9th Annual AIAA/Utah State University
Conference on Small Satellites

Logan, Utah September 18-21, 1995

Technical Session IV: New Mission Concepts

MARS RELAY SPACECRAFT: A LOW-COST APPROACH

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Abstract

The next phase of Mars exploration will utilize a network of small low-cost landers, penetrators, micro-rovers and other surface devices to provide site-diversity. A direct-to-Earth communications link, if required for these landers, will drive the lander design for two reasons: a) mass and complexity needed for a steerable high-gain antenna and b) electric power requirements for a high-power X-Band amplifier (i.e., solar panel and battery mass). Total mass of direct-Earth-link hardware in recent small landers designs exceeds the mass of scientific payload. Alternatively, if communication between landers and Earth is via a Mars-orbiting relay satellite, resource requirements of the local UHF link are comparatively trivial: a simple low-gain antenna and one Watt (DC) power. Clearly, using a Mars Relay Spacecraft (MRS) is the preferred option if a MRS can be accomplished in an affordable and robust way.

Our paper describes a point design of such a mission. We have tried to arrive at the lowest-cost MRS spacecraft configuration that meets mission requirements (relay of data from stations on the Martian surface to Earth and relay of commands in reverse direction). The study requirements were based on the latest IMEWG mission model for the post-2001 era. The typical data return total is 65 Mbit/sol from six landers to be relayed in one 4-hour DSN contact using the 34-m subnet.

The mission design assumes a dedicated MRS launch on the NASA Small Expendable Launch Vehicle to LEO parking orbit. A small spin-stabilized upper stage (Star-27) will inject the spacecraft to the Mars transfer orbit. Capability of this configuration to Mars interplanetary trajectory is 98 kg. The spacecraft provides all propulsion after injection: trajectory corrections, Mars orbit insertion and maintenance, and attitude control. The simple blowdown monopropellant hydrazine system has a Av capability of 1850 m/see.

The spacecraft is spin-stabilized, pointing towards the Earth. The solar panel is behind an optically-transparent (mesh) high-gain antenna and is sized for continuous transmitter operation under the worst-case conditions. All of the highly-integrated electronics are contained in a single unit with the exception of RF hardware and attitude sensors. Attitude determination is performed with simple V-slit star and sun sensors. The standard X-Band Small Deep-Space Transponder (SDST) is used for DSN communications and UHF transceiver is used for in-situ communications at 400 MHz. No antenna switching Is required. The only active thermal control components are the hydrazine heaters. The estimated spacecraft dry mass is 42 kg with 20% margin. This MRS point design requires no post-launch deployments, has no moving parts and its full functional redundancy and expendable budget is compatible with a five year lifetime in Mars orbit. The estimated life-cycle cost of this mission is less than $50M including launch and mission operations.

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1. Introduction

Need for Mars Relay Spacecraft

The next phase of Mars exploration will consist of a network of small low-cost landers, penetrators, microrovers and other surface stations to provide site-diversity. A direct-to-Earth X-Band communication link, if required for these landers, will burden lander design for two reasons:

- mass and complexity of a steerable high-gain antenna, and
- electric power requirements of high-power X-Band amplifier.

The total mass of the direct-Earth-link hardware (antenna and its steering mechanism, power amplifier and heat-removal device, and increased solar panel size, battery capacity and power handling electronics) exceeds the mass of scientific payload in recent small landers (Mars Pathfinder and Mars Surveyor'98 lander).

MRS Concept Study

Alternatively, if the communication between landers and Earth is via a Mars-orbiting relay satellite, resource requirements of the local UHF link are comparatively trivial: a simple low-gain antenna and about one Watt (DC) power (Figure 1). Clearly, using a Mars Relay Spacecraft (MRS) is the preferred option if a MRS can be accomplished in an affordable and robust way. Our paper describes a point design of such a mission.

Lander Parameter	Direct-to-Earth Link	Relay-Spacecraft Link
RF transmit power	≈5.5 W (X-Band)	<0.7 W (UHF)
Transmitter input power	≈15 W	<1.5 w
Antenna	0.2 m dia dish	hemispherical
Antenna pointing	4° accuracy	none required
Data return volume	3 Mb/sol	10 Mb/sol
Time to return data per sol	8 hours	20 minutes

Figure 1. Comparison of Typical Lander Communication Requirements and Performance between Direct Earth Link and MRS Link

2. Mission Goals and Architecture

2.1 *Mission Objectives*

65 Mbit/sol Data Return

In our study, we have attempted to design the lowest-cost Mars Relay Spacecraft configuration that meets mission requirements: data relay from Mars surface stations and command relay in reverse direction. The study requirements were based on the latest IMEWG (International Mars Exploration Working Group) mission model for the post-2001 era. The typical data return total is 65 Mbit/sol from six landers to be relayed in one 4-hour DSN contact every day using the 34-m subnet. The proposed spacecraft is compatible with a 2001 or 2003 launch and 5-year on-orbit lifetime. The single spacecraft is designed to satisfy all data return requirements, although multiple Mars orbiters would enhance system redundancy. The orbit and in-situ communications parameters are based on the earlier studies performed by JPL and STel: sun-synchronous circular orbit with 1700 km altitude and 6:00 am/6:00 pm ascending/descending nodes.

