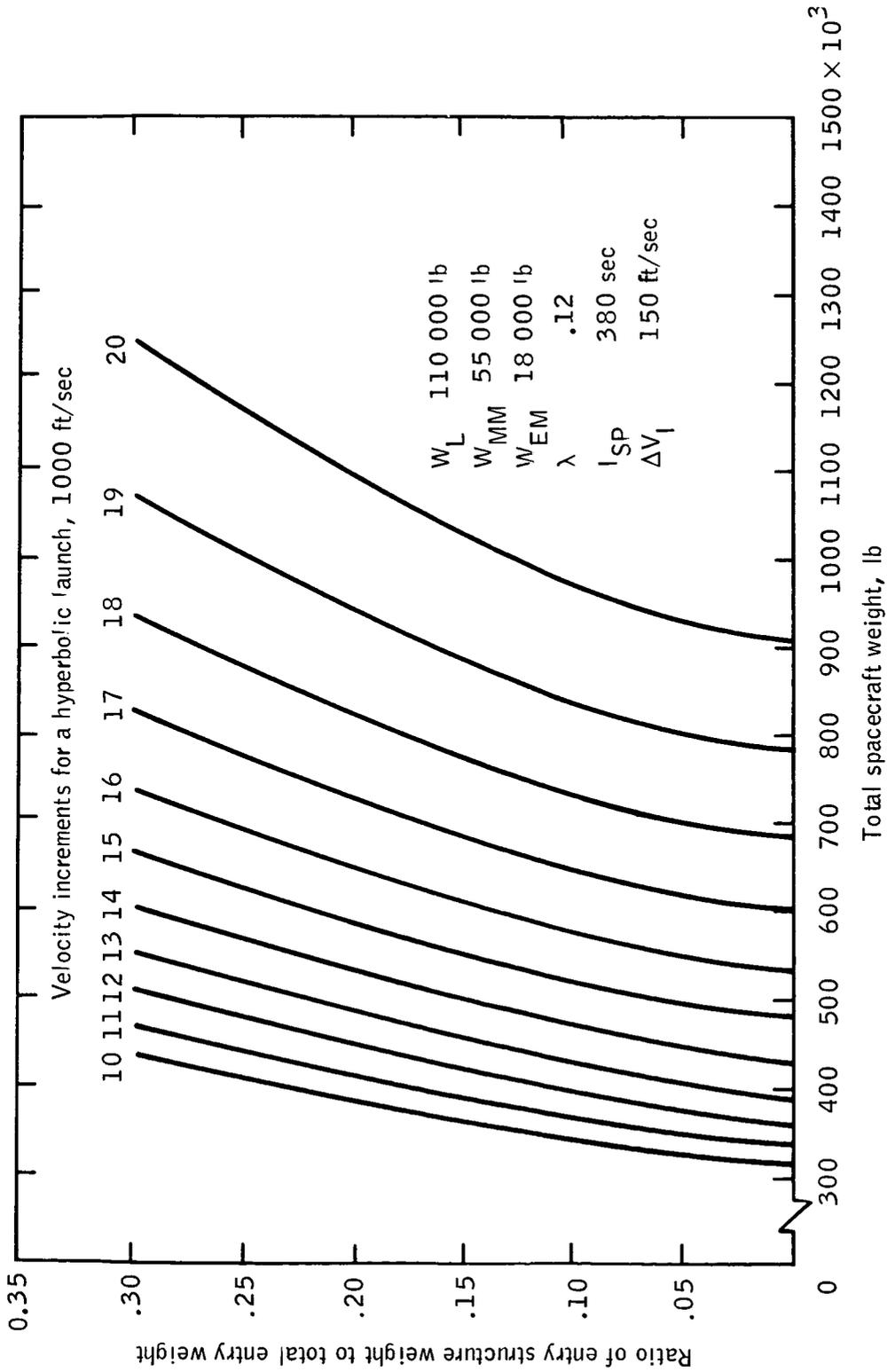


Figure 14. - Total spacecraft weight versus ratio of entry structure weight to total entry weight (Mars landing mission, mode 2).

