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Deborah M. Badi
and Jeffery T. Farmer
Langley Research Center
Hampton, Virginia

Paul A. Garn
Bionetics Corporation
Hampton, Virginia

Gary L. Martin
The George Washington University
Joint Institute for Advancement of Flight Sciences
Langley Research Center
Hampton, Virginia

National Aeronautics and Space Administration
Office of Management
Scientific and Technical Information Division

Abstract

Future missions to Mars will require a communications system to link activities on the Martian surface with each other and with mission controllers on Earth. This paper presents a conceptual design for an areosynchronous communications satellite to provide these links. The satellite provides the capability for voice, data-command, and video transmissions. The mission scenario assumed for the design is described, and a description of the communications requirements is given. The basis for selecting a single areosynchronous satellite is explained. A viable spacecraft design is then presented. Communications band selection and channel allocation are discussed. The communications system conceptual design is presented along with the trade studies used in sizing each of the required antennas. Also, the analyses used to develop the supporting subsystem designs are described, as is the communications impact on each subsystem design.

Introduction

Recent reports have identified missions to Mars as possible future goals for the United States space program (refs. 1 and 2). The missions planned include the following: (1) Mars Rover/Sample Return, which is an unmanned automated mission; (2) Manned Mars Expeditions, which are missions with short Mars stay times; and (3) Manned Mars Evolutionary, which are missions with greater stay times.

Much work has been done on the conceptual design of the various elements required to accomplish such missions, including work on the rovers, transport vehicles, and aerobrakes. One area that has only recently begun to be addressed is communications. For mission success a means of communication between Mars and Earth, as well as among the various mission elements on the Martian surface, is needed. Therefore, the objective of this study was to develop detailed requirements for a communications system and to evolve a conceptual design. This design was restricted to existing and near-term technologies, and it included an effort to minimize overall mass.

Symbols

A	area, m^2
B	bandwidth, Hz
C/N	carrier-to-noise ratio, dB
d	diameter, m
G	gain, dB
g	Earth gravitational acceleration, 9.81 m/s^2
k	Boltzmann constant, 1.38×10^{-23} J/K
L	loss in link
L_p	path loss, dB
n	number of spots in spot-beam coverage
P_{RF}	RF power, W
P_T	transmitted power, W
R	transmission distance, m
T	thermal noise temperature, K
ΔV	change in velocity, m/s
η	antenna efficiency
θ, ϕ	antenna half-power beamwidth angles, deg
λ	wavelength, m

Subscript:

T	transmitter

Acronyms

ATS	Applications Technology Satellite
BER	bit error rate
CTE	coefficient of thermal expansion
D	diplexer
DIPS	Dynamic Isotope Power System
DSN	Deep Space Network
EIRP	equivalent isotropically radiated power
EOL	end of life
EVA	extravehicular activity
GN&C	guidance, navigation, and control
GPS	Global Positioning Satellite
PV	photovoltaic
QPSK	quadrature phase-shift keying
RF	radio frequency
RTG	Radioisotope Thermoelectric Generator
SAFE	Solar Array Flight Experiment
SD	solar dynamic
SMARTS	Synchronous Mars Telecommunications Satellite

SRM	solid rocket motor
T	transponder
TWTA	traveling wave tube amplifier

Communications Requirements

Communication links must be established between Earth and each of the various mission elements on the surface of Mars (see fig. 1). In addition, links must provide communication among these surface elements. The capacity must exist for the transmission of voice, video, and data-command signals. Mars surface coverage must be flexible to accommodate different base locations. Although detailed mission scenarios do not presently exist, a mission scenario was developed, and a mission lifetime of 10 Earth years is assumed for this study.

The communications scenario developed for this study assumes that each manned site has a central base with numerous activities. Each central base is equipped with two rover vehicles. Extravehicular activity (EVA) personnel establish communications via the nearest base or rover antenna. In addition to the manned bases, a variety of automated experiments and activities may be distributed about the Martian surface, both at the manned locations and at other unmanned landing sites. All these elements require communications capabilities with a minimum field of view at each site (manned or unmanned) of 100 km in radius, twice the anticipated maximum rover excursion distance. Up to two manned bases may be active concurrently. Table I summarizes the assumed communications requirements.

Table I. Assumed Communications Requirements

Links required	Earth to Mars Mars to Earth Mars to Mars
Types of links	Voice Video Data-command
Design lifetime, Earth years	10
Active manned bases (100-km radius)	2
Manned rovers per active base	2
Automated experiments (within base areas)	Several

Several locations on the surface of Mars have been identified as potential landing sites for future Mars missions (First Interim Progress Review of the Manned Mars Mission and Program Analysis by SRS Technologies, Huntsville, Alabama, Aug. 1987). Those of particular interest to scientists are listed in table II.

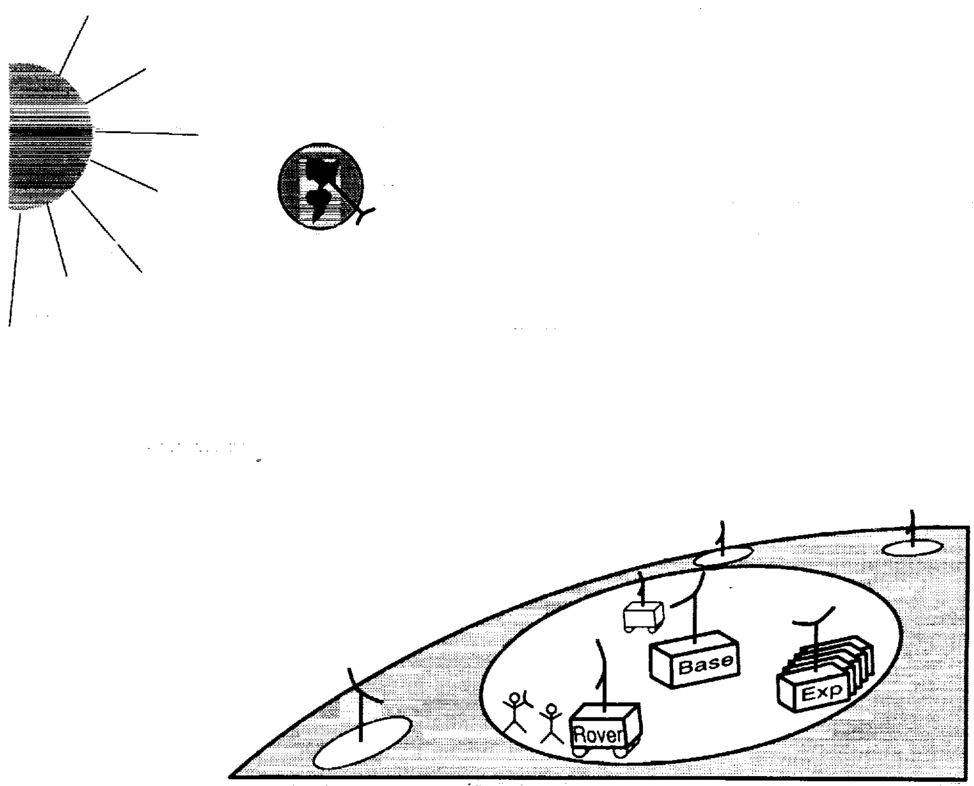


Figure 1. Mars mission elements.

Table II. Potential Landing Sites

Site	Location
Candor Chasma	7.0° S. 74.5° W.
Kasei Valles	15.0° N. 75.0° W.
Mangala Valles	7.0° S. 158.0° W.
Olympus Mons	23.0° N. 130.0° W.
Chasma Boreale	82.0° N. 44.0° W.
Isidia Planitia	15.0° N. 278.0° W.