2.2 *Implementation Options*

We have explored a number of possible configurations, trade-off options and enhancements but ultimately, the concept presented in this paper is the most simple and basic design. The potential enhancements that have been investigated include:

- Increased data-relay performance that is achieved by utilizing either a deployable antenna or a deployable solar panel (the current design uses fixed antenna and fixed solar panel). However, there is no requirements for the enhanced performance and this configuration was not further pursued because of additional risk and expense of deployments.
- Ka-Band downlink for data relay (replacing the current X-Band downlink) would result in a smaller spacecraft due to the diminished antenna size and smaller solar panel (because of reduced transmitter power). However, the current design is already compatible with the smallest existing launch vehicle available to NASA Standard Expendable Launch Vehicle. In addition, there are lower cost methods of reducing the spacecraft mass (bi-propellant propulsion and enhanced X-Band solid-state power amplifier efficiency).
- Shared launch on the MedLite launch vehicle. The Mars Relay Spacecraft is used as the cruise stage to deliver a lander to the Mars entry trajectory. After lander deployment, the cruise stage enters the Mars orbit and starts performing the data relay function.
- Addition of a space physics instrument to the Mars Relay Spacecraft. The spinner (proposed attitude control method in this paper) is an ideal platform for many space physics instruments, like magnetometers or particles detectors.
- Addition of a simple atmospheric observation experiment: synoptic observations with low-resolution camera, mutual radio occultations between two spacecraft, or simple IR or microwave sounder.
- Communications improvements beyond the basic requirements, for example: higher data rates, inter-spacecraft links, reducing DSN contact frequency and more intelligent protocols.
- Tracking and navigation support for other spacecraft and surface stations.

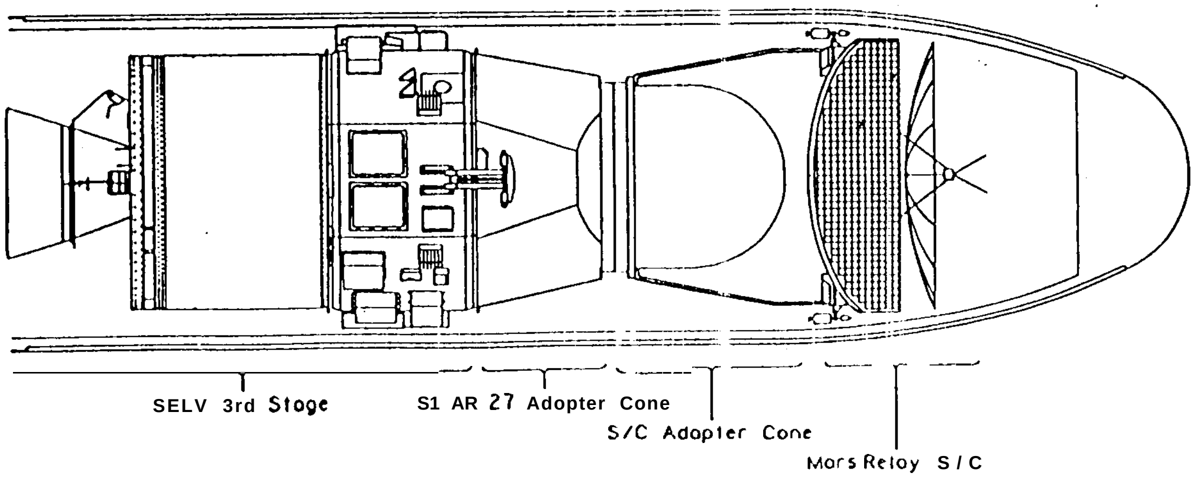

Figure 2. Mars Relay Spacecraft is Designed for Launch by SELV and Star-27 Upper Stage

2.3' *Mission Design*

SELV/Star-27

This MRS concept assumes a dedicated launch on the NASA Small Expendable Launch Vehicle to the LEO parking orbit (185 km circular, O deg inclination) from an equatorial launch site (Kourou or Alcantara range). A small spin-stabilized upper stage (Star-2 7) will inject the spacecraft to the Mars transfer orbit (Figure 2). The launch vehicle is responsible for pointing and spin-up of the upper stage. The upper stage safe & arm function, solid motor ignition and stage separation from the spacecraft are timer activated. There is no active control during the upper stage burn.

2001 Type I Trajectory

The injection capability of this configuration is 98 kg (with upper stage mass subtracted) to trajectory with C3 = 10.2 kg^2/m^2 (2001 Type 1 trajectory with 20-day launch window, Figure 3). The spacecraft provides all propulsion after the interplanetary injection:

- trajectory correction maneuvers (primarily correcting for the Star-27 *in*-jection error, Av = 90 m/see),
- Mars orbit insertion (captured to 72-hours Mars orbit from 200 km flyby altitude, Av = 1290 m/see),
- aerobraking sequence (3 months duration, to 1700 km apoapse, Δv = 45 m/see),
- propulsion maneuver to the final orbit (raising peroapse from the aerobraking altitude of about 200 km to final 1700 km, Av = 285 m/see),
- orbit maintenance and attitude control (total *Av = 50 m/see* over the mission lifetime),
- total requirement: Av = 1760 m/see and
- total capability: Av = 1850 m/see (using the simple blowdown monopropellant hydrazine system).

Mission design specifications for 2001 launch	
Trajectory type	I (no BPM)
Launch window, 20 day	03/1 9-04/08
Required C3	10.2
Mars approach velocity [m/see]	3600
Capture velocity, 3-sol orbit [m/see]	1290
Total MRS Av [m/see]	1720
SELV performance to parking orbit [kg]	486
S3/Star 27 adapter [kg]	7.0
Star 27 + MRS before injection [kg]	479
Star 27 + MRS after injection [kg]	131
Star 27 Av capability [m/see]	3680
S3/Star 27 sep system (S27 side) [kg]	1.4
Star 27 burnout [kg]	24.3
Star 27 ordnance + S&A [kg]	1.8
Star 27/MRS sep system (S27 side) [kg]	1.4
Star 27/MRS adapter [kg]	4.0
Star 27/MRS sep system (MRS side) [kg]	in MRS mass
MRS after separation from Star 27 [kg]	98
Assumed MRS propulsion Isp [see]	225
Required MRS propellant [kg]	53
MRS dry mass capability [kg]	45

