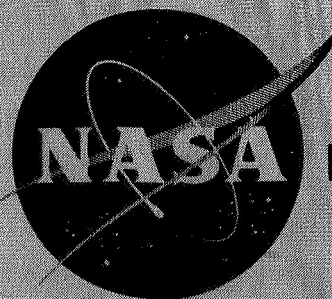


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SOFT LANDER PART III

Parametric Studies Continued

Prepared by

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY
EASTERN DIVISION

Saint Louis, Missouri 63166 (314) 232-0232

for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D.C. • SEPTEMBER 1968

SOFT LANDER PART III

**Mars Soft Lander Capsule Study (Entry From Orbit) -
Parametric Studies Continued**

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Prepared under Contract No. NAS 1-7977 by
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EASTERN DIVISION
Saint Louis, Missouri

for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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3.2.6 THERMAL CONTROL - The thermal control system provides survivable thermal environments for capsule equipment throughout all mission phases. It must be sufficiently flexible to accommodate the mission and environment uncertainties, and be compatible with known external environments, with configuration changes during the mission, and with the operating characteristics of other capsule systems.

The thermal control parametric studies are discussed in two parts: (1) the flight phase, including prelaunch and launch and (2) the post-landed phase. The primary results of the study are definition of thermal control configurations, performance, and sizing data for application to the point design configurations.

3.2.6.1 Flight Phase. - Thermal control during the flight phase provides the required temperature levels within the capsule during the prelaunch period and during the journey from Earth to Mars. Changes in vehicle configuration, orientation, and solar distance during the flight cause varying solar heat input. These are the major thermal control considerations. Other considerations include sterilization compatibility, on-pad checkout, and the thermal effects of the launch transient. Previous studies (NASA CR-89686) have indicated that once sterilization compatibility has been established these other considerations have little effect on design selection. Temperature levels and electrical heater requirements allow selections of recommended approaches and definition of parametric performance and weight effects, as discussed in the following paragraphs.

3.2.6.1.1 Design requirements: The thermal control design must meet the temperature levels required by the on-board equipment and must also be lightweight and impose few additional mission constraints. The design requirements are discussed in terms of required equipment temperature levels, including prelaunch and launch, and external environment.

Equipment Temperature Limits - The required equipment temperature limits (Table 3.2.6-1) are applicable for the several periods throughout the flight. During this time, most equipment is inoperative. The large amount of insulation necessary to protect the lander after touchdown will maintain the lander experiment and supporting subsystems at nearly the same temperature during the flight phase, i.e., above 20°F, the minimum inoperative battery temperature.

TABLE 3.2.6-1
THERMAL CONTROL DESIGN PARAMETERS

PARAMETERS	RANGE OF VALUES	VALUES USED IN ANALYSIS		COMMENTS
		LANDED	FLIGHT	
EQUIPMENT TEMPERATURE LIMITS SCIENCE SENSORS, °F ELECTRONICS, °F TRANSMITTERS, INCLUDING RADAR, °F OTHER ELECTRONICS, °F BATTERIES, °F		AMBIENT 0 TO 100 0 TO 160 0 TO 100 50 TO 125	AMBIENT 0 TO 100 0 TO 160 0 TO 100 20 TO 70	ADDITIONAL FLIGHT PHASE LIMITS: DEORBIT MOTOR, °F 20 TO 40 MINIMUM LANDING ROCKETS, °F 50 HYDRAZINE (MIN.) 30 BIROPELLANT (MIN.)
MARS AMBIENT ENVIRONMENT SOLAR FLUX (MARS), BTU/HR·FT ² ATMOSPHERIC PROPERTIES TEMPERATURE, °F PRESSURE, mb COMPOSITION WINDS, FPS TERRAIN PROPERTIES SOLAR ABSORPTANCE EMITTANCE TEMPERATURE, °F FEATURES LANDING LOCATION	160 TO 180 -190 TO 120 5 TO 20 VM ATMOSPHERES TO 500 0.65 TO 0.95 0.65 TO 1.0 -190 TO 120 FLAT TO 34° SLOPES NEAR EQUATORIAL	180 { CYCLIC DAY (-150 TO 120) CONTINUOUS COLD (-190) { CYCLIC DAY (-150 TO 120) CONTINUOUS COLD (-190) FLAT EQUATOR		NO ATMOSPHERIC ATTENUATION NO SOLAR HEAT INPUT
THERMAL CONTROL COMPONENT PERFORMANCE INSULATION THERMAL CONDUCTIVITY ON MARS (EFFECTIVE) BTU/HR·FT·°F DENSITY (EFFECTIVE), LB/FT ³ HEATER PERFORMANCE ISOTOPE, WATT/LB ELECTRICAL, WHR/LB PHASE CHANGE MATERIAL WITH PACKAGING - (HEAT OF FUSION), WHR/LB EXTERNAL THERMAL CONTROL COATING: SOLAR ABSORPTANCE EFFECTIVE EMITTANCE	.00625 TO .0375 2 TO 4 2.5 TO 10 40 15-25 WIDE RANGE OF MATERIALS AVAILABLE	.00625 TO .0375 4 5 40 20 .5 .9		INSULATION PERFORMANCE IN MARS ENVIRONMENT NOT WELL DEFINED. WEIGHT DEPENDENT ON SHIELDING BATTERY POWER DEPENDS ON CHOICE OF MATERIAL AND PACKAGING CONCEPT EXPECTED VALUES AFTER EXPO- SURE TO MARS AMBIENT ENVIRON- MENT - LONG TERM DEGRADATION EFFECTS MUST BE CONSIDERED.

The propulsion components must be maintained at temperatures that are high enough to avoid, (1) excessive thermal stress in the solid deorbit motor propellant, (2) performance degradation due to low propellant temperature, and (3) excessive cooling of equipment.

Thermal Environment - The environment for the capsule exterior is the natural thermal environment encountered during launch and in Earth orbit and deep space. Near Mars and in Mars orbit the altitude is sufficiently high that radiative heating from the planet to the capsule is small and can be safely ignored for preliminary studies.

The major environmental variable during the interplanetary cruise mission is the change in solar heating. Nominal values of the solar constant are as follows:

<u>Capsule Position</u>	<u>Solar Distance</u> (10^6 miles)	<u>Solar Constant</u> (Btu/hr-ft ²)
Near Earth	92.9	442
Near Mars	141.3	192

These heating rates are expressed in terms of equilibrium temperatures for surfaces facing the sun in Figure 3.2.6-1, showing that for typical conditions the capsule surface temperature will decrease by about 100°F during cruise. A factor tending to reduce the temperature change is stability of thermal control coatings during the extended deep space exposure. If the coatings degrade slowly, Figure 3.2.6-1 shows that increased solar absorptance/infrared emittance (a_s/e) after launch will reduce the cool-down during the cruise from Earth to Mars.

The guiding design requirement for accommodating the change in external thermal environment is to avoid heating the lander batteries above 70°F near Earth, and also maintain a minimum temperature of 20°F near Mars. During this period the capsule equipment is nominally inactive except for battery charging and equipment monitoring activities. Therefore, waste heat from equipment operation is not available, and electrical heaters must be used to maintain equipment temperature as the solar heat declines.

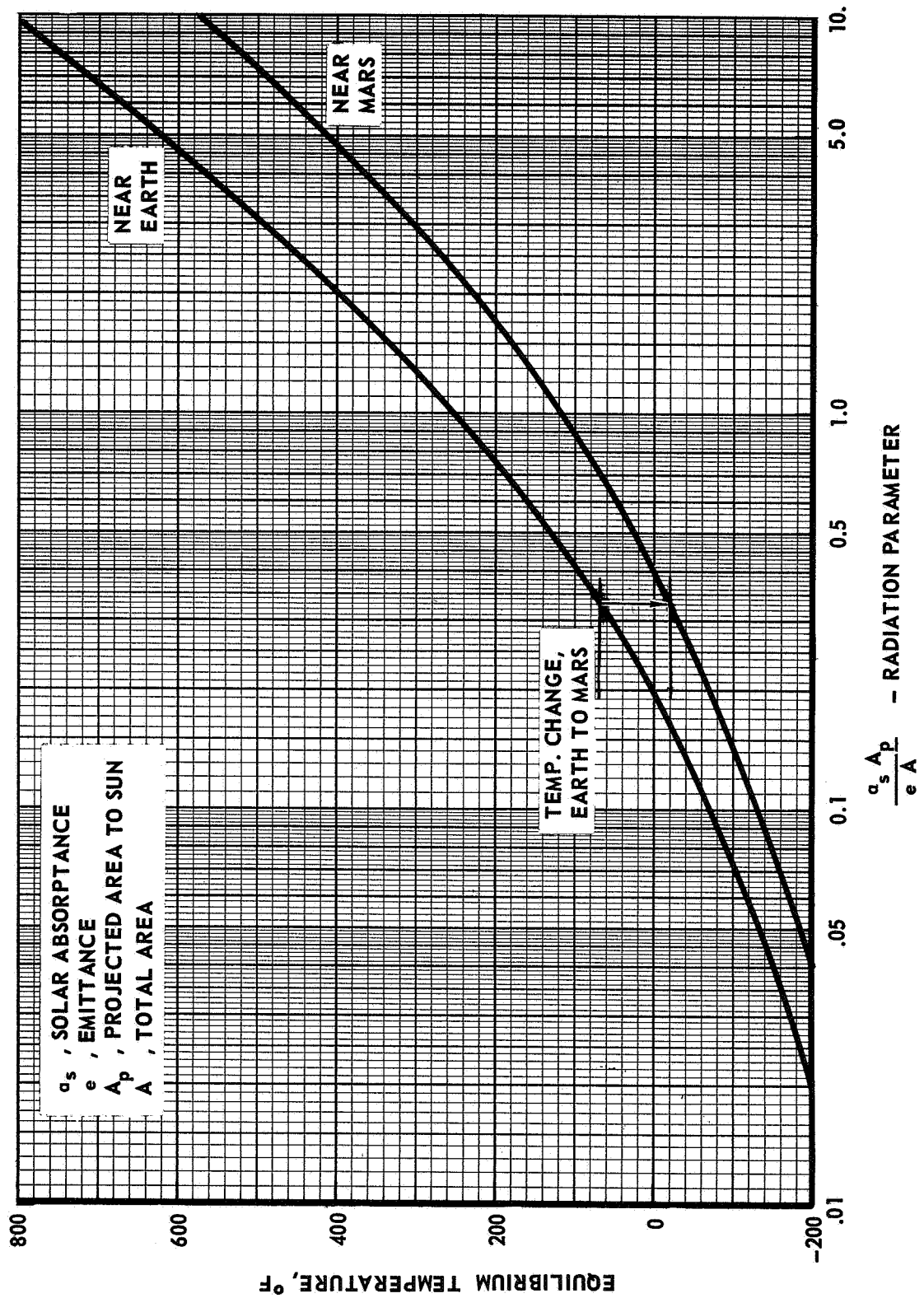


FIGURE 3.2.6-1

3.2.6.1.2 Candidate approaches: During interplanetary cruise and Mars orbit, three basic thermal control design concepts can be used for the capsule, depending on the relative size of the capsule and orbiter. These are: (1) full sun orientation of the capsule, (2) full shade orientation (behind the orbiter), and (3) partial shade orientation. The primary thermal control design objectives are to maintain battery temperature between 20 and 70°F, and avoid excessive cooling of the propulsion equipment, including propellant tankage, lines, and deorbit motor. For orbital descent, the orientation with respect to the sun is again important to avoid overheating or excessive cooling of the exposed capsule and equipment.

A sunlit orientation allows the capsule to be warmed by the sun, but at variable heating rates as the distance from the sun increases. If the orbiter, including its solar panels, is as large in diameter as the capsule, full shade orientation can be obtained, with the orbiter facing the sun and the capsule in the shadow facing deep space. This configuration is similar to that studied for the VOYAGER Phase B effort (NASA CR-89686). Smaller orbiters result in the partial shade configuration, where sunlight impinges on the capsule base outside or between the orbiter solar panels.

The capsule descends from orbit in an exposed configuration. The sterilization canister with its multilayer insulation blanket has been jettisoned, exposing the aeroshell and aft thermal curtain to the combined deep space/solar heating environment. The alternate approaches for thermal control during orbital descent are shown in Table 3.2.6-2 for typical configurations and solar orientation angles.

Several specialized techniques, including thermal control coatings and multilayer insulation blankets, have been examined.

Thermal Control Coatings - As shown in Figure 3.2.6-1, the full sun and partial sun orientations require space-stable thermal control coatings on the sterilization canister. The canister material would be provided with an optically black thermal coating on the inside (for internal heat distribution) and a gray paint on the outside areas facing the sun. The external paint could be one of several applicable materials, including silicone bonded zinc oxide, with lampblack filler to obtain the required α_s/e or silicone bonded

TABLE 3.2.6-2
SUMMARY OF THERMAL CONTROL ALTERNATIVES FOR ORBITAL DESCENT

HEAT SHIELD CONFIGURATION	DE-ORBIT SOLAR ANGLE REQUIREMENTS *	WEIGHT PENALTY LB/FT ² AEROSHELL	LANDING SITE CONSTRAINTS	CANISTER SEPARATION CAPABILITY	
				BEFORE ORBIT	BEFORE CAPSULE SEPARATION
BARE ABLATOR (-150°F)	0 - 50°	0	EARLY MORNING <25° FROM MORNING TERMINATOR BEFORE NOON		✓
BARE ABLATOR WITH SLOW ROLL	0 - 90°	0			✓
INSULATED ABLATOR	0 - 180°	0	NONE	✓	✓
GOLD TAPE OVER ABLATOR	0 - 180°	.071	NONE	MARGINAL	✓
LOW TEMP. ABLATOR (-300°F)	0 - 180°	<.04 (ADDED HEATER POWER)	NONE	✓	✓

*SOLAR ANGLE EQUALS ZERO FOR CAPSULE FACING SUN

aluminum. The lampblack filler with zinc oxide technique was used on the Gemini horizon sensor to obtain a stable a_s/e value of 0.5.

Multilayer Insulation Blankets - These blankets will be necessary for full and partial shade orientation to conserve heat loss from the capsule to deep space. The full shade orientation requires that the blankets completely surround the capsule. Trade-off studies of this configuration during the VOYAGER Phase B study effort (NASA CR-8986) indicated that the blanket should be installed outside the sterilization canister, and the canister retained during Mars orbit until just prior to separation. Evaluation of the ballistic coefficient and Mars orbital parameters indicate that the orbital lifetime of the jettisoned sterilization canister is sufficiently long to avoid violation of planetary quarantine requirements. Use of this configuration provides rapid outgassing of the blanket during launch to remove the interstitial gas, thus obtaining the high performance capability of multi-layer insulation.

A partial shade configuration would employ a multilayer blanket over the forward, shaded portion of the sterilization canister, to retain the solar heating absorbed by the canister base. Removal of the sterilization canister and blanket would be the same as previously described.

In both the full and partial-shade configurations, the multilayer blanket would probably be constructed of from 16 to 35 layers of vacuum-deposited aluminum on Mylar. An alternate material would be gold on Mylar or Kapton, which is more stable in any moisture environment encountered prior to launch.

The sheets would be either crinkled or dimpled to minimize contact between layers. Fabricated thickness of the blanket would be approximately .5 inches, including a protective cover sheet to avoid ground handling damage. Thermal performance has been tested for various blanket configurations under JPL Contract No. 951537. These tests yielded the following vacuum thermal conductance data:

<u>Configuration</u>	<u>Effective Thermal Conductivity</u> <u>10^{-5} Btu-ft/hr-ft²-°F</u>		
35 sheets wrinkled, aluminized 1/4 mil Mylar	3.2	to	5.7
16 sheets dimpled, aluminized 1/2 mil Mylar	10.5	to	13.1

35 sheets wrinkled, aluminized 1/4 mil Mylar (with joint and support post)	4.08	to	6.26
16 sheets dimpled, aluminized 1/2 mil Mylar (with support post and stitching)	13.5	to	27.6

These test results indicate that multilayer insulation blankets with effective thermal conductivities of less than 1×10^{-4} Btu-ft/hr-ft²-°F can be fabricated using wrinkled aluminized Mylar. Similar performance was experienced with gold on Mylar or Kapton. Although this test data shows higher values for the dimpled Mylar blankets, effective conductivities of less than 1×10^{-4} Btu-ft/hr-ft²-°F can be obtained by using more layers. Additional evaluation is required before the optimum material and configuration for the blanket can be selected. However, the test data indicates that an effective thermal conductivity of 1×10^{-4} Btu-ft/hr-ft²-°F is a realistic goal so this value was used in all calculations.

3.2.6.1.3 Analysis techniques: Detailed thermal analysis of the full shade configuration was performed on the VOYAGER configuration using a 44 node computer thermal analysis model programmed for the McDonnell General Heat Transfer Program. Data from this analysis has been ratioed down for the smaller, but similar, configurations of this study on the basis of constant heat loss per unit sterilization canister area. The radiation heat transfer configuration factors developed for the VOYAGER study were simplified for hand calculations of the partial shade orientation. In the full sun orientation, data was obtained assuming the capsule was at constant temperature throughout and that there was no heat loss to the orbiter since the orbiter solar panel will also be in the sun and facing the capsule.

3.2.6.1.4 Performance: The three candidate approaches for the interplanetary cruise and Mars orbit phase have been compared on the basis of performance. Performance during orbital descent has also been analyzed. The influence of isotope and RTG power sources are also considered.

Capsule Sunlit Orientation - Capsule temperature levels in this configuration are determined by selection of appropriate thermal control coatings for the sterilization canister. As the solar distance increases between Earth and Mars, the capsule temperature decreases as shown in Figure 3.2.6-2. Thus, a configuration using $a_s/e = .4$ would cool from 70°F capsule temperature near Earth to -30°F in the vicinity of Mars. This configuration would require little

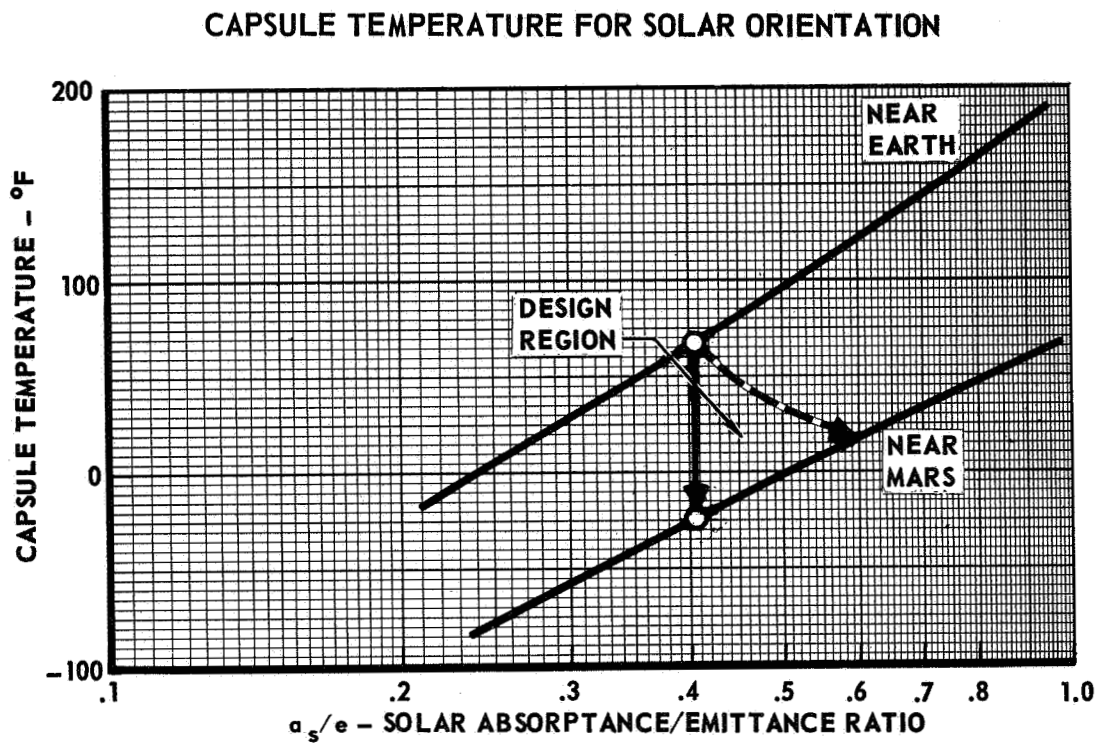


FIGURE 3.2.6-2

electrical heater power until capsule temperature decreases to 20°F. Thereafter electrical heater power would increase to a peak of about 0.13 watts/ft² of canister area near Mars. The thermal control weight requirement is estimated to be 0.150 lb/ft², including thermal coatings on the sterilization canister, aeroshell and capsule equipment, and equipment insulation.

Capsule Fully Shaded Orientation - Capsule equipment temperatures for this orientation are maintained within survival levels by electrical heaters with thermostatic control. Heater power is minimized by use of insulation on temperature-sensitive equipment, and a multilayer insulation blanket covering the entire sterilization canister. For a 1/2 inch thick blanket average heater power is estimated at 0.33 watts/ft² of sterilization canister. Heater power variation with capsule diameter is shown in Figure 3.2.6-3. Thermal control weight is estimated at 0.23 lb/ft² including multilayer insulation, aeroshell equipment insulation, thermal coatings, and electrical heaters.

Capsule Partially Shaded Orientation - This configuration has a weight between those for full sun and full shade orientation because it uses less equipment insulation than the fully shaded configuration, but requires a multilayer insulation blanket over the forward portion of the sterilization canister. Its advantage is in reducing interface constraints between the capsule and orbiter. With the capsule partially shaded, it receives enough solar heating to significantly reduce thermal control weight requirements over the fully shaded option, and does not interfere with capsule solar panels, attitude control sensors, or communication antennas.

Figure 3.2.6-4 shows temperature levels for near-Earth and near-Mars conditions of 70°F and -40°F respectively, using an a_s/e value of 1.4. These temperatures are similar to those for the full sun configuration and will require gradually increasing electrical heater power after the temperature drops below 20°F. Estimated peak heater power near Mars is 0.15 watts/ft². Thermal control weight for capsule equipment is estimated at 0.18 lb/ft² of sterilization canister, including multilayer and aeroshell-mounted equipment insulation, electrical heaters, and thermal coatings.

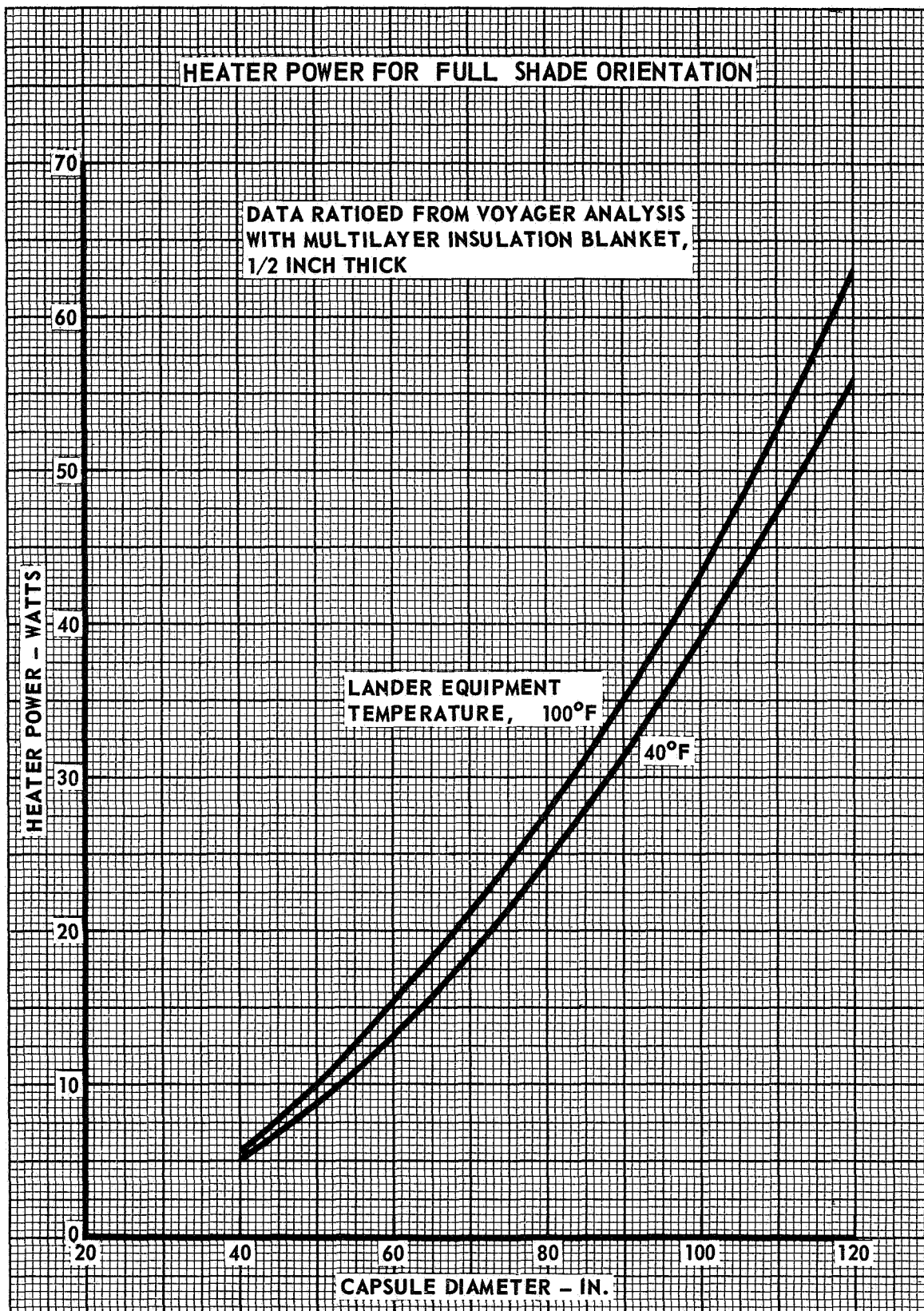


FIGURE 3.2.6-3

CAPSULE PERFORMANCE IN PARTIAL SHADE ORIENTATION INTERPLANETARY CRUISE AND MARS ORBIT

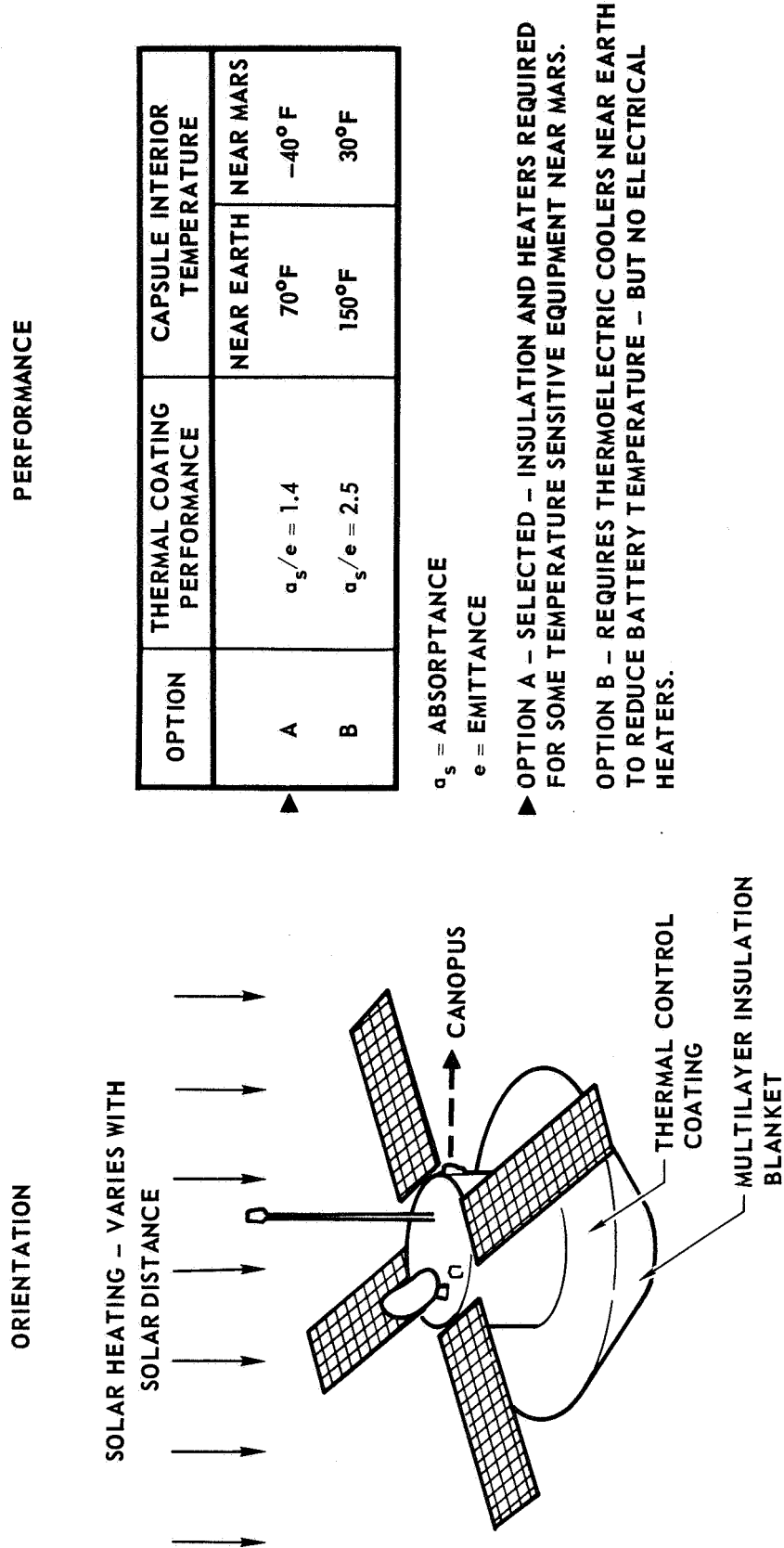


FIGURE 3.2.6-4

Orbital Descent - The capsule orientation which imposes the least constraints on system design is one that allows the capsule to be inertially fixed at the Mars atmospheric entry angle. For landing missions near the Martian morning terminator this orientation would result in the heat shield portion of the capsule facing the sun. Landings which are later in the day result in rotation of the capsule until, for evening terminator landings, the base of the capsule would face the sun. The worst landing time from a thermal control standpoint is near noon, since the sun then strikes one side of the aeroshell, warming it, while the other side is shaded and becomes cold, leading to possible thermal stress problems in the aeroshell structure and heat shield. Figure 3.2.6-5, based on VOYAGER studies, shows that rolling the capsule during orbital descent can help alleviate the excessive cooling problem. However, we are considering a heat shield ablator (McDonnell S-20T) capable of withstanding -300°F which will remove all orientation constraints. Analysis has shown that the Aeroshell can cool to no more than -290°F after 6 hours of descent, assuming the worst case of no solar heating. Preliminary experimental evaluations of the McDonnell S-20T ablator have shown no degradation after exposure to LN₂ temperature -320°F. However additional evaluation is necessary to fully qualify the ablative material.

Isotope Heaters - Isotope heaters have been studied for use in maintaining post-landing temperature levels. Their incorporation into the lander presents a potential problem because the radioactive decay process is continuous, thus overheating the equipment. The heat rejection capability of the lander insulation during space vacuum conditions depends on the insulation ultimately selected, but is lower than on the Mars surface. For example, foam insulation can reject about 60% of the post-landing isotope heat required, while multilayer may only reject a few percent. For this reason, it was assumed that the lander thermal control system would be optimized for the post-landing condition (since it is most weight sensitive) and that supplementary techniques would be used for isotope heat rejection during the flight phases. These supplemental techniques include: 1) providing a movable insulation panel which is open during flight to allow heat rejection and closed prior to landing to maintain post-landing thermal integrity, or 2) incorporating a heat rejection device such as heat

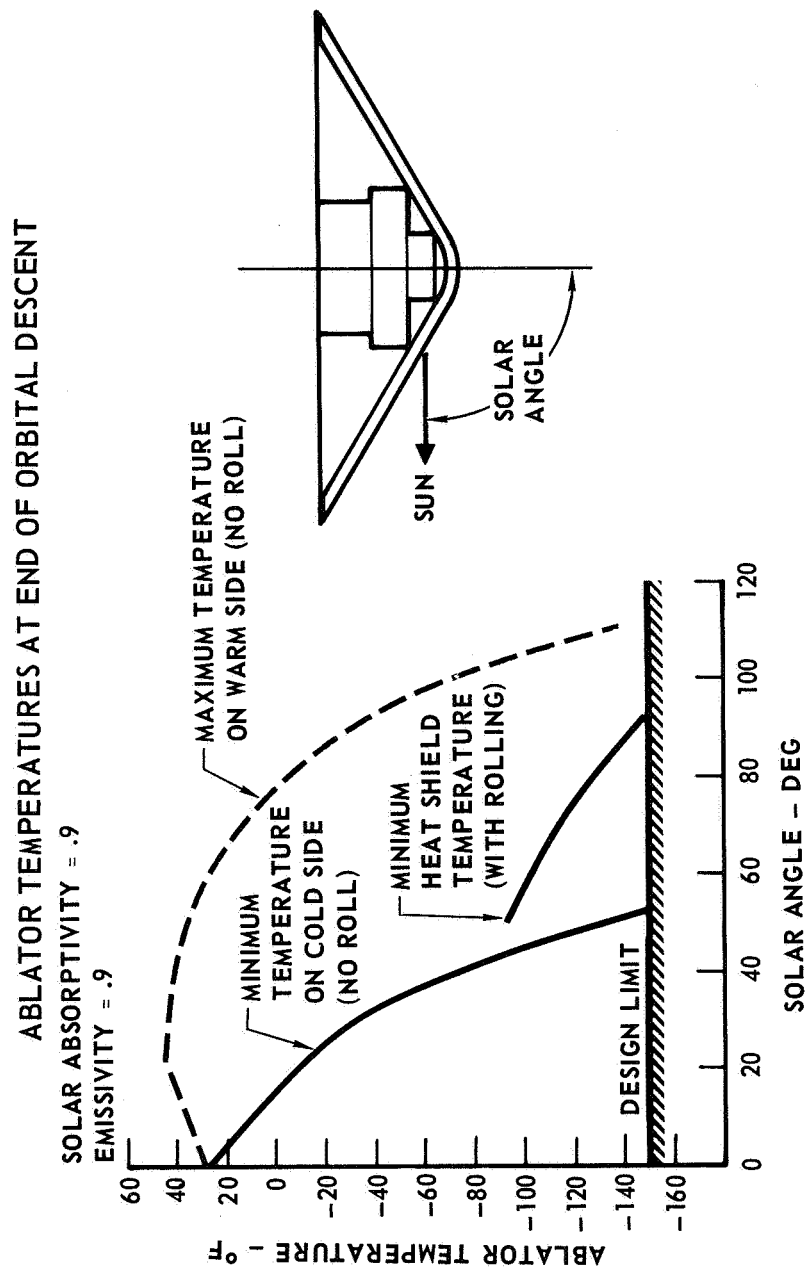
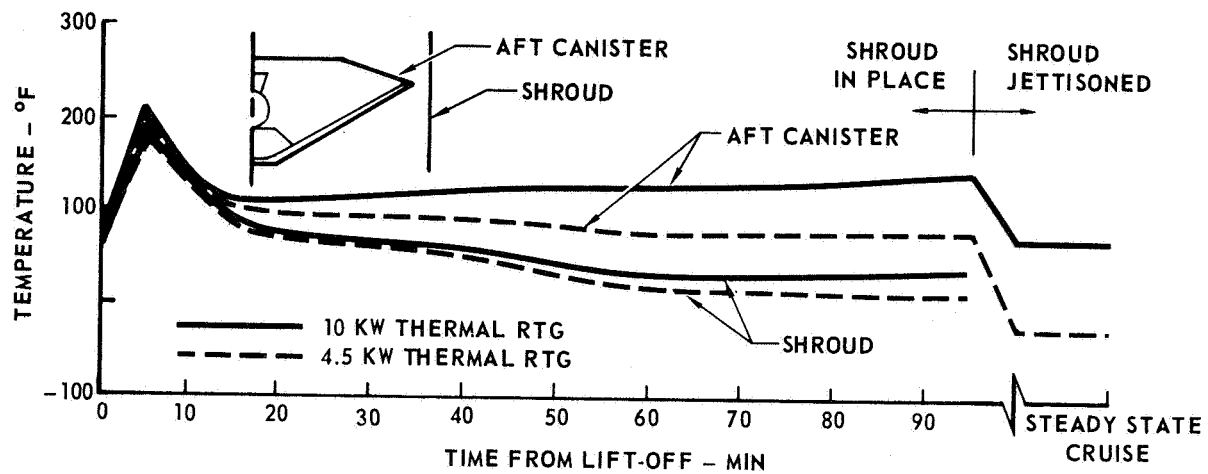


FIGURE 3.2.6-5

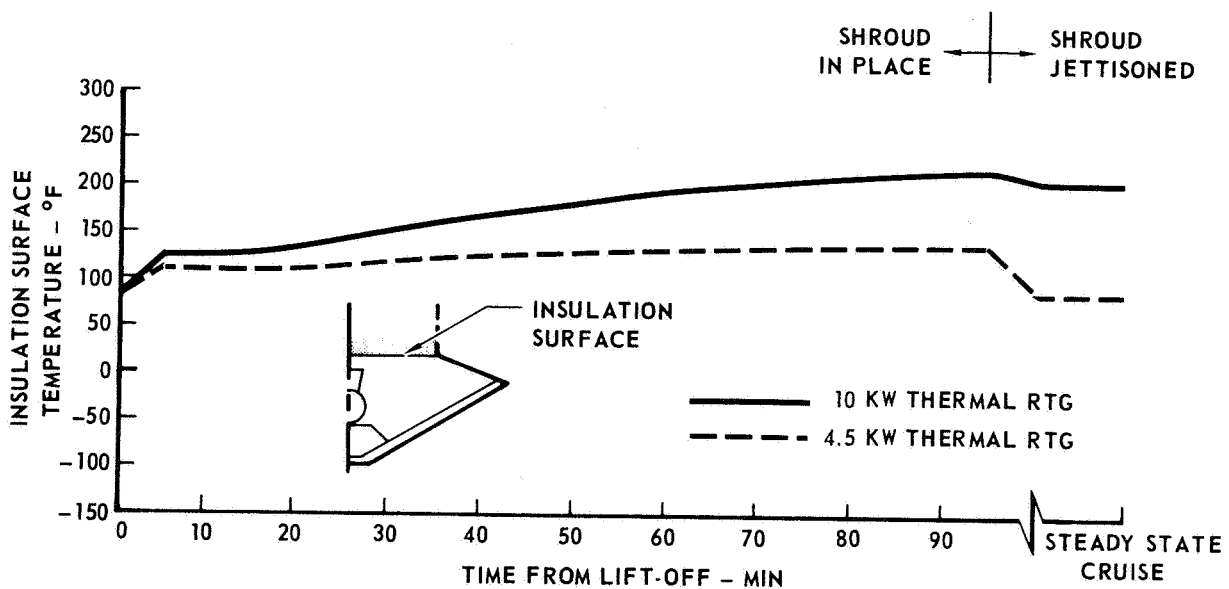
pipes or thermal switches that can also be used after landing. Using isotope heaters, the heater power requirements quoted previously for the various cruise orientations are reduced an amount equal to the isotope heating provided.

Radioisotope Thermal Electric Generator Power Systems (RTG) - These power systems require large heat rejection capability (about 20 times that required by battery-powered systems) throughout the mission. Studies performed during the VOYAGER program indicated that the most critical time period for thermal integration of RTG's is from launch through parking orbit, while the planetary vehicle remains enclosed in the launch shroud. Selected temperature histories for the VOYAGER-size vehicle during this period, Figure 3.2.6-6, indicate that a 10 kW thermal RTG is at or somewhat above the heat rejection capability of that size of vehicle. For other vehicle sizes, the limiting RTG power level will be approximately proportional to the surface area of the sterilization canister.

CAPSULE TEMPERATURES DURING LAUNCH AND ONE EARTH ORBIT AFT CANISTER AND SHROUD TEMPERATURES



INSULATION SURFACE TEMPERATURE



3.2.6.2 Landed phase. - Post-landing thermal control is concerned with maintaining the operative equipment and experiments within their allowable temperature ranges. The portion of surface landed payload attributable to thermal control varies between 15 and 21% and is comprised primarily of insulation necessary to protect equipment from the low temperature Martian night. Lander thermal control is complicated by the range of possible mission objectives and by the uncertainties in component performance after exposure to handling, sterilization, and mission environments. Performance is, therefore, dependent upon: a) heat transfer characteristics of the insulation after landing; b) ambient environments; c) equipment size, weight, arrangement, duty cycle, power level, and temperature tolerance; and d) performance of specialized devices such as phase change materials, isotope heaters, and heat pipes. Additional factors affecting the size and type of thermal control systems include mission duration and choice of electrical power source.

Examination of mission variables and their effects on lander thermal control has resulted in definition of techniques and weights for the lander point designs discussed in Section 4, based on the supporting parametric data summarized in this section.

3.2.6.2.1 Design requirements: The external environment variations, the internal environment imposed by equipment power levels, and the operating temperature limits of the equipment are the primary design constraints for post-landed thermal control design. These are summarized in Table 3.2.6-1.

External Environment - The external environment to which the Martian lander will be exposed is presently highly uncertain. It is, therefore, essential to determine and minimize the lander's sensitivity to these uncertainties, so that as better values are obtained they can be easily incorporated into updated study results.

The near equatorial clear day solar constant for a 1973 Martian landing is approximately $180 \text{ Btu/ft}^2\text{-hour}$. The portion of this irradiation impinging upon a surface depends upon the orientation of the surface, as well as the time of day. A horizontal surface, for example, receives a sinusoidal fraction

of the solar constant, varying over about 12.3 hours and having its maximum at Martian noon. Similarly, a vertical surface oriented normal to the plane of the sun's trajectory receives a quarter cycle sinusoidal portion of the solar constant. Due to the low density of the Martian atmosphere, the effect of absorption and scattering of the solar radiation near sunrise and sunset are not considered. The resulting variation of solar irradiation on vertical and horizontal surfaces is depicted in Figure 3.2.6-7.

The range of expected daily temperature variations for the Martian environment is shown in Figure 3.2.6-8. The average variation between the minimum and maximum expected is appreciable, averaging about 80°F. An approximately average temperature variation for the cyclic day is depicted in Figure 3.2.6-9. This is the assumed nominal cyclic temperature environment that was used for analysis. The nominal environment accentuates the daily temperature variation by incorporating the minimum temperature value of the low range and maximum value of the high range. A continuous cloudy day was also assumed for analysis, in which all direct and reflected solar radiation to the lander was eliminated. A constant soil and surface atmospheric temperature of -190°F was assumed.

The forced convection heat transfer coefficient for the Martian atmosphere, Figure 3.2.6-10, varies between .2 and 2.5 for surface pressures of 5 to 20 mb and wind velocities from 10 to 500 ft/sec. Convection heat transfer coefficients from zero to 2.5 were considered for analysis.

3.2.6.2.2 Candidate approaches: Techniques for providing post-landing thermal control consist of providing sufficient insulation material and internal heat generation to avoid excessive cooling of the lander equipment during the Martian night while avoiding overheating caused by equipment or continuous heater operation during the daytime. Specific techniques studied include thermal insulation, radioisotope and electrical heaters, and phase change material (PCM) i.e., a hydrocarbon wax having a melting temperature near the upper limit of the battery or equipment. In previous McDonnell studies (Voyager Phase B) heat pipes and other devices providing daytime heat rejection

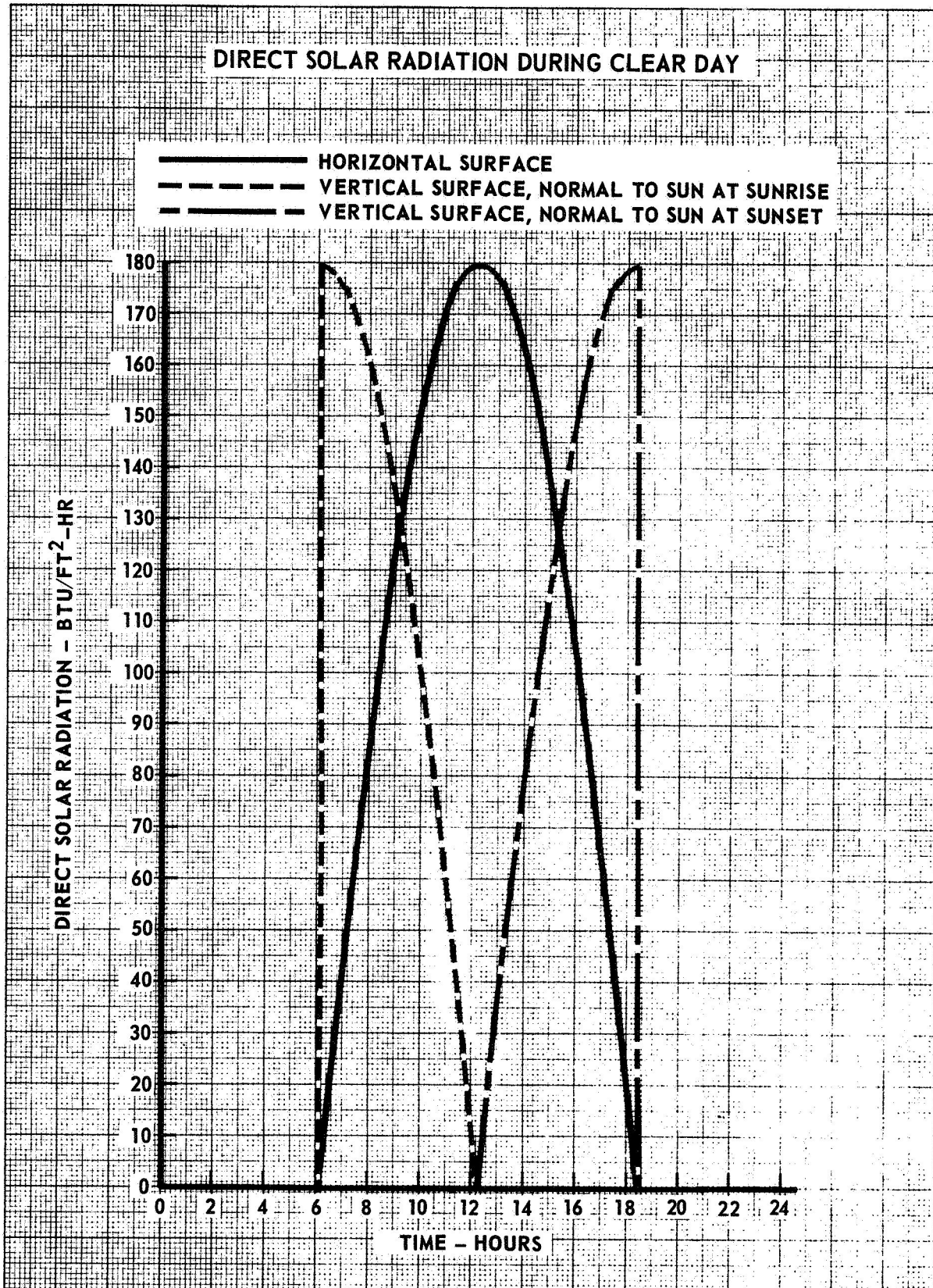


FIGURE 3.2.6-7

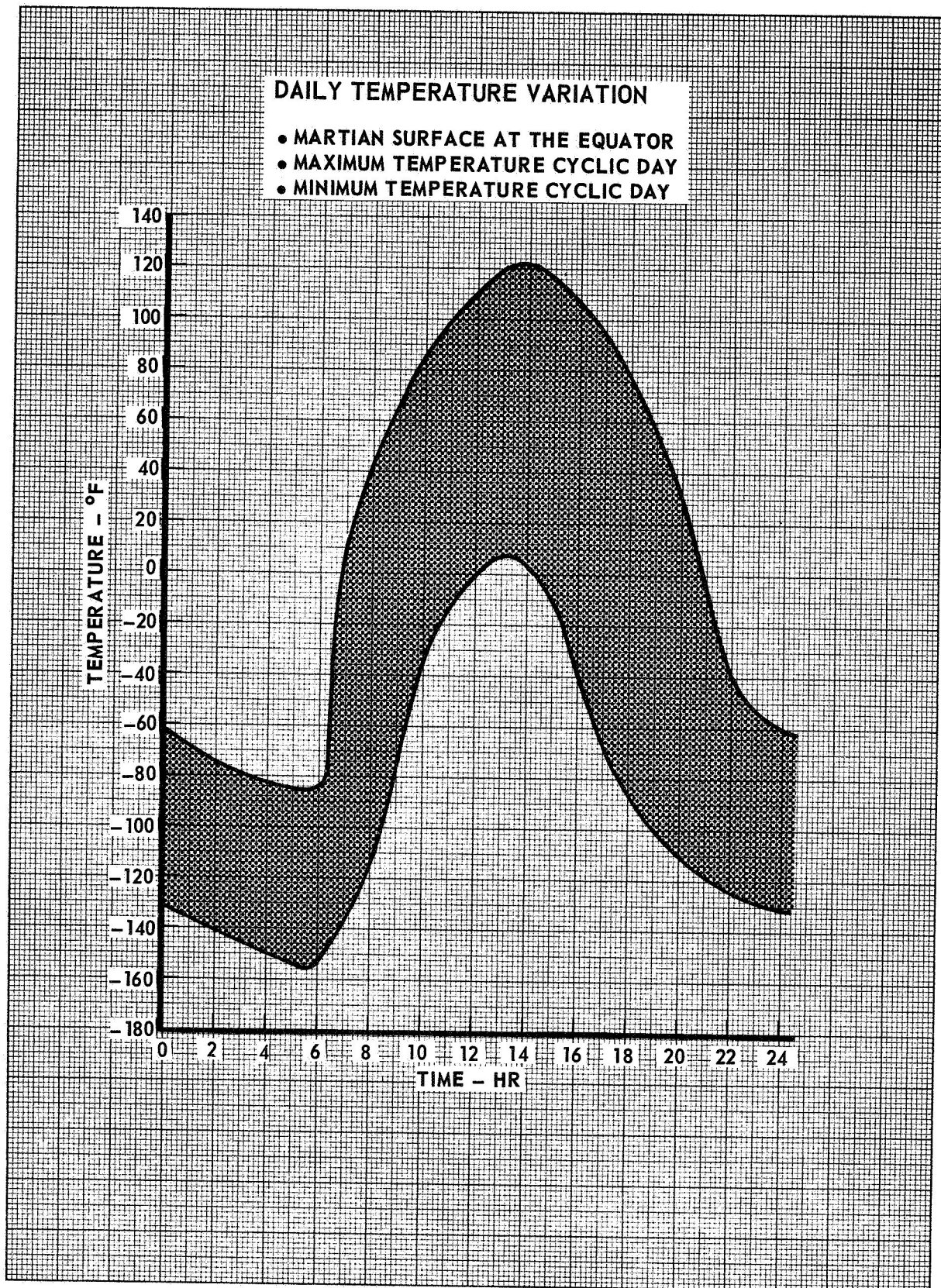


FIGURE 3.2.6-8

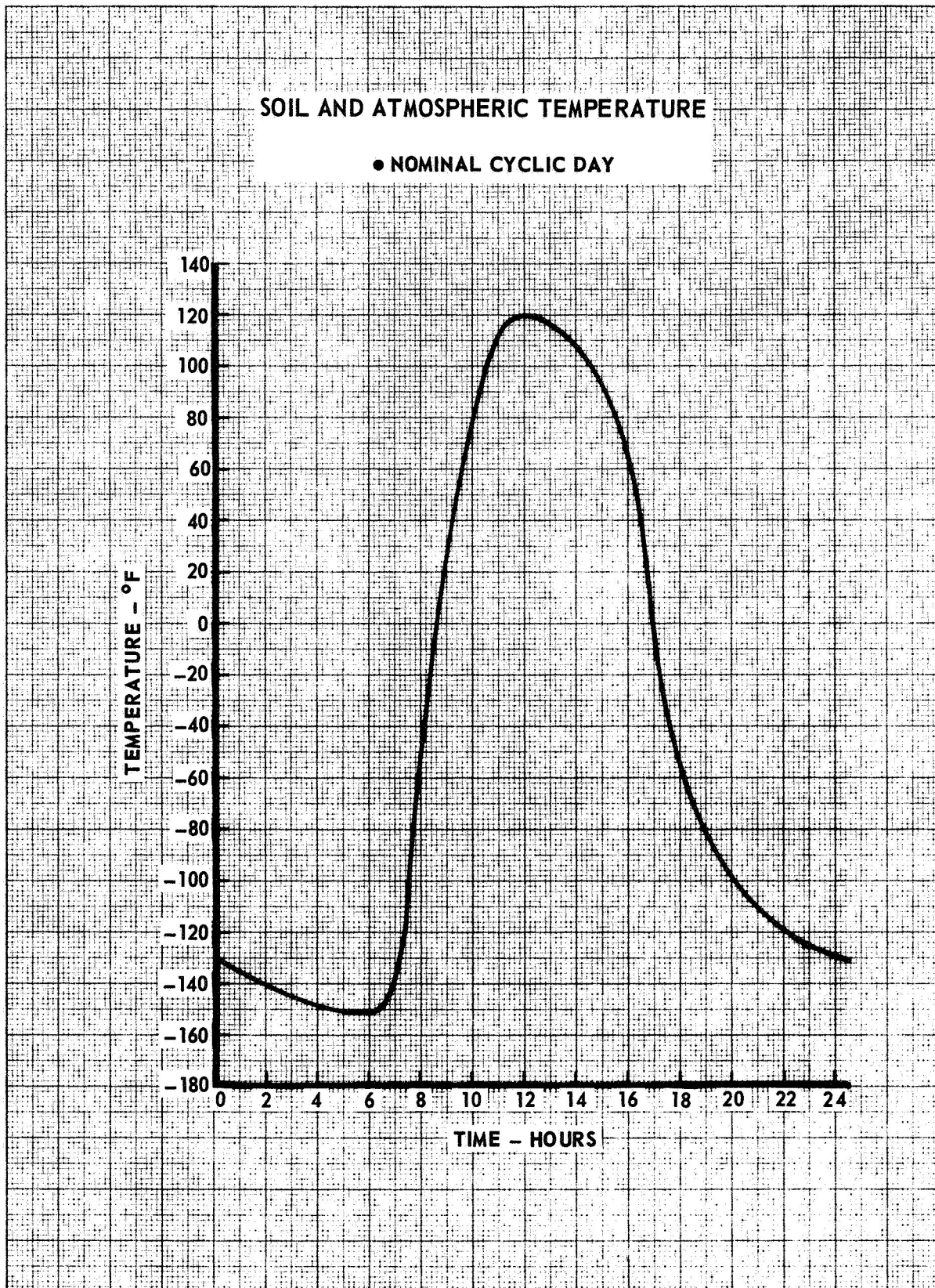


FIGURE 3.2.6-9

CONVECTION HEAT TRANSFER FOR LANDER EXTERIOR

- FORCED CONVECTION
- GAS TEMPERATURE 0°F
- CHARACTERISTIC LENGTH 1 FT

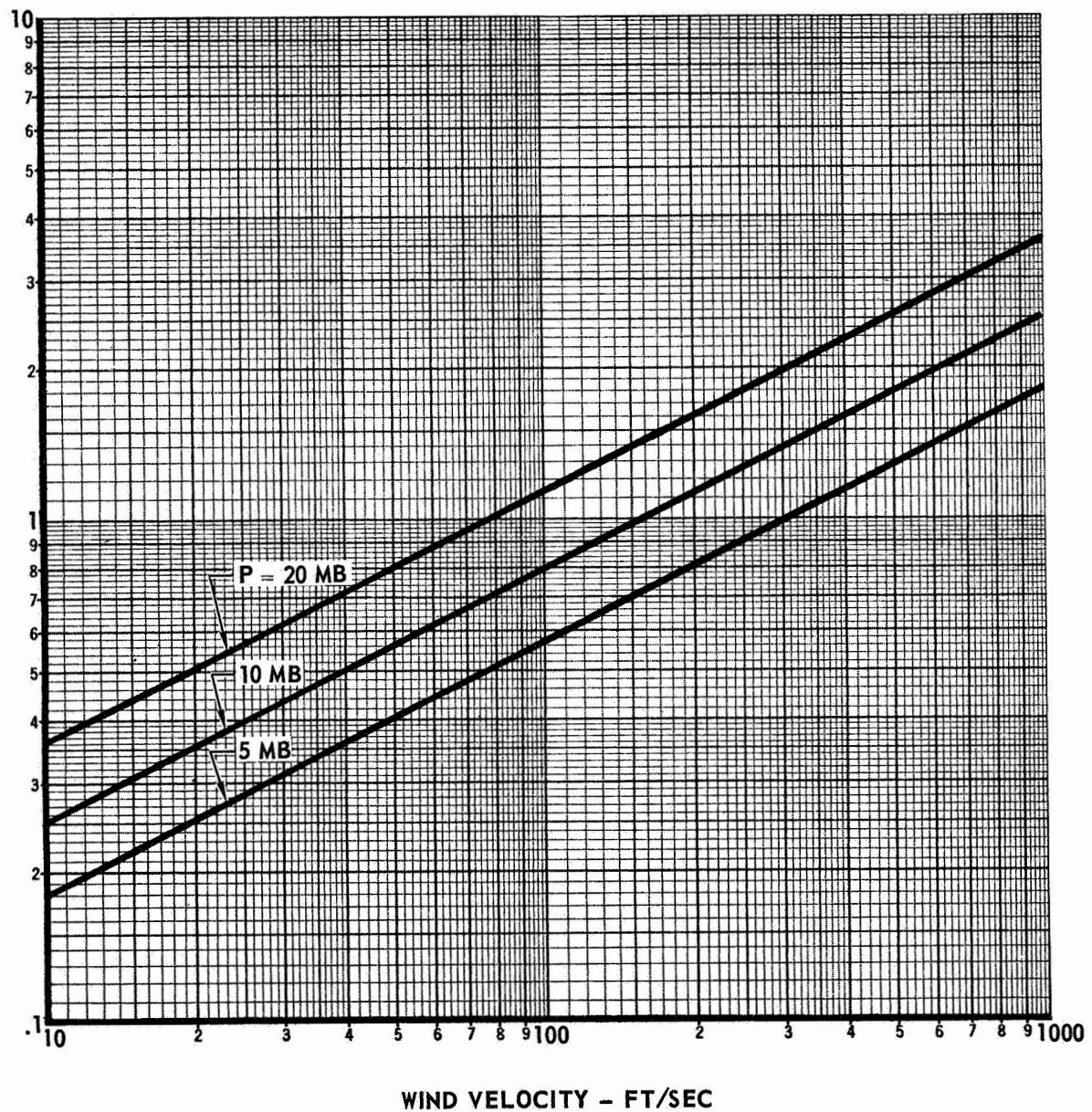


FIGURE 3.2.6-10

were evaluated. The lander configurations in that study were larger, more complex, and included several highly temperature-sensitive components not currently under consideration.

During the analysis it was determined that through judicious selection of the amount of PCM and isotope heaters and the operating characteristics of the insulation, proper lander operation could be achieved for a wide variation in the environmental conditions without employment of heat pipes. However heat pipes continue to remain attractive if the entire range of environmental conditions must be accommodated. While their use may result in some weight savings on small landers, the additional development costs must be considered.

For analysis purposes a generic soft lander was defined; it is shown schematically along with the thermal control techniques to be studied in Figure 3.2.6-11. These included:

- a) Use of two insulated compartments, where equipment can be segregated according to their allowable operating ranges (Inner compartment 50° to 120°F; Outer compartment generally 0° to 100°F).
- b) Phase change material or heat sinks, e.g., Beryllium, provides a large thermal inertia to dampen wide temperature variations.
- c) A thermostat controlled electrical heater(s) to signal the need for heat to the inner compartment when this compartment temperature falls below the minimum operating temperature, 50°F.
- d) Isotope heaters to supply a continuous constant heat generation to either the inner or outer compartment enabling a reduction of battery size and heating requirements for the continuous cold day and eliminating the need for additional heating during the cyclic day environment.
- e) Insulation to dampen the effects of the fluctuating environment and the extreme cold conditions to which the lander is exposed. The insulation reduces the heat losses and the heating required to maintain equipment within operating temperature ranges.

BASIC ANALYSIS MODEL CONFIGURATION

EXTERNAL ENVIRONMENT

COLD DAY - (-190°F CONTINUOUS)

CYCLIC DAY - (-150 TO + 120°F)

HEAT TRANSFER MODES:

RADIATION:

LANDER
EMITTED

DIRECT SOLAR

REFLECTED
SOLAR

GROUND
EMITTED

BATTERY/TRANSMITTER COMPARTMENT
(TEMPERATURE LIMITS, 50 TO 125°F)

- BATTERIES
- TRANSMITTER
- SEQUENCER

CONVECTION

EQUIPMENT COMPARTMENT
(TEMPERATURE LIMITS, 0° TO 100°F)

- SCIENCE

SURFACE AREA =
27 FT²

INSULATION

- ELECTRIC HEATERS WITH THERMOSTATS
 - PHASE CHANGE OR HEAT SINK MATERIAL
 - ISOTOPE HEATERS
- (IF REQUIRED)

EXTERNAL DIMENSIONS

46'' x 39'' x 14.5''

FIGURE 3.2.6-11

The multicompartment lander configuration was selected over a single compartment design in order to reduce total vehicle weight. The battery and equipment such as sequencers, whose proper operation requires strict temperature control, are isolated in the inner compartment from equipment that can operate at significantly lower temperatures. This arrangement reduces the thickness of the outer compartment insulation necessary to protect the equipment from the wide temperature variations of the environment. A portion of this "saved insulation" weight was utilized to protect the inner compartment, but since the inner compartment area is appreciably smaller than that of the outer, a total insulation volume and weight reduction was realized.

Arrangement of equipment in the outer compartment is such that articles requiring higher operating temperatures are clustered around the warmer inner compartment, with the remaining equipment arranged near the cooler exterior. Science equipment may be included in this outer compartment, which simplifies the mechanism for their deployment.

Performance characteristics of the thermal control parameters and equipment limitations are summarized in Table 3.2.6-2.

3.2.6.2.3 Analysis techniques: Evaluation of lander thermal response characteristics was performed for a basic dual compartment design shown in Figure 3.2.6-11. The objective of this analysis was to determine the weight of thermal control techniques necessary to maintain the required temperature limits for two general communication modes, direct and relay, consistent with the external environment and insulation performance. This data was then applied to analytical thermal scaling laws to allow extrapolation of the results to other lander sizes, while retaining the thermal response characteristics of the basic lander.

Complexity of the heat transfer relationships germane to the lander, the number of lander elements together with the transient behavior of the power profile and external environment necessitates development of an analytical model for the lander. This model represented the lander shown

schematically in Figure 3.2.6-11, and utilized the McDonnell General Heat Transfer Computer Program for solution. It was composed of 34 thermal elements (nodes) representing the lander structure, equipment, and thermal control devices which were thermally connected through the appropriate modes of heat transfer (radiation, conduction, and convection), heat storage, and heat addition. A backward difference technique was employed in simultaneously solving the transient heat transfer finite difference equations. Thermal characteristics of this basic analysis model are summarized in Table 3.2.6-3.

The thermal analysis computer model has several special features that are especially applicable to the thermal control methods for the lander. A thermostat routine calculates and supplies to the heat balance the amount of heat necessary to retain the battery compartment at the prescribed lower temperature limit (50°F). Another feature is the capability of properly considering the phase change material (PCM) in the heat balance. As heat is added to the mass, the temperature increases until the fusion temperature of the wax is reached. As additional heat is imposed, the temperature is maintained at the fusion value until the entire mass is melted. Only after total melting has occurred does the temperature again increase with heat addition. The program tabulates the quantity of melted and frozen PCM at each time step, and reversal of the process (change from heating to cooling) is allowable at any single or dual phase condition.

Heat sources considered in the thermal analysis can be either invariant with time or time variant. This first class is especially applicable to the isotope heaters employed onboard the lander; the second class is more descriptive of the heat dissipation associated with the equipment power profile.

The lander configuration was purposely made simple to enhance the efficiency of parametric computer studies and to avoid the pitfalls of establishing an overly specialized configuration for which conclusions are unique to the

TABLE 3.2.6-3
THERMAL CHARACTERISTICS OF BASIC ANALYSIS MODEL

LANDER COMPONENT	AREA FT ²	THICKNESS IN.	TOTAL HEAT CAPACITY (mcp) BTU/°F	DAILY POWER GENERATION WH	HEAT SINK CAPACITY WH	SOLAR ABSORP- TANCE α_s	EFFECTIVE EMITTANCE ϵ	CONVECTIVE HEAT TRANSFER COEFFICIENT, h_c BTU/FT ² -HR-°F
EXTERNAL SURFACE	42	-	-	-	-	.5	.9	1.0
OUTER INSULATION	27	3.0	8.52	-	-	-	.9	-
EQUIPMENT COMPARTMENT	27	-	27.4	628	-	-	.9	-
INNER INSULATION	17.5	.4	.58	-	-	-	.9	-
BATTERY COMPARTMENT PHASE CHANGE MATERIAL AND PACKAGING (17 LB)	17.5	-	18.1	650	-	-	-	-
	-	-	-	-	340	-	-	-

TOTAL SURFACE PAYLOAD WEIGHT - 330 LB EXCLUSIVE OF DEPLOYED INSTRUMENTS

design. The lander geometry can be easily modified to other candidate designs and the operating characteristics established.

For analysis purposes the lander was oriented on the Martian surface so that the Sun's trajectory was parallel to the two longer sides of the craft. Hence, these two sides and the bottom surface receive radiation which has been reflected from the ground, while the other two sides receive in addition direct solar radiation. In this orientation the lander receives minimum direct solar heating during the cyclic day. Since the parametric studies indicate that heating is required during cyclic as well as during continuous cold operation, selection of this minimum solar heating orientation reflects a conservative approximation.

Using the assumed solar heating orientation the physical and thermal symmetry of the lander about a vertical centerline simplifies the analysis. There is no net heat flow across this centerline (adiabatic plane). Therefore, only one half of the vehicle need be considered for analysis, thus simplifying the analysis model and reducing computer usage time.

3.2.6.2.4 Performance: The desired temperature response within the equipment limits can be obtained by many combinations of thermal control techniques. Therefore, an optimization technique was necessary to allow later definition of the point designs. The optimization approach was to:

- a) Minimize weight with respect to most likely or nominal values for the environment and performance of thermal control techniques.
- b) Maximize flexibility for accommodating variations in environmental and performance characteristics with a low weight increment.

Parametric Variables - The nominal lander configuration used for thermal analysis contained the thermal control techniques discussed above. Power profiles representing direct transmission and relay transmission modes, shown in Figures 3.2.6-12 and 3.2.6-13, were investigated. The expected lander insulation performance range, expressed as thermal conductivity divided by thickness (k/x), was used in the analysis. The ratio of outer to inner insulation thickness (7.5) was held constant, based upon maintaining the battery

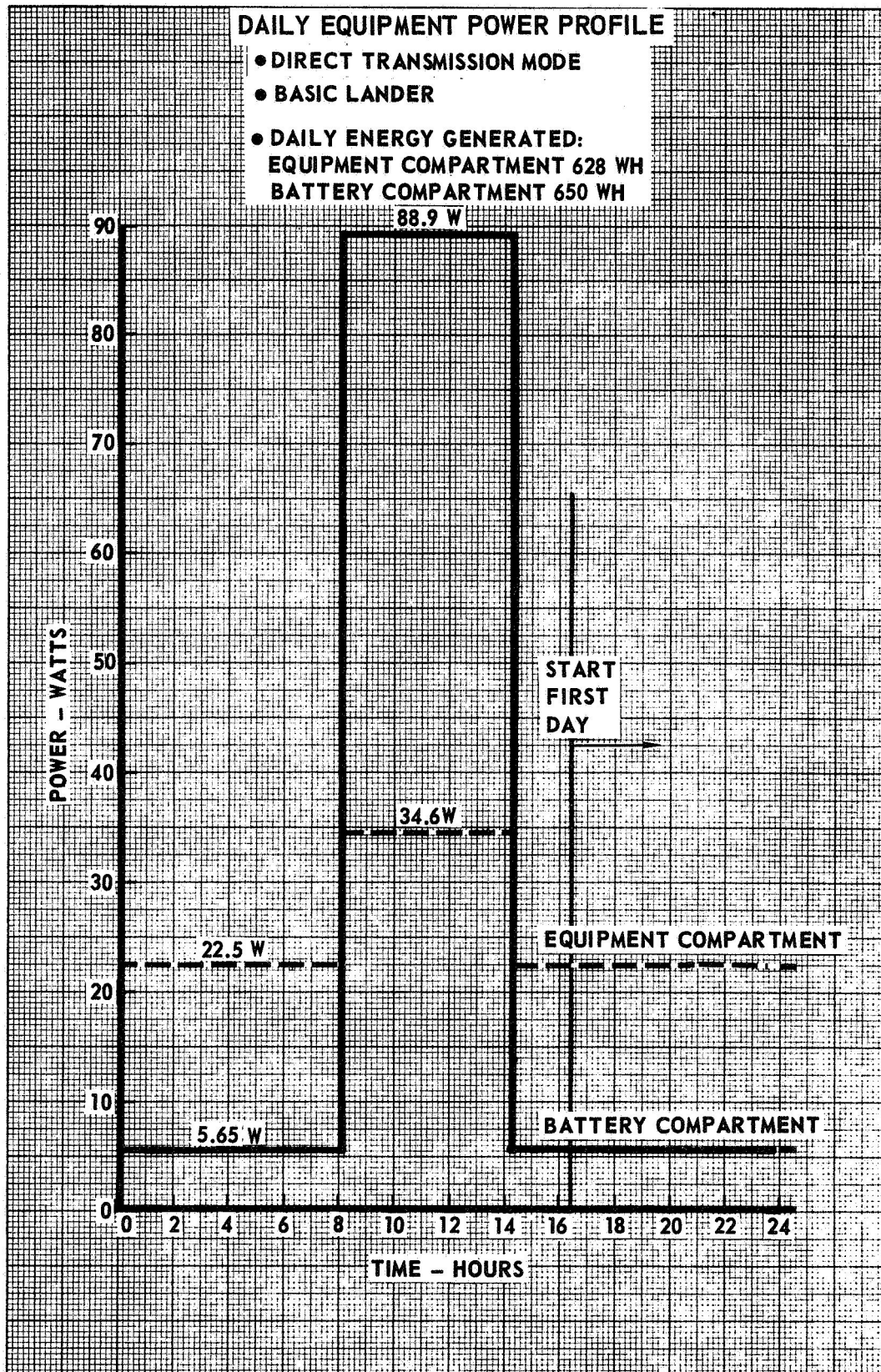


FIGURE 3.2.6-12

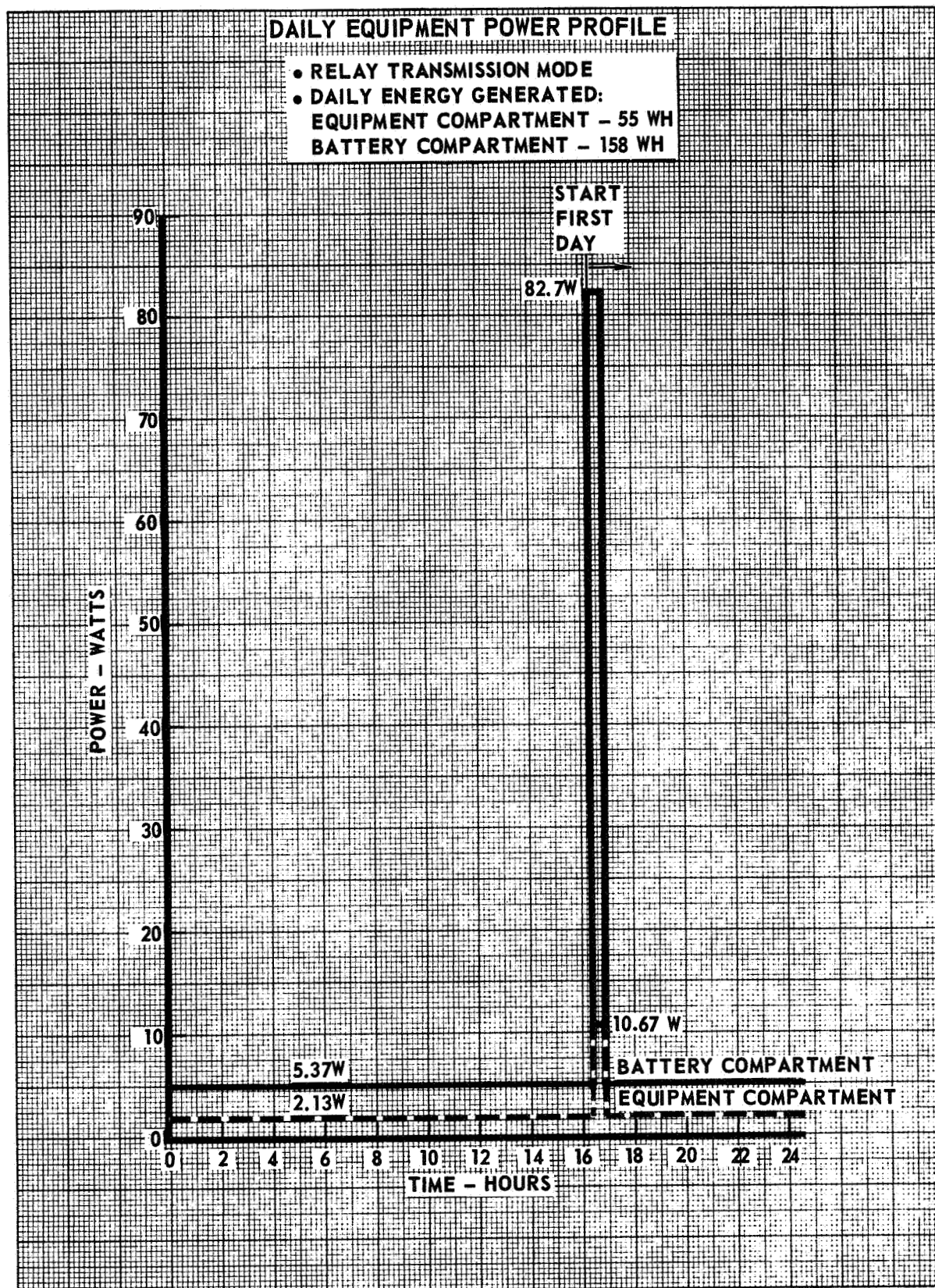


FIGURE 3.2.6-13

and equipment compartments above minimum operating temperature requirements for the continuous cold (-190°F) day. The analysis determined the heater power (both electrical and isotope) necessary to maintain the minimum temperature limits, and the necessary amount of phase change material to avoid over-heating during daytime equipment operations. This data was then extended to other lander sizes through use of thermal scale modeling techniques.

Analysis for Direct Transmission Mode - The direct transmission mode analyzed is characteristic of the high equipment power usage during Martian daytime when the Earth is in view. This type of transmission mode has the greatest adverse effect on thermal control due to potential equipment over-heating, since the high transmitter power levels coincide with daytime environment heatup. Utilizing the direct mode power profile (Figure 3.2.6-12) the allowable heating energy ranges, Figure 3.2.6-14, were determined as a function of insulation performance (k/x) for the nominal cycle and the continuous cold day. Heating energy, as defined, includes both equipment power profile heating and heater energy. The minimum heating energy curve indicates the heating necessary to prevent the battery compartment temperature from falling below 50°F during the night. The minimum heating energy shown for a continuous cold day is that required when there is no solar heating and the ambient temperature is -190°F . The maximum heating energy curve shows the maximum total heating allowable to prevent total melting of the phase change material and subsequent overheating of equipment during daytime operations.

The horizontal line at $Q/A = 24$ is the equipment heat generation in the battery compartment for the direct transmission mode. The curves are interpreted as follows: The portion of the required heating lines lying below the battery compartment power profile line ($24 \text{ whr/ft}^2 \text{ day}$) represents the range of outer compartment insulation performance factor (k/x) for which no additional heating (above the power profile) is needed to maintain the battery compartment temperature above its minimum value. Those portions of the heating curves above the power profile represent the (k/x) range for which heating in excess of the equipment power profile is necessary - either by electrical or isotope heaters. Selection of the outer insulation thickness must be such

ALLOWABLE DAILY HEATING ENERGY TO BATTERY COMPARTMENT - DIRECT MODE

- Q, SUM OF POWER PROFILE AND HEATER ENERGY
- A, LANDER INNER SURFACE AREA = 27 FT²
- EQUIPMENT COMPARTMENT POWER = 23.2 WH/FT²-DAY

NOTE: BATTERY COMPARTMENT TEMPERATURE:
50°F MINIMUM
125°F MAXIMUM

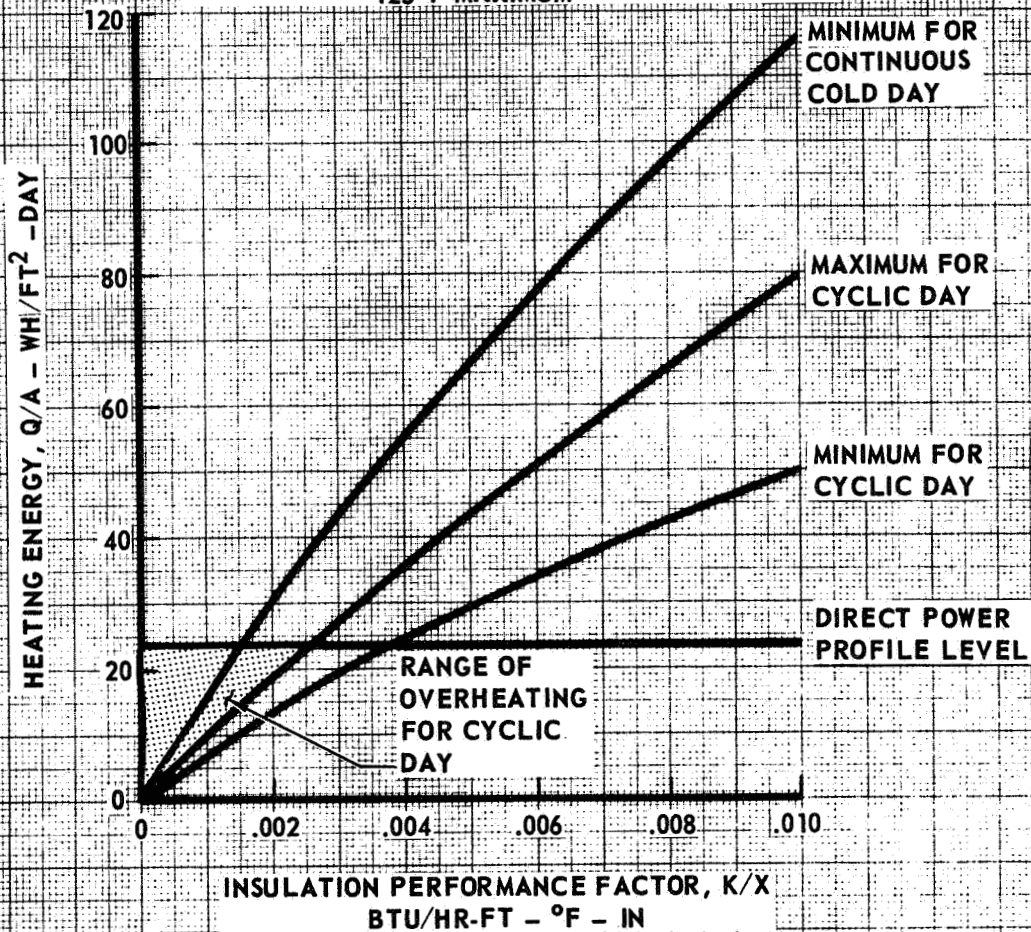


FIGURE 3.2.6-14

that the ratio (k/x) is greater than about .0027 Btu/hr-ft-°F-in, to avoid overheating during the cyclic day operation.

Both "minimum" curves of Figure 3.2.6-14 apply to a controlled heater source which supplies heat only when necessary to maintain temperature at or above 50°F (i.e. electrical heaters with thermostats). The maximum curve, however, is for a continuously operating, constant power source, typically isotope heaters. Based on providing equal amounts of heat energy per day, the isotopes are much more weight efficient than batteries, by a factor of about 3:1. Isotope heaters are, therefore, preferred as long as overheating can be avoided during daytime operations.

The weight advantage of isotopes is demonstrated by the optimization curves of Figure 3.2.6-15 for the direct transmission mode. For the range of insulation performance factors of interest, the optimum insulation thickness necessary to minimize thermal control weight is typically, between 2 and 4 inches for direct mode. Subsequent sizing analyses were conducted using the mean value of 3 inches.

Analysis for Relay Transmission Mode - Thermal characteristics of the direct and relay transmission modes were compared. The typical relay mode power profile (Figure 3.2.6-13) is essentially constant, while the direct mode (Figure 3.2.6-12) has an extended high power level during the Martian day when Earth is in view. The short transmission distance associated with the relay mode is reflected in the lower power profile values.

Comparison of the direct mode power profile with constant power level profiles, characteristic of the relay mode, resulted in the thermal response data of Figure 3.2.6-16. Here the direct transmission mode power profile was restructured to the constant power level having the equivalent amount (watt hours) of energy per day. This redefined profile, characteristic of the relay mode, reduced the lander temperature variations and, consequently, the amount of phase change material required to avoid overheating, compared to the direct transmission mode. This reduction of phase change material is indicated (Figure 3.2.6-16) by the reduced time at the fusion temperature, (117°F) for the relay mode. The allowable heating energy values for the relay mode are

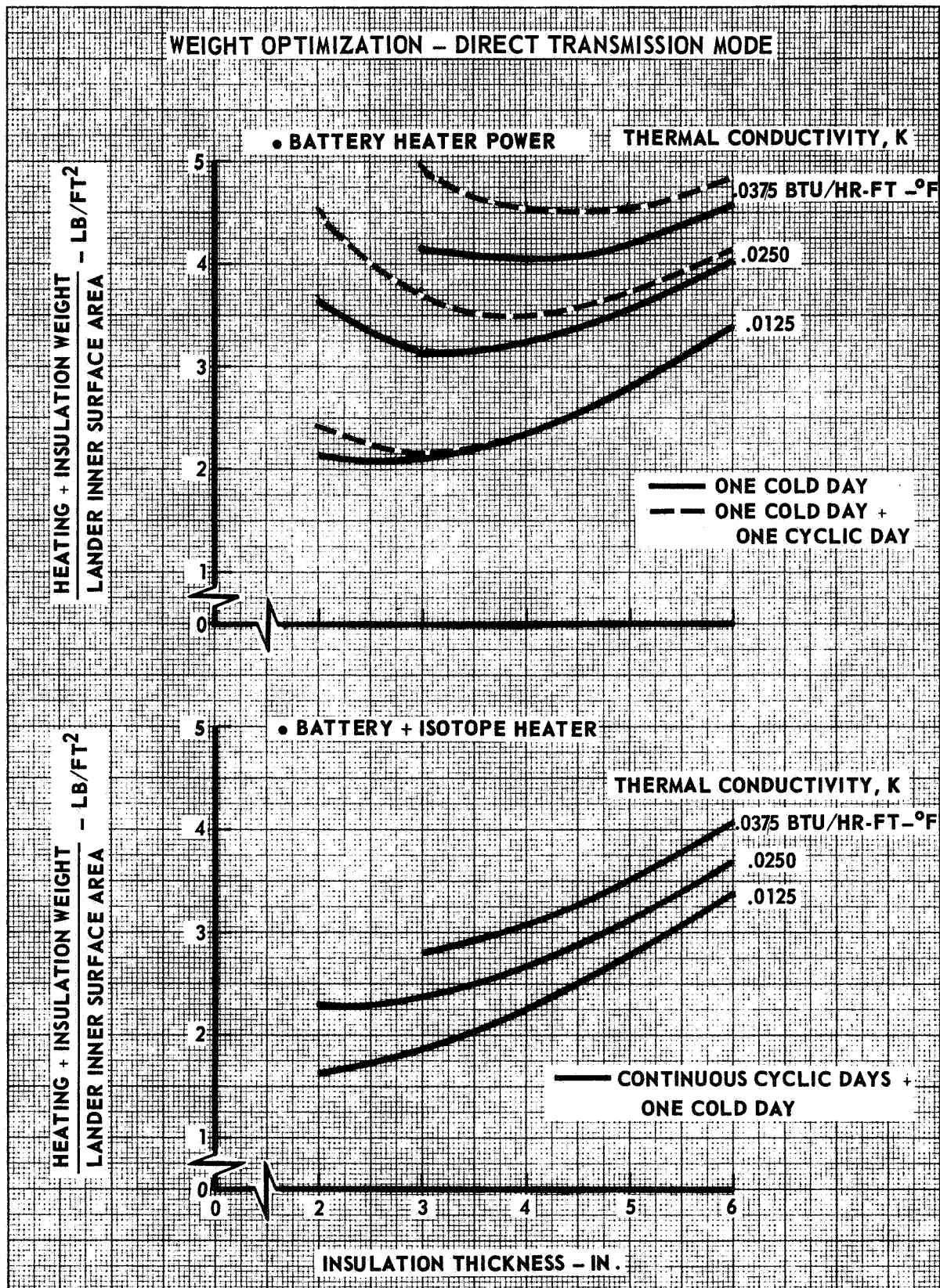


FIGURE 3.2.6-15

TEMPERATURE RESPONSE OF LANDER COMPARTMENTS FOR RELAY AND DIRECT TRANSMISSION MODES

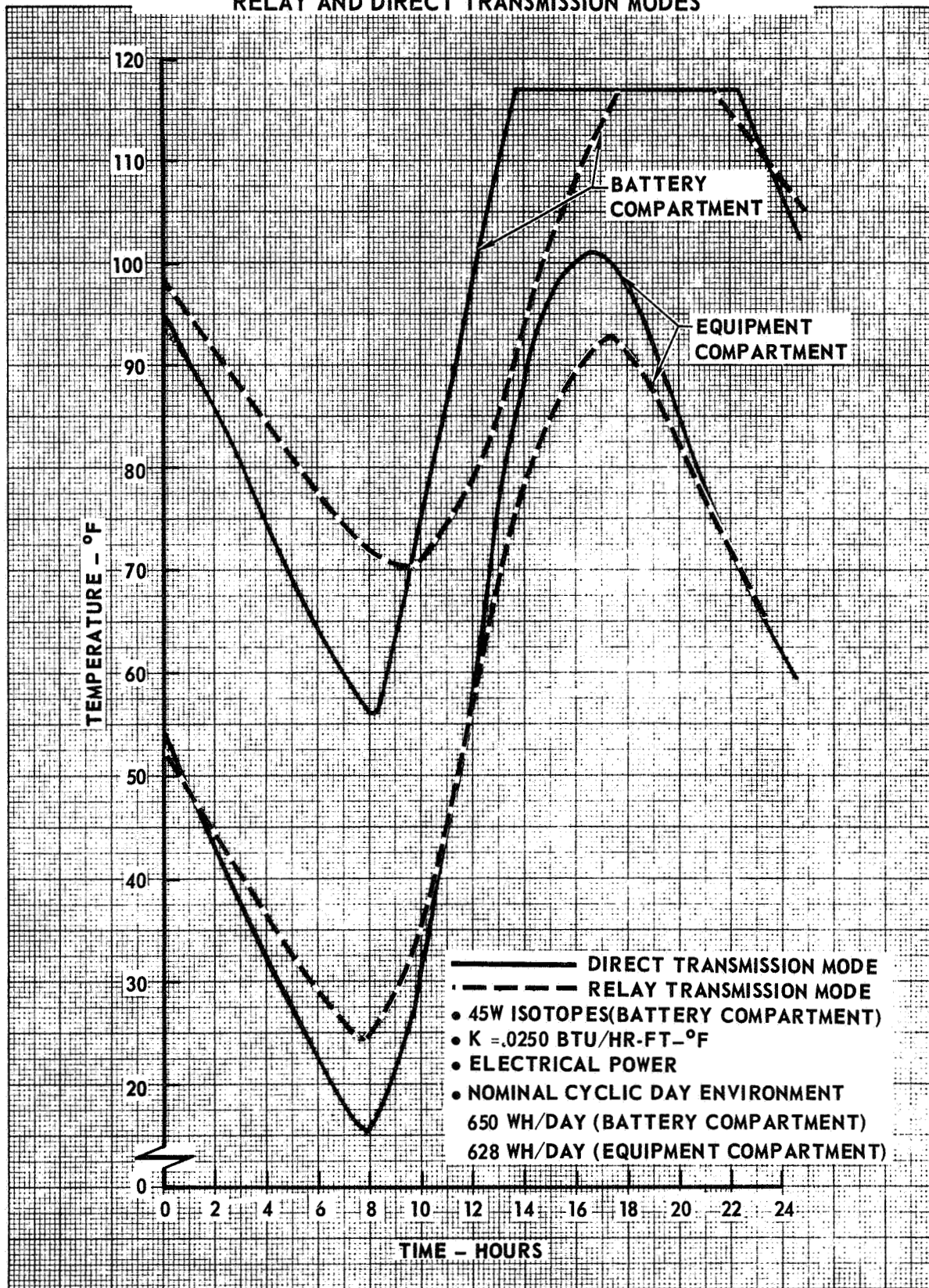


FIGURE 3.2.6-16

shown in Figure 3.2.6-17. The maximum values for cyclic days in the relay mode are higher than those for direct mode (Figure 3.2.6-14). This indicates that, for equivalent equipment power, the constant power level profile can utilize more isotope heating (without overheating) than can the direct mode type profile, and thus the thermal control weight could be reduced.

If the actual power levels associated with the direct and relay mode type profiles is considered, the weight optimization based on insulation thickness is shown in Figure 3.2.6-18, for a nominal insulation thermal conductivity of 0.025 Btu/hr-ft°F. Both modes display optimum thickness in the 2 to 4 inch range. Because the relay mode has a lower power profile, it results in appreciably less equipment heating during Martian night operations, and more heater weight is required to maintain minimum temperature levels.

Mission Duration Effects - The fact that the minimum heating requirement for the continuous cold day is larger than the maximum allowable heating for cyclic days precludes the use of isotopes to accommodate both environments without use of other thermal control techniques, such as heat pipes to dissipate isotope heat during the daytime operations. A more attractive technique for small soft landers would be to provide enough battery capacity to survive one cold day, which could occur at any time in the mission, and to rely on isotope heating for cyclic days. The lower curve, Figure 3.2.6-18, shows that with this approach less thermal control weight is necessary for extended mission operation than for a one or two day mission using no isotope heaters.

Several additional factors must be considered for missions of only several days. As previously explained, the maximum allowable cyclic day heating depicted in Figure 3.2.6-14, is applicable to extended missions. For shorter missions, this maximum heating level can be increased somewhat. This results from the nature of the phase change material operating over extended cyclic days. For example, Figure 3.2.6-19 depicts the increasing amount of phase change material which melts during extended "nominal" cyclic days for a typical insulation (k/x) and isotope heater power level. The amount of melted material increases until the eighth cyclic day, when all of it melts and overheating occurs, as seen in Figure 3.2.6-20. While this example shows that excess

ALLOWABLE DAILY HEATING ENERGY TO BATTERY COMPARTMENT - RELAY MODE

- Q, SUM OF POWER PROFILE AND HEATING ENERGY
- A, LANDER INNER SURFACE AREA - 27 FT²
- EQUIPMENT COMPARTMENT POWER - 23.2 WH/FT² - DAY

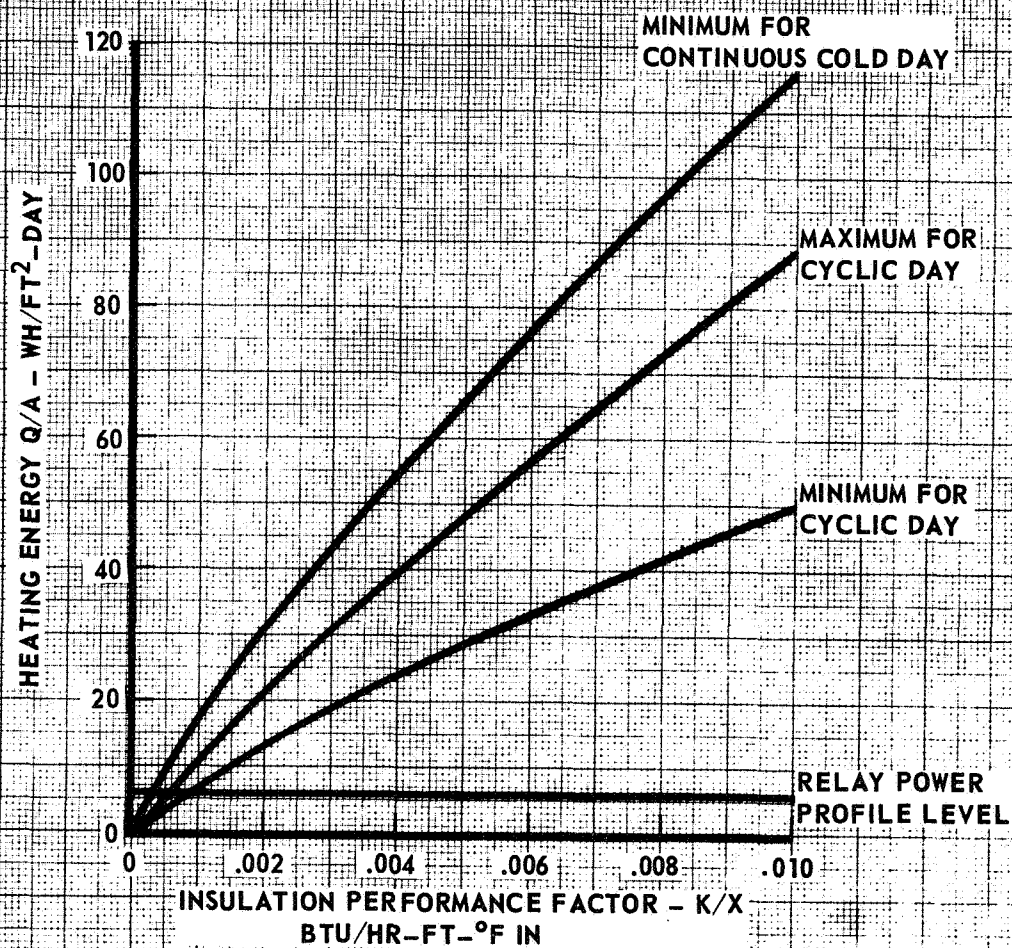


FIGURE 3.2.6-17

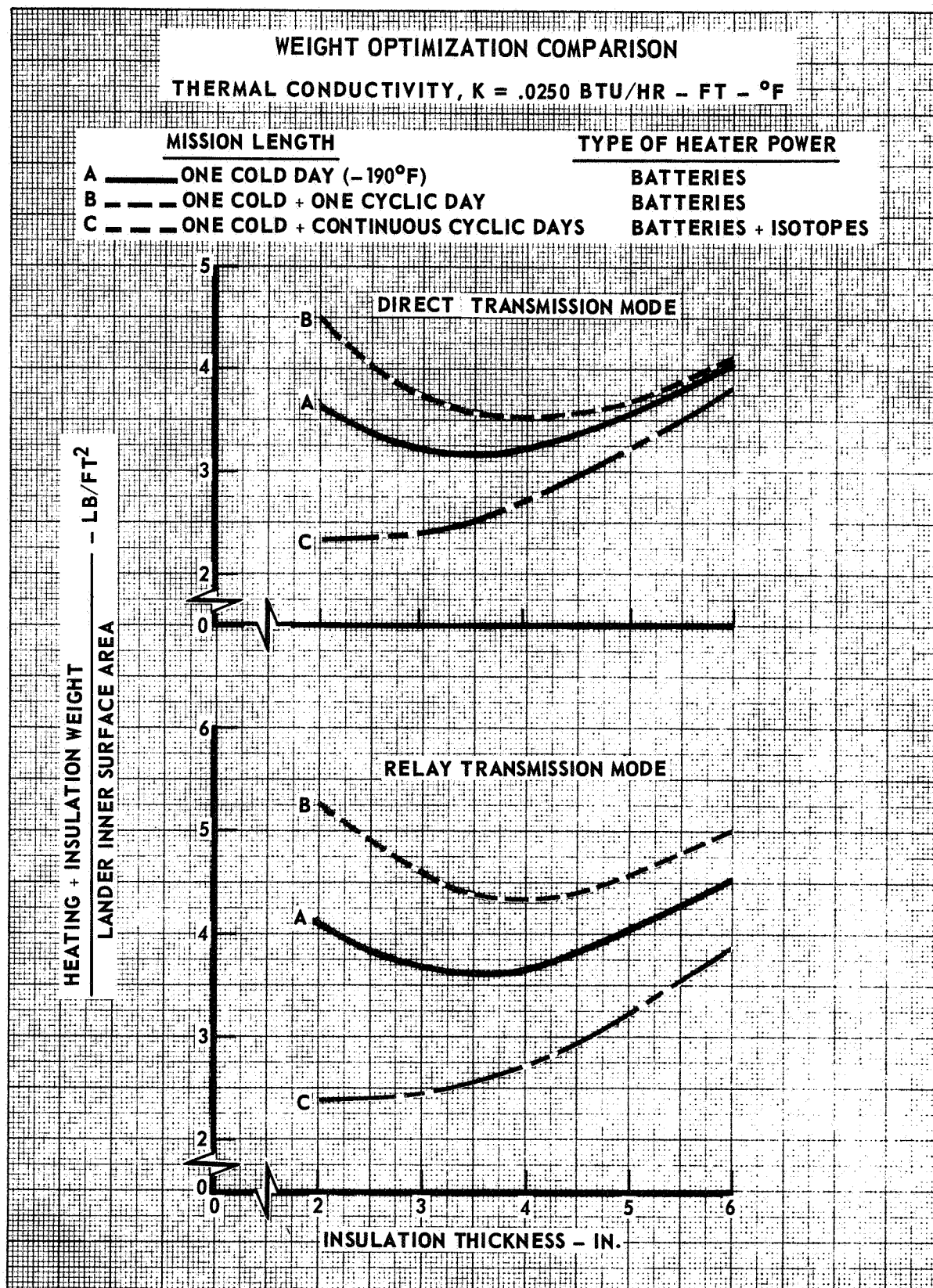


FIGURE 3.2.6-18

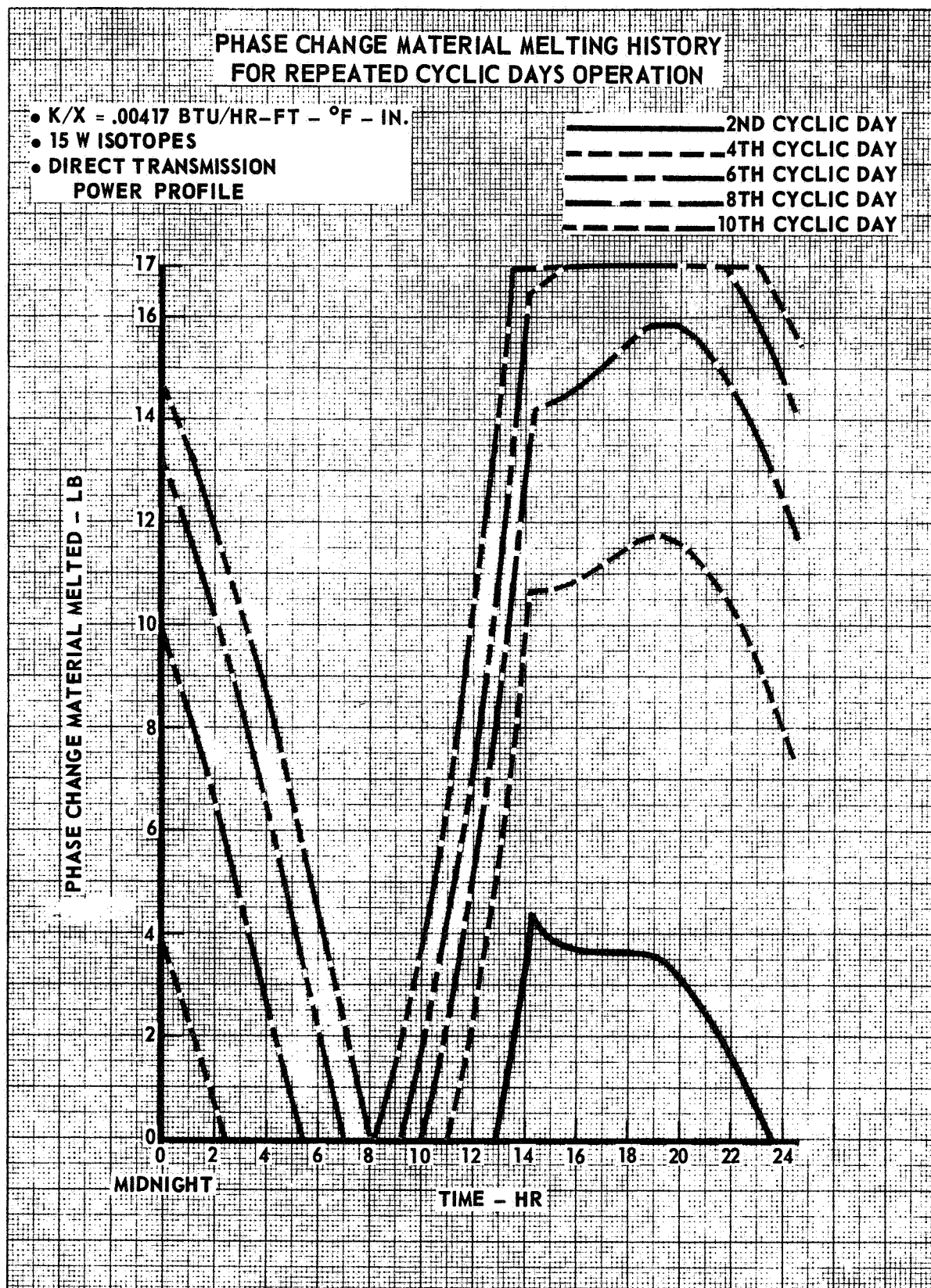


FIGURE 3.2.6-19

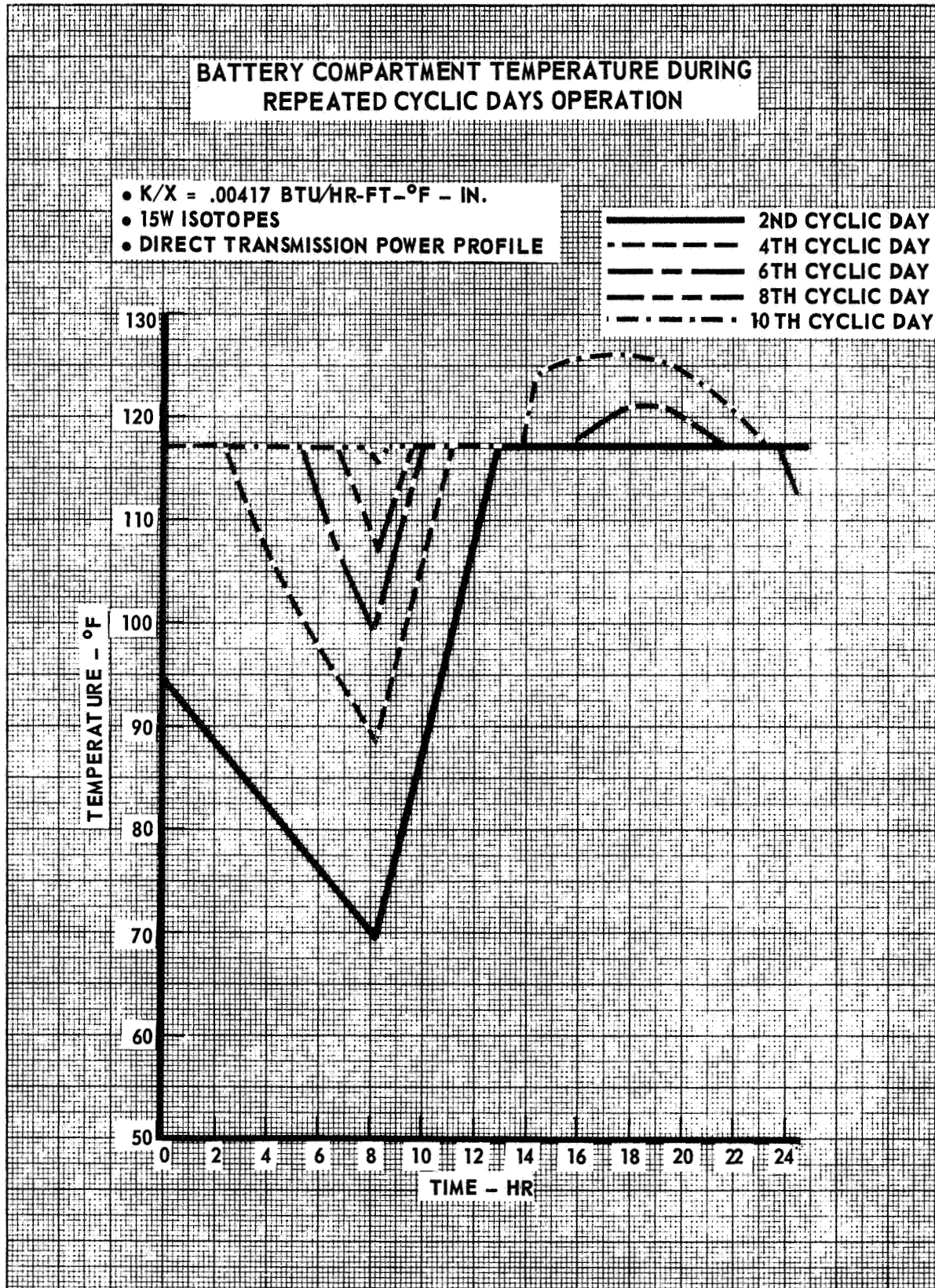


FIGURE 3.2.6-20

isotope power is present for extended operation, the amount is allowable for a six day mission. Additional isotopes, therefore, can be employed for short post-landed periods, to reduce battery weight.

