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VOYAGER SPACECRAFT SYSTEM STUDY

(PHASE II SATURN V LAUNCH VEHICLE)

FINAL REPORT

VOLUME I

SUMMARY

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INTRODUCTION

A study has been conducted by General Electric to investigate the application of the Saturn V launch vehicle to the unmanned exploration of Mars using a Voyager-type spacecraft. Specific objectives of this study were to:

1. Define Landers capable of carrying scientific payloads of 250, 500, 1,000, 2,500 and 5,000 pounds. The study was not intended to define the make-up of the scientific payload, but a range of required electrical power and communication bit rates was assumed for each payload size.

2. Identify problem areas encountered in the design of subsystems for very large, gross-weight Landers.

3. Explore the use of the large available energy of the Saturn V launch vehicle to achieve certain desirable mission features. This included investigation of added weight for reliability, shorter trip time for reliability, adjustment of trip time to arrive at the planet at a particularly desirable time, etc.

4. Identify system configurations that are capable of delivering the various Lander sizes to Mars.

5. Develop cost and schedule information for a Saturn V program.

Ground rules were established for this study to insure that the results would:

1. Be compatible with the Saturn V launch vehicle as presently defined.

2. Cover the range of uncertainty in the definition of the Martian environment.

3. Produce results that can be compared to the two previous Voyager studies using the Saturn IB/SVI and the Titan IIIC launch vehicles.

The Saturn V imposes constraints on the total weight and volume of the system. The weight capability was obtained from JPL, and shroud volume limitations were defined in a general sense through discussions with JPL and MSFC. In system designs that do not involve carrying Landers into orbit, the shroud volume limitation is encountered before the weight limit is reached.

The primary Martian environment of concern in the design of Lander vehicles is the atmospheric density. The range of surface pressure considered in this study is 11 to 30 millibars, which is consistent with JPL model atmospheres G through K. The entry angle corridor used in the basic study is 20 to 35 degrees, which is consistent with the guidance accuracies used in the Saturn IB and Titan IIIC studies. Primary retardation is achieved by parachutes, as was done in the previous studies. The combination of these parameters requires that the ballistic coefficient, W/C_DA, of the entering vehicle be no greater than 15 lbs/ft^2.

As in the previous studies, the ground rule was imposed that the designs would be based on 1965 state-of-the-art. A final ground rule imposed was that major emphasis would not be placed on re-evaluating subsystem design approaches that were arrived at in the previous Voyager studies. Rather, the same design concepts would be used unless factors, such as vehicle size, forced a change in approach.

Results of the study are presented so that selection of a scientific payload weight, power and required communication rate will allow determination of the weight of the remaining gross payload subsystems. Gross payload includes the electrical power subsystem, thermal control subsystem, and communication subsystem plus the scientific payload. Additional curves provide determination of the gross Lander weight as a function of the total gross payload weight; the Lander weight can be broken down into its subsystems, namely: structure, heat shield, retardation, impact attenuation, ground orientation, separation and spin, and delta impulse rocket. Given the gross Lander weight, the weight of the Bus system to deliver some number of these Landers to Mars is presented on additional curves. The Bus subsystem weight breakdown between structure, guidance and control, propulsion, and communication can be determined.

Further results of the study show that scientific payload weights up to 5000 pounds can be carried in Landers that are compatible with the Saturn V booster. However, this is nearly the upper limit of payload weight achievable for Landers restricted to a W/C_DA of 15 lb/ft^2 using extendible flaps to achieve the large drag area required for large weights. Further, restriction of W/C_DA to 15 makes very inefficient use of the Saturn V weight capability, due to the poor packaging efficiency of these Landers. Volume limitations for an aerodynamic shroud of reasonable length are such that only about half the weight capability of the Saturn V can be packaged.

Preliminary investigations conducted during this study show that a large increase in payload carried can be achieved if the W/C_DA is increased. Because of the extreme significance of this parameter, as evidenced in all the Voyager studies to date, several detailed studies are recommended in Section 2 to determine if a larger value can be used in future design studies. In summary, these studies should include:

1. Current estimates of DSIF capability indicate that entry corridor tolerances tighter than the 15 degrees used in this study can be readily achieved. A more detailed guidance analysis is recommended to establish the entry corridor achievable as a function of guidance system implementation.

2. A detailed comparison of the alternate approaches to the retardation system design should be made in terms of weight, reliability, state of development, cost and compatibility with the scientific mission. Parameters would be the entry corridor accuracy, Martian atmosphere and vehicle size.

3. Entry from orbit was investigated on a preliminary basis in this study and shows some advantages in terms of payload weight and mission flexibility. A more detailed study of this approach is required.

Additional conclusions resulting from the study are:

1. Use of the largest vehicles considered in this study may dictate some change in the presently specified requirements for sterilization.

2. Use of radioisotope thermoelectric generators (RTG) in a Lander imposes two serious interface problems. The first is a radiation level problem associated with sensitive scientific instruments. The second is a thermal control problem during ground operations and transit. The thermal problem is further complicated by the sterilization interface, requiring heat removal during thermal sterilization and compatibility of the sterilization barrier with heat removal during transit. Since RTG's appear to be the only feasible approach to the design of a long-life Lander power system, there interface problems should be subjected to a detailed study.

3. The large total energy available from the Saturn V booster can be used to provide considerable flexibility in the conduct of a Voyager mission. Wide launch windows and short trip times can be achieved. Control of these parameters yields reasonable control over the Martian season at planet encounter if arrival at a particular season is desired. Short trips can be used to improve reliability and to achieve shorter communication distances at planet encounter. The value of this flexibility can be assessed in detail only when the specific scientific mission is defined.

Statements regarding the effectiveness of the Saturn V launch vehicle compared to the two previous vehicles studies, i.e., Saturn IB/SVI and Titan IIIC, cannot be made in the absence of more specific definition of the scientific mission. Certainly, if very large integrated scientific payloads are identified, in excess of 1000 pounds, a clear requirement exists for the use of a launch vehicle with Saturn V capability. On the other hand, if the scientific payload is divisible into smaller units, but large numbers of total Landers are required, a cost effectiveness comparison between single Landers launched by the smaller boosters and multiple Landers launched by Saturn V is required before a choice can be made.

1. SUMMARY

This report presents the results of a study, conducted under Contract 950847, Phase 2, to investigate the application of the Saturn V launch vehicle to the unmanned exploration of Mars using a Voyager-type spacecraft. Extensive use has been made of work performed in two previous Voyager studies; Contract NASw-696 which defined the Voyager mission and spacecraft based on the Saturn IB/SVI launch vehicle, and Contract 950847, Phase 1, which investigated Voyager spacecraft compatible with the Titan IIIC launch vehicle.

1.1 STUDY OBJECTIVES

The previous two Voyager studies conducted by General Electric resulted in the design of Orbiters which weighed roughly 2000 pounds and Landers which ranged from 1300 to 2000 pounds. An Orbiter of 2000 to 3000-pounds would appear to be sufficient to carry most of the scientific experiments identified to date as being useful in an orbiting mission. In the case of Landers, however, work is in progress, e.g., the Automated Biological Laboratory study being conducted by Aeronutronics, which may result in the definition of scientific payloads that require a much larger Lander vehicle to place them on the surface of Mars. The purpose of the present study was to perform conceptual design of Landers to carry a range of scientific payload sizes up to a maximum of 5000 pounds. Orbiter work was limited to identification of means of including Orbiters of the type defined in the Titan IIIC study along with one or more Landers to make up a total Saturn V payload.

Specific objectives of this study were:

1. Define Landers capable of carrying scientific payloads of 250, 500, 1000, 2500, and 5000 pounds. No attempt was made to define the make-up of the scientific payload, but a range of required electrical power and communication bit rates was assumed for each payload size.

2. Identify problem areas encountered in the design of subsystems for very large gross weight Landers.

3. Explore the use of the large available energy of the Saturn V launch vehicle to achieve certain desirable mission features. This includes investigation of added weight for reliability, shorter trip time for reliability, adjustment of trip time to arrive at the planet at a particularly desirable time, etc.

4. Identify system configurations that are capable of delivering the various Lander sizes to Mars.

5. Develop cost and schedule information for a Saturn V program.

6. Explore alternate subsystem designs in specific areas.

1.2 STUDY APPROACH AND GROUND RULES

1.2.1 APPROACH

Since the scientific payload to be placed on Mars is not well defined in terms of weight, power required or data rate, this study was aimed at generating design information in a parametric form so that when a specific payload is selected, the results of this study can be used to define a system for that payload compatible with the Saturn V vehicle.

To generate the parametric data, five scientific payload weights were selected at 250, 500, 1000, 2500, and 5000 pounds. For each of these payload weights, a nominal, maximum, and minimum power level required by the payload was defined. Similarly, a nominal, maximum, and minimum data rate was assumed for each payload size. The spread between minimum and maximum power and data rate was made sufficiently large to encompass any likely payload of a given weight range. By trading off antenna size and transmitted power, a minimum weight communication system was defined for the total range of bit rates involved. From this optimization study, curves of communication system weight and power were generated as a function of bit rate. An electrical power system based on the use of radioisotope thermoelectric generators (RTG) was sized to supply the range of power required by the scientific payload, the communication system, and other vehicle needs. A curve of power subsystem weight as a function of power output was prepared. A temperature control system was then designed and a curve of weight versus total vehicle power was prepared.

The above four subsystems: 1) scientific payload, 2) communication subsystem, 3) electrical power subsystem, and 4) thermal control subsystem, which constitute the gross payload were thus defined on a parametric basis. Selection of a specific scientific payload weight, power, and data rate together with the above curves will yield a specific gross payload weight.

The basic Lander vehicle consists of the following subsystems:

1. Structure
2. Heat Shield
3. Retardation
4. Impact Attenuation
5. Delta Impulse Rocket
6. Separation and Spin Up
7. Ground Orientation

Five entry vehicles were designed to carry the scientific payloads of 250, 500, 1000, 2500 and 5000 pounds with nominal levels of power and data rate. From these designs, curves were prepared to show the weight of each of the above subsystems as a function of entry weight or gross vehicle weight as applicable.

In the case of the design of the Bus to deliver the Landers to Mars, the results are not as parametric as in the case of the gross payload or the basic Lander. Because of limitations imposed by the shroud, certain Lander sizes cannot be efficiently packaged. Bus designs were prepared for five Lander sizes that package reasonably well within the shroud, and curves prepared for Bus system weight as a function of number of Landers carried. If "in-between" Lander sizes are to be chosen, some loss in packaging efficiency may result with an attendant reduction in total payload carried.

1.2.2 GROUND RULES

The ground rules imposed on this study arise from the following considerations:

1. Compatibility with the Saturn V launch vehicle.

2. Uncertainty in the definition of the Martian environment.

3. Desirability of producing results that can be compared to the two previous Voyager studies using the Saturn IB/SVI and the Titan IIIC launch vehicles.

The Saturn V imposes constraints on the total weight and volume of the system. The weight capability as a function of required vis viva energy was obtained from JPL and is presented in Section 3.2. Shroud volume limitations were defined in a general sense through discussions with JPL and MSFC and are discussed in Section 6.1. In system designs which do not involve carrying Landers into orbit or using a higher W/C_DA Lander, the shroud volume limitation is encountered before the weight limit is reached.

The primary Martian environment of concern in the design of Lander vehicles is the atmospheric density. The range of surface pressure considered in this study is 11 to 30 millibars, consistent with JPL model atmosphere G through K. The entry-angle corridor used in the basic study is 20 to 35 degrees, consistent with the guidance accuracies used in the Saturn IB and Titan IIIC studies. Primary retardation is achieved by parachutes which imposes a requirement that the Mach number be 2.5 or less at an altitude of 20,000 feet or greater to allow deployment of drogue and main parachutes to achieve deceleration prior to impact. Finally, the maximum entry velocity considered in the design is 26,000 feet per second. The combination of these parameters:

Surface pressure = 11 mb

Maximum Entry Angle = 35 degrees

Mach number $\leq$ 2.5 at altitude $\geq$ 20,000 feet

Maximum Entry velocity $\leq$ 26,000 ft/sec requires that the ballistic coefficient (W/C_DA) of the entering vehicle be no greater than 15 lb/ft^2. In a later section it is pointed out that an increase in the ballistic coefficient, through improved entry corridor tolerances or determination of higher atmospheric pressure, will materially increase the efficiency of the system. However, for the basic parametric study, W/C_DA of 15 lb/ft^2 was used to be consistent with the previous Voyager work.

As in the previous studies, the ground rule was imposed that the designs would be based on 1965 state-of-the-art. The degree to which this has been achieved depends to a large extent upon the definition of state-of-the-art. In general, the technology on which the designs are based is here, but in many cases designs of the specific size range have not been built or even designed in detail. Specific examples that are worthy of note are:

1. While radioisotope thermoelectric generator technology is well developed and small units have been flown, this study recommends units in sizes up to 750 watts which will not have been accomplished in 1965.

2. In the communication area, RF output stages are obviously not available to cover the entire range of bit rates considered. Further, the approach to radiating high power in the low atmospheric density on Mars will not be completely explored in 1965. The use of array antennas, as discussed in this report, should provide a solution to this problem.

3. The retardation system uses parachutes for primary deceleration. This type of system has been flight proven in Earth entry tests. For the large vehicles considered in this study and the low density Martian atmosphere, up to four parachutes of 60-foot diameter are used. This is felt to be feasible but is again extending the design range of a proven concept.

4. Certainly the sterilization requirement and its implications throughout the design will not be thoroughly investigated by 1965.

A final ground rule imposed on this study was that major emphasis would not be placed on re-evaluating subsystem design approaches that were arrived at in the previous Voyager studies. Rather, the same design concepts would be used unless factors such as vehicle size forced a change in approach. Therefore, tradeoffs were not conducted to compare the basic approaches to retardation, etc., but rather the existing designs were sized to cover the range of vehicles being considered.

1.3 BASIC PARAMETRIC STUDY

This section presents a description of the system configuration considered in the basic study, a definition of the mission sequence associated with the system, and a summary of the study results obtained.

1.3.1 SYSTEM CONFIGURATION

For an entry vehicle with W/C_DA of 15 lb/ft^2, using extendable flaps to achieve high drag area for the large vehicles, the shroud volume accommodates a single Lander capable of carrying a 5000-pound payload, and up to 12 Landers capable of carrying 150 pounds of payload. This is described in more detail in Section 6.1.1. The basic 20-foot shroud diameter determines the largest fixed flare Lander weight that can be designed for W/C_DA of 15 lb/ft^2 at 6200 pounds gross weight. Above this weight, extendable flaps must be used to increase drag area. Below this weight, the Lander base diameter can be reduced below the maximum accommodated by the shroud. In configuring systems, three Lander diameters were selected to allow packaging as shown in Figure 1.3-1. The small Landers were packaged in clusters of three or four per level with the number of clusters determined by the available shroud height. The next size Lander considered was of such a diameter that it used the full shroud diameter. Above the size that could be carried using a fixed flare, extendable flaps were added to increase the drag area. The number of Landers that could be carried with flaps was determined by the length of flaps required and the shrould length available.

It is apparent from consideration of Figure 1.3-1 that Lander weights requiring a base diameter between 9 feet and 11.2 feet or between 11.2 feet and 20 feet may package somewhat less efficiently within the available shroud volume than the specific sizes chosen. Thus, some care must be exercised in the application of the parametric data derived in this study.

Determination of Bus configurations to deliver these Landers to Mars is the next consideration. It was concluded early in the study that use of a Bus for each of the small Landers was not feasible. First of all, the operational problem posed for the DSIF by a requirement to simultaneously handle 9 to 12 individual vehicles is nearly inconceivable. Additionally, the cost of Bus hardware for each small Lander is quite high for the very small benefit gained in terms of probable number of successful Lander impacts. Therefore, a decision was made to provide a Bus for each cluster of small Landers rather than for each Lander. For the large Landers, which are not packaged in clusters, a Bus is provided for each Lander.

The degree of integration of the Bus and Lander functions was considered in some degree. In the Saturn IB Voyager study, the Bus which delivered the Lander to Mars subsequently served as an Orbiter so all Bus functions were separate from the Lander and the Lander was inactive during transit. In the Titan IIIC study, due to weight limitations, Landers and Orbiters were launched on separate vehicles. In this case, the Bus that delivered the Lander to Mars had no function to perform after the Lander was delivered. To achieve maximum reliability for a given weight, the Bus uses the power and communication system aboard the Lander during transit instead of having a separate system for the Bus. When several Landers are carried by a single Bus, the problem of integration between Bus functions and Lander functions becomes

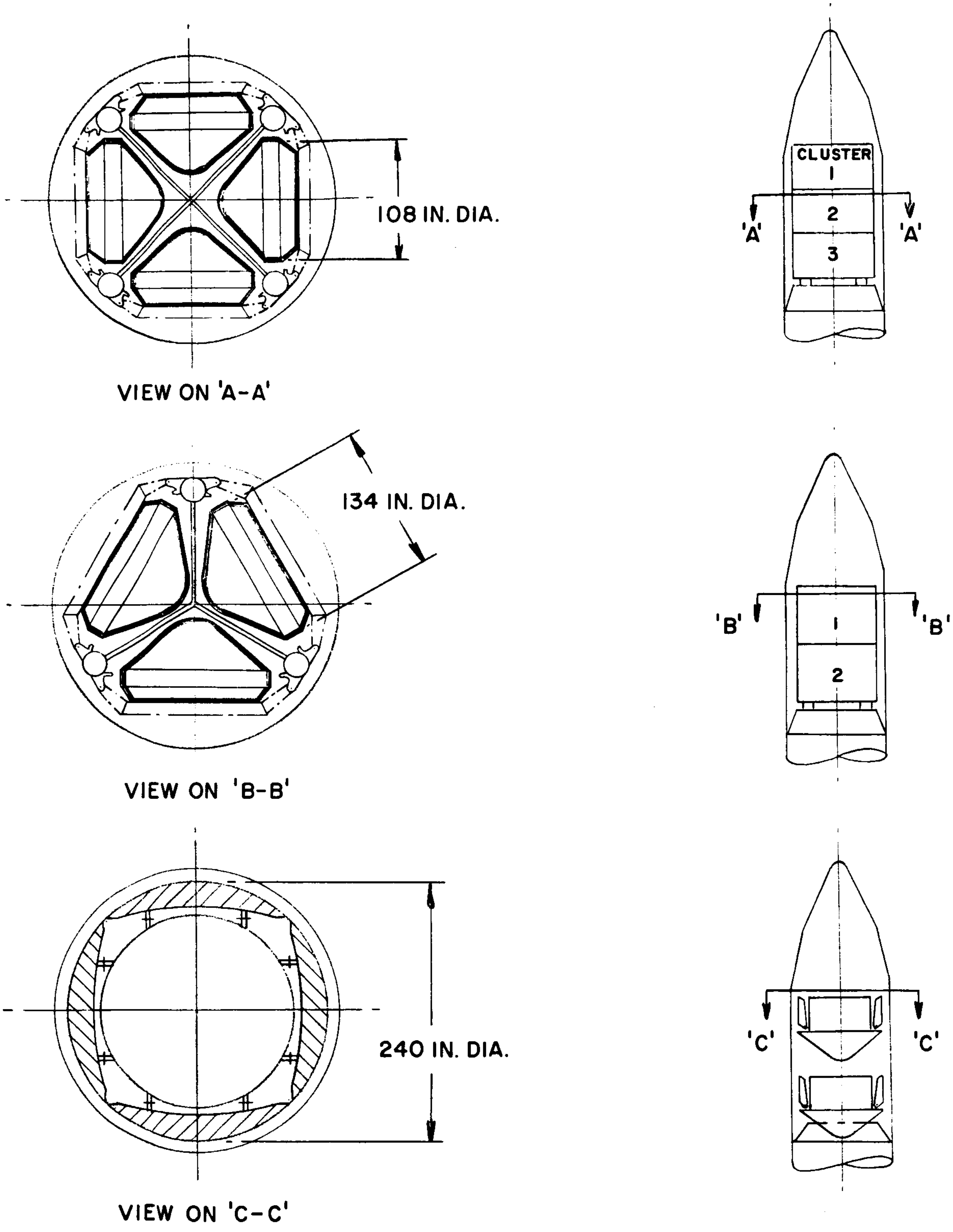


Figure 1.3-1. Lander Packaging

more troublesome. If the communication system within one Lander is to be used to process engineering data from the other Landers, the number of interconnections becomes quite numerous. This is particularly true if flexibility is to be provided to allow use of the communication system aboard any of the Landers to perform this function for all other Landers. It was ultimately decided to provide a communication system in the Cluster Bus rather than use the communication system aboard the Lander. An additional factor contributing to this decision was the fact that there is not a weight limitation on this system. That is, in no case is the Saturn V weight capability approached, and if the weight increase is not accompanied by a volume increase, it can be accommodated. Weight added to the Lander, however, requires an increase in diameter or flap length to maintain the W/C_DA and contributes directly to the volume problem.

