TRW Space and
Technology Group
One Space Park
Redondo Beach, CA 90278

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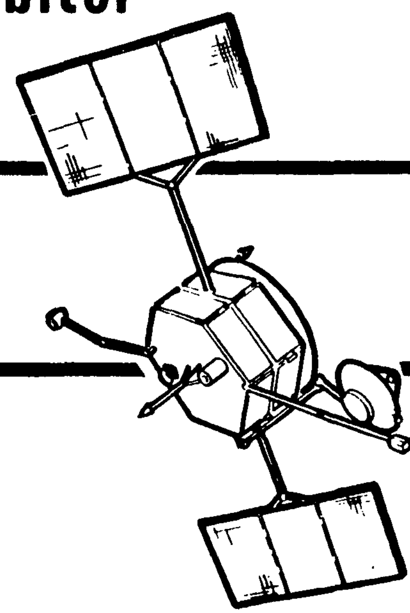
Revision I

Study of Mars Geoscience Orbiter and Lunar Geoscience Orbiter Final Report



Volume I. Technical

Prepared for

Jet Propulsion Laboratory

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Contract 956286

December 1982

VOLUME I - TECHNICAL

FINAL REPORT

MGO/LGO STUDY PROGRAM

BY

R. F. BRODSKY, TRW

DECEMBER 1982

CONTRACT 956286

WITH

JET PROPULSION LABORATORY

CONTENTS

SUMMARY OF CONTENTS

This report describes a study accomplished in the Systems Engineering Directorate of the TRW Space Systems Division which was funded by JPL Advanced Planetary Studies Program Office. The formal portion of the study took place from mid-May through mid-October 1982 and was marked by two major milestones:

1) Preliminary Design Review-type oral report on 22 July 1982

2) Final oral briefing on 23 August 1982

The purpose of this study was to evaluate the feasibility and efficiency of using an existing Earth orbiter to perform planetary missions. This report describes the highlights of the study findings and satisfies all contractual requirements with respect to deliverable items. The technique used in this report to present the results of the study is to verbally describe or discuss each viewgraph most of which were previously presented in the preliminary and final briefings. Dividing tabs are included for easy access in accordance with the Table of Contents.

The viewgraphs were prepared by the writer and with the assistance of the members of the study team listed on page I-4. The writing, for the most part, was done by the Study Manager, Dr. R. F. Brodsky, except for the Communications and Data Handling section which was written by Dr. V. R. Sapojnikoff; the Scientific Instrument section by Dr. Alan Rosen; the Thermal Control section by Henry Pan; the Power Section by Mel Swerdling. The authors of the appendices are noted.

INTRODUCTION AND SUMMARY

THE MGO/LGO PROGRAM

PROGRAM DIRECTION - JET PROPULSION LABORATORY, J. R. FRENCH, STUDY DIRECTOR
- TRW, R. F. BRODSKY, STUDY MANAGER

PROGRAM SCHEDULE - START MAY 19, 1982 - COMPLETE OCTOBER 31, 1982
- FINAL ORAL PRESENTATION, AUGUST 23, 1982

PROGRAM OBJECTIVES (1) ADAPT FLTSATCOM SPACECRAFT TO PERFORM MGO AND/OR LGO MISSIONS
(2) ESTIMATED COST OF SPACECRAFT FOR LAUNCHES IN 1988, 1990, 1992

PROGRAM FUNDING - $65,000

THE MGO/LGO PROGRAM CONTRACT

The submission of this final report completes the contractual requirements of Contract 956280. All objectives of the work statement were met. The final report encompasses material informally presented at the mid-term briefing and the final briefing of 23 August 1982.

The last pages of this report indicate additional studies TRW proposes for the purpose of providing clearer visibility into key problems, approaches, and risk reductions appropriate for continuing work. Estimated costs and schedules for these studies have been provided to Jim French.

The orbital requirement for the one year on orbit Martian and Lunar Geoscience Orbiters (MGO/LGO) may be summarized as follows:

MGO - 300 Km   Sun Synchronous Polar Circular Orbit   (~Noon/Midnight)

LGO - 100 Km   Polar Orbit

MGO/LGO
TRW PERSONNEL INVOLVED

SCIENCE	- Rosen
C&DH	- Roberts/Sapojnikoff
MISSION ANALYSIS	- Meissinger
DESIGN	- Akle
GENERAL	- Kaminskas
POWER	- Swerdling
COSTING	- Dryden/Falletta
CONTRACTS	- Grujich/Hamilton

ADVISORS:

FSC	- P. Melancon/T. Castle
G&C	- J. Stavlo/R. Rose
SENSORS	- R. Cash/R. Schissler
AI&T	- F. Krausz
COST	- G. Shier (Electronics)
SCIENCE	- F. Scarf

THE MGO/LGO PROGRAM

TRW thanks JPL for the opportunity and inputs to perform this point design study since it enabled us to prove our earlier contention that the FLTSATCOM was an ideal earth orbiting spacecraft amenable to fairly painless and reasonably economical conversion to a planetary orbiter, particularly for the Martian mission. Specifically, we wish to acknowledge the help and guidance of the JPL contract manager, J. R. (Jim) French, of the Advanced Planetary Studies Program Office and other JPL personnel who assisted us, in particular Doug Nash, who reviewed and specified the scientific instrument requirements. Internally, the study manager wishes to thank and commend the members of the study program office and the many other TRW personnel who assisted in this effort, and who are listed above.

The two references listed below also present information pertinent to the study. Both are available at JPL:

(1) The proposal to JPL, "Study of Mars Geoscience Orbiter and Lunar Geoscience Orbiter," Volumes I and II, TRW 40425.000, March 9, 1982.

(2) P.O. BX-783 for Lincoln Labs, "Study of Integrating MILSTAR Transition Package with FLTSATCOM Spacecraft," Phase 1 Final Report, April 12, 1982.

- THERE ARE NO FUNDAMENTAL REASONS WHY AN EARTH ORBITER WHICH CONTAINS INTEGRATED PROPULSION FEATURES AND LIMITED AUTONOMY DURING CRUISE-OUT TO FINAL ORBIT CAN NOT BE CONVERTED TO A PLANETARY ORBITER FOR "NEARBY" PLANETS

- THE 7-10 YEAR LIFETIMES OF MANY PRESENT DAY EARTH ORBITERS CAN BE EXTENDED TO LONGER LIFETIMES

- THE MAIN WEAKNESS OF DIRECT CONVERSIONS FALLS INTO THE FAULT TOLERANCE/AUTONOMOUS ACTION AREA. MOST DESIGNS CAN TAKE FAIRLY SIMPLE STEPS TO MAKE LARGE GAINS: HOWEVER, GREAT SOPHISTICATION CAN NOT BE ACHIEVED IF PRICE IS TO BE KEPT DOWN

SUMMARY OF STUDY GROUND RULES

The cardinal rule for the study was to find ways to adapt the FLTSATCOM to the MGO/LGO missions with least impact on subsystems. A problem which clouded this approach is concerned with the launch vehicle. The FLTSATCOM is Atlas-Centaur launched, and, indeed, both missions (the LGO with ease, the MGO marginally) could be accomplished with this launch vehicle, an advanced version of which will be used to launch FLTSATCOM's 6, 7, and 8 later in this decade. If the STS is used to achieve LEO, significant additional expenses are incurred both to make the FLTSATCOM compatible with the STS launch, and to adapt and launch an upper stage booster from the STS, for either/and/or MGO/LGO.

The major adaptations for both spacecraft consist of a complete change in the TT&C system, including the addition of a steerable, earth-tracking, high-gain antenna. Both spacecraft require a Data-Handling system, as well as significant changes in the solar panel size (a 50% reduction) and the power control unit. Other changes are much less significant, ranging from "moderate" to "minor".

POSSIBLE SHORTFALLS OF ADAPTED EARTH ORBITER

- LACK OF POINTABLE HIGH GAIN ANTENNA
- HARDNESS AGAINST NEW PLANETARY OR CRUISE-OUT ENVIRONMENTS
- AUTONOMOUS FEATURES, SUCH AS
  - SAFE HAVEN MODE, KEYED BY ANOMALIES
  - AUTOMATIC RECAPTURE IF LOCK (TM OR NADIR, e.g.) LOST
  - RECOVERY FROM FLAT SPIN, IF APPLICABLE
- ACTIVE NUTATION CONTROL, IF APPLICABLE
- PARTITION OF PROPULSIVE ENERGY BETWEEN LIQUID AND SOLID, IF APPLICABLE
- GENERALLY MAGNETICALLY "DIRTY", IF APPLICABLE

ADAPTATION SHORTFALL IMPACTS

Most earth orbiting spacecraft do not presently contain autonomous "safe haven" features, since anomalies are generally detected and acted upon in real time without great delays. The long communication time delays with distant planets, coupled with the periodic eclipses, make this problem a major one. There are detailed discussions later in this report on remedies. It turns out that the earth environment is probably more severe than most known planetary environments, and the "hardened" FLTSATCOM apparently has no problems in this respect.

Two major elements of a scientific mission program are spacecraft and launch vehicle cost. In today's and the near future environment, these cost elements are of the same order magnitude. Consequently, it is desirable to minimize both. For the overall MGO and LGO mission, the major study avenues explored were:

- Minimize spacecraft cost by adapting an available bus (the FLTSATCOM) that can do the job with comfortable margin.
- Determine if additional program cost savings can be obtained by designing maximum commonality into the MGO and LGO spacecraft or on the other hand, determine if cost savings are possible through selective noncommonality since the LGO mission is not as demanding as the MGO mission: neither in ΔV needed for both launch and lunar orbit insertion; nor in power, because of solar proximity.
- Minimize launch vehicle cost by using an economical launch vehicle and transfer vehicle combination.
- Determine how to accomplish additional cost savings by using less than a fully dedicated STS launch (or launching both missions from the same STS).

THE ADVANTAGES OF THE FLTSATCOM BUS

- THE FSC AFFORDS
  - AMPLE PERFORMANCE MARGINS FOR BOTH INTERPLANETARY LAUNCH AND ORBIT INSERTION, THUS PRECLUDING WEIGHT PROBLEMS, AND PERMITTING GREAT FLEXIBILITY OF OPERATION
  - READY INTEGRATION OF MGO INSTRUMENTS AND BOTH CATEGORY I AND II LGO INSTRUMENTS, ALL WITH CLEAR FIELDS OF VIEW AND NECESSARY RADIATIVE SURFACES
  - THE ABILITY TO INCORPORATE THE TWO NASA ARC PRIME MGO CLIMATOLOGY MISSION INSTRUMENTS NOT CARRIED BY MGO - THUS PERFORMING ALL (i.e. GEOSCIENCE AND CLIMATOLOGY) THE SCIENTIFIC MEASUREMENTS DESIRED BY BOTH GOVERNMENT LABORATORIES
  - BY REASON IF ITS OVER-PERFORMANCE ABILITY, THE MISSION PLAN CAN BE ALTERED TO COMPENSATE FOR GLITCHES

FLTSATCOM ADVANTAGES

Our Study effort resulted in the following conclusions:

- The in-production Atlas-Centaur-launched, geosynchronous-based, three-axis controlled, communication satellite - the FLTSATCOM - can be adapted to almost ideally fulfill the MGO/LGO mission requirements.

- A new fixed price follow-on contract for Units 6, 7, and 8 is under negotiation, therefore, identification of subsystem cost elements will be accurate, permitting a highly credible cost accounting the next study phase.

- The LGO spacecraft (with both Category I and II instruments) can be launched by the same Atlas-Centaur version that will launch FLTSATCOM's 7 and 8, thus eliminating many integration costs. LGO launch is also possible using the STS/PAM-A, and STS/SRM-1.

- The MGO version can be launched by the Titan or STS with the IUS or, if available from the INTELSAT VI Program, the more compatible spinning SRM-1 booster. Launch by Atlas-Centaur, with an added solid stage is also marginally feasible.

- The MGO mission, with its present instrument complement taking measurements in the most scientifically desirable manner cannot be accomplished with an STS/PAM-A launch even with a specifically designed new spacecraft.

These conclusions have shaped our Technical Approach to the following:

- Use the FLTSATCOM spacecraft for both MGO and LGO by removing the present communications payload and replacing it with the MGO/LGO instrument complement and modifying the spacecraft subsystems as required.

- Maintain maximum commonality, consonant with cost effectiveness, between MGO and LGO and the production FLTSATCOMs 7 and 8, launched by the most economical launch system available.

THE ATTRIBUTES THAT WOULD FAVOR CONVERSION

- ✓ STS NORMAL TRANSPORTATION TO LEO; OR HIGH ENERGY ELV/OTV
- ✓ HAVING ITS OWN INTEGRAL (BUT JETTISONABLE) TRANSFER PROPULSION SYSTEM OR EASILY ADAPTABLE TO A STANDARD OTV
- ✓ DESIGNED TO CARRY ITS OWN APOGEE KICK MOTOR OF ENERGY LEVEL CAPABLE OF PLANETARY INSERTION, WITH MANY DESIGNING LOADS RESULTING FROM FIRING OF THIS PROPULSION SYSTEM
- ✓ ABILITY TO PROVIDE ITS OWN POWER DURING CRUISE-OUT
- ✓ ABILITY TO USE ITS TRANSFER ATTITUDE DETERMINATION, ORBIT ADJUST, AND CONTROL SYSTEM AND PROPELLANT SUPPLY, INCLUDING AVAILABLE RCS, FOR CRUISE-OUT ATTITUDE CONTROL AND COURSE CORRECTION. THIS INCLUDES ABILITY TO ADJUST AXIS ORIENTATION PRIOR TO PERIGEE BURNS
- ✓ ABILITY TO ADAPT ITS ON-ORBIT ATTITUDE DETERMINATION AND POINTING SYSTEM TO PLANETARY CONDITIONS
- ✓ ABILITY TO ADAPT ITS THERMAL CONTROL AND ENVIRONMENTAL SYSTEM DESIGN TO NEW CONDITIONS OF CRUISE-OUT AND PLANETARY ORBIT
- ABILITY TO ADAPT ITS TT & C (AND DATA HANDLING) SYSTEM TO INTERPLANETARY COMMUNICATIONS DURING TRANSFER, INJECTION, AND ON-ORBIT MANEUVERS
- ✓ ABILITY TO FULFILL QUARANTINE RESTRICTIONS, IF ANY

THE FLTSATCOM (✓) HAS MOST OF THESE ATTRIBUTES

ADAPTION OF FLTSATCOM GEO EARTH ORBITER

The FLTSATCOM is near perfect because:

- It is a three-axis controlled, nadir facing, communications satellite operating in Earth's geosynchronous orbit (GEO). Removal of its normal communication payload provides more than adequate room and power for all MGO/LGO scientific instruments, and has room, power, and performance to carry additional instruments.

- It reaches GEO in an undeployed spinning mode providing its own power and orbit correction system. Because of its long duration life on-orbit, it carries sufficient consumables to also achieve MGO/LGO propulsive demands.

- It is launched by the economical Atlas-Centaur, and this vehicle, if desired, can be retained for LGO mission launch. The MGO mission can be achieved using other standard launch vehicles.

- Its design loads and environment are strongly based on values induced while its apogee kick motor (AKM) is firing. The same AKM will be used for Martian orbit insertion.

- A new contract for three more units is presently under negotiation, guaranteeing its continued development during the 1980's.

- THE MGO MISSION IS MORE DIFFICULT, BECAUSE
  - COMMUNICATIONS DISTANCE AND VARIATIONS
  - NEW LAUNCH SYSTEMS
  - ORBIT INCLINATION CHANGE AND QUARANTINE MANEUVER
  - TIME REQUIRED TO CORRECT ANOMALIES
- BECAUSE OF THE WIDE WEIGHT AND PERFORMANCE MARGINS AFFORDED BY THE FLTSATCOM, THERE IS AMPLE ROOM, POWER, WEIGHT, ETC. TO CARRY ADDITIONAL EXPERIMENTS AND ADD FAULT TOLERANT/SAFE HAVEN MODES
- THE ABOVE IS OBVIOUSLY TRUE, IN SPADES, FOR THE LGO MISSION
- TRW CAN INTEGRATE THE MGO OR LGO, OR MGO/LGO MISSIONS INTO THE FLTSATCOM 6, 7, 8 PROGRAM WITH FACILITY FOR EITHER A 1988 OR 1990 LAUNCH. A 1992 LAUNCH MIGHT INCUR A SIGNIFICANT INCREASE IN COST (OVER AND ABOVE INFLATIONARY EFFECTS) IF THERE IS NO FOLLOW-ON TO FSC 8

TECHNICAL SUMMARY

All conditions are favorable for not only having available a spacecraft bus that fits MGO/LGO needs, but also one that has parent spacecraft production and test continuing through the period when go-ahead decisions would be made.

The sources of essentially all structure, equipment, and components for the MGO/LGO spacecraft stem from three TRW on-going programs:

1) The FLTSATCOM program's latest add-on is for units 6, 7, and 8. The initial FLTSATCOM launch occurred in 1978. This add-on program is presently planned for manufacturing completion at the end of 1985 with assembly and test to continue thereafter. There is a reasonable probability for a further follow-on order.

2) The power system for MGO/LGO will be partially adapted from the Defense Support Program (DSP) spacecraft bus. This program commenced in the early 1970s. A new order for Flights 14 through 17 is under negotiation, and includes a power system modification. This program will complete manufacturing in the summer of 1986, whereupon assembly and test will begin. The DSP program has a high probability of continuing through the 1990s.

3) The final source of MGO/LGO components comes from the Gamma Ray Observatory (GRO) Program presently in the design phase for NASA GSFC. The steerable high-gain antenna system will be derived from this program, and again timing appears to be ideal. The HGA for GRO is itself a direct descendent of the TRW DSCS-II program. The latter program is essentially completed, although some chance of additional unit add-ons depends on DSCS-III progress.

In summary, the time phasing scenario appears to be perfect if the MGO/LGO start is not delayed much beyond a 1990 opportunity (assuming no new follow-on orders for derivative spacecraft).

COST SUMMARY [1]

(IN 1982 DOLLARS FOR 1988 LAUNCH)

- THE BASELINE MGO MISSION SPACECRAFT COST IS [2]
  - THIS INCLUDES STS ADAPTION (8 $M) AND OTV ADAPTION* (2 $M)
- THE BASELINE LGO MISSION SPACECRAFT COST IS [2]
  - MAJOR SAVINGS COME FROM USE OF NORMAL A-CENTAUR LAUNCH VEHICLE (-8 $M) AND NO HGA (-2.7 $M)
- THE COST OF BOTH SPACECRAFT, IF DELIVERED FOR A COMMON STS LAUNCH IS [2]
  - SPACECRAFT SAVINGS COME FROM SHARED MANAGEMENT AND ENGINEERING SERVICES, EQUIPMENT BUYS, ETC.
  - ADDED EXPENSES ACCRUE DUE TO NEED TO ADAPT TO STS AND PAM-A, WITH ANC ADDED TO SPACECRAFT (+12 $M)
- IT MAY BE POSSIBLE TO LAUNCH THE MGO BY ATLAS-CENTAUR. IF SO, STS INTEGRATION COSTS WOULD BE MINIMIZED

* ASSUMES KNOWN SYSTEM, SUCH AS INTELSAT VI ASE

(1) ADD 10% TO GET PRICE

(2) COSTS ARE PRESENTED IN VOLUME II, WHICH IS PROPRIETARY

COST COMMENTS

The costs of adapting a spacecraft to STS launch are significant, and include:

- Payload Interface Definition
- Test Program Additions (Modal Surveys, Static Load)
- Thermal Analysis
- Fracture Control/Analysis
- Loads Analysis (Preliminary, Final, Components, Verification)
- System Safety/Reliability
- Mechanical/Electrical Interface
- Cleanliness

If spinning upper stage boosters are employed, it is probable that active nutation control systems will have to be added - and addition to the spacecraft (rather than the booster) is the most logical approach. If the upper stage is different from the Centaur, additional adaptation costs are accrued.

Cost details are shown in Volume II. The costing technique was to start with basic FLTSATCOM costs, and add and subtract from the alterations and changes.

MGO/LGO PROGRAMMATIC FACTORS

- WE PROPOSE, AS REQUISITE TO A LATE 1988 LAUNCH
  - RFP FOR NON-COMPETITIVE COSTING IN OCTOBER 1984
  - GFE INSTRUMENTS DELIVERED 1 YEAR PRIOR TO LAUNCH (OCTOBER '87)
- IN HARDWARE PROGRAM, MGO/LGO PROGRAM OFFICE WOULD
  - BUY MODIFIED FLTSATCOM BUS FROM FSC PROGRAM OFFICE
  - INSTALL GFE INSTRUMENTS AND OTHER MODIFICATIONS IN ASSEMBLY AND TEST
  - PERFORM SYSTEM TESTS TO SUPPLEMENT PRIOR TESTING AT BUS LEVEL
- STUDIES/COMPETITION(S) TO DETERMINE A CONTRACTOR AND PROJECTED COSTS IN NEXT TWO YEARS COMMENCING IN LATE 1982 OR EARLY 1983

PROGRAMMATIC FACTORS

TRW can accommodate a FLTSATCOM-based MGO/LGO program for a 1988 launch or 1990 launch with equal facility and without significant total program cost impact except for inflationary factors and length of program.