Incorporation of Radioisotope Thermoelectric Generator (RTG) Power Systems - Although a quantitative analysis of the integration and optimization of RTG's into the thermal control system has not been performed, certain conclusions as to its effects can be discussed. Because of their high ratio of continuous heat dissipation to usable electrical power generated (about 20 to 1), the RTG waste heat can be utilized to provide some warming of lander equipment. Since the analysis considered complete thermal isolation from any RTG power system which might be selected, the results are applicable to both thermally isolated RTG's and other power systems. However, use of a portion of the RTG dissipated heat could be employed to reduce the additional heating requirements of the lander during the cyclic day operation, i.e. that provided by isotope heaters, thereby reducing the isotope weight. Further weight reductions may be attainable through the utilization of controllable heat transfer techniques, such as heat pipes, with the RTG system. Combination of these techniques would result in reduction in the total amount of insulation required for protection of the lander equipment. The additional level of thermal integration and optimization required for these techniques was not included in the current study.

3.2.6.2.5 Performance of Other Lander Sizes - The initial post landing thermal control investigations concentrated on evaluation, parameterization, and optimization of a particular lander configuration for specific insulation weight, heater power, and data transmission capabilities. This point design was demonstrated to be operable over a wide range of external environmental conditions, internal heating inputs, and thermal control techniques. However, the results obtained from the analysis of the lander are of limited value unless the thermal behavior of various other lander sizes, weights, and power profiles can be predicted. In this section, the initial results are demonstrated to be equally valid for other lander sizes.

The approach is to employ thermal scaling modeling techniques to identify other size landers to which the present results may be easily scaled, and to identify the particular design constraints imposed on the "scaled" lander, i.e. volume, internal power levels, and equipment contained in the two compartments.

In order to perform the sizing it is necessary to completely identify the initial lander, its various components, and the controlling heat transfer processes. Table 3.2.6-4 summarizes the components and the heat transfer processes of the basic lander or "prototype" employed in the scaling. The analogous heat transfer and geometric characteristics of other size landers (i.e. models) and the appropriate scaling factors were then be determined. The scaling is based on the philosophy that the transient temperature response of the model and the prototype will be identical at geometrically similar locations and corresponding times. This can be accomplished by scaling the various modes of heat transfer between model and prototype for each element and all times to the same ratio.

Selection of the appropriate method for scaling the basic lander design depends critically upon the various external and internal constraints. The following conditions must be satisfied.

- a) External Environment Constraints - Both the basic lander and the models to be identified must operate in the identical external environment. This includes the temperature variation of both the ground and atmosphere, the solar irradiation, and the atmospheric convection.
- b) Internal Design Constraints - Certain restrictions are placed on the components of the lander, such as the type of insulation, the type of equipment, and the required operating temperatures of the equipment. These remain unchanged from prototype to model. The weight optimization process determined insulation performance factors for minimizing thermal control weights; these will also be unchanged in the model configurations. Similarly, the thermal properties of equipment, battery, and phase change material are expected to be valid for other size landers.

TABLE 3.2.6-4
TABLE 3.2.6-4 BASIC LANDER THERMAL CHARACTERISTICS FOR SCALING

COMPONENT IN THERMAL ANALYSIS	HEAT TRANSFER MODES TO BE SCALED	THERMAL CHARACTERISTIC	NOMINAL CONDITIONS
EXTERNAL ENVIRONMENT	DIRECT & REFLECTED SOLAR HEATING CONVECTIVE HEAT TRANSFER	SOLAR RADIATION WIND VELOCITY PRESSURE TEMPERATURE	180 BTU/HR-FT ² 0-500 FPS TO 20 MB CYCLIC & CONT. COLD
LANDER EXTERNAL SURFACE	ABSORBED SOLAR HEATING RADIATION TO SPACE	SOLAR ABSORPTANCE EMITTANCE	.5 .9
EQUIPMENT COMPARTMENT INSULATION	CONDUCTION HEAT TRANSFER HEAT STORAGE	SURFACE AREA THICKNESS THERMAL CONDUCTIVITY HEAT CAPACITY (mC _p)	27.0 FT ² 3.0 INCHES .0250 BTU/HR-FT-°F 8.52 BTU/°F
EQUIPMENT COMPARTMENT	HEAT STORAGE INTERNAL RADIATION HEAT GENERATION	SURFACE AREA HEAT CAPACITY (mC _p) EFFECTIVE EMITTANCE EQUIPMENT POWER - ISOTOPE HEATING	27.0 FT ² 21.6 BTU/°F .9 VARIABLE
BATTERY COMPARTMENT INSULATION	CONDUCTION HEAT TRANSFER HEAT STORAGE	SURFACE AREA THICKNESS THERMAL CONDUCTIVITY HEAT CAPACITY (mC _p)	17.5 FT ² .4 INCHES .0250 BTU/HR-FT-°F .58 BTU/°F
BATTERY COMPARTMENT	HEAT STORAGE HEAT GENERATION	SURFACE AREA TOTAL HEAT CAPACITY EQUIPMENT POWER - ISOTOPE HEATING	17.5 FT ² 12.7 BTU/°F VARIABLE
PHASE CHANGE MATERIAL AND PACKAGING	HEAT STORAGE	HEAT CAPACITY AT MELTING POINT	123 WH

TOTAL SURFACE PAYLOAD WEIGHT - 280 LBS EXCLUSIVE OF DEPLOYED INSTRUMENTS

External Environment Scaling - The heat transfer modes of direct and reflected solar radiation, lander radiation to space and ground, and convective heat transfer were scaled. The preservation of radiation view factors between the landers and their environments and the preservation of direct and reflected solar radiation reduces this operation to maintaining identical heat transfer rates per unit area.

$$\frac{q_m/A_m}{q_p/A_p} = \frac{a_{s_m}}{a_{s_p}} = 1 \quad \text{Direct and Reflected Solar Radiation} \quad 1)$$

$$\frac{q_m/A_m}{q_p/A_p} = \frac{h_m (T_m - T_o)}{h_p (T_p - T_o)} = 1 \quad \text{Convective Heat Transfer to Atmosphere} \quad 2)$$

$$\frac{q_m/A_m}{q_p/A_p} = \frac{e_m (T_m^4 - T_o^4)}{e_p (T_p^4 - T_o^4)} = 1 \quad \text{Radiation from Lander to Space and Ground} \quad 3)$$

where,

A_m/A_p = surface area ratio of model-to prototype

a_s = solar absorptance

e = emittance

g, m, p, o are subscripts referring to the ground, model, prototype and outside environment condition, respectively.

Preservation of environmental and surface temperatures reduces Eqns. 2 and 3 to:

$$\frac{q_m/A_m}{q_p/A_p} = \frac{h_m}{h_p} = 1 \quad \text{Convective Heat Transfer} \quad 4)$$

$$\frac{q_m/A_m}{q_p/A_p} = \frac{e_m}{e_p} = 1 \quad \text{Radiation from Lander} \quad 5)$$

As long as the sizes of the model and prototype landers do not differ excessively, the convective heat transfer coefficient is approximately preserved, and, hence, model-to-prototype convective heat transfer is preserved.

To assure similitude between model and prototype, the remaining external heat transfer processes must scale similarly, which requires that both surface solar absorptance and emittance be preserved in the model.

Internal Lander Scaling - To insure model-to-prototype temperature similitude the modes of heat transfer within the lander must also scale identically with those at its boundary. This requires that conduction, internal radiation, heat storage, and heat generation per unit area scale as follows:

$$\frac{q_m/A_m}{q_p/A_p} = \frac{k_m}{k_p} \frac{x_p}{x_m} = 1 \quad \text{Conduction} \quad 6)$$

$$\frac{q_m/A_m}{q_p/A_p} = \frac{F_m}{F_p} = 1 \quad \text{Internal Radiation} \quad 7)$$

$$\frac{q_m/A_m}{q_p/A_p} = \frac{(\rho c_p)_m}{(\rho c_p)_p} \frac{V_m}{V_p} \frac{A_p}{A_m} = 1 \quad \text{Heat Storage} \quad 8)$$

$$\frac{q_m/A_m}{q_p/A_p} = 1 \quad \text{Heat Generation} \quad 9)$$

where:

c_p = specific heat

x = thickness

k = thermal conductivity

q = heat generation per unit surface area, (Btu/ft²-hr)

V = volume

F = grey body radiation shape factor, which includes geometry and emissivity

ρ = density

As shown in Table 3.2.6-4, certain of the above heat transfer scaling ratios must be satisfied for each of the lander components and for the insulation, equipment compartment, and battery compartment. The specific scaling technique for each of these constituents is described below.

Insulation Thickness - Both the conduction and heat storage constituents above must be satisfied through the outer insulation. Utilization of the identical insulation in both model and prototype (k, ρ, c_p) results in the following requirements:

$$\frac{x_m}{x_p} = \frac{x_p}{x_p} \quad \text{Insulation Thickness} \quad 10)$$

$$\frac{V_m}{V_p} = \frac{A_m}{A_p} \quad \text{Volume of Insulation} \quad 11)$$

Equation 10 specifies that the insulation thickness must be preserved and Equation 11 requires the volume of the insulation to scale according to the lander area ratio (A_m/A_p). These conditions are consistent with each other, since thickness preservation implies that the volume scales as the area ratio, with the exception of some insulation "corner" and "edge" effects.

Equipment Compartment - As shown in Table 3.2.6-4 the analytical representation of the equipment compartment is dominated by heat storage, heat generation, and radiation to the lander inner surface area and battery compartment, (Equations 7, 8, and 9). The radiative requirement of identical grey body shape factor of Equation 7 can be achieved by approximately preserving geometric similarity and equipment surface emittance. If similar equipment are considered, average density and specific heat are preserved, and the heat storage scaling requirement simplifies from Equation 8 to Equation 11, and the equipment compartment volume scales according to the lander area ratio.

Heating input to the equipment in the basic lander through the sum of power profile, isotope heating, or battery heating is scaled in the model by multiplying by the area ratio, as shown in Equation 12 (Equation 9, rearranged):

$$q_m = q_p \frac{A_m}{A_p}$$

Heat Generation

12)

For example, if a lander model inner surface area is one-half that of the basic lander, the total required heat generation at any time must equal one-half of the corresponding prototype heat generation.

The aforementioned method of modeling the equipment compartment may prove to be overly stringent due to the constraints it places on the battery compartment. While the lander inner volume scales as the third power of the compartment dimensions, the area scales to the second power. Hence, when scaling to a smaller size model, the fraction of the lander volume occupied by the equipment compartment increases and that occupied by the battery compartment decreases since both lander volume and equipment compartment volume must scale as the area ratio. Ultimately, a point may be reached whereby allowable volume for the model battery compartment is inadequate to contain all required equipment. In this case, application of some heat sink material (high specific heat, high density) or PCM in the equipment compartment will reduce the required equipment weight and volume while retaining the required total heat capacity.

Battery Compartment - The battery compartment is idealized as an isothermal composite of battery, equipment, phase change material, and isotopes. As discussed above, for a smaller scale model, the relative size of the battery compartment volume compared to the overall lander volume is reduced. This therefore, can force the use of a greater percentage of heat sink to equipment and battery than contained in the nominal lander, in order to satisfy the heat storage requirement of Equation 8. The relationship between total heat capacity in the prototype battery compartment and that in the model is given by Equation 13:

$$\begin{aligned} [(\rho V c_p)_E + (\rho V c_p)_B + (\rho V c_p)_{PCM}]_m &= [(mc_p)_E + (mc_p)_B + (mc_p)_{PCM}]_m \\ &= \frac{A_m}{A_p} \sum (mc_p)_p \end{aligned} \quad 13)$$

where: E, B, and PCM refer to equipment, battery, and phase change material respectively, and the summation is for the respective prototype quantities.

Since both the area ratio and the prototype lander heat capacities are known, the relative amounts of battery and phase change material can be determined to satisfy both the scaling ratios and the volume restrictions.

Application of the Scaling Results - The preceding describes a method of extending the results of the initial sizing analysis to other size landers. Extension of the technique using modified area/volume relationships between the battery and equipment compartment is also discussed. For some possible configurations, the landers may require the addition of heat sinks, such as phase change material, to the model equipment compartment, and scaling with a summation of the entire heat capacity (mc_p). For the present case, however, the "family" of landers has been defined to be true scale models of the prototype design. Through the proper implementation of scale factors, the lander compartment volumes, weights, power levels, and areas can be derived. The appropriate scaling ratios are shown in Figure 3.2.6-21 to define a range of lander sizes which have identically the same thermal response as a relay mode (constant power profile) lander. The relay mode temperature response shown in Figure 3.2.6-16 has been modified slightly to correspond more closely with direct transmission mode temperature response. This was accomplished through a reduction in the phase change material weight used in the initial evaluation, and total equipment compartment weights, enabling maximum utilization of all thermal control weight. The scaling parameters for the prototype lander are shown in Table 3.2.6-4 and have been used in conjunction with the overall surface payload parametric analysis to define thermal control requirements for the point designs of Section 4.

REQUIRED THERMAL CONTROL SCALING RATIOS

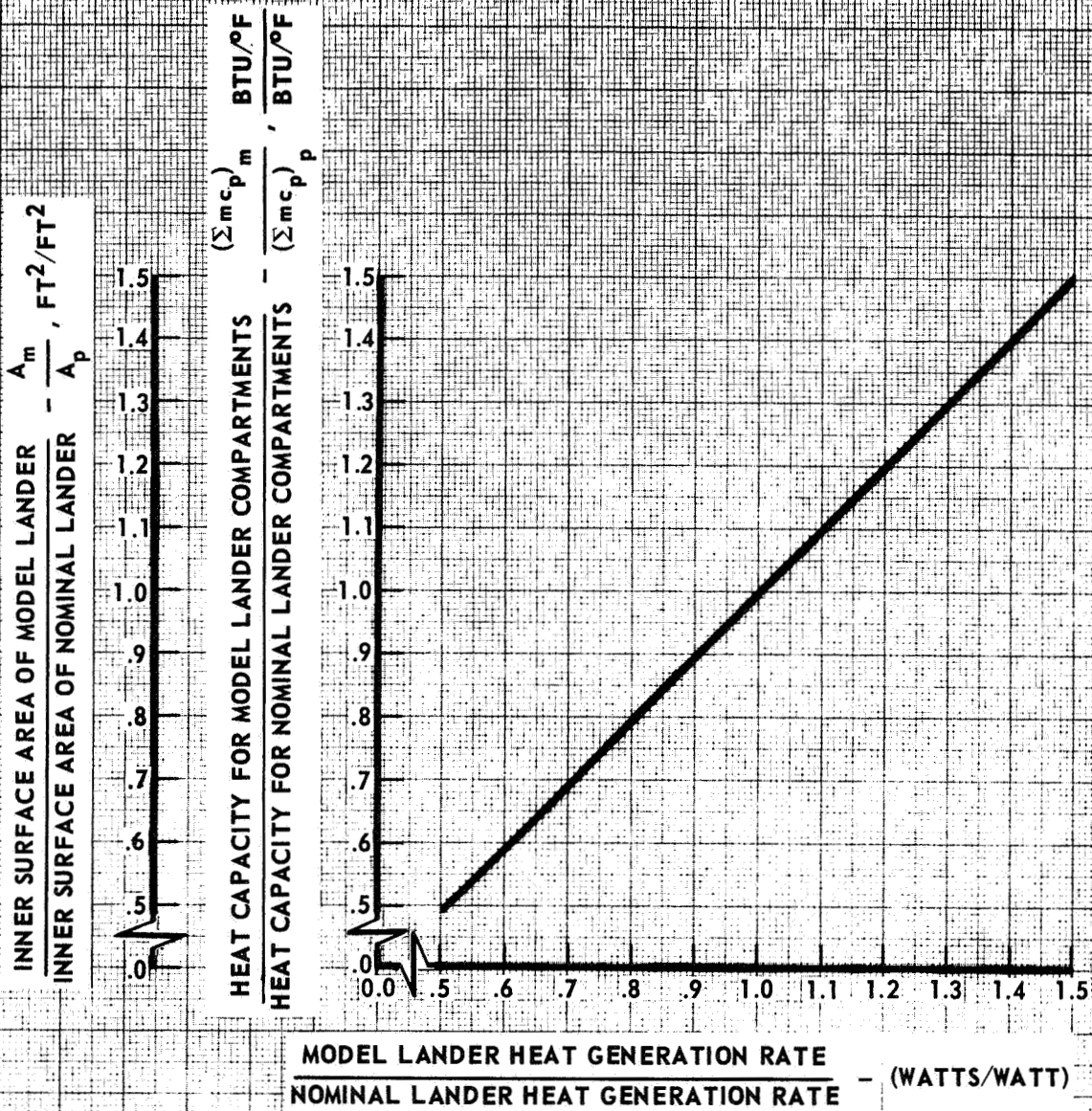


FIGURE 3.2.6-21

3.2.7 PROPULSION - The Mars capsule requires propulsion to deorbit the vehicle from its Mars parking orbit, to control attitude from capsule-orbiter separation to lander touchdown, and to effect the final deceleration required for a soft landing on the Mars surface. Candidate propulsion concepts were evaluated for each of these functions during the VOYAGER Phase B Study and system selections were made for capsules weighing about 5000 pounds. Much of the data and methodology developed during this earlier study are applicable to the Soft Lander Study. However, subsequent focusing of the Mars exploration program on smaller, austere vehicles warrants re-examination of alternate concepts for the propulsive functions, to determine if cost and development risk can be reduced, perhaps at the sacrifice of system performance.

The terminal propulsion system had previously been identified as a long-lead-time item, and the combined cost of the three propulsive systems amounted to roughly 20% of the total capsule cost. Consequently, the propulsion systems offer significant potential for reductions in program cost and risk.

Cold gas, monopropellant, bipropellant, and solid propellant systems were considered, where applicable to the various mission phases. System comparisons were made on the basis of performance, cost, reliability, development status, and systems interaction.

For the deorbit function, solid propellants provide a weight advantage over liquid monopropellant and bipropellant systems, but considering that a liquid propellant system will be required for the terminal propulsion phase, most of the cost and development risk of a liquid deorbit system could be amortized against the terminal propulsion by utilizing the same hardware for both systems. Even so, a solid rocket is very cost-competitive, due to the low recurring costs. Also, the weight penalty introduced by utilizing common hardware for the two systems would result in higher costs to other capsule systems, and this fact must be considered in the selection of a deorbit propulsion concept.

Comparisons of attitude control concepts include cold gas and monopropellant hydrazine active control; cold gas spin/active; and cold gas active control combined with deorbit TVC. The propulsive attitude control system figures

more strongly into the design of other capsule systems, i.e., the guidance electronics and the guidance power requirements, than do the other propulsion systems and a judicious selection can be made only on an integrated systems level.

Performance of the terminal propulsion system was not strongly emphasized except in those cases where significant cost reductions could be realized by optimizing for minimum system weight and throttle ratio. In this vein, several descent guidance modes were evaluated for the terminal deceleration phase. Tradeoffs were made between propellant requirements and the design thrust-to-lander weight ratio of the terminal propulsion system. A new guidance mode was selected on the basis of these studies, making it possible to reduce engine thrust level by a factor of two over that established by earlier studies. A similar reduction in engine throttle ratio from 10:1 to approximately 5:1 significantly reduces development risk. Bipropellant terminal propulsion systems provide a performance advantage over monopropellant systems, but monopropellants offer an edge in simplicity, cost, and reliability.

3.2.7.1 Deorbit propulsion system - A deorbit propulsion system is required for retrograding the capsule from its Mars parking orbit, thereby placing it on a trajectory which intersects the surface at a predetermined landing site. Several propulsion concepts are suitable candidates for this function. Solid propellant, monopropellant, and bipropellant deorbit systems have been evaluated. A comparison based on weight and performance, reliability, cost, development status, and interactions with other systems is provided below. These results provide the information necessary for supporting system selections for capsule point designs. Final system selections in which guidance, etc. are considered jointly with the propulsion system are discussed in Section 3.3.1 for four selected configurations.

3.2.7.1.1 Requirements: As established in Section 3.1.2, the velocity increment required to deflect the capsule from a synchronous Mars parking orbit varies from 187 ft/sec to approximately 950 ft/sec, depending on deorbit anomaly, entry angle, and landing anomaly. Based on our mission analyses, however, a maximum velocity increment of 450 ft/sec is adequate. The difference in landing anomaly achieved by varying the maximum deorbit velocity from 450

ft/sec to 950 ft/sec is only a few degrees. Consequently, most of the deorbit propulsion parametrics presented in this section are based on a 450 ft/sec deorbit propulsion capability.

Thrust termination capability has been assumed for the deorbit system to provide flexibility in the choice of orbit and landing site.

3.2.7.1.2 Candidate system designs: The deorbit propulsion requirements can be satisfied by several types of propulsion systems. Solid propellant, monopropellant, and bipropellant systems were considered for this study.

Five basic configurations were evaluated for the three system types. In three of these configurations, inert system parts are jettisoned after the deorbit function to reduce capsule entry weight. The other two utilize tankage common to the terminal propulsion system. The five configurations are depicted and designated in Figure 3.2.7-1.

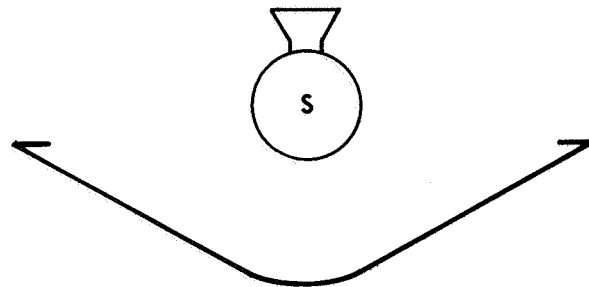
Earth-storable, hypergolic propellants, monomethyl hydrazine (MMH) and nitrogen tetroxide (N_2O_4) were selected for the bipropellant configurations because of their reasonably high performance and their demonstrated ability to survive heat-sterilization. Of the monopropellants, only hydrazine (N_2H_4) has sufficiently high performance to be considered for this application. Saturated polybutadiene/ammonium perchlorate was selected for the solid propellant.

Range dispersion analyses, presented in Section 3.1.2, indicate that deorbit thrust level is not a critical parameter. From a systems design standpoint, however, a low-thrust level is desirable to minimize engine weight and envelope and to maintain deorbit attitude disturbance torques to manageable limits. Thrust for the solid rocket was established on the basis of a minimum propellant burning rate of .3 inch/second and a spherical motor design. The spherical motor offers a high propellant fraction and short centerline length. Thrust level for the liquid propellant concepts was chosen to provide a thrust-to-capsule deorbit weight (Earth) of 0.10 to minimize weight; however, consideration was also given to the use of a single engine design for the deorbit and terminal propulsion functions in evaluating system costs. This latter consideration provides a deorbit thrust-to-weight (Earth) of approximately .26.

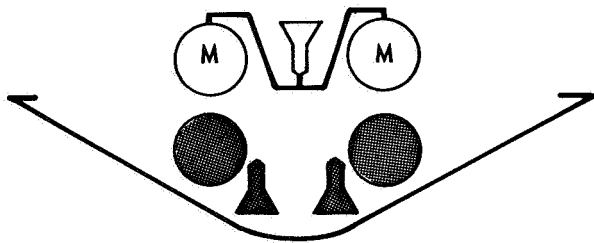
The tolerance on deorbit velocity is one of the largest contributors to downrange touchdown dispersions. Liquid system impulse is simply and

CANDIDATE DEORBIT SYSTEMS

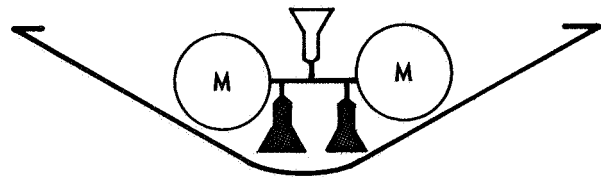
SOLID PROPELLANT



MONOPROPELLANT

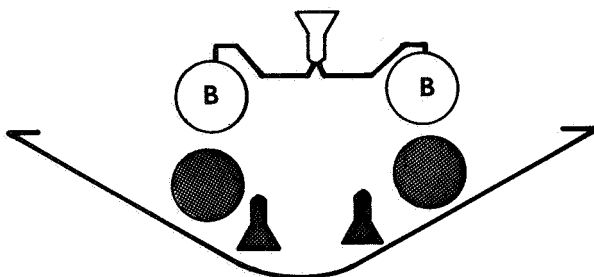


SEPARATE SYSTEMS

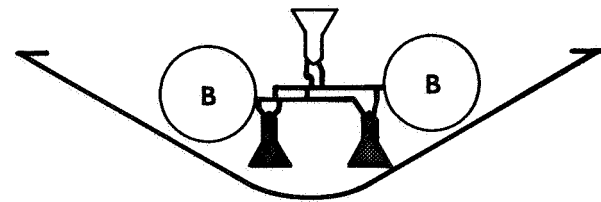


(COMPOSITE SYSTEM)

BIPROPELLANT



SEPARATE SYSTEMS



(COMPOSITE SYSTEM)

FIGURE 3.2.7-1

accurately controlled by closing the engine propellant valves. Total impulse repeatability for the liquid systems is approximately .015% (3σ). On a fixed impulse solid rocket, the variation in total impulse is 1.0% (3σ). This variation can be reduced to .37% and mission flexibility enhanced by incorporating thrust termination capability in the solid rocket. For a deorbit anomaly of 195° , these tolerances contribute to downrange landing dispersions of approximately .6 km and 15.2 km for liquid and solid deorbit systems, respectively. Rapid depressurization thrust termination was therefore assumed for the solid. This would be achieved by jettisoning the nozzle assembly, as was done on the Jupiter and Titan vernier control rockets.

3.2.7.1.3 System trade studies: The candidate design concepts were evaluated for performance, reliability, cost, development status, and system interactions to support system selections for the reference capsule designs.

Performance - The major factors to be considered under system performance are weight and envelope. The candidate systems were compared on the basis of total system weight and incremental surface payload weight. The latter parameter illustrates the penalty incurred with the composite systems in carrying expended deorbit hardware to lander impact. On all other configurations, the inert parts are jettisoned after deorbit.

Since the deorbit motor/engine is mounted on the capsule roll axis, the critical system dimension is centerline length. For the liquid systems, this dimension is assumed to be the maximum propellant tank diameter or the combined length of the engine and pressurant tank, whichever is greater. The propellant tanks are located as close to the thrust axis as possible, to minimize thrust disturbance torques due to radial c.g. shifts.

Solid rocket performance is illustrated in Figure 3.2.7-2, which shows the effect of aluminum content and total solids loading, i.e., aluminum and ammonium perchlorate, on the specific impulse of a typical polybutadiene propellant. The curve also shows the restriction placed on solids loading to provide reasonable assurance that the motor will survive heat sterilization. The corresponding propellant physical properties following sterilization are discussed under Development Status. The propellant formulation considered for this study contains 85% total solids consisting of 16% aluminum and 69%

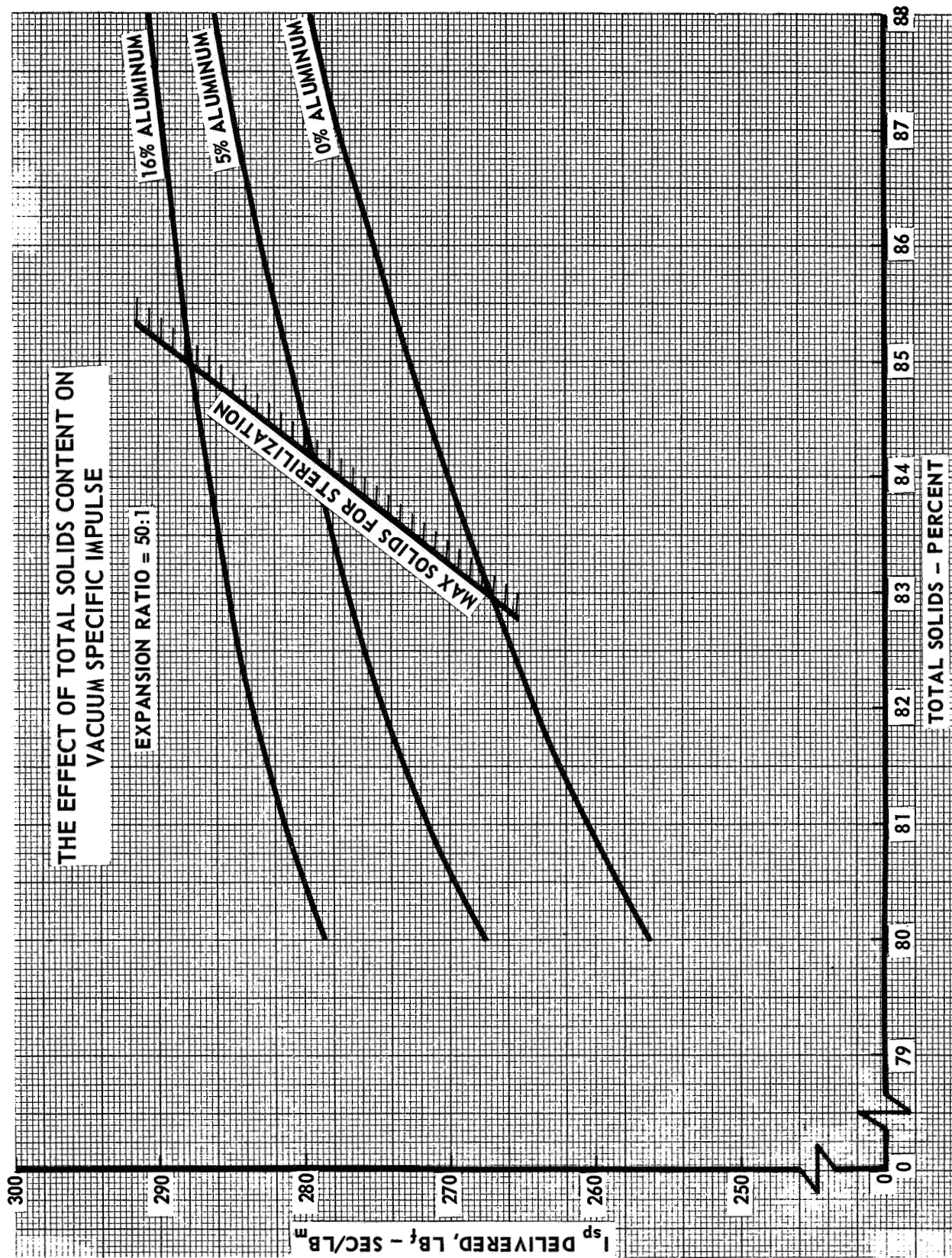


FIGURE 3.2.7-2

ammonium perchlorate. For the assumed aluminum content, a 2% reduction in total solids loading would introduce a performance penalty of only 1%.

Motor weight and length are shown as a function of total capsule weight and nozzle expansion ratio in Figure 3.2.7-3. A nozzle expansion ratio of 50:1 was selected as near-optimum, providing a minimum capsule weight. The estimated vacuum specific impulse is 288 sec.

For liquid propellant system, chamber pressure was determined to be the most critical factor in optimizing performance. Low chamber pressures minimize propellant tank pressurization requirements but require larger, and consequently heavier, engines for the design thrust-to-weight (Earth) of 0.10. The effect of chamber pressure on system weight and engine length is shown in Figure 3.2.7-4 for both the mono- and bipropellant systems. The bipropellant system is seen to optimize at a chamber pressure of approximately 75 psia, while the monopropellant, unaffected by kinetic performance losses at low chamber pressure, optimizes at approximately 50 psia.

Except for the composite systems, propellant orientation for the liquid systems is achieved by firing aft-directed attitude control engines prior to deorbit. This technique is impractical for the composite systems since the thrust loads applied during the deorbit and terminal deceleration functions are opposite in direction. To orient the propellants before burn by auxiliary means and maintain them in that position during engine operation would require outlets on each end of the propellant tanks. Positive expulsion with reinforced metallic diaphragms was selected as a more reasonable approach.

Weight and dimensional characteristics of the candidate designs are summarized in Figure 3.2.7-5 for a deorbit velocity of 450 ft/sec. The effect of deorbit velocity on system weights is shown in Figure 3.2.7-6.

Reliability - The reliability of each of the candidate systems is presented in Table 3.2.7-1. The basic failure rate data utilized in this study was obtained from other programs employing similar components. The reliability figures do not include the effect of heat sterilization; thus, the actual estimates must be considered relative. Aerojet (Aerojet General Rept. 2470-01F, "Space Vehicle Solid Rocket Analysis and Correlation Study", 8 Sept. 1967)

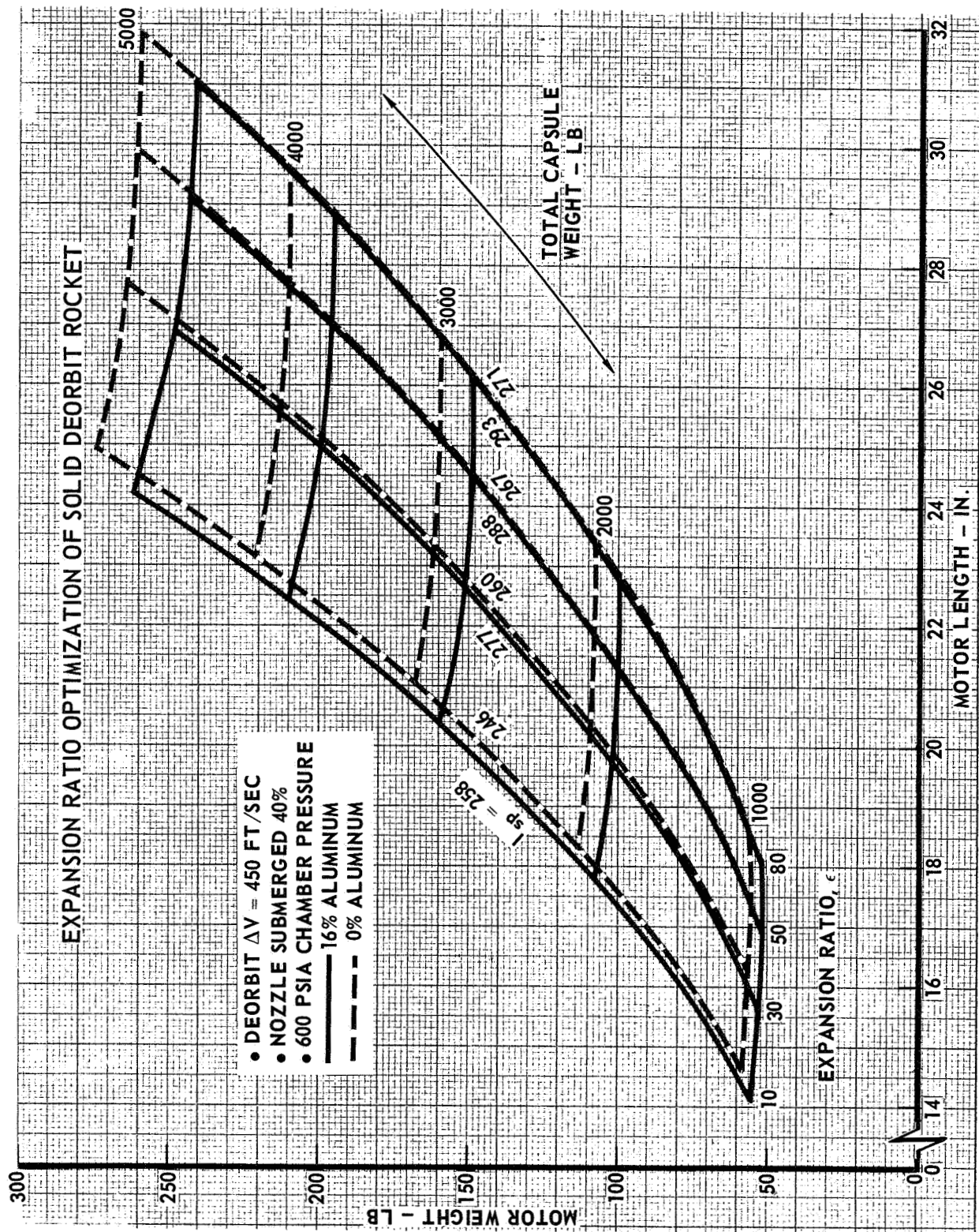


FIGURE 3.2.7-3

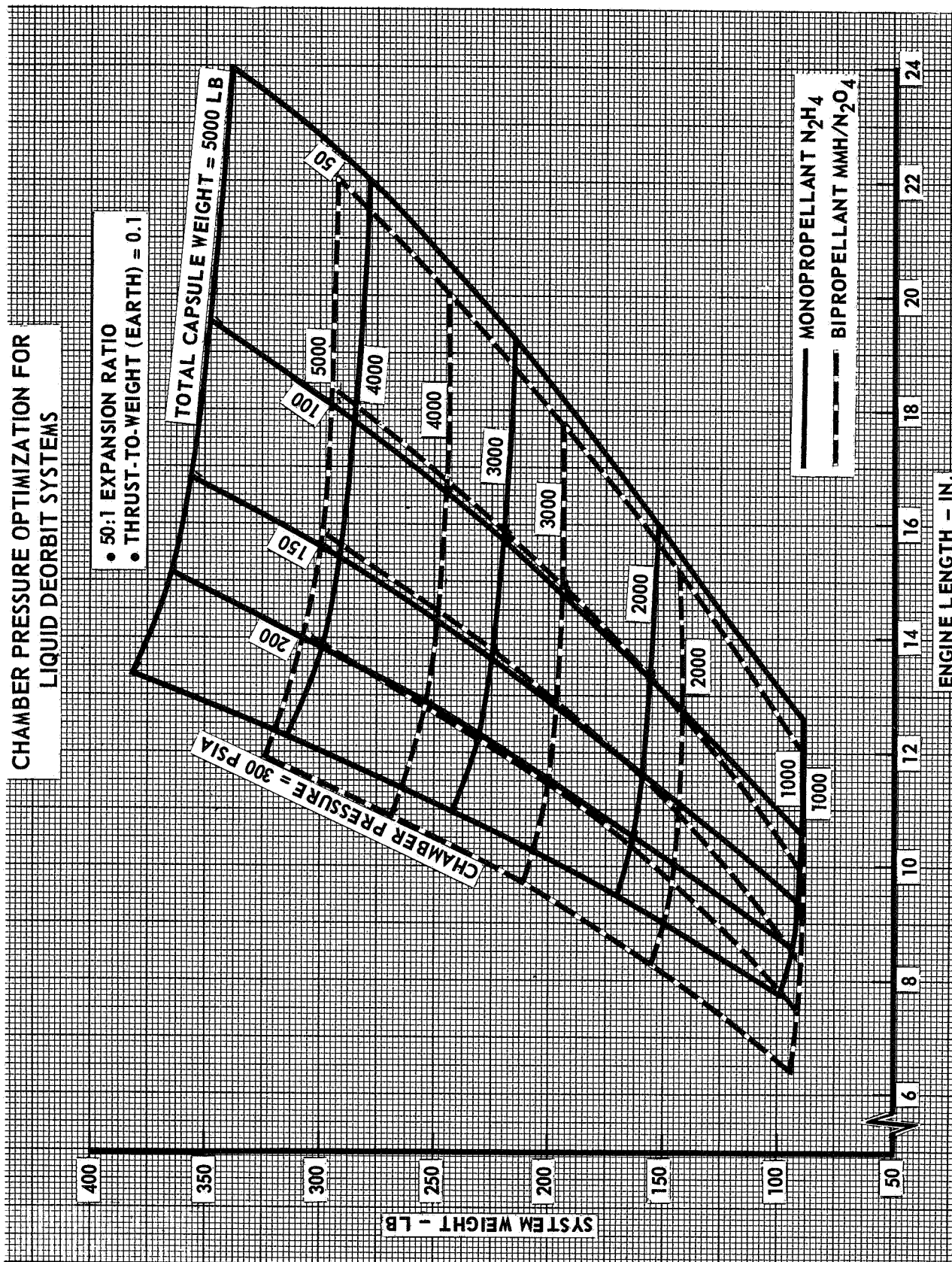


FIGURE 3.2.7-4

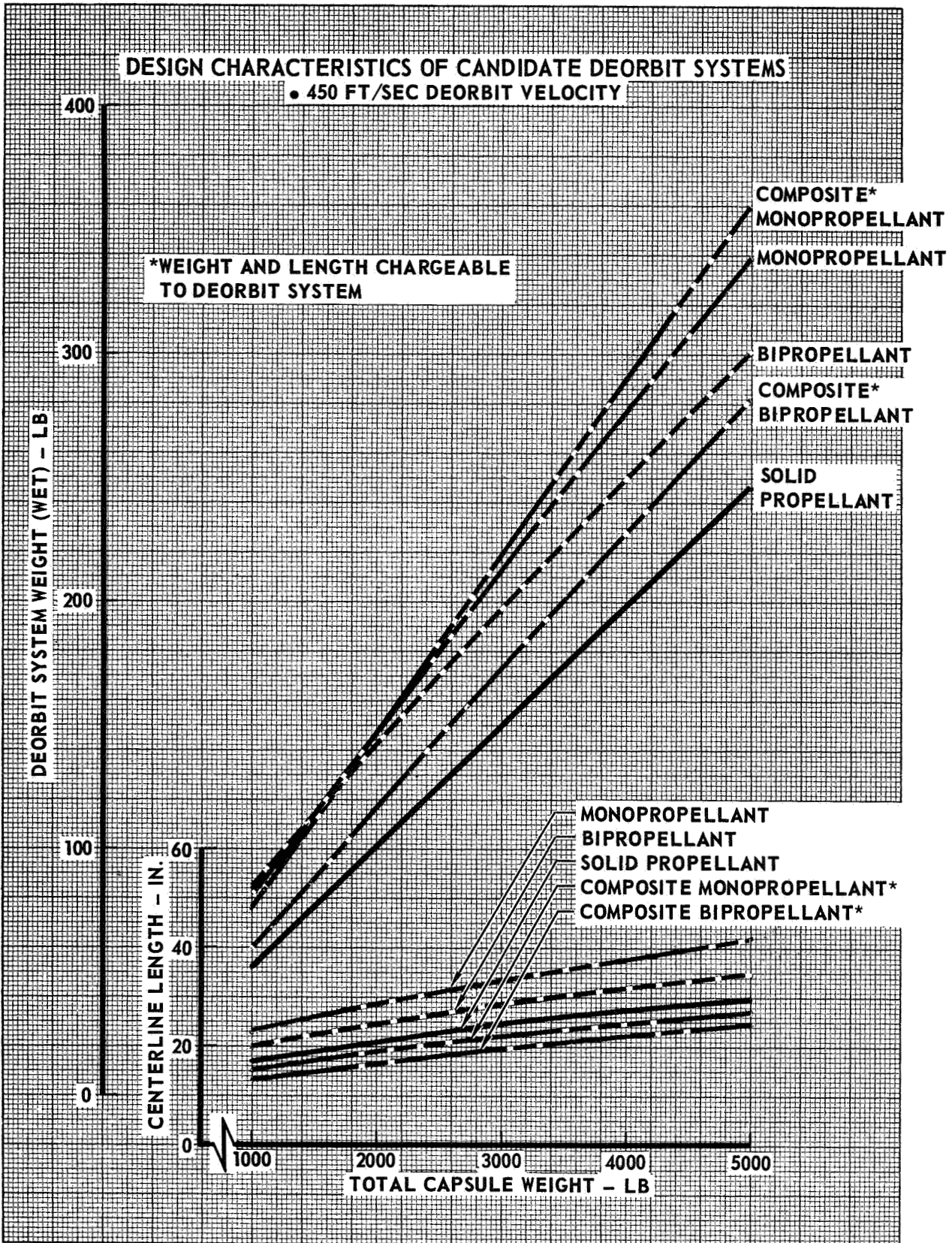


FIGURE 3.2.7-5

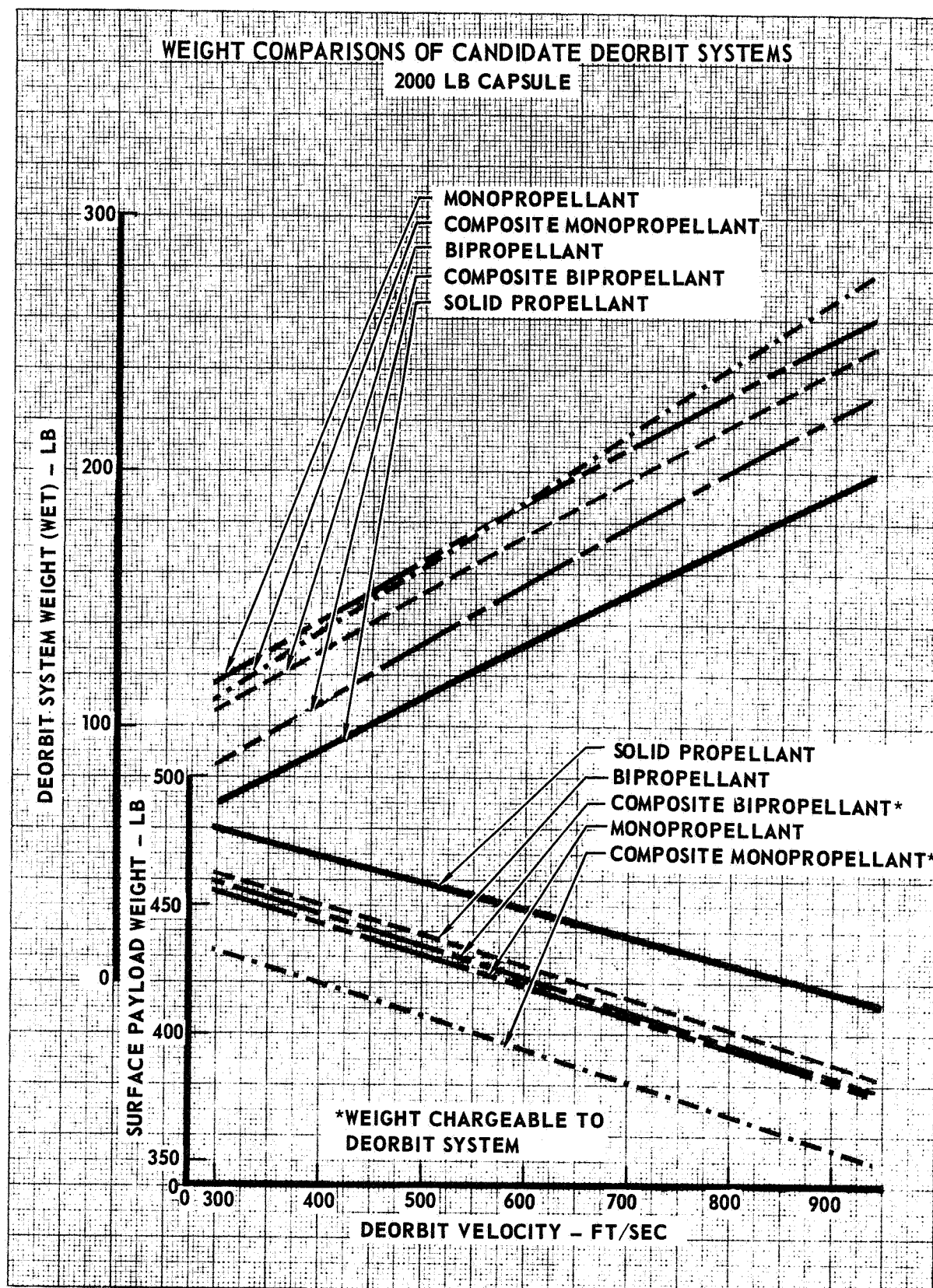


FIGURE 3.2.7-6

TABLE 3.2.7-1
DEORBIT SYSTEM RELIABILITY ANALYSIS

CANDIDATE SYSTEM		SOLID			MONOPROPELLANT			BIROPELLANT			COMPOSITE* MONOPROPELLANT			COMPOSITE* BIROPELLANT		
	FAILURE RATE EACH COMPONENT (λ) · 10 ⁻⁶	NO. OF COMPON (n)	DUTY CYCLE (k · t)	n/λ · t x 10 ⁶	NO. OF COMPON (n)	DUTY CYCLE (k · t)	n/λ · t x 10 ⁶	NO. OF COMPON (n)	DUTY CYCLE (k · t)	n/λ · t x 10 ⁶	NO. OF COMPON (n)	DUTY CYCLE (k · t)	n/λ · t x 10 ⁶	NO. OF COMPON (n)	DUTY CYCLE (k · t)	n/λ · t x 10 ⁶
PROPELLANT (SOLID)	300 CY	1	1 CY	300												
MOTORCASE	20 CY	1	1 CY	20												
INSULATION	650 CY	1	1 CY	650												
NOZZLE	650 CY	1	1 CY	650												
IGNITER	200 CY	1	1 CY	200												
THRUST TERMINATION	127 CY	1	1 CY	127												
PRESSURANT TANK	.08 HR				1	5561 HR	445	1	5561 HR	445						
PRESSURANT FILTER	25 CY	1	1 CY	25	1	1 CY	25	1	1 CY	25						
PRESSURE REGULATOR	16/HR	1	87 HR	1392	1	87 HR	1392	1	87 HR	1392	1	5 HR	80	1	5 HR	80
PYRO VALVE	250 CY	7	1 CY	1750	7	1 CY	1750	8	1 CY	2000	1	1 CY	250	2	1 CY	500
CHECK VALVE	1 CY				2	1 CY	20	2	50 CY	100						
BURST DIAPHRAGM	10 CY				2	1 CY	20	2	1 CY	20						
PROPELLANT TANK	53.5 CY	2	2 CY	214	2	2 CY	214	2	2 CY	214						
PROPELLANT FILTER	52 CY	1	1 CY	52	1	1 CY	52	2	1 CY	104						
ENGINE VALVE	127 SEC	1	140 SEC	17780	1	140 SEC	17780	2	140 SEC	35560	1	36 SEC	4572	2	36 SEC	9144
THRUST CHAMBER (MONOPROPELLANT)	5.2 SEC	1	140 SEC	728	1	140 SEC	728				1	36 SEC	187			
THRUST CHAMBER (BIROPELLANT)	10 SEC							1	140 SEC	1400				1	36 SEC	360
PROPELLANT TANK W EXPULSION DEVICE	357 CY										2	1 CY	714	2	1 CY	714
$\sum n \cdot \lambda \cdot kt \cdot 10^{-6}$				1947			22406			41260			5803			10798
SYSTEM RELIABILITY $R = e^{-\sum n \cdot \lambda \cdot kt}$			.9981			.9776			.9587			.9942			.9892	

* FAILURE RATES ASSOCIATED WITH DEORBIT FUNCTION

has estimated the effect of heat sterilization on the solid rocket as an increase of 680 failures per million firings. Heat sterilization effects on the reliability of liquid systems have not been evaluated.

Based on data available, the candidate systems have been ranked in order of decreasing reliability, as follows: solid, composite monopropellant, composite bipropellant, monopropellant, and bipropellant.

Cost - The relative costs of the candidate concepts are compared in Figure 3.2.7-7, as a function of total capsule weight. The data includes the costs for development, qualification, and delivery of six systems. Since heat sterilization of solid motors remains to be demonstrated, a new development program was assumed for that concept. For the liquid propellant concepts, cost data were derived, first on the basis of new system developments and second on the basis of using the same hardware as the liquid terminal propulsion system. Under the latter assumption, the cost for development, qualification and delivery of the first six systems was amortized against the terminal propulsion system. Hardware commonality between the two systems is seen to provide a very significant reduction in the cost of the liquid systems. Surprisingly, the solid propellant rocket remains cost competitive, primarily because the recurring costs for liquid systems are an order of magnitude higher than the solid system.

Development Status - The most critical development problems are heat sterilization and chemical decontamination.

The feasibility of heat sterilizing liquid systems has been demonstrated by JPL, Martin, and McDonnell. Nitrogen tetroxide is extremely corrosive at sterilization temperatures and titanium is the only known metal suitable for component use. Hydrazine does not attack titanium or 300 series stainless steel, but it has been observed to decompose when in contact with these metals at sterilization temperature. Titanium is preferred since it is less active than stainless steel. Monomethyl hydrazine possesses greater thermal stability than hydrazine and is suitable for use with both titanium and stainless steel.

COST COMPARISONS OF CANDIDATE DEORBIT CONCEPTS

- TOTAL PROGRAM COST
- UNSTERILE SYSTEMS
- DELIVERY OF SIX SYSTEMS

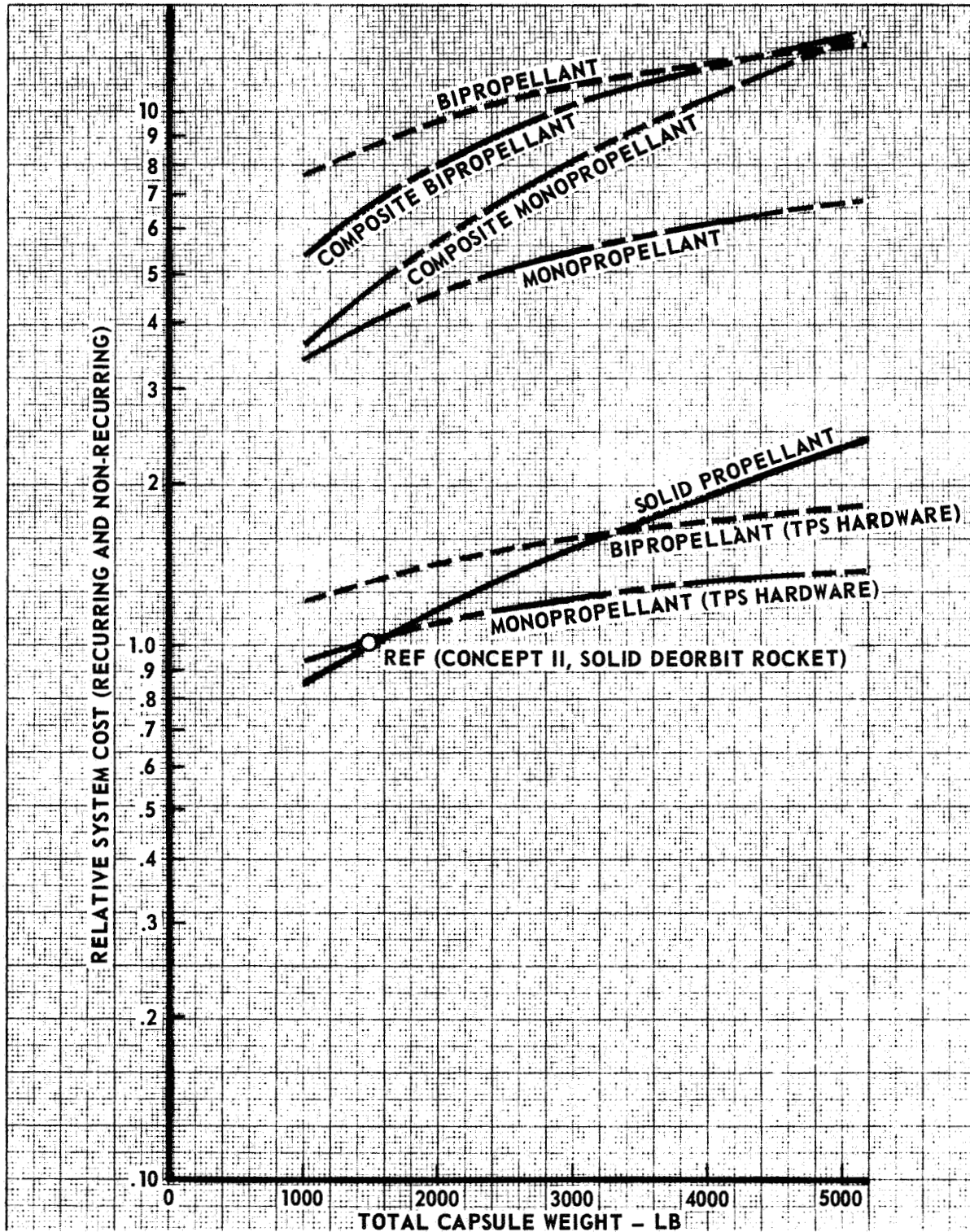


FIGURE 3.2.7- 7

The limited choice of materials which are compatible with earth-storable propellants at sterilization temperature suggests a system design wherein propellant is isolated from fluid control components during heat exposure. This requires new design techniques to overcome shortcomings in the physical properties of propellant-compatible materials.

The effect of chemical decontamination using an ethylene oxide-Freon 12 mixture must also be considered. Tests at McDonnell have shown that Shell 405 catalyst is poisoned by exposure to this mixture. Hence, it will be necessary to hermetically seal the hydrazine reactor in order to isolate the catalyst from the decontaminant.

The requirement for sterilizing a solid propellant system by heating the entire motor assembly for 6 cycles at 275°F has caused serious degradation of current off-the-shelf propellants, liners, and insulations. Material incompatibilities also exist because of differences in thermal expansion between the different materials. Testing has been performed on new propellant formulations developed specifically to withstand thermal sterilization and on modifications of existing propellants. In addition, system components such as liners, insulation, O-rings, nozzle, and igniters have been investigated. Table 3.2.7-2 summarizes the major failure modes associated with the heat sterilization of a solid rocket and the corrective measures most successful in removing the deficiencies. The corrective modifications are mutually compatible. Also shown are the measured physical properties of Aerojet ANB-3289-2 propellant before and after six 275° sterilization cycles. The propellant properties are considered adequate for high performance motor designs.

Appreciable effort has been spent in recent years to develop high energy propellants with increased solids loading, tailored to withstand low temperature strain requirements. The addition of plasticizers and a low curing-agent-to-polymer ratio improved physical properties of propellants at low temperature. The existence of water in the raw materials was not particularly detrimental. However, at high temperature (275°F) all of these conditions have adverse effects on the cured propellant. These conditions can be **eliminated** to enhance

TABLE 3.2.7-2
STERILIZATION FAILURE MODES OF
SOLID PROPELLANT ROCKET

FAILURE MODE	CAUSE OF FAILURE	CONTROL MEASURES
CRACKING	THERMAL STRESSES, EXCESS DEFORMATION	STRAIN RELIEF BOOT, GOOD PROPELLANT PHYSICAL PROPERTIES
VOIDS	VISCOUS MIX, CHEMICAL INSTABILITY	CHOICE OF CURING AGENT, BAYONET CASTING, VACUUM STRIPPING OF LOW MOLECULAR WEIGHT SPECIES FROM RAW MATERIALS
SLUMP	LOW EQUILIBRIUM MODULUS	GOOD PHYSICAL PROPERTIES, ADEQUATE GRAIN SUPPORT
SEPARATIONS	EXCESS MOISTURE, PLASTICIZERS, SHRINKAGE	MATERIAL SELECTION (UNHYDRATED FILLERS, NO PLASTICIZERS)
AUTO IGNITION	DECOMPOSITION EXOTHERMS	RECRYSTALLIZATION OF OXIDIZER, SATURATED PREPOLYMER
IGNITION FAILURE	DESENSITIZED INITIATOR, PROPELLANT DEGRADATION	APOLLO STANDARD INITIATOR, SAME PROPELLANT MODIFICATIONS AS MAIN CHARGE
BALLISTIC EXCURSIONS	OXIDIZER DECOMPOSITION, MICROVOIDS	RECRYSTALLIZED OXIDIZER, HIGH BINDER CROSS-LINK DENSITY

**EFFECT OF HEAT STERILIZATION ON THE MECHANICAL PROPERTIES OF AEROJET
ANB-3289-2 PROPELLANT**

- MECHANICAL PROPERTIES AT 77°F
- 85 PERCENT SOLIDS LOADING

NUMBER OF HEAT STERILIZATION CYCLES	MAX STRESS S_{nm} (PSI)	ELONGATION AT MAX STRESS γ_m (PERCENT)	ELONGATION AT BREAK γ_b (PERCENT)	MODULUS E_o (PSI)
0	136	21	33	1708
6	102	24	44	1305

the high temperature stability of the propellant if the low temperature requirement is removed. This would be accomplished with a small amount of active thermal control during Earth-Mars transit.

Despite potential difficulties, sufficient sterilization testing has been accomplished to indicate that either liquid or solid propellant systems can be exploited for planetary landing missions. However, without early commitment to a sterilizeable solid motor development program, only a liquid system will offer reasonable assurance of meeting a launch opportunity in 1973.

System Interactions - The major interaction between the deorbit propulsion and other systems is related to engine exhaust effects during deorbit burn. Areas of primary concern are: exhaust-induced surface contamination and upsetting blast impingement loads on the orbiter. Contamination of orbiter solar cells, optics, or other sensitive surfaces may be alleviated by utilizing clean-burning propellants or by providing adequate separation between the capsule and orbiter at deorbit ignition. Only a large separation distance is effective in keeping impingement disturbance torques to manageable levels. A minimum separation of 300 meters has been assumed for this study. McDonnell studies on the Apollo Experiment Pallet (AEP) indicate that a liquid propellant system will not present a problem to the orbiter at this distance. On the other hand, alumina in the exhaust of high energy solid propellants presents significant contamination potential. Test observations show that alumina particles in a rocket exhaust release their kinetic energy at impact, causing them to melt and resolidify on the impinged surface. Because of the high performance gains available with aluminized solid propellants (approximately 8%), the deorbit space-time relations between the capsule and orbiter were investigated as part of VOYAGER Phase B studies for a 1973 mission. Based on deorbit anomalies of 190 to 240 deg and deorbit attitudes of -20 to +60 deg, these studies showed that the "look angle", between the orbiter and exhaust nozzle centerline precluded exhaust impingement during the first 280 ft/sec deorbit velocity increment. The range at that point was ≥ 1200 meters (300 meters initial). For a maximum deorbit velocity of 950 ft/sec, the total aluminum accumulation level was less than 9 milligrams/ft².

Alumina contamination levels of this magnitude do not pose a threat to orbiter operation.

3.2.7.1.4 Summary and conclusions: Results from studies which have included solid propellant, monopropellant and storable bipropellant deorbit systems are summarized in Table 3.2.7-3. As indicated in the table, a solid propellant rocket provides advantages in weight and reliability and, for new system developments, it is less costly than liquid propellant systems of comparable performance. On the other hand, a liquid propellant system is required for terminal propulsion and the use of common hardware for both the terminal and deorbit systems significantly reduces the overall cost and requires development of only one system type. Liquid propellant systems will also pose less development risk in meeting the heat sterilization requirements. Sterilization feasibility has been successfully demonstrated on systems containing both monopropellant hydrazine and the earth-storable bipropellants, monomethyl hydrazine and nitrogen tetroxide. The largest solid rocket to survive six cycles at 275°F contained only 16 lb propellant. Solid propellant requirements for a 450 ft/sec deorbit velocity increment vary from 43 lb for a 1000 lb capsule to 212 lb for a 5000 lb capsule.

The composite (deorbit/terminal) systems offer no significant advantage over duplicate systems for the two propulsive functions. Performance is poor, mainly because of the high propellant tank pressures required for the throttleable terminal propulsion engines, and because the inert system parts must be carried to lander touchdown.

The above trade studies were utilized in the selection of deorbit propulsion systems for the point designs. Because of the impact of deorbit system performance and configuration on other capsule systems, these selections may appear inconsistent with selections based on propulsion considerations alone. For instance, a selection based on the lowest system cost may be incompatible with the lowest vehicle cost because of system performance penalties. These factors are discussed in Section 3.5, Cost Trends.

TABLE 3.2.7-3
DEORBIT PROPULSION SYSTEM COMPARISON SUMMARY CHART

TYPE SYSTEM	PROPELLANTS	WEIGHT		RELATIVE COST **		RELIABILITY	DEVELOPMENT STATUS	SYSTEM INTERACTIONS
		1000 LB CAPSULE	5000 LB CAPSULE	1000 LB CAPSULE	5000 LB CAPSULE			
SOLID PROPELLANT	HTPB/AP/AL	52	246	.84	2.21	.9981	LOW - REQUIRES FULL SCALE DEMONSTRATION FOR STERILIZATION	MODERATE - ALUMINA IN ROCKET EXHAUST MUST BE EVALUATED AS TO EFFECTS ON ORBITER. INERTS EASILY JETTISONED AFTER DEORBIT
MONOPROPELLANT	HYDRAZINE	83	339	3.40 (.93)*	6.70 (1.38)*	.9776	GOOD - COMPONENT AND PROPELLANT MATERIAL COMPATI- BILITY MUST BE DEVELOPED FOR 275°F	GOOD - CLEAN EXHAUST
COMPOSITE MONOPROPELLANT	HYDRAZINE	76	358	3.69	13.50	.9942	SAME AS MONOPRO- PELLANT	POOR - SYSTEM INERTS MUST BE CARRIED TO TOUCHDOWN, COMPLI- CATING ENTRY CONFIG- URATION
BIPROPELLANT	NTO/MMH	84	301	7.60 (1.20)*	13.20 (1.82)*	.9587	GOOD - MORE PRO- PELLANT/MATERIAL COMPATIBILITY TESTING REQUIRED BUT PROPELLANTS ARE THERMALLY STABLE @ 275°F	GOOD - CLEAN EXHAUST
COMPOSITE BIPROPELLANT	NTO/MMH	61	283	5.30	13.80	.9892	SAME AS BIPROPEL- LANT	POOR - SYSTEM INERTS MUST BE CARRIED TO TOUCHDOWN, COMPLI- CATING ENTRY CONFIG- URATION

* COSTS BASED ON HARDWARE COMMON TO TERMINAL PROPULSION

** VALUES SHOWN ARE REFERENCED TO COSTS OF SOLID PROPELLANT DEORBIT ROCKET FOR 1450 LB
CAPSULE (CONCEPT II, SECT 4.0)

3.2.7.2 Attitude Control. - The attitude control system provides capsule control after separation from the orbiter through atmospheric entry. Specifically, the attitude control system provides for accurate pointing of the deorbit velocity vector, limited vehicle angle-of-attack excursions upon atmospheric entry, and a stable platform for entry science measurements. Attitude control is not required during the parachute deceleration interval, and control during the terminal propulsion phase is maintained by differential throttling of multiple engines, as was utilized by Surveyor.

Selection of the control scheme which best satisfies these requirements depends upon the type of deorbit system employed. Therefore, control concepts for solid and liquid deorbit systems are analyzed separately. Parametric data are presented to show the effect of total capsule weight and deorbit velocity increment on attitude control system design, and candidate concepts are compared on the basis of weight, reliability, cost and development status.

Analyses of the integrated guidance and control, electrical, and propulsion concepts are required to select the optimum attitude control system. This section contains a description of design constraints and supporting analyses for the various propulsion concepts only.

3.2.7.2.1 Requirements: Attitude control and stabilization requirements established for this study are presented by mission phase in Table 3.2.7-4 and are briefly discussed in the following paragraphs.

Separation - A velocity increment of 1.5 ft/sec was established as adequate to separate the capsule from the orbiter. With a reaction control system, aft-directed pitch and yaw engines can provide the necessary translational impulse. Reverse control logic is used for attitude hold during the separation maneuver. That is, with all thrusters operating, the thruster(s) producing a torque in the same direction as the disturbance is momentarily shut-off.

Deorbit Alignment and Pre-Deorbit Limit Cycle - Following separation from the orbiter, the capsule attitude must be maintained within $\pm .25^\circ$ to avoid excessive drift in the guidance and attitude reference. A 20 minute time interval was provided between separation and deorbit.

TABLE 3.2.7-4
ATTITUDE CONTROL AND STABILIZATION REQUIREMENTS

MISSION PHASE	FUNCTION	CONTROL DEADBAND	DESIGN CONSIDERATIONS
SEPARATION	SEPARATE CAPSULE FROM ORBITER	--	SEPARATION VELOCITY = 1.5 FT/SEC
PRE-DEORBIT LIMIT CYCLE		$\pm .25^\circ$	20 MIN TIME INTERVAL. - MINIMIZE EXCESSIVE DRIFT IN GUIDANCE AND ATTITUDE REFERENCE.
DEORBIT	- LIMIT DEORBIT POINTING ERROR - COUNTERACT DISTURBANCE TORQUE DUE TO THRUST MAL-ALIGNMENT.	$\pm .25^\circ$	THRUST MISALIGNMENT = .25 INCHES (3σ)
DESCENT	MAINTAIN ATTITUDE REFERENCE	$\pm 3^\circ$	MINIMIZE EXCESSIVE EXCURSIONS IN ENTRY ANGLE OF ATTACK.
ENTRY	- COUNTERACT AERODYNAMIC DISTURBANCE TORQUES - PROVIDE STABLE PLATFORM FOR ENTRY SCIENCE.	$\pm 3^\circ/\text{SEC}$	DESIGN FOR DYNAMICALLY UNSTABLE ENTRY CONFIGURATION ($C_{Mq\alpha} = .3$)

Deorbit - Attitude must be controlled within $\pm .25^\circ$ during deorbit thrusting to maintain pointing accuracy. The pitch/yaw disturbance torque during the deorbit maneuver was based on a thrust vector - c.g. eccentricity of .25 inches (3σ).

Descent - The capsule attitude requirement during descent was set at $\pm 3^\circ$, to minimize angle-of-attack dispersion at entry. As an alternative to active control, the capsule may be spin-stabilized during the descent phase to conserve power required by the guidance and control electronics. Total descent time was considered to be no greater than six hours.

Entry - For this study the capsule entry configuration was assumed dynamically unstable ($C_{M_{q_{\text{ref}}}} = .3$), and adequate control capability was provided to accommodate sharp-edged wind gusts in the worst-case, VM-10 atmosphere. Rate damping requirements during entry were set at $\pm 3^\circ/\text{second}$.

In addition to deorbit and entry rate damping requirements, the diameter of the aeroshell influences the thrust levels required to maintain attitude control. In order to provide the maximum moment arm, control thrusters were located about the perimeter of the aeroshell. The diameter of the aeroshell increases with capsule weight, but was limited to 14.5 ft to be compatible with a maximum 16 ft diameter bulbous payload fairing. Required thrust levels are presented in Figure 3.2.7-8 as a function of total capsule weight. The selected thrust level must be adequate to counteract entry aerodynamic disturbances as well as deorbit disturbance torques.

3.2.7.2.2 Candidate design concepts: The attitude control concepts considered are summarized in Table 3.2.7-5. Active, spin/active and deorbit thrust vector control schemes were investigated to satisfy mission attitude control requirements. Concepts for solid and liquid deorbit systems were evaluated separately.

Solid Rocket Deorbit Systems - Two combinations were considered for this application. The first is an all-active reaction control concept employing gaseous nitrogen (N_2) or hydrazine (N_2H_4) as propellant. Nitrogen was selected for the cold gas system because of extensive development experience and usage on other programs. The minor advantages offered by a few alternate gaseous

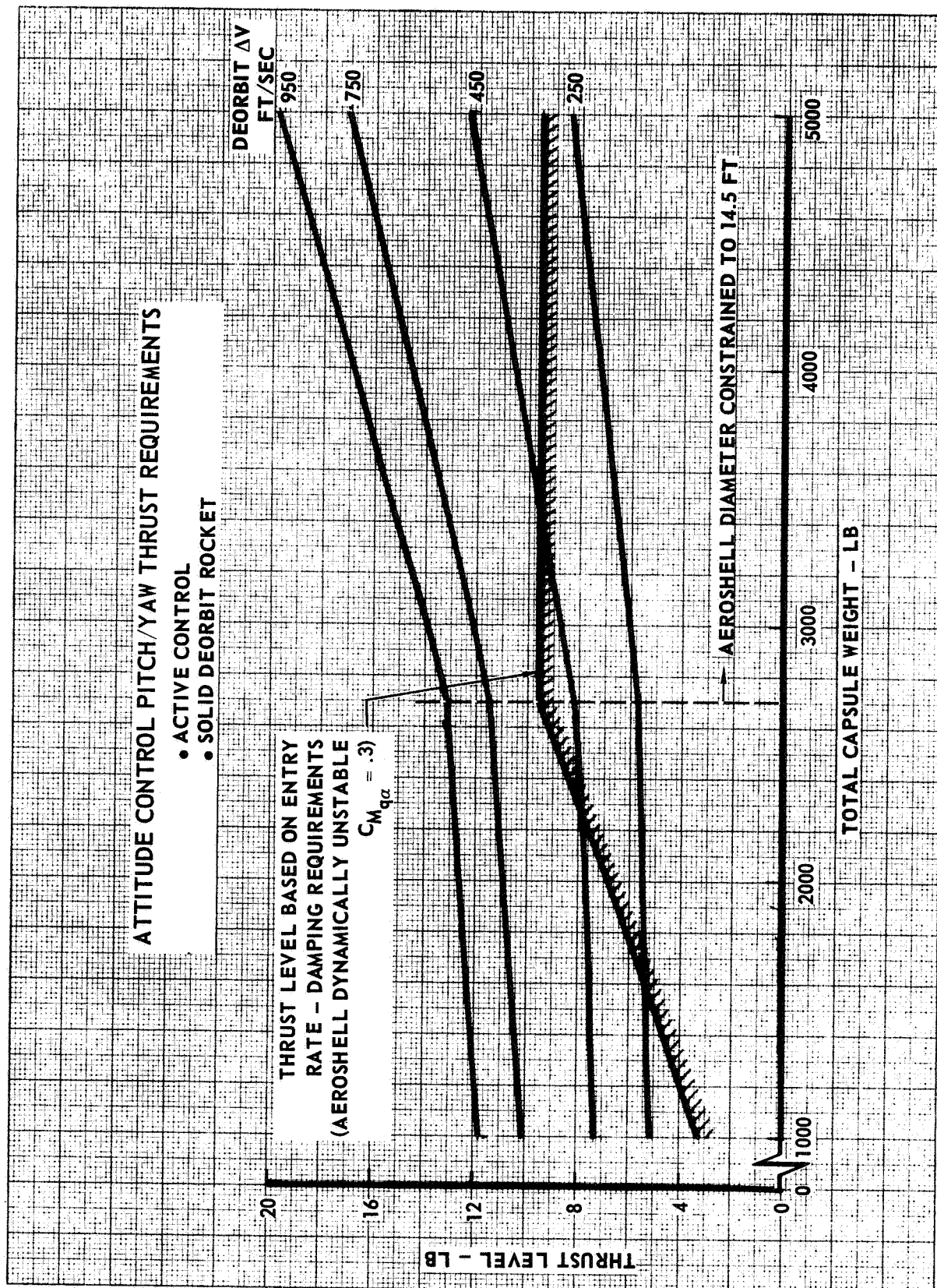


FIGURE 3.2.7-8

TABLE 3.2.7-5

CANDIDATE ATTITUDE CONTROL SYSTEM CONCEPTS

MISSION PHASE	CONTROL CONCEPT			
	SOLID PROPELLANT DEORBIT SYSTEMS		LIQUID PROPELLANT DEORBIT SYSTEMS	
	ACTIVE	SPIN/ACTIVE	ACTIVE	TVC/ACTIVE
SEPARATION	RCS*	RCS	RCS	RCS
DEORBIT	RCS	RCS	RCS	TVC ENGINE GIMBAL
DESCENT	RCS	SPIN	RCS	RCS
ENTRY	RCS	RCS	RCS	RCS

*REACTION CONTROL SYSTEM

propellants did not warrant additional investigation. Hydrazine is the only monopropellant with demonstrated capability for satisfying performance and sterilization constraints. Bipropellants were excluded from consideration due to their high cost, complexity, and relatively small performance margin over N_2H_4 . Schematics of the cold gas N_2 and monopropellant N_2H_4 systems are presented in Figure 3.2.7-9. Four .5 lb_f thrusters were used to provide coupled roll control while pitch/yaw control was obtained by four aft-firing thrusters. A typical installation schematic is presented in Figure 3.2.7-10. For this application it is not practical to utilize pitch and yaw engines in couples, because of problems presented by the forward firing chambers. The exhaust gases would have to expand counter to the supersonic free-stream, creating disturbance torques due to non-uniform pressure distributions at the edge of the aeroshell. Furthermore, the engines so placed would be exposed to high entry heating rates and the possibility of blockage from products of ablation. Propellant storage tanks were located in the nose of the aeroshell to minimize lateral c.g. shift during propellant expulsion.

The second combination considered for the solid rocket deorbit system is a spin/active concept. The configuration is identical to the active system with the exception that two additional thrusters are employed to spin stabilize the capsule immediately following deorbit retrograde. Spin-up is accomplished within 10 sec to minimize coning angles which contribute to angle-of-attack excursions at entry. (Spin thrust requirements are quite low: 2-5 lb_f for a range of capsule weights from 1000-5000 lb.) De-spin is not time critical, and is accomplished using the .5 lb_f roll jets. Spin stabilization prior to deorbit retrograde was not considered in this study due to large angle-of-attack excursions at entry. This problem is discussed in greater detail in Section 3.2.4.

Liquid Deorbit Systems - For the liquid deorbit systems, both all-active control and thrust vector control (TVC) were evaluated. Thrust vector control was employed in the Ranger, Mariner, and Lunar Orbiter propulsion systems. TVC by engine gimbaling was selected for this study since it is the least expensive technique. Furthermore, if a terminal descent type engine is employed for the deorbit maneuver, electromechanical servoactuators qualified

REACTION CONTROL SYSTEM SCHEMATICS

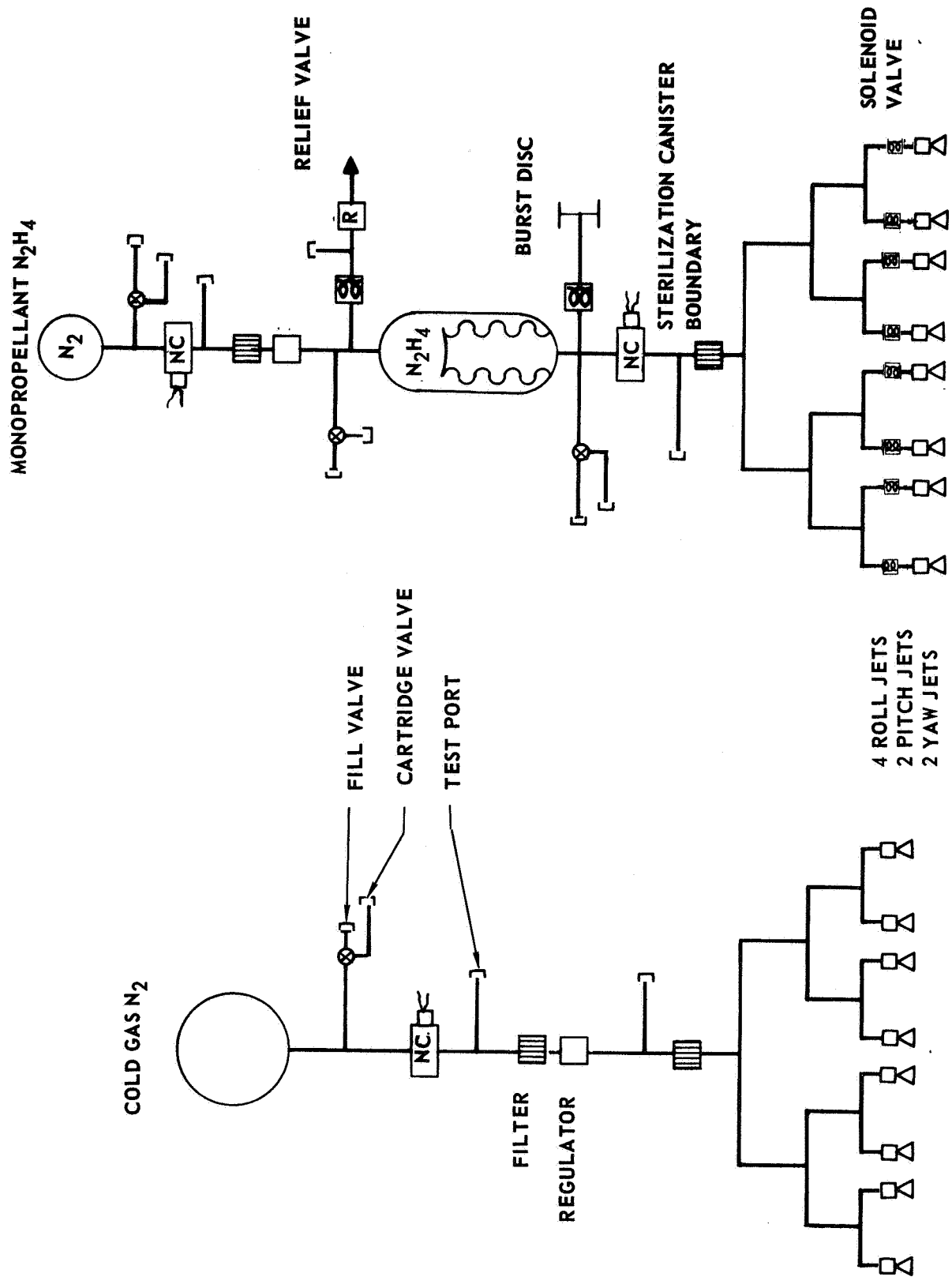
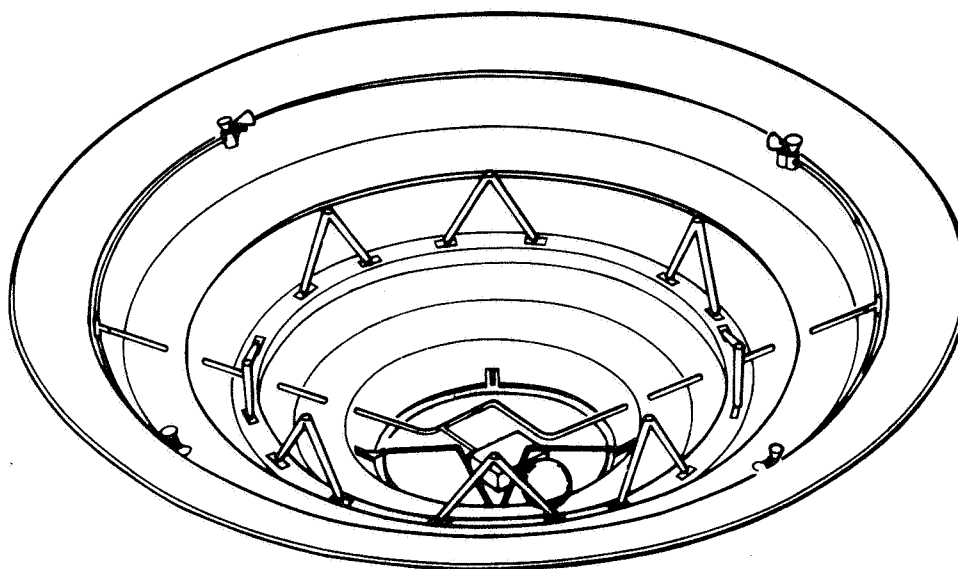


FIGURE 3.2.7-9

REACTION CONTROL SYSTEM INSTALLATION SCHEMATIC

COLD GAS N_2



MONOPROPELLANT
 N_2H_4

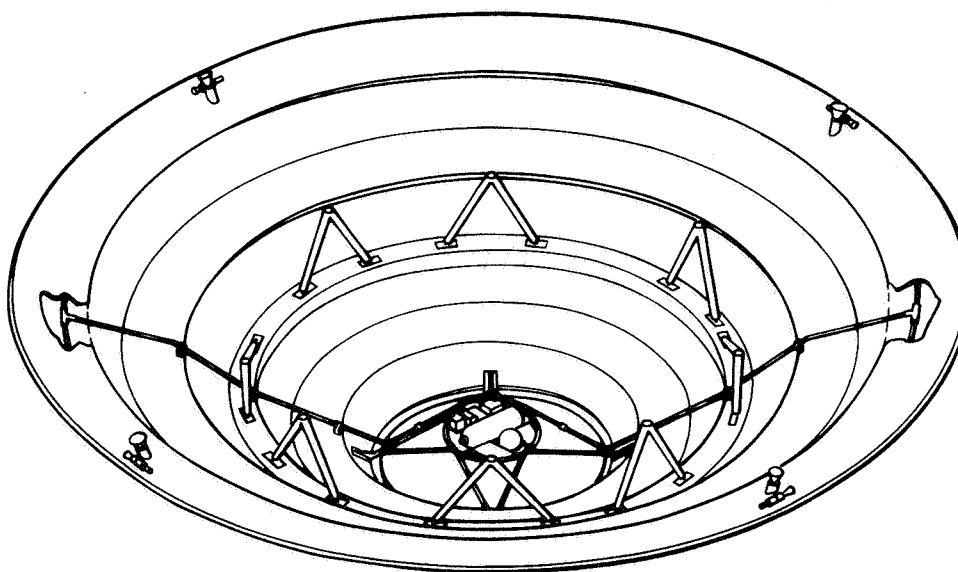


FIGURE 3.2.7-10

for the terminal propulsion system could also be employed in the deorbit system. Spin concepts were not evaluated further, since conclusions reached for solid deorbit systems are applicable to liquid deorbit systems as well.

3.2.7.2.3 Candidate system trade studies: Attitude control concepts for the solid and liquid propellant deorbit systems were compared on the basis of weight, reliability, cost, and development status.

Weight - For the solid propellant deorbit system, active and spin/active control system weights are presented in Figures 3.2.7-11 and -12 for both mono-propellant N_2H_4 and cold gas nitrogen. Hydrazine systems were evaluated due to the relatively high thrust and impulse requirements needed for deorbit attitude control.

The discontinuity in the curves results from the maximum aeroshell diameter limit of 14.5 ft. Using a deorbit ΔV of 450 ft/sec as a baseline, it is seen that the active control systems are lighter than the spin/active systems throughout the full range of capsule weights considered. Although the mono-propellant N_2H_4 systems offer a weight advantage for both active and spin/active control, the system dry weights are nearly identical up to a capsule weight of 2000 lb, with only a six pound advantage in favor of the N_2H_4 system for a 5000 lb capsule. The resulting variation in surface payload weight would be less than 3.5 lb.

Attitude control system weights for the liquid propellant deorbit systems are compared in Figures 3.2.7-13 and -14. A comparison of these figures illustrates the fact that use of gimbaled engine thrust vector control (TVC) results in a weight savings except at low deorbit ΔV 's (250 ft/sec).

Reliability - Failure rate data used in this study were obtained from other programs employing similar components and design. Reliability estimates for each of the attitude control concepts are presented in Table 3.2.7-6 for capsule weights of 1000 and 5000 lb. For the solid propellant deorbit systems, the cold gas concepts provide greater reliability than monopropellant N_2H_4 . Furthermore, the spin/active concept has a reliability advantage over the all-

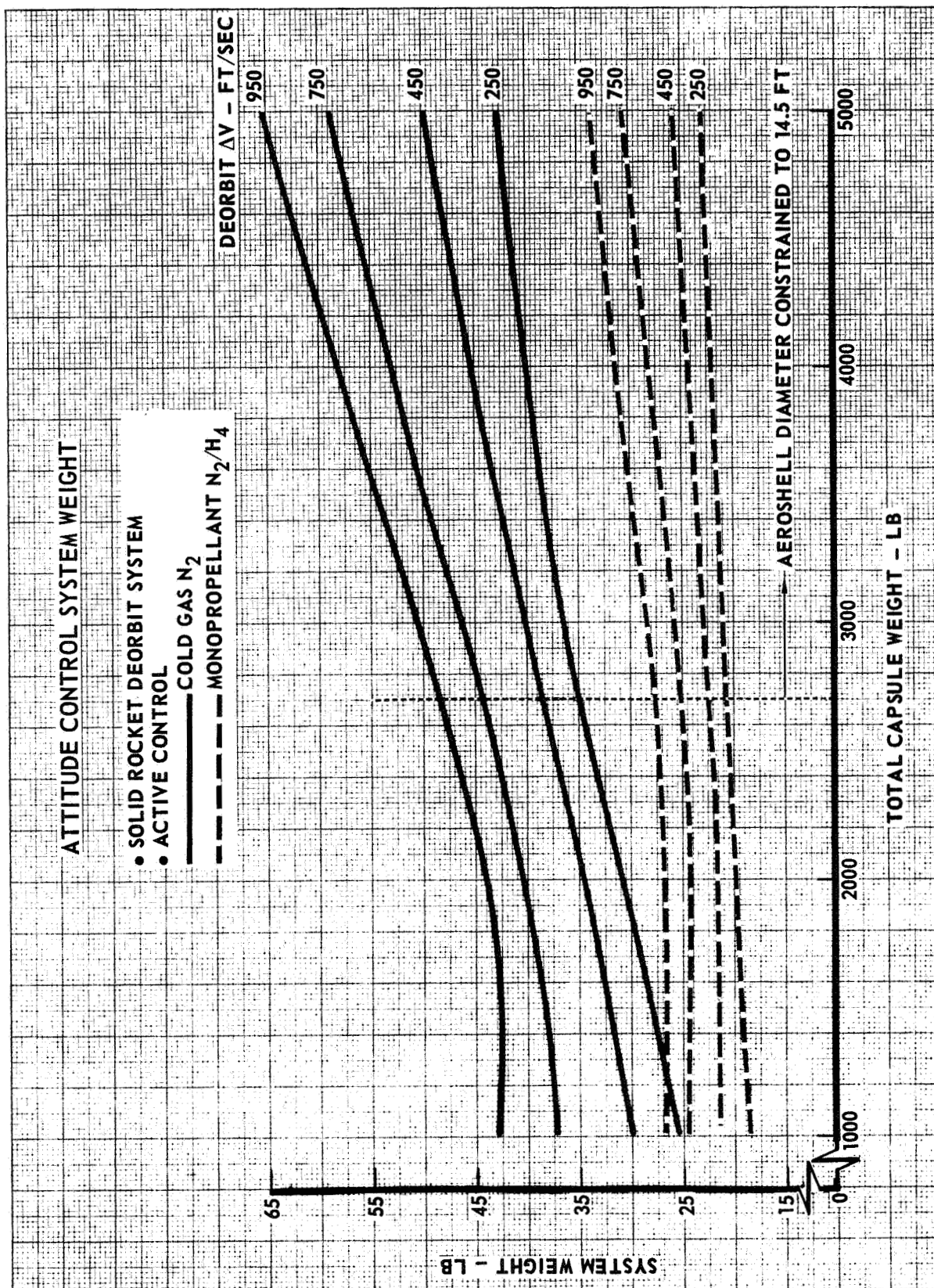


FIGURE 3.2.7-11

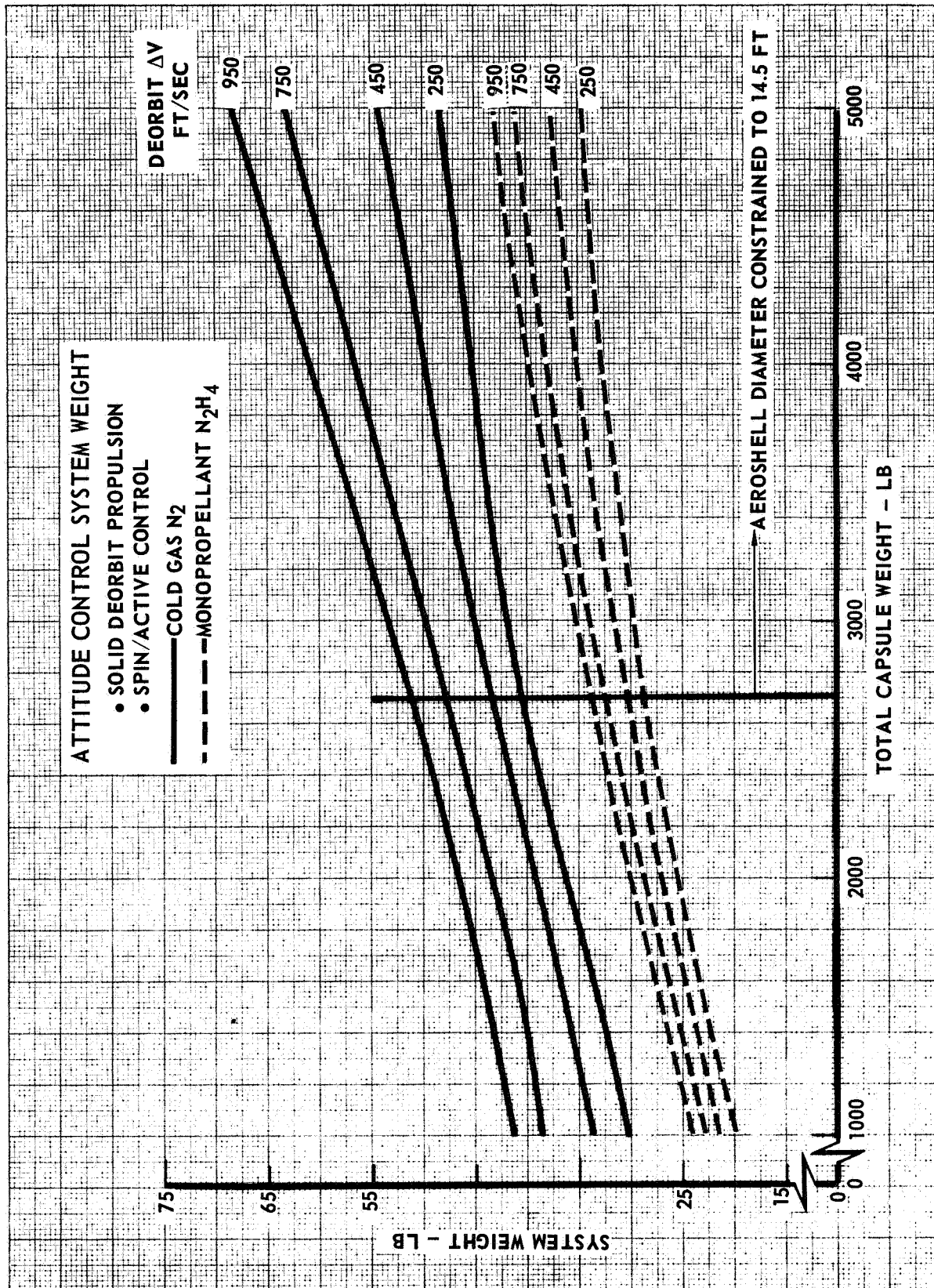


FIGURE 3.2.7-12

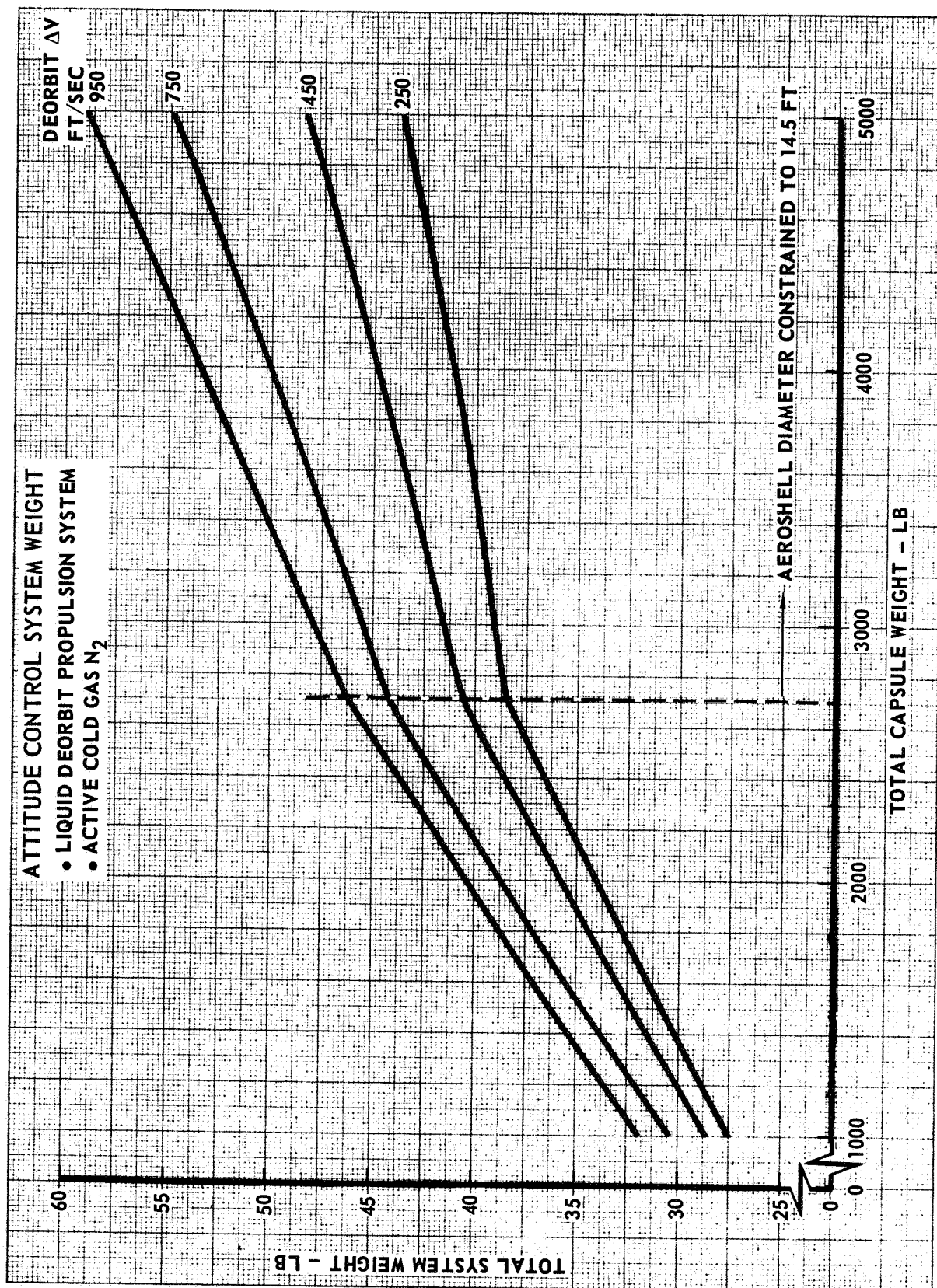


FIGURE 3.2.7-13

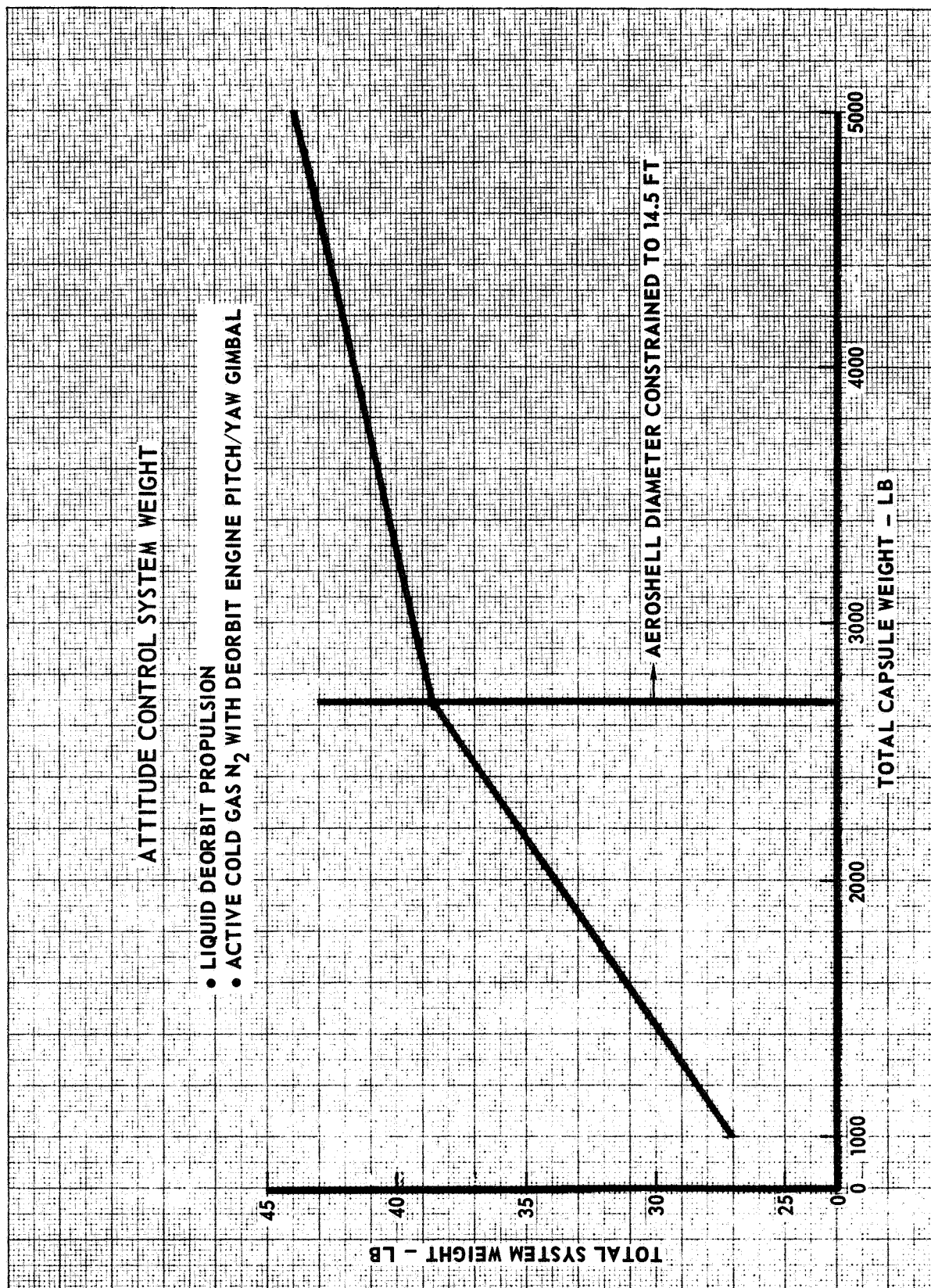


FIGURE 3.2.7-14

TABLE 3.2.7-6

ATTITUDE CONTROL SYSTEM RELIABILITY COMPARISON

SOLID PROPELLANT DEORBIT SYSTEM	RELIABILITY	
	1000 LB CAPSULE	5000 LB CAPSULE
• COLD GAS N_2 - ALL ACTIVE	.9959	.9965
• COLD GAS N_2 - SPIN/ACTIVE	.9964	.9965
• MONOPROPELLANT N_2H_4 - ALL ACTIVE	.9916	.9933
• MONOPROPELLANT N_2H_4 - SPIN/ACTIVE	.9925	.9933
LIQUID PROPELLANT DEORBIT SYSTEM		
• COLD GAS N_2 - ALL ACTIVE	.9963	.9966
• COLD GAS N_2 - WITH GIMBALLED ENGINE TVC	.9959	.9964

active technique, due to a fewer number of limit cycle valve actuations. Comparing the control techniques for liquid propellant deorbit systems, it is seen that the all active concept possesses a slight advantage over the active/TVC concept.

