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"VEHICLE DESIGN FOR MARS LANDING AND RETURN TO MARS ORBIT"

by

DAVID M. HAMMOCK and BRUCE G. JACKSON

1 ref

MANNED SPACECRAFT CENTER, NASA

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PREPRINT (8)

This paper briefly describes three modes for accomplishing the Mars landing mission and compares them on a gross basis to indicate their probable order of merit and to identify design requirements placed on the Mars-excursion module (MEM) by the choice of mode. The paper shows that a flyby-rendezvous mode requiring low weight in earth orbit requires the MEM to enter the Mars atmosphere at velocities ranging from 20,000 to 30,000 ft/sec. The MEM for the flyby-rendezvous mode is not covered in this paper but merits further study.

The MEM for the other modes of mission accomplishment begins its active operational sequence in Mars orbit and need not be greatly influenced by the method of delivery to Mars orbit.

Parametric studies of the entry problem for two vehicles typifying a ballistic-type and a lifting-body-type were conducted to identify the problems associated with design of a MEM to accommodate the extremes of Mars atmospheric density presently predicted.

This brief study indicates that: (a) the presently predicted density extremes of the Mars atmosphere present no serious design problems for a MEM which can operate across the entire band of predicted densities; (b) details of operational requirements and mission objectives will control the choice of configuration rather than entry requirements; and (c) the ballistic-type MEM is lighter and simpler but has less operational flexibility than a high L/D MEM.

INTRODUCTION

Mars is perhaps the most exciting target for space exploration following Apollo and its definition of the lunar characteristics. Mars is particularly interesting because of the possibility of life on its surface and the ease with which men might be supported there. The temperatures of the Mars surface are temperate, when we consider the wide range of temperatures in space; the atmosphere is relatively hospitable. However, atmospheric pressures are such that some type of pressure suits would have to be worn; and the atmosphere for breathing would have to be supplied. The atmosphere contains water vapor and probably oxygen (.1% to 2%) which can be concentrated with relatively simple systems to provide the desired concentration of oxygen and to provide water for drinking and various sanitary purposes. Since it appears that with relatively simple equipment all the life support requirements on the Mars surface could be met, with the exception of food, the logistics support for a Mars scientific station would be small. In addition, if atmospheric braking is used for landing supplies on Mars, the energy requirement is less than that required to deliver cargo to the lunar surface.

Studies indicate that the exploration of Mars is feasible with the present technology. However, certain advances in this technology are highly desirable and would reduce considerably the weight in earth orbit required to accomplish this mission. The Apollo guidance and navigation system techniques are adequate for the Mars mission. Most of the subsystems and equipment of the Apollo command module are adequate for the earth reentry. The greatest advance in technology required is in the area of heat-protection systems for earth reentry for velocities ranging from 45,000 to 60,000 ft/sec.

It appears feasible to accomplish the mission using the Saturn V as the launch vehicle and with chemical propulsion in the spacecraft. Nuclear propulsion for earth-orbit-depart would be highly desirable, but not essential. Nuclear propulsion reduces the number of Saturn V's that would have to rendezvous to provide earth-depart propulsion. This capability, with the present technology and systems now under development, makes the Mars missions of immediate interest.

MARS MISSION MODES

There are a multitude of modes for accomplishing the Mars mission. However, the most attractive modes appear to be variations of three basic modes. Since the modes differ primarily in the use of propulsion or atmospheres for braking and in the place that rendezvous operations occur in the mission, we will refer to these modes as:

1. the flyby-rendezvous mode
2. the aerodynamic-braking mode
3. the propulsive-braking mode

Flyby-Rendezvous Mode

The flyby-rendezvous mode is shown in Figure 1. This mode involves two spacecraft. One spacecraft, the MEM, is launched, proceeds directly to Mars, decelerates and lands, using the Mars atmosphere. The other spacecraft flys by Mars without decelerating and returns to earth.

The sequence briefly is:

1. Rendezvous the Saturn V payloads in earth orbit necessary to assemble the flyby spacecraft and accomplish assembly and checkout.
2. Launch the flyby spacecraft into the Mars flyby trajectory. This launch may occur 50 to 100 days prior to launch of the MEM since its transit time to Mars requires about 200 days.
3. Rendezvous the necessary Saturn V payloads to assemble the MEM in earth orbit. Assemble and checkout the MEM.
4. Launch the MEM into a trajectory designed for direct landing on Mars.
5. The MEM proceeds to Mars, decelerates aerodynamically to elliptical orbit velocity and may establish a circular orbit from which the landing site is chosen. If the approach trajectory is such that a desired landing site may be reached without establishing a circular orbit, the landing is accomplished in a continuous maneuver from initial entry.
6. Choices in the launch times and trajectory variables of the two spacecraft are such that stay-times of about 40 days can be provided without undue increases in the required injection velocities. Scientific and engineering research activities are conducted during the stay on the surface.

7. As the flyby spacecraft approaches Mars, the details of its trajectory are transmitted to the MEM and the detailed procedure for rendezvous is developed by the two spacecraft.
8. As the flyby spacecraft passes Mars, the ascent stage of the MEM is launched into a trajectory matching that of the flyby spacecraft. Rendezvous is accomplished on the return trajectory. A launch window of three to four hours can be provided for at the expense of an added 1,000 ft/sec velocity capability above that required to match spacecraft velocities in the ideal case. The rendezvous would then be accomplished about two days after launch.
9. After rendezvous, the MEM crew transfers to the flyby spacecraft and the return trip is completed.

Aerodynamic-Braking Mode

This mode is shown in Figure 2. The sequence for the mission is:

1. The Saturn V payloads necessary to assemble the spacecraft are launched into earth orbit. The spacecraft is assembled and checked out and the spacecraft is launched into the trans-Mars trajectory.
2. The spacecraft is then deployed to its artificial-gravity configuration and spun up to produce the artificial gravity for the outbound portion of the flight which will cover approximately 120 days. The crew then settles into its normal routine wherein the activity is divided between the operation of the spacecraft, the conduct of scientific experiments, and the collection of engineering information. Course-correction maneuvers are executed as required.
3. At about five to ten days out from Mars, the vehicle is brought back to its entry configuration and final entry-corridor corrections are made.
4. The vehicle enters the Mars atmosphere at velocity of about 25,000 ft/sec, which is typical of missions for the early 1970's. Velocities up to 35,000 ft/sec are typical of missions occurring in the late 1970's. The spacecraft enters the Mars atmosphere and decelerates to orbital velocity, which is about 11,000 ft/sec. The deceleration maneuver is adjusted such that an elliptical orbit is established which has an apogee near the desired circular-orbit altitude from which landings will be conducted.

5. After the desired orbit is established, the Mars-excursion module is checked out, landing sites are selected, landing operations planned, and the Mars-excursion module crew enters the MEM. The MEM is then decelerated from the Mars orbit and descends aerodynamically to the surface.
6. The crew stays on the Mars surface from 10 to 40 days conducting various scientific activities and exploring the surface in the vicinity of the MEM.
7. When the exploration is complete, the launch stage of the MEM is prepared for launch and is injected into a rendezvous trajectory with the earth-return spacecraft.
8. Rendezvous is accomplished in the Mars orbit. The crew and scientific data are transferred to the spacecraft which then prepares for the return journey.
9. The spacecraft is injected into the trans-earth trajectory and the return-trip operations are similar to those conducted on the outbound trip.
10. As the earth is approached, the crew makes the final terminal corrections, enters the earth-reentry module, checks it out, accomplishes the reentry, and completes the mission.

Propulsive-Braking Mode

The propulsive-braking mode is similar to the aerodynamic-braking mode just described except that a propulsion module is used to decelerate the vehicle into a Mars orbit, rather than using the Mars atmosphere for deceleration. Except for this deceleration into the Mars orbit, other phases of the mission are similar. The propulsive-braking scheme has the advantage of considerable flexibility in arrangement of the configuration. There are no requirements that the spacecraft have any heat protection or any special aerodynamic configuration for atmospheric deceleration to a Mars orbit. Therefore, freedom is allowed in the manner in which provisions will be made for packaging the MEM and for producing artificial gravity.

Mission Mode Comparison

Table I is a weight summary for the atmospheric-braking-mode spacecraft in earth orbit. These weights are based on a 40-day launch window and total 1.47 million pounds. It appears operationally feasible to design for a contingency time in earth orbit, therefore eliminating the 40-day launch window requirement. With a one-day launch window, the total weight in earth orbit is reduced to 1.2 million pounds.

Figure 3 shows a comparison of the various modes for accomplishing the Mars mission. It can be seen from the weight requirements for the flyby-rendezvous mode and the atmospheric-braking mode with a one-day launch window that the atmospheric-braking mode requires only slightly more weight than the flyby-rendezvous mode. The all-chemical propulsive-brake into Mars orbit requires a weight in earth orbit which is greater than the atmospheric-braking modes by a factor of about 3. If nuclear propulsion is used for all propulsive phases of the mission, departing earth orbit, braking at Mars and departing Mars, the total weight in earth orbit is reduced to about 1.5 million pounds.