Overall System Architecture

Several types of system architectures could provide the links required for proposed Mars missions. These architectures are illustrated in figure 2 and are as follows: a system of direct Mars-Earth and Mars-Mars links, a series of low-orbiting relay satellites, and a single relay satellite operating in areosynchronous orbit (the Mars equivalent of a geosynchronous orbit about the Earth).

The areosynchronous satellite was selected for this study. The advantages of this type of system include the following: the areosynchronous satellite remains fixed with respect to the surface of Mars, and therefore high-gain, low-power surface antennas can be used; a single relay satellite can service nearly an entire hemisphere of the planet; and the possibility of topographic obstacles impeding the surface-to-surface links is avoided by routing these communications through the satellite.

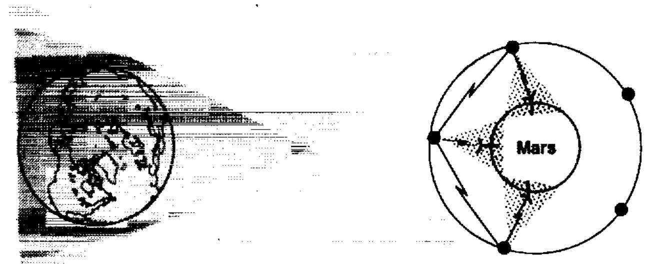
The conceptual design presented in this paper is that of the Synchronous Mars Telecommunications Satellite (SMARTS), also described in reference 3. The satellite operates in areosynchronous orbit in the Mars equatorial plane at an altitude of 17 070 km and with a period equal to 1 Martian day (24.66 hours). This orbit provides constant coverage of a fixed portion of the planet surface within its field of view, and SMARTS is blocked from Earth contact for a maximum of only 1.31 hours in a day.

Of the six landing sites identified previously, four are sufficiently clustered so as to be serviceable from a single areosynchronous satellite located at 0° latitude and 110° W. longitude. These four sites are Mangala Valles, Olympus Mons, Kasei Valles, and Candor Chasma. These sites were used to formulate the SMARTS conceptual design.

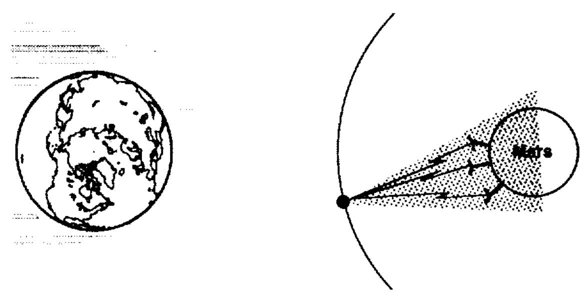
Spacecraft Design

The SMARTS spacecraft, illustrated in figure 3, is powered by three deployable 3.0- by 8.5-m photovoltaic arrays. These arrays provide a total of

(a) Direct links.



(b) Series of low Mars-orbiting relay satellites.



(c) Single areosynchronous relay satellite.

Figure 2. Overall system architecture options.

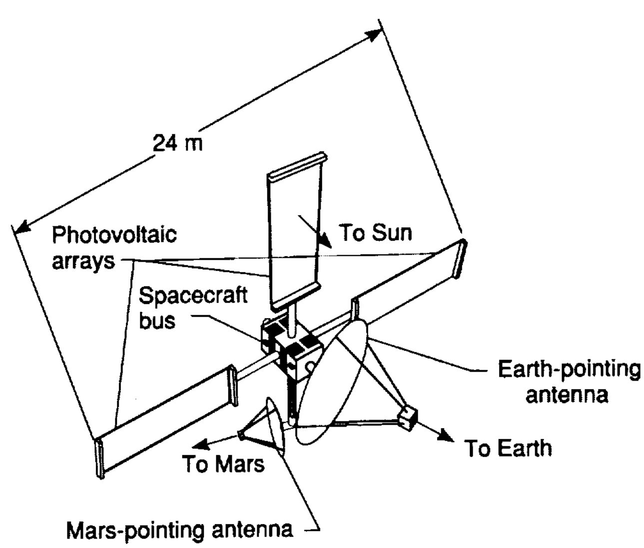


Figure 3. Synchronous Mars Telecommunications Satellite (SMARTS).

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3535 W of end-of-life (EOL) power to the communications system and supporting subsystems. The largest portion of this power requirement is used to support the Earth link.

The spacecraft has a three-axis-stabilized configuration so that it can sustain three independent pointing requirements at all times during its orbit. A 9.1-m-diameter antenna remains pointed toward Earth, a 3.0-m-diameter antenna is directed toward the Martian surface, and the solar arrays are oriented toward the Sun. The solar-fixed attitude also allows for continuous heat rejection through the exterior surfaces of the spacecraft.

The Earth-pointing antenna is positioned so that it may slew as necessary to maintain the Mars-Earth communication link. The Mars-pointing antenna is located on a boom, which affords it a full 360° range of orientation. This positioning of the Mars-pointing antenna also provides a mass balance with the third solar array; thus, overall mass symmetry is increased and the fuel required for attitude maintenance is reduced.

The total estimated spacecraft mass at the beginning of operations is 2405 kg, with hydrazine fuel constituting approximately one-third of the mass. This fuel is used by the attitude control system to achieve and maintain the proper orbit and attitude and to perform station-keeping maneuvers over the 10-year design lifetime of the spacecraft.

Based on the conceptual designs described subsequently in this report, the masses and power requirements of the subsystems onboard SMARTS are listed in tables III and IV, respectively. The total system mass (and volume) is well within the envelopes of many conventional launch systems.

Communications System

The three types of links which SMARTS must provide are Earth to Mars, Mars to Earth, and Mars to Mars. These links can be broken down into four types of link segments: Earth to SMARTS (Earth up link), SMARTS to Earth (Earth down link), Mars to SMARTS (Mars up link), and SMARTS to Mars (Mars down link).

Channel allocations. A 12-channel system is used to establish all the necessary communications links routed through SMARTS. The channels are illustrated in figure 4.

There are 10 Mars-Mars channels (M1 to M10), each of which is 40 MHz wide, with a 4-MHz buffer between channels. Channels M3 to M10 are primary channels, with M1 and M2 reserved as spares for a circuit failure of a primary channel or expansion to meet additional needs. Each active base and each rover is allocated one multipurpose channel which can carry a combination of video, voice, and data transmissions. These are channels M3 to M8. Channel M9 is used for transmission of commands to and data from the automated surface elements. Channel M10 provides up to 300 concurrent two-way voice links.

Channels E1 and E2 are allocated for the Earth-Mars communications. Channel E1 is a 20-MHz-wide multipurpose channel and channel E2 is a highly accurate 10-MHz-wide data channel. Again, a 4-MHz buffer is included between channels. The various elements on the Mars surface time-share these channels for Earth communications.