Figure 3. MRS Mission Design for 2001 Launch Opportunity

3. Spacecraft Reference Design

3.1 System *Requirements*

The systematic effort towards system requirements definition was the first step of our MRS study. The following list is a quantitative summary of requirements that were considered:

- Compatibility with the launch vehicle and Star 27 upper stage and, (because of on-board propulsion) complying with launch site safety rules.
- Autonomous spacecraft initialization after interplanetary injection. The spacecraft is passive during the launch and upper stage burn. After the upper stage separation, the spacecraft must stabilize its attitude, reorient to acquire sun and be ready for subsequent ground commands, The same recovery sequence can be used after the later maneuvers (Mars orbit insertion, for example).
- Sustaining safe-state during interplanetary cruise: maintaining attitude control, thermal control and communication links, and supporting navigation and trajectory corrections.
- Executing Mars orbit insertion maneuver, aerobraking sequence and establishing operational orbit.
- Supporting nominal mission operations: pointing high-gain ant enna towards Earth, providing electric power for spacecraft, and maintaining Earth-spacecraft and spacecraft-surface communications links (both forward and return). Expendable budget is sized for five years in the Mars orbit.
- Fault management and recovery that assures uninterrupted ground commanding capability, prevents electric power loss and implements emergency attitude recovery.

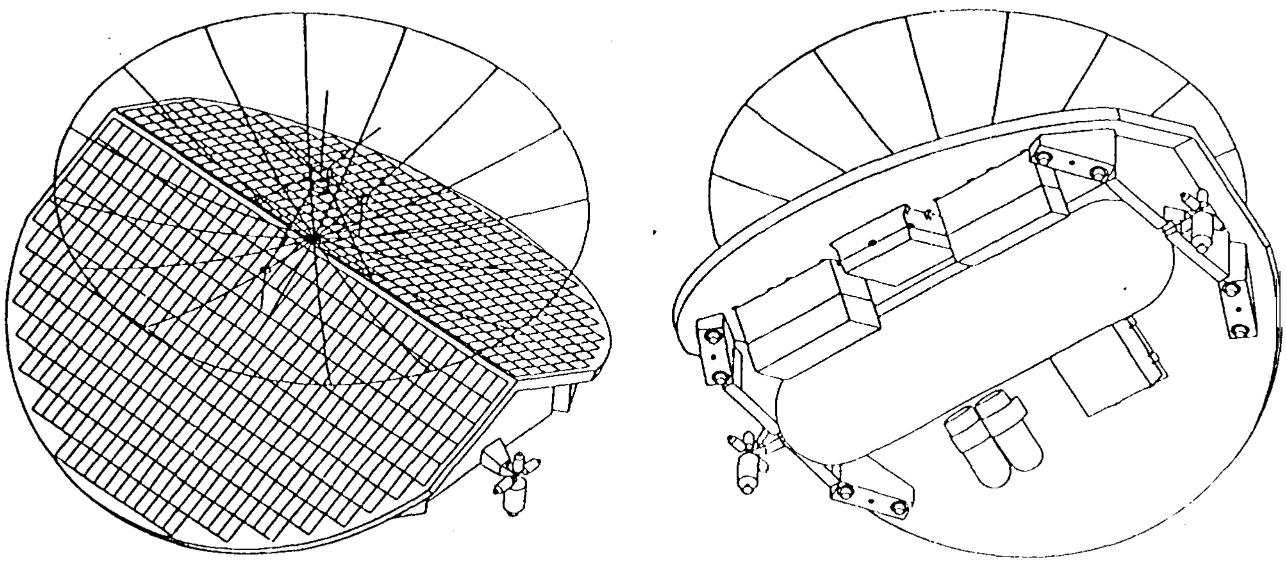

Figure 4. Mars Relay Spacecraft is Designed for Maximum Simplicity

3.2 *Spacecraft System Architecture*

The overriding spacecraft design criteria of the MRS study was maximum simplicity in hardware and operations (Figure 4). Our MRS point design requires no post-launch deployments, has no moving parts and relies exclusively on existing technology. This simple/robust approach is consistent across all subsystems (spin-stabilized attitude control, blowdown monopropellant propulsion, oversized solar panel for continuous operations without relying on battery, no antenna switching, passive thermal control, etc.) This approach is essential for reducing development cost while enhancing reliability.

No Moving Parts

The spacecraft is spin-stabilized, with the spin axis pointing towards Earth. The mesh high-gain antenna is body-fixed and 90% optically transparent. Our design relies on the fact that the angle between the Mars-Earth and Mars-sun directions is never more than 40 deg (Earth is always within 40 deg from Sun, as viewed from Mars). The solar panel is sized for continuous transmitter operation under the worst-case conditions. It turns out the maximum spacecraft transmitter performance is required during the superior conjuction when the Sun is close to Earth (as viewed from Mars) and the solar panel provides maximum power. Minimum solar panel power output occurs when the Earth is angularly farthest from the Sun, but the maximum transmitter performance is not required then.

The highly-integrated electronics contributes towards the small MRS size (Figure 5). All of the MRS electronics is contained in a single unit, with the exception of RF hardware and attitude sensors. Even with this minimalistic approach, the spacecraft provides full functional redundancy.