Figure 3 indicates that an all-chemical Mars mission system is feasible, using Saturn V as a basic launch vehicle since rendezvous of six payloads should be feasible at this time. It also demonstrates the considerable advantage that might be expected if nuclear propulsion is utilized for the earth-departure maneuver of the atmospheric-braking mode. The use of nuclear propulsion for departure at Mars would reduce the total weight in earth orbit; however, the liquid hydrogen propellant for nuclear rockets has a very low density (4.2 lbs per cubic foot) and would require large tanks and therefore large weights for heat-protection material and insulation. The gain for using nuclear propulsion departing the Mars orbit would be relatively small.

MARS EXCURSION MODULE DESIGN CONSIDERATIONS

Mission modes have been described which could result in direct entry of a Mars-excursion module from orbital velocities of 11,000 ft/sec to hyperbolic velocities of 35,000 ft/sec. Detailed considerations of entry from Mars orbit are given in this section, with emphasis on the effects of atmospheric extremes and vehicle configuration extremes. Although the higher velocities will have considerable effect on heat-protection-system requirements, it is believed the operational problems and system design problems for terminal flight, landing and launch are similar.

Figure 4 is a schematic of a ballistic Mars-excursion module. The cylindrical cone-tipped vehicle in the center is the launch and rendezvous vehicle. The accommodations and the crew compartment are designed for four men or two men plus 800 lbs of scientific data, Mars surface samples and various specimens that one might desire to return from the Mars surface. Two compartments are constructed on either side of the launch vehicle on the basic heat-shield structure. One of these compartments is the scientific station and the other compartment is the living area for the crew during the 10- to 40-day stay-time on the Mars surface. Sectors of the heat shield are deployed on shock absorbers to act as landing gear. The heat shield provides an inherent emergency capability if the vehicle lands on soil that has very low bearing strength. The basic deceleration is by atmospheric drag, using the spherical heat shield as the drag device. Terminal deceleration is accomplished with parachutes. Rocket engines provide hover for final touchdown at specific site.

Table II is a detailed weight summary of the major systems and elements of a typical Mars-excursion module.

Figure 5 shows an arrangement of a lifting-body shape typical of some shapes being investigated by the Langley Research Center. This particular arrangement gives a high ballistic coefficient to the vehicle. The equivalent wing loading is such that the equilibrium-glide velocity of the vehicle is supersonic. The low terminal descent requirement of the vehicle is most economically accomplished with parachutes unless engines can be developed which utilize the Mars atmosphere, e.g., the air turbo-rockets. With the development of this type of engine, the vehicle could be decelerated and a hover and landing maneuver accomplished at relatively small expenditures in fuel weight. The weight of the engine systems is probably not much greater than the landing system for the ballistic vehicle if we consider the landing system to include the hover rockets, their fuel as well as the parachutes and their deployment systems. The landing site flexibility and maneuver flexibility of the lifting-body shapes makes them attractive when considering later missions to the Mars surface.

To obtain a better feel for the problems associated with designing one vehicle to accommodate the extremes of atmospheric density presently proposed, a parametric study on two extremes in configurations was chosen. The atmospheric extremes used for this study were taken from Speigel[1], as nearly all predictions of Mars atmosphere fall within these limits. These predicted density extremes are shown in Figure 6.

The effects of atmospheric extremes on the orbital retro requirement for direct entry are presented in Figure 7. The ordinate is the relative velocity at an arbitrarily-chosen altitude, while the abscissa is the direction of application of a retro velocity. The parameter is the incremental velocity. The boundaries that define direct entry for the upper and lower density extremes are shown. These curves are based on zero L/D entry. A body using negative lift would require less V for deceleration, but the reduction is not significant. The difference in atmospheric extremes represents approximately 70 ft/sec in retro velocity requirements.

The entry load factors and ranges are presented in Figure 8 for each configuration entering with zero and positive L/D, and show the effect of atmospheric density extremes. The use of lift has a more significant effect on load factor than density extremes, even for the low-lift-ballistic shape. In any case, the loads are less than two earth g's. The high-lift MEM can easily correct for guidance errors or range errors due to atmospheric differences, whereas the low lift of the ballistic shape does not provide this capability. However, the atmosphere should be well determined from the orbiting spacecraft prior to entry from orbit and the general landing area of interest selected. With the atmosphere determined from the orbiting spacecraft, landing in the chosen area can be accomplished based on present guidance system errors and an L/D = .25.

The stagnation-point convective-heating rates for the two vehicles and atmospheric extremes are shown in Figure 9. The ballistic shape represents relatively modest heat-protection requirements, whereas those of the lifting vehicle are more severe. However, even the lifting vehicle can probably be designed with a radiation-cooled heat-protection system.

Figure 10 shows the effects of atmospheric density extremes on the terminal flight conditions at 50,000 feet for the lifting vehicle and the ballistic shape. The ballistic number $\frac{m}{C_D A}$ for the lifting vehicle is approximately 20, where the $\frac{m}{C_D A}$ for the ballistic shape is approximately 1.5 subsonicly. The ballistic vehicle will reach subsonic terminal conditions which are amenable to large-size parachute deployment. Actually, the diameter of the heat shield was chosen to provide subsonic terminal conditions. The lifting body, however, may have terminal velocities at from 1,400 ft/sec to 2,300 ft/sec, depending on atmospheric extremes, and requiring retro propulsion of from 500 ft/sec to 1,400 ft/sec to provide for the deployment of large subsonic parachutes. If, however, by increasing the vehicle size the loading can be reduced in half, which is a sizable weight penalty, retro propulsion of from 100 ft/sec to 750 ft/sec will be required. Supersonic drogues appear feasible only for relatively small velocity differentials due to size requirements.

Figure 11 shows the ratio of retro braking-propellant mass to total vehicle mass for various velocity increments as a function of varying specific impulse. For a specific impulse of 350 sec, the braking velocity increments indicated from the previous slide, propellant mass from one to eleven per cent of the total vehicle mass will be required. The added weight in earth orbit is about three times the weight of added retro propellant.

Figure 12 indicates the requirements for a parachute descent system. A parachute system would serve three purposes. It would reduce considerably landing retro-rocket propulsion requirements. It would provide time for surveillance of the intended landing area from close in to further verify landing feasibility. It would provide time for checkout of the retro landing propulsion system and checkout of the surface launch propulsion in the event an abort of the mission is required. An altitude of 50,000 feet was chosen as both vehicles could be assured of reaching subsonic terminal conditions; ballistic shape through natural drag and lifting vehicle through propulsive braking. Assuming five minutes are required, it can be seen that an $\frac{m}{C_D A}$ of 0.1 or a velocity at 50,000 feet of 135 ft/sec is required for the lower density extreme. This would require a single chute 175 feet in diameter. To provide redundancy and good pendulum stability, a cluster is more desirable. Assuming a three-chute cluster, where two chutes are required to provide the desired descent rate, the single chute diameter is 125 feet. If all three chutes deploy, an additional minute of descent time is obtained.

The time required under parachute descent is ill-defined and will require extended effort to fix a realistic value. The parachute size may be reduced by deployment at higher altitudes, but this will result in higher terminal velocities at the surface. Therefore, higher retro landing propulsion system weight, by approximately one per cent of the landing vehicle gross weight for every 100 ft/sec of retro velocity, will be required. Minimum weight combinations have not been determined.

Figure 13 shows the total characteristic velocity requirement for launch and rendezvous with the spacecraft in orbit. No advantage occurs for injection altitudes below about 450,000 ft, since below this altitude gravitational-loss reduction due to low injection is more than offset by high aerodynamic drag losses.

CONCLUSIONS

The conclusions drawn from this study are:

1. The presently predicted extremes of the Mars atmosphere present no serious problem of design for a vehicle capable of operation at either extreme of the prediction.
2. Details of operational requirements and mission objectives will be controlling factors in the choice of a MEM configuration rather than special requirements of the atmospheric-braking maneuver.
3. The high-drag, ballistic-type MEM vehicle is the simplest and lightest system.
4. The high-L/D lifting body can accommodate errors in the prediction of the Mars atmosphere and provides the option of landing sites out of the initial orbit plane. This flexibility requires weight increases to increase lifting surface or for added propulsion during the landing maneuver.

REFERENCE

1. Joseph M. Spiegel, AFIAS, California Institute of Technology, "Effects of Mars Atmospheric Uncertainties on Entry Vehicle Design," Aerospace Engineering, vol. 21, No. 12, Dec. 1962, p. 62.

