NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Memorandum No. 33-228

Systems Comparison of Direct and Relay Link Data Return Modes for Advanced Planetary Missions

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PREFACE

Justification of this simultaneous mission approach is based, in part, on an assumed communications advantage when using the orbiter to relay data from the lander to Earth. The total data accrual capability of this approach is thought necessary to support the many high-data-rate experiments proposed for surface exploration of Mars.

Philip Eckman of the Advanced Missions Staff provided the initial and continued drive for this study. Without his help, direction, and support, this study would have been impossible. Discussions with Lloyd Nalaboff of the Telecommunications Division have added much to the preparation of Section 3. Also, special mention goes to Thomas Hamilton, Systems Analysis Section Manager, who helped this report over some rough spots.

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ABSTRACT

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This report analyzes advanced planetary missions using a *Saturn IB* and simultaneous orbiting and landing spacecraft. An analysis of the relative data return capability of two alternate modes is given. One mode (named the relay link) uses transmission of the lander produced data to a planetary relay communications satellite, then transmission to the Earth; the other (named the direct link) uses transmission directly from the lander to the Earth. Numerical results are given for sample advanced planetary missions to Mars.

1. INTRODUCTION

Eventually, an advanced planetary program will try highly sophisticated experiments to find out the esoteric characteristics of the Martian surface. These experiments can be described by their support requirements and, for this study specifically, their data gathering potential. A basic premise of this study is that there will be requirements for the collection of much data (perhaps 10^9 to 10^{10} bits) over a relatively long time (6 mo). Once this premise is set, it is necessary to study the feasibility of systems that might be used to return these data to Earth.

Two such systems proposed are the direct link (where the data is transmitted from the surface of Mars directly to Earth) and the relay link (where the data is sent from the landed capsule to an orbiting spacecraft and then to Earth).

To compare these two system approaches, this study uses three standards: (1) a measure of the effective data-rate capability of each system, (2) a measure of the probability that each system will survive to perform its intended mission, and (3) a combination of items (1) and (2) that gives a measure of the average amount of data to be expected from each system approach. The first standard, effective data-rate capability, demands a knowledge of the bit rate at which each system can operate and the time available for communications at that bit rate. The second standard, success probability, calls for postulating a mission sequence with the functions and hardware needed to successfully perform the mission sequence. Evaluation of these standards calls for geometric analyses of lander–orbiter and lander–Earth view times and ranges, comparison of potential designs of telecommunications systems to bring about transmission, and reliability analyses of the representative systems and subsystems to carry out the stated objectives.

1.1 Outline of Problem

1.1.1 Basis of Comparison

The following paragraphs outline the problem in terms of the mathematical parameters chosen for the system comparisons.

1.1.1.1 Instantaneous bit rate. The instantaneous bit rate is defined as the actual mechanical (or electrical) bit rate at which the system operates when communications are being carried out. For this study, instantaneous bit rate is thought to be dependent on two basic factors: (1) the geometric characteristic, range; and (2) the performance of the telecommunications systems that support a bit rate at a given range. The gain needed from the telecommunications system is expressed as a function of range and desired bit rate in Section 3, and the range characteristics for the direct and relay link are derived in Section 4.

There are bit rates characteristic of each of the two systems, one bit rate for the direct link, and two bit rates for the relay system corresponding to the lander–orbiter and orbiter–Earth legs of the link. It may be desirable to choose several levels of operation for the Mars-to-Earth leg of either link because of the change in the Mars–Earth range over the mission life.

Instantaneous bit rate is not a valid comparison parameter for a mechanization that includes storage capability because the instantaneous bit rate does not directly relate to the overall long-term system cumulative information capability. It is used only as a stepping stone to more representative comparison functions.

1.1.1.2 Effective bit rate. Because neither the direct link nor the relay link system can transmit data when occulted by Mars, the instantaneous bit rate that the system is capable of supporting is not enough to describe the long-term total bit capability of either system. To determine the total bits received at Earth, the instantaneous bit rate must be multiplied by the time that communications are being carried out. If this total number of bits is regarded as the product of some average bit rate times the total time elapsed, then this average might be termed the effective bit rate. Therefore, the effective bit rate is defined as the rate at which a hypothetical system would have to continuously broadcast to accumulate the same data as the real system.

If storage is provided to keep the data being collected during the times when there is no visibility, the communications link may be operated at full instantaneous bit rate when possible and so transfer the most information. The idea of effective bit rate relies on the provision of enough storage. Effective bit rate is defined as:

$$\dot{I}_e = \dot{I}_i \quad v$$

where $\dot{I}_i$ is the instantaneous bit rate and v is the fractional viewing time. Note, there are three effective bit rates defined: one for the direct link, and two for the relay link. By a continuity condition (i.e., finite storage), the two effective bit rates for the relay link are really only one (see Section 6.1.4).

1.1.1.3 Viewing fraction. The average fraction of time two stations are intervisible is a main factor in the calculation of the effective information rate $\dot{I}_e$. As with $\dot{I}_i$, there are three pertinent viewing fractions in this problem: that for the lander–orbiter link, the orbiter–Earth link, and the lander–Earth link. In these applications the viewing fractions are, in general, less than unity because the line of sight between the two bodies of interest is interrupted by the planet Mars. The viewing fraction is discussed more thoroughly in Section 4.1.3.2 where methods for its calculation are given.

1.1.1.4 Reliability function. Reliability functions are produced for all components that have a typical direct link and relay link mission. The reliability functions are either discrete, where a component is estimated to perform satisfactorily with some probability that is independent of time, or time-dependent, where a component is estimated to have some constant failure rate and failures occur randomly with time.

The reliability functions are then combined as the mission requirements dictate so that conclusions about the absolute and relative reliabilities of the direct link and relay link missions can be drawn.

Because the short and long term survivability of the equipment used in the two mission approaches may be different, parameters reflecting this difference are included in the reliability analysis. The inclusion of these degrees of freedom allows measuring the relative effects of the complexity of the hardware used for each of the approaches. Also, the sensitivity of each approach to various reliability levels is assessed.

1.1.1.5 Expected cumulative bits. The average amount of information that can be expected from the direct link and the relay link missions is defined as the expected cumulative bits. This parameter is a measure of the degradation in data gathering potential due to the unreliability of each mission approach and is shown to be a function of the mission reliability function, the effective bit rate, and the length of the surface operation by the following relationship:

$$\bar{I}(T) = \dot{I}_e \int_{t_0}^{T} R(t)\, dt$$

where t_0 is the start of data accumulation mission and T is its termination.

The behavior of this integral as a function of T is of interest because one may identify T directly as the length of the mission. Therefore, one may answer the question: "How much data can I expect over a mission of time T?" and get the answer for all T from zero to infinity. If the expected cumulative data rises rapidly to its asymptotic value, this system of data retrieval may be more desirable than one that exhibits a slow rise. A judgment on the relative value of early and late bits is needed to assess which behavior is more desirable.

1.1.2 Alternate Mission Modes

1.1.2.1 Relay satellite orbit selection. Two classes of orbits are considered for the relay link mission. These classes are: (1) favorable communication orbits and (2) orbits that are selected for scientific experiments (or might be characteristic of partial orbit insertion failures). The communications orbit is chosen mainly to maximize the amount of data obtained from the lander, giving an upper bound on the relay link approach. A variation in the orbit affects two of the comparison parameters: the viewing fraction and the range. Section 4 shows the behavior of these parameters as a function of the selected orbit. The science orbit parameters are chosen to try and satisfy some orbit science constraints as well as to maximize lander data transmittal. By showing the relay data characteristics of the two types of orbits, the penalty paid in lander data to get lander and orbiter science data simultaneously can be assessed. Also, either of these data rate numbers to the direct link data rate may be compared.

1.1.2.2 Data requirements. The amount of data to be sent back to Earth and the time to do so fixes the needed rate of data transmission. As will be seen later, the assumption of 10^{10} bits calls for large antennas and high-power transmitters. The effect of relaxing this requirement to 10^9 bits or lower is qualitatively studied in Section 6.1.1.

1.2 Limitations of Approach

1.2.1 View Fraction Calculations

The major mathematical limitations of the approach are due to the averaging process used in calculating the viewing fraction v. Because the averaging process is used only in calculating the lander–orbiter viewing parameters, the limitations mainly apply to the relay link.

This model does not have the ability to represent adaptivity of the mission sequence, (i.e., a change of scientific experiments) or a change in instantaneous bit rate because of unforeseen circumstances. In this approach, the averaging is performed before the calculation of the system behavior rather than computing an average as a result of many outcomes. Therefore, an increase or decrease in effective bit rate caused by a nonplanned circumstance (i.e., an adaptive policy) has not been represented.

Because the viewing fraction does not converge to the actual percentage view time for several days, short missions are not accurately represented. However, under most circumstances, the value of v is quite well behaved and it is felt that this approach is representative of missions that are more than a week long.

Furthermore, it is possible to set a favorable first phasing between the lander and the orbiter by a wise choice of injection geometry. Thus, the early part of the mission may involve partial view fractions, which are greater than the average; therefore, the value of v calculated here would be pessimistic for short missions.

The actual storage needed to support a given effective bit rate is a function of the input rate (the effective bit rate) and the sequence of view and nonview times. In general, the process that produces the length of the view times acts as an infinite memory Markov source; therefore, in theory, the whole history of the process is needed to properly size the memory to avoid any loss of lander data. The averaging process allows only a calculation of the average length of view time (or average time between views). In general, for a nondegenerate process (passes of differing lengths), this parameter is not sufficient to give any information on the needed size of the storage unit. Thus, a simulation of the ephemeris is necessary for this type of calculation. Section 6.2.3 shows some actual storage requirement calculations using the first 10 days of the orbit as a basis of estimation.

1.2.2 Reliability Calculations

Because the basic tool of the reliability model is the assumption of random failures, no wear-out or infant mortality phenomena are simulated. This assumption may be particularly restrictive for the steering mechanism of the lander high-gain antenna.

Because the reliability analysis is also done on an averaging basis, no failure distributions are available. This approach was adopted instead of a direct simulation because the orbit view time calculations were done on an average basis. The computer time needed for a direct simulation of the mission is too prohibitive to make it attractive as a study tool here.

A rather gross limitation in the reliability approach adopted is the inherent low confidence in the model itself. Because there are no actual designs or design data available for the prediction of system reliability, the model has been constructed as an extension of existing hardware into the needed functions. It is most likely that the hardware cannot be made to operate in the mode needed by the functional sequence, but perhaps there is no better estimate available. For example, none of the existing subsystems are designed to withstand sterilization, yet all the lander subsystems are assumed sterilizable. Also, the survivability of the present day subsystem has not been proved to the potential shock levels to be encountered at landing. In general, a note of caution must be adopted when dealing with the reliability figures produced in this report. Their main function is to provide an equivalent basis for comparing the two system approaches. It is hoped that any failure to represent reality is equal for both the direct and relay links.

There is one area where this assertion is not valid and should be reemphasized. The Mars surface environment certainly has a differing effect on the two approaches. If there is great uncertainty in the surface slope, the strength of the surface winds, and similar surface parameters, the direct link is much more at their mercy than is the relay link. One can say that the comparison is between the uncertainty in the direct link high-gain antenna environment and the complexity of the relay link equipment. Because of this, the confidence in the direct link analysis will be low until the surface and atmospheric data are made available. Once there is a good estimate of what is there, even though the data shows an unfavorable environment, there is a much better chance of designing equipment that can survive with a high level of confidence.

Therefore, one must evaluate the direct link reliability results because a surface suitability of 0.9 means we are 0.9 sure that the surface is suitable as far as the design goes, not that the system is 0.9 reliable operating on the surface. Contrast this with the relay system failure rate that says there is a 0.9 probability that the satellite station will last through the needed time. For the lander, either the surface is suitable or it is not. If not, every lander that we send will fail with 1.0 probability. If the surface is suitable, no degradation will be suffered. A 0.9 probability of surface suitability merely represents an estimate of the ratio of favorable to unfavorable cases. This inequity in comparisons will persist until further definition of the Mars lower atmosphere and surface environment. Now, one can only qualitatively say that there is a larger uncertainty associated with the direct link approach than in the relay link approach.

1.2.3 Cost

There is one more important parameter that has not been included in the comparison, yet needs mentioning: the cost of the two system approaches. This comparison can be made without dependence on the remainder of this report.

2. ASSUMPTIONS AND CONSTRAINTS

2.1 Applicable to Both Direct and Relay Links

2.1.1 Science and Engineering Mission Support

It is assumed that, eventually, in the Mars exploration program, a primary mission objective will be recovery of large amounts (10^{10} bits total) of Mars surface data from a landed spacecraft for 6 mo. This same basic mission objective is assumed for either data recovery approach. For the relay link approach, the spacecraft orbit is chosen mainly to support the relay communications function. Also, it is assumed that there are no scientific experiments that need the simultaneous operation of an orbiter and a lander. Because there is no scientific payload specified now that does need simultaneous operation, this assumption does not appear particularly restrictive.

2.1.2 Equipment State of the Art

To make reliability analyses of both communications schemes on an equivalent basis, it is assumed that both systems need the same equipment state of the art. Because the assumed mission is in the 1975–1977 period (see Section 2.1.7), equipment may be considered to be 1970 state of the art.

2.1.3 Launching Vehicle

This study has not been restricted to only those trajectories or satellite orbits peculiar to a particular launching vehicle. It is assumed that the launching vehicle has enough injection energy to deliver either a direct lander or a combination lander–orbiter in the 1975 or 1977 launch times with equivalent injection success probabilities.

2.1.4 Environment

To compare the reliabilities of the two systems on an equivalent basis, it is assumed that both systems are properly designed to withstand the fabrication, test, injection, and transit environments; and that these phases of the program affect the reliabilities of the two systems equally. Note that this includes lander sterilization. But, the surface environment is assumed to affect the two landers unequally as discussed in Section 6.1.2.

2.1.5 Lander Descent Data

Because this study is mainly concerned with the recovery of data from long-term, high-data-rate surface experiments, neither system has been constrained to get lander descent data.

2.1.6 Landing Site

Some of the interesting features on the surface of Mars are the dark areas lying between 10°N latitude and 30°S latitude. For calculations it is assumed that a landing site at ±10° latitude has been selected.

2.1.7 Trajectory Geometry

The trajectory geometry is chosen to be characteristic of 1975 and 1977 opportunities. This does not mean to imply that such a mission would actually be undertaken during these years, but rather is simply to cite two typical opportunities. Range safety restrictions normally limit the launch azimuth to 90 to 114 deg; this limitation eliminates all Type I trajectories for both 1975 and 1977. However, if it were possible to obtain a waiver to use north–east launch azimuths up to 45 deg, then both 1975 and 1977 Type I trajectories would be available.

The Mars approach geometry characteristics of the 1975 and 1977 trajectories result in minimum inclination orbits (with no injection plane change) in the range of 3 to 20 deg.

Nominal transfer trajectories are assumed corresponding to minimum hyperbolic excess energy at Earth. It seems likely that a launch time for either the 1975 or 1977 opportunities would be approximately centered about these trajectories.

Table 1 summarizes the characteristics of 1975 and 1977 Type II minimum launch energy transit trajectories.

Table 1. Characteristics of 1975–1977 Type II minimum energy trajectories

Year	Flight time, days	Earth–Mars range at encounter km	Earth–Mars–Sun angle at encounter deg	Minimum orbit inclination without plane change, deg
1975	378	376×10^6	10.5	19
1977	340	339×10^6	21.5	2.8

2.1.8 Guidance Requirements

To simplify the reliability analysis, the guidance accuracy requirements for the direct lander and the orbiter-relay lander are assumed to be the same. Therefore, the same subsystems and functional sequence to separation are assumed for either mission profile.

2.1.9 Communications Backup Mode

Engineering diagnostic telemetry is assumed to be carried on a low-bit-rate direct link on both the relay and the direct landers. This link is not dependent on orientation of the capsule or survival of the orbiting spacecraft, so the probability of recovering failure data from either approach will be considered identical. This backup link for the later mission is assumed to be the low-data-rate FSK (frequency shift keyed) system developed for earlier *Voyager* missions.

2.1.10 Sterilization

As mentioned in Section 2.1.4, it is assumed that both approaches use a completely sterile lander. The direct link bus and the relay link orbiter, however, are assumed to be unsterilized.

2.1.11 Storage Requirements

For analytic simplicity, it is assumed that enough storage is available to make efficient use of the bit-rate capability of either the direct or relay link approach. Storage requirements for each approach are shown for typical systems in Section 6.2.3.

2.1.12 Deep Space Network Capabilities

It is assumed that all Deep Space Network stations are equipped with 210-ft-diameter antennas and are continuously available to receive data from either a lander or an orbiter.

2.2 Direct Link

2.2.1 Transmission System

Because of potential ionization breakdown problems in the Mars atmosphere, it is assumed that transmitter power is limited to 20 to 50 w. It will be shown (Section 3) that this power constraint makes necessary a very high-gain antenna because of the data requirements and ranges involved. This direct consequence of the power limitation assumption is fundamental to the whole analysis of this report.

2.2.2 Prime Data Link

For reliability calculations, it is assumed that the prime high-data-rate link from the lander to Earth is a coherent S-band link.

2.3 Relay Link

The largest single difference between the direct link lander and the relay link lander is in the communications equipment. Differences in the equipment used to support the two links, such as power and temperature control, are considered secondary for this analysis.

2.3.1 Lander Radio System

The relay link will incorporate different radio equipment showing the best approach to the solution of the relay problem. This equipment may operate on a VHF band, and may have solid-state components only.

2.3.2 Antennas

It is assumed that the lander antenna is erected to the local vertical and that its pattern is symmetric about the local vertical. It is also assumed that the orbiter has a local vertical tracking antenna to receive the lander signals and a large high-gain antenna to return the data to Earth.

2.3.3 Orbit Considerations

It is assumed that the orbiter will not be sterilized; therefore, all orbits considered must satisfy the 50-yr noncontamination constraint. This limits the lowest orbital altitude to the range of 2000 to 5000 km and places a lower bound on the lander–orbiter communication distances.

It is also assumed that proper orbiter operation is not dependent on continuous onboard tracking of the various reference bodies. Therefore, one or more of the references can be occulted without interrupting Earth–orbiter communications.

3. TELECOMMUNICATIONS

The telecommunications problems divide into three parts, each part corresponding to one of the legs of the data retrieval schemes under consideration. Each of the links has particular design requirements that are quite distinct. In the relay link, data is transmitted to the satellite when a receiver on the ground detects a beacon from the satellite. This data is stored on the orbiting spacecraft and then relayed to Earth by an S-band transmitter and a high-gain antenna. In the direct link, a high-gain steerable antenna on the ground tracks the Earth and transmits on Earth command when the Earth is in view.

The telecommunications parameters given in this section are wholly based on those derived in Ref. 1.

3.1 Lander–Orbiter Communications

3.1.1 Radio System

3.1.1.1 Lander transmitter. Optimization studies show that a high-reliability, low-weight transmitter for the power range under consideration ($\sim$ 10 w) would, most likely, be solid state operating in the 200 Mc frequency band.

3.1.1.2 Orbiter receiver. Because the lander-to-orbiter communications are started by command, the orbiter has a radio beacon that operates continuously. When the lander receives the beacon signal it starts to transmit to the orbiter. The orbiter then automatically acquires the lander carrier, and goes on to detect and store the lander data. Because rapid automatic acquisition is needed, a wide bandwidth receiver is necessary. The data rate from the lander to the orbiter is an order of magnitude higher than the data rate from the orbiter to the Earth; therefore, a storage device on the orbiting spacecraft is needed.

3.1.1.3 Radio system parameters. Table 2 lists the pertinent parameters of the lander-orbiter radio link. The parameters are based on VHF link with enough bandwidth to allow acquisition with the doppler shift because

Table 2. Lander-to-orbiter communication parameters

Parameter	Value	Tolerance	Comment
Transmitting circuit loss (Total losses between transmitter and antenna terminals. Includes losses mentioned in antenna discussion.)	0.5 db	± 0.2 db	
Transmitting antenna gain	Variable	± 2.0 db	
Transmitting antenna pointing loss	Variable	± 1.0 db	
Transmitter frequency	200 Mc		
Polarization loss Axial ratio transmitter = 10 db Axial ratio receiver = 5 db	1.5 db	± 1.2 db	
Receiving antenna gain	Variable	± 1.0 db	
Receiving antenna pointing loss	Variable	± 1.0 db	
Receiving circuit loss	0.8 db	± 0.2 db	
Receiver noise spectral density	−169 dbm/cps	± 1.0 db	5 db noise figure
Carrier automatic phase control noise bandwidth	75 cps	±7.5 cps	
Required carrier signal-to-noise ratio in $2B_{LO}$ for data transmission	6.0 db		
Required data ST/N B[a] $P^b = 5 \times 10^{-3}$ 1.5 db RF losses, 0.5 db miscellaneous losses)	7.2 db	+ 0.5 − 0.0 db	
Total transmitter power	Variable	± 1.0 db	

[a]Signal power per bit divided by noise power per unit bandwidth.

of the radial motion of the orbiter about the lander. The 90% probability acquisition time is approximately 20 sec for the parameters chosen.

3.1.1.4 Lander–orbiter bit rate vs power-gain product. Figure 1 shows the bit rates possible using the system parameters of Table 2. The bit rates are given as a function of power-gain product and lander–orbiter range. Note the low power-gain products give slightly less bit rate than the usual linear relationship. This is due because optimum modulation index needs more power in the carrier at low-bit-rates, lowering the available power for the data.

3.1.2 Antenna System

3.1.2.1 Lander transmitting antenna. Several types of antennas are available for use as a lander transmitting antenna. Typically, the antenna will be a simple turnstile over a ground plane, helix, or crossed slot of low to medium gain and relatively wide beamwidth. The actual choice of beam pattern is dictated by the choice of the orbit, landing site, the orbiter antenna pattern, and (most important) physical constraints such as size and ruggedness. The relative data retrieval capabilities of several beam patterns are discussed in Section 6.2.1.

The most efficient antenna system is one that produces a constant signal level at the input to the receiver throughout the time of communications. This implies that the antenna system should have gains proportional to the square of the range. Although this is generally not physically feasible, the combined shaping of the lander transmitting and the orbiter receiving antenna gives enough degree of flexibility so the lander–orbiter antenna system may be made quite efficient.

3.1.2.2 Orbiter receiving antenna. The orbiter receiving antenna is located on the planetary horizontal platform (PHP) and, therefore, is oriented to the local vertical at all times. Because the orbits are relatively high, and because the lander-orbiter transmission does not occur down to the horizon, it is possible to make the orbiter antenna somewhat directional. The pattern chosen for the orbiter is cosine-squared, operating between highest off axis limits of 35 deg. The theoretical on-axis gain of the antenna is 7.7 db. Note, a polarization loss of 1.5 db (Table 2) is incorporated into the bit rates of Fig. 1.

3.2 Orbiter–Earth Communications

3.2.1 Radio System

The orbiter–Earth leg of the relay link communications can be mechanized using the techniques developed in the *Mariner* series spacecraft. The system will operate at S-band with power levels on the order of 35 w needed to give the desired bit rates. Table 3 lists the radio system parameters, and Table 4 lists the receiving station parameters of the 210-ft-diameter ground antenna.

Table 3. High-gain antenna orbiter and direct communication parameters

Item	Value	Tolerance
Total transmitter power, db	Variable	± 1.0
Transmitting circuit loss, db	2.5	± 0.5
Transmitting antenna gain, db	36.7	± 1.0
Transmitting antenna pointing loss, db	0.5	± 0.5 − 0.0
Antenna axial ratio, db	1.0	
Frequency, Mc	2295	

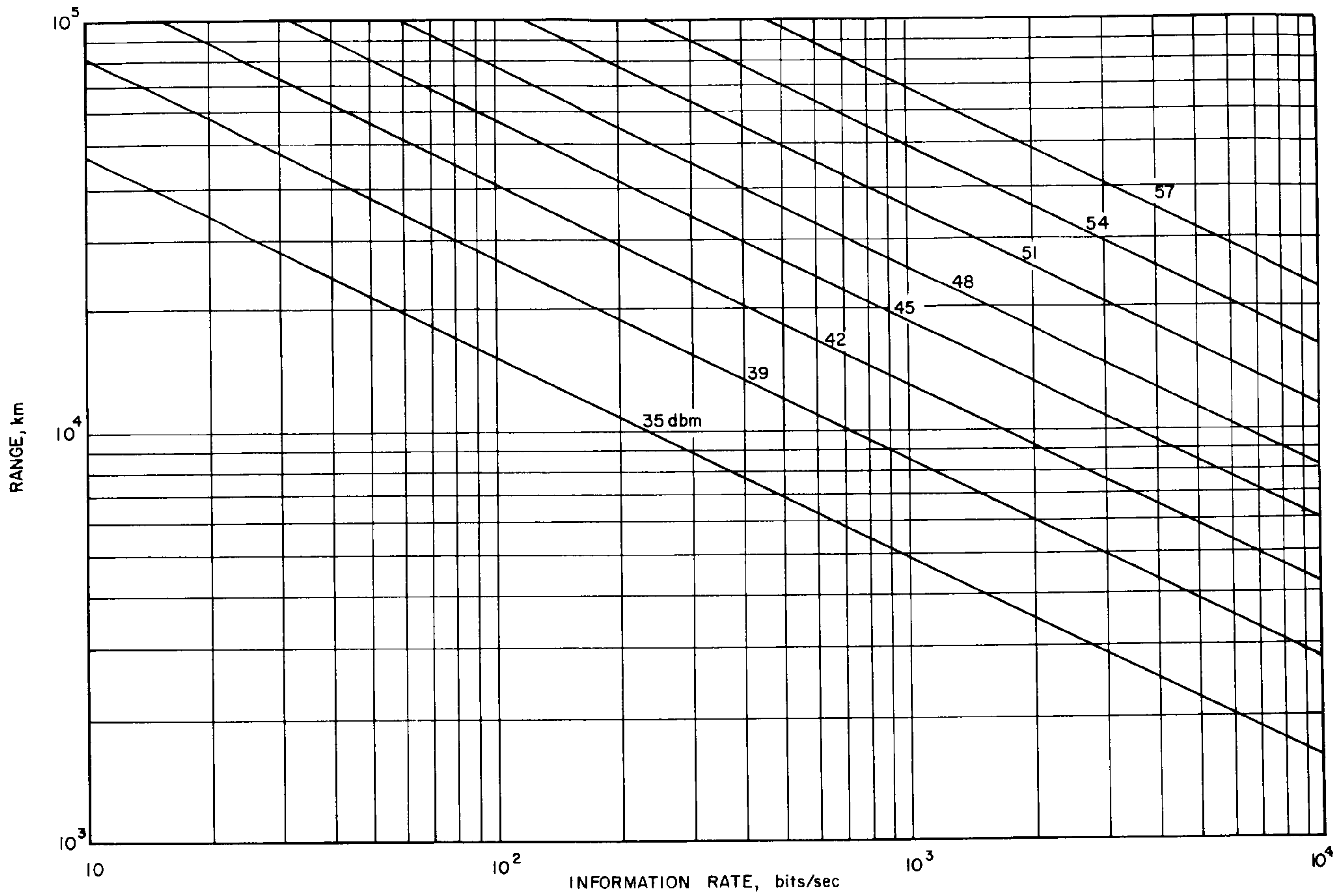


Fig. 1. Information rate vs lander-to-orbiter range for various power-gain products

Table 4. Orbiter-to-Earth deep space instrumentation facility parameters

Item	Standard deep space station 85-ft antenna		Low noise listen station 210-ft antenna	
	Value	Tolerance	Value	Tolerance
Antenna gain, db	53.0	+ 1.0 − 0.5	61.0	± 1.0
Axial ratio, db	0.75	± 0.25	0.5	
Feed line loss to low noise amplifier, db (Includes diplexer loss)	0.18	± 0.05	0.02	+ 0.01
Antenna temperature, °K (zenith)	16	± 3	10	± 2
Maser temperature excess/ second stage, °K	18	± 3	18	± 3
Transmitter noise contribution, °K	10	± 3		
Carrier automatic phase control noise bandwidth, cps	12	+ 0.79 − 0.97	12	+ 0.79 − 0.97
Required carrier signal-to-noise ratio in $2B_{LO}$, db (for data reception)	6.0		6.0	
Required data $ST/N\,B$[a], db ($P^b = 5 \times 10^{-3}$, includes 1.5 db RF losses)	± 7.2	+ 0.5 − 0.0	± 7.2	+ 0.5 − 0.0

[a]Signal power per bit divided by noise power per unit bandwidth.

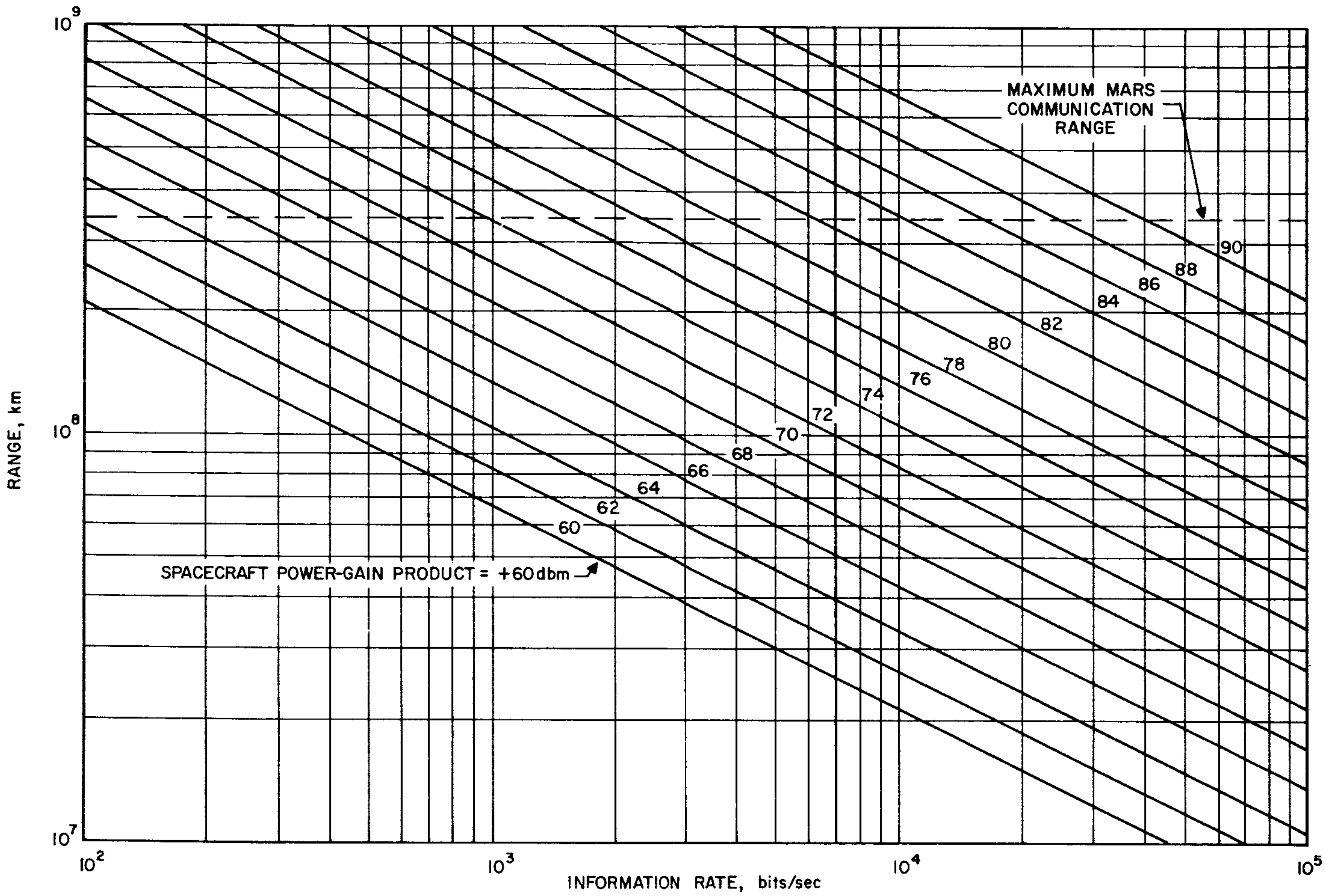


Fig. 2. Information rate vs range with 210-ft-diameter ground antenna

3.2.2 Orbiter Transmitting Antenna

The orbiter transmitting antenna is a 12.5-ft-diameter parabolic dish that is Earth-oriented by an active radio tracking loop. The on-axis gain is 36.7 db, and the maximum pointing loss is 0.5 db.

3.2.3 Bit Rate as a Function of Power-Gain Product and Range

Figure 2 shows the power-gain product needed to transmit at a desired bit rate over a given range. Note the Earth–Mars range is approximately 350 million km for Type II arrivals. The needed power-gain product at this distance for a bit rate of 3300 bits/sec is 80 dbm.

3.3 Lander–Earth Communications

3.3.1 Radio System

Because the direct link system must operate over the same range at approximately the same data rate, the radio system is similar to the orbiter radio system. However, to get a data rate matching that of the orbiter, it is necessary to operate at higher power. The basic unit will operate at S-band and Table 3 displays representative parameters.

3.3.2 Bit Rate vs Power-Gain Product

Because the basic system parameters are assumed to be similar, Fig. 2 may be used to calculate the needed power-gain product as a function of desired bit rate and transmission range.

3.3.3 Antenna System

It is assumed that the antenna structure used on the orbiter may also be used in some form for the lander. Coarse acquisition of the Earth may be made using geometric information such as the position of the Sun. Note that, for the 1975 arrival, the angle between the Earth and the Sun is quite small (10 deg), making it relatively simple to locate the Earth. The fine acquisition and tracking can be done using the same RF pointing system that would be developed for the orbiter.