A 1988 launch would be possible if:

- TRW contract start was on 1 January, 1985
- A long-lead item buy of ~$10M was ready to order on 1 January, 1985 (per spacecraft)
- GFE scientific instruments were accepted by TRW no later than May 1987 (i.e., 1 year prior to spacecraft acceptance)

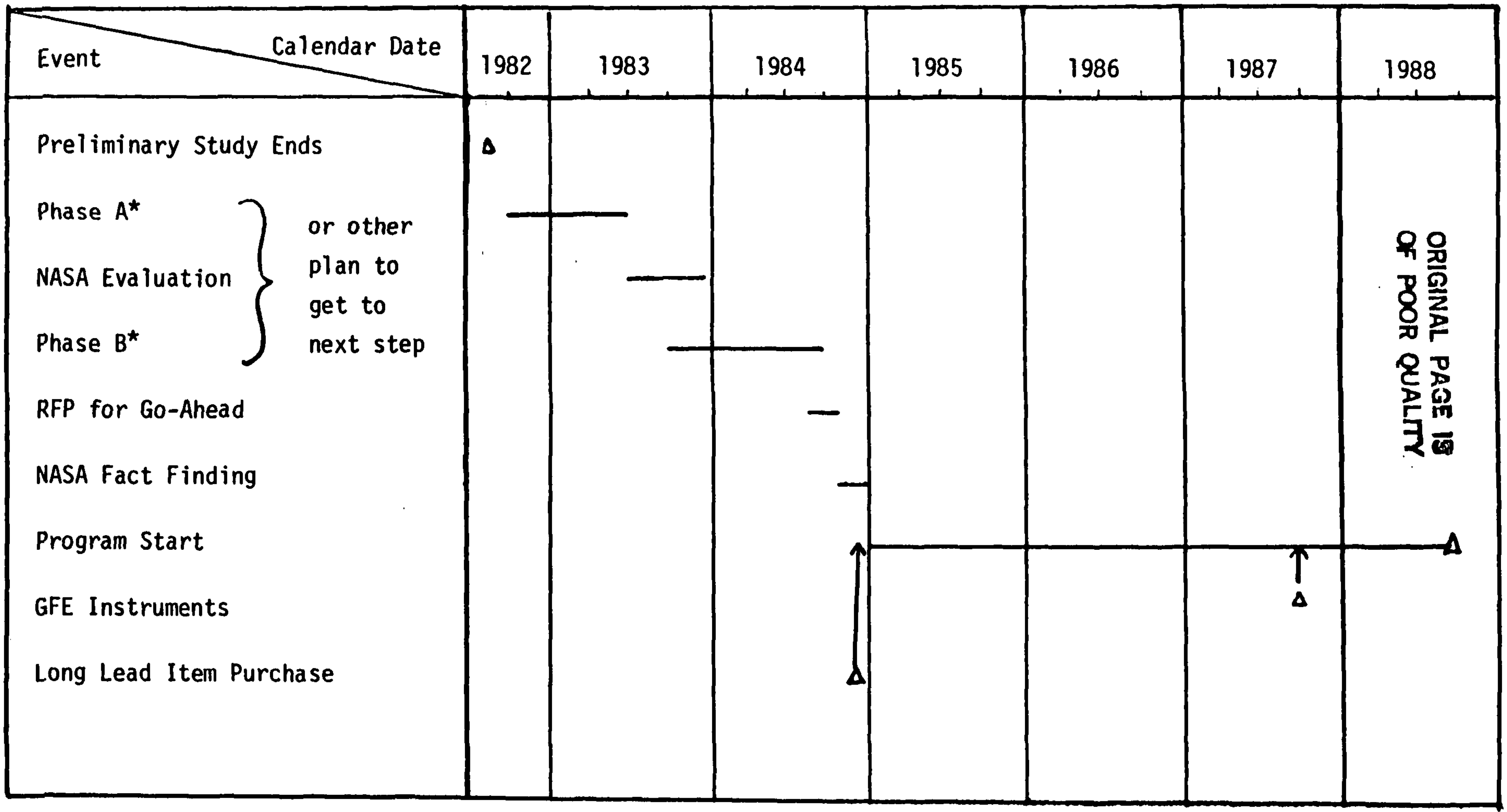
A 1990 launch might be advantageous from several fiscal, mission and work-flow standpoints.

- A long lead item block of dollars might not be required if design decisions can be made early enough to permit normal procurement
- Assuming no new FLTSATCOM orders, the FLTSATCOM Program Office could wind its business down more gracefully (synergism at work) with MGO/LGO following (rather than interleaving) the big FLTSATCOM 7 and 8 push

Conversely, a 1992 launch might yield significant cost increases if the FLTSATCOM Program Office has closed up shop after FLTSATCOM 8 launch.

PROPOSED MGO/LGO PROGRAM FOR LATE 1988 LAUNCH

ALTERNATIVES TO ARRIVE AT LONG LEAD PURCHASE MILESTONE ARE POSSIBLE



*RFP action for Phases A and B not shown.

1988 LAUNCH SCHEDULE

The basic schedule factor upon which the above program is based results from the average of about 3½ years which is necessary to design, fabricate, assemble, and test each one of the FLTSATCOM 6, 7, and 8 units. This schedule is predicated on receipt of authority to spend funds on long-lead items (e.g., AKM motor procurement) being received by 01 January 1985. Prior to that, an approved (by JPL) list of long-lead items for procurement would have to be generated. Since the Air Force funded TRW for FLTSATCOM 6, 7, and 8 long-lead items commencing 01 January 1982, much of the same list will be applicable.

STUDY BACKGROUND

MARS STUDY BACKGROUND*

	1981	1982	
CENTER	ARC	ARC	JPL
MISSION	MARS "WATER" MISSION	MARS ORBITERS: "CLIMATOLOGY" "AERONOMY"	MARS AND LUNAR "GEOSCIENCE" ORBITERS
VALUE	$20K	$175K	$65K
WINNERS	HUGHES RCA BALL MARTIN	TRW HUGHES	TRW RCA
LOSERS	--	RCA BALL	G.E. LOCKHEED BALL +?
PERFORMANCE	JUL-AUG	MAY-AUG	MAY-SEP

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* A LUNAR SCIENCE ORBITER WAS CONSIDERED IN "MISSION SUMMARY FOR LUNAR POLAR ORBITER", JPL 660-41A, 1977

STUDY BACKGROUND

As may be seen from the above, TRW was concurrently engaged in two very similar Martian Orbiter Studies - one being the study of this report, and the other being the NASA ARC sponsored "Climatology" Mission Study. Both studies utilized sun synchronous 300 Km circular orbits, and the JPL study would have almost satisfied its experimenters with the orbit-plane/sun line relationship desired by the ARC experimenters. Two of the four JPL instruments were also carried on the ARC mission.

Since the original ARC RFP required a spinning spacecraft (later ARC relaxed this requirement), TRW, not having a spinning earth orbiting bus to propose for adaptation, chose to propose a new "air frame" to ARC which utilized subsystems and components with good heritage. The resulting design turned out to be very similar to the MGO in weight and performance, but was judged to be significantly more expensive to develop.

An excellent account of the background of these "low cost" planetary orbiter studies appears in the October 1982 issue of Astronautics and Aeronautics, page 28. The article is entitled "Effective Planetary Exploration at Low Cost", by Jesse W. Moore, NASA OSSA.

I have discussed with Tom Young what sort of information his committee will want to receive from the contractors at the late August discussions. He has listed six topics which are outlined below:

1. Describe the basic spacecraft and its current use (include policy on redundancy, fault tolerance, etc.).

2. Describe the proposed planetary application and capabilities of the spacecraft for such missions.

3. What changes are required to the basic spacecraft and why.

4. What are the risks inherent in applying this Earth-orbiter to a planetary mission.

5. Will the systems, subsystems, and parts proposed be available in the time frame of interest.

6. What is the proposed implementation mode. (This part is a JPL responsibility but with inputs from you.)

Plan on three hours total including discussion. This means a pretty tight fast-paced presentation. I would target for a presentation length of two hours without discussion so that we will have time for the committee to ask questions.

I will be discussing the schedule with you by phone and will publish it when it becomes firm. As of now, the Committee plans all meetings to be at JPL. I realize this involves extra work but if we do it right it will add substantially to the credibility of the concept.

CAN EARTH ORBITERS HACK PLANETARY MISSIONS?

Shortly after the JPL and ARC Planetary Mission Studies were negotiated, the SSEC (Solar System Exploration Committee) formed a special summer study group, chaired by Tom Young (then Director of NASA GSFC) to investigate the righteousness of converting earth orbiters into planetary orbiters. The questions they posed are summarized above. Fact finding was conducted late in August 1982 at JPL, with TRW, RCA, Hughes, Ball Brothers as known participants. JPL retains a copy of the two TRW presentations (i.e., JPL Study and ARC Study). Later, interspersed in this report are pages which point to the problems of this conversion insofar as the FLTSATCOM is concerned.

A general conclusion, albeit a subjective one, is that if one reviews the inventory of earth orbiters, there will be one or two current models that can be adapted to any planetary mission, and thus this resource should <u>always</u> be investigated before starting a new development. Less subjective, since this report tells all, is the belief that a FLTSATCOM adaptation is ideal for the MGO mission, and pretty darn good for the LGO mission.

MISSION DESCRIPTION/REQUIREMENTS

MGO/LGO MISSIONS

- CONSIDER 1988, 1990, 1992 LAUNCH OPPORTUNITIES
- CATEGORY 1 INSTRUMENTS MANDATORY: CATEGORY 2 (LGO ONLY) (1)
  OPTIONAL
- LASER ALTIMETER OPTIONAL*
- MISSION LIFE

  MGO - 1 EARTH YEAR ON-ORBIT, 300 KM SUN SYNCHRONOUS, NOON/MIDNIGHT ORBIT
  LGO - 1 EARTH YEAR, 100 KM POLAR ORBIT
- DSN AVAILABLE 8 HOURS PER DAY
  - 64 M ANTENNAS AVAILABLE, AS NECESSARY
  - TRACKING ONCE/WEEK
  - 5 MINUTE ACQUISITION TIME FOR MGO

(1) CATEGORY 1 AND 2 INSTRUMENTS TO BE DEFINED IN SCIENCE SECTION IV

* WE DID NOT LET ITS PRELIMINARY DATA RATE REQUIREMENTS DESIGN C & DH SYSTEM

MISSION DISCUSSION

The Category 1 and 2 instruments for MGO/LGO are described in detail in the next section (IV). The FLTSATCOM bus provides ready accommodation of both mandatory and optional instruments.

The MGO mission requires a 300-km circular, polar, sun-synchronous orbit around Mars near the noon-midnight meridian. One earth year of operation in Mars orbit is planned. The LGO mission will be performed in a 100-km circular polar orbit around the moon with an expected mission duration of 1 earth year. Launch opportunities in 1988, 1990, and 1992 are to be considered for the Mars mission. The Lunar missions can be accomplished at any time during this period.

In both cases, it could be desirable to inject the spacecraft into an elliptical capture orbit with subsequent circularization at the specified low altitude in order to first calibrate the gamma ray instrument far from the planet. In TRW's mission concept, utilization of the solid apogee kick motor on the FLTSATCOM spacecraft for orbit injection at Mars or the moon and the limited maneuver capacity available from the existing monopropellant hydrazine propulsion system favors injection into a low altitude near-circular, rather than a highly eccentric, initial orbit. An alternative method to calibrate the gamma ray instrument was proposed and accepted (see later). An injection altitude of 350 km at Mars, and 250 km at the moon, is selected to allow for guidance errors and, in the case of MGO, to meet Martian quarantine safety requirements. Quarantine safety at Mars also requires an end-of-mission orbit raising maneuver to about 400 km to avoid orbital decay and possible impact on the planet prior to 2039, at the earliest.

10-DAY LAUNCH PERIOD

	BEGINNING	MIDDLE	END	
LAUNCH DATE	244 7347 88/07/04	7352 88/07/09	7357 88/07/14	(JULIAN DATE) (CALENDAR DATE)
ARRIVAL DATE	←	244 7555 89/01/28	→	(JULIAN DATE) (CALENDAR DATE)
TRIP TIME	208	203	198	DAYS
INJECTION ENERGY	11.9	11.85	12.35	KM^2/S^2
ARRIVAL V_∞	2.62	2.60	2.575	KM/S
ΔV_1 (AT EARTH)	3.375	3.73	3.75	KM/S
ΔV_2 AT MARS)	2.075	2.065	2.06	KM/S
TOTAL ΔV	5.81	5.795	5.81	KM/S
ZAP ANGLE (TYPE I)	103^0	104^0	105^0	

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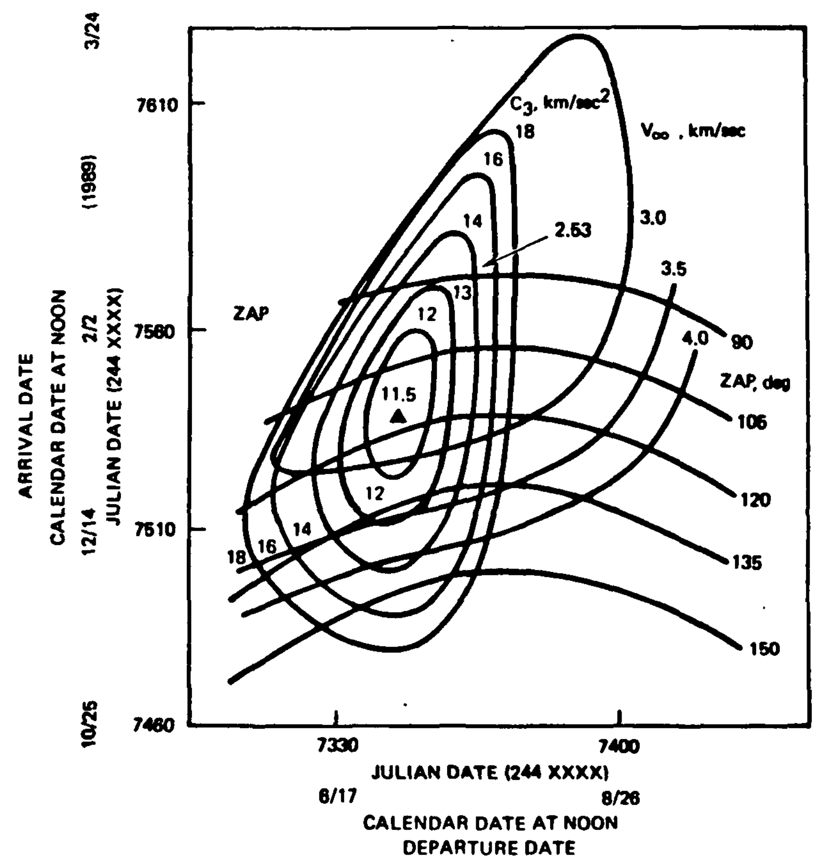
1988 MGO LAUNCH

Earth-to-Mars transfer trajectory characteristics for the 1988 launch opportunity will first be discussed, based on available data on Earth departure energy, Mars arrival velocity, and orientation of the Mars approach asymptote as functions of departure data and flight time. These characteristics are shown as contour plots of C_3 (departure energy), V_∞ (hyperbolic arrival velocity) and ZAP angle (orientation of the approach velocity vector relative to the sun) versus launch and arrival dates. A 10-20 day launch window is selected centered on the date of minimum departure energy, 1 July 1988. The shorter the window selected, the less the energy requirements become. A high ZAP angle value is desired to attain Mars entry conditions as close as possible to the required orbit alignment with the noon meridian. However, since departure energy and arrival velocity grow rapidly with increase in ZAP angle as trip time is decreased, this angle should be restricted to about 135 degrees. That is, for the assumed launch window, the arrival at Mars should not be set earlier than about 12 December 1988, which implies a 150 to 170 day trip time. Up-to-date contour plots, such as the one shown, have been supplied by JPL to include 1990 and 1992 launches.



1990 AND 1992 LAUNCH OPPORTUNITIES

- BOTH MISSIONS FAVOR TYPE II TRAJECTORIES WITH THE FOLLOWING MIDDLE-OF-10-DAY LAUNCH-WINDOW CHARACTERISTICS:

	1988 LAUNCH	1990 LAUNCH	1992 LAUNCH	UNIT
LAUNCH DATE	88/07/09	90/08/27	92/09/26	CALENDAR DATE
ARRIVAL DATE	89/01/28	91/08/18	93/08/31	CALENDAR DATE
TRIP TIME	203	356	339	EARTH DAYS
INJECTION ENERGY	11.85	15.53	11.92	KM^2/S^2
ARRIVAL V_∞	2.60	2.79	2.48	KM/S
ΔV_1 (AT EARTH)	3.73	3.89	3.73	KM/S
ΔV_2 (AT MARS)	2.065	2.16	2.01	KM/S
TOTAL ΔV	5.795	6.05	5.74	KM/S
ZAP ANGLE	104.2°	54.1°	72.6°	DEG
LVI ANGLE	0.1°	35.2°	14.3°	DEG
MOI SOLAR DISTANCE	1.533	1.643	1.591	A.U.
MOI EARTH DISTANCE	1.230	2.480	2.317	A.U.
MOI ORIENTATION ANGLE*	81°(N), 38°(S)	52°(N), 58°(S)	77°(N), 109°(S)	DEG

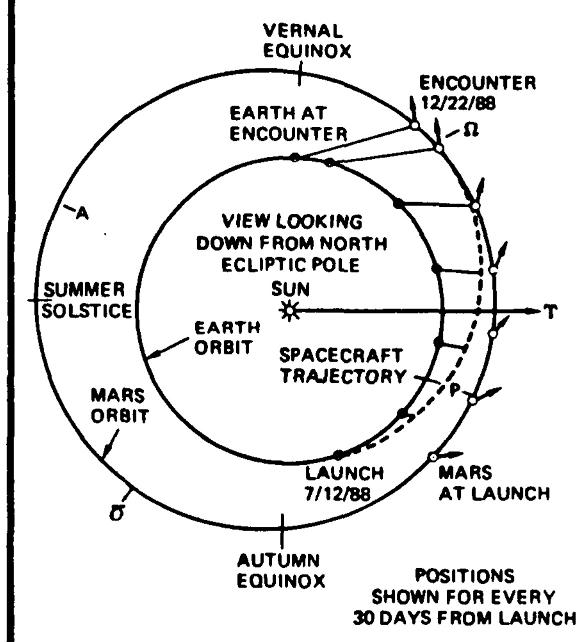
* WITH RESPECT TO EARTH (NORTHERN AND SOUTHERN APPROACHES)

ALTERNATIVE LAUNCH DATES

The 1990 and 1992 opportunities are more demanding as is apparent from a comparison of their velocity characteristics and principal ΔV requirements with those of the 1988 opportunity. The orbit insertion ΔV values listed are those required to initially enter into a near-circular 350 by 500 km altitude orbit. An additional 55 m/sec are required to trim this to a 300-km circular orbit. The two-stage orbit insertion procedure involves a 30 m/sec ΔV penalty compared with direct insertion at 300-km altitude. Note that a +0.5 percent rocket motor impulse dispersion would result in a $\pm$60 km apoapsis altitude variation.

For a 1988 launch, the orbit geometry shown has the following implications on Earth-spacecraft communications during the transfer and arrival phase. The sun-spacecraft-earth angle is close to 80 degrees within a day after launch and decreases steadily. About 30 days after launch, the sun and earth are within 47 degrees (as seen from the spacecraft) and stay within this value for the remainder of the mission. Communications may be degraded when the sun is in the link path. However, with careful planning, we foresee no operational problems. The spacecraft arrives at Mars about 10 days after the planet passes the ascending node and about 50 days before the (northern) vernal equinox. Since at launch the Earth is north of the Mars orbit plane, the spacecraft trajectory has a southerly component. This affects the Mars approach phase and spacecraft visibility from Earth at the time of orbit injection. Similar analyses are necessary to design 1990 and 1992 missions.

HELIOCENTRIC TRANSFER TRAJECTORY TO MARS



IMPLICATIONS OF 1990 OR 1992 LAUNCH DATES

- PERFORMANCE FLEXIBILITY IS SUCH THAT NO PROBLEMS ARE ENCOUNTERED BY LATER (1990 AND 1992) LAUNCH DATES.
- GREATER MARS-TO-EARTH DISTANCE AT MOI LINK BUDGETS DOWN 6.1 DB (1990) AND 5.5 DB (1992) RELATIVE TO 1988 LAUNCH
- LOWER DATA RATES AT MOI ARE NECESSITATED, BUT REMAIN ADEQUATE
- ZAP ANGLE CHANGES AFFECT INITIAL ORBIT PLANE ORIENTATION AND DRIFT PERIODS FOR MGO MISSION

LATER LAUNCHES

On the next page, we will see that the suggested upper stage booster, the SRM-1 Motor, provides adequate performance margins to accomplish launches in 1990 and 1992 (MGO fly-by weight is ~4000 pounds).

Following study initiation, JPL indicated that it was not scientifically necessary to transmit at high data rate (e.g. from multispectral scanner) during the entire mission, since the same ground coverage would be overlapped many times during the year on orbit. Thus, the design of the communication system is relieved, and the importance of the launch date lessened.