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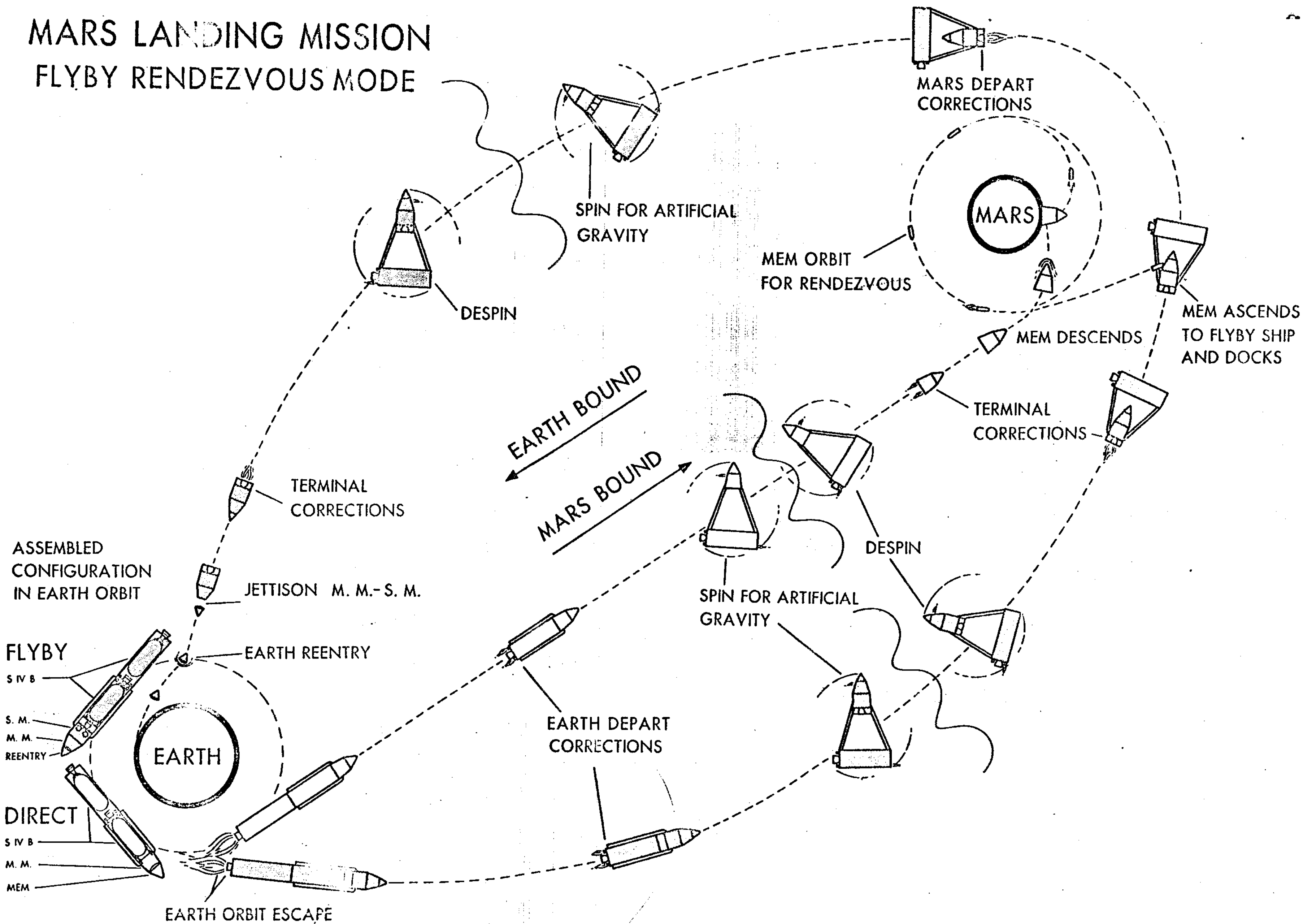
MARS LANDING MISSION
FLYBY RENDEZVOUS MODE