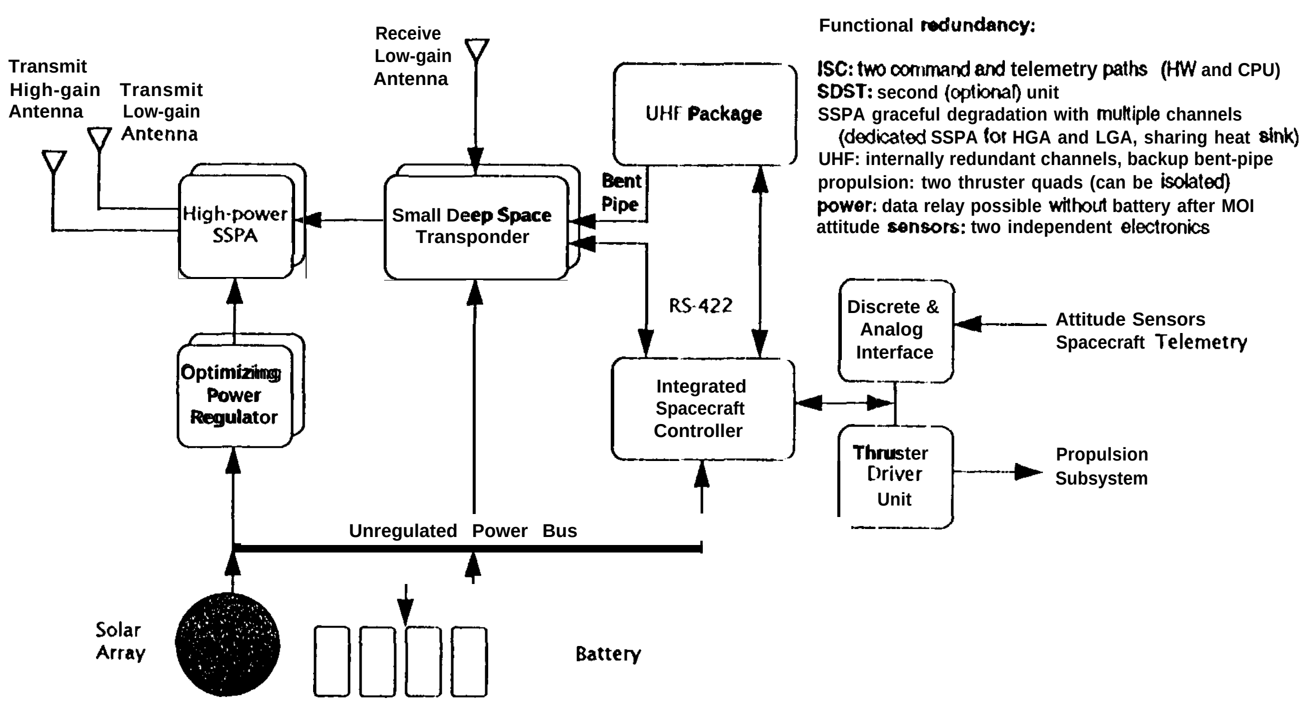

Figure 5. MRS Electronics Relies Exclusively on Existing or Near-term Technology

3.3 Integrated Spacecraft Controller

The Integrated Spacecraft Controller (ISC, Figure 6) incorporates electronics functions that are commonly divided across multiple subsystems. Therefore, small power and mass is achieved at low cost, without relying on advanced exotic technology. Our design approach always prefers simplicity over flexibility.

ISC performs the following functions:

- decode, authenticate, sequence and execute ground commands
- collect and format telemetry for downlink
- perform attitude control functions
- handle contingency and fault recovery situations
- provide interface to all RF units
- provide interface to attitude sensors and propulsion hardware
- control battery charging and electric load switching