In the case of Buses for a single large Lander, the communication system aboard the Lander is used during the transit phase. Because of the large heat dissipation in the final stage RF amplifier, however, a thermal control problem is created if the Lander output stage is used. On the surface of Mars, heat is rejected from the Lander by radiation. Since the aft cover is closed during transit, this means of heat transfer is ineffective and the large dissipation associated with this amplifier creates a problem. Therefore, an output stage is located in the Bus associated with an individual Lander.

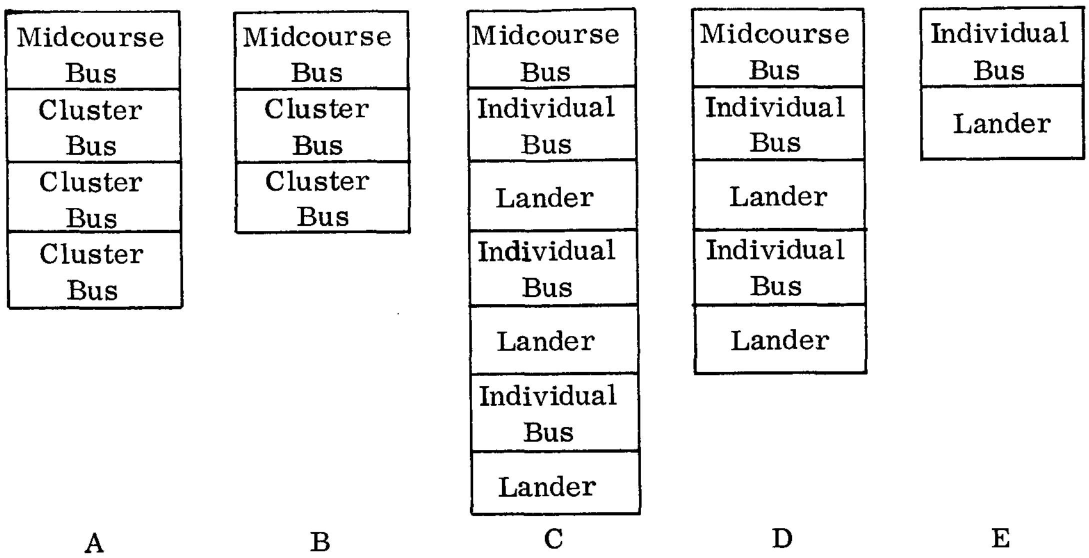
In all cases, power during transit is supplied by the RTG aboard one or more of the Landers.

One system concept which reduces operational problems and provides an increase in reliability is the use of a Midcourse Bus to provide the Bus function for the entire assembly of Landers and Individual/Cluster Buses until late in the transit phase. With this concept, the entire assembly would remain attached through the midcourse maneuver and until the vicinity of the planet is reached. Thus, the DSIF has only one vehicle to control through midcourse, and the operating time required of the Individual/Cluster Buses is reduced resulting in an increase in reliability. The Individual/Cluster Buses are sized, however, so that upon failure of the Midcourse Bus at any point in the trajectory, the system can be separated and the Individual/Cluster Buses can perform the midcourse correction as well as the terminal guidance maneuver. The Midcourse Bus has a communication system independent of the Landers, but uses power from the Lander RTGs.

The system configuration used in this study is shown in Figure 1.3-2. The number of Landers carried as a function of size is indicated. In the case of a single large Lander, of course, the Midcourse Bus is not used. In that case, the Individual Bus subsystems are made redundant to improve overall reliability.

1.3.2 MISSION PROFILE

The mission profile associated with this system is shown in Figure 1.3-3. After injection into the Mars transfer orbit, the entire assembly is separated from the launch vehicle and the Midcourse Bus stabilizes to the sun and canopus. This orientation is maintained throughout the transit phase except when velocity changes are being made. As pointed out previously, failure of the Midcourse Bus at any point will result in separation of the Individual-cluster/Buses which will then accomplish the mission. Power is supplied during transit by the Lander RTGs. Since the Lander aft covers are closed, RTG cooling by radiation is not feasible and a liquid cooling loop is provided to carry RTG heat to a radiator which is exposed to space. Communication is through an omni-directional antenna while the system is near Earth, and through a small dish when the system is near Mars.



System	Landers Per Cluster	Total Landers	Weight Per Lander	Scientific Payload Per Lander
A	4	12	1400	150
B	3	6	2000	370
C	-	3	6200	1760
D	-	2	13,100	3100
E	-	1	26,200	5000

Figure 1.3-2. System Configurations

A midcourse correction is made after sufficient tracking is accomplished to establish the trajectory. Fly-by trajectories are used in all cases to avoid a requirement for sterilization of the Bus or extreme reliability in a propulsion system to deflect the Bus from an impact to a fly-by trajectory.

Sufficient power is available to maintain communication throughout the midcourse maneuver through the omni-antenna.

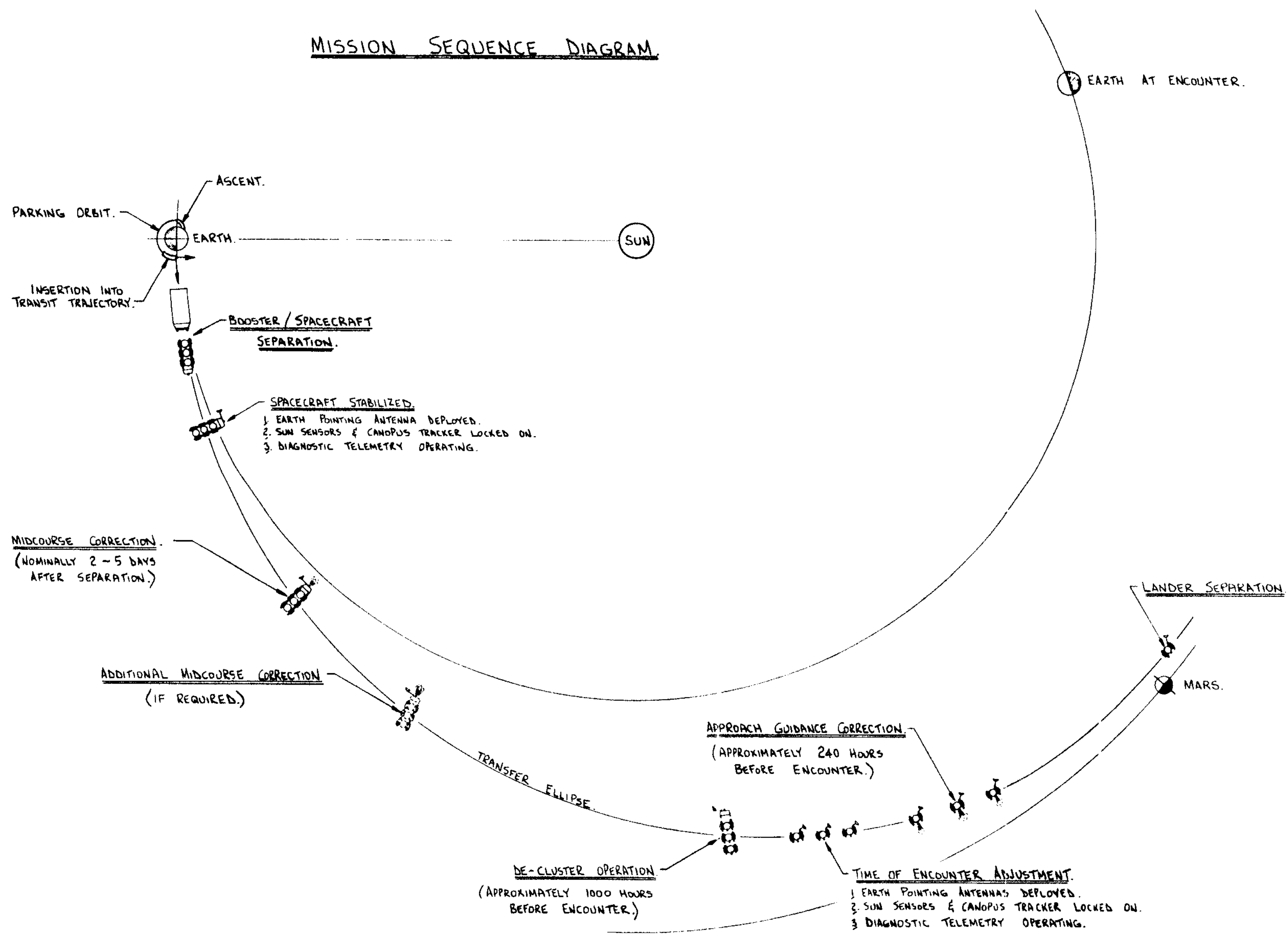


Figure 1.3-3. Mission Profile

Following the midcourse maneuver, the system is "inactive" until the vehicle is approximately 1000 hours from Mars. At this point, assuming longitudinal separation of the Landers is desired on the surface, the individual or clustered Landers are separated and a velocity correction made to adjust time of arrival. The magnitude of the velocity correction as a function of separation time desired and time from encounter is shown in Section 3.2.

Subsequent to the arrival-time separation correction, terminal trajectory measurements are made, either by DSIF only or aided by an on-board planet line-of-sight sensor depending upon the accuracy required and that achievable using DSIF. This is discussed in more detail in Section 3.4. A terminal correction is made to achieve a fly-by trajectory of sufficient accuracy that the required entry corridor can be achieved using a fixed impulse rocket aboard the Lander to divert it from a fly-by to an impact trajectory. After this correction, further control of the entry corridor can be achieved by controlling the separation point of the Lander and the angular orientation of the solid rocket.

At a nominal distance of 150,000 nautical miles from the planet, the Bus orients the solid rocket in the desired direction for imparting the $\Delta \nu$. This is achieved by ground command. The Lander is separated and spun up to provide stability during the solid rocket burn. After some time delay, the solid rocket is fired and the Lander is on an impact trajectory. In the case of several Landers in a cluster, capability is provided for out-of-plane firing of the solid rockets to achieve separation of the Landers on the surface.

During the final approach to the planet, the Lander RTG is still cooled by a liquid loop transferring heat to an external radiator. Just prior to entry, the empty solid rocket case and the radiator are jettisoned to reduce the entry weight as much as possible. From this point until the RTGs are deployed on the surface, cooling is achieved by means of a water boiler.

At an altitude of 20,000 feet or above, a drogue parachute is deployed and the extendable flaps are jettisoned on the large vehicles. The main parachute is then deployed and just prior to impact the retardation rockets are fired. Remaining velocity at impact is absorbed by fiberglass honeycomb crush-up material. After impact the Lander is oriented nose down, the aft cover is opened, RTGs are deployed, and the large antenna deployed and oriented to the Earth. The vehicle is then ready for operation. A more detailed description of the Lander sequence of events is presented in Section 5.

The communication links used for the three types of spacecraft configurations are as shown in Figures 1.3-4, 1.3-5 and 1.3-6.

Each Bus and Lander of the small-Lander configurations contains a complete communications subsystem comprising the deep space transmission subsystem (DSTS), data processing and storage subsystem (DPSS) and command and computer subsystem (CCS). Before Midcourse Bus separation, the Midcourse Bus communication subsystem

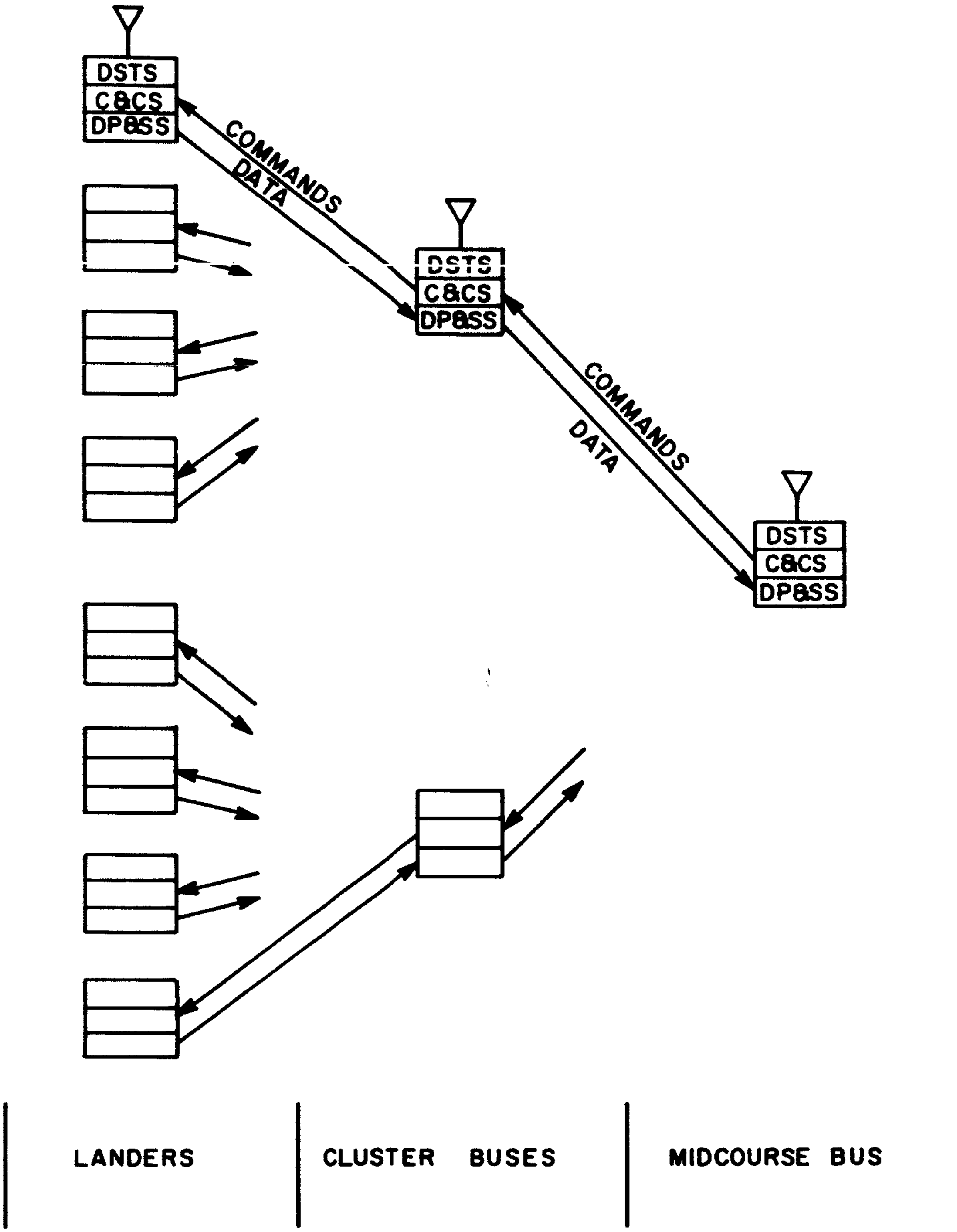


Figure 1.3-4. Bus/Lander Communication Interconnections (1400- and 2000-Pound Landers)

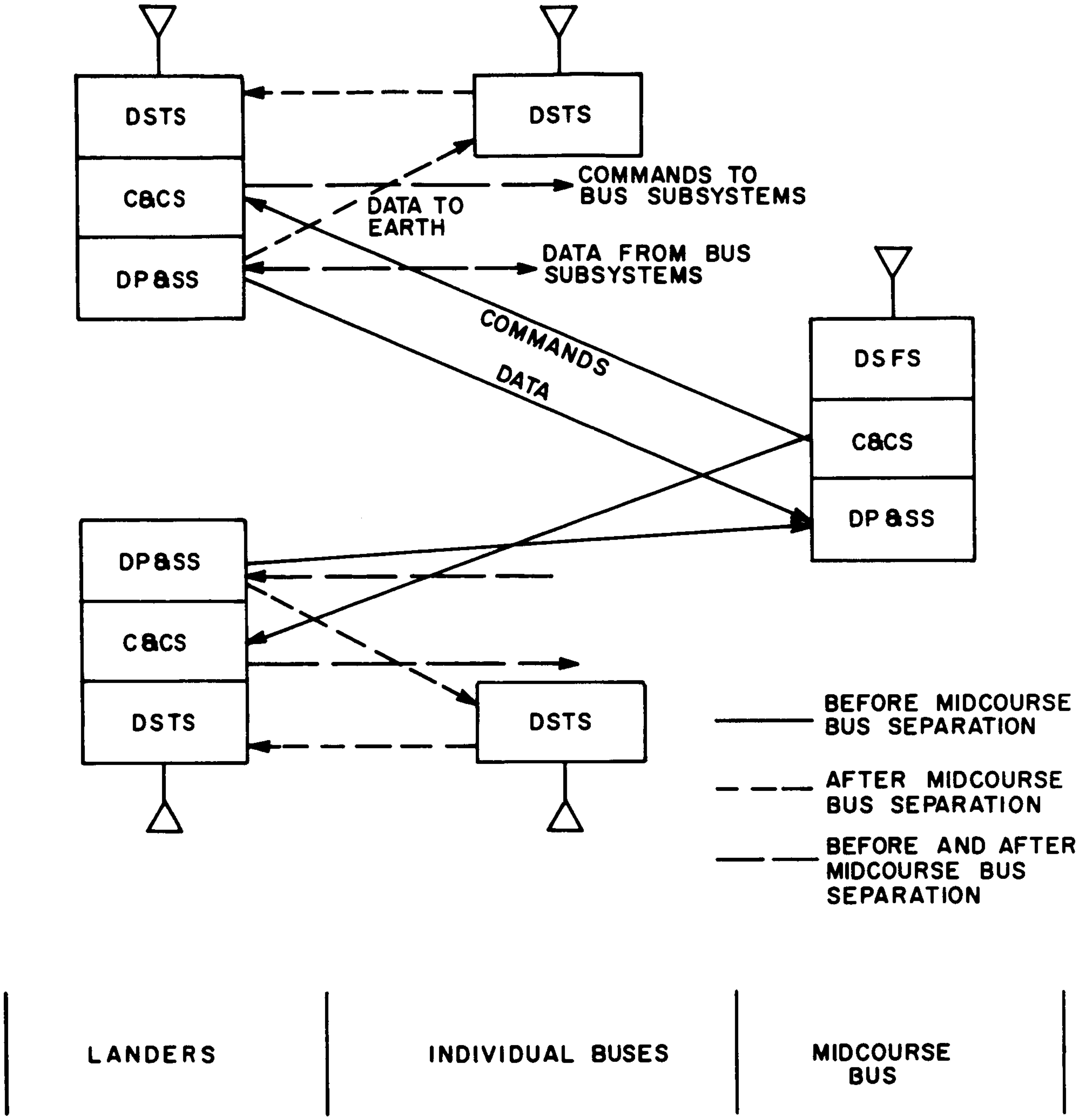


Figure 1.3-5. Bus/Lander Communication Interconnections
(6600- and 13,100-Pound Landers)

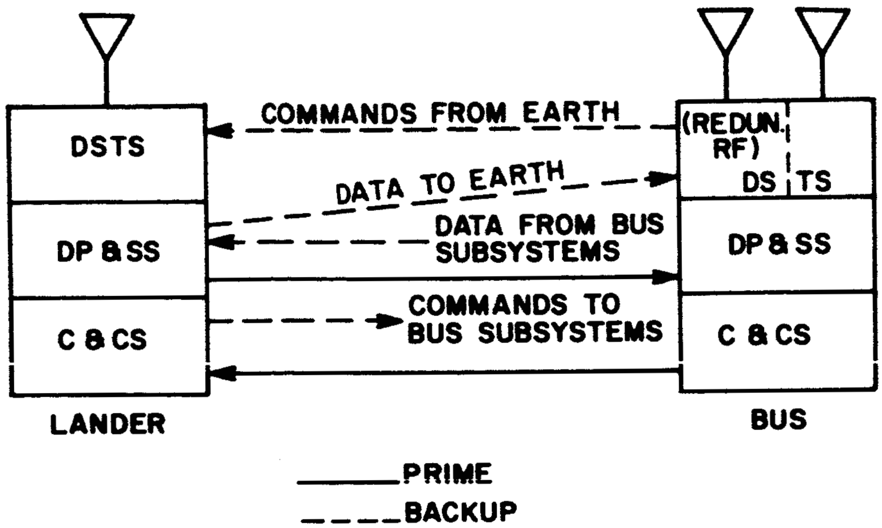


Figure 1.3-6. Bus/Lander Communication Interconnections (26,200-Pound Lander)

provides all communication with Earth, issues commands to the Midcourse Bus subsystems and to the CCS of the Cluster Buses, and collects data from the Midcourse Bus subsystems and from the DPSS of the Cluster Buses. The communication subsystem of each Cluster Bus, in turn, accepts commands from the Midcourse Bus, issues commands to its own Cluster Bus subsystems and to each CCS of its associated Landers, and collects data from its own Bus subsystem and from each DPSS of its Landers. Finally, each Lander communication subsystem accepts commands from its Cluster Bus and issues commands to and collects data from the Lander subsystems. The Midcourse Bus is therefore the central information processing point between Earth and the Cluster Buses, and each Cluster Bus is the central information processing point between the associated Landers and the Midcourse Bus.