NASA-MSC MARS MISSION SYMPOSIUM D M HAMMOCK 18 MAY 63 S-132-83

Fig 1

NASA
S-63-6205

MARS LANDING MISSION
ATMOSPHERIC BRAKING MODE

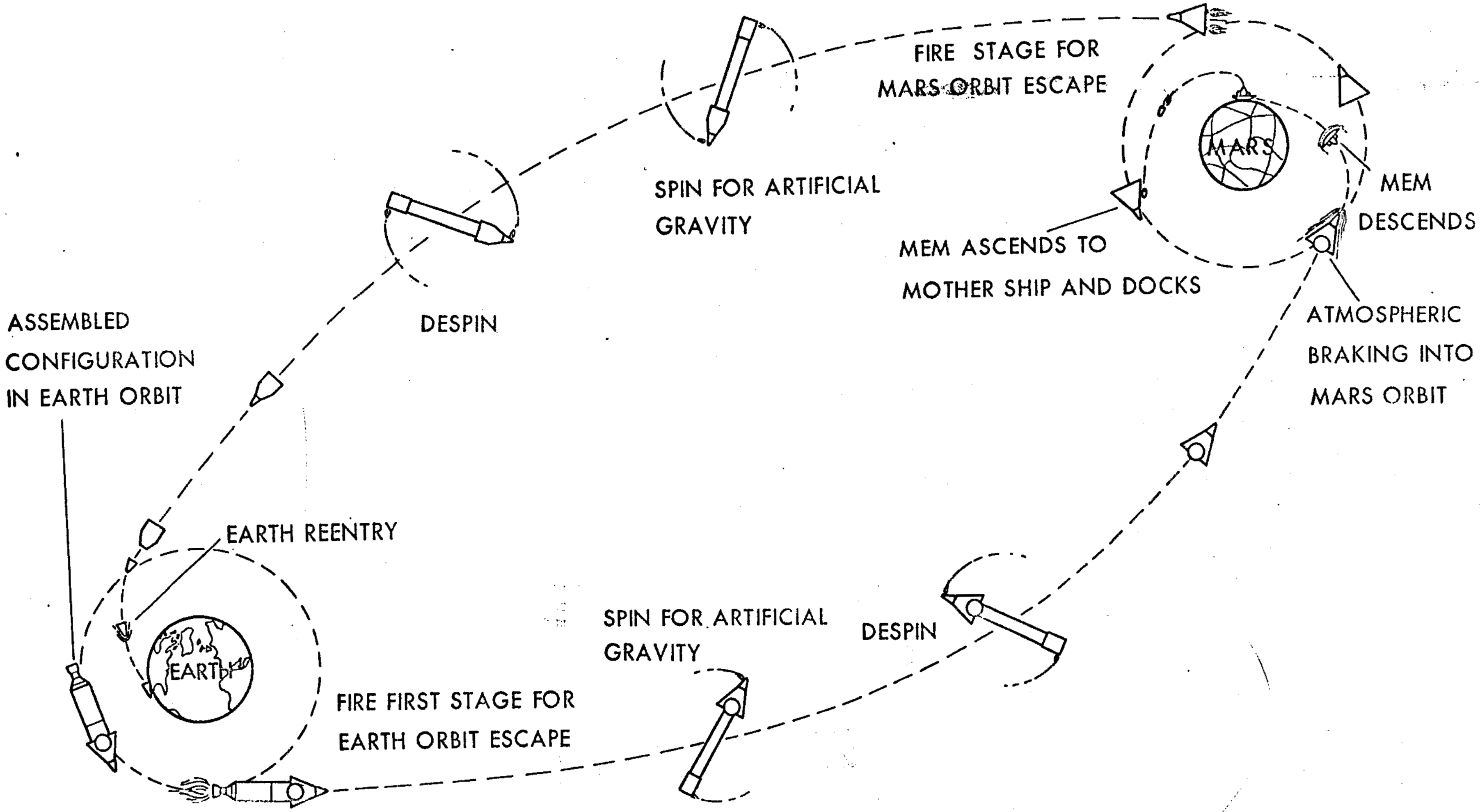


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FIG 2

NASA
S-63-7902

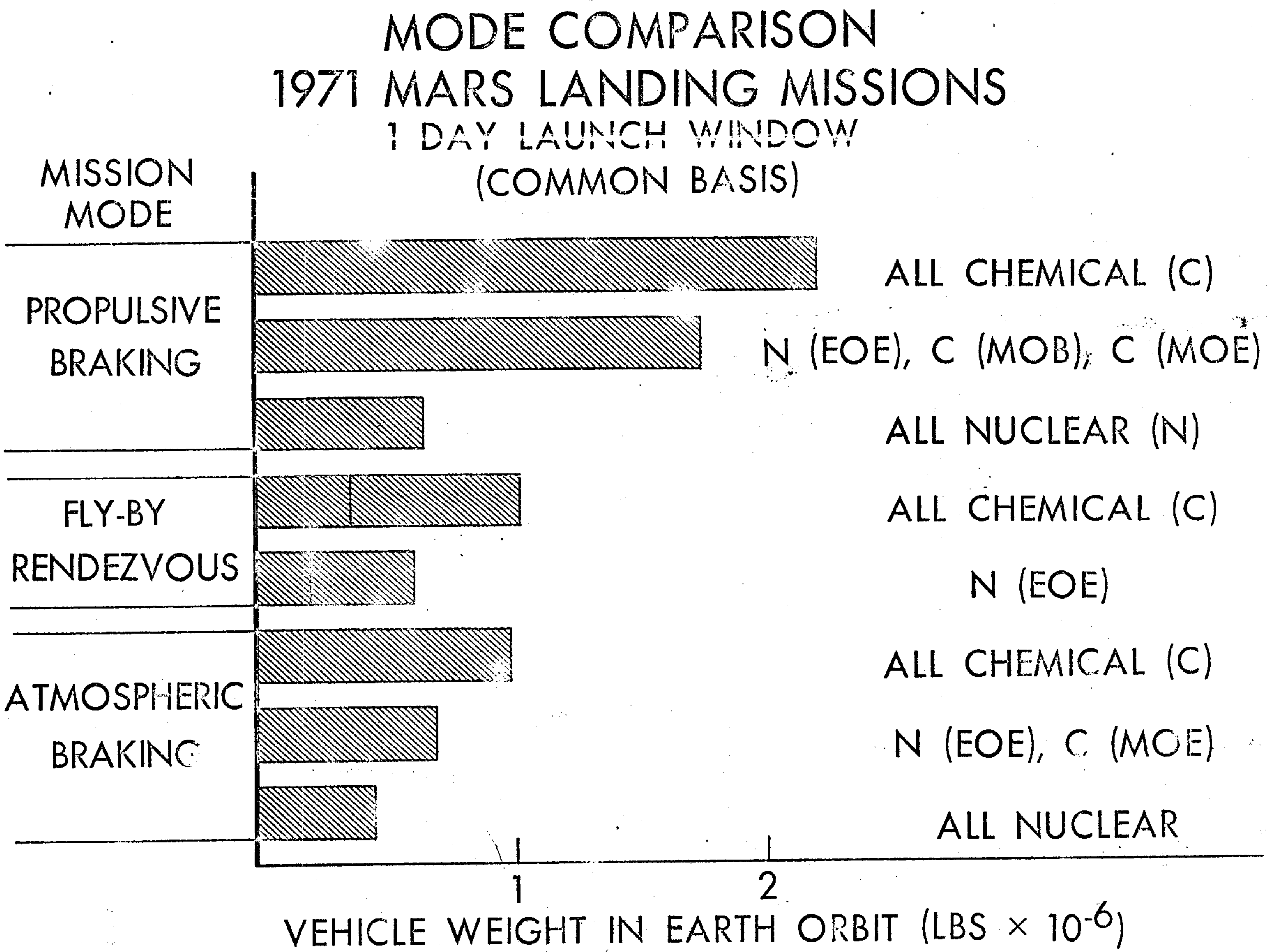

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FIG 3

NASA
S-63-6203

MARS ENTRY HIGH DRAG

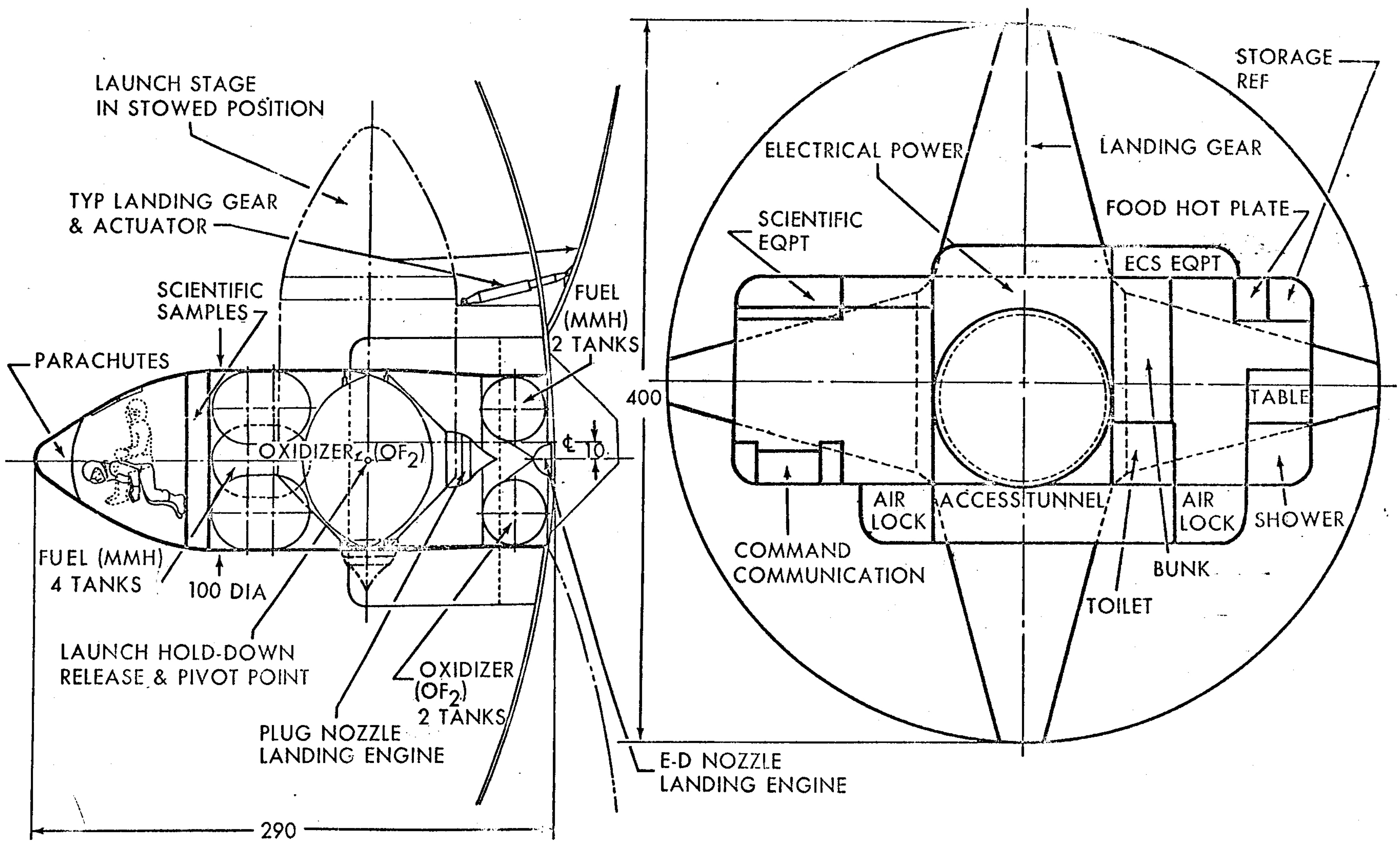

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FIG 4