Table III. SMARTS Mass Breakdown

Subsystem	Mass, kg
Communications payload (including antennas)	275
Data storage	45
Electrical power	500
Thermal control	45
Guidance, navigation, and control	160
Propulsion (including hydrazine fuel)	1100
Structural components	280
Total before orbit circularization	2405
Solid fuel	895
Total launch mass	3300

Table IV. SMARTS Power Budget

Subsystem	Power required, W
Communications payload	2865
Data storage	120
Electrical power	90
Thermal control	335
Guidance, navigation, and control	120
Propulsion	5
Total bus load	3535
Battery charging	220
15-percent contingency	530
Total power required	4285

Band selections. Also included in figure 4 are the operating frequency bands. The bands selected for the Earth up link and Earth down link segments comply with Deep Space Network (DSN) allocations for deep-space research as described in reference 4. The highest frequency bands allocated to space-to-Earth and Earth-to-space deep-space research on a

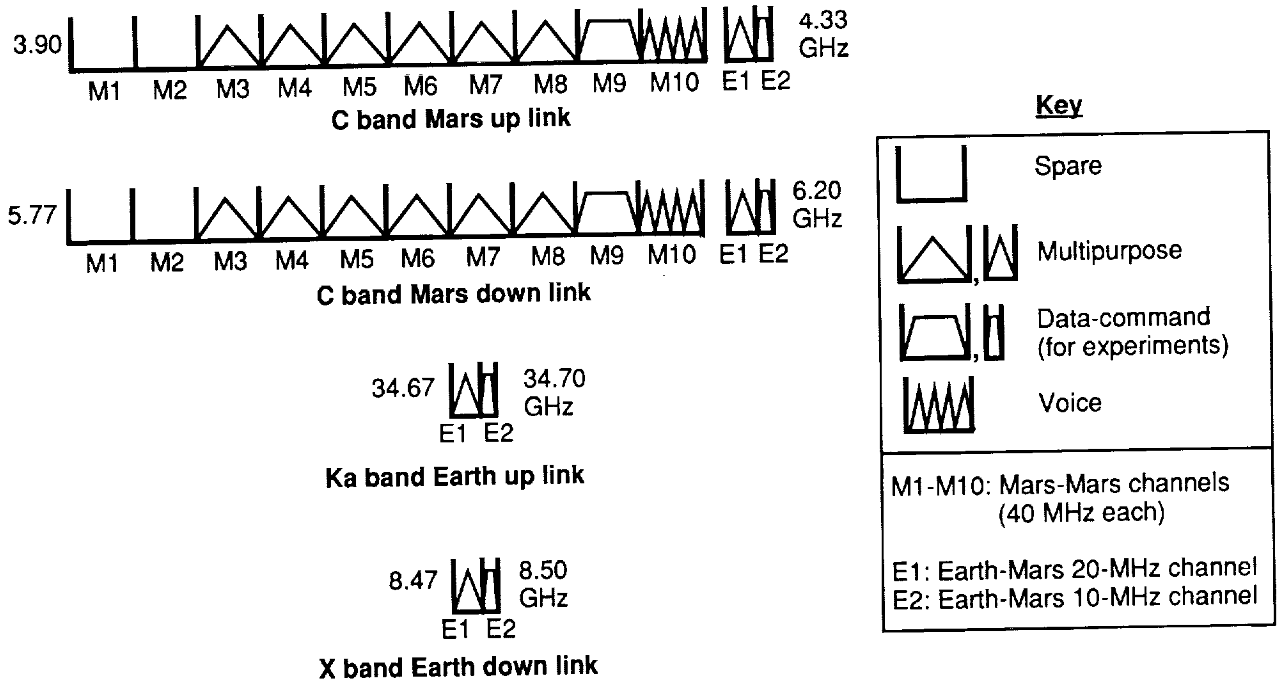

Figure 4. Bandwidth and channel allocations.

primary basis are selected for the Earth down link and up link, respectively. The highest frequencies are used in order to minimize the power required for signal transmission, since high frequencies require less RF power for transmission than low frequencies require for identical antenna sizes and efficiencies.

The highest available band for the Earth down link is in the X band. A 30-MHz bandwidth is required for the two Earth-Mars channels. The selected frequency band is 8.47 to 8.50 GHz, the upper end of the allocated band. The technology exists for support of this X band down link (ref. 4).

The highest DSN band available for the Earth up link is in the Ka band. This is one of the more recent DSN allocations; therefore, the technology to support this frequency does not yet exist. Since the time frame for sending humans to Mars is the first decade of the 21st century (ref. 2), it was assumed for this study that the Ka band up link technology would be available. To allow for the same two Earth-Mars channels, the frequency band of 34.67 to 34.70 GHz is used.

The band selected for the Mars up link and down link segments is the C band. This selection is based on proven lifetime and current availability of supporting technology. The upper and lower extremes of the C band (5.77 to 6.20 and 3.90 to 4.33 GHz, respectively) are selected to avoid interference between the links. The higher frequency band is allocated to the down link to minimize the power load on the satellite.

Assumptions. Table V summarizes various assumptions made in designing the SMARTS communications system. The Mars up link and down link calculations are based on 1-m-diameter parabolic dish antennas on the Martian surface. These are scaled-up versions of the 0.762-m (30-in.) diameter surface antennas of the Viking missions. The DSN Earth ground-station antennas are 70 m (230 ft) in diameter, recently upgraded from 64 m (210 ft) (ref. 5). Antenna efficiencies are assumed to be 70 percent in all link calculations based on an overview of current technology reported in reference 6. Quadrature phase-shift keying (QPSK), a common current digital modulation technique, is employed. Thermal noise temperatures T for the X band and the C band are 28.5 K (ref. 4) and 160 K (ref. 6), respectively.

The bit error rates (BER's) are determined based on the individual channel needs. In the Mars down link all channels except E2 are transmitted with a BER of 10^{-6}. This is a lower, and therefore more nearly error-free, BER than that used on Viking. Channel E2 of the Mars down link is assumed to require a lower BER (10^{-8}) for the relay of commands and data from Earth.

Table V. Communications Systems Assumptions

	SMARTS to Mars	SMARTS to Earth
Receiver diameter, m	1	70
Antenna efficiency, percent	70	70
Encoding technique	QPSK	QPSK
Thermal noise temperature, K	160	28.5
BER:		
Channels M1-M10	10^{-6}	(a)
Channel E1	10^{-6}	10^{-3}
Channel E2	10^{-8}	10^{-8}
C/N, dB:		
Channels M1-M10	13.8	(a)
Channel E1	13.8	10
Channel E2	15	15
Losses, dB:		
Line	.9	.9
Polarization	.3	.3
Pointing	.8	1.9
Rain	(a)	2.0
Cloud cover	(a)	.5

[a]Not applicable.

The Earth down link multipurpose channel (E1) has a BER of 10^{-3}, on the order of that of Viking. This BER is selected to reduce the power required for transmission. The data channel (E2) once again is assumed to require a lower BER (10^{-8}).

The carrier-to-noise ratios C/N shown in the table are determined as described in reference 6, based on the BER's and the use of QPSK encoding.

The line and polarization losses in the links are approximated based on existing hardware characteristics. The pointing losses are based on allowable antenna pointing errors. The Earth rain and cloud cover losses are obtained from reference 6.

Link calculation method. The diameter of a reflector is inversely proportional to the power required to transmit a given signal (for identical antenna efficiencies) and to the beam width of the transmitted signal. Therefore, a large-diameter antenna produces a sharply focused beam and requires a relatively small amount of power. This relationship is obtained from the following method, which is detailed in reference 6 and is used for all link calculations.

The equivalent isotropically radiated power (EIRP) of an antenna is found by

$$\mathrm{EIRP}\Big|_{\mathrm{dBW}} = \frac{C}{N}\Big|_{\mathrm{dB}} - \frac{G}{T}\Big|_{\mathrm{dB}} + k\Big|_{\mathrm{dB}} + L_p\Big|_{\mathrm{dB}} + B\Big|_{\mathrm{dB}} \qquad (1)$$

where C/N is the required carrier-to-noise ratio and is based on the modulation technique employed for the assumed BER (ref. 6, p. 441), G is the gain of the receiving antenna, T is the total receiving system thermal noise temperature (K), k is the Boltzmann constant (1.38×10^{-23} J/K, or -228.6 dB), L_p is the path loss from transmitter to receiver, and B is the signal bandwidth (Hz).

The gain of an antenna is computed by

$$G = \frac{4\pi A}{\lambda^2}\eta \qquad (2)$$

where A is the area of the reflector (m^2), λ is the wavelength (m), and η is the antenna efficiency (assumed to be 70 percent for all cases, as indicated previously). The gain of a transmitting antenna G_T can also be expressed as

$$G_T = \frac{30\,000}{\theta\phi} \qquad (3)$$

where θ and ϕ are the orthogonal half-power beamwidth angles in degrees.