After Midcourse Bus separation, each Cluster Bus performs the same functions; however, it now receives commands from and transmits data to the Earth through its own DSTS. Subsequent to separation from the Cluster Bus, each Lander performs its own communication functions.

In the Medium-Lander configurations, the Midcourse Bus communication subsystem performs the same functions as described previously; however, it is now connected directly to the Landers. The Cluster Buses are replaced by Individual Buses (one for each Lander) which contain only the RF portion of the communication subsystem. All command and data collection associated with a Lander and its Individual Bus is performed by the Lander CCS and DPSS.

After Midcourse Bus separation, the RF portion of an Individual Bus is used in conjunction with the command detectors of the Lander DSTS, and with the Lander DPSS and CCS to perform all required functions. The Lander RF is then actuated after separation from the Individual Bus.

The large-Lander configuration does not include a Midcourse Bus; however, the Individual Bus communication subsystem utilized has redundant, independent modes of operation so that its reliability is at least as great as that of the medium-Lander configurations. A complete Bus communication subsystem is provided for the prime mode. It functions with the Lander in the same manner as the previously described Cluster Buses. In addition, redundant RF equipment is available on the Bus. When used with the Lander command detectors, DPSS and CCS, a completely independent communication backup is formed.

1.3.3 STUDY RESULTS

The results obtained from the basic parametric study are summarized in this section. The approach taken is to start with a scientific payload, size a communication subsystem, power subsystem, and thermal-control subsystem compatible with this payload. A Lander vehicle is then sized to carry this gross payload, and some number of Landers is selected for the overall mission. Based on the Lander size and number of Landers, the Bus system is sized and this weight added to the Lander weight to yield the total Saturn V payload. Based on the total weight, the trajectory curves allow tradeoffs between launch window duration and trip time to Mars within constraints of launch azimuth and arrival velocity. Based on the selected trip time, the reliability curves show probability of mission success, and the communication curves show bit rate upon arrival at Mars.

The power and commurication bit rate required by the scientific payload as a function of payload size is shown in Figures 1.3-7 and 1.3-8. A nominal level and a maximum and minimum are shown in each case in an attempt to bracket the requirements likely to exist for any payload that is defined in the future. The communication bit rate is that available at an Earth-Mars separation of 1.4 AU. The basis for these curves is discussed in Section 4.1. Essentially, the lower end of the curves are based on the previous Voyager studies while the upper end represents an engineering estimate of the growth in both power and bit rate required as the payload size increases.

The bit rate required from Figure 1.3-8 varies from a minimum of 800 bits/second to a maximum of 70,000 bits/second. A weight optimization study was conducted on the communication system required to transmit this data rate and this study is presented in Section 4.2. The weight of those elements that are proportional to transmitted power or antenna size are expressed, and antenna size is traded off against transmitted power to yield a minimum weight radio system. From this study, the prime power required by the communication system as a function of bit rate is shown in Figure 1.3-9, and the weight of the system as a function of bit rate is shown in Figure 1.3-10. The weight and power required by the RF system is accurately

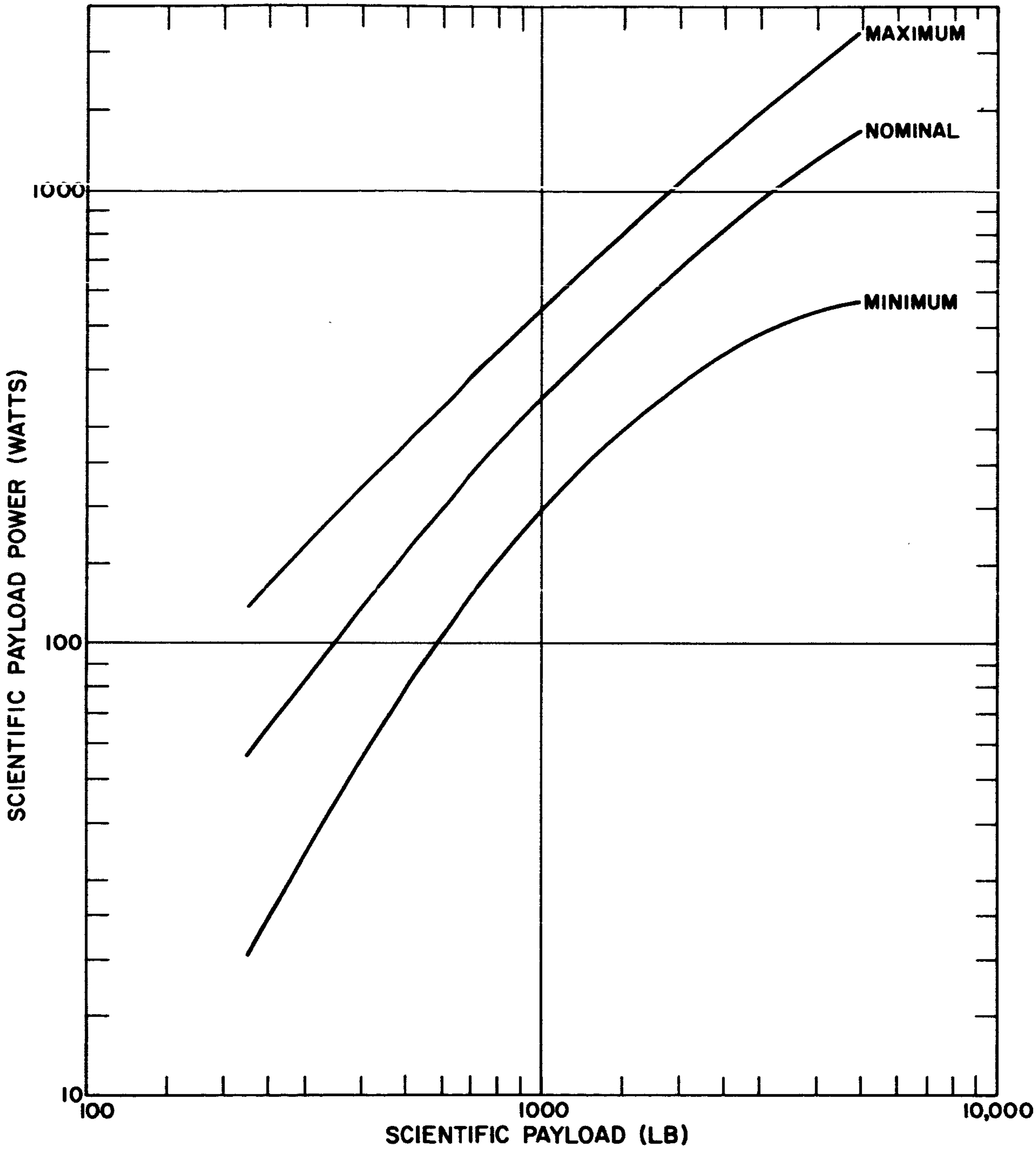


Figure 1.3-7. Scientific Payload Power Versus Weight

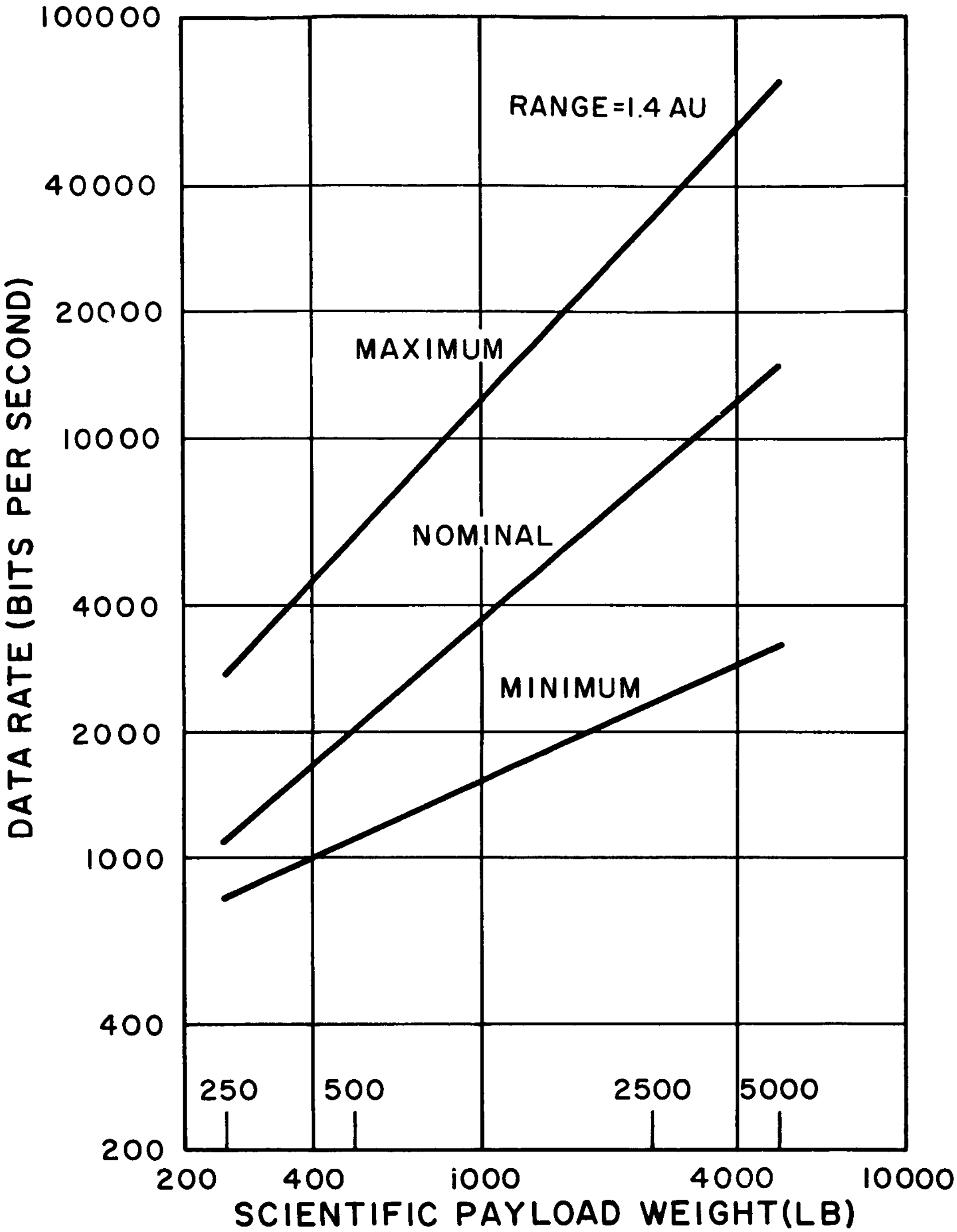


Figure 1.3-8. Scientific Payload Bit Rate Versus Weight

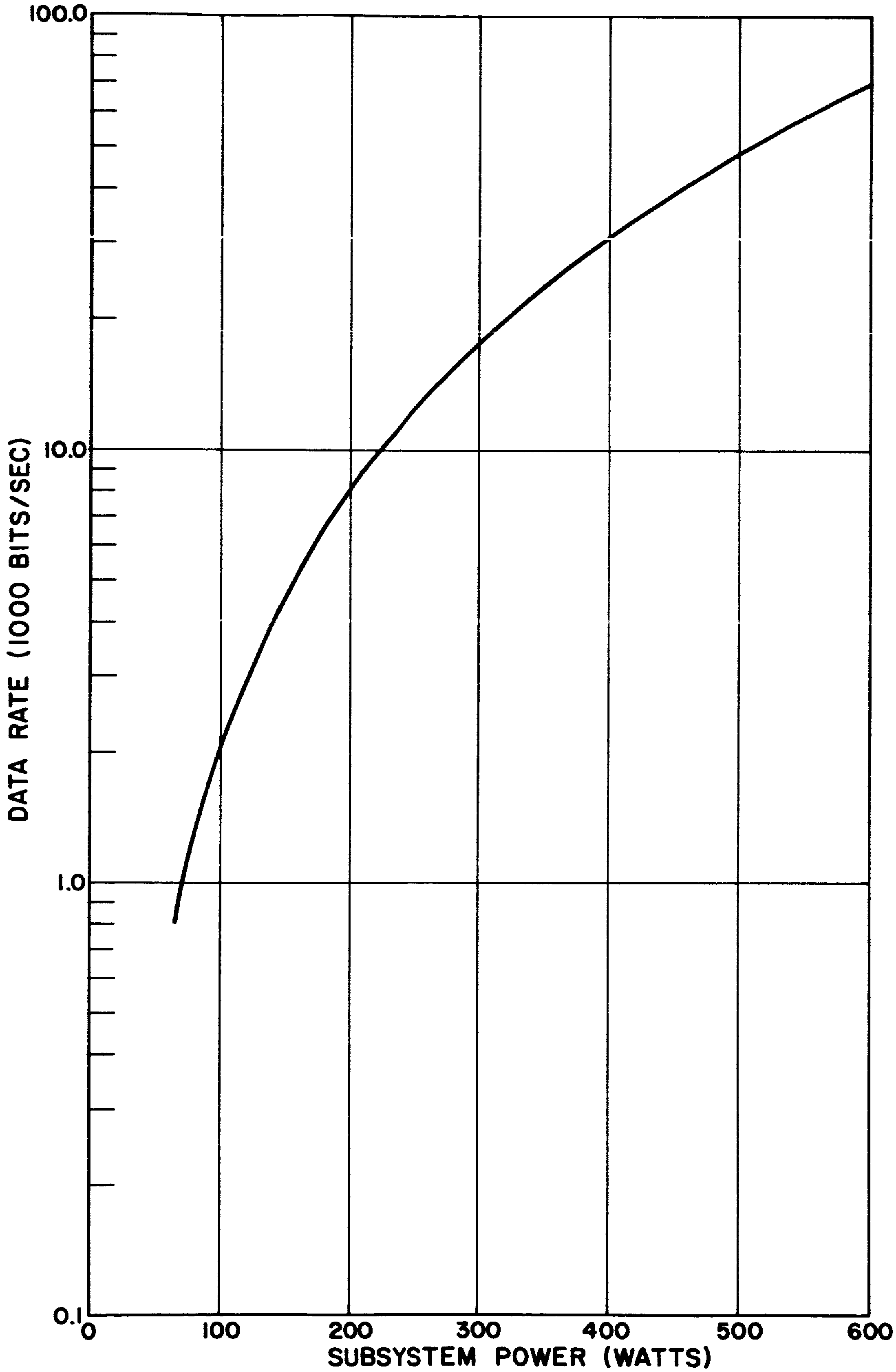


Figure 1. 3-9. Communication Subsystem Data Rate Versus Subsystem Power

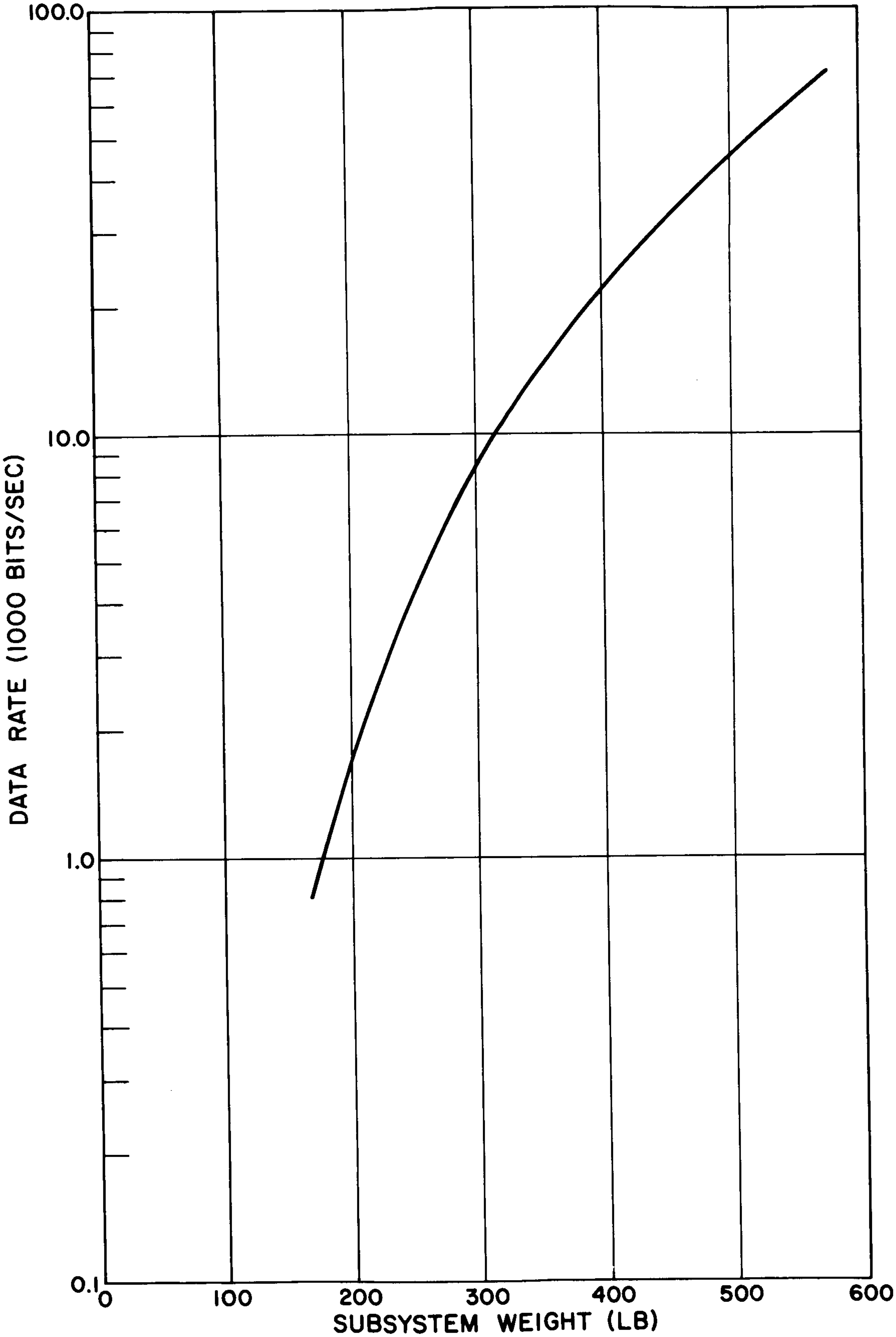


Figure 1.3-10. Communication Subsystem Data Rate Versus Subsystem Weight

determined from the above optimization. The weight and power of the telemetry and command processing portions are engineering estimates based on the type of functions assumed to be required as defined in Section 4.2.