3.4 Backup Communications

In either the relay link or direct link, there will be a relatively low-data-rate backup system that transmits directly to the Earth. A separate antenna system will be used for this transmitter; therefore, the retrieval of the failure and diagnostic data can be made independent of the working of the high-gain antenna in the direct link or the orbiting relay satellite in the relay link. It is quite likely that this direct link system will be the system developed for the earlier *Voyager* missions. If so, it may use frequency shift keying as a data modulation technique.

4. RANGE AND VIEWING CHARACTERISTICS

The following section discusses the effect that the relative geometry of the transmitting and receiving stations has on the performance of both the relay and direct links. The section is broken into two parts: the first treats the lander–orbiter relationships and the second treats the Mars–Earth relationships. In the second part, the similarities and differences of the lander–Earth and orbiter–Earth parameters are delineated.

4.1 Lander-to-Orbiter

4.1.1 Characteristics

The problem of communication from a ground station to a satellite is not new. Television transmission by the Relay, Syncom, and Early Bird satellites is now almost routine. Various authors (e.g., Ref. 1) have proposed satellite retrieval of meteorological data from simple remote ground stations, a situation closely analogous to the lander–orbiter portion of the *Voyager* relay link.

The most important geometric characteristics of the line-of-sight communications of the lander-to-orbiter portion of the link are the distribution and length of the times of intervisibility and the range during these times. The first item is necessary for calculating how much time is available for transmission, and the second sets the rate at which this transmission can be made. A combination of these two factors is used to find the effective information rate, $\dot{I}_e$, one of the bases for comparison discussed in Section 1.1.1.2.

4.1.2 Orbit Selection Standards

The following paragraphs discuss the various constraints and standards for selecting an orbit about the planet. These standards fall into four broad categories with overlapping or conflicting requirements: noncontamination, engineering, communications, and science. An orbit that meets the mandatory requirements of each of these can be found but, beyond that, tradeoffs must be made to raise one capability at the expense of another.

4.1.2.1 Noncontamination. One of NASA's constraints is that the *Voyager* mission should have a probability of contaminating Mars of less than 10^{-4} until manned missions can be flown to the planet. The orbiter is assumed to be unsterile so its lifetime must be greater than the time needed for the high probability of a successful manned flight. This time gap is usually taken to be 50 yr. Using the NASA Model II atmosphere, this implies that a satellite in a circular orbit must have an altitude greater than 5000 km and, in an elliptic orbit of eccentricity of 0.5, minimum elements are $h_p = 4000$ km and $h_a = 20{,}000$ km. Therefore, the NASA Model II atmosphere constrains the orbit to quite a high altitude.

The NASA maximum atmosphere may be unrealistically dense. Mr. R. A. McClatchey of JPL's Space Science Division has formulated an atmosphere model that allows circular orbits down to 2000 km altitude. This low altitude orbit is used as a sample for calculations in Section 6.2.1.1.

4.1.2.2 Engineering. Assumptions have been made (Section 2.3.3) that occultations of the various reference bodies do not preclude proper operation of the orbiter. It would be desirable, however, to avoid occulations if this can be done without harmfully affecting the communications ability. In particular, it may be necessary to avoid interrupting the Earth–orbiter line of sight if this largely decreases the time available for returning data to Earth (it is presumed that by 1975, RF occultation by the Martian atmosphere will no longer be needed). It should be said that, for the postarrival conditions of Type II trajectories, Earth and Sun occultations by the planet Mars are closely correlated because the three bodies are roughly in a straight line. Table 5 shows the threshold inclinations of the bodies of interest for two communications orbits. This only shows the first orbit after injection and does not apply to conditions several days later. Note, a 5000 km altitude circular orbit inclined 40 deg to the equator occults none of the reference bodies on the first several revolutions. Then, no plane change maneuver (Table 1) is needed for injection into this orbit.

Table 5. Threshold inclinations for occultations on the first orbit

Minimum inclination, deg				
Orbit	Year	To avoid Earth occultation	To avoid Sun occultation	To avoid Canopus occultation
5000 km,	1975	24	30	48
circular	1977	10	10	46
2000 km,	1975	39	53	22
circular	1977	18	27	27

4.1.2.3 Communications. If a satellite is to be put in orbit about Mars for relaying information from the planet surface to Earth, then it is desirable that it maintain line-of-sight communications for a large fraction of the mission. Assuming there are n viewing times or lander–orbiter sightings after a time t_n, then the partial viewing fraction v_n is defined as $\sum_{i=1}^{n} \tau_i/t_n$ where τ_i is the length of the ith view time. That is, v_n equals the ratio of the time in view to the total time at t_n and lies between zero and one. The limit of v_n as n goes to infinity is defined as the viewing fraction v and is thoroughly treated in Section 4.1.3.2.

$$v = \lim_{n\to\infty} \sum_{i=1}^{n} \frac{\tau_i}{t_n} \tag{1}$$

The best viewing fraction is obtained from a satellite orbit that matches the planet's rotation time. This orbit has a v of unity if the satellite is in view and a v of zero otherwise.

Disregarding the synchronous satellite, the viewing fractions for circular orbits increase with the altitude as shown in Fig. 3.

Besides the viewing fraction characteristics, it is desirable to have the lander–orbiter range as low as possible to minimize the inverse square power loss in transmission. This implies low altitude orbits to get high data rates. Note, this conflicts with the viewing requirements stated earlier for circular orbits. The tradeoff between gain in view fraction by going to higher orbits vs the inverse square loss is discussed in Section 6.2.1.1.

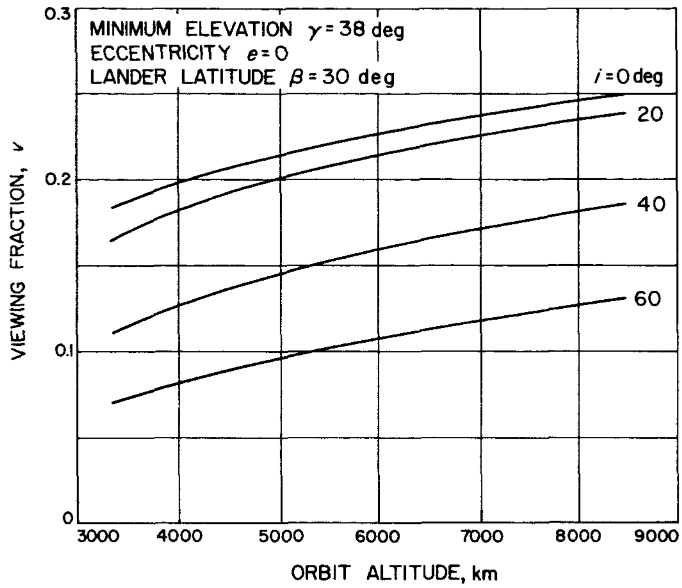


Fig. 3. Viewing fraction vs orbit altitude circular orbit

If the lander-to-orbiter is the saturated link (see discussion on the limiting link in Section 6.1.4), then it may be necessary to eliminate Earth occultations to raise the Earth–orbiter viewing fraction. This implies adjusting the orbit inclination to the ecliptic; the longitude of the ascending mode is nearly fixed by the arrival conditions. Table 5 has shown the minimum inclinations to the Mars equatorial plane for nonoccultation of the Earth.

Another communication standard is the length of the sighting (the size of τ_i in Eq. 1). If the time needed for lockup is significant compared with the total viewing time, then the function v_n is not representative of the fraction of time available for transmission. This would mean raising the satellite's altitude to lower its angular velocity with respect to an observer on the ground. An estimate of the sighting times and the time between sightings is also needed to set data storage requirements for both the lander and the satellite. Actual storage requirements are discussed in Section 6.2.3.

4.1.2.4 Science. The satellite may have functions besides relaying information from the lander to Earth. In particular, it may be needed to gather science data from orbit through such instruments as television, Mars scanner, fields and particles probes, etc. Here, an eccentric orbit may prove more useful.

4.1.3 Computation of Range and Viewing Characteristics

The fraction of time the lander and orbiter are in view and the lander–orbiter range have both been discussed as standards for orbit selection in Section 4.1.2.3. The following section shows methods for calculating these characteristics and numerical results.

*4.1.3.1 **Range characteristics.*** Because range characteristics are needed for communications calculations, it is necessary to develop methods for producing them. This is greatly helped by the concept of the surface on which the satellite always lies, or simply the satellite surface; this concept is developed in the following paragraphs.

4.1.3.1.1 Satellite surface. In an inertial coordinate system, the set of all possible satellite positions can be thought of as a closed curve, an ellipse. Essentially, this curve is the collection of all possible satellite locations. Similarly, in a planet rotating system, the set of all possible satellite positions can be thought of as a surface. Analogously, this surface is the collection of all possible satellite orbits. To an observer on a rotating planet, the motion of a satellite consists of its elliptical movement in its orbital plane and a constant rotation of this plane about the planet axis. Therefore, the spatial locus of all possible satellite positions is produced by rotating the orbit ellipse about the planet axis. The rotation of any curve about a fixed axis produces a surface. The surface produced by the rotation of the orbit ellipse is therefore referred to as the satellite surface. Note again, this is the locus of all positions that the satellite can possibly occupy. Figure 4 is an isometric representation of a typical satellite surface.

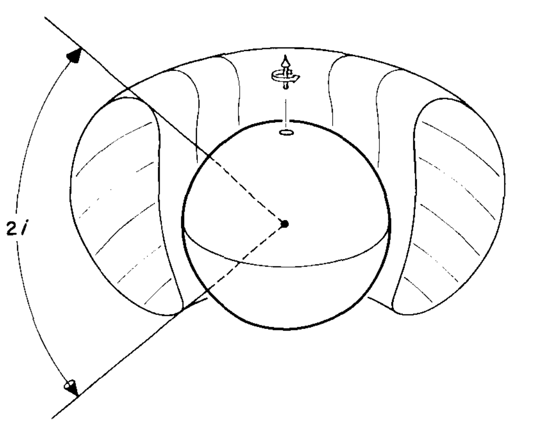


Fig. 4. Satellite surface and planet

Because the satellite surface is a surface of revolution, a complete description may be obtained by studying its intersection with a plane containing the axis of symmetry. This intersection is defined as a cross section of the satellite surface. A cross section may be analytically produced by plotting r, the satellite planet center distance, as a function of its latitude. From celestial mechanics the range, r, is given by the orbital elements and the true anomoly, f. The true anomoly, in turn, is related to the satellite latitude, ϕ, through the inclination, i, and argument of periapsis, ω. These producing equations are:

$$r = \frac{p}{(1 + e \cos f)}$$

$$\sin \phi = \sin i \sin (f + \omega) \, .$$

These equations were programmed on the analog computer and plots of cross sections are shown in Fig. 5, reprinted from Ref. 2. In each of the plots, the value of the eccentricity is shown at the location of the periapsis.

Figure 5 shows that the satellite surface is, in general, double-sided (i.e., there are two values of r at the same latitude, ϕ). It should be emphasized that the satellite never falls within the volume enclosed by the surface but only on its boundary.

Two special satellite surfaces occur. When periapsis lies at minimum or maximum latitude, there is only one satellite radius associated with each latitude and the double surface degenerates to a single one. Figure 6 shows an isometric representation of this. Where periapsis is at minimum latitude (shown in Fig. 5d), the satellite surface cross sections are simply single lines, as opposed to closed loops. The second special surface occurs when the orbit is circular; here the satellite surface is just a spherical zone.

4.1.3.1.2 Lander–orbiter range. Next, consider the possible lander–orbiter ranges. The lander can be thought to have a distorted umbrella over it on which the satellite, when in view, always lies. A different umbrella-like pattern is associated with the inner and outer surfaces but, unlike the circular orbit where the umbrella is just a spherical cap, the surfaces for the general orbit are not necessarily symmetric about the lander's local vertical. The general shape of the umbrella-like surfaces may be seen by imagining a planet, lander, and lander viewing cone superimposed on the satellite surface cross sections of Figs. 5a through 5c. Figure 7 shows the conically shaped volume, where the lander and orbiter are covisible

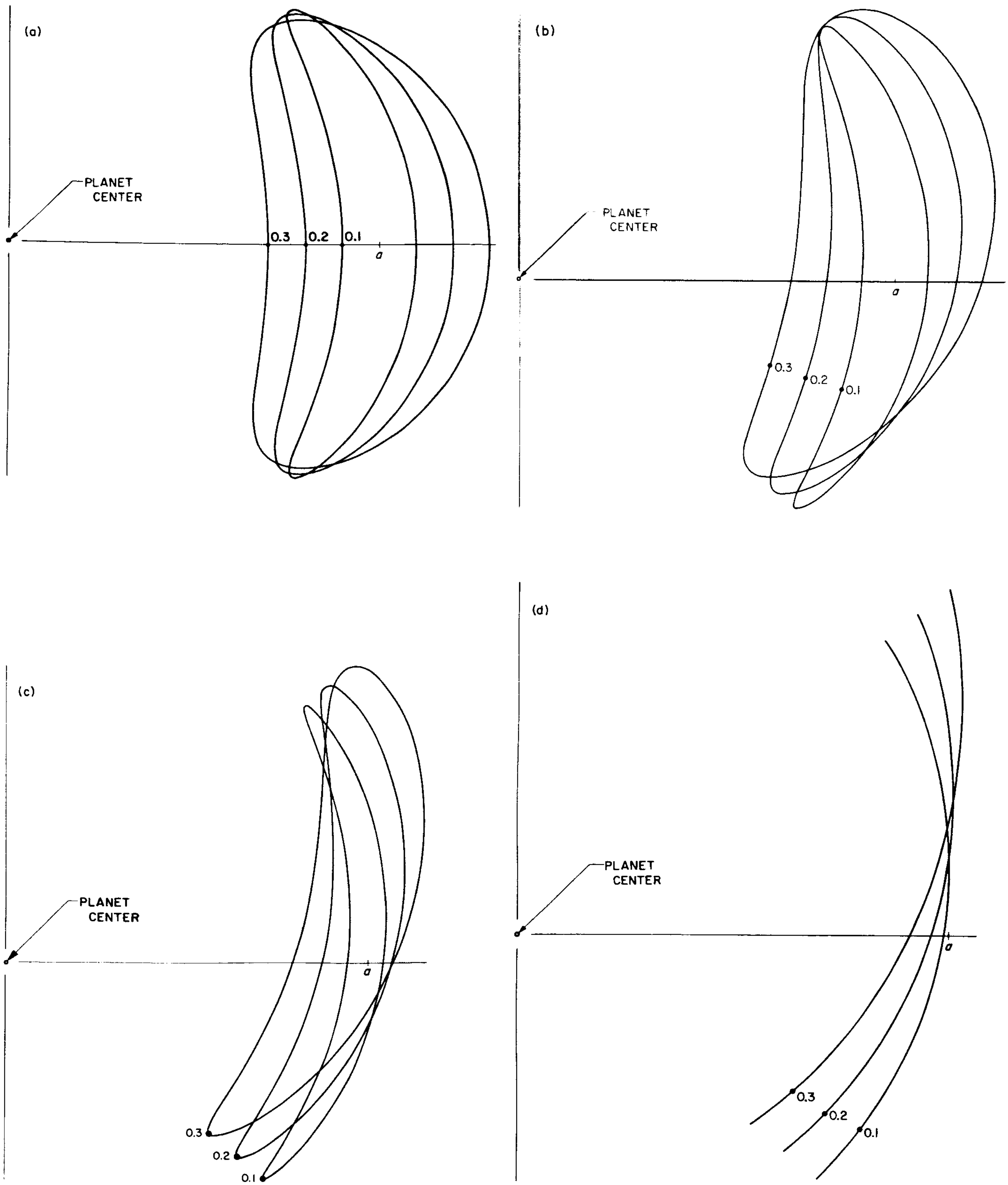


Fig. 5. Satellite surface cross sections for various eccentricities: (a) periapsis argument = 0 deg, (b) periapsis argument = −30 deg, (c) periapsis argument = −60 deg, (d) periapsis argument = −90 deg

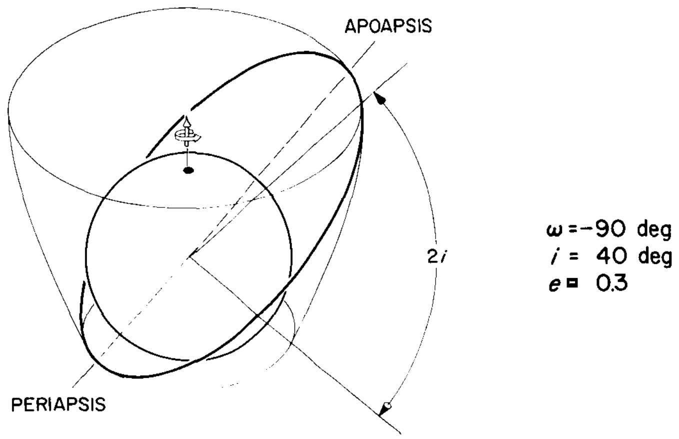


Fig. 6. Degenerate satellite surface periapsis at minimum latitude

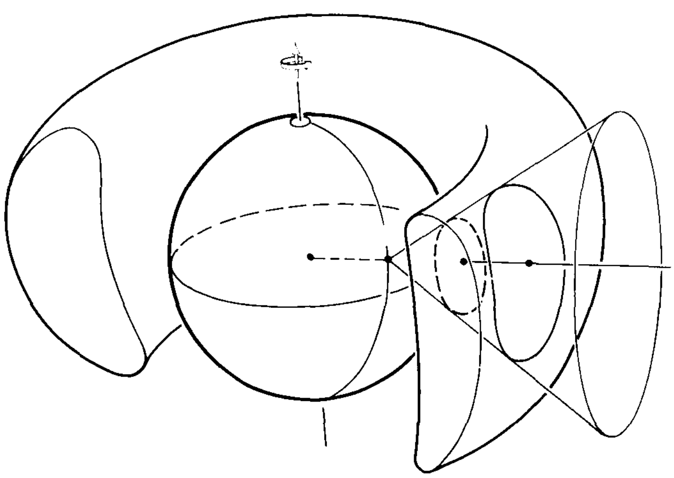


Fig. 7. Viewing cone on satellite surface

(referred to as the viewing cone), and its two intersections with the satellite surface.

4.1.3.2 Viewing characteristics. The partial viewing fraction v_n has been defined in Eq. (1) (this equation is repeated below) as the fraction of time the satellite has been in view at time t_n, and the view fraction v, as the limit of the partial fraction when t_n goes to infinity.

$$v = \lim_{n \to \infty} \sum_{i=1}^{n} \frac{\tau_i}{t^n}$$

The function v depends on the lander latitude and minimum elevation angle, β and γ, and the orbital elements a, e, i, and ω: the semi-major axis, eccentricity, inclination, and argument of periapsis, respectively. (See Fig. 8 for lander–orbiter geometry.)

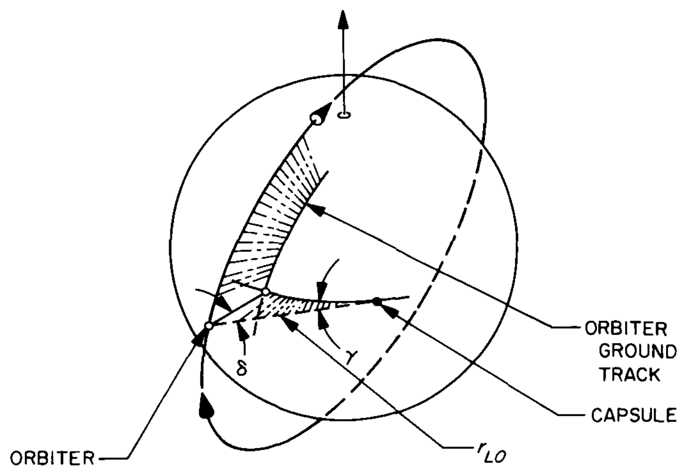


Fig. 8. Lander–orbiter geometry

Reference 2 analyzes this long term average fraction with the aid of the satellite surface concept developed. For view fraction calculations, the satellite surface is separated into two distinct parts (referred to as the inner and outer surfaces) at the minimum and maximum latitudes. Separate view fractions are associated with each of these two surfaces and are called v_1 and v_2; the sum of v_1 and v_2 equals v. The following expressions are derived in Ref. 2 for the inner and outer fractions:

$$v_1 = \frac{(i - e^2)^{3/2}}{2\pi^2} \int_{f_{11}}^{f_{12}} \cos^{-1}$$

$$\left\{ \frac{\cos \alpha (f) - \sin \beta \sin i \sin (f + \omega)}{\cos \beta \, [1 - \sin^2 i \sin^2 (f + \omega)]^{1/2}} \right\} \frac{df}{(1 + e \cos f)} \; 2$$

$$v_2 = \frac{(i - e^2)^{3/2}}{2\pi^2} \int_{f_{21}}^{f_{22}} \cos^{-1}$$

$$\left\{ \frac{\cos \alpha (f) - \sin \sin i \sin (f + \omega)}{\cos \beta \, [1 - \sin^2 i \sin^2 (f + \omega)]^{1/2}} \right\} \frac{df}{(1 + e \cos f)} \; 2$$

$$v = v_1 + v_2$$

where

$$\cos \alpha (f) = \sqrt{1 - \frac{r_{LO}^2 (f)}{r^2 (f)} \cos^2 \gamma}$$

$$r_{LO} = \sqrt{r^2(f) + r_m^2 \cos^2 \gamma} - r_m \sin \gamma$$

$$r(f) = \frac{a (1 - e^2)}{1 + e \cos f}$$

r_m = planet radius

f_{11}, f_{12} = the limits of the true anomoly when the satellite is on the inner surface

f_{21}, f_{22} = the limits of the true anomoly when the satellite is on the outer surface

For a circular orbit of radius R, the integral for v simplifies to:

$$v_{CIRC} = \int_{f_1}^{f_2} \cos^{-1}\left(\frac{\cos\alpha - \sin\beta \sin i \sin f}{\cos\beta\sqrt{1 - \sin^2 i \sin^2 f}}\right) df$$

where

$$\cos\alpha = \sqrt{1 - \frac{r_{LO}^2}{R^2}\cos^2\gamma}$$

$$r_{LO} = \sqrt{R^2 + r_m^2\cos^2\gamma} - r_m \sin\gamma$$

Circular orbit view fraction is shown as a function of $h\ (= R - r_m)$ and the inclination in Fig. 3. The inner and outer view fractions for elliptic orbits are plotted separately in Fig. 9. This figure shows the variation of v with eccentricity for constant periapsis altitude of 4000 km. Note, the view fraction increases with eccentricity up to a certain critical value and then decreases. In all the plots the total view fraction v may be obtained by adding v_1 and v_2. Note, in Fig. 9d the inner and outer view fractions are equal. This is because the argument of periapsis equals 90 deg, periapsis occurs at maximum altitude, and the satellite surface degenerates to the single-sided surface similar to that of Fig. 6.

4.1.4 Communications Implications

The concept of the satellite surface and its implications for possible lander–orbiter ranges allows certain observations to be made about the ground-satellite antenna

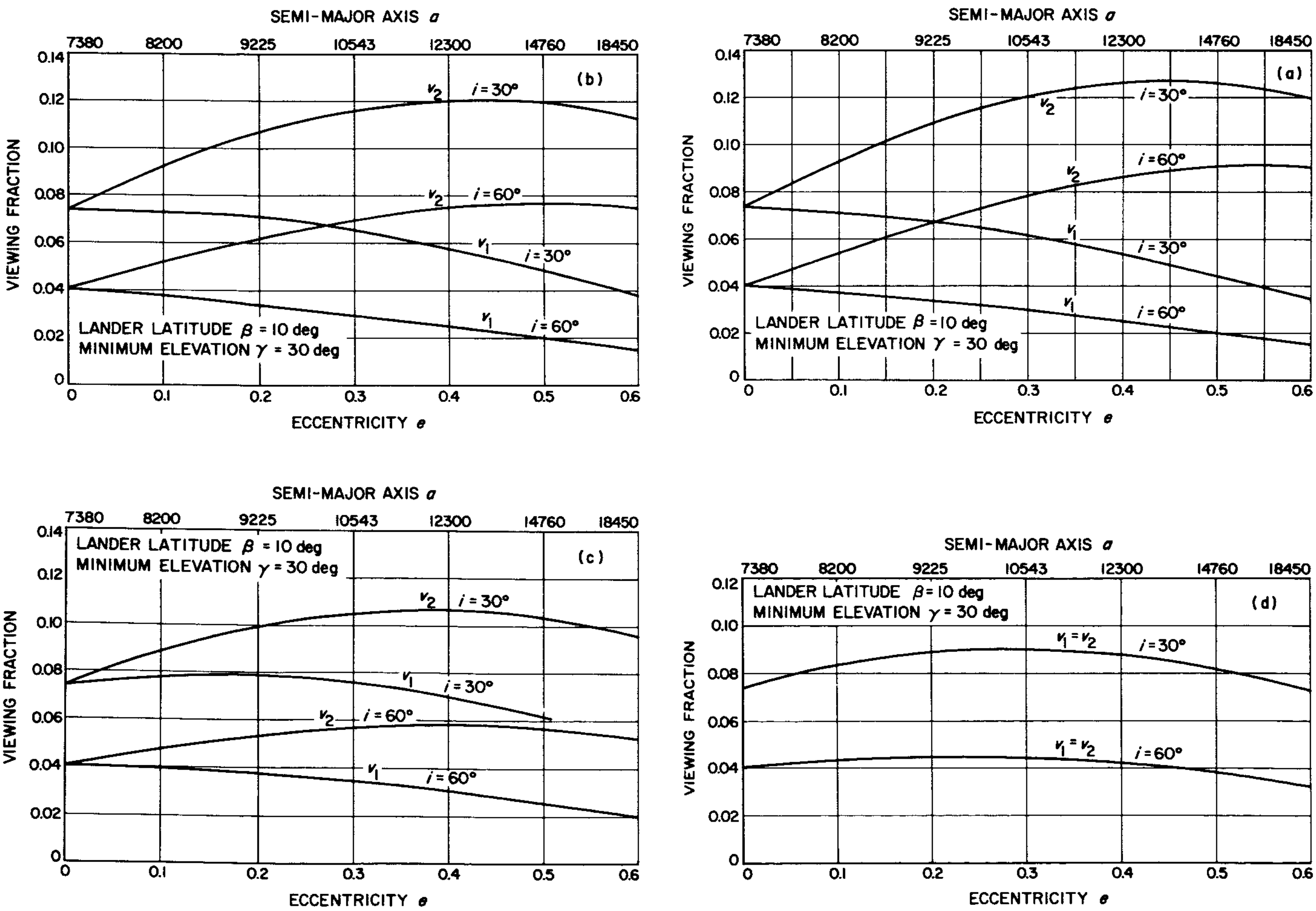


Fig. 9. Viewing fraction vs eccentricity for constant periapsis altitude of 4000 km: (a) periapsis argument = 0 deg, (b) periapsis argument = 30 deg, (c) periapsis argument = 60 deg, (d) periapsis argument = 90 deg

system. Note, the power radiated at the transmitter would be used most efficiently if the received signal-to-noise ratio were at its lowest acceptable value during the whole time of lander–orbiter communications. This implies that the antenna system gain product divided by the range squared should be a constant over the angle of transmission; the product of the antenna gains at any given elevation angle should be proportional to the lander–orbiter range squared. Unfortunately, shaping the transmitter beam exactly to the range profile squared may be impossible, but this can serve as a guideline in the design of the antenna system. It should be noted that there are two antennas involved and only the product of their gains is the parameter of interest; therefore, a wide range of choices is available. In particular, one antenna can have a region of low gain in the center of its pattern if the other matches this hole with a high-gain lobe. This idea has been developed in Ref. 1, where one of the antennas is a four-arm balanced conical spiral.

The combination of the viewing fraction developed earlier and the range profile from the satellite surface allows calculation of the best use of a given antenna system. The figure of merit of the system (and, therefore, the variable to be maximized) is the effective information rate, $\dot{I}_e$, discussed in Section 1.1.1.2. Assuming that the lander antenna is oriented to the local vertical and that the orbiter antenna is pointed at the center of the planet, the power received at the satellite is[1]:

$$P_R = K \frac{G_T(\psi, \sigma)\, G_R(\delta(\psi, \sigma))}{r^2_{LO}(\psi, \sigma)} P_T$$

where

ψ = 90-γ is the angle of the lander zenith

σ = azimuth angle

δ = lander–orbiter-planet center angle (see Fig. 9b)

G_T = transmitting antenna gain

G_R = receiving antenna gain

r_{LO} = lander–orbiter range

K = a proportionality factor

P_T = transmitted power

[1]Two other angle-dependent quantities, the polarization loss and the atmosphere attenuation, have been assumed constant. These quantities, if included, would decrease P_R monotonically with increasing ψ and would make the best value of ψ (calculated above) slightly smaller.

The minimum power received at a given angle from the zenith is:

$$P_{\min}(\psi) = K \min_{0 \le \sigma \le 2\pi} \left\{ \frac{G_T(\psi, \sigma)\, G_R[\delta(\psi, \sigma)]}{r^2_{LO}(\psi, \sigma)} \right\}$$

Therefore, if the information rate $\dot{I}_i$ is constant over the whole angle of transmission ψ_m and is governed by the minimum received power during the transmission time[2], then the effective information rate $\dot{I}_e$ is:

$$\dot{I}_e(\psi_m) = v(\gamma = 90 - \psi_m) \cdot \dot{I}_i \min_{\psi \le \psi_m} [P_{\min}(\psi)]$$

where $\dot{I}_i$ $(\cdots)$ shows that the instantaneous bit rate is some function of the minimum of the received power. In particular, if the constant $\dot{I}_i$ is simply proportional to this minimum received power, then:

$$\frac{\dot{I}_e(\psi_m)}{K' P_T} = v(\gamma = 90 - \psi_m) \min_{\substack{0 \le \sigma \le 2\pi \\ \psi \le \psi_m}} \frac{G_T(\psi, \sigma)\, G_R(\psi, \sigma)}{r^2_{LO}(\psi, \sigma)} \tag{2}$$

This expression can be maximized with respect to ψ_m to find the best angle through which transmissions should be made. This has been done for three representative situations described in Section 6.2.1.1.

4.2 Mars-to-Earth

The following section discusses the geometric aspects of the Mars-to-Earth portions of both the direct and relay communication schemes and delineates similarities and differences between the two.

4.2.1 Range Characteristics

For communications, the distance from a Mars lander to Earth is essentially the same as that from a Mars satellite to Earth; i.e., distances comparable with the dimensions of the planet can be neglected when compared with the astronomical distances involved.

As seen earlier, the arrivals for the proposed trajectories occur with Earth and Mars on nearly opposite sides of the solar system. This is unfortunate because of solar noise and communication range. Figure 10 shows the Earth–Mars range as functions of time. The arrival dates

[2]A much more involved scheme for maximizing the exchanged information is possible. If the bit rate were adaptive with the signal-to-noise ratio, then it could always be adjusted to the value of highest acceptable bit error probability.

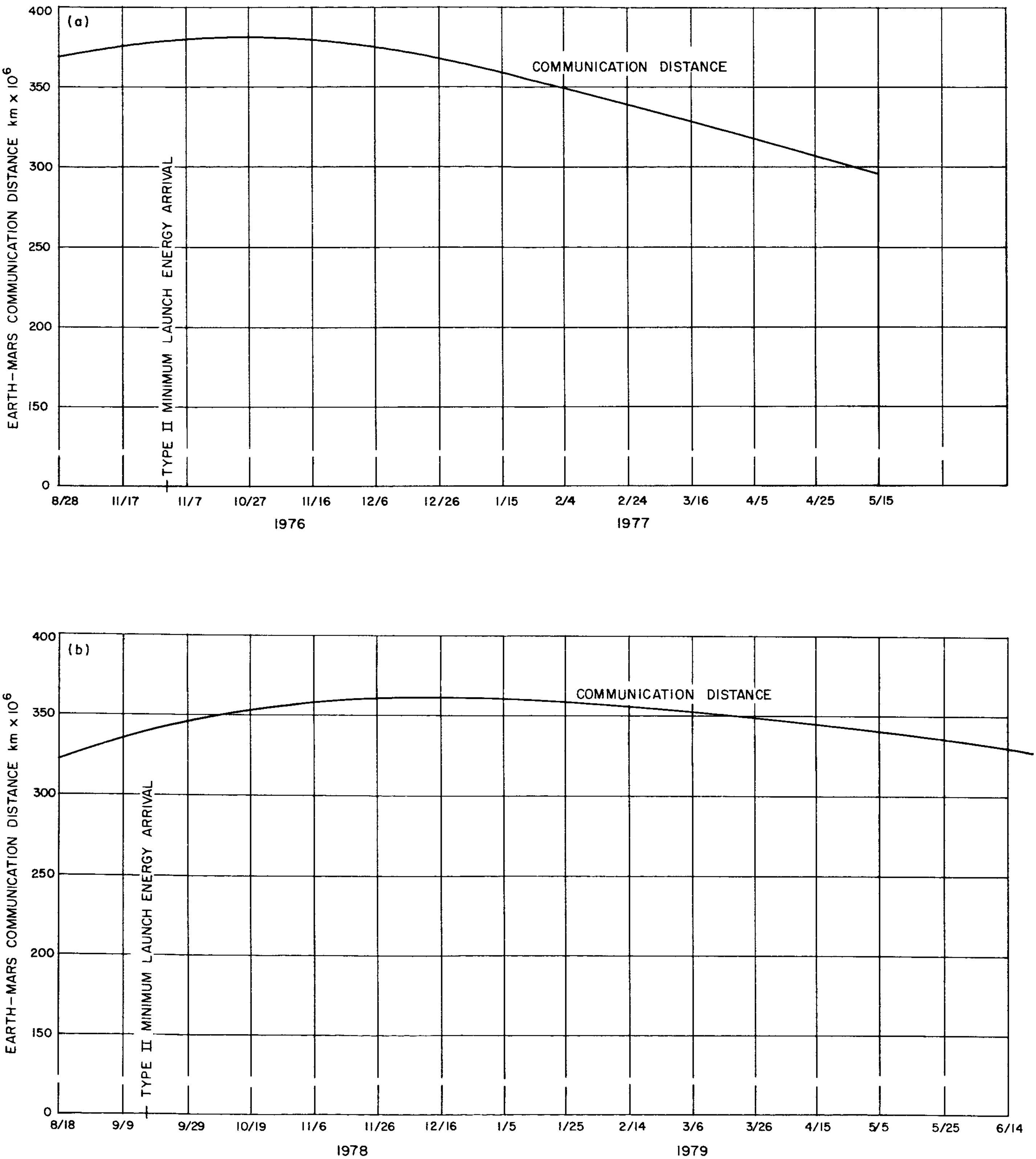


Fig. 10. Earth–Mars range vs time: (a) 1976–1977, (b) 1978–1979

of the two trajectories of interest are marked on each of the plots. Note, distances are in the order of 350 million km. Furthermore, the two ranges do not change appreciably over the 6 mo of expected operation. For the 1975 launch opportunity, where the variation is the largest, the range decreases by only 20%.