Cost - Preliminary cost estimates were obtained from prospective vendors for development, qualification and delivery of six flight systems. These data were adjusted to reflect total program cost (vendor cost plus system integration, mission support and fee).

Concept	Estimated Program Cost-\$ Million
Cold Gas N ₂	2.37
Monopropellant N ₂ H ₄	4.95
Cold Gas N ₂ /TVC	2.87

Although the data do not reflect charges for sterilization, they are considered adequate for the purpose of system comparison. Also, for the range of capsule weights evaluated, these costs are subject to only minor change.

Development Status - Extensive development and flight experience on several applicable programs provide confidence in the applicability of cold gas nitrogen and monopropellant hydrazine to reaction control concepts. Cold gas N₂ was employed in Ranger, Mariner, Lunar Orbiter, Surveyor and Agena. Hydrazine systems were incorporated on Ranger, Mariner, Intelsat, and ATS, and will perform the attitude control function for Transtage. The technology advancement resulting from these programs is applicable to the current study. Developed engines that are near the required thrust levels are available.

The major development problems are related to the in-tank sterilization requirements. Sterilization of a cold gas system will cause high storage tank pressures increasing leakage potential. Control components common to cold gas and hydrazine systems are also subject to performance degradation, requiring careful control of material selection, design tolerances, and process specifications. Testing at JPL and McDonnell has demonstrated the feasibility of

sterilizing hydrazine systems. However, hydrazine is subject to thermal decomposition and care must be exercised in selecting materials and establishing processing operations. Of particular importance is the development of positive expulsion tankage.

3.2.7.2.4 Summary: A summary of the trade studies performed is presented in Table 3.2.7-7. Due to different design requirements, control schemes for solid and liquid propellant deorbit systems were evaluated separately. For a solid propellant deorbit system, the N_2H_4 concepts possess a weight advantage. However, cold gas is more reliable, less costly, and will require less development time. Spin-active mechanizations are slightly more reliable than the all-active concepts and, although heavier from a propulsion standpoint, they may provide significantly lighter flight capsules by eliminating guidance power requirements during descent. For the liquid propellant deorbit systems, gimballed engine thrust vector control was evaluated due to a potential weight advantage over an all-active system. This assumption was confirmed in the analyses. However, installation of the gimbal servo-actuators detracts slightly from the high reliability afforded by the all-active system. Selection of the preferred attitude control system for various conceptual design concepts is made in Section 3.3.1 based upon an integrated evaluation of the guidance and control, electrical, and propulsion concepts.

TABLE 3.2.7-7
ATTITUDE CONTROL SYSTEM TRADE STUDY SUMMARY
DEORBIT $\Delta V = 450$ FT/SEC

CONCEPT	WEIGHT-LB			RELIABILITY		COST \$-MILLION	DEVELOPMENT STATUS
	1000 LB CAPSULE	5000 LB CAPSULE	1000 LB CAPSULE	5000 LB CAPSULE			
SOLID PROPELLANT DEORBIT SYSTEM							
• COLD GAS N ₂ - ACTIVE	30	50	.9959	.9965	2.37	MINIMUM DEVELOPMENT	
• COLD GAS N ₂ - SPIN/ACTIVE	34	55	.9964	.9965	2.37	MINIMUM DEVELOPMENT	
• N ₂ H ₄ - ACTIVE	21	26	.9916	.9933	4.95	EXTENSIVE DEVELOPMENT	
• N ₂ H ₄ - SPIN/ACTIVE	21	38	.9925	.9933	4.95	EXTENSIVE DEVELOPMENT	
LIQUID PROPELLANT DEORBIT SYSTEM							
• COLD GAS N ₂ - ACTIVE	29	49	.9963	.9966	2.37	MINIMUM DEVELOPMENT	
• COLD GAS N ₂ - ACTIVE /TVC	27	44	.9959	.9964	2.87	MINIMUM DEVELOPMENT	

3.2.7.3 Terminal Propulsion System. - The terminal propulsion system (TPS) provides the final braking required for soft landing on the surface of Mars. Previous studies have shown that this is best accomplished using an earth-storable liquid propellant system employing multiple throttleable engines (e.g., Surveyor). Design of the TPS depends upon lander weight, the altitude and velocity at which the engines are ignited, and the type of terminal guidance logic employed. The purpose of this section is two-fold: First, to establish the near-optimum ignition altitude and preferred terminal guidance mode, based on propulsion considerations only, and Second, to evaluate candidate propulsion system designs on the basis of weight, reliability, cost, development status, and interactions with other capsule systems. The results are used to support system selections for the point designs covered in Section 4.0. The following paragraphs contain the system requirements, a description of the candidate design concepts, and a discussion of the concept trade studies.

3.2.7.3.1 Requirements: The TPS must provide the thrust and total impulse needed for the soft landing of instrumented payloads on the Mars surface. The lander vehicle varies from 500 to 3000 Earth lb for the range of capsule weights considered. In addition to the decelerated mass, system requirements are controlled by the altitude and velocity at which the system is initiated. The ignition envelope is presented in Figure 3.2.7-15, where the ignition region is bounded by the VM-10 atmospheric model (with parachute), landing radar limitations, and a maximum engine throttle ratio of 10:1. The velocity at ignition is the resultant of the horizontal wind and vertical descent velocity vectors. For these studies, the minimum ignition velocity is based on entry into the VM-10 atmosphere without winds and the maximum ignition velocity on entry into the VM-7 atmosphere with continuous winds. The throttle ratio limitation is based on demonstrated technology of monopropellant and bipropellant systems. (All throttle ratios specified in this study provide a margin for attitude control by differential throttling of multiple engines; a 20% control margin is provided at deep throttle, while a 10% margin is provided at max. thrust.) The system design must be compatible with the lander configuration and must not cause interactions with other capsule systems to the extent that mission objectives are jeopardized.

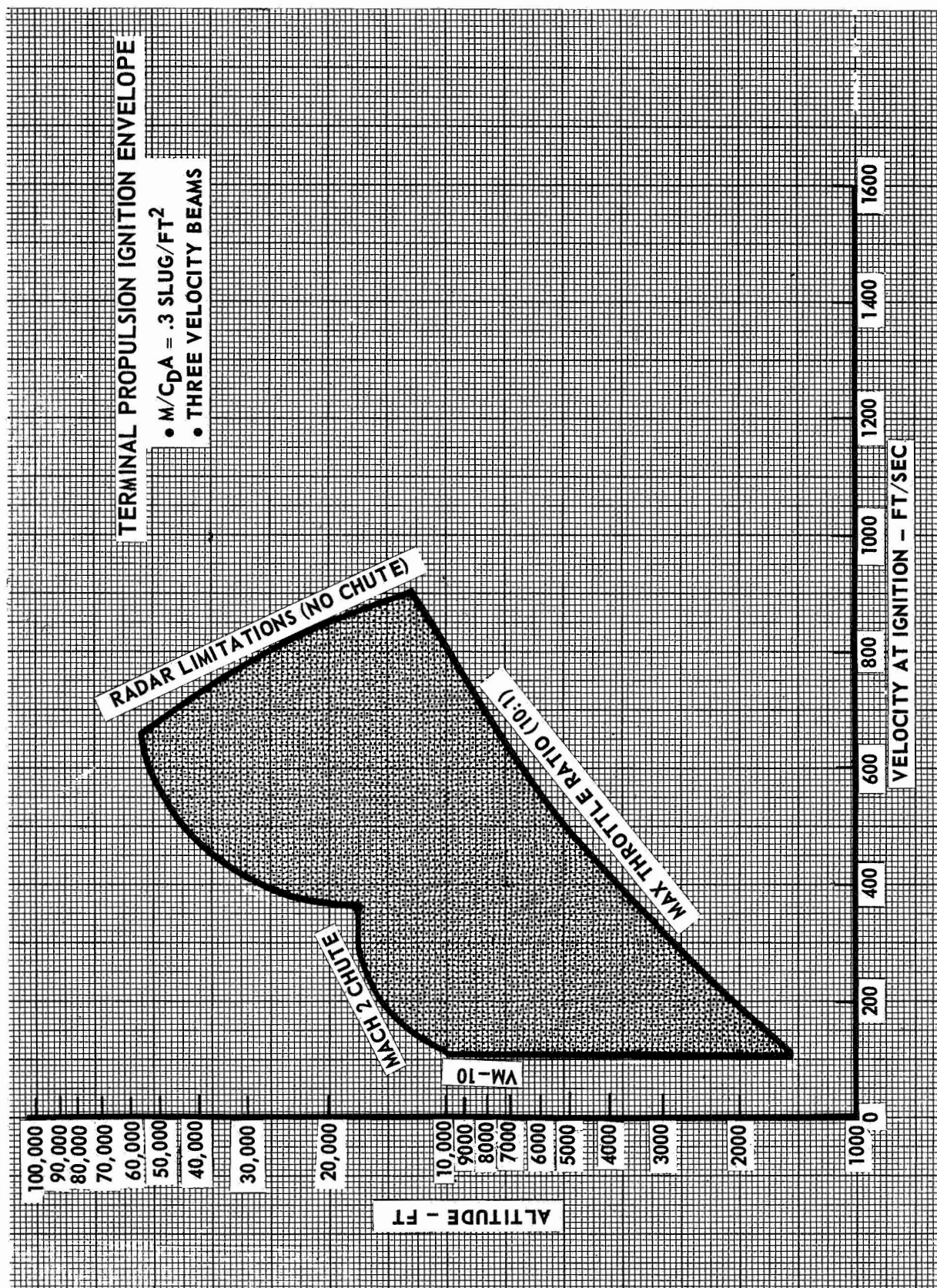


FIGURE 3.2.7-15

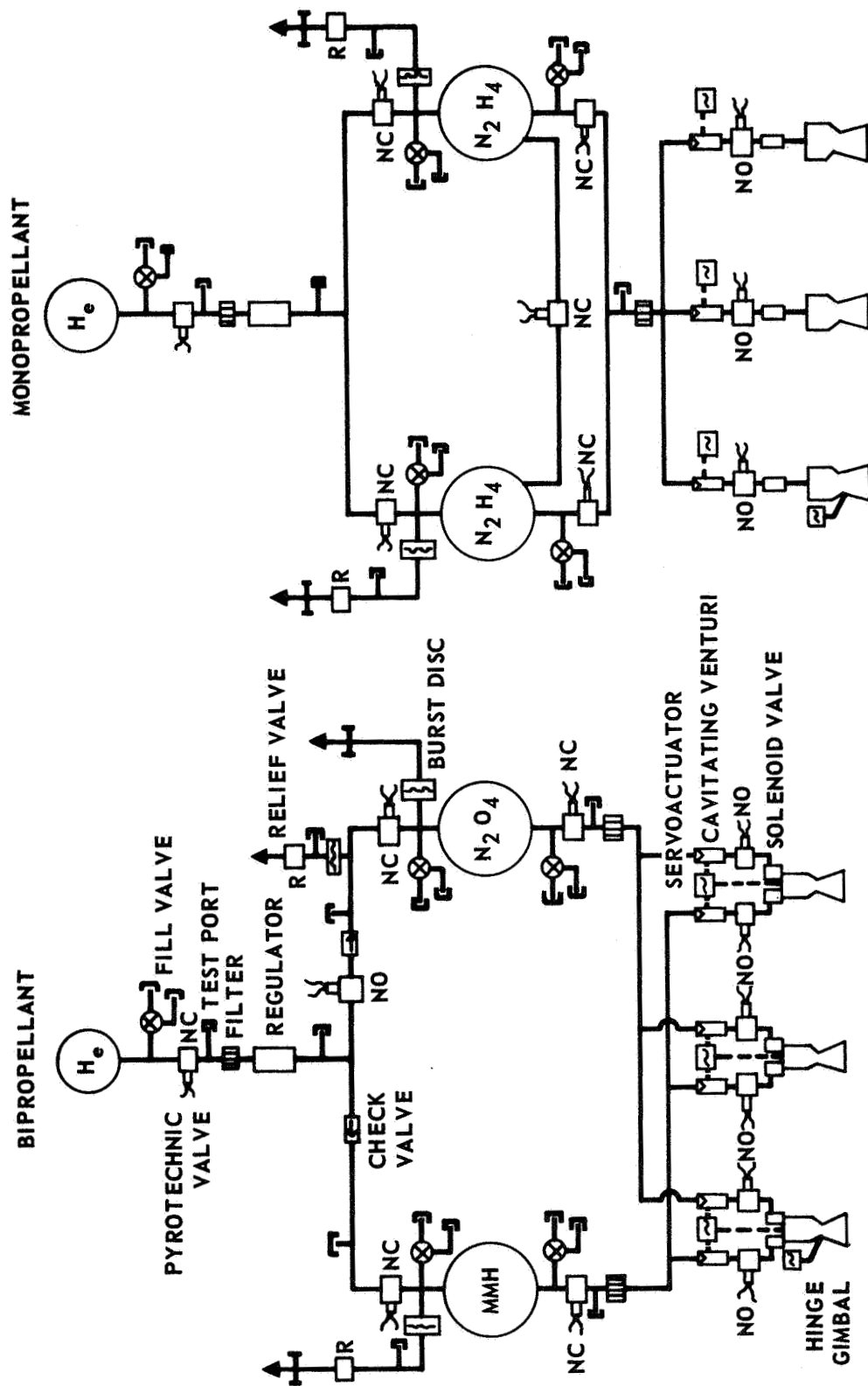
3.2.7.3.2 Candidate design concepts: Monopropellant and bipropellant designs were evaluated for the terminal propulsion system. Of the monopropellants, only hydrazine (N_2H_4) offers sufficiently high performance to be considered for this application. Advantages of monopropellant N_2H_4 systems as compared with bipropellant systems are high reliability, low cost, short development time, and minimum interaction with the landed science. In addition, compatibility tests at McDonnell and systems tests at JPL have demonstrated the feasibility of heat sterilizing N_2H_4 systems.

The bipropellant system offers significant weight advantages over alternate concepts. Furthermore, a firm technological base for bipropellant throttling applications has been established as a result of engine developments for Surveyor and the Apollo Lunar Module. Of several candidate propellant combinations, monomethyl hydrazine (MMH) and nitrogen tetroxide (N_2O_4) were selected for this study. This combination offers good performance and a large backlog of applicable space engine experience. Also, propellant/materials compatibility testing at McDonnell and systems level tests at Martin-Denver have demonstrated the feasibility of heat sterilizing MMH/ N_2O_4 systems.

Detailed schematics of the candidate monopropellant and bipropellant systems are presented in Figure 3.2.7-16. As illustrated, a three-engine configuration was assumed for this study. Pitch/yaw control during terminal descent is achieved by differential engine throttling, and roll control is obtained by hinge gimbaling one engine. This configuration represents the best compromise for achieving the desired performance at minimum complexity.

Both catalytic and thermal decomposition chambers were considered for the candidate N_2H_4 engine designs. The thermal decomposition concept offers a weight advantage for thrust levels in excess of 300 lb_f , and the potential of improved performance throughout the full throttle range. Its primary disadvantage lies in obtaining proper ignition. The chamber must be heated to at least 500°F to promote thermal decomposition of the propellant. If a chemical (oxidizer slug) start is used for chamber preheat, hot fire acceptance testing of the engine is impossible without undue compromise to the design or mission reliability. If a small, catalytic reactor is provided for multi-start capability, the required preheat time tends to be excessive (approximately 5 sec). Preliminary test results on thermal decomposition chambers at TRW are encour-

CANDIDATE TERMINAL PROPULSION SYSTEMS



3.2.7-40

FIGURE 3.2.7-16

aging but the catalytic decomposition chamber possesses the greatest backlog of development experience and was therefore selected for the system trade studies. Weight and envelope comparisons of the monopropellant and bipropellant concepts are shown in Figure 3.2.7-17.

3.2.7.3.3 Trade studies: Trade studies were conducted to compare descent guidance modes and terminal propulsion concepts.

Descent Guidance Mode - Within the ignition altitude - velocity envelope of Figure 3.2.7-15, three candidate terminal guidance modes (adaptive, pre-programmed, and modified pre-programmed) were evaluated.

Terminal propulsion system design depends upon the weight of the lander, the altitude and velocity at which the engines are ignited, and the type of command guidance logic employed. A high ignition altitude requires low engine thrust (hence, small engines) for deceleration, but propellant requirements are large because of the high potential energy of the lander at ignition. Reducing the ignition altitude reverses the trend: propellant requirements decrease but engine thrust levels must increase to remove the same velocity increment over a smaller altitude range. This suggests a trade-off between engine weight and propellant weight to arrive at a minimum weight propulsion system. The descent sequence and propulsion optimization studies are presented below for each of the three guidance techniques.

o Adaptive Guidance - In its simplest form, the adaptive guidance mode uses information from the landing radar to compute the deceleration thrust which must be applied to simultaneously reduce lander range and range rate to zero. The descent profile therefore appears as a constant deceleration on a range-velocity plot. The guidance technique, as used in this analysis, is altered slightly by terminating the deceleration phase when all but 10 ft/sec of the lander velocity is removed at a nominal altitude of 60 ft. At this point, the terminal propulsion system is throttled to a thrust-to-weight of one, permitting the lander to descend at constant velocity to an altitude of 10 ft. The engines are then shut down and the lander falls to the surface.

Normalizing TPS requirements for lander mass, Figure 3.2.7-18 shows the total impulse (propellant) and throttle ratio needed to decelerate the lander

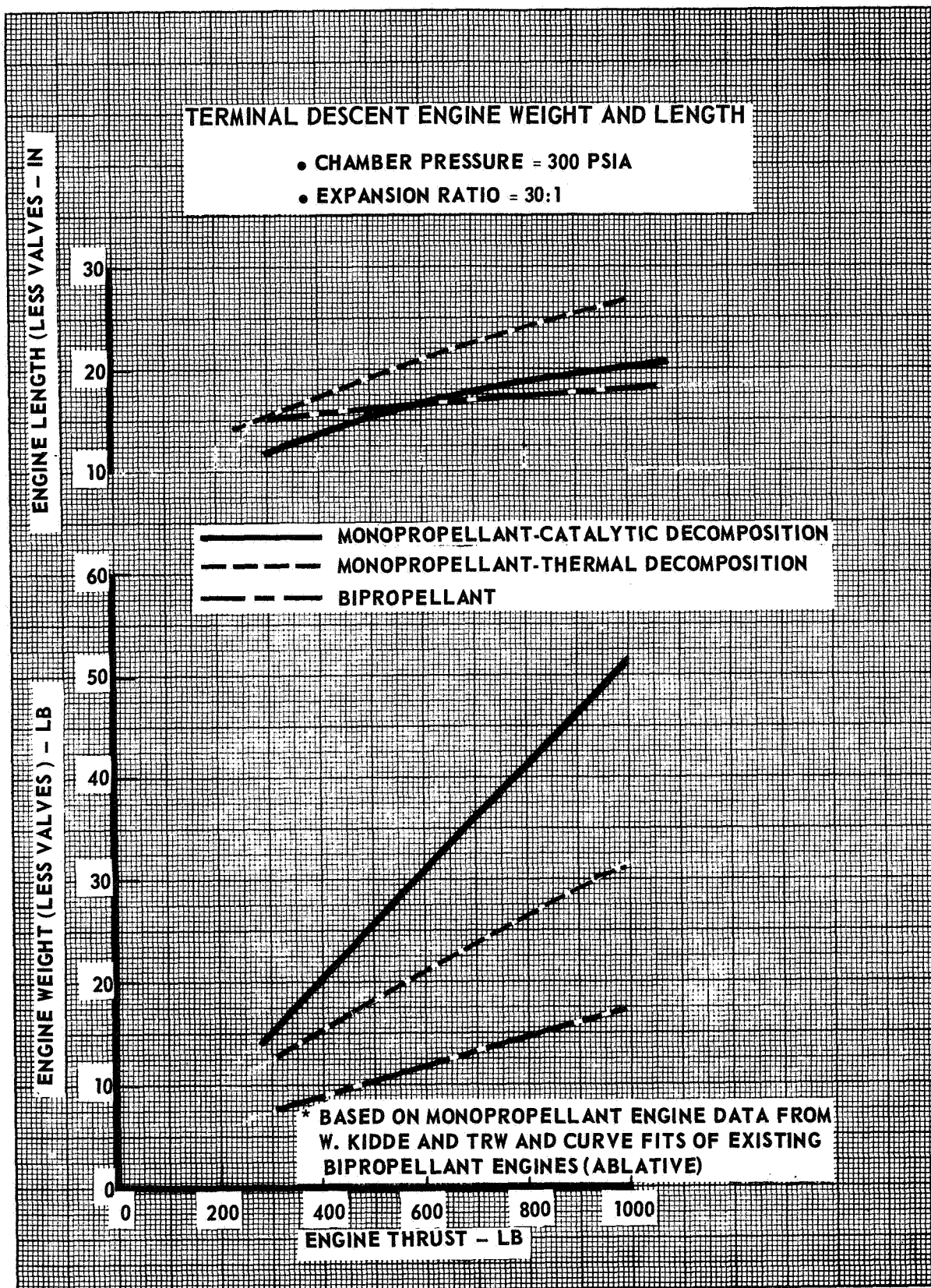


FIGURE 3.2.7-17

TERMINAL PROPULSION REQUIREMENTS

- ADAPTIVE GUIDANCE
- $M/C_D A = .3 \text{ SLUGS/FT}^2$

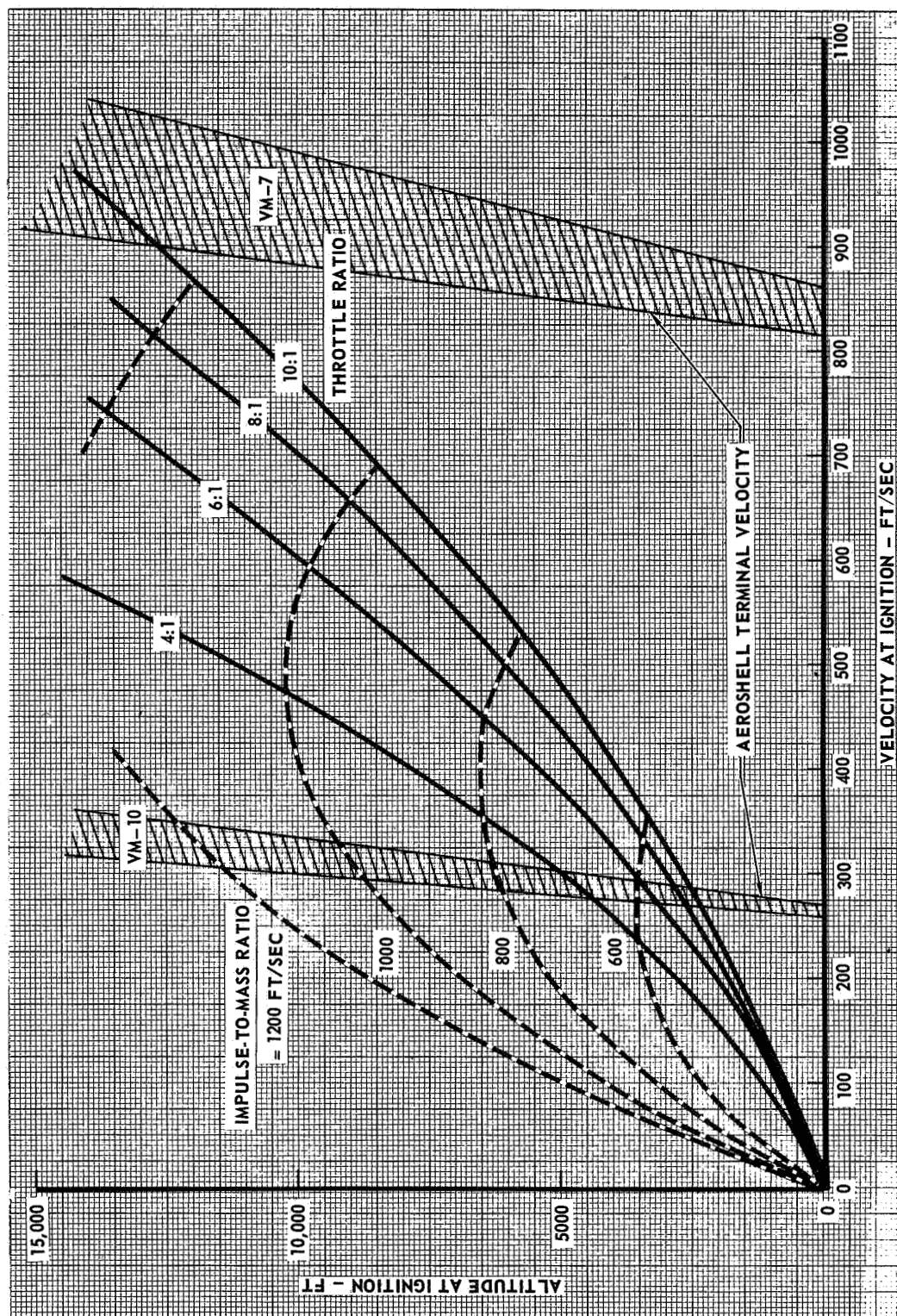


FIGURE 3.2.7-18

to a soft landing from any given set of ignition conditions (altitude and velocity). The correlation between throttle ratio and thrust level is keyed to the thrust requirements for the final, constant-velocity descent. This phase of the descent sequence establishes the minimum commanded thrust, i.e., a thrust-to-weight of one, while the maximum is set by the deceleration requirements from the range and velocity conditions at ignition. The ratio of maximum to minimum thrust is the throttle ratio required for powered descent. To this must be added throttle margins for velocity sensing errors and inaccuracies in the predicted lander touchdown weight. Also, sufficient throttling capability must exist about the commanded thrust-to-weight range to permit pitch and yaw control through differential throttling of the multiple engines. In the analyses which follow, thrust levels were selected to provide a minimum commanded deceleration of .8 g (Mars); attitude control margins about the commanded high and low thrust levels were established at 10% and 20%, respectively.

Figure 3.2.7-19 shows the propulsion system weight optimization for an ignition velocity of 425 ft/sec and lander weights of 1000 and 3000 lb. Utilizing the impulse and throttle ratio requirements presented in Figure 3.2.7-18, ignition altitude was varied in order to identify the point at which minimum propulsion system weight is achieved. It is seen that the optimum throttle ratio is relatively insensitive to lander weight. The optimum throttle ratio is also seen to be less for the monopropellant system than for the bi-propellant. This result reflects the fact that monopropellant engines are characteristically heavier than bipropellant engines, and constitute a greater fraction of total system weight. There is, therefore, a weight advantage in igniting the monopropellant system at a higher altitude than the bipropellant, thereby reducing the thrust requirement at the sacrifice of added impulse.

The above optimization was repeated for a number of ignition velocities and the results are summarized in Figure 3.2.7-20.

- o Pre-Programmed Guidance - The landing approach for this mode is programmed into the guidance computer to provide a scheduled descent based on a constant ratio of range-to-velocity. Ignition of the propulsion system is effected above the descent switching line. Following ignition, the engines

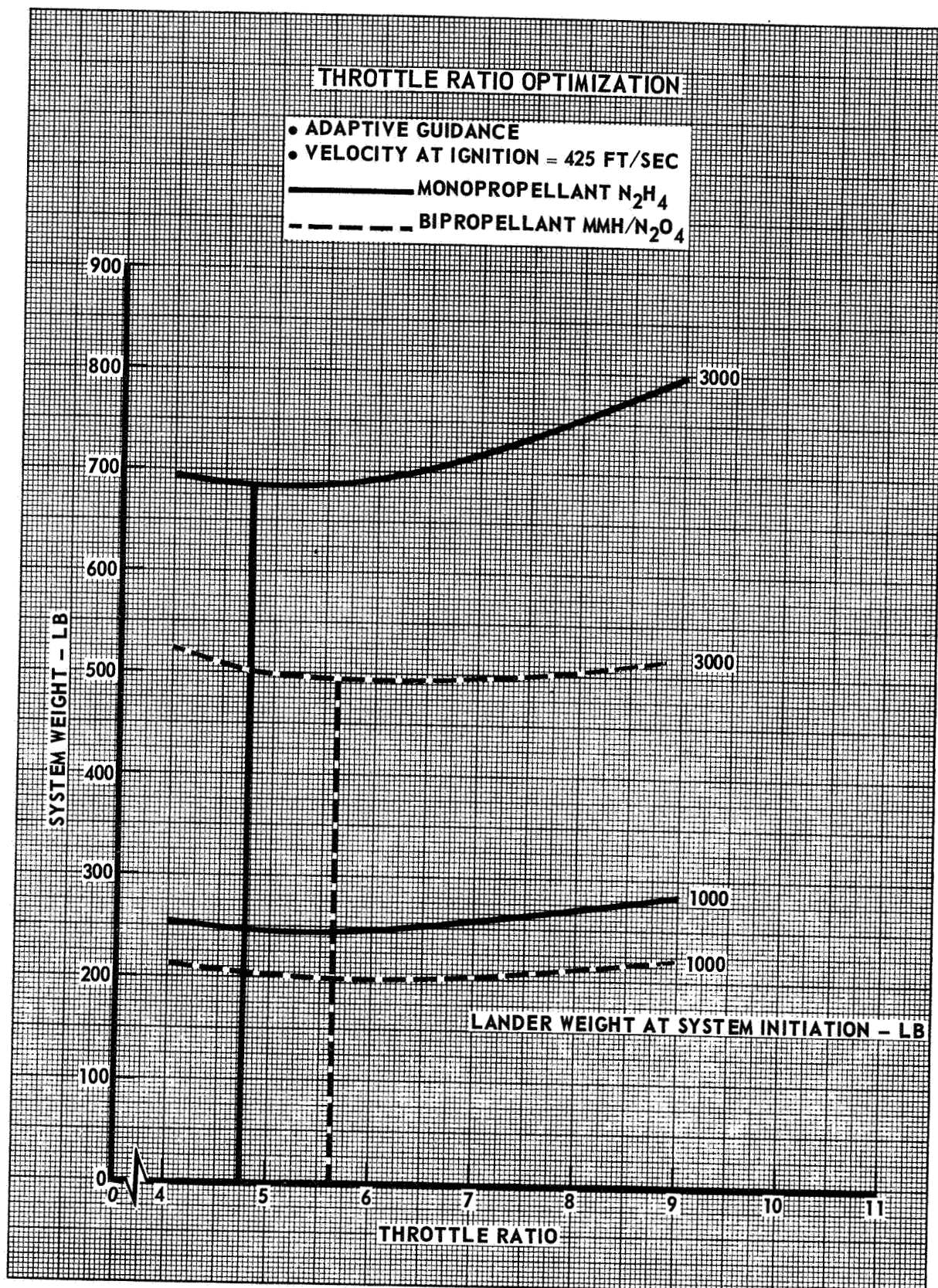


FIGURE 3.2.7-19

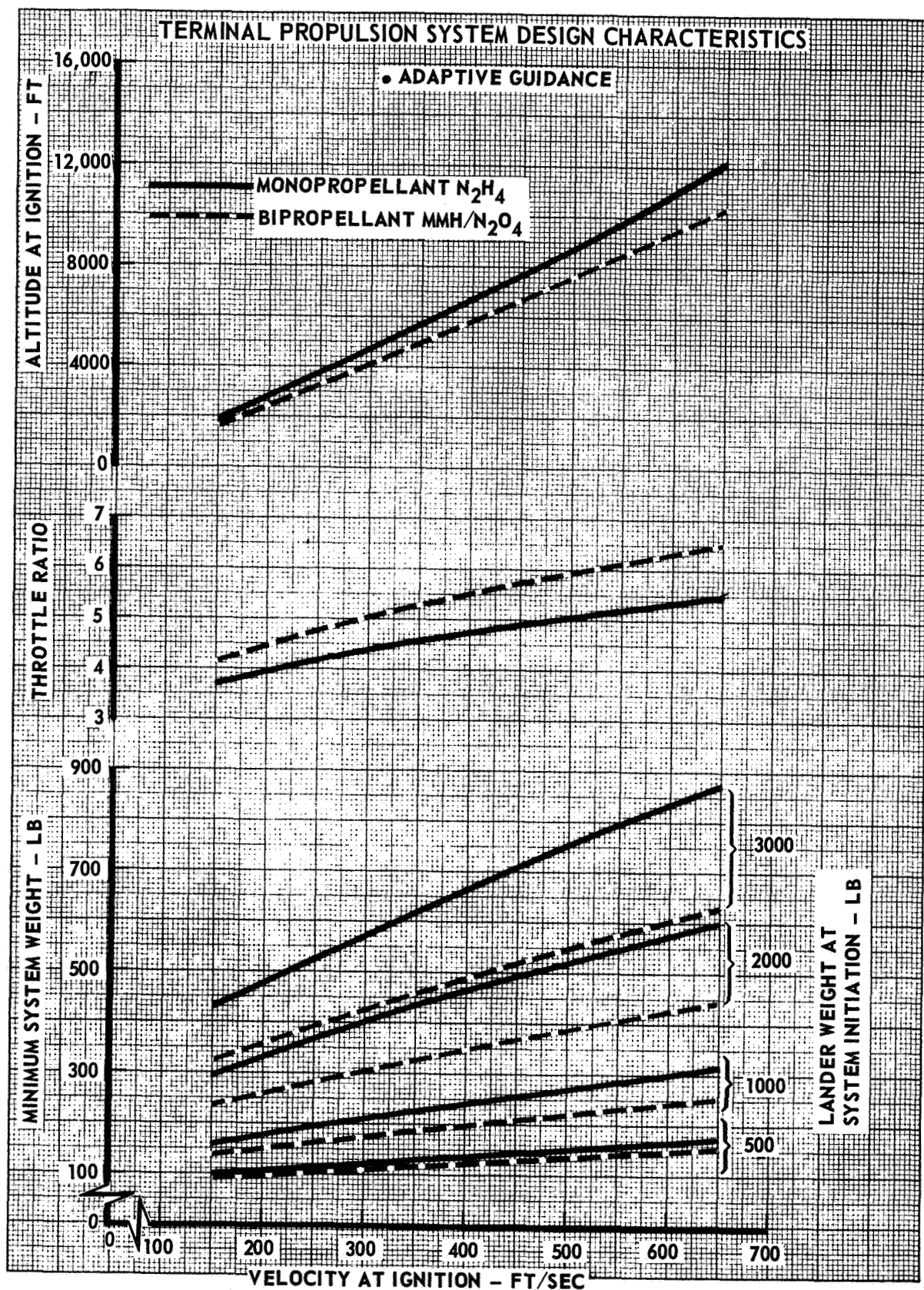


FIGURE 3.2.7-20

are throttled to permit the lander to accelerate at .2g (Mars) until the switching line is attained. Thereafter, the thrust level is continually adjusted in order to maintain the pre-programmed ratio of range-to-velocity. Like the adaptive mode, descent from 60 ft is at a constant 10 ft/sec. A typical descent profile is shown in Figure 3.2.7-21, which also shows the impulse and throttle ratio requirements as a function of ignition conditions.

The optimization technique employed for the pre-programmed guidance mode differs from the adaptive mode, and is illustrated in Figure 3.2.7.3-22 for a 1000 lb lander employing a bipropellant TPS. System weight is plotted as a function of descent profile slope (pre-programmed range-to-velocity ratio) for various ignition velocities. The descent profile slope which results in minimum terminal propulsion system establishes the near-optimum throttle ratio and ignition altitude. This analysis was repeated for the monopropellant system. Results for both propulsion concepts are summarized in Figure 3.2.7-23.

o Modified Pre-Programmed Guidance - In examining the altitude-velocity plots for the above guidance modes (see Figures 3.2.7-18 and 3.2.7-21), it can be seen from the impulse-to-mass requirements that the pre-programmed mode is the more efficient at low velocities while the adaptive is more efficient for high velocities. The reason for this can be explained by reviewing each velocity regime independently. At low velocities, the adaptive guidance logic provides a thrust command which attempts to decelerate the vehicle, increasing the descent time from a given altitude. The propulsion system is therefore required to compensate for the large gravity loss. For the same ignition condition, the pre-programmed guidance allows the vehicle velocity to increase by commanding a thrust-to-weight less than one until intercept of the switching line. Consequently, the descent time, and therefore the gravity loss, is less, resulting in a lighter propulsion system. At high ignition velocities, the relative values on descent time are reversed. The adaptive guidance commands a high constant-thrust burn (except for adjustments to compensate for vehicle mass) for the full deceleration maneuver while the pre-programmed mode requires that the engine be continually throttled-back in order to maintain a linear relationship between range and velocity. Obviously, the latter concept requires a longer burn to remove the same velocity increment.

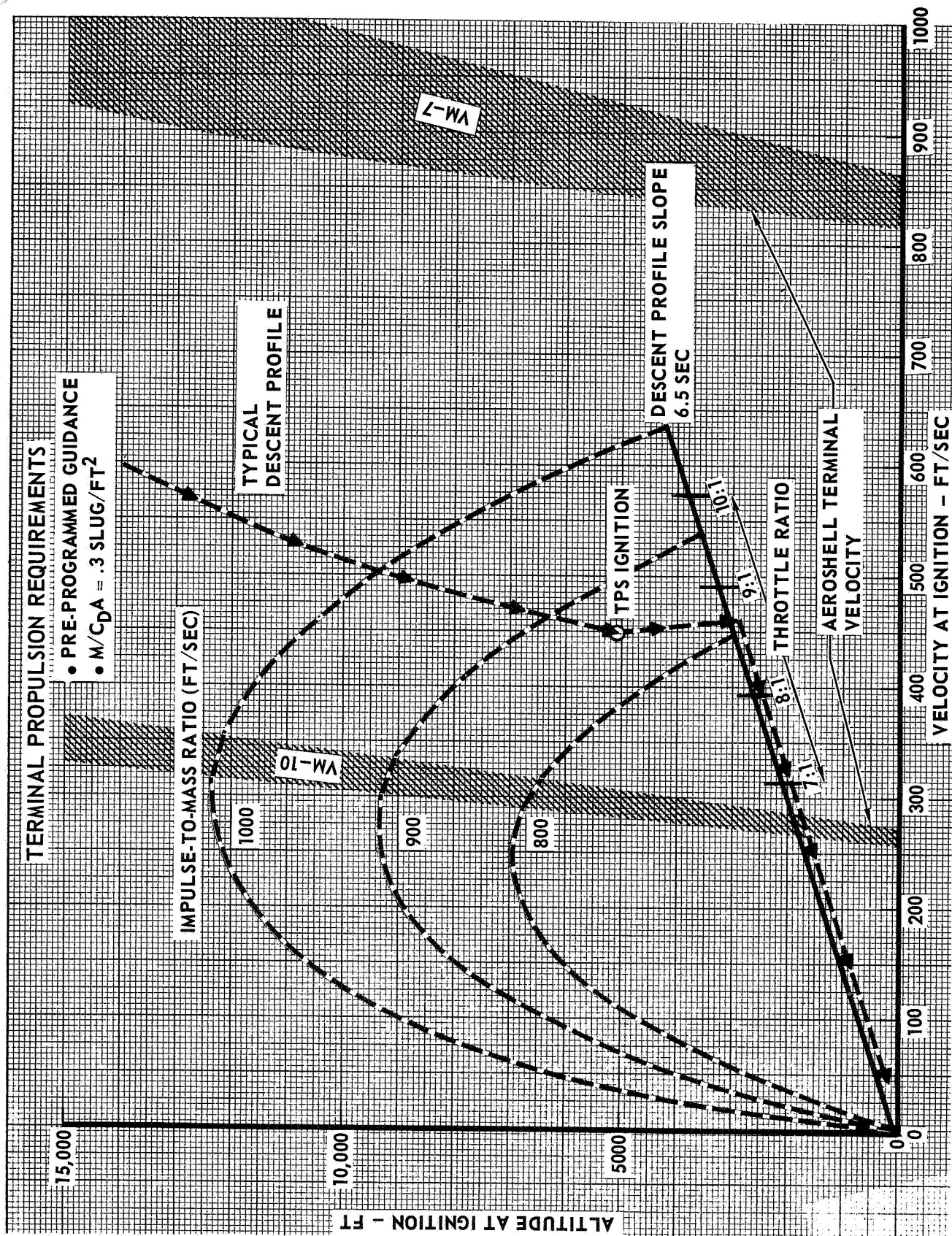


FIGURE 3.2.7-21

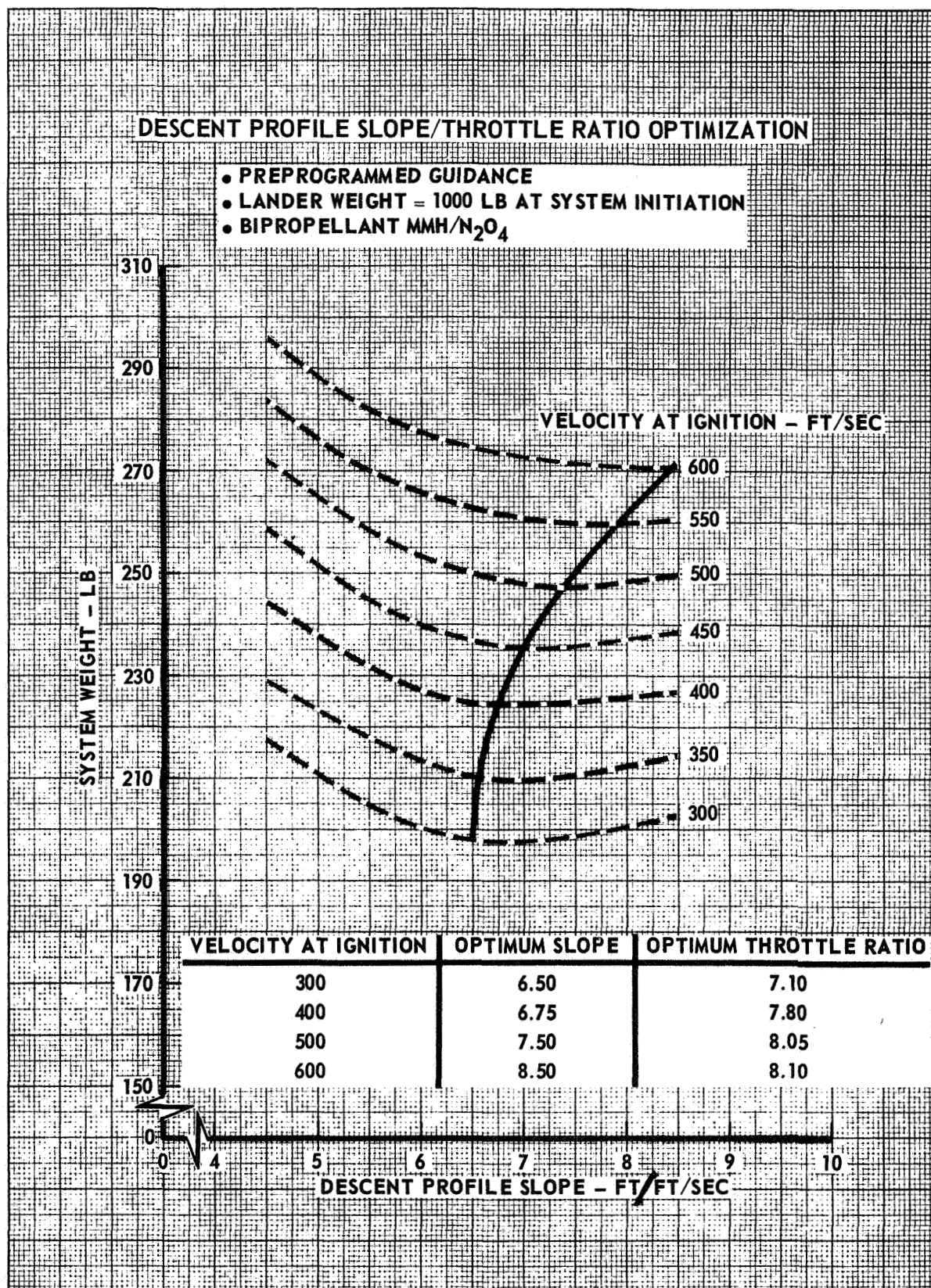
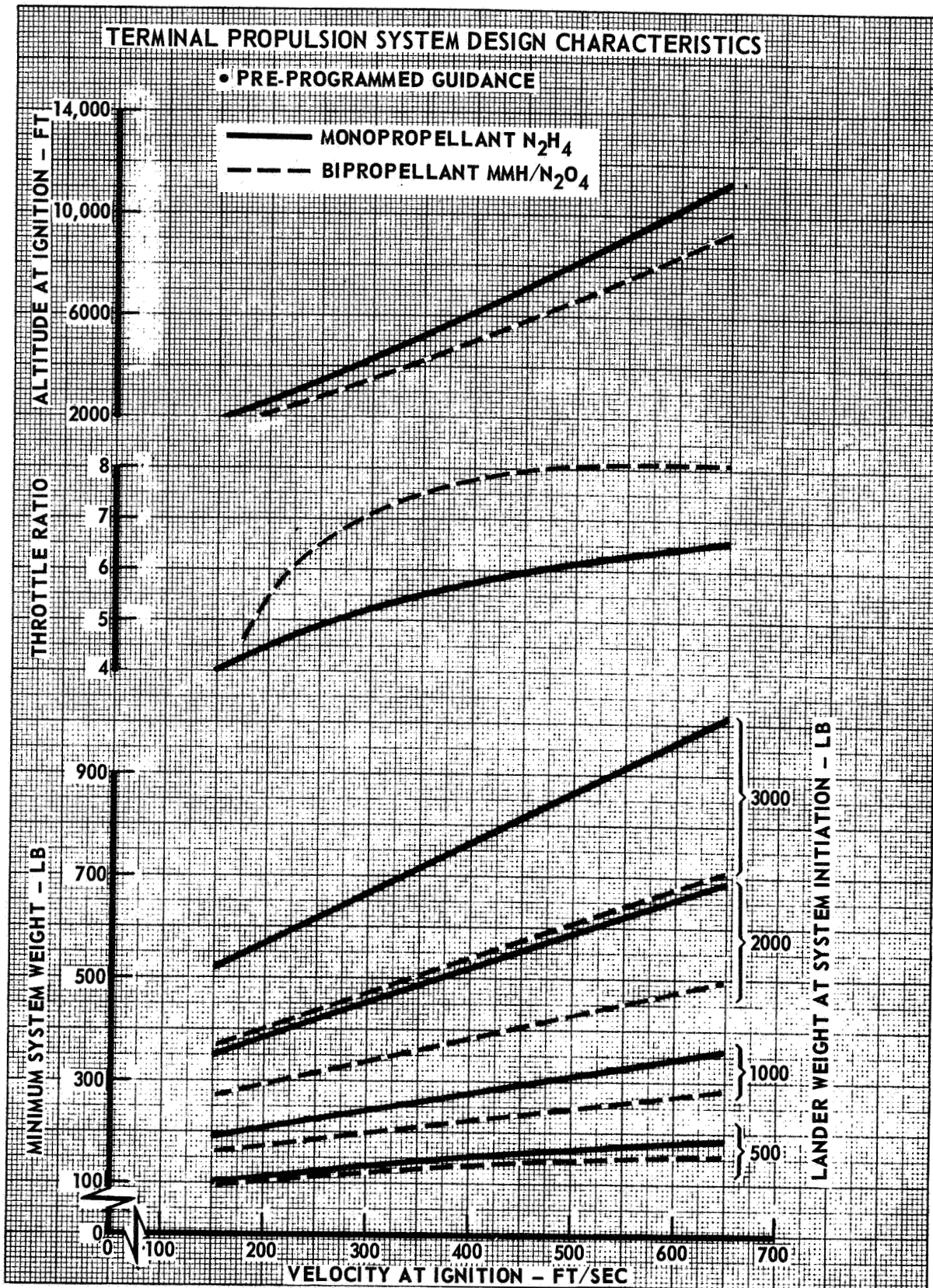


FIGURE 3.2.7-22



The modified pre-programmed mode combines the best characteristics of each of the above guidance schemes. Similar to the pre-programmed guidance mode, a pre-selected relation between range and velocity governs the descent profile. However, instead of a linear relation between range and velocity, a parabolic relation is used. A descent curve corresponding to a thrust-to-weight ratio of two was employed in this study. As was the case for the adaptive mode, no constraints are imposed on ignition altitude. If ignition occurs above the switching curve, the lander is allowed to accelerate at .2g (Mars) until the curve is intersected. Then, a deceleration force equivalent to a thrust-to-weight ratio of two is applied. If, on the other hand, ignition occurs below the switching curve, maximum thrust is commanded until the switching curve is intersected.

The TPS requirements for the modified pre-programmed guidance are shown in Figure 3.2.7-24. The optimization procedure employed for this technique is identical to the adaptive mode and results are summarized in Figure 3.2.7- 25

o Guidance Mode Comparisons - Due to the lack of precise knowledge about the Martian atmosphere, the terminal propulsion system must possess sufficient versatility to satisfy a wide range of velocities. Since velocity measurements obtained while on the parachute would be of questionable value in the presence of wind gusts, it is assumed that system initiation will be triggered by a range mark, only, using the altimeter. For this study a maximum design velocity of 425 ft/sec was established based on the VM-7 atmosphere with continuous winds. The minimum achievable velocity during aerodeceleration was set at 150 ft/sec based on the VM-10 atmosphere without winds. To illustrate the effect of velocity on TPS design for a constant ignition altitude, monopropellant system weight is presented in Figure 3.2.7-26 for each of the three guidance modes. As seen in this figure, for the adaptive guidance modes, use of a TPS designed for the maximum velocity (425 ft/sec) would not satisfy the requirements for a minimum initiation velocity because of the gravity-induced impulse penalties described earlier. Hence, propellant requirements must be established based on the lowest achievable velocities while throttle ratio must be sufficient to satisfy the maximum design velocity. This paradox is not encountered if pre-programmed or modified pre-programmed guidance modes are employed. Of the latter two concepts, the modified pre-programmed mode is preferred from a propulsion standpoint since it affords the lowest TPS weight.

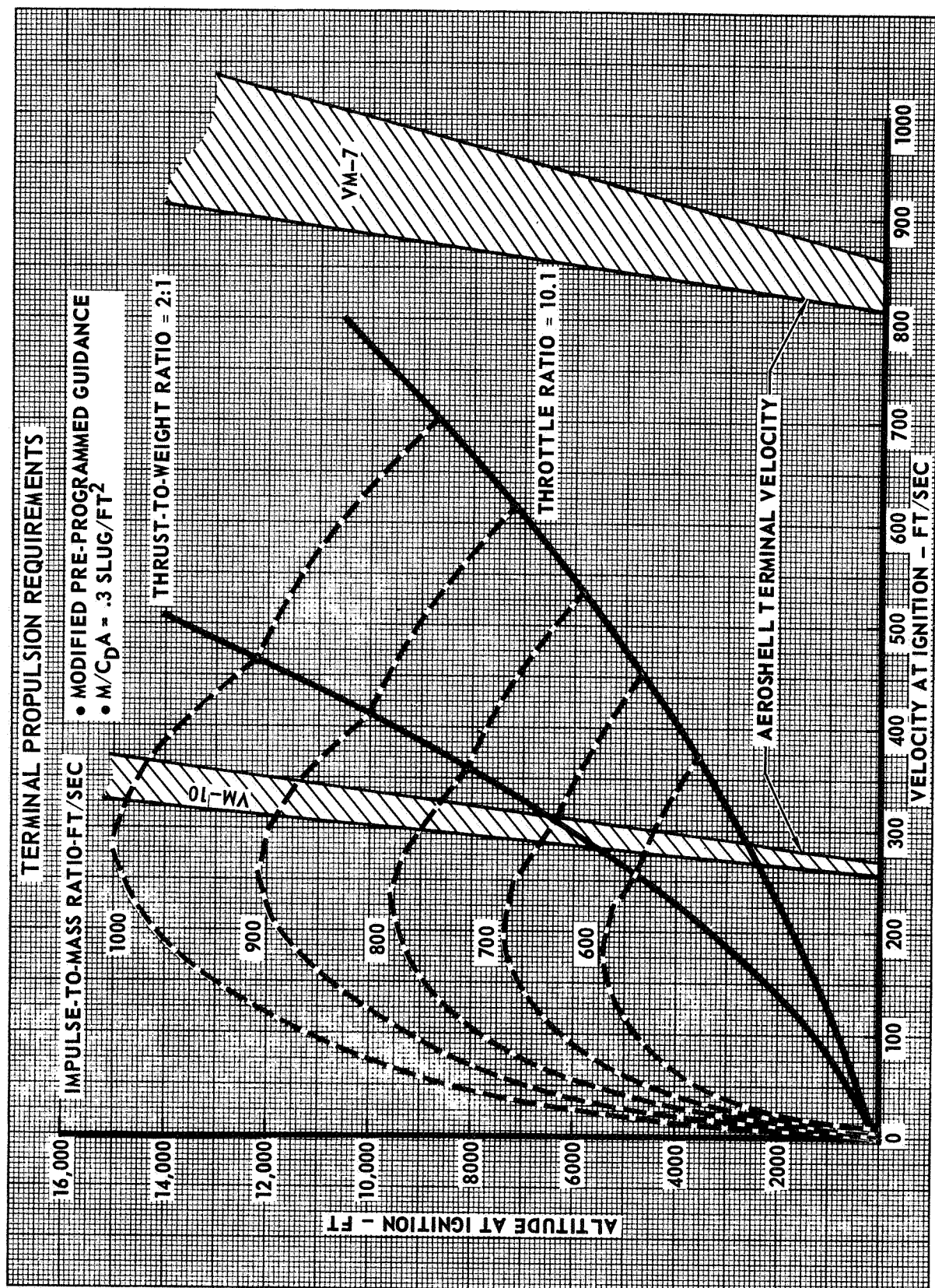


FIGURE 3.2.7-24

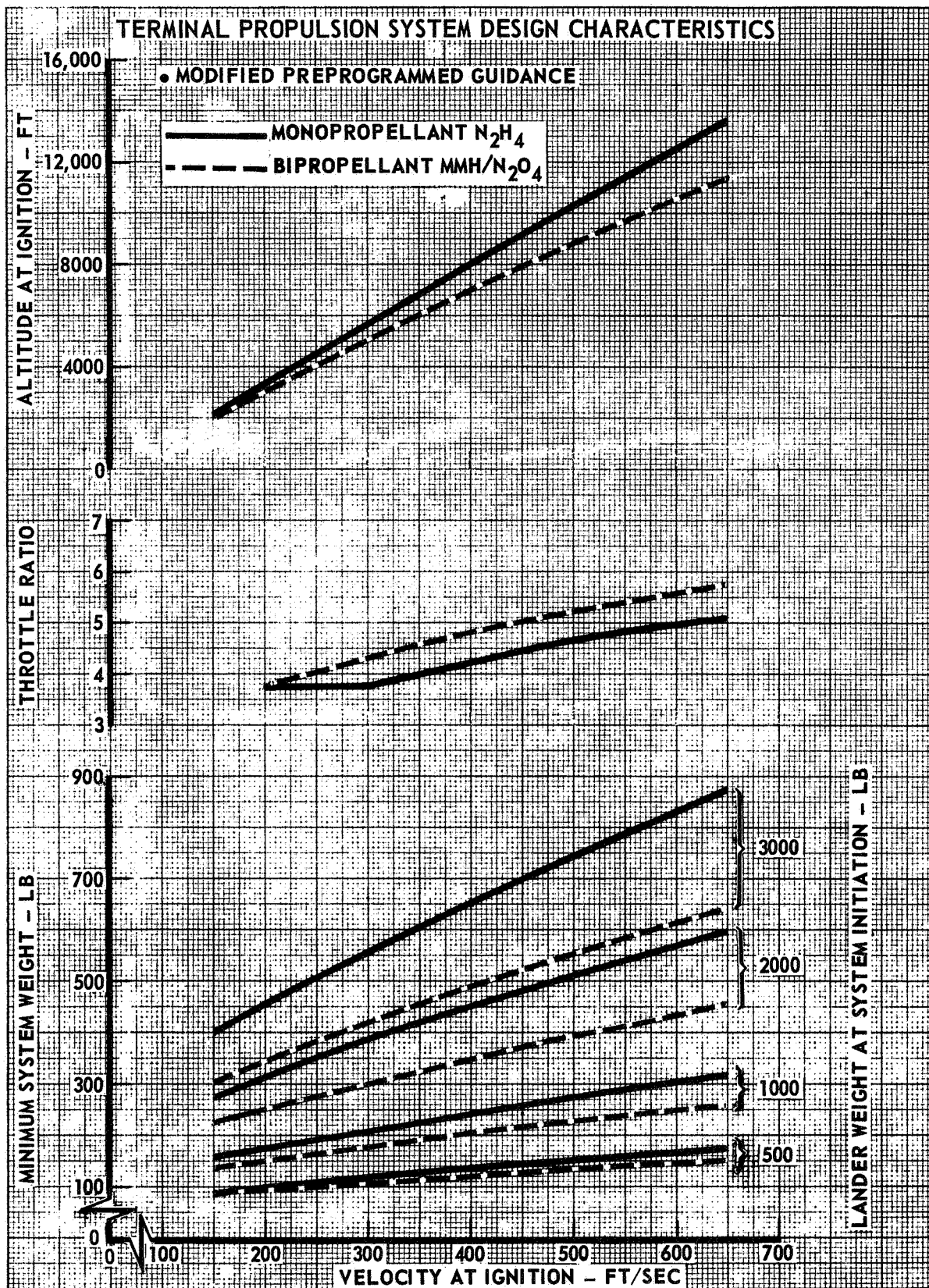


FIGURE 3.2.7-25

TERMINAL PROPULSION SYSTEM WEIGHT FOR CONSTANT ALTITUDE IGNITION

- MONOPROPELLANT N_2H_4
- LANDER WEIGHT = 1000 LBS AT SYSTEM INITIATION
- MAX VELOCITY = 425 FT/SEC

GUIDANCE MODE	NEAR-OPTIMUM IGNITION ALTITUDE	NEAR-OPTIMUM THROTTLE RATIO
ADAPTIVE	7050 FT	4.80
PRE-PROGRAMMED	6600 FT	5.82
MODIFIED PRE-PROGRAMMED	8400 FT	4.31

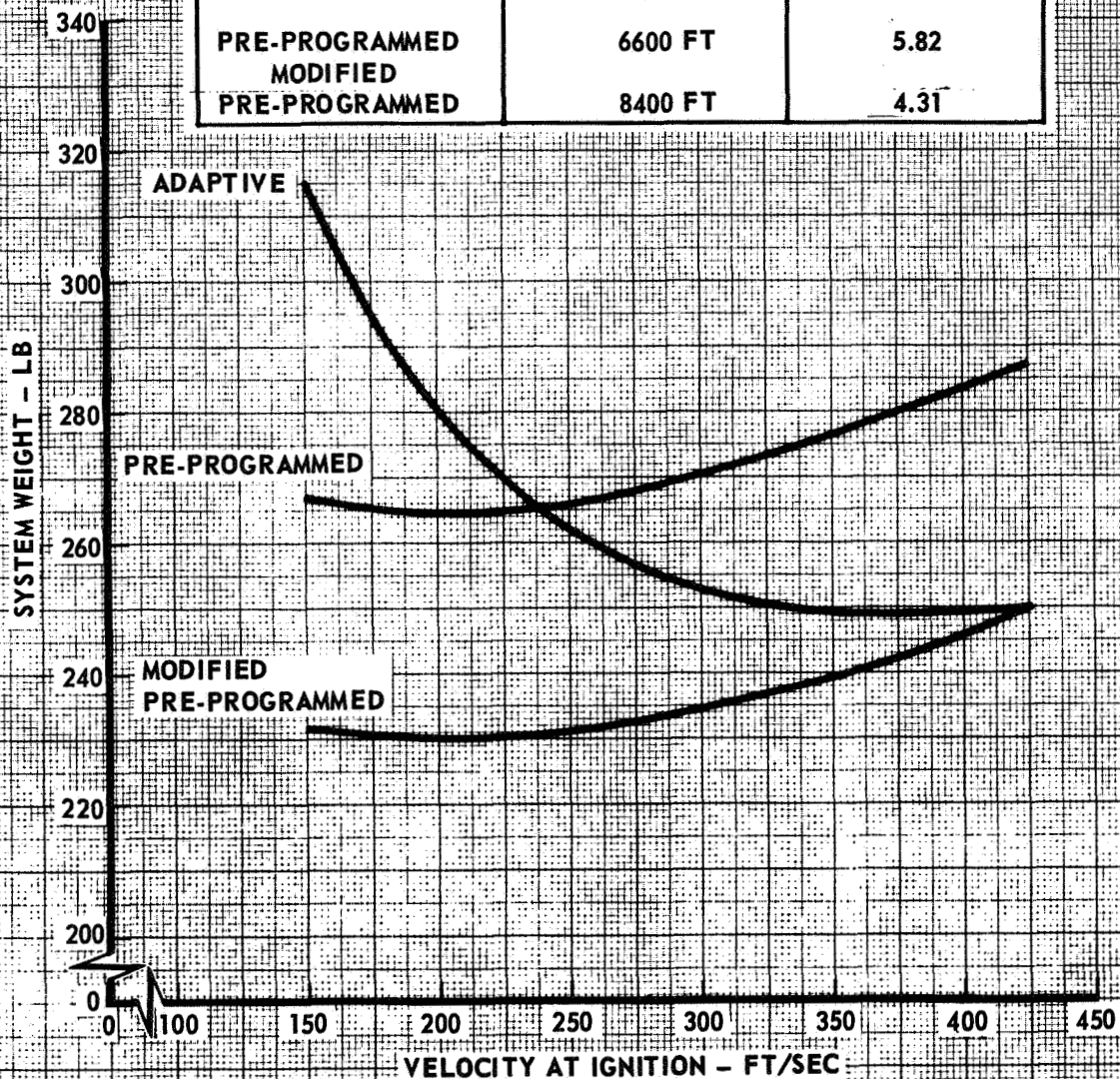


FIGURE 3.2.7-26

System Concept Comparison - The monopropellant and bipropellant configurations are compared in the following sections on the basis of weight and performance, reliability, cost, development status, and system interactions.

o Weight and Performance - System weight for both the monopropellant and bipropellant configurations is presented in Figure 3.2.7-27 as a function of lander weight for the modified pre-programmed guidance mode. Weights are based on the optimum throttle ratio and ignition altitude derived in the previous section. At the request of the NASA (LRC), system weights were also evaluated for a throttle ratio of 10:1 in order to provide margin for capsule growth. Ignition was assumed to occur at lower altitude than for the optimum throttle ratio, to maintain high performance and reduce the total impulse requirement. The results are presented in Figure 3.2.7-28. For the range of lander weights considered, the monopropellant system is significantly heavier than the bipropellant system.

Estimated vacuum specific impulse as used in the system trade studies is shown in Figure 3.2.7-29 as a function of thrust ratio for both the monopropellant and bipropellant systems. These data were adjusted assuming a constant 5 mb ambient pressure, and then used to evaluate mission-average specific impulse for each of the guidance modes previously considered.

o Development Status - The terminal propulsion system has been categorized in this study as a long lead time item. The program requires complete qualification within approximately three years from Phase D go-ahead to meet a launch in 1973.

Heat sterilization and engine throttling represent the greatest development problems. Sterilization testing to evaluate liquid propellant/materials compatibility has been accomplished at JPL, Martin-Marietta, and McDonnell. Furthermore, heat sterilization of a bipropellant (MMH/N₂O₄) propulsion system by Martin and a monopropellant N₂H₄ system by JPL has demonstrated the feasibility of sterilizing flight systems. Material compatibility and propellant decomposition are the major problems associated with liquid propellant sterilization and storage. Titanium is the only metal known to be compatible with nitrogen tetroxide at the sterilization temperature. Titanium should also be

TERMINAL PROPULSION SYSTEM WEIGHT

- CONSTANT IGNITION ALTITUDE
- MODIFIED PRE-PROGRAMMED GUIDANCE
- MAXIMUM VELOCITY = 425 FT/SEC

———— MONOPROPELLANT N_2H_4
 - - - - - BI-PROPELLANT MMH/N_2O_4

	IGNITION ALTITUDE-FT	THROTTLE RATIO
N_2H_4	8400	4.31
MMH/N_2O_4	7300	4.86

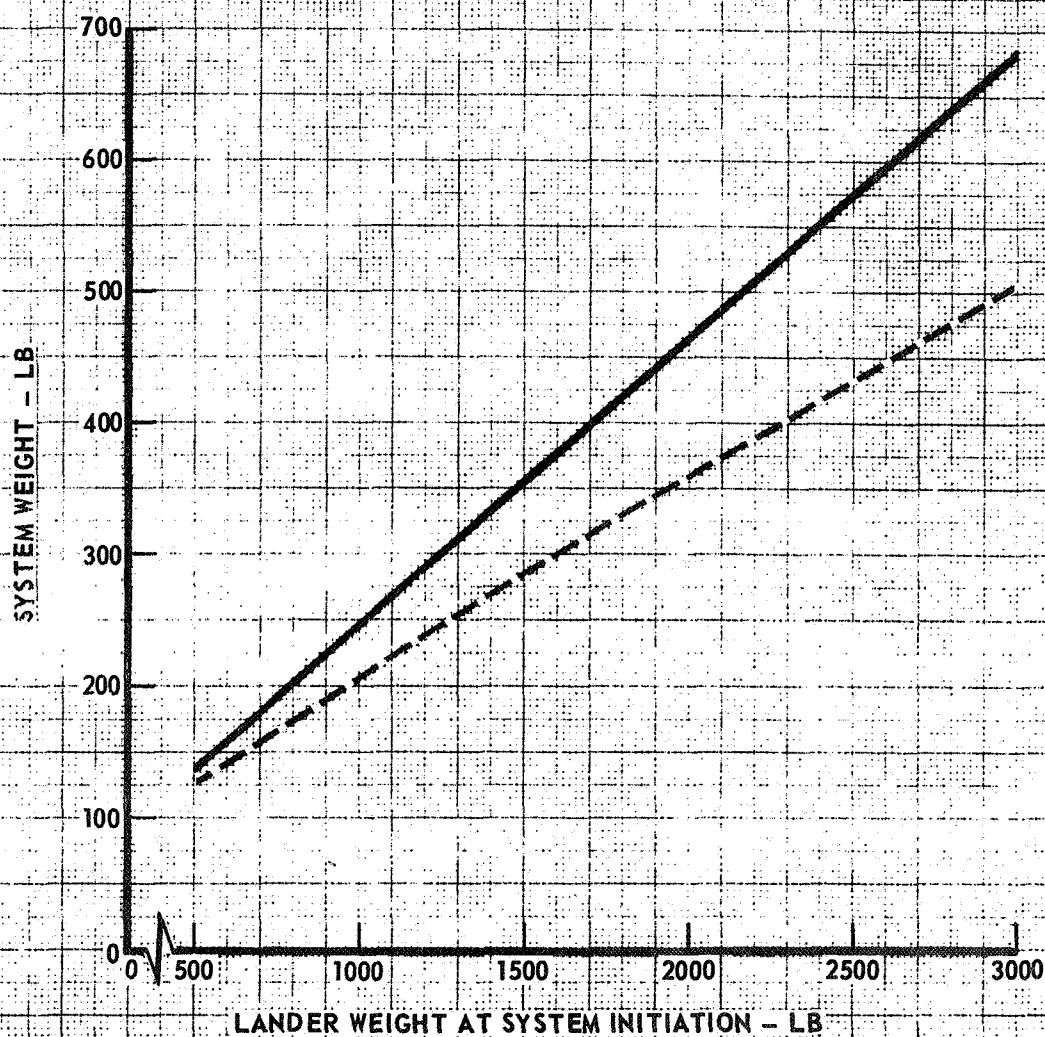


FIGURE 3.2.7-27

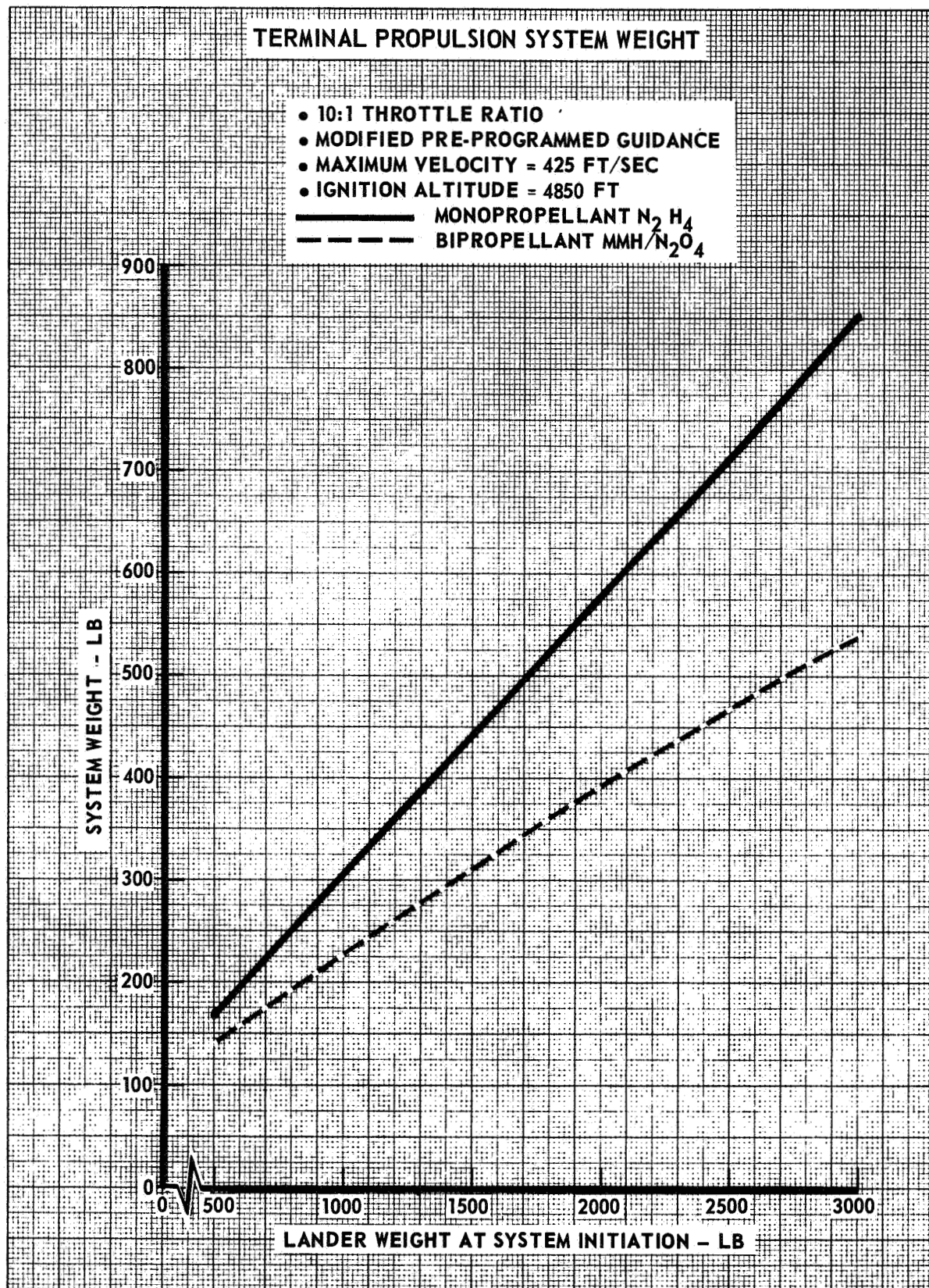


FIGURE 3.2.7-28

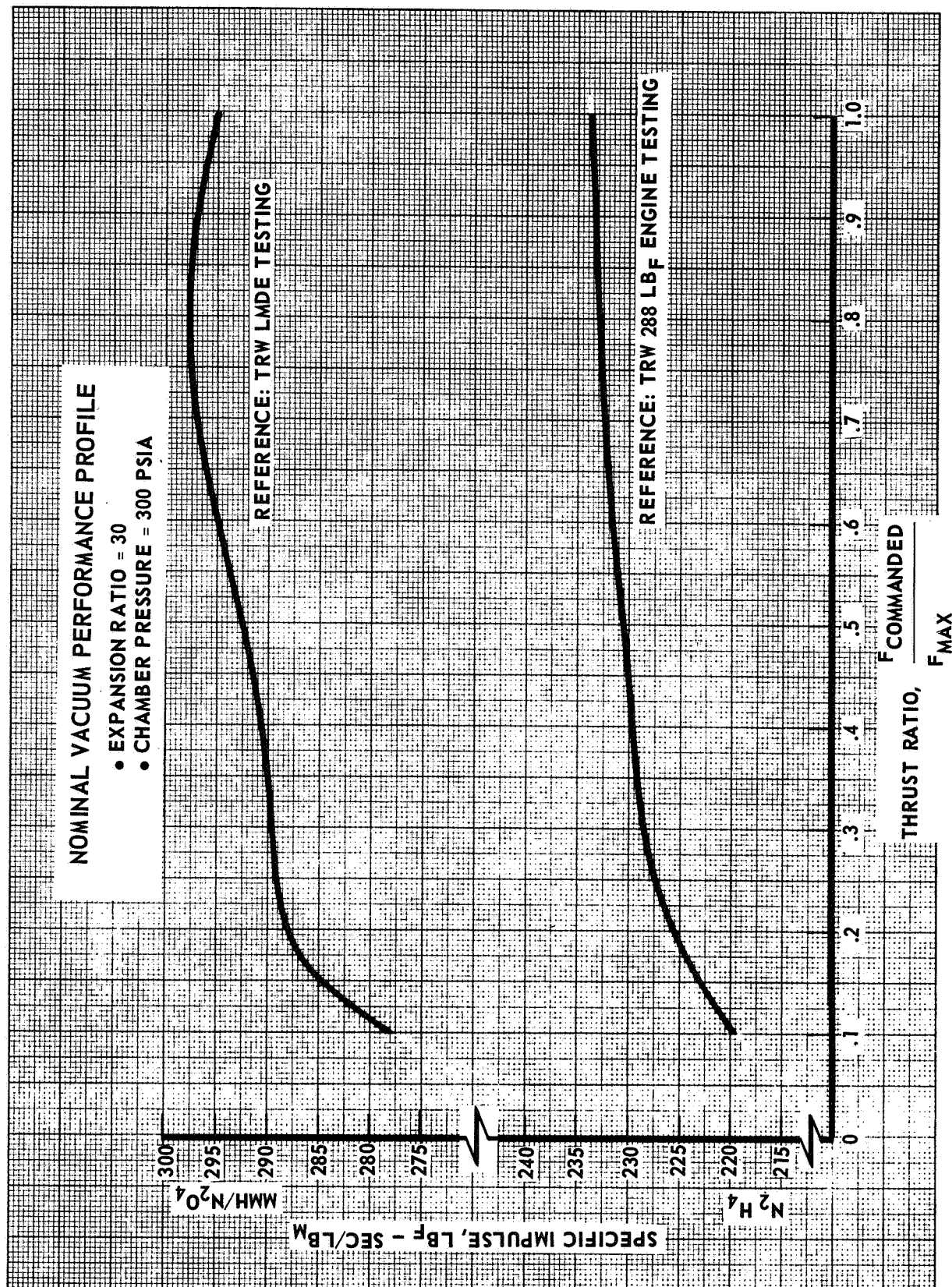


FIGURE 3.2.7-29

used for N_2H_4 and MMH tankage to minimize propellant decomposition. Elastomeric seals exposed to N_2O_4 during sterilization should be avoided. If seals are required, additional development work must be done. Teflon is the best candidate for seal material.

Throttling is considered to be the most critical problem in selection of the engine. Throttling by varying the injector area is the most developed technique for achieving throttle ratios in excess of 5:1 in bipropellant engines. The concept has been flight-proven on the Apollo Lunar Module Descent Engine (LMDE) and the LANCE missile. The LMDE concept represents the more conservative design approach, since the flow control function is separated from the high temperature injector by incorporation of upstream cavitating venturi throttle valves. These valves serve two functions: they control propellant flow rate to the engine, and they decouple the feed system from chamber or injector manifold pressure oscillations, thereby minimizing the possibility of overall system instability. Although the LMDE has been qualified using Aerozine 50 (50/50 N_2H_4 -UDMH) and N_2O_4 , its adaptability to MMH- N_2O_4 has also been demonstrated. The LMDE employs a concentric tube injector element which inherently provides high combustion stability, as demonstrated by "bomb" tests over the full throttle range.

The LANCE sustainer has been throttled over 50:1 using IRFNA and UDMH, but when it was operated with Aerozine 50 and N_2O_4 , severe erosion of the throat and pintle occurred in approximately 11 seconds. Furthermore, without extensive modification, the engine cannot be refired, since heat soak back after engine shutdown warps the injector propellant flow control ring beyond use. (Restart capability is required in lander applications for pre-flight test purposes.)

Due to its high level of design maturity and inherent high stability margins, the LMDE throttle concept is preferred for bipropellant applications.

Monopropellant throttling technology is still in its early stages of development. However, since only one propellant is employed, control of injection velocity is not as critical as it is for bipropellant engines. Feasibility testing performed by several manufacturers has demonstrated throttle ratios

up to 10:1, using simple upstream flow control. Nevertheless, a few design problems have been uncovered which tend to limit throttle range. During full thrust operation, the bed loading must be limited to avoid excessive catalyst bed pressure drop, which could result in crushing of the catalyst particles. Catalyst crushing may result in the generation of "fines" and subsequent loss of catalyst from the reaction chamber which can cause very rough operation. Also, at low thrust the catalyst bed pressure drop is low permitting the propellant to seek preferential paths (channeling) between catalyst particles. This condition results in limited catalyst wetting. Therefore, ignition with a cold bed should be restricted to full throttle settings to avoid flooding or excessive ignition pressure spikes.

- o Reliability - Failure rate data utilized in this study were obtained from other programs employing similar components and design. Although these data cannot be applied directly to planetary landing missions because more stringent requirements will be imposed by sterilization and long term space storage, they are satisfactory for relative system comparisons. Table 3.2.7-8 shows a reliability estimate of .988 for the monopropellant system and .979 for the bipropellant.

- o Cost - Total program costs were estimated for both monopropellant and bipropellant systems based upon development, qualification and delivery of six systems. Relative costs are compared in Figure 3.2.7-30 as a function of total capsule weight. As illustrated for a lander weight of 1000 lb, bipropellant system costs are approximately 60% higher than monopropellant costs.

- o System Interactions - The terminal propulsion system engine exhaust interacts with the capsule and landing site. The primary interactions with the capsule systems result from exhaust plume/radar interference, and vehicle base heating due to exhaust recirculation. The major landing site interactions are exhaust contamination, erosion, and surface heating.

Free electrons in rocket exhaust plumes can cause interference with electromagnetic waves originating from vehicle guidance and communication systems. The terminal engine exhausts are potential sources of free electrons which could interact with the landing radar causing wave attenuation, phase

TABLE 3.2.7-8

TERMINAL PROPULSION SYSTEM RELIABILITY ANALYSIS

CANDIDATE SYSTEM		THREE ENGINE MONOPELLENT			THREE ENGINE BIPELLENT		
COMPONENT	FAILURE RATE EACH COMPONENT (λ) $\times 10^{-6}$	NUMBER OF COMPONENTS (n)	DUTY CYCLE (KT)	$-nR \times 10^6$	NUMBER OF COMPONENTS (n)	DUTY CYCLE (KT)	$-nR \times 10^6$
PRESSURANT TANK	.08/HR	1	5566 HR	445	1	5566 HR	445
PRESSURANT FILTER	25/CY	1	1 CY	25	1	1 CY	25
PRESSURE REGULATOR	16/CY	1	87 HR	1392	1	87 HR	1392
PYROTECHNIC VALVES	250/CY	5	1 CY	1250	5	1 CY	1250
CHECK VALVES	1/CY	-	-	-	2	50 CY	100
BURST DIAPHRAGMS	10/CY	-	-	-	1	1 CY	10
PROPELLANT TANKS	53.5/CY	2	2 CY	214	2	2 CY	214
PROPELLANT FILTER	52/CY	1	1 CY	52	2	1 CY	104
THROTTLING MECHANISM (MONOPELLENT)	50/SEC	3	50 SEC	7500	-	-	-
THROTTLING MECHANISM (BIPELLENT)	100/SEC	-	-	-	3	50 SEC	1500
SOLENOID VALVES	127/CY	3	1 CY	381	6	1 CY	762
THRUST CHAMBER (MONOPELLENT)	5.16/SEC	3	50 SEC	774	-	-	-
THRUST CHAMBER (BIPELLENT)	10/SEC	-	-	-	3	50 SEC	1500
GIMBAL ASSEMBLY	1/HR	1	87 HR	87	1	87 HR	87
GIMBAL ACTUATOR	28.6/CY	1	10 CY	286	1	10 CY	286
$\Sigma n \cdot \lambda \cdot KT = n R \times 10^6$		TOTAL		12406	TOTAL		21175
RELIABILITY (R)				.988			.979

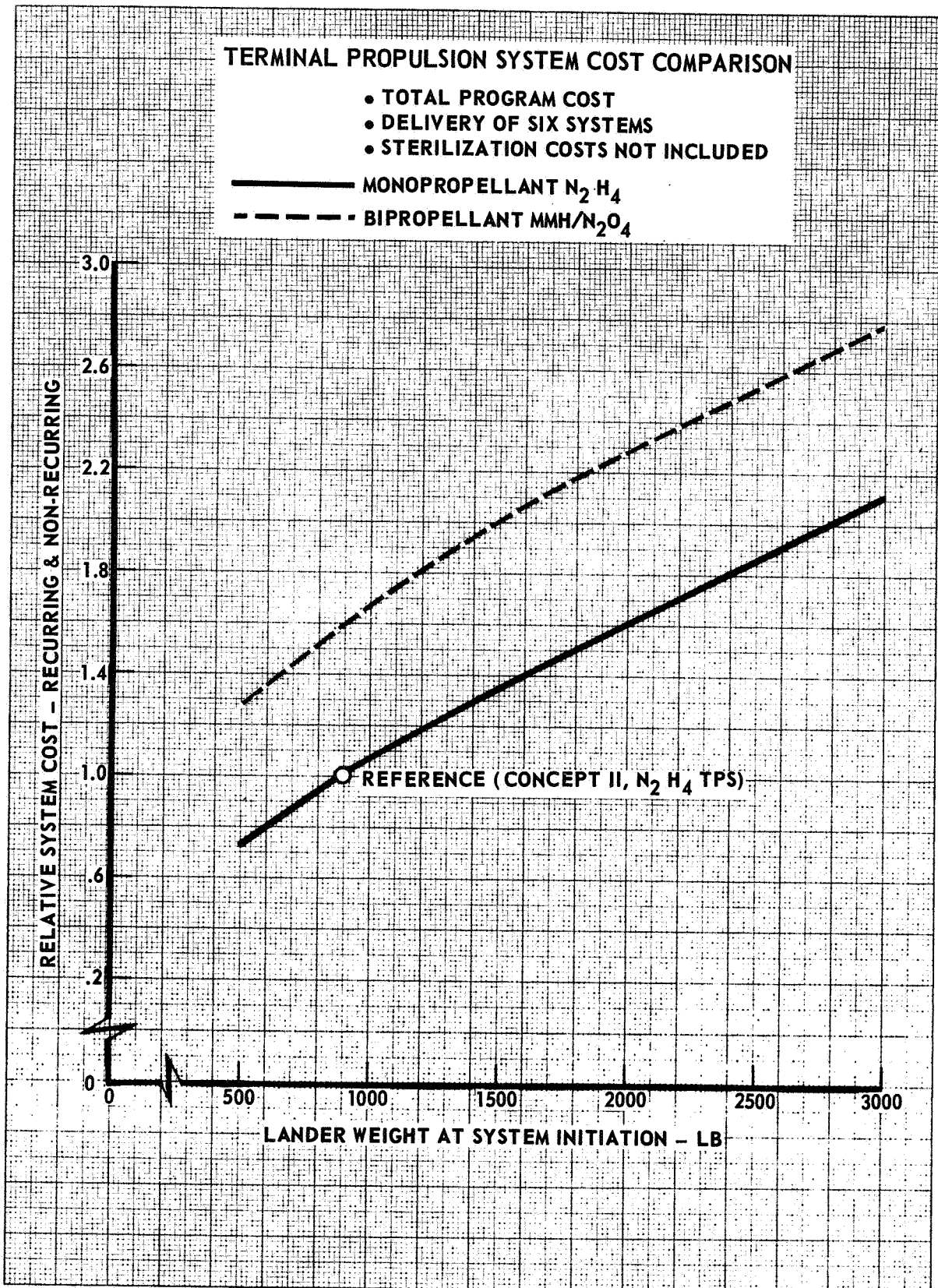


FIGURE 3.2.7-30

shift, noise, reflection, or refraction. The principal factors determining the degree of interference are free electron concentration, electron collision frequency, transmitting frequency, and plasma depth. Analytical calculations of radar wave attenuation in rocket exhausts, assuming only thermal ionization of the gaseous species, have compared favorably with experimental measurements. The predominant contributors to thermal ionization at typical rocket chamber flame temperatures are chemical species that possess low ionization potentials, specifically the alkali metals. Small amounts of sodium and potassium are usually present in the form of propellant and ablative nozzle material contaminants. Their presence, even in concentrations of a few parts per million, usually accounts for the bulk of free electrons. Potassium and sodium concentrations in the Titan II propellants (N_2O_4 and Aerozine 50) are listed below:

	<u>Na (ppm)</u>	<u>K (ppm)</u>
N_2O_4	.04	.04 - .07
Aerozine 50	.2 - .5	1.0 - 1.7

Data on the impurity levels in MMH and N_2H_4 were not available, but it has been assumed that impurity concentrations are similar to Aerozine 50. Due to lower chamber temperatures in the N_2H_4 reactor (1800°F compared with 5100°F for MMH and N_2O_4) the degree of ionization is slight, yielding lower electron densities and collision frequencies than a bipropellant engine of comparable size.

The three-engine configuration evaluated in this study introduces a potential for exhaust recirculation or backflow when the highly expanded jets impinge on one another. As a result of shock patterns and attendant pressure gradients generated by the intersecting plumes, a portion of the rocket exhaust gases in the plume boundaries is forced back toward the vehicle base. This reverse flow gives rise to base heating. Due to lower jet total temperatures, base heating would be significantly less for a N_2H_4 engine system than for a bipropellant.

Analyses conducted during the Voyager Phase B Study and reported in NASA CR-89678 showed that radar wave attenuation and base heating for a four engine bipropellant configuration were within acceptable limits. Therefore, these effects should not pose a significant problem for either of the candidate configurations employed as a baseline in this study.

Contamination and alteration of the landing site cannot be avoided with either of the concepts considered. On the other hand, the bipropellant engine exhaust contains water vapor and undersirable carbonaceous products which could interfere with on-site scientific measurements; the hydrazine engine contains neither. Surface heating is also less for the monopropellant concept due to lower jet stagnation temperatures.

3.2.7.3.4 Summary: Studies reported in this section show that the modified pre-programmed terminal guidance mode, as compared with other candidate guidance schemes, permits a significant reduction in propulsion system weight over the full range of capsule weights studied (1000-5000 lb). Using the modified pre-programmed guidance mode as a baseline, evaluations of candidate monopropellant and bipropellant terminal propulsion systems were conducted considering weight, reliability, cost, development status, and interactions with other capsule systems. A summary of these evaluations is presented in Table 3.2.7-9. For small landers, the hydrazine monopropellant system offers many advantages. Among these are high reliability, low cost, and minimum interaction with the landed science. The major disadvantage is that it is heavier than the bipropellant systems over the full spectrum of capsule weight. The relatively low throttle ratios required for the modified pre-programmed guidance mode should not pose unacceptable development risk for either propulsion concept.

Final selection of a preferred terminal propulsion system must be based on a consideration of combined aerodynamic (parachute or aerodynamic inflatable decelerator - AID) and propulsive deceleration schemes. This evaluation is made in Section 3.3.1.

TABLE 3.2.7-9
TERMINAL PROPULSION SYSTEM TRADE STUDY SUMMARY
(MODIFIED PREPROGRAMMED GUIDANCE)

SYSTEM	OPTIMUM THROTTLE RATIO	OPTIMUM IGNITION ALTITUDE, FT	SYSTEM WEIGHT, LBS		RELIABILITY	RELATIVE COST*		DEVELOPMENT STATUS	SYSTEM INTERACTIONS
			TOTAL CAPSULE WEIGHT			TOTAL CAPSULE WEIGHT			
			1000 LB	5000 LB		1000 LB	5000 LB		
MONO-PROPELLANT N ₂ H ₄	4.31	8400	150	681	.988	.78	2.13	ADDITIONAL THROTTLING EXPERIENCE IS REQUIRED. COMPATIBILITY WITH HEAT STERILIZATION HAS BEEN DEMONSTRATED BUT THE PROPELLANT IS SUBJECT TO EXPLOSIVE DECOMPOSITION IN THE PRESENCE OF CONTAMINATES. CATALYST MUST BE HERMETICALLY SEALED AGAINST ETO STERILANT.	SURFACE EROSION AND CONTAMINATION LEVELS FOR BOTH SYSTEMS ARE ESSENTIALLY THE SAME. HOWEVER, THE N ₂ H ₄ ENGINE EXHAUST CONTAINS NONE OF THE UNDESIRABLE CARBONACIOUS SPECIES OR H ₂ O VAPOR WHICH COULD INTERFERE WITH ON-SITE SCIENTIFIC MEASUREMENTS. ALTHOUGH LEVELS APPEAR ACCEPTABLE WITH EITHER SYSTEM, EXHAUST ELECTRON CONCENTRATION AND VEHICLE BASE HEATING ARE MINIMIZED WITH THE N ₂ H ₄ SYSTEM, DUE TO LOWER JET TOTAL TEMPERATURES.
BIPROPELLANT MMH/N ₂ O ₄	4.86	7300	137	503	.979	1.34	2.80	COMPLIANCE OF MMH AND N ₂ O ₄ SYSTEMS WITH HEAT STERILIZATION REQUIREMENTS HAS BEEN DEMONSTRATED. THESE PROPELLANTS POSSESS GREATER THERMAL STABILITY THAN N ₂ H ₄ BUT ADDITIONAL TESTS ARE REQUIRED ON PROPELLANT COMPATIBILITY WITH STORAGE TANK MATERIALS. REQUIRED THROTTLING CONSIDERED STATE OF THE ART USING LMDE TECHNOLOGY.	

* VALUES SHOWN ARE REFERENCED TO THE COST OF A MONOPROPELLANT TERMINAL PROPULSION SYSTEM FOR AN 873 LB LANDER

3.2.8 AUXILIARY AERODYNAMIC DECELERATOR - Deceleration of the capsule to a velocity (~ 400 ft/sec) compatible with a minimum size terminal propulsion system will limit the maximum capsule ballistic parameter ($m/C_D A$) to values of the order of 0.05 slugs/ft². Since this ballistic parameter does not allow sufficient weight for reasonable diameter capsules, use of an auxiliary aerodynamic decelerator is necessary.

3.2.8.1 Requirements. - The candidate auxiliary decelerator must satisfy certain requirements. These include:

- o The Mach number and thermodynamic performance of the system must be compatible with deployment in the low density Martian environment.
- o Inflation and deceleration must be compatible with the terminal propulsion system required for soft landing.
- o The decelerator and suspended weight must be aerodynamically stable.
- o The decelerator system must be light-weight.
- o The decelerator must package to a small volume in the capsule.
- o The decelerator system must minimize total system cost.
- o Development time must be compatible with the 1973 launch opportunity.

In addition, a promising technique for separating the aeroshell from the landing configuration is to use the auxiliary decelerator to pull the lander out of the aeroshell. If this separation technique is used, the ballistic parameter of the decelerator must be smaller than the aeroshell's in order to prevent recontact.

3.2.8.2 Candidate concepts. - The primary concepts which have been considered are the parachute and a tucked-back Attached Inflatable Decelerator System (AIDS), based on studies made for VOYAGER Phase B, NASA CR-8976. Other concepts, including rigid or inflatable conical aeroshell extension, drag brakes, tori, and rotors, are beyond this study. The AIDS provides a pressurized decelerator operating in the primary flow field with a rear opening to the lander structure for deorbit motor attachment. The parachute operates in the wake flow field. Some parachute experience has been gained from the Planetary Entry Parachute Program (PEPP), to simulate the Martian

operation. Three types of parachutes (disk-gap-hand, cross and ringsail), were used in these tests.

3.2.8.3 Performance. - The primary measure of the decelerator performance is its drag coefficient. Estimates for the AIDS (based on inflated cross-sectional area, excluding fence) are presented in Figure 3.2.8-1. For preliminary design purposes, a parachute drag coefficient of 0.6 (based on total surface area, including gaps) is consistent with PEPP data and is used in this study. The ratio of suspension line length to parachute diameter is 1.3. Analysis of the PEPP data also shows that the parachute drag coefficient is relatively constant with Mach number once full inflation is attained. For a 15 percent porosity parachute such as the ringsail, the drag coefficient (C_{D_0}), based on essential cloth area, is equivalent to 0.7. The drag coefficient of the ringsail has been assumed to be constant with parachute size, so long as the number of sails is kept constant. If the parachute drag coefficient were based on its inflated cross-sectional area as on the AIDS, then it would be about 1.3.

The use of the parachute has been limited to less than Mach 2.0 by the constraints of this study. Very little experimental data exists at higher speeds. The speed limit on the AIDS actually corresponds to a temperature limit on the material strength. The limits on deployment altitude with ballistic parameter are shown in Figure 3.2.8-2. Local equilibrium temperatures of 300°F at the aeroshell-AIDS interface are considered to be limiting values because of the rapid degradation of strength with increasing temperature.

The PEPP program provides experimental data in a low density environment similar to that expected in the Martian atmosphere. This data was used for parametric and point design studies. A summary of the canopy weights, fill time, and opening shock factors is presented in Figures 3.2.8-3 through -5. In our analyses, the parachute nominal diameter, D_0 , is based on the essential canopy cloth area, S_0 (excluding gaps). This definition provided a better data correlation than that based on total surface area (including gaps).

Figure 3.2.8-3 presents canopy weights versus nominal diameter (D_0) for the three types of PEPP parachutes. Weight curves for a ringsail canopy in an Earth environment are provided for comparison. The faired line through the PEPP data was used for canopy weight design and reflects the more conservative

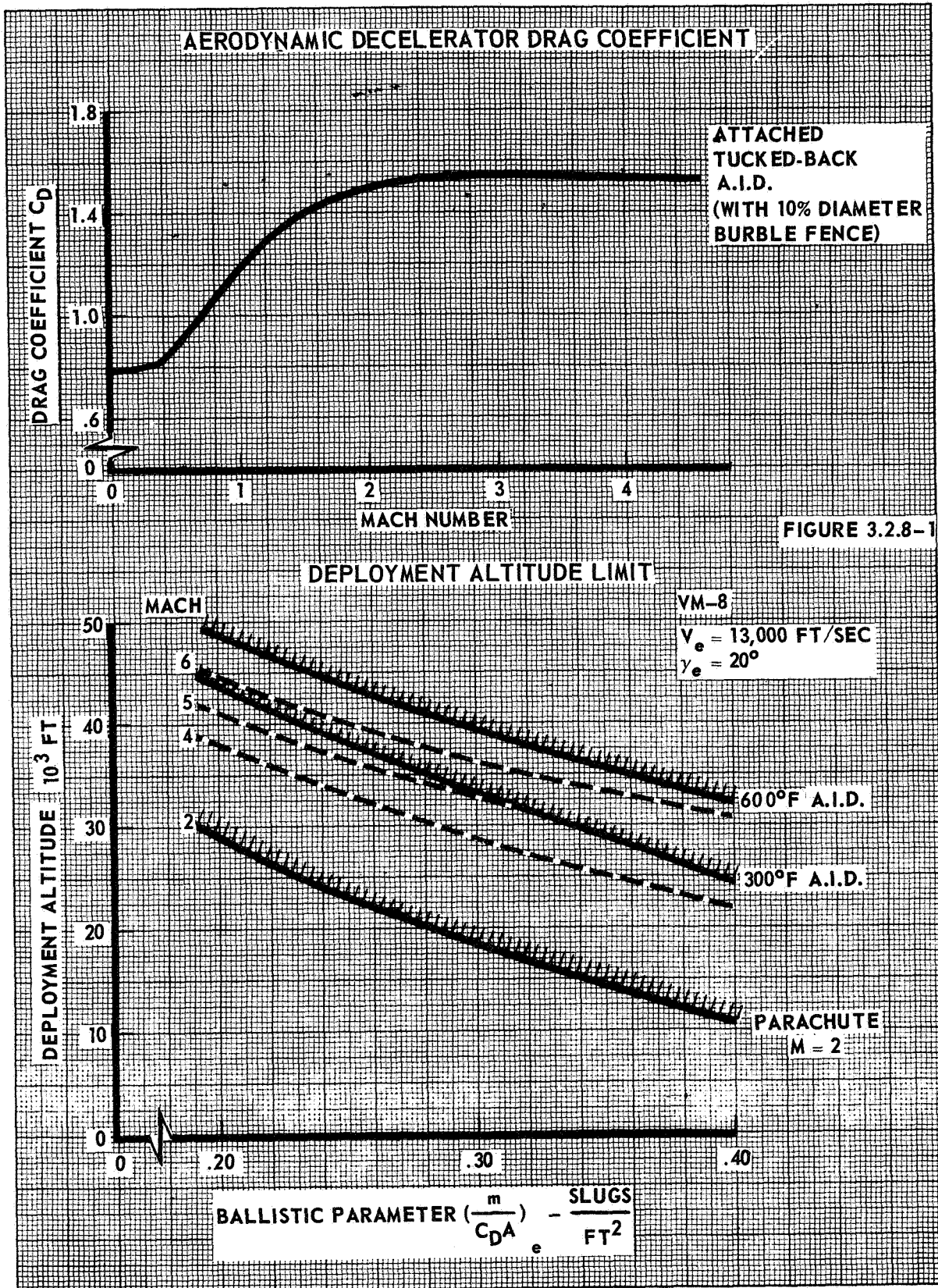
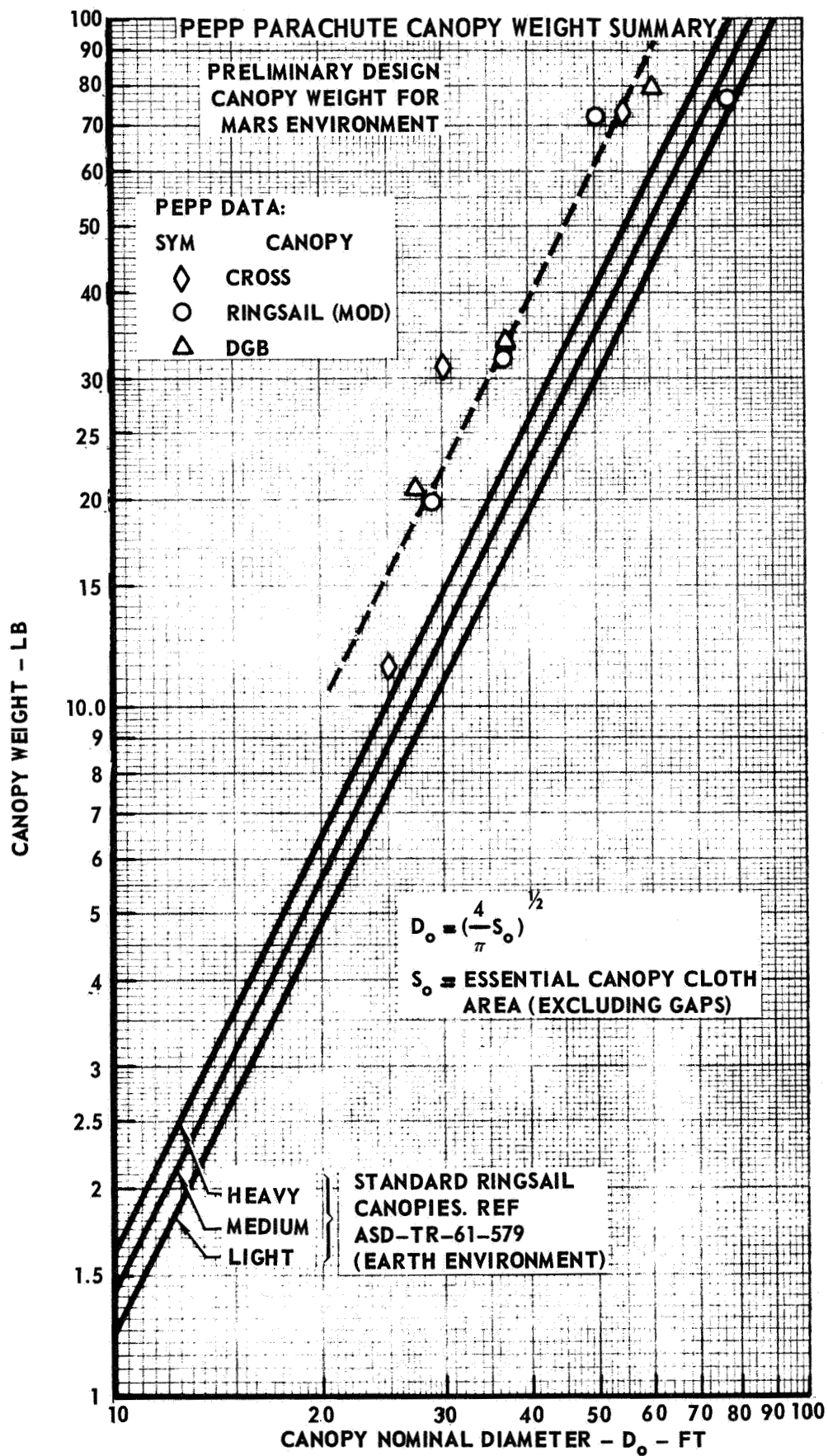


FIGURE 3.2.8-2



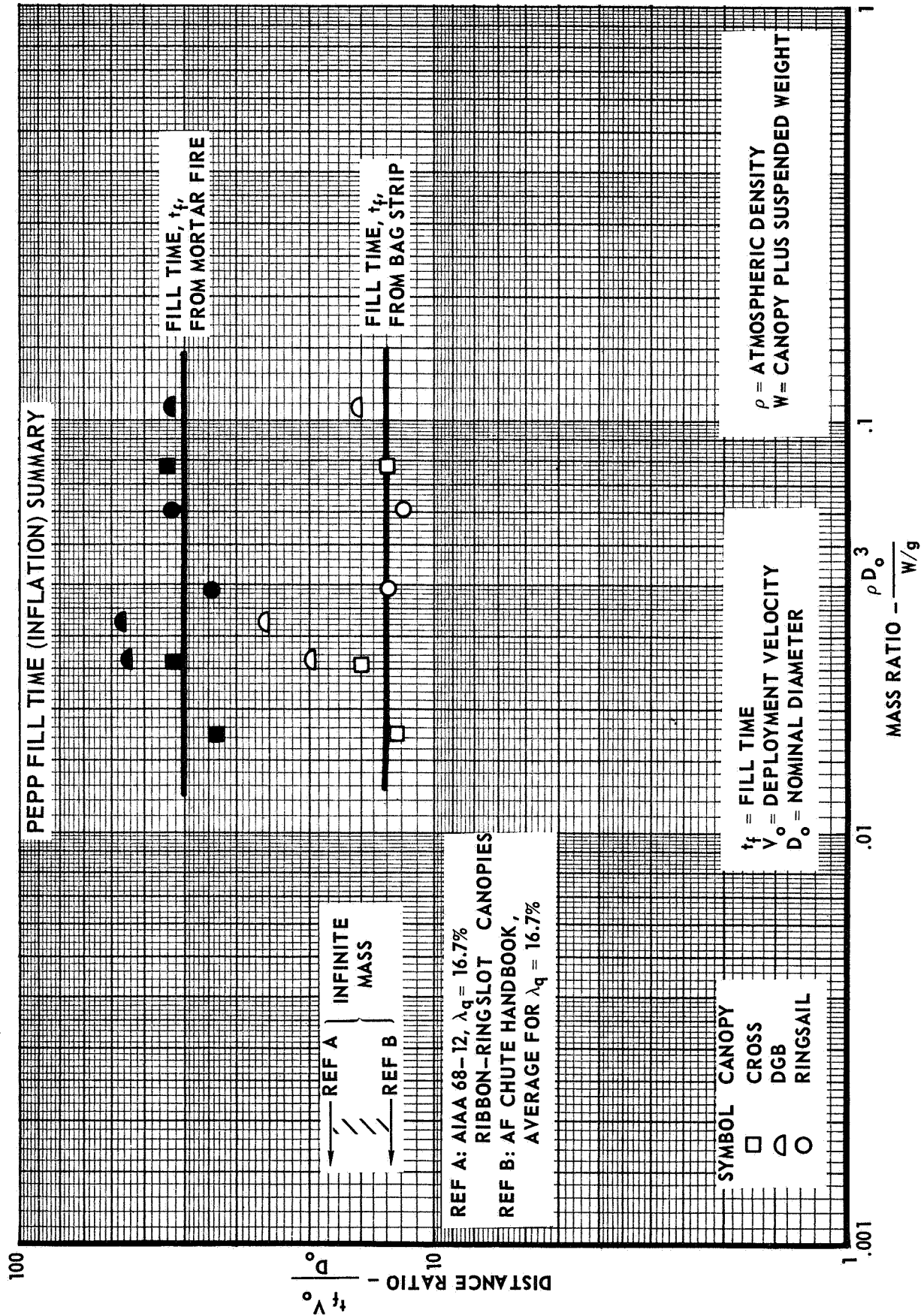
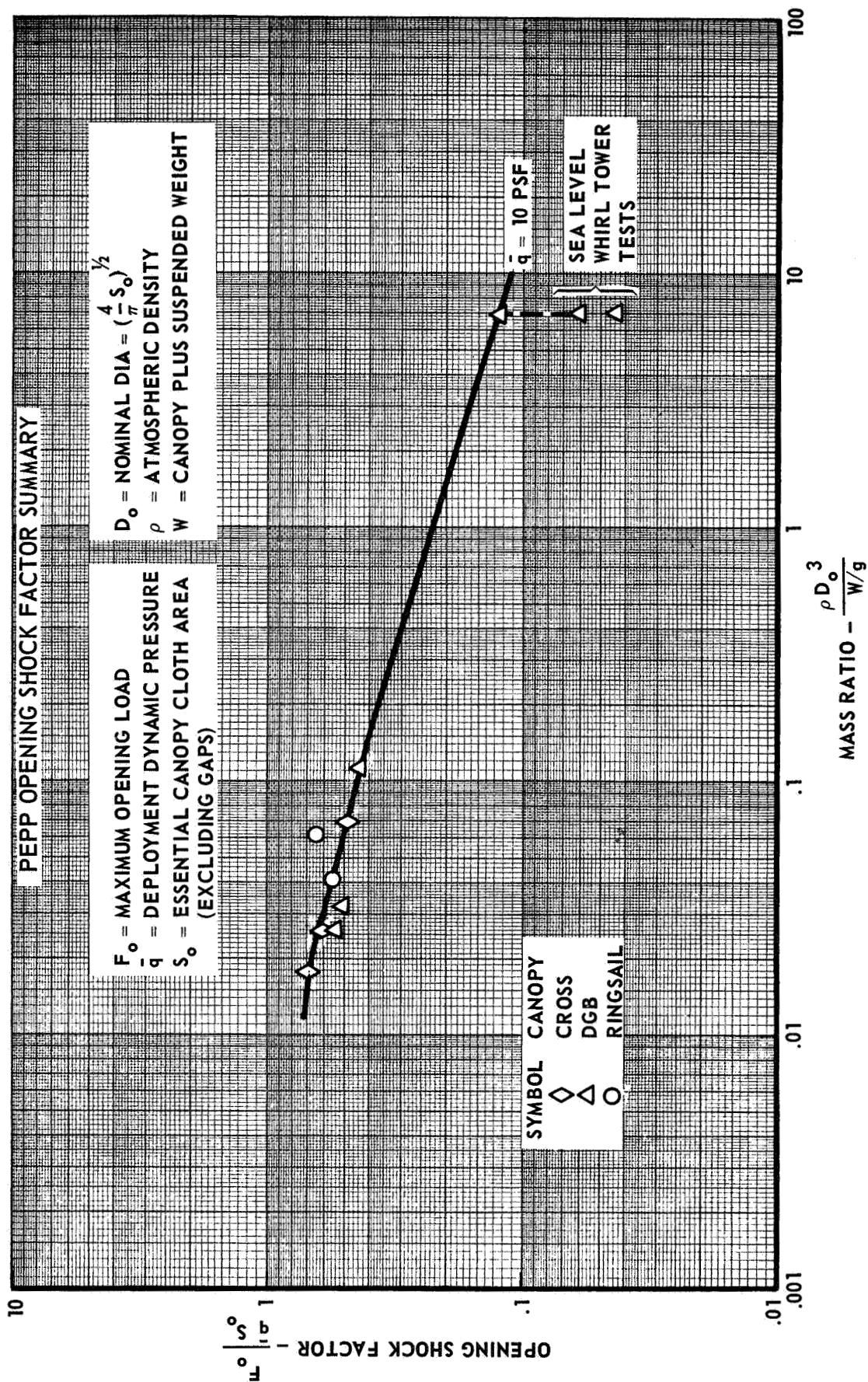


FIGURE 3.2.8-4



design requirements for Martian entry. The PEPP designs were based on dynamic pressures between 7 and 12 psf at full inflation. Maximum dynamic pressure for Mach 2 deployment at 23 000 ft was shown in Figure 3.1.4-13 to be below 15 psf.