The transmission path loss is determined by

$$L_p = \left(\frac{4\pi R}{\lambda}\right)^2 \tag{4}$$

where R is the transmission distance (m).

Once the EIRP is known, the transmitted power P_T is found with the equation

$$P_T\Big|_{\mathrm{dBW}} = \mathrm{EIRP}\Big|_{\mathrm{dBW}} - G_T\Big|_{\mathrm{dB}} \tag{5}$$

The RF power for transmission is thus

$$P_{\mathrm{RF}}\Big|_{\mathrm{dBW}} = P_T\Big|_{\mathrm{dBW}} + L\Big|_{\mathrm{dB}} \tag{6}$$

where L is an approximation of the losses in the link, as listed in table V.

Mars antenna power requirements and sizing. The size of the Mars-pointing antenna onboard SMARTS is selected to minimize the power required to provide flexible coverage of the Martian surface. As indicated previously, four landing sites were selected to demonstrate the ability to provide such coverage.

To establish the necessary coverage of the Martian surface, three methods were examined. These are shown in figure 5, which represents a trade study to determine the optimum technique. The parameters used in implementing the trade study were antenna diameter (to be kept within a manageable range), power required for transmission (to be kept to a minimum), and beamwidth (to provide coverage to a minimum radius of 100 km from the base site).

The first option is to irradiate an entire hemisphere of the planet with a single feed horn. The reflector diameter required for this method is 0.075 m, which is too small to be practical. Therefore, no reflector is used; the feed horn faces the planet and irradiates the surface directly. This technique is, therefore, the simplest of those examined, and it provides very flexible coverage. The load on the power subsystem, however, is excessive (greater than 9000 W).

Another method examined is to irradiate an elliptical area which includes all points of interest on the surface. This coverage is also obtained with a single feed horn; however, a reflector is required. The smallest elliptical coverage of the four sites cited previously requires a 0.3-m-diameter reflector which needs nearly 1000 W of power to operate. This technique provides flexible coverage within its elliptical field of view.

The final option is to cover the various areas of interest with individual spot beams. A feed horn array system, which allows for multiple excitation of

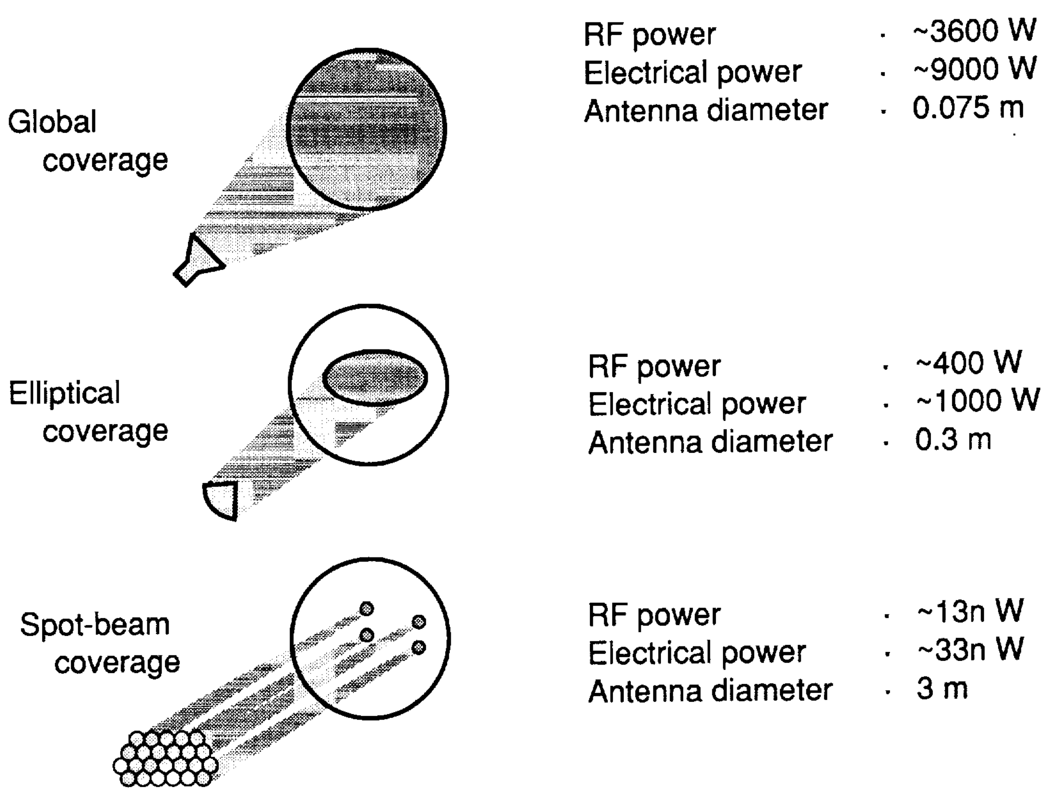

Figure 5. Sizing and feed array design of Mars-pointing antenna.

a single aperture, is used to retain some degree of flexibility of coverage provided by the other methods. The feed horns are positioned so that individual horns can be activated to cover specific areas of the planet. Through an iterative process a reflector diameter of 3.0 m, which requires only 33 W per site, was selected. This antenna provides coverage to a radius of 170 km at each site, a value which is greater than the required 100 km. The spot-beam coverage technique is the method selected because of the significant decrease in the power load on the satellite.

For the selected spot-beam coverage method, the calculation of the Mars down link is based on the method described previously. The calculation for a 3.0-m-diameter parabolic reflector antenna is shown below for the 340-MHz band (5.85 to 6.19 GHz) which includes primary channels M3 to M10 and E1. This calculation assumes a worst case power loading of transmitting all channels simultaneously.

The gain of the Mars-pointing antenna is obtained with equation (2) for a diameter of 3.0-m and the longest wavelength in the band ($\lambda = 0.0513$ m). This yields a gain of

$$G_T = 23\,643 = 43.7 \text{ dB} \tag{7}$$

Setting equation (7) equal to equation (3) and setting $\theta = \phi$ (for a circular beam cross section) yields

$$\theta = \phi = 1.13^\circ$$

which is the half-power beamwidth. This provides coverage to a radius of approximately 170 km.

Equations (1), (2), (4), (5), and (6) are used to determine the RF power required to transmit to the surface. The C/N and T values are listed in table V. The bandwidth in decibels is 85.3.

The gain of the receiving antenna is found from equation (2) with $d = 1$ m, $\lambda = 0.0513$ m, and $\eta = 0.7$. These values result in

$$G = 2627 = 34.2 \text{ dB}$$

The path loss is found from equation (4) with $R = 20\,750\,000$ m (the maximum range to the edge of the hemisphere) and $\lambda = 0.0485$ m (the shortest wavelength in the band). These values yield a path loss of

$$L_p = 194.6 \text{ dB}$$

The total EIRP as determined by equation (1) is therefore

$$\text{EIRP} = 52.9 \text{ dBW} \tag{8}$$

The RF power required for transmission is found by substituting equations (7) and (8) into equations (5) and (6) and using the sum of the Mars link losses in table V. This results in

$$P_{\text{RF}} = 11.2 \text{ dB} = 13.2 \text{ W} \tag{9}$$

The result in equation (9) is the RF power requirement per spot beam. For the four coverage areas cited previously, the total RF power is therefore 52.8 W. The load on the power system, if we assume a 40-percent-efficient system, is 132 W.

A similar calculation was done for the 10-MHz-wide channel E2. This calculation results in an electrical power requirement of roughly 4.0 W (1.6 W of RF power). The total load on the SMARTS power system for the Mars down link is therefore approximately 140 W for a worst case power condition of transmitting all channels simultaneously.