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S-63-5462

MARS EXCURSION MODULE

LANGLEY HL-10 SHAPE

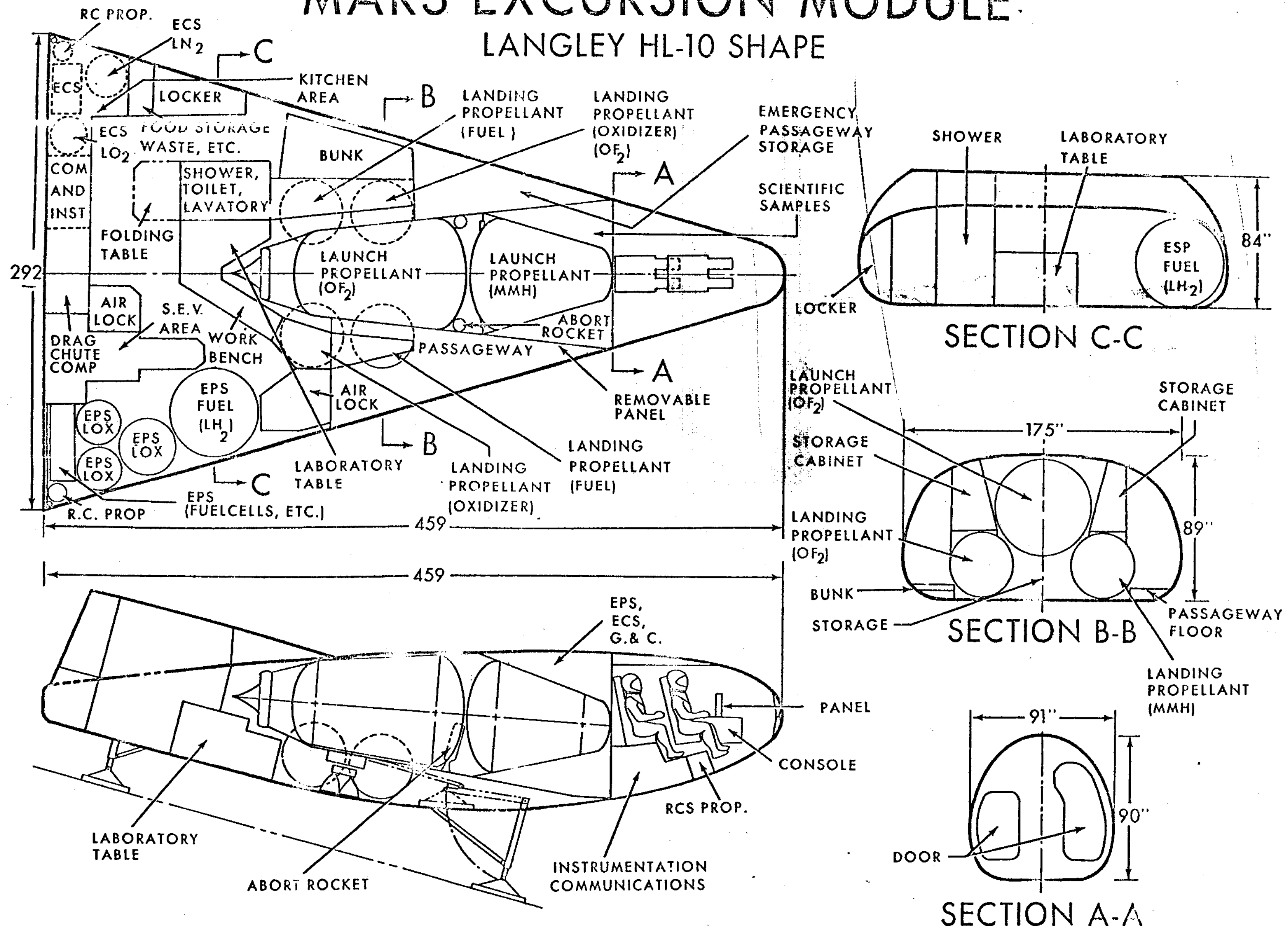

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FIG 5

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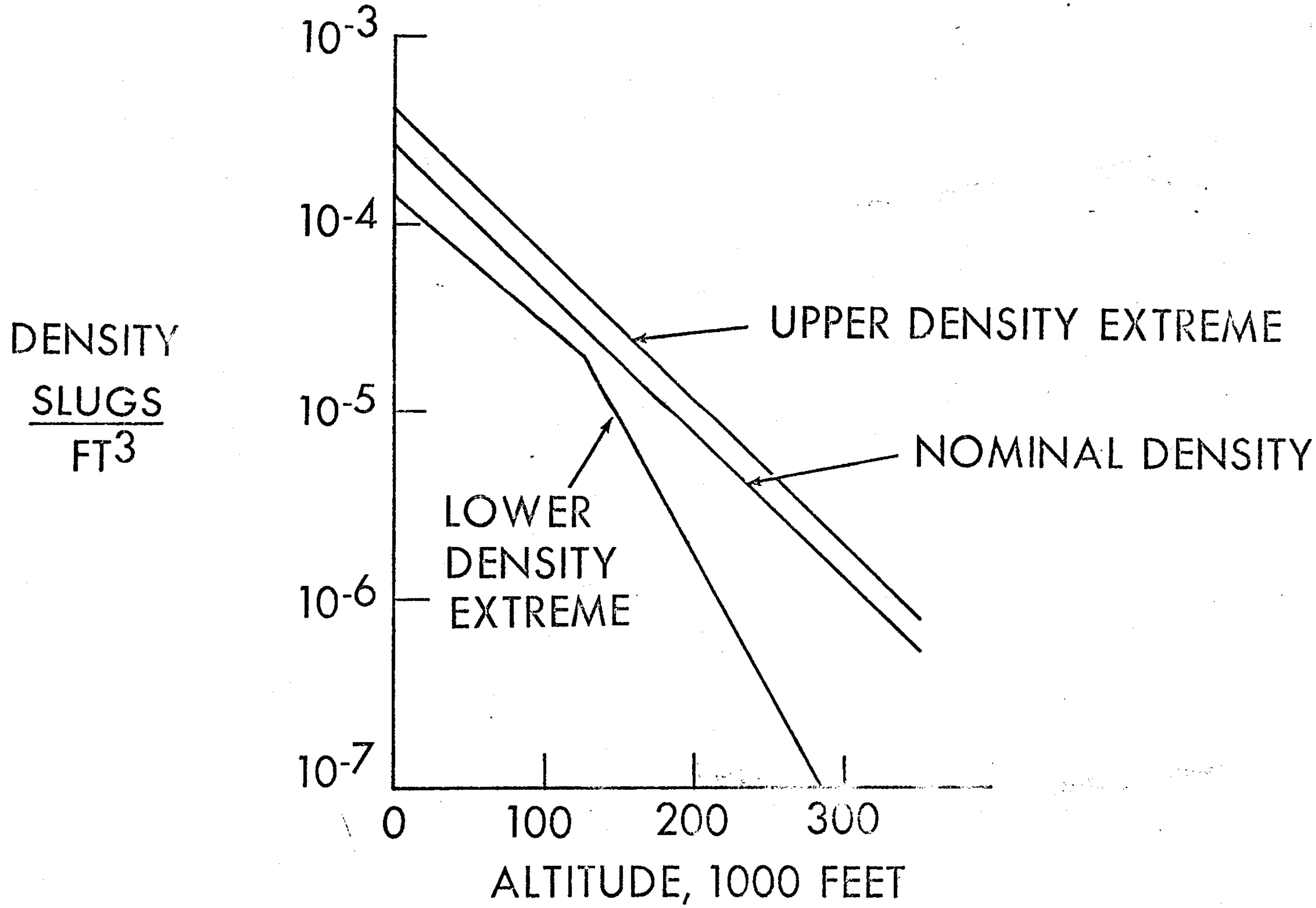
S-63-7906

MARTIAN ATMOSPHERIC EXTREMES



NASA-MSC MARS MISSION SYMPOSIUM D M HAMMOCK 27 MAY 63S-132-94

FIG 6

NASA
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MARS ENTRY, ORBITAL RETRO