4.2.2 Viewing Characteristics

The Earth-viewing characteristics of a lander and an orbiter are quite different. The problem of Mars interrupting the orbiter–Earth line of sight and how this is brought about by inclination has been discussed earlier. For sample calculations it will be assumed that the orbit has been chosen to eliminate Earth occultation, at least in the early parts of the mission. The orbits discussed in Section 6.2.1.1 meet this requirement.

Therefore, the orbiter–Earth viewing fraction as defined in Eq. (1) is equal to unity. This is certainly desirable from a lockup standpoint because one-way communication times are in the order of 20 min and if no data is sent until lock has been made, then a large portion of the view time could conceivably be lost.

Unfortunately, the only way to maintain uninterrupted communication by a direct link is to choose a landing site near one of the poles. These locations have not been of particular interest; therefore, the view fraction for the direct link must be less than unity.

The declination of the sub-Earth point in aerocentric coordinates for both the 1975 and 1977 arrival dates is +21 deg. A plot of view time vs minimum elevation angle is given in Fig. 11 for the geometry and landers at latitudes of +10 and −10 deg. Very little definite knowledge of the Martian topography is available, therefore, the quantitative estimate of the amount of masking by terrain features is impossible. A reasonable guess of the minimum elevation angle governed by terrain protuberances is 30 deg. These conditions yield view fractions of 0.34 and 0.29 for landing sites of +10 and −10 deg, respectively.

During the year, the declination of the sub-Earth point fluctuates while always remaining within ±26 deg. These fluctuations cause the lander viewing fraction to change although it too remains within bounds. Table 6 shows the lowest and highest possible viewing fractions for a lander within 10 deg of the equator for various minimum elevation angles.

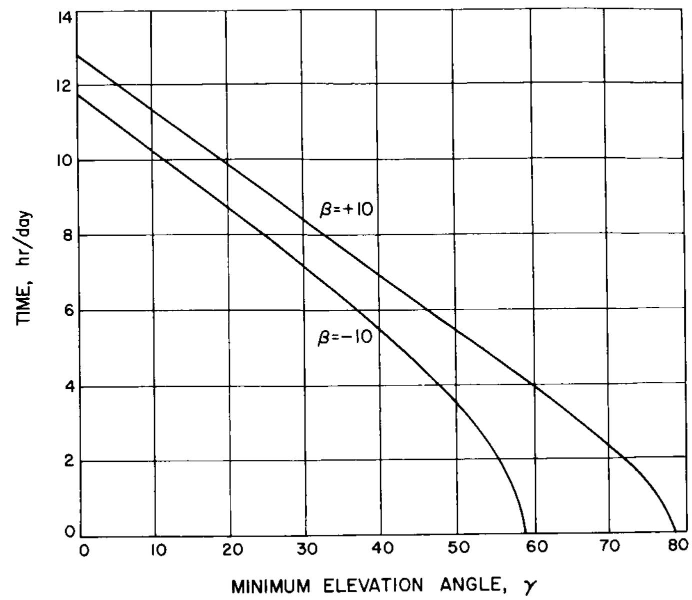


Fig. 11. Communication view time vs minimum elevation angle

Table 6. Viewing fraction extremes for landing site within 10 deg of Martian equator

Minimum elevation angle, deg	10	20	30	40
Highest fraction	0.46	0.40	0.34	0.28
Lowest fraction	0.41	0.34	0.28	0.20

4.2.3 Deep Space Network Requirements

The Deep Space Network (DSN) requirements for the relay and direct links are different. For the relay, a tracking station must maintain continuous communications with the orbiter to assure contiguous data, but communications with a direct lander need only be maintained one-third of the time. One system uses the DSN capabilities all the time, the other does not.

The appearance of the direct lander is very predictable, allowing regular scheduling of the DSN stations. The rotation time of Mars is 24.62 Earth hours, so the viewing will slowly shift from one station to the next. Often, two stations will be needed to track in series, with the third station idle for 8½ days.

4.3 Summary

This section has discussed the geometric aspects of both direct and relay communications. For the relay, the concept of the satellite surface has been developed; this, in turn, has led to certain observations on the fraction of time the satellite and lander are intervisible and the profile of possible lander–orbiter ranges. An averaging analysis rather than an emphemeris simulation has been applied to calculate the viewing fractions, which have been shown to fall in the range of 0.10 to 0.25. The satellite surface and viewing fraction concepts have been used to develop a method to optimize antenna system use. This optimization technique is applied to representative orbits and a sample communications system in Section 6.2.1.

The geometric characteristics of the Mars–Earth portion of both links have been shown to be much simpler than those mentioned earlier. The Earth–Mars range for the situations studied is on the order of 350 million km and the lander–Earth viewing fractions are approximately one-third.

5. DIRECT LINK AND RELAY LINK RELIABILITY

5.1 Introduction

The intent of this section is to evaluate, for designs proposed in current studies (Refs. 3 and 4), absolute and relative reliabilities at critical points in an advanced planetary mission.

Reliability will be used throughout this section to denote the probability that a system successfully satisfies the requirements for which it was designed. It should be emphasized that the reliability models and values used in this analysis are built on very uncertain ground. The hardware is undefined, the future state of the art is not established, and the past analyses are not universally accepted. Therefore, confidence in the numbers used is low. To provide for this uncertainty, a number of parameters are introduced. The parameters are then varied so the sensitivity of the system reliability can be seen.

5.2 Analysis

5.2.1 Summary of Analysis

Because the advanced planetary spacecraft subsystems are not designed now, a reliability analysis cannot be strictly representative. However, there are sequences in the advanced planetary mission that are functionally similar to deep space missions flown previously. Therefore, one can produce a partial reliability model based on these functional similarities. Where no previous design exists because of novel requirements imposed by the advanced planetary mission, subsystem designs were obtained from advanced planetary spacecraft systems studies.

Two typical missions, one using direct link communications and the other using relay link communications, were taken from material in Refs. 3 and 4. The major sequential events that occur throughout the mission were acquired from these mission descriptions. Each event was then analyzed to find the set of subsystems that must work properly at some time before or during the event to assure its accomplishment. Note, the analysis assumes that knowledge of the reliability of all the subsystems at a particular time implies knowledge of the overall system reliability.

5.2.2 Mission Sequence of Events

Here, the typical direct link and relay link missions will be described in general terms. The mission events analyzed are extracted from this description. Possible sequences of events for both the direct and relay link missions are shown in Table 7 and described in the following paragraphs. Functional block diagrams for the

Table 7. Sequence of major events for typical Voyager mission (assuming successful launch and injection)

Direct link	Relay link
Flyby bus	Orbiter
Deployment and acquisition	Deployment and acquisition
Midcourse trajectory corrections	Midcourse trajectory corrections
Lander separation	Lander separation
	Orbiter trim and retro maneuver
	Orbiter deployment and acquisition
	Orbiter operations
Lander	Lander
Flyby bus separation	Orbiter separation
Entry and landing	Entry and landing
Lander deployment and acquisition	Lander deployment and acquisition
Lander surface operations	Lander surface operations

direct and relay link events are shown in Figs. 12 and 13, respectively. The time of occurrence assumed for each event is depicted in Table 8.

5.2.2.1 Launch and transit mission description. Both missions begin by being launched from the Eastern Test Range (ETR) during 1975–77. Trip times to Mars range from 200 to 400 days for Type I and Type II trajectories, respectively.

For this analysis, the sequence from injection through capsule release for both missions is assumed to resemble

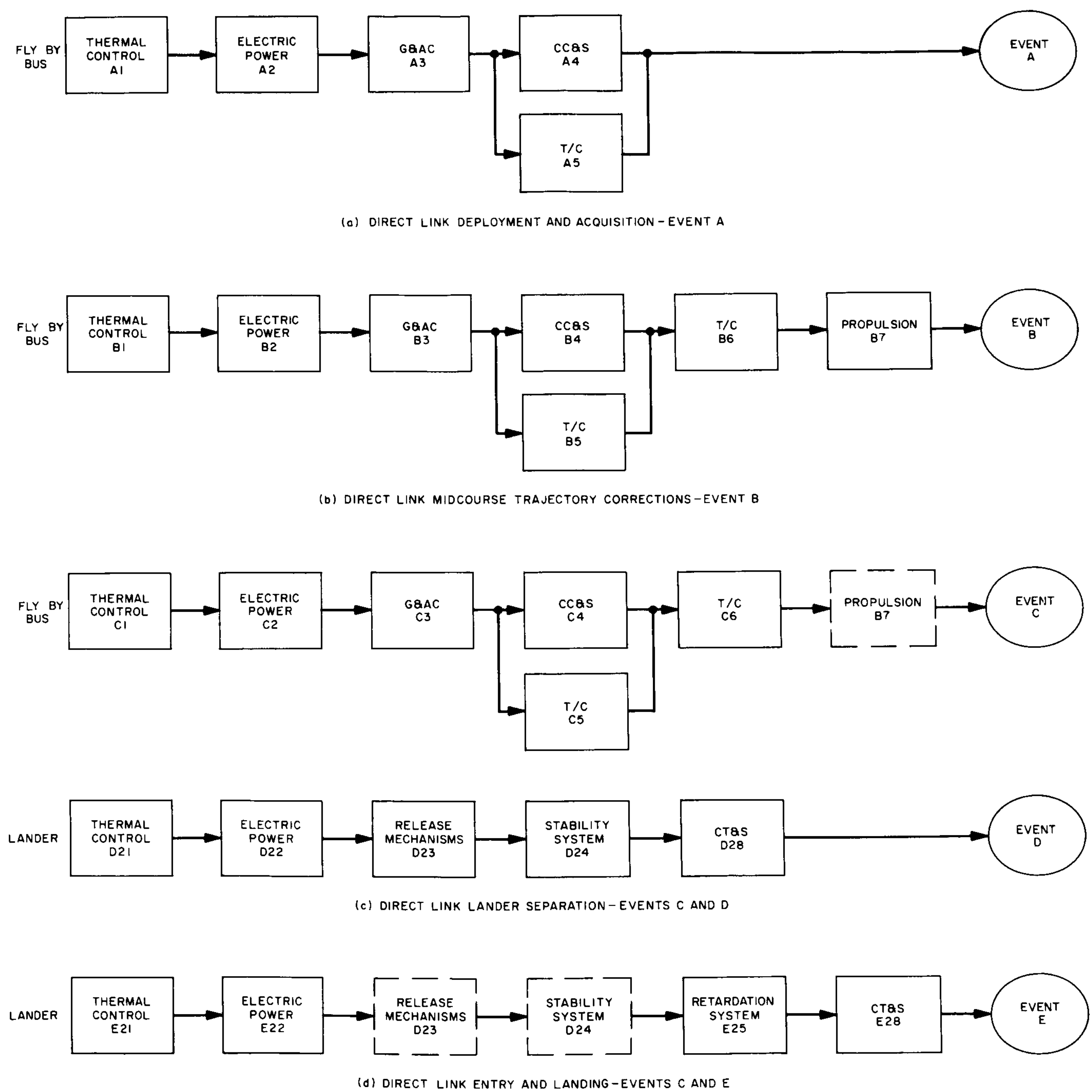


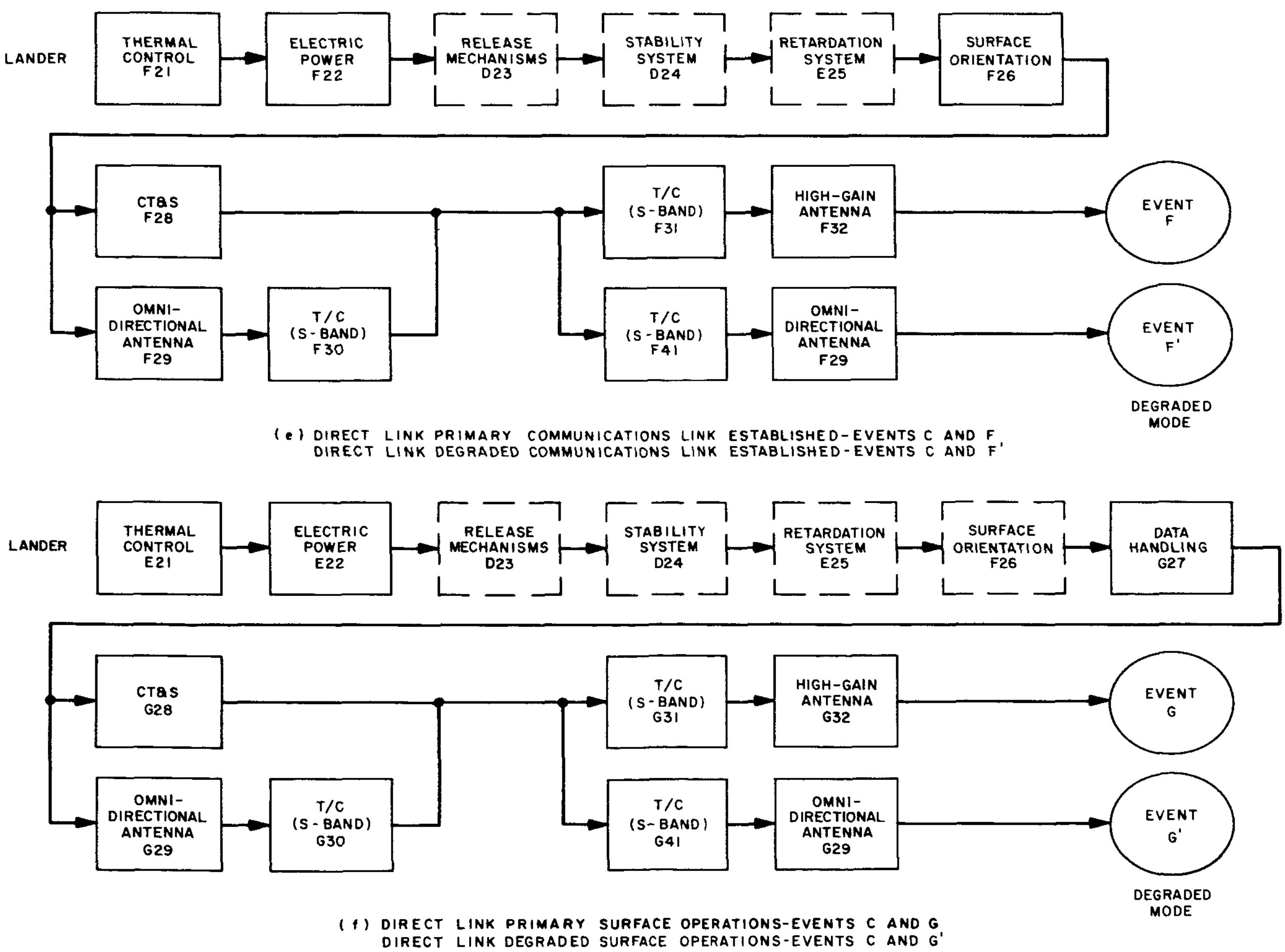
Fig. 12. Direct link events and required operating subsystems

that of the *Mariner* series of spacecraft. The spacecraft is separated and the launch vehicle is retarded by retro rockets. Storage batteries supply electrical power until solar panels are deployed and Sun acquisition occurs; then the principal axis of the spacecraft is pointed toward the Sun, and the solar energy collection and conversion system supplies power.

After the Sun acquisition, a Sun-Canopus sensing system is used for full three-axis attitude control. First Earth–spacecraft communications are conducted over an omnidirectional antenna but, sometime during the transit phase of the flight, a 12.5-ft antenna radiating 35 w at S-band frequencies and pointed toward Earth becomes the main communications link between the spacecraft and Earth.

One or more midcourse maneuvers, based on Earth-based radio tracking data are made during transit; enough maneuvers are performed to ensure that the trajectory is accurate for mission requirements. The midcourse propulsion system, as in the *Mariner* systems, uses storable liquid propellants; however, the thrust level is raised to a level compatible with orbit insertion requirements.

As the spacecraft approaches Mars, trajectory refinements may be made. Again, any maneuver uses Earth-based radio tracking techniques; no onboard approach-guidance systems are considered. About 80 hr before impact, the capsule is separated from the remainder of the spacecraft and set on an impact trajectory with the planet. Then, the direct link and the relay link systems are dissimilar.



(e) DIRECT LINK PRIMARY COMMUNICATIONS LINK ESTABLISHED-EVENTS C AND F
DIRECT LINK DEGRADED COMMUNICATIONS LINK ESTABLISHED-EVENTS C AND F'

(f) DIRECT LINK PRIMARY SURFACE OPERATIONS-EVENTS C AND G
DIRECT LINK DEGRADED SURFACE OPERATIONS-EVENTS C AND G'

Fig. 12. Direct link events and required operating subsystems (cont'd)

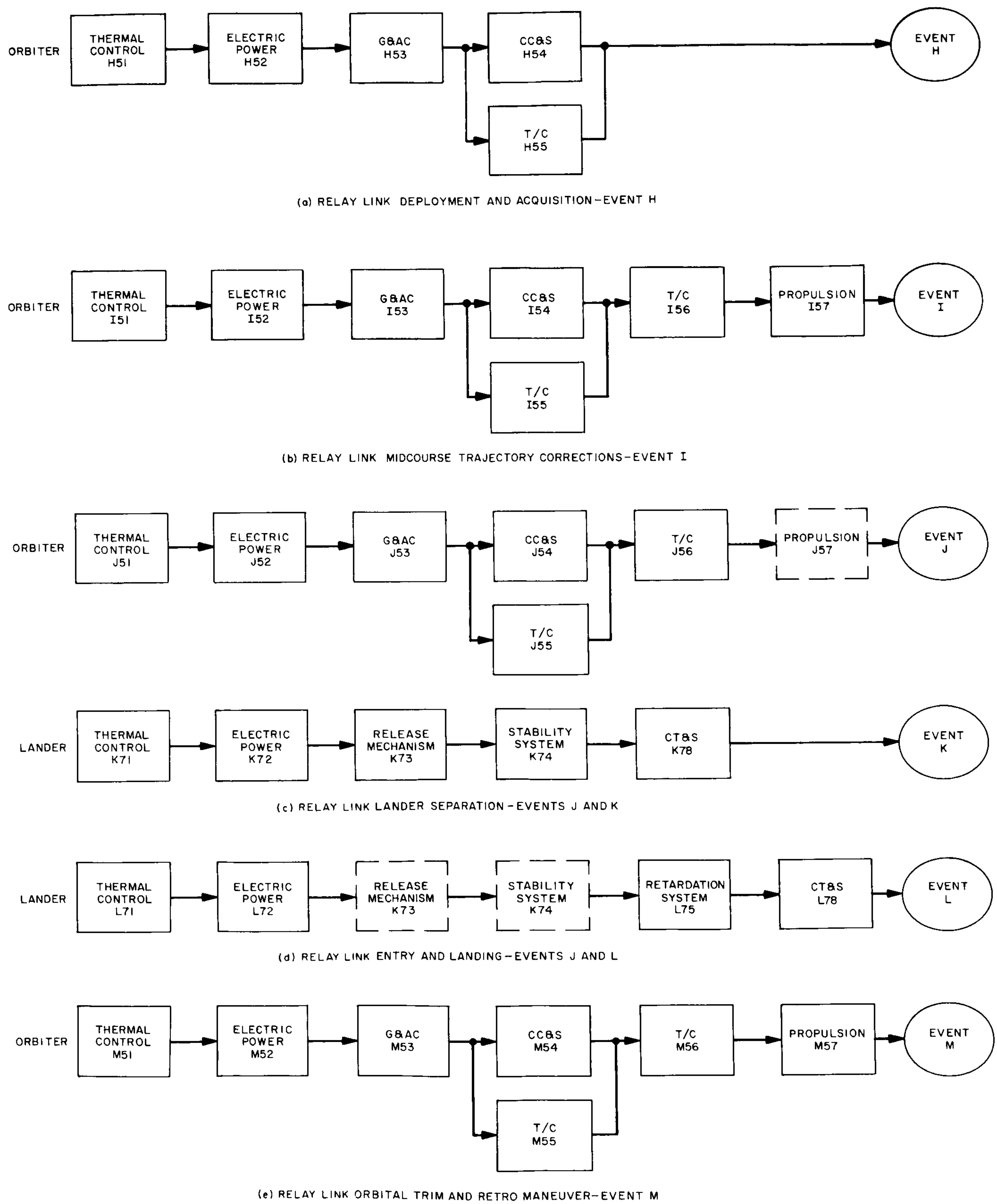
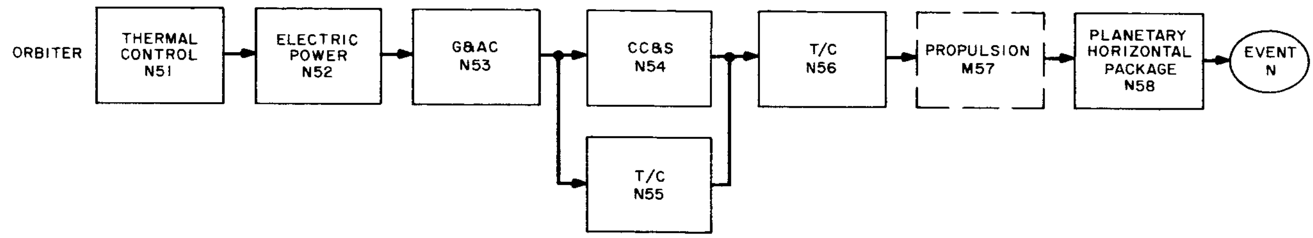
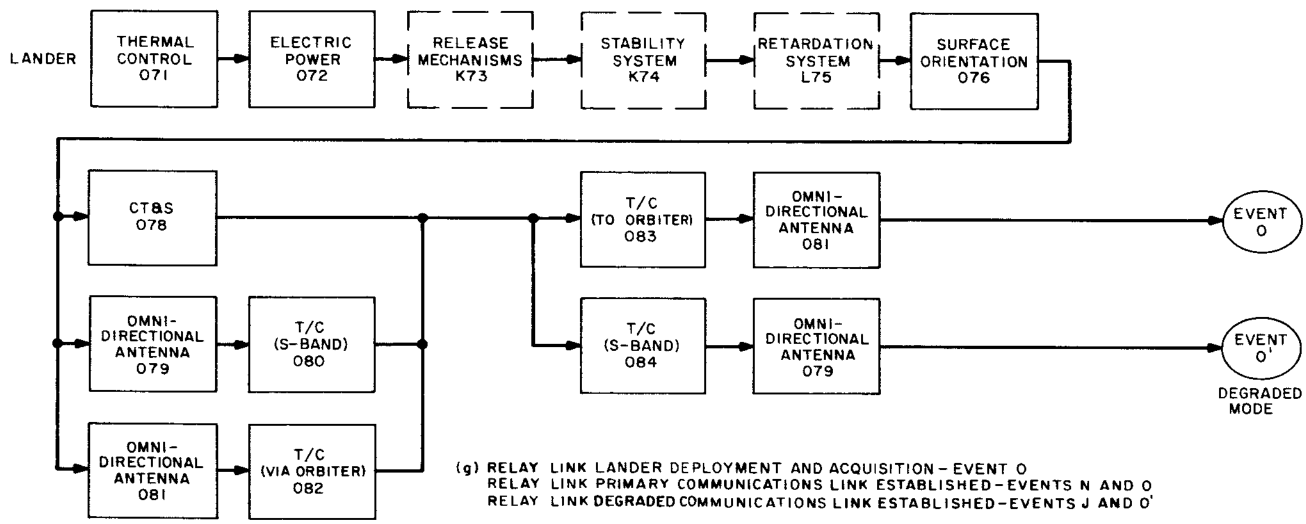


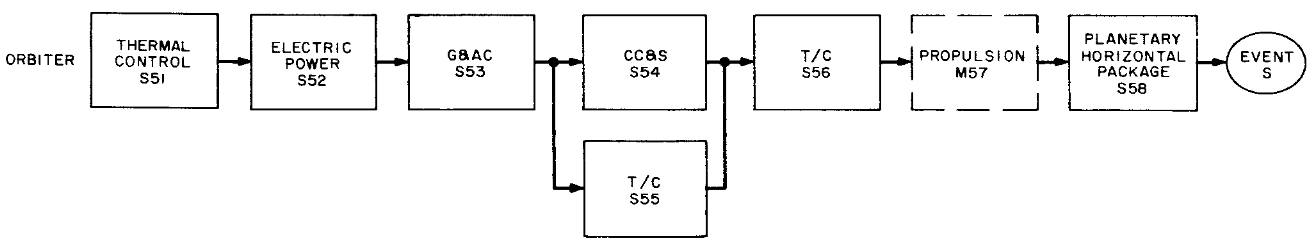
Fig. 13. Relay link events and required operating subsystems



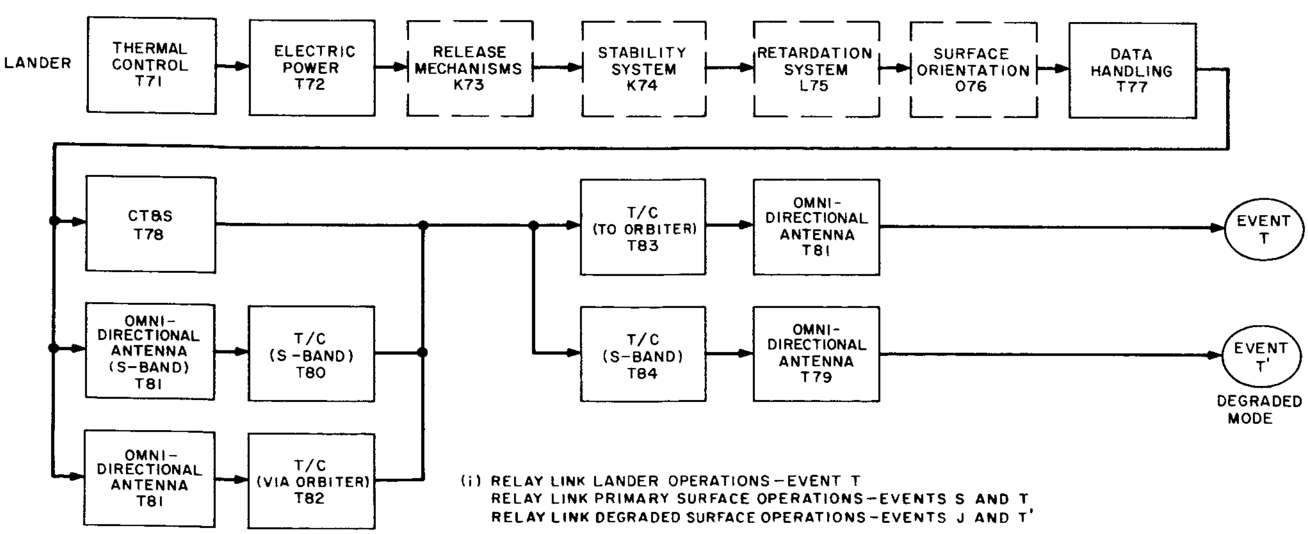
(f) RELAY LINK ORBITER DEPLOYMENT AND ACQUISITION—EVENT N



(g) RELAY LINK LANDER DEPLOYMENT AND ACQUISITION—EVENT O
RELAY LINK PRIMARY COMMUNICATIONS LINK ESTABLISHED—EVENTS N AND O
RELAY LINK DEGRADED COMMUNICATIONS LINK ESTABLISHED—EVENTS J AND O'



(h) RELAY LINK ORBITER OPERATIONS—EVENT S



(i) RELAY LINK LANDER OPERATIONS—EVENT T
RELAY LINK PRIMARY SURFACE OPERATIONS—EVENTS S AND T
RELAY LINK DEGRADED SURFACE OPERATIONS—EVENTS J AND T'

Fig. 13. Relay link events and required operating subsystems (cont'd)

Table 8. Event times (assumed for numerical calculations)

Trip time	200 days	400 days	Symbols (see Appendix B)
Direct link			
Flyby bus			
Deployment and acquisition	6 hr	6 hr	t_A
Midcourse trajectory correction	14 days	14 days	t_B
Lander separation	200 days	400 days	t_C
Lander			
Flyby bus separation	200 days	400 days	t_D
Entry and landing	203 days	403 days	t_E
Deployment and acquisition	*6 hr after landing*	*6 hr after landing*	t_F
Surface operations	Up to 1 yr after deployment and acquisition	Up to 1 yr after deployment and acquisition	t
Relay link			
Orbiter			
Deployment and acquisition	6 hr	6 hr	t_H
Midcourse trajectory correction	14 days	14 days	t_I
Lander separation	200 days	400 days	t_J
Orbiter trim and retro maneuver	203 days	403 days	t_M
Deployment and acquisition	6 hr after retro	6 hr after retro	t_N
Orbiter operations	Up to 1 yr after deployment and acquisition	Up to 1 yr after deployment and acquisition	t
Lander			
Orbiter separation	200 days	400 days	t_K
Entry and landing	203 days	403 days	t_L
Deployment and acquisition	6 hr after landing	6 hr after landing	t_O
Surface operations	Up to 1 yr after deployment and acquisition	Up to 1 yr after deployment and acquisition	t

*5.2.2.2 **Post separation direct link mission description.*** The direct link lander, which is completely sterilized, is assumed to be passive during entry and makes an uncontrolled aerodynamic entry into the atmosphere of Mars. The lander consists of structure, ablation and heat protection material, a parachute for final descent, an impact absorption device, a capsule timer and sequencer, an orientation and erection device, power supply, a data handling system, an S-band telecommunications system with a FSK low-bit-rate backup, a high-gain antenna system with erection and Earth-tracking capabilities, and a scientific payload.

The capsule enters the atmosphere and is slowed by its retardation subsystem. After impact, the capsule discards the remainder of its retardation and impact attenuation system, orients itself, and erects its high-gain antenna and omnidirectional antenna.

All lander systems are then activated and the system transmits science data over the high-gain antenna system until failure. For certain types of failure that incapacitate only the primary telecommunications system or high-gain antenna system, the backup FSK system is assumed to transmit at a low-bit-rate over an omnidirectional antenna. The direct link lander's operations are independent of the spacecraft bus as soon as it is separated.

*5.2.2.3 **Postseparation relay link mission description.*** After lander separation, the relay link lander performs identically (to the direct link lander) through entry, impact, and postimpact erection and deployment with two exceptions. It does not use a high-gain antenna system but communicates with an orbiter over an omnidirectional antenna that is erected to the local vertical only. Also, its telecommunications system does not necessarily operate at S-band, but on whatever frequency is selected for lander-orbiter communications.

After capsule separation, the spacecraft bus is injected into orbit to serve as a relay satellite. To do this, the bus retracts its solar panels and high-gain antenna, performs a retro maneuver, then redeploys its system and reacquires the Sun, Canopus, and Earth. The orbiter receives, stores, and transmits to Earth the data obtained from the lander. The relay link lander detects when the orbiter is visible and transmits the accumulated science data in response to an orbiter command.

The exact time of rising and setting of the lander (from the orbiter's frame of reference) is a random variable and, therefore, the orbiter has enough storage for long intervisibility times as they happen. Also, the relay link and lander data storage system are large enough to hold all the science data accrued when the time between intervisibility becomes large.

The basic orbiter attitude control will be similar to the *Mariner* spacecraft with proper changes to cope with the times of Sun or Canopus occultation. A quasi-omnidirectional antenna for lander–orbiter communications is mounted on a gimbaled platform that points toward the planet center. Orbiter–Earth communications are assumed to be conducted over the same high-gain antenna and S-band telecommunications system used during transit.

As in the direct lander, a backup communication link (using an FSK system transmitting over an omnidirectional antenna oriented only to the local vertical) is used to transmit low-bit-rate information from lander to Earth.

5.2.3 Necessary Subsystems for Major Events

Table 9 shows the subsystems considered necessary to support the mission sequences of Table 7. Combinations of the subsystem functions are assumed to satisfy all mission events. Where similar sets of subsystems are used in both systems, equivalent reliability is realized.