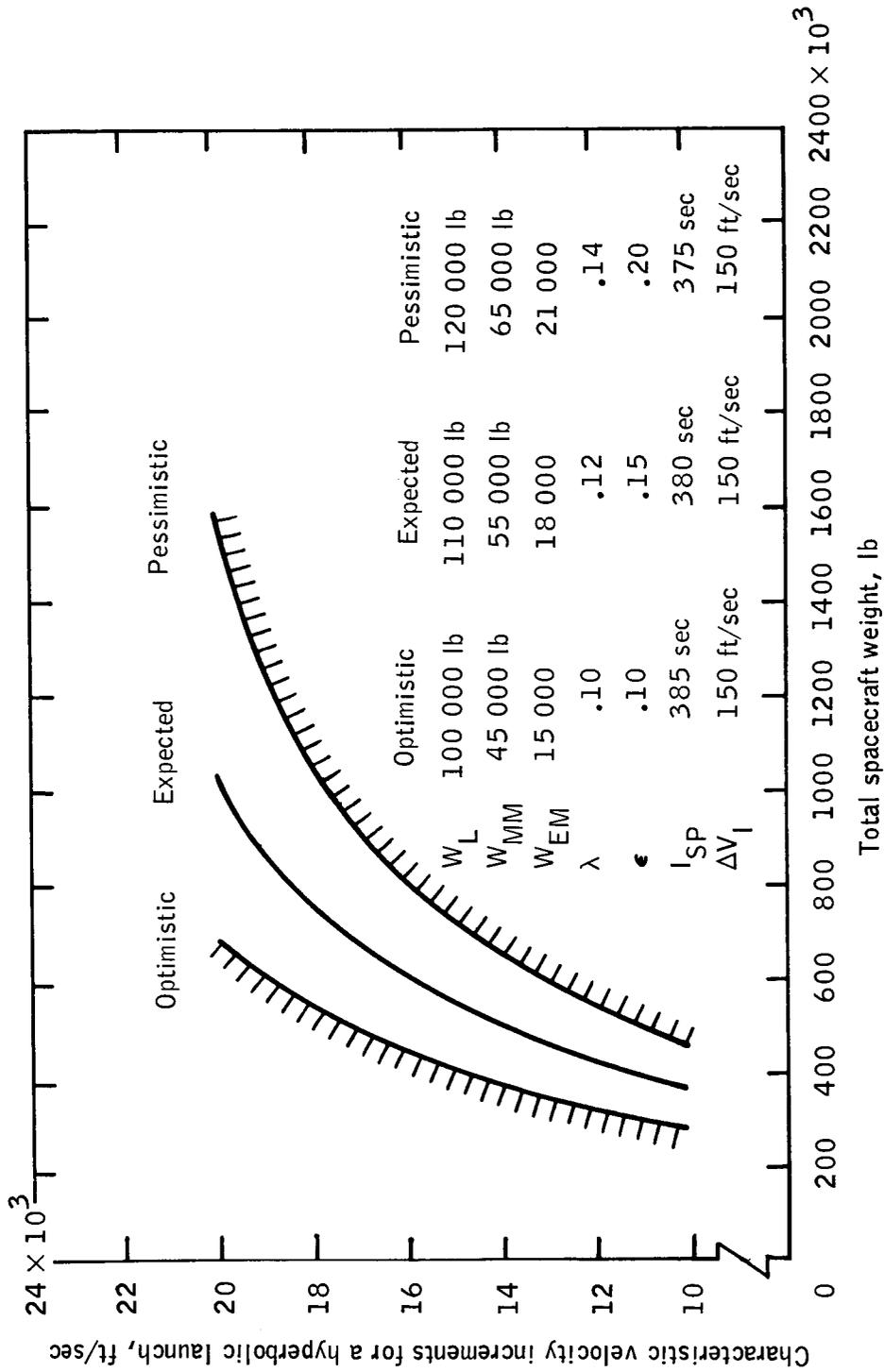


Figure 15.- Design for total spacecraft weight (Mars landing mission, mode 2).