Figure 3.2.8-4 presents a correlation of inflation fill times from the PEPP data in terms of distance ratio and mass ratio parameters. The distance ratio parameter from mortar fire to full inflation is fitted as a constant, 40, and the same parameter referenced to the time from the beginning of inflation (deployment bag strip) to full inflation is 13. The latter agrees very well with predictions for infinite mass conditions.

Correlation of the opening shock factor as a function of the mass ratio parameter at deployment is presented in Figure 3.2.8-5. A more detailed analysis would be required to account for the difference in dynamic pressure gradient from deployment to full inflation in the Martian atmosphere as compared to the gradient in the Earth-based simulated tests. Slightly higher opening loads would occur in the Martian atmosphere. In the event that the opening shock loads as shown in Section 3.3.1.2.4 are excessive, further investigation will be necessary to evaluate the trade between heavier construction and reefing for specific designs.

In addition to the parachute canopy, the decelerator system includes a riser assembly, parachute deployment bag, catapult and attachment structure. The weight of these items is calculated as a function of canopy weight, aeroshell diameter, initial loads and differential drag. Weight distribution of the decelerator system is illustrated for a typical case in Figure 3.2.8-6; detailed description is available in McDonnell's VOYAGER Capsule Phase B Study Report, NASA CR 89673 and CR 89688.

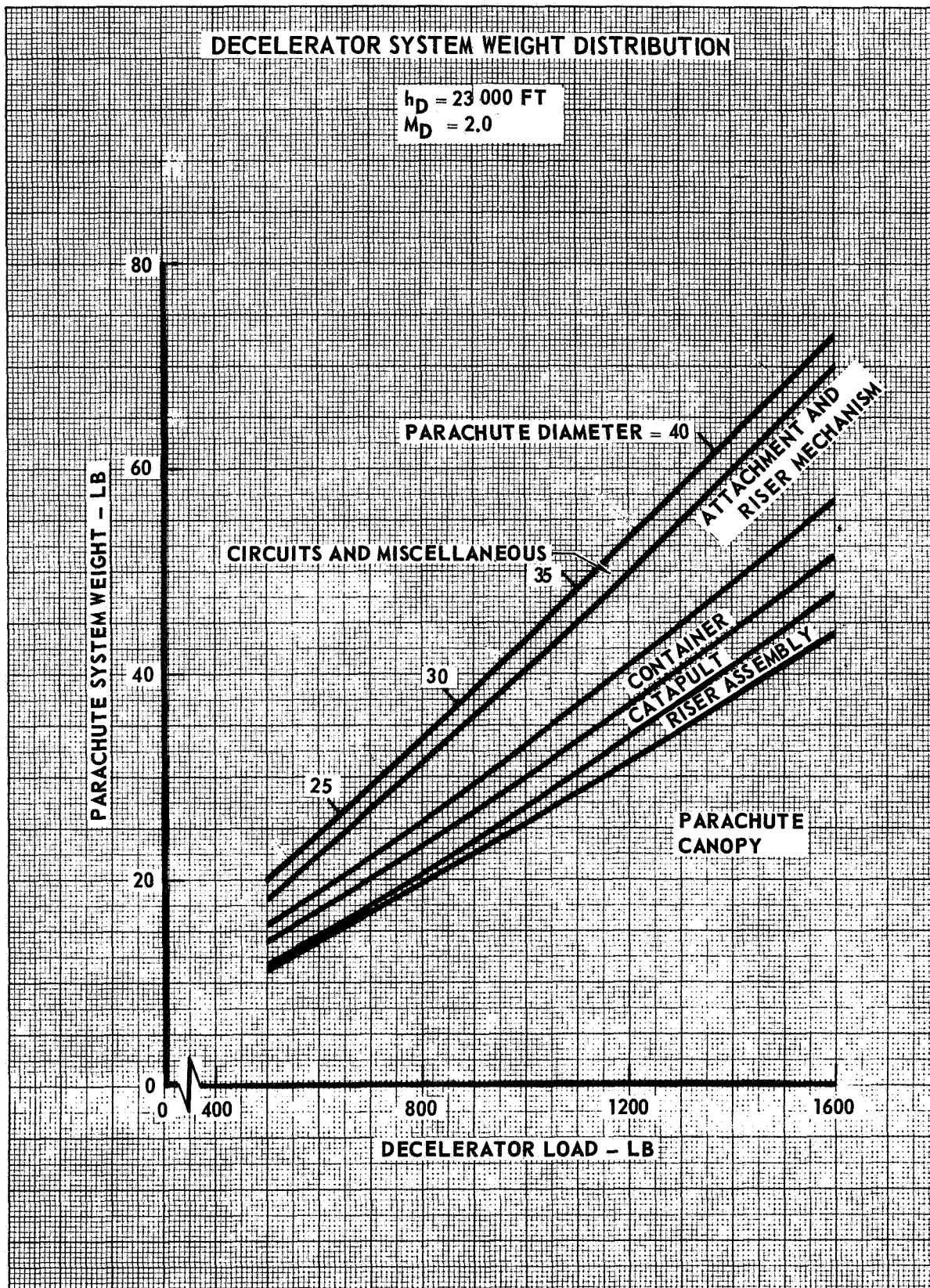


FIGURE 3.2.8-6

3.2.9 AEROSHELL - The aeroshell decelerates the entry capsule and provides thermal protection to the capsule during ballistic entry into the Martian atmosphere. It is a 60 degree half-angle sphere-cone configuration, with a spherical nose radius equal to one-half the base radius. The aeroshell structure has the capability to sustain entry loads and the heat shield provides thermal protection for the aeroshell structure and other systems of the capsule during entry.

An extensive parametric study of both the aeroshell structure and the heat shield was conducted during the VOYAGER Phase B studies (NASA CR-89677). Results of those studies were used as a basis for the parametric studies presented here. Small weight savings could possibly be realized by reinvestigation of Phase B concepts for smaller vehicles, but this detailed analysis was considered unwarranted for this parametric study.

3.2.9.1 Summary of Phase B studies. - Screening of forty structural configurations led to choosing three promising candidates which were studied in greater depth. They were the beryllium sandwich with an aluminum core, magnesium ring-stiffened monocoque, and titanium semi-monocoque with aluminum rings. The lightest configuration was the beryllium sandwich; however, it was not selected because of the cost and difficulty of fabricating and inspecting such a large sandwich structure. The magnesium ring-stiffened monocoque was eliminated because of the numerous rings and the susceptibility of magnesium to corrosion. Our selected concept therefore consisted of a single-faced, longitudinally corrugated titanium shell with internal aluminum rings. We have high confidence in this concept because of successful flight experience with similar spacecraft structure.

Approaches investigated for thermal protection of the aeroshell included metallic radiative (both hot and cold structure), heat sink, insulating non-ablators, high density charring ablators, low temperature sublimers, and low density charring ablators. Each of these approaches was assessed to select the lightest system weight that could meet the wide range of anticipated entry conditions. In addition, the flexibility of each approach to accept an off-nominal entry was considered.

Evaluation of the various approaches relative to weight, sterilization and decontamination susceptibility, low temperature and hard vacuum effects, RF transparency, fabrication, handling, cost, and flight experience showed that the low density charring ablators provide the lightest weight with a minimum number of problems expected in their life cycle preceding entry. For specialized non-ablative applications on the vehicle, such as the area around atmospheric sampling ports, the low density ceramics were preferred. Hardened compacted fibers were selected for the non-ablative nose cap. A silica cloth thermal curtain was used for thermal protection of the aeroshell base areas.

Thirteen low density ablators were selected as candidates. Extensive test programs were conducted for these candidate materials, to aid in the final selection process. A large number of thermal, mechanical and environmental tests were conducted to subject the materials to most aspects of the total mission environment, including fabrication, sterilization, launch vibration, space vacuum pressures, and simulated entry heating. In the final evaluation the recommended ablators included McDonnell S-20T, a chemically blown silicone elastomer in phenolic glass honeycomb that is pre-bonded to the aeroshell structure with HT-424 adhesive. The McDonnell S-20T has been used in this study to make weight predictions for the numerous combinations of entry conditions.

In a similar evaluation process for nose cap materials, hardened compacted fibers (HCF) of aluminosilicate were selected. This fiber mat material has uniform strength throughout its thickness. The HCF material is bonded to the substructure with HT-424, or with HT-435 adhesive when low RF transmission loss is a requirement. Thermal performance of HCF exposed to the Martian entry heating rates has been shown by tests to equal or exceed the performance of the selected low density ablators. For the mild Martian entry heating environment, the low density ablators and the HCF both behave primarily as high temperature insulators, with the vast majority of incoming heat being rejected by surface radiation.

3.2.9.2 Requirements. - In addition to the fundamental requirements stated earlier, the aeroshell also provides structural support for the capsule support equipment (entry science, altimeter antennas, RCS system) and the lander, and provides a clean (non-ablative) and geometrically stable region on the spherical nose cap where the entry science instrument ports are installed.

Entry into the Mars atmosphere is the most significant mission phase in the design of the aeroshell. Definition of the design requirements include:

- a) Selection of the entry corridor and atmospheric models.
- b) Determination of maximum loading and maximum heating conditions.
- c) Description of pressure and heating distributions over the vehicle surface and base regions, including angle-of-attack effects.
- d) Establishment of entry science requirements.

Variables having a significant influence on aeroshell weight are listed in Figure 3.2.9-1 along with the range of values considered in this study and a definition of the entry corridor.

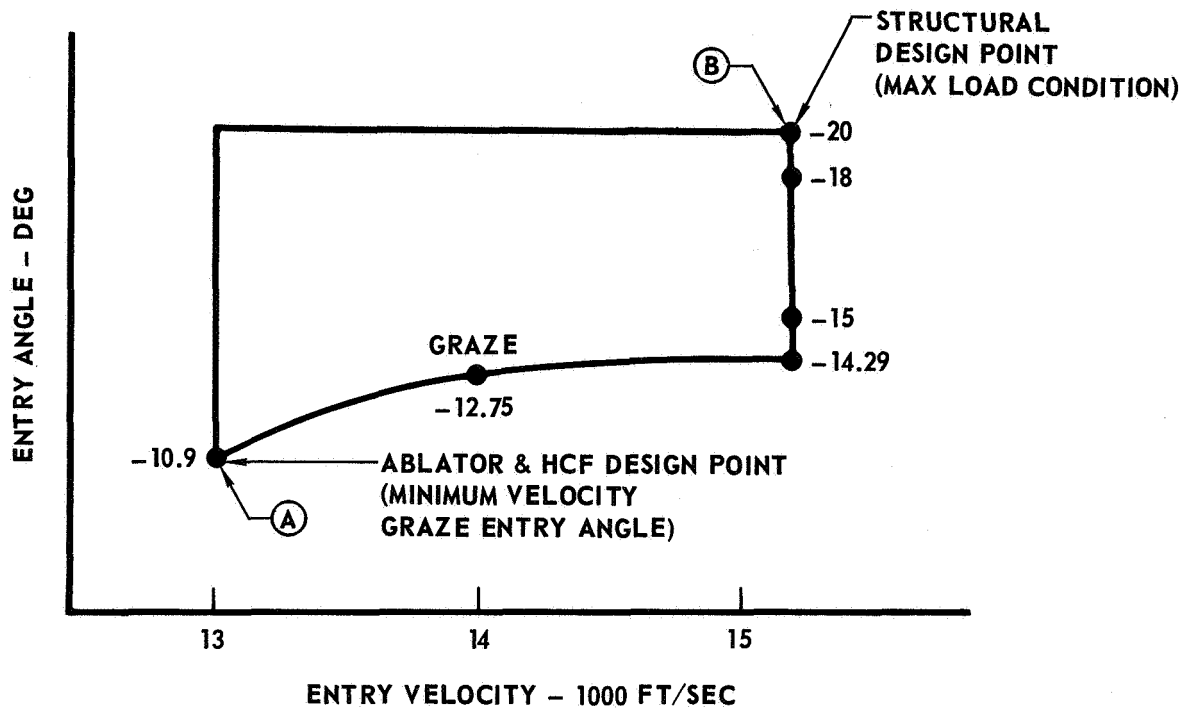
As shown, design conditions for the structure and the heat shield occur at opposite corners of the entry corridor. Furthermore, peak load occurs in the VM-8 atmosphere and maximum total heating in odd numbered VM atmospheres (e.g., VM-9). Thus, the maximum temperature at the bondline of the ablator and structure does not occur simultaneously with maximum loads. To insure structural integrity of the adhesive used to bond the ablator to the structure, the heat shield is designed to limit the maximum bondline temperature to 800°F. VOYAGER studies showed clearly that the selected structural concept is critical for the maximum loading condition rather than for the maximum heating condition.

Weight parametrics discussed below are for the full entry corridor and for the full range of VM atmospheres. The effects of changing the corridor or the environment are discussed in Section 34.

3.2.9.3 Structure. - The structural configuration used for this study is a titanium single-faced, longitudinally corrugated shell with internal aluminum rings, as shown in Figure 3.2.9-2. The conical portion consists of a forward and aft section mechanically joined at the payload ring. Each section consists of a 0.008 inch titanium 6Al-4V outer skin, stitch-welded to a 0.008 inch corrugated inner skin. The ablator is bonded to the smooth outer skin.

This configuration is designed without the need for a hoop load path in the skin. Each corrugation, plus its effective skin, function as an individual

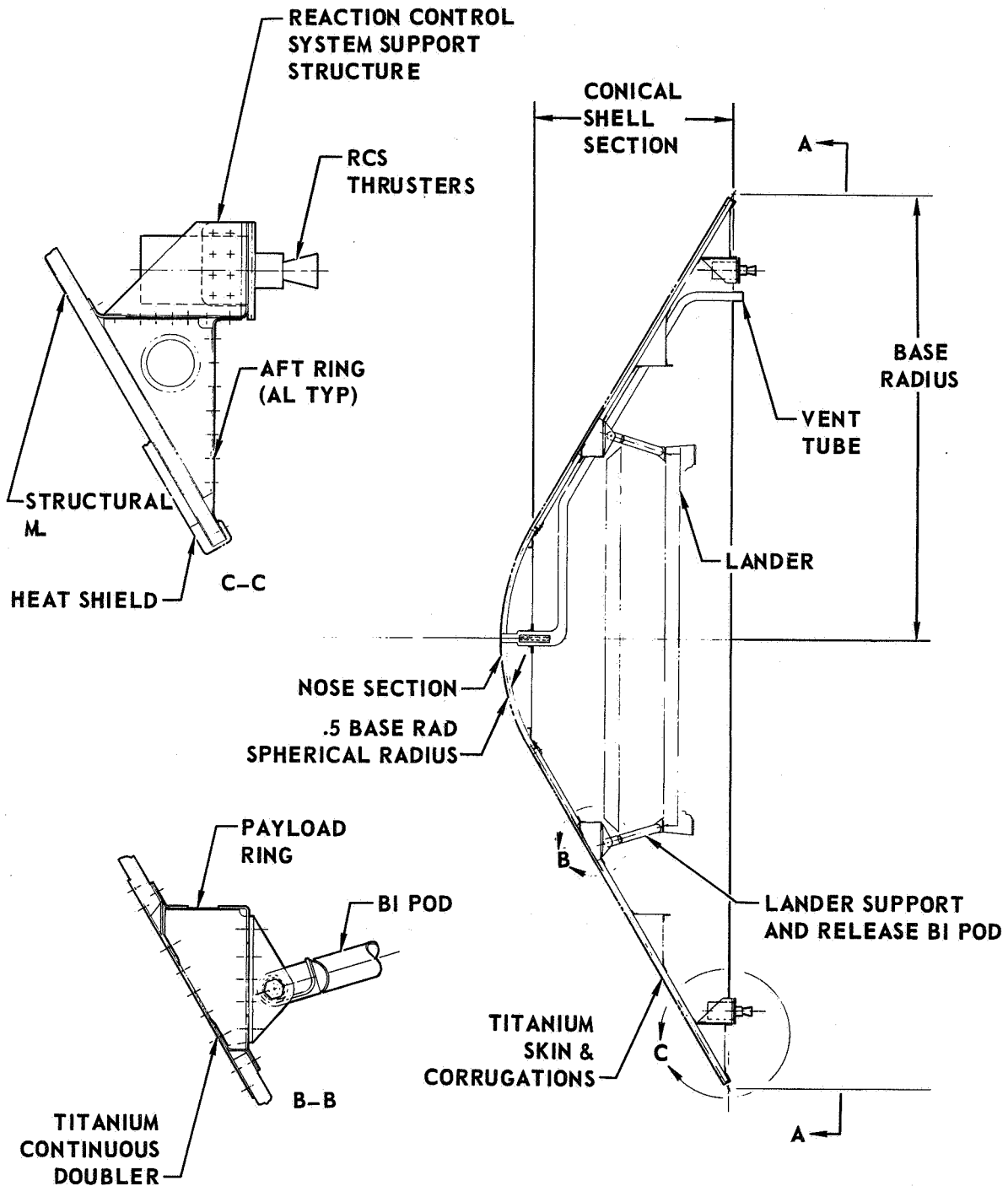
ENTRY CORRIDOR & RANGE OF STUDY VARIABLES



VARIABLE	SYMBOL & RANGE OF VALUES	UNITS
AEROSHELL DIAMETER	$8 \leq D \leq 16$	FT
ENTRY VELOCITY (@ 800 KFT)	$13000 \leq V_e \leq 15\ 200$	FT/SEC
BALLISTIC COEFF	$0.15 \leq m/C_D A \leq 0.45$	SLUGS/FT ²
ENTRY ANGLE (@ 800 KFT)	$10.9 \leq -\gamma_e \leq 20$	DEG
SCALE HEIGHT	$5.025 \leq (H_\rho) \leq 14.2$	KM
MAX. BOND LINE TEMP	$400 \leq T_{BL} \leq 800$	°F
INITIAL ABLATOR TEMP	$-100 \leq T_e \leq 100$	°F

FIGURE 3.2.9-1

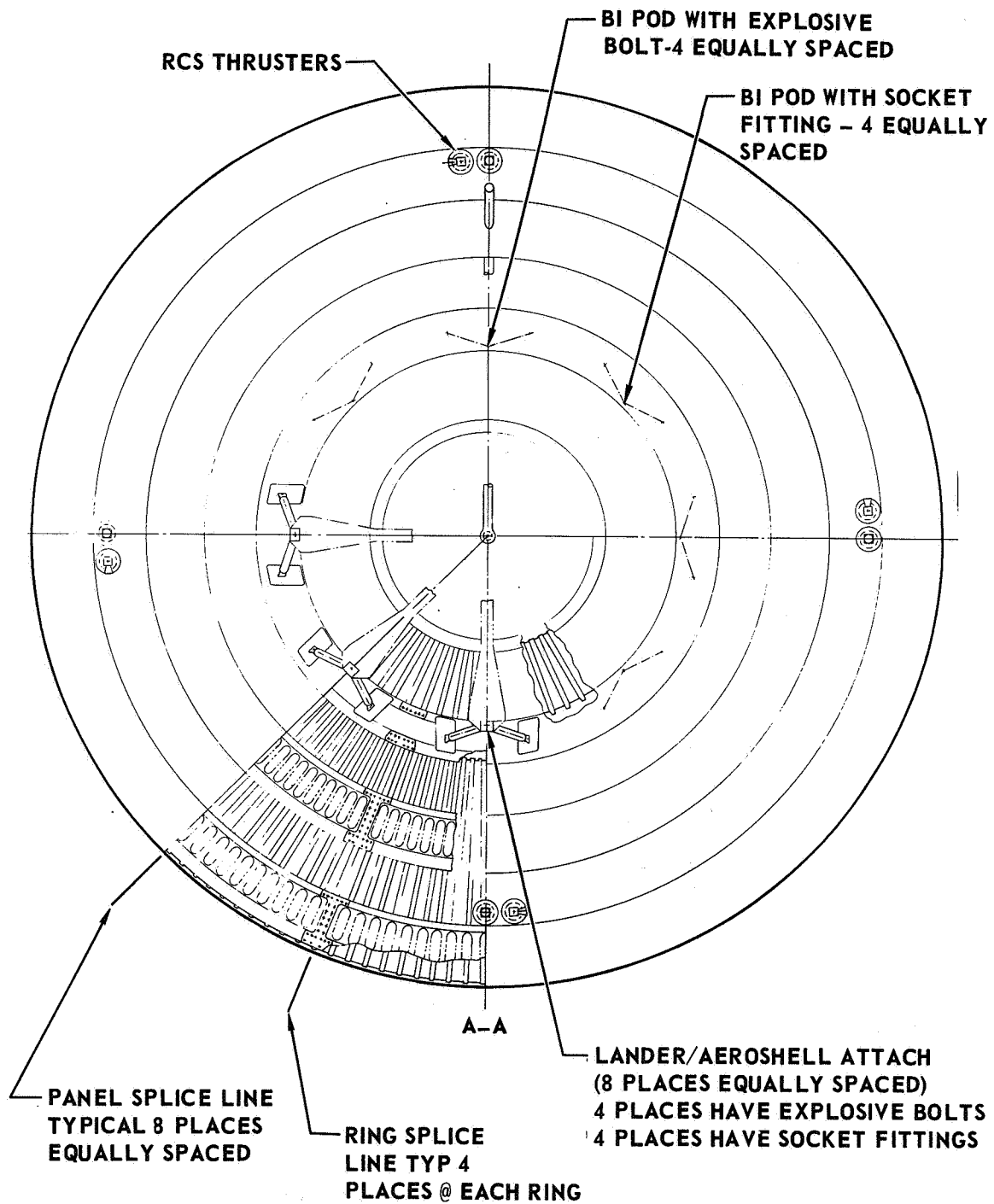
AEROSHELL STRUCTURAL ARRANGEMENT



3.2.9-5

FIGURE 3.2.9-2
SHEET 1 OF 2

AEROSHELL STRUCTURAL ARRANGEMENT



longitudinal beam supported by the rings. Loads normal to the shell surface are therefore beamed longitudinally to the rings.

The rings provide stability for collapsing pressure and redistribute lateral loads from unsymmetrical pressure into the overall shell. They are aluminum triangular torque boxes composed of three caps and two beaded webs, with the shell providing the third web.

Aluminum bipod trusses attach to the lander at eight points and to the aeroshell payload ring at sixteen equally spaced points. The meridional component of the bipod loads at the 16 attach points is distributed into the shell by a splice plate adjacent to the payload ring. The radial component is distributed into the shell by the payload ring.

Fittings and brackets at the aft ring provide support for the reaction control jets, and a truss attached to the ring at the sphere-cone tangency provides support for reaction control system equipment.

The aeroshell structure is completed when the nose cap is attached to the conical shell at the cone-sphere tangency by a radial bolt pattern. For the nose cap in this study, we have used a reinforced plastic sandwich consisting of heat resistant phenolic (HRP) honeycomb core and phenolic fiberglass face sheets as shown in Figure 3.2.9-3. It is assembled with a modified epoxy film adhesive (HT-435). The minimum sandwich section used has a core that is 0.26 inch thick and face sheets that are each 0.020 inch thick. It is designed to be laid up, cured, and bonded directly to the inside surface of the prefabricated heat shield, in one assembly operation.

3.2.9.4 Entry heat protection. - The following is a summary of the baseline heat protection system used in this study:

- a) The spherical nose cap is covered with hardened compacted fibers (HCF) of aluminosilicate (density = 20 to 25 lb/ft³). This material is non-ablative and will not contaminate the entry science ports located in a beryllium plug at the apex of the nose cap.
- b) The conical aeroshell surface is covered with a low density foamed silicone ablator, McDonnell S-20T (density = 18 to 20 lb/ft³).

NOSE SECTION ASSEMBLY

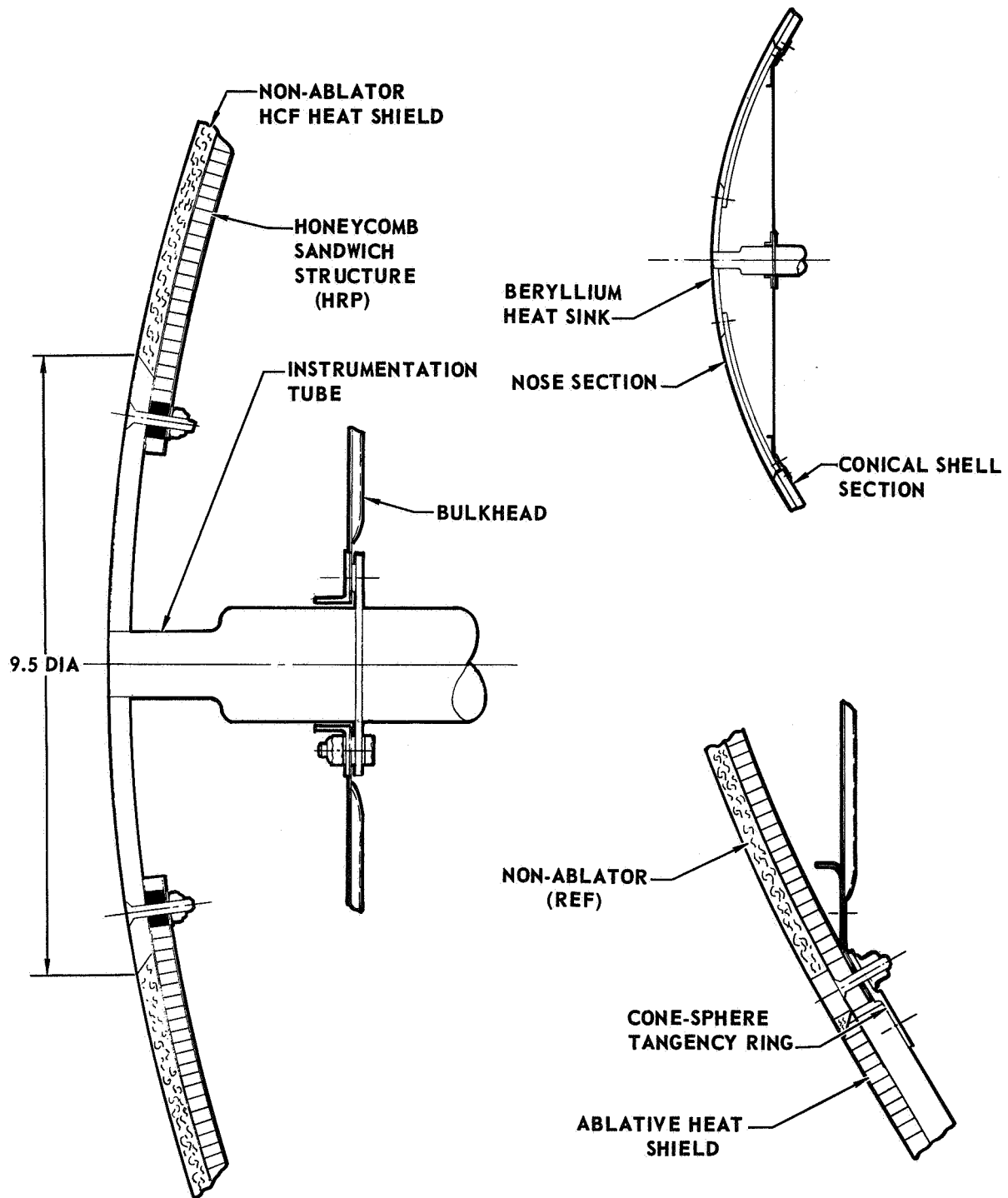


FIGURE 3.2.9-3

- c) The HCF and S-20T are attached to the aeroshell structure with HT-424 high temperature adhesive, using techniques similar to those developed for the Gemini heat shields.
- d) A silica cloth thermal curtain over the aeroshell base area aids heat retention during orbital descent, prevents overheating of the internal equipment during entry, and prevents heating and contamination from the deorbit motor exhaust products.

These components are shown in Figure 3.2.9-4.

3.2.9.5 Parametric Studies. - Weight of the aeroshell structure was determined as a function of applied external loads, size, and entry weight. For the maximum loading condition discussed in Section 3.2.9.2, external loads (dynamic pressure) are proportional to the entry vehicle ballistic parameter, as shown in Figure 3.1.3-4.

Total aeroshell structural weight is primarily attributable to the shell, rings, and lander support structure in the conical section and the fiberglass sandwich structure in the nose cap. The combined weight of these items is shown in Figure 3.2.9-5. These weights are influenced primarily by aeroshell size, because of minimum gage considerations and manufacturing limitations. For a given diameter aeroshell, increases in structural weight with increasing dynamic pressure are due primarily to the rings. Lander support structure weight, although influenced by dynamic pressure, entry weight, and diameter, is a small part of total aeroshell structural weight.

Entry heat protection system weight was determined for several combinations of system and mission variables, using the selected heat protection materials and defined heating environments. The method used to define the weight of local heat protection has been reported by Mezines (S. A. Mezines, "A Semi-Empirical Method for Correlating the Thermal Performance of Charring Ablative Materials," McDonnell Company, AIAA Paper No. 68-757, presented at 3rd Thermophysics Specialist Conference, June 24, 1968). This method relates the required ablative unit weight for a specific initial temperature and maximum bondline temperature to the total heat and heating time as follows:

AEROSHELL ENTRY HEAT PROTECTION

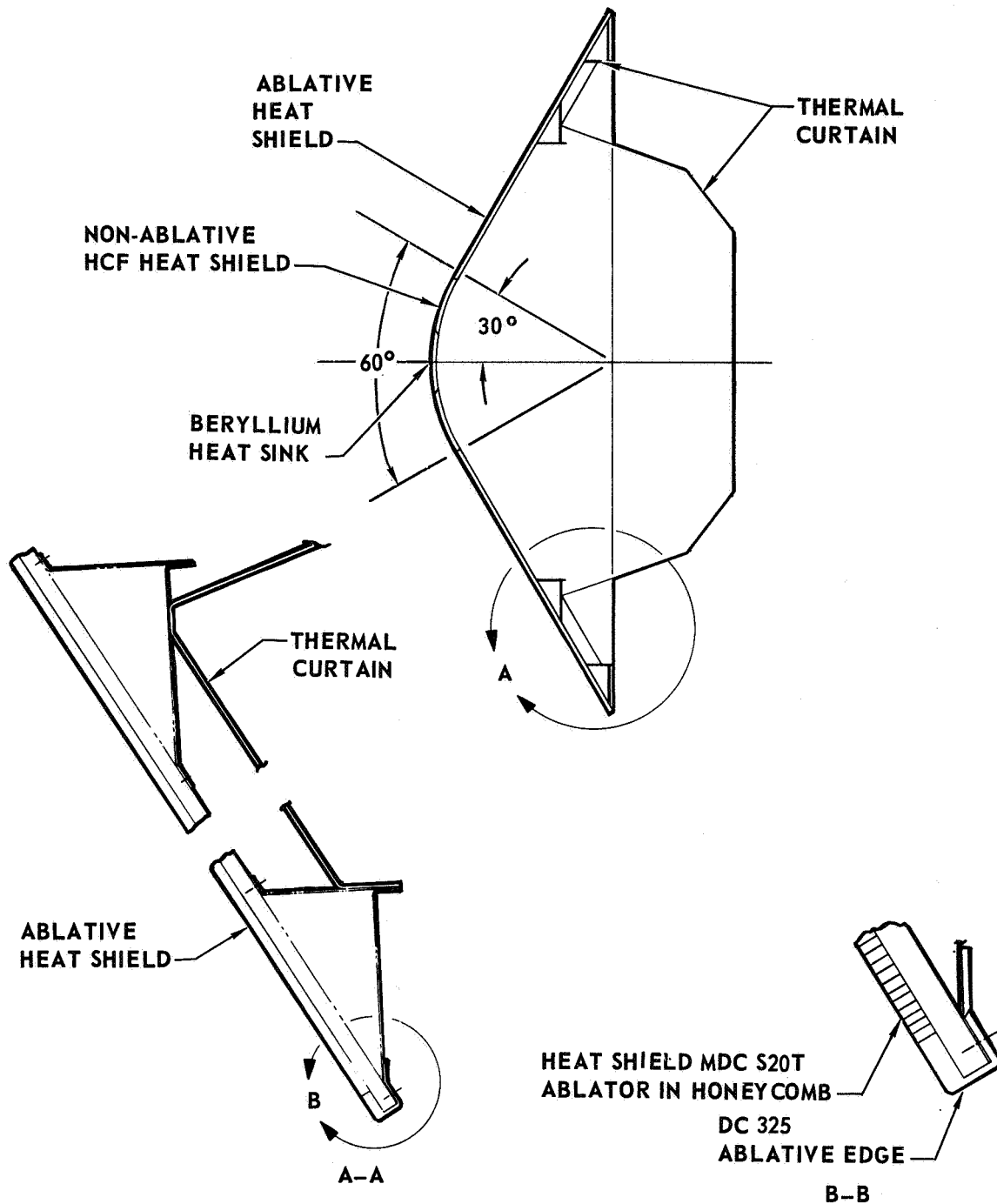


FIGURE 3.2.9-4

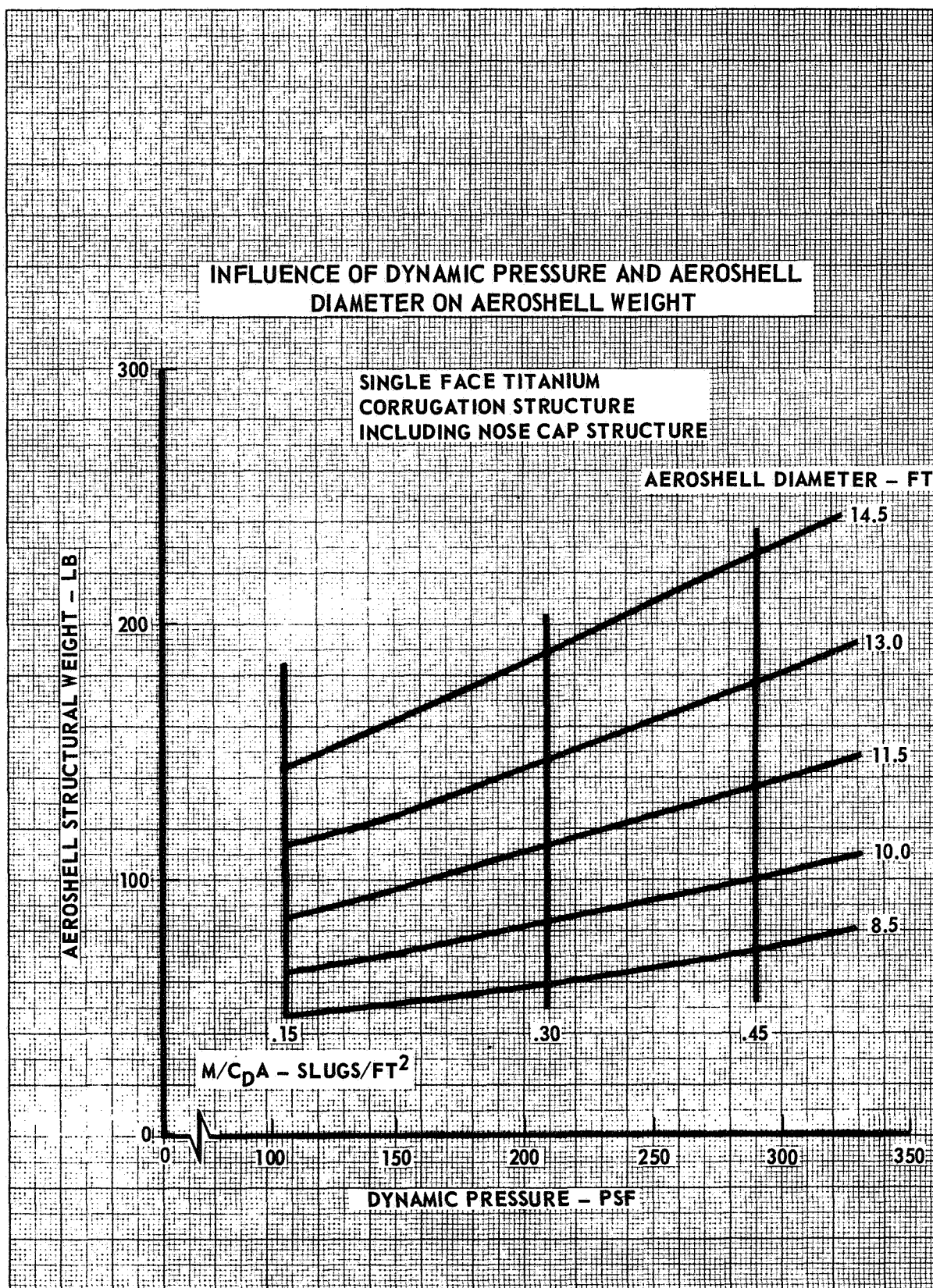


FIGURE 3.2.9-5

$$\frac{W}{A} = f \left[K (Q)^{1/8} (\theta)^{3/8} \right]$$

where

W/A = Unit weight of entry heat protection - lb/ft^2

Q = Local total heat - Btu/ft^2

θ = Duration of heat pulse, sec. Defined as the time that the equilibrium surface temperature of the ablator exceeds the specified limiting bondline temperature.

K = Relationship obtained from arc-plasma test data correlations for specific materials

The powers for total heat and heating time were derived from the theory for composite flat plates. The method has been shown to be satisfactory for non-ablative insulators and several ablative materials, including the McDonnell S-20T low density silicone elastomer. It has been verified by comparing results with those obtained from computerized ablation analyses. The selected materials do not recede when exposed to the Martian heating environment, which is consistent with the assumptions used in the development of the empirical method.

The unit weight for the S-20T ablator was computed at the sphere-cone tangent point, assuming stagnation point heating conditions (angle of attack = 30°), and at the outer edge of the cone assuming the heating equaled 40% of the stagnation conditions. Design convective and radiative heating profiles and total heats are shown in Figure 3.2.9-6 for a nose radius of one foot. These profiles are for the thermal design condition of Point A of Figure 3.2.9-1, in a VM-9 atmosphere. The S-20T weights were computed with a 0°F initial temperature at the beginning of entry (800,000 ft). The predicted bondline temperatures were limited to a maximum of 640°F (maximum design temperature of 800°F). Unit weights for the cone were area-averaged and multiplied by the conical surface area to obtain total ablator weight using a density of 20 lb/ft^3 . The computed ablator weight was increased 10% to allow for heating uncertainties associated with non-equilibrium radiation and likely boundary layer turbulence during the terminal portion of the entry trajectories. Ablator is attached to the structure with HT-424 adhesive having a unit weight of 0.089 lb/ft^2 .

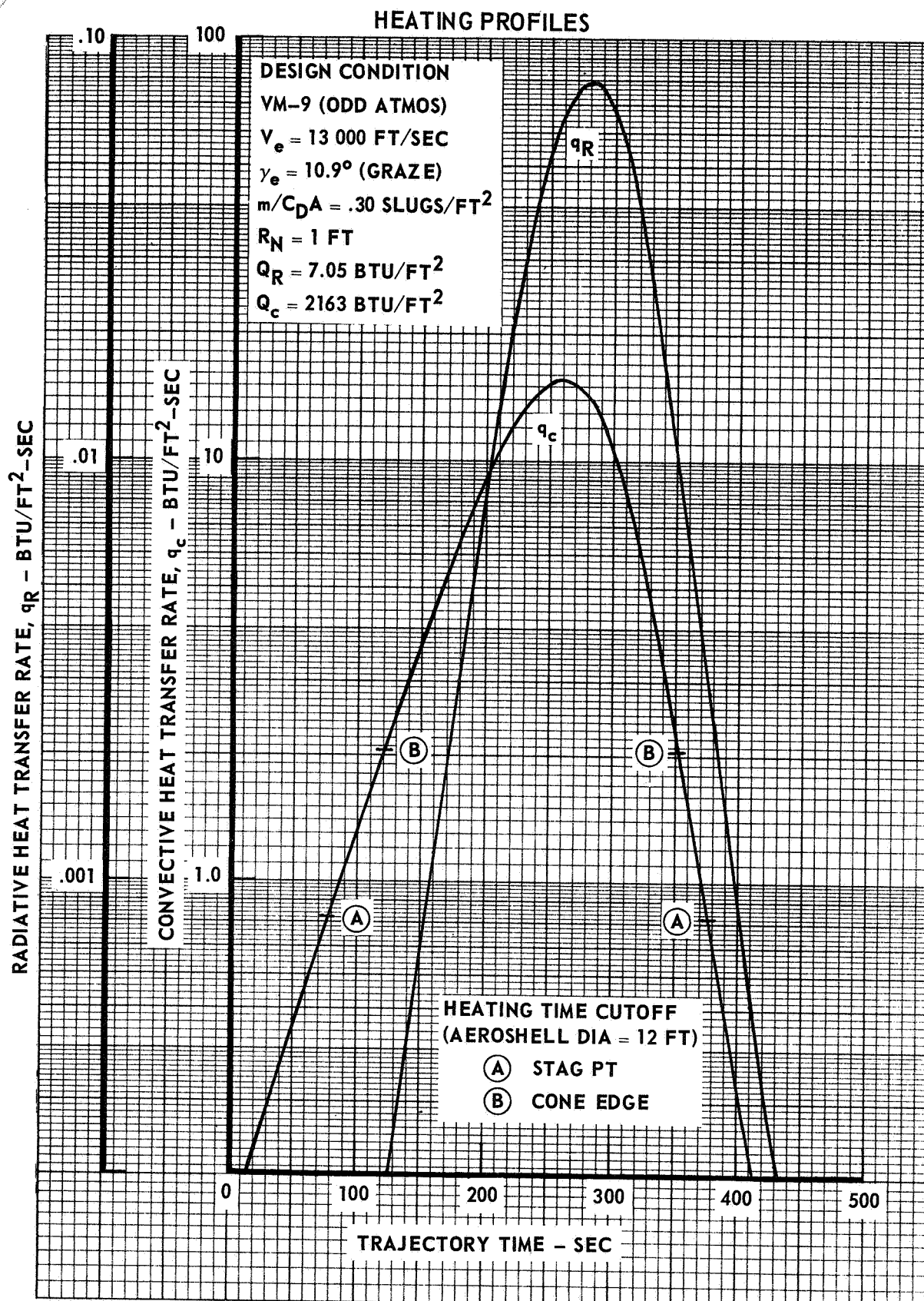


FIGURE 3.2.9-6

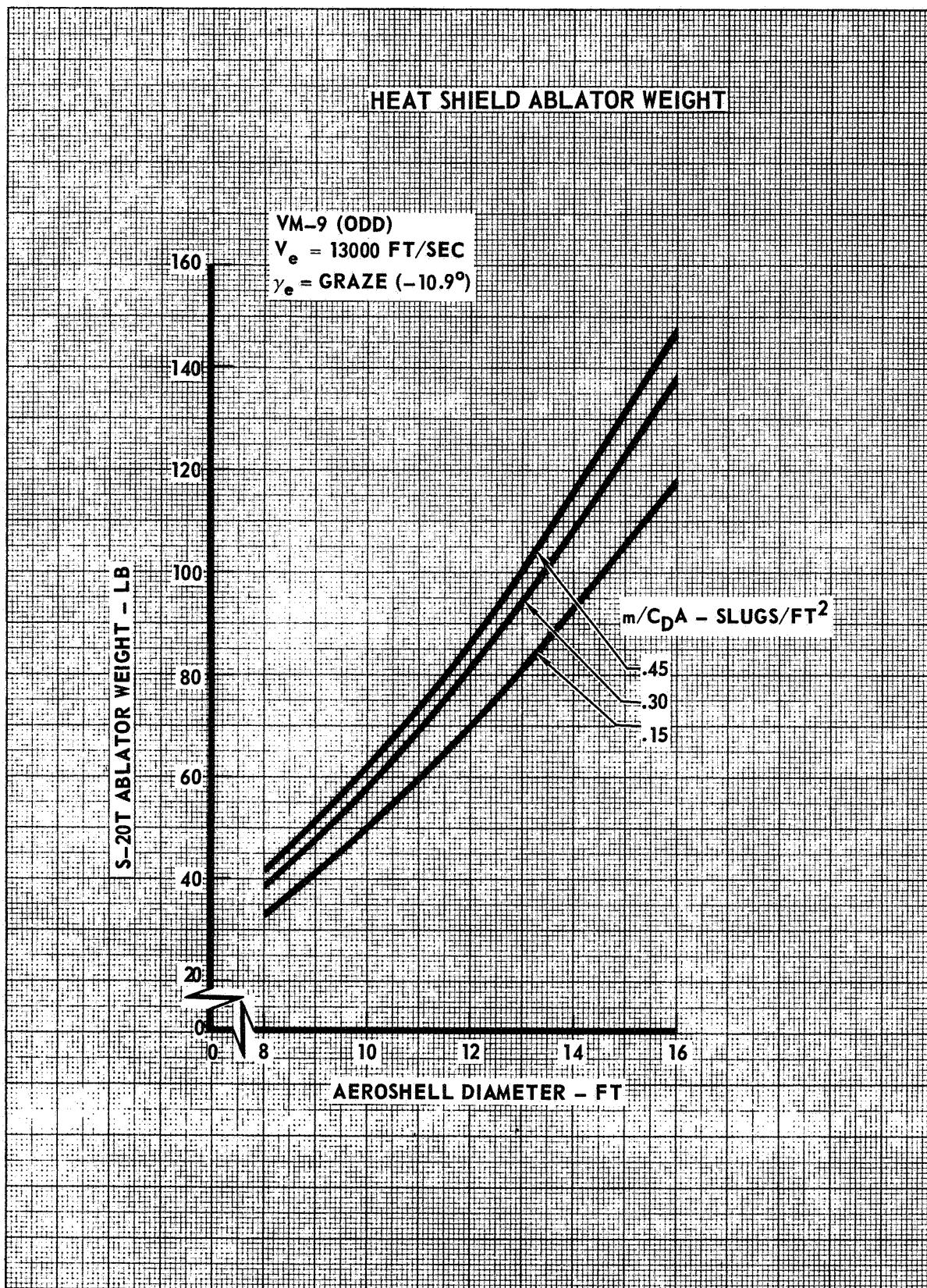


FIGURE 3.2.9-7

Figure 3.2.9-7 presents the ablator weight as a function of the aeroshell diameter and ballistic coefficient for the maximum heating condition (Point A of Figure 3.2.9-1). Aeroshell diameter is the dominant influence on heat protection weight. Nominal changes in $m/C_D A$ also produce modest changes in weight.

Heat protection for the nose cap was computed using the method described earlier. Stagnation point heating conditions were used for the entire cap surface. The thermal performance of HCF was assumed to be equivalent to the S-20T ablative performance, based on available test results. Weights were based on a HCF density of 25 lb/ft³. A beryllium plug, located at the center of the nose cap surface, contains ports for entry science atmospheric sample collection. The plug weight was computed considering the total heat storage and limiting the maximum beryllium temperatures to less than 500°F to minimize degradation of the entry science measurements. The HCF is attached to the structure with HT-424 adhesives.

The nose cap heat shield weight required for the maximum heating condition versus aeroshell diameter and $m/C_D A$ is shown in Figure 3.2.9-8. These weights include the HCF, and beryllium plug. The HCF weights were computed with a 0°F initial temperature at entry (800,000 ft. altitude). The predicted bondline temperatures were limited to a maximum of 640°F (maximum design temperature of 800°F). Figure 3.2.9-9 illustrates the fraction of weight reduction that is possible for either the HCF material or the S-20T ablator when allowable bondline temperatures are changed.

In a similar manner, heat protection weight can be reduced if aeroshell temperatures lower than -150°F can be accepted during cruise and orbital descent. Lower aeroshell temperatures interface with the structure, ablator materials, cruise thermal control, and other aeroshell systems. Establishing a lower limit on aeroshell temperature requires additional study.

The weights of the individual components of the entry heat protection system were combined to obtain a total system weight, as shown in Figure 3.2.9-10 for a combination of aeroshell diameters and $m/C_D A$. These weights include weight of heat protection for nose cap and conical shell, adhesives, and the aft thermal curtain. The aft thermal curtain, made of silica cloth,

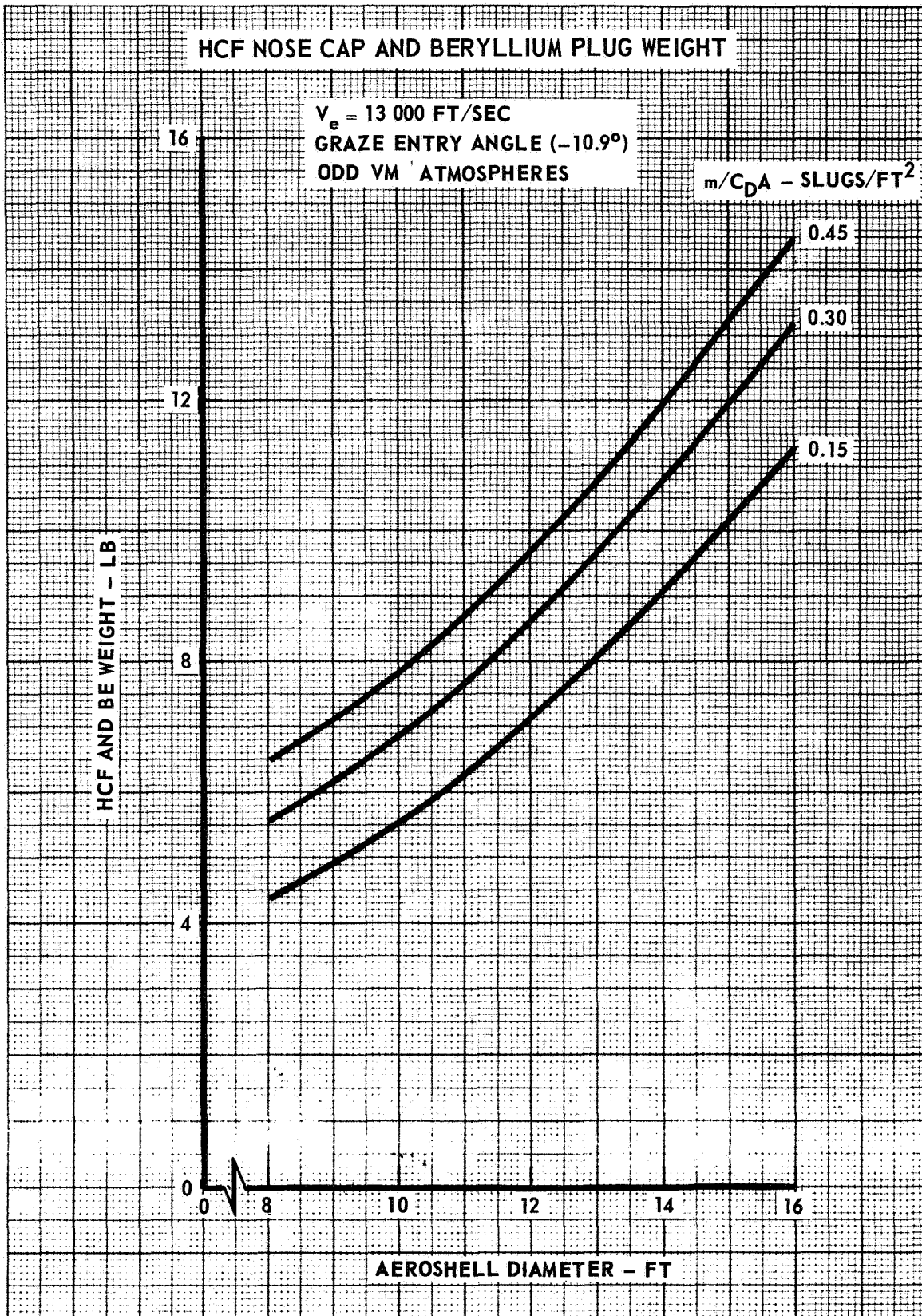


FIGURE 3.2.9-8

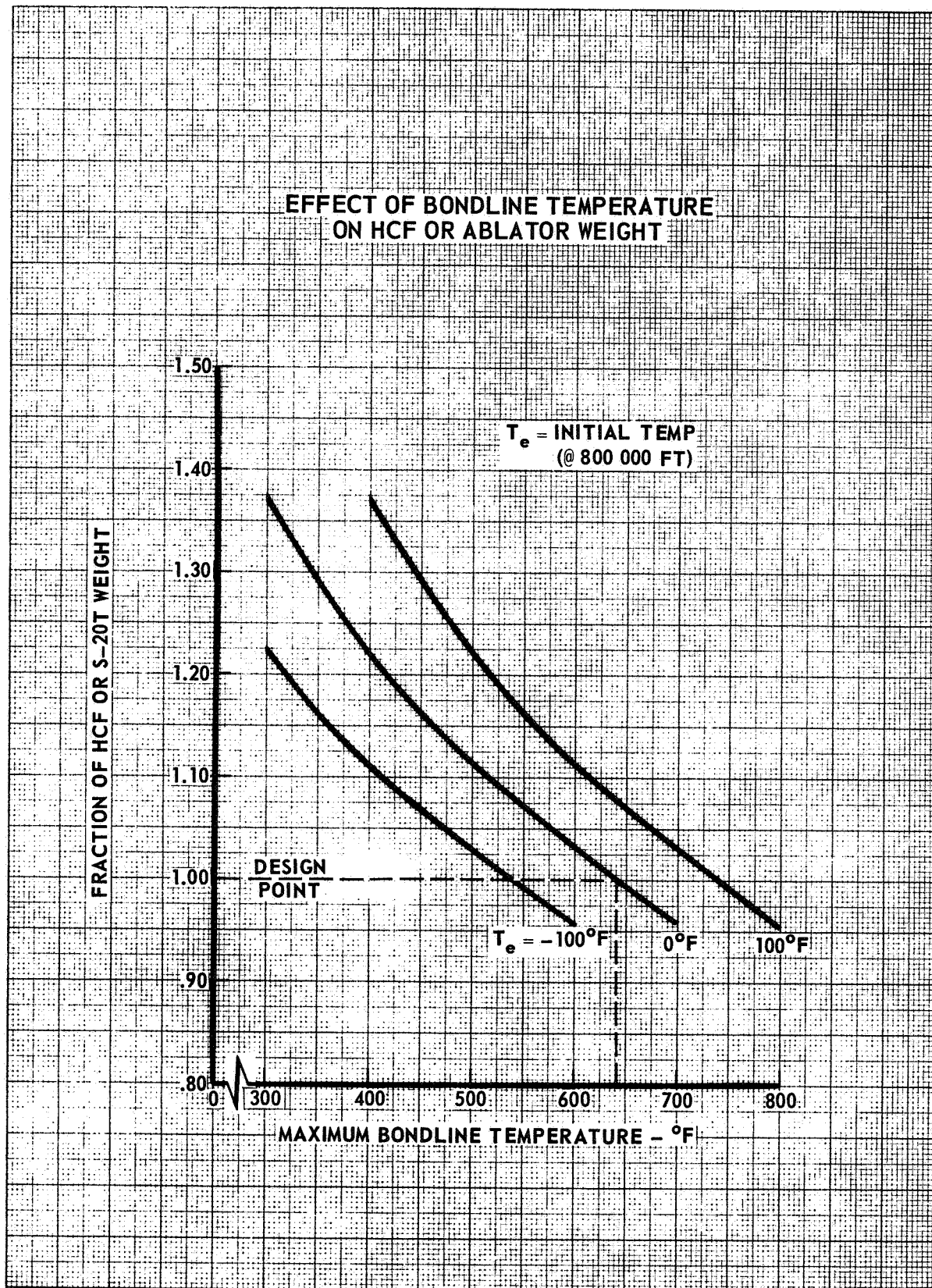


FIGURE 3.2.9-9

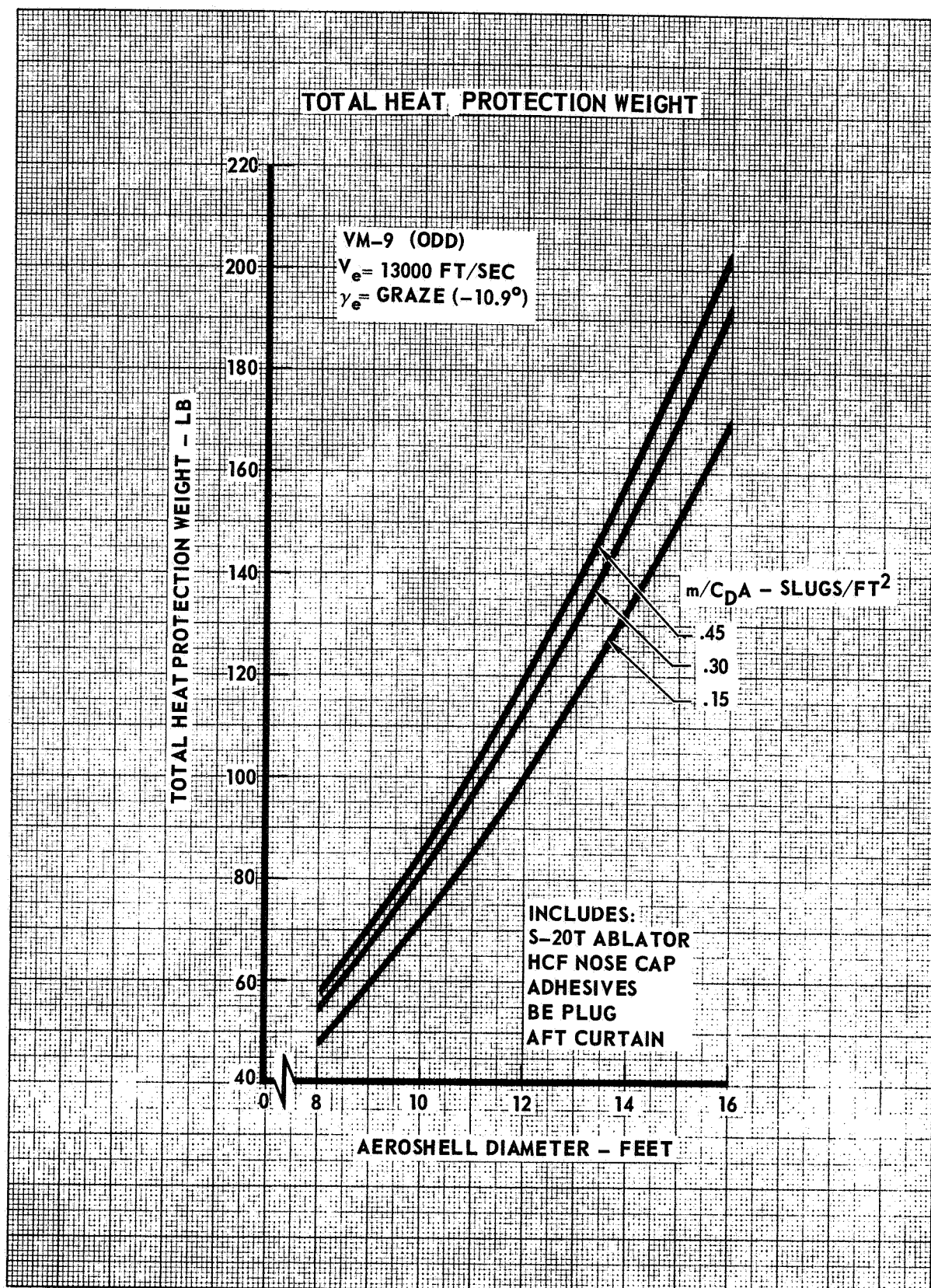


FIGURE 3.2.9-10

has a unit weight estimated to be 0.1 lb/ft^2 , including seams and attachment devices. This estimate is based on the weight of the Gemini adapter thermal curtain.

Finally, the total aeroshell weight - structure plus heat protection system is shown in Figure 3.2.9-11.

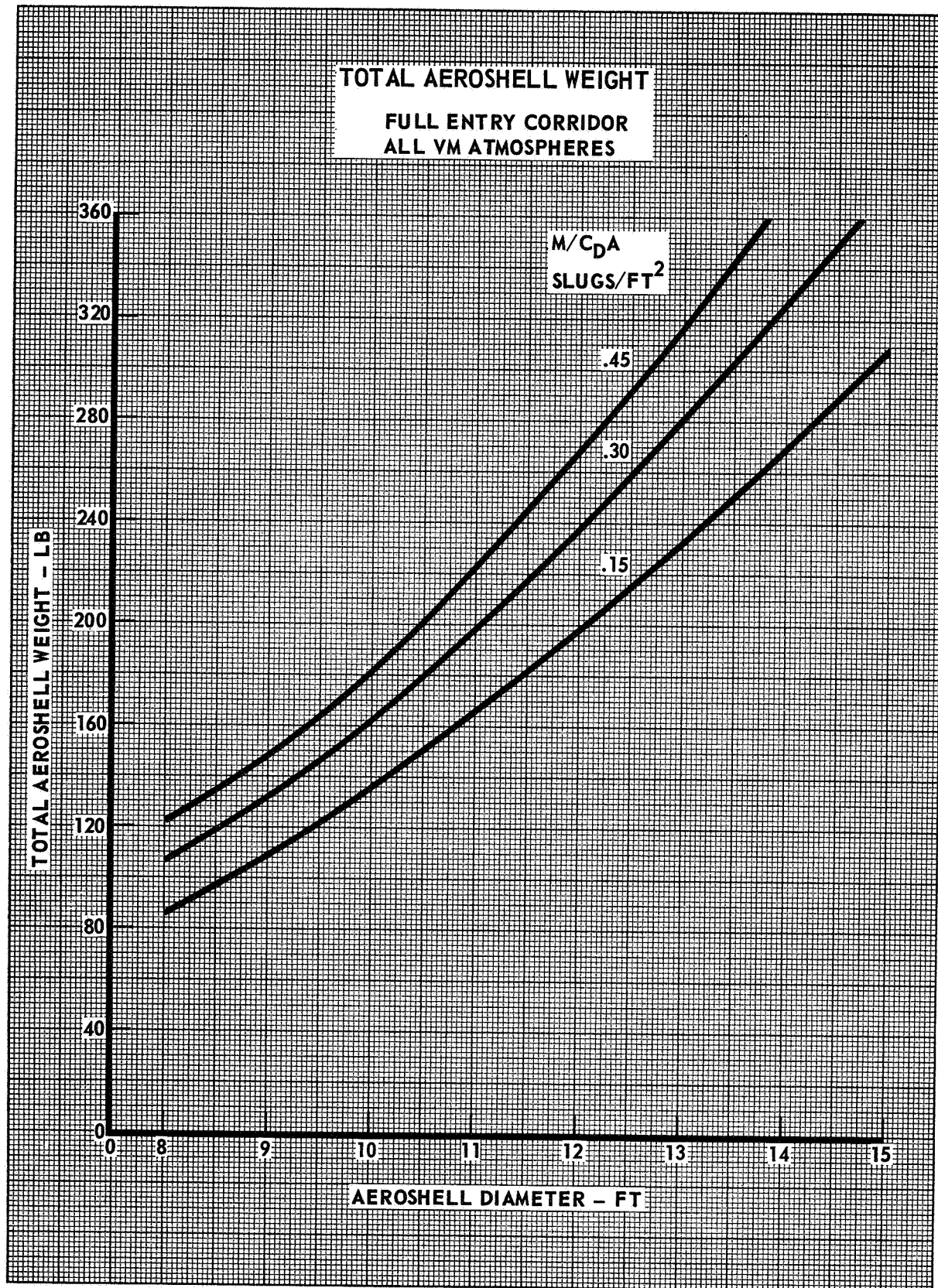


FIGURE 3.2.9-11

3.2.10 LANDING - Safe delivery of the surface payload requires coordinated design of the impact attenuation system, the terminal descent propulsion system, the landing radar altimeter, and the other support systems required to control descent and landing. This section describes the characteristics of the Uni-Disc landing system.

3.2.10.1 Requirements. - The requirements established for this study are as follows:

- a) The lander must remain upright during and after impact.
- b) Landings will be limited to surface slopes of 34° maximum.
- c) The combination of $V_V = 16$ ft/sec and $V_H = 10$ ft/sec for stability, and $V_V = 20$ ft/sec and $V_H = 5$ ft/sec for structural strength will be used throughout. (Determined most critical in VOYAGER Phase B study, NASA CR-89673.)
- d) "G" forces felt by the payload should not exceed 25g.
- e) No attenuation is contributed by the impact surface.
- f) The total landed weights investigated vary from 750 to 1500 pounds.

3.2.10.2 Uni-Disc selection. - Many landing system concepts were investigated prior to and during the Phase B VOYAGER study, including limited directional and omnidirectional landers, mechanical and pneumatic landing systems, and toroidal and spherical shapes. Various legged landers were studied, such as a pendulum payload concept, and others similar to the Lunar Module and Surveyor, as well as flat disc shaped landers. Of all those studied, only the Uni-Disc met the total requirement; therefore, it was selected as the landing system for this study.

This configuration is basically a large disc having a low landing silhouette, which produces a low center of gravity and excellent stability, and which attenuates the lander energy remaining after terminal propulsion cut-off. It consists of the following components (Figure 3.2.10-1):

- a) A base platform, which performs the dual function of 1) supporting the payload and propulsion equipment and 2) providing the backup and carry-through structure for the impact attenuator and tension cable assemblies.

LANDING SYSTEM

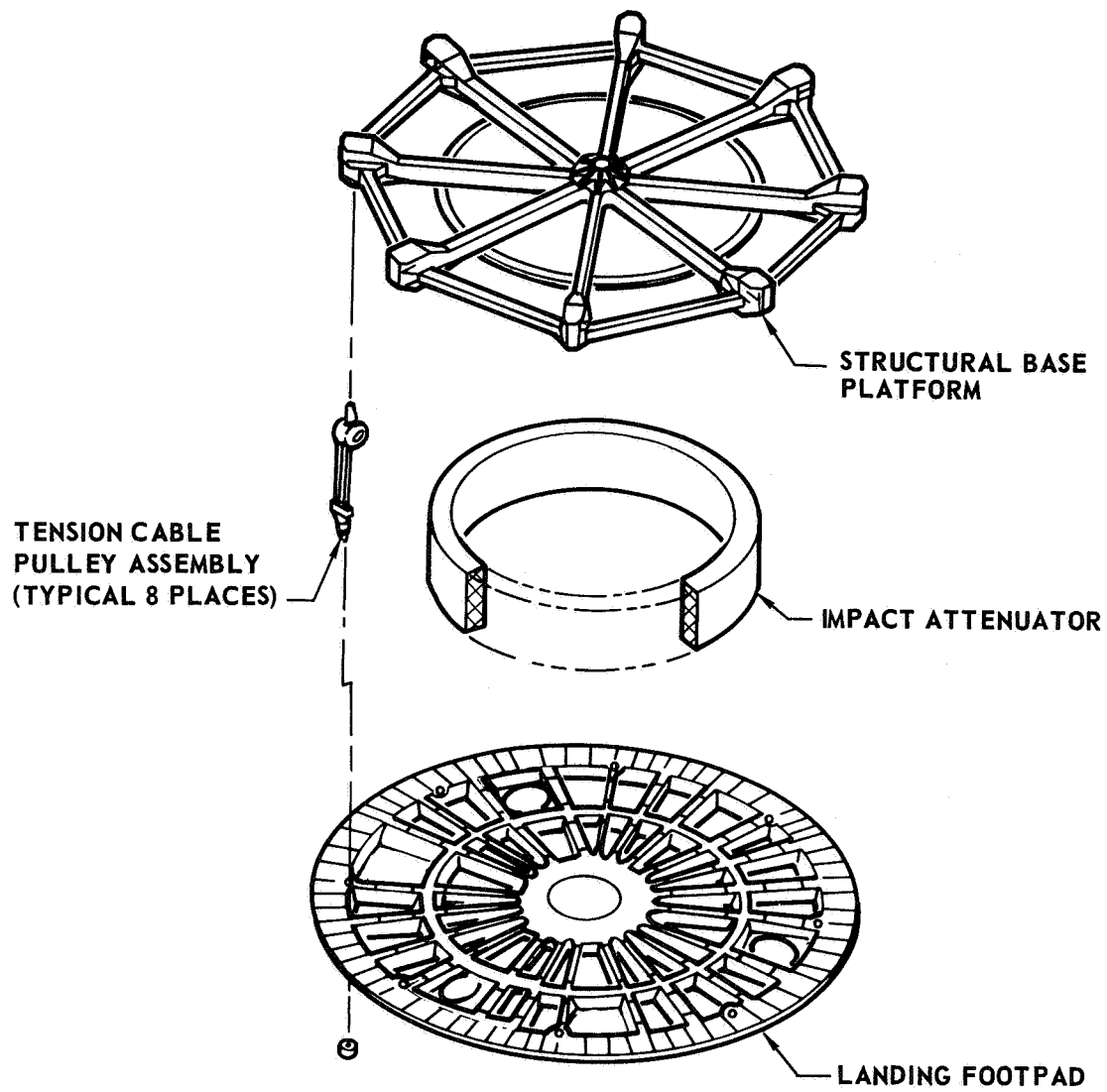


FIGURE 3.2.10-1

- b) An impact attenuator ring, which is a cylindrical ring of aluminum cellular crushable material such as "Trussgrid."
- c) Eight tension cable assemblies.
- d) A landing footpad.

The base platform reacts loads introduced by the attenuator ring and the tension cables by bending of the radial I-beams, and provides support for the surface payload, terminal descent propulsion system, and other support subsystems required to control descent and landing.

3.2.10.3 Landing system operation. - At a point 10 feet above the surface of Mars, terminal descent engines are cut off, allowing the lander to free fall from that height. Because of the $\pm 5^\circ$ variation in trajectory angle and a $\pm 5^\circ$ attitude error due to uneven engine tail off, the lander may be rotated as much as 10° from horizontal on impact. This 10° of lander tilt, when coupled with landing on a 34° slope, imposes an equipment installation clearance line of 44° , as shown in Figure 3.2.10-2.

In the case of a straight flat landing at zero horizontal velocity on level surface, the footpad distributes the loads uniformly to the crushable attenuator. The attenuator absorbs sufficient energy to reduce the loads felt by the surface payload and associated equipment during landing to the desired maximum g level. For this landing condition, the eight tension cable assemblies carry no loads, but merely roll up the slack cable created by the crushing attenuator. The landing radar antenna, which is mounted to the bottom center of the footpad, the frangible terminal descent engine nozzles, and the radar altimeter antenna are crushed during landing.

For the case of an off-center landing on a slope ridge, or peak, or a landing with horizontal velocity, the landing system operates as follows:

The off-center vertical load applied to the footpad is reacted by the tension cables and the attenuator. If the structurally stiff footpad was not restrained by the tension cable assemblies, it would begin to rotate about the attenuator nearest the impact point. This rotation of the footpad would put tension loads on the bond between the attenuator ring and the footpad on the side opposite the impact point, resulting in separation of attenuator and

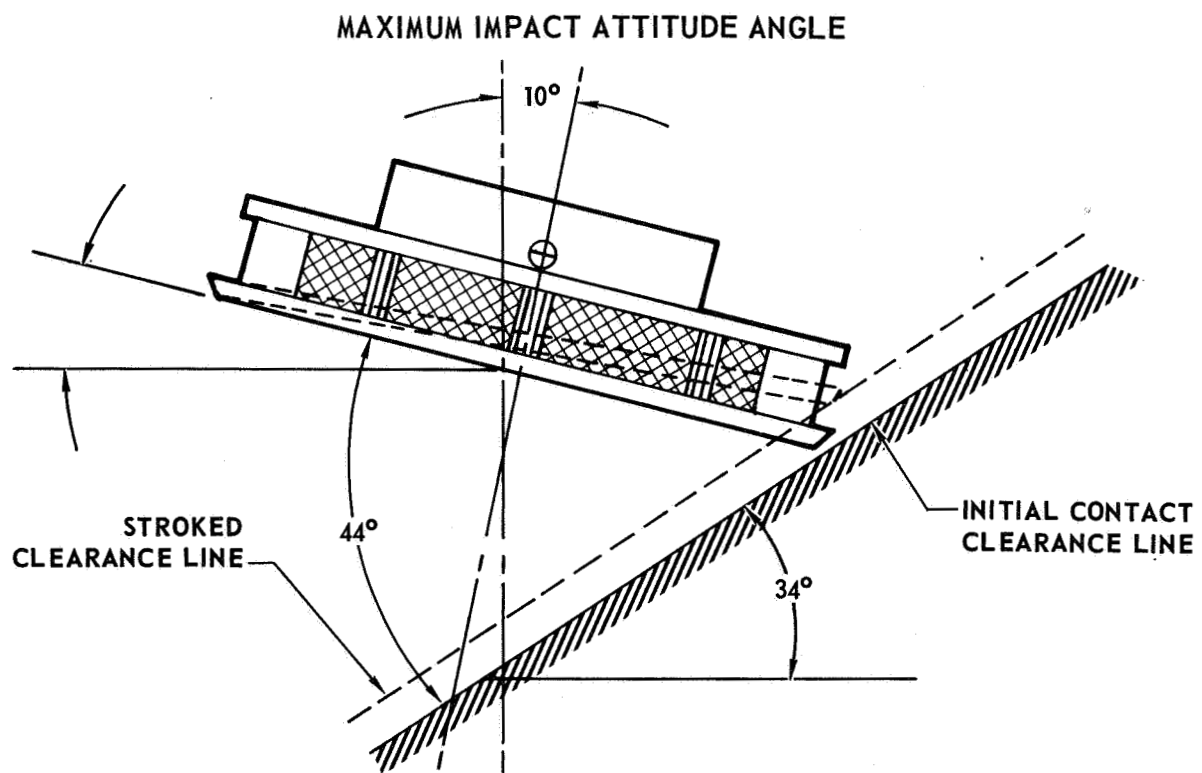


FIGURE 3.2.10-2

footpad, with only local crushing of the attenuator ring. Use of the tension cables prevents this separation and forces the crushable ring to crush over its entire area. The one-way ratchetted cable pulleys keep the footpad tight against the attenuator ring and **keeps** the system operational during bouncing or sliding movements.

3.2.10.4 Performance. - Maintaining stability during touchdown is the most demanding requirement on the landing system. It is assured by limiting the ratio of center-of-gravity weight to base diameter (H/R). The maximum allowable value for this ratio is approximately .6 for successful landing on a 34° slope with the worst approach velocity condition. As the surface slope is decreased, the allowable H/R is increased; see Figure 3.2.10-3. In calculating the limiting H/R values it is necessary to use inertia values of the landing vehicle; for this landing system, the inertias used were those of the capsules described in section 3.3. Since these inertia values do not scale with weight in a manner which provides exact dynamic similarity of different size landers, the H/R limits vary with landed weight. The data presented in Figure 3.2.10-3 is for a landed weight of 780 lb, typical of those which can support the minimum mission for the study and 1500 lb, typical of the largest capsule which satisfies the study constraints. The landing system weight required for these two landed weights is presented in Figure 3.2.10-4 as a function of the maximum slope to be encountered. The effect of landed weight on landing system weight is shown in Figure 3.2.10-5 for 34° and 20° slopes. A difference of about 2% of the landed weight for this 14° slope difference. Landing system weight varies from 20% of the landed weight for small landers to less than 14% of landed weights of 2000 lb. Diameter of the Uni-Disc footpad ranges from 5 to 7 feet. Landing load factors are generally less than 20 g's for the range of capsule sizes being considered in this soft lander study.

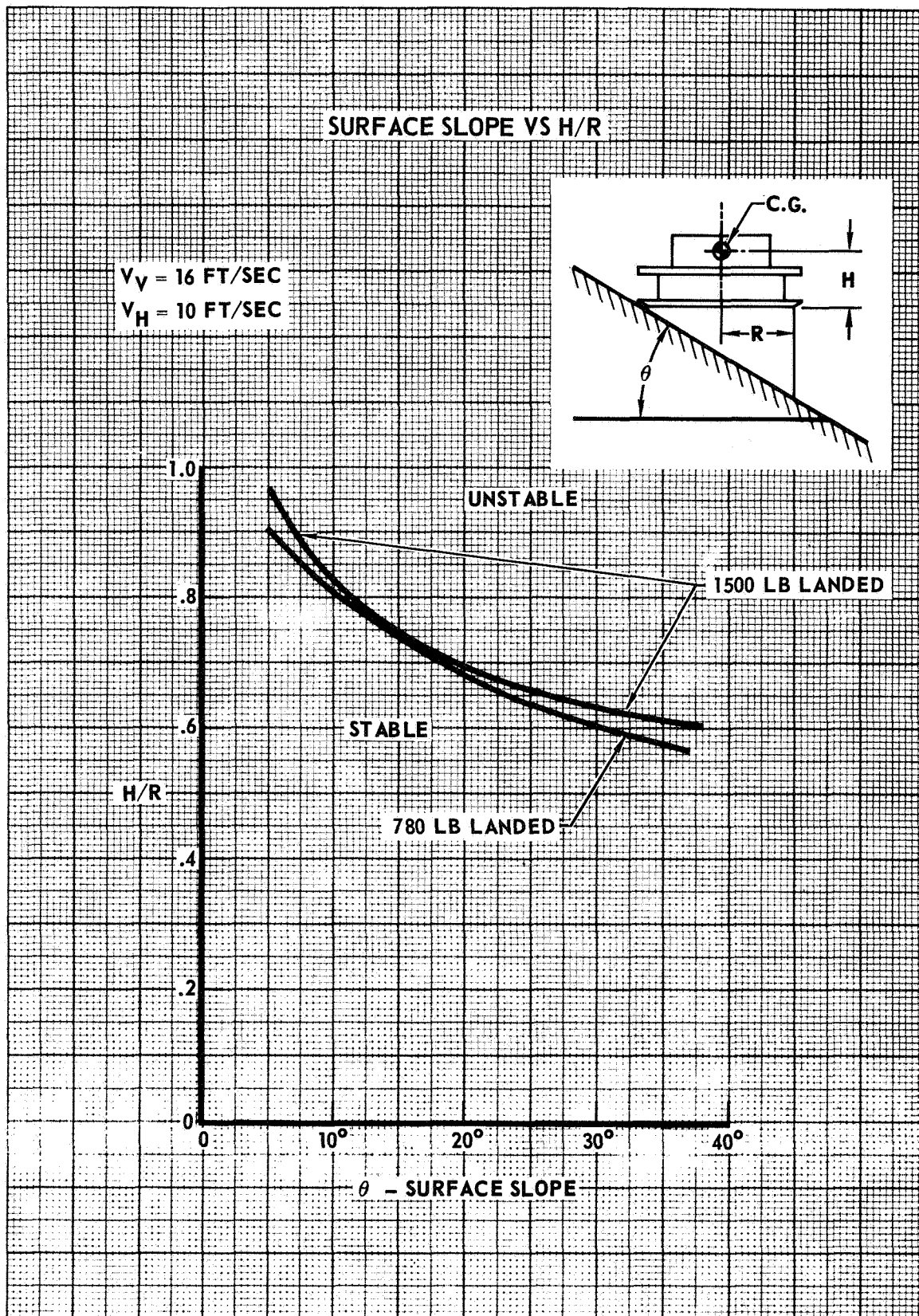


FIGURE 3.2. 10-3

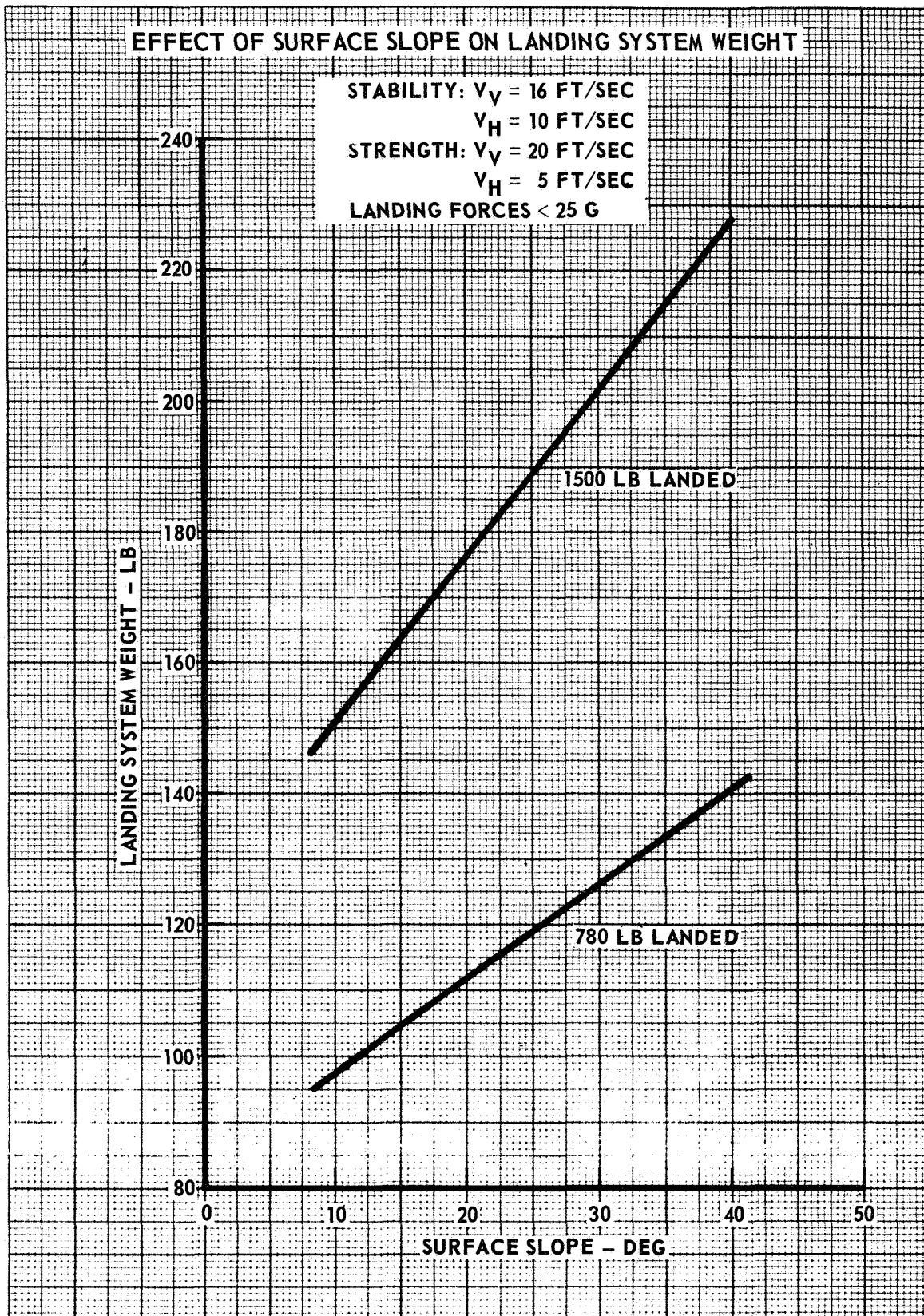


FIGURE 3.2.10-4

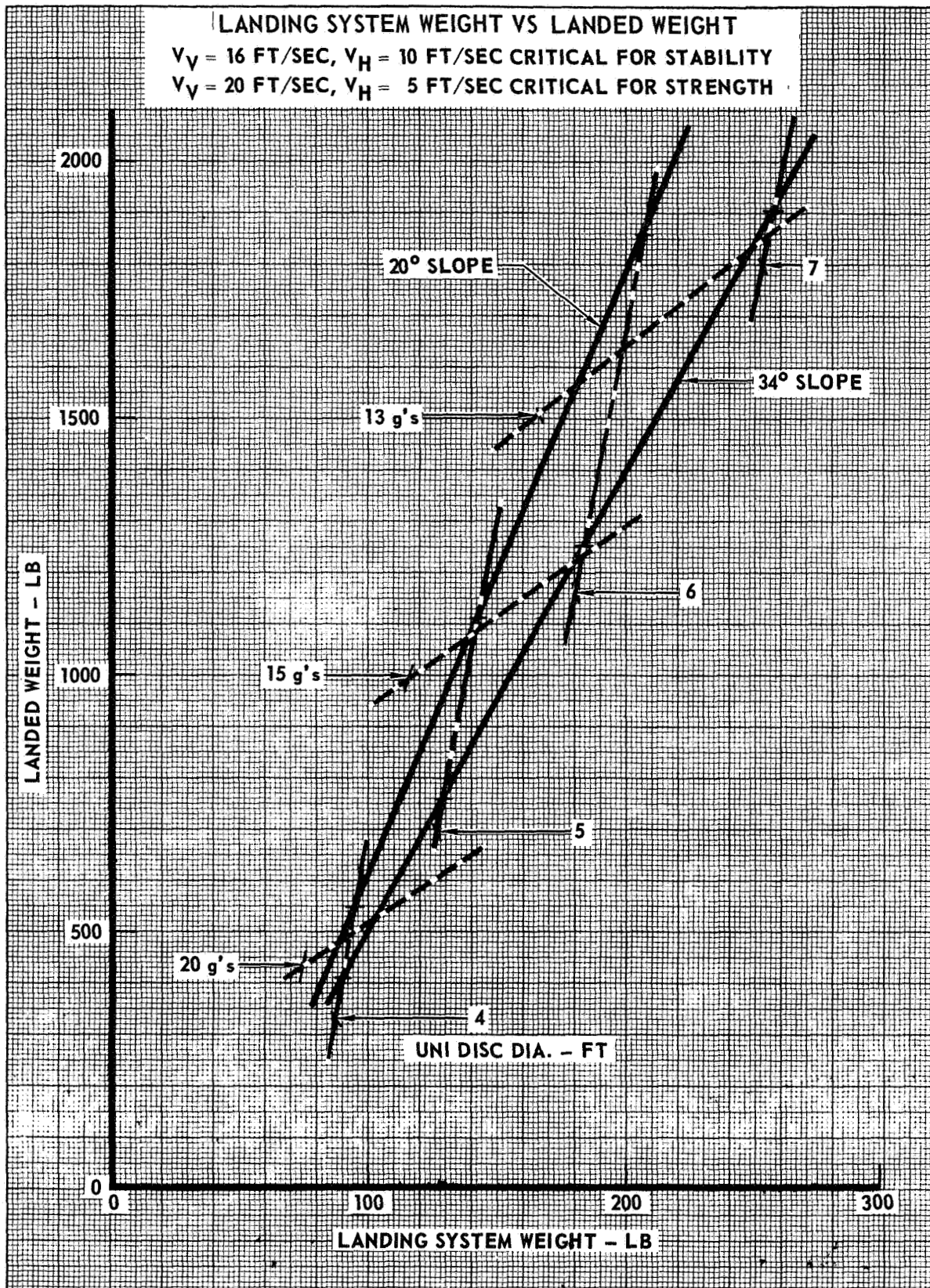


FIGURE 3.2.10-5

3.2.11 CANISTER AND ADAPTER - The function of the sterilization Canister is to maintain sterile conditions of the lander, capsule, and associated equipment from terminal sterilization until separation from the orbiter in Mars orbit. The adapter attaches the capsule to the canister interface cone which supports the flight capsule from the orbiter until separation. The conceptual designs of both the canister and adapter are based on the VOYAGER Phase B studies (NASA CR-89673). The present study shows the influence of flight capsule weight and aeroshell diameter on canister and adapter weight. Results of parametric studies of the canister and adapter are presented in Sections 3.2.11.1.3 and 3.2.11.2.3, respectively.

3.2.11.1 Canister. - The canister configuration, shown in Figure 3.2.11-1, consists of forward and aft aluminum semi-monocoque shell structures, mechanically joined at the field joint ring by a radial bolt pattern. Separation is accomplished at this ring by a dual confined explosive separation device (CESD-the device is redundant) which is sandwiched between the bolt flanges. When the device is fired, bolts fail in tension, causing the forward canister section to separate from the capsule. A pressurization and venting (P&V) system is installed in the canister to prevent excessive internal pressures.

3.2.11.1.1 Requirements: The canister must provide a microbiological barrier to prevent contamination of the enclosed capsule. Its diameter is constrained by the dynamic characteristics of the spacecraft and by the launch vehicle shroud envelope.

A pressurization and venting system maintains a positive pressure differential of at least .5 psi between the inside of the canister and the outside. This pressure differential is limited to 2.25 psi (the structure is designed for a 3.75 psi burst pressure). The P&V system must tolerate temperature build-up during on-pad operations without venting or requiring sterile make-up gas. The internal pressure is vented during launch and ascent, and reduced to zero upon reaching Earth orbit.

A separation system is required that completely severs the canister around its periphery while retaining the debris to prevent contaminated ejecta from impacting on Mars.

CANISTER STRUCTURAL ARRANGEMENT

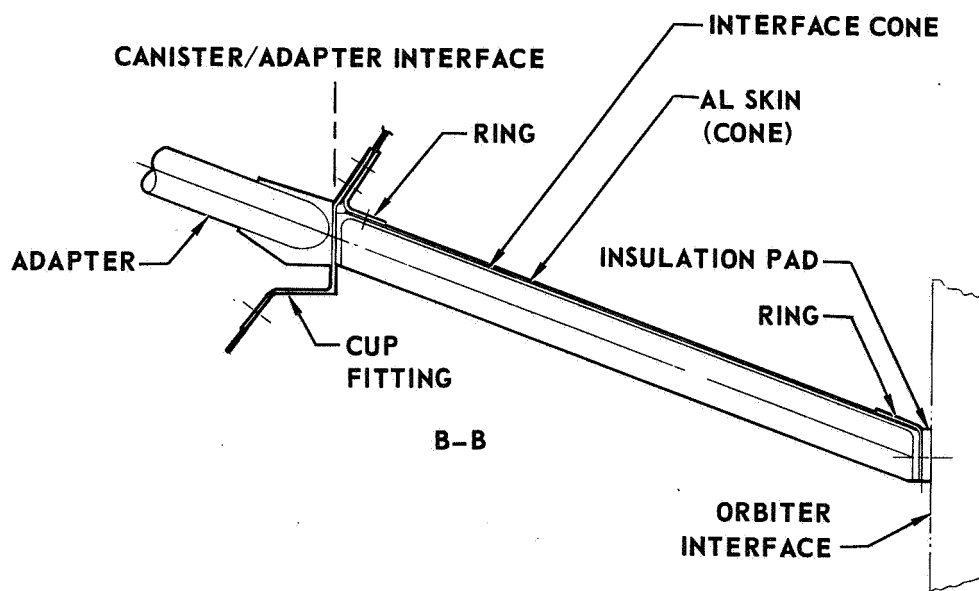
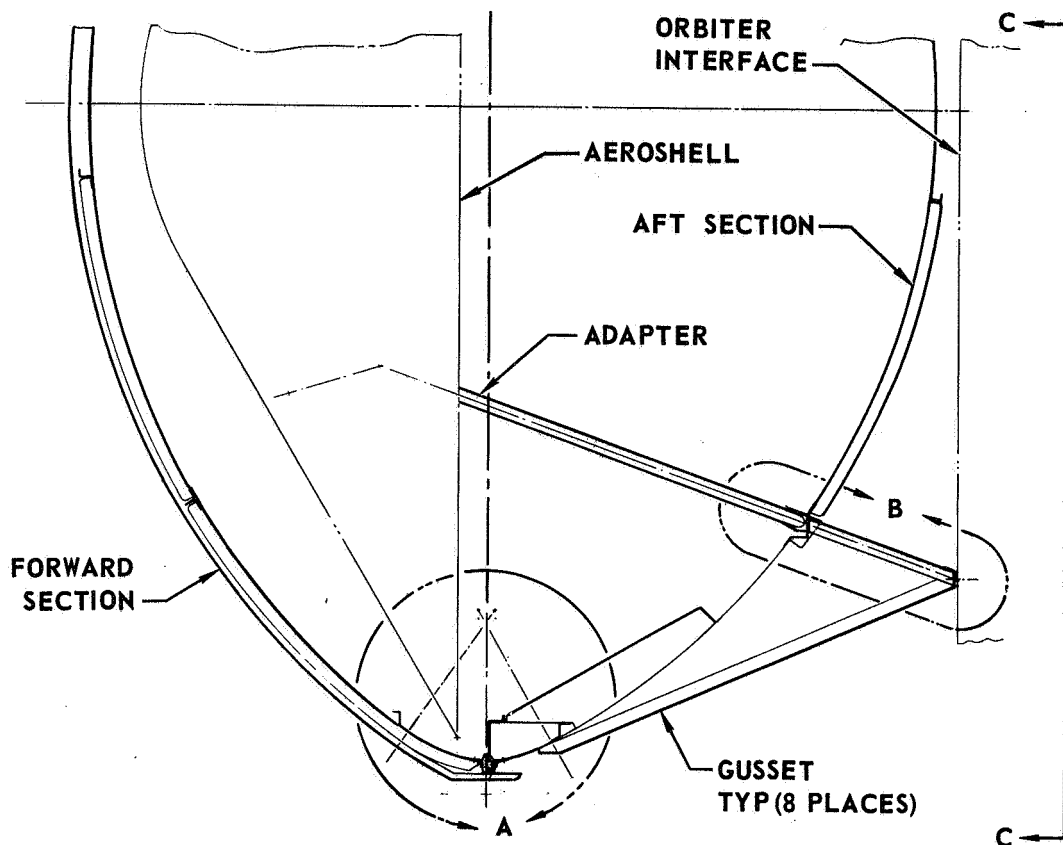
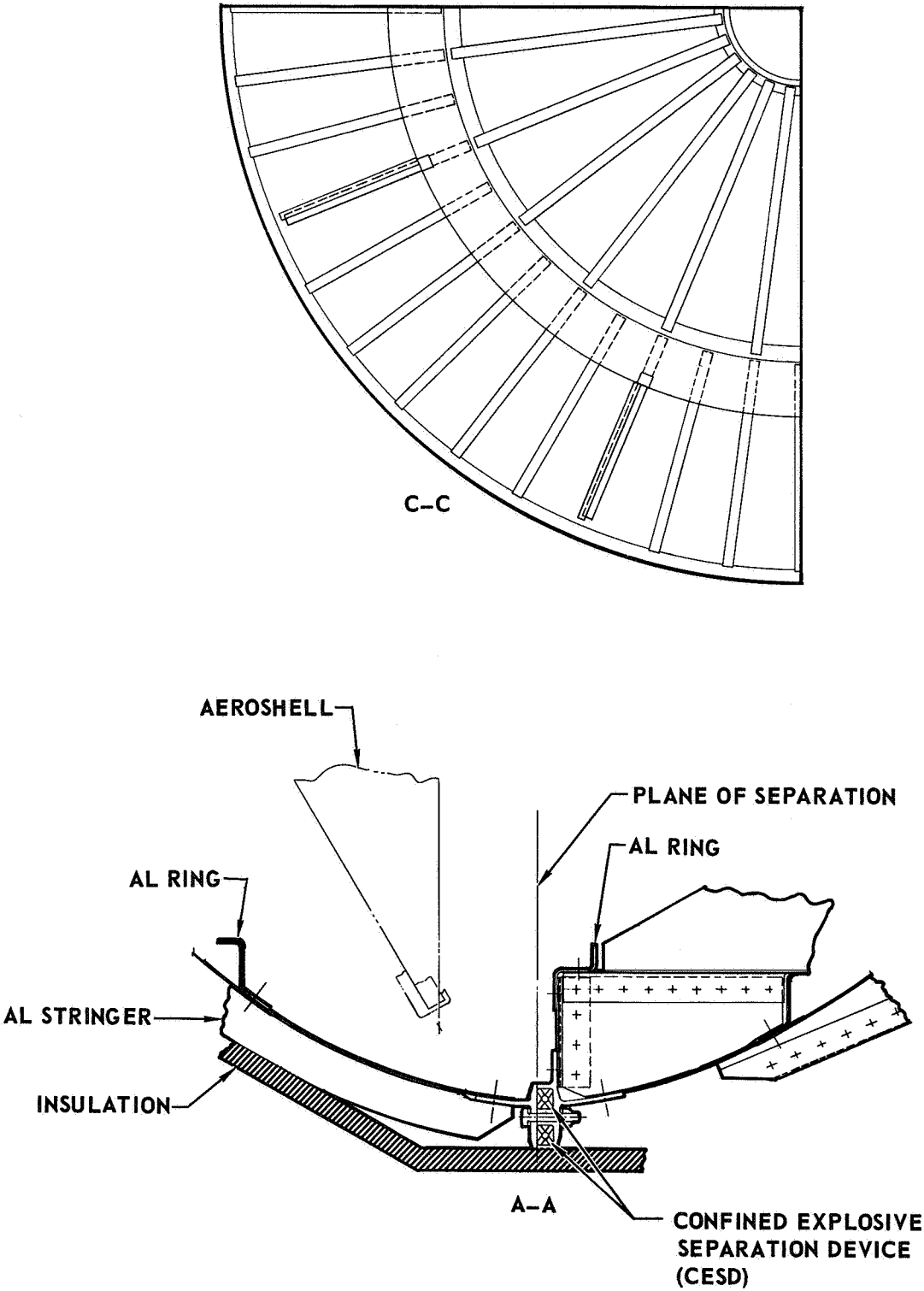


FIGURE 3.2.11-1
SHEET 1 OF 2

3.2.11-2

CANISTER STRUCTURAL ARRANGEMENT



3.2.11.1.2 System description: The canister structure is composed of two major sections or assemblies, as shown in Figure 3.2.11-1. The forward section consists of an 0.012 inch aluminum spherical segment shell and a toroidal segment adjacent to the field joint ring. Stiffening is provided by external rings and meridional stringers. These stringers and rings also provide support for the insulation blanket covering the external surface.

The aft section mates with the forward section at the field joint ring, and its shape and construction are similar to that of the forward section. In addition, the aft section mates with the adapter and the interface cone. Bolts through the canister wall are used to attach the adapter to fittings on the interface cone. The other end of these fittings bolts to the orbiter. The interface cone transfers loads from the canister/adapter interface to the orbiter. Use of the interface cone minimizes penetration, and therefore leakage, of the canister wall. Leakage is further minimized by sealing the structural joints during assembly with RTV-560, a room temperature-curing, elastomeric sealing compound. Most pressurization and vent and separation equipment mounts on or in the aft canister section.

The separation system provides a well proven design to separate and eject the canister forward section from the aft section. Major elements of the separation system are shown in Figure 3.2.11-2 and include the following:

- a) The confined explosive separation device, which is installed in grooves on each side of the bolts that clamp the canister forward and aft sections together. This device consists of a mild detonating cord (MDC), positioned by a Teflon extrusion in a stainless steel tubing which is flattened to the shape of the groove in the field joint flange. The CESD is made up of two 180° sections, sealed at each end for each groove.
- b) The stimulus transfer tube, which contains an explosive for transferring the detonation to the CESD.
- c) The electro-explosive device (EED).

