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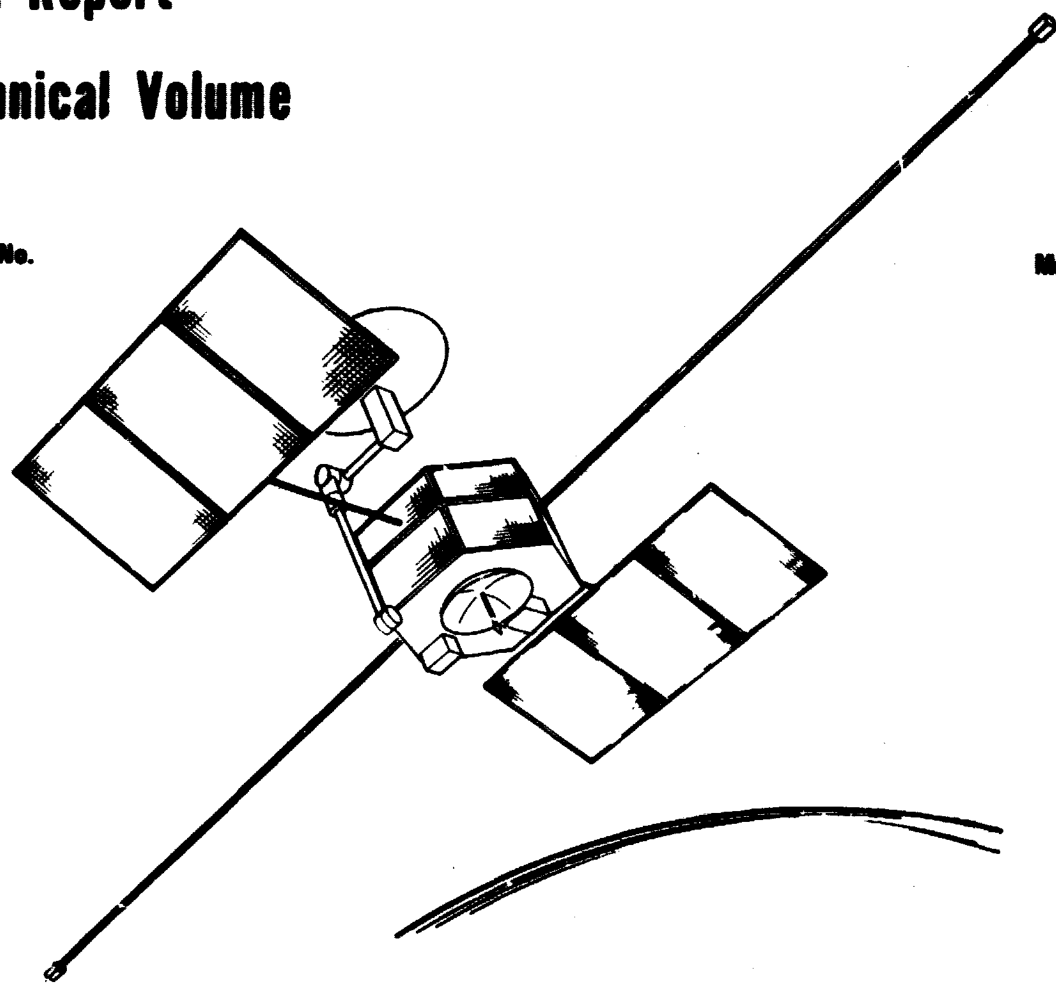
Mars Geoscience Climatology Orbiter

MGCO Extended Study Final Report

Technical Volume

Contract No.
956286

May 30, 1983



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Prepared for:
Jet Propulsion Laboratory
Pasadena, California

TRW INC.
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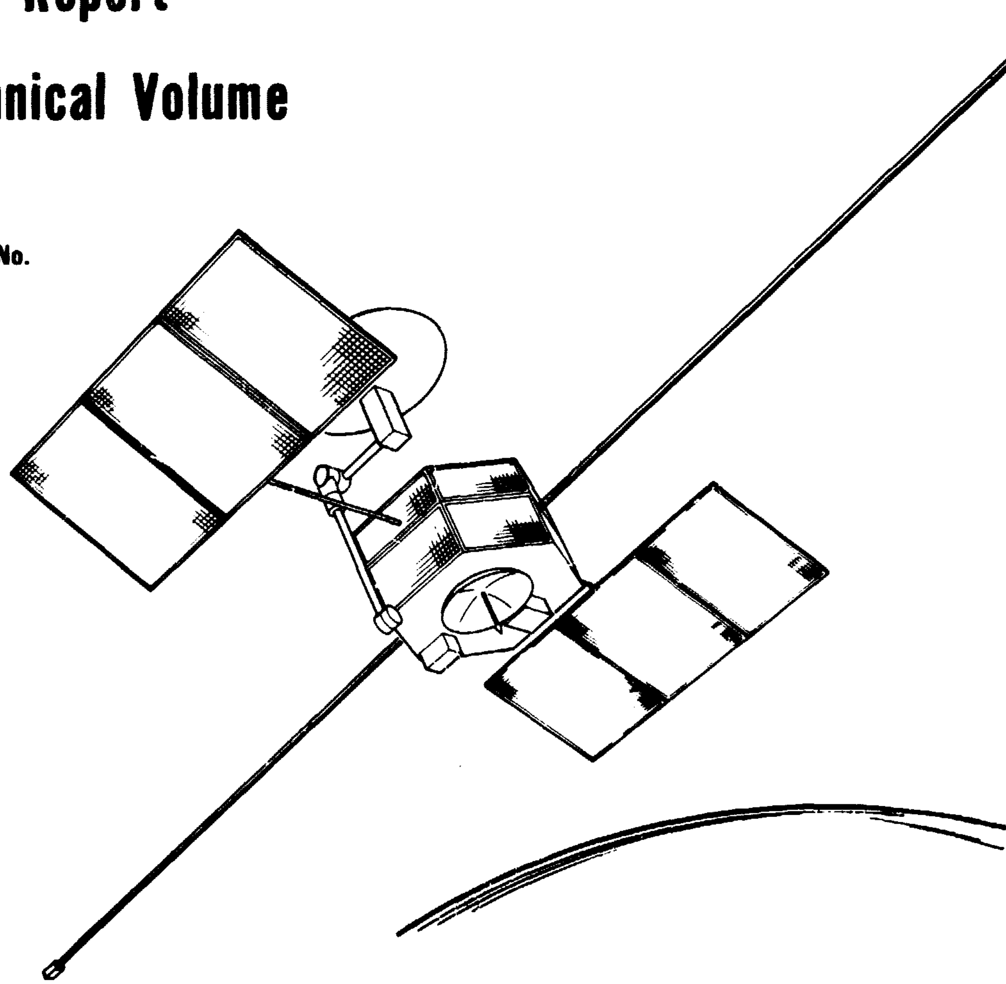

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MARS GEOSCIENCE CLIMATOLOGY ORBITER (MGCO)

EXTENDED STUDY

FINAL REPORT

By

TRW Defense and Space Systems Group

Redondo Beach, California

For

Caltech/Jet Propulsion Laboratory

Pasadena, California

Contract No. 956286

May 30, 1983

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MARS GEOSCIENCE CLIMATOLOGY ORBITER

Extended Study

1. INTRODUCTION AND SUMMARY

In 1982, the Jet Propulsion Laboratory engaged TRW under contract number 956286 to study the applicability of the FLTSATCOM spacecraft (a communications satellite built by TRW for the Air Force) to two missions: The Mars Geoscience Orbiter (MGO) and the Lunar Geoscience Oribter (LGO). TRW's final report was affirmative, and was submitted in December, 1982.

In 1983, the study of the Mars Orbiter was extended to incorporate Climatology objectives as well. The FLTSATCOM spacecraft is again the subject of the applicability study. This is the report of the extended study.

1.1 HISTORY

1.1.1 Background

To end a hiatus in the funding of new planetary exploration space missions since 1977 (when Galileo was approved) NASA, through an appointed Solar System Exploration Committee and supported by NASA Centers and JPL, undertook to identify and outline candidate missions of valuable but limited scientific objective which could be performed within a modest budget.

1.1.2 Candidate Mission Classes

Following these guidelines, missions were identified with widely differing targets, including the terrestial planets, the moon, the outer planets and their satellites. For Mars alone, a number of different missions were examined: penetrators to explore subsurface properties, a "network" of landers to examine surface properties at several locations, and orbiters which emphasize outer atmospheric characteristics (elliptical orbit), surface mapping, and the determination of seasonally varying volatile components (circular orbits).

1.1.3 1981 Studies

In 1981, Ames Research Center supported by the University of Colorado defined a Mars "Water Mission" (now "Climatology") to quantify CO_2 and H_2O in Mars' atmosphere and condensed on the surface and to track these phenomena over the Martian year. Existing earth-orbiting spacecraft were identified which could be modified to perform such a mission.

JPL similarly defined a Mars orbiter mission with emphasis on mapping the surface characteristics in several spectral regions ("Geoscience" Mission). They also felt the adaptation of existing spacecraft was the key to accomplishing the mission within strict budgetary limits.

1.1.4 1982 Studies

In 1982, JPL and ARC each solicited bids for studies of missions which could make maximum use of existing spacecraft designs or exploit the heritage of existing programs.

The ARC studies defined two Mars orbiter missions, the Climatology Mission and the Aeronomy Mission which examines upper atmospheric characteristics from an elliptical orbit.

The JPL studies concentrated on the Mars Geoscience Orbiter (MGO) and a similar lunar mission (LGO). JPL emphasized the existing spacecraft approach; ARC solicited savings by using existing spacecraft or heritage from existing programs.

TRW was awarded study contracts in each case.

1.1.5 Extension of the JPL Study in 1983

By a supplemental agreement, the JPL study was extended into 1983. This study combined objectives and payload instruments from one JPL mission, the MGO, and one ARC mission, the Climatology Mission, into the MGCO mission.

Mission requirements were also combined, and defined to greater detail. The intent is a reaffirmation of the ability of the FLTSATCOM to handle the combined mission.

1.2 STUDY PHILOSOPHY

The philosophy of the study is to maximize the scientific return of a mission whose cost is kept low by exploiting an existing spacecraft design. It is recognized that the spacecraft cost will inevitably rise as more and more changes are forced on it. If the mission and payload requirements are so demanding that extensive changes must be incorporated in the spacecraft, the advantages of using an exising spacecraft can disappear. Indeed, if changes are too extensive, it could even cost more to impose them on an existing design than to custom design a new spacecraft.

In the 1982 studies, existing spacecraft designs were identified which are compatible with the MGO mission as it was defined then. The combining and refining of MGO and climatology mission payloads and requirements should be restrained to remain within the capabilities of these candidate designs.

Section 3 is devoted to more detailed discussion of the MGCO requirements which have been forwarded (hopefully, tentatively in some instances) by JPL for the 1983 study extension.

1.3 CONCLUSIONS

- The MGCO mission, scientifically important and desirable, is a feasible mission as it is outlined for this study extension.

- The FLTSATCOM earth-orbiting communications satellite is a prominent candidate to serve as the MGCO spacecraft. Major aspects are directly applicable, including:
  - -- the incorporation of a solid orbit insertion motor
  - -- the ability to cruise to Mars in the spin-stabilized mode
  - -- ample capability for payload mass and power
  - -- attitude control in orbit tied to nadir and orbit-plane coordinates

- -- exemplary earth-orbital performance record and projected lifetime
- -- existence of an on-going procurement into the MGCO time period

- But scientific requirements, which have shown progressive tightening since 1982 must be iterated and restrained, to keep modifications and costs down.

2. EVOLUTION OF REQUIREMENTS

The evolution of requirements for the MGCO mission is traced from the 1982 study for JPL to the present.

2.1 GOVERNING REQUIREMENTS

Two documents furnished by JPL in the course of the study are accepted as the governing requirements for this report, particularly for use in Sections 4 and 5 where performance capabilities and characteristics are compared with the requirements. These documents are References (2), Standard Mission/System Performance Requirements, and (6), MGCO Science Accommodation Capabilities/Constraints, both received in March, 1983.

Reference 2 supersedes Reference 1, which was received in February, 1983, in describing the mission and system requirements.

Reference 6 follows References 3, 4, and 5 in describing the accommodations desired for the instrument payload. It reflects changes in the payload (starting with the four-instrument payload for the original MGO mission studied in 1982) and refinements in the payload characteristics and requirements.

This illustrates that the mission, science, and instruments are in a period of progressively better and more refined definition. However, as we will see, in general this progression has led to tighter and tougher requirements on the spacecraft.

Recalling that the philosophy of the study, and indeed of the mission itself, is to limit the mission cost by adapting an existing spacecraft design to the MGCO mission, this progression of requirements must be restrained (and in some instances reversed) to meet this goal. As the payload is permitted greater and greater sophistication, the cost penalty is two fold: the instruments themselves cost more; and the modifications necessary to the spacecraft to satisfy more stringent requirement aggravates the spacecraft's cost.

2.2 ITERATIVE NATURE OF REQUIREMENTS

Therefore, to conform to the adopted study and mission approach, the requirements imposed on the spacecraft must be regarded as tentative. Where they create strain and hardship to the spacecraft design they must be iterated to balance the scientific value against the increase in mission cost.

We anticipate that subsequent descriptions of the "provisional payload" will incorporate changes in the instrument characteristics, even if it means a drastic deletion or change of an instrument whose demands are too great. These changes and further refinements will continue through Phase B and into the hardware phase, but it is easier to institute changes earlier in the program.

2.3 IMPORTANT CHANGES

Here we identify and review the major changes in requirements or new statements of them since the 1982 study reported in Reference 7.

The purpose is two fold: to justify further changes to the spacecraft design submitted in that report; and, in other uses where the spacecraft design was not changed, to show why performance margins might have eroded or even given way to deficits.

2.3.1 Delay in Launch Year

The 1983 MGCO study has a schedule leading to a 1990 launch to Mars. The earlier study assumed a launch in the preceding opportunity, 1988. Because of changes in planetary positions, the Type I earth-Mars trajectory in 1988 is changed to a Type II trajectory for the 1990 launch. This launch opportunity change has a number of consequences significant to the mission. These consequences are outlined in Table 2-1.

Table 2-1. Impact of Launch Delay

Characteristic	Launch Opportunity and Trajectory Type 1988 Type I	1990 Type II	Impact
C_3 for launch $(km/s)^2$	12.4	16.3	No impact. SRM-1 stage has adequate margin in either year.
V_∞ at arrival (km/s)	2.62	2.95	Star 37 FM stage provides positive margin for orbit insertion maneuver in either case. However, in 1990 excess ΔV available to effect part of the orbit plane reorientation is reduced.
ZAP angle at arrival	104^0	54^0	In combination these changes greatly degrade spacecraft-earth communications at MOI and in the MOI attitude.
Earth-Mars range at MOI (AU)	1.23	2.48	
Earth aspect angle in MOI orientation	38^0 (s)	52^0 (n)	
Atmospheric density at orbital altitude		Increased because of solar cycle	Selection of higher orbit (350km instead of 300km altitude); greater ΔV indicated for drag compensation.

In addition the delay in launch year causes a gap in the manufacturing schedule which impacts progr[illegible] cost.