FROM 300 NM CIRCULAR ORBIT

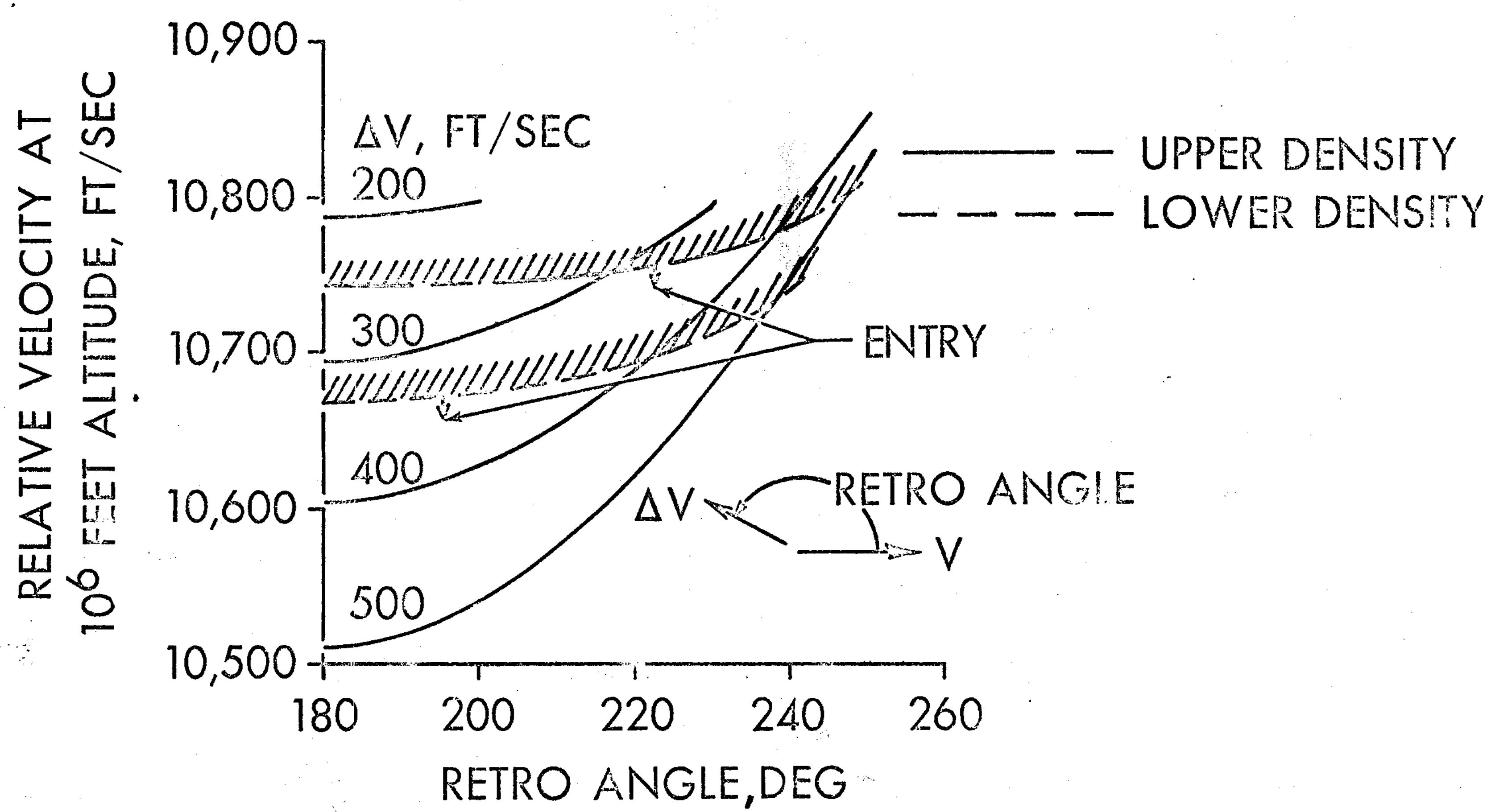

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FIG 7

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S-63-7899

MARS ENTRY LOAD FACTORS

400 FT/SEC RETRO FROM 300 NM ORBIT

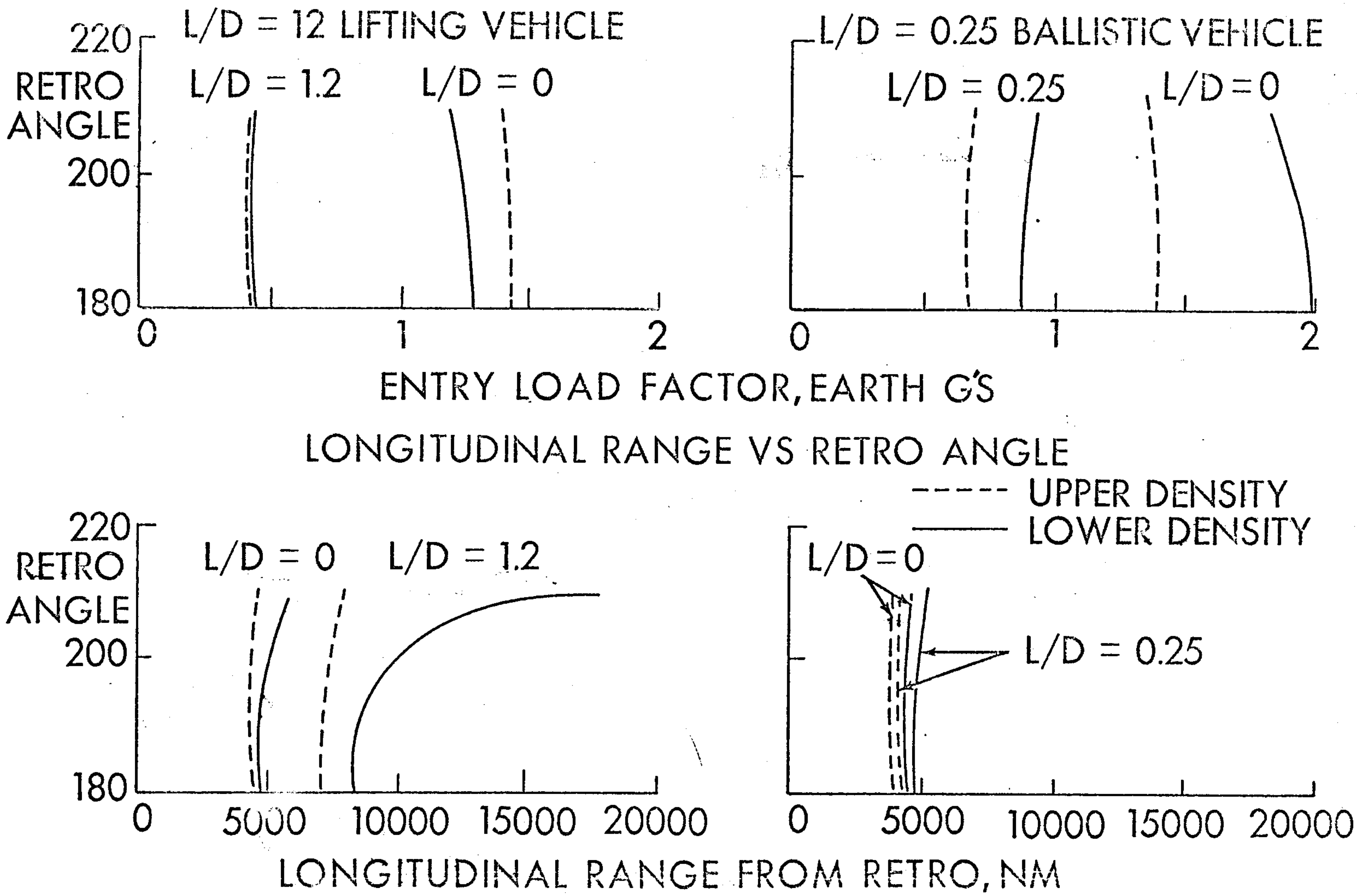

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Fig 8

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S-63-7905

MARS ENTRY HEATING

400 FT/SEC RETRO FROM 300 NM ORBIT

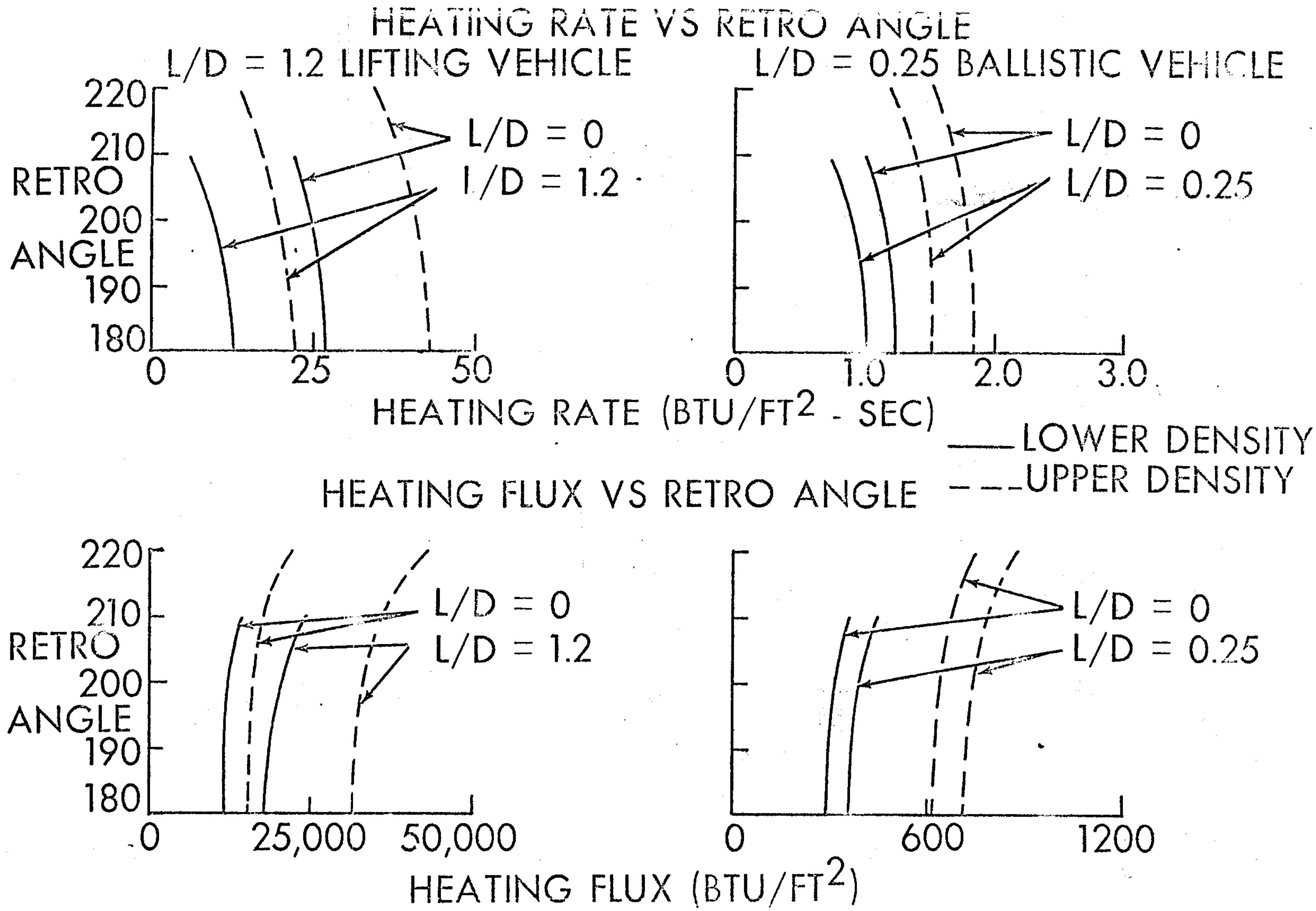

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FIG 9

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MARS LANDING, TERMINAL FLIGHT