SEPARATION SYSTEM COMPONENTS

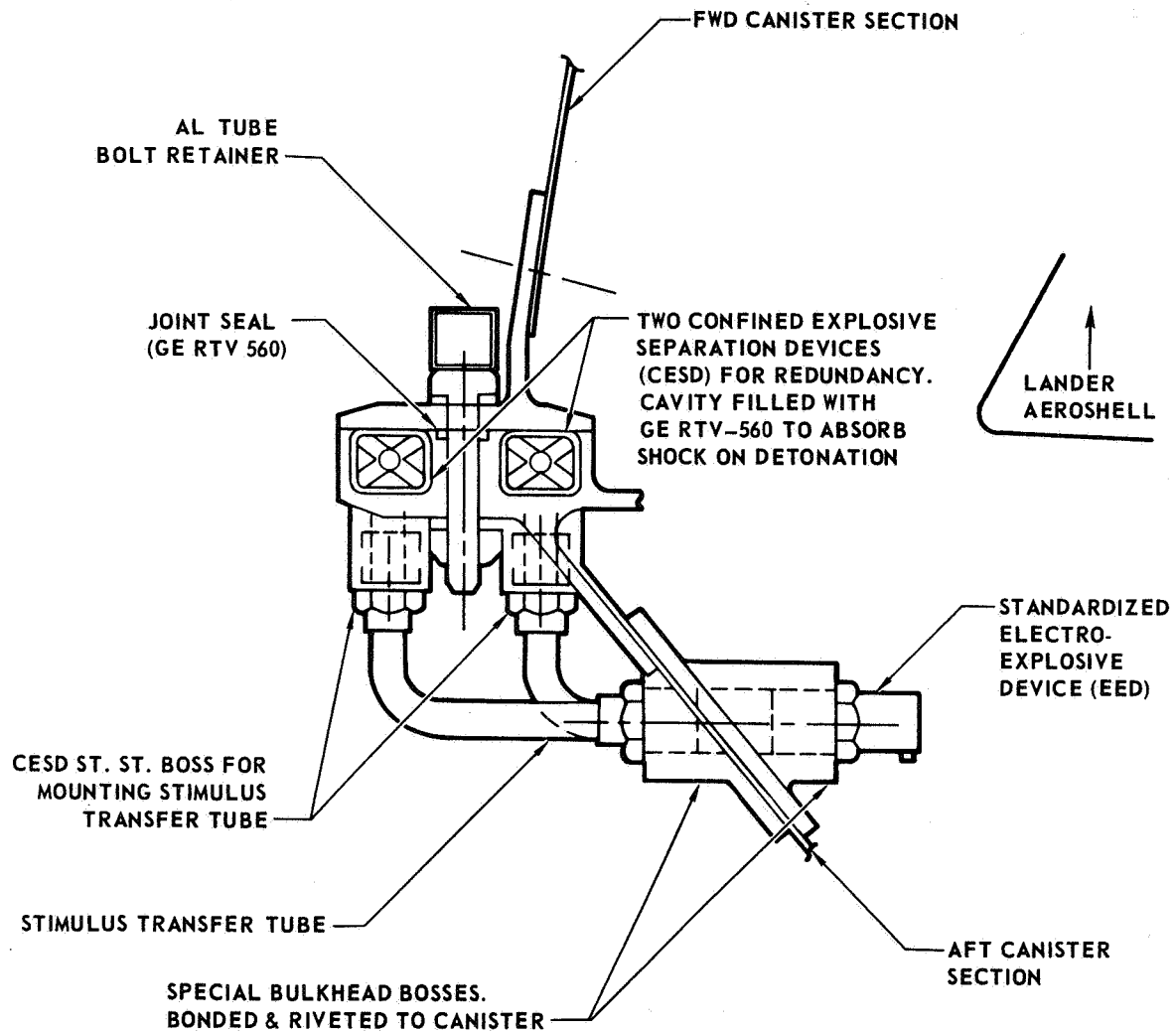


FIGURE 3.2.11-2

- d) The field joint titanium bolts, which have their head end drilled to the depth of the desired break point so that the breaking strength is predetermined.

The EED's are initiated on receipt of an electrical signal, followed by detonation of the stimulus transfer tubes and then the CESD. This detonation expands the steel tube and exerts a force between the clamped flanges of the field joint ring, which causes the bolts to break in tension and the forward section of the canister to be ejected. The broken bolts are constrained by a retainer from leaving the canister forward half, and the nut is retained by nut plates. The canister separates at a velocity of from 1.25 to 10 ft/sec.

The pressurization and vent equipment maintains the internal biointegrity of the canister and prevents the canister structure from being overloaded during terminal sterilization and until capsule separation in Mars orbit. Two redundant components of each major element are incorporated as shown in Figure 3.2.11-3 so that no single failure can have catastrophic results. This redundancy is carried into the valve systems by having filters in series to protect the open valves.

The major elements of the P&V equipment are:

- a) Vent valves to provide the capability to quickly release the canister internal pressure during launch.
- b) Pressure operated relief valves to protect the canister structure internal pressure from exceeding 2.25 psi.
- c) Purge and evacuation valves, which are used during decontamination and sterilization cycles on the ground. They are opened in Earth orbit and remain open to bleed off any residual pressure or outgassing.
- d) Biological filters, placed in series with each of the above valves to protect the valve's penetration into the sterile canister interior. This barrier to recontamination is redundant with the valve to provide a double guard against recontamination.
- e) Differential pressure switches provide a signal to the Operational Support Equipment (OSE) to reduce pressure on the ground and a signal

PRESSURIZATION & VENT SCHEMATIC

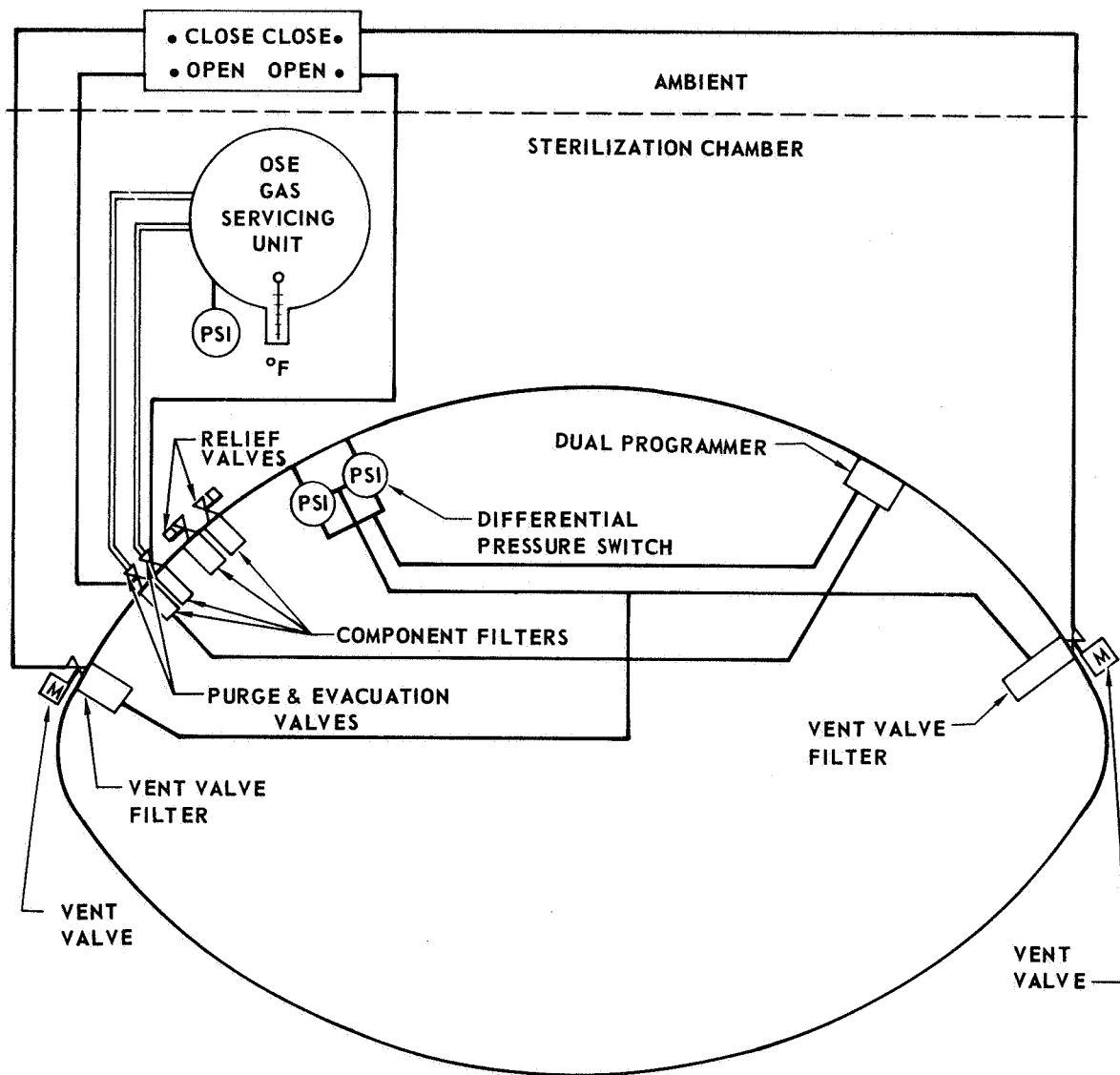


FIGURE 3.2.11-3

after launch to close the vent valves when differential pressure falls below .50 psi.

The OSE Gas Servicing Unit is connected to the purge and evacuation valves as soon as the canister is closed in preparation for terminal sterilization. It maintains internal pressure from .5 psi to 2.25 psi (limit pressure).

3.2.11.1.3 Parametric study: Results of parametric studies to determine the effect of flight capsule weight and aeroshell diameter on canister weight are shown in Figure 3.2.11-4. The canister weight includes the structure, the separation system, and the pressurization and venting system.

Structural weight contributes the largest portion to the total canister weight and is primarily influenced by aeroshell diameter. Structural weight includes weights of the forward and aft shell structures and the field joint ring.

An 0.012 inch aluminum skin (considered to be the minimum material gage practical for handling) is used for the forward and aft shell structure. This thickness is sufficient to withstand the 3.75 psi burst pressure for the range of aeroshell diameters investigated during this study. Therefore, canister structural weight is primarily a function of aeroshell diameter. However, shell weight is also influenced by flight capsule weight, because of the effects on aft canister geometry. As the flight capsule weight is increased, the depth of the aft canister must be increased to provide more clearance, resulting in additional shell weight. Increases in shell weight due to large flight capsule weights are partially offset by a reduction in the weight of the field joint ring, because radial loads applied to the ring decrease as the canister depth is increased.

The separation system contributes only a small part to the total canister weight. Its weight is primarily influenced by aeroshell diameter.

The P&V system also contributes only a small part to the total canister weight. Its weight is primarily a function of canister volume, and therefore aeroshell diameter.

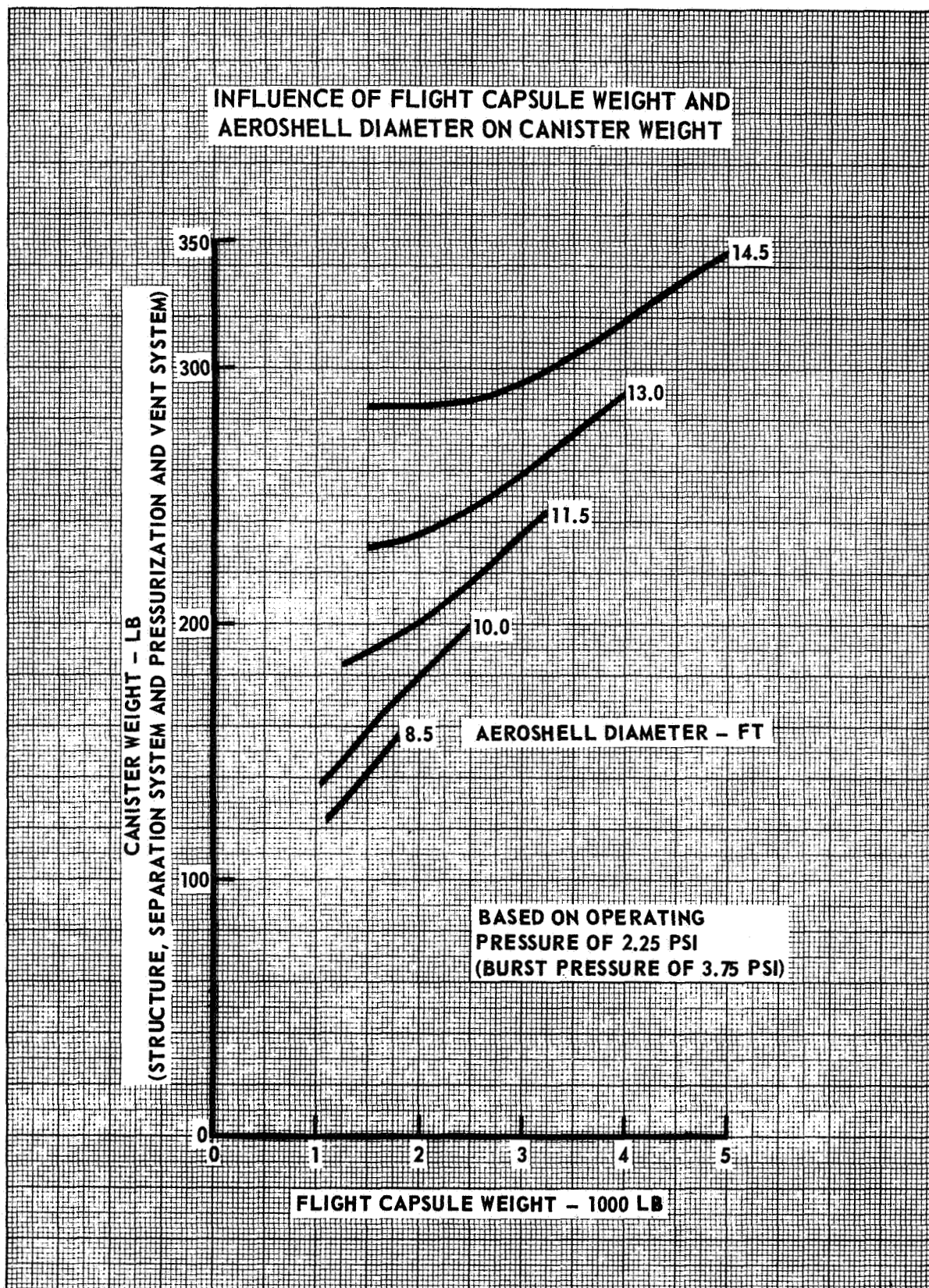


FIGURE 3.2.11-4

3.2.11.2 Adapter. - The adapter is a simple truss assembly which attaches the capsule to the orbiter. Capsule inertia loads which occur during ground handling, launch, and cruise are transferred by the adapter structure to the canister/adapter interface and through the canister interface cone to the orbiter. Parametric studies of the adapter show that its weight is primarily a function of flight capsule weight.

3.2.11.2.1 Requirements: The requirements that have a significant influence on the design of the adapter are:

- a) Attaching adapter to capsule before attaching adapter to canister.
- b) Interchangeability of adapter as a total assembly.
- c) Accurate alignment between the inertial platform of the capsule and that of the orbiter.
- d) Thermal isolation of the orbiter from the capsule.
- e) Limitations on dynamic frequencies and excursions.

3.2.11.2.2 Structure: The adapter is designed to support the capsule for ground handling, launch, and cruise loads, and to allow for joining or separation of the capsule to the canister without access doors in the aeroshell or in the canister. The main truss consists of 16 aluminum tubular members assembled as shown in Figure 3.2.11-5.

The design of the adapter-to-canister endfittings, with fixed plate nuts on the adapter side of the joint, permits the adapter to be attached to the capsule. This assembly can then be installed within the canister aft section. Bolts which attach the adapter to the canister are installed from outside the canister with sealing washers, to complete the installation.

The adapter is made an interchangeable unit by jig-drilling the interface fittings after assembly. The structural interface surfaces are accurately machined to maintain a close alignment between the capsule and orbiter. Thermal isolation between the adapter, capsule, and canister is not required, since all three are in a nearly isothermal environment within the insulation blanket.

ADAPTER STRUCTURAL ARRANGEMENT

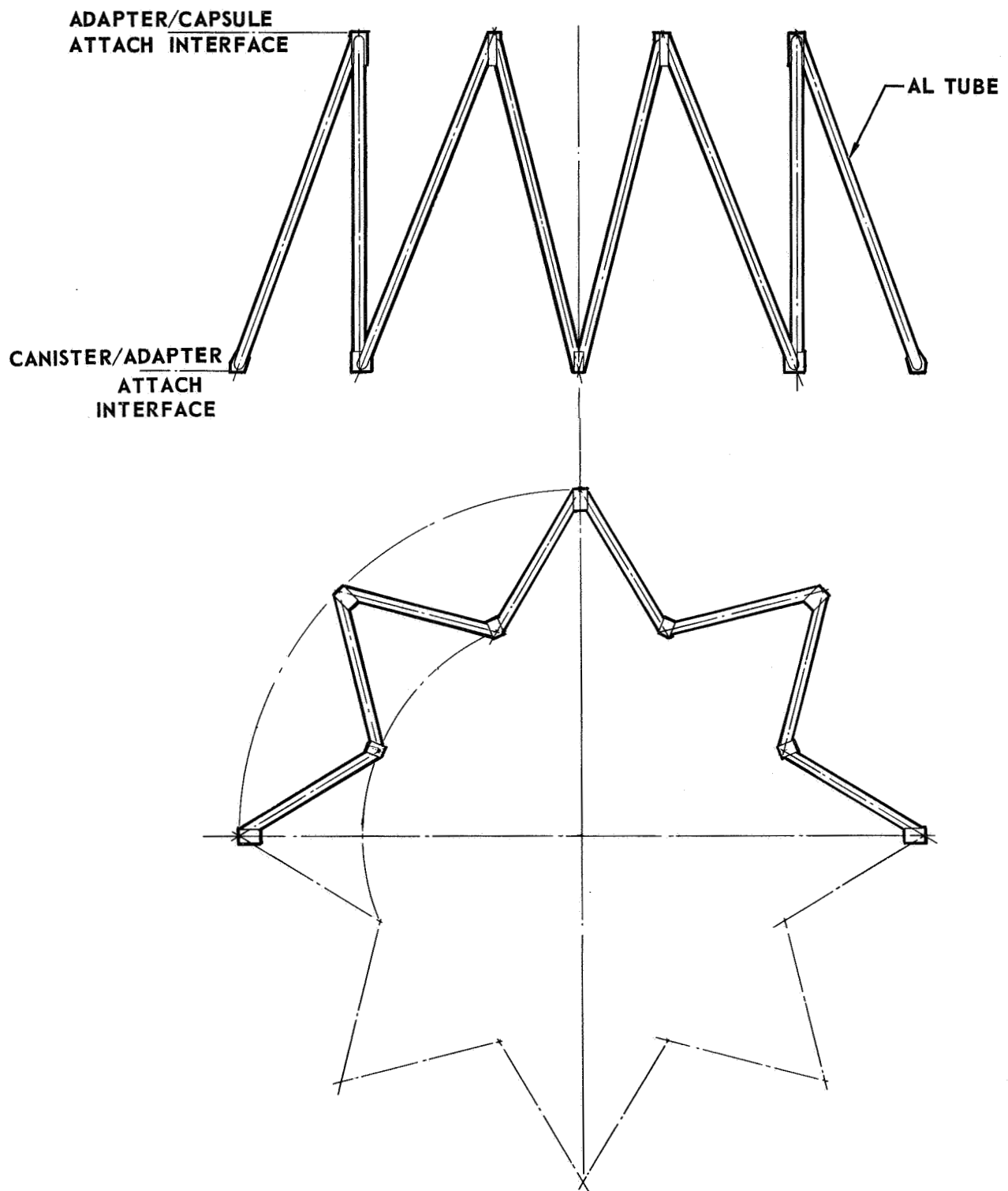


FIGURE 3.2.11-5

3.2.11-11

The truss-type design provides a short and stiff load path between the capsule and the orbiter through the interface cone. No obstructions to circulation of gases during sterilization are presented by the widely spaced tubes. Using eight attachment points to the canister allows fewer support struts and fittings and minimizes weight for the large, lightly-loaded canister structure.

3.2.11.2.3 Parametric study: The results of the adapter parametric studies to determine the influence of flight capsule weight and aeroshell diameter on adapter weight are shown in Figure 3.2.11-6. Adapter weight increases as flight capsule weight increases because the adapter must have greater length and greater load carrying capability.

The orbiter interface diameter was maintained constant at 6.5 ft for these studies. The adapter geometry and weight are influenced by changes in adapter length which is associated with flight capsule weight changes and/or changes in adapter/lander interface diameter. Changes in adapter/lander diameter are associated with changes in aeroshell diameter.

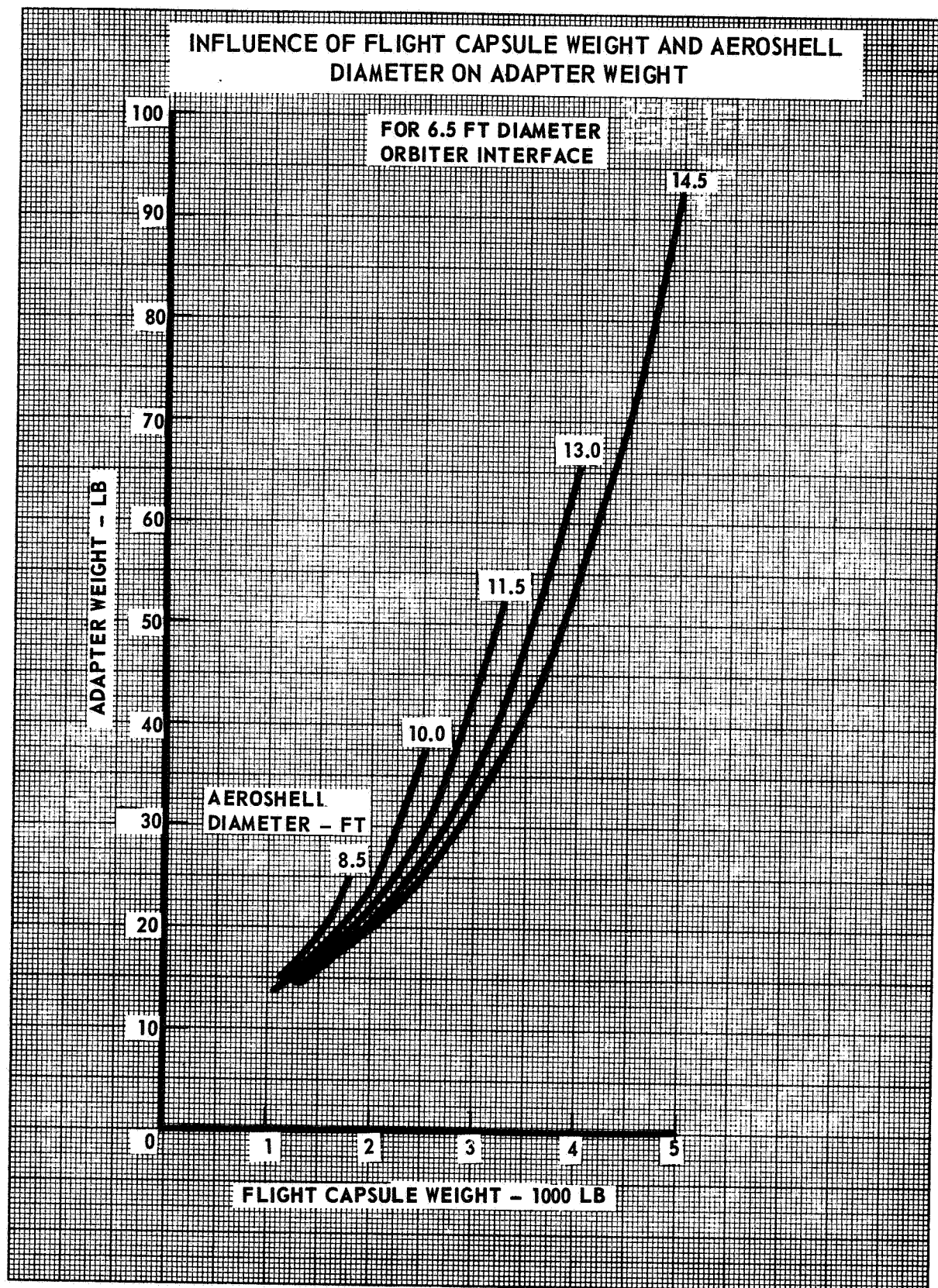


FIGURE 3.2.11-6

3.3 Capsule Parametric Analysis

There are many possible combinations of capsule systems. In order to present meaningful and usable data the number of these combinations must be constrained to some manageable level. The capsule parametrics accomplished during this study cover the most likely system combinations. The selected combinations and some rationale for their selection are discussed below.

The criteria used in the selection were to minimize the capsule cost, weight, or aeroshell diameter. Thus, in some cases, the lightest weight system was not selected.

The parametric analysis was conducted separately on the delivery systems and the surface payload. The delivery system parametrics are made as a function of quantitative values of the surface payload. This provides the capability to integrate selected surface payloads with the appropriate delivery systems. This type of matching has been done for the four designs of Section 4.0.

3.3.1 DELIVERY SYSTEMS - The delivery systems include all capsule systems except the surface payload. These systems were analyzed in terms of the two major mission phases: descent and entry and terminal deceleration.

3.3.1.1 Descent and entry. - Descent and entry begins with capsule separation from the orbiter and ends at deployment of the aerodecelerator or, for the all-propulsive system, at ignition of the terminal propulsion system. The systems considered in this phase are the attitude control system, the reaction control system, the deorbit motor, the aeroshell, the entry science, communications, and thermal control.

3.3.1.1.1 Requirements: The capsule descent and entry systems must provide the capability to separate the capsule from the orbiter, perform the deorbit maneuver, and make an entry into the VM-1 to VM-10 model atmospheres. The important parameters which define the descent and entry phase are the desired landing site, the orbit size, and the entry corridor. Data transmission requirements define additional parameters that affect the descent and entry. These include equipment **and power** requirements, for transmission of entry science data to the orbiter for relay to Earth, minimization of multipath interference, and communication range.

The conditions at the end point, i.e., velocity and altitude at deploy-

ment of the aerodecelerator, must be considered. All of the considerations, i.e., landing anomaly (θ_L), capsule lead angle at entry ($\Delta\theta_e$), entry velocity (V_e), entry angle (γ_e), orbit size, aerodecelerator deployment conditions (V_D & h_D), then establish the descent and entry requirements such as the deorbit velocity increment (ΔV_{DO}), deorbit anomaly (θ_{DO}) deflection angle, and entry relay communication range.

3.3.1.1.2 Candidate system combinations: The candidate descent and entry systems are presented in Table 3.3.1-1. In some cases the system/operating modes are fixed. For other systems they must be determined after considering the functional performance of all the systems. The fixed systems (and the primary reasons for selection) are: relay communications (energy requirements), ablator/composite heat shield (weight, prevention of entry atmospheric sensor contamination, and provision for an RF window for the radar altimeter), entry science, and thermal control.

3.3.1.1.3 Deceleration: The functions to be performed are separation of the capsule from the orbiter, deorbit maneuver, and attitude control of the capsule through entry.

Capsule Separation - Routine separation techniques are used to accomplish separation. Explosive bolts provide disconnect and the reaction control jets provide the thrust for separation.

Deorbit Maneuver - The deorbit maneuver requirements are based on the selection of the deorbit velocity increment. To provide flexibility in landing site selection a trade-off must be made between (1) the deorbit velocity increment, to permit variation in deorbit anomaly, and (2) the orbit insertion velocity increment, to permit orbit rotation. A $\Delta V_{DO} = 450$ ft/sec was chosen for these parametric studies. The effect of this selection on the other descent and entry parametrics is shown in Section 3.1. A 2-3 ft/sec increase in this deorbit velocity increment will allow a reduction of approximately 1 ft/sec of orbit insertion velocity increment.

It should be noted that a small increase in the ΔV_{DO} increases the total capsule weight but has negligible effect on the entry and lander weight. Thus, there is no significant change to a capsule's aeroshell diameter by increasing the ΔV_{DO} . If the ΔV_{DO} is increased by 500 ft/sec ($\Delta V_{DO} = 950$ ft/sec) for a

TABLE 3.3.1-1

DESCENT AND ENTRY CANDIDATE SYSTEMS – CAPSULE PARAMETRICS

SELECTED SYSTEM

SYSTEM	CANDIDATES	REMARKS
DEORBIT MOTOR	SOLID PROPELLANT LIQUID MONOPROPELLANT LIQUID BIPROPELLANT	LOWEST WEIGHT EASE OF SEPARATION HIGH RELIABILITY LOW COST
THERMAL CONTROL	THERMAL CURTAIN INSULATION	LOW WEIGHT SIMPLE
ATTITUDE CONTROL SYSTEM	ACTIVE SPIN/ACTIVE TVC/ACTIVE	MINIMUM DISPERSIONS FROM ZERO ENTRY ANGLE OF ATTACK HIGH RELIABILITY LOWEST WEIGHT
REACTION CONTROL SYSTEM	COLD GAS NITROGEN MONOPROPELLANT HYDRAZINE	HIGH RELIABILITY LOW COST NEGLECTIBLE WEIGHT PENALTY
AEROSHELL/HEAT SHIELD	SINGLE FACED CORRIGATED STRUCTURE/ABLATOR WITH COMPOSITE NOSE CAP	LOW WEIGHT NO ENTRY SCIENCE SENSOR CONTAMINATION
ENTRY SCIENCE	ATMOSPHERICS & MASS SPECTROMETER SAME PLUS IMAGING	
COMMUNICATIONS	RELAY DIRECT	LOW ENERGY REQUIREMENT MOST EFFICIENT

1600 lb capsule using a solid deorbit motor, the capsule weight increases only about 6%.

The solid propellant motor was selected for the parametric studies. This system is lightest in weight, and has the highest reliability and lowest cost (see Section 3.5). In addition, jettisoning the deorbit residual and inerts is much simpler with the solid motor. In order to jettison either of the liquid deorbit systems, the tankage, valves, lines, thrusters, etc. must be assembled on some type of truss. This would result in additional weight and would require additional development (separation testing) costs. The sterilization capability of the solid must be demonstrated; however, no major problems are anticipated.

Attitude Control - Control system electronics and reaction control system parametric studies were made for active, spin stabilized, spin/active, (both solid and liquid spin motors), and TVS/active control modes. The attitude control and capsule separation requirements are satisfied by using the active control mode. This mode, although it involves somewhat higher costs, results in combined systems that are lighter, more accurate, and more reliable.

The spin stabilized mode was eliminated because of the requirement for orbiter maneuvering and entry angle of attack dispersions (see Section 3.2.4). It is lowest in cost, but is the heaviest, has large pointing errors, and is less reliable than the other systems. The control electronics are heaviest for the active mode, but when all systems, (battery increment, sequencing and timing, reaction control system, and radar altimeter) are included it is the lightest for descent times up to 5.5 hours. This overall relationship is shown for a 2000 lb capsule in Figure 3.3.1-1. The active control mode system is 3% lighter than the spin/active mode system for a typical trajectory ($\gamma_e = -16^\circ$, $\Delta V_{DO} = 400$ ft/sec, $\Delta \theta_e = 5^\circ$, and $\theta_{DO} = 214^\circ$) with a descent time of 3.4 hours. Even at a descent time of 10 hours, the spin/active mode systems are only 6.5% lighter than the active mode systems. This weight increment is only 0.4% of the capsule weight.

The cold gas reaction control system was chosen for the descent and entry attitude control. It has a higher reliability and requires much less develop-

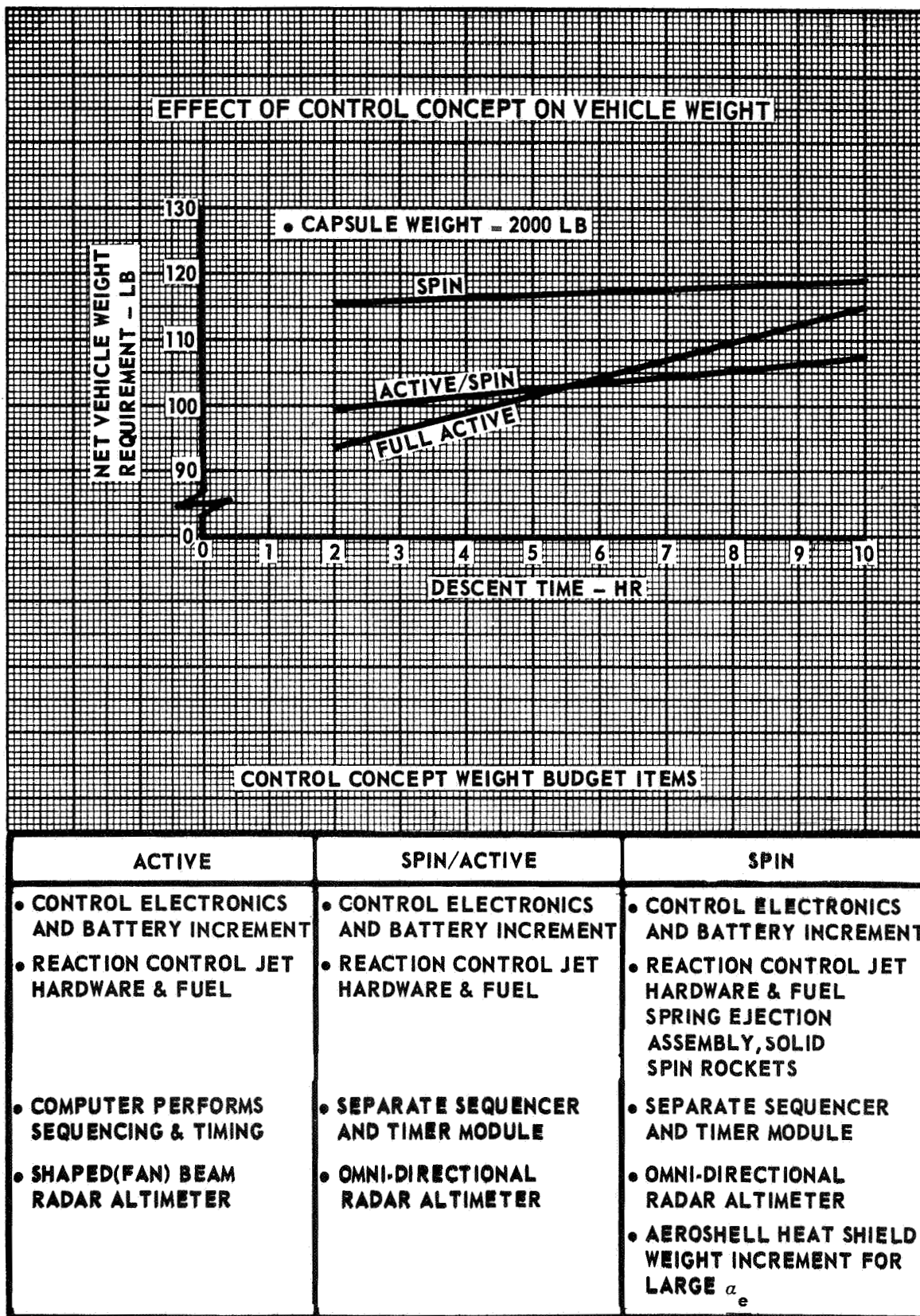


FIGURE 3.3.1-1

ment effort than the monopropellant hot gas system, and the weight penalty is less than 0.25% of the capsule weight, i.e., 1 lb for 1000 lb capsule and up to 12 lb for a 5000 lb capsule. The cold gas system costs less than the hot gas system.

3.3.1.1.4 Communication capability: The capsule relay communication system can transmit engineering and entry science data at approximately 4000 bits per second from the deorbit through the landing phases. The capsule deorbit maneuver is selected to control the capsule-orbiter lead angle to a value of 5° to 10° at the point of entry. This value minimizes the capsule transmitter power required and, at the same time, insures capsule-orbiter visibility for a few minutes after landing to allow immediate transmission of some surface video data.

3.3.1.1.5 Science compatibility: A composite (HCF) nose cap is necessary in order to prevent contamination of the sensors by the ablator material. The selection of the active control mode as discussed above improves the control accuracy (minimizes dispersions from 0° angle of attack) and thereby provides entry data which gives better reproducibility of the measured atmosphere.

3.3.1.1.6 Thermal control: The capsule thermal control system for descent and entry employs a thermal curtain across the back of the aeroshell, insulation, and electrical heaters within the capsule.

3.3.1.2 Terminal deceleration - Terminal deceleration begins with deployment of the aerodecelerator (or for the all-propulsive at TPS ignition) and ends after lander touchdown. Deceleration of the entry vehicle to "soft landing" impact velocities, considering the possibility of strong surface winds, requires the use of a terminal propulsion system. Because of the high efficiency of auxiliary aerodynamic decelerators, use of these devices in combination with terminal propulsion systems has received more attention than the use of all-propulsive systems.

3.3.1.2.1 Requirements: The capsule terminal deceleration system must provide the capability to remove the last 5-10% of the entry velocity. The important parameters which define the terminal deceleration requirements are altitude and velocity at deployment of the aerodecelerator, altitude and velocity at terminal propulsion initiation, surface winds, and surface con-

ditions. The selected entry corridor and the Mars atmosphere define a range of values for the above parameters.

Several constraints have been imposed for combined terminal deceleration systems. For aerodynamic decelerators, the deployment initiation by radar altimeter has been selected, based on Voyager Phase B studies of triggering techniques. The maximum deployment altitude is limited by the velocity constraints on the aerodynamic decelerator (Mach 2 for parachutes and temperature limits on fabric strength for attached inflatable decelerators). The minimum deployment altitude must be sufficient for inflation and deceleration. The size of the auxiliary decelerator is governed by its drag efficiency for deceleration and for differential drag separation of the lander and the aeroshell and by the lander velocity at propulsion ignition, or by a requirement that the aeroshell impact prior to landing radar activation to avoid radar tracking of the aeroshell. Criteria for sizing the parachute are presented in Section 3.1.4 and 3.2.8.

For the terminal propulsion system the trade between total impulse and throttle ratio must be examined, in conjunction with the guidance system, to define the desirable ignition conditions.

3.3.1.2.2 Candidate system combinations: The candidate terminal descent systems are presented in Table 3.3.1-2. As a result of the system parametrics some of the systems are fixed, but the optimum conditions for combining them are not. Various combinations of control mode, terminal propulsion system, and parachute have been investigated.

3.3.1.2.3 Sequence of operations: A typical sequence of operations is shown in Figure 3.3.1-2. This is representative of a parachute deployment at $M_D = 2.0$ and aeroshell impact at the time of landing radar activation. The sequence of events is keyed to the radar altimeter which supplies range (altitude) data to deploy the aerodecelerator and initiate the TPS. The aeroshell is sequenced to separate approximately 5 seconds after decelerator deployment. The landing radar is activated 3 seconds after TPS ignition. The velocity and range signals are employed through the guidance and control system to effect a controlled soft landing.

TABLE 3.3.1-2
TERMINAL DECELERATION CANDIDATE SYSTEMS – CAPSULE PARAMETRICS
SELECTED SYSTEM

SYSTEM	CANDIDATES	REMARKS
RADAR ALTIMETER	NON-COHERENT PULSE	ACCURATE EFFICIENT MOST HIGHLY DEVELOPED
AERODECELERATOR	PARACHUTE AIDS NONE	LOWER WEIGHT HIGH RELIABILITY AIDS LANDER/AEROSHELL SEPARATION
TERMINAL PROPULSION SYSTEM	MONOPROPELLANT BIPROPELLANT ALL PROPULSIVE (SOLID/ LIQUID)	MONO: HIGHER RELIABILITY LOWER COST BI: LOWER WEIGHT
TERMINAL GUIDANCE MODE	MODIFIED PREPROGRAMMED PREPROGRAMMED ADAPTIVE	LOWER WEIGHT
LANDING RADAR	LM TYPE RADAR (LINEAR FM/CW FOR RANG- ING. CW FOR VELOCITY SENSING) BESSEL SIDE-BAND (SINUSOIDAL FM/CW FOR BOTH RANGE AND VELOCITY SENS- ING)	SIMPLE, PROVEN TECHNIQUE REQUIRES MODIFICATION TO EXISTING SYSTEM, BUT LESS ADDITIONAL DEVELOPMENT
LANDING SYSTEM	UNI-DISC	LOW WEIGHT GOOD LANDING STABILITY ON 34° SLOPES

TERMINAL DECELERATION SEQUENCE

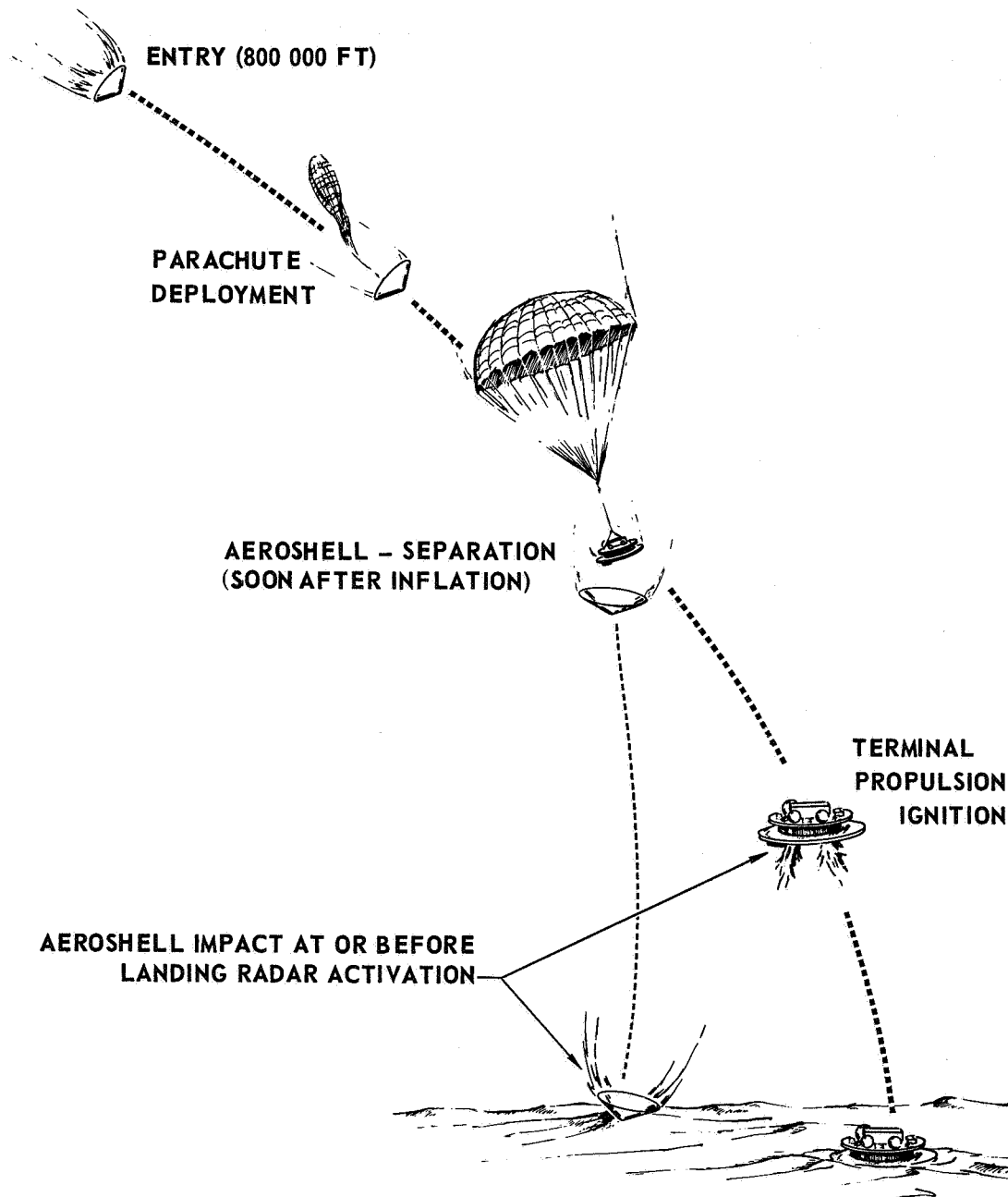


FIGURE 3.3.1-2

3.3.1.2.4 Aerodecelerator performance: Limited parametric studies have been made of the effects of parachute size and $m/C_D A$ on the velocity-altitude history of the lander and the relative position of the lander and aeroshell after separation. Deployment conditions are defined by the VM-8 atmosphere conditions for the specified Mach 2 parachute deployment limit. The lowest density occurs in the VM-7 atmosphere. This defines the parachute size required to reach a desired equilibrium velocity at ignition of the terminal propulsion system. Aeroshell weight and diameter, and lander weight, were determined from a preliminary relation of separated aeroshell weight with entry weight, see Figure 3.3.1-3. Aeroshell and lander masses and aeroshell diameter used in this study were:

$m/C_D A$ (Slug/ft ²)	h for M=2 in VM-8 $V_e = 13000$ $\gamma_e = -20^\circ$	Mass of Lander (Slugs)	Diameter of Aeroshell (Feet)	Mass of Aeroshell (Slugs)
.254	23000	51.2	14.75	14.2
.30	18590	52.2	13.5	12.7
.35	14430	53.2	12.6	11.8

Figure 3.3.1-4 presents velocity-altitude plots of the lander plus aeroshell from deployment to separation, and of the lander with various parachute sizes from separation to impact. For comparison, the trajectories of aeroshell plus lander with no parachute are included.

The time from mortar fire at deployment to full inflation is determined by our analysis of the PEPP data, assuming for preliminary analyses that the ballistic parameter of the parachute, the muzzle velocity, and the relative velocity of the parachute is similar to the PEPP test. Full inflation will be accomplished when the vehicle has traveled a distance equal to 40 times the parachute diameter. Two-thirds of this time is consumed in extension of the riser lines. It is assumed that the parachute contributes zero drag from deployment to line stretch and then the parachute drag area, $C_{D_0} S_0$, varies as the cube of time from line stretch to full inflation (on the basis of PEPP data). At full inflation, C_{D_0} is assumed to be equal to 0.7 for all cases. Aeroshell separation occurs four seconds after full inflation. Maximum opening shock loads occur in the VM-8 atmosphere. Non-reefed opening shock loads were calculated

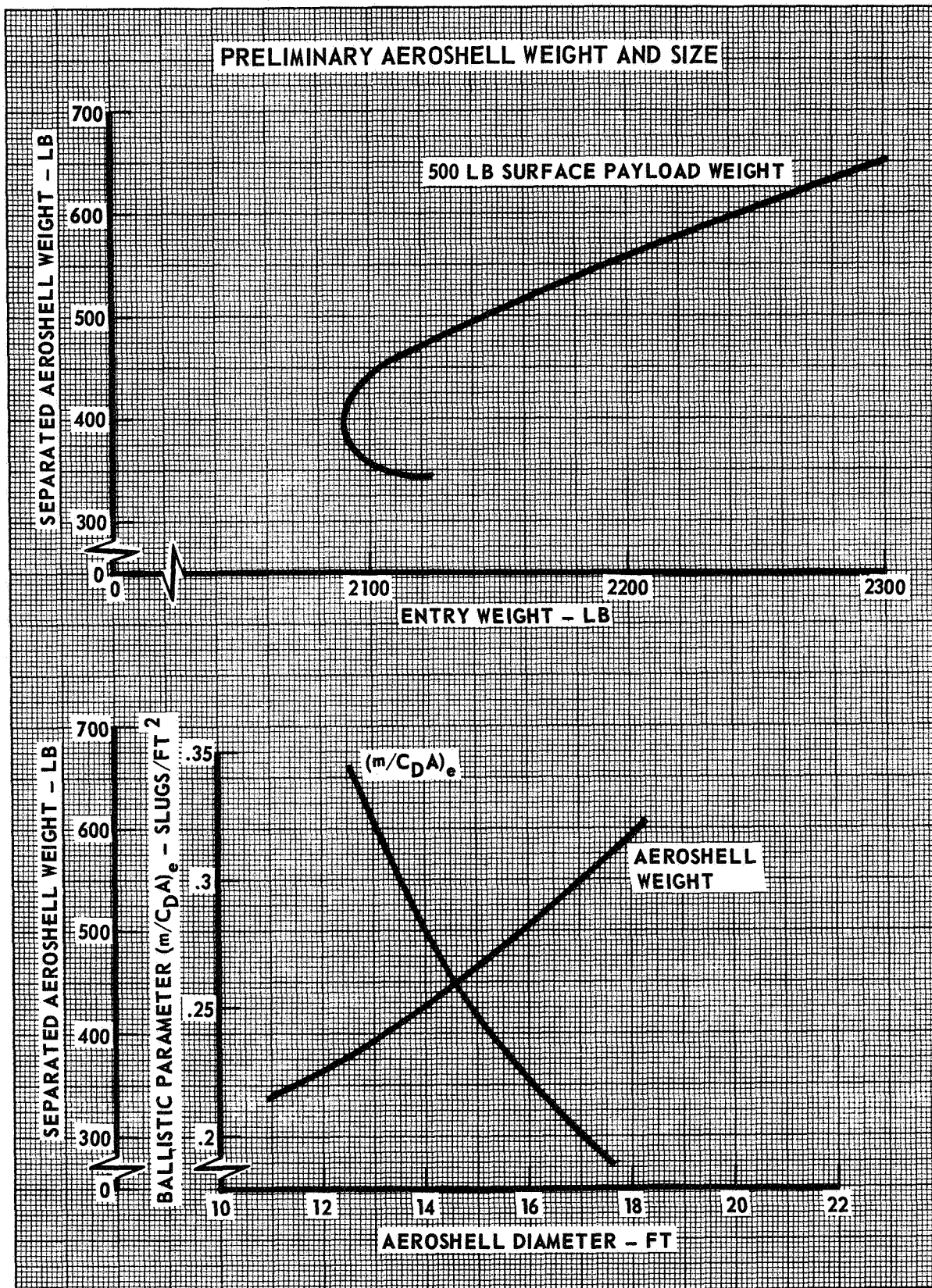


FIGURE 3.3.1-3

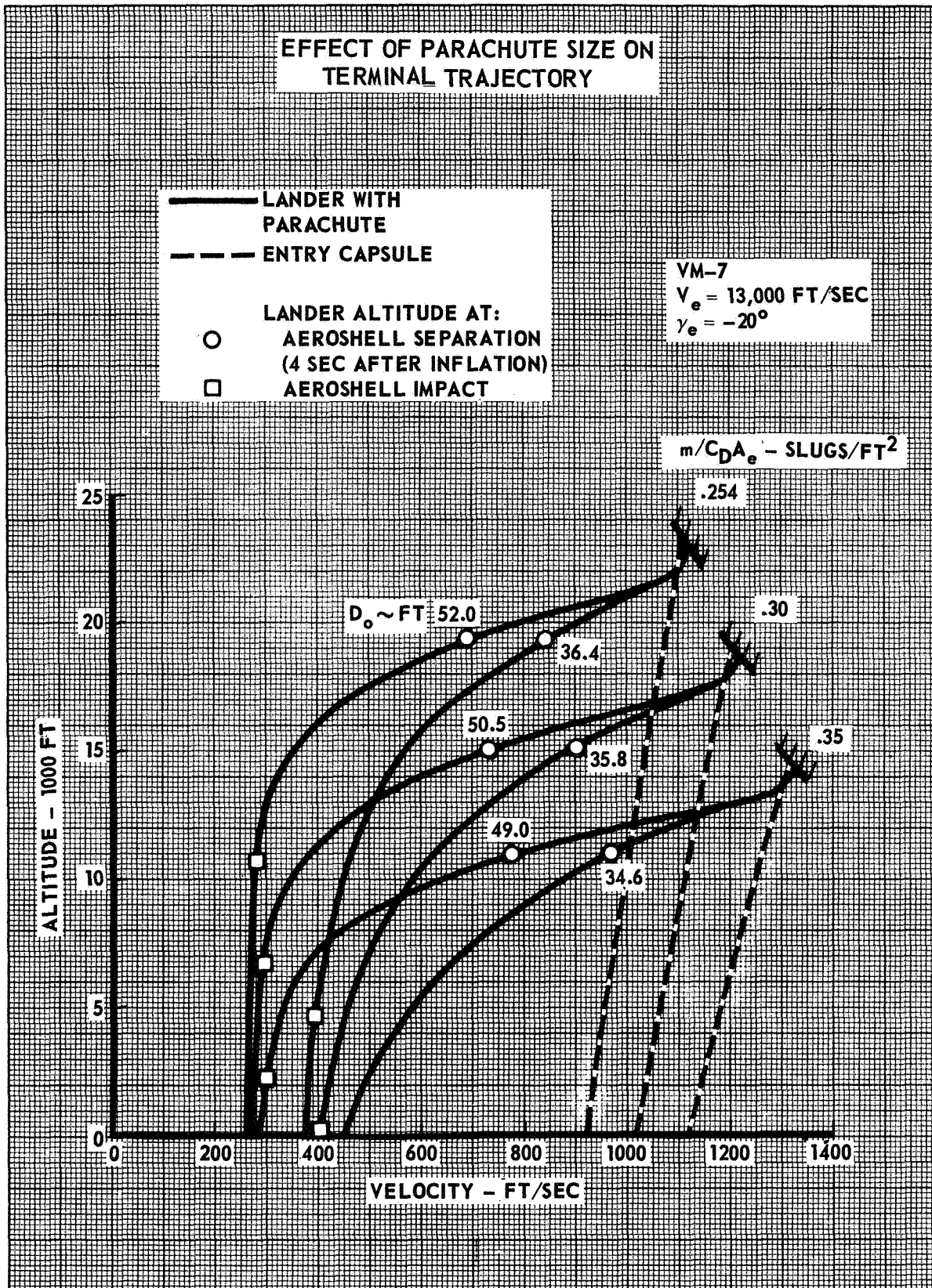


FIGURE 3.3.1-4

by empirical relationships derived from PEPP data and are shown in Figure 3.3.1-5 as a function of parachute nominal diameter for several values of m/C_{DA} . The weight of large parachutes designed for shock loads above 20000 to 30000 lb may be very heavy. Consequently, for point designs, reefing may be required, with further iterations being necessary to determine the parachute size.

Lander equilibrium velocity and altitude were defined as the point during descent at which the vertical descent velocity changed at a rate of 4 ft/sec per thousand feet of altitude change. The altitude and velocity at equilibrium velocity as a function of parachute diameter is given in Figures 3.3.1-6 and -7. To prevent the lander radar from tracking the aeroshell, the aeroshell should impact the surface prior to or at radar activation (3 seconds after terminal propulsion ignition). The relation of lander altitude at aeroshell impact as a function of entry m/C_{DA} and parachute diameter is given in Figure 3.3.1-8. With this information, Figure 3.3.1-9 was constructed, giving the lander equilibrium velocity as a function of entry m/C_{DA} and lander altitude at aeroshell impact. It would be desirable to design the system so that the lander attains its parachute terminal condition before the aeroshell impacts. The smallest parachute size allowed would be that specified by this limiting condition as shown in Figure 3.3.1-9. However, a higher altitude may be imposed by terminal propulsion ignition constraints, thus resulting in a requirement for a larger parachute.

The equilibrium velocity is lower for a more dense atmosphere (VM-9) as also shown in Figure 3.3.1-9. For the same size parachute ($m/C_{D_0} S_0$) the lander is also at a higher altitude than in a less dense atmosphere (VM-7) when the aeroshell impacts. Consequently if the landing radar is initiated by an altitude signal, the minimum altitude defined by the least dense atmosphere (VM-7) is critical.

3.3.1.2.5 Deceleration control concepts: Throttling laws of varying degrees of complexity can be configured to provide the thruster commands and achieve the soft landing objective. Considerations include: 1) propellant usage over the range of atmospheres, 2) capability for correcting possible dispersions, 3) capability of verticalizing the flight path angle prior to the final constant velocity descent, and 4) simplicity in obtaining the required

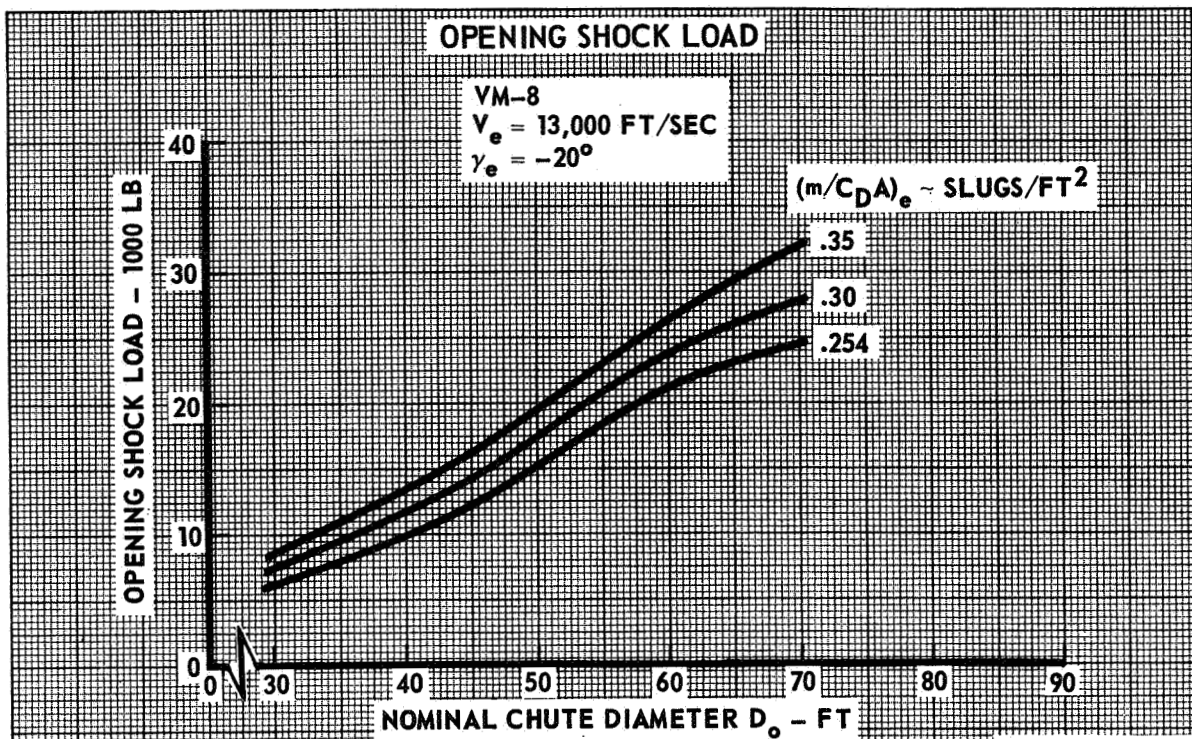


FIGURE 3.3.1-5

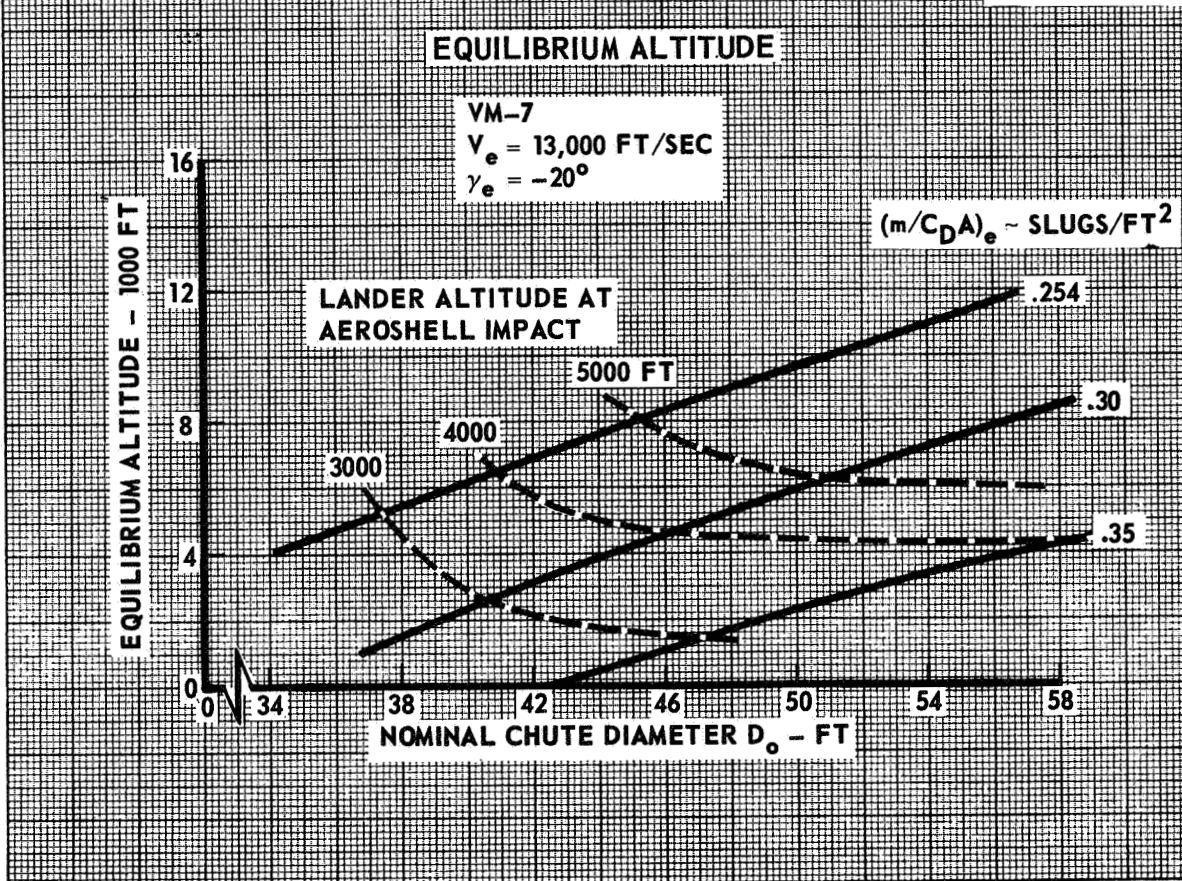


FIGURE 3.3.1-6

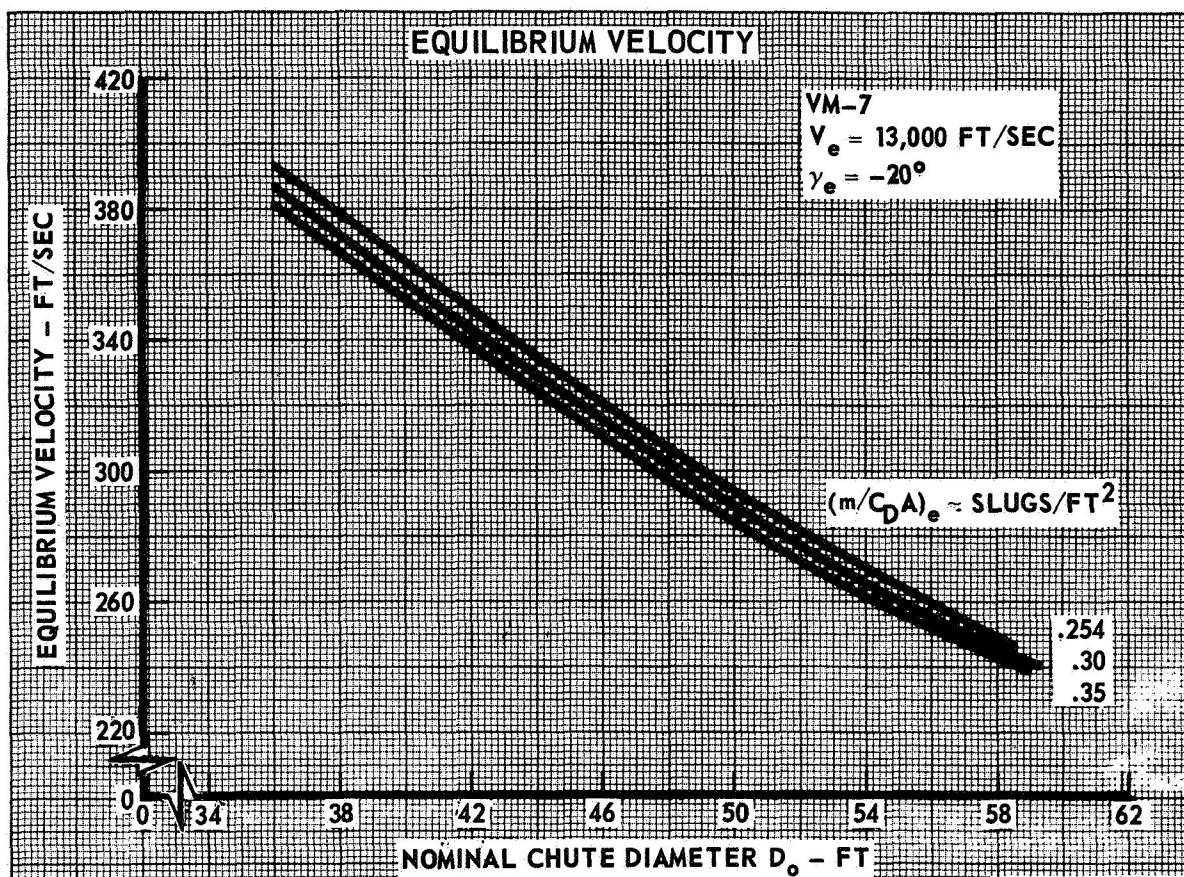


FIGURE 3.3.1-7

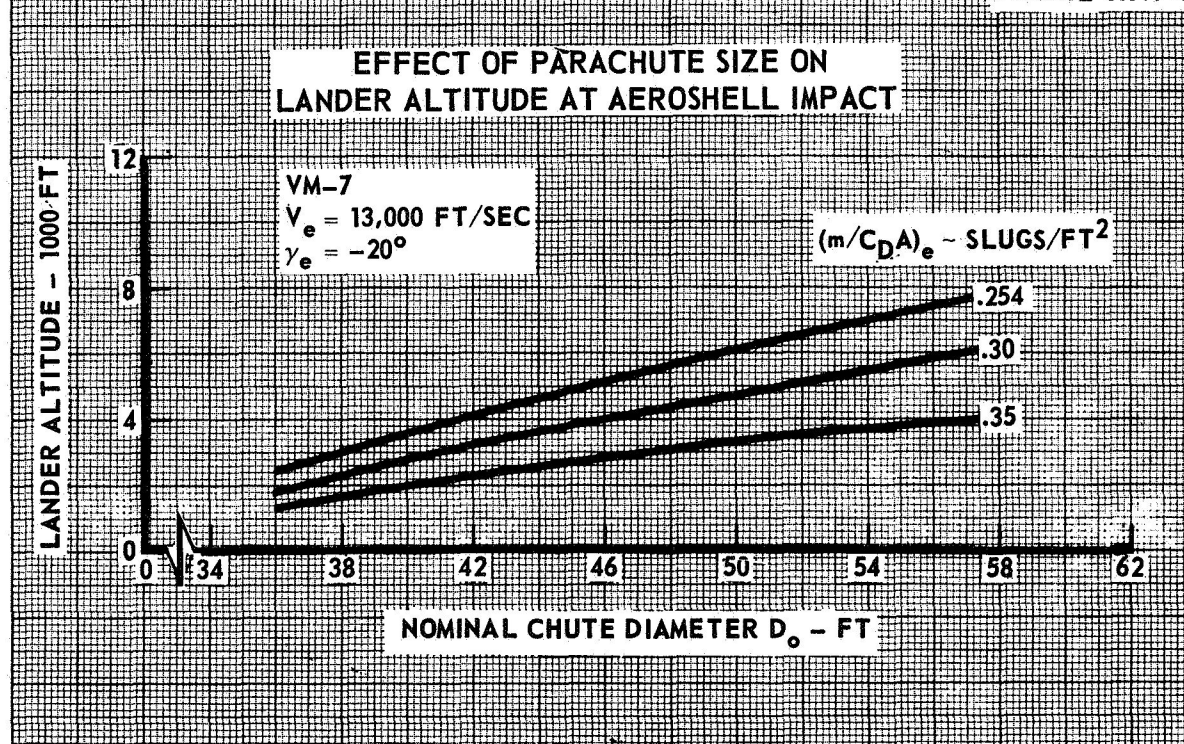


FIGURE 3.3.1-8

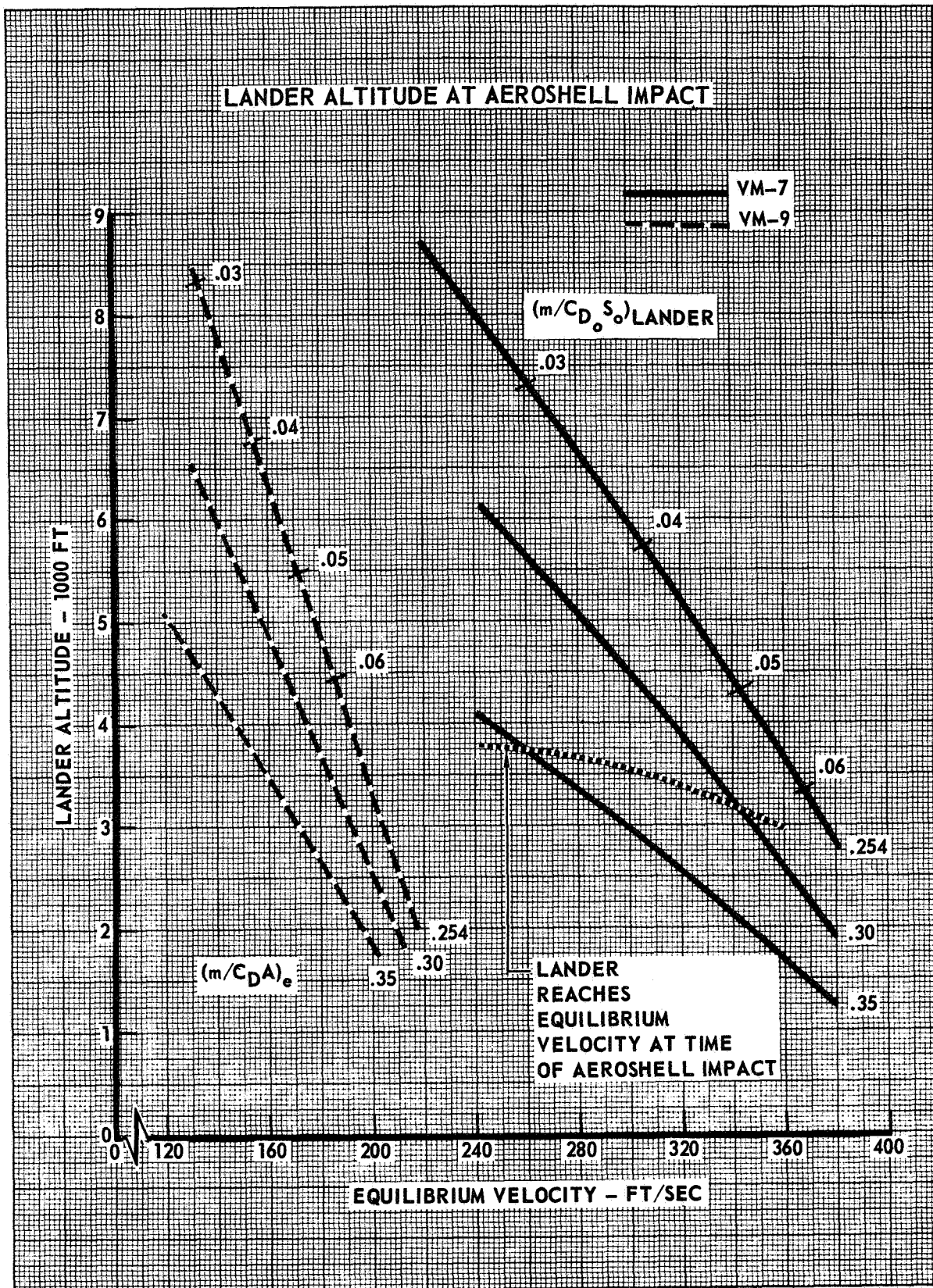


FIGURE 3.3.1-9

data for instrumentation.

Assuming a gravity turn maneuver where the thrust axis is maintained along the velocity vector, two simple throttling laws were investigated.

a) Linear Energy Law (Adaptive)

$$a_c = \frac{V^2}{R} + g_M$$

b) Linear Velocity Law (Surveyor type)

$$a_c = K (V_c - V)_{\text{Limited}}$$

where: V is the measured range velocity

R is the measured slant range

a_c is the command input to the acceleration loop which is closed around the throttleable engines

V_c is the velocity command, which is a preprogrammed function of the measured slant range

K is the velocity loop gain

g_M is the Martian gravity

The linear energy law forces the capsule energy to decrease linearly with range, thereby creating a constant deceleration thrust profile. It utilizes the existing range and velocity information and computes the acceleration to be applied to reduce these parameters to zero.

The linear velocity law, which was utilized in the Surveyor system, forces the capsule velocity to decrease linearly with altitude, based on preprogrammed constants. This law requires the thrust level to begin near maximum and decrease to become equal to the capsule weight near the end of the descent. The velocity command (V_c) mechanization of the previous equation for this system is of the following form:

$$V_c = K_1 R$$

where K_1 is a preprogrammed constant.

A modification to the preprogrammed (Surveyor type) throttling law was also investigated. This modification utilizes a velocity command mechanization of the form:

$$V_c = \sqrt{2} [R (a - g_M)]^{1/2}$$

where a is a constant preprogrammed acceleration. Hence, the system forces the capsule to follow a velocity history corresponding to a constant deceleration.

Typical time histories for each concept are shown in Figure 3.3.1-10 for the same ignition conditions. A six second period following ignition for attitude stabilization and radar acquisition is shown for each concept. During this period the engines are throttled to produce $.8g_M$. In both the preprogrammed concepts, at the end of this six second period $.8g_M$ is maintained if the measured velocity is less than the commanded, and maximum thrust is commanded if it is greater.

The propulsion system weight parametrics of Section 3.2.7 indicated the minimum weight system can be realized by utilizing the modified preprogrammed concept, assuming an optimum throttle ratio is employed.

In addition to propulsion system requirements, another important factor is the flight path verticalization prior to switchover to the final constant velocity descent. Since lateral velocity control is not employed during the constant velocity descent phase, because of the noise uncertainty in these signals, misalignment of the thrust axis from the vertical will result in an increase in horizontal velocity. A gravity turn descent will assure flight path verticalization as the velocity approaches zero for each of the throttling law concepts. However, for thrust decelerations of greater than $2 g_M$'s, the rate of change of the flight path angle approaches infinity as the velocity approaches zero. Keeping the thrust axis aligned with the flight path during the high rates of change is not possible, due to control limitations.

The change in flight path angle as a function of velocity change, for the adaptive, preprogrammed, and modified preprogrammed concepts are shown in Figure 3.3.1-11. The preprogrammed constants are those that produce the velocity command curves of Figure 3.3.1-10. As can be seen, the adaptive system has high rates of change as the velocity approaches zero. This will always be the case as long as the initial conditions are such as to require command accelerations greater than $2 g_M$'s. This situation is alleviated in the other two concepts, since the deceleration commands are $2 g_M$'s or less as the velocity approaches zero. Hence, from the standpoint of flight path verticalization, the adaptive system is not acceptable, but either the pre-

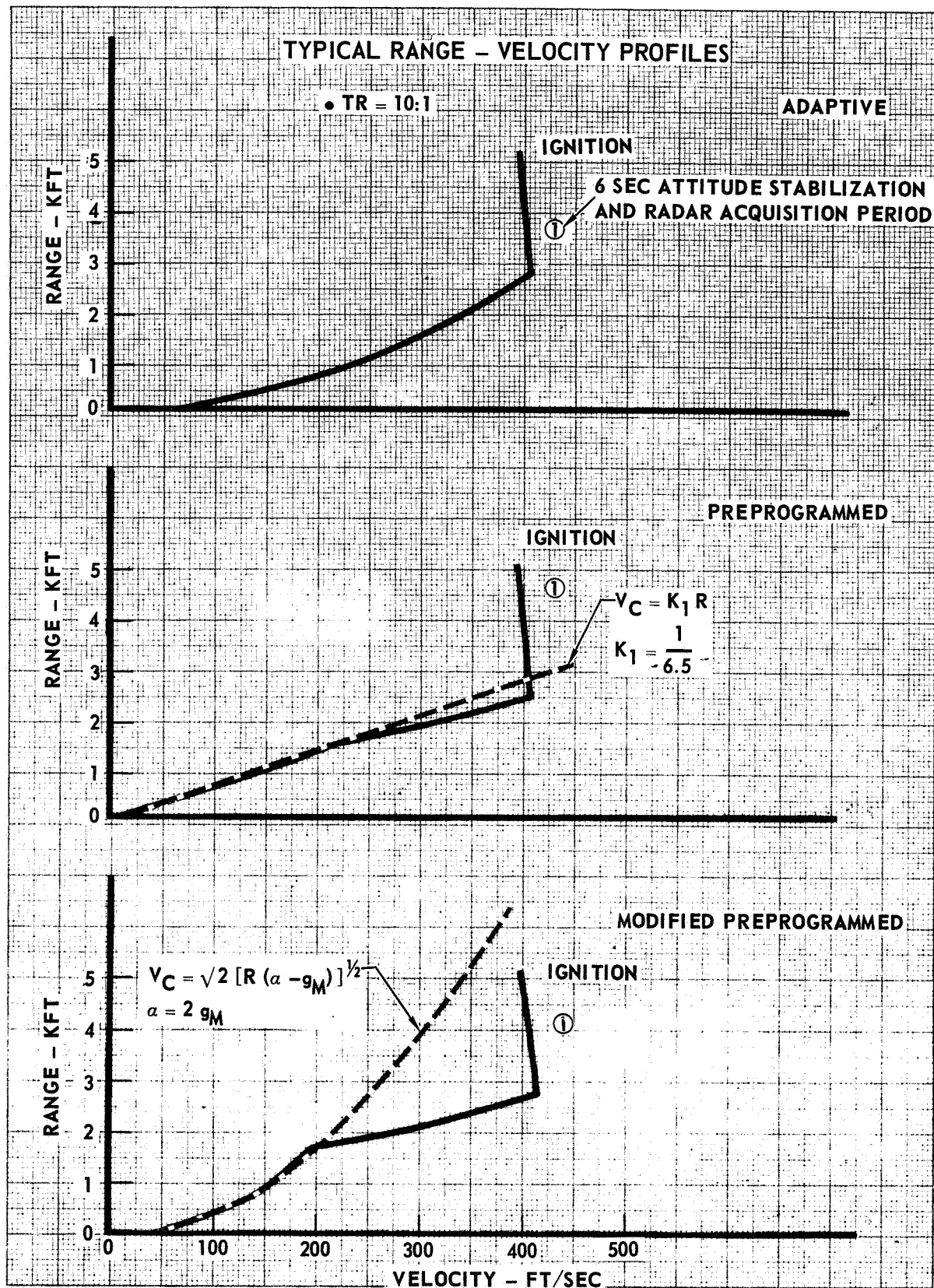


FIGURE 3.3.1-10

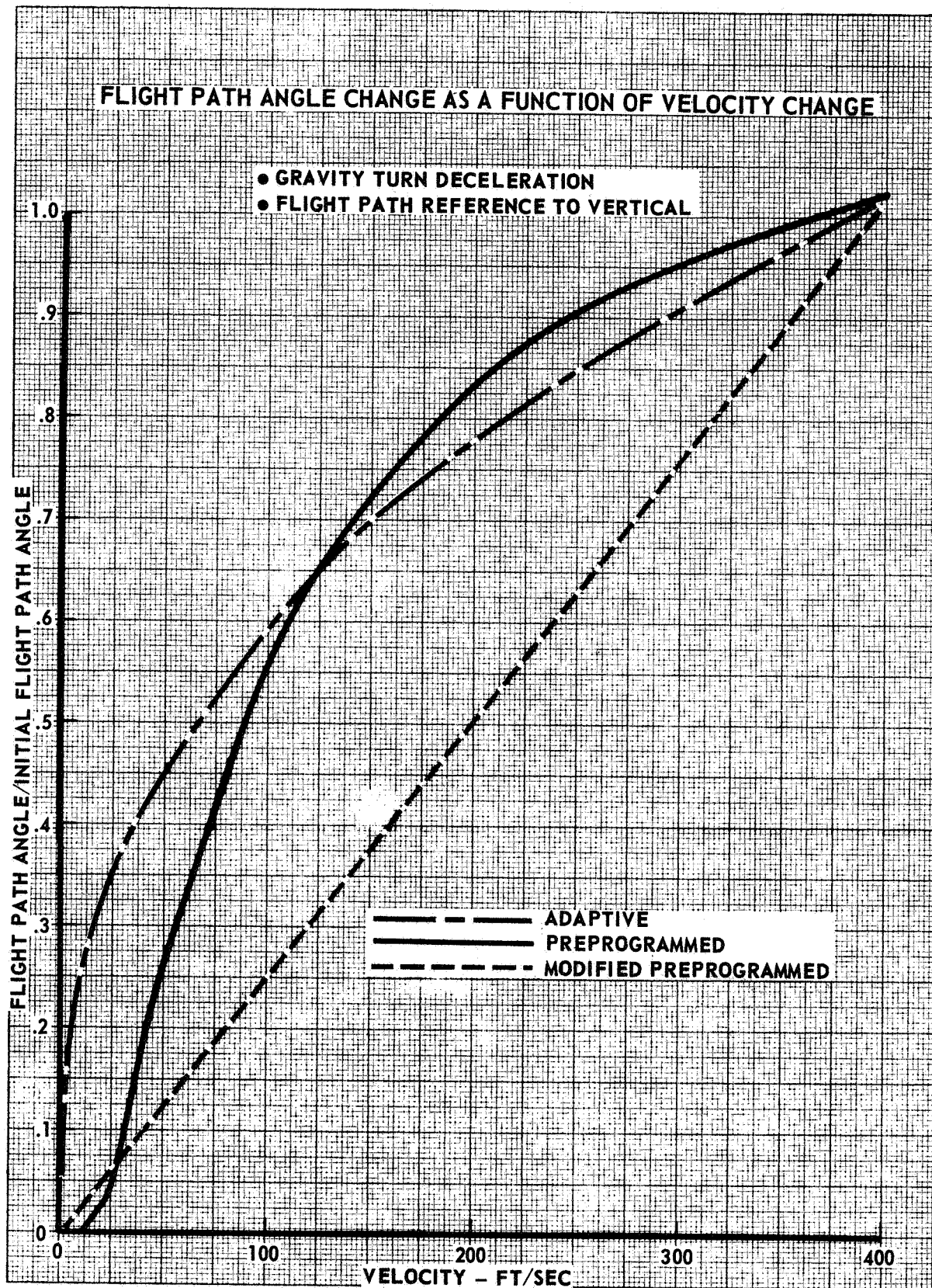


FIGURE 3.3.1-11

programmed or modified preprogrammed will restrict the turning rate within control system limitations.

Mechanization of any of the aforementioned concepts is not a consideration when a digital computer is used. However, with analog instrumentation, the preprogrammed (Surveyor type) technique would be considerably simpler.

3.3.1.2.6 Deceleration system performance: The overall deceleration system performance has been evaluated for a number of system combinations, to insure that the apparent best systems of Section 3.2 gave the best overall deceleration system. The combinations investigated were an all propulsion system and parachute/TPS systems for both optimum TPS throttle ratio and for aeroshell impact requirements.

All Propulsive Deceleration System - In addition to eliminating the aerodecelerator system, an all-propulsive system can employ engines with existing throttle ratios for attitude and vernier control by utilizing a solid propellant rocket to supply a major portion of the ΔV requirements. Hence the deceleration phase for an all-propulsive landing terminal system was investigated to 1) obtain a workable sequence of events, 2) define required equipment characteristics, and 3) obtain impulse requirements to size the propulsion system.

The major considerations which influence the required sequence are:

- a) Aeroshell interference with the radar line of sight
- b) Verticalization of the flight path angle for radar lock
- c) The information available for performing the sequence of events.
- d) Utilization of throttle ratios of less than 4.5:1.

Two methods were investigated, both of which employed solid propellant engines and low-throttle-ratio liquid vernier engines. Method A uses solid rockets to back the lander away from the aeroshell. With this system, the landing radar must furnish velocity and range data prior to ignition and hence have the capability to "see" through the aeroshell. In addition, since the aeroshell is always in front of the lander after separation, a delay period is required following the solid burn, to prevent the radar from locking onto the aeroshell.

Method B utilizes solid rockets on the aeroshell to push the shell away and to the side of the lander. This action will be triggered by an altimeter signal, and hence landing radar information is not required prior to separation. It is assumed that the lander will stay in front of the aeroshell because of the larger $m/C_D A$. (Additional analyses must be performed to verify this in the event the aeroshell is tumbling.) After the shell is clear of the lander, landing radar information is required to initiate and terminate the solid rocket firing and direction of burn. Radar information during the burn is not required.

o Method A - Figures 3.3.1-12 and -13 present the Method A range-velocity histories for the VM-7 and VM-10 atmospheres. The aeroshell trajectory is also included on the figures, to illustrate the sequence of the aeroshell-ground impact. In general, the sequence of events is similar for each atmosphere. However, the times and initiation points are different. The following presents the detail sequence and the manner in which the differences are established through preprogrammed curves in the on-board computer.

a) Ignition is based on the range-velocity curve a of Figure 3.3.1-14. Measured velocity from the landing radar is used to obtain a desired ignition range from the curve. At the desired range both the solid rocket and the liquid verniers are ignited. The acceleration command value is set such that the vernier thrust is approximately half of maximum. Hence, maximum attitude control moment is available.

b) The velocity change command (ΔV_c) is based on the measured velocity at ignition, as shown in curve b of Figure 3.3.1-14. The solid rocket burn is terminated when the measured (accelerometer) velocity change equals the command. At this time the velocity-range relationship of the vehicle must be within the capability of the throttleable vernier engines.

c) At ignition the lander is torqued to align the thrust axis along the inertial velocity vector. The torquing time (T_c) and direction are calculated from the angle of attack measurement ($\alpha = \tan^{-1} \frac{\text{Radar Lateral Velocity}}{\text{Radar Axial Velocity}}$) and based on a 10 deg/sec torquing rate.

RANGE - VELOCITY PROFILE

- VM-7 ATMOSPHERE
- 220 FT/SEC TAILWIND
- METHOD A

TIME	EVENT
0	SOLID AND VERNIER IGNITION, LANDER TORQUED TOWARD INERTIAL VELOCITY VECTOR
1.0	INERTIAL HOLD
5.2	TERMINATE SOLID ENGINE THRUST, LANDER TORQUED TOWARDS VERTICAL
9.0	INITIATE LATERAL VELOCITY CONTROL
12.4	INITIATE VERNIER PROGRAMMED DECELERATION
27.8	CONSTANT VELOCITY DESCENT PHASE

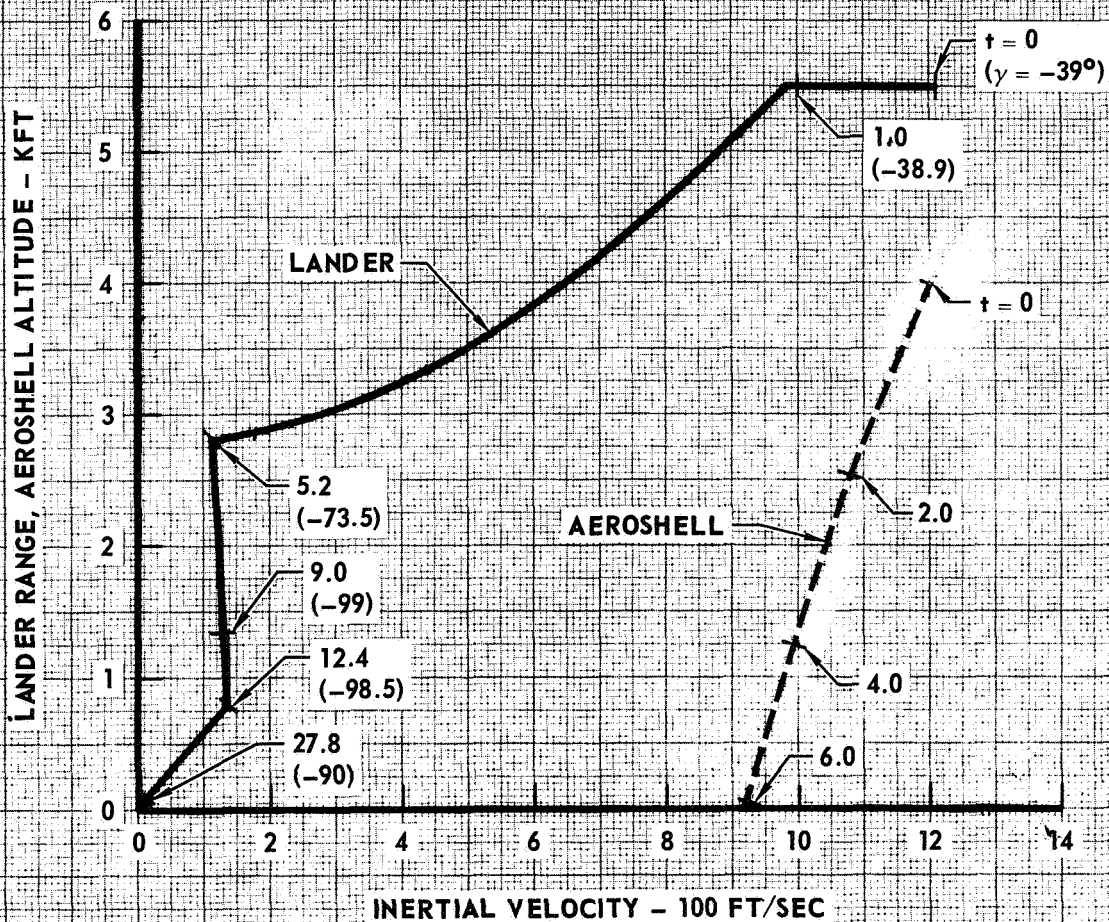


FIGURE 3.3.1-12

RANGE - VELOCITY PROFILE

- VM-10 ATMOSPHERE
- NO WIND
- METHOD A

TIME	EVENT
0	SOLID AND VERNIER IGNITION
2.0	TERMINATE SOLID ENGINE THRUST
9.0	INITIATE LATERAL VELOCITY CONTROL
11.4	INITIATE VERNIER PROGRAMMED DECELERATION
25.4	CONSTANT VELOCITY DESCENT PHASE

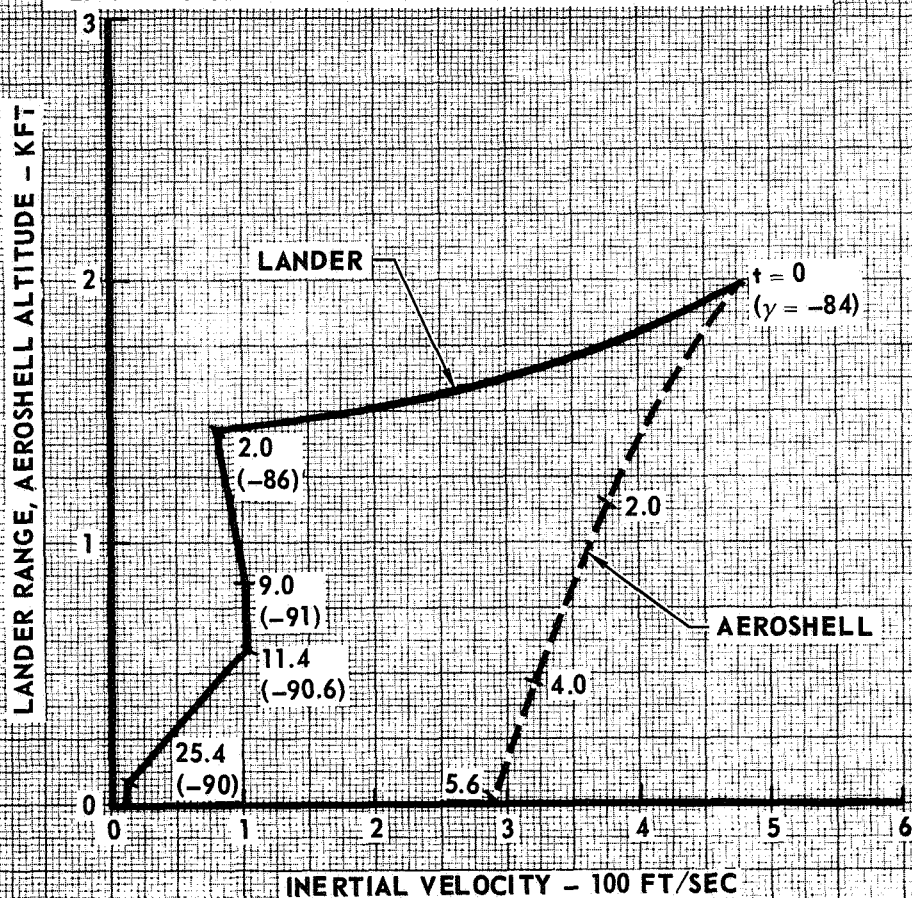


FIGURE 3.3.1-13

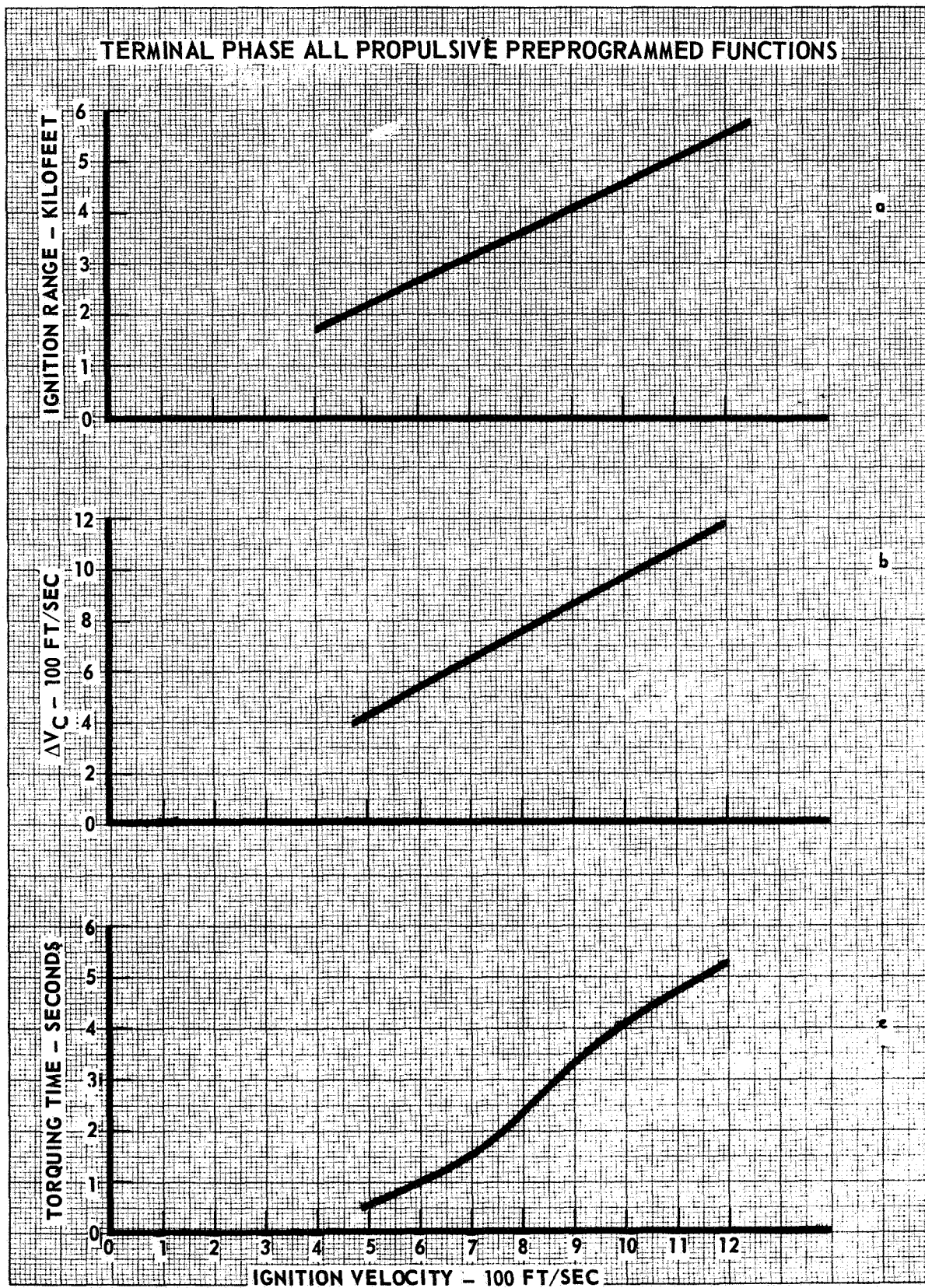


FIGURE 3.3.1-14

d) Inertial hold is initiated following the torquing period and is maintained for the remaining time of the solid rocket burn.

e) Following the solid rocket thrust termination, the lander is torqued to align the radar (thrust) axis near the vertical and the vernier engines are throttled to $.8g_M$. The torquing time is pre-calculated and equal to $-T_c$ plus the value from curve c, Figure 3.3.1-14. The times of curve c are based on the relationship of flight path angle to velocity for the odd numbered (high scale height) atmospheres. In a low scale height atmosphere, the radar axis will still be within 20 degrees of the vertical in some instances, which is well within the radar capability.

f) Following the lander re-alignment, inertial hold control is again engaged until lateral velocity control is initiated at a pre-set time from ignition. The time represents the required time for aeroshell impact plus three seconds for radar acquisition. The time of aeroshell impact will be a constant 6 seconds for each atmosphere because of the ignition range variations with velocity.

g) Intersection of the preprogrammed descent profile and subsequent deceleration with the vernier engines to the constant velocity descent phase completes the control sequence.

For the example range-velocity histories presented in Figures 3.3.1-12 and -13, an approximate thrust to weight (T/W) ratio of 6 was used for the solid rocket. A near spherical motor case can be employed for this T/W with a 1000 pound lander weight. Although extensive parametrics to optimize the T/W were not performed, the value of 6 was a favorable compromise between lower values, which increase impulse requirements due to increased ignition altitudes, and higher values, which create design problems due to required elongation of the motor case.

A cursory weight comparison was made between the Method A all-propulsive terminal deceleration system and the combination aerodecelerator plus TPS of Design Concepts II and III. Increases for the all propulsive system of 45 and 50 pounds, respectively, were required. Possible additional increases in the landing radar weight would be required for a system to "see" through the aeroshell prior to separation.

o Method B - Separation by utilizing auxiliary solid propellant rockets mounted on the aeroshell will be initiated by an altimeter signal. Assuming the lander propulsion to have the same characteristics as in Method A, separation must occur at 10 000 ft for the VM-7 atmosphere, to allow sufficient time for aeroshell clearance and radar acquisition. Lander solid rocket ignition will again be initiated based on the range-velocity of curve a Figure 3.3.1-14. In VM-7, 8 seconds will be needed between aeroshell separation and this ignition and 16 seconds in VM-10. The complete sequence would be as follows:

- a) Aeroshell rocket ignition is at 10 000 ft altitude
- b) Liquid propellant verniers are also ignited at $.8g_M$ level to provide attitude control for the lander.
- c) A three second delay is needed for aeroshell clearance.
- d) Landing radar acquisition is initiated following aeroshell clearance period. In the event all velocity beams do not acquire "lock-on" because of shallow incidence angles, the lander attitude is torqued towards locked-on beams at 10 deg/sec rate. At lock-on, inertial hold is again initiated.
- e) Lander solid rocket ignition is based on the range-velocity of curve a Figure 3.3.1-14. The radar acquisition period (above) will be a minimum of 5 seconds at 10 000 ft separation altitude, using the ignition curve of Figure 3.3.1-14.
- f) The remaining sequence is identical to Method A, except lateral velocity control is initiated 3 seconds following lander realignment since the additional 6 second delay is not required for aeroshell impact.

Additional system weight is required for the aeroshell separation rockets and added impulse in the liquid vernier system (for the period between aeroshell separation and lander rocket ignition) as compared to Method A. This amounts to separation rocket weights of 19 and 22.6 pounds and 29 and 33 pounds added to the liquid system for Design Concepts II and III, respectively. However,

since the landing radar is not required to "see" through the aeroshell, a minimum modification LM radar can be employed.

A summary comparison between the all propulsive terminal deceleration systems and the optimum aerodecelerator/TPS of Design Concepts II and III are shown in Table 3.3.1-3. A severe weight increase is required for the all-propulsive system. The solid propellant engine weighs considerably more than the aerodecelerator and the weight reduction realized in the liquid vernier system because of the lower ΔV requirement is not enough to counterbalance this increase. With proper choice of aerodecelerator size, low throttle ratio engines can also be utilized in the combination system, hence no advantage is gained with the all-propulsive system. Also, release of the aeroshell from the lander-aerodecelerator configuration is believed to be far "cleaner". Hence, other than the requirement for aerodecelerator development, which is already underway with the PEPP program, the aerodecelerator - TPS terminal deceleration system is superior to the all-propulsive system.

Parachute/Terminal Propulsion System (Optimum Throttle Ratio) - Terminal propulsion systems (Section 3.2.7) sized for the optimum throttle ratio were combined with parachute systems deployed at 23 000 ft and sized to give equilibrium velocity at the optimum TPS ignition altitude. This trade study included both weight and cost factors and was made for the monopropellant and bipropellant systems using both the preprogrammed and the modified preprogrammed control modes.

Deceleration system weights, as a function of the lander weight and the equilibrium velocity, are given in Figures 3.3.1-15 and -16. (The 1000 lb lander weight, which is representative of the larger discrete design points, will be used for all subsequent comparisons.) The bipropellant TPS combination gives the lighter decelerator system weight for both control modes. The minimum weight points are shown in Table 3.3.1-4. Thus, on a weight basis the modified preprogrammed, bipropellant TPS/parachute is the best system.