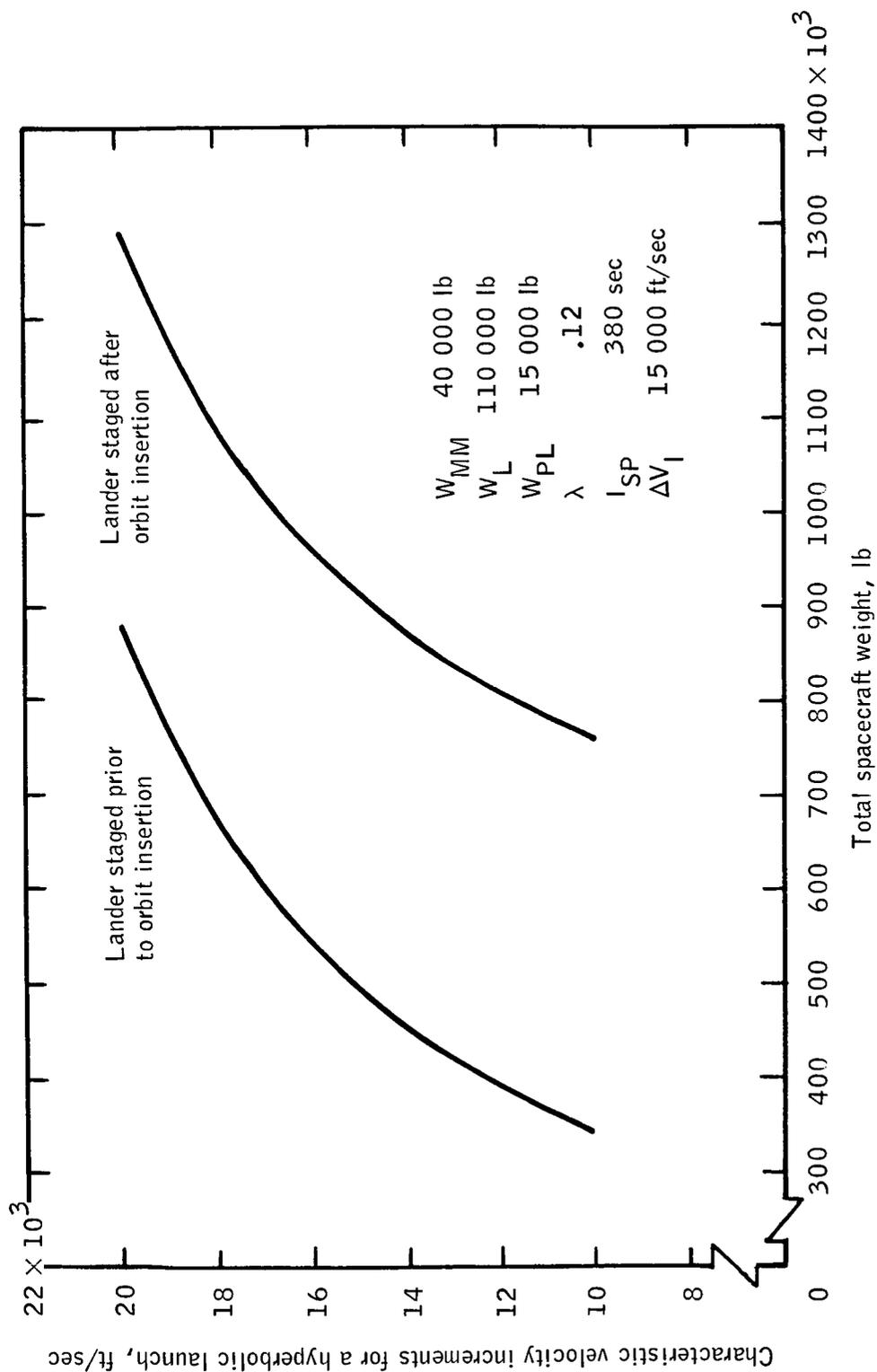


Figure 16. - Spacecraft weight with the Mars lander staged before and after orbit insertion (Mars landing mission, mode 3).