Table 9. *Voyager* subsystems and sources of reliability models

Direct link system	Reliability model source	Relay link system
Flyby bus		**Orbiter**
Thermal control	Passive; reliability assumed unity	Thermal control
Electrical power	Ref. 5 (PRC)	Electrical power
Guidance and attitude control (G&AC)	Ref. 5 (PRC)	Guidance and attitude control (G&AC)
Central computer and sequencer (CC&S)	Ref. 5 (PRC)	Central computer and sequencer (CC&S)
Propulsion	Ref. 5 (PRC)	Propulsion
Telecommunications (T/C)	Internal JPL document	Telecommunications (T/C)
	Ref. 3 (Avco)	Planetary horizontal platform (PHP)
Lander		**Lander**
Thermal control	Ref. 4 (GE)	Thermal control
Electrical power	Ref. 4 (GE)	Electrical power
Release mechanism[a]	Ref. 3 (Avco)	Release mechanism[a]
Stability system	Ref. 3 (Avco)	Stability system
Retardation system	Ref. 4 (GE)	Retardation system
Capsule timer and sequencer[b] (CT&S)	Ref. 5 (PRC)	Capsule timer and sequencer[b] (CT&S)
Ground orientation	Ref. 4 (GE)	Ground orientation
Data handling[c]	Ref. 5 (PRC)	Data handling[c]
Telecommunications (T/C)	Internal JPL document	Telecommunications (T/C)
Omnidirectional antenna	Parameterized	Omnidirectional antenna
High-gain antenna	Parameterized	

[a]If a portion of the release mechanisms are mounted on the flyby bus or orbiter instead of totally on the lander, as depicted here, there is no affect on the probability that the bus (or orbiter) and lander will separate.

[b]The CT&S parts were assumed to be identical to the *Mariner* C CC&S parts for this analysis.

[c]The lander data handling subsystem parts were assumed to be identical to the *Mariner* C data handling system parts.

Figures 12 and 13 show the subsystems needed to operate for successful accomplishment of each of the events in the direct and relay links sequences, respectively. Every solid lined subsystem must operate through the event for a successful event; every dashed lined subsystem is needed to perform adequately at some time before the event.

5.2.4 Parameters

Flexibility is incorporated into the analysis by the introduction of several parameters that modify the more uncertain parts of the model. Areas that are felt to be uncertain enough to warrant representation are: operational component failure rates, stored component failure rates, relay link complexity factor, relay lander power amplifier, relay omnidirectional antenna, direct lander high-gain antenna. These parameters are introduced in detail now.

5.2.4.1 Operational component failure rates. Because the basic mode of analysis uses an exponential failure model, an estimate of a range of representative failure rates is needed. Uncertainty in the failure rates causes a concomitant uncertainty in the reliability. To reflect this uncertainty, a parameter β is introduced to modify operational component failure rates.

5.2.4.2 Stored component failure rates. In general, a stored component does not have the same failure rate as an identical operational component. Therefore, a parameter α is introduced that adjusts the failure rate of stored components to a level more representative of their storage failure rate history.

5.2.4.3 Relay link complexity factor. Because the orbiter telecommunications subsystem carries added components to provide relay communications, there is a greater probability that the relay orbiter will fail during transit. This added potential degradation is represented by the reliability multiplier ν.

5.2.4.4 Relay lander power amplifier reliability. The relay and direct landers do not necessarily carry the same power amplifier. The difference in their reliability is represented by the factor μ, which multiplies the failure rate of the direct lander power amplifier.

5.2.4.5 Omnidirectional antenna reliability. This parameter reflects the uncertainty in the ability of the omnidirectional antenna erection mechanism to work on the Mars surface after landing.

5.2.4.6 Direct lander high-gain antenna reliability. A little understood factor in this analysis is the ability of the high-gain, erectable, steerable antenna to perform properly in an unknown environment. Unlike α, β, and μ (which represent failure rate multipliers), R_{HG} represents the reliability of the high-gain antenna system and is assumed to be independent of time.

5.3 Assumptions and Constraints

To mold the advanced planetary mission into a form that can be represented by a mathematical model, a number of assumptions are made and constraints imposed. The following paragraphs detail each of these assumptions and constraints.

5.3.1 Functional Block Spacecraft Configuration

The advanced planetary spacecraft is assumed to be broken into functional blocks, each of which is operationally independent. The components in each functional block are assumed to be serially connected unless otherwise noted; the failure of any component results in total mission failure unless a redundant functional path is provided.

5.3.2 Failure Model

With the exception of the direct lander high-gain antenna system, all electronic components are assumed to obey a random failure model. In this model it is assumed that the probability of any given component failing in a fixed time interval is constant. One-shot components such as pyrotechnics and mechanical devices operating for a short time are assumed to obey the binomial probability law. (The mathematics used to produce the failure model are described in Appendix A. Appendix B gives the failure model for each of the subsystems.)

Component failures are assumed to be catastrophic; no marginal failure cases are studied. Once a part fails it is assumed to remain so for the length of the mission. No partial mission successes are considered except the acquisition of low-bit-rate information if any part of the primary telecommunication system fails.

5.3.3 Similarity of Subsystems

Whenever subsystems in both the relay and direct systems have enough common functional requirements, their designs are assumed identical for this reliability analysis.

5.3.4 Science Subsystems

Reliability figures are computed exclusive of science subsystems.

5.3.5 Sterilization Requirements

The reliability analysis of the lander system does not includes the effects of sterilization requirements.

5.3.6 Launch Operations

The reliability figures are calculated assuming that the spacecraft has been successfully launched and injected. All probabilities given are then contingent on a successful completion of this phase of the mission.

5.3.7 Trajectory Accuracy Requirements

The trajectory accuracy requirements from launch through lander separation are assumed identical. It is assumed that only Earth-based radio tracking is necessary for trajectory correction; no onboard guidance system reliability is considered.

5.3.8 Lander Reliability

It is assumed that there are only two basic areas of difference between the lander systems: the primary radio systems and the transmitting antennas. Differences in other equipment for the two landers are not considered.

5.3.9 Lander Configuration

The weight, size, shape, and other physical characteristics of both the direct and relay landers are assumed to meet all necessary constraints. For the direct lander, therefore, it is assumed that any high-gain antenna configurations are compatible with the operation of all other subsystems. Also, the application of surplus payload capability to the improvement of reliability through redundancy is not considered.

5.3.10 Degraded Communications

Both the direct link and relay link landers are assumed to carry a backup S-band FSK communications system transmitting over a quasi-omnidirectional antenna that must be erected to the local vertical. This backup system is assumed to work completely independent of the main communications system.

5.3.11 Relay Orbiter

Any reliability degradation due to orbital trim maneuvers is not considered.

5.3.12 Design Adequacy of Subsystems

It is assumed that all spacecraft subsystems are designed well enough to meet their design objectives.

5.3.13 Support System

The DSN equipment, procedure, and personnel are assumed to be perfectly reliable and, therefore, do not degrade the outcome of the mission in any manner.

5.4 Results

5.4.1 Effect of Parameters

The following sections discuss the effect of exercising each of the parameters introduced into the model in Section 5.2.4. Results for all parameter variations exercised can be found in Appendix C. The effects of the seven parameters are detailed in the following paragraphs.

5.4.1.1 Effect of trip time. In Fig. 14, the typical effect of trip time can be seen. The relay link's reliability is 0.95 times the reliability of the direct link because of the assumption of a more complex telecommunications subsystem. At approximately 200 days there is a sharp decrease due to the reliability of the functions corresponding to the landing phase. This comes from the failures that occur from capsule separation, entry, landing, and first activation of systems stored since launch.

The transit reliability function of the 400 day trajectory closely follows the 200-day trajectory until the separation sequence is started. A monotonically decreasing trend continues for the 400-day trajectory until the time when its separation sequence begins. Notice that the sharp degradation after 400 days of transit is larger in the relay link mission than its 200-day counterpart. This increase comes from the requirement placed on the relay link system to delay its retro maneuver and to store a planetary horizontal platform and lander system for an added 200 days.

5.4.1.2 Effect of operational failure rates. The parameter β is used to modify the operational failure rates. A β of 1.0 corresponds to the first estimates of *Mariner R* failure rates and is the most pessimistic estimate of β to date. The current estimate is approximately 0.19; the range in the analysis goes from 1.0 to 0.01, which may be representative of failure rates in the future.

The effect of β is illustrated in two distinct ways. Figure 15 illustrates the effect of the assumed operational

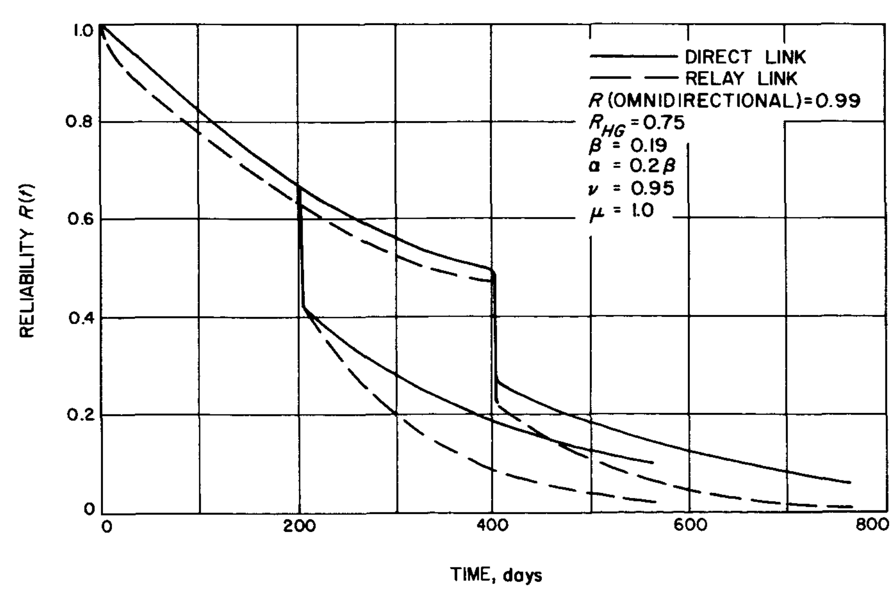


Fig. 14. Effects of flight time on reliability

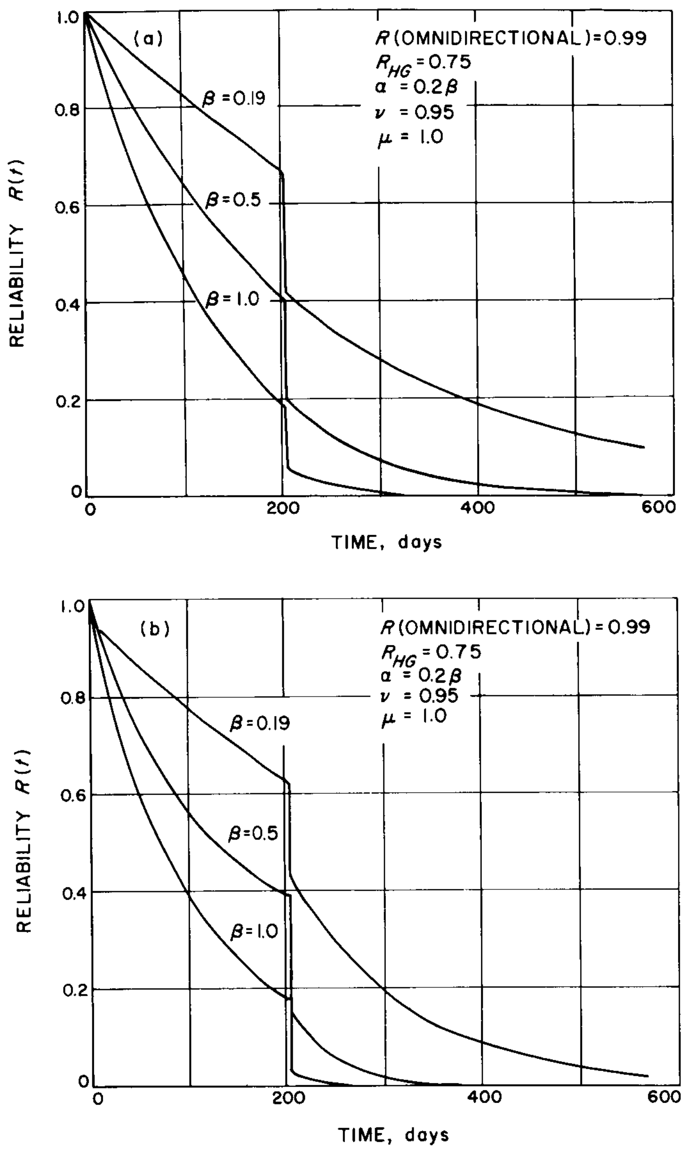


Fig. 15. Effects of operational failure rates: (a) on direct link reliability, (b) on relay link reliability

failure rates on the direct and relay link reliabilities; three distinct values of β were chosen so its influence on the total mission can be seen. It should be noted that a constant change in β produces a greater change in the relay link liability than the direct link reliability. Therefore, higher operational failure rates tend to favor the direct link mission over the relay link mission even though the absolute reliabilities decrease with increasing failure rates. The physical reason for this effect is because the more complex lander–orbiter combination used by the relay link mission is degraded more by poor failure rates than the direct link configuration.

In Fig. 16 the effect of β at certain selected mission times is shown. A remarkable aspect of Fig. 16 is that the reliability functions cross at a particular value of β. As the mission time progresses, this crossover value of β becomes smaller. This happens because it is assumed that the direct link high-gain antenna system reliability

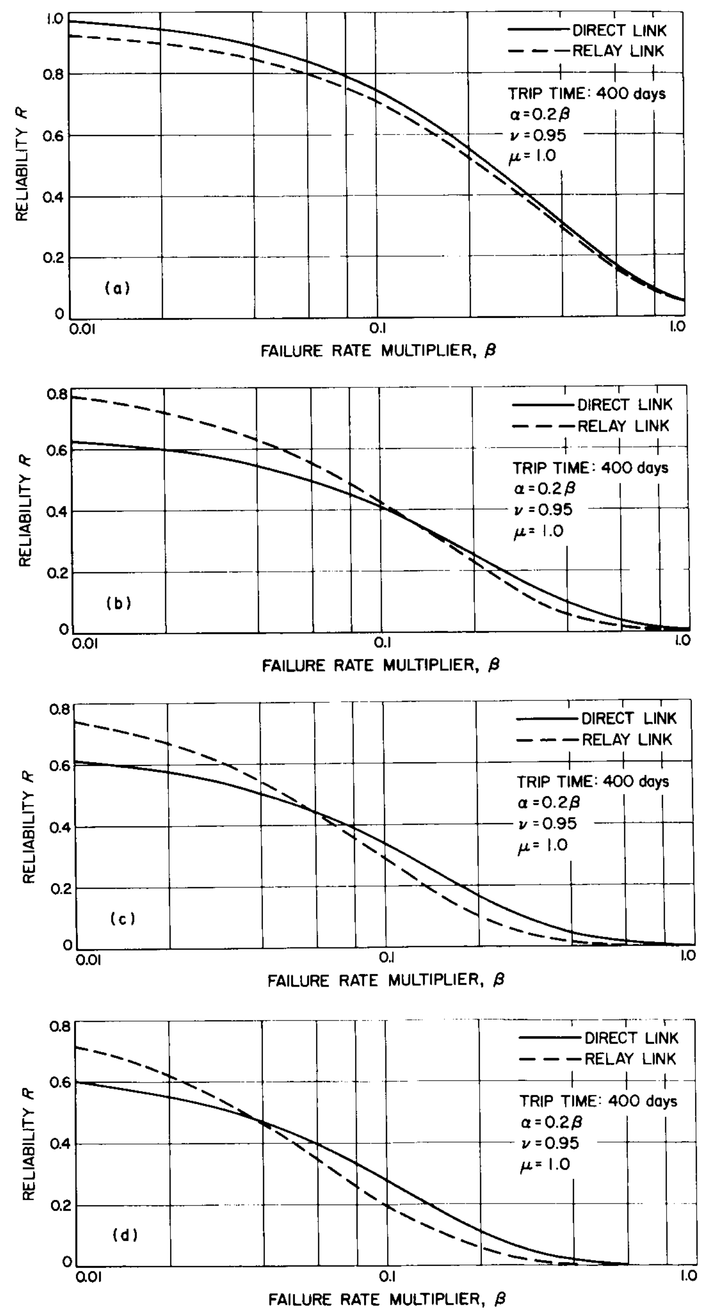


Fig. 16. Reliability of system vs failure rate multiplier: (a) before capsule separation, (b) after 1 day surface operation, (c) after 3 mo surface operation, (d) after 6 mo surface operation

is independent of time, so the direct system reliability asymptotically approaches the high-gain antenna system reliability as β goes to zero; while the relay link system reliability goes asymptotically to unity.

*5.4.1.3 **Effect of storage failure rates.*** The parameter α was used to see the effect that storage failure rates have on the probability of mission success. The highest chosen value of α was 1.0, corresponding to storage failure rates equal to operational failure rates. It was felt that this is a proper upper limit for α because, in general, stored components do not fail at a faster rate than operating ones. An α of zero would correspond to no degradation to components during storage, and is a suitable lower bound.

The effects on reliability that storage failure rates produce are shown in Appendix C. Portions of this data are plotted in Fig. 17, which illustrates the effect of α on the direct and relay link reliabilities. It should be noted that a constant change in α produces a greater reliability change in the relay link than in the direct link reliability. Therefore, higher storage failure rates relatively favor the direct link over the relay link mission even though the absolute reliabilities decrease with increasing α.

The physical reason for this effect is attributed to the requirement for a terminal retro maneuver and storage of the planetary horizontal platform.

*5.4.1.4 **Effect of relay link complexity.*** Because the relay link orbiter telecommunications subsystem is potentially more complex than its direct link counterpart, the relay link system is perhaps more prone to fail during transit. Both were considered to have identical parts and functions during transit; however, the relay link telecommunications subsystem may carry added components for conducting lander–orbiter communications after arrival at Mars. Therefore, the parameter ν represents the added complexity of the relay link telecommunications subsystem.

A value of $\nu = 0.95$ was arbitrarily chosen. It has the effect of degrading all results pertaining to the relay link by 5%. To get results for different values of ν, multiply any relay link results given in this analysis by $\nu/0.95$, because ν is purely a linear multiplier.

*5.4.1.5 **Effect of relay lander power amplifier reliability.*** Because the relay link lander might use a more reliable lander–orbiter power amplifier design in its telecommunications subsystem, the parameter μ was introduced. A value of $\mu = 1.0$ corresponds to the relay link power amplifier having the same failure rate as the S-band power amplifier in the direct link lander. When μ was reduced to 0.2, showing that the relay link's power amplifier's failure rate is estimated at only one-fifth that of the S-band power amplifier, essentially there is no effect on the mission reliability. This can be confirmed by comparing the relay link result in Table C-1 with those in Table C-8, C-5 to C-9, and C-7 to C-10, where all assumptions in each pair of tables (with the exception of the value of μ) are identical.

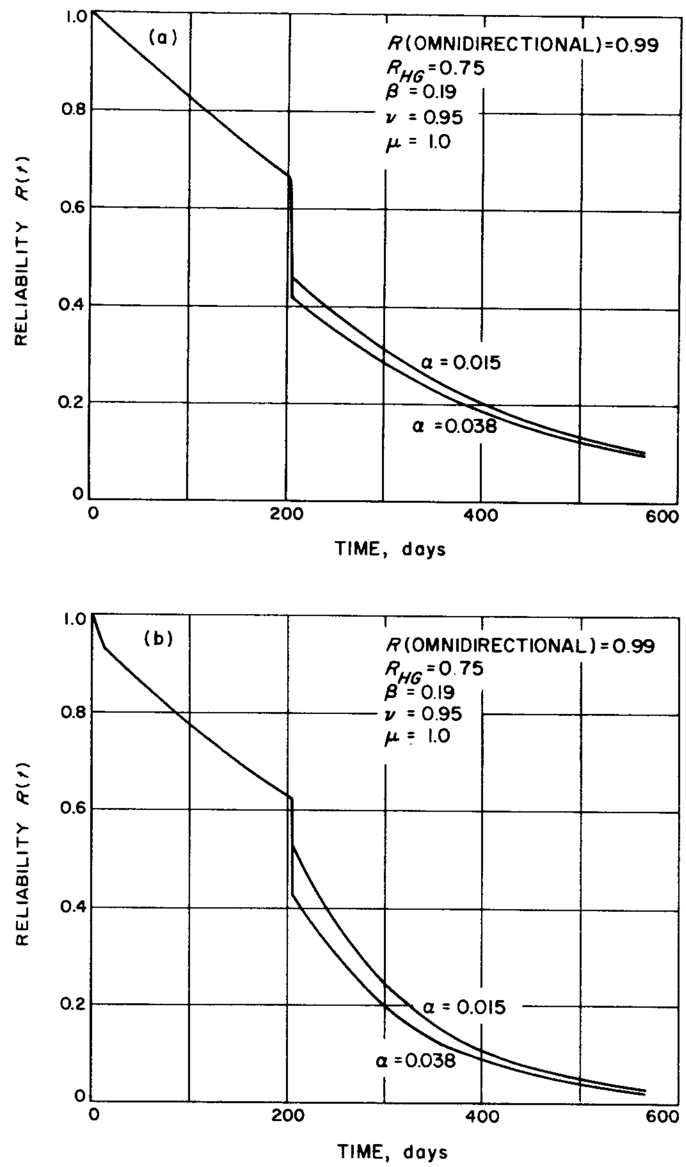


Fig. 17. Effect of storage failure rates: (a) on direct link reliability, (b) on relay link reliability

5.4.1.6 Effect of omnidirectional antenna reliability. Because the relay lander omnidirectional antenna must operate for the lander to communicate with the orbiter, its reliability multiplies the reliability of the relay link primary communications system. This effect can be noted in Tables C-1 through C-11.

5.4.1.7 Effect of direct link high-gain antenna reliability. The reliability of the direct link is directly proportional to the reliability of the high-gain antenna system. Its reliability is a measure of its ability to withstand impact, erect itself in an uncertain environment, and detect and track the Earth daily. Tables C-1 through C-11 show the reliability of direct link for three values of the reliability of the high-gain antenna system.

5.4.2 High-Gain Antenna Requirements for Equivalent System Reliability

Because the high-gain system reliability is a major question, it is instructive to consider what requirement should be placed on it to get equivalent relay and direct link reliability. This condition may be calculated analytically by forming the ratio of the relay system reliability to the direct system reliability. In this ratio, the direct system reliability is calculated exclusive of the high-gain reliability. The validity of this approach can be seen by considering the conditional reliability function:

$$R_{DL} = R_{DH} \times R_{HG}$$

where

R_{DL} = direct link reliability

R_{DH} = direct link/high-gain antenna works reliability

R_{HG} = high-gain antenna reliability

The high-gain antenna system reliability that equates the relay and direct link systems reliabilities is therefore given by:

$$R_{HG} = \frac{R_{RL}}{R_{DH}}$$

where

$$R_{RL} = \text{relay link reliability}$$

Note, the denominator is exactly equivalent to calculating the direct link reliability, assuming that the reliabilities of the high-gain antenna system are unity. Remember this function is sensitive to the assumptions made in the analysis.

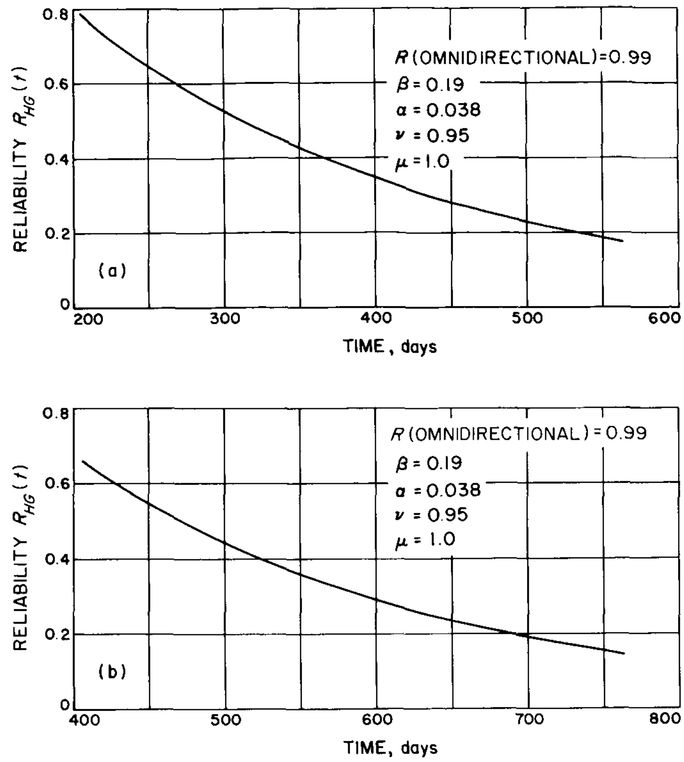


Fig. 18. Minimum high-gain antenna reliability function for direct link reliability ≥ relay link reliability: (a) for 200 day transit, (b) for 400 day transit

In Fig. 18a, the function that corresponds to the assumptions and results of Table C-1 can be found. In Fig. 18b, the function that corresponds to the assumptions and results of Table C-2 is illustrated. The only difference in the assumptions between these two tables is that one refers to a 200-day transit and the other refers to a 400-day trajectory. Note, by comparing Figs. 18a and 18b, that longer trip times reduce the minimum reliability requirement of high-gain antenna; longer trip times cause the direct link system's reliability to improve relative to the relay link system even though both absolute reliabilities drop. Also, the reliability requirement on the antenna system is quite low, never exceeding 0.8 at landing, and dropping to 0.4 or less at 6 mo.

5.5 Summary

Of the several parameters introduced, two seem to dominate the effect on reliability; the failure rate multiplier β, and the reliability of the direct link high-gain antenna system. The failure rate multiple has the greatest effect on the relay link and the high-gain antenna system is the largest contributor to direct link unreliability.

For the assumed value of β; ($\beta = 0.19$) the maximum reliability requirement on the high-gain antenna system (for equivalent direct and relay reliability) is less than 0.8. This requirement is a function of both flight time and the assumed failure rate. Lower failure rates tend to increase the reliability requirement on the high-gain antenna system. This is true because the more complex relay link benefits more by the lower failure rates. Longer flight times have an adverse effect on the relay link reliability; therefore, the high-gain antenna reliability requirement is lessened.

6. SYSTEM COMPARISONS

There are two general classes of comparisons that can be made from the preceding sections. The first class has the results that are direct consequences of (or observations from) the assumptions; they are given in Section 6.1. The second class, discussed in Section 6.2, has the analytic comparison functions: effective bit rate, reliability, and expected cumulative bits.

6.1 Observations

This section has material that, although pertinent to the comparison of the relay and direct mission approach, is not embodied in the mathematical comparisons.

6.1.1 Data Requirements

The design requirements of both the direct and relay missions are dictated largely by the data rate needed to transmit 10^{10} bits of data within a 6-mo mission lifetime. It is instructive to consider the effect on the design of a reduction by a factor of ten in total data needed. A direct result is the reduction in power-gain product by 10 db, making it possible to either reduce the transmitted power or to reduce the high-gain antenna size. Because there is much uncertainty in the direct lander high-gain antenna system, it is likely that the antenna size would be reduced. For antennas below 8 ft in diameter, an active RF pointing system to reduce pointing losses is no longer needed. Therefore, the reduction of the data requirement by a factor of ten allows passive pointing, and gives a potential gain in reliability. Also, it is likely that smaller antennas yield lander designs less sensitive to surface uncertainties and can be made to survive more stringent landing environments. To land a large ($\sim$ 12 ft) high-gain antenna system with a high probability of survival, it is possible that the method of descent would be limited to those needing active retro-propulsion near the surface. A reduction in antenna size makes alternate descent schemes, such as parachutes, more attractive.

Although the relay orbiter also benefits from the reduction in data requirement, the radio steerable antenna is not subjected to the environmental uncertainty of the direct link; therefore, the total reliability is affected less. The relay approach still has the same problems of lander–orbiter link design; for example, automatic acquisition and the planet-oriented antenna. Therefore, it is probable that more uncertainty is removed from the relay approach, showing that lowering data requirements favors the direct lander approach. This is certainly true in the limit when the needed data rate can be supported by low powered transmitter and a nonsteerable antenna; because, here, the orbiter is not useful as a relay satellite and merely adds to the unreliability of the mission.

6.1.2 Hardware Development

It is apparent that there is extensive hardware development to be undertaken for either approach.

A broad categorization of development effort results in the following observations. The direct link requires the development of a large steerable high-gain antenna system and a relatively high-power S-band transmitter that are capable of surviving the landing shock. The lander design configuration will be dictated to a large extent by the requirement of the antenna system. It is possible that an active descent system such as propulsion and descent control might be needed to ensure a high probability of an upright attitude during and after landing. Also, it may be necessary to get detailed information about the Mars surface before committing to a higher data-rate direct link. Therefore, the hardware development of an eventual advanced planetary direct link mission may be keyed to the acquisition of surface data characteristics.

The hardware development for the relay link is characterized by the development of two major systems, the lander and the orbiter, which must be RF compatible.

The relay link lander is not as open to surface uncertainties as its direct link counterpart, because transmission is made without the use of high-gain directive antennas. However, there is some possibility that the relay link development schedule will be dependent on more definite information on the Martian upper atmosphere because the altitude of the relay satellite orbit is limited by a noncontamination constraint. First calculations (see Section 6.2.1.1) show there is some latitude in selection of orbits that have enough communications capability; therefore, this constraint is not restrictive. Data from the *Mariner C* occultation experiment should resolve this uncertainty.

6.1.3 Trajectory Dependent Parameters

It is worthy to note that the flight time for the Type II trajectories in 1975 and 1977 is on the order of 400 days, showing that a premium is placed on reliable long-life operation. Because the relay approach is more complex on a part-count basis, this observation tends to favor the direct lander.

If Type I trajectories are attempted in the 1975 and 1977 opportunities, the injection energy requirements dictate a larger vehicle than now programmed to inject a combined lander–orbiter. Also, the fuel needed to place a relay satellite in a favorable communication orbit is higher for the Type I trajectories, causing an increase in the satellite weight. Thus, an orbiter–lander on a Type I trajectory demands more launching vehicle capability than now planned, and a Type II trajectory results in a reliability penalty.

Therefore, from the standpoint of trajectory considerations, direct landers seem to be favored for both 1975 and 1977. The 1975 Type II arrival is almost diametrically across the solar system from the Earth, so the Earth–Mars–Sun angle is only 10.5 deg at encounter and drops to 0.5 deg after 58 days. This will present a serious solar noise problem; one possible solution might be to fly a higher-energy trajectory that gives an earlier Mars encounter date. However, the higher energy requirement again tends to favor the direct lander approach. This condition does not exist for the 1977 encounter.

6.1.4 Limiting Link Relay Operations

Some general observations about the extremes of operation of the relay link may be made using the definition of the effective bit rate. First, note that the relay link operates most efficiently when the unconstrained effective bit rates of the two legs are equal. In any event, the actual bit-rate capacity of the relay link is dictated by the lowest capacity leg. A mathematical statement of this is:

$$\dot{I}_{e_{RL}} = \min\left(\dot{I}_{e_{LO}}, \dot{I}_{e_{OE}}\right) \tag{3}$$

The regimes of operation of the relay link are seen to fall into two categories from Eq. (3); either the lander–orbiter leg is saturated and the total effective bit rate is equal to $\dot{I}_{e_{LO}}$, or the orbiter–Earth leg is saturated and the total effective bit-rate is equal to $\dot{I}_{e_{OE}}$. Because each of these circumstances present interesting and distinct properties, let us consider them further.

6.1.4.1 Orbiter–Earth leg saturated:

$$\dot{I}_{e_{LO}} > \dot{I}_{e_{OE}}$$

When the orbiter-to-Earth link is saturated, alternate orbits can be chosen that decrease the lander–orbiter viewing time without decreasing the overall lander data retrieval capability of the relay link. This implies that, if the original orbit were chosen at the expense of orbit science, one is free to choose a new orbit more favorable to the orbit science data. This effort is in vain, however, because (by the original hypothesis) the orbiter-to-Earth leg is saturated, and the orbit science data could not be returned anyway. Therefore, the region of operation where the orbiter–Earth leg is saturated is not conducive to the gathering of orbit science data.

One further observation may be made if it is assumed that the lander–Earth direct link and the orbiter–Earth link have equivalent power-gain products. The equation defining the effective bit rate of the direct link is:

$$\dot{I}_{e_{DL}} = \dot{I}_{i_{DL}} \cdot v_{DL}$$

and a similar equation defines the relay link effective bit-rate.

$$\dot{I}_{e_{RL}} = \dot{I}_{e_{OE}} = \dot{I}_{i_{OE}} \cdot v_{OE}$$

Because the orbiter–Earth link is saturated, the effective bit-rate is given by the orbiter–Earth effective bit-rate. Now, by assumption of equal Mars–Earth power-gain product,

$$\dot{I}_{i_{OE}} = \dot{I}_{i_{DL}}$$

It can be seen that the effective bit rates of the direct link and the relay link are related solely by the ratio of the view fractions.

$$\dot{I}_{e_{DL}} = \frac{v_{DL}}{v_{OE}}\,\dot{I}_{e_{OE}}$$

The best view fraction of the direct link is about one-half, with a reasonable value being about one-third. The maximum view fraction of the orbiter is unity because it is likely that an orbit that does not occult the Earth can be chosen (see Section 4.2.2). Therefore, an upper bound on the ratio of effective bit-rates of about three can be derived for the relay link. Note, this ratio of three rests on the assumption that the power-gain product of the direct link and the orbiter–Earth link are equal. Furthermore, this ratio is a function solely of the relative geometries of the relay and direct missions.

6.1.4.2 Lander–orbiter leg saturated:

$$\dot{I}_{e_{LO}} < \dot{I}_{e_{OE}}$$

Here, any change in the selection of the orbit other than the one yielding optimum lander–orbiter communications will result in further degradation of the relay link capacity for lander data transmission. However, if there are some orbit science experiments that can accommodate the low-altitude and low-inclination orbit that is best for communications, their data can be carried on the orbiter–Earth link.