However, the cost factor must also be considered. A normalized cost comparison is given in Figure 3.3.1-17. This indicates that the monopropellant systems are much lower cost (approximately 40%) for a small weight penalty

TABLE 3.3.1-3
TERMINAL DECELERATION SYSTEM COMPARISON

COMPARISON FACTORS	AERODECELERATOR + TPS	ALL PROPULSIVE METHOD A	ALL PROPULSIVE METHOD B
AERODECELERATOR DEVELOPMENT	YES	NO	NO
SOLID ENGINE	NO	YES	YES
THROTTLE RATIO	4.0 (II)	3.5 → 4.5	3.5 → 4.5
LIQUID SYSTEM	MONO	MONO	MONO
RADAR DEVELOPMENT	LM WITH MINOR MOD	LM WITH MAJOR MOD TO "SEE" THROUGH AEROSHELL	LM WITH MINOR MODS
ADDITIONAL DEVELOPMENTS		FIRE IN HOLE SEPARATION	AEROSHELL PULL AWAY SEPARATION
WEIGHT (TERMINAL) DECELERATION SYSTEM)	BASELINE	+ 45 (II)	+ 93 (II)
	BASELINE	+ 50 (III)	+ 105.6 (III)

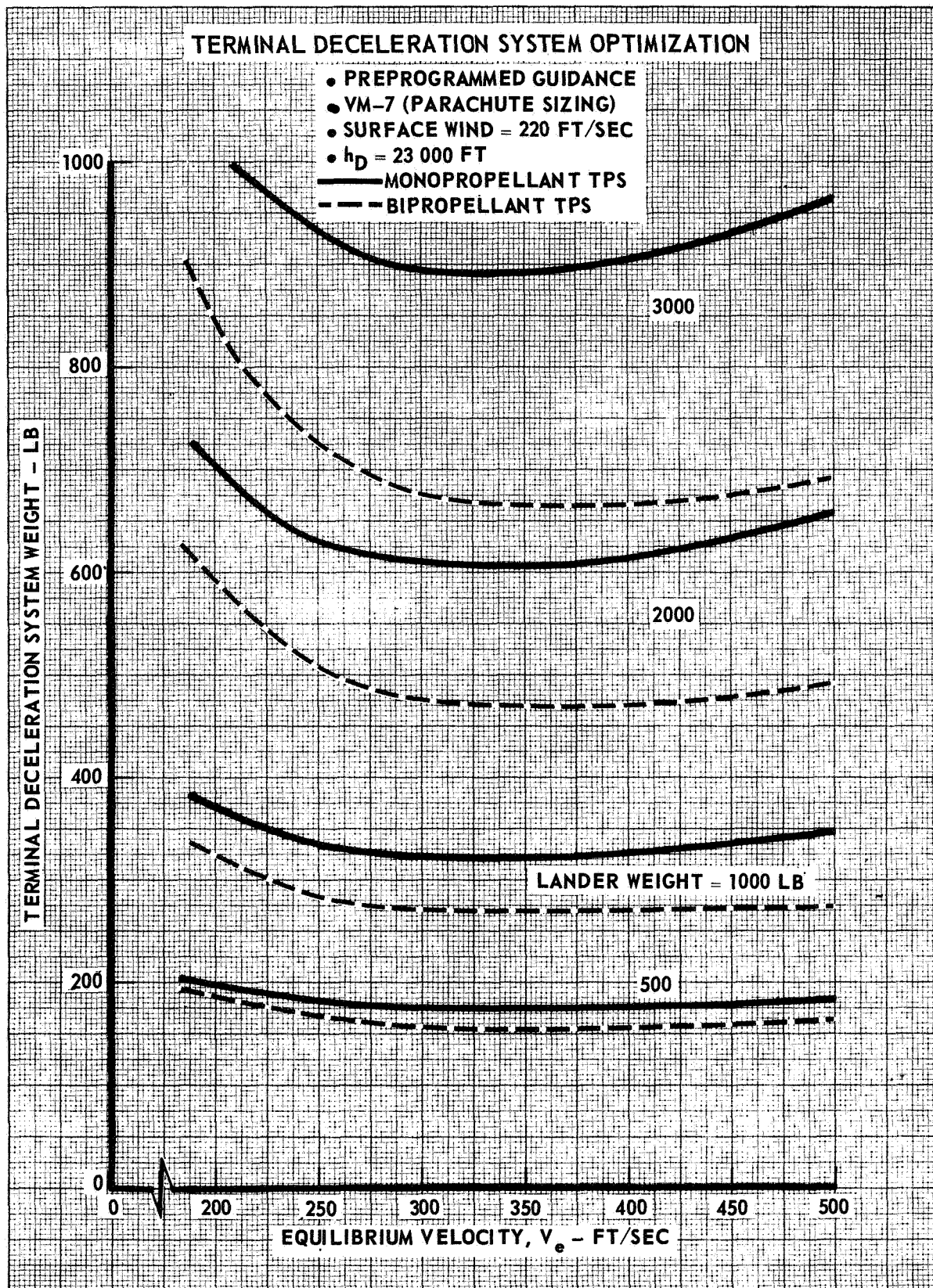


FIGURE 3.3.1-15

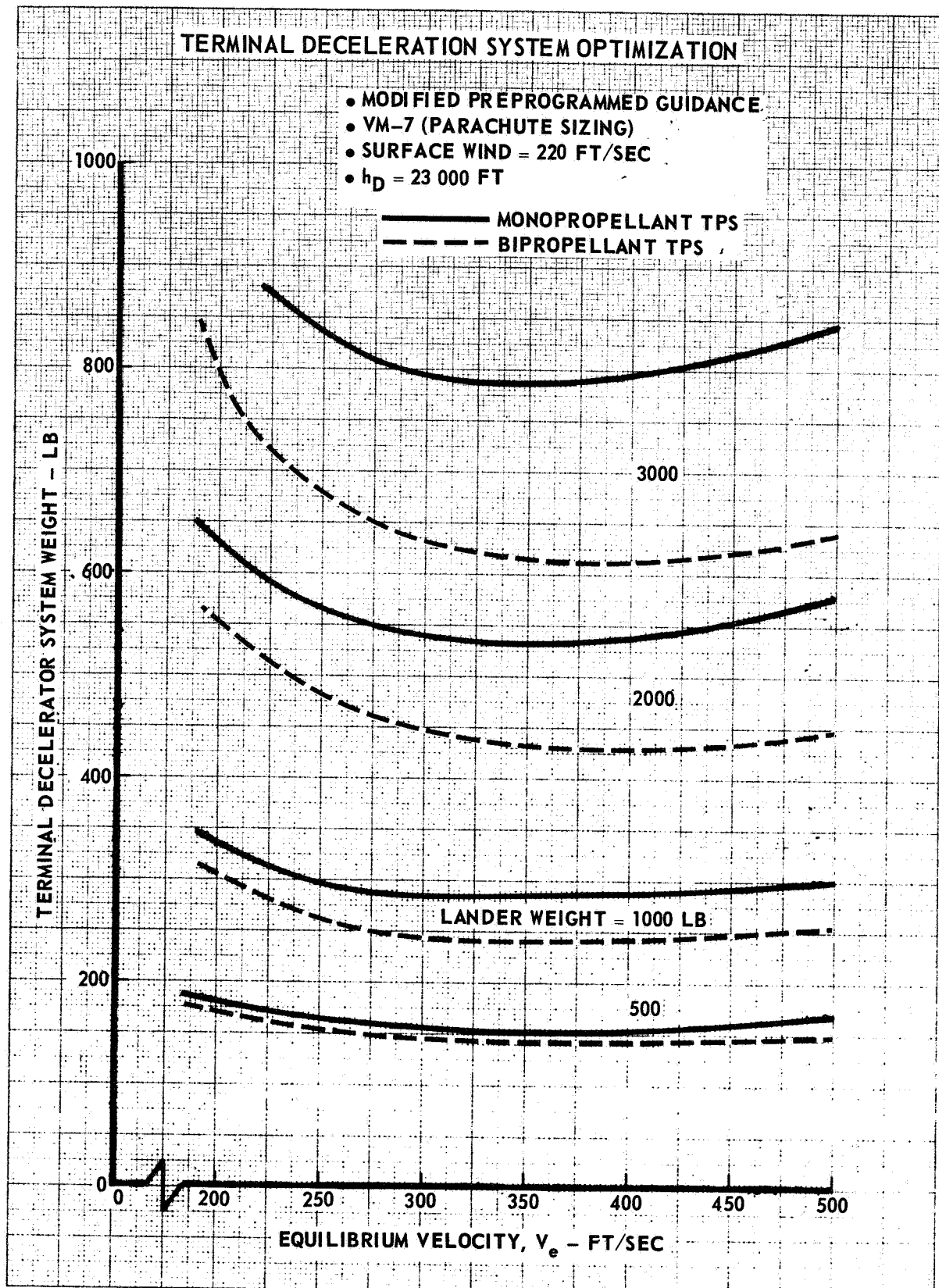


FIGURE 3.3.1-16

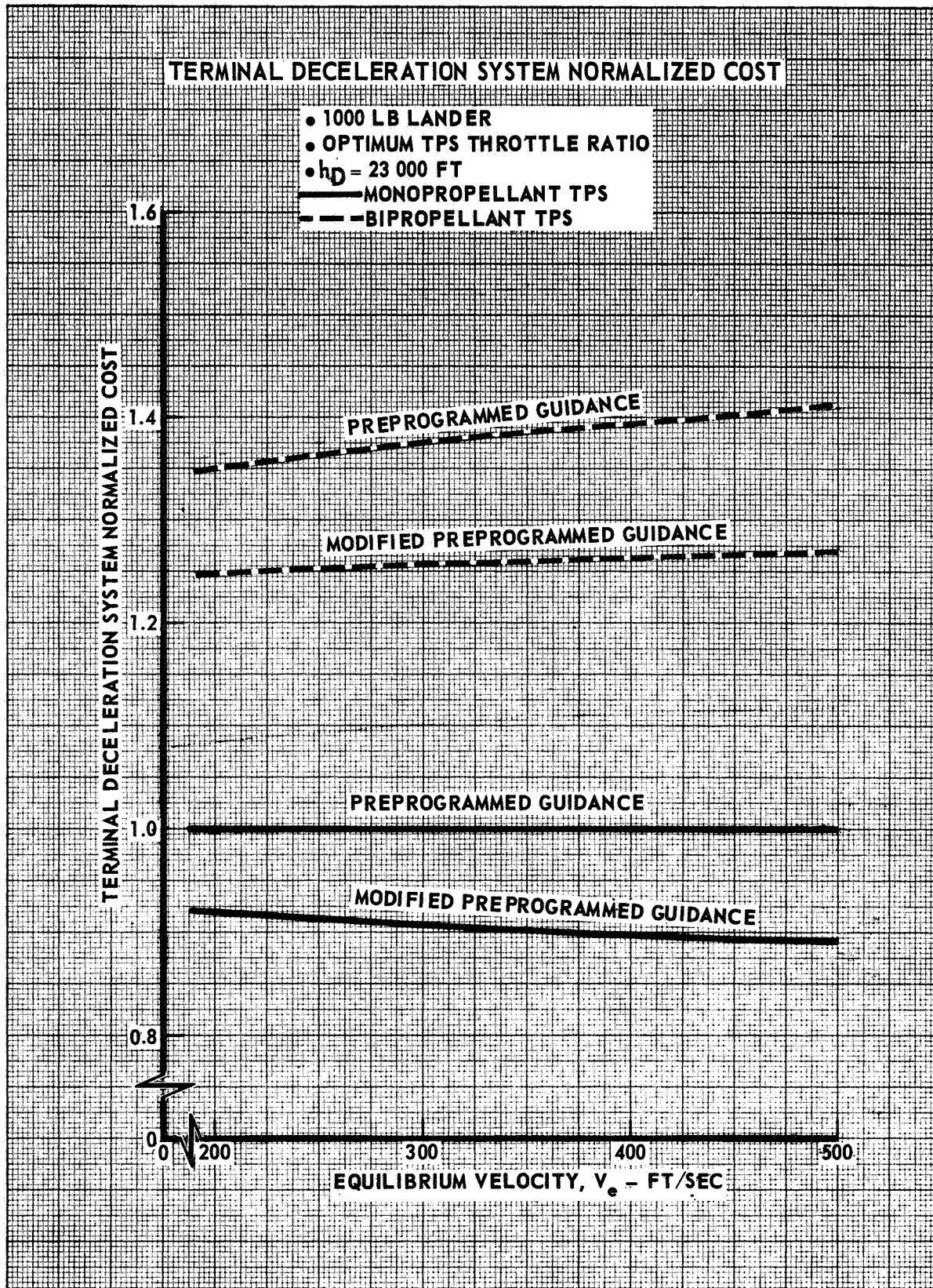


FIGURE 3.3.1-17

TABLE 3.3.1-4 TERMINAL DECELERATION SYSTEM WEIGHT COMPARISON

TPS TYPE	MODIFIED	PREPROGRAMMED	COMPARISON BY CONTROL MODE
MONO	285 LB	325 LB	14%
BI	241 LB	270 LB	12%
EQUILIBRIUM VELOCITY	325 FT/SEC	300 FT/SEC	---
COMPARISON BY TPS TYPE	15.3%	17%	---

(approximately 15%). The modified preprogrammed guidance mode has small weight (14%) and cost (10%) advantages over the preprogrammed guidance mode.

Parachute/Terminal Propulsion System with Aeroshell Impact Requirement -

The above study defines the optimum (weight and cost) decelerator system; however, the aeroshell effect on the landing radar must be considered. Presently the landing radar systems do not have the capability to lockout the aeroshell. Therefore, the aeroshell must impact on the Mars surface prior to landing radar activation.

In addition, attention was given to means of reducing the aeroshell diameter to eliminate the booster shroud hammer head. The diameter can be reduced by increasing the value of $m/C_D A$. This can be accomplished by reducing the deployment altitude (to maintain $M_D \leq 2.0$). The deployment altitudes of 23 000 (Phase B study), 18 590 ft ($m/C_D A = 0.3$) and 14 430 ft ($m/C_D A = 0.35$) were chosen and included in this study. As a result of the previous optimization study, only the monopropellant TPS was considered.

The aerodecelerator data of Section 3.3.1.2.4 was used to size the parachute for a range of equilibrium velocities and thereby specify the equilibrium altitude and the lander altitude at aeroshell impact for each of the three deployment altitudes. Monopropellant terminal propulsion systems were sized to remove the total ΔV (equilibrium velocity with surface wind of 220 ft/sec) with ignition at the lander altitude when the aeroshell impacts.

The resultant terminal deceleration system weights and normalized costs for a 1000 lb lander are given in Figure 3.3.1-18. The lightest weight and lower cost terminal deceleration systems occur for the deployment altitude of 23 000 ft. The system weight is essentially the same, but the cost is about 7% higher for deployment at 18 590 feet. The modified preprogrammed guidance mode results in lighter weight and lower cost terminal deceleration systems than the preprogrammed guidance mode. The optimum point for 23 000 ft deployment altitude is approximately $V_E = 290$ ft/sec and $h_{ign} = 6500$ ft. The optimum point for 18 590 ft deployment altitude is approximately $V_E = 270$ ft/sec and $h_{ign} = 5500$ ft.

The combination of a modified preprogrammed guidance mode, monopropellant TPS, and a parachute was selected for the capsule parametrics of Section 3.3.1.3, (Figures 3.3.1-22, -23, -27, and -28). The deployment altitude parametric, Figure 3.3.1-29, indicates that the lower deployment altitudes result in lower capsule weights. Deployment at 23 000 ft gave slightly lower capsule cost. The lowest deployment altitude (14 430 ft) resulted in an aeroshell diameter of 9.3 ft, which is not small enough to prevent hammer heading. Thus to obtain maximum benefit from the computer program and the other capsule parametric data the deployment altitude of 23 000 ft and TPS ignition altitude of 6500 ft were chosen.

Follow-on point designs with new or modified environments should be analyzed for various deployment altitudes.

3.3.1.3 Delivery system capability - The value of the delivery system is largely determined by the weight of surface payload it is capable of landing. Therefore a method of calculating this capability parametrically has been developed and programmed for automatic computation. Data from the systems studies was included where appropriate.

The surface payload capability is determined as a function of the capsule weight, and aeroshell diameter for selected combinations of the capsule systems. The surface payload systems and weights are allocated using another program, discussed in Section 3.3.2. The capsule weight parametric developed in this section includes the adapter and adapter mounted equipment. (Note that the

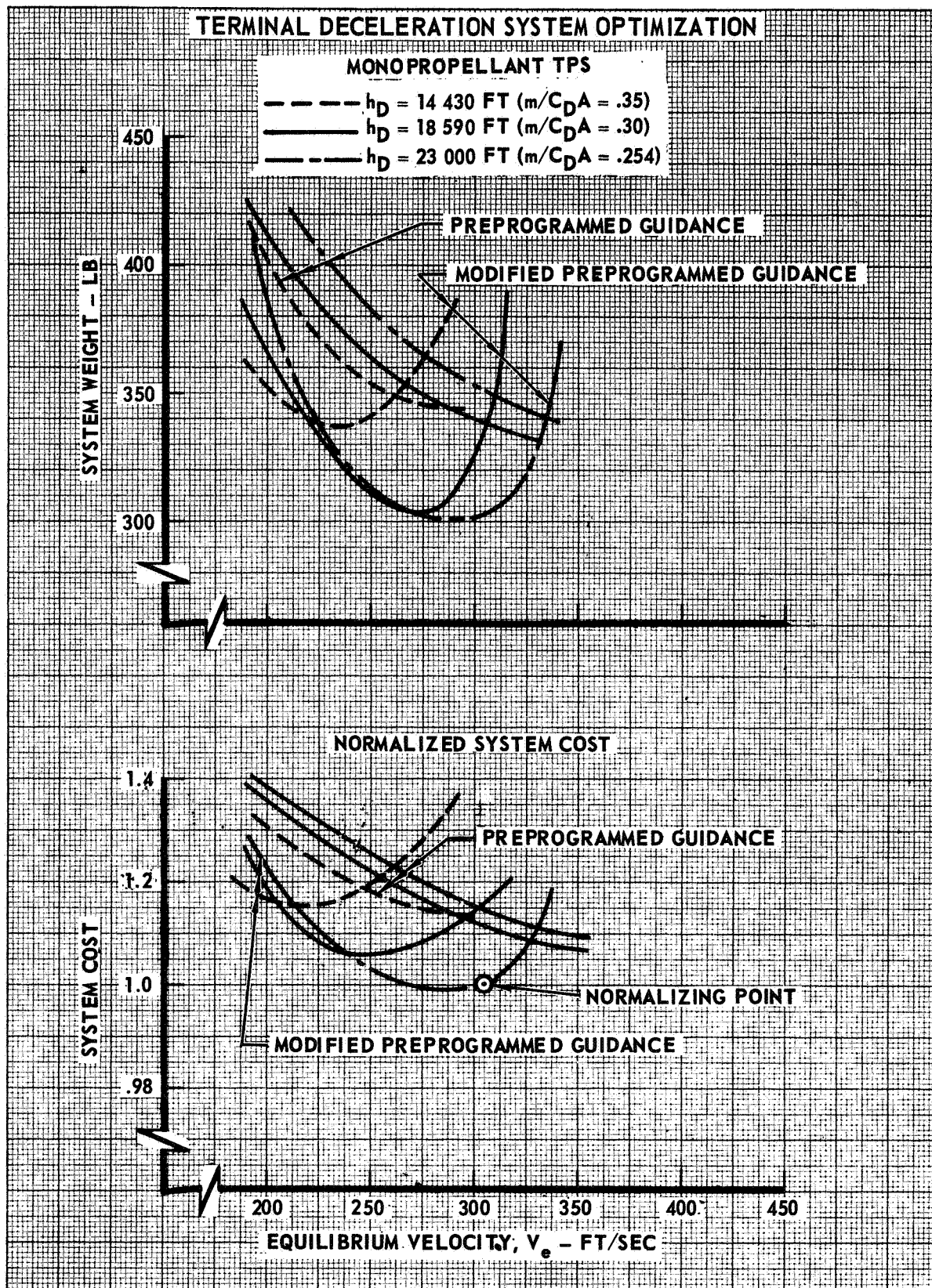


FIGURE 3.3.1-18

capsule support equipment mounted on the orbiter is not included. This equipment consists mainly of telemetry equipment and the total weight is 38.9 lb.)

The Mars planetary entry capsule weight program provides the capability to calculate mission history group weight breakdowns using the calculation order presented in Table 3.3.1-5.

The capsule surface payload capability was determined for the total entry corridor, total range of Mars atmospheres, winds up to 220 fps, surface slopes up to 34° , and a deployment altitude (h_D) of 23 000 ft. This data is presented in Figures 3.3.1-19 through -28. Tables 3.3.1-6 and -7 present the criteria used for these parametrics. The surface payload parametrics were done in parallel with the other capsule parametrics. Thus, the first data presented is for preprogrammed guidance mode with the TPS ignition at 4900 ft. The later parametrics are for the conditions selected in Section 3.3.1.2.6, modified preprogrammed guidance mode with TPS ignition at 6500 ft.

Surface payload capability versus capsule weight for various aeroshell diameters is shown in Figures 3.3.1-19 and 3.3.1-24. These figures are then cross plotted to illustrate the aeroshell diameter requirements (see Figures 3.3.1-20 and 3.3.1-25). Mission profile weight breakdowns are shown for capsules which deploy a parachute at Mach 2 (see Figures 3.3.1-21, -22, -26 and -27). Capsules which deploy an attached inflatable decelerator at the optimum Mach number (approximately Mach 3.5) are also shown (see Figures 3.3.1-23 and -28).

Figure 3.3.1-29 shows the effect of changing deployment altitude on aeroshell diameter and capsule weight, for the preprogrammed guidance mode with TPS ignition altitudes below 5000 ft. The capsule aeroshell diameter relationship for the optimum (with aeroshell impact requirement) terminal deceleration system is also shown. The aeroshell diameter reduction is approximately the same in both cases; however, the capsule weight penalty for deployment is only about 30 lb for the optimum system. The total capsule (with optimum terminal deceleration system) cost is slightly lower for deployment at 23 000 ft. The parachute is sized to provide a minimum separation distance equal to the terminal propulsion ignition altitude less the distance traveled during the 3 second delay (landing radar activation). The criteria for this parametric

TABLE 3.3.1-5
MARS PLANETARY ENTRY CAPSULE WEIGHT PROGRAM
CALCULATION ORDER

- A. ALL DATA IS READ INTO A CRD MATRIX
- B. ITEMS NOT DEORBITED ARE CALCULATED.
 1. AN INITIAL ASSUMPTION ON SIZE (GEOMETRY) IS MADE FROM THE INPUTS (LATER CALCULATED AND COMPARED)
 2. DESCRIPTIVE TERMS ARE ASSIGNED TO THE INPUT MATRIX VALUES.
 3. PRE-DEORBIT WEIGHT IS ESTIMATED (INPUT, LATER TO BE CALCULATED AND COMPARED).
 4. INPUT VALUES OF CONSTANTS, PAYLOAD EQUIPMENT, ETC. ARE ASSIGNED LABELS.
 5. STERILIZATION CANISTER WEIGHT IS CALCULATED IN A SEPARATE SUBROUTINE.
 6. INTERNAL STRUCTURE JETTISONED WITH THE STERILIZATION CANISTER IS CALCULATED.
 7. PAYLOAD LEFT WITH THE STERILIZATION CANISTER IS CALCULATED.
 8. PART OF TEMPERATURE CONTROL SYSTEM LEFT WITH THE STERILIZATION CANISTER IS CALCULATED.
 9. ATTITUDE CONTROL PROPELLANT USED IN ORBIT IS CALCULATED.
 10. ADAPTER WEIGHT IS CALCULATED.
 11. SUMMATION OF DECREMENTS TO PREDEORBIT CONDITION.
- C. PRE-DEORBIT WEIGHT IS CALCULATED (INPUT FLIGHT CAPSULE MINUS SUM OF DECREMENTS TO PRE-DEORBIT CONDITION).
 1. EARLIER ESTIMATED PRE-DEORBIT WEIGHT (B-3) IS CHECKED AGAINST THE CALCULATED PRE-DEORBIT WEIGHT. FOR A DIFFERENCE GREATER THAN 10 LBS., THE NEW VALUE IS ASSIGNED AND THE PROGRAM RETURNS TO STEP B-8.
 2. FOR A NEGATIVE OR ZERO CALCULATED PRE-DEORBIT WEIGHT THE PROGRAM GOES TO STEP R.
- D. ITEMS EXPENDED DURING DEORBIT ARE CALCULATED.
 1. ATTITUDE CONTROL PROPELLANT WEIGHT USED DURING DEORBIT IS CALCULATED.
 2. DEORBIT PROPULSION PROPELLANT USED DURING DEORBIT IS CALCULATED.
 3. DECREMENTS DURING DEORBIT ARE SUMMED.
- E. POST-DEORBIT WEIGHT IS CALCULATED (DEORBIT DECREMENTS ARE SUBTRACTED FROM PRE-DEORBIT WEIGHT).
- F. ITEMS EXPENDED FROM POST-DEORBIT TO ENTRY ARE CALCULATED (ITEMS NOT ENTERED).
 1. INPUT INDICATOR DETERMINES WHETHER DEORBIT MOTOR IS JETTISONED OR RETAINED.
 2. IF THE DEORBIT IS JETTISONED, THE WEIGHT OF THE JETTISONED PART IS CALCULATED.
 3. DECREMENTS FROM POST-DEORBIT TO ENTRY ARE SUMMED.
- G. ENTRY WEIGHT IS CALCULATED.
 1. FOR A NEGATIVE OR ZERO CALCULATED ENTRY WEIGHT THE PROGRAM GOES TO STEP R.
- H. ITEMS EXPENDED DURING ENTRY ARE CALCULATED.
 1. ATTITUDE CONTROL PROPELLANT USED DURING ENTRY IS CALCULATED.
 2. GUIDANCE AND CONTROL WEIGHT LEFT WITH AEROSHELL IS INPUT.
 3. VEHICLE MAXIMUM BALLISTIC AERODYNAMIC LOAD (Q) IS CALCULATED BASED ON $M/C_D A$.
 4. VEHICLE NOSE CAP IS CALCULATED BASED ON Q AND AEROSHELL BASE DIAMETER.
 5. AEROSHELL FRUSTRUM WEIGHT IS CALCULATED.
 6. AEROSHELL RING WEIGHT IS CALCULATED.
 7. AN INPUT INDICATOR DETERMINES WHETHER OR NOT PROVISIONS FOR A LANDER ARE REQUIRED.
 8. LANDER SUPPORT STRUCTURE IS CALCULATED IF THERE IS A LANDER.
 9. AEROSHELL WEIGHT IS CALCULATED (AS THE SUM OF PIECES PREVIOUSLY CALCULATED).
 10. INTERNAL STRUCTURE LEFT WITH THE AEROSHELL IS CALCULATED.
 11. THE HEAT SHIELD WEIGHT IS CALCULATED.
 12. THE PAYLOAD LEFT WITH THE AEROSHELL IS INPUT.
 13. THE GUIDANCE AND CONTROL WEIGHT LEFT WITH THE AEROSHELL IS INPUT.
 14. THE WEIGHT OF THE LANDED ITEMS IS ESTIMATED (LATER CALCULATED AND COMPARED).
 15. THE WEIGHT OF THE THERMAL PROTECTION CURTAIN, INSULATION, HEATERS, AND COATINGS IS CALCULATED.
 16. THE AERO-DECELERATOR DEPLOYMENT LOAD (QD) IS CALCULATED FROM ONE OF THREE EQUATIONS, DEPENDING ON THE $M C_D A$ RANGE, OR IS INPUT, DEPENDING ON AN INPUT INDICATOR.
 17. THE ATMOSPHERIC DENSITY AT THE AERODECELERATOR DEPLOYMENT IS CALCULATED FROM INPUT DATA.
 18. THE SPEED OF SOUND AND MACH NUMBER AT THE AERODECELERATOR DEPLOYMENT ALTITUDE ARE CALCULATED.
 19. BASED ON THE MACH NUMBER, A PARACHUTE OR ATTACHED INFLATABLE DECELERATOR (AIDS) DRAG COEFFICIENT IS CHOSEN FROM THE INPUT DATA.
 20. THE ADIABATIC WALL TEMPERATURE IS CALCULATED FOR DEPLOYMENT CONDITIONS.
 21. THE FABRIC STRENGTH TO WEIGHT RATIO FOR THE AERODECELERATOR, BASED ON THE ADIABATIC WALL TEMPERATURE, IS CALCULATED.
 22. THE DECELERATOR DIAMETER FOR DIFFERENTIAL DRAG, FOR THE OPTIMUM TERMINAL VELOCITY FOR TERMINAL PROPULSION SYSTEM (TPS) INITIATION, AND FOR AEROSHELL SEPARATION DISTANCE EQUAL TO TPS INITIATION ALTITUDE IS CALCULATED AND THE LARGER OF THREE IS SELECTED. TERMINAL VELOCITY FOR THE SELECTED DIAMETER IS CALCULATED.
 23. HORIZONTAL WIND VELOCITY AND TERMINAL DECELERATOR VELOCITY ARE VECTORALLY SUMMED.
 24. THE PROJECTED AREAS OF THE AIDS AND OF THE AEROSHELL ARE CALCULATED.
 25. A PARACHUTE OR AIDS IS SELECTED BASED ON THE MACH NUMBER.
 26. IF AN AIDS IS SELECTED, THE AIDS WEIGHT IS CALCULATED.
 27. IF A PARACHUTE IS SELECTED, THE PARACHUTE AND EQUIPMENT WEIGHT IS CALCULATED.
 28. THE DECREMENTS FROM ENTRY TO TERMINAL DESCENT ARE SUMMED.
- I. THE TERMINAL DESCENT WEIGHT IS CALCULATED.
 1. FOR A NEGATIVE OR ZERO TERMINAL DESCENT WEIGHT, THE PROGRAM GOES TO STEP R.

TABLE 3.3.1-5
MARS PLANETARY ENTRY CAPSULE WEIGHT PROGRAM
CALCULATION ORDER (Continued)

- J. ITEMS EXPENDED FROM TERMINAL DESCENT TO TOUCH-DOWN ARE CALCULATED.
 - 1. VEHICLE GEOMETRY IS CALCULATED FROM WEIGHTS AND PREVIOUSLY INPUT OR DERIVED DIMENSIONS.
 - 2. PRE-PROGRAMMED OR ACTIVE GUIDANCE IS SELECTED BASED ON MINIMUM IMPULSE REQUIREMENT, (A BIAS TOWARD ONE SYSTEM MAY BE INPUT.) OR AN INDICATOR MAY BE USED TO CALL INPUT IMPULSE AND THROTTLE RATIO INFORMATION.
 - 3. IMPULSE TO MASS RATIO AND OPTIMUM THROTTLE RATIO ARE SELECTED BASED ON THE HORIZONTAL VELOCITY AND THE PARACHUTE (OR AIDS) TERMINAL VELOCITY.
 - 4. BI-PROPELLANT AND MONO-PROPELLANT TERMINAL PROPULSION SYSTEM WEIGHTS ARE CALCULATED IN TWO SEPARATE SUBROUTINES, AND THE LIGHTER ONE IS SELECTED (A BIAS TOWARD ONE SYSTEM MAY BE INPUT).
 - 5. THE DECREMENTS TO TOUCHDOWN WEIGHT ARE SUMMED.
- K. THE TOUCHDOWN WEIGHT IS CALCULATED.
- L. THE WEIGHTS OF THE LANDED ITEMS ARE CALCULATED.
 - 1. IF THE TOUCHDOWN WEIGHT IS POSITIVE, THE IMPACT SYSTEM IS CALCULATED.
 - 2. THE IMPACT SYSTEM WEIGHT CALCULATION INCLUDES A GEOMETRY, C.G. AND INERTIA CALCULATION WHICH ARE USED TO DETERMINE MINIMUM WEIGHT AND DIAMETER WITHIN THE STABILITY LIMITS.
 - 3. THE TEMPERATURE CONTROL SYSTEM WEIGHT LANDED IS CALCULATED.
 - 4. THE LANDED ITEMS WEIGHTS ARE SUMMED.
 - 5. THE LANDED ITEMS WEIGHT IS CHECKED AGAINST THE VALUE ASSUMED IN H-14. IF THE DIFFERENCE IS GREATER THAN TEN POUNDS THE PROGRAM RETURNS TO J.
- M. THE REMAINING WEIGHT CAPABILITY, POSITIVE OR NEGATIVE, IS CALCULATED (THE DIFFERENCE BETWEEN THE INITIAL ASSUMED VEHICLE WEIGHT AND THE SUM OF THE WEIGHTS OF THE PIECES).
- N. EACH SYSTEM IS CALCULATED (THE SUM OF THE DECREMENTS PLUS THE FINAL WEIGHT).
- O. THE MATRIX NEEDED FOR PRINT OUT IS GENERATED.
- P. THE INERTIA CALCULATION AND ITERATION IS CONSIDERED.
 - 1. THE NUMBER OF ITERATIONS IS CHECKED AGAINST AN INPUT VALUE TO SEE IF SUFFICIENT ITERATIONS HAVE BEEN COMPLETED. IF THEY HAVE, THE PROGRAM GOES TO THE PRINT-OUT STEP (Q). IF THEY HAVE NOT, THE PROGRAM GOES TO THE NEXT STEP.
 - 2. THE VEHICLE INERTIA IS CALCULATED IN A SEPARATE SUBROUTINE.
 - 3. THE PROGRAM IS RETURNED TO STEP B-4.
- Q. RESULTS ARE PRINTED-OUT.
- R. PROGRAM STOPS OR RETURNS TO FIRST STEP (A) FOR NEW CASE, IF A PARAMETRIC IS BEING RUN.

TABLE 3.3.1-6
CRITERIA FOR FIGURES 3.3.1-19 THROUGH 3.3.1-23

ENTRY ALTITUDE			800 000 FT
ENTRY VELOCITY (V_e)	FROM 13 000	TO	15 200 FT/SEC
ENTRY FLIGHT PATH ANGLE (γ_e)	FROM GRAZE	TO	-20°
ATMOSPHERES			
SCALE HEIGHT (H_ρ)	FROM 5	TO	14 KM
SURFACE PRESSURE (P_0)	FROM 5	TO	20 MB
DEORBIT VELOCITY INCREMENT			450 FT/SEC
FIXED EQUIPMENT WEIGHT			
GUIDANCE & CONTROL & RADARS - ON AEROSHELL			2.9 LB
GUIDANCE & CONTROL & RADARS - LANDED			112.9 LB
SCIENCE, TCM, ELECTRICAL & MISC - TOTAL			168.0 LB
SCIENCE, TCM, ELECTRICAL & MISC - ON ADAPTER			15.0 LB
SCIENCE, TCM, ELECTRICAL & MISC - ON AEROSHELL			15.5 LB
SCIENCE, TCM, ELECTRICAL & MISC - LANDER			137.5 LB
DRAG COEFFICIENTS (C_D)			
AEROSHELL - SUPERSONIC C_D			1.51
AEROSHELL - SUBSONIC C_D			.78
ATTACHED INFLATABLE DECELERATOR - SUPERSONIC C_D			1.00
ATTACHED INFLATABLE DECELERATOR - SUBSONIC C_D			.80
PARACHUTE - SUPERSONIC C_D (BASED ON TOTAL SURFACE AREA)			.60
PARACHUTE - SUBSONIC C_D (BASED ON TOTAL SURFACE AREA)			.60
MINIMUM ACCELERATION OF AEROSHELL RELATIVE TO LANDER			.6 g
PARACHUTE DEPLOYMENT MACH NO. LIMIT			2
DEPLOYABLE DECELERATOR DEPLOYMENT ALTITUDE (h_D)			23 000 FT
MAXIMUM DEPLOYABLE DECELERATOR VERTICAL VELOCITY			
AT TERMINAL PROPULSION INITIATION			350 FT/SEC
HORIZONTAL VELOCITY VECTOR DUE TO WINDS			220 FT/SEC
TERMINAL PROPULSION IGNITION ALTITUDE			4868 FT
MAXIMUM SURFACE SLOPE	34° ABOVE LOCAL HORIZONTAL		
TERMINAL PROPULSION SYSTEM TYPE			MONO
TERMINAL GUIDANCE MODE			PREPROGRAMMED
FOR FIGURES 3.3.1-22 & 3.3.1-23			
IGNITION ALTITUDE IS CHANGED TO			6500 FT
TERMINAL GUIDANCE MODE			MODIFIED

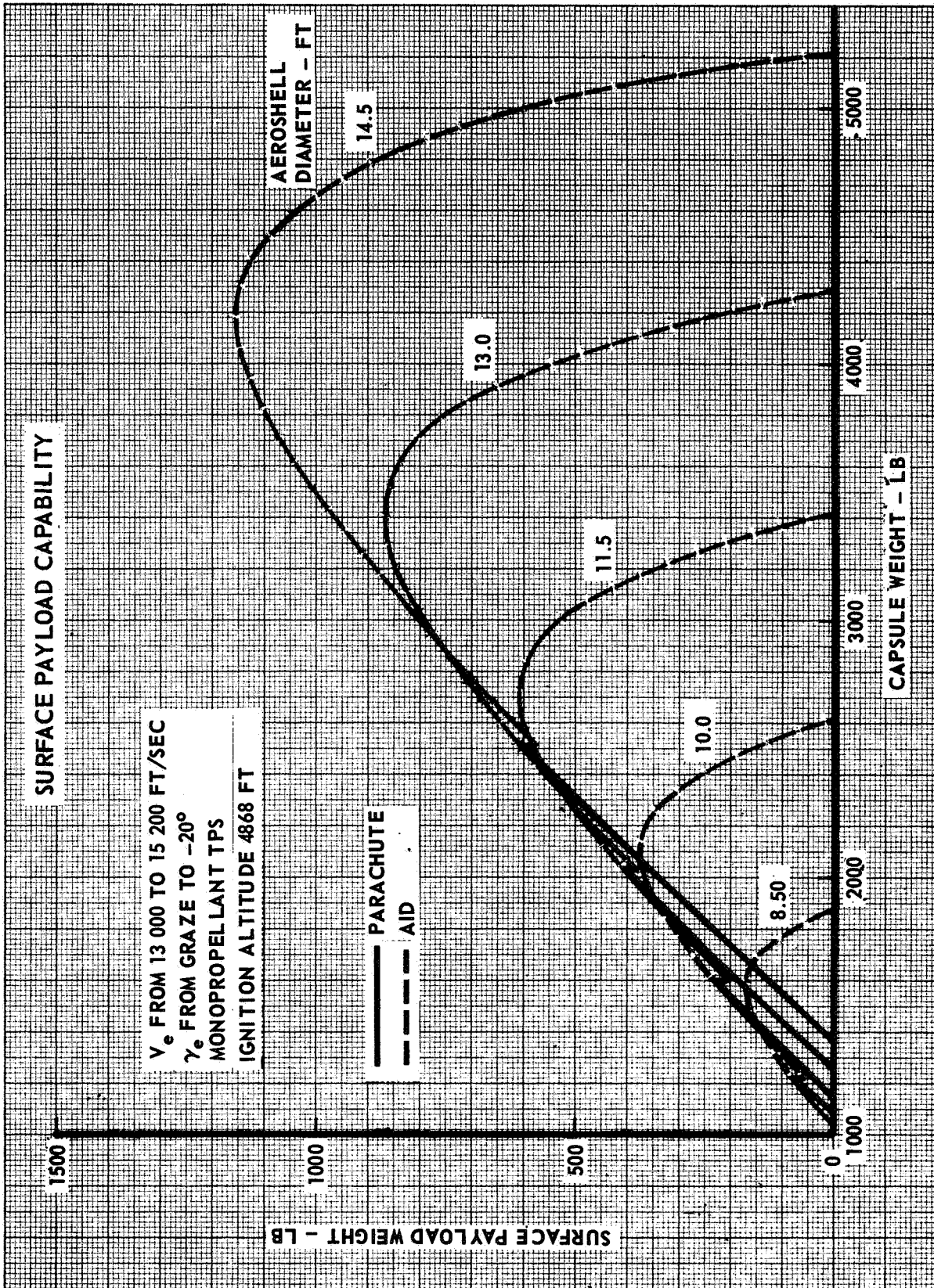


FIGURE 3.3.1-19

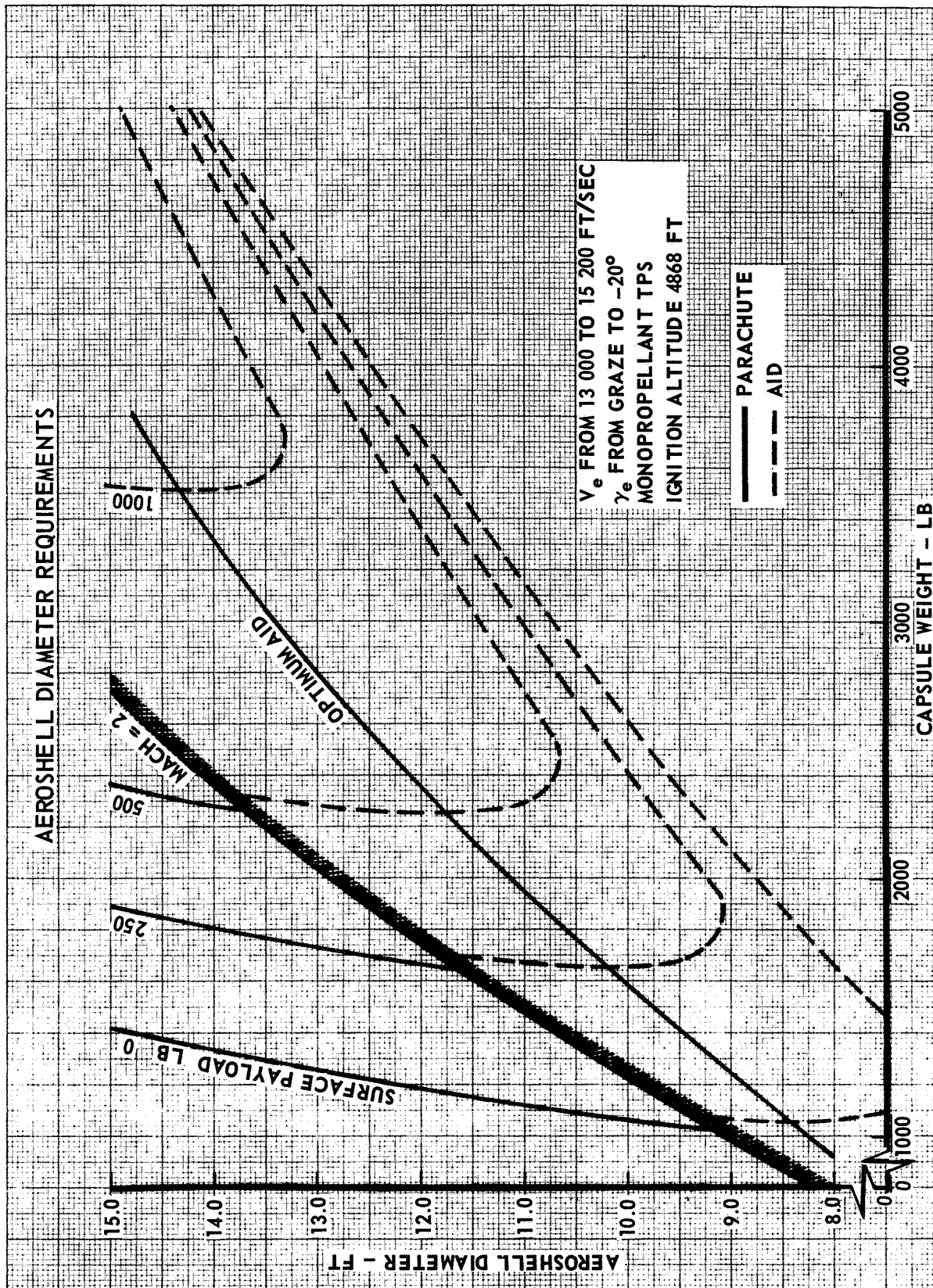


FIGURE 3.3.1-20

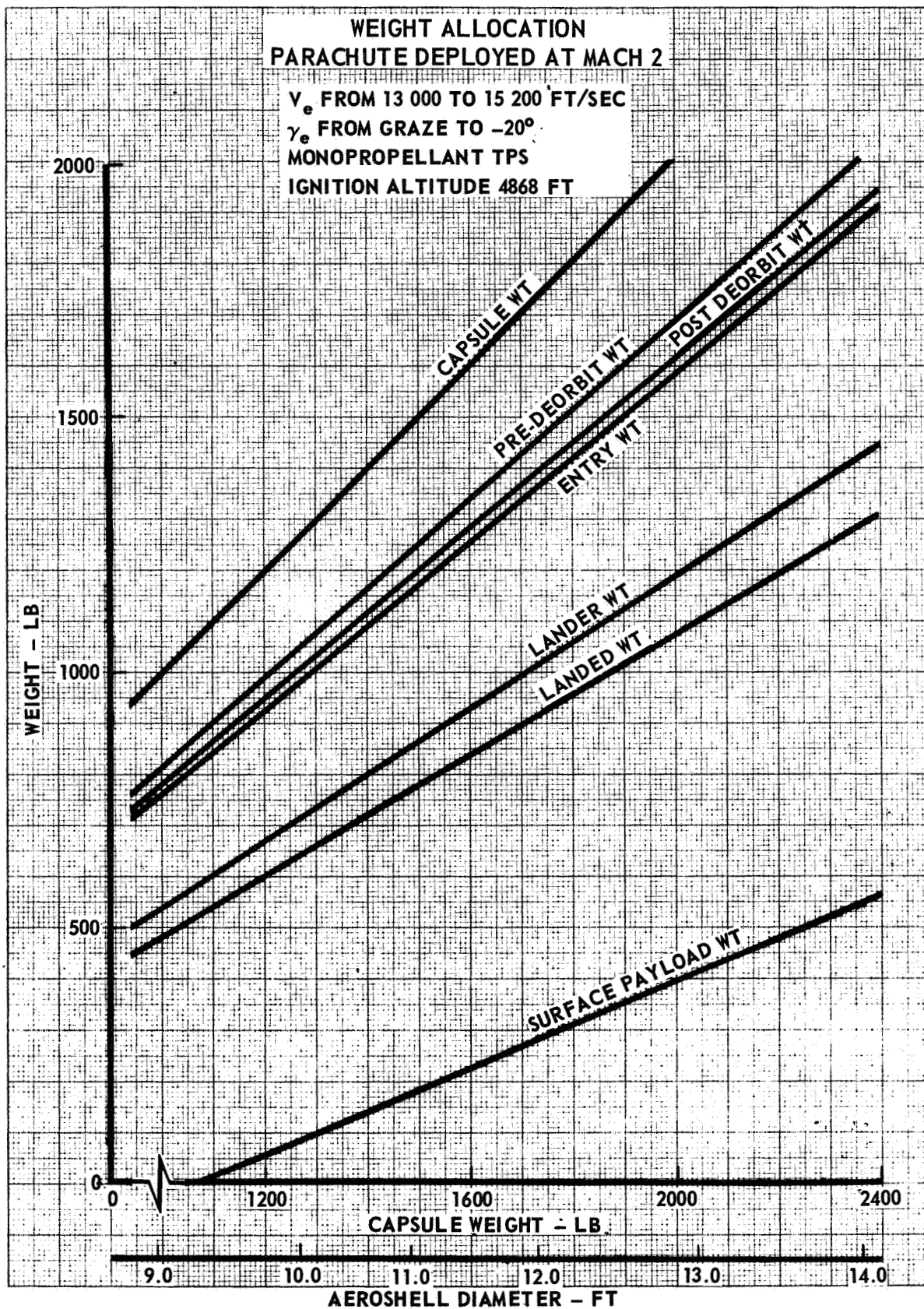


FIGURE 3.3.1-21

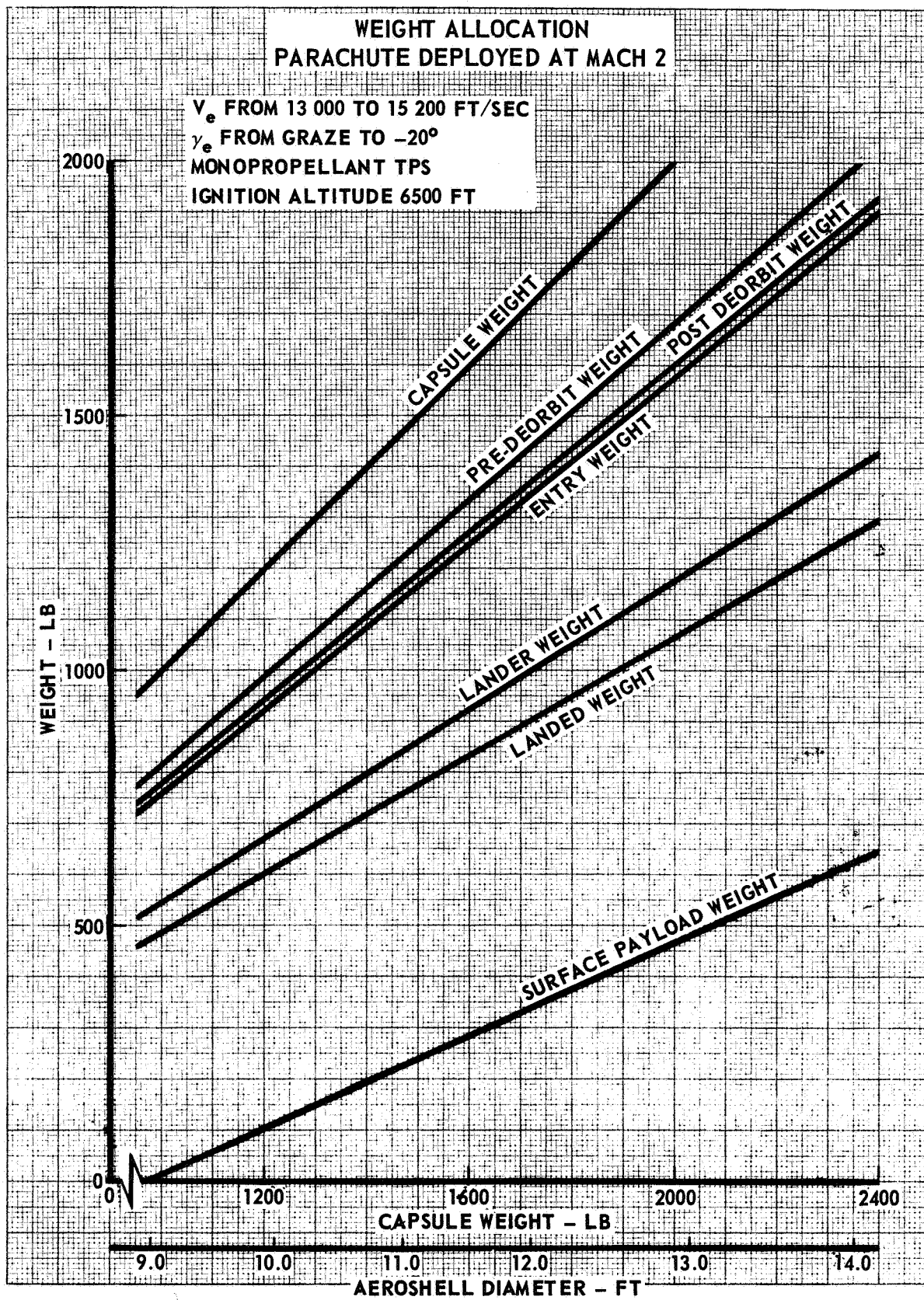


FIGURE 3.3.1-22

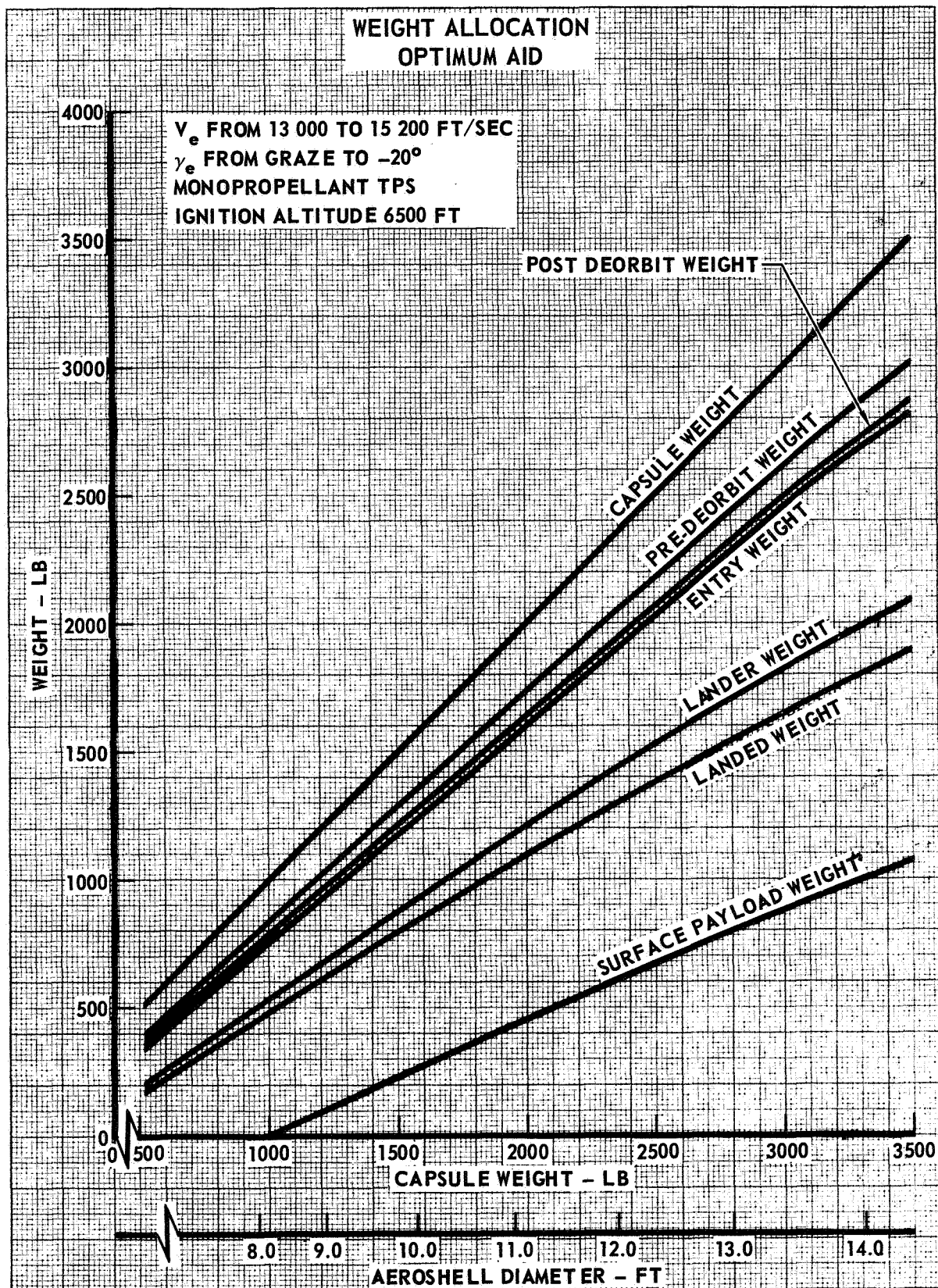


FIGURE 3.3.1-23

TABLE 3.3.1-7
CRITERIA FOR FIGURES 3.3.1-24 THROUGH 3.3.1-28

ENTRY ALTITUDE			
ENTRY VELOCITY (V_e)	FROM 13000	TO	<u>800000 FT</u> <u>15200 FT/SEC</u>
ENTRY FLIGHT PATH ANGLE (γ_e)	FROM <u>GRAZE</u>	TO	<u>-20°</u>
ATMOSPHERES			
SCALE HEIGHT (H_ρ)	FROM 5	TO	<u>14 KM</u>
SURFACE PRESSURE (P_o)	FROM 5	TO	<u>20 MB</u>
DEORBIT VELOCITY INCREMENT			<u>450 FT/SEC</u>
FIXED EQUIPMENT WEIGHT			
GUIDANCE & CONTROL & RADARS - ON AEROSHELL			<u>2.9 LB</u>
GUIDANCE & CONTROL & RADARS - LANDED			<u>112.9 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - TOTAL			<u>168.0 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - ON ADAPTER			<u>15.0 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - ON AEROSHELL			<u>15.5 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - LANDER			<u>137.5 LB</u>
DRAW COEFFICIENTS (C_D)			
AEROSHELL - SUPERSONIC C_D			<u>1.51</u>
AEROSHELL - SUBSONIC C_D			<u>.78</u>
ATTACHED INFLATABLE DECELERATOR - SUPERSONIC C_D			<u>1.00</u>
ATTACHED INFLATABLE DECELERATOR - SUBSONIC C_D			<u>.80</u>
PARACHUTE - SUPERSONIC C_D (BASED ON TOTAL SURFACE AREA)			<u>.60</u>
PARACHUTE - SUBSONIC C_D (BASED ON TOTAL SURFACE AREA)			<u>.60</u>
MINIMUM ACCELERATION OF AEROSHELL RELATIVE TO LANDER			<u>.6 g</u>
PARACHUTE DEPLOYMENT MACH NO. LIMIT			<u>2</u>
DEPLOYABLE DECELERATOR DEPLOYMENT ALTITUDE (h_D)			<u>23000 FT</u>
MAXIMUM DEPLOYABLE DECELERATOR VERTICAL VELOCITY			
AT TERMINAL PROPULSION INITIATION			<u>350 FT/SEC</u>
HORIZONTAL VELOCITY VECTOR DUE TO WINDS			<u>220 FT/SEC</u>
TERMINAL PROPULSION IGNITION ALTITUDE			<u>4868 FT</u>
MAXIMUM SURFACE SLOPE			<u>34° ABOVE LOCAL HORIZONTAL</u>
TERMINAL PROPULSION SYSTEM TYPE			<u>B I</u>
TERMINAL GUIDANCE MODE			<u>PREPROGRAMMED</u>
FOR FIGURES 3.3.1-27 & 3.3.1-28			
IGNITION ALTITUDE IS CHANGED TO			<u>6500 FT</u>
TERMINAL GUIDANCE MODE			<u>MODIFIED</u>

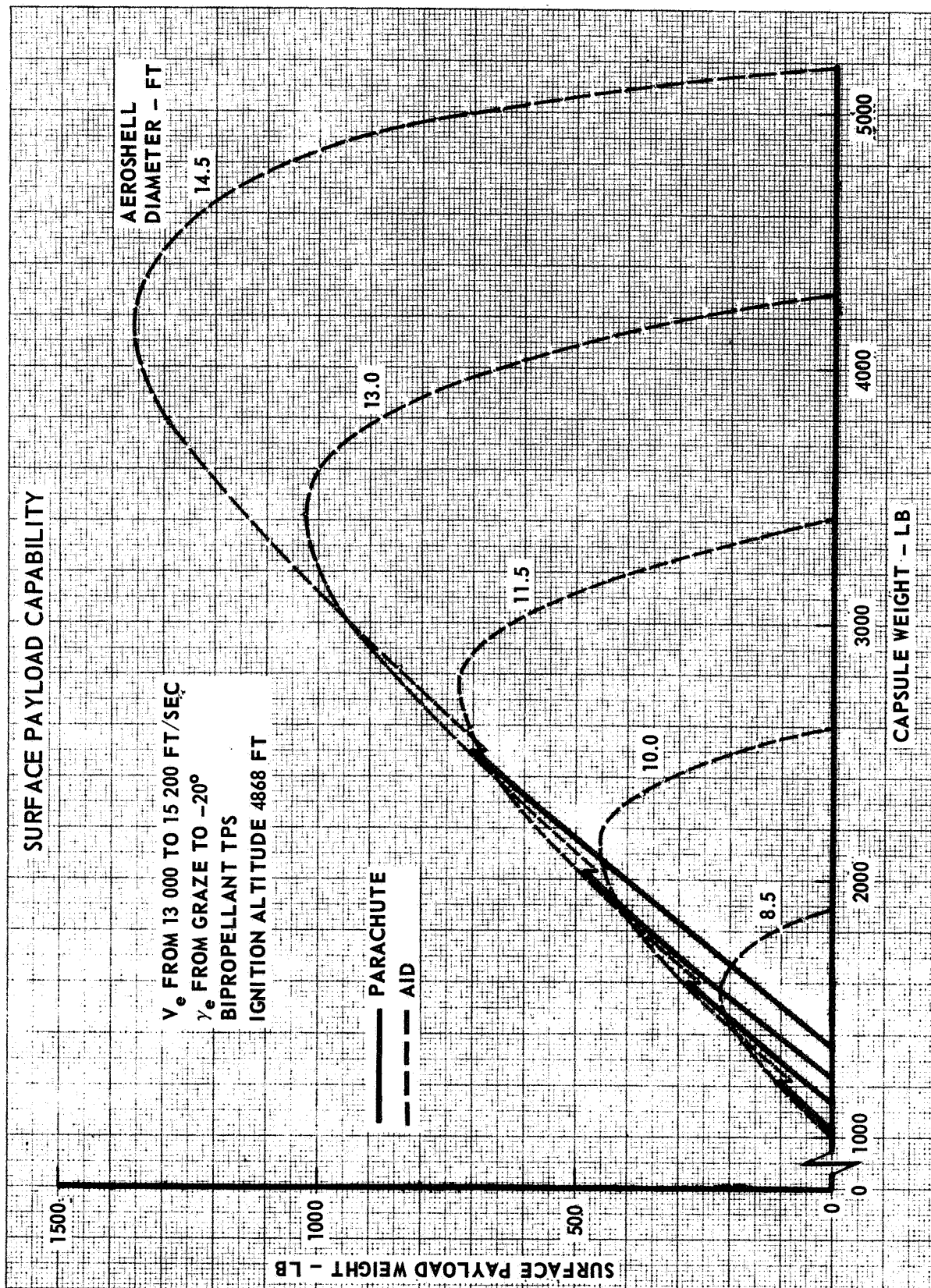


FIGURE 3.3.1-24

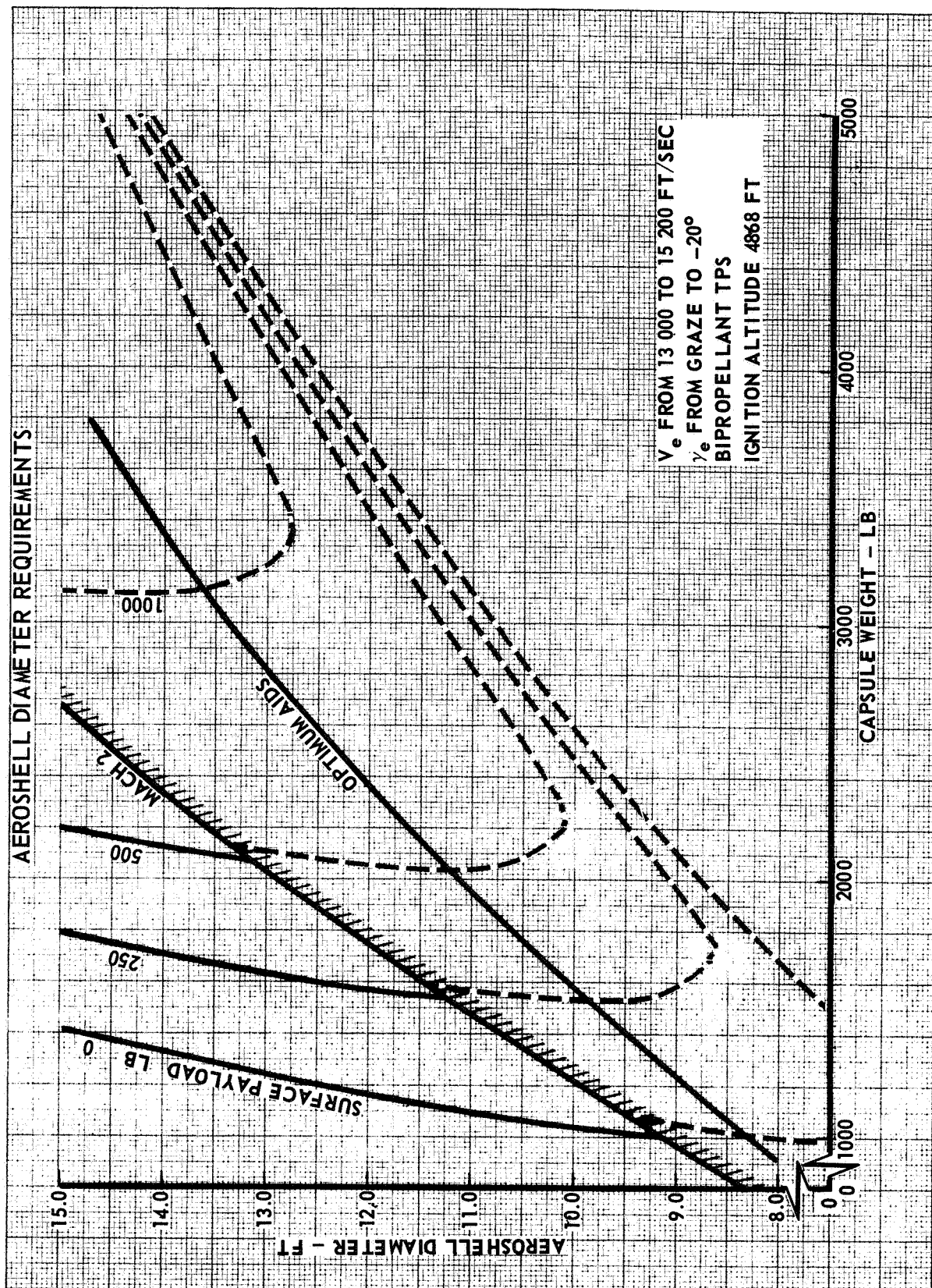


FIGURE 3.3.1-25

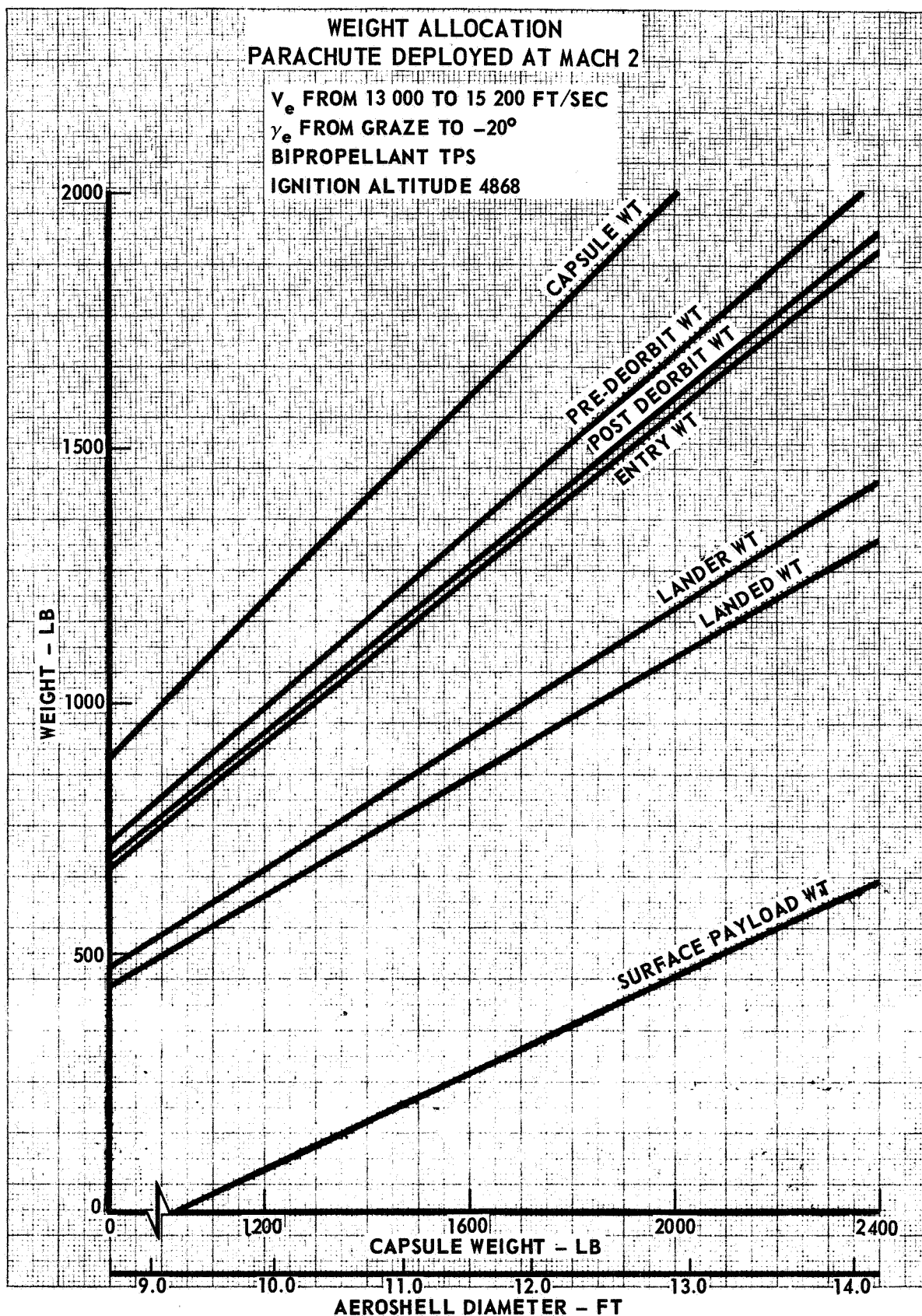


FIGURE 3.3.1-26

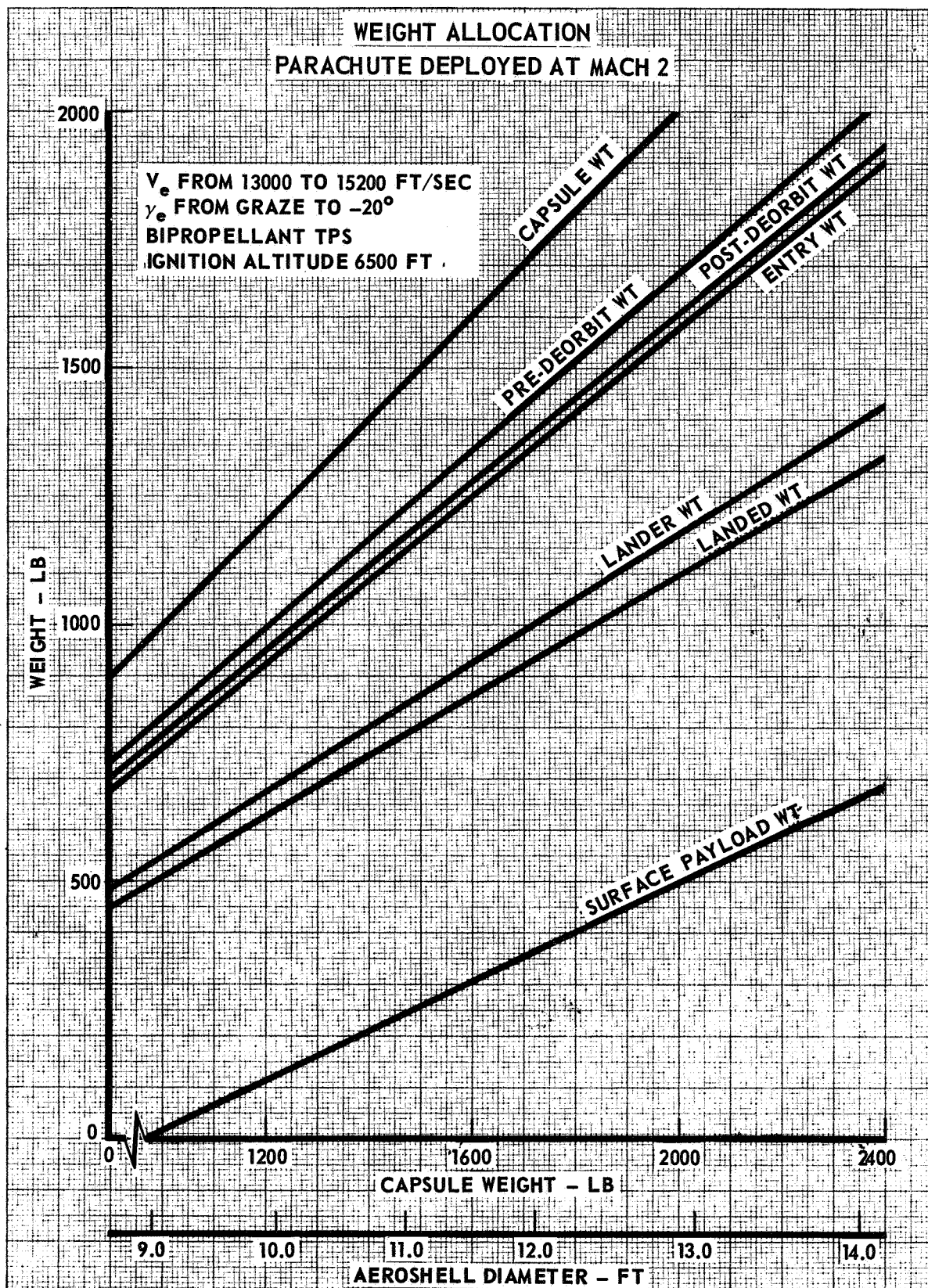


FIGURE 3.3.1-27

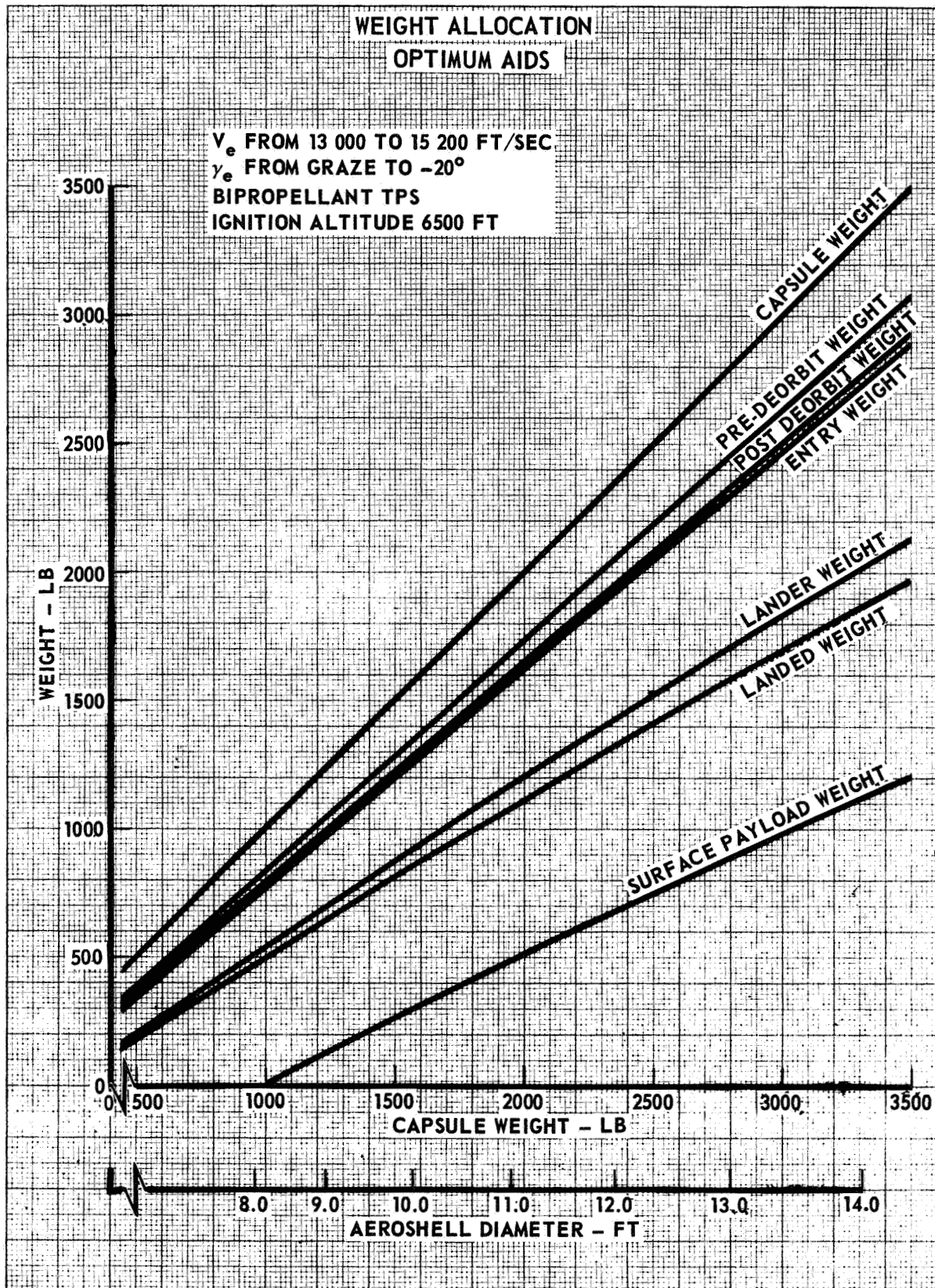


FIGURE 3.3.1-28

is given in Table 3.3.1-8.

Surface payload capability versus capsule weight for various aeroshell diameters is shown in Figure 3.3.1-30 for reduced entry corridor, scale height, winds, and surface slopes. Figure 3.3.1-31 is a cross plot which illustrates the aeroshell diameter requirements. Figure 3.3.1-32 is a mission profile weight breakdown. Table 3.3.1-9 presents the criteria for these parametrics.

Weight exchange functions were determined along the Mach = 2.0 line for the parachute systems and along the optimum weight line for the AIDS. The conditions for these three sets of weight exchange functions were:

- a) Modified preprogrammed guidance
- b) $h_D = 23\ 000\ \text{ft}$
- c) Full entry corridor
- d) $h_{\text{igh}} = 6500\ \text{ft}$
- e) Solid deorbit motor
- f) Parachute $M_D = 2.0$; Monopropellant TPS
- g) AIDS $M_D = 3.5$; Monopropellant TPS

These equations are presented in Tables 3.3.1-10 and 3.3.1-11.

TABLE 3.3.1- 8
CRITERIA FOR FIGURE 3.3.1-29

ENTRY ALTITUDE			800 000 FT
ENTRY VELOCITY (V_e)	FROM 13 000	TO	15 200 FT/SEC
ENTRY FLIGHT PATH ANGLE (γ_e)	FROM <u>GRAZE</u>	TO	<u>-20°</u>
ATMOSPHERES			
SCALE HEIGHT (H_ρ)	FROM 5	TO	14 KM
SURFACE PRESSURE (P_0)	FROM 5	TO	20 MB
DEORBIT VELOCITY INCREMENT			450 FT/SEC
FIXED EQUIPMENT WEIGHT			
GUIDANCE & CONTROL & RADARS - ON AEROSHELL			2.9 LB
GUIDANCE & CONTROL & RADARS - LANDED			112.9 LB
SCIENCE, TCM, ELECTRICAL & MISC - TOTAL			168.0 LB
SCIENCE, TCM, ELECTRICAL & MISC - ON ADAPTER			15.0 LB
SCIENCE, TCM, ELECTRICAL & MISC - ON AEROSHELL			15.5 LB
SCIENCE, TCM, ELECTRICAL & MISC - LANDER			137.5 LB
DRAG COEFFICIENTS (C_D)			
AEROSHELL - SUPERSONIC C_D			1.51
AEROSHELL - SUBSONIC C_D			.78
ATTACHED INFLATABLE DECELERATOR - SUPERSONIC C_D			1.00
ATTACHED INFLATABLE DECELERATOR - SUBSONIC C_D			.80
PARACHUTE - SUPERSONIC C_D (BASED ON TOTAL SURFACE AREA)			.60
PARACHUTE - SUBSONIC C_D (BASED ON TOTAL SURFACE AREA)			.60
MINIMUM ACCELERATION OF AEROSHELL RELATIVE TO LANDER			.6 g
PARACHUTE DEPLOYMENT MACH NO. LIMIT			2
DEPLOYABLE DECELERATOR DEPLOYMENT ALTITUDE (h_D)			*, FT
MAXIMUM DEPLOYABLE DECELERATOR VERTICAL VELOCITY			
AT TERMINAL PROPULSION INITIATION			350 FT/SEC
HORIZONTAL VELOCITY VECTOR DUE TO WINDS			220 FT/SEC
TERMINAL PROPULSION IGNITION ALTITUDE			** FT
MAXIMUM SURFACE SLOPE			<u>34°</u> ABOVE LOCAL HORIZONTAL
TERMINAL PROPULSION SYSTEM TYPE			MONO
TERMINAL GUIDANCE MODE			<u>PREPROGRAMMED</u>
FOR OPTIMUM SYSTEM			
TERMINAL PROPULSION IGNITION ALTITUDE			***
TERMINAL GUIDANCE MODE			<u>MODIFIED</u>
* h_D = 23 000 FT	**4868 FT	***6500 FT	
h_D = 18 590 FT	4460 FT	5200 FT	
h_D = 14 430 FT	3700 FT	4000 FT	

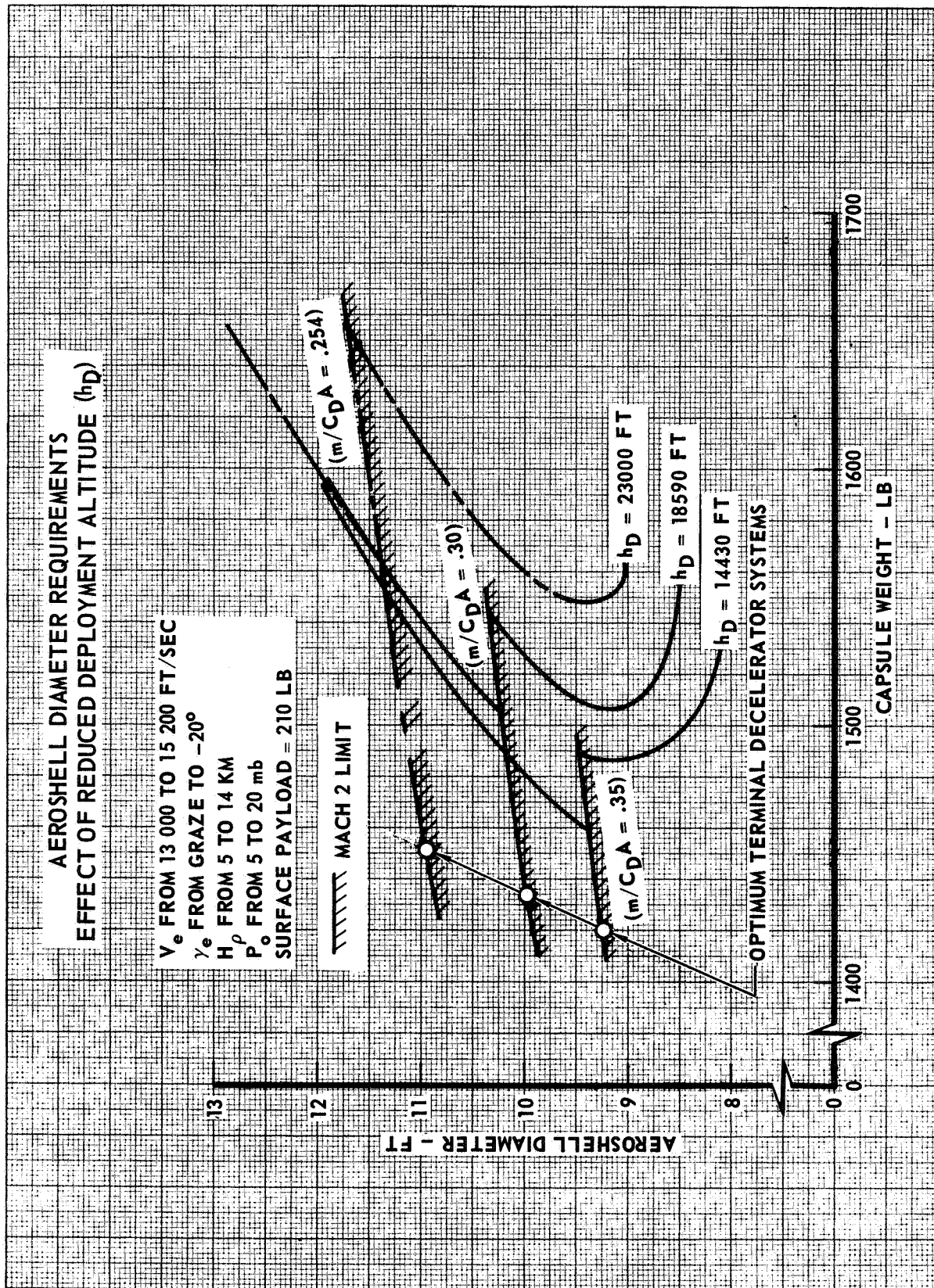


FIGURE 3.3.1-29

TABLE 3.3.1-9
CRITERIA FOR FIGURES 3.3.1-30, 3.3.1-31 & 3.3.1-32

ENTRY ALTITUDE		
ENTRY VELOCITY (V_e)	FROM <u>14000</u> TO	<u>800000 FT</u> <u>15 200 FT/SEC</u>
ENTRY FLIGHT PATH ANGLE (γ_e)	FROM <u>-15°</u> TO	<u>-18°</u>
ATMOSPHERES		
SCALE HEIGHT (H_p)	FROM <u>8</u> TO	<u>14 KM</u>
SURFACE PRESSURE (P_0)	FROM <u>5</u> TO	<u>20 MB</u>
DEORBIT VELOCITY INCREMENT		<u>450 FT/SEC</u>
FIXED EQUIPMENT WEIGHT		
GUIDANCE & CONTROL & RADARS - ON AEROSHELL		<u>2.9 LB</u>
GUIDANCE & CONTROL & RADARS - LANDED		<u>112.9 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - TOTAL		<u>168.0 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - ON ADAPTER		<u>15.0 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - ON AEROSHELL		<u>15.5 LB</u>
SCIENCE, TCM, ELECTRICAL & MISC - LANDER		<u>137.5 LB</u>
DRAG COEFFICIENTS (C_D)		
AEROSHELL - SUPERSONIC C_D		<u>1.51</u>
AEROSHELL - SUBSONIC C_D		<u>.78</u>
ATTACHED INFLATABLE DECELERATOR - SUPERSONIC C_D		<u>1.00</u>
ATTACHED INFLATABLE DECELERATOR - SUBSONIC C_D		<u>.80</u>
PARACHUTE - SUPERSONIC C_D (BASED ON TOTAL SURFACE AREA)		<u>.60</u>
PARACHUTE - SUBSONIC C_D (BASED ON TOTAL SURFACE AREA)		<u>.60</u>
MINIMUM ACCELERATION OF AEROSHELL RELATIVE TO LANDER		<u>.6 g</u>
PARACHUTE DEPLOYMENT MACH NO. LIMIT		<u>2</u>
DEPLOYABLE DECELERATOR DEPLOYMENT ALTITUDE (h_D)		<u>23000 FT</u>
MAXIMUM DEPLOYABLE DECELERATOR VERTICAL VELOCITY		
AT TERMINAL PROPULSION INITIATION		<u>310 FT/SEC</u>
HORIZONTAL VELOCITY VECTOR DUE TO WINDS		<u>118 FT/SEC</u>
TERMINAL PROPULSION IGNITION ALTITUDE		<u>6500 FT</u>
MAXIMUM SURFACE SLOPE	<u>20°</u> ABOVE LOCAL HORIZONTAL	
TERMINAL PROPULSION SYSTEM TYPE		<u>MONO</u>
TERMINAL GUIDANCE MODE		<u>MODIFIED</u>

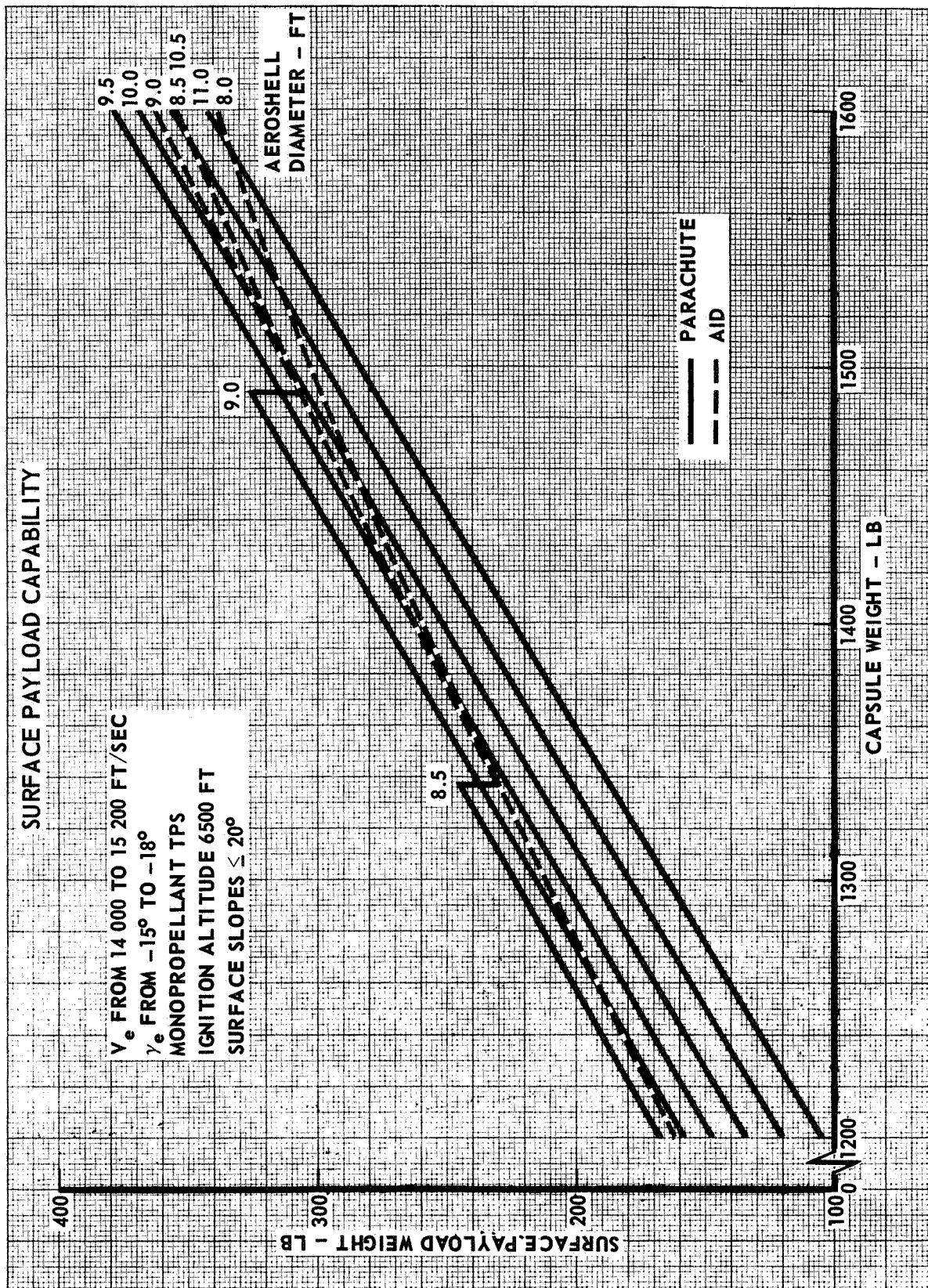


FIGURE 3.3.1-30

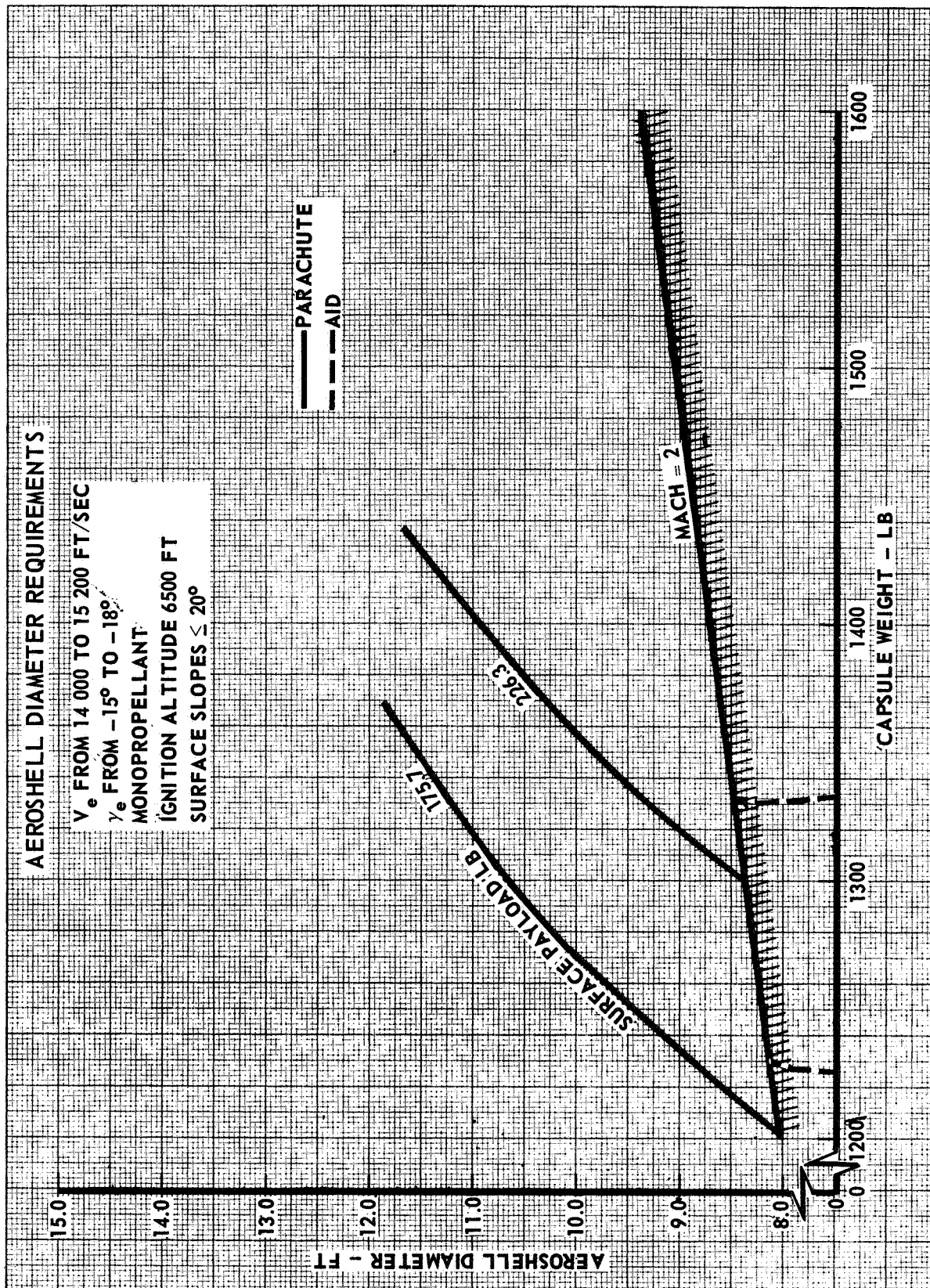


FIGURE 3.3.1-31

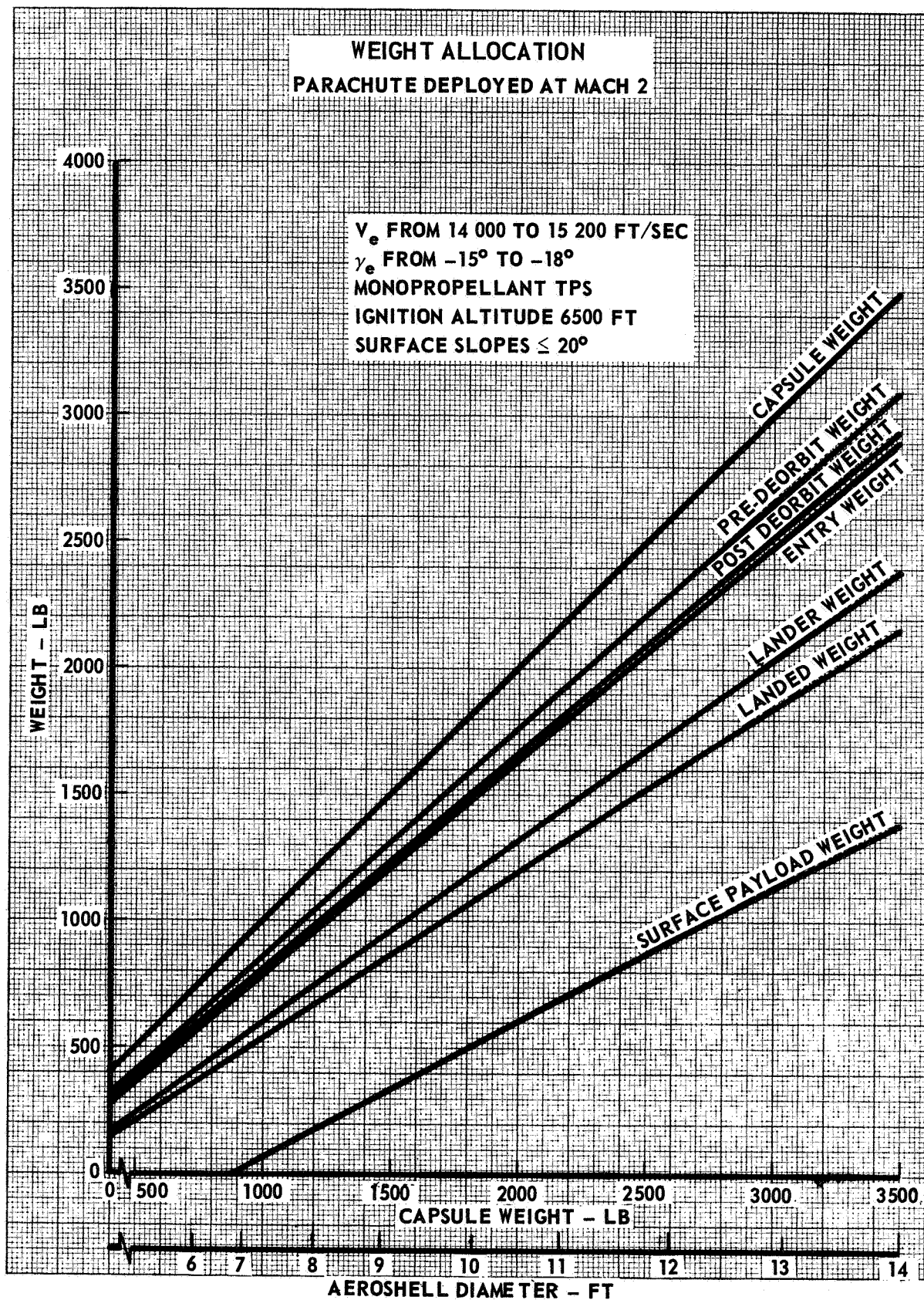


FIGURE 3.3.1-32

TABLE 3.3.1-10
CAPSULE WEIGHT EXCHANGE FUNCTIONS

CONFIGURATION	NOMINAL ENVIRONMENT	MODIFIED ENVIRONMENT
SOLID PROPELLANT DEORBIT PARACHUTE ($M_D = 2.0$)	$W_C = 980 + 2.19 W_{SP}$	$W_C = 875 + 1.90 W_{SP}$
MONOPROPELLANT TPS	$W_L = 528 + 1.40 W_{SP}$	$W_L = 515 + 1.35 W_{SP}$
MODIFIED PREPROGRAMMED GUIDANCE	$W_{DL} = -5 + 1.05 W_{SP}$	$W_{DL} = -3 + 1.05 W_L$
AEROSHELL IMPACT REQUIREMENT	$W_E = 70 + 1.22 W_{DL}$	$W_E = 90 + 1.12 W_{DL}$
IGNITION ALTITUDE = 6500 FT	$W_C = 80 + 1.22 W_E$	$W_C = 50 + 1.2 W_E$

TABLE 3.3.1-11
CAPSULE WEIGHT EXCHANGE FUNCTIONS

CONFIGURATION	NOMINAL ENVIRONMENT
SOLID PROPELLANT DEORBIT AIDS (OPTIMUM $M_D \sim 3.5$)	$W_C = 1000 + 2.16 W_{SP}$
MONOPROPELLANT TPS	$W_L = 540 + 1.45 W_{SP}$
MODIFIED PREPROGRAMMED GUIDANCE	$W_{DL} = -8 + 1.12 W_L$
AEROSHELL IMPACT REQUIREMENT	$W_E = 80 + 1.16 W_{DL}$
IGNITION ALTITUDE = 6500 FT	$W_C = 110 + 1.17 W_E$

NOTE:
SP - SURFACE PAYLOAD
L - LANDER
DL - DECELERATOR LOAD
E - ENTRY WEIGHT
C - CAPSULE

3.3.2 SURFACE PAYLOAD ANALYSIS - In the analysis of the overall capsule capability, the payload element which is delivered to the surface of Mars can be considered independently from the other capsule elements, since the capsule delivery system is essentially independent of the performance or operational characteristics of the surface payload.

The surface payload includes the surface science instruments, communications, electrical power, sequencing, thermal control, and the structure required to carry out the experimental program after landing. The primary variables include the mission duration, the science instruments, the communication mode, the power source, and the thermal control approach. To permit rapid and accurate evaluation of the payload capability as a function of these variables, the computations were mechanized in a computer program which incorporates parametric design characteristics of each system. The analysis therefore provides a relatively detailed design solution which simultaneously satisfies the design and operating requirements of all systems.

3.3.2.1 Summary. - The surface payload changes as the mission changes from short (1 to 3 days) to extended (90 days) to long (6 months to 2 years) duration. Regardless of the mission duration, certain conclusions are applicable.

- a) Surface payloads designed for a continuous cold (-190°F) are considerably heavier than vehicles designed for a cyclic environment.
- b) The relay mode of data transmission allows lighter and less complex surface payloads than the direct mode. However, its selection as the principal mode of data transmission is dependent upon the availability of the orbiter.
- c) Radioisotope heaters are the most attractive means of providing heat to the surface payload during the cold periods of the mission. They do represent somewhat of a problem during the warm portions of the mission because they increase the temperature of the equipment.

- d) The weight of the science payload has a moderate effect upon the total surface payload. Generally, a 10 pound increase in science increases the surface payload weight by 20 to 40 pounds.

In addition to the general results, which are applicable regardless of mission duration, the following conclusions apply to the three classes of missions:

Short Duration Missions (Battery Power) - With a minimum science payload (30 pounds, 10^7 bits per day), surface payloads of approximately 175 pounds are possible. This weight results from using isotope heaters, the relay transmission mode, and by designing for a cyclic environment and a one day mission. A more conservative approach is to design for a constant cold environment and to use only electrical heaters. This approach results in a surface payload weight of about 210 pounds.

If the surface payload is designed for more than one day of landed operations, its weight increases rapidly for each additional day of operation if a constant cold environment is assumed, and electrical heaters are used. If a cyclic environment is assumed or if isotope heaters are used, the daily weight increase is gradual (approximately 25 pounds per day).

The relay transmission mode is preferred for a short duration mission. A command system is not considered to be required in a mission of a few days. Direct readout of the facsimile camera can be used to eliminate the requirement for a tape recorder.

Extended Duration Missions (Solar Power) - A surface payload for extended duration missions, and with the minimum approved science payload, can be designed in several ways. A conservative approach is to provide the capability for surviving one cold day with a combination of electrical and isotope heaters, and cyclic days with only isotope heaters. This approach yields a small solar panel area (20 ft^2), a large battery (50 pounds) and a total payload weight of about 280 pounds. Another approach is to consider only the cyclic environment and to use only isotope heaters. This results in a surface payload weight of 226 pounds, and a solar panel area of less than 20 ft^2 . These approaches have

a minimum solar panel area with no margin for decreased solar panel performance.