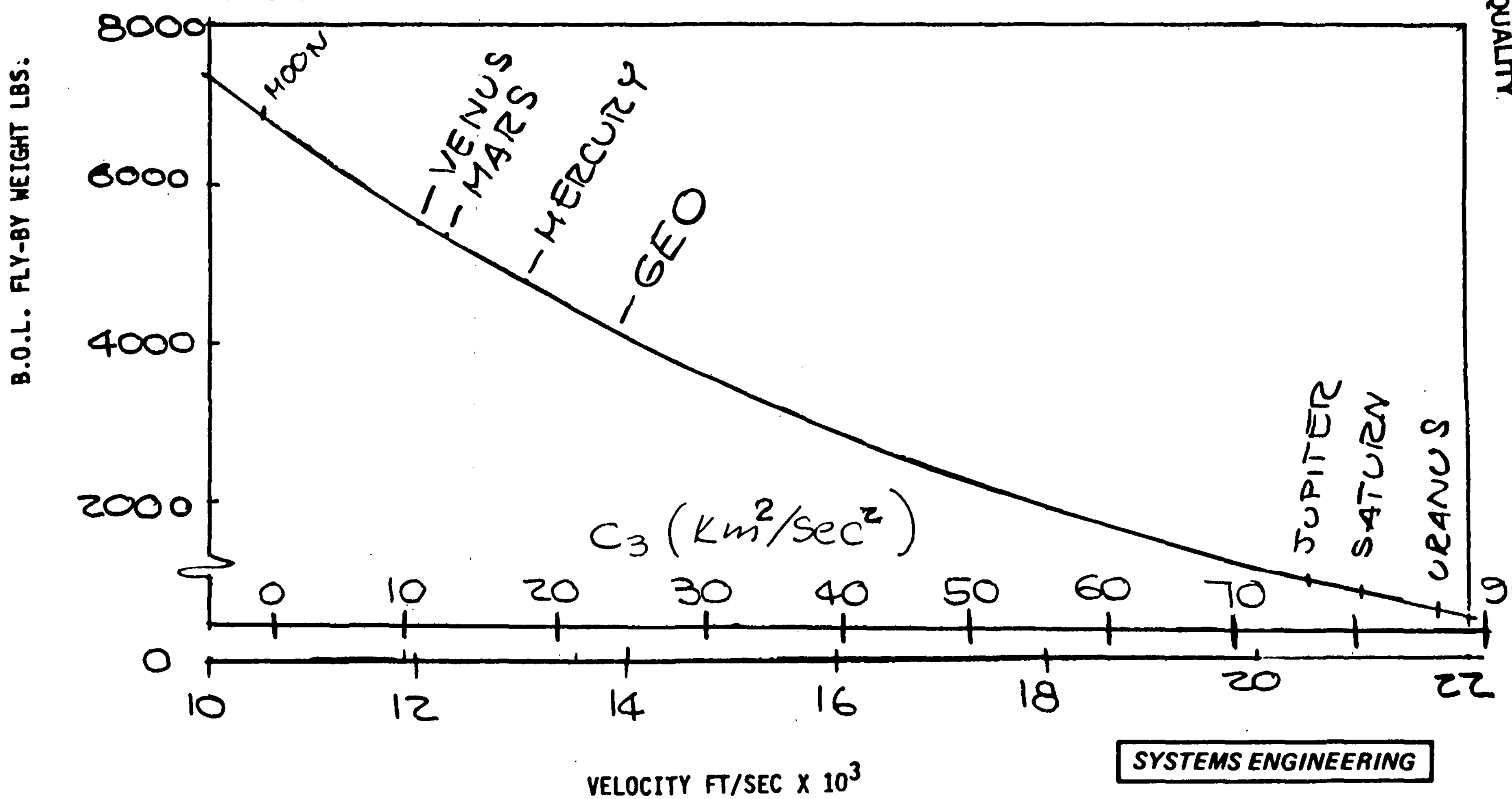
Later, it will be shown that a 1990 launch might best facilitate arriving at the desired noon-midnight orbit plane due to its favorable injection point and ZAP angle.

TRW
SPACE AND TECHNOLOGY GROUP

FLTSATCOM BUS INTERPLANETARY CAPABILITY
WITH SRM-1 MOTOR
(KSC LAUNCH)

MGO PERFORMANCE MARGINS

- AT BOOST TO MARS WITH 100% SRM-1 LOAD
  - WORST CASE WEIGHT MARGIN OF ~1000 LBS, IF NO BALLAST
  - IF ORBIT INSERTION MOTOR IS OFF-LOADED, MARGIN BECOMES ~1800 LBS
- AT INSERTION TO MARS
  - WORST CASE HAS EITHER 1500 FT/SEC EXCESS VELOCITY OR OFF-LOAD MOTOR BY 800 LBS OR USE STAR 37F (ALSO SAVES ~800 POUNDS)



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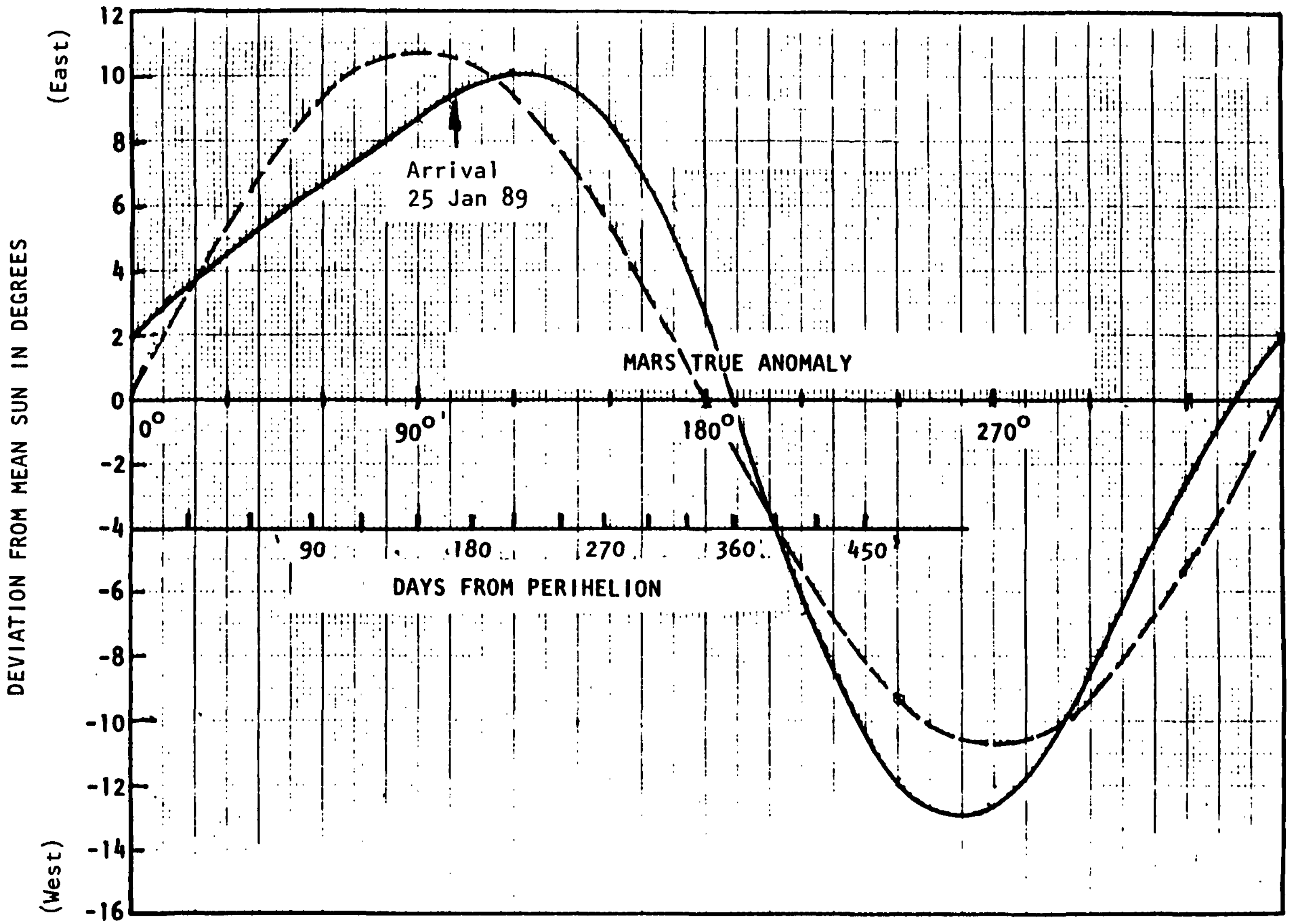
SYSTEMS ENGINEERING

THE SRM-1 PROVIDES ADEQUATE C_3

The above chart vividly illustrates that the total ΔV requirements to orbit the earth at GEO are more demanding than those to achieve fly-by of Venus, Mercury and Mars. This is, of course, due to the large energy requirement necessary to make the $\sim 23^0$ dog-leg maneuver from a KSC launch.

The inherent weight and size of the FLTSATCOM MGO bus precludes launch by Atlas, Titan, or STS based PAM-A boosters. The only presently available upper stage vehicle capable of meeting required booster performance is the IUS. This, however, is somewhat of an overkill and calls for off-loading of both motors. The same performance overkill would be true (even more so) if a Centaur version upper stage were employed. Neither of these approaches is very cost effective, and illustrate the need for an inexpensive transfer vehicle system intermediate to the PAM-A and IUS capabilities. A possible candidate, which will be developed as part of the INTELSAT VI Program, is a spinning boost system featuring the large SRM-1 first stage IUS motor. STS compatible ASE is being developed to launch INTELSAT VI and such equipment might be available to buy or rent from INTELSAT (or Hughes) at a reasonable cost. Since the FLTSATCOM spacecraft could supply the RCS for control, spin up, and active nutation control for an SRM-1/FLTSATCOM launch combination, a cost-effective integrated system could be developed based on the spinning version of the SRM-1. The spinning version SRM-1 motor will be qualified for the INTELSAT VI Program.

MGO 1988 LAUNCH



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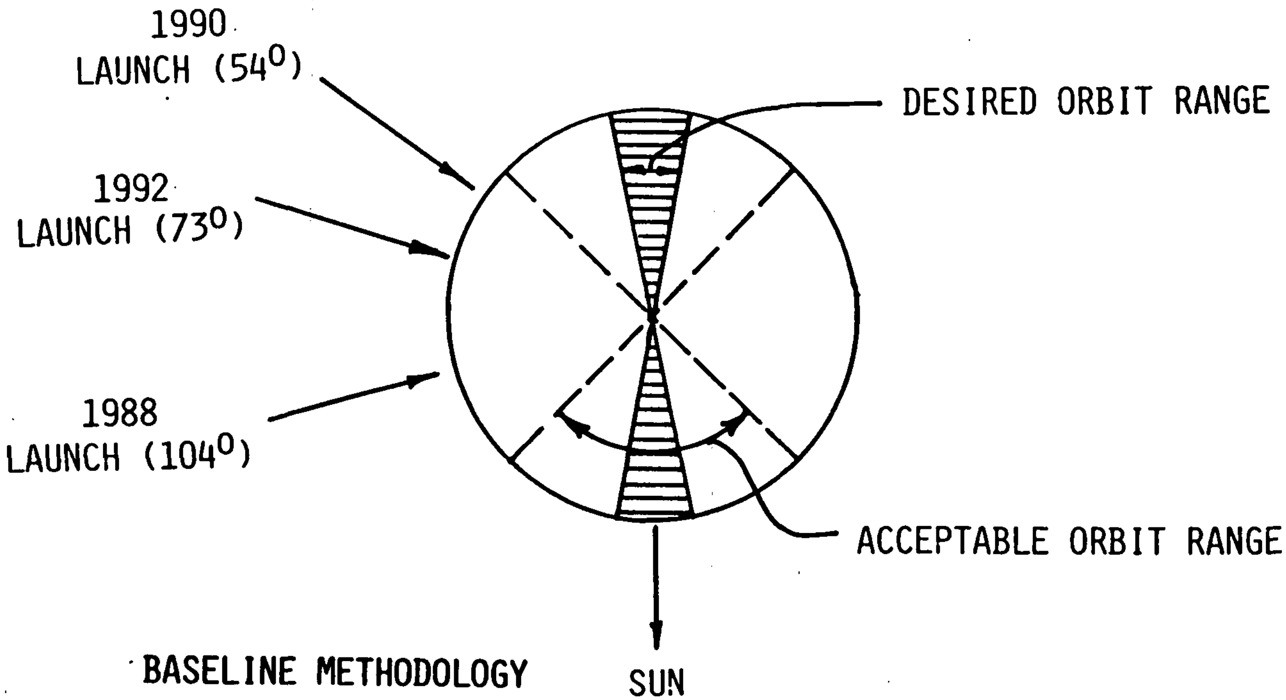
APPARENT MOTION OF SUN

The above figure depicts the Martian analemna for a 1988 launch. The broken curve shows the Martian equation of time if only orbit eccentricity is included. Thus, the sun deviation ranges over a total $\sim 23^{\circ}$ during a Martian year. A "sun synchronous" orbit, say noon-midnight, can be selected to limit this angle to $\pm \sim 6^{\circ}$ during an Earth year, if the initial orbit plane is established in the "middle" of the apparent longitudinal movement zone.

MARS ARRIVAL GEOMETRY



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BASELINE METHODOLOGY

- INITIAL ORBIT IS 300 $\pm$ 100 KM CIRCULAR, i = 92.6 $\pm$ Δ
  (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°
  (ΔV = 59.3 $\cdot$ Δi M/SEC)

ARRIVAL GEOMETRY VIS-A-VIS DRIFT

The figure shows the relationship between the desired Mars orbit orientation near the noon-midnight meridian and two alternate approach velocity directions.

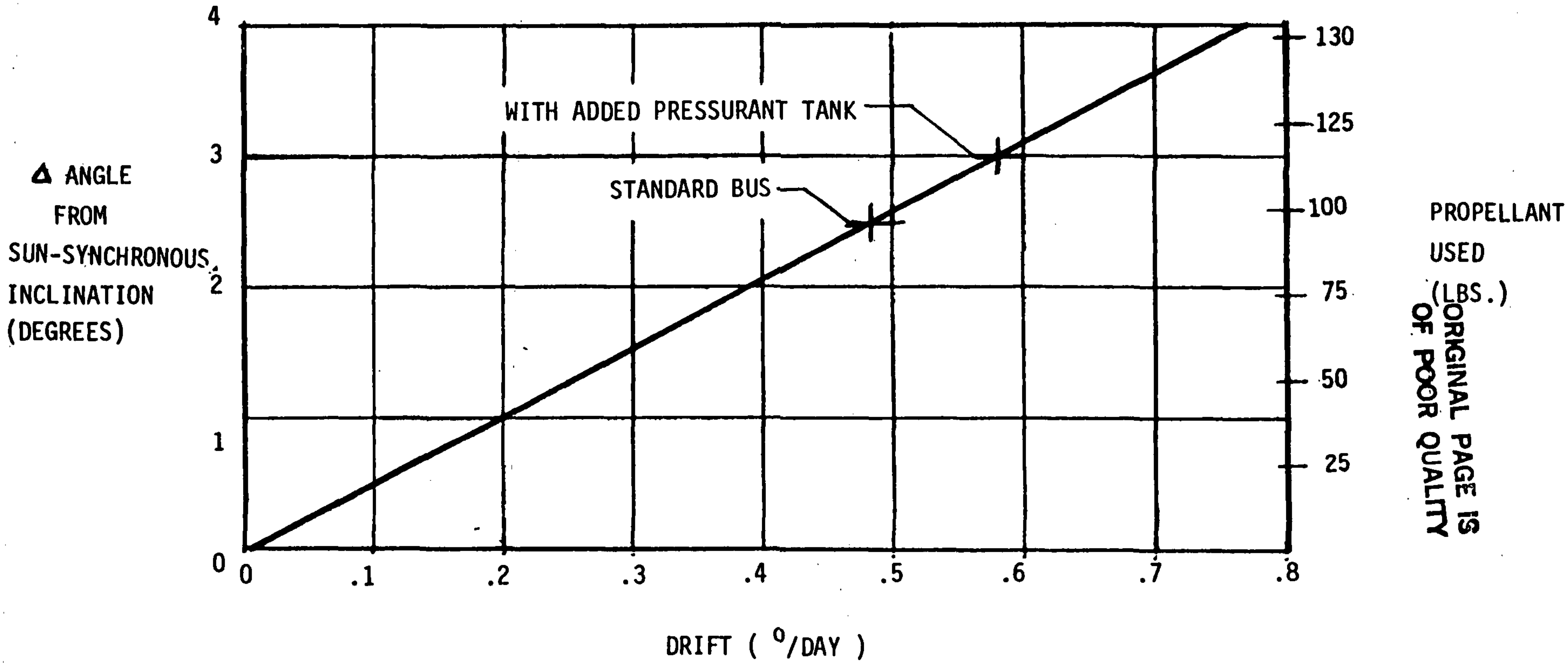
The shaded sector shown is a ± 45 minute time zone on both sides of the subsolar point. During the Martian year, the actual sun position varies periodically from its mean position because of the Mars orbit heliocentric eccentricity, as was shown on the previous page set.

The figure is presented as if the sun were fixed. In this coordinate system, the desired nominal noon-midnight orbit appears to oscillate once per year with respect to the sun as shown. At arrival for the 1988 opportunity, this orbit appears to the west of the sun and the required nodal drift is as shown. The drift for the 1990 and 1992 opportunities are also shown. By selecting the correct orbit inclination, drift can occur in either direction.

An alternate drift strategy (1, 2), is also possible, if the MOI motor is fully loaded and its excess energy is used to also rotate the line of nodes at insertion. Finally, strategy 3 provides the inclination which limits the yearly drift to $\sim\pm 45^{\circ}$ around the mean noon-midnight sun line:

1. Choose initial orbit with "minimal" Δi
2. Use excess MOI propellant capability to rotate line of nodes, or
3. Choose inclination with $\Delta i = 0.7^{\circ}$, allowing for 90° drift in node during one Earth year (save up to 100% of hydrazine requirement for orbit finalization; e.g., 1990 encounter)

PROGRESSION OR REGRESSION OF THE ORBIT PLANE



DRIFT TIME VS PROPELLANT USAGE

Since the FLTSATCOM design is propellant limited, it is essential to plan the best strategy to control propellant usage to stop drift as soon as a favorable location, vis-a-vis the sunline, is attained. By carefully monitoring propellant use during cruise-out, one can budget the amount of propellant available to stop drift. If the mid-course corrections are small, a "high" drift rate can be established. Conversely, if mid-course corrections require more propellant than budgeted, a low (unto zero) drift rate must be established. If only modest mid-course corrections have been made, then drift rates can be high.

In later study phases, TRW plans to investigate several alternative operating modes to reduce orbit drift time without requiring excessive ΔV maneuvers such as:

1) Perform some plane change as part of the orbit insertion maneuver, using excess orbit insertion motor energy if available, to reduce subsequent nodal drift requirements.

2) Adopt an MGO orbit orientation short of full alignment with the mean sun, e.g., at 11:00 (for Type I arrival) or 1:00 (for Type II arrival), thus saving about one third of the drift period. (This is probably within the range of acceptable mission performance.)

3) Start science payload operations while still in the drift mode, thus making a slow drift rate acceptable and avoiding large ΔV requirements.

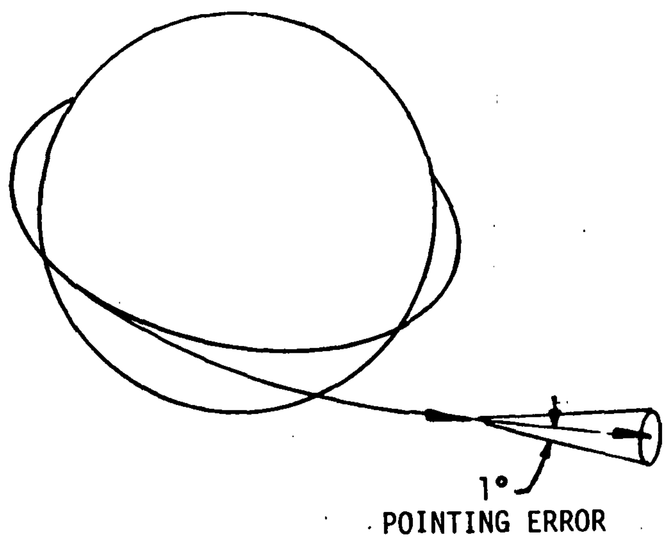
With respect to alternative 1, a 10-degree plane change during the MOI maneuver would produce an 8 to 10 degree nodal shift, thus reducing nodal regression (drift) time by 20 to 30 percent at an MOI maneuver penalty of less than 3 percent (about 70 m/sec in the 1988 mission example).

Initiation of science payload operation during the drift mode (Alternative 3) is feasible immediately since the solar array is overdesigned because of requirements during the pre-MOI phase when the array is folded.

3 σ ΔV'S REQUIRED TO CORRECT INJECTION ERRORS RESULTING FROM INITIAL POINTING ERRORS AND MOTOR TOTAL IMPULSE VARIATIONS

MGO

EARTH DEPARTURE



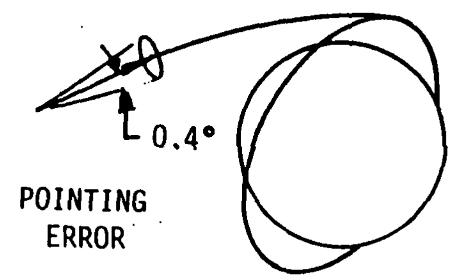
ΔV = 0.75% of ΔV PERIGEE BURN, 3σ

TOTAL VELOCITY ERROR =

$[(4 \text{ km/sec} \cdot \sin 1°)^2 + (.0075 \cdot 4 \text{ km/sec})^2]^{\frac{1}{2}}$

$\longrightarrow$ 76 m/sec, (250 ft/sec)

MARS INJECTION



ΔV = 0.75% OF ΔV INJECTION, 3σ

TOTAL VELOCITY ERROR =

$[(2.1 \text{ km/sec} \cdot \sin .4°)^2 + (2.1 \text{ km/sec} \cdot .0075)^2]^{\frac{1}{2}}$

$\longrightarrow$ 21.5 m/sec, (70 ft/sec)

PROPELLANT REQUIREMENTS - INJECTION ERRORS

Hydrazine propellant will be needed for mid-course corrections (LGO, if launched by Centaur, should be inserted with relatively high accuracy), and for correcting planetary insertion errors and deliberate safety biases.