Assuming equal power-gain products as in Section 6.1.4.1, when $\dot{I}_{e_{LO}}$ drops to one-third $\dot{I}_{e_{OE}}$ the two approaches are equivalent in lander data transmission capacity. The lander data transmission capacity of the relay link can drop below the direct link capacity only when the lander–orbiter link is the saturating link.

6.2 Mathematical Comparisons

This section is to evaluate representative systems for their effective bit rates, expected cumulative bits, and storage requirements. For the relay link, three possible orbits have been chosen and their communications capability analyzed. Expected cumulative bit calculations have been made for both systems with reliability as a parameter. An approximate analysis of the data storage requirements for both systems is given in Section 6.2.3.

6.2.1 Effective Bit-Rate Calculations

6.2.1.1 Relay link.

6.2.1.1.1 Lander-to-orbiter. Three particular orbits have been chosen for a detailed study of their information exchange characteristics. The first two of these are desirable from a relay communications standpoint and are circular with altitudes of 5000 and 2000 km. The third, a 4000- by 20,000-km orbit, is considered for nonstandard orbit injection, or possible utility for scientific measurements.

6.2.1.1.1.1 Communications orbits. The two orbits selected correspond to the lowest altitude circular orbits allowable by two proposed noncontamination constraints (see Section 4.1.2.1). They are both at an inclination of 40 deg so that neither occults the Earth on the first orbit using the nominal approach conditions (see Table 6). Therefore, for a time at least, uninterrupted transmissions can be made to the Earth without time losses due to occultation and relock.

The fractional viewing time v, the largest maximum possible lander–orbiter range, and the angle δ (Fig. 8) are plotted vs the elevation angle γ for the 5000- and 2000-km orbits in Fig. 19. Figure 20 shows elevation angle vs time past arrival for the 5000-km orbit and the nominal 1975 approach conditions.

Optimum transmission policies for these two orbits have been developed using the method outlined in Section 6.1.4. It is assumed that the satellite has an antenna pointed at the planet center with a cosine-squared radiation pattern. The angle off axis for the orbiter antenna, δ, for a given elevation angle has been shown in Fig. 19. In Case A the lander has a similar antenna whose gain is proportional to $\cos^2(90\text{-}\gamma)$. Case B will treat a lander with an ideal turnstile over ground plane at height $0.35\,\lambda$ where λ is the wavelength, and Case C an ideal turnstile over ground plane at height $0.45\,\lambda$. Cases A, B, and C have progressively wider radiation patterns on the ground antennas. Figure 21 shows the radiation pattern of the transmitting antenna for Cases A, B, and C. Optimization of the transmission policies has been carried out using Eq. (2) and assuming the communications parameters listed in Table 2 and the bit-rate vs range plot of Fig. 1. Recall the Eq. (2) assumed that bit rate is proportional to lowest received power. The results of the calculations for the two orbits under consideration are given in Table 10 assuming the transmitter power is 10 w.

Several general properties of the optimum lander–orbiter communications policy are revealed in Table 10. Note, the view fraction tends to become larger for higher altitude orbits. At the same time, the communication range becomes larger resulting in lower effective bit-rate capability for higher orbits. It can also be seen that the general tendency is to transmit over a smaller angle for the higher orbits, showing that more directional antennas are better.

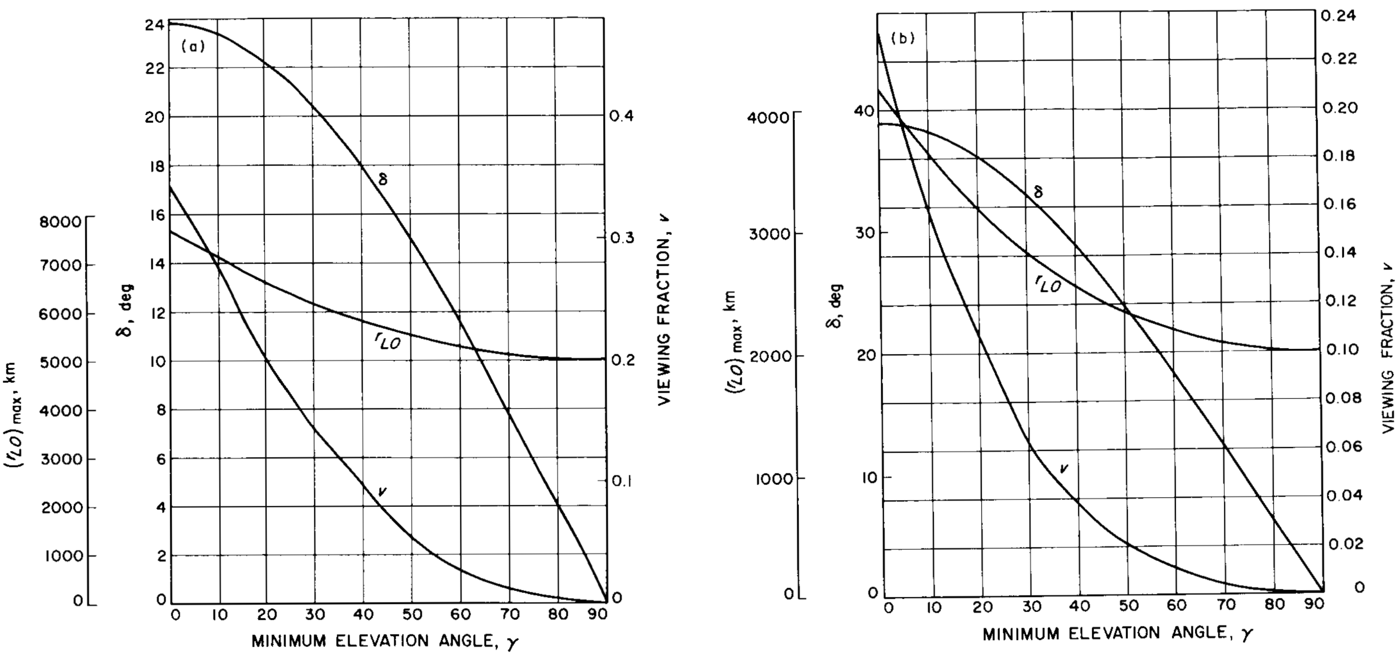


Fig. 19. Geometric quantities vs minimum elevation angle: (a) 5000 km orbit, (b) 2000 km orbit

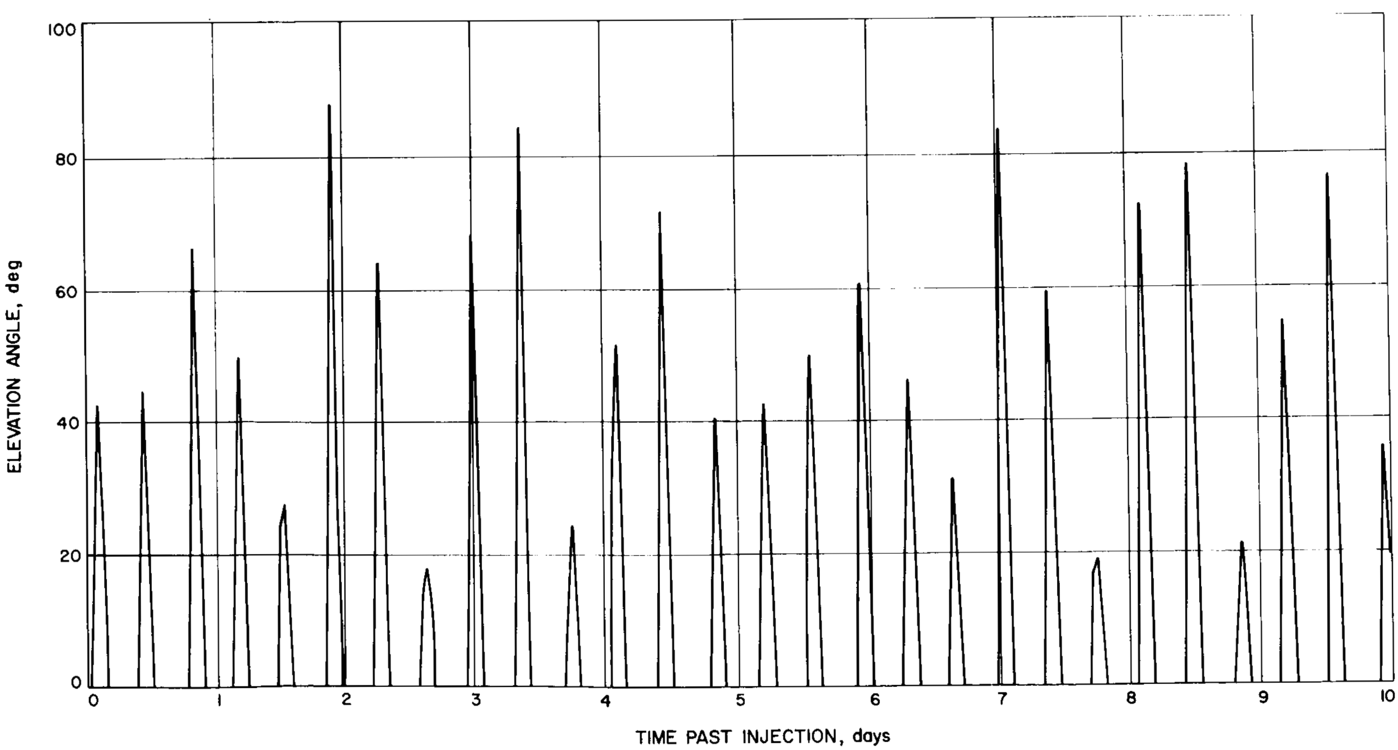


Fig. 20. Elevation angle vs time for 5000 km circular orbit

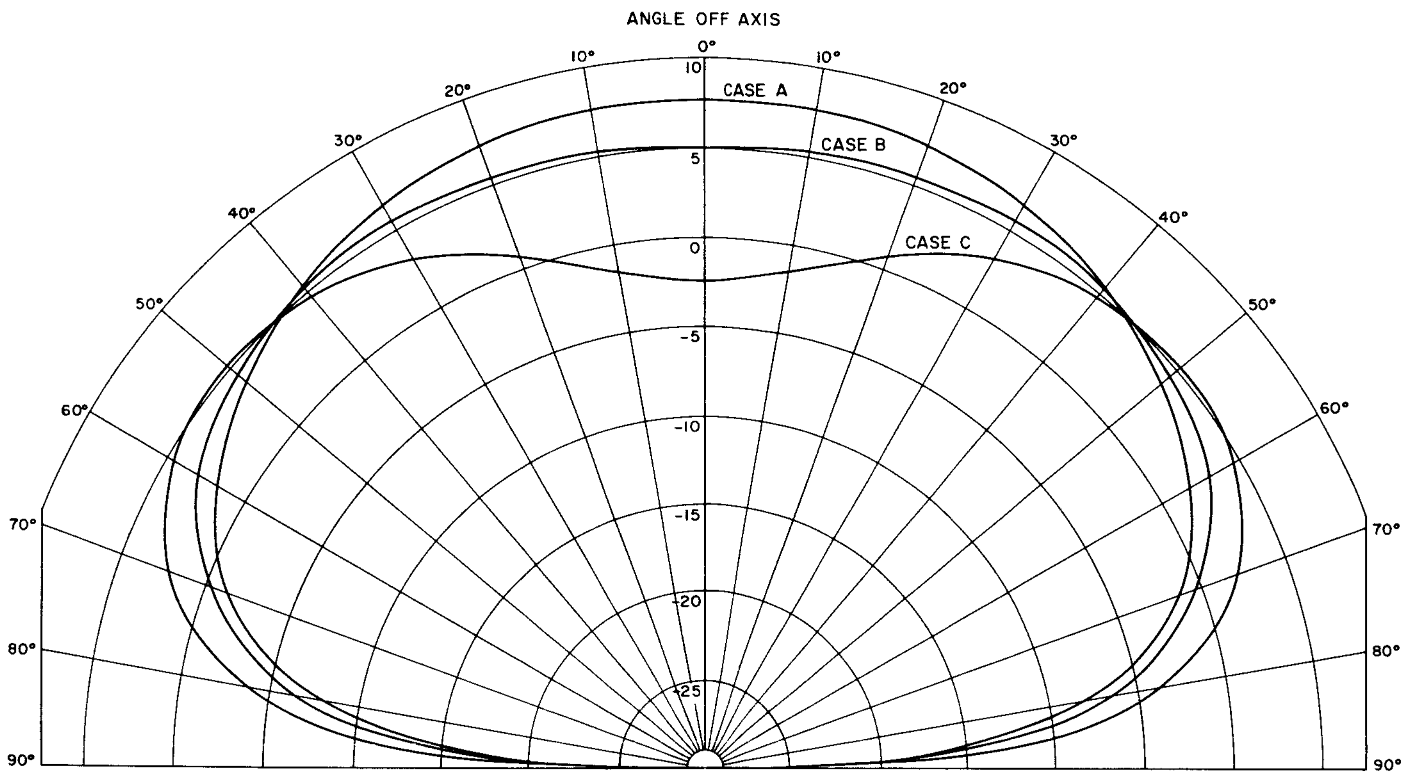


Fig. 21. Antenna radiation patterns (gain over isotropic, db)

Table 10. Effective bit rates for 10-w transmitter and 40-deg inclination orbit

Orbit	Case	ψ_m, deg	δ, deg	$v(\gamma = 90 - \psi_m)$	$G_T(\psi)$, db	$G_R(\delta)$, db	(r_{LO}) max, km	I_i, bits/sec	I_e, bits/sec
5000 km circular	A	50	18	0.096	3.9	7.4	5800	28,900	2780
	B	54	19	0.118	4.1	7.3	5950	28,300	3340
	C	70	22	0.202	2.1	7.1	6000	12,300	2480
2000 km circular	A	43	25	0.023	5.1	6.9	2360	210,000	4830
	B	51	29	0.038	4.5	6.6	2560	143,000	5450
	C	65	35	0.084	3.6	6.0	2990	72,900	6120
4000 × 20,000 km	A	41	5.5	0.099	5.3	7.7	20,700	3470	343
	B	48	6.2	0.124	4.6	7.7	21,000	2930	364
	C	68	7.8	0.212	2.8	7.7	21,900	1430	304

Conversely, lower altitude orbits would need broader beam antennas and transmission over a larger angle.

It is interesting to note that the effective bit rates for a given orbit are nearly equal when the antennas are used optimally. Suboptimal use degrades the results considerably. Had the cosine-squared antenna of Case A and the 5000-km orbit, for instance, been used to transmit down to 20 deg off the horizon ($\psi = 70$ deg), then its effective bit rate is only 1200 bits/sec.

6.2.1.1.1.2 4000-×20,000-km orbit. An analysis similar to the one mentioned has been run on an elliptical orbit. This orbit has the following elements:

a	15,380 km
e	0.52
i	30 deg
ω	−90 deg
h_p	4000 km
h_a	20,000 km
periapsis	30°S latitude

Using the three antennas of Cases A, B, and C mentioned, an optimization was made; the results are given in Table 10.

6.2.1.1.2 Orbiter-to-Earth. The set of orbits chosen have the property that line-of-sight communications with the Earth can be maintained continuously for at least several days after injection. Therefore, the viewing fraction for all of these is unity and the effective orbiter-to-Earth information rate is the same as the instantaneous information rate. The range is about 350 million km. Assuming that the transmission system on board the satellite has the parameters shown in Table 3 and a 50-w transmitter, the information rate (from Fig. 2) is 4800 bits/sec. Note, for a 5000-km circular orbit around Mars (Table 10), the lander–orbiter and the orbiter–Earth effective information rates are matches if the lander has a 10-w transmitter and the satellite has a 35-w transmitter.

6.2.1.2 Direct link. The direct link range and viewing characteristics have been discussed in Section 4.2, where a lander at 10° North latitude may view the Earth for approximately 8.4 hr per Martian day, or one-third of the time. The Earth–Mars range is about 350 million km. Assuming that the lander antenna transmission system has the same parameters as those of Table 3, then the effective information rate for a 50-w transmitter is 1600 bits/sec. Note, for this direct link to have the same effective information rate as the 5000-km orbit relay link cited earlier, it must have a transmitter power of 100 w. The comparison is shown in Table 11. Note further, that an effective bit-rate of 3340 bits/sec is enough to return the needed data over the time of the mission. Table 12 shows the effective information rates and total accumulated bits over 6 mo for equal Mars-to-Earth S-band power for the direct and relay links.

Table 11. Necessary transmitter powers for $\dot{I}_e$ of 3340 bits/sec, w

Direct link	Relay link 5000 km, 40-deg inclined circular orbit	
Lander	Lander	Satellite
100	10	35

Table 12. Effective information rates and total bits for 35-w S-band transmitters

System	Bit-rate, bits/sec	Total bits
Direct link	1110	1.7×10^{10}
Relay link: 5000-km circular orbit (10-w transmitter on lander)	3340	5.2×10^{10}

6.2.2 Expected Cumulative Bit Calculations

In this section, two derivations of the average amount of information expected from a mission will be given, numerical results will be shown, and the particular solutions of Table 12 will be treated.

Because, in Section 5, time-dependent reliability functions of both the direct link and relay link configurations were found, it is now possible to use them to derive the average total bits expected over missions of fixed duration.

6.2.2.1 Derivation of $\bar{I}(T)$. From Section 5, $R(t)$ was found to have the general shape of Fig. 22. Recall that $R(t)$ represents the probability that the system will work properly to time t. From the definition of $R(t + \Delta t)$, the difference, $R(t) - R(t - \Delta t)$, represents the probability that the system will fail in the interval, t to $t + \Delta t$. Letting Δt approach 0, and taking the limit, $-R(t)$ is obtained. Therefore, the probability that the systems will fail during the time interval dt is $-R(t)$ and this function is shown in Fig. 23.

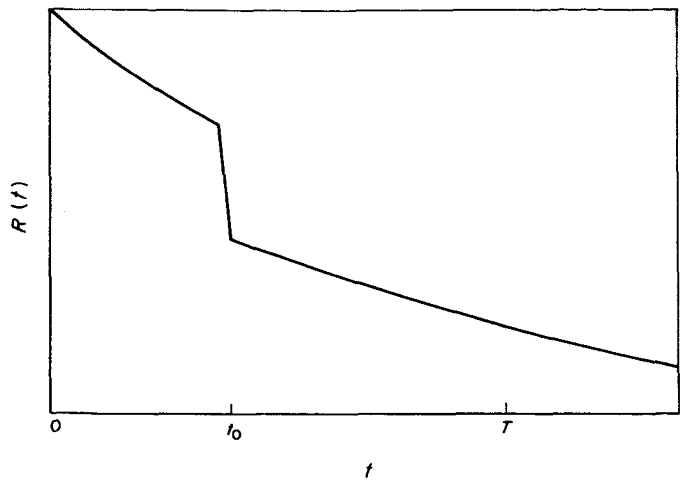


Fig. 22. Typical reliability functions

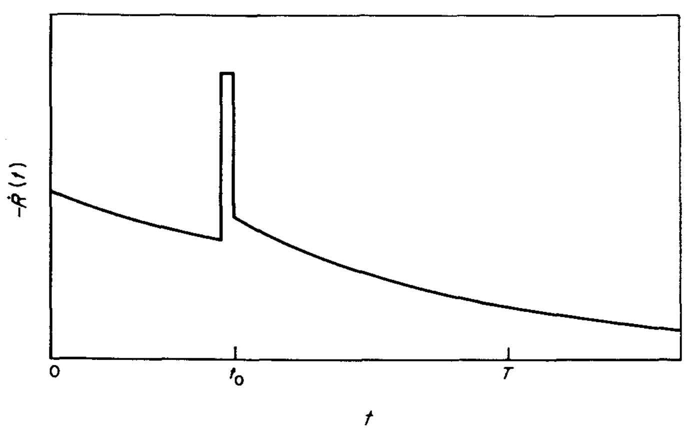


Fig. 23. Typical failure density function

The average information expected from a system transmitting data at a fixed rate of $\dot{I}_e$ is:

$$\bar{I}(T) = \int_{t_0}^{T} \dot{I}_e\,(t - t_0)\,(-\dot{R}(t))\,dt$$
$$+ \int_{T}^{\infty} \dot{I}_e\,(T - t_0)\,(-\dot{R}(t))\,dt \text{ for } t > t_0 \qquad (4)$$
$$= 0 \quad \text{for } t < t_0$$

where

$t = 0$ represents mission start

$t = t_0$ represents beginning of surface data accumulation

$t = T$ represents total mission time

The first term on the right side of the equation represents the average information obtained from that percentage of the systems that fail between t_0 and time T. The second term represents the information obtained from that percentage of the systems that work for the length of the mission. Not given in the expression is the average information obtained from the systems that fail before time t_0, because they accrue no information at all. This equation can be reduced to:

$$\bar{I}(T) = \dot{I}_e \int_{t_0}^{T} R(t)\,dt \qquad \text{for } T > t_0 \qquad (5)$$
$$= 0 \qquad \text{for } T \leq t_0$$

Another argument that yields the results of Eq. (5) follows. The expected increase in information between t and $t + dt$ is $d\bar{I}$ where:

$$d\bar{I} = \dot{I}_e\,dt\,R(t) \qquad \text{for } t_0 \leq t \leq T$$
$$= 0 \qquad \text{elsewhere}$$

In other words, the expected increase of cumulative bits is equivalent to the bit rate, times the increment of time over which the system transmits at this bit rate, times the probability that the system is working at this time. Solving for I:

$$\bar{I} = \int_{-\infty}^{\infty} d\bar{I}$$

$$\bar{I}(T) = \int_{t_0}^{T} d\bar{I}$$

$$\bar{I}(T) = \dot{I}_e \int_{t_0}^{T} R(t)\,dt\,, \qquad \text{for } T > t_0$$
$$= 0\,, \qquad \text{for } T \leq t_0\,.$$

Note, as $T \to \infty$, $\bar{I}(T) \to E[I]$, where $E\,[I]$ is the statistical expectation of information returned by a system allowed to transmit until failure.

6.2.2.2 Results. For every set of model parameters studied in Section 5, the normalized average information, $\bar{I}(T)/\dot{I}_e$ was computed. Results can be found in Appendix C. Note, $\bar{I}(T)$ is normalized for effective information rate given in bits/hr. To get $\bar{I}$ for a 6-mo mission, merely multiply the value corresponding to average accrued bits in 6 mo by the effective bit rate in bits/hr.

Figure 24 depicts results given in Tables C-1 and C-2. With everything else equal, this figure shows the effect of the trip time on the average information returned. Although the absolute sizes of the expected information decreases with increasing flight times, it can be noted that the ratio of expected information between the direct link and relay link systems favors the direct link approach as flight times are lengthened.

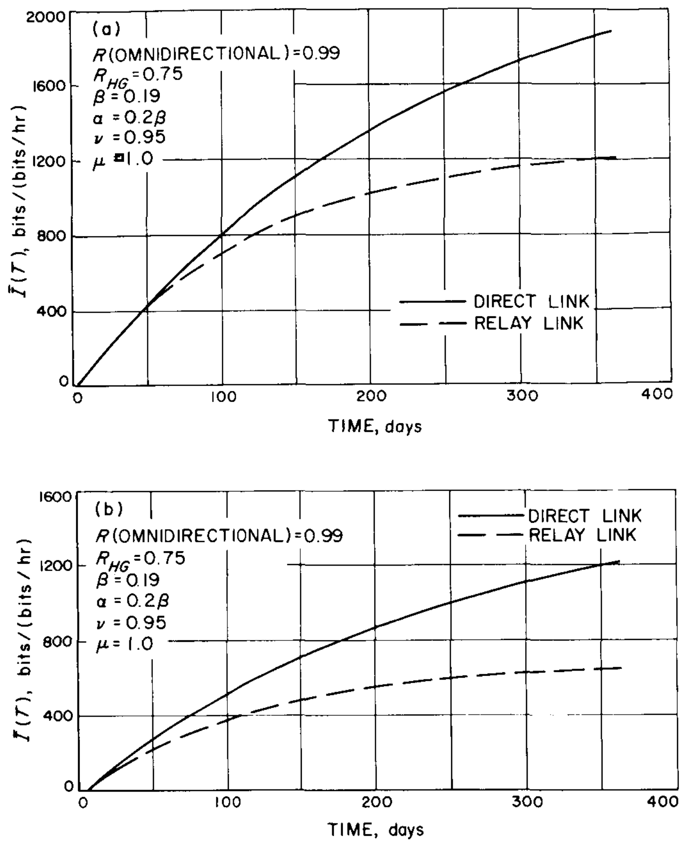


Fig. 24. Normalized expected information from N day surface mission: (a) for 200 day trajectory, (b) for 400 day trajectory

Another interesting result of the analysis is obtained from manipulating Eq. (5). From the discussion in Section 6.1.4, it can be said that $\dot{I}_e$ for the relay link will not go beyond $\dot{I}_e$ for the direct link by more than a factor of three for equal Mars-to-Earth radiated powers. Now the average information expected from a direct link mission can be equated to that expected from a relay link mission whose effective bit rate is n times that of the direct link. In this manner the minimum reliability function for the direct link high-gain antenna can be obtained. If the high-gain antenna designer can assure that its reliability function exceeds this lowest value that will be found, then the direct link will always yield a greater expectation of total bits than the relay link. From Eq. (5):

$$\bar{I}(T)_D = \dot{I}_{e_D} \int_{t_0}^{T} R(t)_D \, dt$$

$$\bar{I}(T)_R = \dot{I}_{e_R} \int_{t_0}^{T} R(t)_R \, dt$$

where:

$$\dot{I}_{e_R} = n \dot{I}_{e_D}$$

Equating $\bar{I}(T)_D$ to $\bar{I}(T)_R$ yields:

$$\int_{t_0}^{T} R(t)_D \, dt = n \int_{t_0}^{T} R(t)_R \, dt$$

or:

$$\int_{t_0}^{T} \left[R(t)_D - n\, R(t)_R \right] dt = 0$$

The only condition under which this expression holds for all $T > t_0$ is for:

$$R(t)_D = n\, R(t)_R$$

Now, if the time dependent reliability function for the direct link system is separated into two parts, $R_{HG}(t)$, which is the time dependent reliability function of the high-gain antenna, and $\hat{R}(t)_D$, which is the reliability function of the direct link system excluding the high-gain antenna:

$$R_{HG}(t) = \frac{n\, R(t)_R}{\hat{R}(t)_D}$$

This relationship is shown in Fig. 25 for 200- and 400-day trajectories. The middle curve of these two plots represents the smallest reliability function to which the high-gain antenna must be designed to make the direct link mission more favorable whenever both systems' effective bit rates are equal, i.e., $n = 1$. The upper curve represents this minimum function when $n = 3$, and the lower for $n = 1/2$. From the results of Section 6.1.4 it seems unlikely that n will be less than $1/2$; therefore, the direct link mission should not be considered if a high-gain antenna system cannot be made more reliable than the lower curve. If the high-gain antenna system reliability function falls anywhere within the shaded areas of Fig. 25, then further refinement of the relative effective bit rates is necessary before a conclusion can be drawn as to which mission is more favorable, based on expected information. One further comment is in order about Fig. 25. Because the reliability of the high-gain antenna system must exceed unity for the direct link to be the more favorable over certain surface mission times (all missions less than 208 days in the surface for a 200-day trajectory), this shows that missions of less than 7 mo should use the relay link approach to get more average total information. For surface missions greater than 7 mo, the direct link mission should be advocated whenever the high-gain antenna system reliability is shown to go beyond the upper curve.

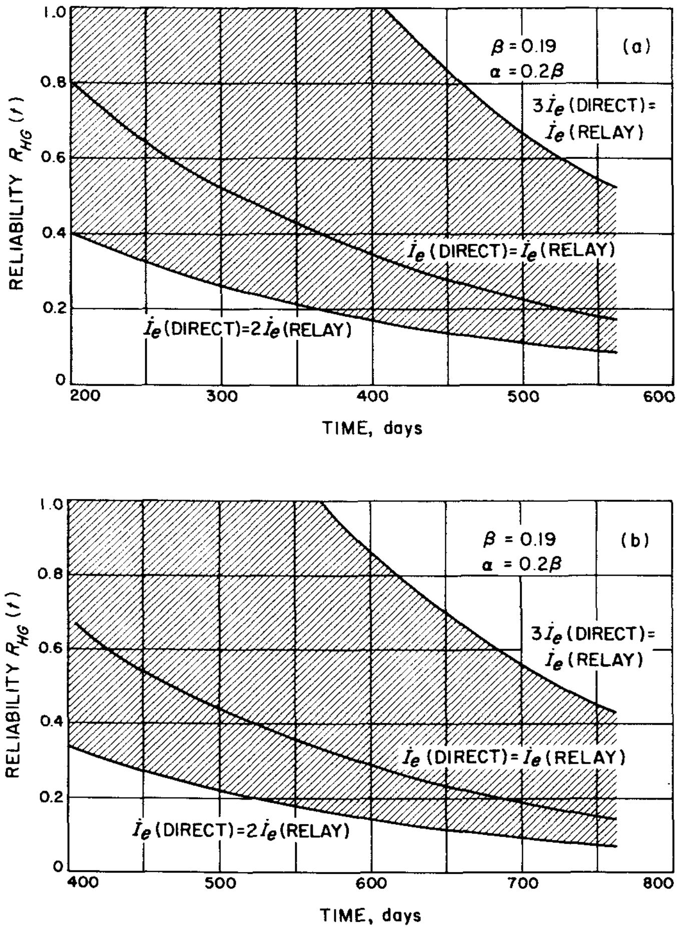


Fig. 25. Reliability goals for high-gain antenna: (a) for 200 day trajectory, (b) for 400 day trajectory

6.2.2.3 Sample calculations. Section 6.2.1 has given effective bit rates for a sample direct link system and three sample relay link systems (Tables 10 and 12). Tables C-1 and C-2 have given results of the expected cumulative bit calculations and these are repeated in Fig. 24. From these various data it is possible to calculate expected total information for several sample systems. Recall that the information in Fig. 24 is based on the assumption that the S-band amplifier on board the orbiter and the direct link lander are, from a reliability standpoint, identical. (The multiplier, $\mu_r = 1$.) Effective information rates for the direct link and a 5000-km circular orbit relay link using a 35-w, S-band transmitter are 1110 bits/sec (4.0×10^6 bits/hr) and 3340 bits/sec (12×10^6 bits/hr), respectively. The effective information rate for the relay link having a 20,000-km elliptic orbit is 364 bits/sec (1.27×10^6 bits/hr) for S-band Mars–Earth radiated power greater than 4 w. It should be noted that the information capacity of a 10-w relay link lander can be handled by the orbiter with only 4 w radiated power for this particular orbit. If higher power is used, the remaining capacity would most likely be filled with orbiter science data.

The expected cumulative bits and the total bits (reliability not included) for a 6-mo mission for these three systems is given in Table 13 for 200- and 400-day transit trajectories and the following assumptions:

1. The reliability of the direct link high-gain antenna is 0.75 and time independent.

2. With the exception of the high-gain antenna and parts of the telecommunication subsystem, the relay lander is identical to the direct lander from a reliability standpoint.

Table 13. Expected cumulative lander bits over 6-mo mission using 35-w S-band Mars–Earth transmitter

Mission	Direct link, high-gain antenna reliability = 0.75	Relay link, 5000 km circular[a]	Relay link, 4000 × 20,000 km[a]
200-day transit	5.1×10^9	1.18×10^{10}	1.29×10^9
400-day transit	3.3×10^9	6.3×10^9	6.6×10^8
Total bits[b]	1.7×10^{10}	5.2×10^{10}	5.7×10^9

[a] 10 w lander radiated power.
[b] Perfect reliability.

6.2.3 Storage Requirements

6.2.3.1 Relay link. The length and spacing of the lander–orbiter viewing times have been mentioned in Section 4.1.2.3. These two quantities fix the data storage requirements for both the landed capsule and the satellite.

If the system has a viewing fraction v and an average time between sightings $\bar{T}$, then the fraction of time out of sight is (1-v) and the average amount of data stored at the beginning of a sighting is approximately $\bar{T}(1-v)\,\dot{I}_c$ where $\dot{I}_c$ is the rate of data collection. The maximum data collection rate $\dot{I}_c$ is equal to the effective bit rate $\dot{I}_e$, but it seems likely that information would be gathered at a somewhat lesser rate to assure complete return of all data.

Unfortunately, a closed analytic expression for the quantity T has not yet been found, but average storage requirements for the example systems can be produced.