2.3.2 Orbit Orientation and Altitude

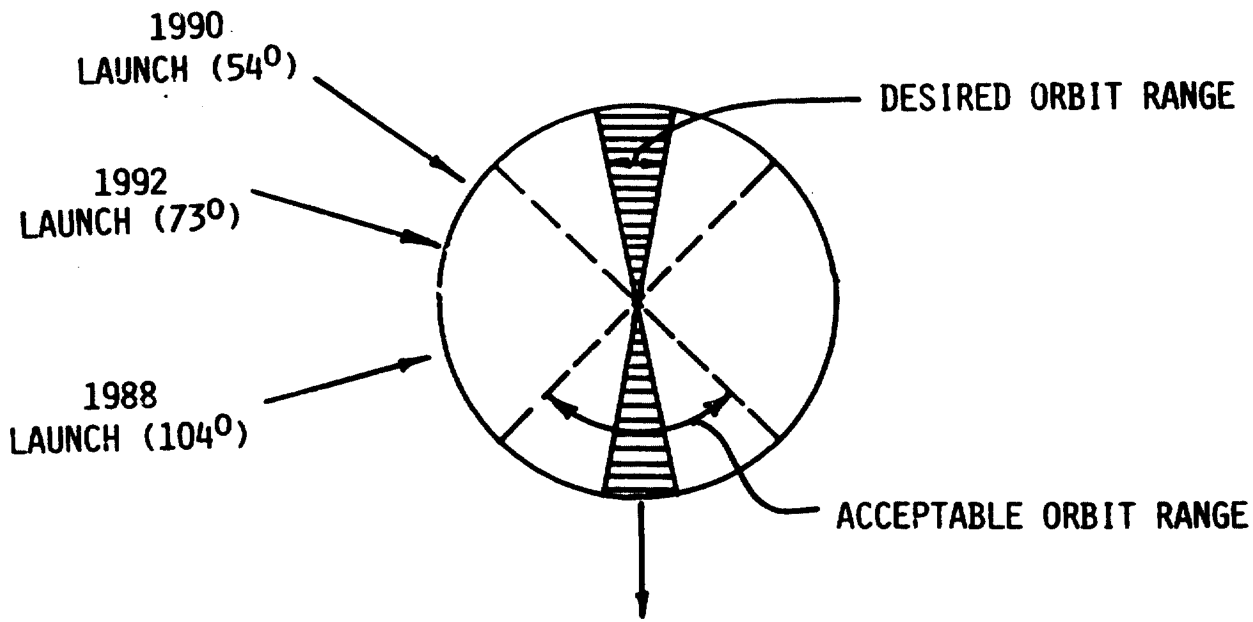
The orbit altitude has changed from 300km (1982 study) to 350km. This has a very small effect on the orbit insertion ΔV (reduction from 2244 to 2239 m/s). But it may permit a reduction in the altitude bias of the approach trajectory, and thereby a reduction in ΔV for trim maneuvers.

The mapping orbit orientation has been changed from a noon-midnight orbit in the 1982 study to a 2:00pm orbit. (These are equatorial crossings, in mean solar time). The direction of the spacecraft's approach to Mars also varies with launch year, described by the ZAP angle listed in Table 2-1. Figure 2-1 shows the relation of the approach direction to the noon-midnight orbit. For the 1988 launch, the most efficient orbital drift during the phasing orbit apparently is eastward, relative to the mean solar direction. For the 1990 launch and a 2:00 orbit, the most efficient drift is definitely westward, and of the order of 28 degrees in magnitude. (However, a portion of this can be attained in the orbit insertion maneuver by using excess MOI ΔV).

A further effect of the change to a 2 pm mapping orbit is to introduce a significant cosine loss in electrical power generation by solar arrays. This is because the sun line is now outside the orbit plane, but the solar array drive axis is perpendicular to the orbit plane.

2.3.3 ΔV Requirements

Table 2-2 shows the propulsive velocity increments associated with the MGCO mission. The existing FLTSATCOM system has a fixed limit to the liquid hydrazine mass which can be carried, so the ΔV's performed by N_2H_4 propellant are of most concern. The first column of numbers gives the ΔV magnitudes suggested in the TRW 1982 study report, Reference 7. The other two columns list values given or implied by the February and March, 1983, requirements documents, References 1 and 2.



BASELINE METHODOLOGY

- INITIAL ORBIT IS 300 ± 100 KM CIRCULAR, $i = 92.6 \pm \Delta$
  (ΔV = 2.098, 2.135, 2.005 KM/SEC)

- GO TO CIRCULAR ORBIT AT 300 KM
  (ΔV = 25 M/SEC)

- DRIFT TO APPROPRIATE SUN LINE

- FINALIZE AT $i = 92.6^{\circ}$
  ($\Delta V = 59.3 \cdot \Delta i$ M/SEC)

Figure 2-1. Mars Arrival Geometry

Table 2-2. Propulsive Velocity Increments (m/s)

Maneuver	Stage/ Propellant	Suggested in TRW's 1932 study report (1988 launch)	Attachment 1 (1990 launch) Feb 1983	Mar 1983
Injection from low earth orbit	SRM-1 solid	3730	(3922)$^{(1)}$	(3922)$^{(1)}$
Trajectory corrections in earth-Mars cruise	N_2H_4	46	60$^{(3)}$	100$^{(4)}$
Mars orbit insertion (MOI)	Star 37FM solid	2065	2239$^{(2)}$	2239$^{(2)}$
MOI error correction	N_2H_4	21	TBD	TBD
Inclination change$^{(7)}$	N_2H_4	148	164 ref	164 reference mission
Orbit maintenance (drag compensation)	N_2H_4	25	100 est	100 estimated 150 maximum
Raise to quarantine orbit	N_2H_4	45 (5)	76$^{(6)}$	76$^{(6)}$ 80 maximum
Post MOI Total	N_2H_4	239	340+	394+

(1) Corresponding to injection at 160 nmi altitude to C_3 = 16.29 km^2/sec^2.

(2) Corresponding to V_∞ = 2.954 km/s at Mars.

(3) Estimate for a 3-axis injection stage.

(4) SRM-1 injection stage without pointing update.

(5) Raises orbit from 300 to 400 km altitude.

(6) Raises orbit from 350 to 525 km altitude.

(7) This requirement can be reduced by accepting a longer drift period.

2.3.4 Instrument Mass

The instrument masses are given in Table 2-3. The first column of mumbers shows requirements provided for the 1982 MGO study. The next two columns are for the present MGCO study, as given in References 4 and 6. The table shows how requirements have increased by: (a) substitution of instruments; (b) addition of instruments; and (c) addition of contingency.

2.3.5 Instrument Power

The instrument power requirements are also given in Table 2-3, reflecting the same sources as for mass. Note particularly the large contingency requirement which has been introduced; the total required power now is almost triple that of 1982 study.

2.3.6 Pointing Accuracy

The pointing accuracy required by the provisional payload was only briefly addressed in the 1982 MGO study. In 1983 the requirements for the MGCO instruments have been defined and refined. Table 2-4 shows the values of pointing accuracy required for control and for knowledge, in milliradians. Recognizing that the FLTSATCOM capability was not the same about all axes, the instrument pointing requirements were stated separately about each axis.

Many instruments are less exacting in pitch and in yaw than in roll. However, the pressure-modulated infrared radiometer, which evidenced the tightest pointing requirmeent in February, retained that value in yaw motion in the March requirements. The FLTSATCOM attitude control system is least accurate in yaw.

2.3.7 Command

Requirements for commands were not stated for the 1982 study. For 1983, command requirements are definitive, generally in these categories:

Table 2-3. Mass and Power Requirements of the Instruments

INSTRUMENT	MASS, kg MGO 1982	MASS, kg MGCO 1983 Jan	MASS, kg MGCO 1983 Mar	POWER, W MGO 1982	POWER, W MGCO 1983 Jan	POWER, W MGCO 1983 Mar
Gamma Ray Spectrometer (GRS)	12	14	14	10	10	10
Multi spectral Mapper (MSM)	17	--	--	10	--	--
Mapp Vis. IR Spectrom (MVIRS)	--	18	18	--	12	12
Radar Altimeter (RA)	10	10	12	18	18	25
Magnetometer (MAG)	3	4	4	4	3	3
Press. Mod IR Radiometer (PMIRR)	--	13	13	--	7	15
Ultra Violet Spectrometer (UVS)	--	4	4	--	3	3
Ultra Violet Photometer (UVP)	--	2	2	--	1	4
Totals	42	65	67	42	54	72
Contingency		15	13		66	48
Total Req'd.	42	80	80	42	120	120

Table 2-4. Evolution of MGCO Pointing Accuracy Requirements

[3σ Pointing Accuracy in mrad Control/Knowledge]

				MAR		
INSTR.	FOV	JAN	FEB	X(roll)	Y (pitch)	Z (yaw)
GRS	2π Sr	50/50	50/50	50/50	50/50	100/50
MVIRS	$8^{o} \times 0.2^{o}$	9/3	9/3	9/3	25/3	25/5
PMIRR	$0.2^{o} \times 0.7^{o}$	8/3	3/2	3/2	25/2	3/2
RA	2^{o} circ	30/10	10/5	10/5	10/5	20/5
UVS	$0.1^{o} \times 1^{o}$	3/3	3/3	3/3	25/3	25/5
UVP	$0.1^{o} \times 1^{o}$	3/3	3/3	3/3	25/3	25/5
MAG	--	9/9	9/9	9/9	9/9	9/9

- Command sequences
  - -- Stored on board
  - -- Modified or replaced from the ground
  - -- Write protected decoding
  - -- Interrupts
  - -- Macros (repetitive sequences)
  - -- Checksum
  - -- Telemetering commands

- Command Classes
  - -- Immediate action
  - -- Stored for future execution
  - -- Forwarded to subsystems and instruments for further decoding

- Command rate
  - -- 7.8125 b/s backup
  - -- 31.25 (or higher) b/s normal

- Command timing and storage
  - -- Up to 1024 commands stored
  - -- Timetag capacity 96 hours, resolution 1 sec
  - -- Up to 5 commands executed in 1 sec

- Critical command protection

- Timing signals to instruments
  - -- No rollover in mission
  - -- 1 msec absolute accuracy

2.3.8 Data Handling

The 1982 study requirements were limited to the ranges of data rates to be supplied by each instrument. In 1983 the individual instruments' rates are identified for reference, but more significant to the spacecraft, the combination of instruments has a data rate limited to 1500 bits per second under normal orbital operations, and 32 kb/s available for high resolution instrument data when the telecommunications link permits (i.e., at ranges considerably less than the maximum earth-Mars distance).

Other new requirements for the 1983 study extension are:

- Operate with NASA standard X/X band deep space transponder
- Be compatible with a JPL-furnished integrated payload data system (IPDS) which interfaces with the instruments, both in distributing commands to them and in acquiring data from them and forming it into a single bit stream to be accepted by the spacecraft.*
- Support six telemetry modes, differing in the source of the data (instruments or engineering), whether it is real time or stored data, and what is being recorded.
- Implement four data rates:

Low rate	100 b/s
Medium rate	1500 b/s
Intermediate rate	8192 b/s
High rate	32 kb/s

- Conform to Packet Telemetry Guidelines per JPL Document 663-9.
- Implement an error correcting code (Reed-Solomon suggested) on top of the standard convolutional code.

---

* And JPL has verbally solicited our opinion on how TRW would implement the system if there were no JPL-supplied IPDS, but the spacecraft would have to perform its functions.

- Carry two operational tape recorders, with capabiliby for simultaneous record and playback

- Form data into transport frames

- Provide for 24 hours of autonomous operation

2.3.9 Environment

The significant environmental level imposed by the requirements is the meteoroid fluence level which must be withstood by the spacecraft. Without detailed analysis, this appears to be a relatively severe environment, and compliance could call for modification of the FLTSATCOM spacecraft structure.

2.3.10 Government Furnished Equipment

JPL has stated the intention to furnish the X-X band transponder and the integrated payload data system (IPDS) for the MGCO, as well as to assume responsibilities in the communications system. The IPDS is to serve as the spacecraft's command and data unit for interfacing with all scientific instruments on board.

This action may or may not affect the overall spacecraft requirements. But it certainly changes the interfaces observed by the spacecraft contractor, and the equipment list which spacecraft cost estimates will encompass.

3. THE BASELINE SPACECRAFT

As a result of the changes in mission and science requirements as discussed in Section 2, the baseline spacecraft for the 1983 study extension incorporates certain changes from that described in the 1982 study report (Reference 7). This section details these changes.

3.1 SIGNIFICANCE

The significance of this baseline spacecraft description is:

- To identify the spacecraft analyzed for the performance assessment (Tasks B and C of Modification No. 1 of the contract).
- To identify the spacecraft which is the subject of the cost estimates (Task A).

3.2 MGCO SPACECRAFT

Reference 7 describes the MGCO spacecraft configuration at the end of the 1982 study of the Mars Geoscience Orbiter. Figure 3-1 illustrates the MGCO spacecraft at the end of the extended study, with the high gain antenna and solar panels in the stowed and deployed configuration. A system block diagram of MGCO is shown in Table 3-1.

3.2.1 The FLTSATCOM

The MGCO uses the spacecraft bus of the FLTSATCOM, but carries the MGCO payload and data and communications subsystems appropriate to the Mars mission. The FLTSATCOM spacecraft was designed for military satellite communications from a geosynchronous equatorial orbit. In this role it carries a communications payload -- primarily operating at UHF frequencies -- and large antennas which provide earth coverage. It is launched by the Atlas/Centaur, which puts it on a transfer trajectory to synchronous altitude. Circular orbit is attained by firing a solid apogee kick motor.

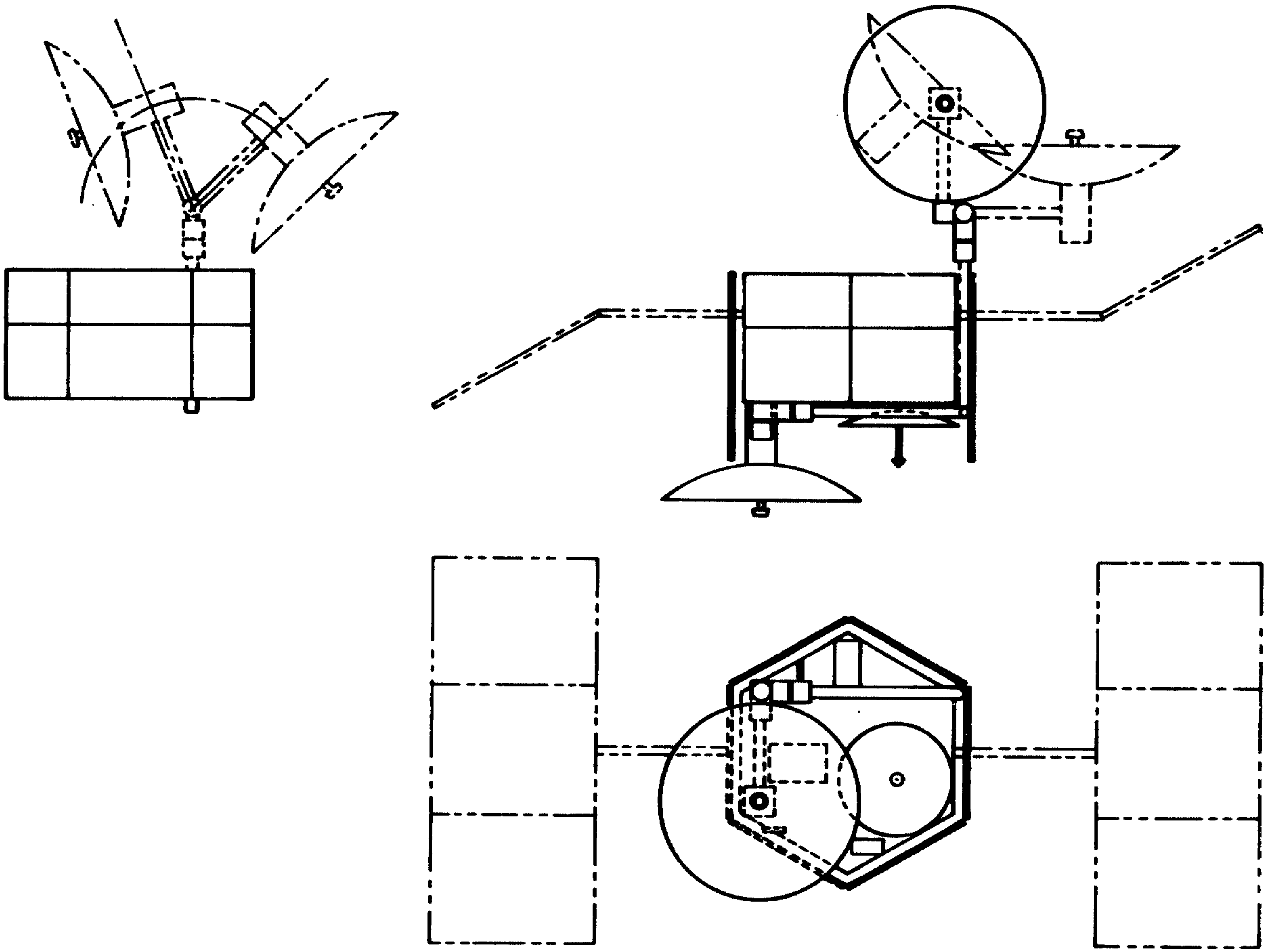


Figure 3-1. The MGCO Spacecraft Showing Solar Panels and High Gain Antenna in the Stowed and Deployed Configuration.