If solid motors are used for both insertion maneuvers, then total injection errors will be due to two major causes:

(1) Variation in total impulse delivered - taken to be 0.75%

(2) Initial pointing errors of mean spin axes

Typical type (2) errors might be about 1° for insertion into planetary transfer orbit (this is an open loop value and includes STS pointing error at MGO release, and subsequent errors accrued in the spin-up process) and 0.4° for planetary insertion. The latter error, for MGO, is generous, since a stellar scanner based attitude determination system could be utilized.

The resulting rms ΔV's are 250 ft/sec and 70 ft/sec. These values will be used later in determining the hydrazine-required budget.

FINAL BOOST TO SAFE ORBIT

REQUIREMENT

PPP REQUIRES 10^{-4} PROBABILITY OF SURFACE IMPACT TO 2009 AND 0.05 PROBABILITY TO 2039

RESULTS

ALTITUDE	LIFETIME (TO 250 KM)
300	4.1 YEARS
350	22 YEARS
400	165 YEARS

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CONCLUSION: RAISE TO 400 KM ⟶ ΔV = 148 FT/SEC (~ 27 LBS N_2H_4)

DRAG MAINTENANCE ⟶ ΔV ~ 30 FT/SEC AT 300 KM (MAX)

PROPELLANT REQUIREMENTS - QUARANTINE MANEUVER

After study initiation, JPL suggested that planetary quarantine restrictions might still apply to inadvertently "landing" foreign objects on Mars. According brief studies were made to determine the circular altitude above Mars where life-in-orbit after the year 2039 could be assured. The above indicates that even allowing for poor planetary environment data and drag/cross section estimates, a circular 400 km orbit, achieved by a periapsis burn followed by an apoapsis burn, should be adequate.

Similarly, worst case estimates of drag effects led to relatively small ΔV requirement to maintain orbit altitude.

MOI LOCATIONS AND SPACECRAFT POINTING ANGLES (I = 92.6°)

- MOI (POINT OF CLOSEST APPROACH) OCCURS AT:

  51°, 87°, 63° N FOR NORTHERN APPROACHES

  51°, 17°, 35° S FOR SOUTHERN APPROACHES

- FOR 1988 ENCOUNTER, ANGLE OF SPACECRAFT AXIS TO SUN LINE IS 81° (NORTHERN APPROACH) AND 77° (SOUTHERN APPROACH). TO EARTH LINE THESE ARE 69° AND 38°.

- IN 1988 EARTH OCCULTATION OCCURS 39° BEFORE MOI NORTH, BUT 15° AFTER MOI SOUTH

  IN 1990 THESE ARE 170° AFTER AND 234° AFTER

  IN 1992 THESE ARE 108° AFTER AND 278° AFTER

  (IN LATTER CASES OCCULTATION TIME IS $<$ 2 MINUTES)

WHICH MARTIAN INJECTION - NORTH OR SOUTH?

We will now begin a short series of pages basically dealing with the time it will take to arrive at a noon-midnight MGO orbit for any of the launch years under consideration.

First we must consider Northern vs Southern hemisphere insertions. Factors to be considered are:

(1) Omni-antenna numbers needed and types

(2) Earth viewing

(3) Folded solar array capability

(4) Ability to rotate line of nodes by excess OIM motor energy (i.e., nearness of injection point to pole)

The spacecraft will be targeted to approach Mars over the North or South pole for a MOI maneuver either at a northern or southern latitude. For Type I arrival trajectories, the perihelion will be located on the far side of the planet as seen from earth; for Type II arrival it will be on the near side.

Since the approach velocity vector generally will be inclined relative to the Martian equator, (in the 1988, Type I transfers it will have a southerly component) the northern and southern MOI locations generally will be asymmetrical with respect to the poles, and the spacecraft sun and earth pointing geometry will differ in the two approaches. Selection of the northern or southern MOI approach will depend on which one provides the most favorable pointing conditions for spacecraft-to-earth communication and solar array illumination. Both approaches are equally suitable from the mission scientific objectives standpoint since the resulting sense of orbital revolution, i.e., with the ascending or descending node being located near the subsolar point, is immaterial in payload operations.

RATIONALE FOR PLANE CHANGE AT MOI

- LARGE SPARE ΔV CAPACITY (450 M/SEC FOR 1988) OF OIM MOTOR PERMITS MAJOR NODE SHIFT BY PLANE CHANGE DURING MOI, REDUCING TOTAL DRIFT TIME TOWARD SUN SYNCHRONOUS ORBIT ALIGNMENT.

- IMPROVE NODE CHANGE EFFECTIVENESS BY SHIFTING MOI LOCATION TOWARD POLE, OFF APPROACH TRAJECTORY PERIAPSIS.

- THIS ADDS IN-PLANE FLIGHT PATH ANGLE CHANGE, ϵ, BUT ONLY HAS MINOR EFFECT ON TOTAL DOG-LEG ANGLE (SEE NEXT PAGE SET).

- SMALL Δi CAUSED BY OFF-POLAR PLANE CHANGE TO BE TAKEN INTO ACCOUNT IN TARGETING THE AIM POINT IN B-PLANE.

- BOTTOM LINE EXAMPLE (SEE NEXT 3 PAGE SETS) -- IF MOI OCCURS @ $\epsilon \approx 10°$, THE LINE OF NODES OF THE ORBIT CAN BE "CRANKED" AROUND BY $\sim 15°$ AT LITTLE ΔV PENALTY TO ROUNDING OUT AT 300 KM.

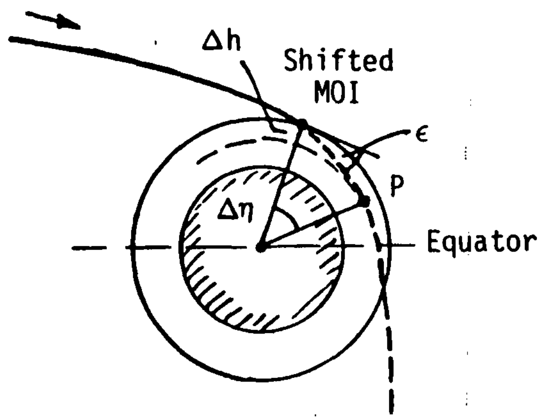
USE OF EXCESS MOI* MGO MOTOR CAPABILITY

The highly efficient STAR 37FM motor to be utilized by FLTSATCOM 7 and 8 can be off-loaded by ~800 pounds (see page 111-10) and still achieve Martian circular orbit. On the other hand, since this weight saving is not essential due to the over-capability of the SRM-1 boost motor, the excess energy could be used to rotate the injection plane to get closer to the noon-midnight orientation.

The next three page sets will present details on maneuver trade-offs. A detailed explanation of this work, by Hans Meissinger, is given in Appendix A-1.

* Mars orbit injection.

COMPOSITE DOG-LEG MANEUVER



$\Delta\eta$ = true anomaly shift
ϵ = local flight path angle
Δh = altitude change

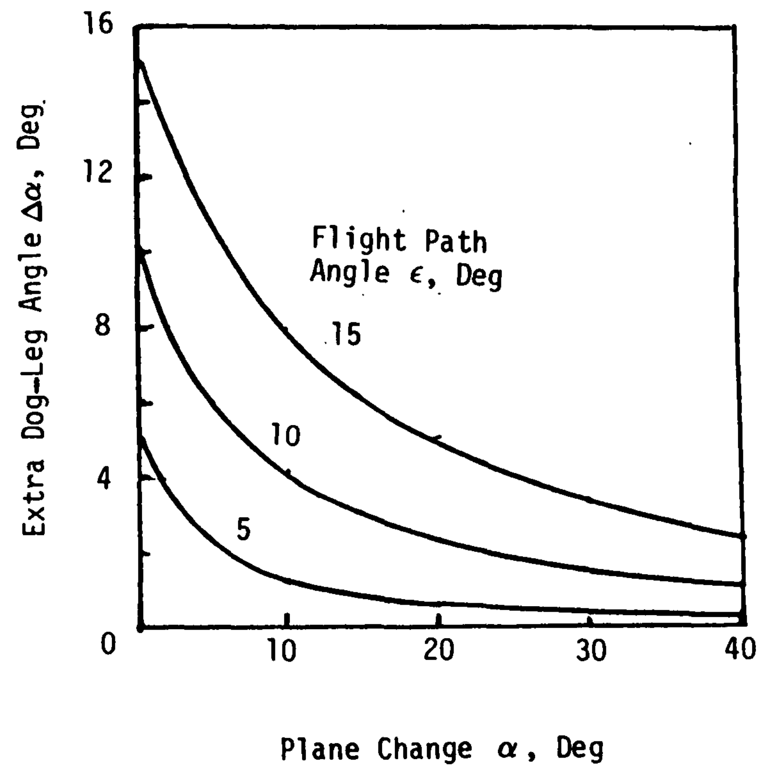
1. Shift of MOI toward pole

   - Increases node change effectiveness
   - Adds in-plane dog-leg component (ϵ) to out-of-plane α
   - Raises MOI altitude, adds Δh orbit trim requirement

2. Composite dog-leg angle:

$$\alpha_1 = \cos^{-1} (\cos \alpha \cos \epsilon)$$

Extra angle $\Delta\alpha = \alpha_1 - \alpha$ due to ϵ is small if α large



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PROCEDURE TO USE EXCESS MOI MOTOR ENERGY

The "crank around" maneuver requires motor firing at other than true periapsis (where the flight path angle, ϵ, is zero, point P) and at a higher altitude than for $\epsilon = 0$. These changes shift the MOI burn point towards the pole where the "cranking" maneuver is most effective in producing a larger plane change, α.

EFFECTS OF MOI OFFSET FROM PERIAPSIS

(1988 MISSION, h_p = 300 KM, e = 1.0614)

TRUE ANOMALY AT MOI η (DEG)	ALTITUDE CHANGE Δh (KM)	FLIGHT PATH ANGLE ε (DEG)	ΔV FOR SUBSEQUENT ALTITUDE ADJUSTMENT (M/SEC)
0	-	-	-
-10	29	5.3	13
-20	118	10.6	53
-30	274	15.5	119
-40	506	20.6	211

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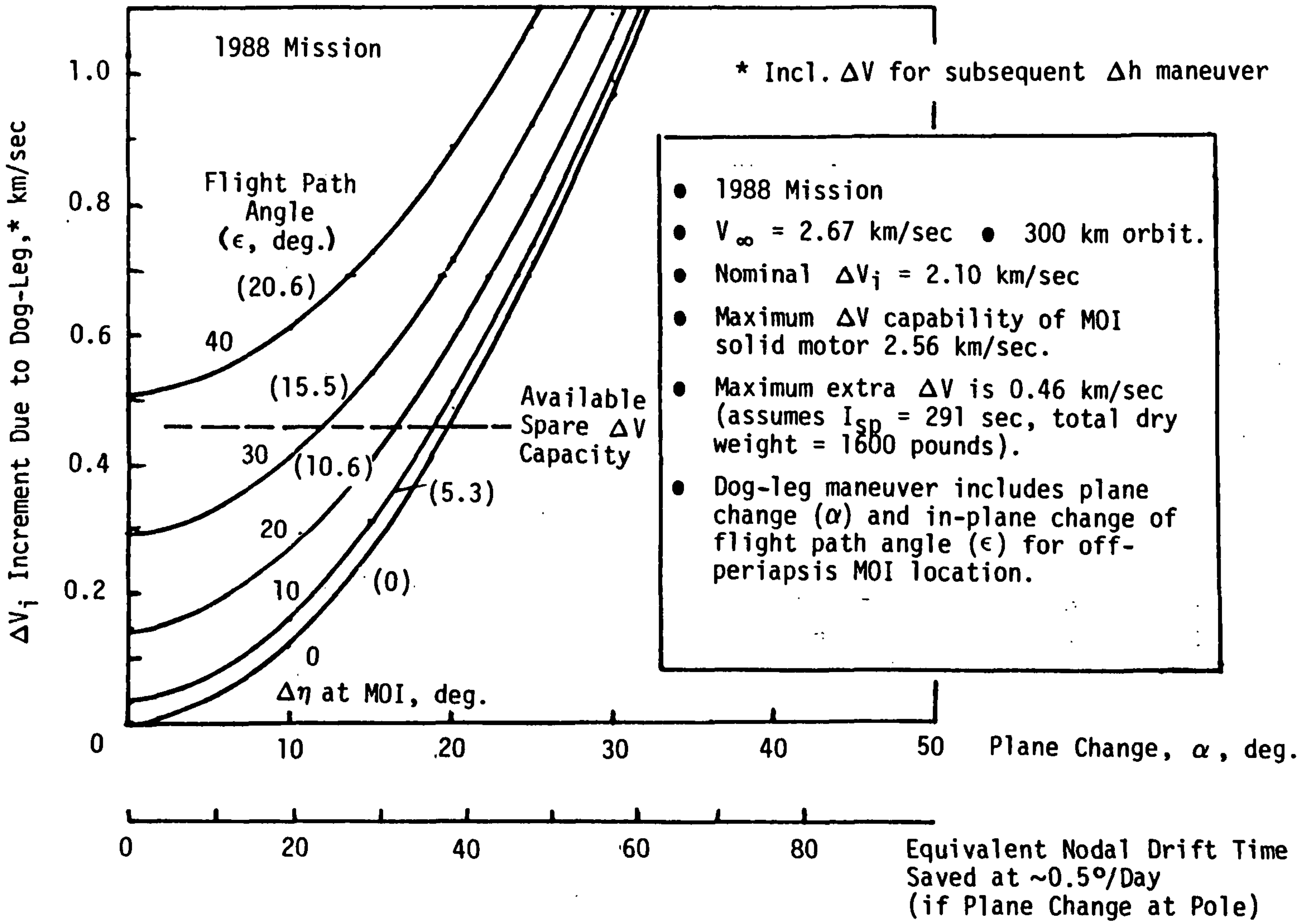
TYPICAL CRANKING STATISTICS

The above chart shows, for example, that if a flight path angle of $\sim 10°$ is selected (1988 - northern hemisphere), injection altitude will be ~ 120 km above 300 km (with no biases) and this will later require a ΔV correction of ~ 50 m/sec. Injection will take place $\sim 20°$ prior to the "no-crank" zero flight path angle position. Obviously, if correction ΔV's get too large, the "cranking" scheme loses viability.

PLANE CHANGE BY MOI DOG-LEG MANEUVER



- 1988 Mission
- V_∞ = 2.67 km/sec
- 300 km orbit.
- Nominal ΔV_i = 2.10 km/sec
- Maximum ΔV capability of MOI solid motor 2.56 km/sec.
- Maximum extra ΔV is 0.46 km/sec (assumes I_{sp} = 291 sec, total dry weight = 1600 pounds).
- Dog-leg maneuver includes plane change (α) and in-plane change of flight path angle (ϵ) for off-periapsis MOI location.

FINAL "CRANKING" RESULTS AND MISSION CONCLUSIONS

The final "cranking" results are summarized above, and may be illustrated by an example. It may be seen that if (ϵ) is selected $\sim 10^{\circ}$, the orbit could be rotated ("cranked") by $\sim 17^{\circ}$ closer to the noon-midnight orbit, by taking advantage of all the available spare ΔV capacity (in 1988).

The proposal (previously cited) includes considerable information on arrival geometry and spacecraft/earth look angles during pre-MOI maneuvers. This work is not reported in the present final report.

Moreover, since the proposal included considerable detail concerning the mission aspects of the LGO mission, and the effect of the injection maneuvers on arrival geometry and spacecraft/earth-look angles are not stressful, they are not discussed again in this report.

Note that an MOI mode using a highly eccentric initial capture orbit, e.g., 350 km by 5 Mars radii, would permit a comparatively inexpensive node change maneuver near apoapsis. However, since the relatively inefficient monopropellant would be required for this maneuver and subsequent orbit trim, the required design change would not be cost effective.

SCIENCE REQUIREMENTS AND INSTRUMENTS

DETERMINATION OF SCIENCE REQUIREMENTS

METHODOLOGY FOLLOWED

- REVIEW MISSION REQUIREMENTS DOCUMENTS
- AMPLIFY REQUIREMENTS WITH ADDITIONAL SOURCE DOCUMENTS, OTHER MISSION DESIGNS
- DERIVE ADDITIONAL REQUIREMENTS (FURNISHED BY JPL)
- IDENTIFY QUESTIONS AND PROBLEM AREAS
- ITERATE WITH JPL AND CANDIDATE INSTRUMENT SPONSORS
- FINALIZE EXPERIMENT REQUIREMENTS

RESULTS

- EACH INSTRUMENT'S REQUIREMENTS WERE REVIEWED WITH SPONSOR(S) AND PERFORMANCE AGREED UPON
- γ-RAY INSTRUMENT CALIBRATION PROCEDURE AGREED UPON
- ALL INSTRUMENTS (PRIME AND OPTIONAL) SUCCESSFULLY PLACED ON SPACECRAFT
- REGULATED VOLTAGE SUPPLIED AS REQUIRED

INSTRUMENT REQUIREMENTS

The most important factor leading to a realization of the MGO and/or LGO missions is providing a bus that satisfies the instrument requirements and the operational goals. Adaptation of a bus presents different technical challenges than designing a new spacecraft to accommodate the instrument requirements. Using either approach, compromises would be necessary. FLTSATCOM can physically accommodate all instruments. The operational requirements, however, lead to compromises in the amount of data taken and stored, and the rate at which data is dumped. However, this compromise would be necessary whether an existing bus was adapted or a new spacecraft was designed for this application.

PERMISSABLE SCIENTIFIC/OPERATING LEEWAY

- MGO SCIENCE DOES NOT NEED TO BE DONE SOLELY IN SUN SYNCHRONOUS ORBIT - ORBIT CAN DRIFT $\pm$ 45° FROM MEAN SUN LINE
- MSM DATA NEED NOT BE TAKEN CONTINUOUSLY - CAN WAIT UNTIL MARS IS NEARER TO EARTH TO EASE REQUIREMENT ON THE TT&C SYSTEM
- 64M DSN DISHES CAN BE MADE AVAILABLE FOR WORST CASE SITUATIONS
- γ-RAY INSTRUMENT CAN BE SATISFACTORILY CALIBRATED DURING CRUISE ORBIT OUT TO MARS AND THE MOON EVEN THOUGH SOLAR PANELS ARE NOT EXTENDED.

OPERATING LEEWAY

Since the scientific requirements are not rigid, the missions, particularly MGO, can be adjusted to account for many contingencies, thus considerably enhancing success chances. For example:

- If due to excessive propellant usage during mid-course and/or post MOI orbit adjustments, a continually drifting orbit plane would be acceptable, thus precluding propellant usage to stop the drift
- High data rate data dumps can wait until the Mars/Earth distances are near minimum, thus limiting TT&C antenna size and EIRP
- To limit size of data storage, high data rate data can be taken in short "bursts", since the ground track will repeat itself many times during a Martian season
- The omni transmitting antennas can be sized for operation with the 64m DSN stations, permitting TT&C sizing for worst case distances and relative orientations

EXPERIMENT ACCOMMODATION REQUIREMENT DRIVERS

- MSM IS ATTITUDE CONTROL DRIVER
  - POINTING ACCURACY 0.2°
  - POINTING STABILITY 5.7×10^{-4} $^{\circ}$/MIN *
  - OTHER XPTS 1.50° AND 5.7×10^{-3} $^{\circ}$/MIN
- TWO ARE BOOM MOUNTED (MAG, GRS)
- TWO REQUIRE PASSIVE COOLING (MSM, GRS)
- MSM HAS HIGH DATA RATE CAPABILITY (TO 12KBPS)
  - SENSOR DAMAGED BY PARTICULATE CONTAMINANTS
  - DIRECT EXPOSURE TO SUNLIGHT (COMMANDABLE APERTURE COVER)
- MAGNETOMETER REQUIRES THAT SPACECRAFT UNDERGO A MAGNETIC CLEANLINESS PROGRAM

* THIS REQUIREMENT MAY BE RELAXED BY A FACTOR OF 50-100. HOWEVER, THIS RELAXATION DOES NOT IMPACT DESIGN.

MGO MISSION SCIENCE

The experiment accommodation requirement drivers for the MGO mission are in the areas of attitude control, thermal control, data management and communication requirements, and mechanical accommodation. The Multispectral Mapper (MSM) imposes driving requirements on data rate as well as attitude control and stability. Since the MSM sensor is susceptible to damage by particulate contaminants and direct exposure to sunlight, it will be fitted with commandable aperture covers. All of these requirements can be accommodated by the FSC-derived system.

INSTRUMENT POWER, WEIGHT AND DATA REQUIREMENTS

INSTRUMENT	WEIGHT (KG)	POWER AVERAGE (WATTS)	DUTY CYCLE %	DATA COLLECTION RATES PER SECOND (BITS/SEC)	PER DAY (BITS/DAY)
GAMMA RAY SPECTROMETER (GRS)	12	10	100	1.5×10^3	1.3×10^8
MULTISPECTRAL MAPPER (MSM)	17	8	33	$(1.5\text{-}12) \times 10^3$ (6×10^3 AVG)	1.7×10^8
RADAR ALTIMETER (RADAR)	10	18	100	0.6×10^3	0.5×10^8
MAGNETOMETER (MAG)	3	4	100	0.4×10^3	0.35×10^8
TOTAL	42	40		8.5×10^3	3.85×10^8

From the above, it may be seen that the MSM is the communications and data handling subsystem driver. Its operational use will have to be carefully considered in finalizing C&DH design.