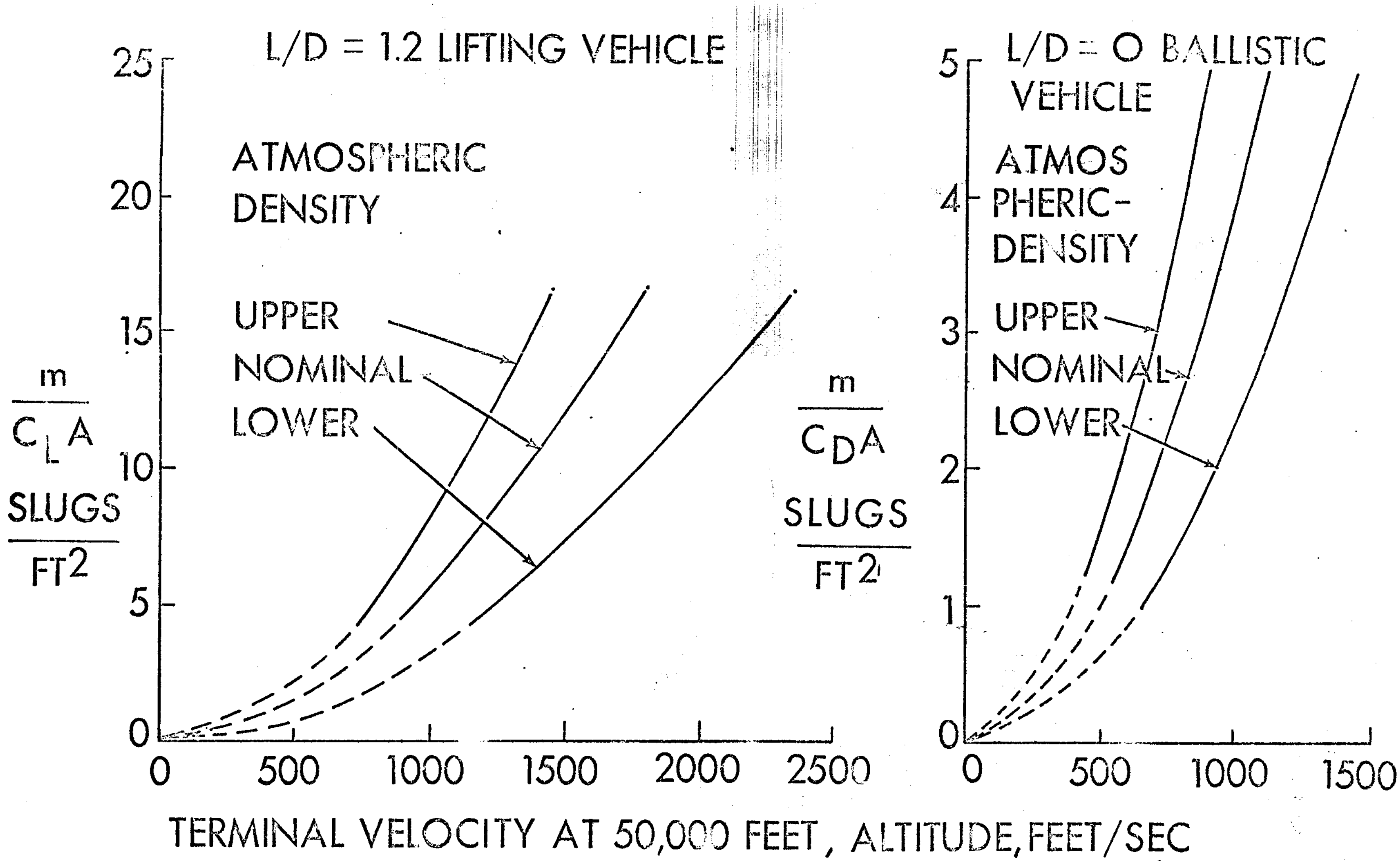

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FIG 10

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MARS LANDING RETRO BREAKING

PROPELLANT WEIGHT — LANDING WEIGHT RATIO
VS BRAKING VELOCITY

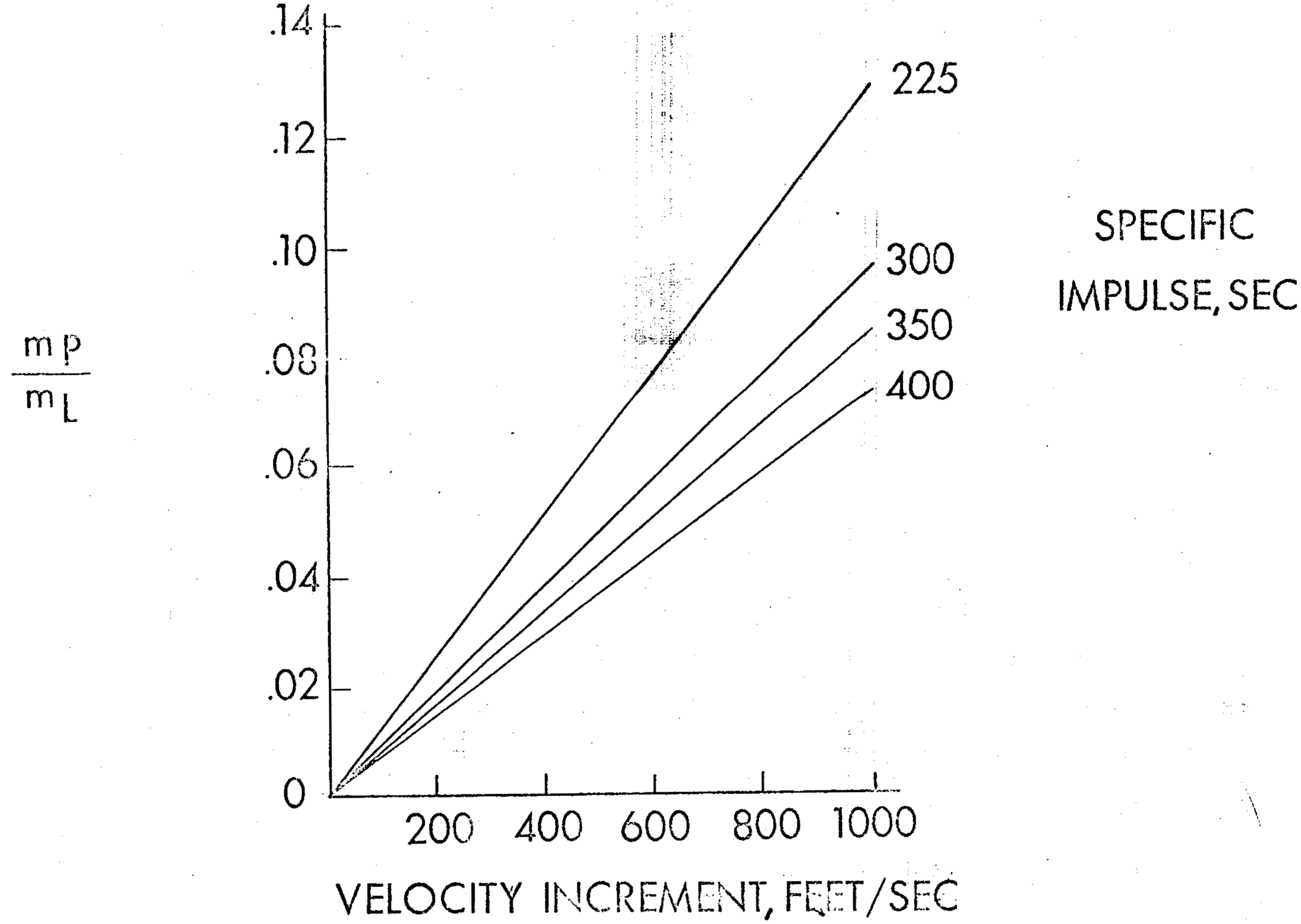

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FIG 11

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S-63-6154

MARS LANDING, PARACHUTE DESCENT

VEHICLE MASS = 1850 SLUGS

SUBSONIC PARACHUTE DRAG COEFFICIENT = 0.72

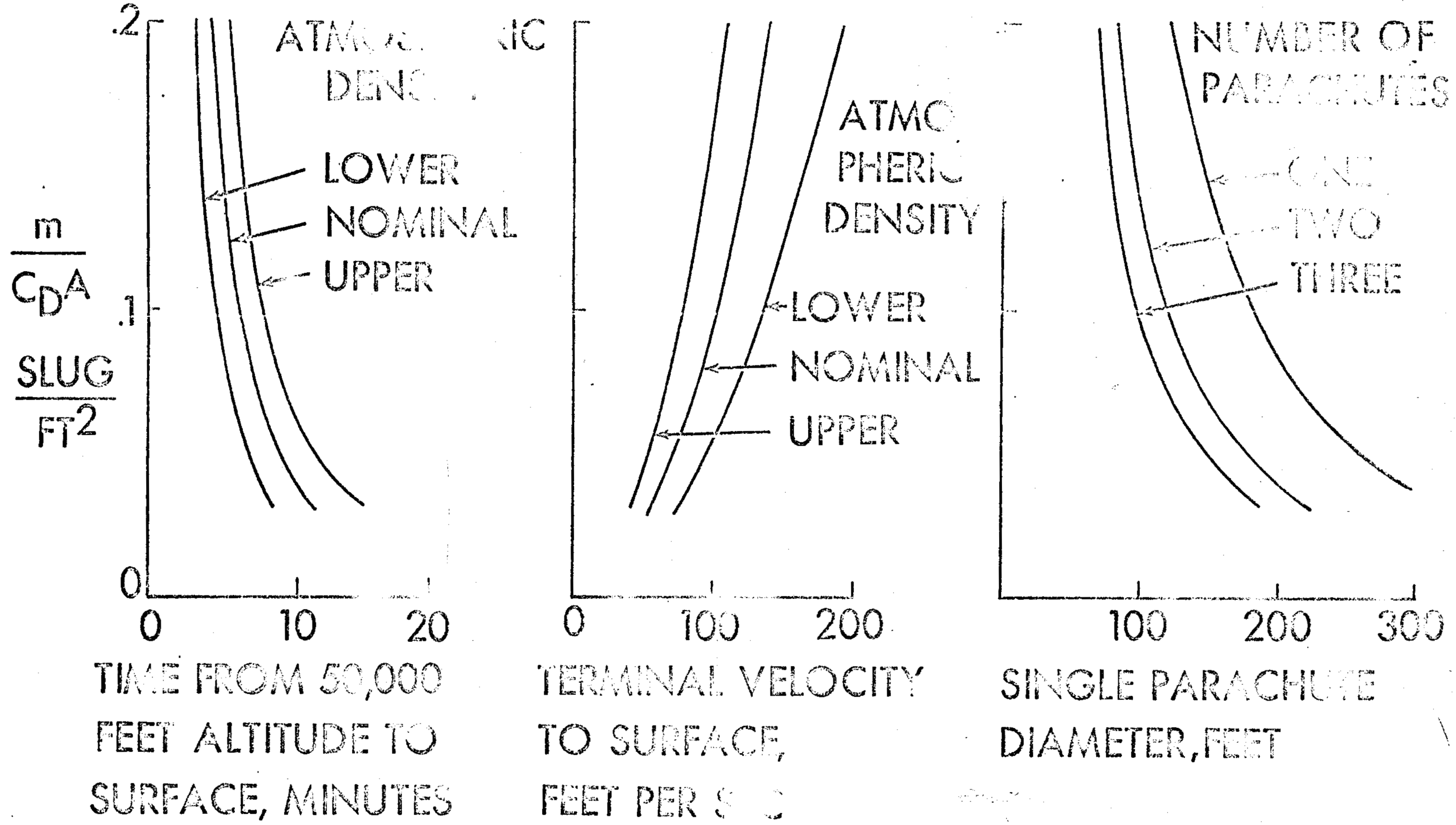

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FIG 12

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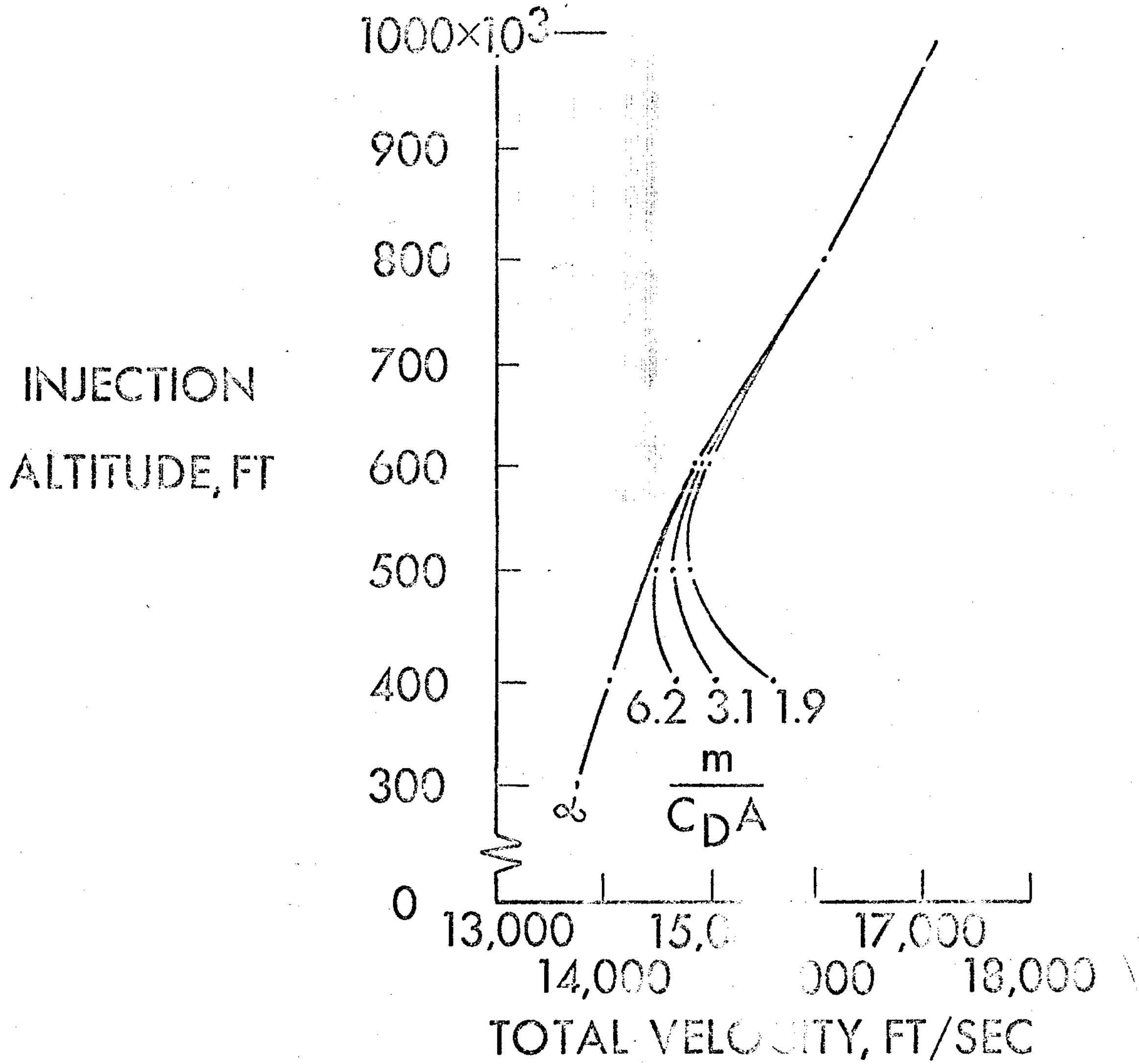
S-63-6151

MARS LAUNCH, ORBITAL ALTITUDE [illegible] NM



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VEHICLE WEIGHT SUMMARY

1971 - DIRECT - ATMOSPHERIC BRAKING AT MARS

LH_2/LO_2 OF_2/MMH

ITEM	ΔV	I_{SP}	LBS
EARTH REENTRY MODULE			17,400
MISSION MODULE			38,200
RETURN MIDCOURSE STAGE WT PROP 10,616 WT DRY 1,873	1,500	320	12,489
MARS ORBIT ESCAPE STAGE 2 WT PROP 63,102 WT DRY 10,272	7,362.5	386	73,374
MARS ORBIT ESCAPE STAGE 1 WT PROP 126,507 WT DRY 15,636	7,362.5	386	142,143
M O E PROP MODULE HEAT PROTECTION	---	---	7,000