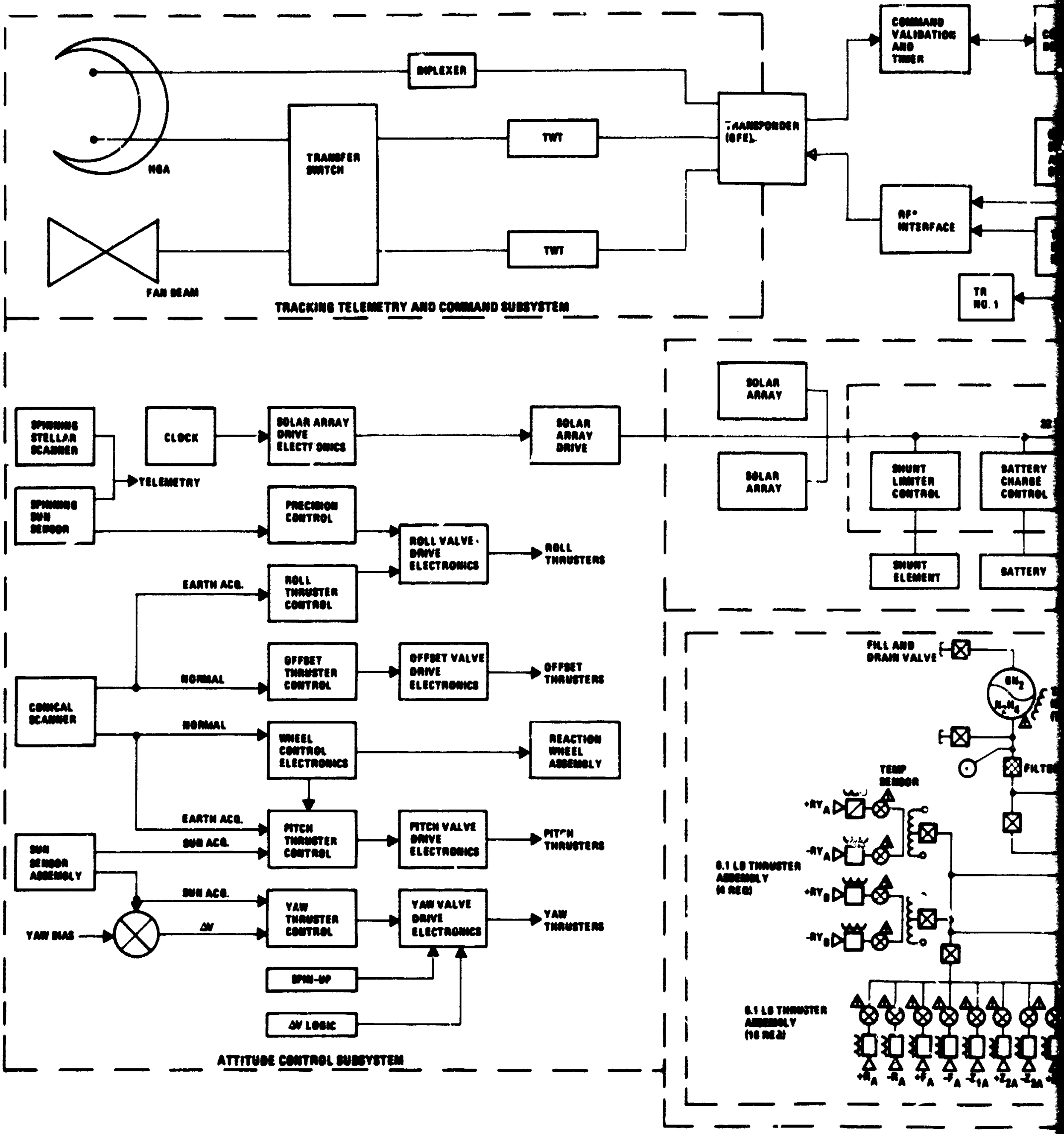


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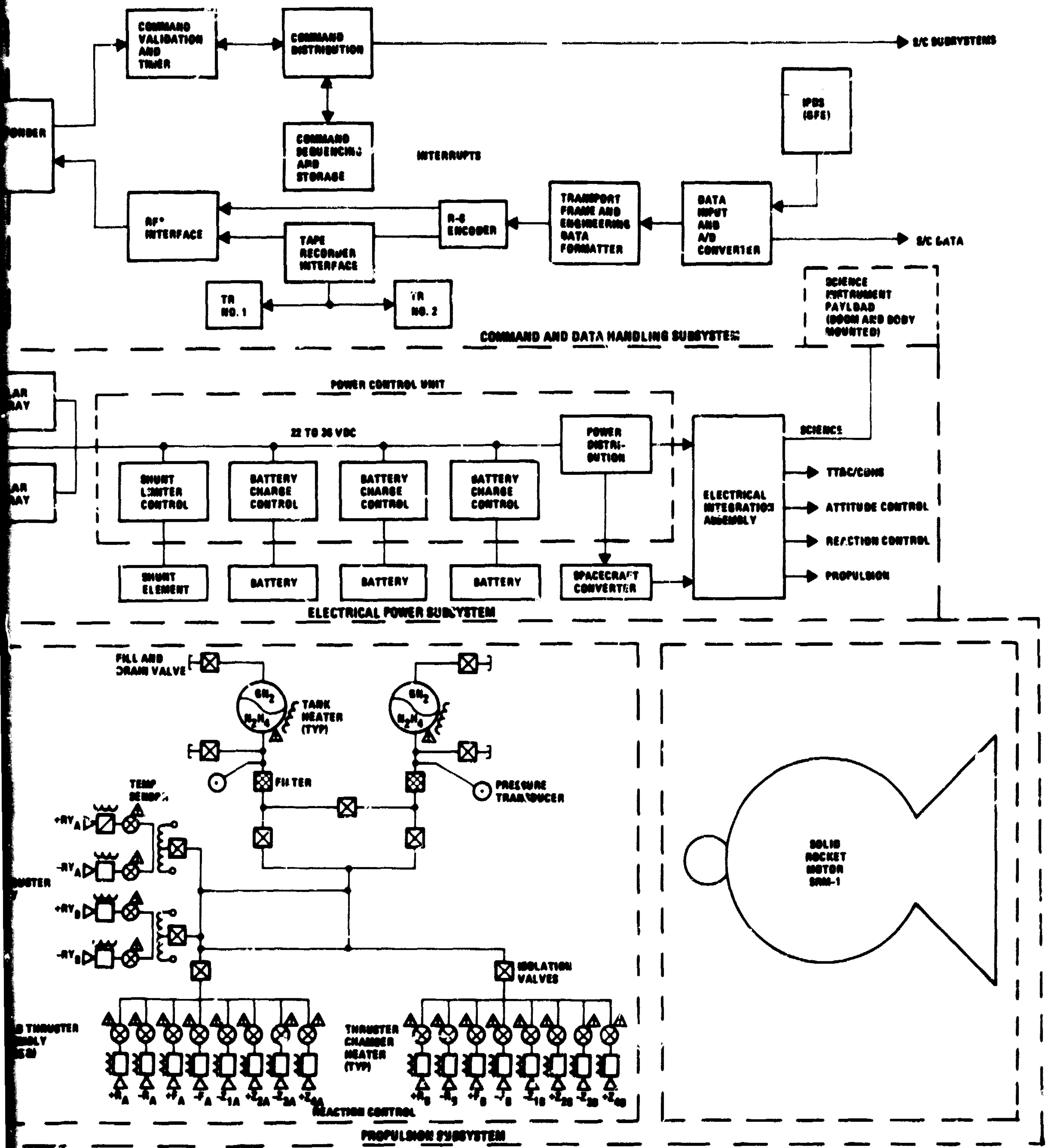
Table 3-1. System Block Diagram of MGCO



tem Block Diagram of MGCO

3-1.2

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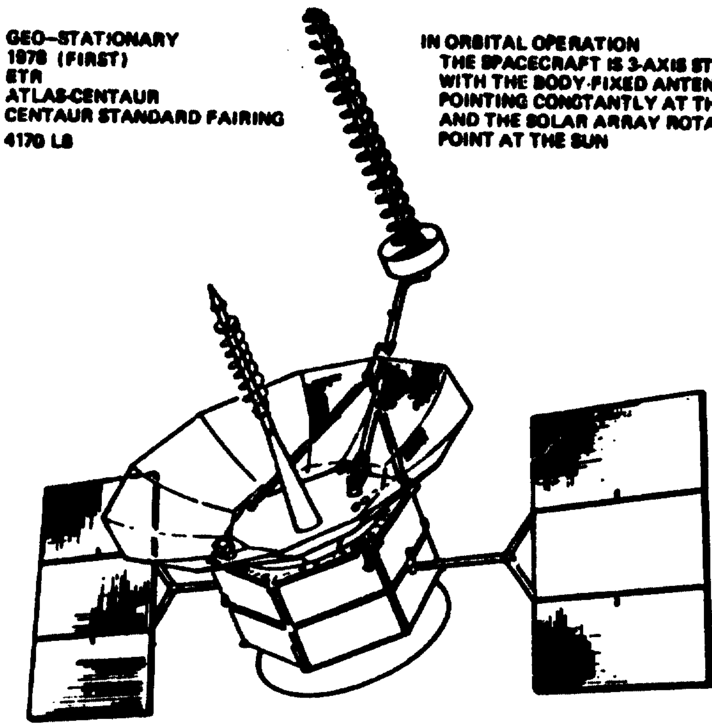
Figure 3-2.

PRINCIPAL FLTSATCOM CHARACTERISTICS

UNITS 1-6

NORMAL ORBIT:	GEO-STATIONARY
LAUNCH:	1978 (FIRST) ETR ATLAS-CENTAUR CENTAUR STANDARD FAIRING
SPACECRAFT WEIGHT:	4170 LB

IN ORBITAL OPERATION
THE SPACECRAFT IS 3-AXIS STABILIZED WITH THE BODY-FIXED ANTENNA POINTING CONSTANTLY AT THE EARTH AND THE SOLAR ARRAY ROTATED TO POINT AT THE SUN



In orbital operations, the configuration is as shown in Figure 3-1, with the antennas pointed toward the earth and the solar array drive axes north-south, i.e., perpendicular to the equatorial orbit plane.

Table 3-1 gives the major configuration details and status of the five spacecraft which have been built and flown, the three additional spacecraft which are on order, and the one which would be furnished to serve the MGCO mission.

3.2.1.1 Program Status. Five FLTSATCOM spacecraft have been built and launched. The first four are in orbit and are successfully operational, with the first one starting its sixth year. The fifth suffered damage to the antenna during launch, due to a launch vehicle's nose fairing failure. It is not operational.

The Air Force is procuring three additional units, 6, 7 and 8. Long lead parts procurement has been underway for over a year. Launches are scheduled for 1985 and 1986.

3.2.1.2 Payload. The communications payload of units 1-6 consists of earth pointing antennas and a module of electronic components on the forward (nadir-facing) side of the spacecraft bus module. For units 7 and 8 an additional payload module is added on the aft side. This module is for EHF communications, using a small additional earth pointing antenna.

For MGCO, both of these payload modules and their associated antennas are removed, and the MGCO instrument module is added in place of the UHF module. This location provides nadir viewing from the Martian orbit and permits the deployment of boom-mounted instruments, the gamma ray spectrometer and the magnetometer.

3.2.1.3 Electrical Power. The solar cells used in Units 1-5 are no longer available. They will be replaced with a newer, more efficient cell on Units 6-8, and on MGCO. The power control and storage capability is adequate through Unit 6, but must be augmented for the increased requirements of the EHF payload on Units 7 and 8. MGCO, on the other hand, has a much lower power demand, and power control and batteries are downsized accordingly, using components from other space programs.

Table 3-2. FLTSATCOM SPACECRAFT CONFIGURATION AND STATUS

<table>
<tr><th rowspan="2"></th><th colspan="2">PAYLOAD</th><th colspan="2">ELECTRICAL</th><th rowspan="2">AKM/OIM</th><th rowspan="2">LAUNCH YEAR</th><th rowspan="2">STATUS</th></tr>
<tr><th></th><th></th><th>SOLAR CELLS</th><th>POWER CONTROL</th></tr>
<tr><td>FSC 1</td><td rowspan="8">UHF FORWARD MODULE PLUS ANTENNAS</td><td rowspan="6"></td><td rowspan="5">OLD</td><td rowspan="6">OLD</td><td rowspan="6">STAR 37F</td><td>1978</td><td rowspan="4">BUILT, FLOWN, AND OPERATING</td></tr>
<tr><td>2</td><td>1979</td></tr>
<tr><td>3</td><td>1979</td></tr>
<tr><td>4</td><td>1981</td></tr>
<tr><td>5</td><td>1981</td><td>BUILT AND FLOWN</td></tr>
<tr><td>6</td><td rowspan="4">NEW</td><td>1985</td><td rowspan="3">ON ORDER</td></tr>
<tr><td>7</td><td rowspan="2">EHF AFT MODULE PLUS ANTENNA</td><td rowspan="2">AUG-MENTED</td><td rowspan="3">STAR 37FM</td><td>1986</td></tr>
<tr><td>8</td><td>1986</td></tr>
<tr><td>MGCO</td><td colspan="2">MGCO INSTRUMENTS AND BOOMS, FWD MODULE</td><td>REPLACED</td><td>1990</td><td>CANDIDATE</td></tr>
</table>

3.2.1.4 AKM/OIM. The large solid motor which is designed integrally into the FLTSATCOM is used for the apogee maneuver. It will also be used for the orbit insertion maneuver for the MGCO mission. The Star 37F has been used in Units 1-5, and it will also be used in Unit 6. Because of the heavier payload for Units 7 and 8, the larger Star 37FM is required. The Star 37FM is also very well sized for the MGCO mission, and will be employed.

3.2.2 Modifications to FLTSATCOM

We list here the modifications to the FLTSATCOM which were proposed in the 1982 MGO Study. The next 4 pages -- Table 3-2 -- comprise this listing.

3.2.3 Summary Description

The FLTSATCOM modified as discussed above is suitable for performing the MGO mission of the 1982 study, with the sequence of events as outlined in Section 4.2.

3.3 FURTHER MODIFICATIONS IN 1983

In response to the changes in requirements discussed in Section 2, further modifications to the FLTSATCOM-based MGCO spacecraft are adopted and described below.

3.3.1 Criteria for Adoption

Some changes in requirements do not demand any further changes in the spacecraft. One example is the relaxation of requirements. Another is the tightening of requirements but not exceeding the existing capabilities of the spacecraft.

Other changes can be met only by changes in the spacecraft. Where such spacecraft changes can be implemented within the FLTSATCOM approach, and where the impact is not extensive, such changes have been adopted.

Table 3-3.

Details of Changes to FLTSATCOM Which Were Proposed in the 1982 MGO Study

I. MANDATORY CHANGES (Exclusive of launch vehicle, and with no autonomy)

SUBSYSTEM	MODIFICATIONS	REASON	MGO/LGO		IMPACT	
					MAJOR	MODERATE
STRUCTURE/ MECHANISMS	• Build New Earth Facing Cover Platform (Non-Structural Member) to Mount Experiments	Replaces Removed FSC Communication Equipment and Antenna	X	X		
	• Hard Points For Mounting High Gain Antenna and Stem Devices	New Equipment Added	X X	 X		
	• High Gain Antenna and Extension/Drive Mechanisms	New Equipment - Very Similar to System That Will Be Developed For NASA GSFC/TRW GRO (Gamma Ray Observatory)	X		X	
	• Upper Stage/Spacecraft Adapter	For Fit and To Handle STS Lateral Loads	X			
PAYLOAD	• Integration of Scientific Instruments	Physical, Power, Housekeeping, Command, Localized Heaters (As Required)	X	X		X
	• STEM Devices	To Mount Scientific Instruments and to Maintain Balance - Housekeeping Integration	X	X		
	• Umbilical/Harness Changes	To Account For New (Scientific) Payload and STS Integration	X X	X		 X

Table 3-3.