The main thrust in this study was the selection of a low-cost spacecraft with capabilities to accommodate in a broad sense a large fraction of the science requirements. Additional constraints including individual instrument location on the spacecraft to achieve favorable viewing geometries, duty cycling, cooling, and attitude control requirements were satisfied.

The gamma ray spectrometer, radar altimeter, and magnetometer are independent of sun direction and will be gathering data on both the dark and light side of the planets. The instrumental area resolution of the gamma ray spectrometer is approximately 50-km. However, the measured areal resolution and analytical precision depends on the dwell time per unit area. The count rate observed by the gamma ray spectrometer is sufficiently low that data must be accumulated over many successive passes to provide adequate statistics. To achieve the nominal mission objectives, a few months of data are required. The planetary coverage and swath sizes for typical fields of view are discussed in the proposal. For the selected polar orbit, the gamma ray spectrometer will obtain the best areal resolution near the poles. The Multispectral Mapper (MSM) on the other hand, will have poorer lighting conditions near the poles and will obtain the best data at the lower latitudes.

γ-RAY SPECTROMETER FOR MGO/LGO

CAT. 1

TITLE: γ RAY SPECTROMETER

P.I.: AL METZGER JPL; JIM ARNOLD UCSD

SCIENTIFIC OBJECTIVE

MEASURE NEAR SURFACE ABUNDANCE OF ELEMENTS AND IDENTIFY MAJOR RESERVOIRS OF SURFACE AND SUBSURFACE WATER

DIM.

20x32x60 CM

MASS

12 KG (INCLUDES ELECTRONICS & HTRS)

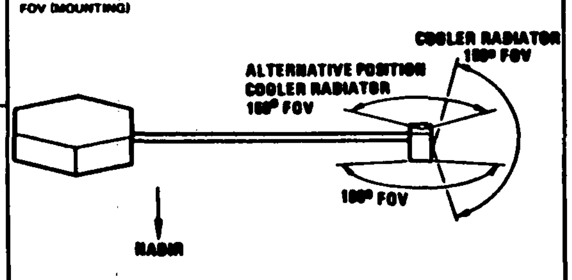
FOV

DETECTOR 180° (2π STER)
COOLER ~160°
EXCLUSION CONE: 180°

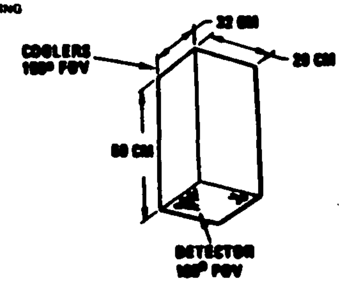
MOUNTING CONSTRAINTS

MOUNTED ON EXTENDABLE BOOM
>1 S/C DIAMETER

FOV (MOUNTING)



SKETCH/DRAWING



MEASUREMENT

MEASURE EMISSION OF GAMMA RAYS FROM THE SURFACE AS A FUNCTION OF LOCATION AND TIME

GRD. OPS, GSE, FACILITY, HANDLING, TESTING

INFLIGHT CALIBRATE

DURING CRUISE WITH BOOM EXTENDED AND IN MID POSITION OR AT DISTANCES >10 PLANETARY RADII (AT APOAPSIS)
NOTE: IN DEPLOYED POSITION TAKES ~3 DAYS TO COOL TO 110°K

ELECTRICAL

POWER:
PK –
AVE – 10 WATTS

COMMANDS
ON/OFF

DUTY CYCLE
CONTINUOUS DAY/NIGHT

TLM TRANSMISSION
STORE DATA AT 1.5 KBPS

CLOCK/TIME RQMTS
FOR POSITION DETERMINATOR

POSITION DATA RQMTS
YES

POINTING & CONTROL

ACCURACY/STABILITY 50 MRAD
NOT SENSITIVE TO SMALL BOOM OSCILLATIONS

KNOWLEDGE
50 MRAD

POINTING DIRECTION
DETECTOR – NADIR
COOLER – SPACE

REFERENCE
NOT REQUIRED

INTERNAL MOTION
NONE

ENVIRONMENTAL DATA

SUSCEPTIBILITY

RADIO ACTIVITY ON S/C
NO THORIATED ALLOYS, POTTASSIUM, PAINTS, ETC.
SUSCEPTIBLE TO STRONG MAGNETIC FIELDS
GENERATES SOME WEAK MAGNETIC FIELDS FROM PM TUBES.

THERMAL CONTROL

TEMPERATURE

NOTES

SENSOR COOLED BY PASSIVE RADIATOR ~100-110°K
- RADIATOR SURFACE SHOULD POINT TO FREE SPACE

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GAMMA RAY SPECTROMETER

The Gamma Ray Spectrometer data chart, outlining accommodation requirements for the MGO and LGO missions is presented on the facing page. Calibration of the gamma ray spectrometer is performed during cruise by taking data in the stowed, mid, and fully deployed position; and in orbit for the MGO mission, by taking data in the stowed, mid, and fully deployed position.

The Gamma Ray Spectrometer has special radiation shielding problems. Because radioactive emissions from the spacecraft can completely mask the desired planetary signal, the instrument should be placed on a boom to lessen the impact of spacecraft gamma and charged particle radiation. Material shielding is relatively ineffective for gamma radiation. Thus, a long boom, equal to or greater than the maximum spacecraft dimension, is required. Several other questions concerning the gamma ray spectrometer must be answered during the study. For example, data rates appear to be high for an instrument that may only be counting a few gamma rays per second from the surface of Mars. Data compression may prove advantageous.

MULTISPECTRAL MAPPER FOR MGO/LGO

CAT. 1

TITLE: MULTISPECTRAL MAPPER

P.I.: TOM McCORD U.HAWAII, BOB CARLSON JPL

SCIENTIFIC OBJECTIVE

GLOBAL MAP OF MINERAL
AND SURFACE COMPOSITION
AND FROST DISTRIBUTION
AS A FUNCTION OF
LOCATION AND SEASON

DIM.

OPTICS: 83 x 37 x 30 CM
ELECTRONICS 20 x 25 x 13 CM

MASS

17 KG

FOV

OPTICS: 4 x 0.2 MRAD – EXCLUSION CONE 30°
COOLER: 180°

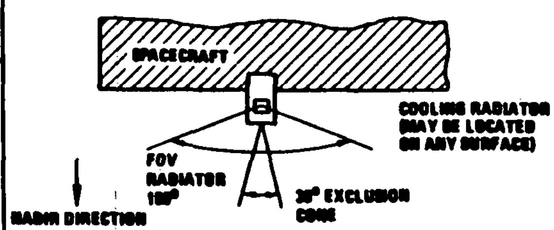
MOUNTING CONSTRAINTS

BUS MOUNTED

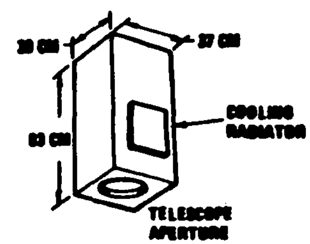
MEASUREMENT

MEASURE SPECTRAL
DISTRIBUTION OF SOLAR
RADIATION DIFFUSELY
REFLECTED FROM THE
SURFACE

FOV (MOUNTING)



SKETCH/DRAWING



GRD. OPS, GSE, FACILITY, HANDLING, TESTING

INFLIGHT CALIBRATE

PERIODIC VIEW OF 2 REFERENCE
TARGETS: ONE REFLECTIVE,
ONE ACTIVE THERMAL

ELECTRICAL

POWER:
PK – 12 WATTS
AVE – 8 WATTS
120 WATTS TRANSIENT

COMMANDS
ON/OFF, SELECT DATA
RATES CALIBRATE,
PURGE, COVER
ON/OFF, COVER
ON/OFF

DUTY CYCLE
DAYLIGHT SIDE OF PLANET ONLY
ASSUME 33%

TLM TRANSMISSION
COLLECT DATA AT 1.5, 3, 6, 12 KBPS

CLOCK/TIME RQMTS
YES

POSITION DATA RQMTS
YES

POINTING & CONTROL

ACCURACY/STABILITY
ACCURACY 3 MRAD
STABILITY 10μ RAD/MIN

KNOWLEDGE
3 MRAD

POINTING DIRECTION
OPTICS NADIR
COOLER SPACE

REFERENCE

INTERNAL MOTION
OPTIC DRIVE, SCANNING

ENVIRONMENTAL DATA

SUSCEPTIBILITY
SENSITIVE TO GAS
AND PARTICULATE
CONTAMINANTS ON
OPTICS AND THERMAL
CONTROL SURFACES

THERMAL CONTROL

TEMPERATURE
SENSOR COOLED BY
PASSIVE RADIATOR
TO 80°K

NOTES
RADIATOR MAY BE
LOCATED ON ANY
SURFACE SO AS TO
VIEW FREE SPACE
WITH A FOV 180°

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OF POOR QUALITY

MULTISPECTRAL MAPPER

The Multispectral Mapper data chart, outlining accommodation requirements for the MGO and LGO missions is presented on the facing page.

Since data will be taken in the visual bands, daylight operation in mid-latitudes will afford the best information. High data rate data will be taken over portions of particular interest only. We have preliminary-sized the system for an average of 6000 bps at 33% duty cycle (i.e., ~2000 bps). Later, however, we reduced this average to 1000 bps to utilize a less expensive data storage unit. The trade-off here is strictly between what is perceived as adequate science versus dollars for more capable communications and data handling equipment.

For the MGO mission, the orbital coverage of the sunlit polar region is extremely poor; especially during the winter season. The proposal shows the observable boundaries around the north pole for instruments requiring solar illumination. It is possible to alleviate the situation by designing the optical system of the MSM to gather data in the polar region. This may imply a heavier instrument In this proposal we have assumed that the multispectral mapper obtains data throughout the sunlit portion (nadir-sunline angle $\leq$ 90 degrees).

RADAR FOR MGO/LGO
CAT. 1

TITLE: RADAR ALTIMETER

P.I.: S. SAUNDERS, C. ELACHI JPL

SCIENTIFIC OBJECTIVE

OBTAIN HIGH RESOLUTION ALTIMETRY DATA

MEASUREMENT

MEASURE LASER PULSES REFLECTED OFF LUNAR SURFACE

GRD. OPS, GSE, FACILITY, HANDLING, TESTING

INFLIGHT CALIBRATE

INTERNAL AUTOMATIC OR ON COMMAND

DIM.

ANTENNA 120 x 120 x 5 CM
ELECTRONICS 120 x 60 x 10 CM

MASS

10 KG

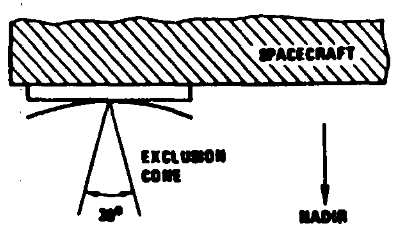
FOV

2°
EXCLUSION CONE 30°

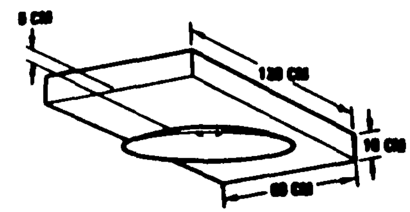
MOUNTING CONSTRAINTS

BUS

FOV (MOUNTING)



SKETCH/DRAWING



ELECTRICAL

POWER:
PK –
AVE – 10 WATTS

COMMANDS

DUTY CYCLE

CONTINUOUS

TLM TRANSMISSION

COLLECT DATA AT 0.6 KBPS

CLOCK/TIME RQMTS

POSITION DATA RQMTS

POINTING & CONTROL

ACCURACY/STABILITY

ACCURACY 30 MRAD
STABILITY 100μ RAD/MIN

KNOWLEDGE

30 MRAD

POINTING DIRECTION

ANTENNA NADIR

REFERENCE

INTERNAL MOTION

NO MOVING PARTS

ENVIRONMENTAL DATA

SUSCEPTIBILITY

THERMAL CONTROL

TEMPERATURE

NOTES

NO COOLING REQUIRED

ORIGINAL PAGE IS OF POOR QUALITY

MAGNETOMETER

The Radar experiment data chart, outlining accommodation requirements for the MGO and LGO missions is presented on the facing page.

The pointing accuracy for the radar altimeter is based on a range resolution requirement of 25 meters. If a finer range resolution is required, the attitude control and stability requirements will again be refined. Note, that with the proposed spacecraft concept which has an attitude control system of ~ 0.2 degrees, it is possible to improve the range resolution of the radar altimeter instrument by more than a factor of 10 without impacting the TRW design approach.

MAGNETOMETER

CAT. 1, MGO - CAT. 2, LGO

TITLE: MAGNETOMETER

P.I.: C. SONNETT U. ARIZONA; C. RUSSEL UCLA

SCIENTIFIC OBJECTIVE

MAP SURFACE FIELD, DETERMINE PERMANENT DIPOLE AND MULTI-POLE MOMENTS AND SCAN THE DEEP INTERIOR

DIM.

SENSOR 5 x 5 x 5 CM
ELECTRONICS 22 x 11 x 15 CM

MASS

SENSOR 1 KG
ELECTRONICS 2 KG

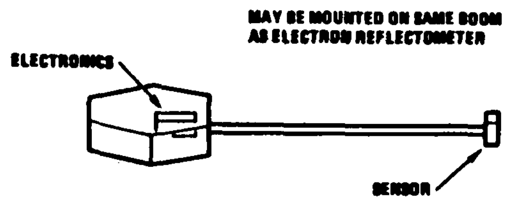
FOV

N.A.
(ORTHOGONAL SENSORS)

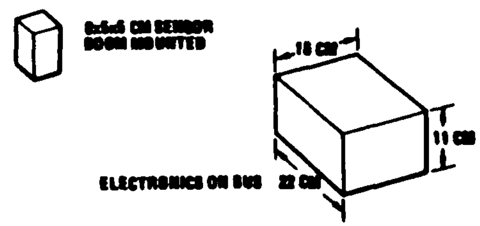
MOUNTING CONSTRAINTS

SENSOR ON BOOM ~ 3 S/C DIAMETERS
ELECTRONICS ON BUS

FOV (MOUNTING)



SKETCH/DRAWING



MEASUREMENT

MEASURE MAGNETIC FIELD VECTOR AS A FUNCTION OF LOCATION AND TIME

GRD. OPS, GSE, FACILITY, HANDLING, TESTING

GROUND CALIBRATION OF CONTRIBUTION OF S/C

INFLIGHT CALIBRATE

ELECTRICAL

POWER:
PK –
AVE – 4 WATTS

COMMANDS

DUTY CYCLE
CONTINUOUS

TLM TRANSMISSION
STORE DATA AT 0.4 KBPS

CLOCK/TIME RQMTS
YES

POSITION DATA RQMTS
YES

POINTING & CONTROL

ACCURACY/STABILITY

KNOWLEDGE
20 MRAD

POINTING DIRECTION

REFERENCE

INTERNAL MOTION
NONE

ENVIRONMENTAL DATA

SUSCEPTIBILITY
NO S/C MAGNETIC FIELDS > 0.01 GAMMA AT SENSOR

THERMAL CONTROL

TEMPERATURE

NOTES
NO COOLING REQUIRED

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OF POOR QUALITY

MAGNETOMETER

The Magnetometer experiment data chart, outlining accommodation requirements for the MGO and LGO missions is presented on the facing page. For the LGO mission, the magnetometer is a lower priority instrument (Category 2), however it is easily accommodated by the FLTSATCOM derived configuration. In order to assure that the spacecraft magnetic field at the magnetometer is $\leq$ 0.01 gamma, a minimal magnetic control program would be required for the spacecraft.

For the magnetometer, the spacecraft's magnetic field is typically much larger than the fields to be measured (particularly for Mars where no magnetic field mapping has been performed). The magnetometer is mounted on a long boom to remove it from the effects of spacecraft magnetic fields. In general, magnetometers mounted on booms require attitude knowledge determination $\leq$ 0.5 degrees.

MGO MAGNETIC CLEANLINESS

- FLTSATCOM HAS NO REQUIREMENT FOR MAGNETIC CLEANLINESS; MGO HAS REQUIREMENT
- BY TRIED AND TRUE TEST/COMPENSATION PROGRAM, AND MOUNTING/TEST OF MAGNETOMETER ON BOOM OF SUFFICIENT LENGTH, SCIENTIFIC REQUIREMENT CAN BE READILY ACHIEVED
- APPROACH WILL BE TO COMPENSATE FOR "HARD" FIELD AND MOVE MAGNETOMETER SUFFICIENTLY FAR FROM S/C BODY
- THIS COMPENSATION PROGRAM WILL CONSIST OF
  - CALCULATION OF MAGNETIC MOMENTS IN BUS AND GFE INSTRUMENTS
  - DEGAUSS IN COIL AVAILABLE FROM GRO PROGRAM
  - MEASURE AND COMPENSATE IN JURY-RIGGED COIL
- APPROACH IS A PRIMARY COST FOR MGO; BUT OPTIONAL FOR CAT. 2 LGO MAGNETOMETER

MAGNETIC CLEANLINESS

The magnetic moment of the FLTSATCOM spacecraft was evaluated in order to determine the magnetic torque imposed on the spacecraft at geosynchronous altitudes. The magnetic moment of FLTSATCOM does not appear to be excessive, however no precautions have been taken to date to minimize the spacecraft magnetic moment. In order to assure that the field at the magnetometer is $\leq$ 0.01 gamma, the field of the spacecraft will be measured, and a magnetic compensation program will be undertaken. This effort will be both analytical and experimental in nature.

EXPERIMENT ACCOMMODATION REQUIREMENT DRIVERS

- MSM (AND LASER ALTIMETER) IS ATTITUDE CONTROL DRIVER
  - POINTING ACCURACY 0.2°
  - POINTING STABILITY 5.7×10^{-4} $^{\circ}$/MIN
- TWO ARE BOOM MOUNTED (GRS, XRS)
- THREE REQUIRE PASSIVE COOLING (GRS, XRS, MSM)
- MSM HAS HIGH DATA RATE CAPABILITY (TO 12KBPS)
- TWO MAY REQUIRE COMMANDABLE APERTURE COVERS (MSM, XRS)

ENHANCED LGO MISSION (CATEGORY II)

- ADD TWO BOOM MOUNTED EXP. (4 TOTAL)
- ADD COMMANDABLE APERTURE (E.R.) (3 TOTAL)
- ADD REQUIREMENT FOR MAGNETIC CLEANLINESS PROGRAM (MAG)
- CHARGE NEUTRALIZATION SURFACE REQUIRED (E.R.)

LGO SCIENCE

The scientific objective of the MGO and LGO is to provide a detailed study of surface characteristics, surface and near-surface composition, volatile location and amount, and internal structure of Mars and the moon, respectively. Four instruments are common to the MGO and the LGO with two additional instruments proposed for the LGO. In addition, an alternate LGO instrument, the laser altimeter, which may replace the radar altimeter is considered.

On the LGO, nadir pointing instruments will map surface chemical and mineralogical composition using gamma-rays, X-rays, and multispectral imagery. A nadir pointing radar altimeter will measure the lunar figure, and provide data on surface profile, roughness, and reflectivity. In addition, if Category 2 instruments are included in the payload, the magnetic fields will be mapped using a magnetometer and electron reflection techniques. The payload requirements for the MGO discussed earlier also apply to the LGO mission since the MGO set of instruments are also included in the LGO mission. However, the low temperature requirements of the detectors of the gamma ray spectrometer and the multispectral mapper become more critical because of increased lunar heating and solar intensity (compared to Mars).

The FLTSATCOM has the potential to meet all LGO instrument requirements.

INSTRUMENT POWER WEIGHT AND DATA REQUIREMENTS

INSTRUMENT	WEIGHT (KG)	POWER AVERAGE (WATTS)	DUTY CYCLE %	DATA COLLECTION RATES PER SECOND (BITS/SEC)	PER DAY (BITS/DAY)
GAMMA RAY SPECTROMETER (GRS)	12	10	100	1.5×10^3	1.3×10^8
X-RAY SPECTROMETER (XRS)	11	10	50	0.3×10^3	0.13×10^8
RADAR ALTIMETER (RADAR)	10	18	100	0.6×10^3	0.5×10^8
MULTISPECTRAL MAPPER (MSM)	17	8	33	$(1.5\text{-}12) \times 10^3$ (AVG 6×10^3)	1.7×10^8
TOTAL	50	46		8.4×10^3	3.6×10^8
ALTERNATE #1 ADD					
MAGNETOMETER (MAG)	3	4	100	0.4×10^3	$.35 \times 10^8$
ELECTRON REFLECTOMETER (E.R)	5	5	100	0.3×10^3	$.26 \times 10^8$
TOTAL (CUM)	58	55		9.1×10^3	4.24×10^8

ALTERNATE #2 NOTE: REPLACING RADAR ALTIMETER BY LASER ALTIMETER HAS MINIMAL IMPACT ON EXPERIMENT ACCOMMODATION REQUIREMENTS (POSSIBLY AN INC. IN DATA RATE)

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LGO MISSION SCIENCE

The experiment accommodation requirement drivers for the LGO mission are in the areas of attitude control, thermal control, data management and communication requirements, and mechanical accommodation. The enhanced LGO mission spacecraft will have four out of six experiments mounted on booms. The requirement for passive cooling on three experiments leads to requirements for large unobstructed FOV's for the passive radiator coolers. Electric, magnetic and particulate cleanliness requirements lead to the requirement of commandable aperture covers (3 total), and the institution of a magnetic control (if magnetometer is added) and electrostatic charge neutralization program.