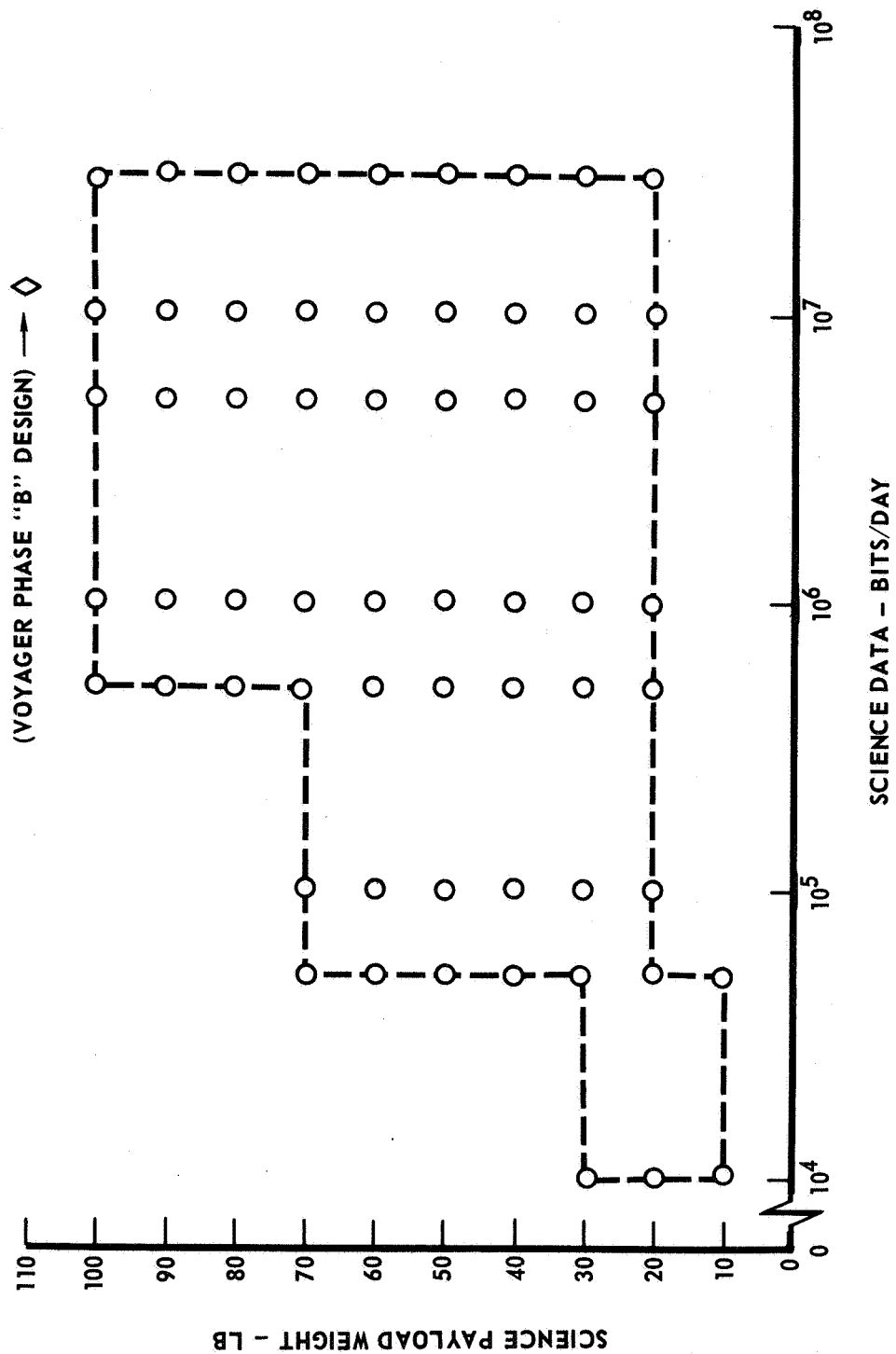
As with short duration missions, the relay mode is preferred as the basic technique for data transmission. However, since it is probable that the orbit of the orbiter will be changed a few days after the start of the mission, making the relay mode inoperative shortly after this change, a low rate direct link should be used to transmit non-imaging data. The length of the mission makes a command link desirable. If images at a fixed lighting condition are acceptable, the facsimile camera can be read out directly, thus eliminating the need for a tape recorder.

Long Duration Missions (RTG Power) - The long duration missions were analyzed although no conceptual design point was derived. A combination of an RTG and a battery should be used for the mission because the use of only an RTG results in a heavier vehicle and an extremely large RTG. The direct transmission mode is preferred, because the relay mode is operationally unfeasible throughout most of the mission. The transmission of a large amount of data, on a daily basis, throughout the mission is practical only if the payload is designed to support the operation of only one radio link at a time. Simultaneous operation of two or more radio links, e.g., high rate and low rate S-band, results in a large power system and a large surface payload.

3.3.2.2 Study alternatives. - The investigation of the surface payload simultaneously considered several design variables, primarily the science payload, mission duration, electrical power source, communication mode, environmental design limits, and thermal control techniques.

Science Payload - The science payload, the basic independent variable in the study, is a function of the specific instruments selected, the mode of operation, and the data produced. For this study, 63 discrete science payloads were considered. A matrix of the weight and data gathering capabilities of these payloads is presented in Figure 3.3.2-1. The specific science payloads used for the analysis are shown in Figure 3.3.2-2. It should be noted that although these payloads may differ from those used for the conceptual designs, the difference is not significant in terms of effects upon the total surface payload.

MATRIX OF SCIENCE PAYLOADS



3.3.2-4

FIGURE 3.3.2-1

SCIENCE PAYLOADS

SYSTEM WEIGHT	LOW DATA RATE ($< 5 \times 10^5$ BITS/DAY)		HIGH DATA RATE ($> 5 \times 10^5$ BITS/DAY)	
	INSTRUMENTS	WT (LB)	INSTRUMENTS	WT (LB)
10 LB	METEOROLOGY TOTAL	<u>10</u> 10	- -	- -
20 LB	METEOROLOGY GAS CHROMATOGRAPH TOTAL	<u>10</u> <u>8</u> 18	METEOROLOGY FACSIMILE (1) TOTAL	<u>10</u> <u>7.5</u> 17.5
30 LB	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE RADIOMETER TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>5</u> 29	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE FACSIMILE (1) TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>7.5</u> 31.5
40 LB	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE LIFE DETECTOR TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>18</u> 42	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE FACSIMILE (2) TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>15</u> 39
50 LB	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE LIFE DETECTOR RADIOMETER TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>18</u> <u>5</u> 47	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE LIFE DETECTOR FACSIMILE (1) TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>18</u> <u>7.5</u> 49.5
60 LB	METEOROLOGY SURFACE SAMPLER GAS CHROM W OVEN SUBSURFACE PROBE LIFE DETECTOR TOTAL	<u>10</u> <u>15</u> <u>15</u> <u>6</u> <u>16</u> 62	METEOROLOGY GAS CHROMATOGRAPH SUBSURFACE PROBE LIFE DETECTOR FACSIMILE (2) TOTAL	<u>10</u> <u>8</u> <u>6</u> <u>18</u> <u>15</u> 57
70 LB	METEOROLOGY SURFACE SAMPLER GAS CHROM W OVEN SUBSURFACE PROBE LIFE DETECTOR RADIOMETER TOTAL	<u>10</u> <u>15</u> <u>15</u> <u>6</u> <u>16</u> <u>5</u> 67	METEOROLOGY SURFACE SAMPLER GAS CHROM W OVEN LIFE DETECTOR SUBSURFACE PROBE FACSIMILE (1) TOTAL	<u>10</u> <u>15</u> <u>15</u> <u>16</u> <u>6</u> <u>7.5</u> 69.5
80 LB	NONE USED FOR STUDY		METEOROLOGY SURFACE SAMPLER GAS CHROM W OVEN LIFE DETECTOR " SPECTROMETER SUBSURFACE PROBE FACSIMILE (1) TOTAL	<u>10</u> <u>20</u> <u>15</u> <u>16</u> <u>10</u> <u>6</u> <u>7.5</u> 84.5
90 LB	NONE USED FOR STUDY		80 LB SUBSYSTEM (ABOVE) 2ND FACSIMILE TOTAL	<u>84.5</u> <u>7.5</u> 92
100 LB	NONE USED FOR STUDY		90 LB SUBSYSTEM (ABOVE) X-RAY FLOURESCENCE (PART OF " SPEC) RADIOMETER TOTAL	<u>92</u> <u>2</u> <u>5</u> 99

3.3.2-5

FIGURE 3.3.2-2

Initially in this study, the entire spectrum of science payloads was examined. As the study progressed, particular emphasis was given to a science payload of 30 pounds and 10^7 bits per day, because it represented significant capability (imaging, soil analysis, and meteorology) and incurred modest payload weight and cost.

Mission Duration - As mentioned before, mission duration was categorized into three classifications; (1) short term (1, 2, or 3 days), (2) extended life (90 days), and (3) long term (6 months to 2 years). The overall study is segregated into the three basic classes of missions.

Electrical Power Source - The electrical power source is closely related to the mission duration. For short term missions, batteries were used; for extended life missions, a combination of batteries and solar panels; and for long term missions, a combination of batteries and RTG's.

Communication Mode - The alternatives in the area of communications are direct (S-band) and relay (UHF). A variable common to both transmission modes is the daily transmitting time, which is limited by the viewing opportunity of the Earth or the orbiter. In the direct mode, the antenna type (low gain or pointed) is a variable. For the relay mode of operation, a choice exists between transmitting when the orbiter is near periapse or near apoapse. For the direct mode, the communication distance is a variable, although it is a function of the landing date and the mission length. Additional options, such as whether a tape recorder, command reception link, or low rate engineering data radio link are used, are functions of other engineering requirements and the mission concept.

Environmental Design Limits - Two basic types of environment were used for this study: constant cold and cyclic. The constant cold environment, a steady state of -190°F , was used as a worst case for calculating the heating requirements of the vehicle. The cyclic environment, -150°F to $+120^{\circ}\text{F}$, was used to determine if the vehicle or its components would overheat. Also, the cyclic environment was used as a "nominal" condition for calculating the heating requirements of the vehicle.

Thermal Control - The design options available for thermal control were found to have a large effect upon the total vehicle and due to interactions with other disciplines (power, structure, etc.). The principal options include insulation thickness and efficiency, temperature limits of the insulated compartment of the vehicle, heating technique (isotopes or electrical heaters), and equipment cooling technique (phase change material or heat pipes).

3.3.2.3 Analysis technique. - The analysis of the surface payload was conducted in three discrete steps; (1) short duration missions, (2) extended duration missions, and (3) long duration missions. Short duration missions were studied first because their similarity to our VOYAGER Phase B design allowed a point for comparison. Short duration missions were analyzed for the 1973 opportunity only; extended duration missions for the 1973 and 1975 opportunities; and long duration missions for the 1975 and 1977 opportunities.

This analysis was guided by various cost considerations. The over-riding consideration was to minimize the weight of the surface payload, thereby reducing the overall weight and cost of the flight capsule. Another consideration employed was to avoid, if possible, technologies which would involve considerable cost or development. Technologies affected by this consideration were thermal control and telecommunications. For thermal control, the use of heat pipes and radiators was minimized. For telecommunications, the use of tape recorders and large S-band transmitters (output >50 watts RF) was minimized.

To construct the overall computational flow of this analysis, a model configuration of the surface payload was used. The basic dimensional characteristics of the payload used the form factors of our Phase B design in order to determine the relationships of volume, surface area, and insulation weight. The equipment of the surface payload was considered as being installed either "internal" or "external" to the basic insulated compartment of the payload. Equipment "external" to the insulated compartment included antennas, various science instruments, solar panels, and RTG's. "Internal" equipment is located inside the insulated compartment of the payload, and its temperature is more rigidly controlled. Nominal temperature limits used were +50°F to 120°F for batteries, and 0°F to 160°F for transmitters. Equipment inside the insulated

compartment included the batteries, telemetry, sequencer, transmitters, command receiver/decoder, certain science instruments, and, when required, phase change material and isotope heaters.

The computational methods employed for this analysis were similar to those used for the point designs although a computer was used for expediency. The computational process, shown in the flow chart on Figure 3.3.2-3, began with the choice of the science payload and design constraints (communication distance, insulation thickness, etc.) to be examined. The first elements determined were the sequencer, the telemetry, the data storage, and engineering instrumentation, and the command link.

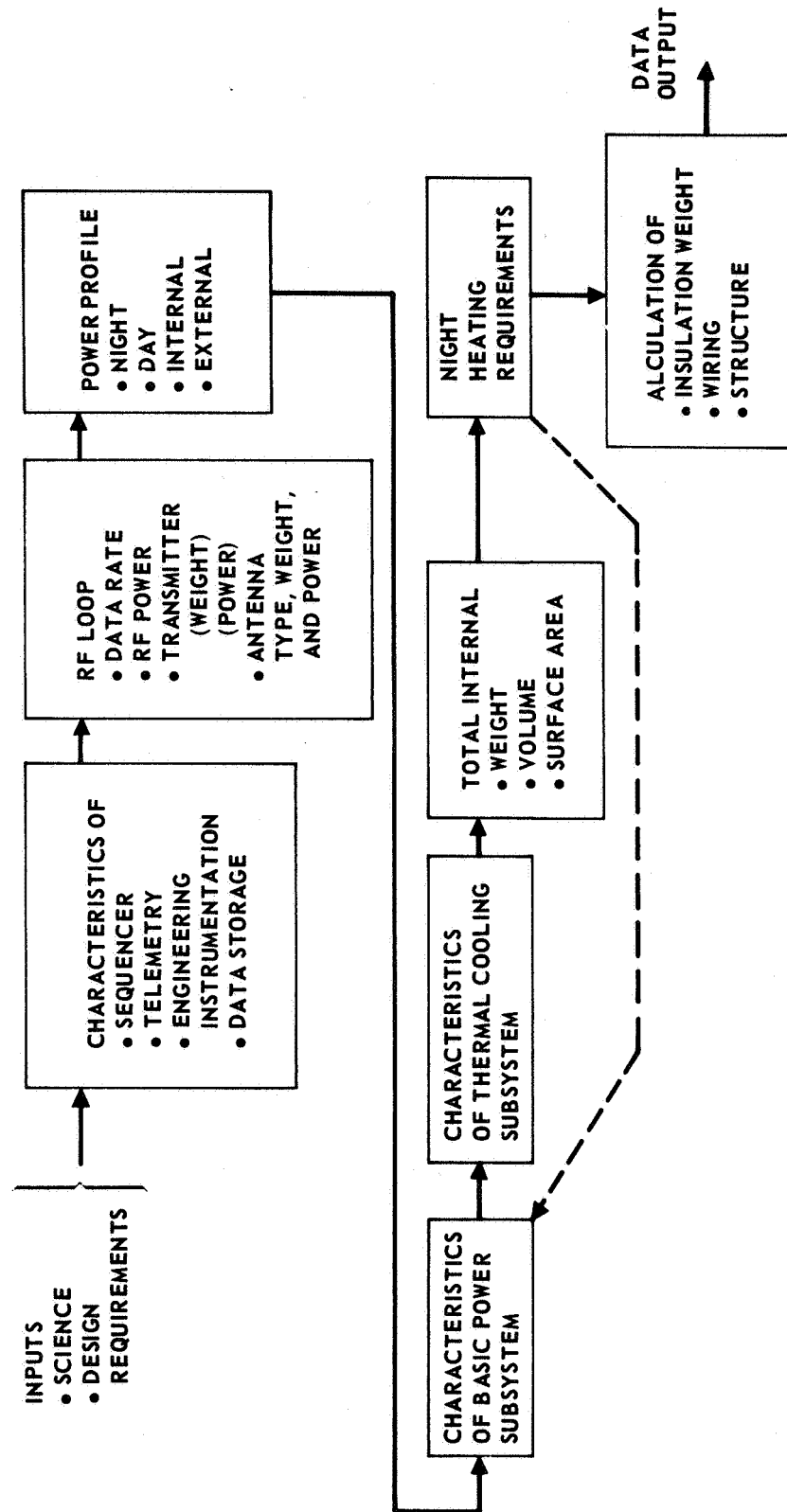
The next step was the computation of the radio link(s) characteristics. First, the data rate and radiated power were determined. If, for S-band, the radiated power exceeded 50 watts RF with a low gain antenna, a pointed antenna was used. The final step in the RF calculation was the determination of transmitter and antenna characteristics. It should be noted that in most cases, two radio links are used: one for engineering data and one for science data.

A power profile was then constructed and categorized in relationship to the Martian day and to the power distribution, internal or external, to the insulated compartment of the vehicle. A detailed definition of the power profile was required to establish the relationship of power dissipation relative to the Martian temperature cycle and, for designs employing solar panels, to the solar energy available. The power profile and the mission length were then used to establish a first estimate of the electrical power system.

The next step was a preliminary computation of the peak temperature reached by the equipment. Considerations were equipment heat dissipation, the Martian environment, the properties of the insulation, and the mass of the equipment. If overheating occurred, provisions for limiting the temperature, usually by phase change material, were calculated and incorporated.

At this point, all basic equipment requirements had been determined. By use of relationships developed from our VOYAGER Phase B design, the basic size of the vehicle was estimated. Using the surface area of the vehicle, the properties of the insulation, and the heat available from equipment operation,

SURFACE PAYLOAD ANALYSIS COMPUTER FLOW DIAGRAM



the requirements for heating the payload during the cold portions of the mission were determined. If isotopes were selected as the means of heating the vehicle, the power system remained intact from the original estimate, but several iterations for equipment cooling requirements, vehicle size, and heating were made until a thermally balanced design was achieved. If electrical heaters were used, the power system was re-calculated, resulting in successive iterations of cooling provisions, volume, surface area, etc., until thermal equilibrium was achieved.

The final step involved computation of insulation weight, wiring and structure requirements, and a summation of the characteristics of the total vehicle.

3.3.2.4 Short duration missions (battery power) Payloads designed for a short duration mission (1, 2, or 3 days) were initially examined with mission duration, communication technique, and the science payload as the principal variables. Once the effects of these basic variables were determined, other areas such as thermal control design, environmental effects, and overall payload capability were explored.

3.3.2.4.1 Effect of mission duration: The missions studied were one, two or three days in length. However, the definition of the "first day" depended on the type of vehicle. For designs using the UHF relay mode, the first Martian day was exactly 1 diurnal cycle (approximately 24.6 hours). For designs using the direct communications mode, the "first day" was a function of the landing site. A landing near the morning terminator allows a few hours of communications to Earth immediately after landing. For morning landings, the "first day" was defined as one diurnal cycle plus the few hours of data transmission which occurred immediately after landing, thus making the "first day" about 28 hours long. For landing near the evening terminator, direct communications to Earth are not possible until 13 to 15 hours after landing. The "first day" for landings near the evening terminator is defined as 22 hours, thus is slightly less than one diurnal cycle. The second and third days were considered as full diurnal cycles for all types of surface payloads.

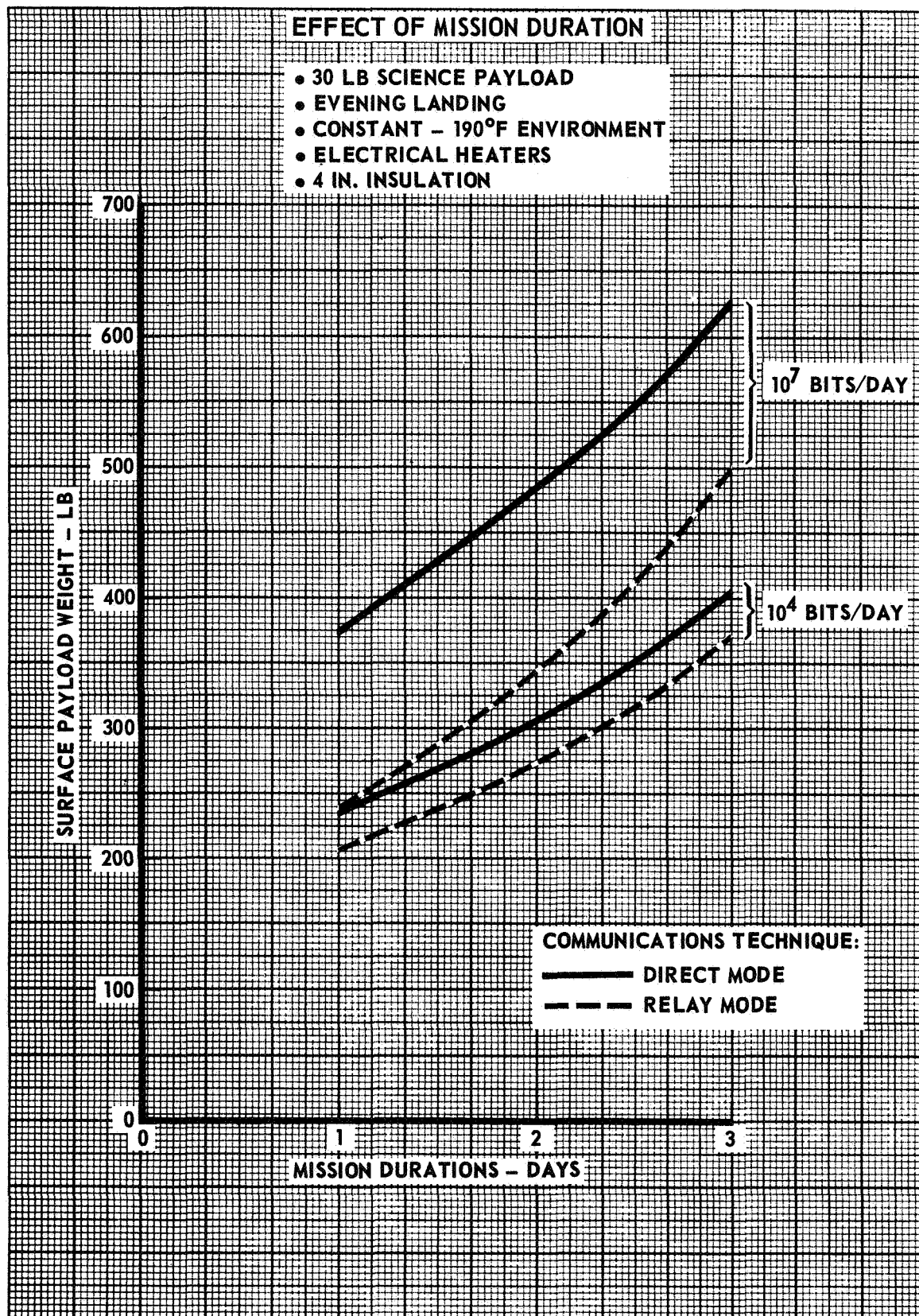
All short duration missions employed silver-zinc (Ag-Zn) batteries as the

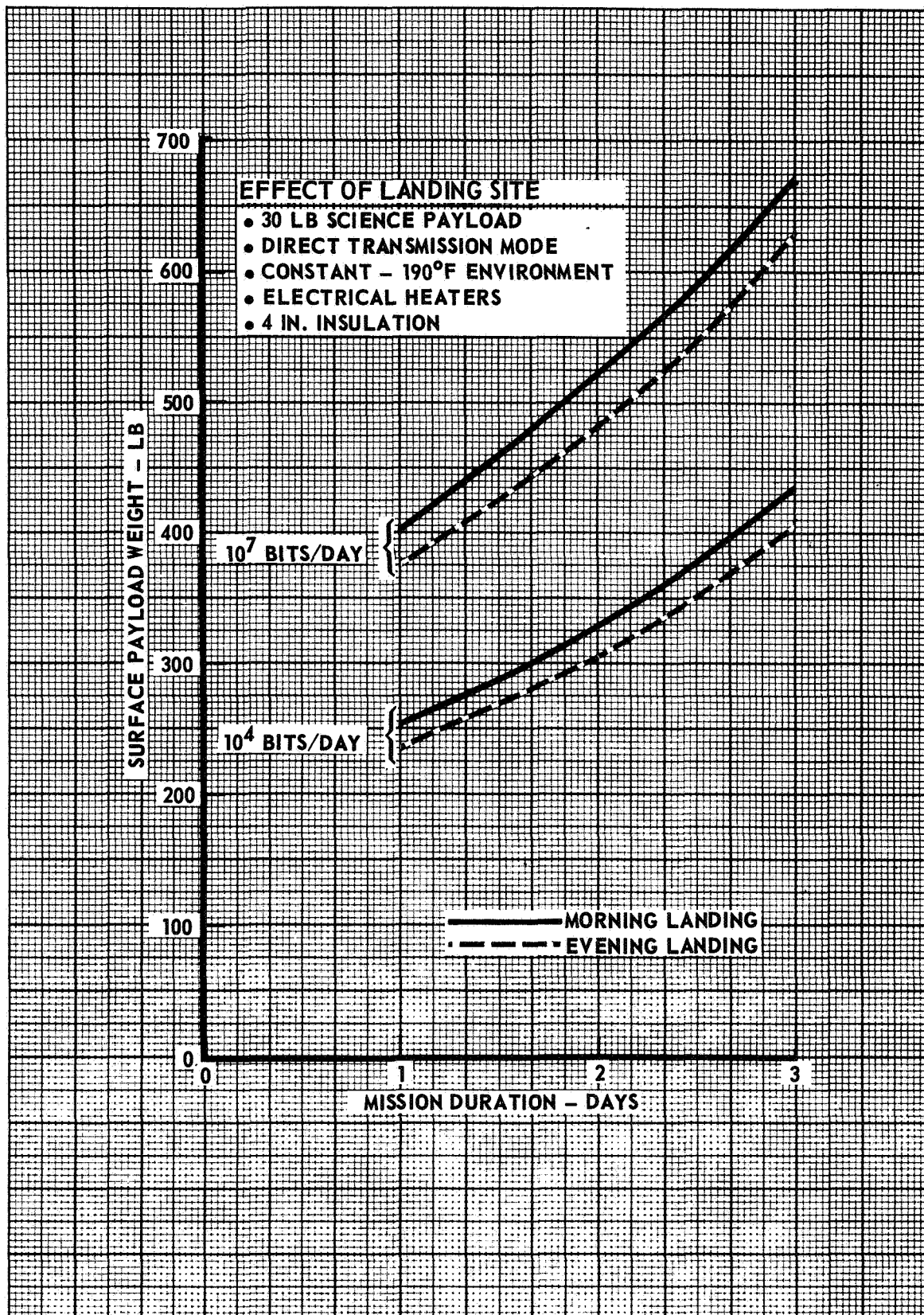
source of electrical power. Alternate power sources, solar panels or RTG, yield extended mission capability, hence were not considered for this section. For the initial examination, a vehicle of conservative design was used: insulation thickness was kept fixed at 4 inches, only electrical heaters were used, and a command reception link, and if required, a tape recorder were employed.

The principal effect of mission duration is that the surface payload weight increases rapidly with mission duration if the vehicle is designed for operation in a constant cold (-190°F) environment. This effect, shown in Figure 3.3.2-4, is prevalent regardless of the communication technique, and is relatively independent of the science payload. As mission duration increases, more energy is required for the operation of the equipment and, if electrical heaters are used, for vehicle heating. The increase in electrical energy requirements necessitates a larger battery, which in turn necessitates a larger overall vehicle. As the vehicle size increases, the radiating surface of the vehicle increases, necessitating more heat to maintain the temperature inside of the insulated compartment. The net result is that the heating requirements of the useful payload increase with mission duration because the vehicle increases in size. As will be discussed later, this avalanching relationship of surface payload weight and mission duration can be circumvented somewhat by different heating techniques, by the use of more insulation or by design to a less severe environmental criteria.

The effect of morning vs. evening landings is shown in Figure 3.3.2-5. Since only S-band (direct) data transmission is influenced by the landing time, relay designs are not shown. As can be seen in Figure 3.3.2-5, a penalty in surface payload weight results if a morning landing is used. This penalty is compounded by the penalties that must be paid to position the orbit to allow a morning landing.

3.3.2.4.2 Effect of communication mode: The basic alternatives of transmitting the data back to Earth are direct (S-band) and relay (UHF).



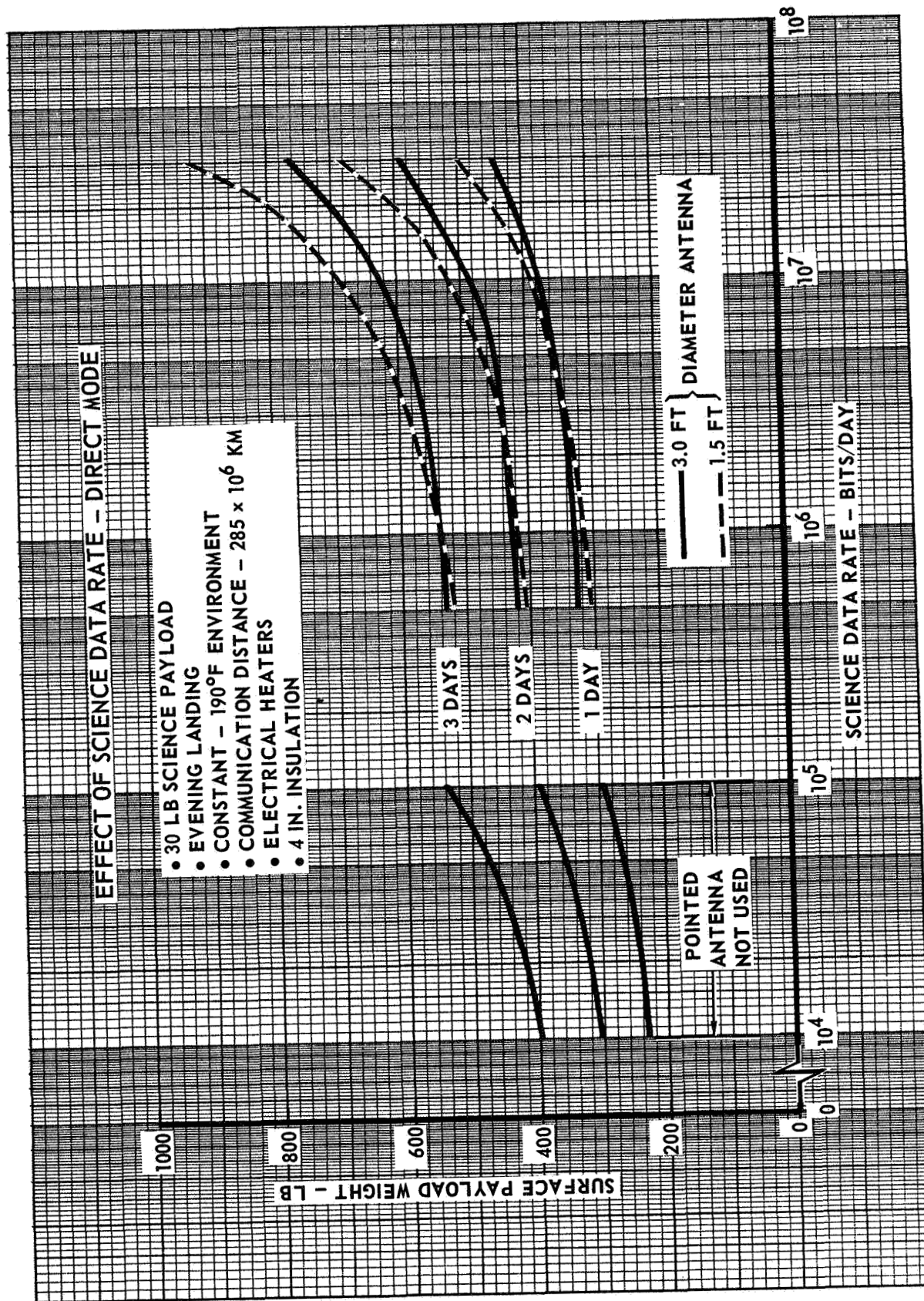


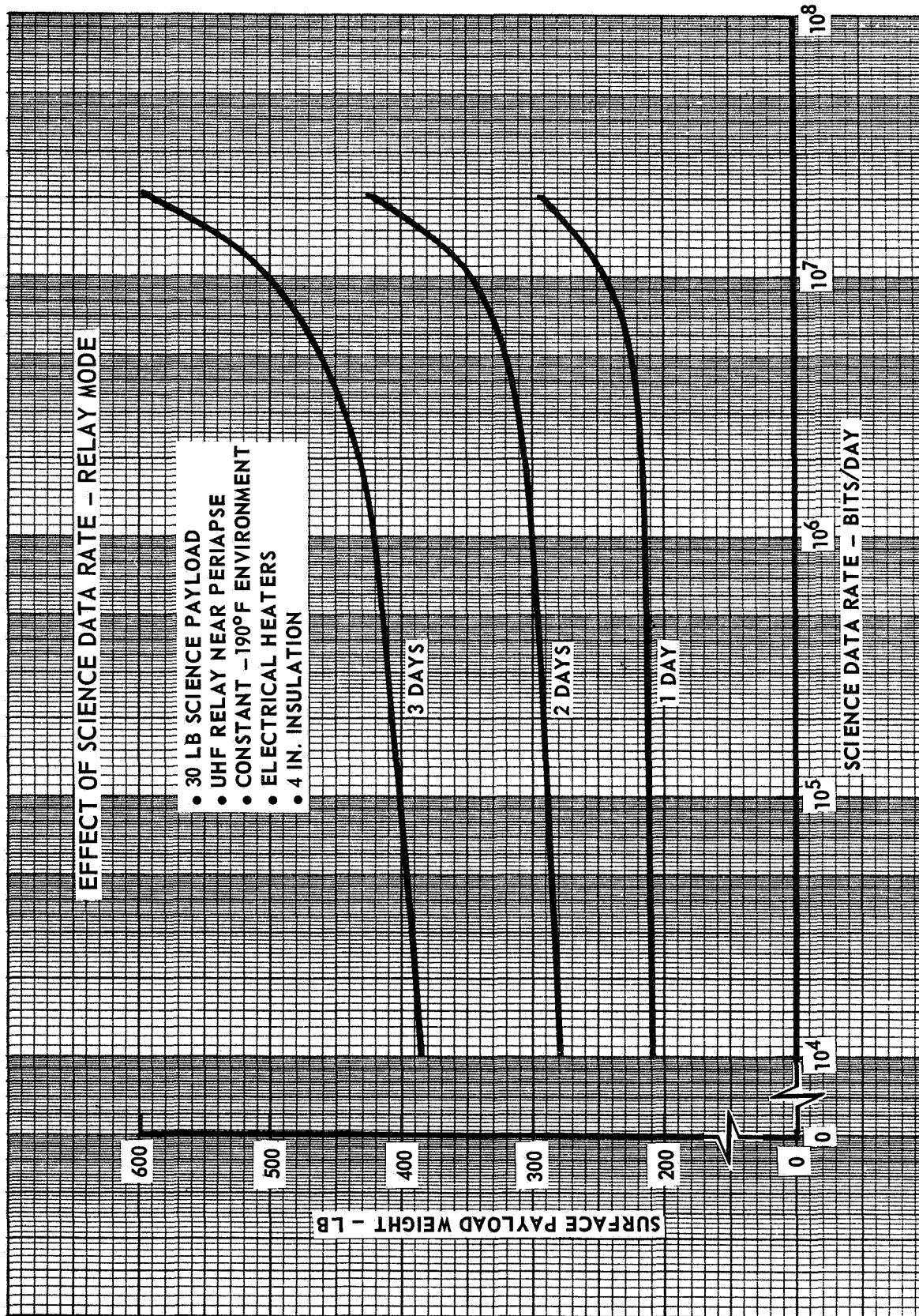
The direct mode of data transmission was employed for our design in the VOYAGER Phase B Studies, and allows the surface payload and the orbiter to operate independently of each other for the landed phase of the mission. Unfortunately, the long communications distance from the Earth to Mars (150 million to over 300 million kilometers) necessitates high levels of radiated power. As shown in Figure 3.3.2-6, the surface payload weight is a strong function of the amount of science data to be transmitted via the direct mode. This trend of weight vs. science data rate is valid regardless of the weight of the science payload, the environment encountered, or the thermal control design used.

Also shown in Figure 3.3.2-6 is a comparison of three different antenna configurations. A low gain antenna was employed for transmitter output power requirements below 50 watts RF. Two sizes of pointed parabolic antennas were considered: a 1.5 foot and a 3.0 foot diameter. Since these antennas must be designed to withstand the Martian winds, and launch and landing loads, a considerable amount of structure is required for the antenna mount, which holds and steers the antenna proper. The 3 foot diameter antenna system, which affords a large area to the potential winds, is about 21 pounds heavier than the 1.5 foot diameter system. The difference in the system weights makes the 1.5 foot diameter preferable for data rates below 10^7 bits per day. Beyond 10^7 bits per day, the 3 foot antenna is preferable on the basis of total surface payload weight, but because of its size, it may be restricted in its use due to the size limitations of the flight capsule.

The relay mode of data transmission can be used whenever the orbiter is in view of the payload. This system can be designed to transmit data either when the orbiter is near periapse or near apoapse. For short duration missions, only the periapse case was analyzed because it requires extremely small amounts of RF energy, does not require special high gain receiving antennas on the spacecraft, and is feasible for a few mission days.

As shown in Figure 3.3.2-7, the weight of the surface payload is not strongly related to the science data rate when the relay mode is used. Most of





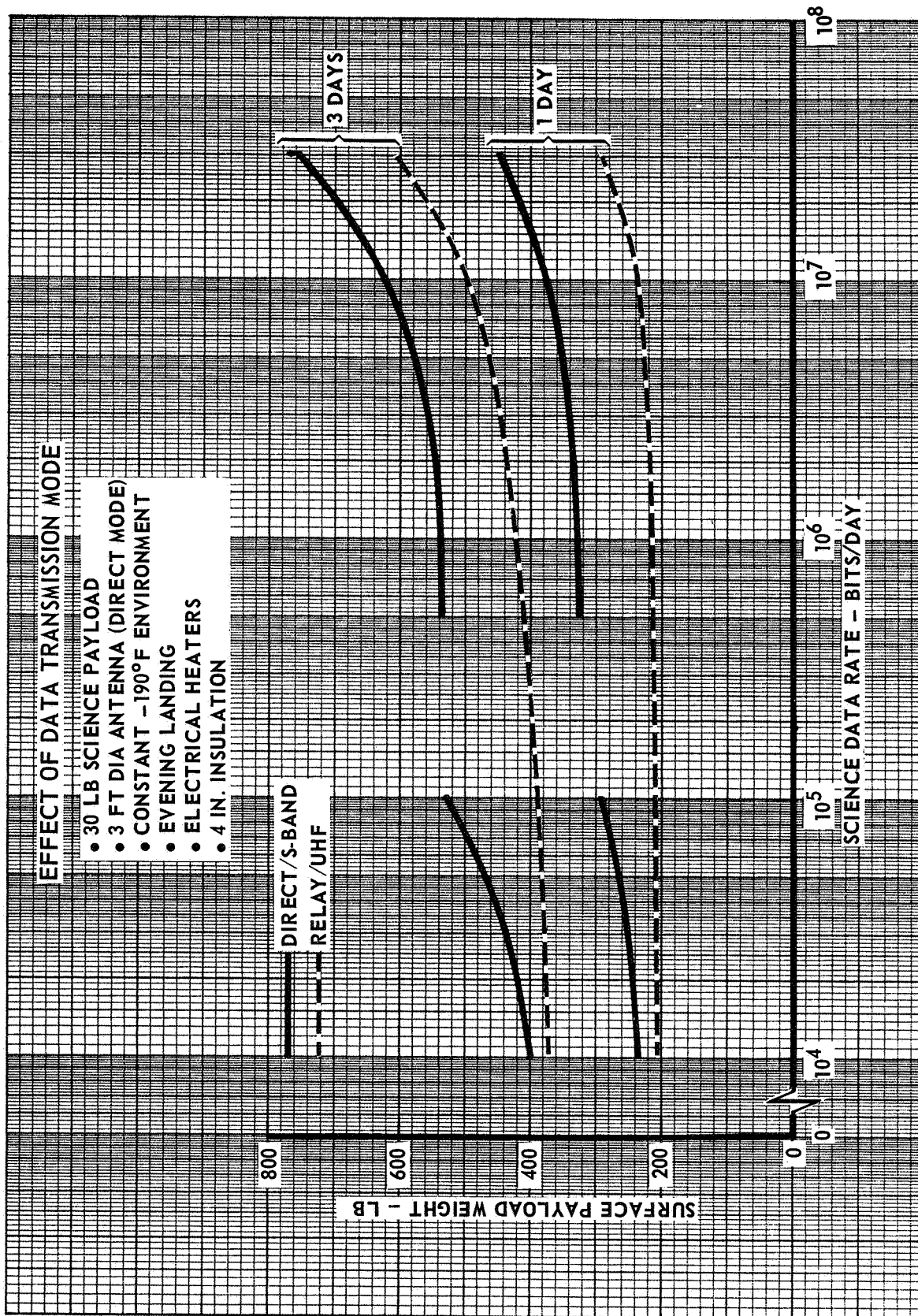
the weight increase at the high data rates (3×10^7 bits per day) is attributable to the large tape recorder used to store the imagery data.

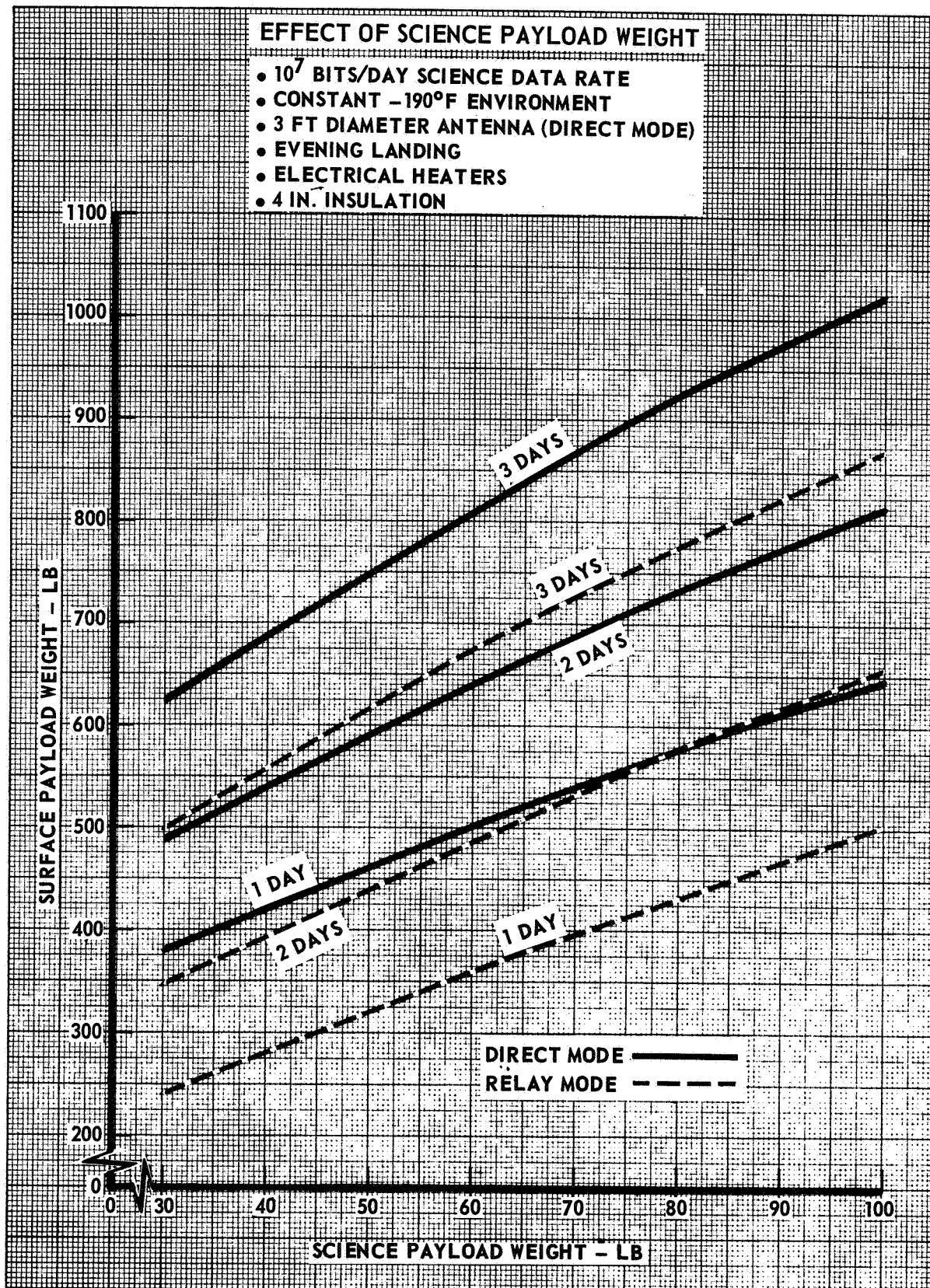
The direct and relay data transmission techniques are compared in Figure 3.3.2-8. The large advantage in total surface payload weight that the relay mode has over the direct mode is a clear indication of the preferred communication technique for short duration missions. This preference is dependent, however, upon the use of the orbiter for a few landed mission days.

3.3.2.4.3 Effect of the science payload: One characteristic of the science payload, data rate, was examined in the previous paragraph. The other major characteristic of the science payload is weight. The weight of the surface payload increases almost linearly with the weight of the science payload, regardless of mission length and communication techniques. It is clear from Figure 3.3.2-9 that one means of minimizing the surface payload weight, which in turn minimizes the flight capsule weight, is to use small science payloads (20 to 40 pounds).

3.3.2.4.4 Design alternatives: To this point in the study, the surface payloads were of conservative design; only electrical heaters, insulation thickness kept fixed at 4 inches, and worst case environment (constant -190°F). In addition, equipment such as a command link and a tape recorder was utilized, although it may not be necessary for a short duration mission. The designs to this point were heavy and used a large amount of batteries, especially for missions longer than one day. To refine the basic vehicles, different design techniques can be employed. In order to limit the amount of vehicles to be studied, a basic design which offers the most obvious advantage has been used. This nominal design uses a relatively small science payload (30 pounds), and the UHF relay mode of data transmission.

Command Link and Tape Recorder Evaluation - Although it is always advantageous to have some reprogramming capability after landing, the usefulness of a command link is diminished in a short duration mission. When the operational aspects of transmitting the data to Earth, having the data analyzed, making a decision on what programming changes are required, and then transmitting the necessary commands back to the surface payload are considered, it is unlikely





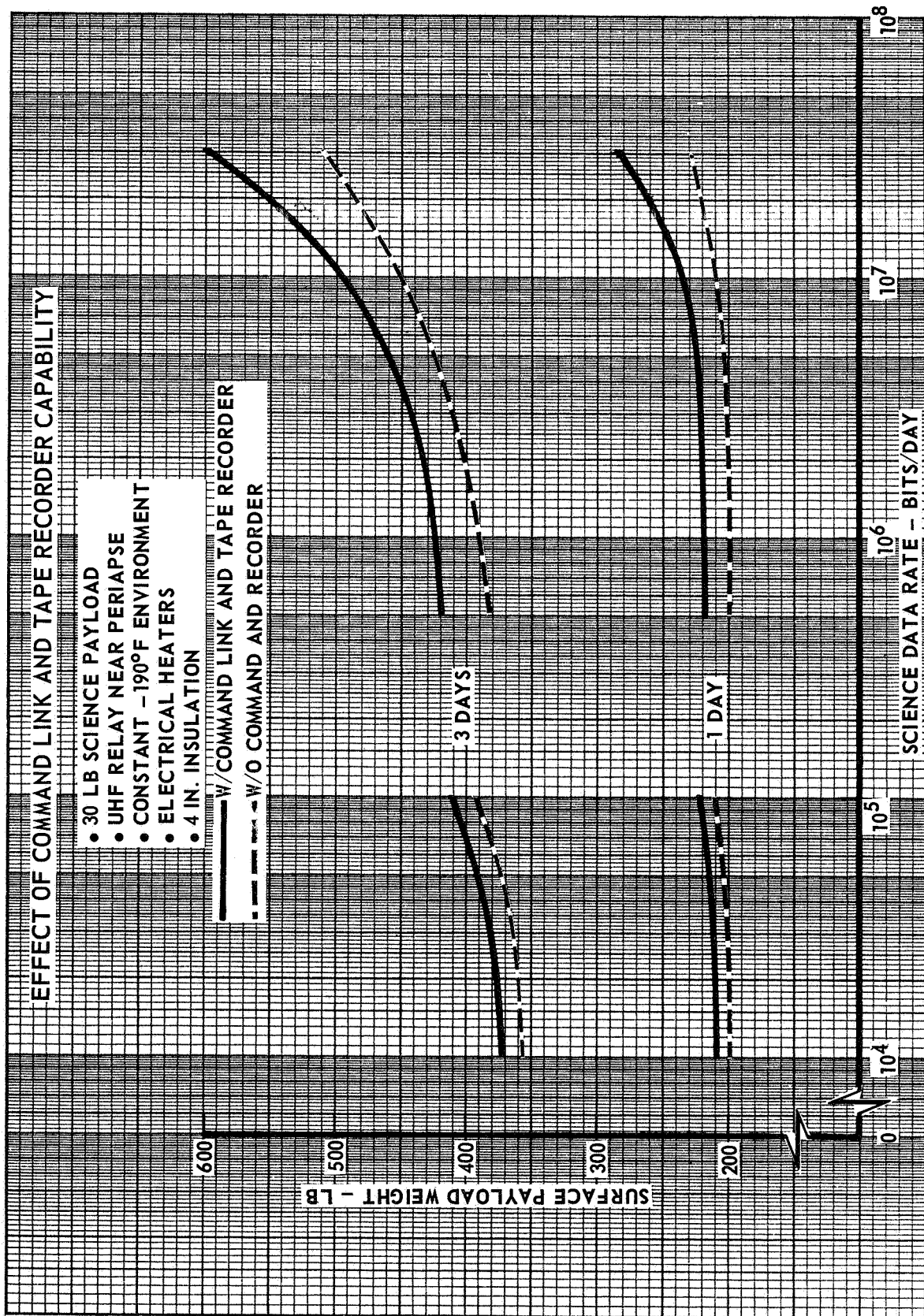
that any important change can be made before the mission is completed. This is especially true for missions of 1 or 2 days.

Aside from imaging data, all science and engineering data can be stored in a small ferrite core (or equivalent) memory. The imaging data from the facsimile camera can be transmitted in "real time" if the lighting at the landing site is acceptable. Since the landing site will probably be near the morning or evening terminator, the lighting (approximately 30° solar elevation) should be acceptable for "real time" transmission of imaging data when the orbiter is near periapse. Therefore, it is concluded that a tape recorder is not an absolute necessity.

A comparison of the surface payload weight for designs with and without command and tape recorder capability is shown in Figure 3.3.2-10. As can be noted, the difference in weight between the two concepts becomes significant when the data rate becomes large. A design without command or tape recorder capability would be somewhat less expensive, lighter, and, since sterilizable tape recorders are of doubtful status, less of a development risk. Therefore, the command link and tape recorder are not used for the following studies, which examine the effects of environment and thermal control design.

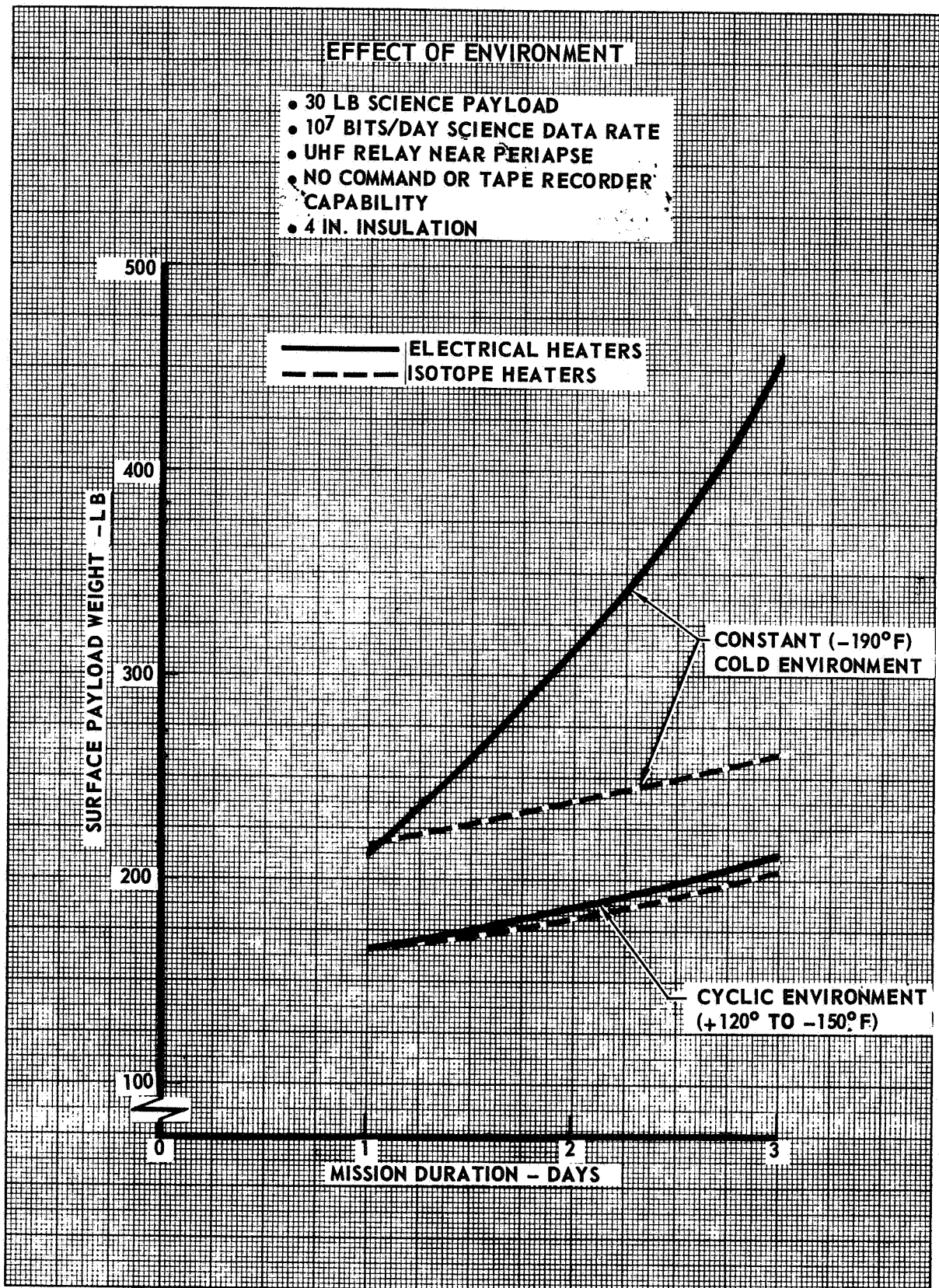
Environment Effects - A design for an environment of a constant -190°F was required for the VOYAGER Phase B study. The constant -190°F necessitates considerable insulation or large heating requirements. A constant cold environment, although possible, is not considered as probable as a cyclic environment. As shown in Figure 3.3.2-11, a vehicle designed for a cyclic environment is clearly lighter than one designed for a constant -190°F. This difference in weight results from a lower heating requirement because of a less severe environment. The average temperature for the nominal cyclic environment of Figure 3.3.2-11 is approximately -45°.

Thermal Control Alternatives - For the designs described to this point, a nominal thermal control design was used. A design option is insulation thickness. Thinner insulation results in less insulation weight, but a higher heating requirement.



3.3.2-21

FIGURE 3.3.2-10



The effect of insulation thickness, shown in Figure 3.3.2-12, is relatively small for one day missions. However, as mission duration increases, a lower surface payload weight can be achieved, for designs employing electrical heaters, by increasing the thickness of the insulation. This is particularly true for the continuous cold environment.

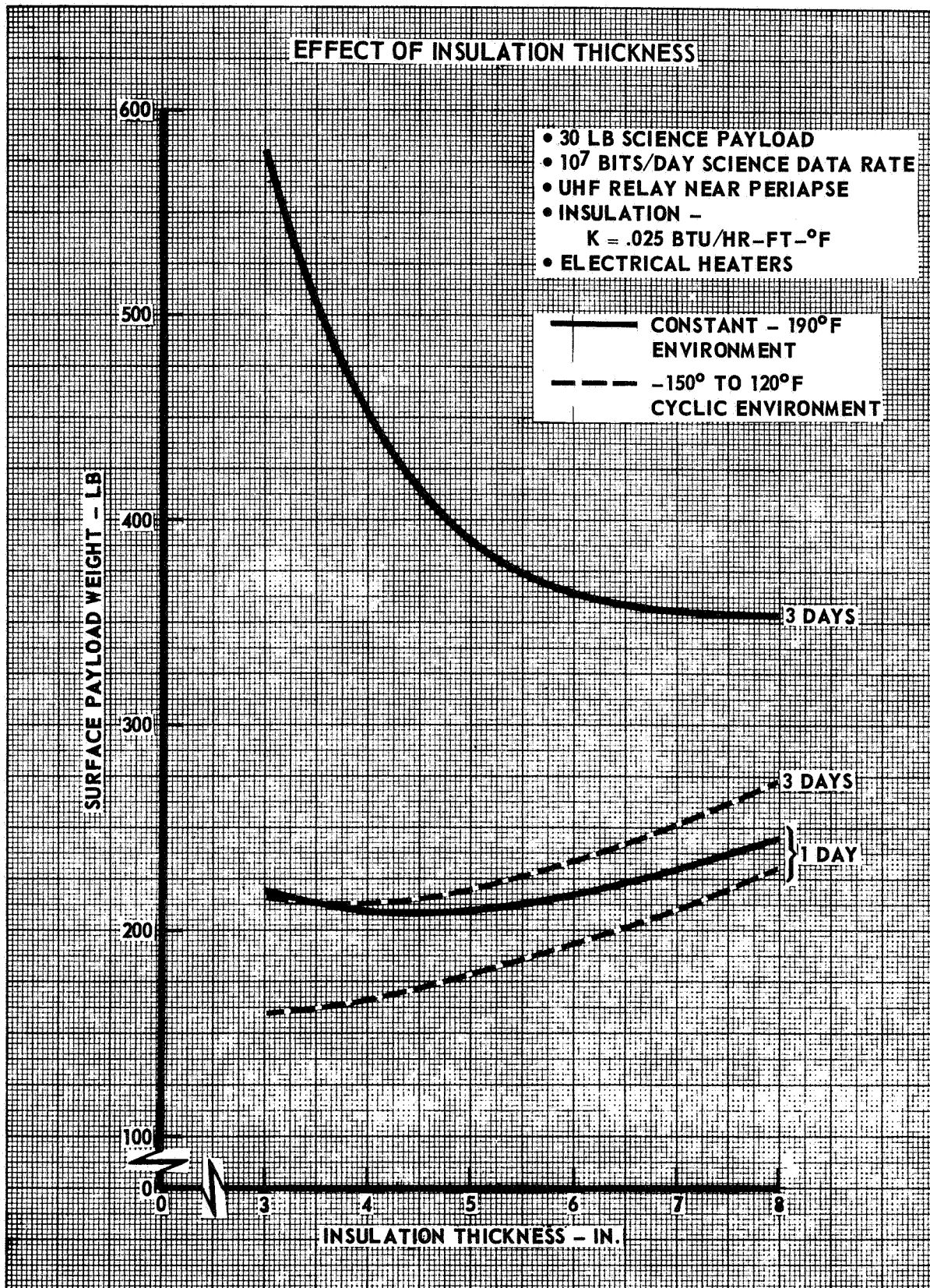
Another thermal control option is the use of radioisotopes as a source of heat. Aside from the cost and integration problems associated with isotopes, they present a technical problem; they provide a continuous, uncontrolled heat. Thus, the equipment can overheat if a warm Martian day occurs. Thermal control techniques include the use of heat pipes/radiators, which were used in the Phase B VOYAGER design, phase change material, or movement of the isotopes in and out of the insulated compartment of the surface payload as the equipment temperature varies. Phase change material, because of its lower cost and higher reliability, was used for this analysis.

As shown in Figure 3.3.2-11, isotopes offer an attractive means of minimizing the weight of the surface payload for missions longer than one day, and for a design environment of -190°F .

3.3.2.4.5 Conclusions ~ Short Duration Missions: A surface payload designed for a short duration mission should have the following characteristics:

- a) The principal communications mode should be UHF relay because the direct mode necessitates a much heavier, and larger, vehicle.
- b) The command link and tape recorder (for imaging data) should not be employed because they increase weight without providing any significant performance advantage.
- c) If the mission duration is longer than one day, isotope heaters or a design to a cyclic environment are required to obtain a realistic vehicle weight.

A surface payload that has the above characteristics, and satisfies the requirements for short duration missions (30 lb science, 10^7 bits/day, one diurnal cycle) can be constructed in several ways. A conservative approach is



to use electrical heaters, assume that a constant -190°F environment prevails, and design for one diurnal cycle. This approach, in a parametric version (Figure 3.3.2-13), weighs about 210 pounds. A lighter weight vehicle results if isotope heaters are used, and the design is for a nominal cyclic environment. This approach, expressed parametrically in Figure 3.3.2-14, weighs about 176 pounds. Once committed to isotopes as source of heat, mission durations of longer than one diurnal cycle can be achieved with modest increase (25 pounds per day) in surface payload weight.

3.3.2.5 Extended duration payloads (solar power). - A vehicle designed for an extended duration mission (90 days) must be evaluated with a set of criteria that is slightly different from that used for the short duration mission. With solar panels, one criterion becomes the area of the panels. Excessive area creates problems in deployment after the vehicle has landed and in storage in the aeroshell. To minimize these problems, it is desirable to keep the solar panel area less than 20 ft^2 . The maximum feasible area is about 40 ft^2 , although this maximum is a function of the overall flight capsule design.

Another criterion to be applied to extended duration missions is the reliability aspects of the mission length. Most of the equipment is subject to increased probability of failure as the mission progresses, but is not highly susceptible to deterioration or "wear-out." The exceptions are the solar panels and the batteries. The solar panel performance may deteriorate with time due to the abrasive effects of the Martian wind, sand and soil, and the accumulation of sand or soil upon the panels. For this analysis, it has been assumed that any deterioration of the solar panels is negligible.

Batteries (Ag-Zn) are subject to wear-out due to discharge/charge cycling. The depth of discharge (% of battery capacity expended before it is recharged) is the dominant factor in the life of the battery. To obtain a 90 day life, the depth of battery discharge must be limited to less than 60% of the battery's capacity.

3.3.2.5.1 Initial design considerations: Although a vehicle designed for an extended duration mission is evaluated differently than one designed for a

SURFACE PAYLOAD – BATTERY POWER

- MINIMUM MISSION
- NOMINAL ENVIRONMENT
- 1 DAY MISSION
- PARAMETRIC COMPUTATION

SCIENCE PAYLOAD: 30 LB, 10 ↑ 7 BITS/DAY

BASIC DESIGN CONSTRAINTS

MISSION LENGTH: 24.6 HR
 BASIC POWER SOURCE: BATTERIES
 ENVIRONMENTAL LIMITS: 120 TO -190°F
 COMMUNICATION DISTANCE: 285 x 10 ↑ 6 KM
 MAXIMUM ANTENNA SIZE: 3 FT DIAMETER
 TRANSMITTING FREQUENCY: 400 MHz
 TRANSMITTING TIME
 LOW RATE RADIO: 0 HR/DAY
 HIGH RATE RADIO: .33333 HR/DAY

SIGNIFICANT CHARACTERISTICS

TOTAL PAYLOAD WEIGHT: 210.219 LB
 RADIO LINKS
 LOW RATE RADIO: 0 BPS, 0 WATTS RF
 HIGH RATE RADIO: 19,198. BPS, 15.3584 WATTS RF
 HIGH GAIN ANTENNA NOT REQUIRED
 BATTERY WEIGHT: 46.3241 LB
 EXT DIMENSIONS: 14.6487 x 34.0524 x 39.004 IN.

WEIGHT AND POWER SUMMARY

ITEM	WEIGHT (LB)		POWER (WATTS)	
	INTERNAL	EXTERNAL	DAY	NIGHT
SCIENCE	7.6	22.4	7.56	7.56
SEQUENCER	7.21	0	5.91	2.43
LOW GAIN ANTENNA	0	0	0	0
RECEIVE ANTENNA	0	0	0	0
HIGH GAIN ANTENNA	0	0	0	0
LR TRANSMITTER	0	0	0	0
HR TRANSMITTER	4.48	0	70.3	0
CMD RECEIVER/DECODER	0	0	0	0
TELEMETRY	6.69	0	5.01	5.01
ENGINEERING INSTRUMENTATION	2.1	2.1	2.1	2.1
CORE STORE	2.27	0	.431	.431
TAPE RECORDER	0	0	0	0
EED MODULES	0	6.5	0	0
SWITCHING LOGIC	6.82	0	0	0
BATTERIES	46.3	0	0	0
SOLAR PANELS	0	0	0	0
RTG	0	0	0	0
INSULATION	0	32.5	0	0
PHASE CHANGE	0	0	0	0
HEAT PIPES	0	0	0	0
ISOTOPES	0	0	0	0
WIRING, ETC.	18.3			
STRUCTURE	44.9			

SURFACE PAYLOAD – BATTERY POWER

- MINIMUM MISSION
- MODIFIED ENVIRONMENT
- 1 DAY MISSION
- PARAMETRIC COMPUTATION

SCIENCE PAYLOAD: 30 LB, 10^7 BITS/DAY

BASIC DESIGN CONSTRAINTS

MISSION LENGTH: 24.6 HR
 BASIC POWER SOURCE: BATTERIES
 ENVIRONMENTAL LIMITS: 120 TO -150°F
 COMMUNICATION DISTANCE: 285×10^3 KM
 MAX ANTENNA SIZE: 1.5 FT DIAMETER
 TRANSMITTING FREQUENCY: 400 MHz
 TRANSMITTING TIME
 LOW RATE RADIO: 0 HR/DAY
 HIGH RATE RADIO: .333333 HR/DAY

SIGNIFICANT CHARACTERISTICS

TOTAL PAYLOAD WEIGHT: 175.684 LB
 RADIO LINKS
 LOW RATE RADIO: 0 BPS, 0 WATTS RF
 HIGH RATE RADIO: 19,197.8 BPS, 15.3582 WATTS RF
 HIGH GAIN ANTENNA NOT REQUIRED
 BATTERY WEIGHT: 18.7034 LB
 EXT DIMENSIONS: 14.3551 x 32.9018 x 37.6347 IN.

WEIGHT AND POWER SUMMARY

ITEM	WEIGHT (LB)		POWER (WATTS)	
	INTERNAL	EXTERNAL	DAY	NIGHT
SCIENCE	7.6	22.4	7.56	7.56
SEQUENCER	7.21	0	5.91	2.43
LOW GAIN ANTENNA	0	0	0	0
RECEIVE ANTENNA	0	0	0	0
HIGH GAIN ANTENNA	0	0	0	0
LR TRANSMITTER	0	0	0	0
HR TRANSMITTER	4.48	0	70.3	0
CMD RECORDER/DECODER	0	0	0	0
TELEMETRY	6.69	0	5.01	5.01
ENGINEERING INSTRUMENTATION	2.1	2.1	2.1	2.1
CORE STORE	2.27	0	.431	.431
TAPE RECORDER	0	0	0	0
EED MODULES	0	6.5	0	0
SWITCHING LOGIC	5.44	0	0	0
BATTERIES	18.7	0	0	0
SOLAR PANELS	0	0	0	0
RTG	0	0	0	0
INSULATION	0	30.2	0	0
PHASE CHANGE	5.44	0	0	0
HEAT PIPES	0	0	0	0
ISOTOPES	2	0	0	0
WIRING, ETC.	12.6			
STRUCTURE	40			

short duration, the same basic design alternatives are available. In order to minimize the number of alternatives, certain of them can be eliminated from the analysis.

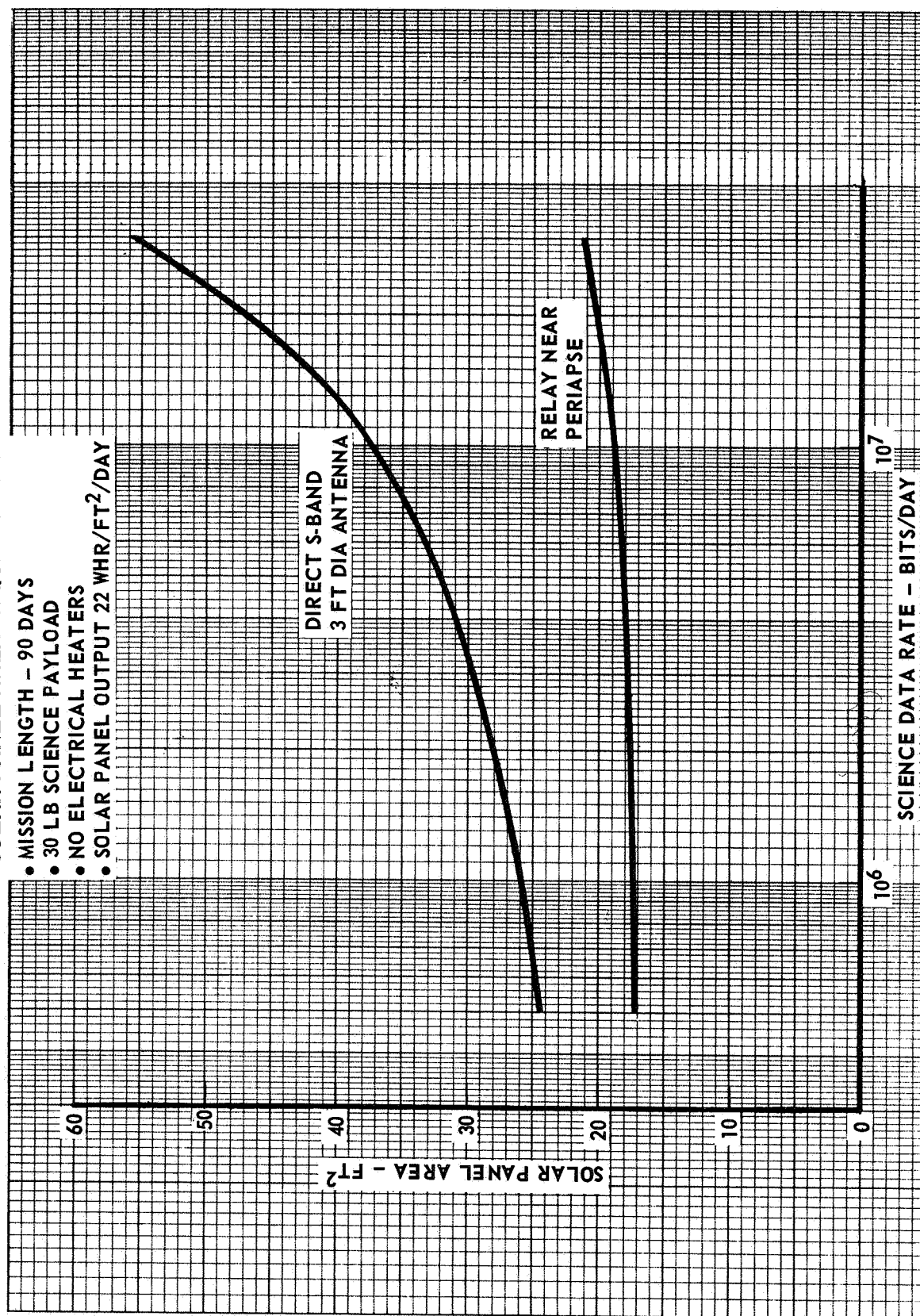
Telecommunications Considerations - A comparison of relay and direct transmission modes for short duration missions showed that the relay mode is preferred on a weight basis. The large weight advantage results from a lower antenna weight, and lower electrical energy requirements, hence lower battery weight. Figure 3.3.2-15 shows that the direct mode requires considerably more solar panel area than the relay mode. This comparison was made for the most favorable operating condition of the direct mode; all data transmission power was less than output of the solar panels during the entire transmission period, hence electrical energy was used at maximum efficiency. Also, the probable shadowing of the solar panels by the 3 foot diameter antenna, thus reducing the output of the panels, was not considered.

On the basis of solar panel area, the relay mode is preferred. However, daily use of the relay mode, whether employed when the orbiter is near periapse or apoapse, is dependent upon the synchronization of the orbit period with the diurnal cycle of Mars. Probably, after a few days of landed operations, the orbit will be altered to allow more efficient mapping of the surface by the orbiter. Therefore, the relay mode near periapse will probably be unusable on a regularly scheduled basis after a few days of landed operations. Several weeks of operation are possible if the relay is employed near apoapse. The requirements of this study state that 10^8 bits per mission is a design goal. The relay mode can easily obtain this goal if the orbit remains synchronous for 3 to 10 days after the start of landed operations. Once the imaging experiment is completed, the remainder of the science data is extremely low (a few thousand bits per day). This low data rate can be transmitted via the direct mode with a low gain antenna. Therefore, for the extended duration missions, the following groundrules were used:

- (a) All imaging data would be transmitted in the first few days of the mission using the relay mode. Further operation of the relay mode would be dependent upon the availability of the orbiter.

SOLAR PANEL AREA REQUIREMENTS

- MISSION LENGTH - 90 DAYS
- 30 LB SCIENCE PAYLOAD
- NO ELECTRICAL HEATERS
- SOLAR PANEL OUTPUT 22 WHR/FT²/DAY



- (b) After the imaging portion of the experimental program was completed, all remaining data would be transmitted via the direct mode using a low gain antenna.

Another telecommunications consideration is the command link. For short duration missions a command link is of limited value. However, for extended duration missions considerable re-programming of the surface payload will probably be desired throughout the mission. Therefore, all configurations in this section will have command capability.

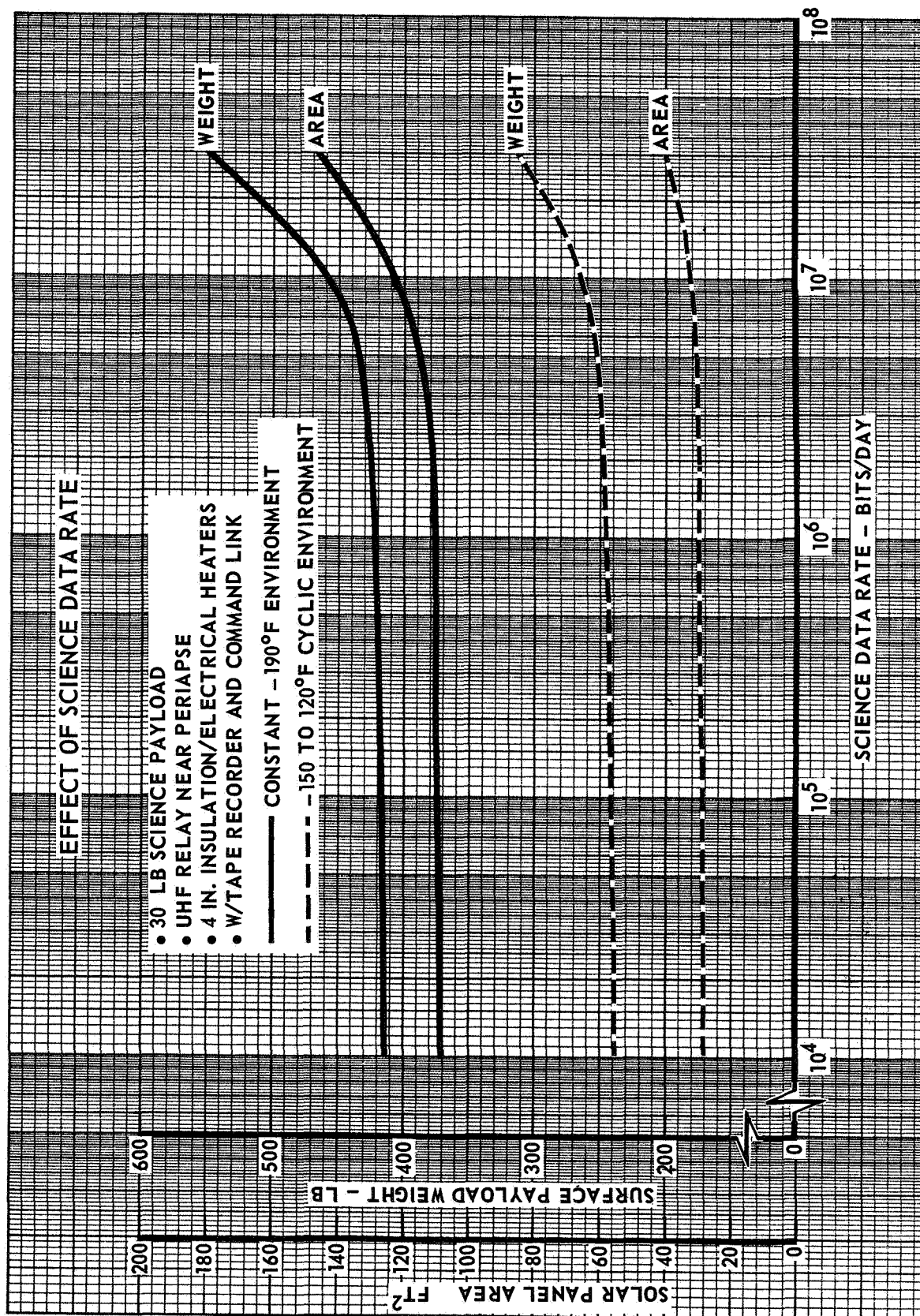
Environmental Considerations - Since it is probable that an extended duration mission will be the first landed operations on Mars, it is assumed that no further data is available toward the definition of a Mars environment. Therefore, the implications of the constant cold and nominal cyclic environments must be examined.

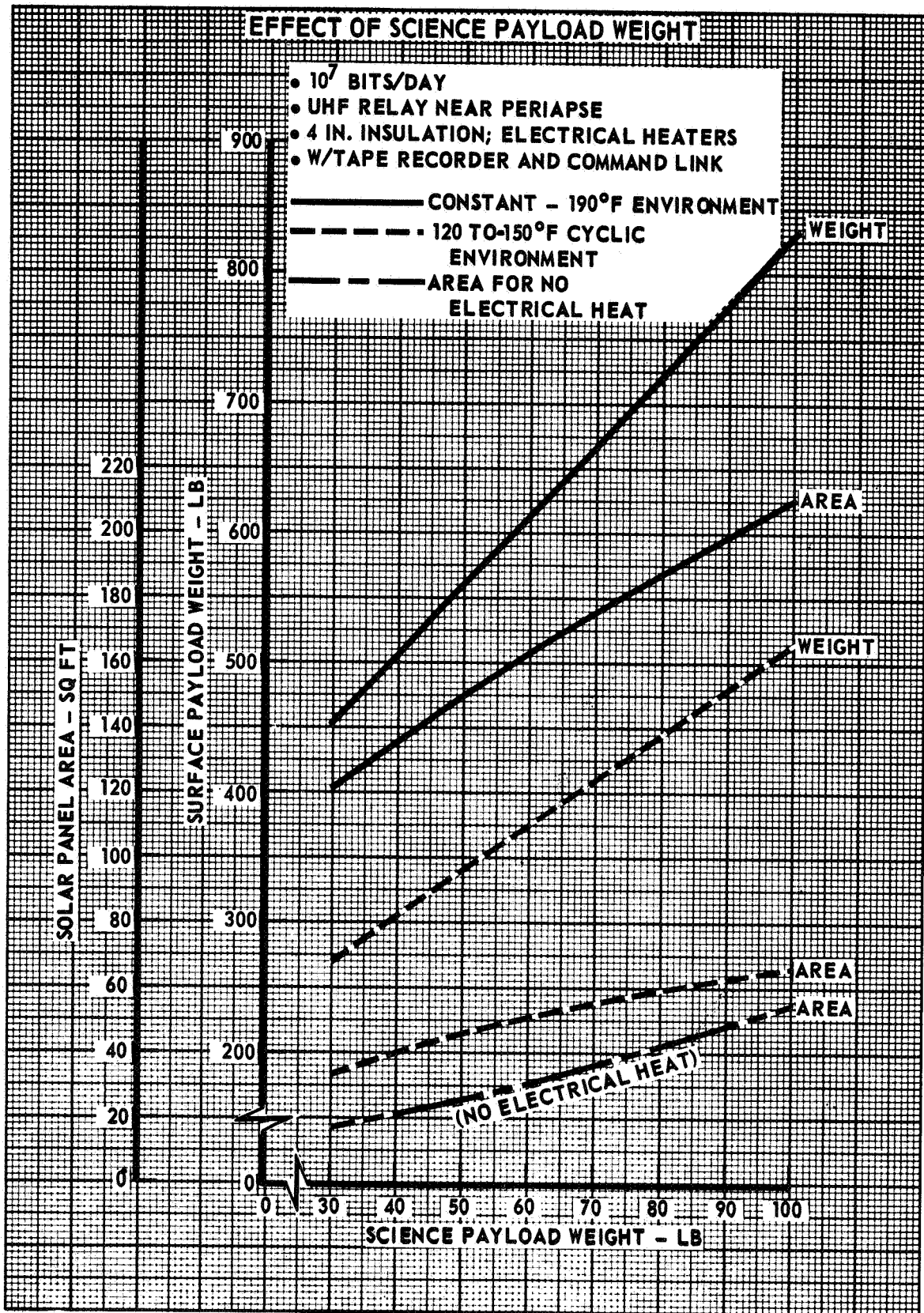
Thermal Control - For reasons of cost and/or development problems, the use of heat pipes/radiators for extended duration missions will be excluded.

3.3.2.5.2 Effect of the science payload: Since mission duration (aside from reliability considerations) and telecommunications techniques are not major variables for extended duration missions, the first concern was to examine the effects of the science payload. As shown in Figure 3.3.2-16, the data rate of the science payload does not have a significant effect upon either the weight or the solar panel area of the surface payload. This result is similar to that obtained for short duration designs, because in both cases the electrical energy requirements for data transmission are small when the relay mode is employed.

The weight of the science payload has a moderate effect upon both vehicle weight and solar panel area. The data of Figure 3.3.2-17 shows an almost linear increase in vehicle weight and solar panel area with science weight. Although Figures 3.3.2-16 and 3.3.2-17 do not indicate any unexpected trend in either science weight or data rate, they do show the impact of the environment and thermal control design.

3.3.2.5.3 Effect of the environment and thermal control design: To this point conservative designs (electrical heaters and 4 inches of insulation) have



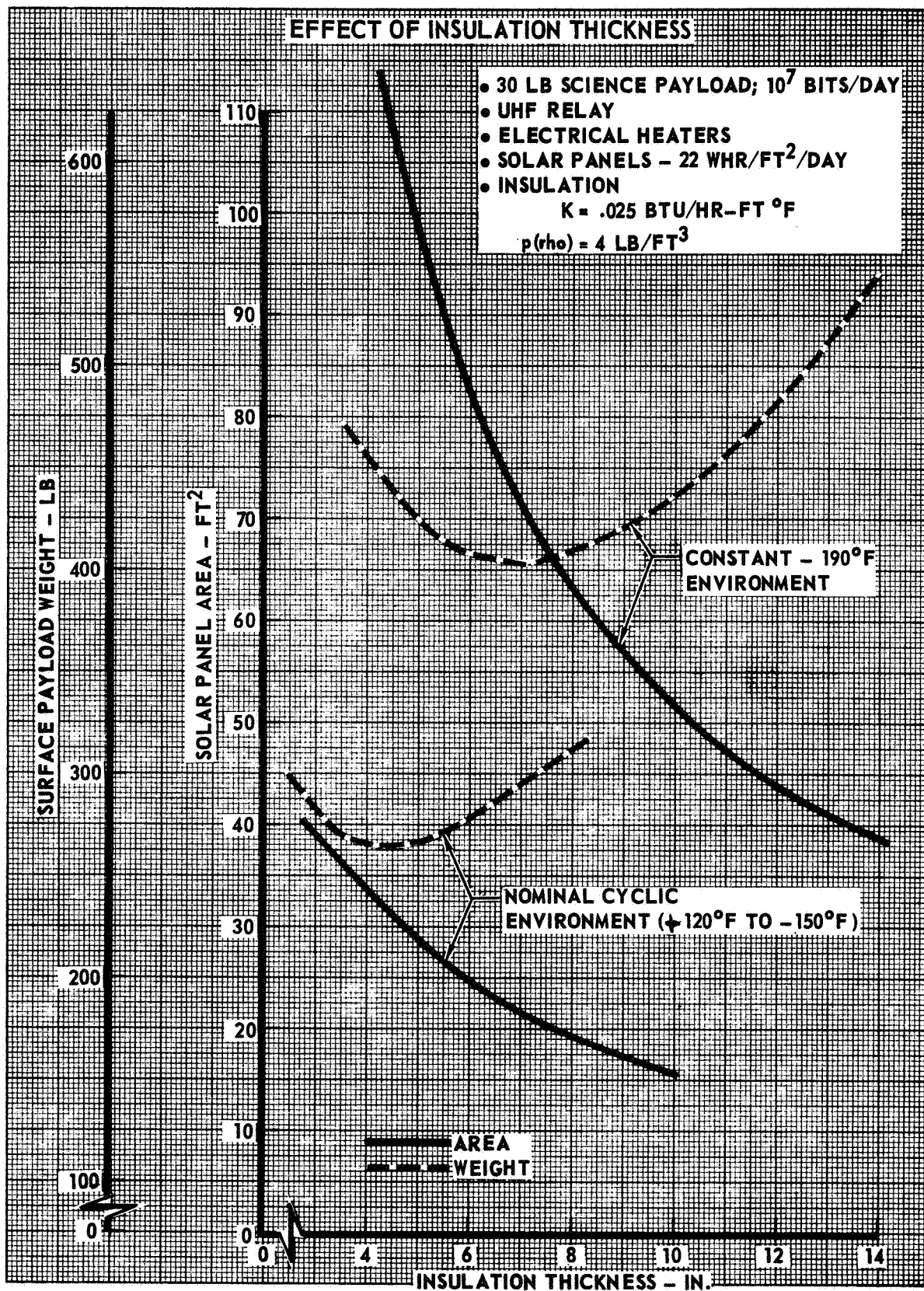


been employed. However, Figures 3.3.2-16 and 3.3.2-17 show that solar panel area and vehicle weight become overwhelming if these conservative practices are used for a design at the continuous cold environment. In fact, these practices result in large solar panel areas even if the design is for a cyclic environment. It is therefore clear that the thermal control design must be refined.

When thermal control provisions are excluded entirely, the resulting values for solar panel area and vehicle weight are practical. As shown in Figure 3.3.2-17, the solar panel area remains under 40 ft² for science payloads up to 75 pounds if electrical heaters are not used. Unfortunately, electrical heaters are desirable because they are controllable and therefore minimize the probability of equipment overheating on a "warm" day. One technique of reducing the amount of electrical heat needed is to increase the amount of insulation used on the basic equipment compartment of the payload. As insulation thickness is increased, the solar panel area required decreases. However, as is shown in Figure 3.3.2-18, the minimum payload weight does not occur at the point of minimum solar panel area. Also, if the vehicle is designed for a constant cold (-190°F) environment, even large amounts of insulation do not reduce the solar panel area to a reasonable value. It is concluded that in a constant cold environment, electrical heaters alone are impractical.

Another design alternative is to use radioisotope heaters to supplement, or totally eliminate, the electrical heaters. The initial effect of isotope heaters is to reduce solar panel area and payload weight. This trend is shown in Figure 3.3.2-19. However, the isotopes required to provide all of the heating requirements for a constant cold environment will cause the vehicle to overheat if a cyclic environment is encountered unless supplemental heat rejection is employed. A practical design alternative is to use isotope heaters for the cyclic environment, and to supplement them with electrical heaters for the continuous cold environment.

3.3.2.5.4 Effect of landing site: Other parameters affecting the surface payload for extended missions are the landing site latitudes and the date of landing. The landing site and date of landing provide the basic constraints on the relative power and energy output of the solar panels. As is shown in



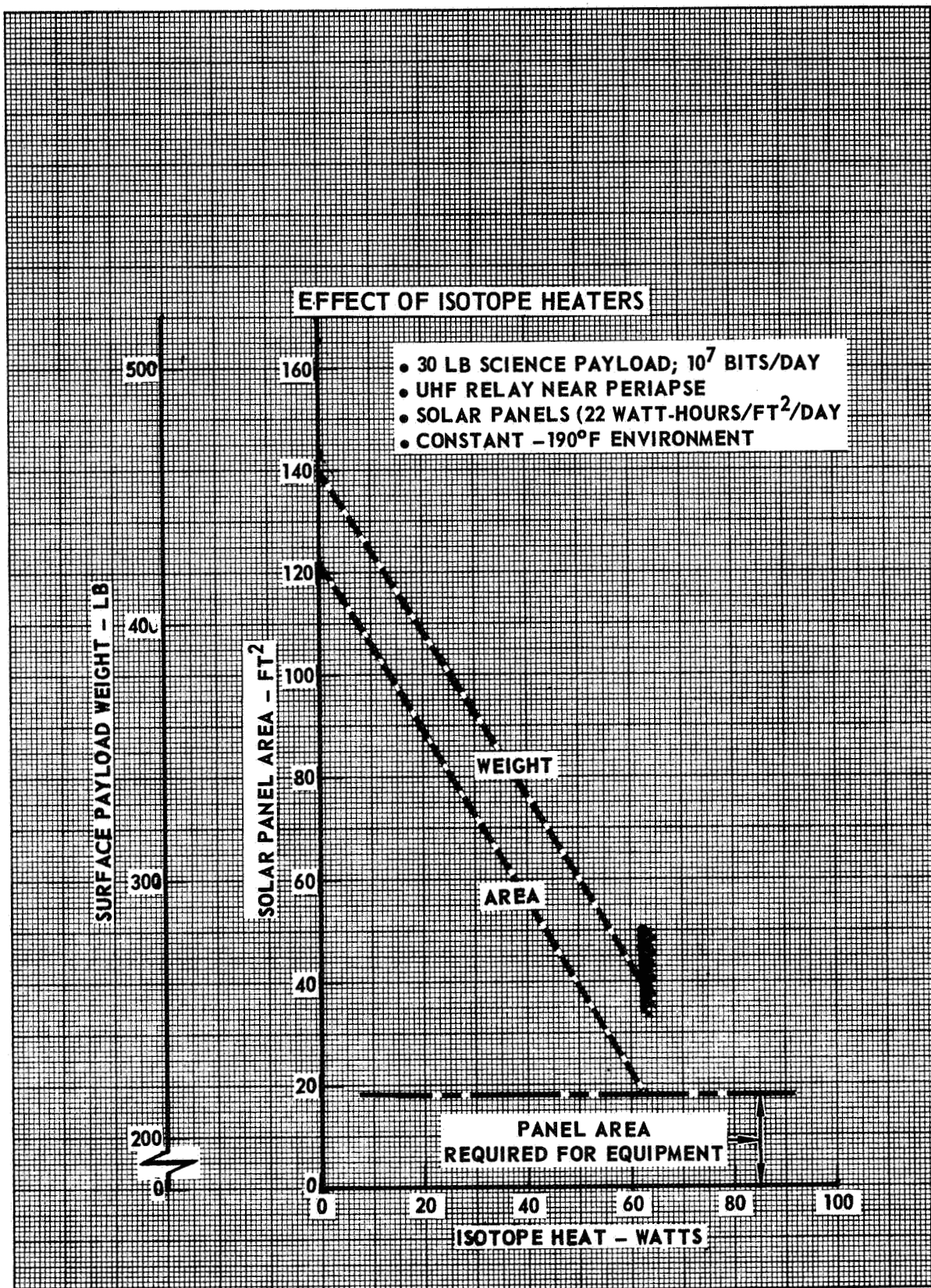


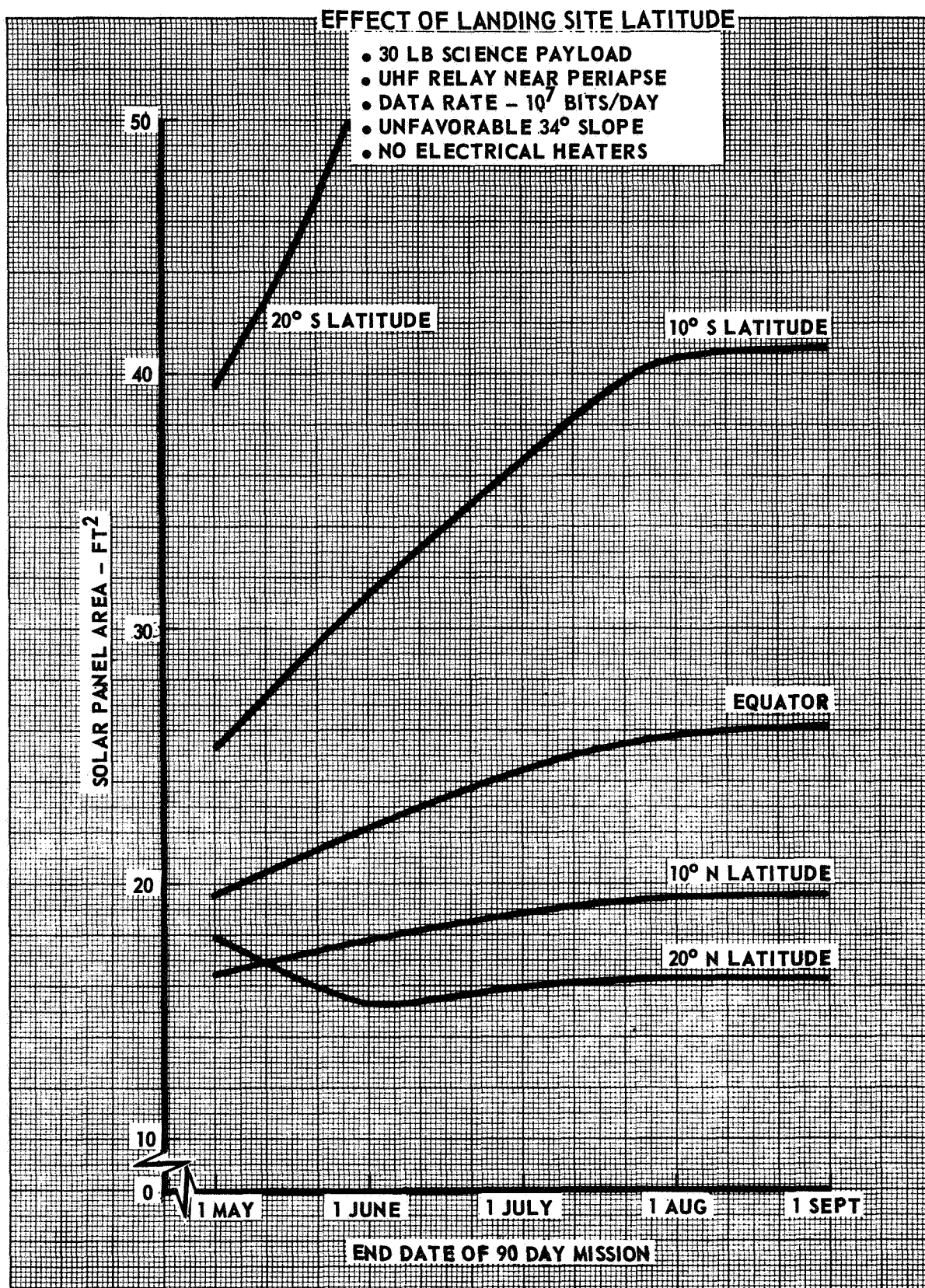
Figure 3.3.2-20, the landing site and date have a considerable influence on the solar panel area requirements, even if electrical heaters are not used. For the 1973 opportunity, sites north of the Martian equator are preferred, to minimize the area of the solar panels.

3.3.2.5.5 Battery considerations: For designs employing isotope heaters, the worst case depth-of-discharge is approximately 55% regardless of what environment is used as a design criteria. Since 55% is below our limit of 60%, it is not a matter of great concern.

If electrical heaters are employed, the depth of battery discharge is a function of the environment and the properties of the insulation. If the surface payload is designed for operation in a continuous cold (-190°F) environment, the worst case depth of discharge is about 55% before the solar panels start recharging the battery. If the vehicle is designed for the cyclic environment ($+120^{\circ}\text{F}$ to -150°F), the depth of discharge is a function of the properties of the insulation. For a highly insulated vehicle, typically 6 inches of insulation whose performance factor is $.025 \text{ Btu/hr-ft-}^{\circ}\text{F}$, the battery depth of discharge is about 60%. If the insulation is decreased to 3 inches, more heat is required. Since the heating requirements at night are intensified, the battery discharges, at night, at a higher relative rate. The worst case depth of discharge for this condition is about 70%, which exceeds our limit of 60%. Therefore, for typical surface payloads designed only for a cyclic environment, and using electrical heaters and minimum insulation, greater battery capacity, hence more vehicle weight, is required than the basic parametric computations have shown to this point.

3.3.2.5.6 Conclusions - extended duration missions: A surface payload designed for an extended duration should have the following characteristics;

- (a) The principal communications mode should be the UHF relay because the direct mode for high data rates necessitates large, and heavy, solar panels. This principal mode should be supplemented with a low rate direct link to transmit non-imaging data throughout the 90 day mission after the orbiter is desynchronized. An alternate approach, although not fully explored, would be to utilize the relay mode to



transmit when the orbiter is near apoapse. After the orbit is de-synchronized, a relay near apoapse remains operational for about 50 days if the orbit period is approximately 24.42 hours.

- (b) The command link should be used to allow periodic updating of the landed mission profile.
- (c) Electrical heaters should not be used as the primary means of heating the vehicle in a continuous cold environment because the solar panel area becomes exceptionally large.

A surface payload that has the above characteristics and satisfies the requirements for extended duration missions (30 lb science, 10^8 bits total, 90 day lifetime) can be constructed in several ways. A conservative approach is to insure that at least one day of operational life is possible regardless of the environment encountered. This approach, in parametric version (reference Figure 3.3.2-21), weighs about 278 pounds, has less than 20 ft^2 of solar panel, uses isotope heaters for operation in a cyclic environment, and a combination of isotope and electrical heaters if one day of continuous cold is encountered. As can be noted in Figure 3.3.2-21, no tape recorder is used, thus allowing a lower vehicle weight but limiting the imaging data to one lighting condition. A lighter surface payload results if no provisions are made for one continuous cold day. This approach, expressed parametrically in Figure 3.3.2-22, weighs about 226 pounds, and is extremely similar to the one shown in Figure 3.3.2-21, except for the lower battery weight. Neither of the two parametric designs have accounted for solar panel deterioration, and they are capable of 90 day missions only if the landing site is north of the equator.

3.3.2.6 Long duration missions (RTG power). - The surface payload for a long duration mission (6 months to 2 years) will, by previous definition, use an RTG as the principal source of electrical power. As for extended duration missions, the basic vehicle design considerations must be analyzed with respect to the mission goals before a parametric analysis can be initiated.

3.3.2.6.1 Initial design considerations: The weight of the surface payload and its major supporting systems remains as a principal means of evaluating

SURFACE PAYLOAD – SOLAR POWER

- MINIMUM SCIENCE
- NOMINAL ENVIRONMENT
- 90 DAY MISSION
- PARAMETRIC COMPUTATION

SCIENCE PAYLOAD: 30 LB, 10 ↑ 7 BITS /DAY

BASIC DESIGN CONSTRAINTS

MISSION LENGTH: 90 DAYS

BASIC POWER SOURCE: SOLAR PANELS

ENVIRONMENTAL LIMITS: 120 TO -150 °F (1 DAY AT -190°F)

COMMUNICATION DISTANCE: 285 x 10 ↑ 6 KM

MAXIMUM ANTENNA SIZE: 1.5 FT DIAMETER

TRANSMITTING FREQUENCY: 400 MHz*

TRANSMITTING TIME

LOW RATE RADIO: 0 HR/DAY*

HIGH RATE RADIO: .3333 HR/DAY

SIGNIFICANT CHARACTERISTICS

TOTAL PAYLOAD WEIGHT: 278.232 LB

RADIO LINKS

LOW RATE RADIO: 0 BPS, 0 WATTS RF*

HIGH RATE RADIO: 19,199.7 BPS, 15.3598 WATTS RF

HIGH GAIN ANTENNA NOT REQUIRED

BATTERY WEIGHT: 46.135 LB

SOLAR PANEL AREA: 17.6782 SQ FT

EXT DIMENSIONS: 15.4232 x 37.0872 x 42.6155 IN.

WEIGHT AND POWER SUMMARY

ITEM	WEIGHT (LB)		POWER (WATTS)	
	INTERNAL	EXTERNAL	DAY	NIGHT
SCIENCE	7.6	22.4	2.1	2.1
SEQUENCER	7.21	0	5.91	2.43
LOW GAIN ANTENNA	0	1*	0	0
RECEIVE ANTENNA	0	.598	0	0
HIGH GAIN ANTENNA	0	0	0	0
LH TRANSMITTER	9*	0	0	0
HR TRANSMITTER	4.48	0	70.4	0
CMD RECEIVER/DECODER	5.2	0	.5	.5
TELEMETRY	7.87	0	1.64	1.64
ENGINEERING INSTRUMENTATION	2.1	2.1	2.1	2.1
CORE STORE	2.27	0	.431	.431
TAPE RECORDER	0	0	0	0
EED MODULES	0	6.8	0	0
SWITCHING LOGIC	11.8	0	0	0
BATTERIES	46.1	0	0	0
SOLAR PANELS	0	26.5	0	0
RTG	0	0	0	0
INSULATION	0	39	0	0
PHASE CHANGE	0	0	0	0
HEAT PIPES	0	0	0	0
ISOTOPES	2.3	0	0	0
WIRING, ETC.	23.1			
STRUCTURE	50.8			

*DIRECT LINK OPERATED, AT LOW DATA RATE, AFTER 1ST FEW DAYS OF MISSION

SURFACE PAYLOAD – SOLAR POWER

- MINIMUM SCIENCE
- MODIFIED ENVIRONMENT
- 90 DAY MISSION
- PARAMETRIC COMPUTATION

SCIENCE PAYLOAD: 30 LB, 10 + 7 BITS/DAY

BASIC DESIGN CONSTRAINTS

MISSION LENGTH: 90 DAYS
 BASIC POWER SOURCE: SOLAR PANELS
 ENVIRONMENTAL LIMITS: 120 TO -150 °F
 COMMUNICATION DISTANCE: 285 x 10 + 6 KM
 MAXIMUM ANTENNA SIZE: 1.5 FT DIAMETER
 TRANSMITTING FREQUENCY: 400 MHz*
 TRANSMITTING TIME
 LOW RATE RADIO: 0 HR/DAY*
 HIGH RATE RADIO: .3333 HR/DAY

SIGNIFICANT CHARACTERISTICS

TOTAL PAYLOAD WEIGHT: 226.338 LB
 RADIO LINKS
 LOW RATE RADIO: 0 BPS, 0 WATTS RF*
 HIGH RATE RADIO: 19,199.7 BPS, 15.3598 WATTS RF
 HIGH GAIN ANTENNA NOT REQUIRED
 BATTERY WEIGHT: 13.1145 LB
 SOLAR PANEL AREA: 17.6311 SQ FT
 EXT DIMENSIONS: 14.9074 x 35.0659 x 40.2101 IN.

WEIGHT AND POWER SUMMARY

ITEM	WEIGHT (LB)		POWER (WATTS)	
	INTERNAL	EXTERNAL	DAY	NIGHT
SCIENCE	7.6	22.4	2.1	2.1
SEQUENCER	7.21	0	5.91	2.43
LOW GAIN ANTENNA	0	1*	0	0
RECEIVE ANTENNA	0	.598	0	0
HIGH GAIN ANTENNA	0	0	0	0
LR TRANSMITTER	9*	0	0	0
HR TRANSMITTER	4.48	0	70.4	0
CMD RECEIVER/DECODER	5.2	0	.5	.5
TELEMETRY	7.87	0	1.64	1.64
ENGINEERING INSTRUMENTATION	2.1	2.1	2.1	2.1
CORE STORE	2.27	0	.431	.431
TAPE RECORDER	0	0	0	0
EED MODULES	0	6.8	0	0
SWITCHING LOGIC	10.2	0	0	0
BATTERIES	13.1	0	0	0
SOLAR PANELS	0	26.4	0	0
RTG	0	0	0	0
INSULATION	0	34.6	0	0
PHASE CHANGE	2.43	0	0	0
HEAT PIPES	0	0	0	0
ISOTOPEs	1.6	0	0	0
WIRING, ETC.	15.8			
STRUCTURE	43.6			

*DIRECT LINK OPERATED, AT LOW RATE, AFTER 1ST FEW DAYS OF MISSION

the effect of the varied parameters. Another evaluation criterion is the electrical power of the RTG power supply. Increased electrical power requirements result in heavy, and expensive, RTG units. In addition, a large RTG compounds the problems of thermal control, both during interplanetary cruise and after landing.

The choice of a data transmission mode for a long duration mission is dictated by operational considerations and not necessarily by performance factors. As it was for short and extended duration surface payloads, the relay mode holds a clear edge in performance over the direct mode. The weight, size, electrical power and complexity of payloads using the relay mode are considerably smaller than for payloads using the direct mode. However, for mission durations of longer than 6 months, continuous availability of the orbiter for relay communications purposes is not considered to be consistent with the mission requirements of the orbiter. The question of the data transmission mode is therefore a function of the average daily quantity of data to be transmitted throughout the mission. For extended duration missions (90 days), it was concluded that a satisfactory mission could be obtained if a high data rate was available for only a few days of the mission. For the long duration missions, a large data rate (10^6 bits per day) is probably required on any given day of the mission. Since the relay mode is not considered to be feasible on a day-to-day basis for 6 months or longer, it is concluded that the direct mode must be used as the primary communications technique for long duration missions. However, for the parametric designs, the relay mode will be included as a back-up or supplement to the direct mode during the early stages of the mission.