Details of Changes to FLTSATCOM Which Were Proposed in the 1982 MGO Study

(Continued)

SUBSYSTEM	MODIFICATIONS	REASON	MGO/LGO		IMPACT		
					MAJOR	MODERATE	MINOR
AVCS	• Add Spinning Stellar Field Sensor and Add to T/M Stream	To Replace Spirning Earth Sensor	X				X
	• Remount Spinning Earth Sensor	To Assure Earth Crossings In Cis - Lunar Space		X			X
	• Add Conical Horizon Scanners and (Possibly) Adapter To Assure No Basic Change In Signal Character Into AVCS Electronics	To Replace Linear Horizon Scanners	X	X			X
	• Add Adapter To Solar Panel Input Drive Signal To Point HGA In Elevation. Add Separate Command To Point In Azimuth	To Permit Open Loop Pointing Of HGA. Azimuth Commands Will Be At Low Frequency (Once/Week?)	X				X
COMMUNICATIONS AND DATA HANDLING	• Install New TT&C System With Antennas; Add Data Handling System	FSC Has No Data Handling/ Processing System, and Its TT&C Is Not Compatible With DSN Or Martian Mission	X	X	X		

Table 3-3.

Details of Changes to FLTSATCOM Which Were Proposed in the 1982 MGO Study

(Continued)

SUBSYSTEM	MODIFICATIONS	REASON	MGO/LGO		IMPACT		
					MAJOR	MODERATE	MINOR
POWER	• Reduce Height Of Solar Panels To ½ Height of FSC	To Account For Smaller Power Needs, And To Permit Movement Of STEM -Attached Instruments During Cruise Out For Calibration Without Making A "Hole" In Solar Panel	X	X			X
	• Adapt DSP/HEAO Modified Power Control And Limiting Units	To Maintain Voltage Within Narrow Limits (FSC Permits Voltages Out Of Eclipse To 70V)	X	X	X		
	• Add New Batteries	FSC Capacity Not Needed. Lighter More Efficient (DSCS-II) Batteries Available	X	X			X
	• Revise Power Harness	To Accommodate System Design Changes	X	X			X
	• Revise Solar Panels To Be 45° Out Of "Plane" Of Rotating Arms	To Maintain Solar Power Throughout The Year		X			X

Table 3-3.

Details of Changes to FLTSATCOM Which Were Proposed in the 1982 MGO Study

(Continued)

SUBSYSTEM	MODIFICATIONS	REASON	MGO/LGO		IMPACT		
					MAJOR	MODERATE	MINOR
PROPULSION/ ORDNANCE	• No Essential Changes	May Require One Additional Propulsion Valve And Added Ordnance To Achieve STS Safety Requirements	X				
THERMAL CONTROL	• No Essential Changes	May Require Reduction In Second Surface Mirror Size To Accommodate Lesser Instrument Load	X	X			
II. CHANGES POSSIBLE BY JPL DICTATE							
ACTIVE NUTATION CONTROL	• Add Two Or Three Accelerometers And Add Logic To AVCS Electronics	Might Be Necessary If Upper Stage Booster Is Spin Stabilized. Alternatives, Including Doing Nothing, Must Be Studied	X			X	
"SAFE HAVEN" MODE	• Possible Modifications Varying From Addition Of Battery Capacity To Varying Degrees Of Sophistication For Autonomous Action	To Prevent Spacecraft From Expiring During Undetected Fault Period Up To 20 Hours, Plus An Addition 10-20 Hours (Minimum) Or Longer To Analyze And Correct Fault	X		? (As Studied) - Ranges From Minor (Battery Addition) To Major (Moving "Auto Safe" System From Ground To Spacecraft)		

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Where the impact is extensive (and expensive) to accommodate the new requirement, or the means to accomplish it are inconsistent with the FLTSATCOM usage, then the change has not been adopted and the result is a performance deficiency.

To summarize, further spacecraft changes have been incorporated only where necessary and feasible within reasonable limits.

3.3.2 Solar Array Expansion

Because of the increased power required for scientific instruments, the solar array area which had been reduced by 50% for the 1982 study is increased by 37% from 10.8 m^2 (116.4 ft^2) to 14.9 m^2 (160 ft^2). It is still 31% below the area of the FLTSATCOM solar array.

Because of the change from a noon-midnight orbit to a 2 pm orbit, the array must now be deployed with a cant angle of 30° to 40° (angle between solar array plane and $\pm$ Y axis) to avoid excessive cosine loss.

3.3.3 RF Subsystems

The imposed changes here are the deletion of S-band communications link which reduces the system to X-band up and X-band down, the fact that the X-X transponder is to be furnished by JPL, and the increased communication distance during the cruise phase resulting from the 1990 launch. The resulting effects on the spacecraft are:

- deletion of all S-band components from the power amplifier to the antennas
- the addition of a medium-gain X-band fan beam antenna with gain increased to 11 dB in the spacecraft's XY plane
- revision of the X-band low-gain antenna
- removal of the transponder from the list of components costed as part of the spacecraft.

The deletion of S-band components means that the high-gain antenna no longer required a dual S-X feed; it is now entirely X-band.

3.3.4 Data Handling

The data handling requirements have been largely defined since the 1982 study. Many of these requirements are not satisfied by the data handling system described in the 1982 study report. The major aspects of the new requirements which are not consistent with that system are:

- adoption of telemetry according to the JPL packetization standard
- the creation of an integrated payload data system (IPDS) which is furnished by JPL and which serves to distribute commands from the spacecraft to the payload instruments and to organize the acquisition of data packets from the instruments into a single fixed-rate bit stream forwarded to the spacecraft.

In addition, the science data rates and volume which must be processed, stored, and transmitted by the spacecraft have been somewhat reduced from the 1982 study. As a result, the complement of command and data handling components (except for the tape recorder) has been entirely revised.

The details of the proposed Command and Data Handling Subsystem are discussed in Section 4.7.

3.3.5 Hydrazine Propulsion

In the case of the liquid propulsion subsystem the requirements call for increased capability. The baseline FLTSATCOM system can support the mission, but it will not meet the maximum requirements listed. A modest increment to the design increases propellant capacity, and, while not meeting all maximum requirements simultaneously, we will see in Section 4 that the mission can be conducted much more comfortably with this addition. Therefore, both propulsion subsystem designs are presented:

Baseline:	
Propellant	Two 22.5 inch diameter tanks
Pressurization	Blowdown (internal ullage)
Thrusters	16 at 1.0 Lbf thrust 4 at 0.1 Lbf thrust
Optional:	
Propellant	Same two tanks
Pressurization	External pressurant tank added. Pressure regulated to 350 psi most of mission, decreasing to 150 psi at end of mission.
Thrusters	Same complement

No further option is presented. To call for an increase in propellant capacity beyond that of the optional propulsion subsystem would be a major change to the FLTSATCOM design. Such a major modification has not been entertained in this study.

3.3.6 Safe Haven Mode

There has been discussion that the MGCO spacecraft needs a capability for fault detection and protection (safe haven mode) and an autonomous operating period in such a mode which are greater than the FLTSATCOM has.

While such an enhancement has not been completely defined, an approach is identified which suggests what triggers should enable the safe haven mode, and what routine should be followed after enabling. (See Table 3-3 and 3-4).

Considering the features already available on the FLTSATCOM, the necessary additional "safe-haven mode" provisions include a dedicated sequencer for the command routines which are to be triggered when the fault is detected, and increased battery capacity to sustain the spacecraft until on-board routines plus ground commands can restore full operations.

However, the MGCO design presented and analyzed in this report does not incorporate any such provision. No weight has been allocated for battery growth, and cost estimates do not include battery growth, sequencer installation, or analyses necessary for the safe haven mode.

Table 3-4.

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AUTOMATIC SAFE HAVEN OPTIONS

- "SAFE HAVEN" INVESTIGATIONS WILL BE PERFORMED DURING A LATER STUDY

- MAIN MALFUNCTION TRIGGERS:
  - LOSS OF HORIZON
  - FAILURE TO ACKNOWLEDGE RECEIPT OF COMMAND
  - LOSS OR ABNORMAL DECLINE OF SOLAR POWER
  - PROPULSION ALLIED TRIGGERS (ABNORMAL LOSS OF PROPELLANT)

- SAFE HAVEN MODES:
  - IN EVERY CASE POWER DOWN SPACECRAFT TO A MINIMUM
  - LATCH PROPELLANT VALVES
  - UNPOWER MOMENTUM WHEEL

- ALTERNATIVE POWER SOURCES DURING THE SAFE HAVEN MODE
  - NEW REDUNDANT BATTERIES TO EXTEND SPACECRAFT LIFE IN A POWERED DOWN STATE 30 - 40 HOURS
  - SIMPLE PROGRAMMABLE SEQUENCER TO DIRECT S/C TO A SUN SEEKING SAFE HAVEN MODE

PROGRAM MANAGEMENT DIVISION

Table 3-5.

THE SUN SEEKING SAFE HAVEN MODE

- UPON A SUITABLE TRIGGER, THE MOMENTUM WHEEL DRIVE IS DISABLED AND THE PROPELLANT TANK(S) ISO VALVE IS LATCHED. THEN, A SEQUENCER (PROGRAMMABLE TO ACCOUNT FOR CONFIGURATION CHANGES TO REDUNDANT AVCS HARDWARE THAT MAY HAVE OCCURRED) WILL

  1) DISABLE NORMAL AVCS MODE

  2) ENABLE A SELECTED (PITCH AND YAW) SET OF 1.0 POUND THRUSTERS

  3) ENABLE THE SUN-POINTING CONTROL MODE, WHICH IS PART OF THE NORMAL NADIR CAPTURE SEQUENCE

  4) CAGE THE SOLAR ARRAY NORMAL TO THE SUN-POINTING AXIS

  5) SWITCH TRANSMITTERS TO OMNI

4. MISSION/SYSTEM PERFORMANCE CHARACTERISTICS

4.1 ORGANIZATION

The purpose of this section is to indicate the performance characteristics of the MGCO spacecraft, to compare this performance with requirements, and thereby determine and note the excess capability (or deficiency, if such is the case).

The order of topics considered is roughly that of the standard Mission/System Performance Requirements (Reference 2).

However, some characteristics can not be treated entirely independent of other characteristics. For example, the propulsive capability for spacecraft maneuvers is related to the payload mass which can be carried. Accommodating one objective may impose limits on the satisfaction of the other.

So in addition to determining excess (or deficient) capability we will attempt to indicate the pertinent tradeoffs to be considered.

4.2 FLIGHT SEQUENCE

It is necessary to establish as a framework for capability determinations, the general manner in which the FLTSATCOM-derived spacecraft is to perform the MGCO mission. This is best done in relation to the sequence of major events comprising the mission.

The following discussion is based on the mission description of Section 1.2, Reference 2.

4.2.1 Launch

The MGCO spacecraft and payload are launched in August or September, 1990, making use of a Space Shuttle flight which carries it into low earth orbit and an injection stage using the SRM-1 solid rocket engine to put it on an earth-Mars trojectory.

When deployed from the Shuttle cargo bay, the spacecraft with the injection stage is spun at about 2 rpm about the longitudinal axis, and oriented with this axis parallel to the ΔV necessary at injection by the SRM-1 firing. The spacecraft's high-gain antenna is stowed, folded in, and the solar arrays are stowed in the form of a hexagonal prism surrounding the spacecraft body, with solar cells facing out. (The HGA and solar arrays will remain stowed in this manner until the spacecraft has reached Mars and has been inserted into orbit about Mars.)

Before the SRM-1 is fired, the spacecraft may be precessed to assume the desired spin axis orientation more accurately, and it will be spun up to $\sim$ 30 to 40 rpm. With the SRM-1 firing, the spacecraft is put on a trajectory which departs from the earth and embarks on an interplanetary trajectory to Mars.

Between the deployment of the spacecraft and injection stage from the Shuttle and the separation of the spacecraft from the spent injection stage several minutes after firing, a number of <u>autonomous spacecraft actions</u> are necessary, including the spin up, SRM-1 ignition, separation from the spent injection stage, and spin down.

These actions are handled by an onboard sequencer/timer (which may be a part of the spacecraft command system). This sequencer is enabled only if <u>both</u> of these signals are received: "breakwires" triggered mechanically by the act of separation from the Shuttle; and a radio command to the separated spacecraft. This approach satisfies Shuttle safety requirements--these actions would have to be considered catastrophic to the Shuttle if they took place in the cargo bay.

This calls for a reliable radio uplink to the separated spacecraft. A radio downlink from the spacecraft may also be needed, for one of these reasons: if it is desired to confirm the enabling of the spacecraft sequencer and the subsequent events it controls; or if it is necessary to command pointing maneuvers to improve the accuracy of the injection firing.

4.2.2 Cruise

The cruise phase lasts from the earth to Mars. Because of the 1990 Type II trajectory this takes almost 365 days, and takes the spacecraft out to $\sim$ 1.69 AU from the sun before arriving at Mars at 1.64 AU from the sun and 2.48 AU from the earth.

The spacecraft is spin stabilized during this phase at about 5 rpm--spun down from injection--with the spin axis perpendicular to the ecliptic plane except when reoriented temporarily for trajectory correction maneuvers.

This orientation permits adequate solar power generation with the still undeployed solar panels, and two-way communications with the Deep Space Network at X band via an antenna with 11 dB gain and its fan beam in the plane perpendicular to the spin axis. The primary purposes of communication are for engineering data transmission and for Doppler tracking--instruments' data are not yet continuously available.

Of three anticipated trajectory correction maneuvers during cruise, two will be executed near earth ($\sim$10 and 30 days after launch) and one near Mars ($\sim$10 days before arrival).

4.2.3 Phasing Orbit

The phasing orbit is initiated by the Mars orbit insertion maneuver, which places the spacecraft in a near polar, circular, 350km orbit about Mars with an initial orientation relative to the sun line corresponding to an equatorial crossing at 3:00 to 4:00, local solar time.

It is desired to reduce the sun/orbit plane angle to a 2 PM orbit. The phasing orbit duration of 2 to 4 months allows for this amount of relative orbit plane rotation, the exact rate being proportional to the extent to which the phasing orbit inclination is less than 92.78^{0}, the sun-synchronous inclination.

Soon after arrival in Mars orbit, the spacecraft is commanded from the ground to undergo changes in attitude control, communications and in power generation.

The spacecraft is despun, and its attitude is 3-axis stabilized relative to orbit plane and nadir directions. Attitude references are furnished by Mars oriented horizon sensors, and stability is augmented with a biased momentum (wheel) about the pitch axis. Roll and yaw corrections are made with the hydrazine propulsion system.

With the spacecraft despun, two deployments are effected. The solar array is unfolded and deployed, so that instead of six panels comprising a hexagonal prism, the array has two large plane segments, one on each side of the orbit plane, rotatable on a shaft parallel to the pitch axis. This improves the area of solar radiation intercepted by a factor of almost π. The increased power production can now support full scientific operations. Figure 4-1 shows how the solar array deploys for the FLTSATCOM in synchronous orbit about the earth.

The second deployment is that at the 2 meter, two-axis gimbaled, high gain antenna (HGA). Figure 4-2 shows the deployment sequence of the HGA. It may then be kept pointing at the earth to support the higher downlink data rates required by the scientific payload.