Earth antenna power requirements and sizing. Another trade study was performed to size the Earth antenna. This sizing is shown in figure 6, which represents the relationship between the antenna diameter, the power required, and the resulting mass. The mass is shown as a function of antenna diameter on three curves representing (1) the mass of the antenna of the given diameter, (2) the mass of the power system needed to provide the necessary power for that diameter, and (3) the sum of these two masses. Power requirements at selected diameters are also shown.

As the diameter of the antenna is increased, the power required decreases. The mass of the power system therefore decreases while the mass of the antenna increases. The sum of these masses decreases initially, then levels off and begins to increase as the antenna diameter increases. The graph shows an area of minimum mass in the 8- to 20-m-diameter region with an electrical power requirement in a manageable range (3660 to 700 W electrical power and 1465 to 280 W RF power). A 9.1-m diameter is selected in order to utilize the existing strongback technology of the 9.1-m wrapped-rib antenna which flew successfully onboard the Applications Technology Satellite (ATS-6) and operated at up to 6.425 GHz (ref. 7).

Technology development may be required in the area of mesh surfaces and shape control for the SMARTS application because of its very high operating frequency. The Ka band up link requires a surface accuracy of approximately 7 mils, which is beyond the state of the art. Possibly accuracies of this order will be feasible in the near term. These technologies may evolve as the DSN develops the technologies to support its Ka band up link allocation.

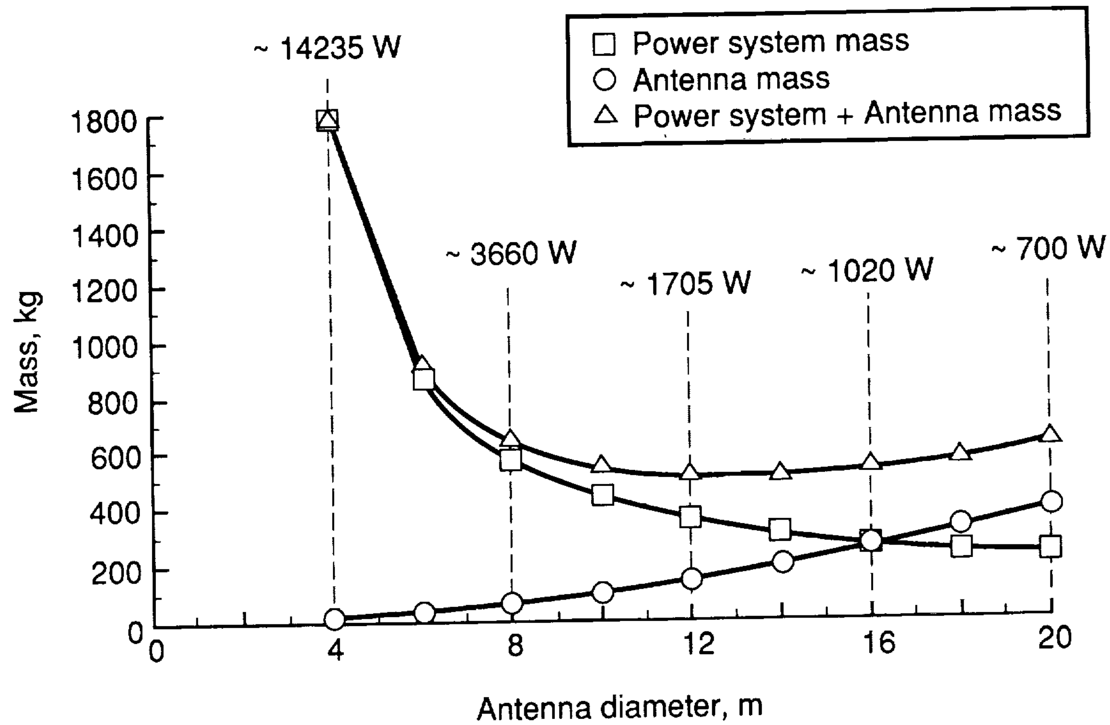

Figure 6. Sizing of Earth-pointing antenna.

The link calculation for the Earth antenna is similar to that done for the Mars antenna. The calculation for the selected 9.1-m-diameter antenna for the 20-MHz-wide channel E1 is as follows.

The gain of the Earth antenna is found from equation (2) for a diameter of 9.1 m and the longest wavelength in the band (0.0354 m). The gain is

$$G_T = 456\,042 = 56.6 \text{ dB} \tag{10}$$

By setting equation (10) equal to equation (3) and setting $\theta = \phi$ (for a circular beam cross section) it is found that

$$\theta = \phi = 0.26^\circ$$

which is the width of the emitted beam.

Equations (1), (2), (4), (5), and (6) are used to determine the RF power required to transmit to Earth. The C/N and T values are listed in table V. The bandwidth in decibels is 73.0.

The gain of the DSN receiving antenna is found from equation (2) with $d = 70$ m, $\lambda = 0.0354$ m, and $\eta = 0.7$. These values yield

$$G = 26\;984\;736 = 74.3 \text{ dB}$$

The path loss is found by use of equation (4) with $R = 380 \times 10^9$ m and $\lambda = 0.0353$ m (the shortest wavelength in the band). These values result in a path loss of

$$L_p = 282.6 \text{ dB}$$

The total EIRP as determined by equation (1) is therefore

$$\text{EIRP} = 77.3 \text{ dBW} \tag{11}$$

The RF power required to transmit is found by substituting equations (10) and (11) into equations (5) and (6) and using the sum of the losses in table V. This yields

$$P_{\text{RF}} = 26.3 \text{ dBW} = 424 \text{ W} \tag{12}$$

Based on a 40-percent efficient system, the load on the power system is approximately 1060 W.

A similar calculation was done for the 10-MHz-wide channel E2. This calculation results in a power requirement of 1664 W (666 W of RF power). For simultaneous transmission of both channels, approximately 2725 W (1090 W of RF power) is required. The various calculated parameters of both antennas are summarized in table VI.

Link calculations were also done for the Earth and Mars up links and result in an approximate electrical power requirement of 1320 W for the Earth ground station and 72 W for each Mars surface antenna.

Table VI. Antenna Parameters

Parameter	9.1-m Earth-pointing antenna channel—		3.0-m Mars-pointing antenna channel—	
	E1	E2	M1 to M10 and E1	E2
Gain, dB	56.6	56.6	43.7	44.2
EIRP, dBW	77.3	79.2	52.9	38.4
RF power, W	424	666	52.8	1.6

Communications flow description. Conventional Earth-orbiting communications satellites generally must only perform simple up link and down link conversions. However, three types of communications links must be relayed through SMARTS: communication from Earth is relayed to Mars, some communication from Mars is relayed to Earth, and the rest of the communication from Mars is relayed back down to the surface. This routing is accomplished by one diplexer and two transponders. The diplexer either splits or combines signals as necessary. The transponders convert the up link signals to their down link frequencies, and 40-percent-efficient traveling wave tube amplifiers (TWTA's) provide the RF power for transmission.

Figure 7 is a block diagram of the routing and processing scheme. In this figure, the 12-channel series is divided into three subsets: Earth-Mars channels E1 and E2, Mars-Mars channels M1 to M10, and the entire series.

Communication from Earth, on channels E1 and E2, is received by the larger Earth-pointing antenna on the Ka band up link frequency. It is routed to transponder T1, which converts the signal to the C band down link frequency for these channels. From transponder T1, the signal is then combined with channels M1 to M10 of the C band down link (as explained below) in a diplexer and routed to the Mars-pointing antenna for transmission to the surface of the planet.