To the power required by the scientific payload and the communication system, a small amount must be added for other functions such as thermal control. While this power is not directly related to payload weight, it can be approximately related since vehicle size, which affects thermal control, and communication complexity vary with payload size. Figure 1.3-11 shows the housekeeping power estimated as a function of payload size. Since this power is quite low, small errors in this curve are insignificant compared to the communication and payload power. Total power required for the payload, communication system, and housekeeping varies from 100 watts to 3000 watts for the various Lander sizes. An RTG was selected to supply power in all Landers for reasons discussed in Section 4.3. An innovation over conventional RTG designs was developed; two types of thermoelectric elements were used in the generator. Germanium silicide thermocouples operate from the generator hot plate temperature to an intermediate temperature while lead telluride thermocouples operate from the intermediate temperature to the cold plate temperature. This allows the two well-proven types of devices to operate at their most efficient temperature range and results in an efficiency of nearly 10 percent, approximately double that of current RTGs. For the large powers being considered, an appreciable reduction in required isotope inventory and cost is achieved. The isotope selected is Curium-244 for reasons explained in Section 4.3. Shielding to protect the payload to a total dose of 10^4 rads was included as part of the power supply weight. The weight of the power system as a function of total Lander power is shown in Figure 1.3-12. While the communication system operates only 10 hours per day, minimum weight and maximum reliability result if the RTG is sized to supply the total power needs while the communication system is on rather than make major use of batteries. Batteries are provided to allow large power drains, such as may be associated with drills, in accordance with a power profile as described in Section 4.3.

A temperature control system has been designed that will provide suitable operating temperatures for the equipment on Mars, and will provide for removal of excess RTG heat during the transit phase. As stated previously, with the Lander aft cover closed during transit, radiation from the RTG is not feasible. During the lift-off period, prior to shroud separation, heat is removed by a water boiler. After shroud separation, an active coolant loop transfers the heat to a radiator external to the Lander for radiation to space. This radiator is used until just prior to entry at which time it is separated to minimize entry weight. Heat removal is again by means of the water boiler until the vehicle is opened on the surface of Mars and the RTGs are deployed to radiate to space.

There is essentially constant heat dissipation within the Lander on the planet surface. When equipments are not operating, the power they normally consume is dissipated in the shunt regulator which provides voltage control. Thus, the basic temperature control on the surface can be achieved by designing for a fixed vehicle emittance

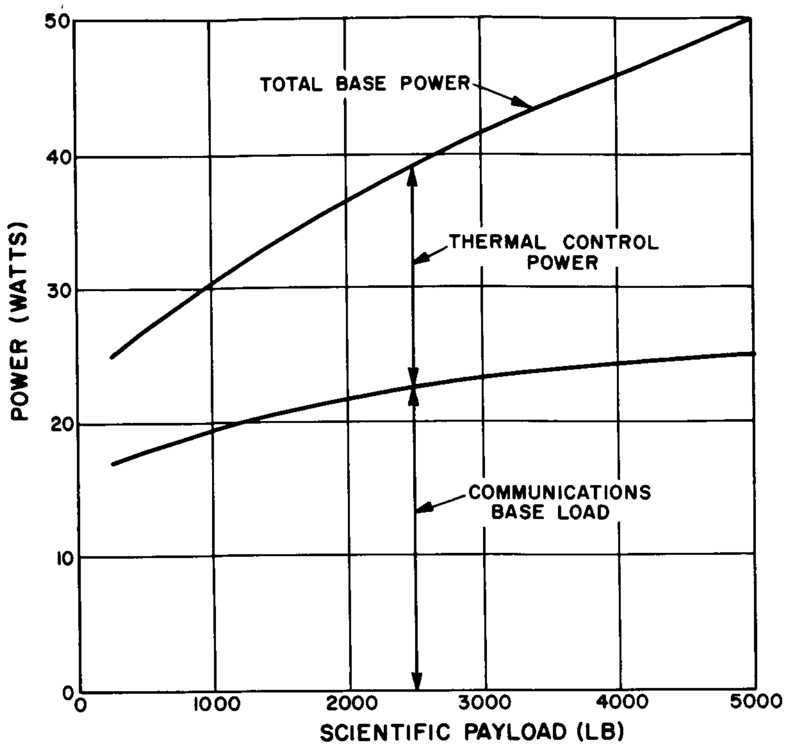


Figure 1.3-11. Thermal Control Power and Communications Base Load Versus Scientific Payload Weight

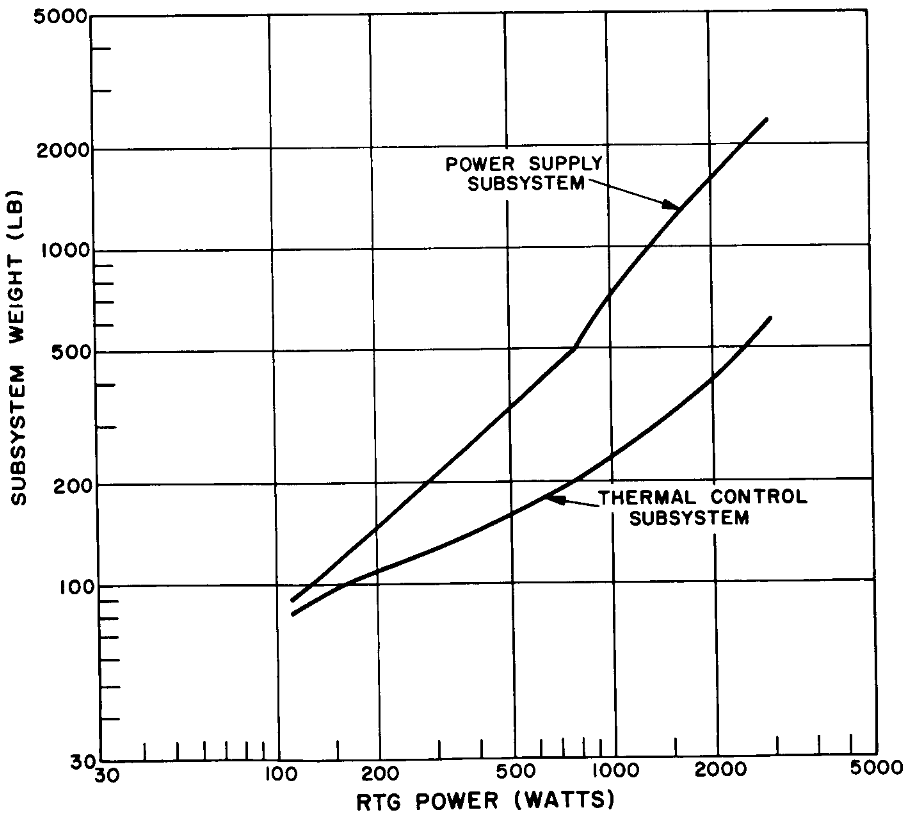


Figure 1.3-12. Power Supply and Thermal Control Subsystem Weights Versus RTG Power

and minimizing the solar heat input during the Martian day by maintaining low solar absorptivity. It has been assumed that some portions of the payload may require rather precise thermal control. Therefore, an active thermal-control loop is provided to transfer heat as required from the RTG to maintain payload temperature.

The weight of the temperature control subsystem is basically a function of the power supplied by the RTG. While some weight elements, such as insulation, are dependent on vehicle size, these are rather small and size is within limits related to total power. Figure 1.3-12 shows the temperature control subsystem weight as a function of total vehicle power. The transit radiator weight is not included as part of the gross payload since this unit is not part of the Lander weight at entry. It is added after the total entry weight has been determined to arrive at the gross Lander weight.

Selection of a scientific payload weight, power, and data rate together with the above curves will yield the gross payload weight. The Lander gross weight required to carry a given gross payload weight can be determined from Figure 1.3-13. The Lander subsystems included in this weight include the structure, heat shield, retardation, impact attenuation, and ground orientation. A breakdown of the weight between the various subsystems is given in Figures 1.3-14 and 1.3-15. The subsystem design approaches are summarized as follows:

1. The heat shield design is based on use of GE Elastomeric Shield Material (ESM) which was determined to be optimum in the Saturn IB Voyager study.

2. The retardation system uses a decelerator chute deployed at Mach 2.5, main parachutes (size and number determined by vehicle weight and deployed sub-sonically), and terminal retrorockets. The system is designed to yield zero impact velocity in an atmosphere with 30 millibar surface pressure. Remaining velocity in lower pressure atmospheres is absorbed by crushable energy absorbing material.

3. Impact studies have indicated that Landers will tumble and roll if adverse combinations of wind velocity and surface slope are encountered. Wind velocities up to 40 mph and surface slopes up to 30 degrees were used in this study. Impact attenuation material is provided on the aft cover of the Lander to absorb the secondary shocks of tumbling. The energy absorbing material used is fiberglass honeycomb.

4. The ground orientation system has been designed so that the aft cover can be opened and the vehicle oriented nose down even if the Lander comes to rest upside down. Stabilization on the surface is by four legs which extend through the heat shield to contact the surface.

5. Radiation shielding is included in the Landers to limit the total dose seen by the payload and the other electronic equipment to 10^4 rods for a 3 year mission. Most of this dose is received during the transit phase prior to deployment of the RTGs. It is recognized that some payload items will require a lower radiation environment for proper operation, and approaches to this problem are discussed in Section 8.3.

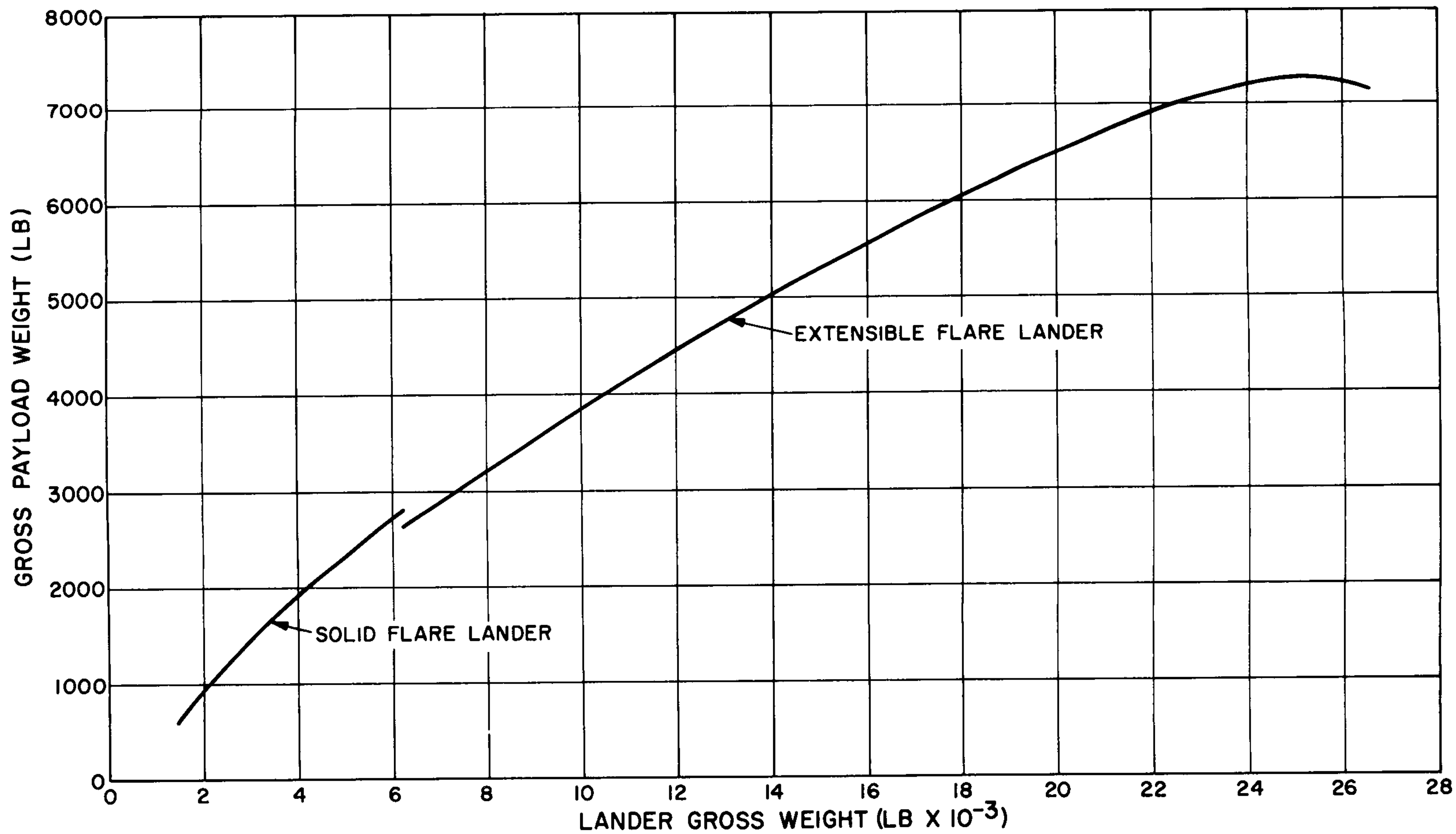


Figure 1.3-13. Gross Payload Weight Versus Lander Gross Weight

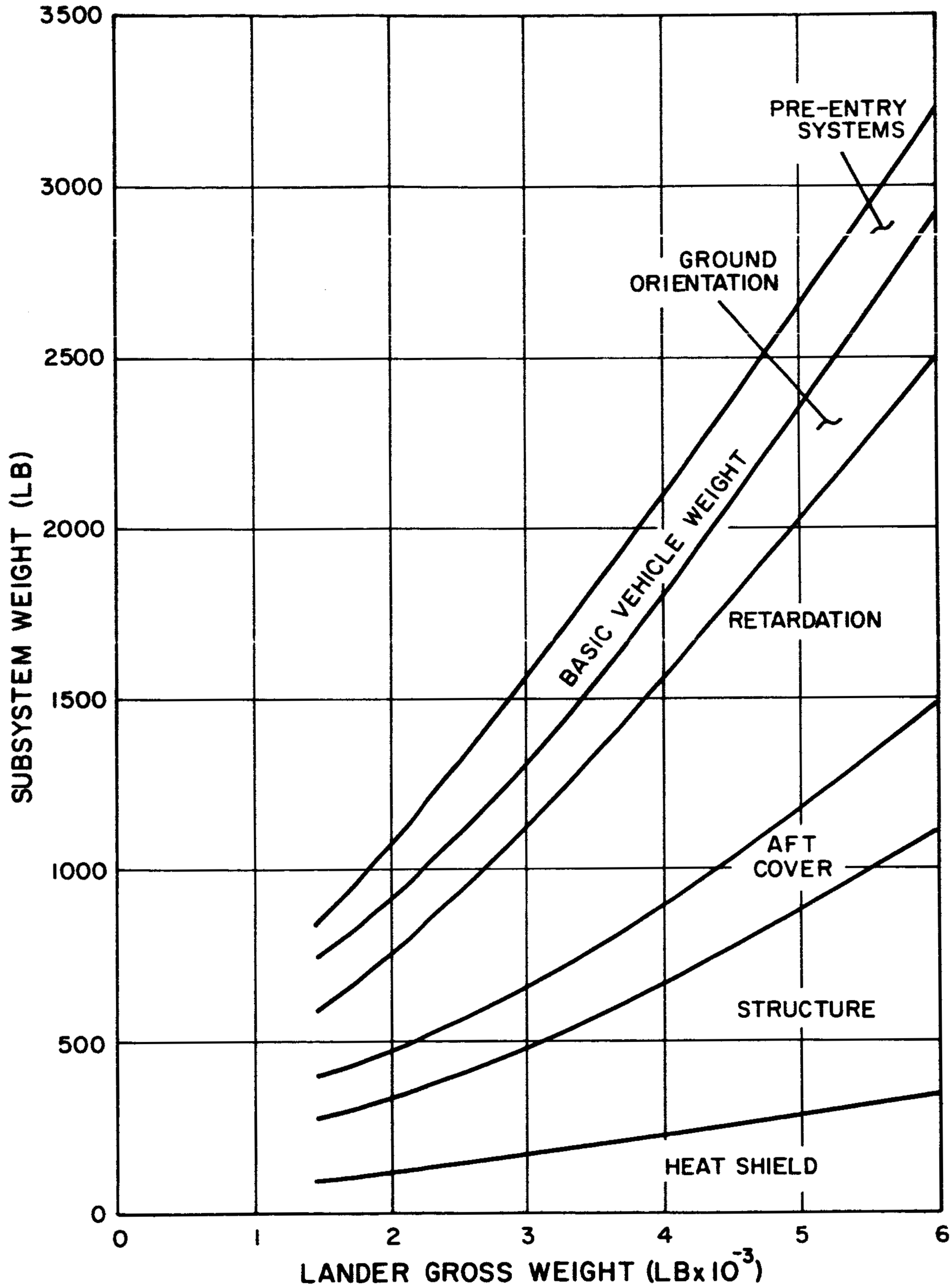


Figure 1.3-14. Lander Subsystem Weights (Solid Flare Lander)

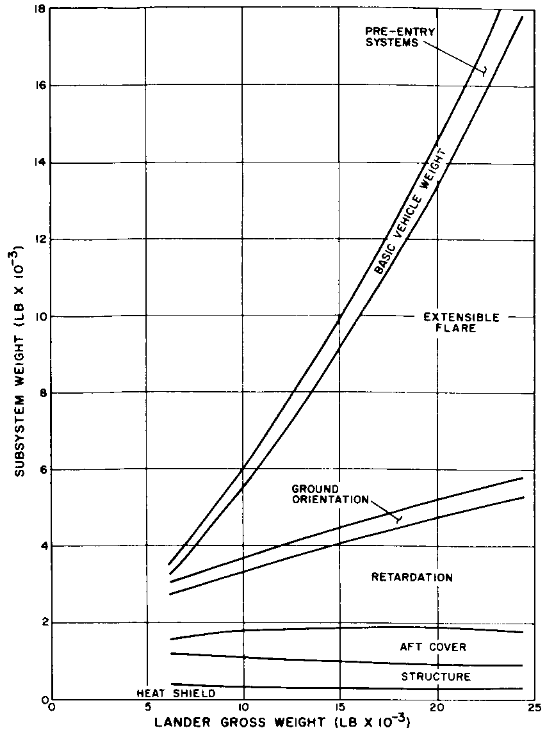


Figure 1.3-15. Lander Subsystem Weights (Extensible Flare Landers)

Given the total entry weight of the Lander, the gross weight is obtained by adding the pre-entry systems. These consist of the solid rocket which changes the Lander trajectory from flyby to impact, the spin and separation system, and the transit radiator of the thermal-control system. The weight of these subsystems as a function of Lander weight is also shown in Figures 1.3-14 and 1.3-15. These weights are based on the following considerations:

1. The specific impulse used for the solid rocket is 230 seconds, and the mass fraction is 90 percent. These values are felt to be consistent with the requirement for sterilization of the solid rocket.

2. The spin up system uses nitrogen as a propellant.

3. The thermal radiator is only approximately related to entry weight since the total power level within a given entry weight can vary somewhat. The size of the radiator is based on an operating temperature of 500°F.