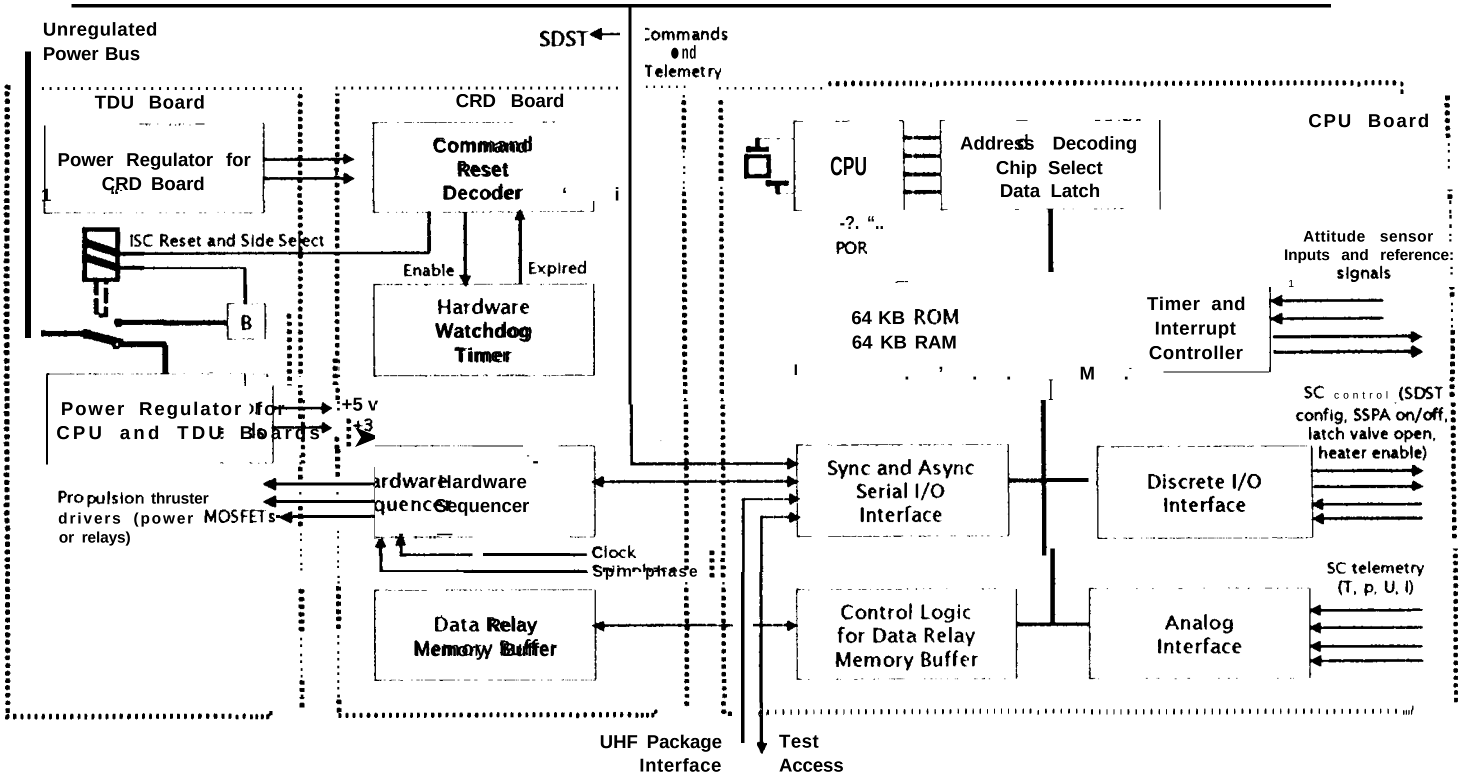

Figure 6. Integrated Spacecraft Controller Design Emphasizes Inherent Robustness

Computer

The actual spacecraft housekeeping and data processing computational **requirements** are minimal, therefore a proven 16-bit microprocessor is acceptable (80C86). 128 Mbit on-board data storage is implemented with state-of-the-art low-power solid-state memory with embedded error-correction codes. All mission-critical commands (i.e., propulsion) are executed by hardware sequencer. This allows more comprehensive testing (finite state machine). The hardware sequencer is synchronous and deterministic, not susceptible to SEU and requires minimum power (thus resistant to power bus transients).

Software

The software design aims for maximum utilization of existing proven spacecraft housekeeping and data relay code, reducing the number of spacecraft modes and states, minimizing on-board data-relay handling (limited to a simple FIFO/LIFO buffer), and reducing the complexity of attitude control functions.

Fault Recovery

The fault management approach is designed to prevent permanent spacecraft damage during unexpected events. The safe-state mode is designed to be a subset of the nominal operational mode. The preferred approach is passive autonomy and endurance (with few exceptions: upper stage burn, Mars orbit insertion and aerobraking sequence). Otherwise, the spacecraft can remain unattended indefinitely in the safe state mode.

Typical fault situations and response actions are

- command time-out: slew towards Earth, switch to LGA downlink
- unexpected computer mode: reinitialize ISC and cent inue operations
- power bus undervoltage/overcurrent or excessive bat tery discharge: autonomous shedding of non-essential electric loads
- attitude spin-rate anomaly: close propulsion latch valves
- propellant pressure loss: close propulsion latch valves

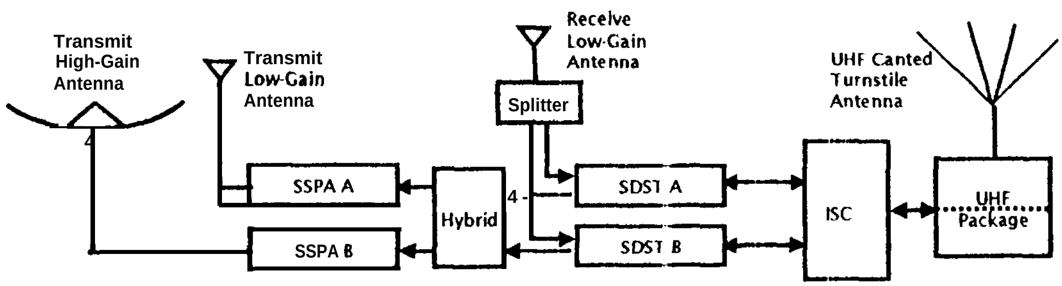

Figure 7. MRS Communication Subsystem Design is Straightforward

3.4 *Communication Design*

The communication subsystem provides links for both the spacecraft housekeeping functions and for the surface stations data relay. It consists of the X-Band system for deep-space communications between the spacecraft and DSN and the UHF system for spacecraft-surface links (Figure 7). No antenna switching nor reconfiguration is required anywhere in the MRS design. The communication subsystem can be implemented in the fully block-redundant configuration.

DSN Compatibility

The X-Band link meets ail DSN and MRS requirements. It is designed to operate exclusively over the 34-m subnet, although the 70-m subnet could be used during Mars orbit insertion and serious emergency. Link calculation assumes concatenated coding (block coding with rate 1/6 convolution