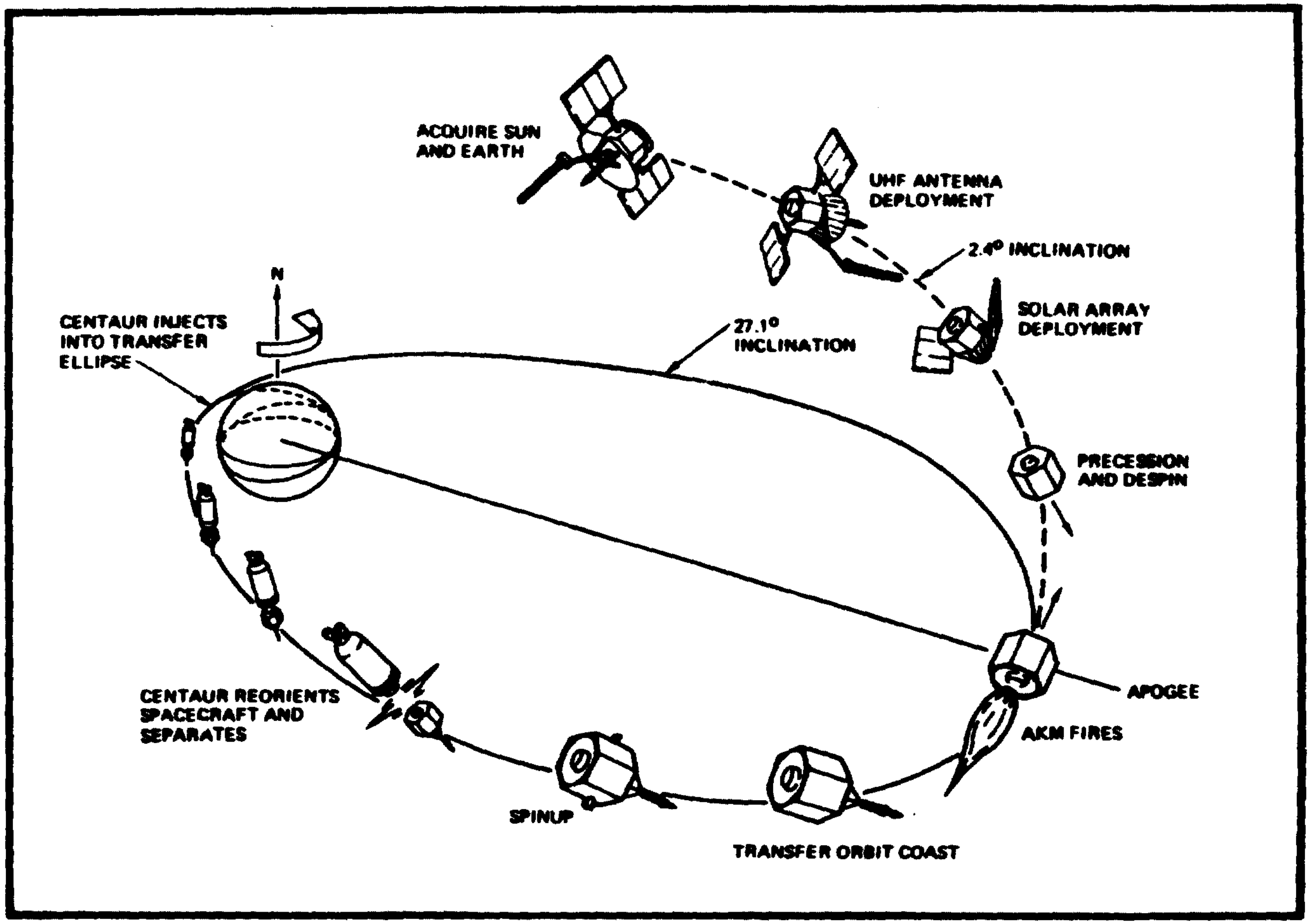

Figure 4-1. FLTSATCOM Mission Profile

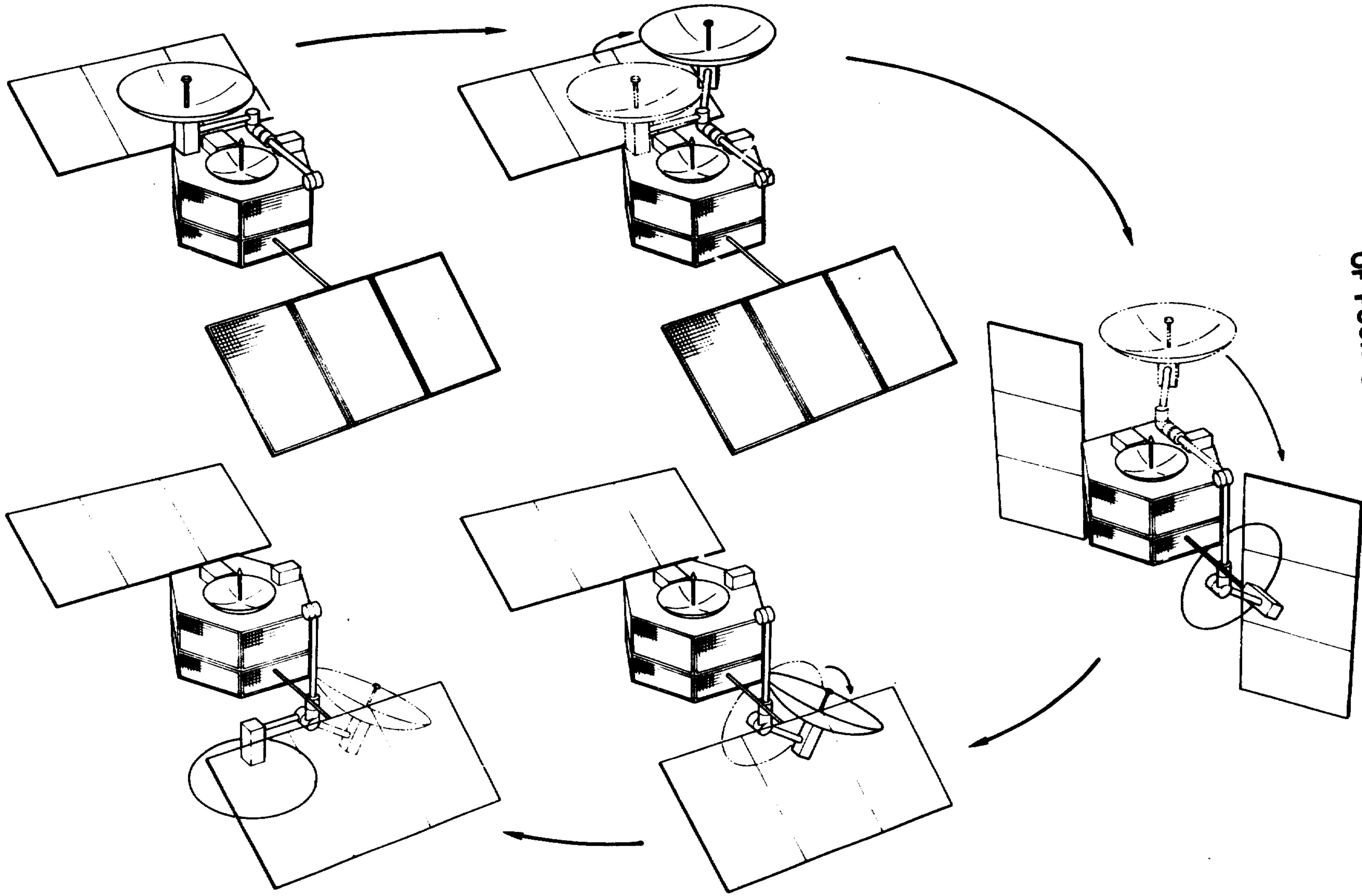

Figure 4-2. The Deployment Sequence of the Two Meter High Gain Antenna

4.2.4 Mapping Orbit

The mapping orbit is the same as the nominal phasing orbit, except for a slight change in inclination to 92.78° to make it synchronous to the mean sun direction.

This inclination change is achieved by easterly (or westerly) ΔV executed in a ∿ ± 20 degree region of the orbit encompassing the equatorial crossing. The scientific emphasis is changed somewhat from gravity and polar region observations to a more systematic mapping of the entire planetary surface. But the spacecraft subsystems continue to operate the same way they did during the phasing orbit.

4.2.5 Quarantine Orbit

At the end of the mission, ΔV maneuvers are conducted to raise the orbit altitude enough to postpone drag decay until after the year 2019.

4.3 FLIGHT PATH AND PROPULSIVE CHARACTERISTICS

The propulsive performance of the MGCO spacecraft depends on certain assumptions about the propulsive components carried and the mass of the spacecraft. The assumptions used are:

SRM-1 Injection Stage	
Propellant mass (full)	9710 kg
Propellant mass (offloaded for MGCO)	9059 kg
Spent stage mass + adapter	1097 kg
Specific impulse	291 sec
Star 37 FM OIM	
Propellant mass	1052.6 kg
Spout stage mass	73.4 kg
Specific impulse	292 sec

Liquid Hydrazine System

Propellant mass (baseline: FLTSATCOM blowdown system)	121.6 kg
Propellant mass (option: modified by added tank for external ullage and pressure regulation)	172.4 kg
Specific impulse	220 sec
Propellant reserved for attitude control and spin control:	
• before MOI	7.7 kg
• after MOI	3.2 kg

Spacecraft Mass

	baseline	(optional propulsion)	
Subsystems (1982 report)	522.7 kg		
10% contingency	52.3 kg		
Δ mass for solar array increase	+10.4 kg		
Δ mass for optional propulsion system	---	(+ 8.4)	kg
Subtotal	585.4	(593.6)	kg
Spent Star 37FM stage	73.4	(73.4)	kg
Science payload	80.0	(80.0)	kg
End of mission, spacecraft depleted of all usable N_2H_4	738.8	(747.0)	kg

The SRM-1 stage is offloaded 10% to tailor it to the MGCO mission needs (C_3 = 16.29 km^2/sec^2). There is no advantage to having a greater performance at injection.

On the other hand, the star 37 FM motor is filled with propellant. This gives an excess ΔV capability for Mars orbit insertion, which can be used to advantage in causing part of the orbit plane orientation change required during the phasing orbit.

The insertion of a 10% contingency on spacecraft subsystem mass and the increase in the science payload mass from 42 to 80 kg make the spacecraft heavier and reduce the propulsive performance from that presented in the 1982 report, Reference 7.

In the baseline spacecraft, the N_2H_4 propulsion system is exactly that of the FLTSATCOM. It employs blowdown pressurization, so that its two 22.5 inch tanks are initially partly occupied by 121.6 kg (268 lbs.) of hydrazine and the remainder by N_2 pressurant.

In the optional hydrazine propulsion system, the same tanks are filled with 172.4 kg (380 lbs.) of N_2H_4. The N_2 pressurant is stored in an added external 10.4 inch tank initially at 3000 psi, but it is regulated to $\leq$ 350 psi where it enters the propellant tanks. The additional 50 kg of N_2H_4 increases the ΔV capability significantly.

The ΔV which can be provided by the hydrazine propulsion system depends on when it is executed. Because of the large mass of the OIM solid propellant, it takes ~2.4 times as much N_2H_4 propellant to produce a certain ΔV before MOI as it does after. Therefore the ΔV performance calculations must distinguish between maneuvers before and after MOI.

We identify two cases. In one, the pre-MOI ΔV is 60 m/sec, associated with an accurate injection from earth with the spacecraft orientation trimmed after separation from the Shuttle. In the other, it is 100 m/s, assuming no such pointing correction.

For the baseline N_2H_4 system, we rely on attitude trimming before firing the SRM-1 stage and assume 60 m/s for the pre-MOI ΔV. For the optional N_2H_4 system, 100 m/s can be tolerated before MOI, so we look at both cases.

Table 4-1 shows the post-MOI ΔV capability estimated for each case, a suggested allocation of it, and the resulting mission parameters. The appropriate requirements of Reference 2 are given for comparison. Some comments on performance follow.

Table 4-1. Propulsive Capability

	N_2H_4 Propulsion System			Ref. 2	Units
	Baseline	Optional			
Usable N_2H_4 propellant	121.6	172.4	172.4		kg
Pre-MOI ΔV	60	60	100		m/s
MOI:					
ΔV	2404.2	2295.0	2361.9	2239	m/s
Excess ΔV (last launch day)	165.2	56.0	122.9	--	m/s
Post-MOI ΔV capability	162.9	286.1	195.6	≤394	m/s
Suggested allocation:					
MOI error correction	15	22	15	TBD	m/s
Δ inclination after drift	40	100	57	≤164(?)	m/s
Orbit maintenance	43	88	59	150 max	m/s
Raise to quarantine	65	76	65	80 max	m/s
Consequences:					
Total plane change required	28°	28°	28°	31°	
Portion attained by excess MOI	11.5°	6.6°	9.9°	--	
Portion attained by drift	16.5°	21.4°	18.1°	31°	
Mapping orbit inclination	92.78°	92.78°	92.78°	92.78°	
Phasing orbit inclination	92.10°	91.08°	91.82°	90.00°	
Phasing orbit duration	129	67	100	59	days
Quarantine orbit altitude	500	525	500	525	km

- MOI Error Correction. The allocations of 15 or 22 m/s to correct for the execution error of Mars orbit insertion amount to 0.7 to 1.0% of the ΔV magnitude. This is consistent with typical solid motor impulse accuracies of 0.5 to 0.7%, 3σ. However, an analysis of all errors of the orbit insertion process, including orbit determination and attitude errors as well, has not been performed.

- Δ Inclination After Drift. The consequence of this allocation is to change the duration of the phasing orbit. The suggested 164 m/s (Reference 2) will permit a rotation of the orbit plane of 31° (relative to the mean solar meridian) in 59 days. The lesser ΔV allocations for the MGCO spacecraft cases indicated produce slower rotation rates. However, since part of the rotation is is attained by using the excess ΔV available at the MOI maneuver, the phasing orbit duration is not increased very much -- to 67 to 129 days. See Figure 4-2. (The phasing orbit inclination is adjusted accordingly.)

- Orbit Maintenance. Reference 2 is ambiguous as to the ΔV necessary to compensate for drag in orbit: 100 m/s is given as an estimate and 150 m/s as a maximum. Our calculations of the drag force suggests 42 m/s will be adequate. Our allocation is 43 to 88 m/s. In any event, if greater than anticipated drag forces are encountered in orbit, the altitude could be raised a small amount -- $\sim$25 km -- to reduce propellant usage.

- Raise to Quarantine Orbit. ΔV has been allocated to raise a 350 km orbit to 500 or 525 km. For a 500 km initial altitude, orbit life is estimated to exceed the required 26-year life (1993 to 2019) even if the atmospheric density is at its solar cycle peak all the time. The actual solar cycle variation represents a large margin.

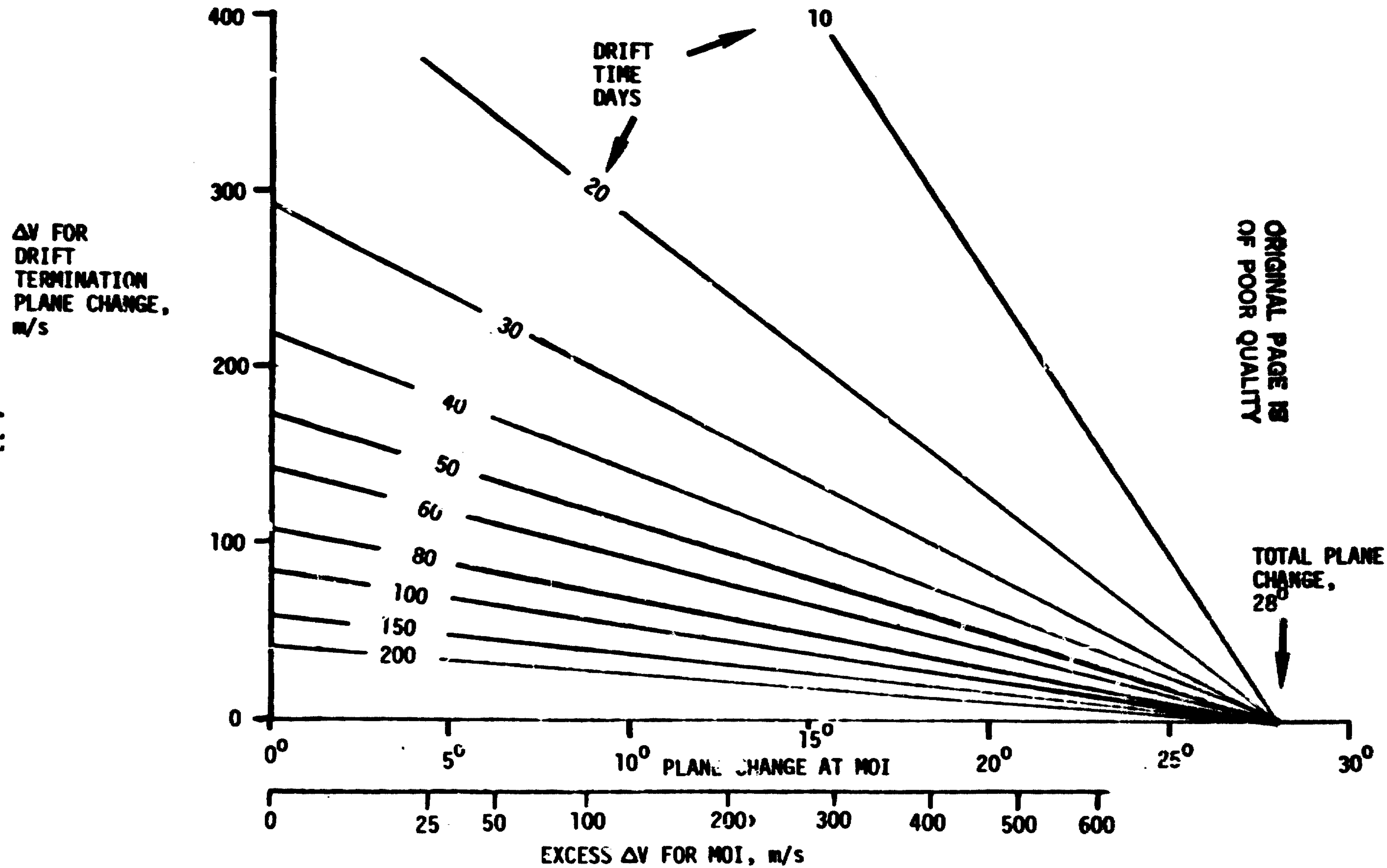

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Figure 4-3. Drift Time and Plane Change ΔV

While the ΔV capability for operations in Mars orbit falls somewhat short of the values indicated in Reference 2, it is evident that the capabilities can easily support the MGCO mission as described.