MARS EXCURSION MODULE	---	---	52,500
OUTBOUND MIDCOURSE AND ORBIT CIRCULARIZATION WT PROP 61,652 WT DRY 10,036	2,000	386	71,688
EARTH ORBIT ESCAPE STAGE 2 WT PROP 258,934 WT DRY 19,490	6,475	430	278,424
EARTH ORBIT ESCAPE STAGE 1 WT PROP 432,739 WT DRY 32,572	6,475	430	465,311
			1,158,532

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TAB I

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MARS EXCURSION MODULE WEIGHT SUMMARY

2 MEN DESCENT - 40 DAY STAY

	LANDER	LAUNCH
ENVIRONMENTAL CONTROL SYSTEMS	1,250	225
FOOD AND CONTAINERS	120	10
COMMUNICATIONS	425	345
INSTRUMENTATION	500	200
ELECTRICAL POWER	1,200	300
CREW SYSTEMS	512	215
REACTION CONTROL	600	500
STRUCTURE (INCL HEAT PROT & LANDING GEAR)	7,000	3,000
ENGINE	500	500
TANKS, LINES, FITTINGS AND INSULATION	120	220
HELIUM PRESSURIZATION	200	200
SCIENTIFIC EQUIPMENT	2,000	
SCIENTIFIC SAMPLES		800
LANDING DRAG DEVICE	600	
LAUNCH INITIATION ROCKETS	635	
CONTINGENCY 10%	1,354	630
DRY WEIGHT TOTAL		7,735 (WITH SCIENTIFIC LOAD) 6,935 (W/O SCIENTIFIC LOAD)
PROPELLANT	4,300	24,250

SUMMARY - MEM

LAUNCH STAGE	
WEIGHT LEAVING ORBIT	31,147
WEIGHT @ LIFT-OFF	31,947
LANDING STAGE	
WEIGHT LEAVING ORBIT	21,354
VEHICLE WEIGHT IN ORBIT	52,500

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TAB II