Based on the five scientific payload weights with three levels of power and bit rate associated with each, a total of 45 Landers can be designed. A matrix of the 6 nominal vehicles is presented in Table 1.3-1 with a summary weight breakdown and other pertinent data. The largest vehicles, 5000-pound payload with maximum power

TABLE 1.3-1. LANDER AND PAYLOAD SUMMARY TABULATION

	Unit						
General							
Scientific Payload	lb	250	500	1000	1760	2500	5000
Gross Payload	lb	739	1138	1850	2842	3978	7235
Lander Gross Weight (W_G)	lb	1675	2396	3846	6192	10,400	24,260
Entry Weight (W_E)	lb	1563	2231	3576	5900	9850	22,800
Base Diameter (D_B)	ft	10.30	12.33	15.60	20.00	25.70	39.35
Nose Radius (R_N)	ft	2.42	2.90	3.67	4.70	6.04	9.25
Scientific P/L Nominal Power	watts	60	162	345	576	805	1425
Communication Nominal Data Rate	bps	1100	2000	3700	6000	8000	15,000
Antenna Dish Diameter	ft	4.90	5.70	6.70	7.55	8.10	9.50
Weight Estimates	lb						
Heat Shield	lb	(103)	(143)	(205)	(360)	(352)	(310)
Structure	lb	(207)	(245)	(423)	(806)	(735)	(605)
Shell (Frustrum)	lb	74	108	218	461	383	255
Internal Structure	lb	133	137	205	345	352	350
Aft Cover	lb	(124)	(154)	(218)	(395)	(690)	(900)
Honeycomb Structure	lb	35	51	86	145	175	275
Heat Shield	lb	16	21	40	65	45	48
Aft Crushup	lb	73	82	92	185	470	577
Retardation	lb	(239)	(377)	(618)	(1158)	(1615)	(3230)
Crushup	lb	122	204	353	708	972	2135
Deceleration Chute	lb	26	42	67	110	171	343
Main Chute	lb	39	68	108	202	313	508
Retro	lb	32	43	68	110	131	214
Hardware and Control	lb	20	20	22	28	28	30
Ground Orientation	lb	(152)	(174)	(262)	(339)	(380)	(470)
Stabilization Legs	lb	71	78	137	165	175	210
Strongback & Deployment	lb	53	65	90	120	140	185
Drives, Controls & Hardware	lb	28	31	35	54	65	70
Extensible Flare	lb	-	-	-	-	(2100)	(10,050)
Flare Structure (Incl. Heat Shield)	lb	-	-	-	-	1561	7325
Support Structure & Deployment	lb	-	-	-	-	539	2725
Gross Payload	lb	(739)	(1138)	(1850)	(2842)	(3978)	(7235)
Scientific Payload	lb	(250)	(500)	(1000)	(1760)	(2500)	(5000)
Power Supply	lb	(123)	(206)	(287)	(475)	(642)	(1080)
RTG	lb	40	74	131	195	260	437
Battery	lb	37	62	109	162	217	363
Controls & Hardware	lb	46	70	100	118	165	280
Communication	lb	(268)	(303)	(348)	(400)	(422)	(496)
Electrical Components	lb	151	170	191	218	222	253
Antenna & Mounting	lb	72	88	112	133	148	183
Hardware	lb	45	45	45	49	52	60
Thermal Controls	lb	(98)	(129)	(162)	(207)	(244)	(359)
Electrical Components & Coolant	lb	65	85	108	140	160	235
Redundant Pumps	lb	10	14	15	16	16	20
Heat Sink Wax-Battery	lb	10	11	19	28	38	64
Hardware	lb	13	19	20	23	30	40
RTG Radiation Shielding	lb	-	-	-	-	(170)	(300)
Moments of Inertia (Estimated)							
I_{xx} (Roll)	slug-ft^2	189	474	1382	3271	12,243	101,870
I_{zz} (Yaw)	slug-ft^2	161	318	731	1799	7385	34,635
I_{yy} (Pitch)	slug-ft^2	186	350	817	1861	7675	34,623
Longitudinal Center of Gravity (from Stagnating Point)	ft	2.9	3.4	3.4	4.6	5.8	9.4

or maximum bit rate cannot be designed using the approach taken in this study. Using an extensible flap to maintain a W/C_DA equal to 15 lb/ft^2 becomes extremely inefficient for large gross weight Landers. Eventually, the point is reached where gross weight increases faster than gross payload weight. This can be seen from inspection of Figure 1.3-13. System approaches which allow a higher W/C_DA, and hence higher gross payloads are summarized in Section 1.4, and discussed in more detail in Section 9.

More detailed discussion of the design of all Lander subsystems is presented in Section 5.

Based on the system configurations described in Section 1.3.1, Bus systems to deliver the various size and number of Landers have been designed. The weight of the total Bus system as a function of size and number of Landers is presented in Figure 1.3-16.

While some interpolation is possible using this curve, it is again pointed out that the Lander sizes chosen for packaging within the shroud were those which packaged most efficiently. For Lander gross weights above 6200 pounds, packaging is reasonably straightforward since only flap length is varied as weight changes. For weights less than 6200 pounds, the number of Landers that can be packaged per level varies with weight. Figure 1.3-17 shows fixed flare diameter and flap length versus Lander entry weight and can be used for estimating the number of Landers of an arbitrary size that can be packaged within the shroud.

The Bus weights shown in Figure 1.3-16 are based on the following:

1. In-transit velocity adjustments for both the Midcourse and Individual-cluster/ Buses are made using a monopropellant hydrazine system. Since the injection accuracy of the Saturn V is not known, a conservative capability of 300 ft/sec is provided by the Midcourse Bus. A total of 600 ft/sec is provided by the Individual-cluster/Buses since they must provide time of arrival adjustments and terminal corrections as well as midcourse corrections should the Midcourse Bus fail.

2. Attitude control propulsion is provided by Freon 14. The "redundant" approach used by Mariner C is employed.

3. Bus power is provided in all cases by the RTGs aboard the Landers.

4. While the guidance analysis presented in Section 3.4, coupled with present predictions of DSIF capability, indicate that the 20 to 35 degree entry corridor can probably be met by DSIF alone, a planet line-of-sight sensor is included in the Individual/Cluster Buses. This has been done to provide a comparable system to the previous Voyager studies and to guard against the existing uncertainties in DSIF capability.

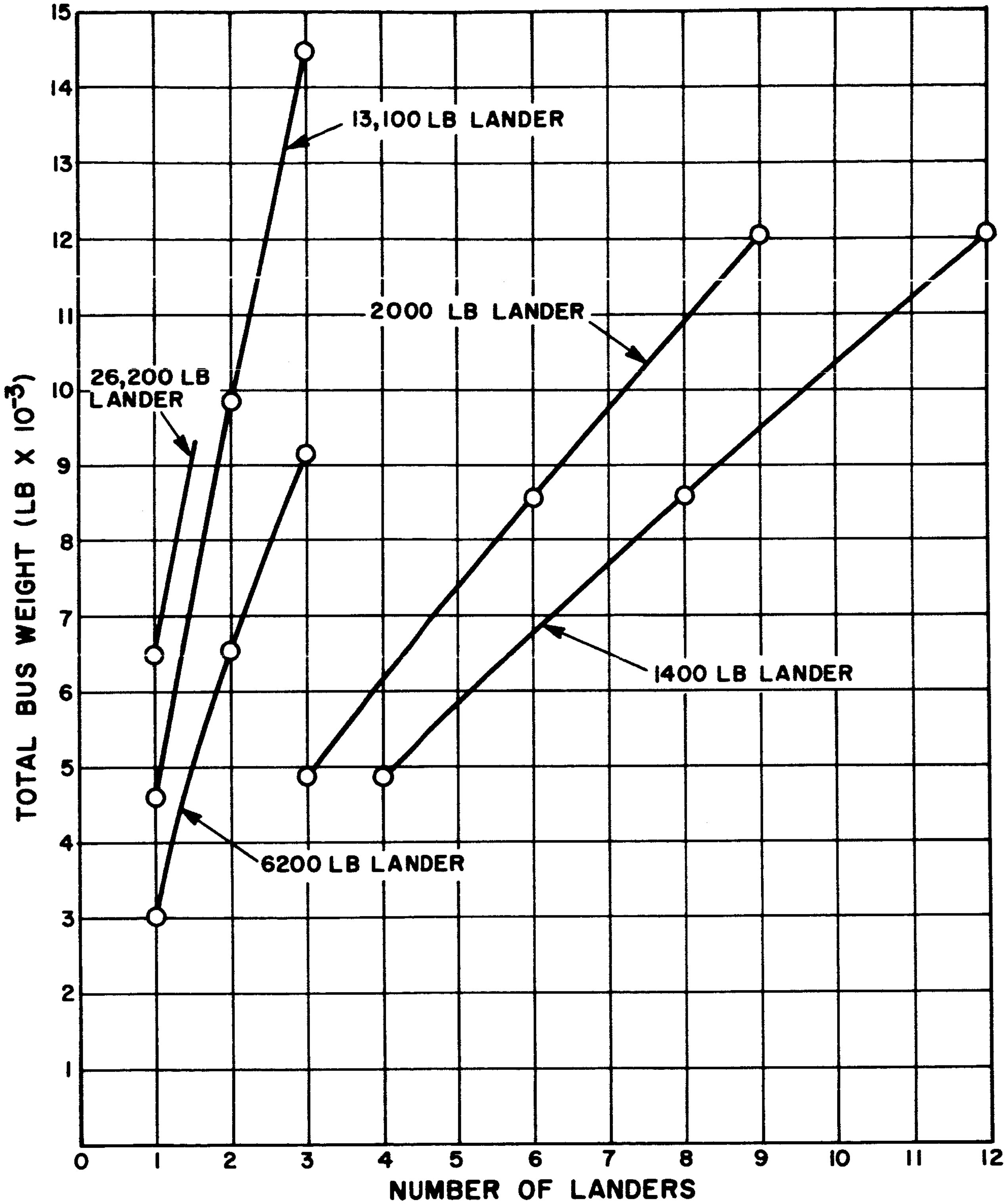


Figure 1.3-16. Bus System Weight

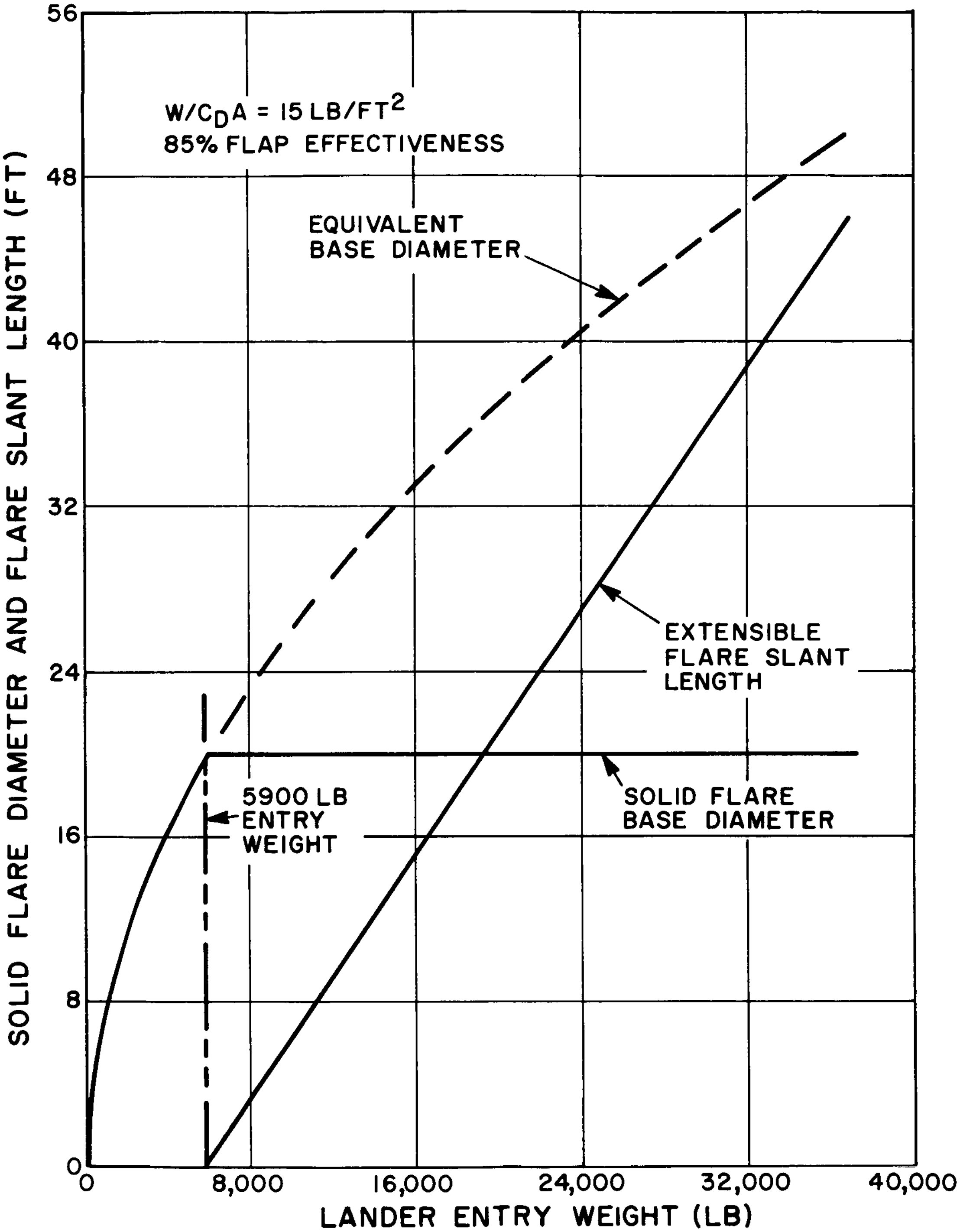


Figure 1.3-17. Lander Base Diameter and Extensible Flare Length

5. The degree of integration between the Bus and Lander communication systems varies with the different systems and is described in Section 6.

6. A high degree of redundancy is used in the Bus subsystems since weight is not critical in this system as long as the weight increase does not require an accompanying volume increase.

Further description of all Bus subsystems is presented in Section 6.

The curves presented thus far will allow for sizing of a total Saturn V payload based on the initial selection of a scientific payload weight, required power, and required bit rate. Having determined the total system weight, the trajectory information presented in Section 3.2 can be used to trade off such things as trip time and desired launch-window duration. For example, Figure 1.3-18 shows trip time as a function of injected weight for a launch window of 30 days for a 1971 type I trajectory.

Considerable latitude is available in this trade-off area since the volume limitation is such that the weight capability of the Saturn V is not approached in most cases.

This flexibility in trip time can be used to good advantage in several ways:

1. Short trips will decrease the probability of failure during the transit phase. The effect of trip time on the probability of successfully landing "n" Landers on Mars and operating for 100 hours is shown in Figure 1.3-19 for one system configuration. The same data for other system configurations is shown in Section 3.3.

2. In some missions it may be desirable to arrive at the planet at a particular time of the year. For instance, it may be desirable to place a Lander just ahead of the wave of darkening to monitor the change in conditions as the wave passes. The variation in time of arrival that is possible for a 1975 mission is indicated by Figure 1.3-20. For reasonable launch-window durations, six months variation in time of arrival is possible using the Saturn V.

3. In many missions it may be desirable to have a high communication bit rate at the time of arrival at the planet to "get the lay of the land," after which a lower bit rate can be tolerated. That is, the amount of information required to monitor change in some parameter of interest is usually less than that required for initial definition of the parameter. The basic communication bit rate has been established for an Earth-planet separation distance of 1.4 AU which corresponds to a typical arrival separation for a Type I minimum energy trajectory. Control of trip time will obviously control the separation distance at time of arrival as well. For example, Figure 1.3-21 shows the improvement obtainable above the basic bit rate as a function of trip time for a 1971 mission. Improvement by a factor of 6 or greater is possible for a 120-day trip and a 30-day launch window which can be achieved by the Saturn V for a total system weight greater than 65,000 pounds in 1971.

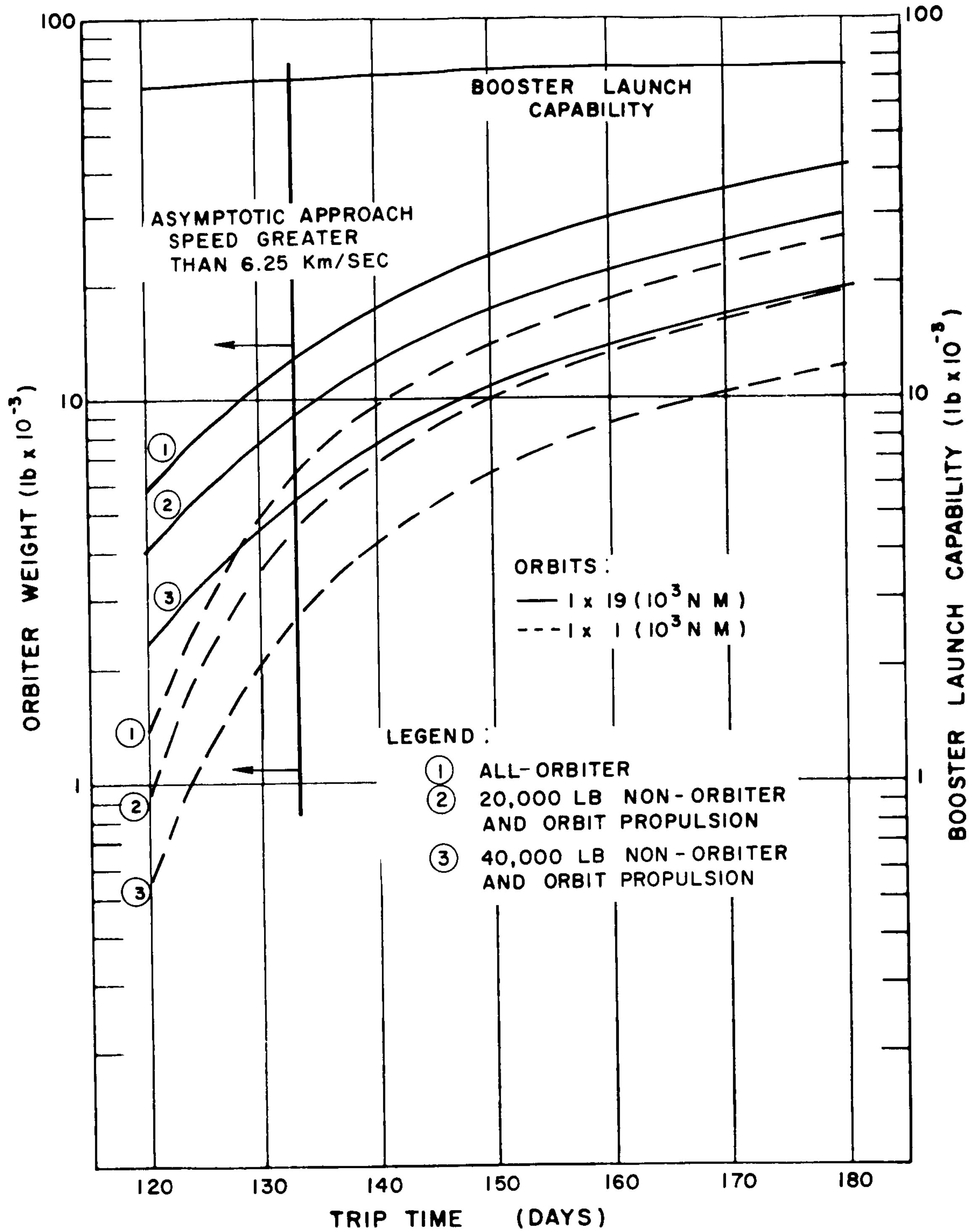


Figure 1.3-18. Mission Weight Capability, 1971 - Type I Trajectory (30-Day Launch Window)

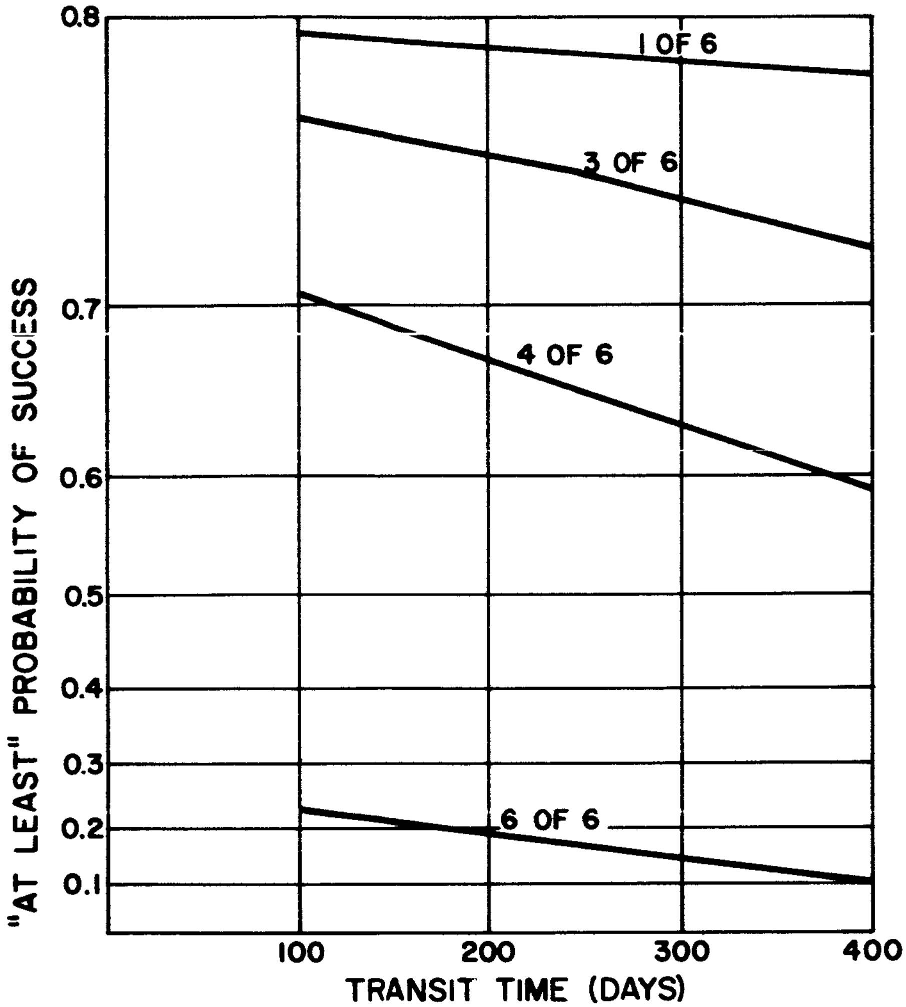


Figure 1.3-19. Probability of Success - 2000-Pound Lander

The specific mission value associated with the control of trip time and launch window duration associated with use of the Saturn V booster can only be identified after the specific mission objective and payload is defined.