4.4 ELECTRICAL POWER

The 21.7 m^2 (233.5 ft^2) solar array area of the FLTSATCOM is more than adequate for the power requirements of the MGCO mission. There are good reasons to reduce the solar array size: to reduce spacecraft mass; to make space available to deploy and retract experiment booms during cruise; to limit infringements on instruments' fields of view.

For the 1982 study, the FLTSATCOM solar array area was cut in half to 10.8 m^2 (116.4 ft^2). However for the current work, two factors require restoring some power capacity: the cruise portion of the mission, which derives power with the array stowed and spinning, now reaches ~1.69 AU from the sun vs. only 1.53 AU for the 1988 launch; and the science power requirements have increased from 42 to 120W.

Other factors which influence the design are the deletion of the S-band power amplifier (as the transponder now handles only X-band up and X-band down) and the change in the angle the sun line makes with the orbit plane, now 50 to 61 degrees at arrival and 20 to 43 degrees during the mapping phase.

The result is that the present array area has increased to 14.9 m^2 (160 ft^2), and the two major panels when deployed are canted 30 to 40 degrees to the solar array drive (pitch) axis.

Two conditions drive the array to this size: (1) cruise, tracking, science off, 1.69 AU from the sun, sun line perpendicular to spin axis; and (2) orbit operations, science and tape recorder, 1.67 AU from the sun, angle of incidence = 21 degrees, battery charging for eclipse operation.

The calculated power available from the solar arrays, required power (including battery charging) and the excess capability are shown in Table 4-2.

Table 4-2. Electrical Power (watts)

	Condition	
	(1) Cruise	(2) Mars Orbit
Power available	203	654
Power required	195	597
Excess capability	8	57

Of course the design condition for cruise is at the maximum sun-Mars distance. And the design condition for Mars orbit combines maximum sun-Mars distance and greatest angle of incidence. Both assume end-of-mission solar cell efficiency.

Also, the array area could be further increased with only a small mass increase, if necessary. The increase from 10.8 to 14.9 square meters adds 10.4 kg to the spacecraft mass. The batteries and power control unit have sufficient capacity that they did not need to be augmented for this purpose.

4.5 ATTITUDE CONTROL

The critical attitude control pointing accuracies occur in Mars orbit. Before then, attitude accuracies better than 1 degree attainable with celestial sensors are easily adequate for solar power acquisition, medium gain antenna pointing, and executing injection, trajectory correction, and orbit insertion maneuvers.

In Mars orbit, pointing of the instruments and the HGA are more demanding. But the biased pitch momentum system, employing horizon sensors, improves the spacecraft capability. Table 4-3 summarizes requirements and capability in this mission phase. The requirements of the instruments are a condensation of Table 2-4, reflecting the requirements of Reference 6. The HGA requirement derives from a 3 dB full beamwidth of 1.05 degrees for

Table 4-3. Pointing Accuracy

3σ Pointing Accuracy (mrad) Control/Knowledge			
	X (roll)	Y (pitch)	Z (yaw)
Requirements			
Payload			
PMIRR	3/2	25/2	3/2
Envelope of other instruments	3/3	9/3	9/5
HGA	5/5	5/5	5/5
Estimated performance	5/3 (.3°/.2°)	3/3 (.2°/.2°)	5/5 (.3°/.3°)

the 2-m antenna and the acceptance of a 0.9 dB pointing loss in the communications link.

It should be pointed out that the estimated performance of the MGCO spacecraft is based on the FLTSATCOM performance at synchronous altitude at earth. No detailed analysis has been made to account for the change to the horizon sensors and the different disturbing moments appropriate to the low orbit at Mars.

4.6 COMMUNICATION

The nominal downlink communication requirement is to transmit 8192 b/s to the earth. Under what conditions this is to be done is given in Reference 2, Paragraph 3.9.2: "The required downlink bit error rate shall be less than 10^{-5} assuming 90% weather and an overall 3 dB design margin," and "The communications system shall be sized for a downlink data rate of 8.192 KBPS to a 64-m ground station antenna (34m goal) at maximum Earth-Mars distance."

The influencing parameters are:

Frequency: X-band	8.415 GHz
Transmitted power	20 W
HGA gain (2m diam.)	41.9 dB
HGA pointing loss	0.9 dB
Maximum range (1990 launch)	2.55 AU

The estimated link performance to the ground station is given in Table 4-4. This table assumes convolutional coding, R = 1/2, K = 7. With Reed-Solomon coding superimposed, performance would be increased.

It shows that communications to the 64-m antenna beat the requirement by a factor of 2.5 with the strictest interpretation. Using the 34-m DSN antenna meets the 8.192 kb/s requirement with a weather loss of -2.6 dB and other adverse link tolerances rss'd giving -1.4 dB.

Table 4-4. Downlink Bit Rate Capability (kb/s)

2.55 AU = Maximum Range	Spacecraft HGA to: 64-m DSN Antenna	34-m DSN Antenna
Bit Rate Capability:		
Nominal link parameters, no weather loss	73	20.6
Adverse link tolerances (-1.4 db rss), -2.6 db weather loss	29.1	8.20
Link margin (-3 db) -2.6 db weather loss	20.1	5.67

If a 3 db overall margin is required as well as the weather loss, then the link to the 34-m antenna is adequate only when the range falls below 2.12 AU.

4.7 COMMAND AND CONTROL

The existing FLTSATCOM TT and C Subsystem used for command and control cannot meet the MGCO requirements for either command and control or data handling (Paragraph 4.8). Therefore, it is replaced by a combined Command, And Data Handling Subsystem (CADH). The FLTSATCOM TT and C Subsystem is designed for real time commanding through the Air Force Satellite Control Facility (SCF), using data decryptors, a SCF compatible command format and providing no on-board sequence storage. It is clear that this equipment could not provide the type of spacecraft command and control required for a Mars Orbiter mission.

For purposes of supplying a rational estimate, the initial study included representative low cost equipment which was non-programmable (i.e., hardwired). When the CADH requirements were defined, as discussed in section 2.3.7 and 2.3.8 an attempt was made to provide off-the-shelf equipment which would meet the requirements with minimal modifications. The prime intent was to define a basic subsystem configuration which could meet the basic requirements, and would be flexible enough to accommodate changing and updated requirements as the mission became better defined.

For the CADH subsystem the following command and control requirements were the prime drivers:

- On-board Sequence Storage of 1024 commands
- Interrupts
- 1 msec absolute accuracy
- Autonomy
- Tape Recorder (TR) commanding

From the detailed requirements in each area, it was felt that existing hardwired designs required too much redesign (approaching a new design) and in addition, would require redesign every time a requirement was updated. This decision was based on examination of the Solar Mesosphere Explorer (SME) Command Decoder Unit (CDU). This unit does provide command decode, command storage and sequencing and interrupt functions, but does lack a TR interface. However, in order to meet the increased MGCO requirements the functions would have to be redesigned as a result of the inflexibility of a hardwired design.

Two programmable CADH units were available: one from a classified TRW spacecraft and the CADH unit from an Air Force program. Since data was readily available for the Air Force CADH unit and it represents a simpler approach, it was selected as a baseline. However, the TRW CADH unit remains a viable approach if the requirements exceed the capabilities of the Air Force unit.

The baseline CADH unit is controlled by an Intel 8086 microprocessor which provides control of the CADH unit through the computer data bus. In addition the microprocessor provides command storage and sequencing as well as telemetry format control. Specific, fixed functions are assigned to dedicated cards within the CADH unit. Figure 4.3 shows a preliminary configuration for both command and telemetry. This figure also defines the CADH interfaces.

It was only possible to make an overall assessment of the capabilities versus requirements for a programmable CADH unit. This is due to the fact that, as in most microprocessor driven CADH units, both the command and data handling functions are combined into a single physical unit, and it was not possible to allocate specific capabilities to either the control function or the data handling function. In addition, in trying to size the CADH equipment, specific requirements are needed, such as number of discreet commands, number of interfacing subsystems, etc, which require a more detailed study to define.

It is important to note that the proposed equipment has some hard upper bounds on its capability such as physical size (number of card slots and connector capacity) while others, such as microprocessor capability are more flexible.

The baseline configuration meets all of the command requirements specified in section 2.3.7. A capabilities versus requirements list is supplied as Table 4.5. A discussion of the table is presented below.

4.7.1 Command Uplinking

The CTU is designed to accept both mode change commands and data loads from the Space Shuttle Payload Command channel at 1000 bps. The MGCO requirements are much less, with 31.3 bps specified as the present maximum.

Figure 4.4. A Preliminary Configuration For Command And Telemetry

•	POWER SUPPLY
•	MEMORY: 16K OR 32K BYTES FROM 8K BYTES RAM + PARITY
•	8086 CPU
•	ANALOG INPUT (64 SINGLE ENDED)
•	DISCRETE MUX (256 INPUT)
•	SERIAL DIGITAL I/O -(4 CH.)
•	TAPE RECORDER I/F (ODETICS -2 EA)
	REED SOLOMON ENCODER
	XPONDER I/F (MOTOROLA STD)
•	TELEMETRY FORMATTER
•••	UTC WITH CLOCK

Notes:

- • QUALIFIED UNDER TALON GOLD
- ••• DESIGN EXISTS IN WIRE WRAP
- ● SANDIA IS DEVELOPING 8085 CPU FOR RADIATION TOLERANCE TO 10^5 RADS. INTEL CLAIMS 10^4 FOR 8086.
- ● EDAC MEMORY UNDER DEVELOPMENT
- ● STANDARD DMS CHASSIS IS FULLY POPULATED. SCI CONSIDERING DEVELOPMENT OF LARGER CHASSIS.
- ● REDUNDANT I/F DRIVERS IN EXISTING DESIGN
- ● PHYSICAL PARAMETERS (WAG):
  - WEIGHT - 15 LBS
  - POWER - 35 WATTS MAX
  - VOLUME - 10x10x10 IN

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The dedicated transponder interface which validates the commands, is not the limit. The upper bound on the CADH command capability is dependent on the microprocessor loading required by the MGCO mission.

4.7.2. Command Sequencing

The command sequences are stored in the microprocessors RAM memory. The only limit to the number of commands being stored is the size of the available RAM. 8K bytes of RAM provide storage for the required 1024 commands (3 bytes time and 4 bytes command). Storage capacity may be increased by increasing the amount of RAM, up to the chassis limit.

Command sequencing again is only limited by the microprocessor capabilities. Adequate capacity exists to process a main sequence, macros (reuseable command blocks) and service interrupts. It is assumed that the autonomy requirements to maintain uplink command capability are included in the macro and interrupt capability provided.

Command sequences are updated by rewriting RAM via ground commands.

4.7.3. Time Accuracy

The required time correlation between UTC and spacecraft time is 1 msec, and must be supplied to the instruments every 10 sec. This requirement can only be met by using an oven stabilized crystal oscillator (oxco).

Table 4.6 summarizes the rationale used in deriving the proposed configuration. A xco with a stability of 1 part in 10^{10} is oven stabilized, and the equipment list is amended to include a oxco. With a 10^{10} oxco, the spacecraft clock need only be correlated once every 30 days.

4.7.4 Autonomy

At this point the driving autonomous requirement is the ability to maintain a real time uplink command capability. This includes both

TABLE 4.5
MGCO COMMAND REQUIREMENTS VS. CAPABILITY

REQUIREMENT COMMANDING	COMPLY
● SEQUENCES - MODIFIED - WRITE PROTECTED DECODING - INTERRUPTS - MACROS - CHECKSUM - DOWNLINK OF ALL RECIEVED CMMDS	YES
● BOTH REAL-TIME AND STORED COMMANDS	YES
● ABILITY TO CHANGE SEQUENCES	YES
● COMMAND RATE OF 7.8 OR 31.3 BPS	YES
● MAXIMUM SEQUENCE TIME OF 96 HRS.	YES
● 1024 COMMAND STORAGE	YES
● 1 SEC COMMAND TIMING	YES
● UP TO 5 COMMANDS PER SEC.	YES
● CRITICAL COMMAND PROTECT	YES (through operational procedures)

TABLE 4.5
MGCO COMMAND REQUIREMENTS VS. CAPABILITY
(CONTINUED)

TRW

	COMPLY
• SUPPLY INSTRUMENT WITH S/C DATA	YES, (direct to IPDS)
• 1 MSEC ABSOLUTE ACCURACY	YES, (in conjunction with DSN operations)
AUTONOMY	
• 36 MO. OPERATION FOR BASIC MISSION	YES
• 24 HR. AUTONOMOUS OPERATION	YES

TABLE 4.6 REQUIREMENT FOR A OVEN STABILIZED CRYSTAL OSCILLATOR

- REQUIREMENT
  - Absolute accuracy of 1 msec, with time unambiguous to end of Science Acquisition Phase.
  - Distributed to instruments at least every 10 sec.

- INTERPRETATION
  - Spacecraft time correlated to UTC within 1 msec.

- CORRELATION
  - S/C DMS time tags specific telemetry bit and transmits time tag to DSN
  - DSN time tags specific bit in NASCOM frame, and time of arrival of S/C bit can be calculated
  - Mission Operations calculates and uploads S/C time delta once per day
  - Ground time tag is $\pm$ 1 bit

    At 32 Kbps - 0.03 msec

    At 1500 bps - 0 67 msec
  - Spacecraft Time Maintenance

    Allowable Drift = (1 - 0.67) msec = 0.33 msec per day

    msec/days $\sim 10^{8}$

    Oven-controlled XCO has 1 part in 10^{10} stability.

    This provides accuracy of 0.01 msec per day.

the capability to receive a command from the ground via the transponder and the capability to execute the command through the CADH subsystem. This requirement is being met by watchdog timers on both the uplink commanding and the CADH unit. The timer on the uplink commanding is reset via a recieved command through the transponder. The timer on the CADH unit is reset by the microprocessor. The sequence to be followed in each case will be defined during detailed design.

4.8 DATA HANDLING

The CADH unit must provide data input from both the science instruments via the IPDS and the spacecraft subsystems. These data must then be formatted, encoded, stored on the TR and played back, modulated on the subcarrier and supplied to the transponder. Again, a non-programmable CADH design would require an extensive redesign to accommodate a packet input mode, transport frame header generation, and implementing the various counters and other accounting fields required by the NASA packet standards. This reinforced the decision to baseline a programmable CADH subsystem discussed in Paragraph 4.7.

The effort in defining the CADH telemetry function has been to meet the full packet format requirements while, at the same time, minimizing the number of separate data handling functions performed by the CADH subsystem. This latter effort is aimed at providing adequate design margins, especially for the computer throughput and memory requirements.

The driving data handling requirements are:

- Implementing the packet format requirements
- Recording and playing back the transport frames
- Generating a separate packet for engineering data

The requirement vs. capability list is Table 4.7

TABLE 4.7

MGCO TELEMETRY REQUIREMENTS VS. CAPABILITY

REQUIREMENT	COMPLY
● OPERATE WITH NASA STD X/X DEEP SPACE TRANSPONDER	YES
● INPUT FROM IPDS	YES
● 6 TLM MODES	YES
● 4 DATA RATES	YES
● PACKET STANDARD	YES
● REED - SOLOMON ENCODING	YES
● TWO OPERATIONAL TR'S SIMULTANEOUS RECORD/ PLAYBACK	YES
● CONVOLUTIONAL ENCODING	YES
● TRANSPORT FRAMES	YES
● INSTRUMENT TO S/C DATA BUS	YES

4.8.1 Packet Format

The CADH subsystem implements the full packet format requirements. This is done by using a synchronous packet input via the IPDS. All buffering and storage is assumed to be done by the IPDS or the science instruments, with the CADH subsystem providing a dwell mode for packet input. Any packet error coding will be done by the IPDS.