Reliability is one of the major design considerations in a mission duration of up to two years. Two of the weakest links in the systems of the surface payload are the tape recorder and high gain pointed antenna of the telecommunications system. Since it is not practical to consider either "block" or "functional" redundancy for the high gain antenna, the basic mechanisms of the antenna must be designed to provide the desired reliability. For the tape recorder, "block" redundancy (two tape recorders; only one operational at any given time) has been incorporated into the parametric analyses. Block redundancy for the high rate transmitter is also included in the analysis. For batteries, the analysis used only a 50% depth of discharge, to insure a long cycle life for the batteries.

As with other types of missions, there is a considerable interaction between the environment, the thermal control design, and the characteristics of the surface payload. It is probable that the knowledge of the Martian environment will be expanded prior to the commencement of a long duration mission. Whether this increased data can be incorporated into the design of the vehicle is questionable, due to the short time span between the end of one mission opportunity and the start of the next one. In addition, the environment at any given landing site will change considerably during the mission as a result of seasonal and general climatic variance. The environment variance over a period of months is expected to differ considerably from the nominal cyclic environment used for analysis of the short and extended duration missions. Also, the power profile/operating mode of the surface payload will vary considerably on a day-to-day basis because of programming directions from the Earth. To accommodate both a varying environment and power, hence heat, profile, the thermal control system must be highly flexible. A heat pipe/radiator concept similar to our VOYAGER Phase B design provides an acceptable solution. To further insure mission success, the current study considered the continuous cold (-190°F) environment as well as the worst case anticipated cyclic or warm environment.

The RTG provides sufficient heat to warm the surface payload if this heat can be controllably transferred into the basic equipment compartment of the surface payload. An idealized thermal control design could use a fully active design by pumping heat, via heat pipes, from the RTG to the equipment during the night and radiating excess heat from the equipment during the day through a heat pipe/radiator system. While this idealized concept is probably the most efficient of the alternatives, little is known about the practical aspects of this approach. For the parametric analysis, the required heat was supplied with either isotopes or electrical heaters. It can be assumed that the weight of isotope heaters would correspond to the weight of the equipment required to implement a fully active thermal control system.

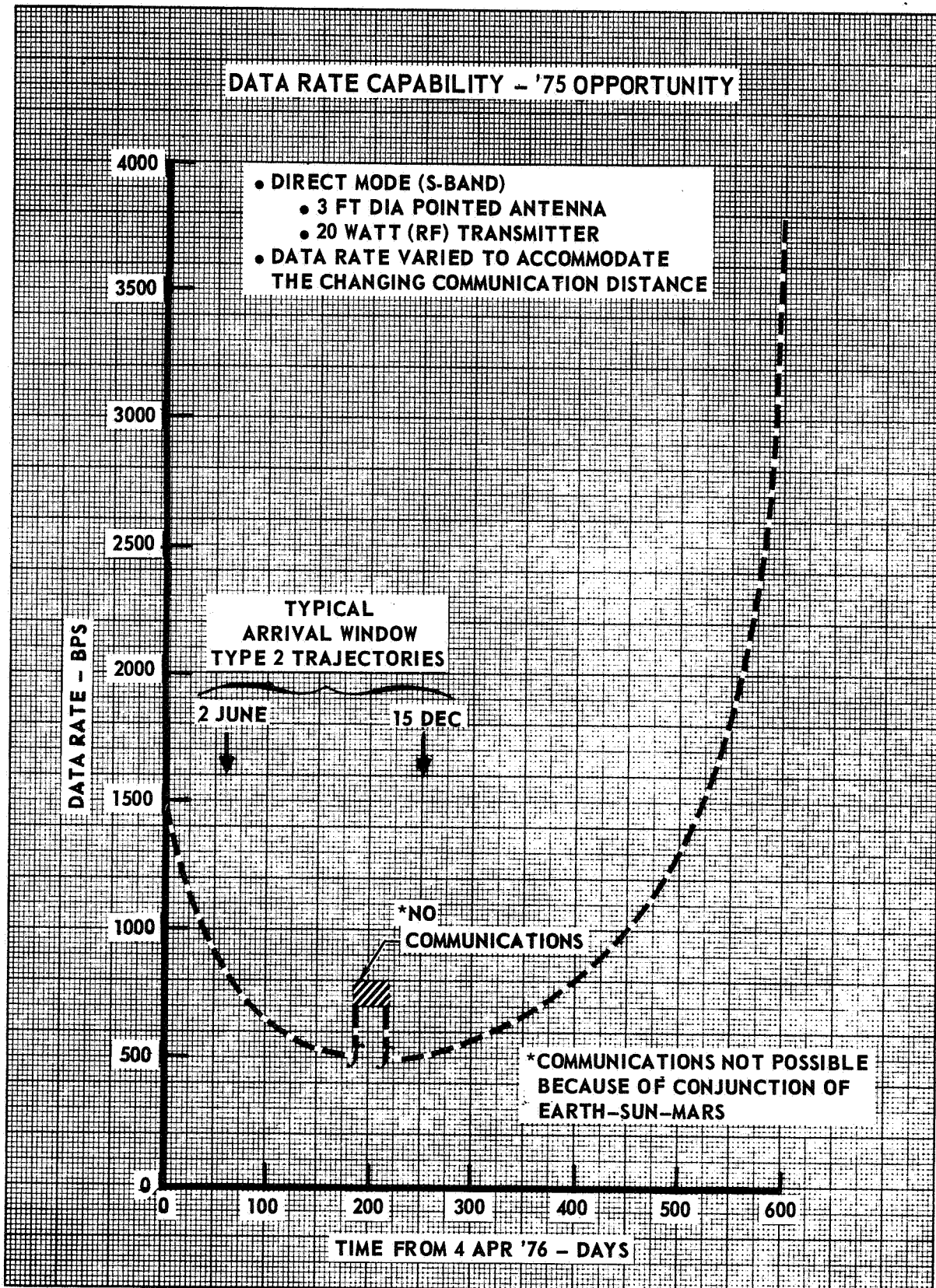
3.3.2.6.2 Design alternatives: Of the alternatives to be explored for payloads designed for a long duration mission, telecommunications, electrical power, science payload are the most significant.

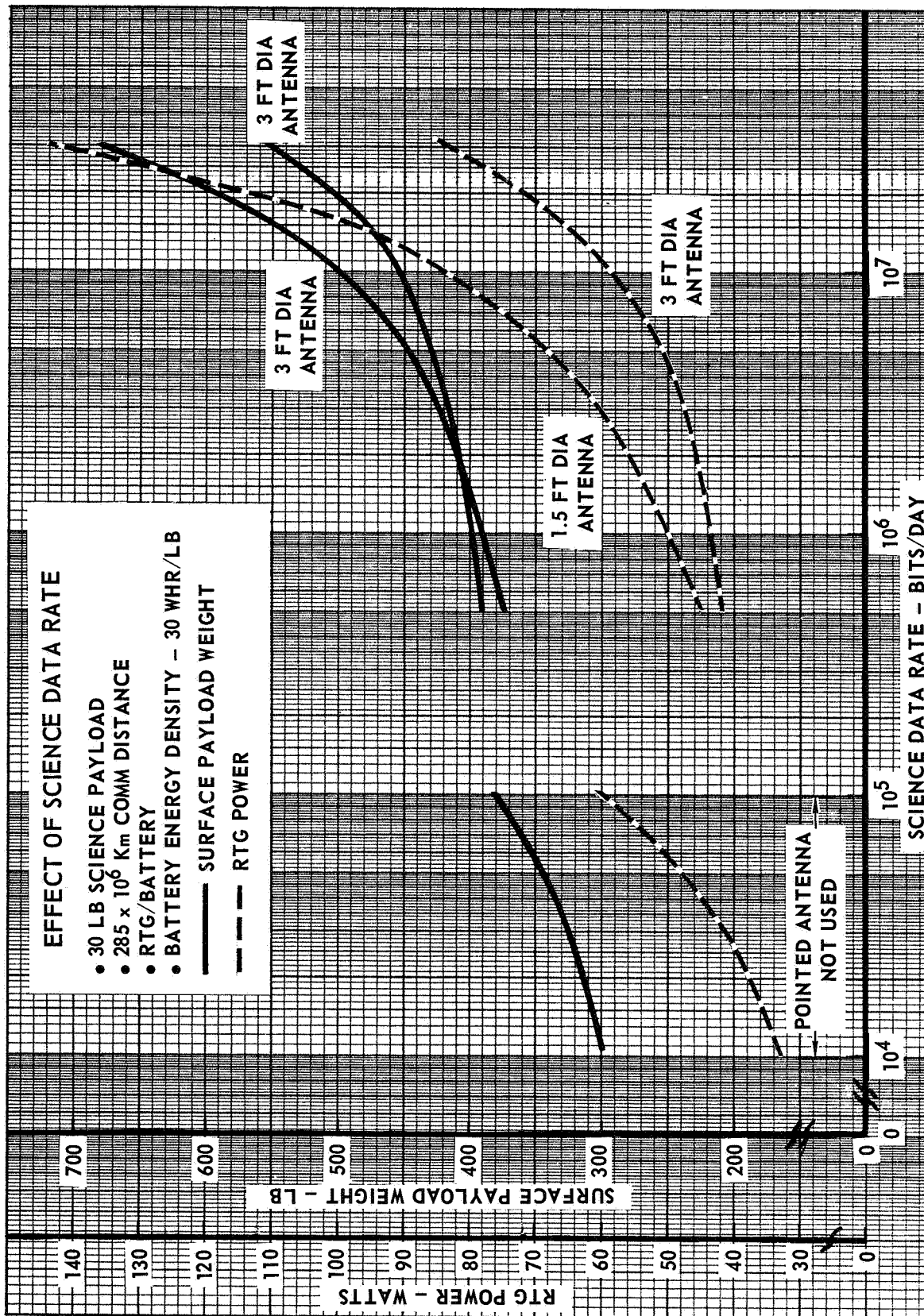
Telecommunications Design Alternatives - With the direct transmission mode, a choice of antenna characteristics is available. Another variable, which interacts with a selection of antennas, is the data rate desired at a given day of the mission. After landing, the distance between Mars and Earth increases. If data rate is kept constant, transmitter power must be increased to maintain adequate signal strength to Earth. Several hundred days into the mission, the communication distance decreases, and less transmitter power is required. An alternate approach, shown in Figure 3.3.2-23, is to keep transmitter power constant and adjust the data rate to the communication distance. This technique represents a practical method of adapting to the varying communication distances. Two design approaches are possible: (1) design for a large data rate at the start of the mission, and let the data rate decrease as communications distance increases, or (2) design for a large data rate capability at the worst communications distance.

For the examination of the telecommunications design alternatives, nominal designs for the other system are used. For electrical power an RTG/battery combination, with the battery having a yield of 30 watt-hours per pound, is used. For thermal control, isotopes to provide heat, heat pipes/radiators for heat dissipation, and 4 inches of insulation are used. In addition, the electrical power system is designed to allow simultaneous operation of the high rate and low rate (engineering) radio links.

These analyses have shown that for data rates greater than 10^6 bits per day, the 3 foot diameter antenna provides an advantage in vehicle weight and RTG power required as compared to the 1.5 foot diameter antenna. As shown in Figures 3.3.2-24 and 3.3.2-25, this advantage is valid regardless of the nominal design communication distances, 285 or 380×10^6 kilometers. However, as in the case of short duration missions, the actual selection may not be based on a weight or RTG power advantage, but on the overall integration problems of installing a large RTG and antenna in the aeroshell of the flight capsule.

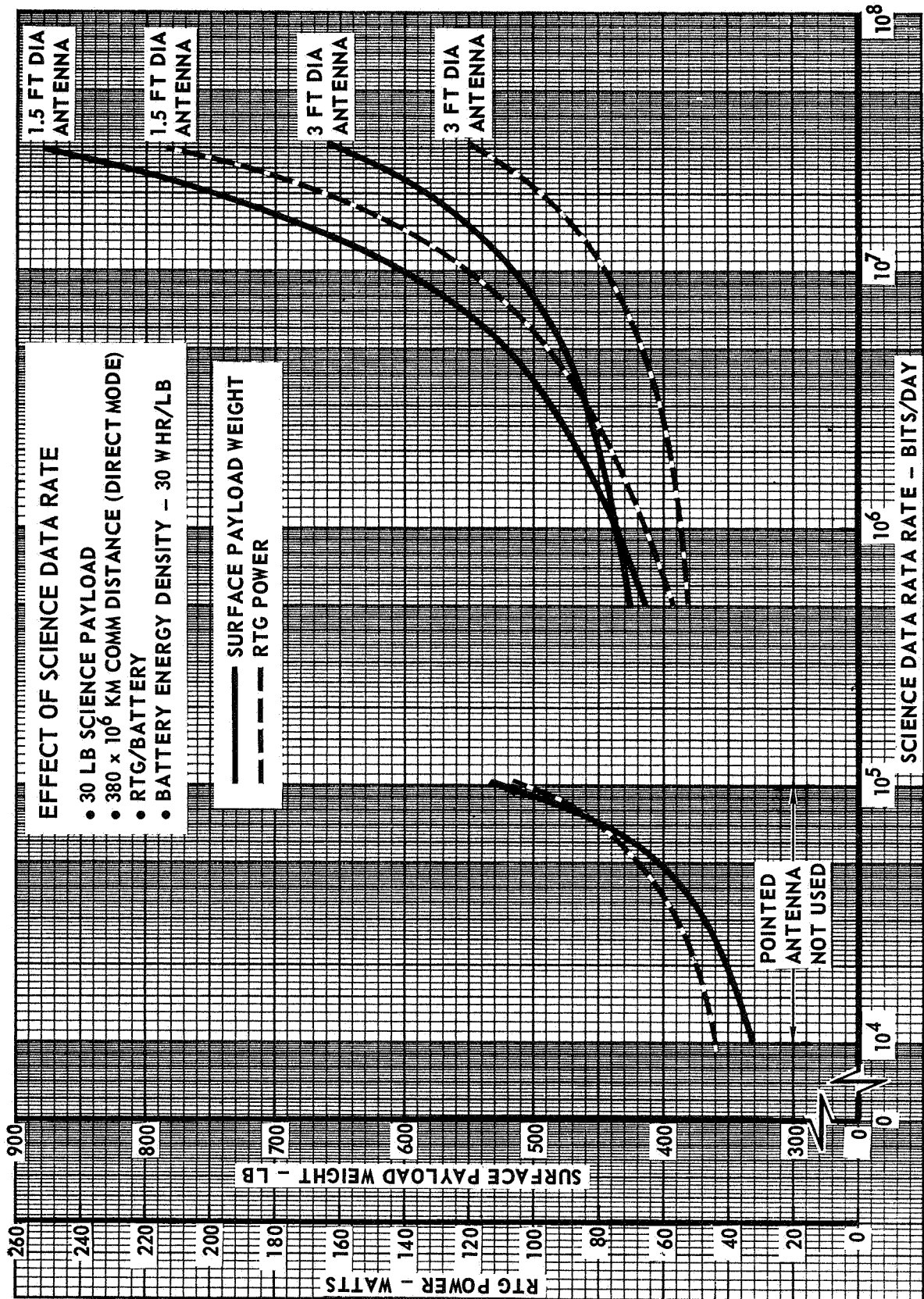
The effect of communication distance is shown in Figures 3.3.2-24 and 3.3.3-25. At this point in the analysis, it would be safe to conclude that a design providing a high data rate capability throughout the mission is very heavy and requires a large RTG. However, this conclusion is applicable only for the design constraints employed at this point, and is subject to re-evaluation later on in this analysis.





3.3.2-45

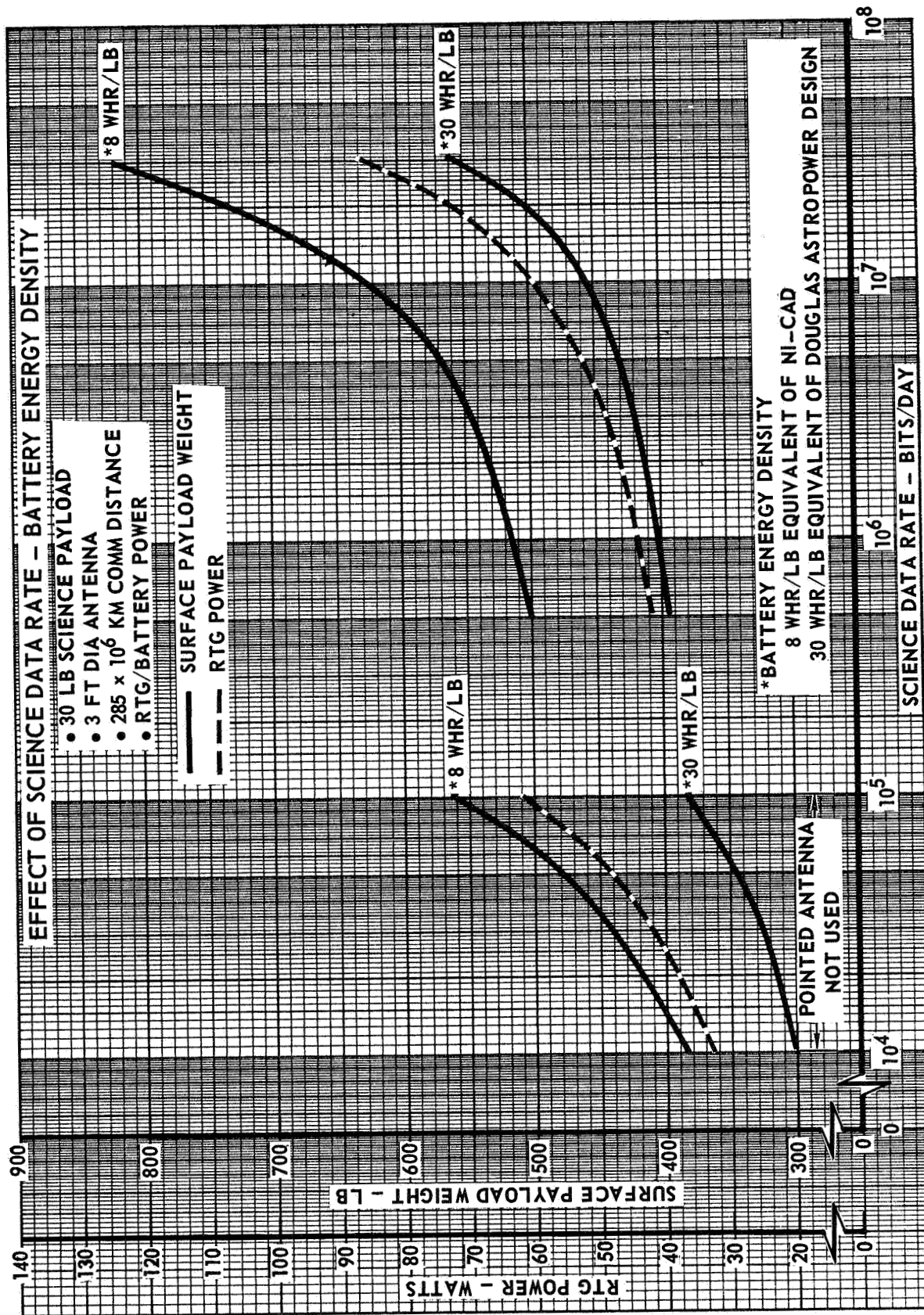
FIGURE 3.3.2-24

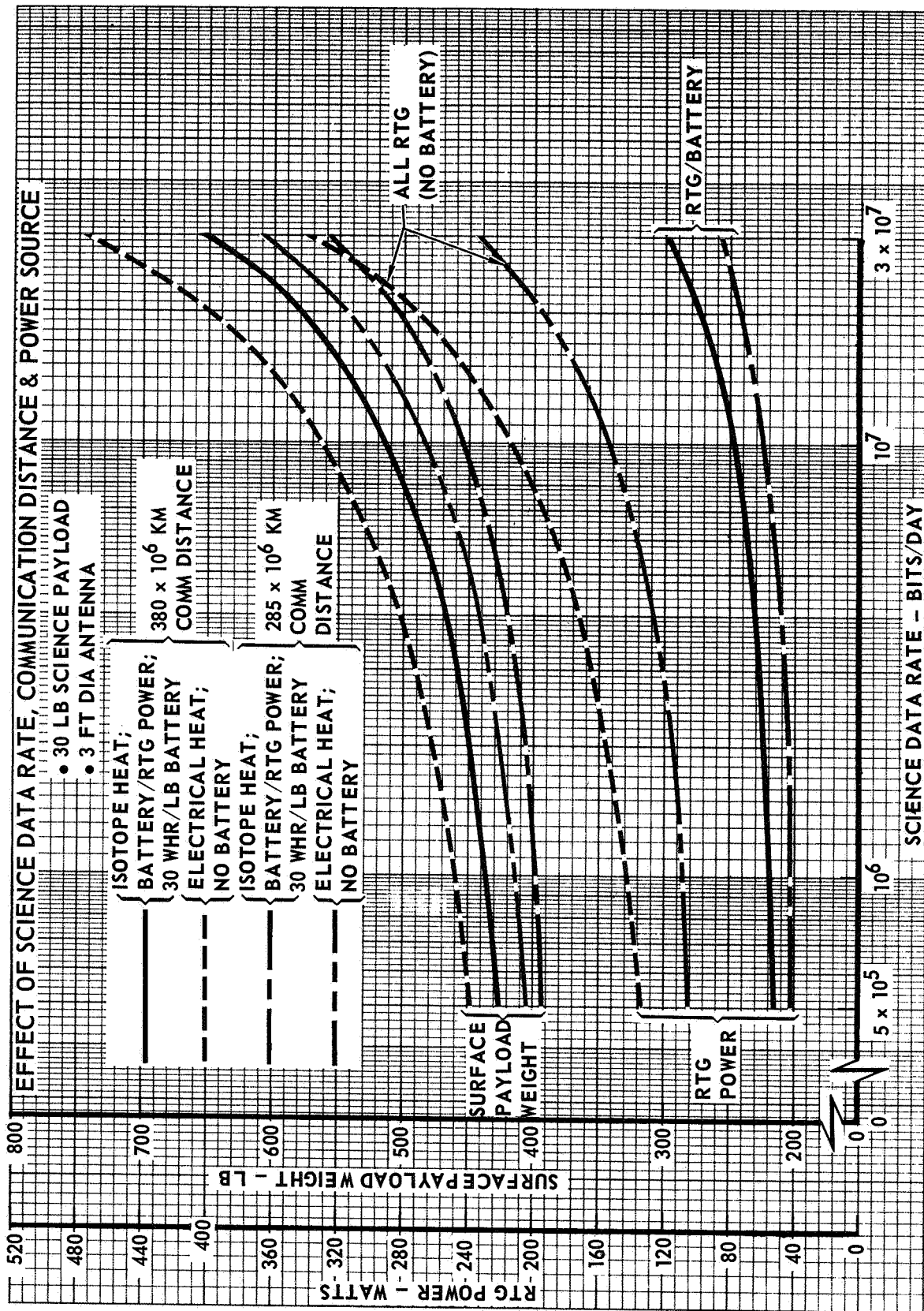


Electrical Power Design Alternatives - The variables of electrical power design interrelated with other subsystems. The conventional design approach is to use an RTG and a battery in combination. The RTG recharges the battery and provides the equipment with power during most of the mission; the battery provides power beyond the basic capability of the RTG during peak demands, such as during data transmission. The critical factor in this approach is the efficiency of the battery. For the data of Figures 3.3.2-24 and 3.3.2-25, and Ag-Zn battery of long life and high efficiency, 30 watt-hours per pound, was used. While this battery is technically feasible, it has not been fully developed. A more conservative approach would use a lower yield battery of demonstrated long life. Such a battery, nickel-cadium, has an efficiency of only 8 to 10 watt-hours per pound. Figure 3.3.2-26 demonstrates the severe penalties in surface payload weight if a low efficiency battery is used.

Another alternative is to use only an RTG. In this case, the RTG is sized to meet the peak power requirements of the payload. When the power requirements are lower than the peak, the excess power of the RTG is dissipated through diodes and large resistors. Since this power is not required by the equipment, it could be used for heating the vehicle during the cold portions of the mission. This approach simplifies the thermal control design of the surface payload because the heat is controllable, and eliminates the need for a battery. Unfortunately, as is shown in Figure 3.3.2-27, this approach not only causes a heavier overall vehicle, but necessitates an extremely large RTG. The size of the RTG, over 300 electrical watts for the worst case shown in Figure 3.3.2-27, creates extreme thermal control problems for the total flight capsule 6000 watts of heat. It is concluded that battery and RTG, in combination, represents the most practical and economical method of designing the electrical power system.

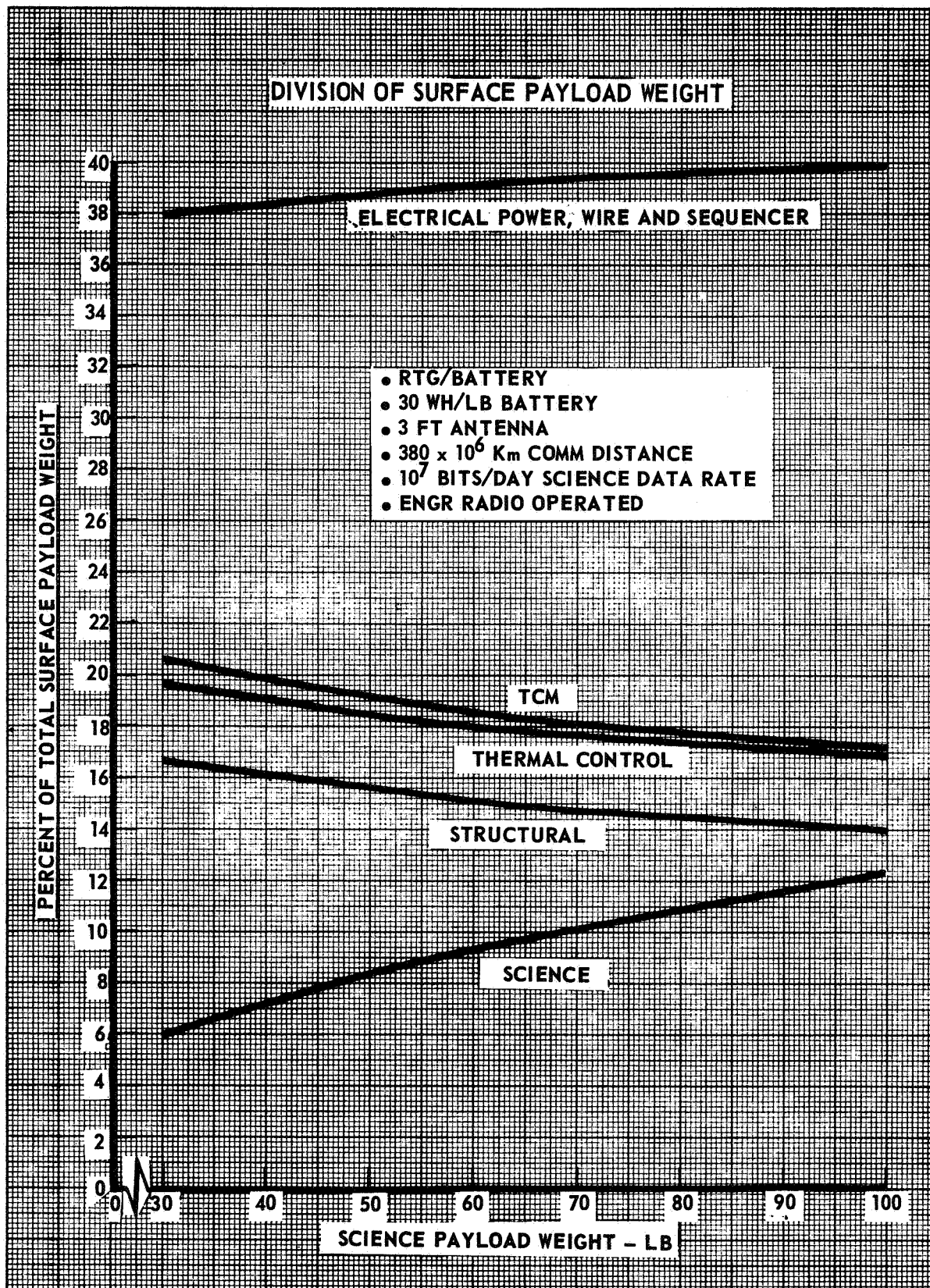
Effect of the Science Payload - The effect of the science data was discussed previously in the telecommunications design alternatives. As it was for other types of surface payloads, the relationship of science weight to total payload weight is virtually linear. However, it is interesting to examine the distribution of the systems of the surface payload as science weight is varied. As shown in Figure 3.3.2-28, as science weight is increased, its percentage of the total vehicle weight increases. The opposite is true for





3.3.2-49

FIGURE 3.3.2-27

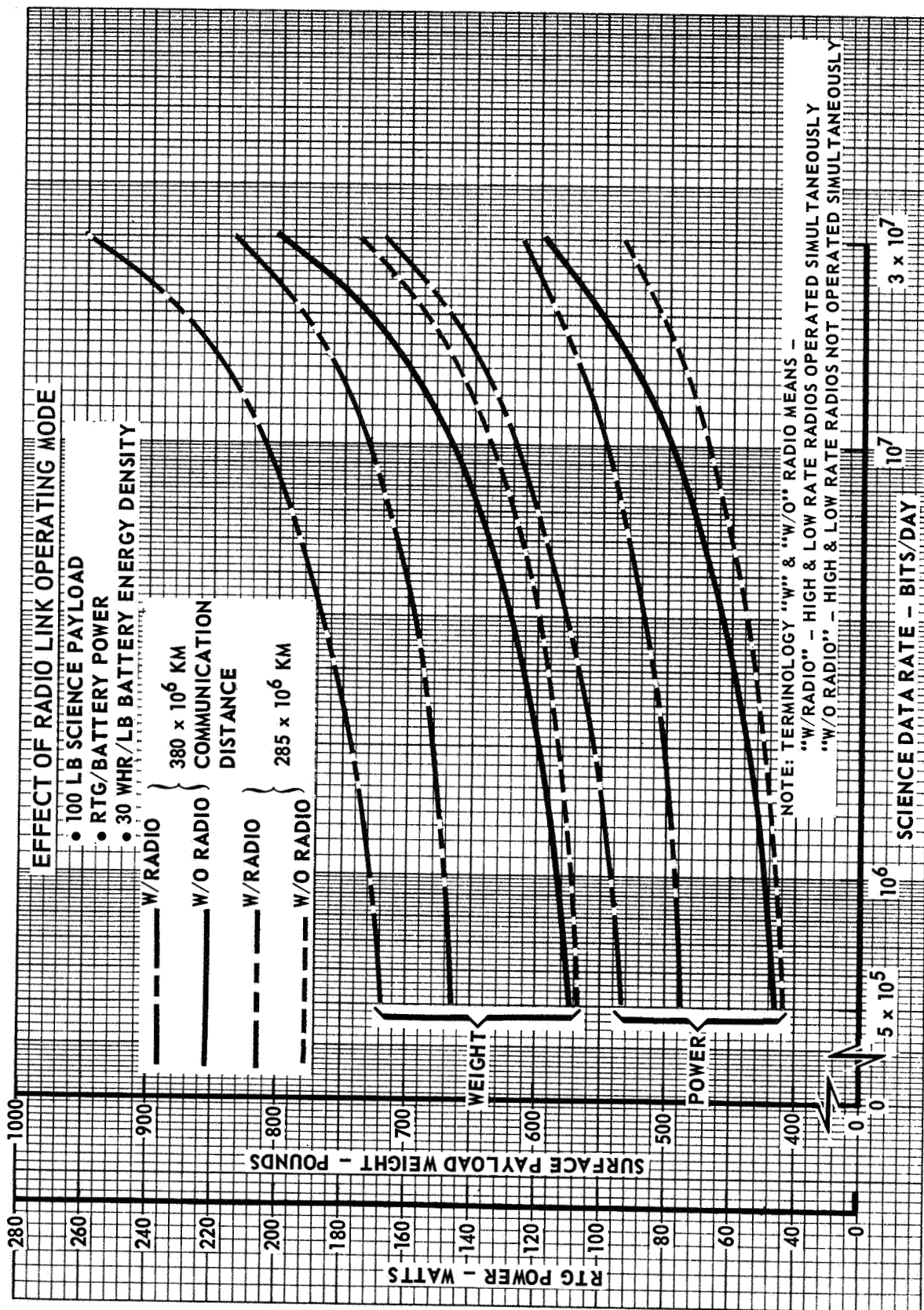


the telecommunications, thermal control and structural systems; they decrease, on a percentage basis, as science weight is increased.

3.3.2.6.3 Refined design considerations: So far the effects of the principal variables upon a surface payload of nominal design have been discussed. Since the weight and power of the RTG are of importance, it would be of value to reduce the power and energy requirements of the vehicle in order to reduce the size of the RTG. A candidate for reducing the power profile is to change the operating profile of the surface payload. To date, the energy and power to allow the simultaneous operation of a high rate radio link, used principally for science data, and a low rate radio link has been provided. The low rate radio link uses an omnidirectional antenna and a relatively large transmitter to transmit engineering data on the operation of the surface payload. This engineering data could be transmitted via the high rate radio link with virtually no change to the basic high rate radio. For short duration missions, it is good policy to operate both links simultaneously because if the high rate link fails, some data is still returned to Earth without having to utilize either a command link or internal diagnostic equipment to activate the low rate link. For long duration missions, assuming that they are not the first Martian mission, a high degree of confidence in the success of the mission can be assumed. It can also be assumed that if the high rate radio is inoperative, the low rate radio can be commanded to operate until sufficient data is received to allow a corrective command to be sent. Therefore, designs providing for simultaneous operation of low and high rate radios are probably too conservative for long duration missions.

A comparison of vehicles, designed with and without simultaneous operation of both radio links, is shown in Figure 3.3.2-29. As can be noted, the power and energy requirements resulting from simultaneous operation are considerable. Therefore, it can be concluded that although some form of low rate radio should be provided, it should not be operated simultaneously with the high rate link.

Assuming that a low rate radio is not operated simultaneously with the high rate, the effects of communication distances can be re-examined. As shown in Figure 3.3.2-29, if the low rate radio is not operated unless commanded, the communication distance has a negligible effect upon the total payload. Thus, although the transmitter power must be increased, the surface payload can be designed to provide the desired data rate throughout the length of the mission with only a modest penalty in vehicle and RTG weight.



3.3.2-52

FIGURE 3.3.2-29

3.3.2.6.4 Conclusions - long duration missions: A surface payload designed for a long duration mission should have the following characteristics:

- a) The preferred electrical power systems is partially dependent upon the development of a high efficiency (30 watt-hours per pound), long life Ag-Zn battery. Assuming that one is developed, the electrical power system should utilize this tbattery to allow reasonable vehicle weights and minimum size RTG's. If this high yield battery is not available, the preferred system is an all RTG (no battery) configuration if the peak electrical requirements are not large.
- b) The principal communications mode should be the direct (S-band) mode because the relay mode is not operationally practical throughout the mission. It is also concluded that the low rate (engineering) radio should not be operated simultaneously with the high rate radio, in order to minimize the size and weight of the electrical power system.

The characteristics of a typical surface payload for long duration missions are shown in Figure 3.3.2-30. Since there are no requirements to select a conceptual design for long duration missions, this payload was selected in order to represent a near maximum capability.

SURFACE PAYLOAD – RTG POWER

- MAXIMUM SCIENCE
- NOMINAL ENVIRONMENT
- 6 MONTH TO 2 YEAR MISSION
- PARAMETRIC COMPUTATION

SCIENCE PAYLOAD: 100 LB, 10 + 7 BITS/DAY

BASIC DESIGN CONSTRAINTS

MISSION LENGTH: 8000 HR

BASIC POWER SOURCE: RTG

ENVIRONMENTAL LIMITS: 120 TO -190°F

COMMUNICATION DISTANCE: 380 x 10 + 6 KM

MAXIMUM ANTENNA SIZE: 3 FT DIAMETER

TRANSMITTING FREQUENCY: 2300 MHz

TRANSMITTING TIME

LOW RATE RADIO: 6 HR/DAY*

HIGH RATE RADIO: 6 HR/DAY*

SIGNIFICANT CHARACTERISTICS

TOTAL PAYLOAD WEIGHT: 671.164 LB

RADIO LINKS

LOW RATE RADIO: .7 BPS, 12.7441 WATTS RF*

HIGH RATE RADIO: 532.407 BPS, 21.2039 WATTS RF*

BATTERY WEIGHT: 30.2193 LB

RTG WEIGHT: 79.448 LB

EXT DIMENSIONS: 17.6313 x 45.7396 x 52.9124 IN.

WEIGHT AND POWER SUMMARY

ITEM	WEIGHT (LB)		POWER (WATTS)	
	INTERNAL	EXTERNAL	DAY	NIGHT
SCIENCE	43.5	56.5	7.3	7.3
SEQUENCER	15.8	0	11.2	10.8
LOW GAIN ANTENNA	0	1	0	0
RECEIVE ANTENNA	0	.747	0	0
HIGH GAIN ANTENNA	9	36	6	0
LR TRANSMITTER	6	0	70.0*	0
HR TRANSMITTER	7.11	0	88.3*	0
CMD RECEIVER/DECODER	6.5	0	1	1
TELEMETRY	29.4	0	6.13	6.13
ENGINEERING INSTRUMENTATION	7	7	7	7
CORE STORE	3.23	0	.479	.479
TAPE RECORDER	6.9	0	8.06	.26
EED MODULES	0	11	0	0
SWITCHING LOGIC	21.5	0	0	0
BATTERIES	30.2	0	0	0
SOLAR PANELS	0	0	0	0
RTG	0	79.4	0	0
INSULATION	0	60.7	0	0
PHASE CHANGE	0	0	0	0
HEAT PIPES	0	28.9	0	0
ISOTOPEs	24.8	0	0	0
WIRING, ETC.	66.9			
STRUCTURE	99.9			
REDUNDANCY	12.0 (TAPE RECORDER; TRANSMITTER)			

*HIGH & LOW RATE RADIO NOT OPERATED SIMULTANEOUSLY

3.4 Martian Environment Effects

The capsule parametric studies of Section 3.3 were based on the same environmental criteria used in the VOYAGER Phase B studies, which are summarized below. This section presents the effects of the environment definition on the capsule design and performance. The analysis is divided into two parts: entry environment and surface environment.

3.4.1 ENTRY ENVIRONMENT DEFINITION - Two aspects of the entry environment affect capsule systems design and performance: The initial entry conditions (velocity and flight path angle at 800 000 ft altitude), and the definition of the Martian atmosphere, the primary design parameters being the density scale height and the atmospheric pressure at the planet surface. Figure 3.4-1 shows the nominal entry corridor and design atmosphere envelope.

As discussed in Section 3.1.3, the critical conditions for capsule system design tend to occur at the upper right or lower left corners of both the entry corridor and the atmospheric envelope. It is therefore interesting to determine the effects on capsule design of relieving these extremes.

The sensitivity of the capsule to the atmosphere definition was evaluated in two steps, shown in Figure 3.4-2: a) the minimum entry velocity was raised to 14 000 ft/sec, and b) the entry flight path angle spread was reduced to -15 to -18 deg.

Three variations in the atmospheric definition were considered, as shown in Figure 3.4-3: a) the minimum density scale height was raised to 8 km, b) the surface atmospheric pressure uncertainty was reduced to 7-14 mb, and c) the combined effects of a) and b) were investigated.

Finally, the reduced entry corridor was combined with the increased minimum scale height, the same environmental relief used for two of the discrete point designs (Concepts I and V of Section 4).

3.4.1.1 Capsule system sensitivity. - The variations cited above affect the aeroshell weight and diameter more than other vehicle characteristics. The remaining capsule systems are either insensitive or experience only secondary effects, due to reduced aeroshell diameter requirements.

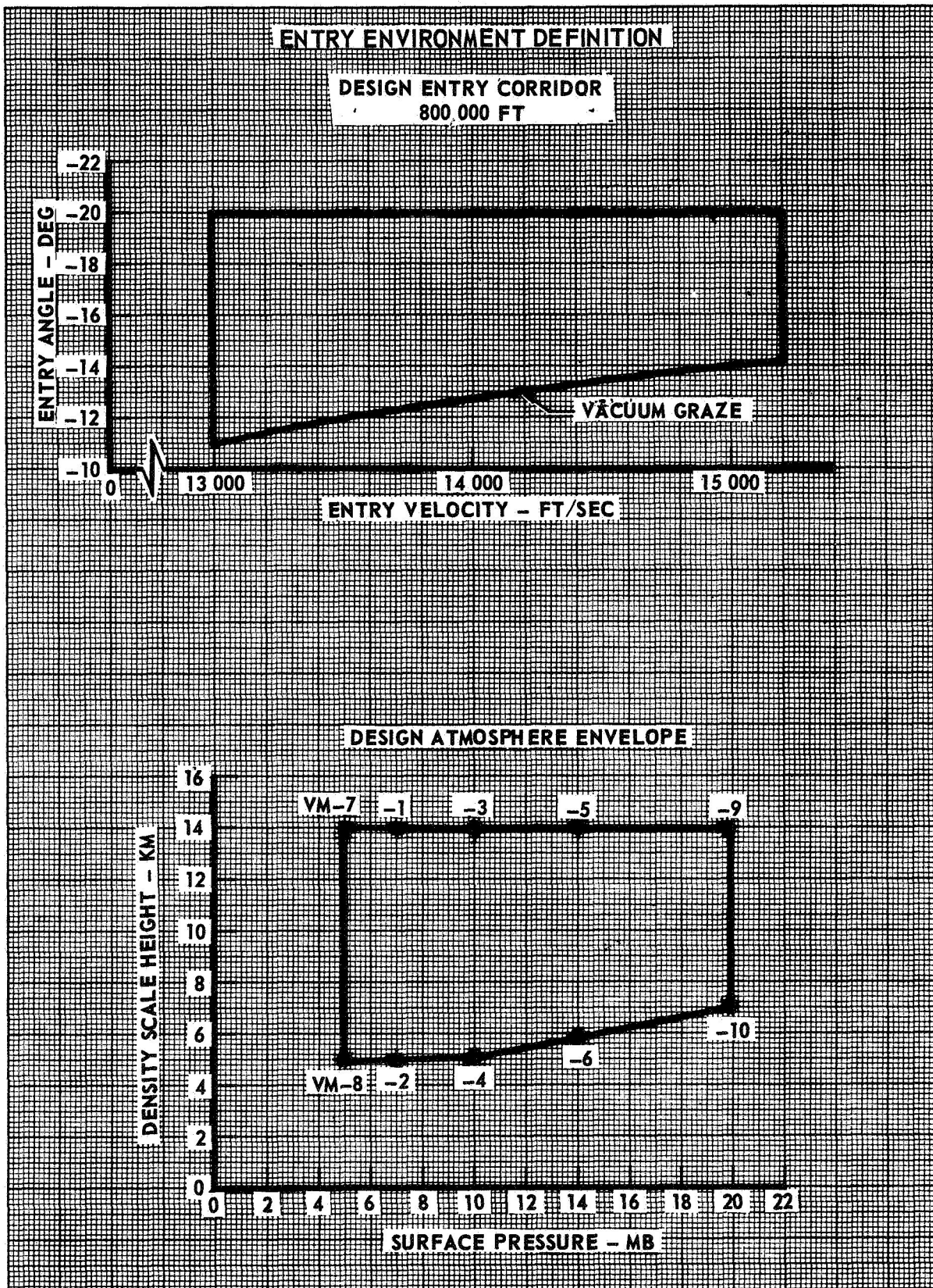


FIGURE 3.4-1

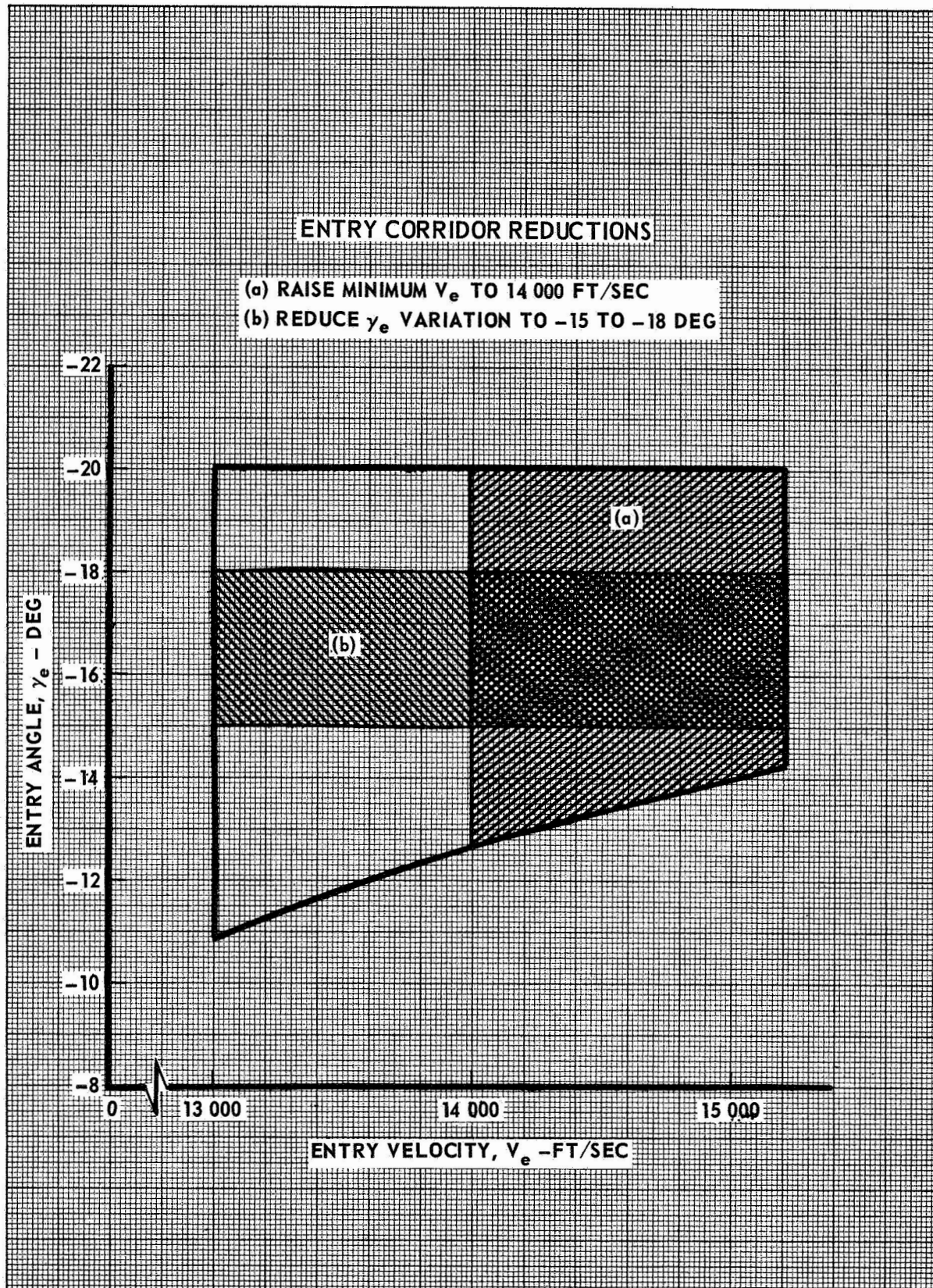


FIGURE 3.4-2

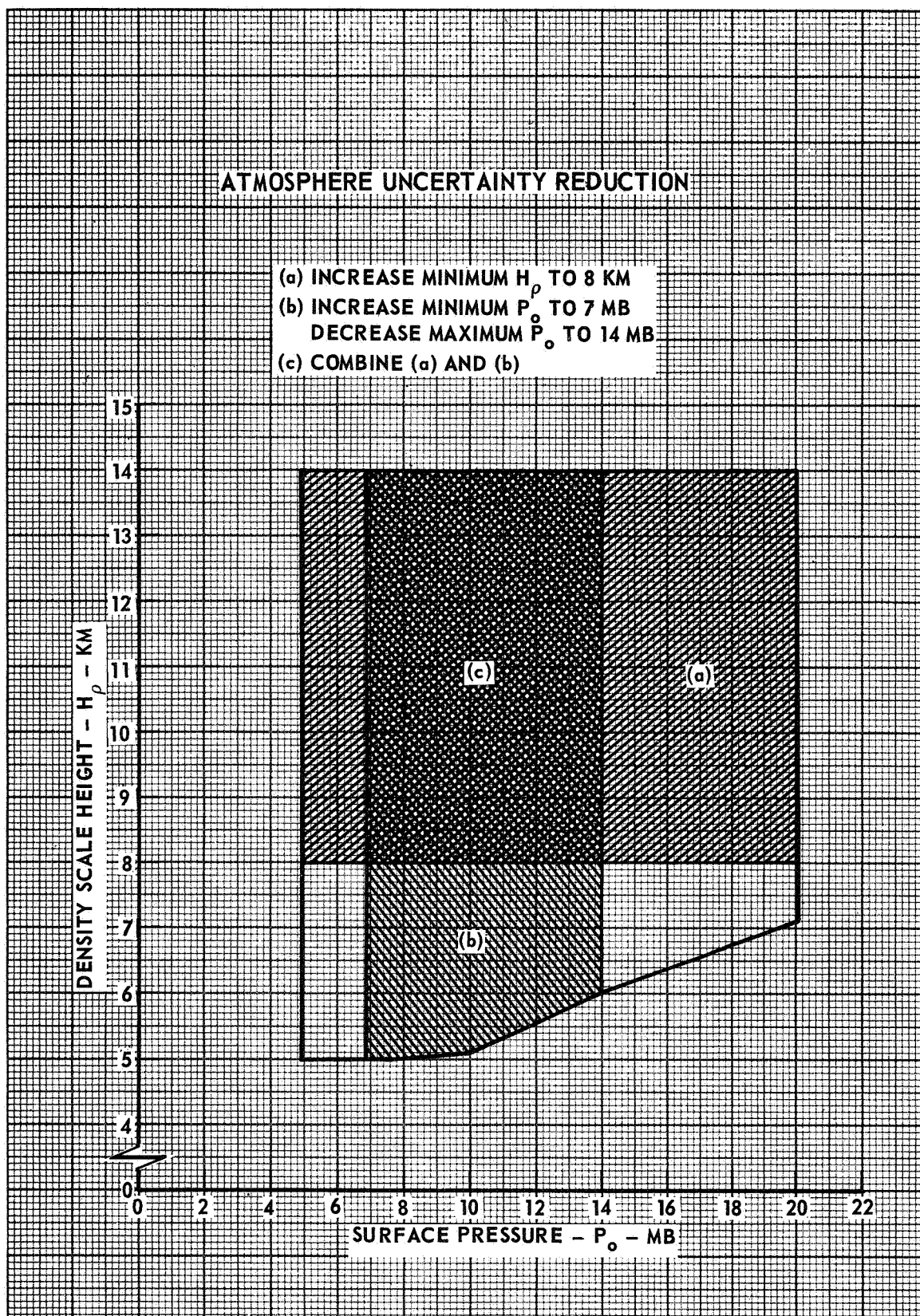


FIGURE 3.4-3

The aeroshell structure and heat shield are sensitive to opposite regions of both the entry corridor and the atmosphere: aeroshell structure is designed by maximum dynamic pressure, which is critical at high entry velocity, steep entry angle, and low scale height. The heaviest heat protection system using ablators and high temperature insulators (such as HCF) will result when the largest scale height is combined with the minimum velocity trajectory entering at vacuum graze angles. If the graze entry angle boundary is excluded from the corridor definition, the heaviest heat protection design conditions for all materials will be the combination of maximum scale height and entry velocity, and minimum initial entry angle. These characteristics are shown in Figure 3.4-4 for a single aeroshell diameter and $m/C_D A$. The total aeroshell weight is the sum of the structure weight at the minimum scale height being considered and the heat protection weight at the maximum scale height. Thus, designing for the full entry corridor and for VM-1 through VM-10 atmospheres results in an aeroshell weight of 226 lb. Reducing the scale height uncertainty to 8-14 km and limiting the entry corridor to γ_e between -15° and -18° and V_e between 14 000 and 15 200 ft/sec permits a reduction to a weight of 197 lbs if the same aeroshell diameter were used. However, the environmental relief allows the use of a higher $m/C_D A$, which permits a reduction in aeroshell diameter. Figure 3.1.4-16 showed that the reduced entry corridor and increased minimum H_p combined to allow an $m/C_D A$ of .385 slugs/ft² for a Mach 2, 23 000 ft parachute deployment condition. This $m/C_D A$ allows the aeroshell diameter to be reduced to 9.7 ft which results in a total aeroshell weight of 144 lb.

The modified environment thus allows a reduction of 82 lb (about 36%) from the 226 lb aeroshell weight required for the nominal environment. The reduced diameter and weight also permit savings in the canister, insulation, TPS, and other capsule systems.

3.4.1.2 Capsule parametrics. - The data presented in this section shows the effects of the entry environment variations described above. Table 3.4-1 lists the input data used in these studies. Table 3.4-2 summarizes the results for the point of most interest, i.e., the minimum diameter and weight for which a parachute can be used within the Mach 2 deployment constraints. The table also keys Figures 3.4-5 through 3.4-12 to the entry corridor or atmosphere

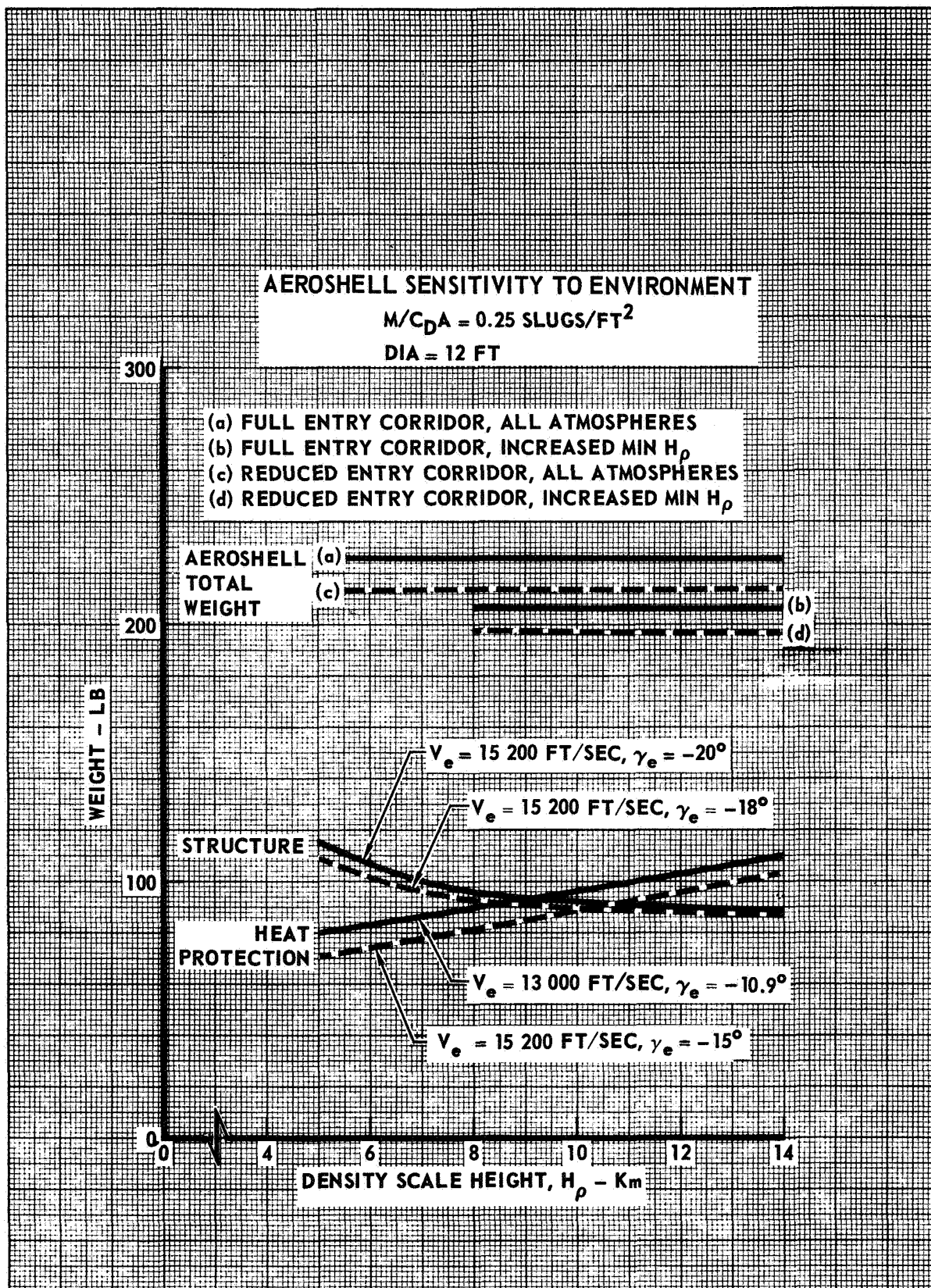


FIGURE 3.4-4

TABLE 3.4-1
ENVIRONMENTAL EFFECTS PARAMETRICS
INPUT DATA

FIGURES 3.4-5 THROUGH 3.4-12 ARE BASED ON THE FOLLOWING:

ENTRY ALTITUDE 800,000 FT
DEORBIT VELOCITY INCREMENT 450 FT/SEC
FIXED EQUIPMENT WEIGHT
 GUIDANCE AND CONTROL AND RADARS - ON AEROSHELL 2.9 LB
 GUIDANCE AND CONTROL AND RADARS - LANDED 112.7 LB
 SCIENCE, TCM, ELECTRICAL AND MISC. - TOTAL 168.0 LB
 SCIENCE, TCM, ELECTRICAL AND MISC. - ON ADAPTER 15.0 LB
 SCIENCE, TCM, ELECTRICAL AND MISC. - ON AEROSHELL 15.5 LB
 SCIENCE, TCM, ELECTRICAL AND MISC. - ON LANDER 137.5 LB
DRAG COEFFICIENTS (C_D)
 AEROSHELL - SUPERSONIC C_D 1.51
 AEROSHELL - SUBSONIC C_D 78
 ATTACHED INFLATABLE DECELERATOR - SUPERSONIC C_D 1.00
 ATTACHED INFLATABLE DECELERATOR - SUBSONIC C_D 80
 PARACHUTE - SUPERSONIC C_D (BASED ON TOTAL SURFACE AREA) .60
 PARACHUTE - SUBSONIC C_D (BASED ON TOTAL SURFACE AREA)... .60
MINIMUM ACCELERATION OF AEROSHELL RELATIVE TO LANDER6g
PARACHUTE DEPLOYMENT MACH NO. LIMIT 2.0
DEPLOYABLE DECELERATOR DEPLOYMENT ALTITUDE (H_D) 23,000 FT
MAXIMUM DEPLOYABLE DECELERATOR VERTICAL VELOCITY
 AT TERMINAL PROPULSION INITIATION 350 FT/SEC
HORIZONTAL VELOCITY VECTOR DUE TO WINDS 220 FT/SEC
TERMINAL PROPULSION IGNITION ALTITUDE 4868 FT
MAXIMUM SURFACE SLOPE - 34° ABOVE LOCAL HORIZONTAL
TERMINAL PROPULSION SYSTEM TYPE MONO
TERMINAL GUIDANCE MODE PREPROGRAMMED

TABLE 3.4-2
SUMMARY OF ENTRY ENVIRONMENT EFFECTS

ENVIRONMENT CHANGE	FIGURE REFERENCE	FOR $M_D = 2.0$	
		AEROSHELL DIAMETER (FT)	FLIGHT CAPSULE WEIGHT (LB)
NOMINAL	3.4-5	11.35	1562
$14,000 < V_e < 15,200$ FT/SEC	3.4-5	11.10	1540
GRAZE $< \gamma_e < -18^\circ$	3.4-6	10.55	1508
$-15^\circ < \gamma_e < -20^\circ$	3.4-7	11.25	1544
$-15^\circ < \gamma_e < -18^\circ$	3.4-8	10.55	1500
$8 < H_\rho < 14$ KM	3.4-9	9.65	1462
$7 < P_o < 14$ mb	3.4-10	9.25	1448
$8 < H_\rho < 14$ KM } $7 < P_o < 14$ mb }	3.4-11	7.95	1392
$14,000 < V_e < 15,200$ FT/SEC } $-15^\circ < \gamma_e < -18^\circ$ } $8 < H_\rho < 14$ KM }	3.4-12	8.8	1425

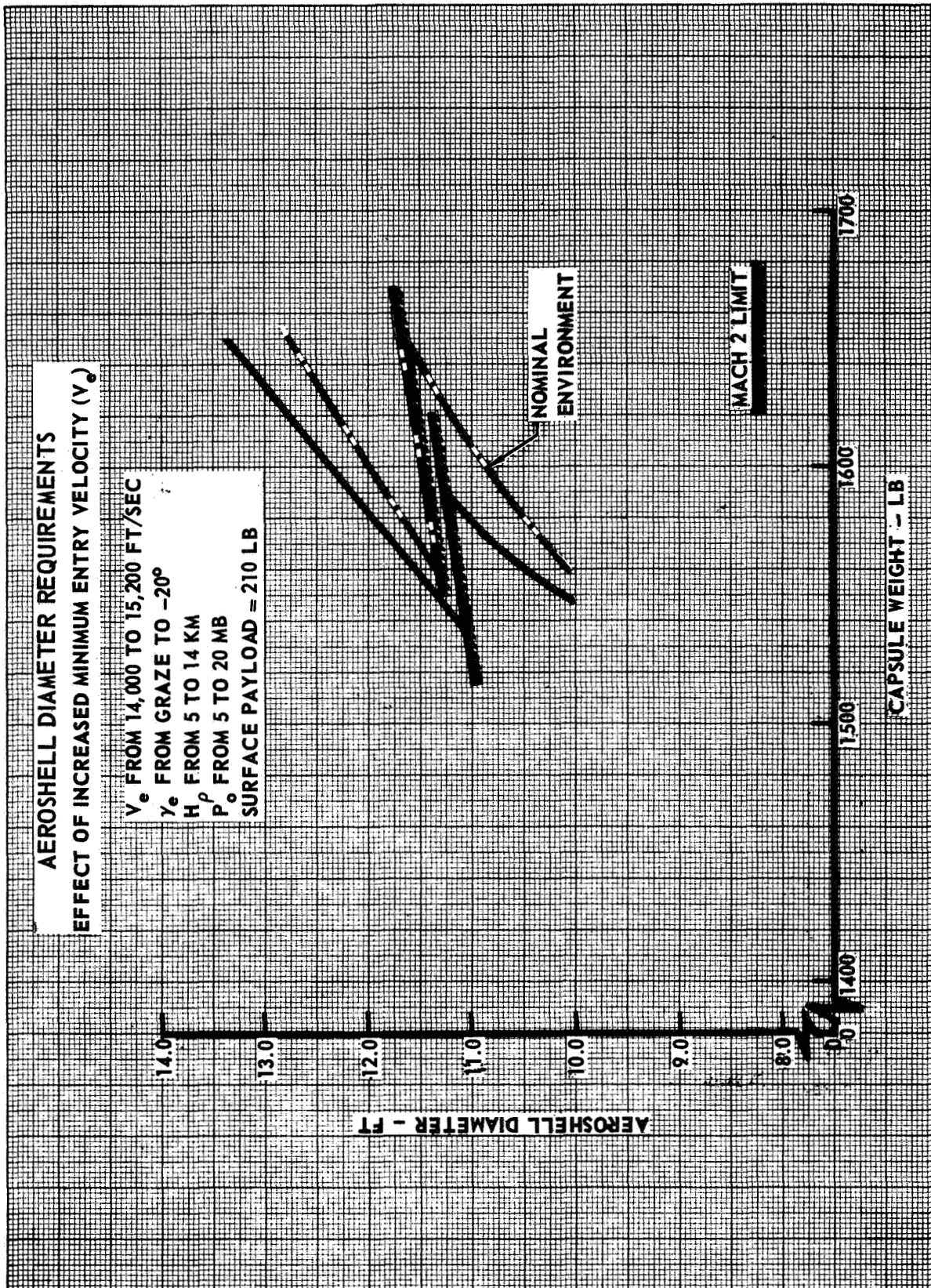


FIGURE 3.4-5

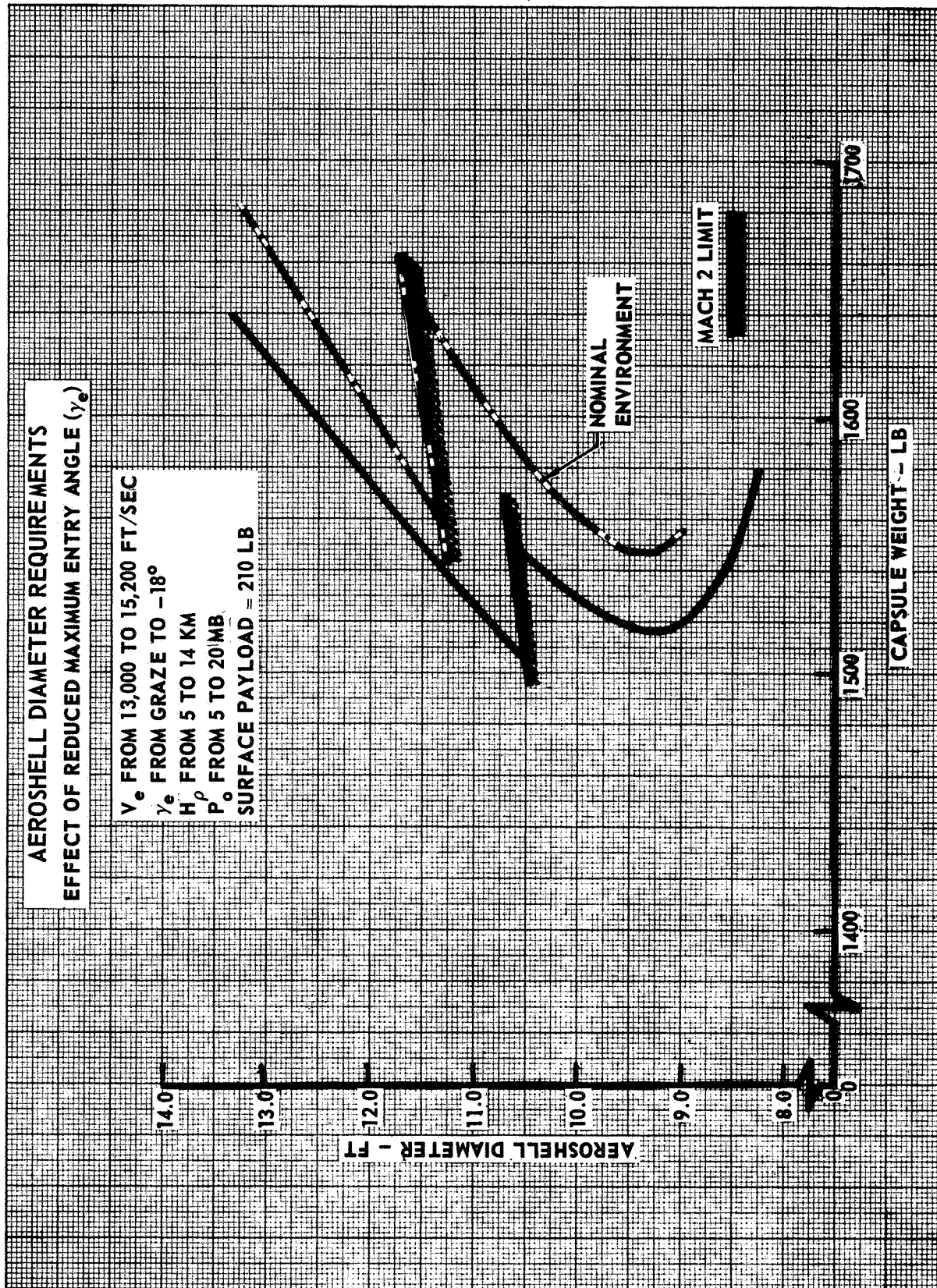


FIGURE 3.4-6

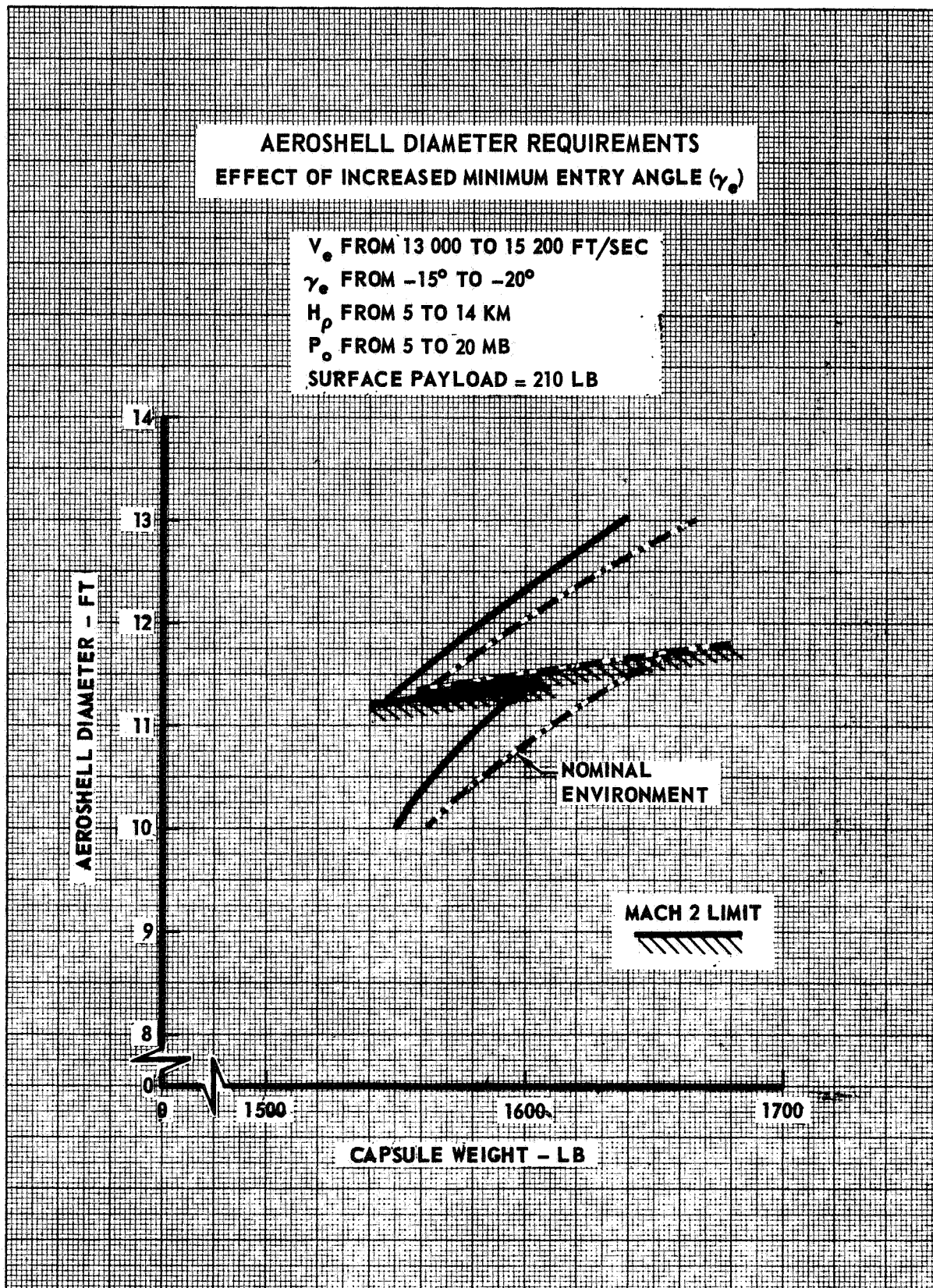


FIGURE 3.4-7

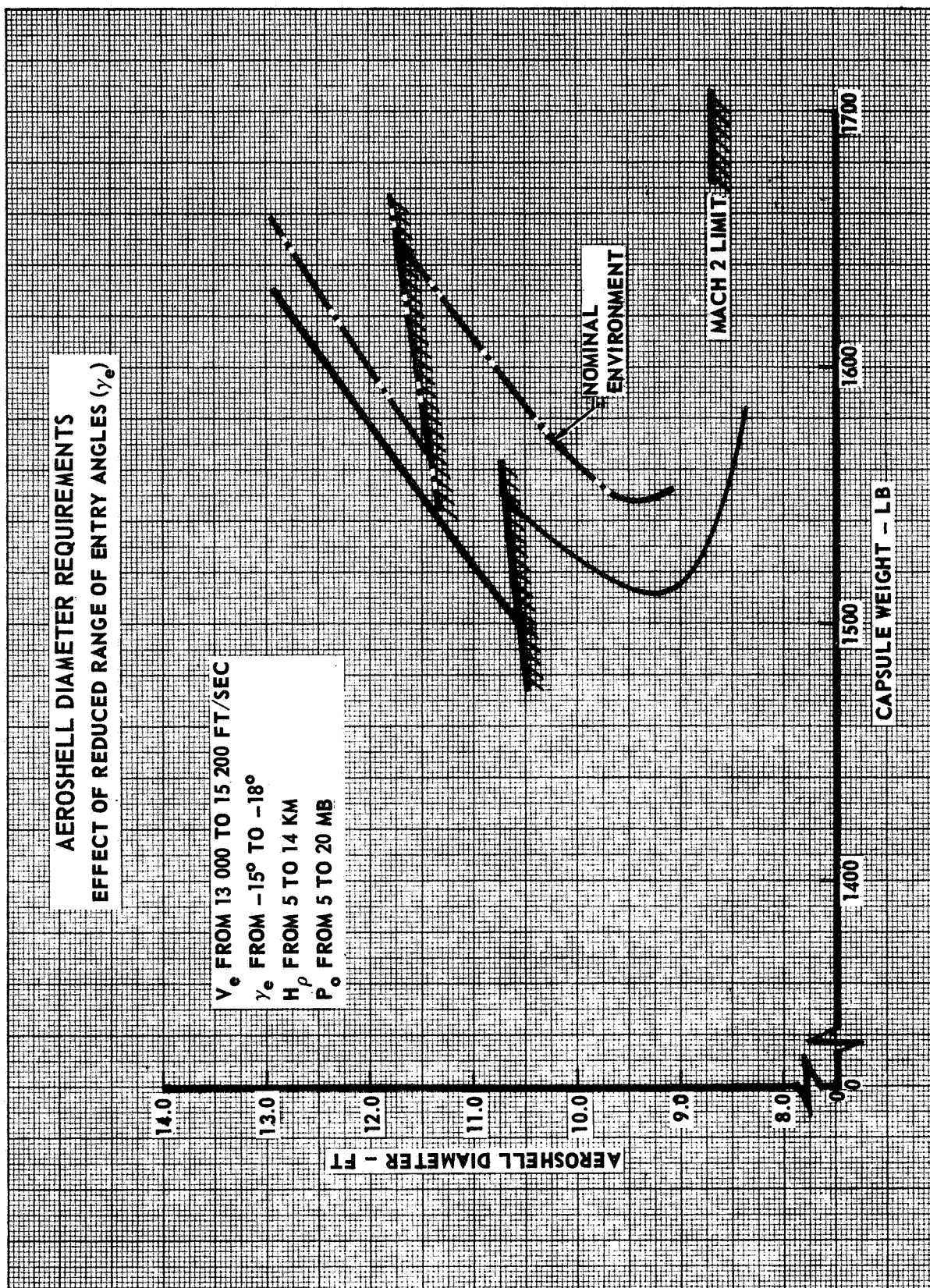


FIGURE 3.4-8

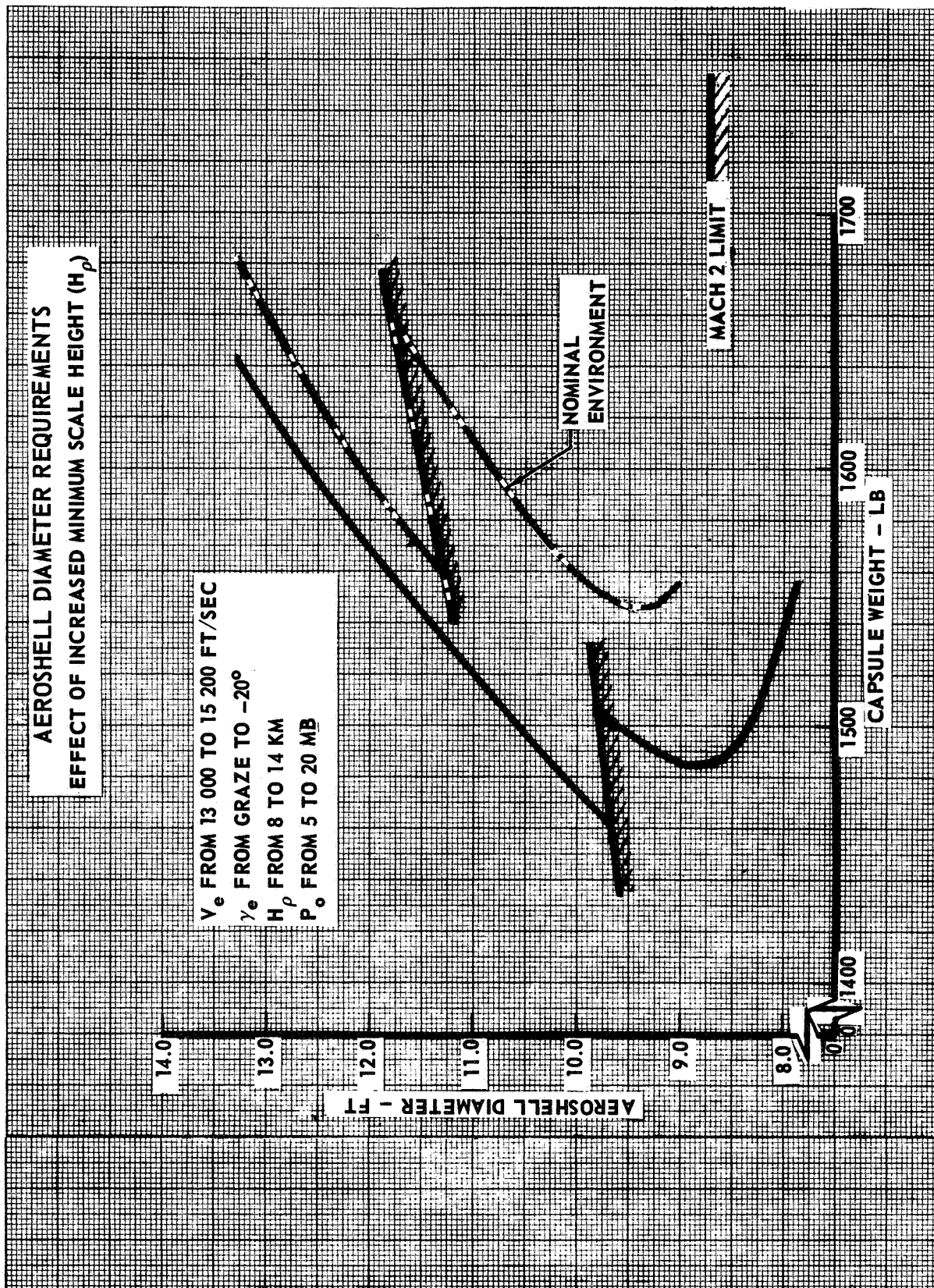


FIGURE 3.4-9

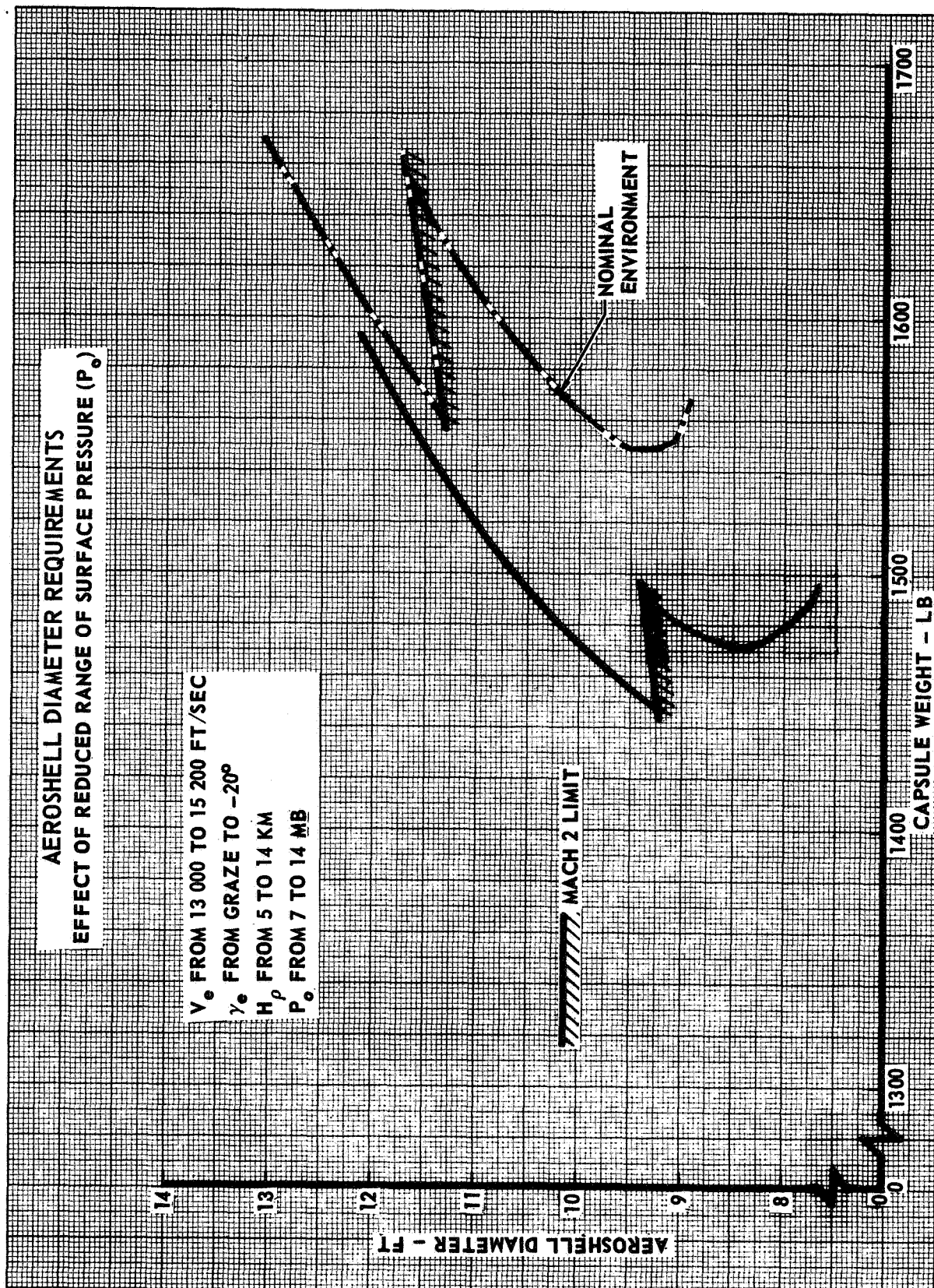


FIGURE 3.4-10

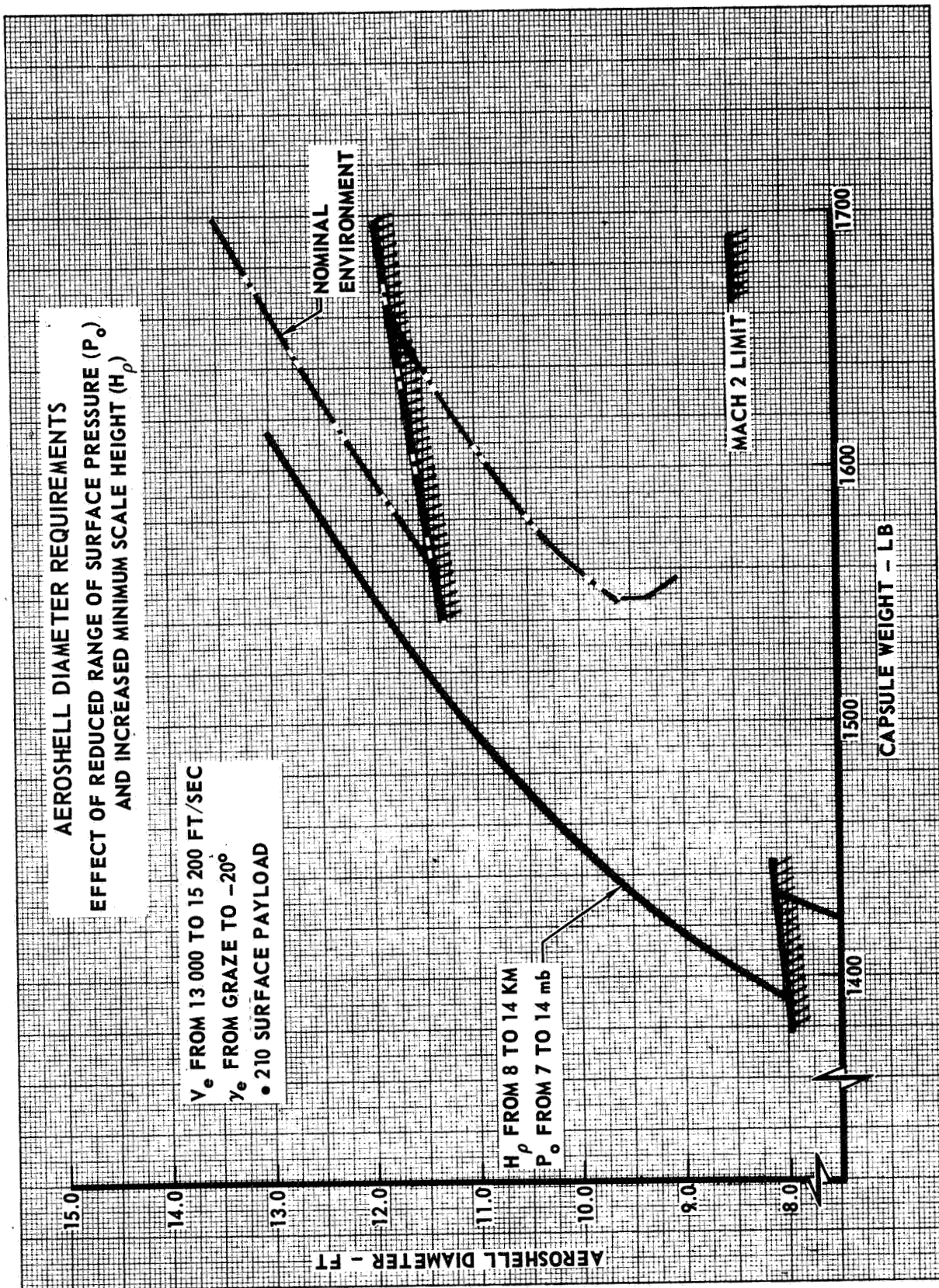


FIGURE 3.4-1.1

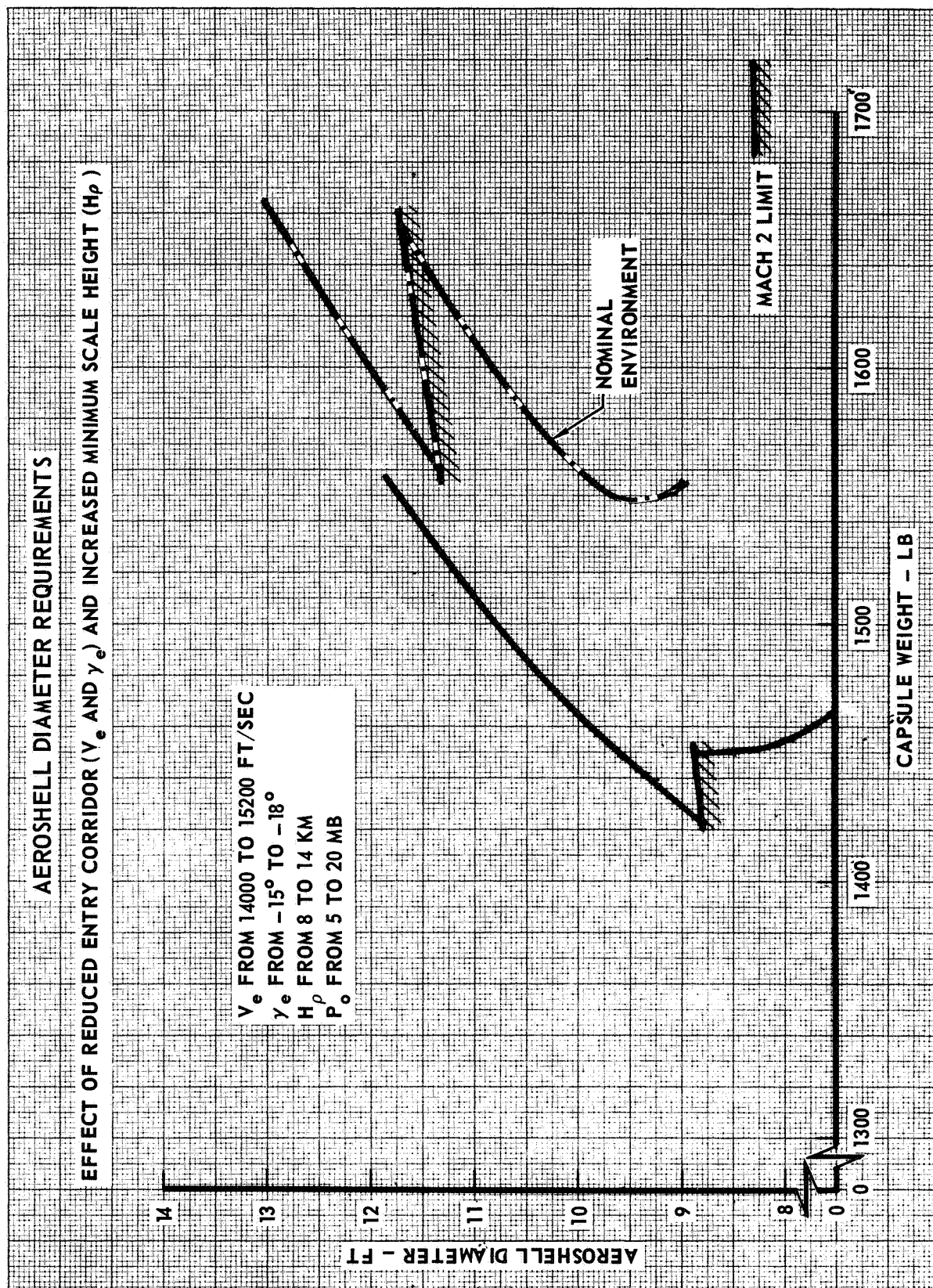


FIGURE 3.4-12

variations considered. All of these curves are for a constant surface payload weight of 210 lb. Each figure also contains the curve for the nominal environment, to provide a convenient reference. The data above the Mach 2 limit line is based on the use of parachutes; AIDS are used below the line where higher deployment Mach numbers are required.

By comparing Figures 3.4-5 through -8 (the entry corridor variations) with Figures 3.4-9 through -11 (the atmosphere variations), it is clear that the atmospheric definition has a more powerful influence on flight capsule diameter and weight than the entry conditions. This means that a wide entry corridor can be retained at nominal penalty to provide operational flexibility while very close attention must be given to the problem of better Martian atmosphere definition.

3.4.2 SURFACE ENVIRONMENT DEFINITION - The surface environment defined for the capsule system parametric studies was as follows:

Continuous Slopes, deg	± 34
*Abrupt Slope Changes, deg	± 68
Bearing Capacity, psi	6 to
Friction Coefficient	.3 to 1.0
Surface Rocks, max. dia., in.	5.0
Length of Surface Slope, ft	328 for 34 deg slope 6562 for 10 deg slope
Temperature	-190°F continuous, or +120°F to -150°F diurnal variation, whichever is worse
Winds 1 m above surface, ft/sec	220

* Local slopes shall not exceed ± 34 degree relative to the horizontal

The most important of these are the slopes, the temperature, and the winds. In order to determine the sensitivity of the capsule systems and operations to this environment definition, these three parameters were assigned less restrictive values:

Slopes	± 20 degree
Temperature	+ 120°F to -150°F only
Winds	118 ft/sec

3.4.2.1 Capsule system sensitivity. - The surface environment definition affects two capsule delivery systems (landing and TPS) as well as the surface payload. Surface slopes have a direct effect on the landing system weight, as shown in Figure 3.2.10-3. A reduction in maximum surface slope reduces the landing system weight by about 20%.

The most important effect of reduced near-surface winds (220 to 118 ft/sec) is to decrease the TPS ΔV requirement by about 50 ft/sec. Using the curves of Section 3.2.7 for a given TPS and guidance mode combination, we can determine that the TPS weight can be reduced by about 4%.

Elimination of the continuous cold day (-190°F) and variations in the cyclic day surface temperature profile have a major impact on the surface payload thermal control system. The thermal control system parametrics of Section 3.2.6 are based on a temperature profile consisting of both a continuous cold day temperature and a nominal cyclic day temperature profile. Three cyclic day temperature profiles, consisting of a maximum expected, a minimum expected, and the nominal profile used for the reference parametrics, are plotted in Figures 3.2.6-8 and -9. Two design alternatives for thermal control were analyzed; i.e., electrical heaters which supply controllable heating, and radioisotopes which supply constant heating. Active cooling was not an acceptable design alternative. Of interest now is the sensitivity of the thermal control system design to variations in the nominal temperature profile of Section 3.2.6. Figure 3.4-13 presents the required thermal control heating energy in $\text{Wh/ft}^2\text{-day}$ as it is affected by eliminating the cold day requirement and by imposing the three cyclic day temperature environments, by themselves and in combination with the cold day requirement. The heating energy is that needed to assure a battery compartment temperature between 50°F and 125°F throughout one diurnal cycle. The reference design case prescribes a heating energy of $122 \text{ Wh/ft}^2\text{-day}$ provided by electrical heating only. Eliminating the cold day requirement reduces this energy to $59 \text{ Wh/ft}^2\text{-day}$, which represents a savings of 31 lb in thermal control weight. Minimum thermal control weight is achieved for the maximum cyclic temperature day: $29 \text{ Wh/ft}^2\text{-day}$, resulting in a net weight reduction in thermal control system of 35 lb. Imposing a requirement to meet the full range of cyclic temperatures in combination with the

ENVIRONMENT SENSITIVITY OF THERMAL CONTROL RELATED COMPONENTS

• ONE DAY MISSION

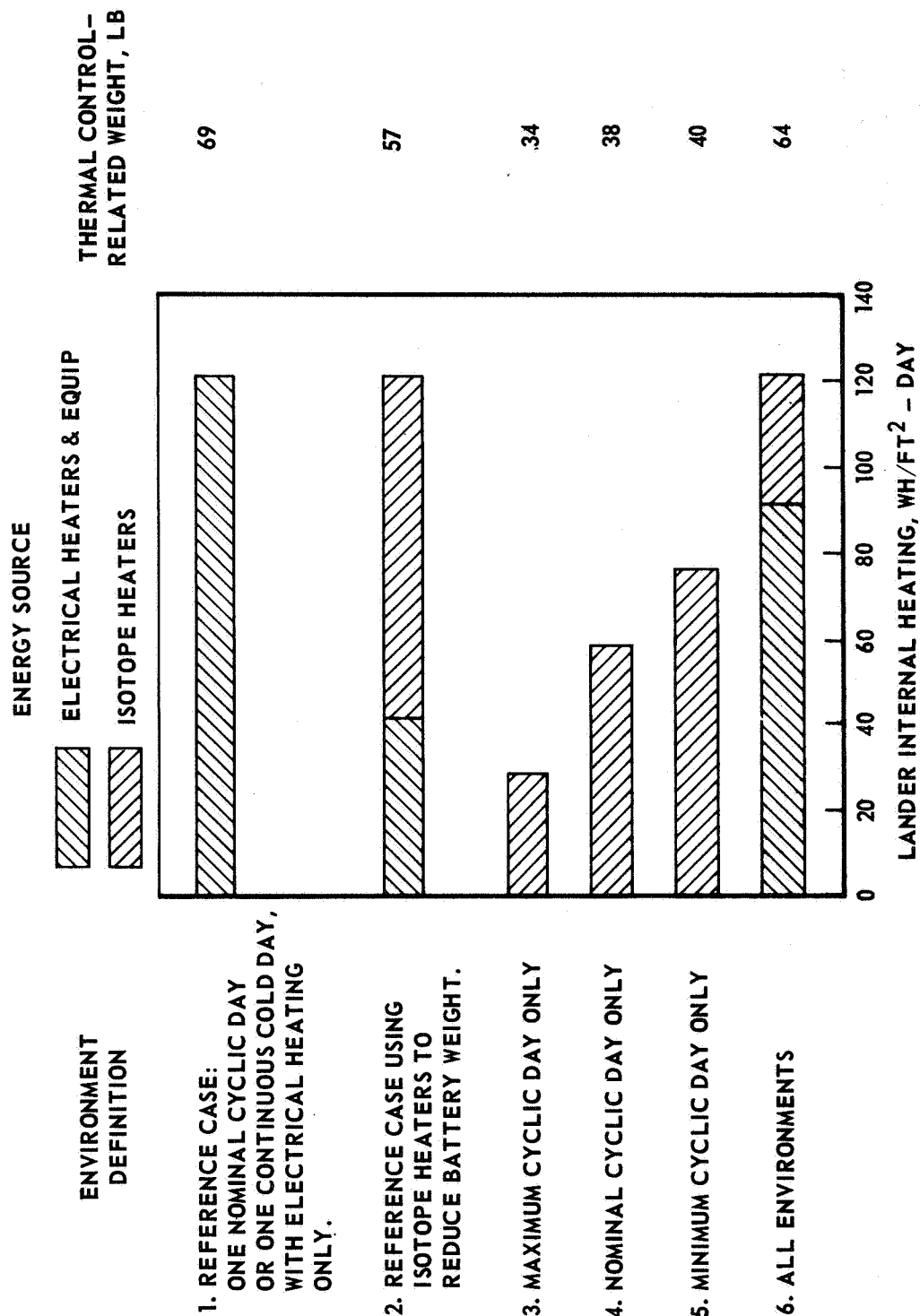


FIGURE 3.4-13

continuous cold day is the most stringent situation which results in an increase in thermal control system weight of 7 lb from the reference case using isotopes. It is important to note that the optimum makeup of the thermal control system between controllable heat (supplied by electrical heaters) and constant heat (supplied by radioisotopes) is in itself dependent upon the external temperature profile which is assumed to exist on Mars.

3.4.3 FLIGHT CAPSULE SENSITIVITY - The total effect of reducing the entry corridor and the atmospheric and surface environments can be seen in Figure 3.4-14 for the two missions for which conceptual designs are presented in Section 4: the one day and 90 day duration missions with the minimum science payload of 30 lb. The surface payload weights are noted for reference.

Table 3.4-3 shows how the capsule system weights vary for the one day duration cases. The major changes are in the canister (due mostly to the smaller aeroshell diameter), the aeroshell (smaller diameter due to higher $m/C_D A$), the TPS (lower lander weight and lower wind velocity), the impact system (lower lander weight and lower slopes), and the landed power supply (no cold day). The total capsule weight is reduced by 224 lb, or about 16%.

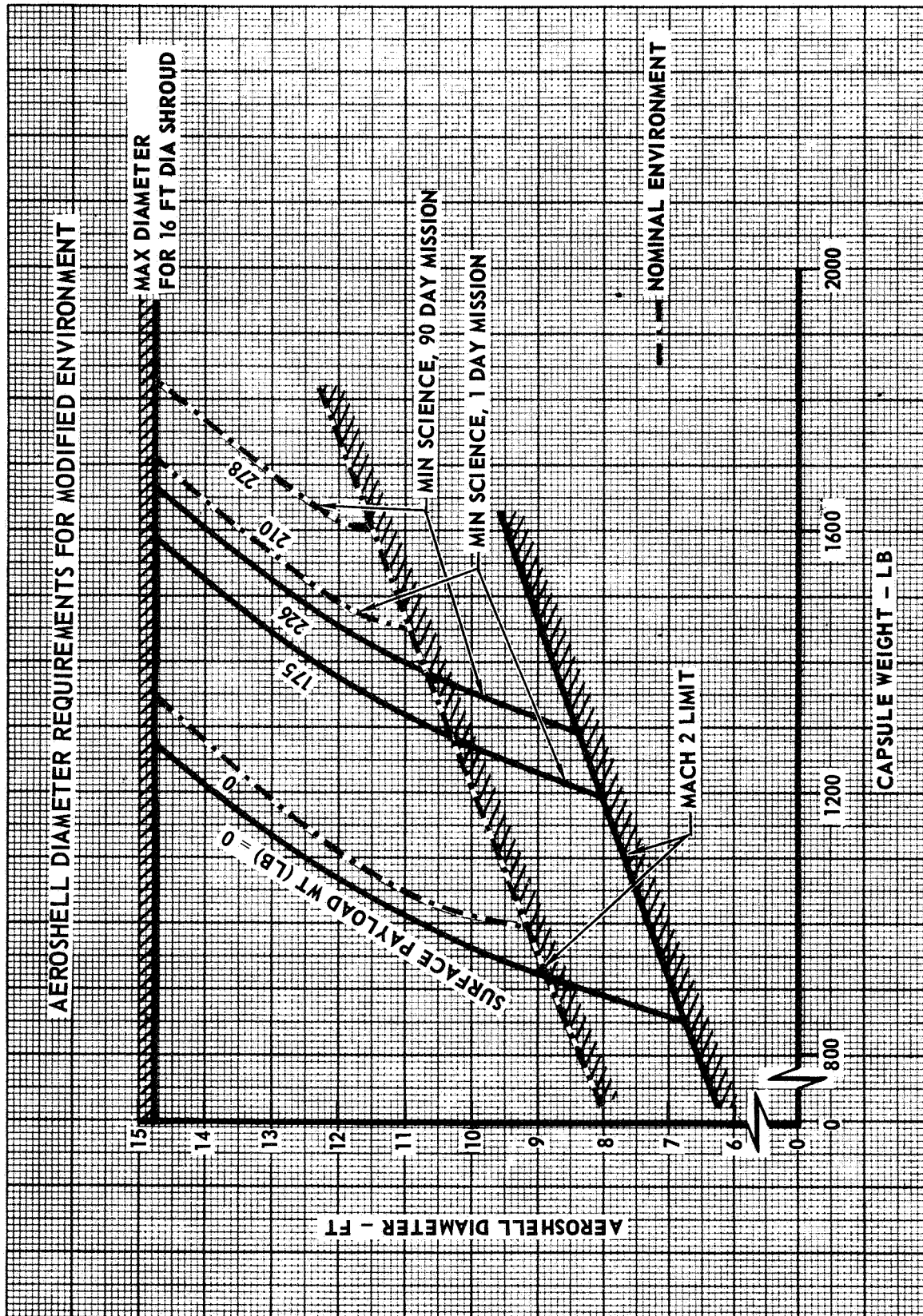


FIGURE 3.4-14

TABLE 3.4-3
FLIGHT CAPSULE SENSITIVITY TO ENVIRONMENT
WEIGHT SUMMARY

	FULL ENVIRONMENT	MODIFIED ENVIRONMENT	ΔW
DELIVERY SYSTEMS			
STERILE CANISTER	174.0	129.6	44.4
ADAPTER	16.2	15.2	1.0
DE-ORBIT PROPULSION	81.0	71.0	10.0
AEROSHELL	197.1	131.0	66.1
TEMP CONTROL SYS	70.0	58.2	11.8
ATTITUDE CONTROL SYS	40.6	41.1	- 0.5
PARACHUTE	40.1	34.3	5.8
TPS	208.7	179.8	28.9
IMPACT SYS	128.4	105.6	22.8
FIXED WEIGHT	283.8	283.8	-
SURFACE PAYLOAD *			
STRUCTURE & MOUNTING	44.6	40.9	3.7
THERMAL CONTROL	32.5	37.6	- 5.1
POWER & SEQUENCER	78.2	43.4	34.8
FIXED WEIGHTS	54.6	54.6	-
FLIGHT CAPSULE	1449.8	1226.1	223.7

* 30 LB SCIENCE PAYLOAD, 1 DAY MISSION

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

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