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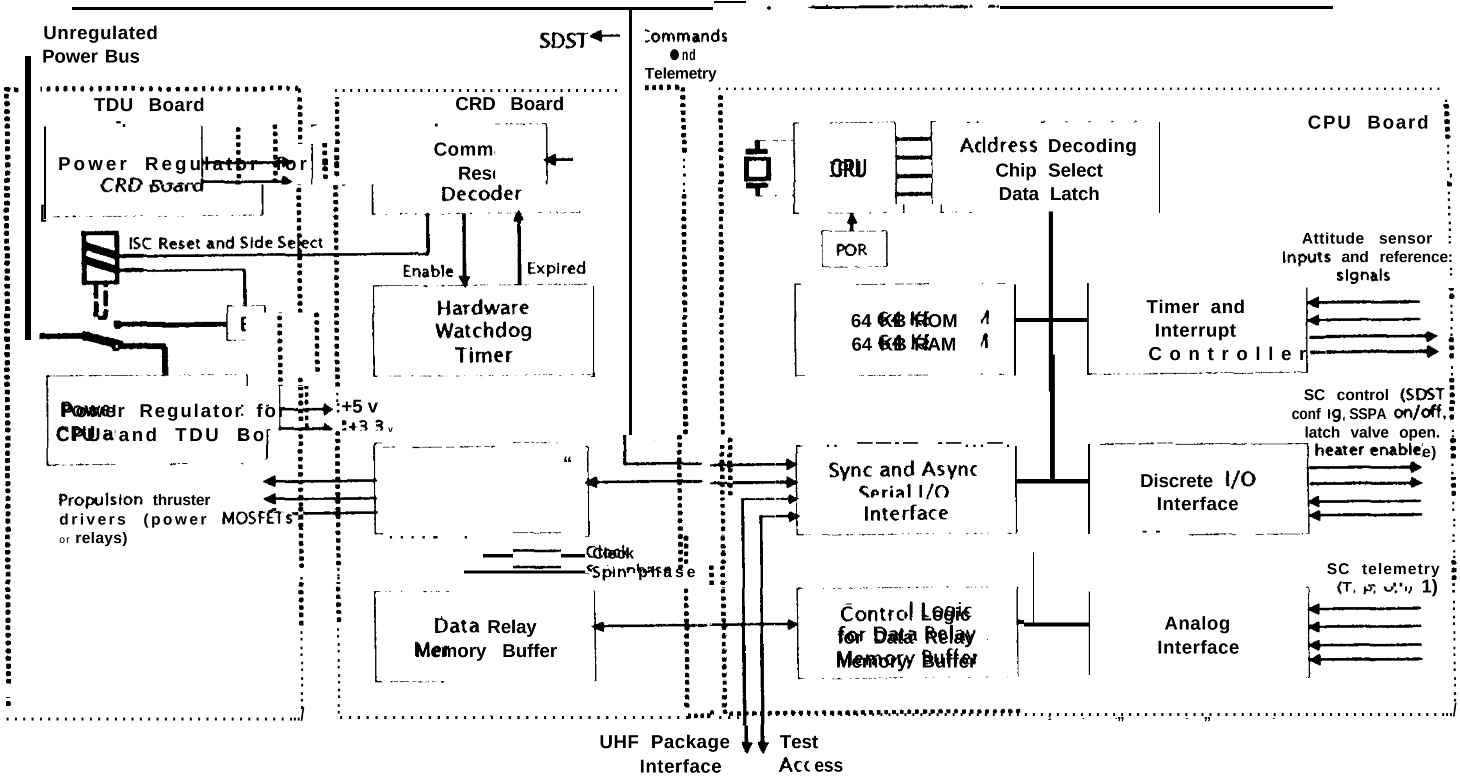

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SDST

Two standard Small Deep-Space Transponder (SDST) arc used in the MRS design. This transponder is currently under development and will be used by the Mars Surveyor'98 mission. Two identical solid-state power amplifiers can be driven by either SDST to produce RF output between 5-12 W with 3096 efficiency. Power amplifiers share the heat sink (heaviest component of the amplifier) to save mass.

Two Downlink Antennas

One power amplifier is connected to the high-gain antenna (37 dB gain) and supports data rates between 4-80 kbps (depending on the Earth-Mars distance) to be used for relay of lander data to Earth. The second power amplifier is connected to the low-gain antenna (6 dB gain) and supports low data rate (10-40 bps) that is sufficient for the interplanetary cruise, Mars orbit insertion, aerobraking and contingency. The uplink is exclusively received over the low-gain antenna for maximum robustness (high data rate on the uplink is not required).

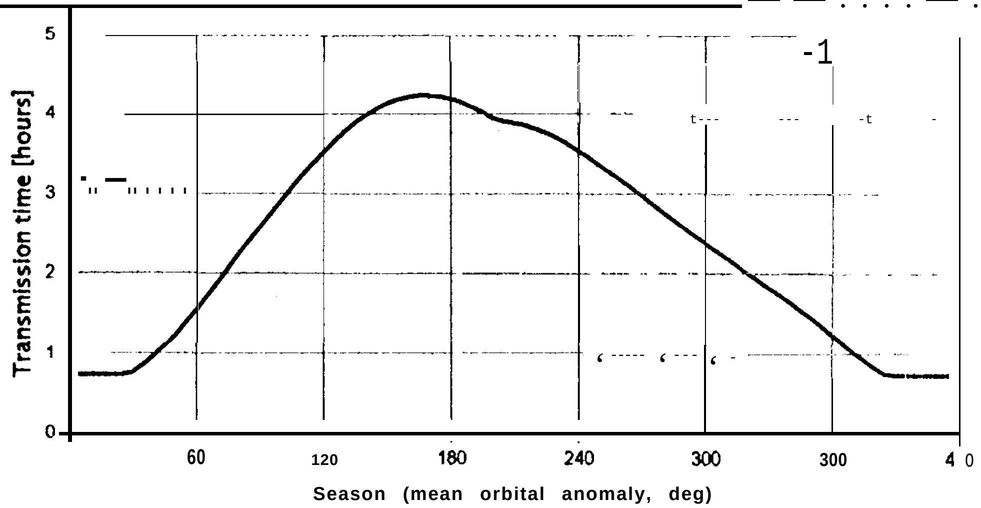

Figure 8. Data-Relay Performance: Transmission Time for 65 Mbit from MRS to DSN as Function of Time of Year

UHF In-situ Link

The UHF transceiver is used. for in-situ communications. It operates at 400 MHz, with 8 kbps data rate and BPSK modulation, and is internally redundant. It typically receives 10 Mbit from a lander every sol. Optionally, it can maintain Mars Balloon Relay backward compatibility, with the penalty of larger power consumption (6.5 W vs. 2.5 W DC power), The proven canted turnstile antenna provides nearly omnidirectional coverage (however, with varying polarization as the spacecraft spins).