In conjunction with the packet input is the requirement for the spacecraft to supply telemetery measurements to the IPDS. It is assumed that all data required by the science instruments will be supplied to the IPDS directly from the source.

With synchronous packet input the transport frame header now can also be generated on a synchronous basis, eliminating any need for buffering telemetry data in the CADH equipment. As the transport frames are generated, they will be output directly to the subcarrier modulator or the tape recorder.

4.8.2 TR Recording and Playback

The packetization requirements require that the transport frame headers be transmitted FIFO (MSB first). Playback packet data is to be synchronous with the transport frames, but may be LIFO (played back in reverse) or FIFO. In order to avoid the necessity of synchronizing the packet data to the transport frame (so that it may be inserted into the transport frame synchronously) during TR playback, the transport frame will be generated prior to recording, and the TR rewound prior to playback.

A second concern is with simultaneously transmitting playback and real time transport frames. In order to interleave data on transport frame boundaries, the TR data must be frame synchronized, a task requiring both buffering and synch. pattern identification. In order to eliminate this function, it is recommended that only real time or only playback data be transmitted. This may be the MGCO intent anyway since simultaneous record and playback capability is required. As an alternative, dedicated playback and real time data channels, per the DSN standard, should be investigated.

4.8.3 Engineering Packet

In order to minimize the processing load on the computer, it is recommended that the requirement for a separate engineering data packet be eliminated, and instead a data field be inserted in the transport frame. This method of handling engineering data is being implemented on the NASA Gamma Ray Observatory (GRO) spacecraft, which uses a packet format.

4.8.4. Data Rate Margin

The existing Talon Gold system is capable of handling telemetry at 128 Kbps., a factor of 2 above the highest TR playback rate envisioned and a factor of 4 above the highest requirement. Again this would be limited by the ability of the computer to supply format commands to the telemetry interleaver.

4.8.5 Encoding

The proposed CADH subsystem provides two levels of telemetry data encoding: a convolutional code for the downlink and Reed-Solomon encoding for the transport frames. This latter code is an option, and should be investigated regarding communication link performance. The R-5 encoder design assumed is that developed for JPL by Cyclotomics, which only requires 40 IC chips.

5. SCIENCE ACCOMMODATION CAPABILITIES

This section also reviews the performance and capabilities of the MGCO spacecraft. But instead of comparing them with the Reference 2 mission/system requirements, the comparison is now with the Reference 6 science accommodation capabilities and constraints.

5.1 ORGANIZATION

The subsections address topics in roughly the order of their treatment in Reference 6. Again, the pertinent capability is described, the requirement is stated, and the excess capability is noted. In many instances, there will be reference to Section 4 because the pertinent parameters have already been discussed.

5.2 MASS

The capability of the MGCO spacecraft to carry the payload mass is subject to a soft limit and a harder (but not rigid) limit.

The soft limit has been alluded to in Section 4.3. Because of the finite propellant mass limitations, the heavier the spacecraft the lower the ΔV capability. With a given pre MOI ΔV requirement, (we have used 60 and 100 m/s as representative values) the post MOI ΔV capability is reduced by 1 m/s for each 1 kg increase in dry spacecraft mass. For a given spacecraft, it is the science payload mass which can be traded for post MOI ΔV, 1 kg for 1 m/s. This ΔV penalty must be subtracted from the results given in Section 4.3.

The harder limit is set by the capability of the Star 37 FM solid motor to provide the full 2239 m/s necessary to insert the spacecraft in a 350 km circular orbit at Mars. (Because of the limited capacity for N_2H_4, and the relatively low specific impulse of that propellant, it is undesirable to accept less than 2239 m/s from the solid with the expectation of completing the maneuver by using N_2H_4.) With this limit, and retaining the masses listed in Section 4.3 for the spacecraft subsystems and contingency, the OIM case, and N_2H_4 for 60 m/s pre MOI ΔV, the remaining mass available for the science payload is:

	Baseline Propulsion	Optional Propulsion
Science Mass		
Capability (kg)	167	110
Requirement (kg)		
Allocated	67	67
Contingency	13	13
Total	80	80
Excess Capability (kg)	87	30

But note again, as this excess capability is utilized, degradation in post MOI ΔV will be sustained as discussed in the preceding paragraph.

A more rigid mass limit is imposed by the ability of the SRM-1 injection stage to send the spacecraft to Mars. But there is enough margin in this process that the limits discussed earlier apply before the SRM-1 limit is reached.

5.3 VOLUME

The central body of the FLTSATCOM spacecraft (Figure 5-1) is hexagonal in shape, and consists of two sections. Spacecraft subsystems occupy one module of this hexagonal prism (the zenith end), and the communications payload occupies the second at the earth-pointing end.

In the conversion to the MGCO spacecraft, the communications payload module (and its associated antennas) is removed and replaced by the science payload module. The envelope for this module is a 7.5 foot hexagon in plan, and undetermined in height.

UNITS 1-6

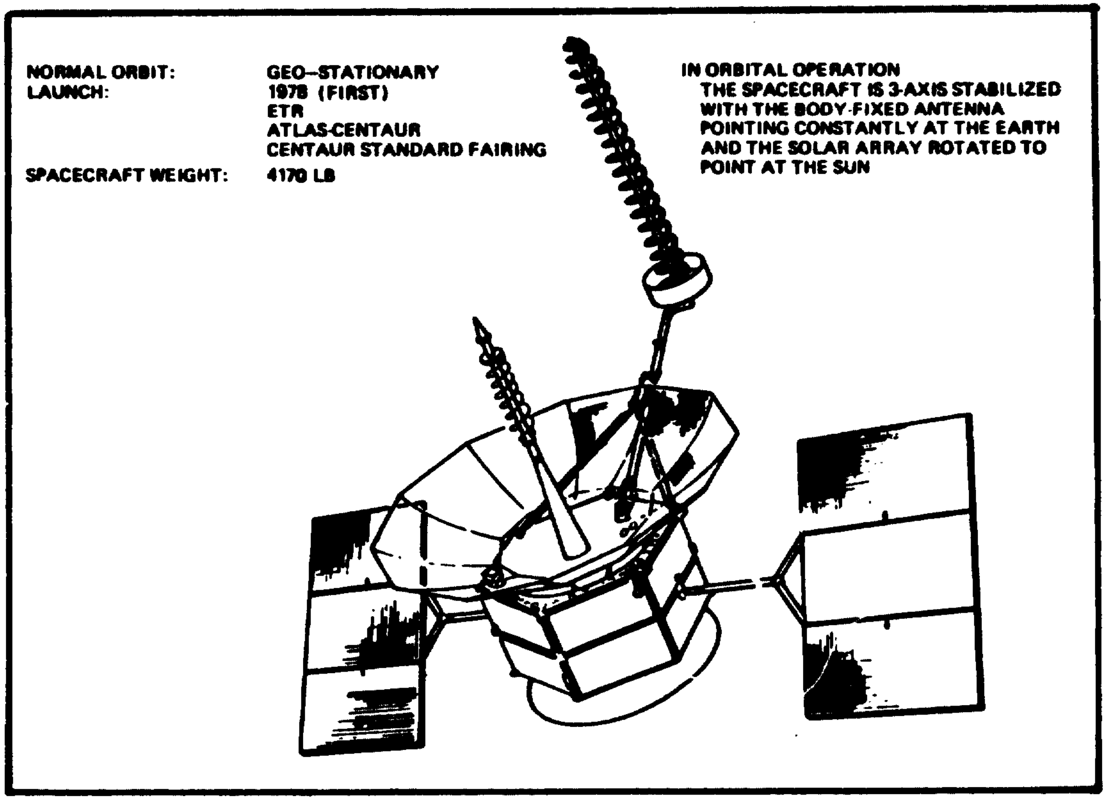

Figure 5-1. Principal FLTSATCOM Characteristics

5.3.1 Instrument Mounting Plan

Figure 5-2 shows to scale a possible layout of the instrument sensors and radiators on the hexagonal platform which conforms to this plan. It is evident the hexagon is not cluttered, and a significant margin exists in mounting area.

The MAG sensor and GRS insturment are mounted on booms which are deployed in the +X and -X directions, respectively, even when the solar array and HGA (not shown) are stowed as they are in the interplanetary cruise phase. These boom deployments are reversible, and these instruments will be restowed for the MOI maneuver. After this maneuver, they will be permanently deployed.

The height necessary for the instrument volume is no problem, and can be adjusted as necessary. Instrument electronics boxes and the IPDS are mounted on the interior of the instrument module. The 2-m HGA dish, when stowed, will overlay a large area of the hexagon shown in Figure 5-2.

5.3.2 Viewing Directions

Figure 5-3 shows some of the significant viewing directions for the instruments and their radiators during the mapping portion of the mission. The surface of Mars occupies the region within ~ 65 degrees of the nadir direction (+Z). The boundary of this region defines the limb. The sun may only be seen in directions in a band 47 to 70 degrees from the +Y direction, circling from limb to limb through the -Z hemisphere once each orbit. Location in the band between the 47 and 70 degree lines is determined by seasonal variation.

Nadir and zenith directions are noted. Scanning to the limb is in the XZ plane for the along-track limb (+X, forward or down track limb) and in the YZ plane for the crosstrack limb (+Y, starboard).

The figure is also a guide to determine thermal radiator orientations to avoid including the sun or the surface of Mars in the fields of view.

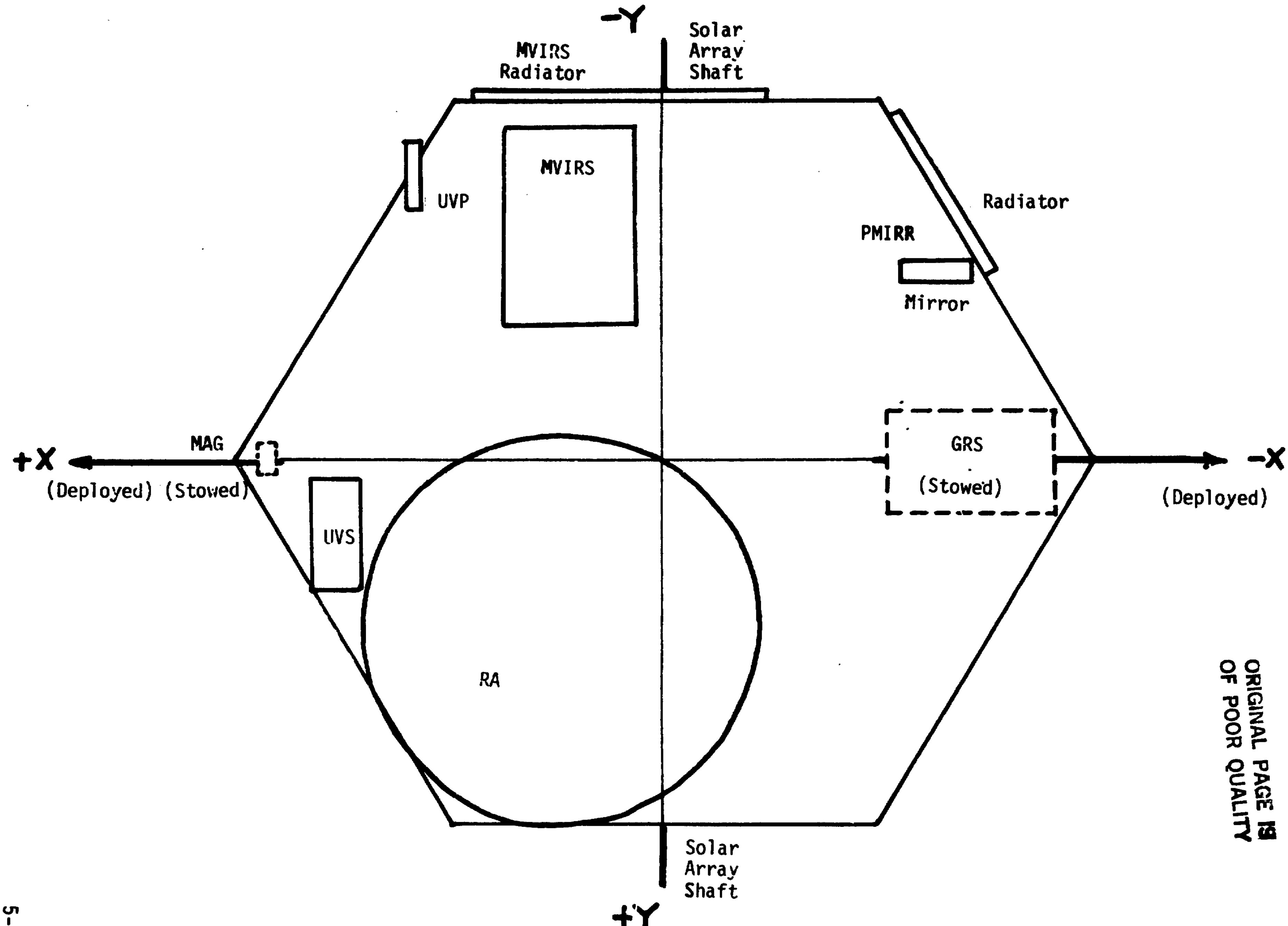

Figure 5-2. Layout of MGCO Instruments

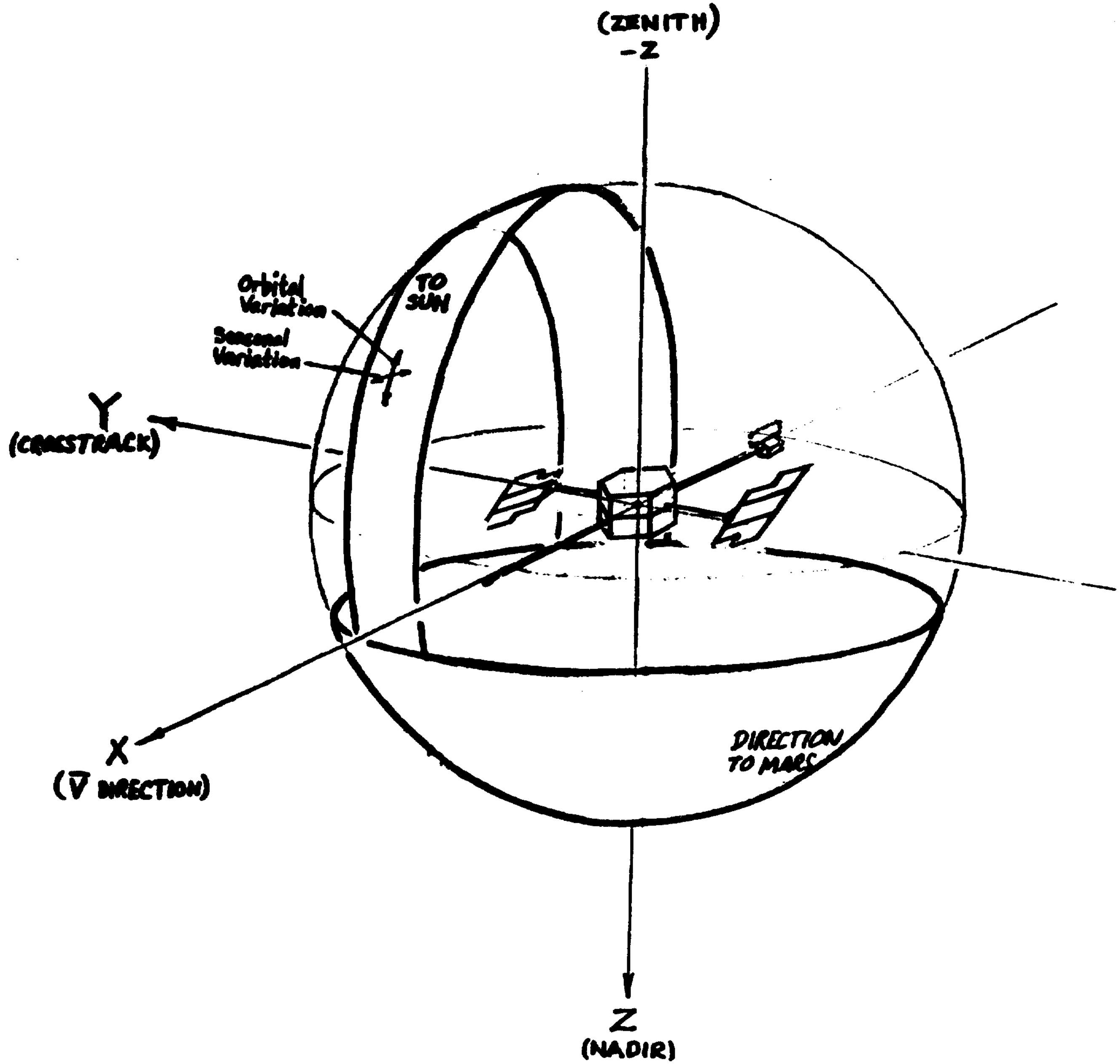

Figure 5-3. Directions from the Spacecraft in the Mapping Orbit

Figure 5-4 shows how this is applied in the case of the GRS instrument, deployed on the -X boom. The radiator is oriented to avoid seeing the surface of Mars, and a sun shield is located in the XZ plane. In Figure 5-2 the MVIRS and PMIRR radiators are shown generally on the -Y side of the instrument module. They should also be tilted to view into the (-Y, -Z) quadrant, but this tilting is more apparent in Figure 5-5.