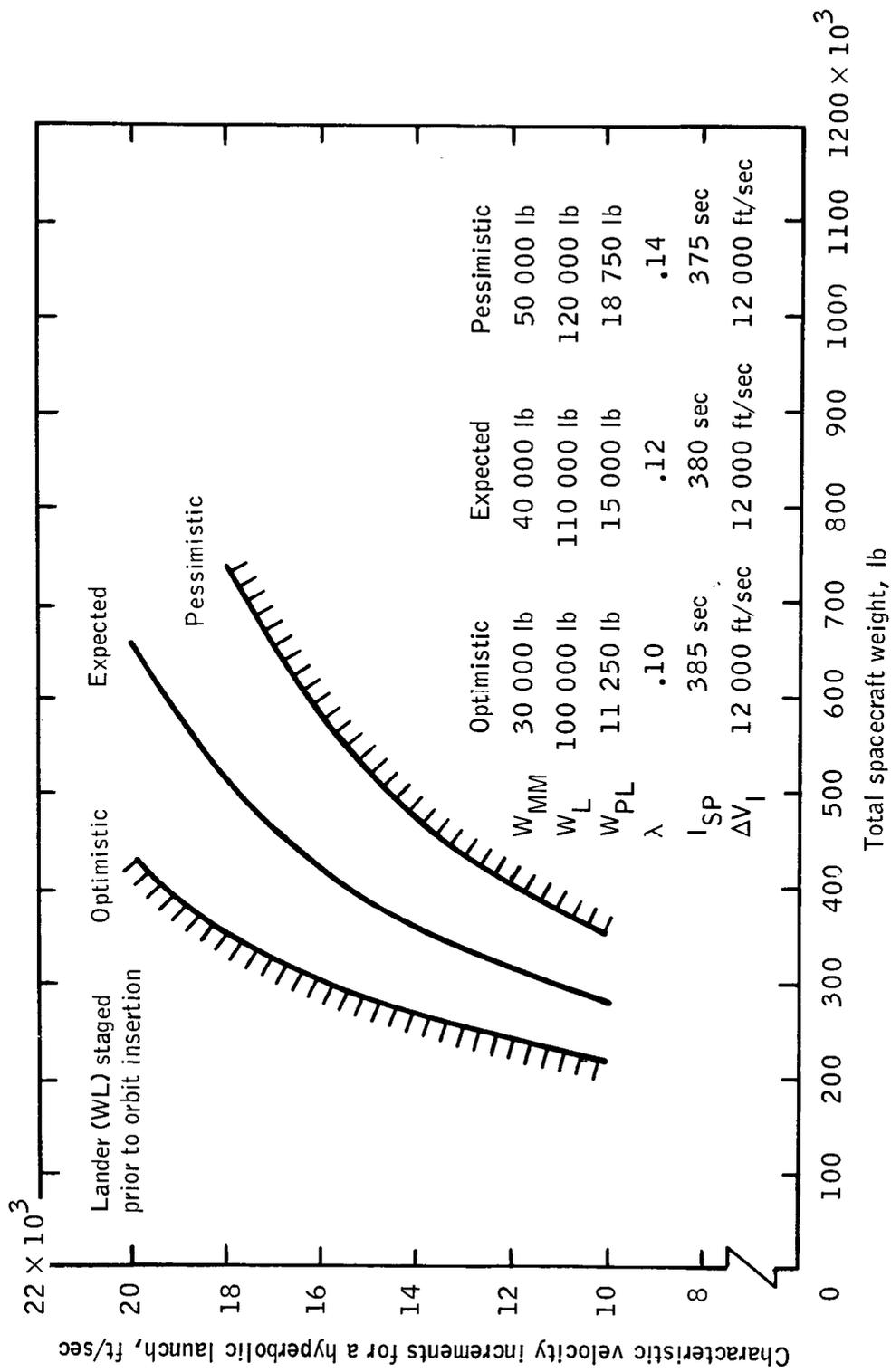


Figure 17. - Design range for total spacecraft weight — $\Delta V_I = 12\ 000\ \text{ft/sec}$
(Mars landing mission, mode 3).

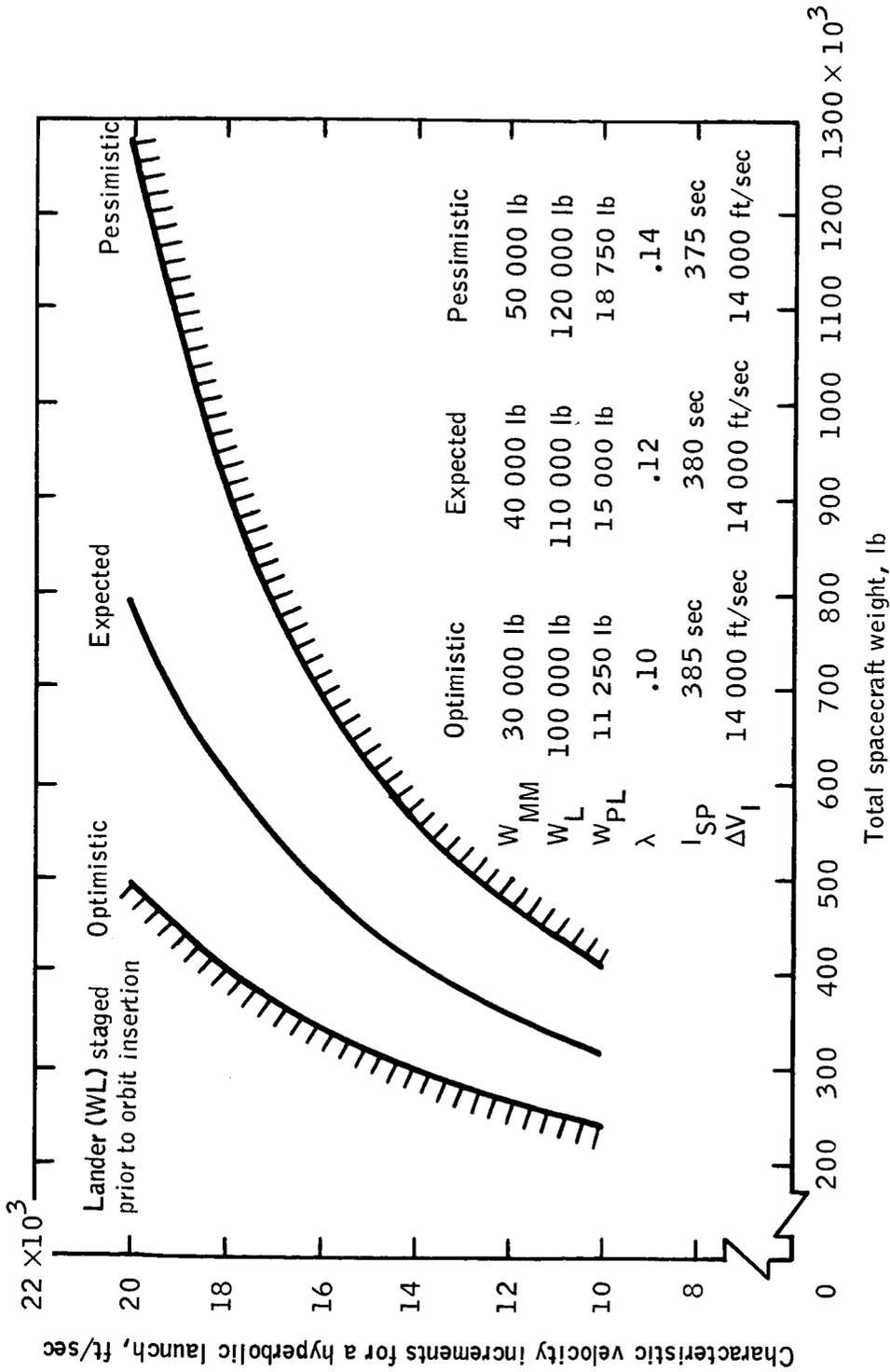


Figure 18. - Design range for total spacecraft weight --- ΔV_I 14 000 ft/sec
(Mars landing mission, mode 3).