3.5 *Spacecraft Subsystems*

Electric Power

The electric power subsystem provides critical power (for command receiver, computer and hydrazine heaters, <10 W) and non-critical power (typically 50 W). power is supplied by a single fixed GaAs/Ge solar panel sized for the maximum diameter allowed by the launch vehicle and illuminat ed by sun during all nominal mission phases. The" spacecraft power budget is dominated by the X-Band solid-state amplifier, which is powered from the solar panel through a combined peak-power-tracker and optimum-RF-bias switching power regulator. This approach provides maximum efficiency while maintaining simplicity and reliability y. The remaining spacecraft loads are powered through the unregulated +28 V bus (ISC, SDST and UHF). The backup rechargeable Li-ion battery lifetime is enhanced by powering only essential loads during eclipse. This reduces depth-of-discharge and requires only safe slow battery charging after eclipse.

Attitude Control

We especially focused on the attitude control approach during our study because it frequently drives the development cost, operational complexity and spacecraft lifetime. The MRS is spin-stabilized for robustness, simplicity and long-lifetime. Nominal spin-axis orientation is towards Earth except during the early interplanetary cruise, propulsion maneuvers and aerobraking sequence. Attitude determination is performed with simple V-slit sun and star sensors (Figure 9). Attitude is controlled by two thruster quads (8 thrusters total). Hydrazine fuel slosh is used for nutation damping. Accuracy requirements are modest (1.0 deg for control and 0.5 deg for knowledge) and are easily met.

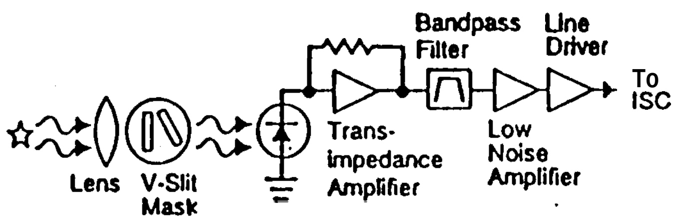

Figure 9. Simple V-slit Star Sensor is Used for Attitude Determination

Propulsion

The low-cost reliable monopropellant blowdown propulsion system is used for both axial propulsion maneuvers as well as for attitude reorientation and spin rate control. A single low-pressure propellant tank with 55 kg hydrazine capacity uses an internal manifold for propellant load equalization between two quads. Two large thrusters (90 N) are used primarily for Δv maneuvers, whereas six small thrusters (1 N) are used for attitude maneuvers. The robust system prevents loss of propellant due to a software or hardware malfunction: the latch valve must be opened and thruster drivers enabled before thruster firing, and both latch valves are closed if time-out occurs or unexpected spin rate change is detected.

Thermal Control

Thermal control design is simplified by the spacecraft compact design, spin stabilization, power-efficiency and limited number of operational modes. Particular attention must be paid to the following issues:

- heat pulse from upper stage burn,
- solar input during early interplanetary cruise,
- orientation during Mars orbit insertion,
- thermal design of the solid-state power amplifier,
- maintaining propulsion system above hydrazine freezing temperature,

- hydrazine thrusters plume effects on spacecraft structure, and
- degradation of thermal surfaces over mission lifetime.

Maintaining temperature within prescribed limits is achieved primarily by passive techniques (multi-layer" insulation blankets and radiator sizing). Preliminary thermal analysis indicates that electronics temperature can be maintained passively within -18° C and +33° C (except for the power amplifier which is between +25° C and +63° C). The only active thermal control components are hydrazine heaters.

Structure

Structural design of the MRS is driven by integration of two dominating structural components for the launch and upper stage injection environment: propellant tank (represents 6096 of initial launch mass) and solar panel with the attached high-gain antenna. Other structural issues are related to the requirement for spin balancing and to high axial loads at the end of the upper stage burn.

3.6 Spacecraft Performance Summary

Figure 10 summarizes important resource characteristics of the MRS design. It is important to remember that, for many resources, multiple bookkeeping of margins occurs. For example, power margin can be translated into enhanced transmitter RF power, which improves communication link margin and enables higher data rate for increased science data return (that is: power margin equals com link margin equals data margin). 'l"he same is true for the hydrazine propellant budget, which can be translated to the Av margin, and that corresponds in turn to the dry mass budget.

Improving Mass Margin

Finally, it must be remembered that this is the conceptual design and the mass margin for the actual development could be improved by several measures:

- switching to spacecraft hi-propellant propulsion,
- using a hi-propellant upper stage instead of Star-27,
- broken-plane-maneuver interplanetary trajectory,
- improved SELV performance,
- shared launch on MedLite,
- Ka-Band data relay downlink,
- reduced redundancy (SDST, UHF), and
- not fully circularizing the final orbit.

3.7 *Ground Segment Considerations*

The MRS ground segment approach was an integral part of this study. Mission operations were analyzed and divided into generic tasks that are shared with other spacecraft and can be handled by multi-mission operational teams or derived from existing established procedures (e.g., navigation, TCM or MOI planning, aerobraking sequencing) and MRS-specific tasks (e.g., evaluating spacecraft state of health, maintaining synchronous planetary orbit, command delivery to surface stations).