Figure 5-5 shows the viewing and scanning directions of the five instruments on the instrument module. Each line of sight is shown with an associated "stray light field of view," which moves with the scan motion, and represents a region where intrusions are to be minimized.

It is not possible to avoid all intrusions, and Table 5-1 shows the extent to which this goal is satisfied. For detector fields of view, all requirements are met, except that the GRS FOV is not quite the 2 π steradians (hemisphere) desired. One solar array will intrude partially into this region. Also, the PMIRR may also conflict with the -Y solar array when it is aimed near or beyond the limb, but only at some portions of the orbit when the array is rotated into the (-X, -Y) quadrant.

Stray light fields of view are satisfied except for the PMIRR scanning near or beyond the limb, and the appearance of the deployed magnetometer near the UVS and UVP lines of sight when they are scanning near the limb. (But the MAG has a very low cross section.)

5.4 BOOMS

Booms which are deployable and retractable are available for the MAG and GRS instruments. We envision the use of the STEM type of boom. The length of the boom can be negotiated. There are no tight limitations on this length, but orientation accuracy degrades as the length increases.

If necessary, there will be corners notched in the solar array panels to permit deployment and retraction while the array is still stowed.

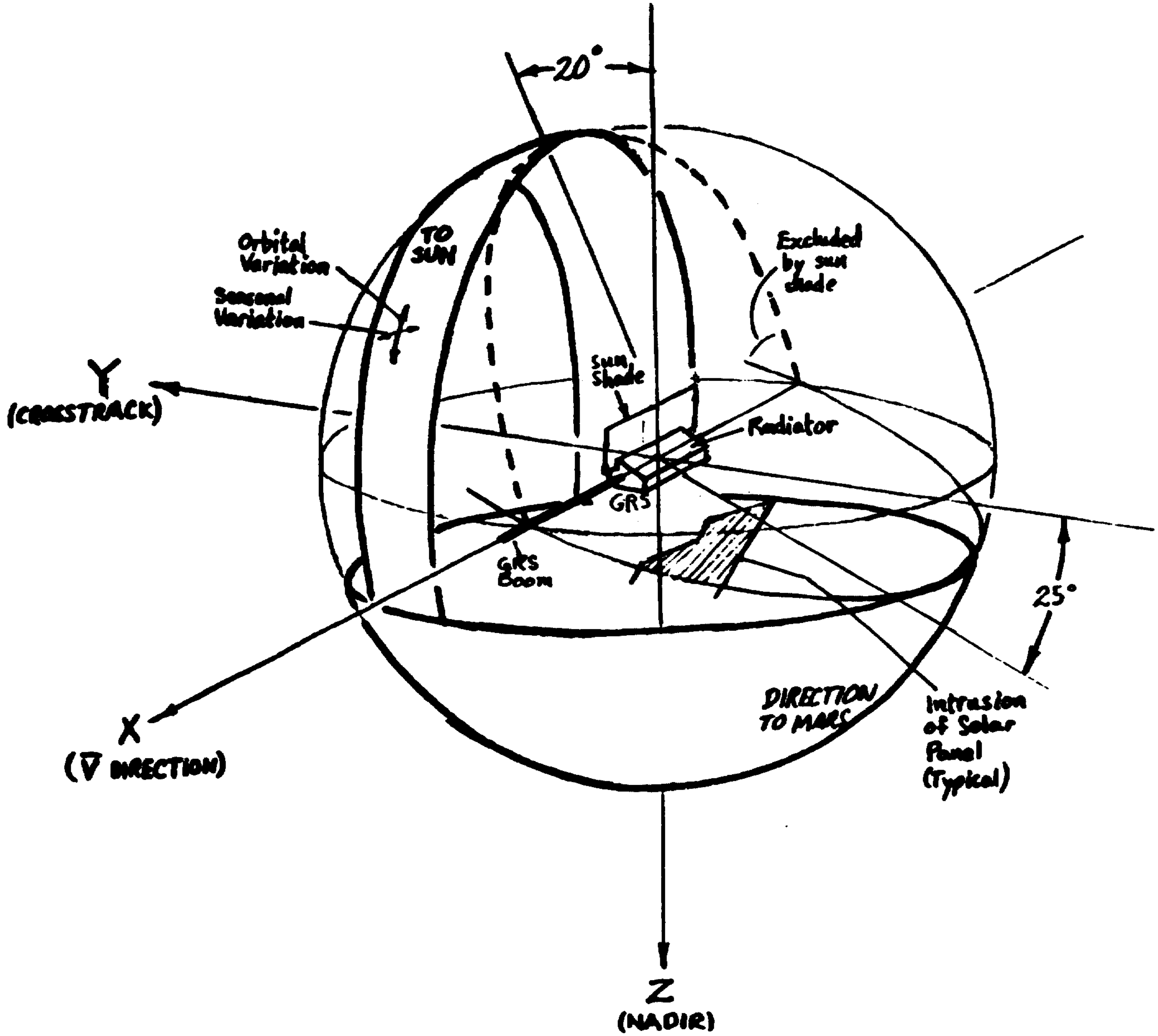

Figure 5-4. GRS Radiator FOV

Directions from the spacecraft
in the Mapping Orbit

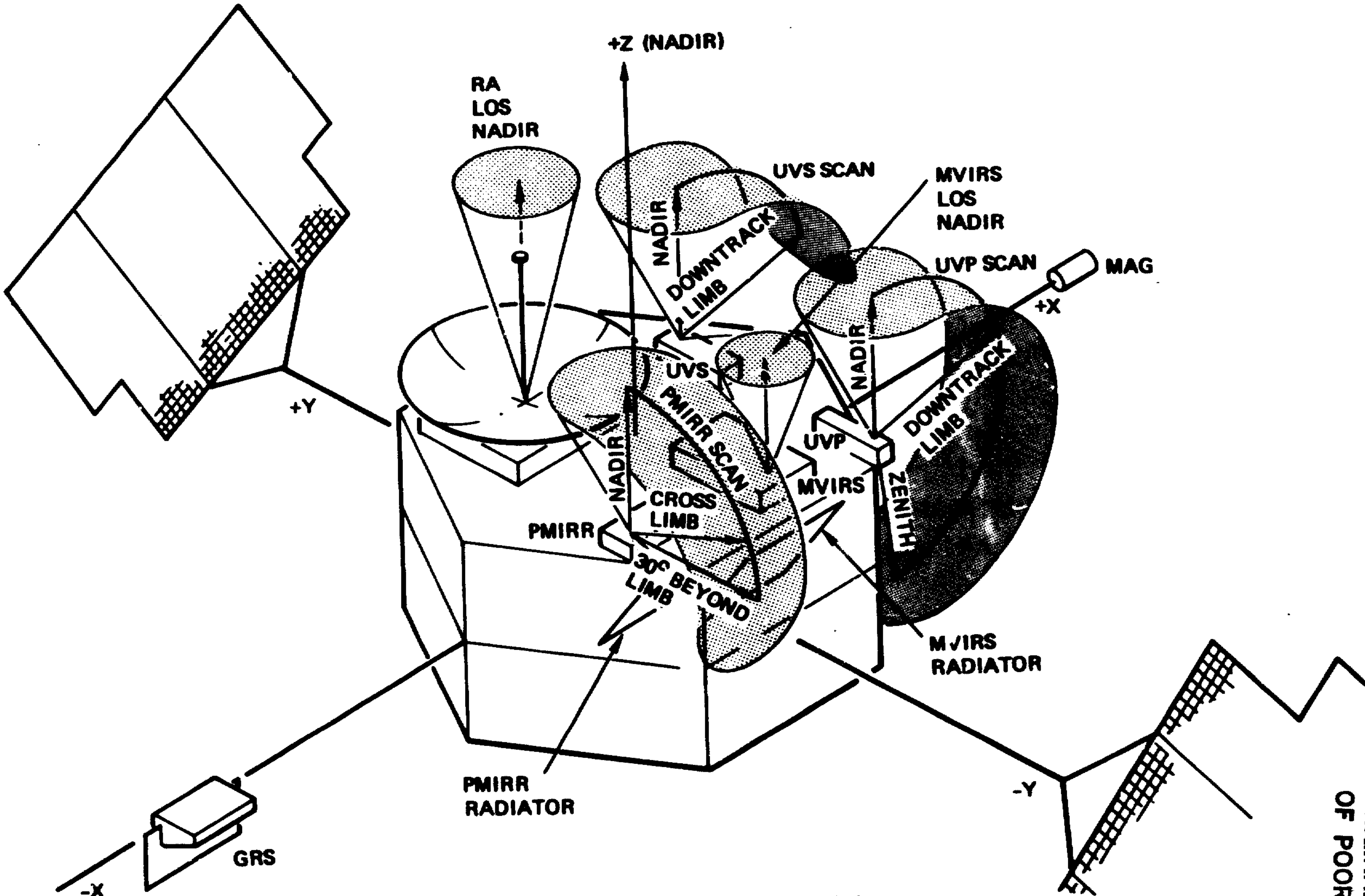

Figure 5-5. Scans and Stray Light Fields of View (Body Mounted Instruments)

TABLE 5-1. SATISFACTION OF INSTRUMENT VIEWING REQUIREMENTS

	INSTANTANEOUS FOV	SCAN	STRAY LIGHT FOV
<u>GRS</u>			
SENSOR	2 π sr SOME INTRUSION BY SOLAR ARRAY PANELS, AT A DISTANCE (VARIES THROUGH ORBIT)	(NONE)	-
RADIATOR	WITH SUN SHIELD, ONLY INTRUSION IS -Y SOLAR PANEL (VARIES THROUGH ORBIT)		
<u>MVIRS</u>			
SENSOR	NADIR - CLEAR	(NONE)	20° HALF < - CLEAR
RADIATOR	INTRUSION IS -Y SOLAR PANEL (VARIES THROUGH ORBIT)		
<u>PMIRR</u>			
SENSOR (MIRROR)	CLEAR, EXCEPT WHEN SCANNED BEYOND LIMB, -Y SOLAR PANEL MAY INTRUDE	NADIR TO 10° DOWNTRACK NADIR TO CROSS TRACK LIMB AND 30° BEYOND OK	CLEAR EXCEPT BEYOND LIMB. -Y SOLAR PANEL WILL INTRUDE
RADIATOR	MAY NEED SUN SHIELD. INTRUSION BY -Y SOLAR PANEL		
<u>RA</u>	NADIR - CLEAR	(NONE)	20° HALF < - CLEAR
<u>UVS</u>	CLEAR IN ENTIRE SCAN	NADIR TO DOWNTRACK LIMB. OK	30° HALF < - CLEAR EXCEPT FOR DISTANT MAG
<u>UVP</u>	CLEAR IN ENTIRE SCAN	NADIR TO DOWNTRACK LIMB TO ZENITH. OK	30° HALF < - CLEAR EXCEPT FOR DISTANT MAG WHEN AIMED 11 X DIRECTION, AND SC BODY WHEN AIMED TO ZENITH (COULD BE CORRECTED BY BOOM DEPLOYMENT)
<u>MAG</u>	(NONE)	(NONE)	(NONE)

5.5 DATA AND COMMAND TRANSFER TO INSTRUMENTS

The CADH subsystem presently is assumed to interface to the science instruments through the IPDS. Four types of data are provided to the IPDS to route to the instruments:

- Serial digital data. This may be commands or computer loads, and is supplied to the IPDS via a serial port with the instrument address.

- Time data. This is supplied as serial digital data through the serial port, with an address indicating it is to be broadcast to all instruments. A separate synchronization line is provided to establish when the time data is valid.

- Discrete commands. This is also supplied as serial digital data also through the serial port to the IPDS. It is assumed that the IPDS decodes this data, and issues the specific command.

- Spacecraft Data for Inclusion in Science Packets. These data are supplied directly from the source to the IPDS. It is assumed that the IPDS inputs the required reading, converts and formats the data as required and forwards the reading to the instruments.

If the IPDS is eliminated, it is assumed that each instrument would interface directly to the CADH subsystem, with a single serial data port and some allotment of discrete commands. This would increase the number of CADH subsystem I/O ports and discrete commands. At this point in the design maturity it is impossible to ascertain the impact of this change to the CADH unit, though it should be minor. A new function, that of providing digitized telemetry data to the instruments would be added. This function could have a major impact. An alternative procedure of having the CADH subsystem insert the required data into the science packet is recommended.

5.6 DATA FROM INSTRUMENTS

The science data packets are input from the IPDS as required by the CADH subsystem at the required rates. This is defined as synchronous packet input. All buffering is assumed done in either the IPDS or science instruments. The CADH subsystem will provide a "dwell" data input mode, wherein a given number of bytes are input from a single port. These science packet data are then placed into a transport frame and either recorded or transmitted in real time.

Instrument data which monitors instrument health may be supplied directly to the CADH subsystem for inclusion in the engineering data. Otherwise it is assumed that such data comes from the IPDS.

5.7 COMMUNICATIONS LINK PERFORMANCE

This subject is treated in Section 4.6.

5.8 ELECTRICAL POWER

Section 4.4 shows that the electrical power subsystem capability is designed more by the cruise phase, when instruments are off (except perhaps occasional exercising of them) and the array is still stowed, than by the orbital phases when instruments are operating.

It also shows an excess power capability of 57 watts in the orbital phase, even when the instruments are taking 120 W. Considering losses incurred in power conditioning and distribution and battery charging requirements, the 57 W margin translates into about 30 W which could be added to the full time instrument power load.

It is also noted in Section 4.4 that the solar array area could be increased if necessary to meet added power requirements.

5.9 THERMAL

At this time it is not possible to determine the temperatures which will prevail in the instruments nor what margins will exist. To the extent that spacecraft thermal control measures establish these temperature, the -20 to +40°C requirement is not difficult to satisfy. And to the extent that the instruments contribute to the satisfaction of their own temperature requirements by radiating to space, the spacecraft design provides clear fields of view for this purpose, except for the possible small intrusion into the 2π steradians requested by the GRS.

In addition to the maintenance of temperatures in the required range at an instrument there is the question of how much replacement heater power -- beyond the electrical power to operate the instrument -- is necessary to do this, both with the instrument on and with it off. The off temperature range should distinguish instrument dormancy from the stand-by state implying readiness to be turned on. Because the designing power condition is during cruise with the instruments off, replacement heater power is important. However, this determination can not be made without additional information, including aperture characteristics which govern heat leaks.

5.10 SPACECRAFT FIELDS AND RADIATION

No analysis has been made of fields and radiation emanating from the spacecraft. But since the only instruments which are identified as having sensitivities of this kind, the MAG and the GRS, are deployed on booms which are not limited to short lengths, it is likely that the magnetic and particle requirements can be satisfied with very little change to the existing spacecraft.

6. REFERENCES

(1) Attachment 1. Standard Mission/System Performance Requirements. Received February, 1983.

(2) Attachment 1. Standard Mission/System Performance Requirements. Received March, 1983.

(3) Table 1, MGO Instruments, from JPL Contract 956386 dated May 19, 1982.

(4) Attachment 1B to M. H. Jacobs' letter 626-MHJ; is dated January 21, 1983. MGCO Science Requirements for Provisional Payload.

(5) Attachment 2. MGCO Science Accommodation Capabilities/Constraints. Dated February, 1983.

(6) Attachment 2. MGCO Science Accommodation Capabilities/Constraints. Dated March, 1983.

(7) Study of Mars Geoscience Orbiter and Lunar Geoscience Orbiter. TRW's Final Report, Revision 1, December 1982. Contract 956286.