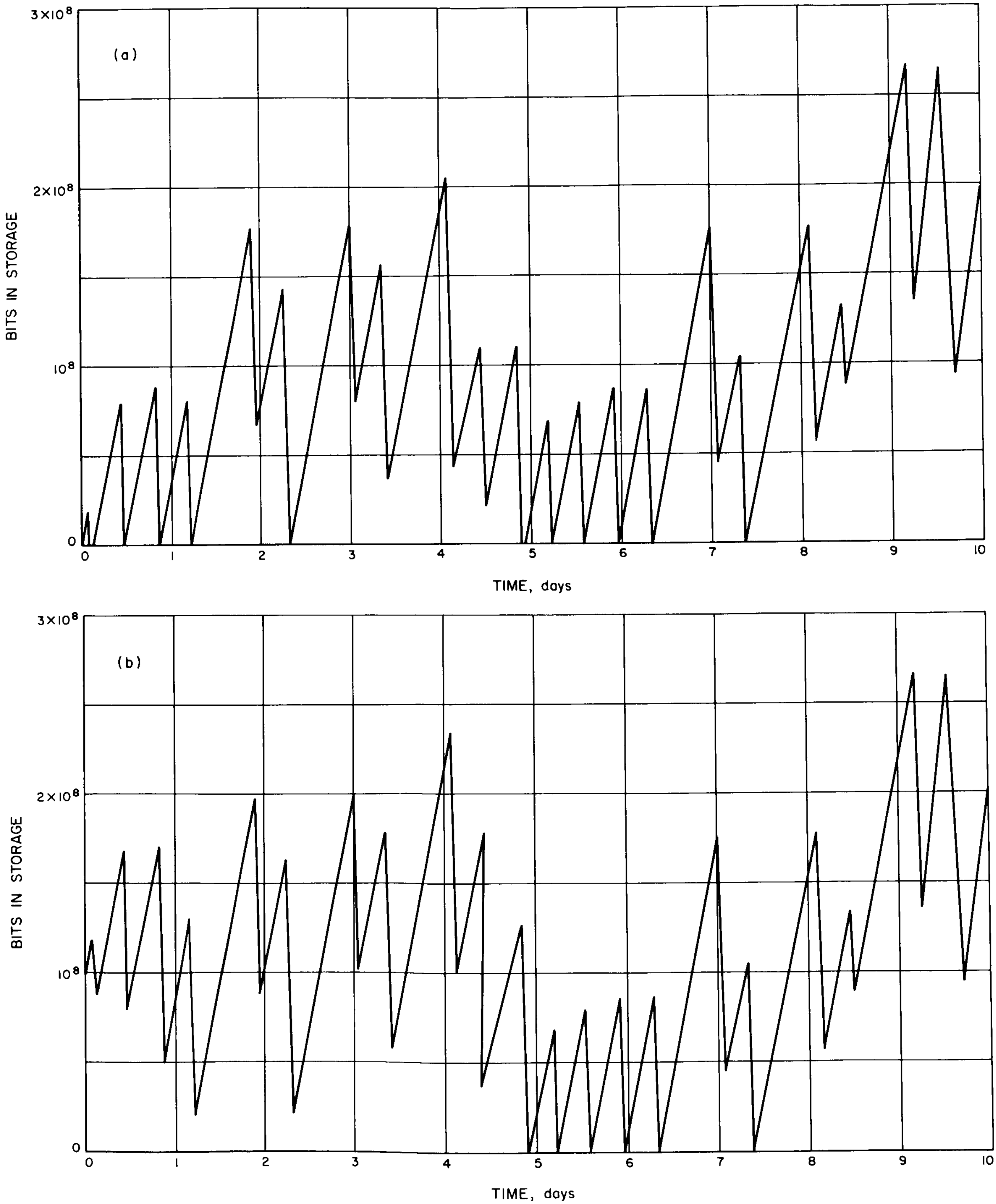


Fig. 26. Time history of storage unit status: (a) no initial bits, (b) 10^8 initial bits

Figure 20 has shown the time history of the elevation angle for the 5000-km orbit, and Table 10 has shown that the Case B antenna can transfer an average of 3340 bits/sec by starting transmission at 36 deg off the horizon. Assuming the lander collects data at a constant rate of 3000 bits/sec, the status of the storage unit for the first 10 days can be derived from Fig. 20 noting that the unit is discharged at 28,300 bits/sec (Table 10 and 10-w transmitter) when the satellite is above the minimum elevation angle.

The initial condition of the storage unit is the subject of some question. It seems likely that by 1975 atmospheric entry data will no longer be needed, but there may be some descent television system storing rather than immediately transmitting the pictures. If, for instance, ten pictures were taken before landing, the initial complement of data would be in the order of 10^8 bits. Figure 26 shows the time history of the status of the storage unit assuming no initial data and 10^8 bits for the nominal 5000-km circular orbit. Note that the most data stored for both is 2.65×10^8 bits.

The storage requirements for the satellite are similar to those of the relay lander. Here the storage unit is discharged rather than charged at a constant rate with occasional inputs of data. Therefore, its time history is more or less the inverse of Fig. 26a.

The 4000- × 20,000-km orbit has not been treated in detail because of its low effective bit rate. Figure 27 shows, however, that the mean time $\bar{T}$ is roughly four times greater than that for the 5000-km orbit.

6.2.3.2 *Direct link.* The storage necessary for the direct link can be calculated immediately. The relation for the stored data at the start of transmission, $\bar{T}(1\text{-}v)\ \dot{I}_e$, can be used because $\bar{T}$ is just the time of the planet in rotation. Assuming that $\dot{I}_e = 3000$ bits/sec and $v = 1/3$ (Section 4.2), then the necessary capacity is about 1.8×10^8 bits.

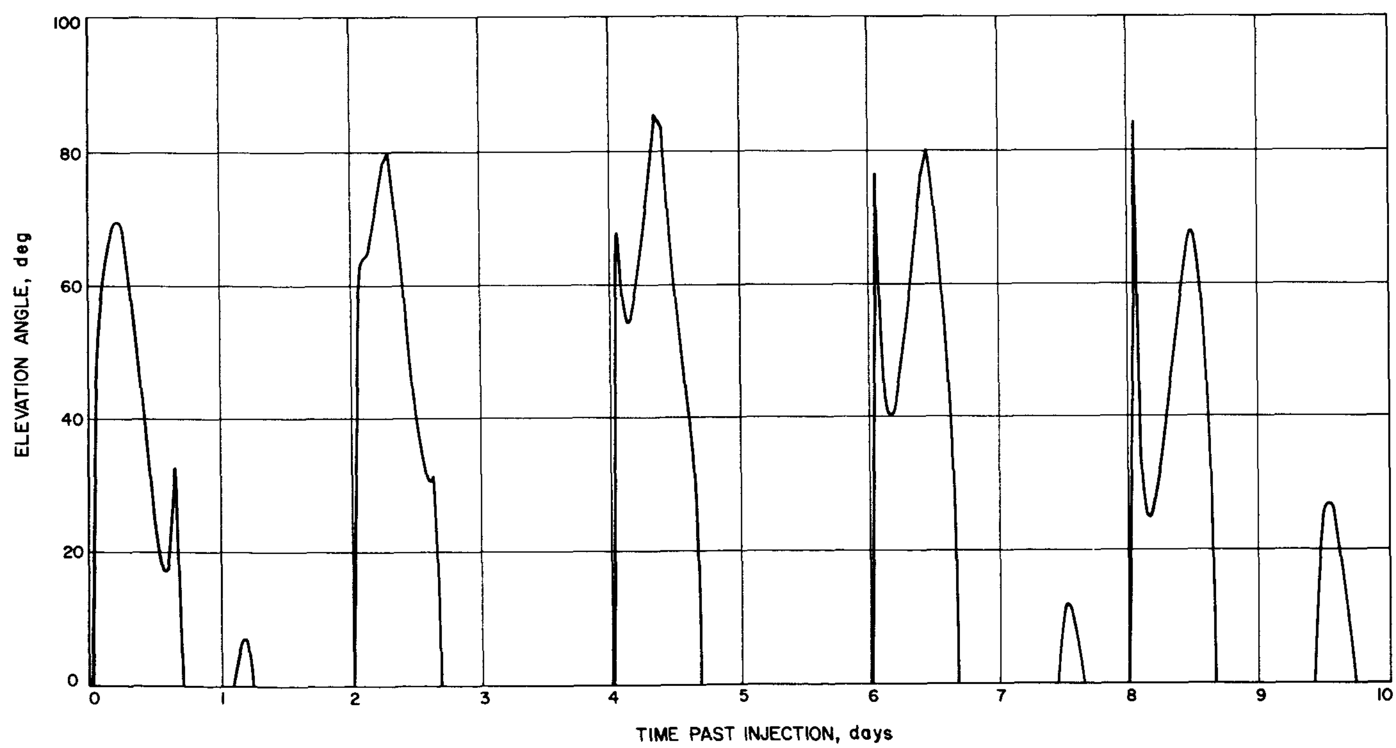


Fig. 27. Elevation angle vs time for 4000 × 20,000 km elliptic orbit

7. CONCLUSIONS

The following general conclusions can be drawn in terms of the three comparison parameters: effective bit rate, reliability, and expected cumulative bits. For equal radiated power to the Earth, the relay link has about three times the data retrieval capability of the direct link. The reliability of the relay link system is always lower than the direct link, and is twice as likely to fail within the 6-mo mission as the direct link. However, the increased data rate of the relay link causes the expected number of bits recoverable from the relay link to be higher than the direct link. But, this difference is negligible when the accuracy of the data used to produce the answers is considered. Therefore, the relay link is better (by a factor of three) in absolute data rate capability; worse by (at most) a factor of two in reliability, and within a factor of two in terms of expected cumulative bits. Thus, neither data retrieval approach shows a clear advantage in terms of the comparison parameters.

Several observations specific to the relay link may be made. The lander transmitter power needed to saturate the orbiter–Earth leg is only 10 w for a 5000-km altitude orbit dictated by the present contamination constraint; it is clear that there is no degradation suffered in the relay link potential due to this constraint. There is no reason to seek a lower altitude orbit because the orbiter–Earth leg cannot handle the added bit-rate capability.

The major constraint on the data retrieval capability of the relay link is the orbiter–Earth communications. If the problems inherent in the orbiter–Earth communications are solved, some of the problems of the direct link are also solved. The main areas in common are the pointing of an extremely directional high-gain antenna and the construction of long-life, high-power, S-band transmitters.

The effect of departing from an optimum communication orbit on the relay link data-rate capability can also be seen. Section 6.2.1 shows that the selection of an orbit to meet some science requirements (4000- $\times$ 20,000-km altitude) decreases the lander–orbiter data-rate capability by almost a factor of ten. This is not as serious as it might first seem because some data-rate capability must be forfeited for the orbiter science data. One might think of raising the lander transmitter power but, unless one is considering a VHF transmitter on the order of 100 w, the goal of 10^{10} bits of lander data must be set aside.

Interesting and important results have been obtained in the area of optimizing the use of antenna beam patterns. Table 10 shows that, contrary to intuition, it is not generally best to transmit over as large an angle as possible. Rather, the transmission policy is dependent on the beam shape of the lander and orbiter antennas and the elements of the orbit. The general results of Section 6.2.1.1 may be applied to any given pattern, or may be used to derive wanted pattern shape.

It is also seen that the longer flight time associated with the Type II trajectories penalize the success probability of the relay link mission more harshly than the direct link mission. This is true because more components are needed for the relay link communications system to operate successfully. This larger failure potential must be weighed against the uncertainty in the ability of the direct link lander to operate in the relatively unknown Mars surface environment.

Preliminary calculations show that the storage unit bit requirements are about the same for both the relay and direct links and are on the order of 3×10^8 bits for a mission objective of 10^{10} bits. The size of the storage unit scales directly with the wanted total bits for a given length mission.

In summary, no definitive conclusion can be drawn between the direct and relay links when using the standards of effective bit rate, reliability, and expected cumulative bits. Therefore, the decision should be made on different and, possibly, more pertinent considerations.

APPENDIX A

Background Mathematics

A.1 INDEPENDENT EVENTS

Let A and B represent the successful operation of two subsystems that are physically and functionally independent over a given time gap. The probability (Pr) that both subsystems operate successfully over the time gap is given by:

$$Pr\,(A \text{ and } B) = Pr\,(A) \times Pr\,(B) \tag{A-1}$$

where $Pr\,(A)$ is the probability that the first subsystem works properly and $Pr\,(B)$ is the probability that the second subsystem works properly.

A.2 DEPENDENT EVENTS

If C_1 and C_2 represent the successful operation of two subsystems over a given time gap, and if these subsystems are interdependent, then

$$Pr\,(C_1 \text{ and } C_2) = Pr\,(C_1) \times Pr\,(C_2/C_1) \tag{A-2}$$

where $Pr\,(C_1)$ is the probability that the first subsystem works properly over the time gap and $Pr\,(C_2/C_1)$ is the probability that the second subsystem works properly over the time gap, if the first subsystem has worked properly over the same time gap. If the second subsystem has the same components as the first, $Pr\,(C_2/C_1)$ is unity.

A.3 REDUNDANCY

If several subsystems are designed so they are functionally redundant in a given spacecraft system, the spacecraft will work successfully as long as at least one of the sets of functionally redundant subsystems continues to operate successfully. Let D and E represent the successful operation of two redundant, but independent, subsystems. Then the probability that at least one of the subsystems will work properly is given by:

$$Pr\,(D \text{ or } E) = Pr\,(D) + Pr\,(E) - Pr\,(D)\,Pr\,(E) \tag{A-3}$$

where $Pr\,(D)$ = probability that the first subsystem works properly over the time gap

$Pr\,(E)$ = probability that the second subsystem works properly over the time gap

$Pr\,(D)\,Pr\,(E)$ = probability that both will work properly over the time gap (Section A.1).

Equation (A-3) holds whenever both subsystems are activated at the outset of the mission. If the redundant subsystem is activated only on failure of the primary, the expression is obtained through a technique calling for the integration of their failure density functions and consideration of the presence of failure detection and switching circuitry. Usually, where redundancy techniques are used in this analysis, they will be of the first type.

A.4 RELIABILITY MODELS

The reliability models used in this analysis come from two basic probability distributions. One is the exponential distribution that is used to represent the reliability model of electronic components; the other is the binomial distribution that is used to represent the reliability model of short time mechanical and chemical devices.

A.4.1 Exponential Reliability Model

If an electric component is known to have a failure density function of $\lambda e^{-\lambda t}$, then the probability that the component will not fail between $t = 0$ and $t = t_1$ is:

$$R(t_1) = 1 - \int_0^{t_1} \lambda \exp(-\lambda t)\, dt$$

$$= \exp(-\lambda t_1) \tag{A-4}$$

Parameters β and α have been introduced (Sections 5.2.4.1 and 5.2.4.2) to modify the failure rate λ in the stored and energized modes, respectively. If the time a component is energized is t_E and the time stored t_S where $t_S + t_E = t_1$, then the application of α and β changes Eq. (A-4) to:

$$R(t_S, t_E) = \left[1 - \int_0^{t_S} \alpha\,\lambda \exp(-\alpha\,\lambda\, t)\, dt\right]$$

$$\left[1 - \int_0^{t_1 - t_S} \beta\,\lambda \exp(-\beta\,\lambda\, t)\, dt\right] \qquad \text{(A-5)}$$

$$= \exp(-\alpha\,\lambda\, t_S - \beta\,\lambda\, t_E)$$

A.4.2 Binominal Reliability Model

A component obeying the binomial probability law will work properly with some probability p, where: $0 \leq p \leq 1$, and will fail to work with probability, $1-p$.

A.5 CONDITIONAL PROBABILITIES

If the probability that a component will work at least for time t_i is $R(t_i)$, then the probability that it will work for time t_2 is if it has successfully operated for time t_1 is:

$$R(t_2/t_1) = \frac{R(t_2)}{R(t_1)} \qquad \text{(A-6)}$$

APPENDIX B

Subsystem Reliability Models

B.1 INTRODUCTION

The following sets of functions represent the reliability models of the individual subsystems appearing in Figs. 12 and 13.

The reliability of a time-dependent subsystem can be obtained by solving the functions at the time of interest. The probability that subsystem X will work through Event A is denoted by R_{AX}. The times at which the major events take place are a function of the trajectory chosen, therefore the models are derived for arbitrary event times. The time of occurrence of Event A is represented by t_A. For a given time of occurrence of Event A, the reliability of all subsystems that are functions of t_A can be obtained.

The reliability models of the direct link and relay link subsystems were obtained from various independent sources, and, therefore, the assumptions on which the several models are based are not necessarily consistent. Because the authors are not familiar with every subsystem configuration on board a typical direct link and relay link spacecraft, it was necessary to accept these models as best estimates of the subsystems. Further refinements of subsystem definition for advanced planetary missions will be made in the future and, later, the reliability models used to represent them can be made more accurate.

An IBM 1620 program was used to get numerical results from the analysis. The program was written with enough flexibility so that any subsystem model thought unrepresentative can be removed and a better model put in with little effort.

B.2 BUS AND ORBITER SYSTEMS

The following models cover all subsystems in the direct link flyby bus and the relay link orbiter. Note that numerical subscripts 1–7 refer to the subsystems of the direct link configuration, and subscripts 51–58 refer to the subsystems of the relay link configuration.

B.2.1 Thermal Control Subsystem

A passive thermal control subsystem was assumed to be ample to meet the advanced planetary mission requirements. Such a passive system, when designed well, should be a very reliable device. Therefore, it was assumed (for this analysis) that the thermal control system in both the direct link configuration and relay link configuration was perfectly reliable.

$$R_{A1} = 1 \qquad R_{J51} = 1$$
$$R_{B1} = 1 \qquad R_{M51} = 1$$
$$R_{C1} = 1 \qquad R_{N51} = 1$$
$$R_{H51} = 1 \qquad R_{S51} = 1$$
$$R_{I51} = 1$$

B.2.2 Electrical Power Subsystem

The model for a solar photo-voltaic electrical power subsystem was taken from Ref. 5, which analyzed the *Mariner C* electrical power (EP) system.

$$R(EP) = P_1 P_2 (P_3P_4 + P_5 - P_3P_4P_5)P_6 (P_7 + P_8P_9 - P_7P_8P_9)P_{10}$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Solar panel circuitry	29.36
P_2	Power distribution	2.96
P_3	38.4 kc CC&S	12.42
P_4	Synchronous source transfer	4.85
P_5	Local oscillator	2.421
P_6	2.4 kc synchronous	4.49
P_7	Booster regulator 1	16.10
P_8	Switch for booster regulator 2	8.26
P_9	Booster regulator 2	16.85
P_{10}	2.4 kc main inverter	11.44

$$P_A = \exp(-48.25 \times 10^{-6}\,\beta t_A)[\exp(17.27 \times 10^{-6}\,\beta t_A) - \exp(-19.69 \times 10^{-6}\,\beta t_A) + \exp(-2.42 \times 10^{-6}\,\beta t_A)] [\exp(-16.10 \times 10^{-6}\,\beta t_A) + \exp(-25.11 \times 10^{-6}\,\beta t_A) - \exp(-41.21 \times 10^{-6}\,\beta t_A)]$$

$$P_{B2} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_B$$
$$P_{C2} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_C$$
$$P_{H52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_H$$
$$P_{I52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_I$$
$$P_{J52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_J$$
$$P_{M52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_M$$
$$P_{N52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t_N$$
$$P_{S52} = \text{same as } P_{A2} \text{ with } t_A \rightarrow t$$

B.2.3 Guidance and Attitude Control Subsystem

The model for a guidance and attitude control (G&AC) subsystem was taken from Ref. 5 (*Mariner C*).

$$R(G\&AC) = P_1P_2P_3P_4P_5P_6P_7P_8P_9P_{10}$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Canopus sensors and electronics	24.64
P_2	Canopus gate	8.89
P_3	Cone angle update circuits	5.71
P_4	Derived rate damping (roll)	1.54
P_5	Switch amplifiers and switch (roll)	10.22
P_6	Cruise sun sensors and regulator	3.68
P_7	Attitude control transformer/ rectifier	1.94
P_8	Derived rate damping (pitch and yaw)	2.33
P_9	Switch amplifiers and switch (pitch and yaw)	32.81
P_{10}	Valves, nozzles, and gas	0.00

$$R_{A3} = \exp(-91.76 \times 10^{-6}\,\beta t_A)$$
$$R_{B3} = \exp(-91.76 \times 10^{-6}\,\beta t_B)$$
$$R_{C3} = \exp(-91.76 \times 10^{-6}\,\beta t_C)$$
$$R_{H53} = \exp(-91.76 \times 10^{-6}\,\beta t_H)$$
$$R_{I53} = \exp(-91.76 \times 10^{-6}\,\beta t_I)$$
$$R_{J53} = \exp(-91.76 \times 10^{-6}\,\beta t_J)$$
$$R_{M53} = \exp(-91.76 \times 10^{-6}\,\beta t_M)$$
$$R_{N53} = \exp(-91.76 \times 10^{-6}\,\beta t_N)$$
$$R_{S53} = \exp(-91.76 \times 10^{-6}\,\beta t)$$

B.2.4 Central Computer and Sequencer Subsystem

The model for a central computer and sequencer (CC&S) subsystem was taken from Ref. 5 (*Mariner C*).

$$R(CC\&S) = P_1P_2P_3P_4P_5P_6P_7P_8$$

Symbol	Unit	Failure rate times 10^{-6}/hr
P_1	CC&S transformer/rectifier	4.60
P_2	Oscillator	7.80
P_3	1 pps, 25 pps, 1 ppm counter	25.44
P_4	Magnetic/1000	6.33
P_5	Magnetic/2000	8.19
P_6	Master time matrix	10.09
P_7	Driver	1.70
P_8	Relay	0.75

$$R_{A4} = \exp(-64.90 \times 10^{-6}\,\beta t_A)$$

$$R_{B4} = \exp(-64.90 \times 10^{-6}\,\beta t_B)$$

$$R_{C4} = \exp(-64.90 \times 10^{-6}\,\beta t_C)$$

$$R_{H54} = \exp(-64.90 \times 10^{-6}\,\beta t_H)$$

$$R_{I54} = \exp(-64.90 \times 10^{-6}\,\beta t_I)$$

$$R_{J54} = \exp(-64.90 \times 10^{-6}\,\beta t_J)$$

$$R_{M54} = \exp(-64.90 \times 10^{-6}\,\beta t_M)$$

$$R_{N54} = \exp(-64.90 \times 10^{-6}\,\beta t_N)$$

$$R_{S54} = \exp(-64.90 \times 10^{-6}\,\beta t)$$

B.2.5 Propulsion Subsystem

The model for a propulsion subsystem was taken from Ref. 5 (*Mariner C*). For this analysis, the propulsion subsystem's reliability is degraded only when it is stored. If it has not failed during storage, it will work properly with probability 1.0 when activated. A failure rate of 187.00×10^{-6}/hr was given in the reference. Therefore:

$$P_{B7} = \exp(-187.00 \times 10^{-6}\,\alpha t_B)$$

$$P_{I57} = \exp(-187.00 \times 10^{-6}\,\alpha t_I)$$

$$P_{M57} = \exp(-187.00 \times 10^{-6}\,\alpha t_M)$$

B.2.6 Telecommunications Subsystem

Besides fulfilling the normal communications requirements for a typical direct link and relay link spacecraft, the telecommunication subsystem (in conjunction with the Deep Space Network, DSN) is also assumed to back up the CC&S. So, where more than one telecommunication subsystem appears in the block diagram for a successful event (Figs. 12 and 13), they are not independent. Wherever this happened, the technique described in Section A.2 (Appendix A) was applied in the analysis. Whenever the same component was found in more than one telecommunications subsystem, its reliability was assumed to be unity in all subsystems except for the one serially connected to the rest of the spacecraft. The model for the various telecommunications (T/C) subsystems was extracted from a JPL internal document.

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Omnidirectional antenna (S-band)	4.0
P_2	High-gain antenna (S-band)	7.0
P_3	Circulator (S-band)	2.0
P_4	Receiver (S-band)	70.8
P_5	Command detector and decoder	43.7
P_6	Power supply (S-band)	5.2
P_7	Power amplifier (S-band)	16.3
P_8	Exciter (S-band)	26.4
P_9	RF switch	2.0
P_{10}	Diplexer (orbiter–lander)	3.0
P_{11}	Receiver (orbiter–lander)	36.4
P_{12}	Buffer and storage (orbiter–lander)	157.0
P_{13}	Modulator	2.54
P_{14}	Data select switch	2.0
P_{15}	Exciter 2 (S-band)	26.4
P_{16}	Power supply (orbiter–lander)	5.2
P_{17}	Tone generator and power amplifier (orbiter–lander)	16.3
P_{18}	Detector (orbiter–lander)	16.54

$$\begin{aligned} R_{A5} &= (P_1 + P_2 - P_1P_2)\,P_3P_4P_5P_6 \\ &= \exp(-120.7 \times 10^{-6}\,\beta t_A)\,[\exp(-4.0 \times 10^{-6}\,\beta t_A) \\ &\quad + \exp(-7.0 \times 10^{-6}\,\beta t_A) - \exp(-11.0 \times 10^{-6}\,\beta t_A)] \end{aligned}$$

$$R_{B5} = 1$$

$$R_{C5} = 1$$

$$\begin{aligned} R_{H55} &= (P_1 + P_2 - P_1P_2)\,P_3P_4P_5P_6 \\ &= \exp(-120.7 \times 10^{-6}\,\beta t_H)\,[\exp(-4.0 \times 10^{-6}\,\beta t_H) \\ &\quad + \exp(-7.0 \times 10^{-6}\,\beta t_H) - \exp(-11.00 \times 10^{-6}\,\beta t_H)] \end{aligned}$$

$$R_{I55} = 1$$

$$R_{J55} = 1$$

$$R_{M55} = 1$$

$$\begin{aligned} R_{N55} &= P_4P_5 \\ &= \exp[-114.5 \times 10^{-6}\,\beta(t_N - t_M)] \end{aligned}$$

$$R_{S55} = \exp[-114.5 \times 10^{-6}\,\beta(t - t_M)]$$

$$\begin{aligned} R_{B6} &= (P_1 + P_2 - P_1P_2)\,P_3P_4P_5P_6P_7P_8 \\ &= \exp(-164.4 \times 10^{-6}\,\beta t_B)\,[\exp(-4.0 \times 10^{-6}\,\beta t_B) \\ &\quad + \exp(-7.0 \times 10^{-6}\,\beta t_B) - \exp(-11.0 \times 10^{-6}\,\beta t_B)] \end{aligned}$$

$R_{C6} =$ same as R_{B6} with $t_B \rightarrow t_C$

$$R_{I56} = \nu (P_1 + P_2 - P_1P_2) P_3P_4P_5P_6P_7P_8P_9$$
$$= \nu \exp(-166.4 \times 10^{-6} \beta t_I) \left[\exp(-4.0 \times 10^{-6} \beta t_I) + \exp(-7.0 \times 10^{-6} \beta t_I) - \exp(-11.0 \times 10^{-6} \beta t_I)\right]$$

$R_{J56} =$ same as R_{I56} with $t_I \rightarrow t_J$

$R_{M56} =$ same as R_{I56} with $t_I \rightarrow t_M$

$$R_{N56} = \nu P_1P_2P_3P_6P_7P_9P_{10}P_{11}P_{12}P_{13}P_{14}P_{15}P_{16}P_{17}P_{18}$$
$$= \exp(-140.9 \times 10^{-6} \beta t_M)\{\exp[-32.5 \times 10^{-6} \beta t_N - (253.1 + 16.3\mu) \times 10^{-6} \alpha t_N]\}$$

$$R_{S56} = \nu \exp(-140.9 \times 10^{-6} \beta t_M)(\exp\{-32.5 \times 10^{-6} \beta t - (253.1 + 16.3\mu) \times 10^{-6} [\alpha t_N + \beta (t - t_N)]\})$$

B.2.7 *Planetary Horizontal Platform Subsystem*

The model for a planetary horizontal platform (PHP) subsystem was taken from Ref. 3, which analyzed a proposed *Voyager*-class mission.

$$R(PHP) = P_1 P_2 P_3 P_4 P_5$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Accelerometer block and associated electronics	64.36
P_2	Computer	196.90
P_3	Planet tracker	98.80
P_4	Horizon scanner	62.50
P_5	Auxiliary star tracker	28.53

$$R_{N58} = \exp(-451.09 \times 10^{-6} \alpha t_N)$$

$$R_{S58} = \exp\{-451.09 \times 10^{-6} [\alpha t_N + \beta(t - t_N)]\}$$

B.3 LANDER SUBSYSTEMS

The following models cover all subsystems in the direct link and relay link landers. Note, numerical subscripts 21–41 refer to direct link lander subsystems and subscripts 71–84 refer to the relay link configuration.

B.3.1 *Thermal Control Subsystem*

The model for a lander thermal control (TC) subsystem was taken from Ref. 4 (*Voyager*).

$$P(TC) = R_1R_2R_3R_4R_5R_6R_7 (1 + \lambda_6 t + \lambda_7 t) R_8 [1 - (1 - R_9)(1 - R_{10})][1 - (1 - R_{10})(1 - R_{11})] R_{12} R_{13} R_{14} R_{15} (1 + \lambda_{14} t + \lambda_{15} t) R_{16} R_{17} R_{18}$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
R_1	Water tank (not used on surface)	0.35
R_2	Water boiler (not used on surface)	1.10
R_3	Solenoid valves (not used on surface)	1.13
R_4	Radioisotope thermoelectric generator heat exchanger	1.10
R_5	Liquid-to-liquid heat exchanger	1.10
R_6	Pump	1.12
R_7	Direct current motor	1.12
R_8	Solenoid valve	1.13
R_9	Solenoid valve	1.13
R_{10}	Squib valve and guillotine	1.13
R_{11}	Check valve	0.11
R_{12}	Intransit radiator (not used on surface)	1.10
R_{13}	Reservoir	0.35
R_{14}	Modulation valve	1.45
R_{15}	Direct current motor	1.12
R_{16}	Temperature sensor	0.15
R_{17}	Temperature controller	0.47
R_{18}	Plumbing, fittings, tubing, and component surface plates	1.10

$$R_{D21} = \exp(-13.89 \times 10^{-6} \alpha t_D)(1 + 2.24 \times 10^{-6} \alpha t_D)[2 \exp(-1.13 \times 10^{-6} \alpha t_D) - \exp(-2.26 \times 10^{-6} \alpha t_D)] [\exp(-1.13 \times 10^{-6} \alpha t_D) + \exp(-0.11 \times 10^{-6} \alpha t_D) - \exp(-1.24 \times 10^{-6} \alpha t_D)] (1 + 2.57 \times 10^{-6} \alpha t_D)$$

$$R_{E21} = (\exp\{-3.68 \times 10^{-6} \alpha t_D - 10.21 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\}) \{1 + 2.24 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\} (2 \exp\{-1.13 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\} - \exp\{-2.26 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\}) (\exp\{-1.13 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\} + \exp\{-0.11 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\} - \exp\{-1.24 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\}) \{1 + 2.57 \times 10^{-6} [\alpha t_D + \beta(t_E - t_D)]\}$$

$R_{F21} =$ same as R_{E21} with $t_E \rightarrow t_F$

$R_{G21} =$ same as R_{E21} with $t_E \rightarrow t_G$

$R_{K71} =$ same as R_{D21} with $t_D \rightarrow t_K$

$R_{L71} =$ same as R_{E21} with $t_E \rightarrow t_L$

$R_{O71} =$ same as R_{E21} with $t_E \rightarrow t_O$

$R_{T71} =$ same as R_{E21} with $t_E \rightarrow t$

B.3.2 *Electrical Power Subsystem*

The model for a lander electrical power (EP) subsystem was taken from Ref. 4 (*Voyager*).

$$R(EP) = P_1\, P_2\, P_3\, P_4\, P_5$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Radioisotope thermoelectric generator	0.28
P_2	Battery charging regulator	2.11
P_3	Nickel-cadmium battery	0.50
P_4	Harness, cabling, connectors	1.00
P_5	Power conversion and control	1.75

$$R_{D22} = \exp\left(-\,5.64 \times 10^{-6}\, \alpha t_D\right)$$

$$R_{E22} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_D + \beta(t_E - t_D)\right]\right\}$$

$$R_{F22} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_D + \beta(t_F - t_D)\right]\right\}$$

$$R_{G22} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_D + \beta(t - t_D)\right]\right\}$$

$$R_{K72} = \exp\left(-\,5.64 \times 10^{-6}\, \alpha t_K\right)$$

$$R_{L72} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_K + \beta(t_L - t_K)\right]\right\}$$

$$R_{O72} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_K + \beta(t_O - t_K)\right]\right\}$$

$$R_{T72} = \exp\left\{-\,5.64 \times 10^{-6}\left[\alpha t_K + \beta(t - t_K)\right]\right\}$$

B.3.3 *Release Subsystem*

The reliability estimate for a capsule release subsystem was taken from Ref. 3, Volume IV, page 316. The components of the release subsystem (RS) are time-independent. When energized, they will work properly with the following probabilities:

$$R(RS) = P_1\, P_2\, P_3\, P_4\, P_5\, P_6\, P_7\, P_8\, P_9$$

Symbol	Unit	Reliability
P_1	Flexible linear shaped charges	0.98200
P_2	Explosive bolts	0.99974
P_3	Ball lock bolt	0.99790
P_4	Mechanical latches	0.97850
P_5	Marmon clamp	Not available
P_6	Spring	0.99160
P_7	Gas generator	0.99435
P_8	Cartridge activated device	0.98900
P_9	"Fly-away"	0.98188

Reliability estimates are based on Avco reliability demonstration tests and the FARADA Handbook (Ref. 6).