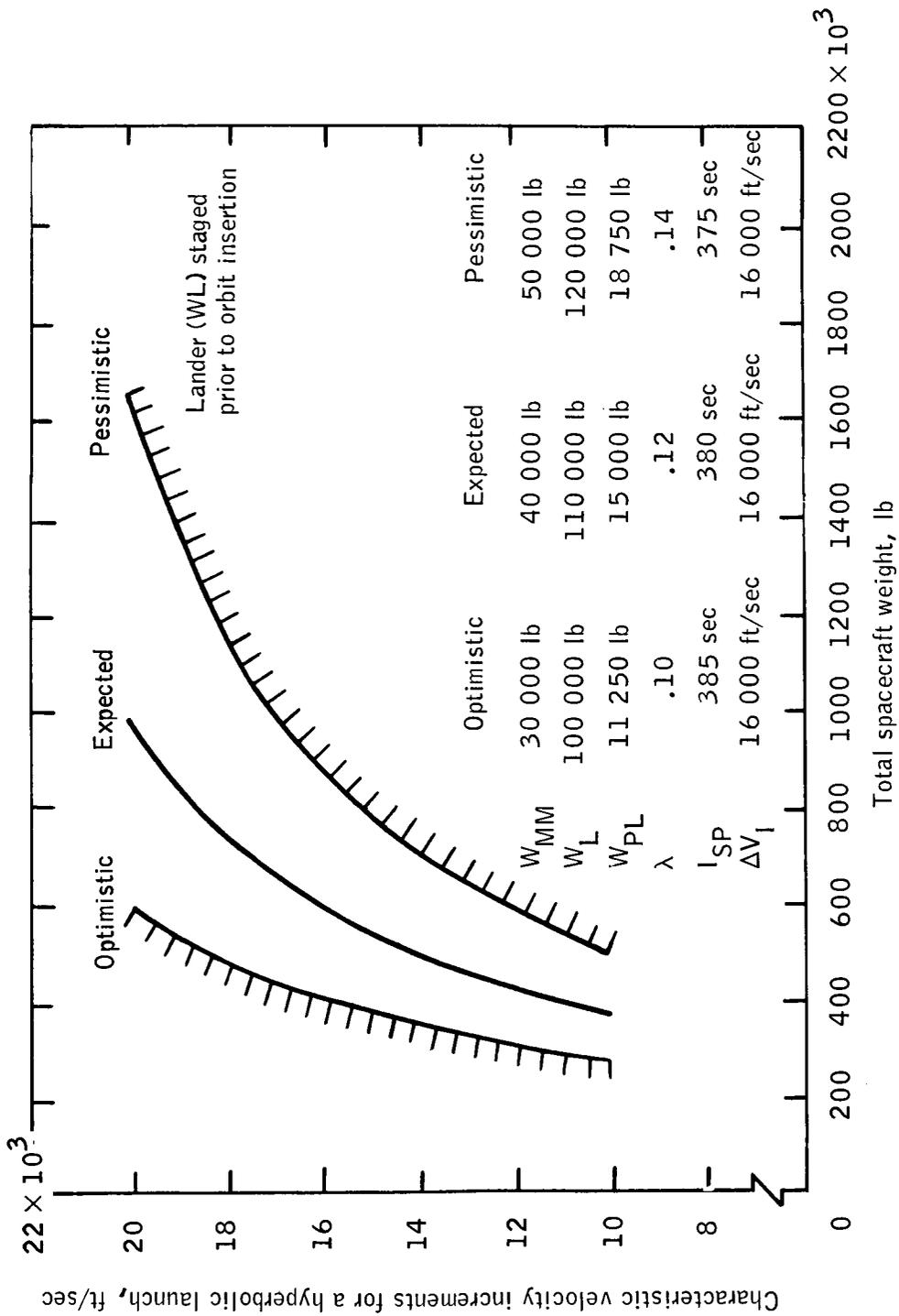


Figure 19. - Design range for total spacecraft weight — ΔV_I 1600 ft/sec (Mars landing mission, mode 3).