Communication from Mars (channels M1 to M10, E1, and E2) is received at the C band up link frequency by the Mars-pointing antenna. The signal is sent through the diplexer to transponder T2. There

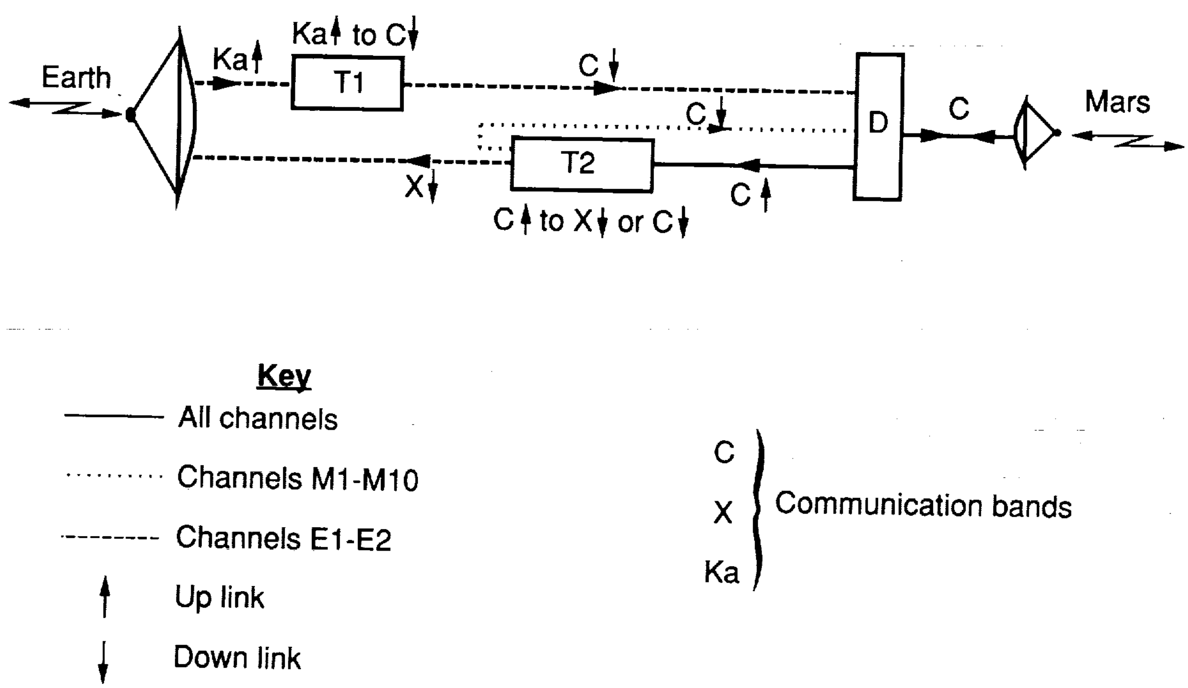

Figure 7. Communications system block diagram.

it is split into two parts, channels M1 to M10 (to be sent back down to the surface) and channels E1 and E2 (to be transmitted to Earth), and is converted to either the C band Mars down link frequency or the X band Earth down link frequency, as appropriate. The C band channels M1 to M10 are combined with the C band channels E1 and E2 in the diplexer and then sent back to the Mars antenna for transmission to the surface. The X band signals from transponder T2 (channels E1 and E2) are routed to the Earth-pointing antenna for transmission to the DSN receivers on Earth.

Data Storage Subsystem

SMARTS remains in constant contact with the surface sites on Mars because of its areosynchronous orbit. There are periods of occultation, however, when Mars is between SMARTS and Earth, during which time direct communication between SMARTS and Earth is not possible. During these periods, data must be stored for transmission at a later time. The maximum blackout duration is approximated to be the same as the longest time that SMARTS is eclipsed from the Sun by Mars. This occurs near the Martian equinox and is 1.31 hours in duration. The maximum amount of data which could be transmitted during an occultation period is 600 gigabits and is based on continuous transmission from both Earth down link channels for the entire blackout period. Data storage is done via a near-term optical disk system (ref. 8).

Solar occultation, when the Sun is between Mars and Earth, occurs once every 2.3 years. During this 17-day period, communication between Mars and Earth is completely blocked. It is assumed that humans will not be on the surface of the planet during this period because of this inability to communicate with mission controllers on Earth; therefore, this issue was not addressed in this design.

Electrical Power Subsystem

The total continuous load on the power subsystem is 3535 W. Included in this are the power required by the communications system (2865 W) and the power for the supporting subsystems.

The distance between Mars and the Sun decreases the effectiveness of photovoltaic (PV) arrays because of the reduced solar flux density at Mars (which ranges from 493 to 718 W/m^2) compared with that at Earth (which averages 1350 W/m^2). Therefore, several alternative power generation concepts were considered along with two types of PV cells (silicon and gallium arsenide). These include a solar dynamic (SD) system, the Radioisotope Thermoelectric Generator (RTG) system, and the Dynamic Isotope Power System (DIPS). Power subsystem selection parameters include mass, launch volume, and housekeeping impacts (i.e., the effect on other subsystems). Power subsystem selection is also affected by spacecraft configuration and system reliability. A discussion of each alternative with respect to these issues follows.

Solar dynamic system. A system consisting of free-piston Stirling engines and solar collectors for power generation with thermal phase change materials for energy storage (ref. 9) was examined. The mass of the solar dynamic system is the lowest of the systems examined (405 kg). Its launch volume, however, is one of the highest, roughly estimated at 4 m^3. This power system would best be served by a three-axis-stabilized configuration, which would provide an orientation in the general direction of the Sun. However, the fine-pointing requirement of the SD collectors ($\pm 0.1°$) imposes a third stringent pointing requirement on the guidance, navigation, and control (GN&C) subsystem in addition to the antenna requirements. The reliability of the SD system over a 10-year lifetime was questioned because of its moving parts and fluids, the complexity of its solar energy collection process, and the fact that SD systems have had no verification or testing in space.

Radioisotope Thermoelectric Generator system. The RTG system (ref. 10) provides constant power generation independent of solar input. This system has the highest mass of those considered (800 kg). Its launch volume is relatively small, about 2 m^3, and its impact on the other subsystems is minimal, although thermal and radioactive isolation from the rest of the spacecraft is required. Unlike the SD system, configuration selection has little impact on the RTG system. The RTG system is relatively simple in design, has no moving parts, and has been flight proven, and it is therefore considered highly reliable.

Dynamic Isotope Power System. The DIPS (refs. 10 to 12) is a heat engine which, like the RTG, provides constant power generation independent of the Sun. Its mass (420 kg) is almost as low as that of the SD system, but its launch volume (15 m^3) is by far the largest of the power systems considered. Like the RTG, the DIPS has minimal impact on housekeeping if thermally isolated from the rest of the spacecraft, and it is independent of configuration selection. However, DIPS has not been space qualified for NASA missions.

Photovoltaic arrays. Two types of PV array cells (refs. 6, 13, and 14) were initially considered: silicon (Si) and gallium arsenide (GaAs). In this type of system, planar deployable PV arrays provide power generation, while nickel hydrogen (NiH_2) batteries provide energy storage for use during periods of eclipse. Although Si and GaAs cells have mostly similar characteristics, near-term GaAs cells were selected for further consideration over state-of-the-art Si cells because of their higher estimated EOL efficiency. Although Si cells have been much more prevalent in previous spacecraft, GaAs cells are a very near-term technology and they have been flight-tested.

The mass of a GaAs PV system for this application is 500 kg, including the solar cells, supporting structure, electronics, and batteries. The launch volume is the smallest of the concepts considered (approximately 1.5 m^3). A three-axis-stabilized, solar-oriented configuration easily provides the required pointing accuracy for the PV arrays. Therefore, the system has minimal impact on the other subsystems if the three-axis-stabilized configuration is selected. Photovoltaic arrays with NiH_2 batteries comprise a highly reliable system with both a significant amount of in-flight experience and a high degree of technology readiness.

Power system and configuration selection. The quantitative parameters presented in the previous sections (mass and launch volume), normalized to the lowest values and multiplied for each system alternative, are shown in table VII.