B.3.4 *Stabilization Subsystem*

The reliability estimate for a capsule stabilization subsystem was taken from Ref. 3, Volume IV, page 316. When activated, the stabilization subsystem (SS) will work properly with the following probability.

$$R(SS) = P_1\, P_2\, P_3$$

Symbol	Unit	Reliability
P_1	Spin rockets	0.99070
P_2	Momentum wheels	Not available
P_3	Attitude control system	0.98430

Reliability estimates are based on Avco reliability demonstration tests and the FARADA Handbook (Ref. 6).

$$R_{D24} = 0.9751$$

$$R_{K74} = 0.9751$$

B.3.5 *Retardation Subsystem*

The reliability estimate for a capsule retardation (R) subsystem was taken from Ref. 4, which analyzed a heat shield and parachute retardation technique. All components of the retardation subsystem have a discrete probability of working properly when energized.

$$R(R) = \left[1-(1-P_1P_2P_3^7P_4^4)^2\right] P_5P_6P_7P_8P_9P_{11}^4P_{12}P_{13} \left[1-(1-P_{14})^4\right] P_{15}$$

Symbol	Unit	Reliability
P_1	Remote activated batteries (2)	0.9980
P_2	Arming relays (2)	0.9999
P_3	Acceleration switches (14)	0.9980
P_4	Timers (8)	0.9990
P_5	Trajectory time delay (1)	0.9999
P_6	Drogue mortar (1)	0.9990
P_7	Deceleration chute (1)	0.9990
P_8	In-flight disconnect (1)	0.9990
P_9	Time delay (1)	0.9999
P_{10}	Aft cover explosive bolts (4)	0.9990
P_{11}	Deceleration chute explosive bolts (4)	0.9990
P_{12}	Main parachute (1)	0.9990
P_{13}	Swivel (1)	0.9999
P_{14}	Reef line cutters (4)	0.9999
P_{15}	Cutoff fittings (4)	0.9999

$$R_{E25} = 0.9872$$

$$R_{L75} = 0.9872$$

B.3.6 Surface Orientation Subsystem

The reliability estimate for a capsule surface orientation (SO) subsystem was taken from Ref. 4. All components of the surface orientation subsystems are time-independent.

$$R(SO) = P_1P_2P_3P_4P_5P_6P_7P_8P_9P_{10}P_{11}P_{12}P_{13}$$

Symbol	Unit	Reliability
P_1	Impact acceleration-switch (1)	0.9990
P_2	Arm relay (1)	0.9999
P_3	Disarm relay (1)	0.9999
P_4	Mercury switches (3)	0.9999
P_5	Motion detectors (3)	0.9990
P_6	Time delay (1)	0.9999
P_7	Motion and gear train (1)	0.9990
P_8	Deployment mechanisms (10)	0.9990
P_9	Electromechanical activator (1)	0.9990
P_{10}	Tilt bar (1)	0.9999
P_{11}	Harpoons (2)	0.9980
P_{12}	Solid rockets (2)	0.9980
P_{13}	Shaped charges (1)	0.9990

$$R_{F26} = 0.9895$$

$$R_{O76} = 0.9895$$

B.3.7 Data Handling Subsystem

The lander data handling (DH) subsystem was assumed to consist of a *Mariner C* tape recorder and real-time data automation system.

$$R(HD) = P_1P_2$$

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Tape recorder (*Mariner C*)	187.00
P_2	Real-time data automation system (*Mariner C*)	537.34

$$R_{G27} = \exp\left\{-724.34 \times 10^{-6}\left[\alpha t_F + \beta(t - t_F)\right]\right\}$$

$$R_{T77} = \exp\left\{-724.34 \times 10^{-6}\left[\alpha t_O + \beta(t - t_O)\right]\right\}$$

B.3.8 Capsule Timer and Sequencer Subsystem

The capsule timer and sequencer (CT&S) was assumed to consist of the same components as the *Mariner C* central computer and sequencer (CC&S). So, the same reliability model as described in Ref. 5 for the *Mariner C* CC&S was picked to represent the CT&S (Section B.2.4).

$$R_{D28} = \exp\left(-64.90 \times 10^{-6}\,\alpha t_D\right)$$

$$R_{E28} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_D + \beta(t_E - t_D)\right]\right\}$$

$$R_{F28} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_D + \beta(t_F - t_D)\right]\right\}$$

$$R_{G28} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_D + \beta(t - t_D)\right]\right\}$$

$$R_{K78} = \exp\left(-64.90 \times 10^{-6}\,\alpha t_K\right)$$

$$R_{L78} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_K + \beta(t_L - t_K)\right]\right\}$$

$$R_{O78} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_K + \beta(t_O - t_K)\right]\right\}$$

$$R_{T78} = \exp\left\{-64.90 \times 10^{-6}\left[\alpha t_K + \beta(t - t_K)\right]\right\}$$

B.3.9 Telecommunications Subsystem (Excluding Antennas)

The telecommunications (T/C) subsystem consists of components that have a primary communication link and components that form a backup FSK S-band link. No component in the lander is shared by both the primary and backup configurations. The model for the T/C subsystems for the various lander events was taken from an internal JPL document.

Symbol	Unit	Failure rate, times 10^{-6}/hr
P_1	Receiver (S-band)	70.8
P_2	Capsule command detector and decoder	5.2
P_3	Circulator (S-band)	2.0
P_4	Power amplifier (S-band)	16.3
P_5	Exciter (S-band)	26.4
P_6	Power supply (S-band)	5.2
P_7	Data encoder (*Mariner C*)	100.0
P_8	Diplexer (lander–orbiter)	3.0
P_9	Power amplifier (lander–orbiter)	16.3
P_{10}	Exciter (lander–orbiter)	10.5
P_{11}	Power supply (lander–orbiter)	5.2
P_{12}	Receiver (lander–orbiter)	36.4

$$
\begin{aligned}
R_{F30} &= P_1 P_2 \\
&= \exp\left(-75.7 \times 10^{-6}\, \alpha t_F\right) \\
R_{G30} &= P_1 P_2 \\
&= \exp\left\{-75.7 \times 10^{-6}\, [\alpha t_F + \beta(t - t_F)]\right\} \\
R_{O82} &= P_1 P_2 \\
&= \exp\left(-75.7 \times 10^{-6}\, \alpha t_O\right) \\
R_{T82} &= P_1 P_2 \\
&= \exp\left\{-75.7 \times 10^{-6}\, [\alpha t_O + \beta(t - t_O)]\right\} \\
R_{F31} &= P_3 P_4 P_5 P_6 \\
&= \exp\left(-49.9 \times 10^{-6}\, \alpha t_F\right) \\
R_{F41} &= P_3 P_4 P_5 P_6 \\
&= \exp\left(-49.9 \times 10^{-6}\, \alpha t_F\right) \\
R_{G31} &= P_3 P_4 P_5 P_6 P_7 \\
&= \exp\left\{-149.9 \times 10^{-6}\, [\alpha t_F + \beta(t - t_F)]\right\} \\
R_{G41} &= P_3 P_4 P_5 P_6 P_7 \\
&= \exp\left\{-149.9 \times 10^{-6}\, [\alpha t_F + \beta(t - t_F)]\right\} \\
R_{O84} &= P_3 P_4 P_5 P_6 \\
&= \exp\left(-49.9 \times 10^{-6}\, \alpha t_O\right) \\
R_{T84} &= P_3 P_4 P_5 P_6 P_7 \\
&= \exp\left\{-149.9 \times 10^{-6}\, [\alpha t_O + \beta(t - t_O)]\right\} \\
R_{O80} &= P_1 P_2 P_3 P_6 \\
&= \exp\left(-83.2 \times 10^{-6}\, \alpha t_O\right) \\
R_{T80} &= P_1 P_2 P_3 P_6 \\
&= \exp\left\{-83.2 \times 10^{-6}\, [\alpha t_O + \beta(t - t_O)]\right\} \\
R_{O83} &= P_2 P_8 P_9 P_{10} P_{11} P_{12} \\
&= \exp\left[-(60.3 + 16.3\mu)\, \alpha t_O\right] \\
R_{T83} &= P_2 P_7 P_8 P_9 P_{10} P_{11} P_{12} \\
&= \exp\left\{-(160.3 + 16.3\mu)\, [\alpha t_O + \beta(t - t_O)]\right\}
\end{aligned}
$$

B.3.10 Omnidirectional Antenna Subsystem

The omnidirectional antenna used in the relay link primary communication mode and in the direct link and relay link backup mode was assumed to have a discrete probability of working. The following probabilities are used to represent the operation of the lander omnidirectional antenna where it is needed to support an event: R_{F29}, R_{G29}, R_{O79}, R_{O81}, R_{T79}, R_{T81}. These reliabilities assume different values, and the effect on the overall probability of mission success is studied. It was assumed that an omnidirectional antenna on board a direct link lander was the same as one on board a relay link lander even though their operating frequencies, and then their physical dimensions, may be different.

B.3.11 High-Gain Antenna Subsystem

The high-gain antenna subsystem has been discussed in Sections 5.2.4.6. The following functions are used in the models to represent the probability of the high-gain antenna subsystem surviving impact, erecting, and tracking the Earth: R_{F32}, R_{G32}.

B.4 DERIVATION OF MISSION EVENT PROBABILITIES

From the block diagrams of Figs. 12 and 13, and the probability analysis techniques of Section A.1 (Appendix A), the functions representing the probabilities that the mission events will happen may be produced. Note, they are expressed in the subsystem reliabilities. This permits use of alternate subsystem reliability estimates if new information were to become available.

B.4.1 Direct Link Event Probabilities

Probability of deployment and acquisition (Event A)

$$Pr(A) = R_{A1} R_{A2} R_{A3} \left(R_{A4} + R_{A5} - R_{A4} R_{A5}\right)$$

Probability of midcourse trajectory corrections (Event B)

$$Pr(B) = R_{B1} R_{B2} R_{B3} \left(R_{B4} + R_{B5} - R_{B4} R_{B5}\right) R_{B6} R_{B7}$$

Probability of lander separation (Events C and D)

$$
\begin{aligned}
Pr(CD) = {} & R_{C1} R_{C2} R_{C3} \left(R_{C4} + R_{C5} - R_{C4} R_{C5}\right) \\
& \times R_{C6} R_{C7} R_{D21} R_{D22} R_{D23} R_{D24} R_{D28}
\end{aligned}
$$

Probability of entry and landing (Events C and E)

$$
\begin{aligned}
Pr(CE) = {} & R_{C1} R_{C2} R_{C3} \left(R_{C4} + R_{C5} - R_{C4} R_{C5}\right) \\
& \times R_{C6} R_{C7} R_{E21} R_{E22} R_{D23} R_{D24} R_{E25} R_{E28}
\end{aligned}
$$

Probability that primary communications link is established (Events C and F)

$$
\begin{aligned}
Pr(CF) = {} & R_{C1} R_{C2} R_{C3} \left(R_{C4} + R_{C5} - R_{C4} R_{C5}\right) \\
& \times R_{C6} R_{C7} R_{F21} R_{F22} R_{D23} R_{D24} R_{E25} R_{F26} \\
& \left(R_{F28} + R_{F29} R_{F30} - R_{F28} R_{F29} R_{F30}\right) R_{F31} R_{F32}
\end{aligned}
$$

Probability that degraded communications link is established (Events C and F′)

$$
\begin{aligned}
Pr(CF') = {} & R_{C1} R_{C2} R_{C3} \left(R_{C4} + R_{C5} - R_{C4} R_{C5}\right) \\
& \times R_{C6} R_{C7} R_{F21} R_{F22} R_{D23} R_{D24} R_{E25} R_{F26} \\
& \left(R_{F28} + R_{F30} - R_{F28} R_{F30}\right) R_{F29} R_{F41}
\end{aligned}
$$

Probability of surface operations (Events C and G)

$$
\begin{aligned}
Pr(CG) = {} & R_{C1}R_{C2}R_{C3}\,(R_{C4} + R_{C5} - R_{C4}R_{C5}) \\
& \times R_{C6}R_{C7}\;R_{G21}R_{G22}R_{D23}R_{D24}R_{E25}R_{F26}R_{G27} \\
& \quad (R_{G28} + R_{G29}R_{G30} - R_{G28}R_{G29}R_{G30})\,R_{G31}R_{G32}
\end{aligned}
$$

Probability of degraded surface operations (Events C and G′)

$$
\begin{aligned}
Pr(CG') = {} & R_{C1}R_{C2}R_{C3}\,(R_{C4} + R_{C5} - R_{C4}R_{C5}) \\
& \times R_{C6}R_{C7}\;R_{G21}R_{G22}R_{D23}R_{D24}R_{E25}R_{F26}R_{G27} \\
& \quad (R_{G28} + R_{G30} - R_{G28}R_{G30})\,R_{G29}R_{G41}
\end{aligned}
$$

B.4.2 Relay Link Event Probabilities

Probability of deployment and acquisition (Event H)

$$
Pr(H) = R_{H51}R_{H52}R_{H53}\,(R_{H54} + R_{H55} - R_{H54}R_{H55})
$$

Probability of midcourse trajectory corrections (Event I)

$$
Pr(I) = R_{I51}R_{I52}R_{I53}\,(R_{I54} + R_{I55} - R_{I54}R_{I55})\,R_{I56}R_{I57}
$$

Probability of lander separation (Events J and K)

$$
\begin{aligned}
Pr(JK) = {} & R_{J51}R_{J52}R_{J53}\,(R_{J54} + R_{J55} - R_{J54}R_{J55}) \\
& \times R_{J56}R_{J57}R_{K71}R_{K72}R_{K73}R_{K74}R_{K78}
\end{aligned}
$$

Probability of entry and landing (Events J and L)

$$
\begin{aligned}
Pr(JL) = {} & R_{J51}R_{J52}R_{J53}\,(R_{J54} + R_{J55} - R_{J54}R_{J55}) \\
& \times R_{J56}R_{J57}R_{L71}R_{L72}R_{K73}R_{K74}R_{L75}R_{L78}
\end{aligned}
$$

Probability of orbital trim and retro maneuver (Event M)

$$
\begin{aligned}
Pr(M) = {} & R_{M51}R_{M52}R_{M53}\,(R_{M54} + R_{M55} - R_{M54}R_{M55}) \\
& \times R_{M56}R_{M57}
\end{aligned}
$$

Probability that primary communications link is established (Events N and O)

$$
\begin{aligned}
Pr(NO) = {} & R_{N51}R_{N52}R_{N53}\,(R_{N54} + R_{N55} - R_{N54}R_{N55}) \\
& \times R_{N56}R_{M57}R_{N58} \\
& \times R_{O71}R_{O72}R_{K73}R_{K74}R_{L75}R_{O76}\,(R_{O78} + R_{O79}R_{O80} \\
& + R_{O82} - R_{O78}R_{O79}R_{O80} - R_{O78}R_{O82} - R_{O79}R_{O80}R_{O82} \\
& \times R_{O82} + R_{O78}R_{O79}R_{O80}R_{O82})\,R_{O81}R_{O83}
\end{aligned}
$$

Degraded communications link is established (Events J and O′)

$$
\begin{aligned}
Pr\,(JO') = {} & R_{J51}R_{J52}R_{J53}\,(R_{J54} + R_{J55} - R_{J54}R_{J55}) \\
& \times R_{J56}R_{J57}\;R_{O71}R_{O72}R_{K73}R_{K74}R_{L75}R_{O76} \\
& \quad (R_{O78} + R_{O82} - R_{O78}R_{O82})\,R_{O79}R_{O84}
\end{aligned}
$$

Surface operations (Events S and T)

$$
\begin{aligned}
Pr\,(ST) = {} & R_{S51}R_{S52}R_{S53}\,(R_{S54} + R_{S55} - R_{S54}R_{S55}) \\
& \times R_{S56}R_{M57}R_{S58} \\
& \times R_{T71}R_{T72}R_{K73}R_{K74}R_{L75}R_{O76} \\
& \quad (R_{T78} + R_{T79}R_{T80} + R_{T82} \\
& - R_{T78}R_{T79}R_{T80} - R_{T78}R_{T82} - R_{T79}R_{T80}R_{T82} \\
& + R_{T78}R_{T79}R_{T80}R_{T82})\,R_{T77}R_{T81}R_{T83}
\end{aligned}
$$

Degraded surface operations (Events J and T′)

$$
\begin{aligned}
Pr\,(JT') = {} & R_{J51}R_{J52}R_{J53}\,(R_{J54} + R_{J55} - R_{J54}R_{J55}) \\
& \times R_{J56}R_{J57}\;R_{T71}R_{T72}R_{K73}R_{K74}\,R_{L75}R_{O76} \\
& \quad (R_{T78} + R_{T82} - R_{T78}R_{T82})\,R_{T77}R_{T79}R_{T84}
\end{aligned}
$$

APPENDIX C

Reliability Tables

Table C-2. Case 2 reliability results* (trip time = 400 days; $\beta = 0.19$, $\nu = 0.95$, $\alpha = 0.038$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99				
		Relay link	Direct link			
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.93	0.98	0.98	0.98	0.98
Of lander separation		0.47 (0.87)	0.50 (0.87)	0.50 (0.87)	0.50 (0.87)	0.50 (0.87)
Of entry and landing		0.47 (0.86)	0.49 (0.86)	0.49 (0.86)	0.49 (0.86)	0.49 (0.86)
That primary communication link is established		0.32 (0.84)	0.48 (0.84)	0.46 (0.81)	0.44 (0.77)	0.37 (0.64)
Of primary surface operations for:	1 day	0.23 (0.62)	0.36 (0.62)	0.34 (0.60)	0.32 (0.56)	0.27 (0.47)
	1 wk	0.22 (0.60)	0.35 (0.61)	0.33 (0.58)	0.31 (0.55)	0.26 (0.46)
	1 mo	0.18 (0.55)	0.32 (0.55)	0.30 (0.53)	0.27 (0.50)	0.24 (0.41)
	2 mo	0.14 (0.48)	0.28 (0.49)	0.27 (0.47)	0.25 (0.44)	0.21 (0.37)
	3 mo	0.11 (0.42)	0.25 (0.43)	0.24 (0.42)	0.22 (0.40)	0.19 (0.33)
	4 mo	0.09 (0.37)	0.22 (0.38)	0.21 (0.37)	0.20 (0.35)	0.17 (0.29)
	5 mo	0.07 (0.33)	0.19 (0.34)	0.19 (0.33)	0.18 (0.31)	0.15 (0.26)
	6 mo	0.05 (0.29)	0.17 (0.30)	0.17 (0.29)	0.16 (0.27)	0.13 (0.23)
	9 mo	0.02 (0.20)	0.12 (0.21)	0.11 (0.20)	0.11 (0.19)	0.09 (0.16)
	12 mo	0.01 (0.14)	0.08 (0.14)	0.08 (0.14)	0.07 (0.13)	0.06 (0.11)
That degraded communication link is established		0.46 (0.84)	0.48 (0.84)	0.48 (0.84)	0.48 (0.84)	0.48 (0.84)
Of degraded surface operations for:	1 day	0.34 (0.62)	0.36 (0.62)	0.36 (0.62)	0.36 (0.62)	0.36 (0.62)
	1 wk	0.33 (0.61)	0.35 (0.61)	0.35 (0.61)	0.35 (0.61)	0.35 (0.61)
	1 mo	0.30 (0.55)	0.32 (0.55)	0.32 (0.55)	0.32 (0.55)	0.32 (0.55)
	2 mo	0.27 (0.49)	0.28 (0.49)	0.28 (0.49)	0.28 (0.49)	0.28 (0.49)
	3 mo	0.23 (0.43)	0.25 (0.43)	0.25 (0.43)	0.25 (0.43)	0.25 (0.43)
	4 mo	0.21 (0.38)	0.22 (0.38)	0.22 (0.38)	0.22 (0.38)	0.22 (0.38)
	5 mo	0.18 (0.34)	0.19 (0.34)	0.19 (0.34)	0.19 (0.34)	0.19 (0.34)
	6 mo	0.16 (0.30)	0.17 (0.30)	0.17 (0.30)	0.17 (0.30)	0.17 (0.30)
	9 mo	0.11 (0.21)	0.12 (0.21)	0.12 (0.21)	0.12 (0.21)	0.12 (0.21)
	12 mo	0.08 (0.14)	0.08 (0.14)	0.08 (0.14)	0.08 (0.14)	0.08 (0.14)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	6	8	8	7	6
	1 wk	39	59	56	53	44
	1 mo	150	240	230	218	181
	2 mo	268	452	434	411	343
	3 mo	359	640	614	582	485
	4 mo	430	806	774	733	611
	5 mo	486	953	915	866	722
	6 mo	529	1083	1040	984	821
	9 mo	609	1391	1335	1264	1054
	12 mo	646	1603	1539	1457	1215

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-1. Case 1 reliability results* (trip time = 200 days; $\beta = 0.19$, $\alpha = 0.038$, $\nu = 0.95$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.93	0.98	0.98	0.98	0.98	0.93	0.98	0.98	0.98
Of lander separation		0.63 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.63 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)
Of entry and landing		0.62 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.62 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)
That primary communication link is established		0.51 (0.85)	0.65 (0.85)	0.62 (0.82)	0.59 (0.78)	0.49 (0.65)	0.49 (0.82)	0.62 (0.82)	0.59 (0.78)	0.49 (0.65)
Of primary surface operations for:	1 day	0.43 (0.73)	0.55 (0.73)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)	0.41 (0.70)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)
	1 wk	0.41 (0.71)	0.54 (0.71)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)	0.40 (0.68)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)
	1 mo	0.34 (0.64)	0.49 (0.65)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)	0.33 (0.62)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)
	2 mo	0.27 (0.57)	0.43 (0.58)	0.42 (0.55)	0.40 (0.52)	0.33 (0.44)	0.26 (0.55)	0.42 (0.55)	0.39 (0.52)	0.33 (0.44)
	3 mo	0.21 (0.50)	0.38 (0.51)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)	0.20 (0.48)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)
	4 mo	0.16 (0.44)	0.34 (0.45)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)	0.16 (0.43)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)
	5 mo	0.13 (0.39)	0.30 (0.40)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)	0.12 (0.38)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)
	6 mo	0.10 (0.35)	0.27 (0.35)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)	0.09 (0.33)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)
	9 mo	0.05 (0.24)	0.18 (0.24)	0.18 (0.23)	0.17 (0.22)	0.14 (0.19)	0.04 (0.23)	0.18 (0.23)	0.17 (0.22)	0.14 (0.18)
	12 mo	0.02 (0.16)	0.13 (0.17)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)	0.02 (0.16)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)
That degraded communication link is established		0.61 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.59 (0.82)	0.62 (0.82)	0.62 (0.82)	0.62 (0.82)
Of degraded surface operations for:	1 day	0.52 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.50 (0.70)	0.53 (0.70)	0.53 (0.70)	0.53 (0.70)
	1 wk	0.51 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.49 (0.68)	0.52 (0.68)	0.52 (0.68)	0.52 (0.68)
	1 mo	0.47 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.45 (0.62)	0.47 (0.62)	0.47 (0.62)	0.47 (0.62)
	2 mo	0.41 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.40 (0.55)	0.42 (0.55)	0.42 (0.55)	0.42 (0.55)
	3 mo	0.37 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.35 (0.49)	0.37 (0.49)	0.37 (0.49)	0.37 (0.49)
	4 mo	0.32 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.31 (0.43)	0.33 (0.43)	0.33 (0.43)	0.33 (0.43)
	5 mo	0.29 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.28 (0.38)	0.29 (0.38)	0.29 (0.38)	0.29 (0.38)
	6 mo	0.25 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.24 (0.34)	0.26 (0.34)	0.26 (0.34)	0.26 (0.34)
	9 mo	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.17 (0.23)	0.18 (0.23)	0.18 (0.23)	0.18 (0.23)
	12 mo	0.12 (0.17)	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	10	13	13	12	10	10	13	12	10
	1 wk	72	91	87	83	69	69	87	83	69
	1 mo	280	373	358	339	282	268	357	339	282
	2 mo	498	703	674	639	532	478	674	638	532
	3 mo	668	995	955	905	754	641	954	904	753
	4 mo	800	1254	1203	1140	950	768	1202	1139	949
	5 mo	904	1483	1423	1348	1123	867	1421	1346	1122
	6 mo	984	1685	1617	1532	1277	945	1615	1530	1275
	9 mo	1134	2164	2077	1967	1639	1088	2073	1964	1637
	12 mo	1205	2495	2394	2268	1890	1156	2390	2264	1887

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-3. Case 3 reliability results* (trip time = 200 days; $\beta = 0.5$, $\nu = 0.95$, $\alpha = 0.1$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
			Direct link					Direct link		
		Relay link	Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75	Relay link	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.90	0.94	0.94	0.94	0.94	0.90	0.94	0.94	0.94
Of lander separation		0.39 (0.86)	0.41 (0.86)	0.41 (0.86)	0.41 (0.86)	0.41 (0.86)	0.39 (0.86)	0.41 (0.86)	0.41 (0.86)	0.41 (0.86)
Of entry and landing		0.39 (0.85)	0.40 (0.85)	0.40 (0.85)	0.40 (0.85)	0.40 (0.85)	0.39 (0.85)	0.40 (0.85)	0.40 (0.85)	0.40 (0.85)
That primary communication link is established		0.23 (0.83)	0.40 (0.84)	0.38 (0.80)	0.36 (0.76)	0.30 (0.63)	0.23 (0.79)	0.38 (0.80)	0.36 (0.76)	0.30 (0.63)
Of primary surface operations for:	1 day	0.15 (0.55)	0.26 (0.55)	0.25 (0.53)	0.24 (0.50)	0.20 (0.42)	0.15 (0.53)	0.25 (0.53)	0.24 (0.50)	0.20 (0.42)
	1 wk	0.13 (0.51)	0.25 (0.52)	0.24 (0.50)	0.23 (0.47)	0.19 (0.39)	0.13 (0.49)	0.24 (0.50)	0.23 (0.47)	0.19 (0.39)
	1 mo	0.08 (0.40)	0.19 (0.41)	0.19 (0.39)	0.18 (0.37)	0.15 (0.31)	0.08 (0.38)	0.19 (0.39)	0.18 (0.37)	0.15 (0.31)
	2 mo	0.04 (0.29)	0.14 (0.29)	0.13 (0.28)	0.13 (0.27)	0.11 (0.22)	0.04 (0.28)	0.13 (0.28)	0.13 (0.27)	0.11 (0.22)
	3 mo	0.02 (0.21)	0.10 (0.21)	0.10 (0.20)	0.09 (0.19)	0.08 (0.16)	0.02 (0.20)	0.10 (0.20)	0.09 (0.19)	0.08 (0.16)
	4 mo	0.01 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.14)	0.06 (0.12)	0.01 (0.14)	0.07 (0.15)	0.07 (0.14)	0.06 (0.12)
	5 mo	0.01 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.10)	0.04 (0.08)	0.01 (0.10)	0.05 (0.11)	0.05 (0.10)	0.04 (0.08)
	6 mo	0.00 (0.08)	0.04 (0.08)	0.04 (0.08)	0.03 (0.07)	0.03 (0.06)	0.00 (0.07)	0.04 (0.08)	0.03 (0.07)	0.03 (0.06)
	9 mo	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)
	12 mo	0.00 (0.01)	0.01 (0.01)	0.01 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.01 (0.01)	0.00 (0.01)	0.00 (0.01)
That degraded communication link is established		0.38 (0.84)	0.40 (0.84)	0.40 (0.84)	0.40 (0.84)	0.40 (0.84)	0.36 (0.80)	0.38 (0.80)	0.38 (0.80)	0.30 (0.80)
Of degraded surface operations for:	1 day	0.25 (0.55)	0.26 (0.55)	0.26 (0.55)	0.26 (0.55)	0.26 (0.55)	0.24 (0.53)	0.25 (0.53)	0.25 (0.53)	0.25 (0.53)
	1 wk	0.24 (0.52)	0.25 (0.52)	0.25 (0.52)	0.25 (0.52)	0.25 (0.52)	0.23 (0.50)	0.24 (0.50)	0.24 (0.50)	0.24 (0.50)
	1 mo	0.18 (0.41)	0.19 (0.41)	0.19 (0.41)	0.19 (0.41)	0.19 (0.41)	0.18 (0.39)	0.19 (0.39)	0.19 (0.39)	0.19 (0.39)
	2 mo	0.13 (0.29)	0.14 (0.29)	0.14 (0.29)	0.14 (0.29)	0.14 (0.29)	0.13 (0.28)	0.13 (0.28)	0.13 (0.28)	0.13 (0.28)
	3 mo	0.10 (0.21)	0.10 (0.21)	0.10 (0.21)	0.10 (0.21)	0.10 (0.21)	0.09 (0.20)	0.10 (0.20)	0.20 (0.20)	0.10 (0.20)
	4 mo	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)	0.07 (0.15)
	5 mo	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)
	6 mo	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)	0.04 (0.08)
	9 mo	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)
	12 mo	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.00 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	4	6	6	6	5	4	6	6	5
	1 wk	24	43	41	39	32	23	41	39	32
	1 mo	83	163	157	149	124	79	156	148	124
	2 mo	126	282	270	256	214	121	270	256	213
	3 mo	148	368	353	334	279	142	352	333	278
	4 mo	159	430	412	391	326	153	411	390	325
	5 mo	165	475	455	431	360	158	454	430	359
	6 mo	168	507	487	461	380	161	485	460	383
	9 mo	171	560	537	509	424	164	535	507	423
	12 mo	171	579	556	527	439	164	554	525	

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-4. Case 4 reliability results* (trip time = 200 days; $\beta = 1.0$, $\nu = 0.95$, $\alpha = 0.2$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.85	0.89	0.89	0.89	0.89	0.85	0.89	0.89	0.89
Of lander separation		0.18 (0.83)	0.19 (0.83)	0.19 (0.83)	0.19 (0.83)	0.19 (0.83)	0.18 (0.83)	0.19 (0.83)	0.19 (0.83)	0.19 (0.83)
Of entry and landing		0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)
That primary communication link is established		0.07 (0.79)	0.18 (0.79)	0.18 (0.78)	0.17 (0.73)	0.14 (0.61)	0.06 (0.76)	0.18 (0.77)	0.17 (0.73)	0.14 (0.61)
Of primary surface operations for:	1 day	0.03 (0.35)	0.08 (0.35)	0.08 (0.34)	0.07 (0.32)	0.06 (0.27)	0.03 (0.33)	0.08 (0.34)	0.07 (0.32)	0.06 (0.27)
	1 wk	0.02 (0.30)	0.07 (0.31)	0.07 (0.30)	0.06 (0.28)	0.05 (0.24)	0.02 (0.29)	0.07 (0.30)	0.06 (0.28)	0.05 (0.23)
	1 mo	0.01 (0.18)	0.04 (0.19)	0.04 (0.18)	0.04 (0.17)	0.03 (0.14)	0.01 (0.18)	0.04 (0.18)	0.04 (0.17)	0.03 (0.14)
	2 mo	0.00 (0.09)	0.02 (0.10)	0.02 (0.09)	0.02 (0.09)	0.02 (0.07)	0.00 (0.09)	0.02 (0.09)	0.02 (0.09)	0.02 (0.07)
	3 mo	0.00 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.04)	0.00 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.04)
	4 mo	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.01 (0.03)	0.01 (0.02)	0.00 (0.02)
	5 mo	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)
	6 mo	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.01 (0.01)	0.00 (0.01)	0.00 (0.00)	0.00 (0.01)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
That degraded communication link is established		0.17 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.18 (0.81)	0.17 (0.78)	0.18 (0.78)	0.18 (0.78)	0.18 (0.78)
Of degraded surface operations for:	1 day	0.08 (0.35)	0.08 (0.35)	0.08 (0.35)	0.08 (0.35)	0.08 (0.35)	0.07 (0.34)	0.08 (0.34)	0.08 (0.34)	0.08 (0.34)
	1 wk	0.07 (0.31)	0.07 (0.31)	0.07 (0.31)	0.07 (0.31)	0.07 (0.31)	0.06 (0.30)	0.07 (0.30)	0.07 (0.30)	0.07 (0.30)
	1 mo	0.04 (0.19)	0.04 (0.19)	0.04 (0.19)	0.04 (0.19)	0.04 (0.19)	0.04 (0.18)	0.04 (0.18)	0.04 (0.18)	0.04 (0.18)
	2 mo	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)	0.02 (0.10)
	3 mo	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)	0.01 (0.05)
	4 mo	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)
	5 mo	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)
	6 mo	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)	0.00 (0.01)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	1	2	2	2	2	1	2	2	1
	1 wk	4	13	12	12	10	4	12	12	10
	1 mo	12	43	41	39	33	11	41	39	33
	2 mo	15	66	63	60	50	14	63	60	50
	3 mo	16	78	74	71	59	15	74	70	59
	4 mo	16	84	80	76	63	15	80	76	63
	5 mo	16	87	83	79	66	15	83	79	66
	6 mo	16	89	85	80	67	15	85	80	67
	9 mo	16	90	86	82	68	15	86	82	68
	12 mo	16	90	87	82	68	15	86	82	68