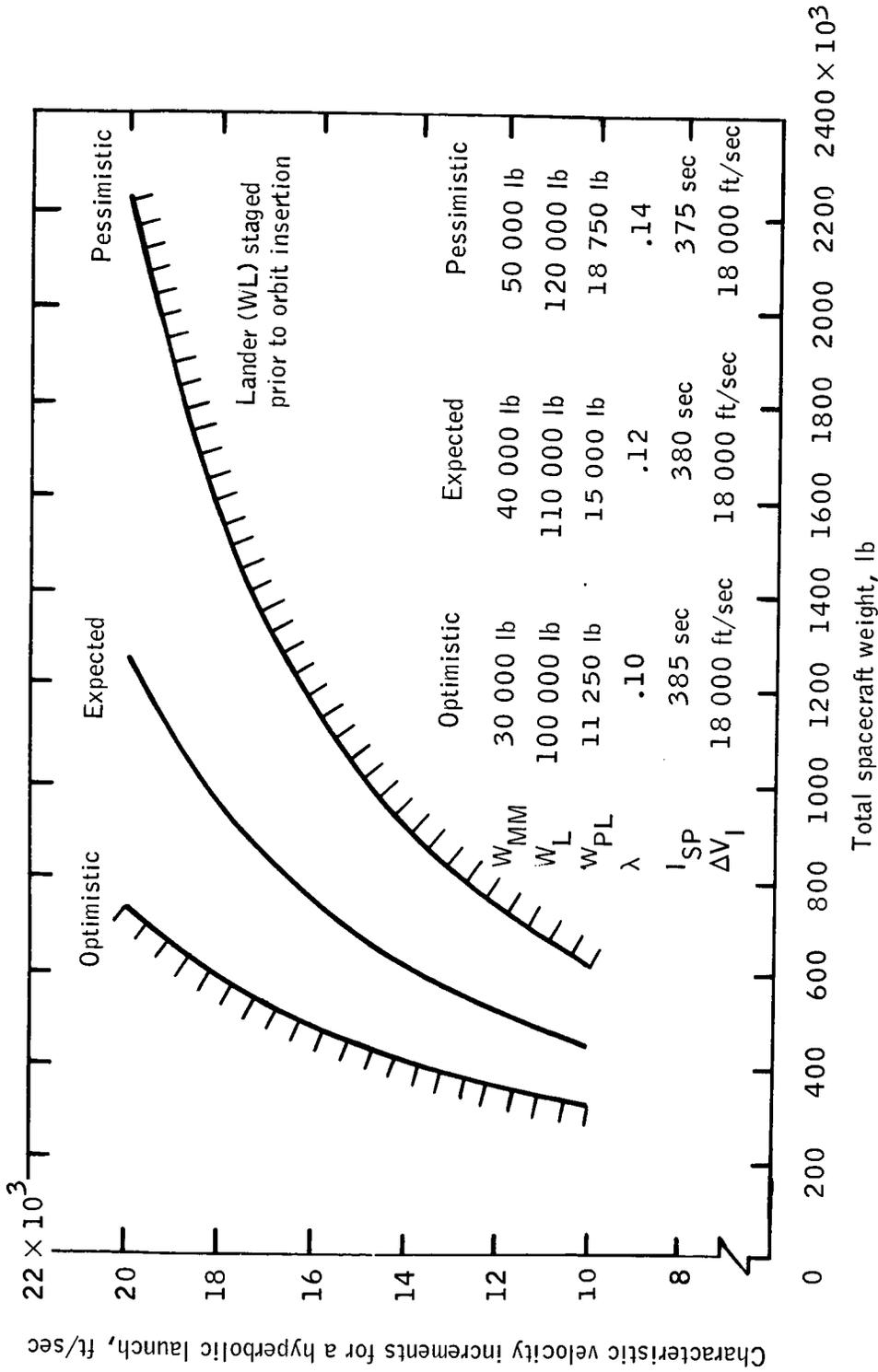


Figure 20. - Design range for total spacecraft weight — $\Delta V_I = 18\ 000\ \text{ft/sec}$
(Mars landing mission, mode 3).