Spacecraft-produced electromagnetic interference (EMI) is of particular concern for the magnetometer. Standard measures must be taken to assure electromagnetic compatibility (EMC) with the distribution of power and data signals on the spacecraft. Interference can be reduced by properly placing the magnetometer so that locally produced magnetic fields do not dominate the ambient signals.

Contamination from spacecraft outgassing must also be considered in spacecraft design. Instrument location would isolate cooled detectors from the spacecraft so that condensation of outgassed material does not pose any problems. However, gaseous contamination in the line-of-sight must be minimized.

X-RAY SPECTROMETER FOR LGO

CAT. 1

TITLE: X-RAY SPECTROMETER

P.I.: AL METZGER JPL, JACK TROMBKA GSFC

SCIENTIFIC OBJECTIVE
MAP CHEMICAL COMPOSITION OF MOON'S SURFACE

DIM.
20 x 20 x 40 CM

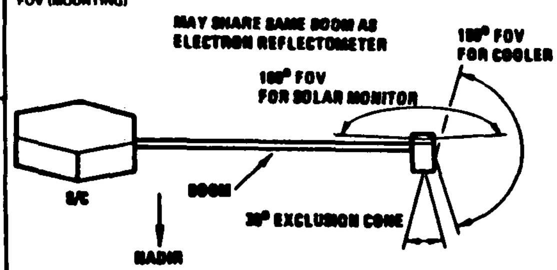
MASS
11 KG (INC. ELECTRONICS & COOLER)

FOV
COLLIMATED 20° ± 10°
SOLAR MONITOR EXCLUSION CONE 180° (2π STER)
COOLER ~150° FOV/SPACE

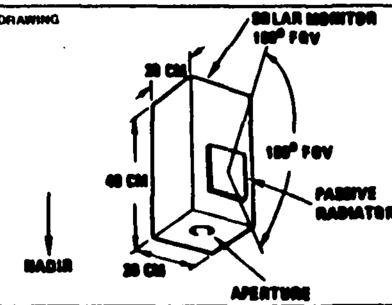
MOUNTING CONSTRAINTS
BOOM MOUNTED – MAY SHARE SAME BOOM AS X-RAY SPECTROMETER

MEASUREMENT
MEASURE SECONDARY X-RAYS PRODUCED WHEN SOLAR X-RAYS INTERACT WITH LUNAR SURFACE

FOV (MOUNTING)



SKETCH/DRAWING



GRD. OPS, GSE, FACILITY, HANDLING, TESTING

INFLIGHT CALIBRATE
ROTATING BAR MOVES IN FRONT OF SENSOR PERIODIC-ALLY WHEN SURFACE IS NOT ILLUMINATED

ELECTRICAL

POWER:
PK –
AVE – 10 WATTS

COMMANDS
ON/OFF
APERTURE # 1 ON/OFF
APERTURE # 2 ON/OFF

DUTY CYCLE
COLLECT DATA ON DAY SIDE ONLY

TLM TRANSMISSION
STORE DATA AT 0.3 KBPS
(2.6×10^7 BITS/DAY)

CLOCK/TIME RQMTS
YES

POSITION DATA RQMTS
YES

POINTING & CONTROL

ACCURACY/STABILITY
30 MRAD: 100 μ RAD/MIN

KNOWLEDGE
30 MRAD

POINTING DIRECTION
DETECTOR – NADIR
COOLER – SPACE
SUN MONITOR – SUN

REFERENCE
NOT REQUIRED

INTERNAL MOTION
COOLER SHIELD
CLOSE/OPEN APERTURE

ENVIRONMENTAL DATA

SUSCEPTIBILITY
SENSORS SENSITIVE TO SUNLIGHT
PASSIVE RADIATOR DEGRADED BY PARTICULATE CONTAMMINATNTS

THERMAL CONTROL

TEMPERATURE
SENSOR COOLED BY PASSIVE RADIATOR ~130°K

NOTES
PASSIVE RADIATOR SHOULD POINT TO FREE SPACE

X-RAY SPECTROMETER

Two instruments on the LGO are sensitive to the direction of the sun with respect to nadir: the X-ray spectrometer and the multispectral mapper. The X-ray spectrometer utilizes the solar X-radiation as the exciting flux and measures the secondary X-ray flux generated on the surface. For a typical instrument, the total sensor FOV is 40 degrees with full width at half maximum of approximately 20 degrees and nearly circular response pattern. In a crossed wire array proportional counter detector, it is possible to achieve an areal resolution of 10 km on the lunar surface. At 100 km altitude, the angular beam at the spacecraft, for a 10 km swath, is 6.0°. If we assume a pointing accuracy control requirement one-tenth the swath size, the pointing control requirement is 0.6 degree.

The X-ray spectrometer instrument will also monitor the sun via an irradiated standard target to make corrections for solar variations. During periods of increased solar activity, the solar X-ray flux increases in intensity and maximum spectral energy. At such times secondary X-rays are produced in the higher atomic number elements, and the abundances of elements such as K and Ca may be determined. The secondary target must view the sun with a FOV of 90 degrees. To meet this requirement, it will be necessary to place the solar X-ray package on one side of the spacecraft and the nadir viewing secondary X-ray detection package on an opposite side.

The areal and temporal coverage of the lunar surface by the X-ray spectrometer is constrained by the requirement that the angles between the sun line and the instrument nadir be $\leq$ 60 degrees. There is a region, emcompassing approximately 30 degrees around the lunar polar cap which will not meet the nadir-sun line angular requirement of $\leq$ 60 degrees. This region corresponds to approximately 6.8 percent of the lunar surface at each of the north and south poles. A discussion of orbit versus lunar surface coverage under a variety of conditions was presented in the proposal.

ELECTRON REFLECTOMETER FOR LGO

CAT. 2

TITLE: ELECTRON REFLECTOMETER

P.I.: K. ANDERSON, BOB LIN UC BERKELEY

SCIENTIFIC OBJECTIVE

IN CONJUNCTION WITH MAGNETOMETER MAP MAGNETIC FEATURES OVER ENTIRE LUNAR SURFACE

DIM.

2 SENSOR CYLINDERS 15 CM DIAMETER x 9 CM LONG
ELECTRONICS 20 x 20 x 10 CM

MASS

5 KG

FOV

CROSSED FAN
5° x 300° EACH SENSOR IN NADIR–ZENITH PLANE

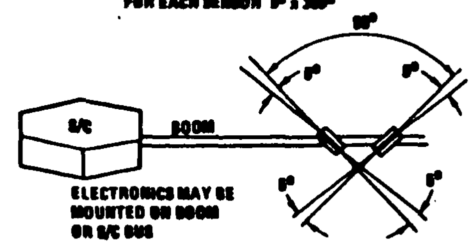
MOUNTING CONSTRAINTS

MAY BE MOUNTED AS SINGLE INTEGRATED PACKAGE OR SEPARATELY. BOOM PERFERRED (FOR SENSORS). BUS OK, BUT AWAY FROM NON-CONDUCTING SURFACES.

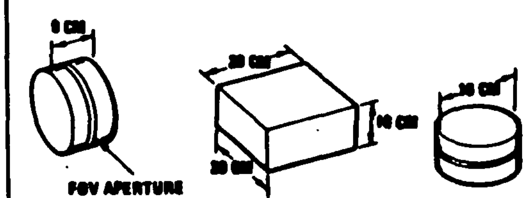
MEASUREMENT

MEASURE ENERGY SPECTRUM AND ANGLUAR DISTRIBUTION OF SOLAR, AMBIENT, AND TERRESTRIAL IONS AND ELECTRONICS IN THE 10 EV TO 20 KEV ENERGY RANGE

FOV (MOUNTING)



SKETCH/DRAWING



GRD. OPS, GSE, FACILITY, HANDLING, TESTING

NONE

INFLIGHT CALIBRATE

ELECTRICAL

POWER:
PK –
AVE – 5 WATTS

COMMANDS
ON/OFF
APERTURE COVER ON/OFF

DUTY CYCLE
CONTINUOUS

TLM TRANSMISSION
STORE AT 0.3 KBPS

CLOCK/TIME RQMTS
YES

POSITION DATA RQMTS
YES

POINTING & CONTROL

ACCURACY/STABILITY
ACCURACY: 30 MRAD
STABILITY: 100 μRAD/MIN

KNOWLEDGE
30 MRAD

POINTING DIRECTION
CROSSED FAN
IN NADIR-ZENITH PLANE

REFERENCE

INTERNAL MOTION
NO MOVING PARTS

ENVIRONMENTAL DATA

SUSCEPTIBILITY
SENSITIVE TO S/C ELECTROSTATIC CHARGING.
SENSITIVE TO S/C MAGNETIC FIELDS.
SENSITIVE TO PARTICULATES CONTAMINATION.
(APERATURE COVER INCLUDED WITH SENSORS).

THERMAL CONTROL

TEMPERATURE
COOLING: NONE

NOTES

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ELECTRON REFLECTOMETER

The electron reflectometer experiment data chart, outlining accommodation requirements for the LGO mission is presented on the facing page. In order to meet the sensors FOV requirements, (crossed fans of 5° x 360°) it is necessary that the sensors be boom mounted. The electron reflectometer instrument is a lower priority experiment (Category 2) on LGO. In order to obtain useful data from the electron reflectometer, a magnetometer (also Category 2 instrument) must be included in the payload complement.

LASER ALTIMETER FOR LGO
CAT. 2 (REPLACES RADAR)

TITLE: LASER ALTIMETER

P.I.: M. KOBRICK, C. ELACHI JPL

SCIENTIFIC OBJECTIVE
MEASURE SURFACE PROFILE ROUGHNESS, REFLECTIVITY, AND SATELLITE ALTITUDE AS A FUNCTION OF LOCATION

MEASUREMENT
MEASURE RADAR PULSE REFLECTED OFF LUNAR SURFACES ALONG ORBITER GROUND TRACK

GRD. OPS, GSE, FACILITY, HANDLING, TESTING
NONE

INFLIGHT CALIBRATE

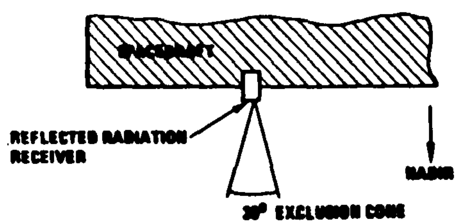
DIM.
40 x 20 x 20 CM

MASS
10 KG

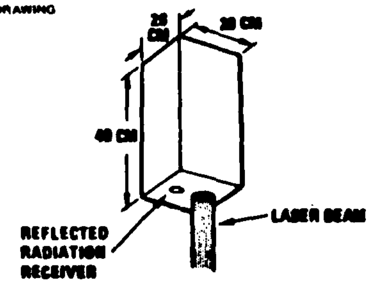
FOV
5 MRAD
EXCLUSION CONE 30°

MOUNTING CONSTRAINTS
BUS

FOV (MOUNTING)



SKETCH/DRAWING



ELECTRICAL

POWER:
PK –
AVE – 10 WATTS

COMMANDS
ON/OFF

DUTY CYCLE
CONTINUOUS

TLM TRANSMISSION
COLLECT DATA AT 10 KBPS

CLOCK/TIME RQMTS

POSITION DATA RQMTS

POINTING & CONTROL

ACCURACY/STABILITY
ACCURACY: 3MRAD
STABILITY: 10μ RAD/MIN

KNOWLEDGE
3 MRAD

POINTING DIRECTION
NADIR

REFERENCE

INTERNAL MOTION
NONE

ENVIRONMENTAL DATA

SUSCEPTIBILITY
NONE

THERMAL CONTROL

TEMPERATURE

NOTES
NO COOLING REQUIRED

LASER ALTIMETER

The Laser Altimeter experiment data chart, outlining accommodation requirements for the LGO mission is presented on the facing page. The laser altimeter is in an early stage of development, however it has the potential of becoming greatly superior to the radar altimeter in measuring surface roughness, reflectivity and satellite altitude as a function of location. It can be accommodated on the LGO spacecraft as easily as the radar altimeter, unless development complications arise. Therefore if the laser altimeter is developed in time for the LGO mission, it will most likely replace the radar altimeter instrument. We have not exhibited concern about this instrument since its development is probably downstream of a 1988 launch, at least.

INITIAL THOUGHTS ON CALIBRATION OF γ-RAY INSTRUMENT (MAY 1982)

- FSC BUS DESIGN MAKES IT NON-COST EFFECTIVE TO CALIBRATE γ-RAY INSTRUMENT WITH FULLY DEPLOYED CONFIGURATION FAR FROM A PLANET
- THE ALTERNATIVE (BASELINE) APPROACH WILL BE TO CALIBRATE THE INSTRUMENT AT 50%, 100%, AND 150% OF ITS ON-ORBIT DISTANCE FROM THE BUS BODY BY EXTENDING IT ON A "STEM" ROLL-UP DEVICE BOOM
- THIS WILL BE ACCOMPLISHED DURING CRUISE-OUT TO MARS AND THE MOON WITH THE BUS SOLAR PANELS IN THEIR FOLDED CONDITION AND WITH THE SPACECRAFT SPINNING AT ~5 RPM
- BALANCE WILL BE MAINTAINED BY EXTENDING A DIAMETRICALLY OPPOSITE BOOM WITH INSTRUMENT(S) (AND, POSSIBLY, BALLAST) AS FAR AS NECESSARY TO MAINTAIN ROLL BALANCE

GAMMA RAY CALIBRATION

JPL scientists Nash and Metzger were still somewhat unhappy about the proposed May 1982 calibration "solution". Consequently, an altered plan was arrived at in August. Basically, it has the following features:

- Taking data during cruise-out at undeployed, deployed, and halfway in between positions
- Taking data on orbit at above 3 positions

The gamma ray instrument is operable in both of these situations, including proper cooling. Moreover, there is sufficient power to take and transmit scientific data almost all the way to Mars, thus enhancing the mission value.

IMPACT OF SCIENTIFIC INSTRUMENTS ON THERMAL DESIGN

- IT WILL BE DIFFICULT TO PROVIDE 80^{o}K HEAT SINK TO MSM IF MOUNTED ON BODY. A PLANETARY AND SUN SHADE(S) WILL BE NECESSARY

- THE γ-RAY AND X-RAY EXPERIMENTS WILL REQUIRE SUN, EARTH, AND SPACECRAFT SHADES

- THE ONLY WAY TO MEET EXPERIMENT COOLING REQUIREMENTS ON LGO IS TO ALWAYS HAVE AVAILABLE A RADIATOR FACING AWAY FROM THE SUN. THIS DICTATES ROTATING THE SPACECRAFT 180^{o} AT LEAST ONCE PER YEAR. THIS REQUIREMENT REDUCED THE LUNAR SOLAR PANEL CONFIGURATION OPTIONS FROM 3 TO THE ONE SATISFACTORY SOLUTION - MAINTAIN THE PANELS AT 45^{o}

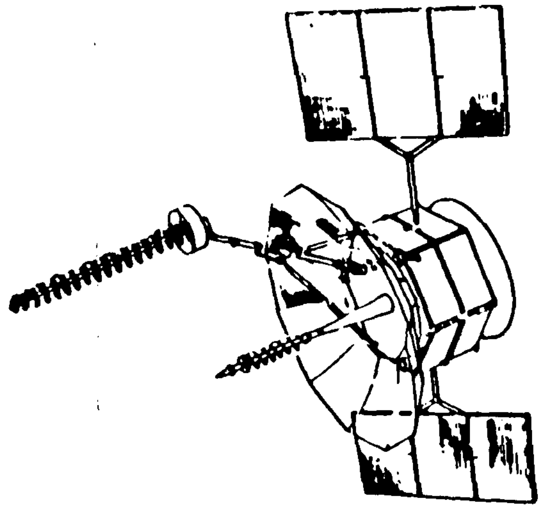
MGO/LGO INSTRUMENT THERMAL DESIGN IMPACT

One concern for this payload is the desire to cool certain detectors to a temperature well below 273°K to suppress thermal noise. The gamma ray spectrometer and multispectral mapper detectors will be cooled with passive radiators that must view deep space and are probably adversely affected by occasional solar viewing. Detector cooler requirements, including power demands and radiator surfaces, will be studied. Duty cycles will also be examined to see if compromises can be made between data taking periods, power consumption, and radiator-look directions.

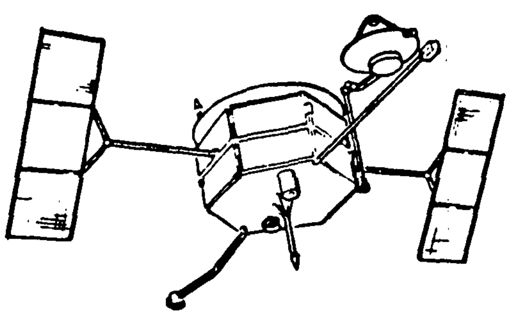
Passive thermal radiators, in order to cool to the $\leq$ 100°k temperature range, require a large FOV into deep space, unobstructed by primary or secondary radiation sources. (The sun, lunar or planetary bodies, or reflecting surfaces). The LGO spacecraft will be rotated 180° to always keep radiator faces away from the sun. Other than this, the FSC installation appears free of thermal problems for all instruments in both MGO and LGO.

SPACECRAFT DESCRIPTIONS

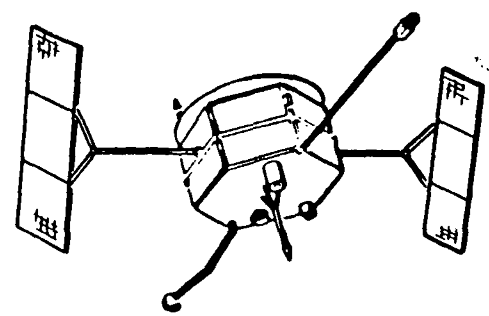
MGO/LGO AND PARENT SPACECRAFT



FLTSATCOM



FLTSATCOM MGO ADAPTATION



FLTSATCOM LGO ADAPTATION

SECTION CONTENTS

Section V consists of five major subsections:

VA - • The FSC Bus

VB - • Major Trade-off Considerations

VC - • Launcher Considerations

VD - • Mission Planning

VE - • MGO/LGO Subsystem Descriptions

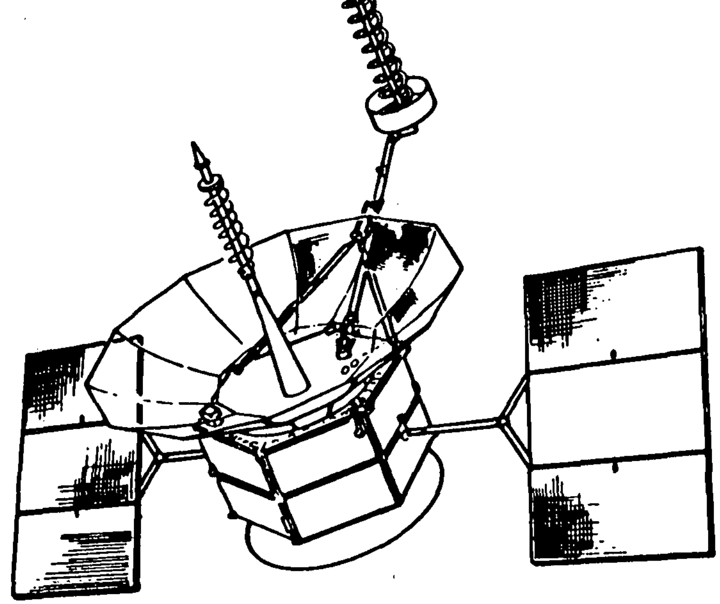
It should be noted that there are two distinct divisions in FLTSATCOM history. Units 1-5 have been completed, and four units are operating at full capacity at the present time. At the time of this report writing (October, 1982), a negotiation for Units 6, 7, and 8 is underway. The latter two units are newer versions; and MGO/LGO will be derived from both old and new designs. FSC 6 will essentially be a copy of FSC 5.