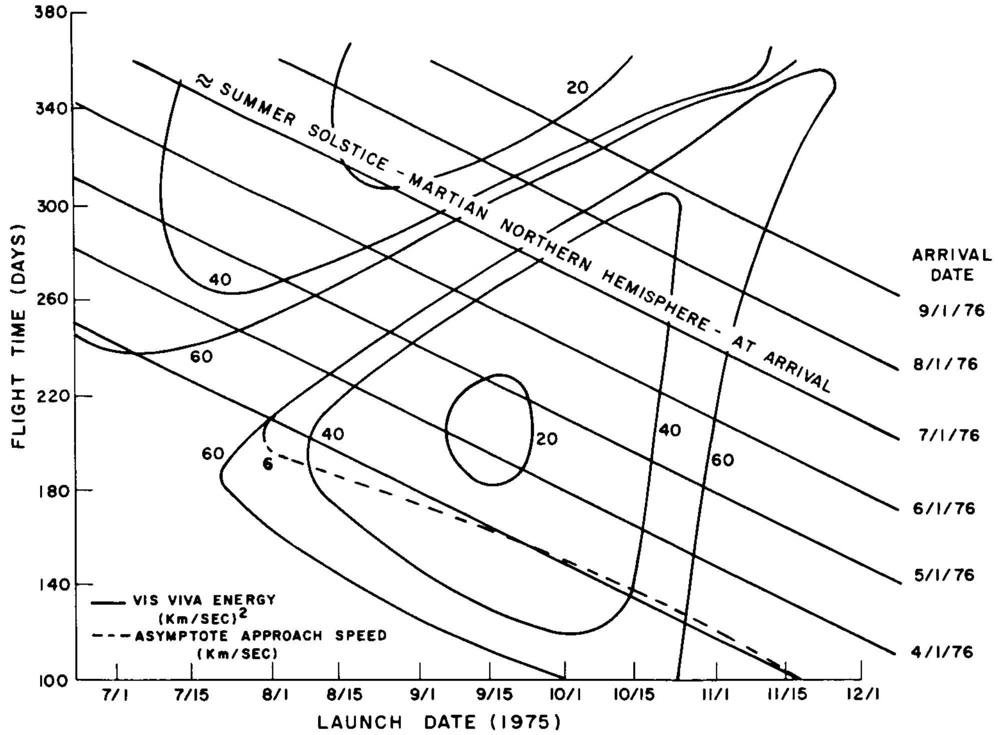


Figure 1.3-20. Constant Arrival 1975 Mars Opportunity

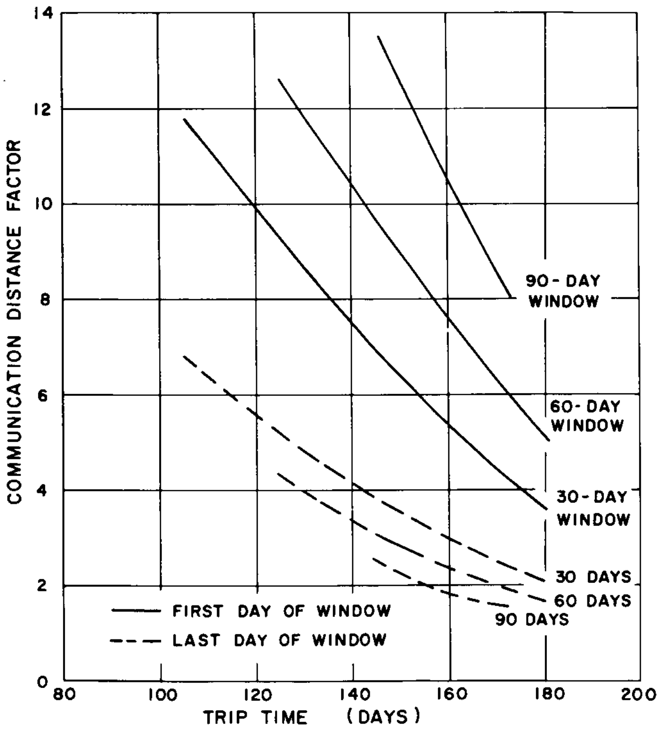


Figure 1.3-21. Communication Distance Factor
1971 - Type I Trajectory

1.4 ALTERNATE SYSTEM APPROACHES

As mentioned previously, it became apparent during the conduct of this study that the entry vehicles designed to carry large gross payload weights are extremely inefficient when the ballistic coefficient, W/C_DA, is limited to 15 lb/ft^2 or less. The reason for this limitation was previously discussed. Several approaches are available which would allow removal of this restriction:

a) An entirely different approach to the retardation system design, such as a Surveyor-type retardation system, could remove the requirement that the vehicle decelerate to a Mach number of 2.5 or less at an altitude of 20,000 feet or greater. Time and funding were not available to pursue this possibility during this study. A rather detailed comparison on the basis of weight, reliability, state of development, and compatibility with the scientific mission is required.

b) Reduction of the entry corridor below the 20 to 35 degrees used in the basic study will allow a higher W/C_DA vehicle to be compatible with the basic retardation system used in the study. The maximum allowable W/C_DA as a function of entry angle is shown in Figure 1.4-1 with atmospheric surface pressure as a parameter. Two approaches to reduction of the entry corridor are: 1) tighter guidance requirements for direct entry, and 2) Lander entry from orbit. Entry from orbit has the added effect of reduced entry velocity. These two approaches have been investigated on a preliminary basis during this study and are discussed in the following sections.

c) Determination that the atmospheric surface pressure is higher than 11 mb will allow the W/C_DA to be increased above 15 lb/ft^2 for the same entry corridor and retardation system used in the basic study. This is indicated in Figure 1.4-1 and is discussed in detail in Section 9.0.

1.4.1 LANDERS OUT OF ORBIT

Ejecting Landers from orbit around Mars rather than direct entry from the approach trajectory was considered during this study and the analysis is presented in Section 9.1. Entry from orbit allows a tighter entry corridor to be achieved with guidance accuracies comparable to those used in the previous Voyager studies. In addition, entry velocities are less - on the order of 15,000 ft/sec compared to 21,000 ft/sec for direct entry. Both of these effects combine to allow higher ballistic coefficients to be used resulting in higher payload weights for a given total Lander weight.

Only the larger Lander sizes were considered, and to limit the extent of the study a constant Lander diameter of 20 feet was used; the maximum fixed flare vehicle allowed by the Saturn V shroud was also considered. For the various opportunities, the total injected weight and the hyperbolic excess velocity at arrival was determined as a function of trip-time and launch-window duration. From the arrival velocity, the ratio of non-propulsive weight to total arrival weight could be determined for the orbit selected - nominally 1000 nautical mile periapsis and a 19,000-nautical mile apoapsis. Assuming two Landers per system, the total adapter weight and Bus weight required to accommodate the arrival weight could be determined. The arrival weight is, of course, the injected weight minus the midcourse propulsion fuel.

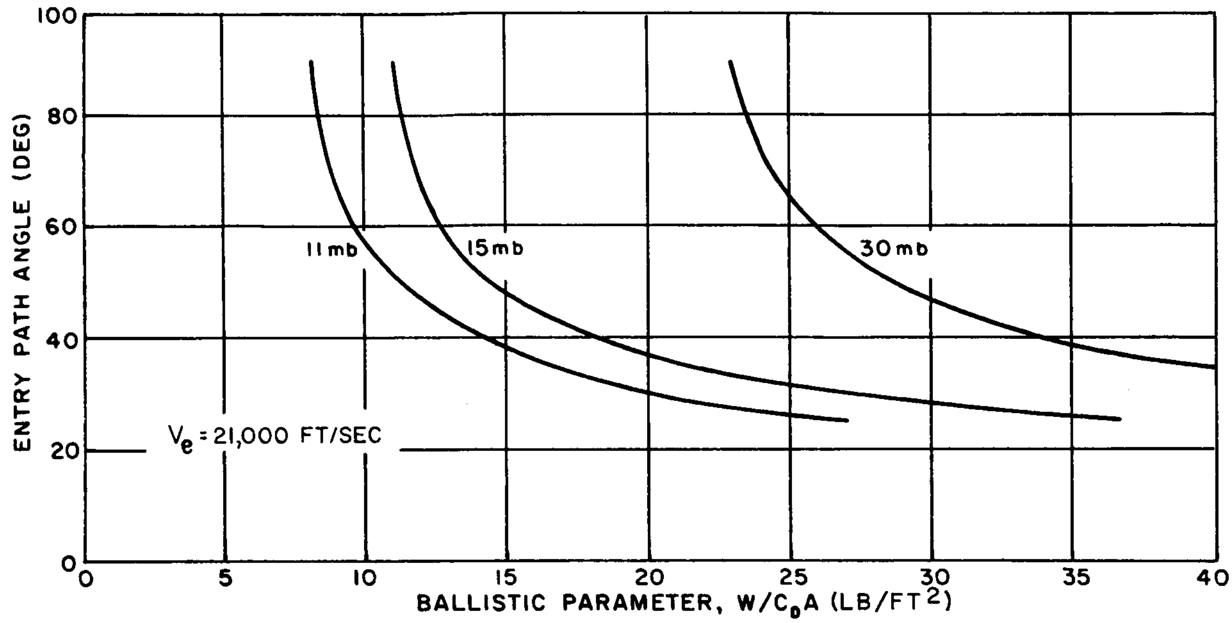


Figure 1.4-1. Allowable Entry Corridor and Atmospheric Limits to Obtain Mach 2.5 at 20,000 Feet

Subtracting from the injected weight the midcourse fuel, the orbit insertion propulsion system, the Bus weight, and the adapter weights, yields the weight allowed for the two Landers. Based on the 20-foot-base diameter, the Lander W/C_DA is then determined.

The results of this analysis are presented in Figures 1.4-2, 1.4-3 and 1.4-4. These figures show the scientific payload achievable per Lander as a function of trip time with launch-window duration as a parameter. It is assumed that the scientific payload requires nominal power and bit rate as defined in the Section 1.3. If other power levels or bit rates are desired, the payload weight in these figures can be adjusted to yield the same gross payload weight. For the 1973 opportunity, for example, a total of 8000 pounds of payload can be placed on Mars with a 175-day trip time and launch-window duration of 30 days. For a Type II trajectory in 1975, 13,000 pounds of payload can be achieved with a 30-day launch window and 320-day trip. Thus, the total payload achievable is increased substantially, in spite of the weight devoted to propulsion, because of the increased efficiency of the Lander design.

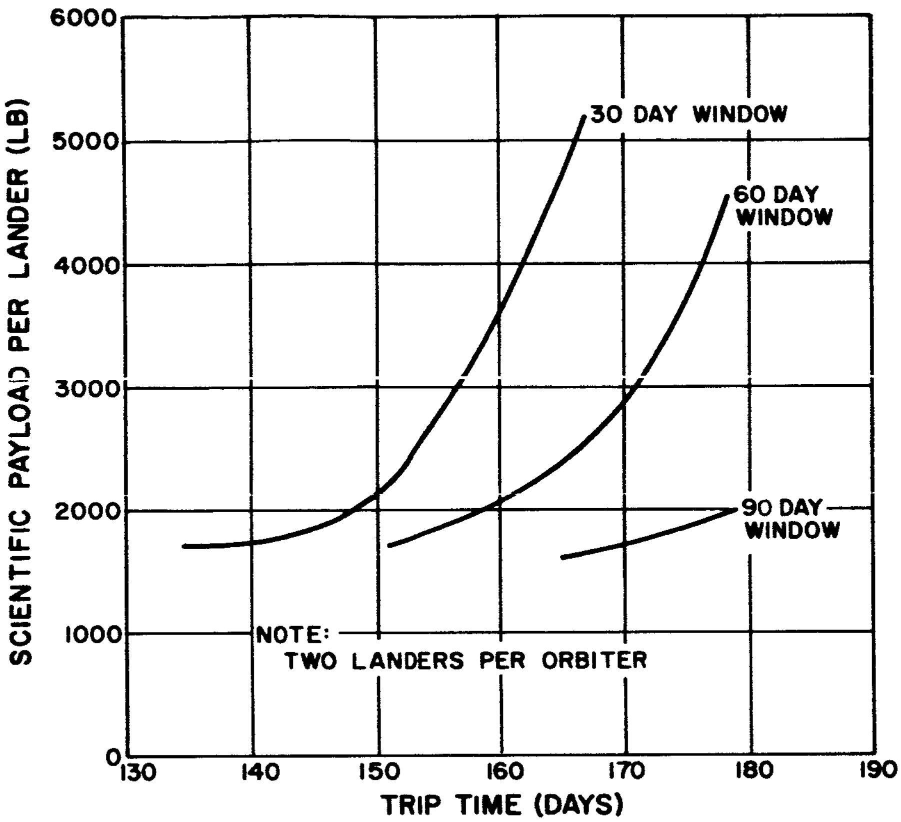


Figure 1.4-2. Landers Out of Orbit, 1971-Type I Trajectory

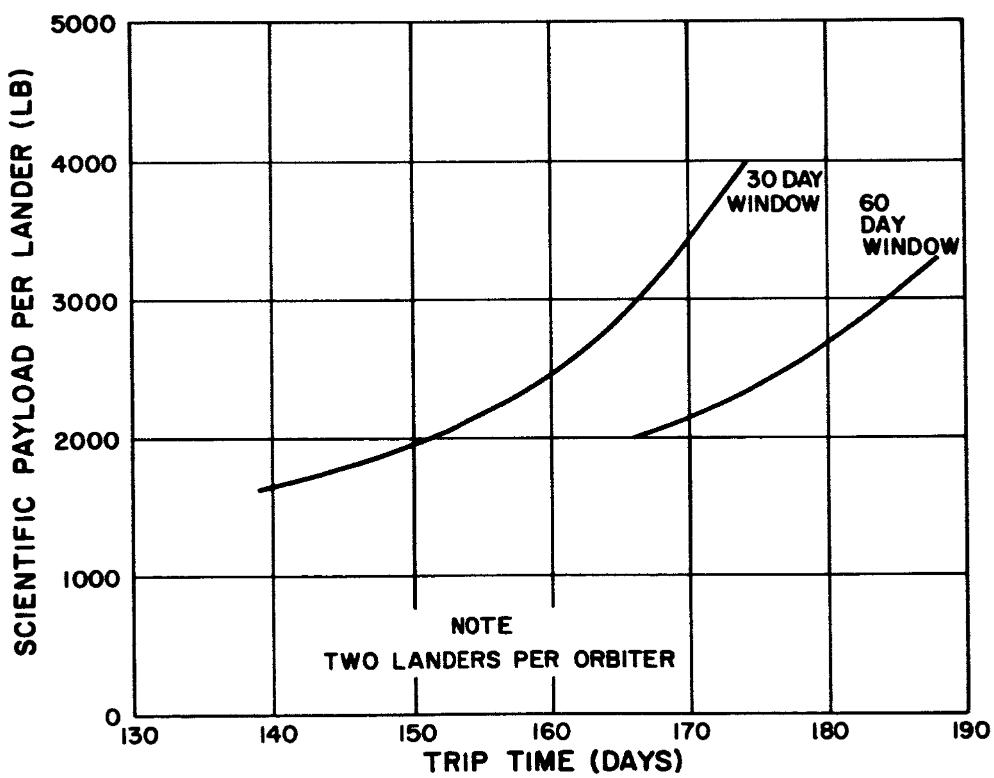


Figure 1.4-3. Landers Out of Orbit, 1973-Type I Trajectory

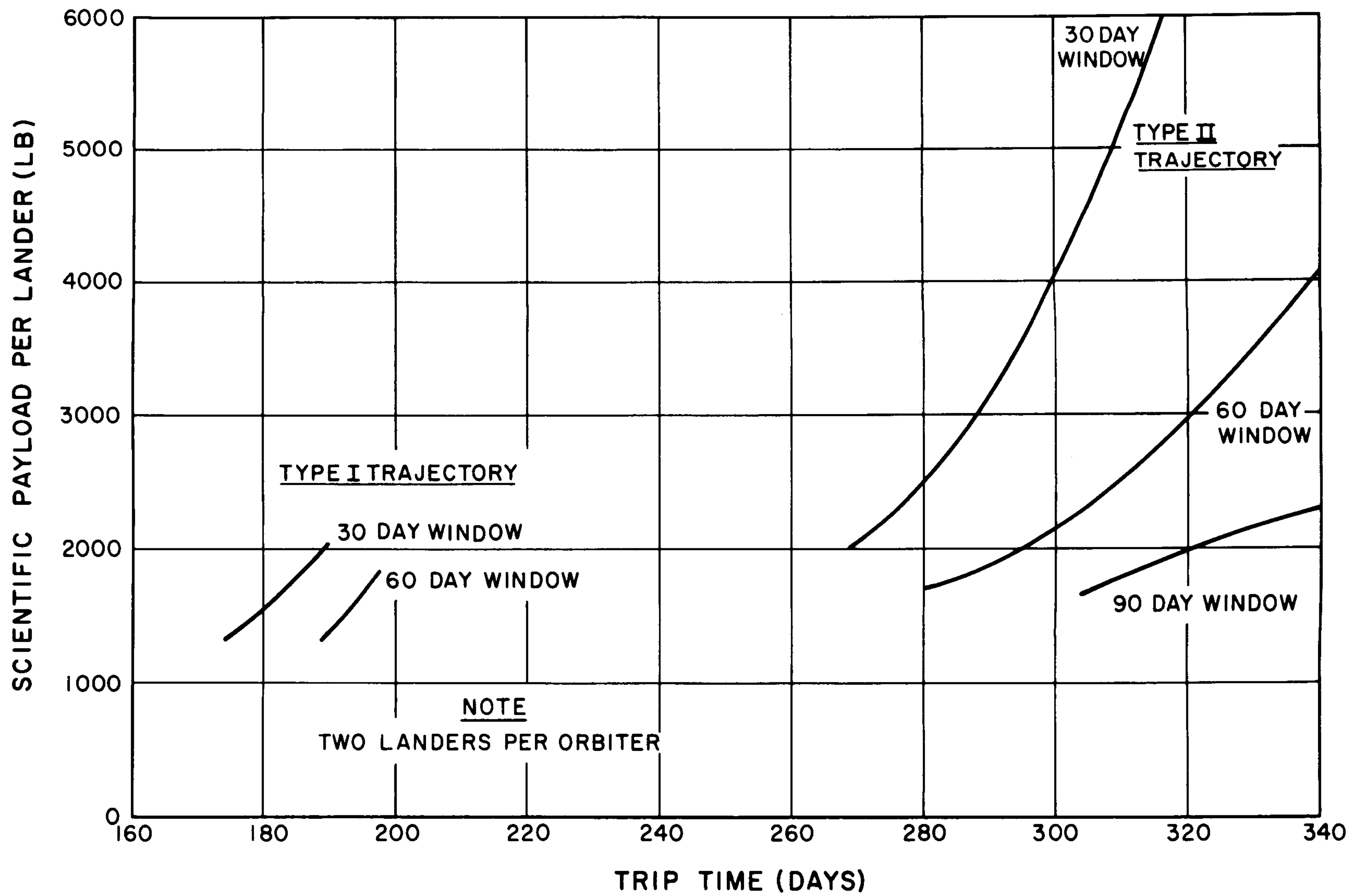


Figure 1.4-4. Landers Out of Orbit, 1975-Types I and II Trajectories

As pointed out, the base diameter was maintained constant with variable W/C_DA for this study. This is not the most efficient approach for small payloads as can be seen from Figure 1.4-5. In this figure, the scientific payload weight, for nominal power level and bit rate, as a function of gross Lander weight is shown for the constant base diameter. It is apparent that for gross Lander sizes less than 14,000 pounds, the rate of increase in scientific payload for a gross Lander weight increase is rather low. This is due to the fact that the W/C_DA is low and the structure and heat shield weight required for the large drag area are excessive. Since the large vehicles are of most interest, this limitation on the results is not felt to be significant.