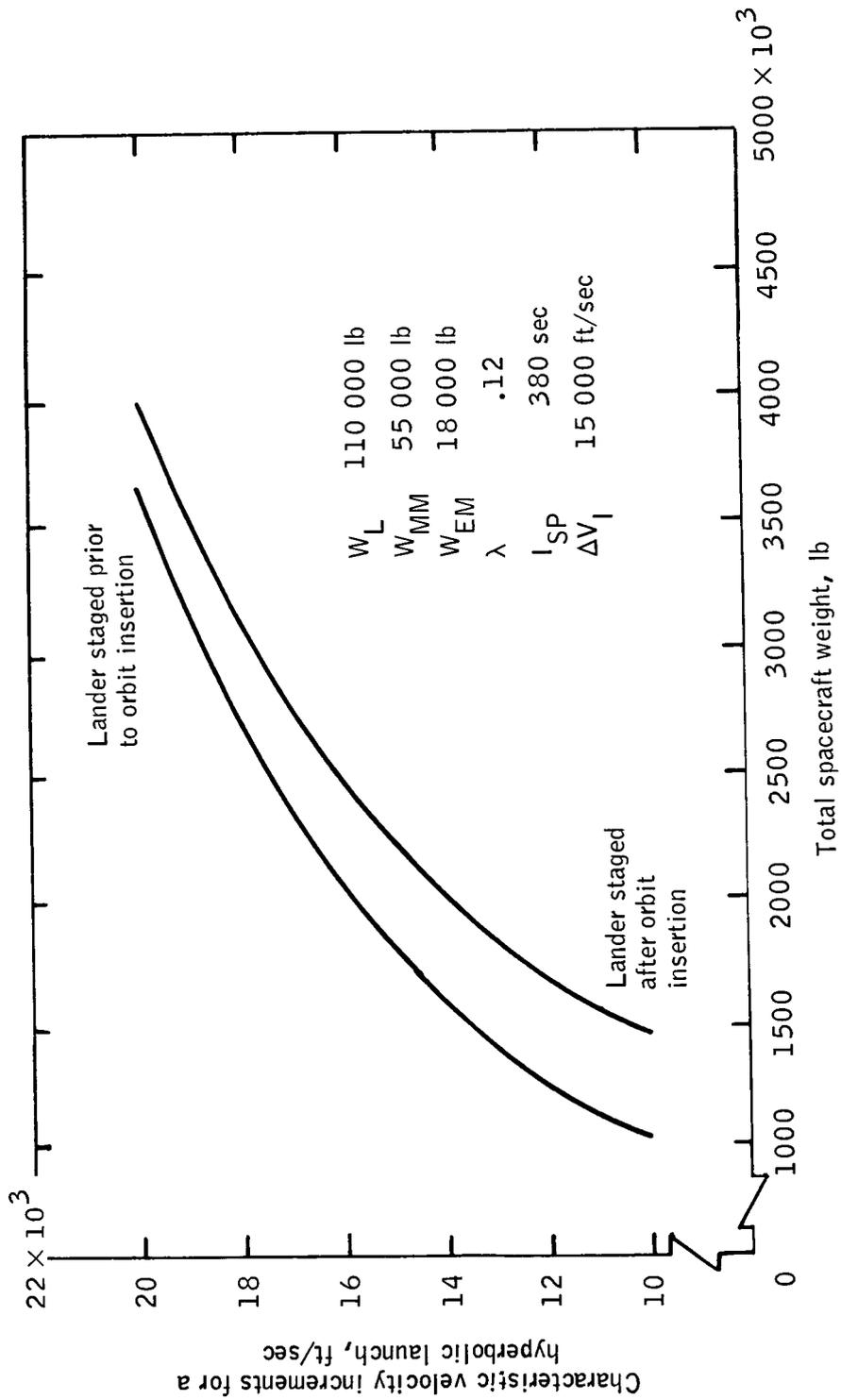


Figure 21. - Total spacecraft weight with the Mars lander staged before and after orbit insertion (Mars landing mission, mode 4).