Importance of Standards

Several projects have already focused on reduction of the mission operations cost (Clementine, EUVE, Pathfinder, Discovery and Pluto Express). Guidelines have been proposed, based on these experiences: design spacecraft with built-in flexibility (passive autonomy), use exist ing multi-mission software (treat DSN as a service-oriented organization), minimize functional interface (message-based rather than bit-based service), and utilize commercial standards. For example, TCP/IP communication protocol can be embedded in the CCSDS space-link protocol so that one SFDU (CCSDS data unit) received on the ground equals one CCSDS packet plus the receive-time stamp and is delivered as one TCP/IP packet.

Specification	Current Estimate	Acceptable	Margin
Mass			
ISC and attitude sensors	3.6 kg		
RF electronics	9.7 kg		
Antennas (HGA, LGA, UHF)	3.3 kg		
Power system	4.0 kg		
Propulsion (dry w/pressurant)	7.4 kg		
Structure, harness, thermal	7.2 kg		
Total (dry spacecraft)	3S.2 kg	42.3 kg	20%
Propulsion			margin
Hydrazine propellant	56 kg	56 kg	in Av budget
He pressurant	0.3 kg	0.5 kg	70%
Av budget	1760 m/see	1850 m/see	5%
Power capability			
Solar panel cross-section area	1.0 m^2		
GaAs/Ge solar cell efficiency (BOL)	18.5%		
Total losses (packing, wiring, radiation)	21%		
Sun cosine loss	O-26%		
High-gain antenna shadowing ioss	1 o-25%		
Power capability (Mars apohelium)	65 W		
Power capability (worst-case geometry)	40 w		
Power requirements			
Spacecraft (receiver, ISC, heaters)	10.5 w		
UHF package (assumption)	6.5W		
SDST X-Band exciter (spec)	6W		
Total spacecraft housekeeping	23W		
X-Band amplifier (design spec)	40 w	42 W	5%
X-Band amplifier (worst-case geometry)	17W	17W	margin in
Communications (worst case)			data rate
Uplink data rate	40 bps	20 bps	100%
Uplink link margin	5.3 dB	3.0 dB	2.3 dB
Emergency downlink data rate	10 bps	10 bps	margin in
Emergency downlink link margin	0.6 dB	0.0 dB	DSN G/T
Data relay downli nk data rate	6300 bps	4s00 bps	40%
Data relay downlink link margin	3.0 dB	0.0 dB	worst-case
Data relay buffer size	128 Mbit	65 Mbit	1 00%
Attitude			
Control	0.8°	1.00	25%
Knowledge	0.3°	0,5°	70%
Lifetime limiting factors		5 years allocation	design margin
Consumables (propellant, pressurant)		"	1 00%
Solar panel degradation (from solar flares)		"	1 00%
Battery degradation (DOD, cycles, temperature)		"	200%
Thermal surface degradation		"	200%
Electronics TID failure		"	1 00%
Electronics thermally-induced failure		"	300%
Moving components		"	none

Figure 10. Spacecraft Performance Summary Demonstrates Robustness of Design

Passive Autonomy

We have focused on the uplink process that is estimated to consume about 70% of the labor for the classical mission operations process. The key prerequisite for reduced mission operations costs is eliminating the requirement for continuous real-time operations. Our spacecraft design adheres to the passive autonomy principle and can be left unattended indefinitely with no permanent damage (except for loss of science data).

Small Mission Operations Team

Our analysis concluded that a dedicated 6-person team would be able to operate the single MRS spacecraft in the routine data relay operations, assuming that tracking, navigation, and trajectory planning is performed by the multi-project support office. During the early cruise phase and around the Mars orbit insertion and aerobraking sequence, this team would have to be increased to about 12 persons to support more real-time operations.

4. Conclusions Path towards Low-Cost Mars Relay Spacecraft

Minimalistic Approach

Our study has adopted a "ruthless minimalism" approach towards designing the basic and simple Mars Relay Spacecraft. We have shown that such a minimal system is compatible with the SELV/Star 27 launch and can easily support 6 landers simultaneously. This design requires no post-launch deployments, has no moving parts, and its full functional redundancy and expendables budget are compatible with a five year lifetime in Mars orbit.

cost Reduction

A Mars Relay Spacecraft mission will be feasible only if the total mission lifetime cost is reduced dramatically, when compared to previous estimates. Multiple paths towards reducing mission cost must be attacked simultaneously to achieve necessary cost reduction:

- design: ruthless minimalism, narrow focus on mission requirements,
- development: innovative methods of qualifying electronic components, elimination of all non-value-added tasks,
- technology: utilization of advanced commercial technology, not for performance improvement but for power reduction, higher integration and simpler design,
- lifetime assessment: higher-fidelity more-realistic models, and
- management: integrated team, accelerated development.

<$50M Lifetime cost

The estimated total life-cycle cost of this mission is less than $50M (including launch vehicle and launch and mission operations). The Mars Relay Spacecraft would significant] y relieve DSN scheduling constraints and eliminate substantial mass and power burden of the direct X-Band link for the Mars surface landers. We are convinced that the Mars Relay Spacecraft is enabling technology for planned future mini/micro-landers and -rovers.

Acknowledgments: **During the past 18 months, countless number of our colleagues and friends have contributed ideas, expertise and time for this study. Incomplete list includes Gary Noreen, Roger Bourke, Bob Ciserone and Dave Murrow from JPL, William Horn from STel, Denise Ponchak from NASA Lewis and Eric Eng, Mike Dobbles, Gordon Hardman, Brent Helleckson, Sally Richardson, Sawat Tantiphanwadi, Trung Phan, Mark Callis and Tom Dragone from OSC. Research described in this document was carried out by Jet Propulsion Laboratory, California Institute of Technology, under contract with National Aeronautics and Space Administration, and by Orbital Sciences Corporation and Stanford Telecommunications, Inc, under subcontract 9S8741.**