1.4.2 DIRECT ENTRY-HIGH W/C_DA

A more efficient approach from a weight standpoint than Landers out of orbit would be to improve guidance accuracies such that high W/C_DA vehicles could be used with direct entry from the approach trajectory. The improvement attainable through this approach was investigated and is discussed in Section 9.1.

The scientific payload that can be carried in a 20-foot-base diameter vehicle as a function of maximum entry angle is shown in Figure 1.4-6. Also shown is the maximum W/C_DA that can be used for this entry angle within the limitations of the retardation system. This curve is drawn for a nominal entry velocity of 21,000 ft/sec and assumes the scientific payload requires the nominal power level and bit rate. Again, the scientific payload weight with nominal power and bit rate can be "juggled" to yield any gross payload weight desired.

From Figure 1.4-6, a scientific payload weight of 4750 pounds can be carried with a vehicle W/C_DA of 32 lb/ft^2, requiring an entry angle of less than 23 degrees. The Lander entry weight at this W/C_DA with a 20-foot-base diameter is 12,600 pounds, or a gross Lander weight of approximately 13,150. A total system to carry three such Landers, with Buses for each and a Midcourse Bus, would weigh less than 60,000 pounds, well within the Saturn V capabilities presented in Section 3.2. Thus, the total payload on Mars is roughly 3 times that achieved using a W/C_DA of 15 lb/ft^2.

The guidance analysis presented in Section 3.4 indicates that a maximum entry angle of 23 degrees can be nearly achieved, for example, if the uncertainty of the impact parameter due to DSIF tracking is 50 km (1σ), and a planet line of sight sensor is employed aboard the spacecraft with an accuracy of 0.3 milliradians (1σ).

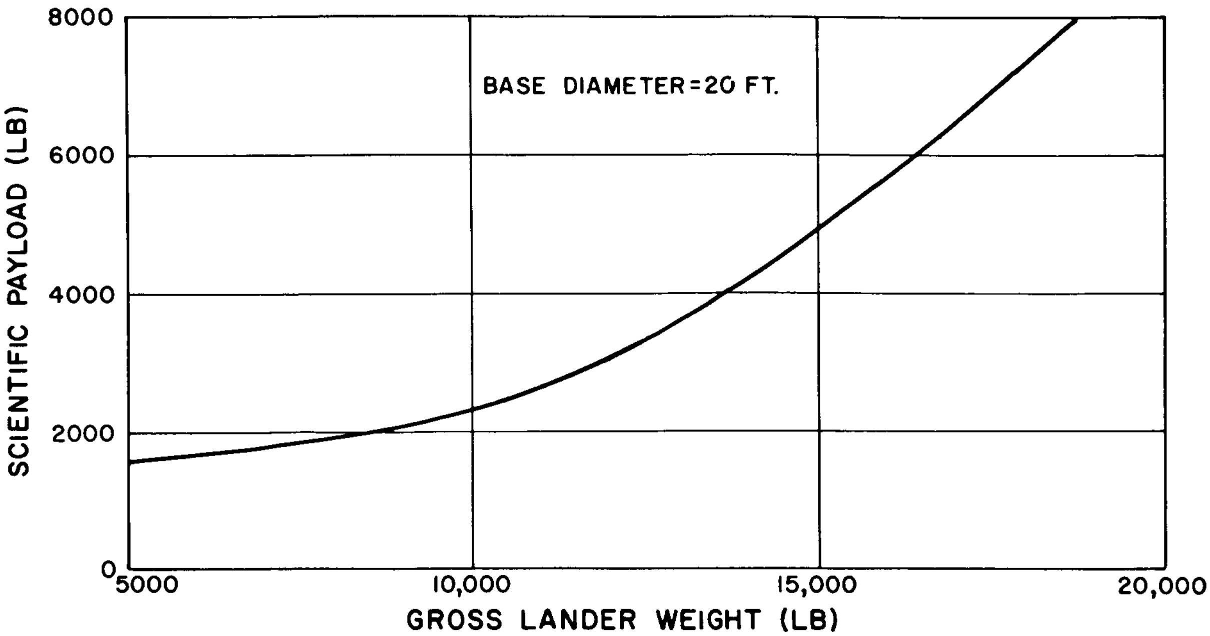


Figure 1.4-5. Gross Lander Weight Versus Scientific Payload Weight

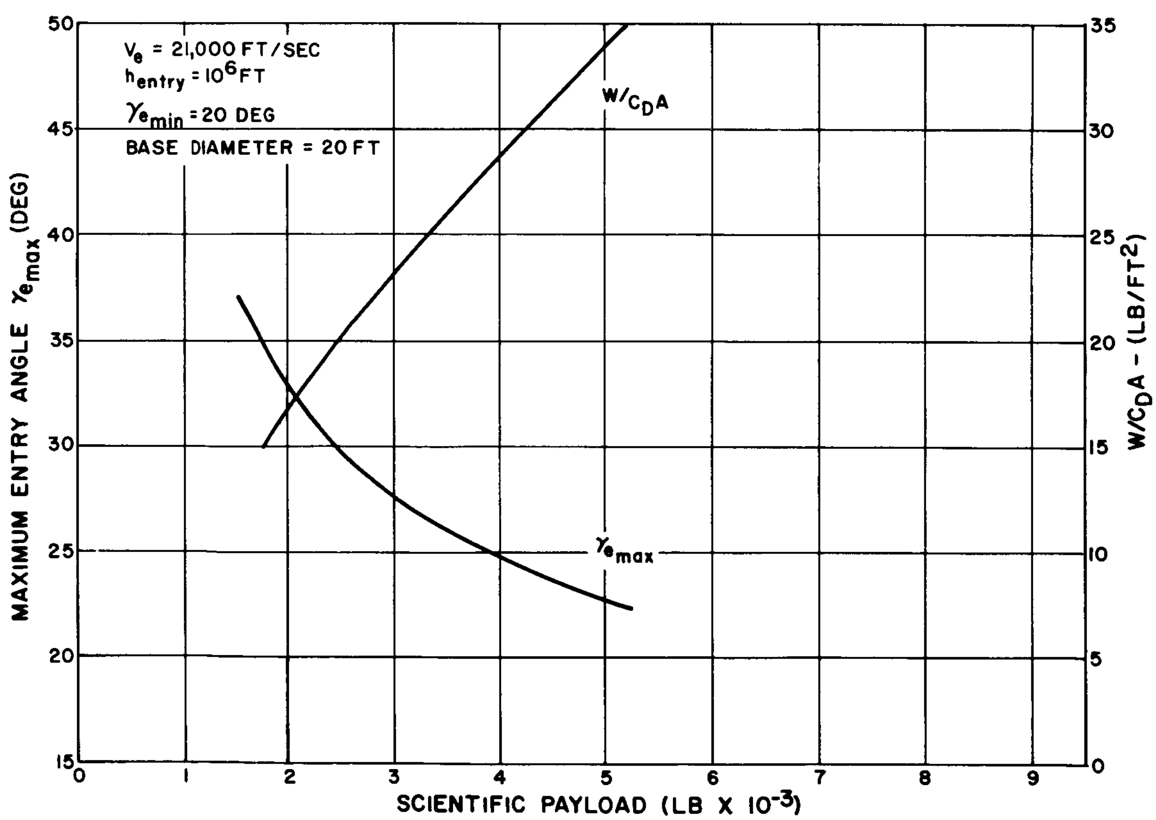


Figure 1.4-6. Scientific Payload Versus Allowable Maximum Path Angle and W/C_DA

1.5 ADDITIONAL CONSIDERATIONS

In addition to the primary parametric study and the consideration of alternate system approaches, several other design areas were investigated in some detail and are reported in this study. These included considerations of sterilization, effects of uncertainties in the Martian atmosphere, approaches to achieving a higher degree of radiation protection for the scientific payload, incorporation of roving capability in the large Landers, and means of including Orbiters within the total Saturn V payload. It has also included alternate approaches in some subsystem areas such as the impact subsystem, and alternate Lander designs suitable for surface winds up to 200 ft/sec. Results of these studies are summarized in the following sections.

1.5.1 ALTERNATE IMPACT ATTENUATION

One of the most challenging technical problems on a Martian Lander is providing a retardation and impact system. This problem is aggravated by the very thin atmosphere which may be encountered on Mars and results in a large percentage of Lander weight being required for retardation. Earlier studies have indicated that a parachute and honeycomb crush up impact attenuation system are very inefficient and would be impractical for very large vehicles. Therefore, terminal retrorockets have been employed in the past and on the prime parametric study to reduce the parachute and impact attenuation weight required. This led to the requirement for an accurate sensing and firing system with their attendant tolerances and reliability problems.

This study undertook to identify a more passive impact attenuation system which might be combined with a parachute system to obtain a more reliable approach. Several concepts were considered, but most of the effort was expended on the analysis of a "blowout" type pneumatic bag, reported in detail in Section 9.2.1. The analysis resulted in a design and weight estimate for impact velocities up to 200 ft/sec for a heavy Lander in the 5000 pound scientific payload class. Figure 1.5-1 presents results of the analysis which indicate the blowout-bag system is significantly lighter than a retrorocket/honeycomb crush-up combination and without the problems attendant to retrorocket systems. In addition, the possibility of contamination of the planet surface by the retrorocket is eliminated.

1.5.2 EFFECT OF BETTER DEFINITION OF THE MARTIAN ATMOSPHERE

The current lack of the exact characteristics of the Martian atmosphere imposes a severe strain on the design of a Lander vehicle. While the vehicle can be designed to operate in the complete range now postulated, it is not the most efficient vehicle possible for any atmosphere. To indicate the potential gain if specific characteristics were available, three Lander systems were synthesized to operate specifically in the 11 mb, 15 mb and 30 mb atmospheres, with the same entry angle tolerances used in the basic study.

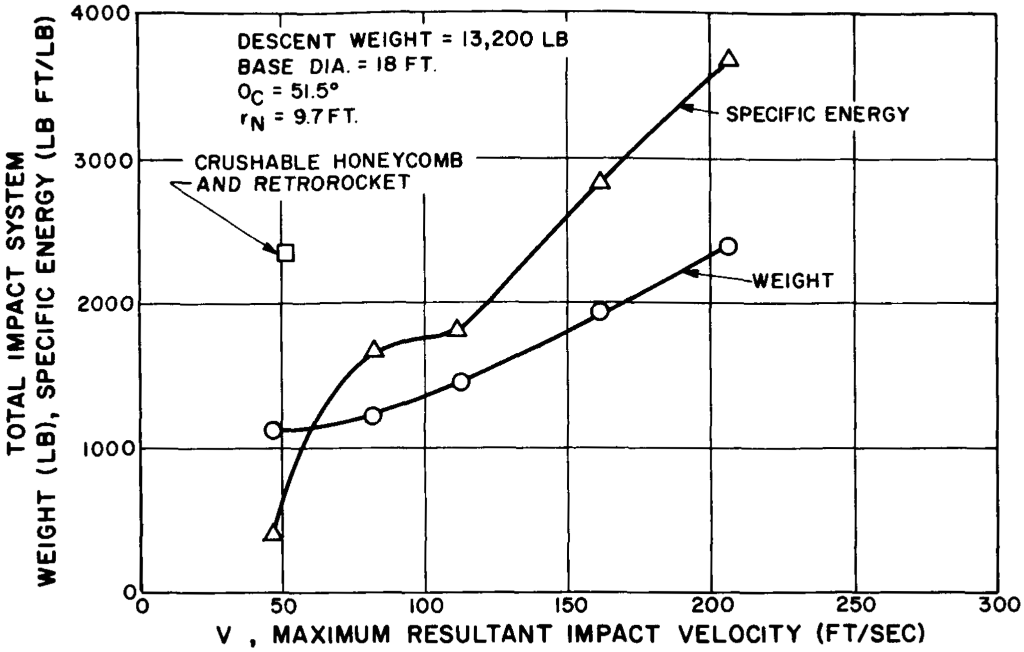


Figure 1.5-1. Pneumatic Bag Characteristics

Significant gains are possible in the retardation system where the retrorocket can now be sized to yield a nominal zero impact velocity, thus removing impact attenuation material which is much less efficient in removing terminal velocity. The retro firing sensing system is also simplified and made more reliable since only altitude need be sensed rather than altitude and velocity. The largest gains result however, if an atmosphere above the 11 mb minimum is to be identified. In this case the W/C_DA could be increased and structure area and weight decreased. The conceptual study identified Landers to carry a 5000 pound scientific payload with nominal subsystems and to operate in each of the three identified atmospheres. Results are shown in Table 1.5-1, the prime system is designed to operate in the full range and is shown for comparison.

TABLE 1.5-1. ATMOSPHERIC COMPARISON OF A 28,000-POUND LANDER

Atmosphere (mb)	11-30	11 (only)	15 (only)	30 (only)
Base Diameter Equivalent (in.)	39.4	35.4	29.1	18.4
W/C_DA (lb/ft^2)	15	15	19.5	36
Entry Weight (lb)	22,800	18,400	16,100	11,900
Gross Weight (lb)	24,260	19,570	17,080	12,460
Scientific Payload/Entry Weight (%)	21.9	27.2	31.1	44.8
Gross Payload/Entry Weight (%)	32.2	40.0	45.6	61.8

These results are plotted in Figure 1.5-2 versus atmospheric surface pressure (although this cannout be considered a strictly linear parameter). In addition to the reduction in weight, a volume reduction is obtained particularly where W/C_DA is significantly changed; this may lead to more Landers being carried within the Saturn V shroud.

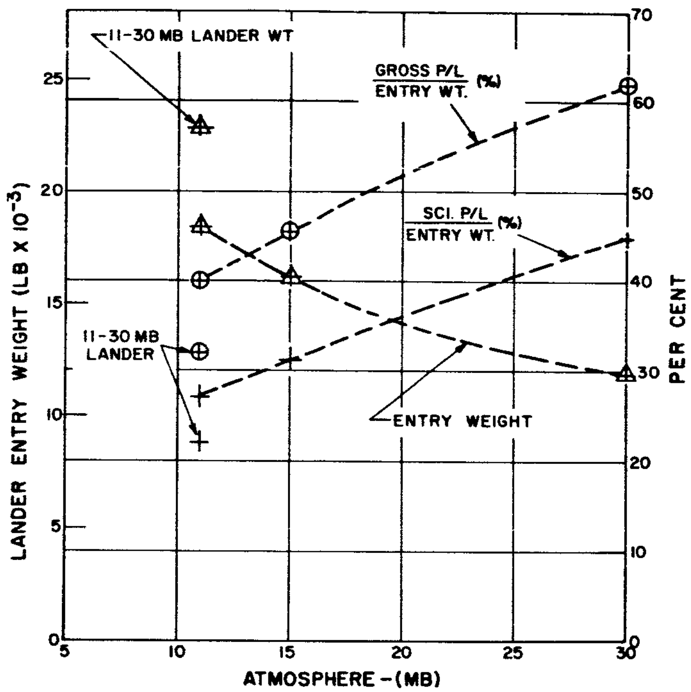


Figure 1.5-2. Effect of Atmosphere on Lander Entry Weight
(5000-Pound Payload at Nominal Conditions)

It is clear that identification of the specific characteristics of the Martian atmosphere would be of major benefit in designing a vehicle of the type considered in this study; therefore, all possible efforts should be made to obtain this data.

1.5.3 DESIGN FOR A 200 FT/SEC WIND

Identification of the possibility that a wind as high as 200 ft/sec might be encountered at or near the surface has led to interest in a vehicle designed to survive this condition. A conceptual study was made to select a possible approach and is described in detail in Section 9.2. The pneumatic impact bag, Section 1.5.1, was used as a basis for the approach selected. It is proposed that the heat shield be dropped after entry and flexible bags, which would completely envelop the landing vehicle, be inflated to protect the vehicle from shock and the inevitable tumbling resulting from impact at a high lateral velocity. While this approach may involve a slightly higher element of risk for the landing event, it must be recognized that a 200 ft/sec wind represents a very adverse landing condition, particularly for an unmanned vehicle. While it is possible to utilize sensors to obtain direction and velocity of drift and negate this with lateral propulsion, the control system sensors and propulsion necessary to implement this approach seem too complex for an unmanned Voyager system.

The study resulted in a Lander weighing 22,800 pounds at entry and capable of carrying 3450 pounds of scientific payload and nominal subsystems. While this vehicle has not been given as much analysis as the prime parametric study Landers, this is a realistic first concept of a vehicle designed to meet this adverse wind condition. It is not felt that the use of a rigid impact attenuation system (i.e., fiberglass honeycomb) is realistic because of the long stroke required for the high velocity involved.

1.5.4 ROVER STUDIES

Since one of the most desirable payloads for a 5000-pound scientific payload class Lander has been a surface roving vehicle, this study considered several approaches to meet this requirement (See Section 8.2). Of these, the three types listed below were evaluated by design layout and preliminary weight estimates.

1. Air-drop Rover in which the entry vehicle shield is dropped after entry and the Rover vehicle lands on its own running gear.

2. Separate Rover which is contained within the cocoon of the Lander and emerges only after impact, stabilization and opening of the aft cover.

3. Integrated Rover which is completely unitized with the Lander and obtains mobility by extending running gear after impact and stabilization.

On the basis of preliminary analysis, the Integrated Rover was selected as the most desirable approach since it avoids the problems and sequence of operations associated with separation from the Lander either in the air or after impact. This should reduce complexity and enhance reliability. An additional factor is that Lander structure need not be duplicated by Rover structure so lower integrated weight should result.

1.5.5 INCLUSION OF ORBITERS

Consideration was given to means of including Orbiters in the overall Saturn V payload and is discussed in Section 8.1. A relatively easy way of achieving this is to add the necessary additional equipment to the Midcourse Bus in the system configuration where the Midcourse Bus is achieved. The additions required are electrical power, orbit injection propulsion, terminal guidance sensor (if required), added stabilization fuel, some added communication equipment, and the scientific payload. The dimensions of the Bus are such that body mounted solar cells could likely provide sufficient power. At worst, small deployable paddles would be required. Sufficient volume is available within the shroud for incorporation of the orbit insertion propulsion system.

For the large Lander which does not use a Midcourse Bus, the Individual Bus could be modified to serve as an Orbiter as well. The additional communication system would be somewhat higher in this case since the Bus communication is more highly integrated with the Lander.

1.5.6 STERILIZATION

The sterilization problem was considered to determine if there is a significant perturbation due to the large size of the Saturn V Voyager system. This is reported in Section 7. Primary conclusions are:

1. The large Landers with correspondingly large surface area may require some extension of the presently recommended thermal sterilization time of 24 hours. This is due to the increase in probable number of viable organisms existing at the start of thermal sterilization.

2. The small Lander designs fit within the range of vehicle sizes being considered in current NASA studies. They would probably be individually sealed in a rigid sterilization barrier, subjected to thermal sterilization, and then assembled to the Bus.