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-5. Case 5 reliability results* (trip time = 200 days; $\beta = 0.19$, $\nu = 0.95$, $\alpha = 0.015$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.93	0.98	0.98	0.98	0.98	0.93	0.98	0.98	0.98
Of lander separation		0.64 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)	0.64 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)
Of entry and landing		0.63 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)	0.63 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)
That primary communication link is established		0.57 (0.86)	0.65 (0.86)	0.63 (0.83)	0.59 (0.78)	0.49 (0.65)	0.55 (0.82)	0.63 (0.83)	0.59 (0.78)	0.49 (0.65)
Of primary surface operations for:	1 day	0.53 (0.81)	0.61 (0.81)	0.59 (0.77)	0.56 (0.73)	0.46 (0.61)	0.51 (0.77)	0.59 (0.77)	0.56 (0.73)	0.46 (0.61)
	1 wk	0.51 (0.79)	0.60 (0.79)	0.57 (0.76)	0.54 (0.72)	0.45 (0.60)	0.49 (0.75)	0.57 (0.76)	0.54 (0.72)	0.45 (0.60)
	1 mo	0.42 (0.71)	0.54 (0.72)	0.52 (0.69)	0.49 (0.65)	0.41 (0.54)	0.40 (0.69)	0.52 (0.69)	0.49 (0.65)	0.41 (0.54)
	2 mo	0.33 (0.63)	0.48 (0.64)	0.46 (0.61)	0.44 (0.58)	0.36 (0.48)	0.31 (0.61)	0.46 (0.61)	0.44 (0.58)	0.36 (0.48)
	3 mo	0.26 (0.56)	0.43 (0.56)	0.41 (0.54)	0.39 (0.51)	0.32 (0.43)	0.25 (0.53)	0.41 (0.54)	0.39 (0.51)	0.32 (0.43)
	4 mo	0.20 (0.50)	0.38 (0.50)	0.36 (0.48)	0.34 (0.45)	0.29 (0.38)	0.19 (0.47)	0.36 (0.48)	0.34 (0.45)	0.29 (0.38)
	5 mo	0.16 (0.43)	0.33 (0.44)	0.32 (0.42)	0.30 (0.40)	0.25 (0.33)	0.15 (0.42)	0.32 (0.42)	0.30 (0.40)	0.25 (0.33)
	6 mo	0.12 (0.38)	0.30 (0.39)	0.28 (0.37)	0.27 (0.35)	0.22 (0.30)	0.12 (0.37)	0.28 (0.37)	0.27 (0.35)	0.22 (0.30)
	9 mo	0.06 (0.26)	0.20 (0.27)	0.20 (0.26)	0.19 (0.26)	0.15 (0.20)	0.05 (0.25)	0.20 (0.26)	0.19 (0.25)	0.15 (0.20)
	12 mo	0.03 (0.18)	0.14 (0.19)	0.14 (0.18)	0.13 (0.17)	0.11 (0.14)	0.03 (0.17)	0.14 (0.18)	0.13 (0.17)	0.11 (0.14)
That degraded communication link is established		0.62 (0.86)	0.65 (0.86)	0.65 (0.86)	0.65 (0.86)	0.65 (0.86)	0.59 (0.83)	0.63 (0.83)	0.63 (0.83)	0.63 (0.83)
Of degraded surface operations for:	1 day	0.58 (0.81)	0.61 (0.81)	0.61 (0.81)	0.61 (0.81)	0.61 (0.81)	0.56 (0.77)	0.59 (0.77)	0.59 (0.77)	0.59 (0.77)
	1 wk	0.57 (0.79)	0.60 (0.79)	0.60 (0.79)	0.60 (0.79)	0.60 (0.79)	0.54 (0.76)	0.57 (0.76)	0.57 (0.76)	0.57 (0.76)
	1 mo	0.52 (0.72)	0.54 (0.72)	0.54 (0.72)	0.54 (0.72)	0.54 (0.72)	0.49 (0.69)	0.52 (0.69)	0.52 (0.69)	0.52 (0.69)
	2 mo	0.46 (0.64)	0.48 (0.64)	0.48 (0.64)	0.48 (0.64)	0.48 (0.64)	0.44 (0.61)	0.46 (0.61)	0.46 (0.61)	0.46 (0.61)
	3 mo	0.40 (0.56)	0.43 (0.56)	0.43 (0.56)	0.43 (0.56)	0.43 (0.56)	0.39 (0.54)	0.41 (0.54)	0.41 (0.54)	0.41 (0.54)
	4 mo	0.36 (0.50)	0.38 (0.50)	0.38 (0.50)	0.38 (0.50)	0.38 (0.50)	0.34 (0.48)	0.36 (0.48)	0.36 (0.48)	0.36 (0.48)
	5 mo	0.32 (0.44)	0.33 (0.44)	0.33 (0.44)	0.33 (0.44)	0.33 (0.44)	0.30 (0.42)	0.32 (0.42)	0.32 (0.42)	0.32 (0.42)
	6 mo	0.28 (0.39)	0.30 (0.39)	0.30 (0.39)	0.30 (0.39)	0.30 (0.39)	0.27 (0.37)	0.28 (0.37)	0.28 (0.37)	0.28 (0.37)
	9 mo	0.19 (0.27)	0.20 (0.27)	0.20 (0.27)	0.20 (0.27)	0.20 (0.27)	0.19 (0.26)	0.20 (0.26)	0.20 (0.26)	0.26 (0.26)
	12 mo	0.13 (0.79)	0.14 (0.19)	0.14 (0.19)	0.14 (0.19)	0.14 (0.19)	0.14 (0.18)	0.13 (0.18)	0.13 (0.18)	0.13 (0.18)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	13	15	14	13	11	12	14	13	16
	1 wk	88	101	97	92	76	84	97	92	76
	1 mo	343	413	396	375	313	329	396	375	312
	2 mo	611	778	747	707	589	586	746	707	589
	3 mo	819	1102	1057	1001	834	786	1056	1001	834
	4 mo	982	1388	1332	1262	1051	942	1331	1261	1051
	5 mo	1109	1642	1575	1492	1244	1064	1574	1491	1242
	6 mo	1208	1867	1791	1697	1414	1159	1789	1694	1412
	9 mo	1392	2396	2300	2178	1815	1335	2296	2175	1813
	12 mo	1478	2763	2651	2512	2093	1419	2647	2508	2090

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-6. Case 6 reliability results* (trip time = 200 days; $\beta = 0.50$, $\nu = 0.95$, $\alpha = 0.015$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.90	0.95	0.95	0.95	0.95	0.90	0.95	0.95	0.95
Of lander separation		0.40 (0.89)	0.43 (0.89)	0.43 (0.89)	0.43 (0.89)	0.43 (0.89)	0.40 (0.89)	0.43 (0.89)	0.43 (0.89)	0.43 (0.89)
Of entry and landing		0.40 (0.88)	0.42 (0.88)	0.42 (0.88)	0.42 (0.88)	0.42 (0.88)	0.40 (0.88)	0.42 (0.88)	0.42 (0.88)	0.42 (0.88)
That primary communication link is established		0.35 (0.86)	0.41 (0.86)	0.40 (0.83)	0.38 (0.78)	0.31 (0.65)	0.34 (0.82)	0.40 (0.83)	0.38 (0.78)	0.31 (0.65)
Of primary surface operations for:	1 day	0.33 (0.80)	0.38 (0.80)	0.37 (0.77)	0.35 (0.73)	0.29 (0.61)	0.31 (0.77)	0.37 (0.77)	0.35 (0.73)	0.29 (0.61)
	1 wk	0.27 (0.75)	0.36 (0.75)	0.35 (0.72)	0.33 (0.68)	0.27 (0.57)	0.28 (0.72)	0.35 (0.72)	0.33 (0.68)	0.27 (0.57)
	1 mo	0.17 (0.58)	0.28 (0.59)	0.27 (0.56)	0.26 (0.54)	0.21 (0.45)	0.17 (0.56)	0.27 (0.56)	0.26 (0.53)	0.21 (0.45)
	2 mo	0.09 (0.42)	0.20 (0.43)	0.20 (0.41)	0.19 (0.39)	0.16 (0.32)	0.09 (0.40)	0.20 (0.41)	0.19 (0.39)	0.15 (0.32)
	3 mo	0.05 (0.30)	0.15 (0.31)	0.14 (0.30)	0.14 (0.28)	0.11 (0.23)	0.04 (0.29)	0.14 (0.30)	0.13 (0.28)	0.11 (0.23)
	4 mo	0.02 (0.22)	0.11 (0.22)	0.10 (0.21)	0.10 (0.20)	0.08 (0.17)	0.02 (0.21)	0.10 (0.21)	0.10 (0.20)	0.08 (0.17)
	5 mo	0.01 (0.16)	0.08 (0.16)	0.07 (0.16)	0.07 (0.15)	0.06 (0.12)	0.01 (0.15)	0.07 (0.15)	0.07 (0.15)	0.06 (0.12)
	6 mo	0.01 (0.11)	0.06 (0.12)	0.05 (0.11)	0.05 (0.11)	0.04 (0.09)	0.01 (0.11)	0.05 (0.11)	0.05 (0.11)	0.04 (0.09)
	9 mo	0.00 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.03)	0.00 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.03)
	12 mo	0.00 (0.02)	0.01 (0.02)	0.01 (0.02)	0.01 (0.01)	0.01 (0.01)	0.00 (0.01)	0.01 (0.02)	0.01 (0.01)	0.01 (0.01)
That degraded communication link is established		0.39 (0.86)	0.41 (0.86)	0.41 (0.86)	0.41 (0.86)	0.41 (0.86)	0.38 (0.83)	0.40 (0.83)	0.40 (0.83)	0.40 (0.83)
Of degraded surface operations for:	1 day	0.36 (0.80)	0.38 (0.80)	0.38 (0.80)	0.38 (0.80)	0.38 (0.80)	0.35 (0.77)	0.37 (0.77)	0.37 (0.77)	0.37 (0.77)
	1 wk	0.34 (0.75)	0.36 (0.75)	0.36 (0.75)	0.36 (0.75)	0.36 (0.75)	0.33 (0.72)	0.35 (0.72)	0.35 (0.72)	0.35 (0.72)
	1 mo	0.27 (0.59)	0.28 (0.59)	0.28 (0.59)	0.28 (0.59)	0.28 (0.59)	0.26 (0.57)	0.27 (0.57)	0.27 (0.57)	0.27 (0.57)
	2 mo	0.19 (0.43)	0.21 (0.43)	0.21 (0.43)	0.21 (0.43)	0.21 (0.43)	0.19 (0.41)	0.20 (0.41)	0.20 (0.41)	0.20 (0.41)
	3 mo	0.14 (0.31)	0.15 (0.31)	0.15 (0.31)	0.15 (0.31)	0.15 (0.31)	0.13 (0.30)	0.14 (0.30)	0.14 (0.30)	0.14 (0.30)
	4 mo	0.10 (0.22)	0.11 (0.22)	0.11 (0.22)	0.11 (0.22)	0.11 (0.22)	0.10 (0.22)	0.10 (0.22)	0.10 (0.22)	0.10 (0.22)
	5 mo	0.07 (0.16)	0.08 (0.16)	0.08 (0.16)	0.08 (0.16)	0.08 (0.16)	0.07 (0.16)	0.07 (0.16)	0.07 (0.16)	0.07 (0.16)
	6 mo	0.05 (0.12)	0.06 (0.12)	0.06 (0.12)	0.06 (0.12)	0.06 (0.12)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)	0.05 (0.11)
	9 mo	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)	0.02 (0.04)
	12 mo	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.01)	0.01 (0.02)	0.01 (0.02)	0.01 (0.02)	0.01 (0.02)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	8	9	9	8	7	8	9	8	7
	1 wk	52	62	60	57	47	50	60	57	47
	1 mo	176	238	228	216	180	169	228	216	180
	2 mo	268	411	394	373	311	257	394	373	311
	3 mo	315	536	514	487	406	302	514	487	405
	4 mo	339	627	601	570	475	326	600	569	474
	5 mo	352	692	664	629	524	338	663	628	524
	6 mo	358	740	710	672	560	344	708	671	559
	9 mo	364	817	784	723	619	349	782	741	617
	12 mo	365	846	811	769	641	350	809	767	639

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-7. Case 7 reliability results* (trip time = 200 days; $\beta = 1.0$, $\nu = 0.95$, $\alpha = 0.015$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.86	0.90	0.90	0.09	0.90	0.86	0.90	0.90	0.90
Of lander separation		0.19 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)	0.19 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)
Of entry and landing		0.18 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)	0.18 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)
That primary communication link is established		0.16 (0.86)	0.19 (0.86)	0.19 (0.83)	0.18 (0.78)	0.15 (0.65)	0.16 (0.82)	0.19 (0.83)	0.18 (0.78)	0.15 (0.65)
Of primary surface operations for:	1 day	0.15 (0.79)	0.18 (0.79)	0.17 (0.76)	0.17 (0.72)	0.14 (0.60)	0.14 (0.76)	0.17 (0.76)	0.17 (0.72)	0.14 (0.60)
	1 wk	0.11 (0.69)	0.16 (0.70)	0.15 (0.67)	0.15 (0.63)	0.12 (0.53)	0.11 (0.67)	0.15 (0.67)	0.15 (0.63)	0.12 (0.53)
	1 mo	0.04 (0.42)	0.10 (0.43)	0.09 (0.41)	0.09 (0.39)	0.07 (0.32)	0.04 (0.40)	0.09 (0.41)	0.09 (0.39)	0.07 (0.32)
	2 mo	0.01 (0.22)	0.05 (0.22)	0.05 (0.21)	0.05 (0.20)	0.04 (0.17)	0.01 (0.21)	0.05 (0.21)	0.05 (0.20)	0.04 (0.17)
	3 mo	0.00 (0.11)	0.03 (0.12)	0.03 (0.11)	0.02 (0.10)	0.02 (0.09)	0.00 (0.11)	0.03 (0.11)	0.02 (0.11)	0.02 (0.09)
	4 mo	0.00 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.05)	0.00 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.05)
	5 mo	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)
	6 mo	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.01)	0.00 (0.01)	0.00 (0.02)	0.00 (0.02)	0.00 (0.01)	0.00 (0.01)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
That degraded communication link is established		0.19 (0.86)	0.20 (0.86)	0.20 (0.86)	0.20 (0.86)	0.20 (0.86)	0.18 (0.83)	0.19 (0.83)	0.19 (0.83)	0.19 (0.83)
Of degraded surface operations for:	1 day	0.17 (0.79)	0.18 (0.79)	0.18 (0.79)	0.18 (0.79)	0.18 (0.79)	0.16 (0.76)	0.17 (0.76)	0.17 (0.76)	0.17 (0.76)
	1 wk	0.15 (0.70)	0.16 (0.70)	0.16 (0.70)	0.16 (0.70)	0.16 (0.70)	0.14 (0.67)	0.15 (0.67)	0.15 (0.67)	0.15 (0.67)
	1 mo	0.09 (0.43)	0.10 (0.43)	0.10 (0.43)	0.10 (0.43)	0.10 (0.43)	0.09 (0.41)	0.09 (0.41)	0.09 (0.41)	0.09 (0.41)
	2 mo	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.21)	0.05 (0.21)	0.05 (0.21)
	3 mo	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.03 (0.11)	0.03 (0.11)	0.03 (0.11)	0.03 (0.11)
	4 mo	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)
	5 mo	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)
	6 mo	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	4	4	4	4	3	3	4	4	3
	1 wk	22	29	28	26	22	21	28	26	22
	1 mo	62	98	94	89	74	59	94	89	74
	2 mo	78	150	144	136	113	75	144	136	113
	3 mo	82	177	170	161	134	79	169	160	134
	4 mo	83	191	183	173	145	80	183	173	144
	5 mo	84	198	190	180	150	80	190	180	150
	6 mo	84	202	194	184	153	80	193	183	153
	9 mo	84	206	197	187	156	80	197	186	155
	12 mo	84	206	198	187	156	80	197	187	156

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-8. Case 8 reliability results* (trip time = 200 days; $\beta = 0.19$, $\nu = 0.95$, $\alpha = 0.038$, $\mu = 0.2$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
			Direct link					Direct link		
		Relay link	Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75	Relay link	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.93	0.98	0.98	0.98	0.98	0.93	0.98	0.98	0.98
Of lander separation		0.63 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.63 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)
Of entry and landing		0.62 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.62 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)
That primary communication link is established		0.51 (0.85)	0.65 (0.85)	0.62 (0.82)	0.58 (0.78)	0.49 (0.65)	0.49 (0.82)	0.62 (0.82)	0.59 (0.78)	0.49 (0.65)
Of primary surface operations for:	1 day	0.44 (0.73)	0.55 (0.73)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)	0.42 (0.70)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)
	1 wk	0.41 (0.71)	0.54 (0.71)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)	0.40 (0.68)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)
	1 mo	0.35 (0.65)	0.49 (0.65)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)	0.33 (0.62)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)
	2 mo	0.27 (0.57)	0.43 (0.58)	0.42 (0.55)	0.40 (0.52)	0.33 (0.44)	0.26 (0.55)	0.42 (0.55)	0.39 (0.52)	0.33 (0.44)
	3 mo	0.21 (0.51)	0.38 (0.51)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)	0.20 (0.49)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)
	4 mo	0.17 (0.45)	0.34 (0.45)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)	0.17 (0.43)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)
	5 mo	0.13 (0.40)	0.30 (0.40)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)	0.12 (0.38)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)
	6 mo	0.10 (0.35)	0.27 (0.35)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)	0.10 (0.34)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)
	9 mo	0.04 (0.24)	0.18 (0.24)	0.18 (0.23)	0.17 (0.22)	0.14 (0.19)	0.05 (0.23)	0.18 (0.23)	0.17 (0.22)	0.14 (0.19)
	12 mo	0.02 (0.17)	0.13 (0.17)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)	0.02 (0.16)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)
That degraded communication link is established		0.61 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.59 (0.82)	0.62 (0.82)	0.62 (0.82)	0.62 (0.82)
Of degraded surface operations for:	1 day	0.52 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.50 (0.70)	0.53 (0.70)	0.53 (0.70)	0.53 (0.70)
	1 wk	0.51 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.49 (0.68)	0.52 (0.68)	0.52 (0.68)	0.52 (0.68)
	1 mo	0.47 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.45 (0.62)	0.47 (0.62)	0.47 (0.62)	0.47 (0.62)
	2 mo	0.41 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.40 (0.55)	0.42 (0.55)	0.42 (0.55)	0.42 (0.55)
	3 mo	0.37 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.35 (0.49)	0.37 (0.49)	0.37 (0.49)	0.37 (0.49)
	4 mo	0.32 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.31 (0.43)	0.33 (0.43)	0.33 (0.43)	0.33 (0.43)
	5 mo	0.29 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.28 (0.38)	0.29 (0.38)	0.29 (0.38)	0.29 (0.38)
	6 mo	0.25 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.24 (0.34)	0.26 (0.34)	0.26 (0.34)	0.26 (0.34)
	9 mo	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.17 (0.23)	0.18 (0.23)	0.18 (0.23)	0.18 (0.23)
	12 mo	0.12 (0.17)	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	11	13	13	12	10	10	13	12	10
	1 wk	72	91	87	83	69	69	87	83	69
	1 mo	281	373	358	339	282	270	357	339	282
	2 mo	502	703	674	639	532	481	674	638	532
	3 mo	674	995	955	905	754	647	954	904	753
	4 mo	809	1254	1203	1140	950	776	1202	1139	949
	5 mo	915	1483	1423	1348	1123	878	1421	1346	1122
	6 mo	997	1685	1617	1532	1277	957	1615	1530	1275
	9 mo	1152	2164	2077	1967	1639	1105	2073	1964	1637
	12 mo	1226	2495	2394	2268	1890	1176	2390	2264	1887

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-9. Case 9 reliability results* (trip time = 200 days; $\beta = 0.19$, $\nu = 0.95$, $\alpha = 0.015$, $\mu = 0.2$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.93	0.98	0.98	0.98	0.98	0.93	0.98	0.98	0.98
Of lander separation		0.64 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)	0.64 (0.89)	0.67 (0.89)	0.67 (0.89)	0.67 (0.89)
Of entry and landing		0.63 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)	0.63 (0.88)	0.66 (0.88)	0.66 (0.88)	0.66 (0.88)
That primary communication link is established		0.57 (0.86)	0.65 (0.86)	0.63 (0.83)	0.59 (0.78)	0.49 (0.65)	0.55 (0.83)	0.63 (0.83)	0.59 (0.78)	0.49 (0.65)
Of primary surface operations for:	1 day	0.53 (0.81)	0.61 (0.81)	0.59 (0.77)	0.55 (0.73)	0.46 (0.61)	0.51 (0.77)	0.59 (0.77)	0.56 (0.73)	0.46 (0.61)
	1 wk	0.51 (0.79)	0.60 (0.79)	0.57 (0.76)	0.54 (0.72)	0.45 (0.60)	0.49 (0.76)	0.57 (0.76)	0.54 (0.72)	0.45 (0.60)
	1 mo	0.42 (0.72)	0.54 (0.72)	0.52 (0.69)	0.49 (0.65)	0.41 (0.54)	0.41 (0.69)	0.52 (0.69)	0.49 (0.65)	0.41 (0.54)
	2 mo	0.33 (0.63)	0.48 (0.64)	0.46 (0.61)	0.44 (0.58)	0.36 (0.48)	0.32 (0.61)	0.46 (0.61)	0.44 (0.58)	0.36 (0.48)
	3 mo	0.26 (0.56)	0.43 (0.56)	0.41 (0.54)	0.39 (0.51)	0.32 (0.42)	0.25 (0.54)	0.41 (0.54)	0.39 (0.51)	0.32 (0.43)
	4 mo	0.20 (0.50)	0.38 (0.50)	0.36 (0.48)	0.34 (0.45)	0.29 (0.38)	0.19 (0.48)	0.36 (0.48)	0.34 (0.45)	0.29 (0.38)
	5 mo	0.16 (0.44)	0.33 (0.44)	0.32 (0.42)	0.30 (0.40)	0.25 (0.33)	0.15 (0.42)	0.32 (0.42)	0.30 (0.40)	0.25 (0.33)
	6 mo	0.12 (0.39)	0.30 (0.39)	0.28 (0.37)	0.27 (0.35)	0.22 (0.30)	0.12 (0.37)	0.28 (0.37)	0.27 (0.35)	0.22 (0.30)
	9 mo	0.06 (0.27)	0.20 (0.27)	0.20 (0.26)	0.19 (0.25)	0.16 (0.20)	0.06 (0.26)	0.20 (0.26)	0.19 (0.25)	0.15 (0.20)
	12 mo	0.03 (0.18)	0.14 (0.19)	0.14 (0.18)	0.13 (0.17)	0.11 (0.14)	0.03 (0.18)	0.14 (0.18)	0.13 (0.17)	0.11 (0.14)
That degraded communication link is established		0.62 (0.86)	0.65 (0.86)	0.65 (0.86)	0.65 (0.86)	0.65 (0.86)	0.59 (0.83)	0.63 (0.83)	0.63 (0.83)	0.63 (0.83)
Of degraded surface operations for:	1 day	0.58 (0.81)	0.61 (0.81)	0.61 (0.81)	0.61 (0.81)	0.61 (0.81)	0.56 (0.77)	0.59 (0.77)	0.59 (0.77)	0.59 (0.77)
	1 wk	0.57 (0.79)	0.60 (0.79)	0.60 (0.79)	0.60 (0.79)	0.60 (0.79)	0.54 (0.76)	0.57 (0.76)	0.57 (0.76)	0.57 (0.76)
	1 mo	0.52 (0.72)	0.54 (0.72)	0.54 (0.72)	0.54 (0.72)	0.54 (0.72)	0.49 (0.69)	0.52 (0.69)	0.52 (0.69)	0.52 (0.69)
	2 mo	0.46 (0.64)	0.48 (0.64)	0.48 (0.64)	0.48 (0.64)	0.48 (0.64)	0.44 (0.61)	0.46 (0.61)	0.46 (0.61)	0.46 (0.61)
	3 mo	0.40 (0.56)	0.43 (0.56)	0.43 (0.56)	0.43 (0.56)	0.43 (0.56)	0.39 (0.54)	0.41 (0.54)	0.41 (0.54)	0.41 (0.54)
	4 mo	0.36 (0.50)	0.38 (0.50)	0.38 (0.50)	0.38 (0.50)	0.38 (0.50)	0.34 (0.48)	0.36 (0.48)	0.36 (0.48)	0.36 (0.48)
	5 mo	0.32 (0.44)	0.33 (0.44)	0.33 (0.44)	0.33 (0.44)	0.33 (0.44)	0.30 (0.42)	0.32 (0.42)	0.32 (0.42)	0.32 (0.42)
	6 mo	0.28 (0.39)	0.30 (0.39)	0.30 (0.39)	0.30 (0.39)	0.30 (0.39)	0.27 (0.37)	0.28 (0.37)	0.28 (0.37)	0.28 (0.37)
	9 mo	0.19 (0.27)	0.20 (0.27)	0.20 (0.27)	0.20 (0.27)	0.20 (0.27)	0.19 (0.26)	0.20 (0.26)	0.20 (0.26)	0.20 (0.26)
	12 mo	0.13 (0.19)	0.14 (0.19)	0.14 (0.19)	0.14 (0.19)	0.14 (0.19)	0.13 (0.18)	0.14 (0.18)	0.14 (0.18)	0.14 (0.18)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	13	15	14	13	11	12	14	13	11
	1 wk	88	101	97	92	76	84	97	92	76
	1 mo	344	413	396	375	313	303	396	375	312
	2 mo	614	778	747	707	589	589	746	707	589
	3 mo	825	1102	1057	1001	834	791	1056	1001	834
	4 mo	990	1388	1332	1262	1051	950	1331	1261	1051
	5 mo	1119	1642	1575	1492	1244	1074	1574	1491	1242
	6 mo	1220	1866	1791	1696	1414	1171	1789	1694	1412
	9 mo	1490	2396	2299	2178	1815	1352	2296	2175	1813
	12 mo	1499	2763	2651	2512	2093	1439	2647	2508	2090

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-10. Case 10 reliability results* (trip time = 200 days; $\beta = 1.0$, $\nu = 0.95$, $\alpha = 0.015$, $\mu = 0.2$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
			Direct link					Direct link		
		Relay link	Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75	Relay link	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.86	0.90	0.90	0.90	0.90	0.86	0.90	0.90	0.90
Of lander separation		0.19 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)	0.19 (0.89)	0.20 (0.89)	0.20 (0.89)	0.20 (0.89)
Of entry and landing		0.19 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)	0.19 (0.87)	0.20 (0.87)	0.20 (0.87)	0.20 (0.87)
That primary communication link is established		0.16 (0.86)	0.20 (0.86)	0.19 (0.83)	0.18 (0.78)	0.15 (0.65)	0.16 (0.83)	0.19 (0.83)	0.19 (0.83)	0.15 (0.65)
Of primary surface operations for:	1 day	0.15 (0.79)	0.18 (0.79)	0.17 (0.76)	0.17 (0.72)	0.14 (0.60)	0.14 (0.76)	0.17 (0.76)	0.17 (0.76)	0.14 (0.60)
	1 wk	0.11 (0.70)	0.16 (0.70)	0.15 (0.67)	0.15 (0.63)	0.12 (0.53)	0.11 (0.67)	0.15 (0.67)	0.15 (0.67)	0.12 (0.53)
	1 mo	0.04 (0.42)	0.10 (0.43)	0.09 (0.41)	0.09 (0.39)	0.07 (0.32)	0.04 (0.41)	0.09 (0.41)	0.09 (0.41)	0.07 (0.32)
	2 mo	0.01 (0.22)	0.05 (0.22)	0.05 (0.21)	0.05 (0.20)	0.04 (0.17)	0.01 (0.21)	0.05 (0.21)	0.05 (0.21)	0.04 (0.17)
	3 mo	0.00 (0.12)	0.03 (0.12)	0.03 (0.11)	0.02 (0.11)	0.02 (0.09)	0.00 (0.11)	0.03 (0.11)	0.03 (0.11)	0.02 (0.09)
	4 mo	0.00 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.05)	0.00 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.05)
	5 mo	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)	0.00 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.02)
	6 mo	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.01)	0.00 (0.01)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.01)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
That degraded communication link is established		0.19 (0.86)	0.20 (0.86)	0.20 (0.86)	0.20 (0.86)	0.20 (0.86)	0.18 (0.83)	0.19 (0.83)	0.19 (0.83)	0.19 (0.83)
Of degraded surface operations for:	1 day	0.17 (0.79)	0.18 (0.79)	0.18 (0.79)	0.18 (0.79)	0.18 (0.79)	0.16 (0.76)	0.17 (0.76)	0.17 (0.76)	0.17 (0.76)
	1 wk	0.15 (0.70)	0.16 (0.70)	0.16 (0.70)	0.16 (0.70)	0.16 (0.70)	0.14 (0.67)	0.15 (0.67)	0.15 (0.67)	0.15 (0.67)
	1 mo	0.09 (0.43)	0.10 (0.43)	0.10 (0.43)	0.10 (0.43)	0.10 (0.43)	0.09 (0.41)	0.09 (0.41)	0.09 (0.41)	0.09 (0.41)
	2 mo	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)	0.05 (0.22)
	3 mo	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.03 (0.12)	0.02 (0.11)	0.03 (0.11)	0.03 (0.11)	0.03 (0.11)
	4 mo	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)	0.01 (0.06)
	5 mo	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)	0.01 (0.03)
	6 mo	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)	0.00 (0.02)
	9 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
	12 mo	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)	0.00 (0.00)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	4	4	4	4	3	3	4	4	3
	1 wk	22	29	28	26	22	22	28	26	22
	1 mo	62	98	94	89	74	60	94	89	74
	2 mo	79	150	144	136	113	76	144	136	113
	3 mo	83	177	170	161	134	80	169	160	134
	4 mo	85	191	183	174	145	81	183	173	144
	5 mo	85	198	190	180	150	82	190	180	150
	6 mo	85	202	194	184	153	82	193	183	153
	9 mo	85	206	197	187	156	82	197	186	155
	12 mo	85	206	198	187	156	82	197	187	156

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

Table C-11. Case 11 reliability results* (trip time = 200 days; $\beta = 0.19$, $\nu = 1.0$, $\alpha = 0.038$, $\mu = 1.0$)

Probabilities		Probability that omnidirectional antenna will work = 0.99					Probability that omnidirectional antenna will work = 0.95			
		Relay link	Direct link				Relay link	Direct link		
			Probability that high-gain antenna will work = 0.99	Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75		Probability that high-gain antenna will work = 0.95	Probability that high-gain antenna will work = 0.90	Probability that high-gain antenna will work = 0.75
Of initial deployment and acquisition		1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Of midcourse trajectory corrections		0.98	0.98	0.98	0.98	0.98	0.98	0.98	0.98	0.98
Of lander separation		0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)	0.67 (0.88)
Of entry and landing		0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)	0.66 (0.87)
That primary communication link is established		0.54 (0.85)	0.65 (0.85)	0.62 (0.82)	0.59 (0.78)	0.49 (0.65)	0.52 (0.82)	0.62 (0.82)	0.59 (0.78)	0.49 (0.65)
Of primary surface operations for:	1 day	0.46 (0.73)	0.55 (0.73)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)	0.44 (0.70)	0.53 (0.70)	0.50 (0.66)	0.42 (0.55)
	1 wk	0.44 (0.71)	0.54 (0.71)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)	0.42 (0.68)	0.52 (0.68)	0.49 (0.65)	0.41 (0.54)
	1 mo	0.36 (0.64)	0.49 (0.65)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)	0.35 (0.62)	0.47 (0.62)	0.45 (0.59)	0.37 (0.49)
	2 mo	0.28 (0.59)	0.43 (0.58)	0.42 (0.55)	0.40 (0.52)	0.33 (0.44)	0.27 (0.55)	0.42 (0.55)	0.39 (0.52)	0.33 (0.44)
	3 mo	0.22 (0.50)	0.38 (0.51)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)	0.21 (0.48)	0.37 (0.49)	0.35 (0.46)	0.29 (0.39)
	4 mo	0.17 (0.44)	0.34 (0.45)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)	0.16 (0.43)	0.33 (0.43)	0.31 (0.41)	0.26 (0.34)
	5 mo	0.13 (0.39)	0.30 (0.40)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)	0.13 (0.38)	0.29 (0.38)	0.27 (0.36)	0.23 (0.30)
	6 mo	0.10 (0.35)	0.27 (0.35)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)	0.10 (0.33)	0.26 (0.34)	0.24 (0.32)	0.20 (0.27)
	9 mo	0.05 (0.24)	0.18 (0.24)	0.18 (0.23)	0.17 (0.22)	0.14 (0.19)	0.05 (0.23)	0.18 (0.23)	0.17 (0.22)	0.14 (0.18)
	12 mo	0.02 (0.16)	0.13 (0.17)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)	0.02 (0.16)	0.12 (0.16)	0.12 (0.15)	0.10 (0.13)
That degraded communication link is established		0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.65 (0.85)	0.62 (0.82)	0.62 (0.82)	0.62 (0.82)	0.62 (0.82)
Of degraded surface operations for:	1 day	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.55 (0.73)	0.53 (0.70)	0.53 (0.70)	0.53 (0.70)	0.53 (0.70)
	1 wk	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.54 (0.71)	0.52 (0.68)	0.52 (0.68)	0.52 (0.68)	0.52 (0.68)
	1 mo	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.49 (0.65)	0.47 (0.62)	0.47 (0.62)	0.47 (0.62)	0.47 (0.62)
	2 mo	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.43 (0.58)	0.42 (0.55)	0.42 (0.55)	0.42 (0.55)	0.42 (0.55)
	3 mo	0.39 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.38 (0.51)	0.37 (0.49)	0.37 (0.49)	0.37 (0.49)	0.37 (0.49)
	4 mo	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.34 (0.45)	0.33 (0.43)	0.33 (0.43)	0.33 (0.43)	0.33 (0.43)
	5 mo	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.30 (0.40)	0.29 (0.38)	0.29 (0.38)	0.29 (0.38)	0.29 (0.38)
	6 mo	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.27 (0.35)	0.26 (0.34)	0.26 (0.34)	0.26 (0.34)	0.26 (0.34)
	9 mo	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.24)	0.18 (0.23)	0.18 (0.23)	0.18 (0.23)	0.18 (0.23)
	12 mo	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.13 (0.17)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)	0.12 (0.16)
Average accrued bits (normalized by I_e in bits/hr) in:	1 day	11	13	13	12	10	11	13	12	10
	1 wk	75	91	87	83	69	72	87	83	69
	1 mo	294	373	358	339	282	282	357	339	282
	2 mo	524	703	674	639	532	503	674	638	532
	3 mo	703	995	955	905	754	675	954	904	753
	4 mo	843	1254	1203	1140	950	809	1202	1139	949
	5 mo	951	1483	1423	1348	1123	913	1421	1346	1122
	6 mo	1036	1685	1617	1532	1277	994	1615	1530	1275
	9 mo	1194	2164	2077	1967	1639	1146	2073	1964	1637
	12 mo	1268	2495	2394	2268	1890	1217	2390	2264	1887

*Numbers in parentheses show lander probabilities, if it was successfully separated from the bus.

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