As table VII shows, the GaAs PV concept mass is similar to the other alternatives while its launch volume is the lowest. These results, along with the minimal housekeeping impacts and high space-proven reliability of this system, led to the selection of planar deployable PV arrays with GaAs cells for the power subsystem onboard SMARTS. The selection of PV arrays, in turn, drove the selection of a three-axis-stabilized configuration.

Description of GaAs PV subsystem. SMARTS is powered by three 3.0- by 8.5-m planar deployable GaAs photovoltaic arrays. During periods of minimum solar flux density (493 W/m^2) these arrays provide a total power of 4285 W at EOL, which is allocated at 3535 W for the spacecraft bus, 220 W for battery charging, and a 15-percent load contingency. The maximum cell efficiency is 17 percent, while the EOL design efficiency is estimated to be 12.7 percent. This decrease in efficiency is due to increases in electrical losses, contamination of cell coatings, thermal degradation, and radiation degradation over the duration of the mission. The arrays are equipped with partial shunt electronics to limit the power supplied to the spacecraft when the available power exceeds the spacecraft requirements.

Table VII. Power System Comparison

Power system	Mass	Launch volume	Mass × Volume
SD	1.00	2.67	2.67
RTG	1.98	1.33	2.64
DIPS	1.04	10.00	10.40
GaAs PV	1.23	1.00	1.23

The battery assembly is composed of six parallel strings containing 32 cells each. Each cell is rated at 1.1 V and 50 A-hr and operates at a 60.9-percent depth of discharge, based on the number of cycles over a 10-year lifetime. In order to increase the reliability of the battery assembly, this configuration is sized to provide the required 3535 W during the daily 1.31-hour eclipse in a worst case failure mode of three failed cells per string plus one completely failed string. A maximum charge power of 220 W at 50.7 V for the entire sunlit portion of the orbit is required to recharge the batteries after the 1.31-hour eclipse if a 90-percent charge efficiency is assumed (ref. 6).

The central bus voltage varies from 28.6 to 50.7 V, a standard operating range for conventional communications satellites (ref. 6). A single bus design is used to match the concentrated load distribution, which is dominated by the Earth-link power requirement. Load monitoring and battery charge-discharge control are also provided by the central bus.

Thermal Control Subsystem

The thermal control subsystem onboard SMARTS employs primarily passive control techniques, with active techniques employed when passive means are inadequate. Passive techniques are employed because of the lower solar flux at Mars and the SMARTS solar inertial configuration, which yields minimum external heat loads on the satellite. The major spacecraft components are thermally isolated from one another, thus enhancing the simplicity of the subsystem design. A description of the thermal control of each of the major sections follows.

Spacecraft bus. The four sides of the spacecraft bus parallel to the solar flux are highly conductive and provide ample area to reject excess heat from the onboard equipment. A moderately high surface emissivity of approximately 0.9 is required over these portions of the external surface, and this value is obtained through use of various conventional surface coatings. The solar absorptivity of these surfaces is not critical for the operational stages of the mission because the surface orientation is away from the Sun. A surface solar absorptivity of 0.3 satisfies the requirements prior to acquisition of the proper orbit and attitude. Internal heat transport is controlled through use of standard surface coatings, highly conductive materials, and proper placement of equipment, such as mounting the high-load, high-temperature TWTA's directly on the inner surface of the external walls to allow more efficient thermal transfer to the heat-rejection surfaces. Insulation is provided for the batteries and high-temperature equipment to inhibit heat transfer to other components.

Active thermal control techniques are implemented during portions of the mission, such as shadow periods or periods of reduced internal heat dissipation, to keep equipment temperatures from falling below design levels. Bimetallic louvers are used to reduce the heat rejection rate of the external surfaces, thereby moderating temperature levels in the spacecraft bus (ref. 6). Electrical resistive heaters are used to maintain battery and propellant tank temperatures.

Solar arrays. The solar array temperatures are properly maintained by rejecting excess heat through both sides of the arrays. This is accomplished with front and back emissivities of 0.7.

Antennas. Two primary thermal control requirements for the antennas are (1) to minimize thermal stresses to maintain proper reflector surface accuracy and (2) to keep feed array temperatures low to reduce thermal noise contamination of the received signals. Structural deformations and stresses are kept to a minimum through use of graphite composite structural elements with a coefficient of thermal expansion (CTE) of approximately -1.4×10^{-7} (ref. 15). These stresses are further reduced by insulation of structural elements with a Teflon or aluminum coating. The low solar absorptance of the gold-plated molybdenum mesh surface minimizes reflector deformations. As stated previously, some technology development may be required to ensure that the reflector surface accuracy is within the acceptable limit of 7 mils. Thermal coatings are also used to minimize the temperature of RF components at the feed array.

Guidance, Navigation, and Control (GN&C) Subsystem

Although the SMARTS three-axis-stabilized configuration results in relatively simple electrical power and thermal control subsystems, it does impose complex pointing requirements on the GN&C subsystem. The GN&C subsystem designed for SMARTS is a modified version of the GN&C subsystem for the three-axis-stabilized geosynchronous Intelsat 5 (ref. 16), a system which has three types of hydrazine thrusters. There are four modes of operation for the GN&C subsystem: acquisition, spinning, normal, and station keeping. These modes are used at various times during the spacecraft mission.

Preoperational phase. A scenario outlined in the Manned Mars System Study by Martin Marietta Astronautics (First Quarterly Report, NASA Contract NAS8-3712, Aug. 1987) was adopted as the means of placing SMARTS in its operational areosynchronous orbit. Figure 8 illustrates this method of orbit acquisition. SMARTS arrives at Mars onboard a cargo ship on a hyperbolic trajectory, traveling at approximately 5.6 km/s with respect to the planet. The cargo ship aerobrakes through the Martian atmosphere, decelerating into a capture orbit. The load experienced by the ship from the atmospheric drag is predicted to be $4.9g$. The capture orbit has an eccentricity of 0.85 and is inclined 75° to the Martian equator. During this orbit, the systems on SMARTS are checked out and the satellite is prepared for launch.

The apoapse of the capture orbit is located in the equatorial plane at an altitude of 33 536 km. It takes 12.1 hours to travel from the atmospheric encounter to the capture orbit apoapse. As the cargo ship nears apoapse, SMARTS is launched from the ship with its antennas and arrays stowed and its GN&C subsystem in the acquisition mode. This mode utilizes a coarse Sun sensor and telemetry information from the nearby cargo ship to determine the proper attitude for injection into the transfer orbit to areosynchronous orbit. Thrusters are used to achieve the correct orientation.

Once the proper attitude is obtained, the spinning mode of the GN&C subsystem is used. The satellite is equipped with an active nutation control system which uses a nutation sensor and thrusters to maintain stability. The nutation sensor consists of an axial accelerometer located at right angles to the thrusters in the thruster plane. Spinning Sun and Mars sensors are used to determine the spacecraft attitude. This active nutation control system is capable of driving a 3° nutation angle to 0° in 92 s.

The satellite spins up to a rate of 45 rpm about the axis of the spacecraft bus of minimum inertia. This rotation provides stability for SMARTS during the transfer orbit. SMARTS then fires its solid rocket motor (SRM), which provides the necessary ΔV of 910 m/s to perform the 75° plane change and insert SMARTS into a transfer orbit. The SRM is the same type as that used by the Global Positioning Satellite (GPS 12, ref. 17). There are 895 kg of fuel consumed during the transfer orbit insertion. The maximum load on the satellite during the SRM firing is approximately $3.0g$.