3. The large flapped Landers might use a plastic film type of sterilization barrier and to avoid handling problems sterilization at the launch pad should be considered.

1.5.7 INCREASED RADIATION PROTECTION

Recognizing that some payload items may require extremely low radiation levels during operation, an investigation of radiation levels achievable through a combination of separation from and shielding of the RTG power source was conducted. Achievement of radiation levels as low as those received from JPL and listed in Table 4.1-1 is not practicable using RTGs of the power level considered within the limits of reasonable deployment distances for either the RTG or the scientific instrument. This investigation is reported in more detail in Section 8.2.

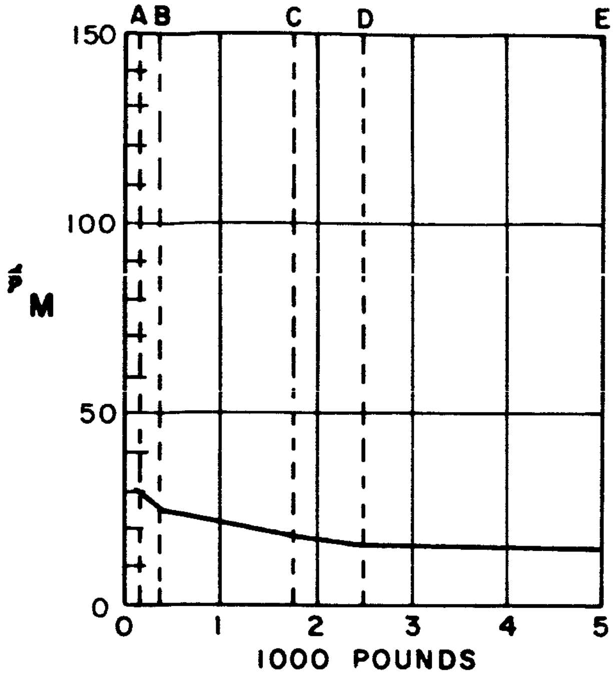
1.6 COSTS AND SCHEDULE

Costs of a Voyager program based on the Saturn V launch vehicle have been estimated and are described in Section 12. The costs through one opportunity for the various system configurations are shown in Figure 1.6-1 based on the following major ground rules.

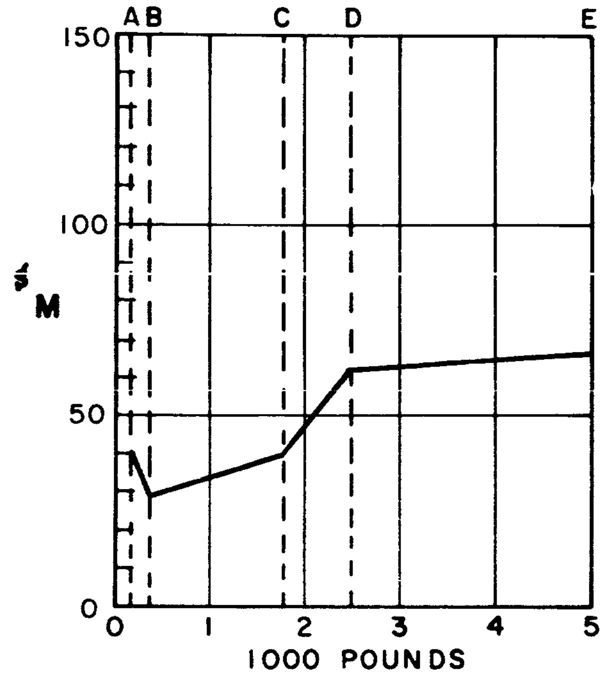
1. 1971 opportunity
2. Two launch vehicles plus spares
3. Scientific payload costs are excluded
4. Costs include RTG units and isotope fuel
5. No launch vehicle or shroud costs are included

More detailed description of the ground rules and basis for the cost estimates is given in Section 12.

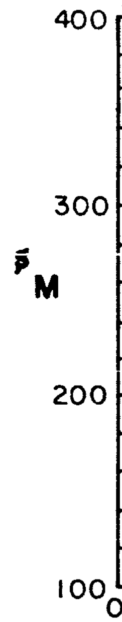
The estimated schedule for development and delivery of the Saturn V Voyager system is presented in Section 11. The required length of the program is slightly in excess of 5 years.

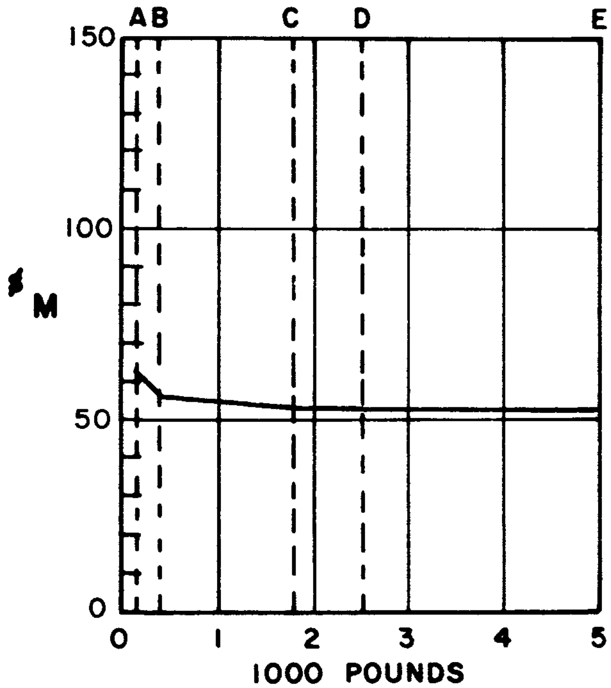


RELIABILITY

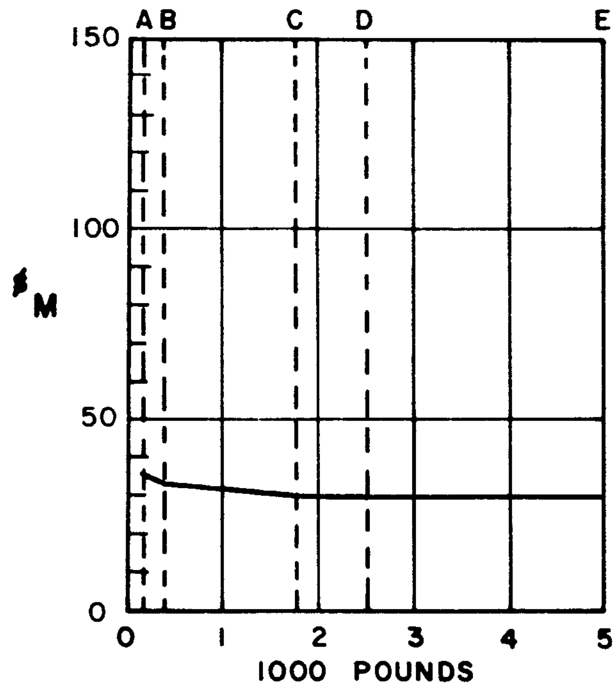


SYSTEM TEST & C/O

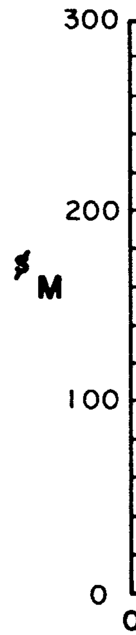




SUPPORT EQUIPMENT

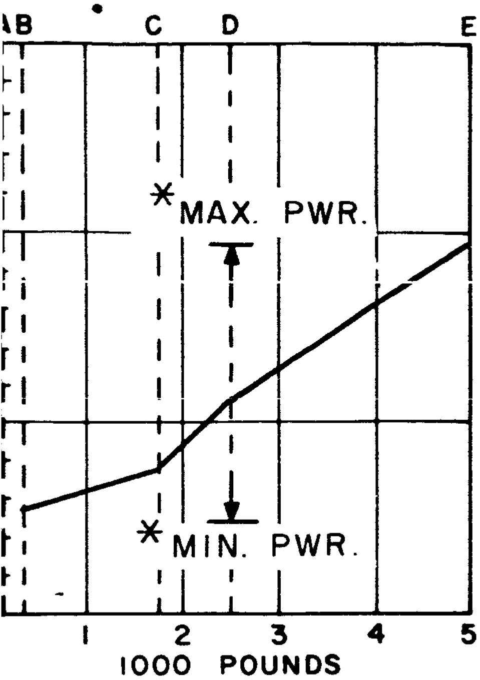


PROGRAM MANAGEMENT

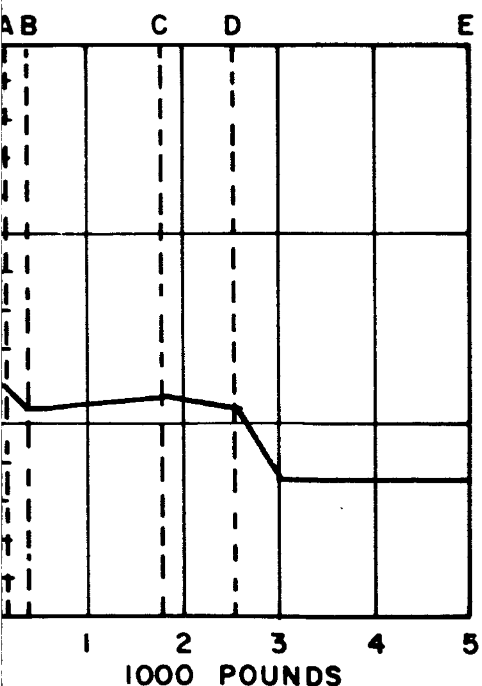


ALL COST PLOTS ARE COSTS IN MILLIONS OF DO
VERSUS SCIENTIFIC PAYLOAD WEIGHT PER LAN
IN THOUSANDS OF POUNDS

LETTERS DESIGNATE FIVE SPACECRAFT PAYLOAD



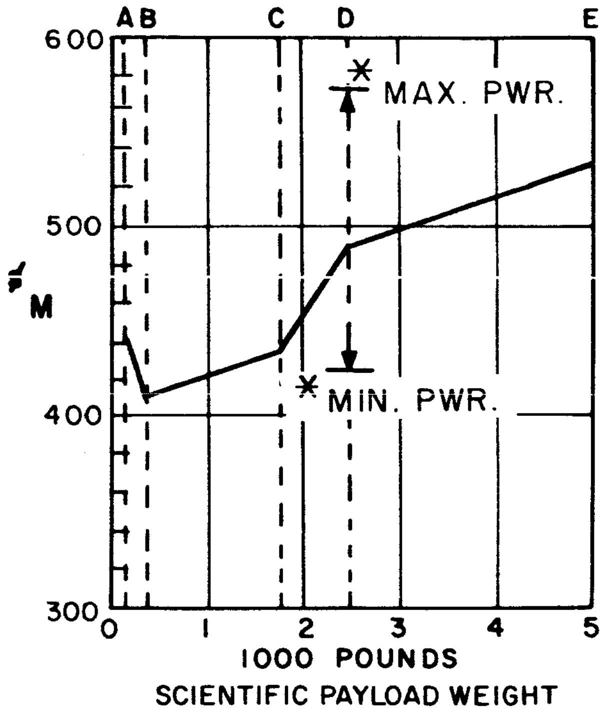
TOTAL LANDER COSTS



TOTAL BUS COSTS

LLARS
DER

WEIGHT OPTIONS



TOTAL SPACECRAFT PROGRAM COST

THE TOTAL COSTS SHOWN ABOVE ARE SUBDIVIDED BY MAJOR COST CATEGORIES IN THE PLOTS SHOWN TO THE LEFT

TOTAL LANDER AND BUS COSTS SHOWN TO THE LEFT ARE SUBDIVIDED IN FOLLOWING FIGURES INTO NON-REPETITIVE AND REPETITIVE COSTS BY SUBSYSTEMS.

* SHOWS THE EFFECT OF VARIATION IN POWER REQ'TS ON COST FOR 2500 POUND PAYLOAD SIZE

Figure 1.6-1. Total Spacecraft Program Costs

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2. CONCLUSIONS AND RECOMMENDATIONS

This study has shown in a parametric fashion how the Saturn V launch vehicle can be used for conducting a Voyager mission. Statements regarding the effectiveness of this launch vehicle compared to the two previous vehicles studies, i.e., Saturn IB/SVI and Titan IIIC, cannot be made in the absence of more specific definition of the scientific mission. Certainly, if very large integrated scientific payloads are identified, in excess of 1000 pounds, a clear requirement exists for use of a launch vehicle with Saturn V capability. On the other hand, if the scientific payload is divisible into smaller units, but large numbers of total Landers are required, a cost effectiveness comparison between single Landers launched by the smaller boosters and multiple Landers launched by Saturn V is required before a choice can be made.

The following are more specific conclusions and recommendations resulting from this study. Some are related specifically to the use of Saturn V for performing a Voyager mission, while others are applicable irrespective of the launch vehicle employed.

2.1 CONCLUSIONS

1. The restriction of the Lander ballistic coefficient, W/C_DA, to 15 lb/ft^2 or less results in poor utilization of the Saturn V weight capability. Using a reasonable shroud length, the volume limitation is such that about half of the weight that the Saturn V can inject into a nominal minimum energy trajectory can be packaged.

2. The maximum gross payload that can be packaged in a Lander of W/C_DA equal to 15 is about 7300 pounds using extensible flaps to achieve the required drag area. As vehicle size increases above this point, the weight of the extensible flaps increases faster than the allowable total weight for constant W/C_DA, forcing a reduction in the weight of the gross payload.

3. Using a W/C_DA of 15, the maximum gross payload that can be landed on Mars with a single Saturn V is approximately 9600 pounds.

4. Design of a Lander for a specific Martian atmosphere results in significant weight reductions as compared to designing for the range of atmospheres currently defined.

5. The guidance analysis conducted,together with current predictions of DSIF capability, indicate that an entry corridor of less than 15 degrees, the value used in this study, should be easily achievable. This will allow an increase in Lander W/C_DA. The exact value of the entry corridor that is achievable, however, is unknown.

6. Taking Landers into orbit and subsequently sending them in to Mars shows an improvement in scientific payload landed on Mars by a factor of 1.5 to 2.5, depending upon the year, trip time, and launch window,

compared to direct entry using $W/C_DA = 15$.

7. Reduction of the entry corridor from 15 degrees to 3 degrees results in an increase of scientific payload by a factor of 3 or better for direct entry Landers. The W/C_DA allowable for maximum entry angle of 23 degrees is 32 lb./ft^2 based on the same atmosphere and retardation system.

8. The large total energy available from the Saturn V booster can be used to provide considerable flexibility in the conduct of a Voyager mission. Wide launch windows and short trip times can be achieved. Control of these parameters yields reasonable control over the Martian season at planet encounter if arrival at a particular season is desired. Short trips can be used to improve reliability and to achieve shorter communication distances at planet encounter. The value of this flexibility can be assessed in detail only when the specific scientific mission is defined.

9. Use of the largest vehicles considered in this study may dictate some change in the presently specified requirements for sterilization.

10. Use of radioisotope thermoelectric generators in a Lander imposes two serious interface problems. The first is a radiation level problem associated with sensitive scientific instruments. The second is a thermal control problem during ground operations and during transit. This thermal problem is further complicated by the sterilization interface, requiring heat removal during thermal sterilization and compatibility of the sterilization barrier with heat removal during transit.

11. The alternate impact system studied, blow-out bags rather than the fiber glass honeycomb and terminal retrorocket combination, shows good promise. Preliminary design data shows it to be a lighter approach and it is a more passive system that does not require the accurate timing and controls required for the retrorockets. It is possible to attain the long strokes needed for high impact velocities and yet retain minimum packaging volume.

12. The blow-out bag concept also provides an approach to the design of a Lander capable of landing safely in a 200 ft/sec wind. This concept uses an impact bag attached on all sides to form a spherical shape that can withstand the shock of impact from any direction.

13. Facilities do not currently exist to simulate flight vibration levels for a total Saturn V Voyager system. Similarly, thermal vacuum facilities do not exist to test an assembled system. Thermal vacuum facilities that should be adequate are under construction by NASA at Houston and at Lewis Laboratories. To our knowledge, adequate vibration facilities are not currently under construction.

2.2 RECOMMENDATIONS

1. Design of Landers using low W/C_DA has been a severe limitation in all the Voyager studies to date. (In the Saturn IB study the problem was less severe since the minimum surface pressure during most of that study was taken to be 41 millibars.) The problem becomes much more severe for very large vehicles, but even in the Titan IIIC study the use of low W/C_DA Landers caused recommendations of bulbous shrouds or the use of extendible flaps. Since the allowable Lander W/C_DA has such a major influence on the total system, the following series of studies is recommended to define more precisely the W/C_DA that should be used in future work:

   a. It is obvious that any steps that can be taken to reduce the uncertainty in the Martian atmosphere will be extremely helpful. Particularly, if the minimum atmosphere can be determined to be less severe than that presently specified, improvement will result.

   b. A detailed analysis of the entry corridor achievable with direct entry should be conducted. It is very likely that the 15 degree entry corridor currently used is too conservative, but the value that should be used is unknown. The analysis should utilize the actual covariance matrix of guidance uncertainties achievable based on Earth-based tracking and computation. Addition of spacecraft planet line of sight information would be factored in, considering both random and bias errors in such measurements, to determine the improvement achievable in entry corridor as a function of sensor type and complexity. Results of this analysis should yield a more realistic mutually acceptable entry corridor tolerance to be used in future studies.

   c. Since the allowable W/C_DA is a function of the retardation system design as well as the atmosphere and the entry corridor, consideration of the approaches available is in order. A detailed comparison is required between the Surveyor-type retardation system and the parachute/crush-up system as a function of Martian atmosphere, entry corridor achievable, and vehicle size. The comparison would include weight, reliability, development status, compatibility with the scientific mission, and cost. The blow-out bag method of impact attenuation would be considered as well as the fiber glass honeycomb/terminal retrorocket system. Also, the effect of deploying the decelerator chute at higher Mach numbers would be considered.

   d. A more detailed comparison of carrying Landers into orbit versus direct entry should be conducted. This comparison would consider a range of Martian atmospheres and entry corridors achievable to determine the crossover point at which more payload can be landed on Mars by carrying Landers into orbit. A comparison of mission reliability would also be made.

2. For long life missions on the surface of Mars, RTGs appear to be the only feasible power source available in the near future. More detailed study of the two major interface problems mentioned earlier is needed. The degree of compatibility between the RTG and selected scientific instruments from a radiation level standpoint should be investigated in detail. Approaches achieving compatibility would be investigated, and a limit to what is feasible would be defined.

   Investigation of the interface problem between the RTG, thermal control system, and the sterilization requirement also requires considerable study. Results of this study would be a handling procedure for RTGs compatible with thermal and sterilization requirements and design guidelines for the Lander, and a sterilization barrier compatible with the recommended procedure.

3. The large Landers considered in this study need further investigation from the sterilization standpoint. The sterilization requirements being used in current NASA studies should perhaps be modified to cover changes likely to be required by vehicles of this size. The definition of required facilities and the general approach to sterilization of vehicles of this size is not well defined, nor is it being covered in current NASA studies.

4. Requirements for RTG power should be made known to the AEC early so that design of generators can be initiated, and availability of isotope fuel can be effected.

5. Close attention should be paid to the design effort being conducted under the Automated Biological Laboratory Contract. As the payload definition begins to evolve, vehicle design inputs should be factored in to their study. Similarly, as results become available from that study, iterations on the vehicle designs described in this report should be made to determine whether major changes in the general approach are required. More specific system designs would eventually result that are compatible with payloads defined in the Automated Biological Laboratory study.

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Application of the Saturn V launch vehicle to the unmanned exploration of Mars using a Voyager-type space craft was studied. To generate design information in parametric form, five scienfific payloads weights were selected at 250, 500, 1000, 2500, and 5000 pounds. and their communication subsystem, electric power subsystem, and thermal subsystem were defined. The basic Lander vehicle was analyzed for its structure, heat shield, retardation, impacy attenuation, delta impulse rocket, separation and spin up, and ground orientation. Bus designs for delivering of the Landers to Mars were deigned prepared for five Lander sizes that package reasonable well within the shroud