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

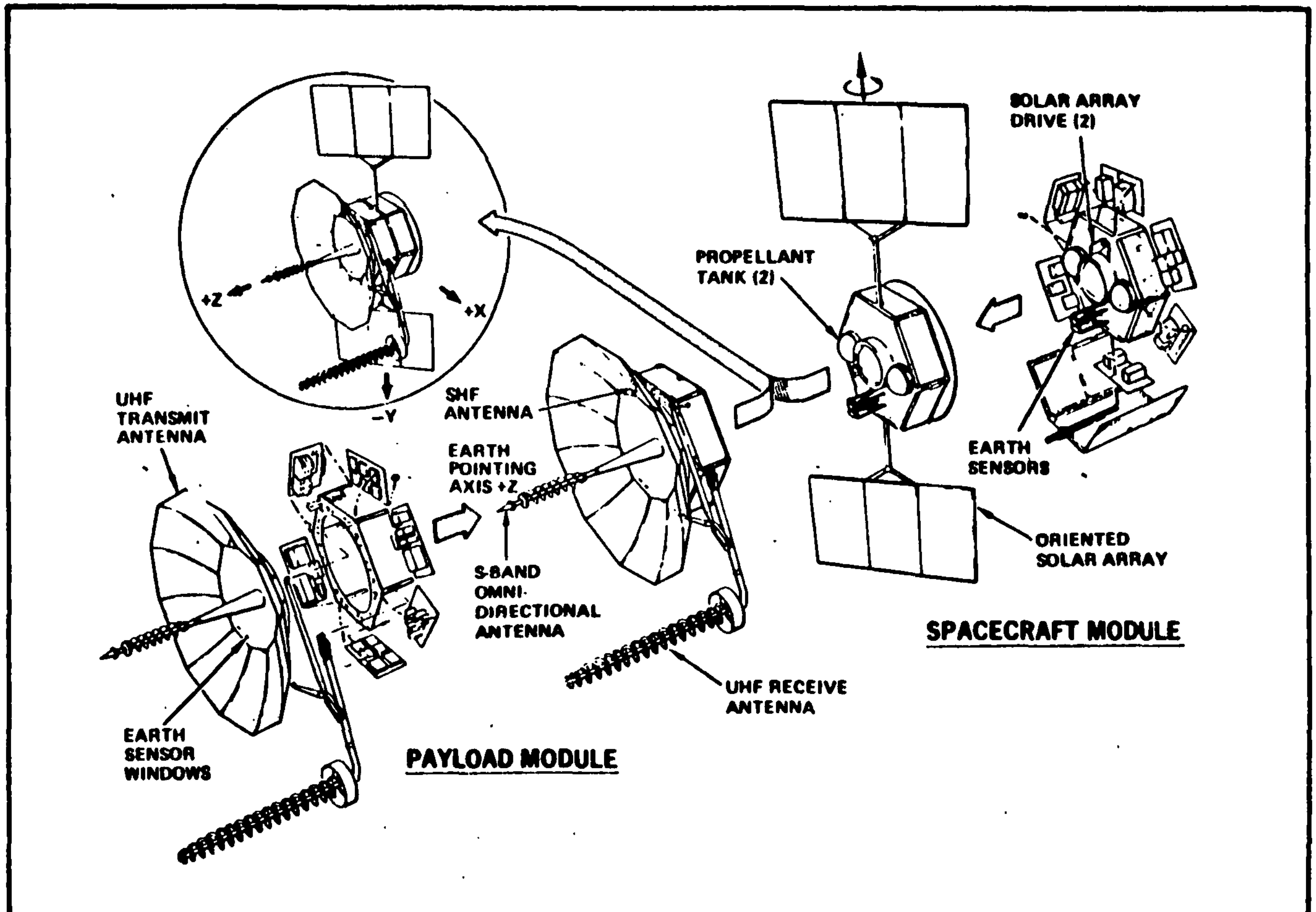


UNITS 1-6

The basic mission of the FLTSATCOM spacecraft is to provide satellite communication capability to improve Navy fleet communications and to make provision for operational use by priority DoD users. The communication links which implement this mission provide 23 separate changes. These 23 communication channels operate at UHF with the exception of Channel 1 which utilizes as X-band uplink. The communication system is a channelized limiting repeater, i.e., the spectrum intercepted at the receive antenna is channelized, amplified, hard-limited, translated in frequency, and radiated via the transmit antenna. The communication antennas on FLTSATCOM are Earth coverage. The total communication payload will be removed for the MGO/LGO buses.

The above shows the spacecraft configuration and some of its general characteristics. The spacecraft's overall size is 43.4 feet between solar panel ends. Electronic equipment is housed in the spacecraft body which is a 7.5 foot hexagonal cylinder. The parabolic transmit antenna is 16 feet in diameter and the receive antenna is an 18-turn helix approximately 12 feet long, mounted to the side of the parabolic transmit antenna. In the launch configuration, the spacecraft is folded to fit within the 10-foot diameter NASA standard Centaur fairing. The spacecraft weighs ~4100 pounds. In this launch configuration, however, approximately half of this weight is the STAR 37F solid rocket that is used for injection into synchronous orbit. The spacecraft is designed to operate in synchronous equatorial orbit and is placed there by the Atlas-Centaur boost vehicle launched eastward from the Eastern Test Range.

SPACECRAFT CONFIGURATION



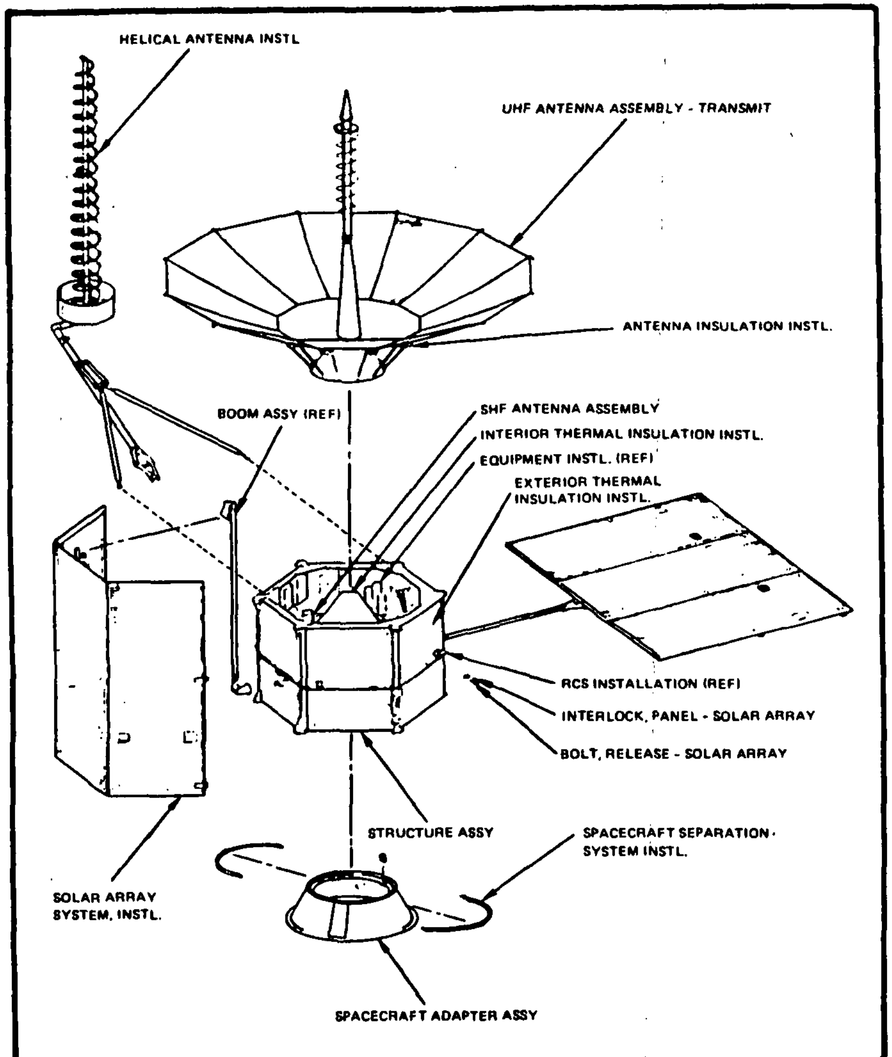
Additional technical features of the spacecraft design are: the spacecraft power budget is slightly over 1,200 watts. This power is provided by 22,632 2 x 4 cm solar cells that generate about 2 kw at beginning of life (BOL) but degrade due to the space environment to 1,200 watts at the 5-year design life point. The spacecraft is attitude controlled on orbit by a combination of a body fixed momentum wheel and direct decomposition hydrazine jets. It points the communication antennas to within a 0.25 degree of nadir. The spacecraft has a design lifetime of 5 years and the design incorporates complete electrical and mechanical redundancy.

Above is an exploded view of the spacecraft configuration. The spacecraft body consists of two compartments that are assembled separately during construction. In the compartment closest to the parabolic transmit antenna is the payload module and houses the payload equipment. The compartment aft of the transmit antenna is the spacecraft module and contains the housekeeping equipment. Each compartment consists of a hexagonal longeron and stringer truss to which honeycomb panels are mounted. The spacecraft electronic equipment is mounted on the inside of the honeycomb panels. The exterior surface of the panels provides thermal radiation area, i.e., heat rejection from the spacecraft. The spacecraft appendages, that is solar panels, the UHF receive antenna, and the UHF transmit antenna, are mounted as shown above.

The modularity features of the FLTSATCOM make it highly desirable as a bus. The spacecraft module remains essentially intact for the MGO/LGO modification. The payload module basic structure is retained, and the communications equipment is replaced by the MGO/LGO science instruments and their support equipment, as well as antenna mounts and booms.

* Unit 6 has different solar cells and 7 year life.

SPACECRAFT STRUCTURAL ASSEMBLY



PRIMARY LOAD PATH	CENTRAL CYLINDER 0.050 INCH MAG 0.7 INCH STIFFENERS
	AKM INSIDE CENTRAL CYLINDER "Z" RING MOUNT
SECONDARY STRUCTURE	HORIZONTAL PLATFORM – PROPELLANT TANKS
	LONGERON AND STRINGER FRAME
EQUIPMENT MOUNTING	EXTERNAL HONEYCOMB PANELS
TRANSMIT ANTENNA	0.750 Aℓ RIBS CRES STAINLESS – Ag COMPOSITE BRAZED MESH
	Cu FACE SHEET HONEYCOMB CENTER DISH
	TUBULAR Aℓ BIFILAR HELICAL FEED
RECEIVE ANTENNA	1.25 INCHES x 0.010 Aℓ RIBBON HELIX
	FIBER GLASS STANDOFFS Aℓ CENTRAL TUBE
	GFRP SUPPORT BOOMS
SOLAR ARRAY	0.625 INCH Aℓ HONEYCOMB 0.005 FACE SHEET

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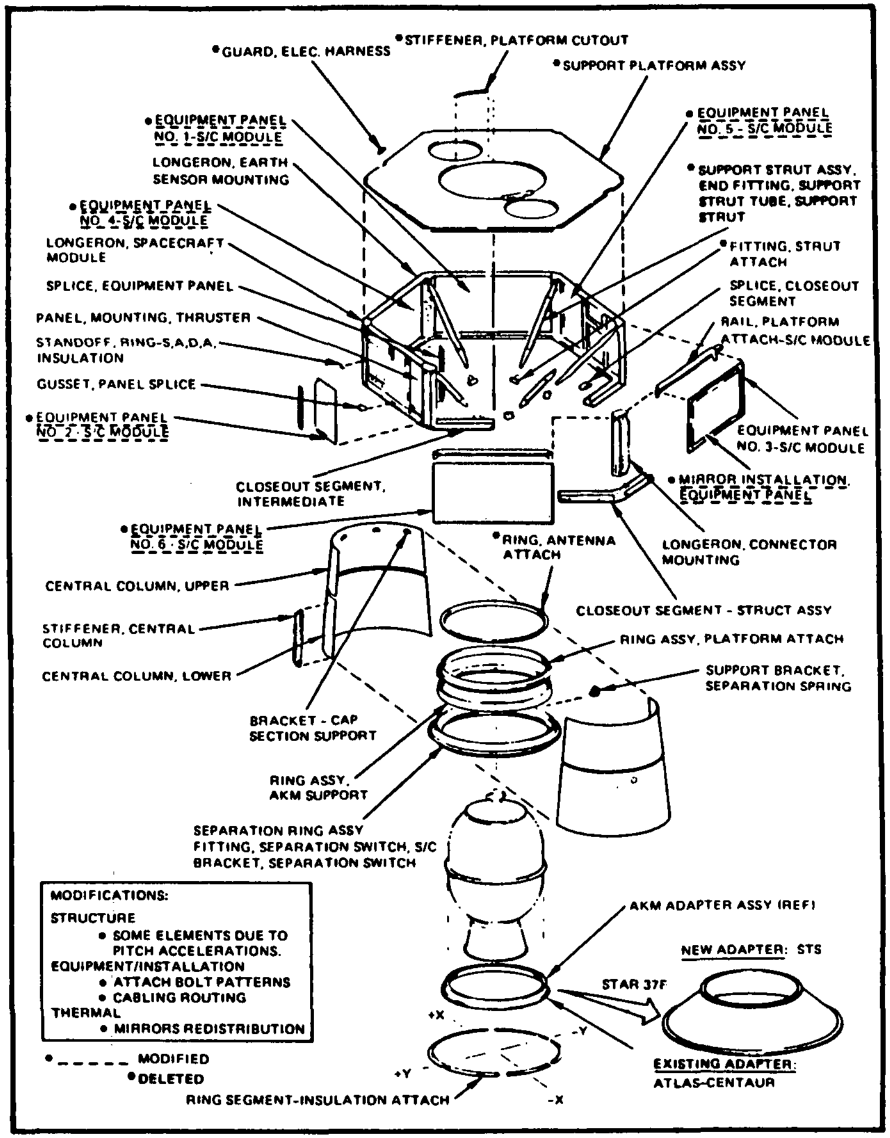
STRUCTURAL DESIGN (1-6)

The principal features of the spacecraft structural design are shown in the exploded view of the spacecraft structure. The principal load path is a central cylinder attached to the boost vehicle by a conical adapter section. The apogee kick motor (AKM) weighs slightly over a ton and is mounted inside this central cylinder attached to the boost vehicle by a conical adapter. This annular Z-ring carries the rocket motor weight during boost, transmits motor thrust during firing, and provides a degree of thermal isolation between the motor casing and the satellite interior.

The solar array substrate is 0.625-inch aluminum honeycomb with 0.005-inch facesheets. The backup structure and solar array booms are rectangular tubular aluminum. Thermal control of the spacecraft is accomplished by radiation of heat directly from the spacecraft external panels. Second surface mirrors (SSM) bonded to the exterior surface of the honeycomb panels provide the required thermal emissivity while reflecting incident solar radiation. The panel areas that are not used as radiators are insulated with multilayer super-insulation.

EQUIPMENT COMPARTMENT STRUCTURAL ASSEMBLY



MORE STRUCTURAL DETAILS

A honeycomb platform mounted at the top of the spacecraft module that carries the two hydrazine tanks. These tanks operate in the blow-down mode, and zero G feed is assured by bladders. Total propellant load capability is about 268 pounds of hydrazine. The longerons and stringers support the external spacecraft panels and are carried by this horizontal equipment panel and diagonal struts. The payload module is of similar construction.

The FLTSATCOM design is easily adaptable because:

- It is spin stabilized in cruise mode without an active nutation control and in this undeployed configuration has ability to continuously communicate with Earth, to provide more than sufficient solar power, to determine its orientation at all times via Earth link, to make midcourse trajectory corrections on command, and to maintain proper thermal control.

- AKM is ideally sized for Martian insertion and this maneuver can be accomplished with suitable Earth coverage by TT&C system.

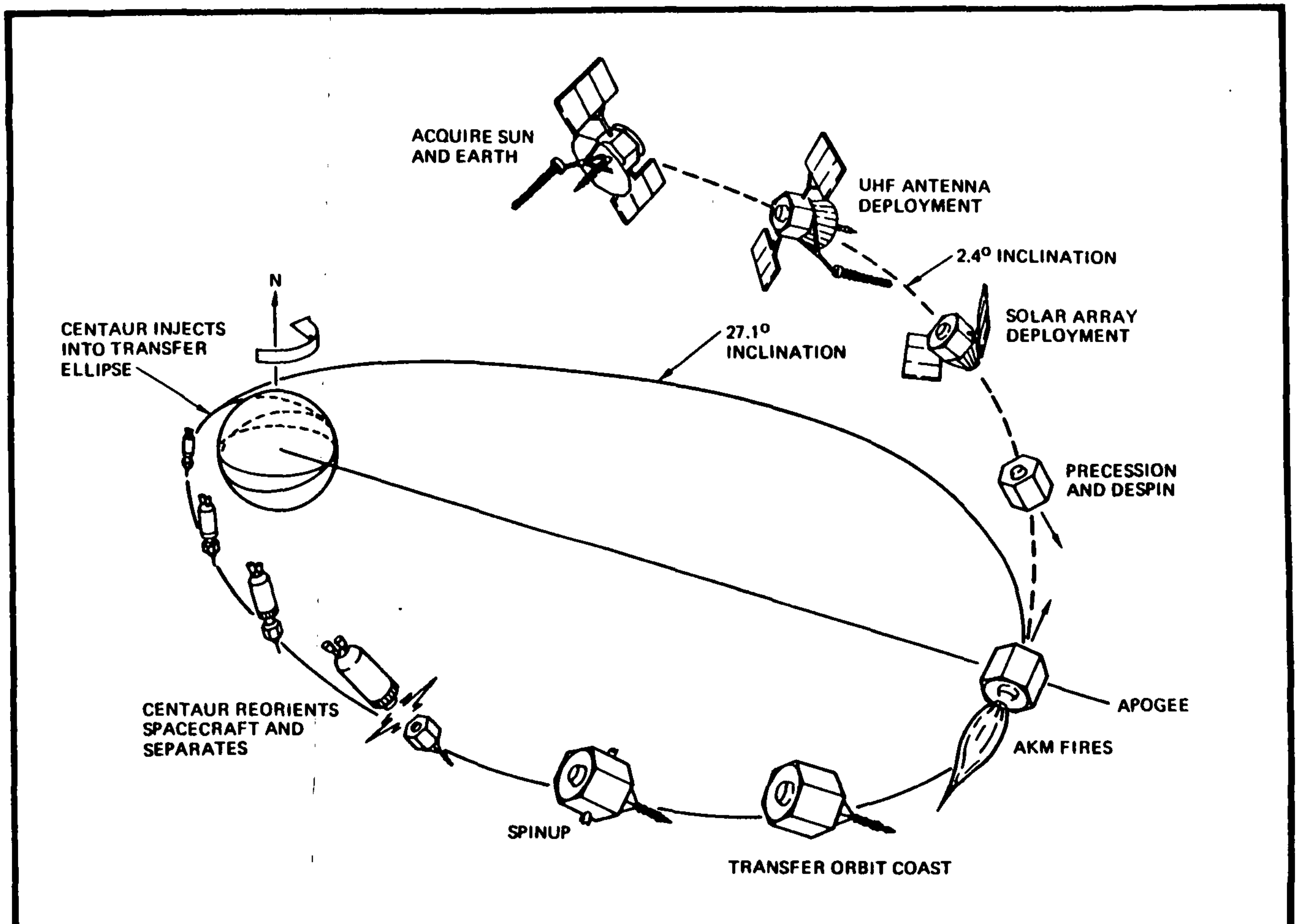
- Thrusters and control inputs are available for orbit trim and plane changes following insertion, either before or after deployment in planetary orbit. FLTSATCOM consumable load appears adequate for MGO/LGO missions.

- Identical Earth nadir pointing transition maneuver to go from spinning cruise configuration to three-axis, biased momentum on-orbit control configuration can be used to establish Martian and Lunar nadir pointing.

SUBSYSTEM	WEIGHT (LB_M)
STRUCTURE	326.4
INTEGRATION HARDWARE	18.5
THERMAL CONTROL	35.8
ELECTRICAL POWER AND DISTRIBUTION	719.1
ATTITUDE AND VELOCITY CONTROL	129.6
COMMUNICATIONS	491.3
TELEMETRY, TRACKING, AND COMMAND	55.7
REACTION CONTROL, DRY	64.7
DRY WEIGHT	1841.1
EXCESS PROPELLANT LOADED (MARGIN)	+46.2
DRY WEIGHT PLUS MARGIN	1887.3
AKM – FIRED CASE	135.3
DRY WEIGHT – IN ORBIT	2022.6
RESIDUAL FLUIDS	7.6
WEIGHT AT END-OF-MISSION	2030.2
RCS EXPENDABLES (REQUIRED)	180.6
AKM EXPENDABLES	1916.7
WEIGHT AT SEPARATION	4127.5
BOOSTER ADAPTER	43.0
WEIGHT AT LAUNCH	4170.5

- On-orbit attitude orientation around Mars and moon will be identical to normal Earth GEO operation. The normally nadir-pointing communication equipment compartment is utilized to hold nadir viewing instruments. Power available as required will be more than adequate in all situations.

FLTSATCOM MISSION PROFILE



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MISSION PROFILE - UNITS 1-8

During transfer orbit and AKM firing, the spacecraft is spin stabilized about its Z-axis. The spin-up is produced by firing the yaw 1-pound thrusters after separation from the Centaur boost vehicle. The orientation of the spin vector in space can be controlled by precessing the spin vector through pulse operation of the roll and pitch 1-pound thrusters. The chart above depicts the launch sequence. The AVCS includes a spinning sun and a spinning Earth sensor that provide attitude data on the position of the spacecraft spin axis and timing data for measurement of spin speed and timing of the precession pulses. When the spacecraft is deployed on orbit, an Earth sensor assembly provides pitch and roll attitude information for normal control. A reaction wheel in the spacecraft provides a momentum component along the Y-body-axis that gives the spacecraft a small amount of gyroscopic stability and precludes the necessity for active yaw control. A sun sensor assembly provides information about the direction of the sun line and this data is used to periodically correct the solar array position and control of the spacecraft during velocity correction maneuvers. As noted previously, the yaw 1-pound thrusters can be used in a thrust aiding mode to impart horizonal velocity to the spacecraft. The equipment is all redundant and capable of extensive cross-connection by ground command.

Once the spacecraft is injected into synchronous orbit, it is deployed and oriented relative to the Earth and the orbit plane. Spacecraft orientation is summarized by the directions forward, south, and down which are the reference directions for the body X, Y, and Z axes, respectively. The receive antenna is located on the +X side of the spacecraft, the solar array axis defines the Y-axis of the spacecraft, and the central axis of the downlink antenna defines the Z spacecraft axis. On orbit, the spacecraft is oriented so that the X-axis is forward in the direction of flight.