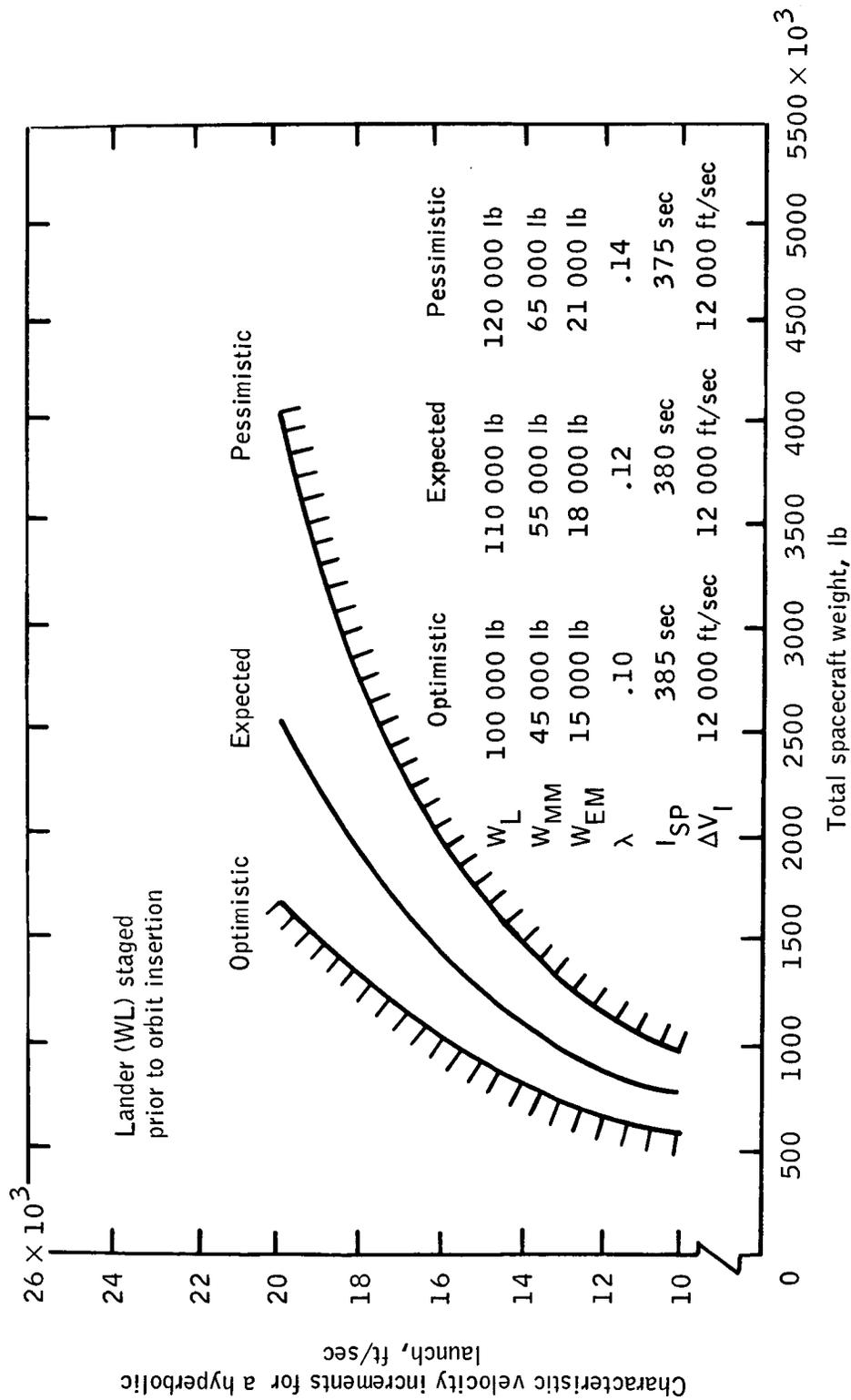
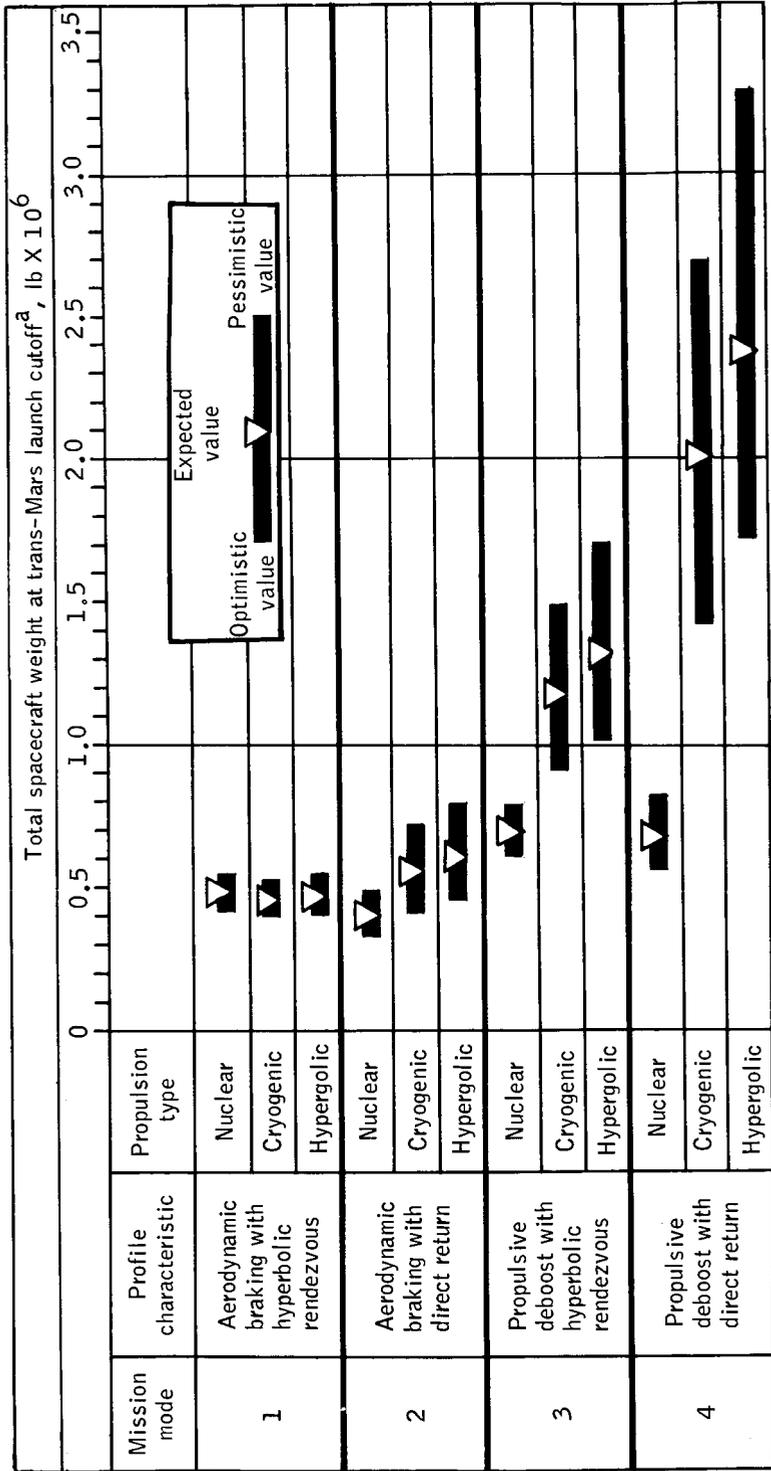


Figure 22. - Design range for total spacecraft weight (Mars landing mission, mode 4).



^aWeights include second vehicle where applicable.

^bLander staged after Mars orbit insertion.

Figure 23. - Comparison of Mars mission modes.