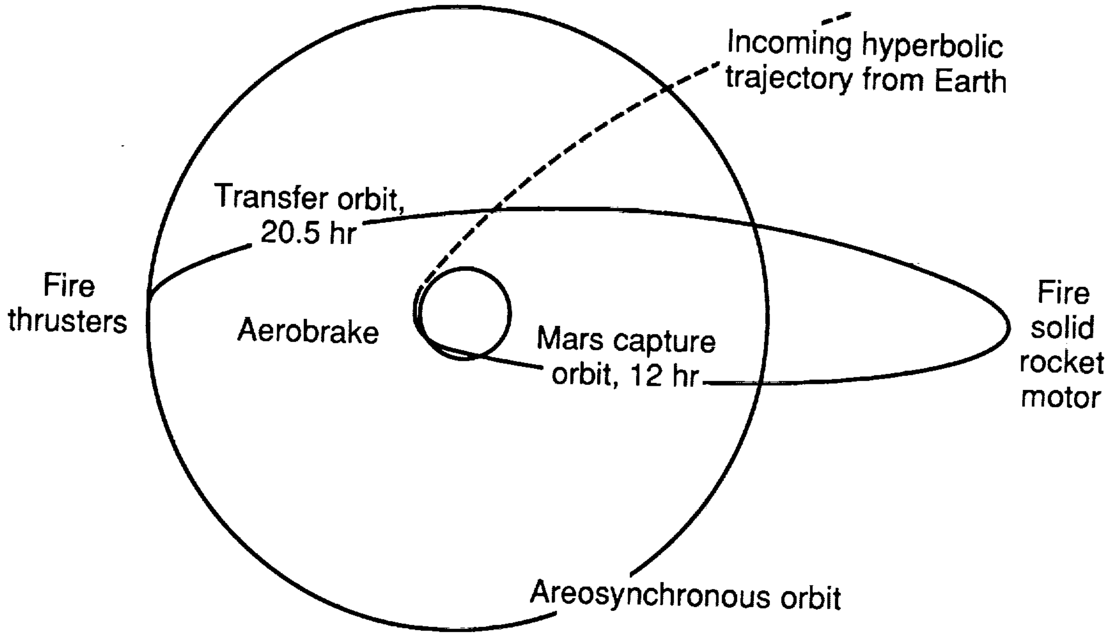

Figure 8. Areosynchronous orbit acquisition.

The periapse of the transfer orbit occurs 20.5 hours after deployment from the mother ship and is located at areosynchronous altitude (17 070 km). As SMARTS nears this point, thrusters are actuated to produce a ΔV of 190 m/s to circularize the orbit. Once in areosynchronous orbit, SMARTS uses its GN&C acquisition mode again, this time to despin to zero and orient itself toward the Sun. The satellite assumes a three-axis-stabilized attitude, deploys its antennas and solar arrays, and then repositions itself over the predetermined location within 20 days using the GN&C station-keeping mode described below. Following this repositioning, two momentum wheels are spun up, and the spacecraft assumes its normal control mode.

Operational phase. The normal control mode is the everyday attitude maintenance mode on SMARTS for the duration of its stay in areosynchronous orbit. This includes maintaining the three pointing requirements already discussed: (1) the solar arrays track the Sun, (2) the 9.1-m antenna points toward the Earth, and (3) the 3.0-m antenna points toward the subsatellite point on the Martian surface. The antennas must point within $\pm 0.1^\circ$.

In evaluating the disturbance environment in areosynchronous orbit, it was found that solar pressure is one-third to one-half that of geosynchronous orbit. Third-body gravitational effects are most significant because of Phobos, followed by minimal effects from Deimos and Jupiter. The oblateness of Mars causes orbit perturbations as well. The largest torques occur at the Mars solstice and, because of symmetry, are on the order of 6×10^{-5} N-m. All these factors are included in the sizing of the GN&C subsystem.

While in the normal control mode, SMARTS operates as a momentum-biased system with two fixed momentum wheels aligned along the pitch axis. Yaw and roll attitude are coupled and controlled autonomously through use of roll error signals from the Mars sensors. These Mars sensors are adapted from infrared Earth sensors which use the carbon dioxide spectral band to identify the location of the planet. The error signals are followed by short thruster firings that are used to limit SMARTS to a small-angle cycle. The pitch attitude is controlled in a similar manner. A Sun sensor is used to maintain the proper pointing of the photovoltaic arrays. The momentum wheels are desaturated by thruster firings.

The station-keeping mode is used to control inclination and longitudinal variations and is used during the repositioning in areosynchronous orbit, as described previously. Eleven inclination corrections and 47 longitudinal maneuvers are predicted for the 10-year design lifetime. These maneuvers are accomplished with thrusters fired in pairs and use the same sensors as those utilized in the normal control mode. The momentum wheels are commanded to a preset speed to prevent saturation from the thruster firings.

Structures

The structural components of SMARTS utilize conventional, lightweight, flight-proven technologies.

Spacecraft bus. The main bus of SMARTS is constructed of an aluminum honeycomb core sandwiched between thin sheets of aluminum. This type of structure is used for the exterior panels, interior thrust bulkhead, equipment shelves, and various stiffeners. The bus is designed to withstand the various acceleration loadings experienced by SMARTS throughout its mission.

Solar arrays. The solar array blanket design is a scaled version of the Solar Array Flight Experiment (SAFE, ref. 18), which flew successfully onboard the Space Shuttle. In the stowed configuration of SMARTS, the solar arrays are packaged to withstand acceleration loadings. In the deployed state, the solar arrays dominate the lower frequency dynamic modes. The frequencies of these modes (the first frequency is 0.29 Hz) are adequately separated from the thruster firing frequencies.

Antennas. Both antenna strongbacks are versions of the 9.1-m wrapped-rib mesh antenna which flew successfully onboard ATS-6 (ref. 7). The stowed configuration is able to withstand acceleration loadings. The stowed shape is long and thin and therefore lends itself well to the storage in the Space Shuttle orbiter cargo bay while providing the necessary reflector diameter on orbit. The materials and structural design of the Mars antenna boom are based on current Space Station Freedom truss technology (ref. 19).

Concluding Remarks

A flexible, high-quality communications network between Mars and Earth is vital to the success of future Mars missions. A comparison of potential network architectures showed that a single satellite orbiting about Mars at areosynchronous altitude best meets the assumed communications network requirements.

A Synchronous Mars Telecommunications Satellite (SMARTS) has been designed to provide this communications network. The satellite meets all the established design requirements summarized below. SMARTS establishes all the necessary links (between Mars and Earth as well as among the various Mars components) and has data storage capabilities for periods of occultation when direct communication between Mars and Earth is not possible; it provides flexible coverage of the Martian surface for various landing site selections and mission scenarios; and it transmits and receives communication on a multiple-channel system and is thus capable of accommodating very large data rates as well as three types of links (voice, data-command, and video).

SMARTS was designed to operate within a set of constraints, including communication with Deep Space Network ground stations on Earth, a 10-year design lifetime, and the use of current or near-term technology. All these constraints were met in this conceptual design study.

NASA Langley Research Center
Hampton, VA 23665-5225
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Deborah M. Badi and Jeffery T. Farmer: Langley Research Center, Hampton, Virginia.
Paul A. Garn: Bionetics Corporation, Hampton, Virginia.
Gary L. Martin: The George Washington University, Joint Institute for Advancement of Flight Sciences, Langley Research Center, Hampton, Virginia.

16. Abstract

Future missions to Mars will require a communications system to link activities on the Martian surface with each other and with mission controllers on Earth. This paper presents a conceptual design for an areosynchronous communications satellite to provide these links. The satellite provides the capability for voice, data-command, and video transmissions. The mission scenario assumed for the design is described, and a description of the communications requirements is given. The basis for selecting a single areosynchronous satellite is explained. A viable spacecraft design is then presented. Communications band selection and channel allocation are discussed. The communications system conceptual design is presented along with the trade studies used in sizing each of the required antennas. Also, the analyses used to develop the supporting subsystem designs are described, as is the communications impact on each subsystem design.

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