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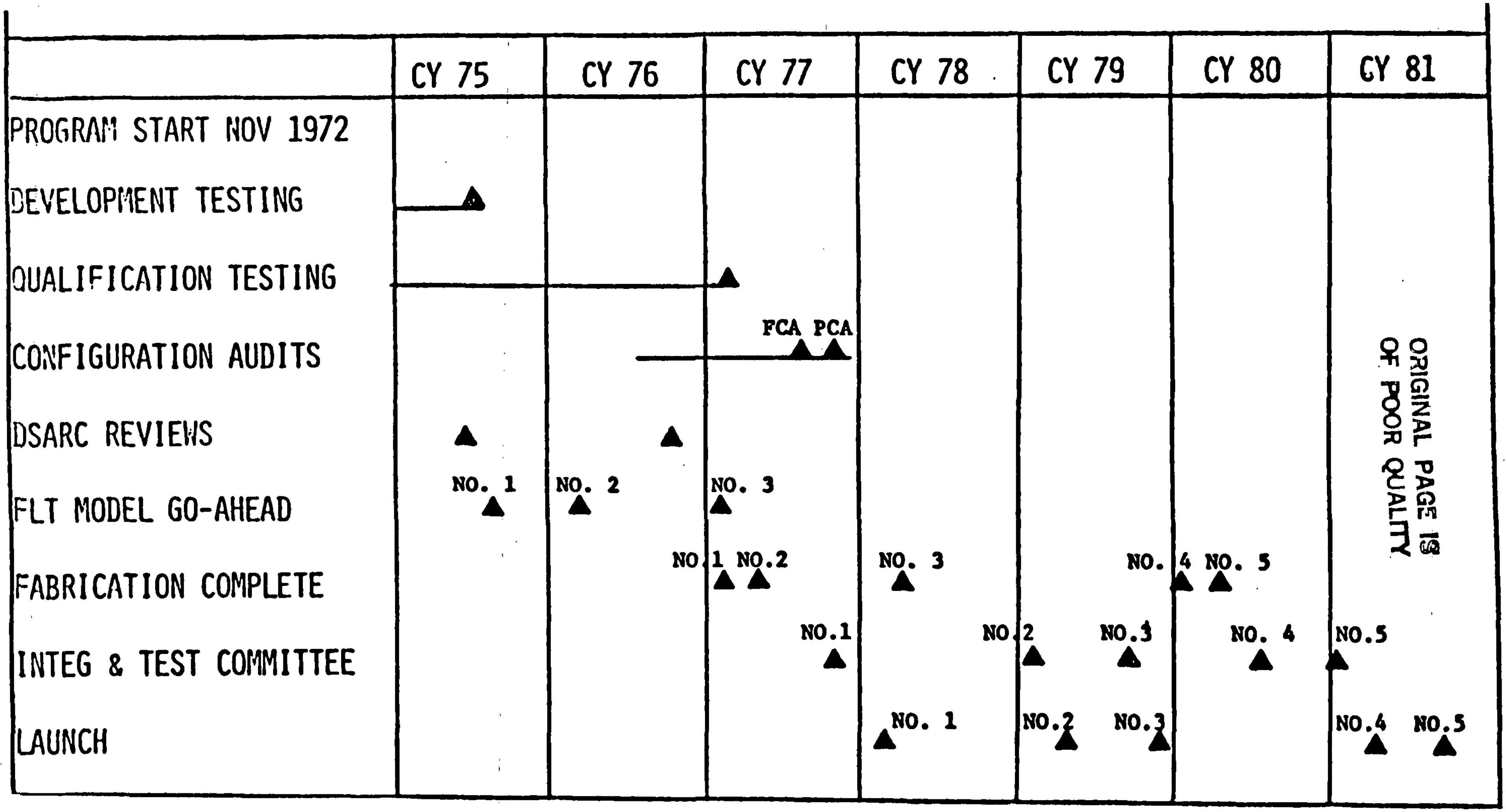
SPACE AND TECHNOLOGY GROUP

MASTER SCHEDULE
FLTSATCOM



DEVELOPMENT SCHEDULE FOR UNITS 1-5

Note that a little over 4 years was required between contract go-ahead and first launch. Over two years were required for testing prior to "buy-off" (see next page). As will be seen later, a compressed schedule is possible for units 6, 7, and 8, due to this earlier learning experience and the availability of the special test equipment required.

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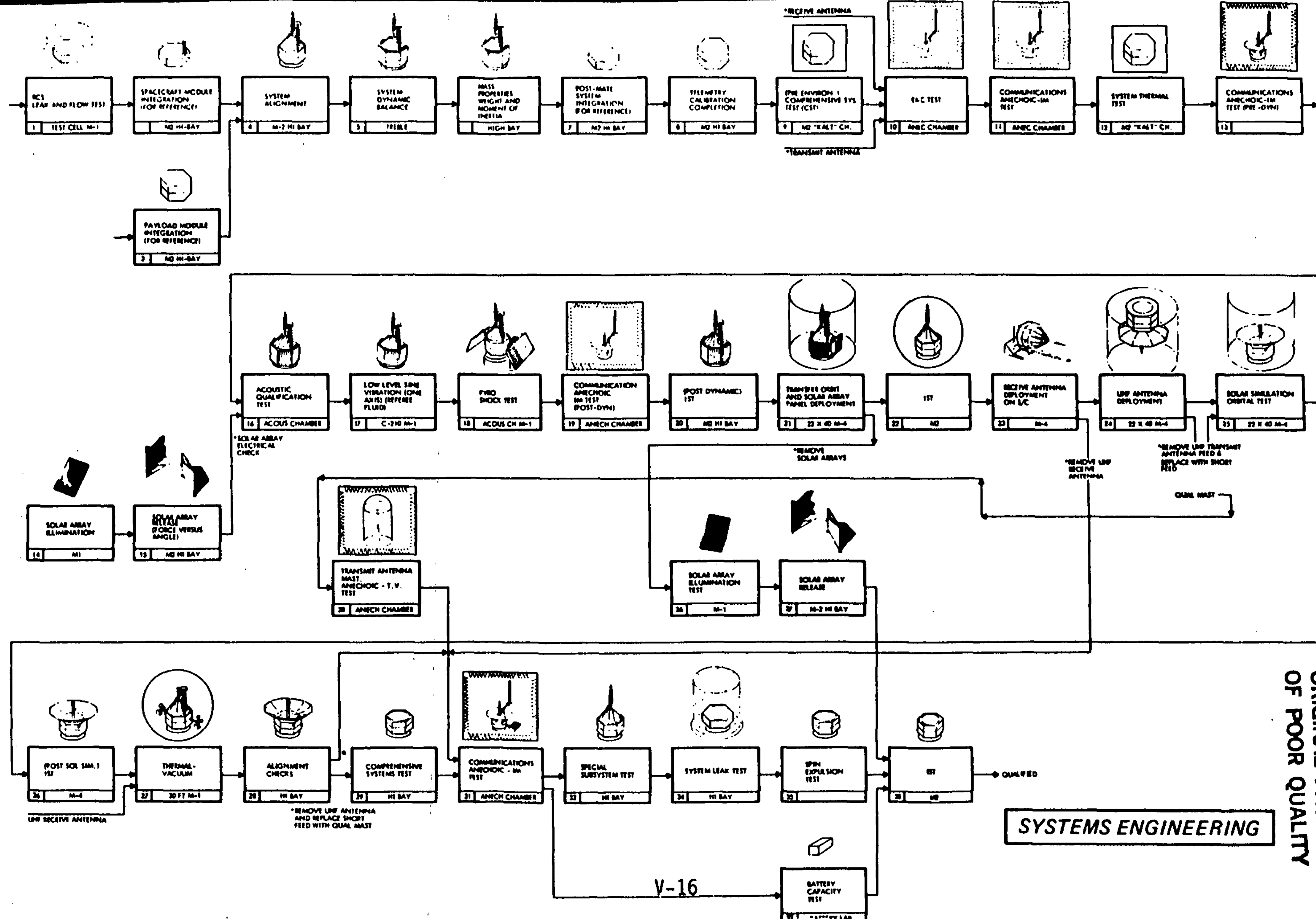
FLTSATCOM QUALIFICATION
SPACECRAFT TEST FLOW
UNITS 1-6



SYSTEMS ENGINEERING

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QUAL TEST FLOW

The purpose of this chart is to (1) indicate the thoroughness of the test program, and to (2) indicate that a great deal of testing set-up and time is devoted to the multi-channel communications payload. The program for units 7 and 8 will be augmented by tests of the new EHF payload. For MGO/LGO, it is believed that test time and cost can be reduced. Moreover, it will not be necessary to perform the difficult and expensive "hardness" (particularly radiation hardness) tests. However, for MGO, at least, magnetic testing and compensation tests will have to be added as both development and qual/acceptance tests.

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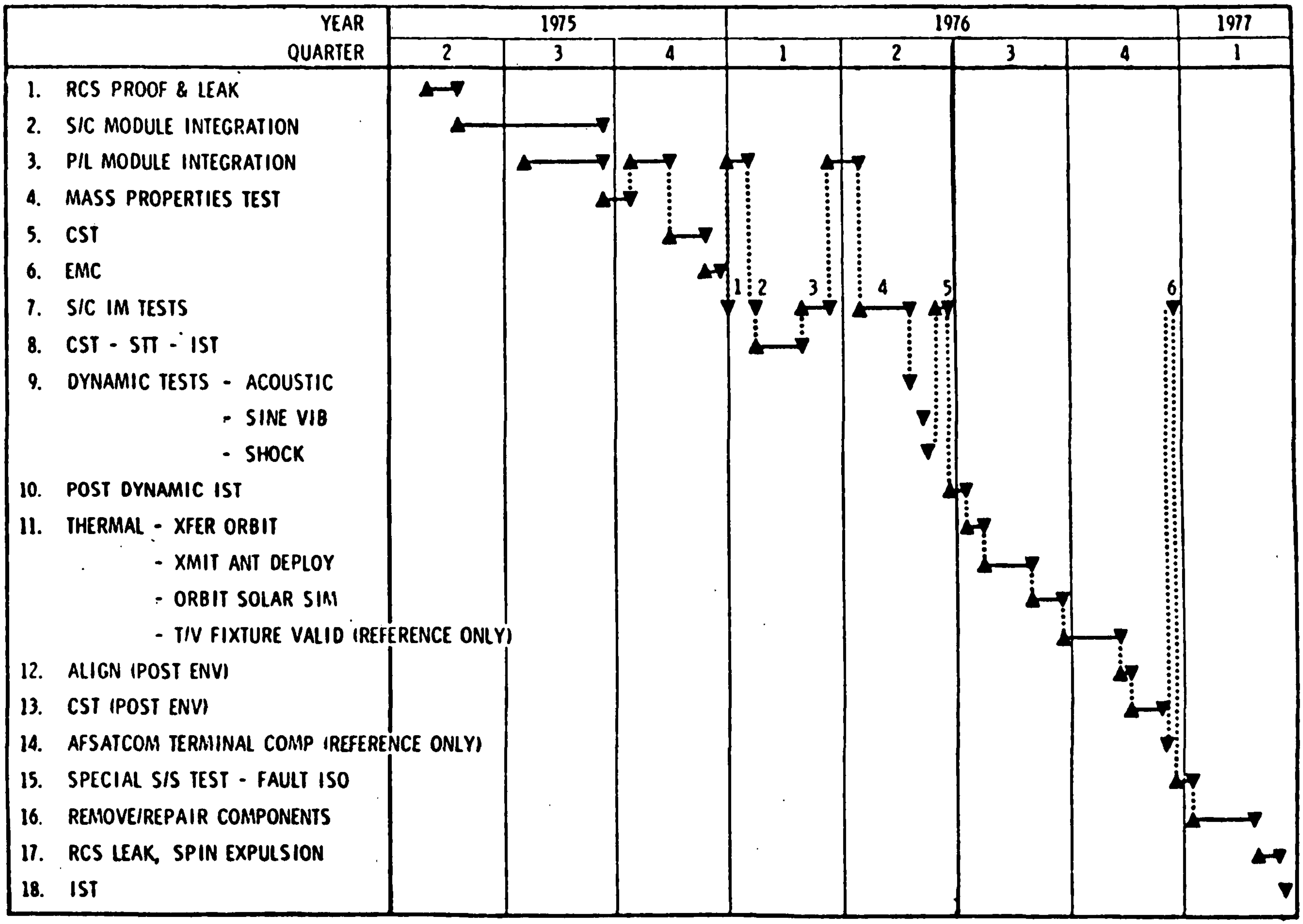
SPACECRAFT QUALIFICATION SCHEDULE



DETAIL PLAN SCHEDULE - UNIT 1

To strengthen the point made on the previous page, note item 7, spacecraft intermod tests, in which 6 different tests were made sequentially. Of course, the "new" C&DH system will require its own set of tests, but the system is considerably less complicated, and will be comprised of "off-the-shelf" components. MGO (and possibly LGO) will also require special tests for compatibility with the STS and a booster vehicle. In all, however, testing time should be reduced.

If the MGO/LGO program is interleaved with the FLTSATCOM program, it will be possible to obtain optimum usage of the various work stations, test set-ups, and crews, thus permitting more efficient operation.

FLTSATCOM RELIABILITY

- DESIGN FEATURES
  - NO SINGLE POINT FAILURE (EXCEPT FOR STRUCTURE) WILL COMPROMISE THE MISSION
  - ALL MAJOR SYSTEMS AND COMPONENTS ARE REDUNDANT

- FLIGHT EXPERIENCE
  - 4 SPACECRAFT NOW IN 100% OPERATION IN GEO
  - FIRST SPACECRAFT LAUNCHED IN 1978
  - TO DATE, 2 SWITCHES MADE TO REDUNDANT EQUIPMENT

- DESIGN LIFE
  - 5 YEARS (7 YEARS FOR EXPENDABLES)
  - RELIABILITY AT 5 YEARS (BUS ONLY) = 0.866

  } ORIGINAL ESTIMATES (SEE BELOW)

A GREAT PERFORMANCE RECORD

The first FLTSATCOM flew in 1978. At this date, there are four spacecraft that are all 100 percent in operation in geosynchronous orbit. There have been a total of two switches to redundant equipment. In the summer of 1981, a fifth unit was launched and achieved orbit. However, during launch the payload shroud inner skin imploded and damaged the spacecraft antenna essentially rendering the spacecraft inoperative.

As a result of flight performance, it has been possible to favorably reassess the intrinsic reliability; viz:

TRW
DEFENSE AND SPACE SYSTEMS GROUP
ONE SPACE PARK • REDONDO BEACH • CALIFORNIA 90278

INTEROFFICE CORRESPONDENCE

FSCOS-81-350-006

TO: P. S. Melancon CC: DATE: 17 February 1981

SUBJECT: Update of Extended Service Life for the FLTSATCOM Satellite System

FROM: T. F. Castle

BLDG.	MAIL STA.	EXT.
R5	2161	64260

"CONCLUSION

Based on all the data gathered to date, there are no expendable or wearout items which indicate a mission truncation of less than ten years. Analysis, to date, of the hardware shows demonstrated and/or calculated life spans well in excess of ten years. Data gathered for these analyses, without exception, substantiate rather than cast doubt on a much longer life span for the FLTSATCOM satellite that was originally forecast.

Statistical analyses (Reference 3) based on the new MIL-HDBK-217C data predict the probability of a totally successful mission at ten years to be on the same order as the old failure rates provided for a five year mission."

FLTSATCOM PROGRAM FACTS

UNITS 6, 7, 8

- WORK ON FSC 6, 7, 8 COMMENCED 1 JANUARY 1982, WITH PERMISSION TO ACQUIRE LONG LEAD ITEMS UP TO $47M
- FSC 6 WILL ESSENTIALLY BE A COPY OF FSC 5 (WITH STAR 37F MOTOR) EXCEPT FOR IMPROVED EFFICIENCY SOLAR CELLS
- FSC 7, 8 WILL BE ~1000 LBS HEAVIER BY REASON OF ADDITION OF EHF PAYLOAD, ADDED AFT MODULE TO CONTAIN PAYLOAD, AND STAR 37FM MOTOR. STRUCTURAL BEEF-UP TO PERMIT MAX HYDRAZINE LOAD (268 LBS)
- FSC 6 IS SCHEDULED FOR LAUNCH ON 1 MAY 1985 (3-1/3 YEARS); FSC 7 ON 1 FEBRUARY 1986 (4 YEARS) AND FSC 8 ON 1 DECEMBER 1986 (~5 YEARS). A 2-MONTH LAUNCH ACTIVITY IS SCHEDULED FOR EACH
- ASSEMBLY, INTEGRATION AND TEST REQUIRES ~2 YEARS
- ONE OR TWO GEOSCIENCE ORBITERS CAN BE INTERLEAVED WITH THIS PROGRAM, ASSUMING AN EARLIEST 1 JANUARY 1985 START

THE NEW FLTSATCOM PROGRAM

In February, 1982, it was announced that the TRW Space and Technology Group had received a $47 million dollar contract from the Air Force's Space Division for procurement of critical long-lead parts and high-technology components for three additional fleet satellite communications system spacecraft.

The three satellites are scheduled to be launched between mid-1985 and early 1987 to fill an envisioned gap in communications services that would have occurred during the mid-1980's.

The FltSatCom units 6, 7, and 8 that are to be built will be somewhat different from Units 1 through 5:

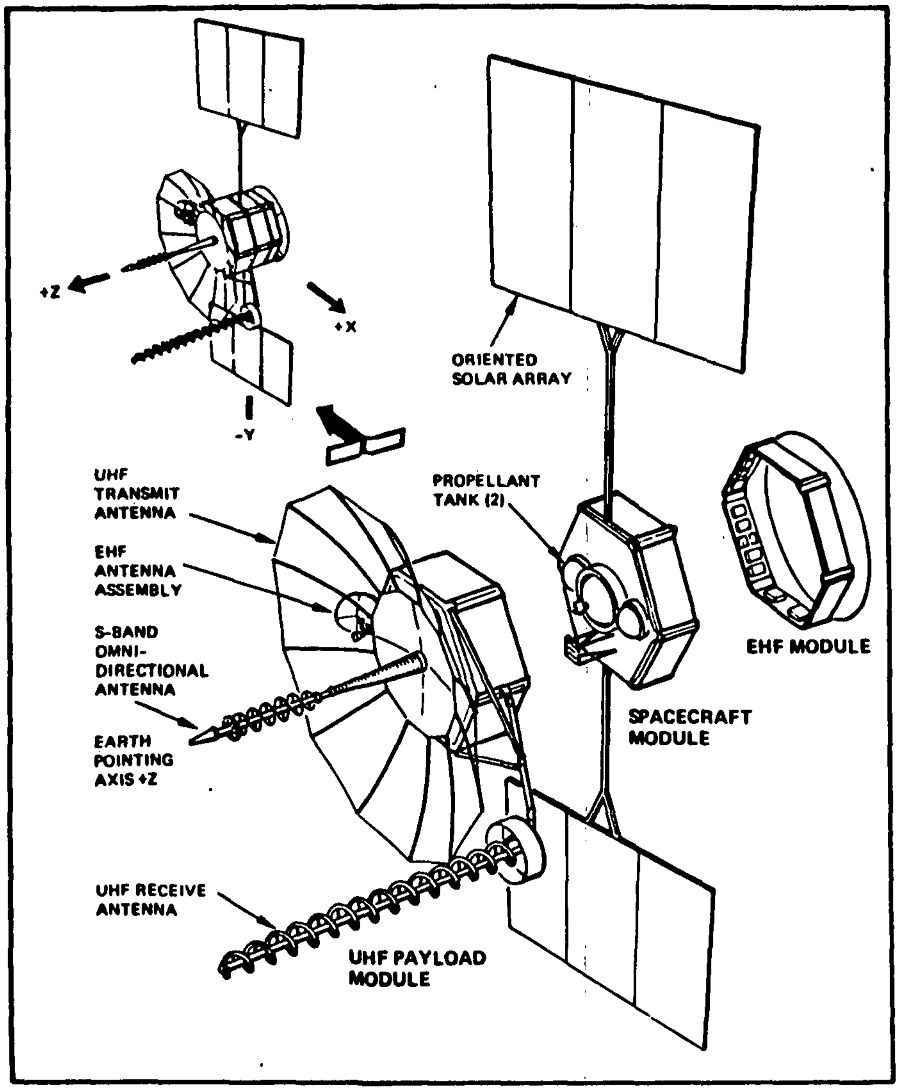
- Unit 6 will be an exact copy of Unit 5 except that the existing solar cells are no longer available and will be replaced by more efficient cells.
- Units 7 and 8 will carry an additional EHF payload necessitating a more powerful AKM and the structure will be modified for both payload and motor accommodation.

The solid motor will change from the Star 37F AKM that is used in Units 1 through 6 to the "new" Star FM motor. The Star 37 FM will have to be qualified by the FltSatCom 7 and 8 programs. Because the energy requirements for the LGO insertion are much less than for the MGO, the less energetic Star 37 N motor could be used as an alternate for the LGO spacecraft, rather than off-loading the Star 37 F or FM motors.

FltSatComs 7 and 8 will thus be heavier than Units 1 through 6 because of the heavier motor; the structural modification necessary to accommodate the motor and other weight increases; and payload addition associated weight increases. Weight increases other than motor increase result in a 415-pound mass addition. Only the spacecraft structure weight increase will be added to the MGO/LGO bus weight. The 415-pound weight increase is tentatively accounted for as follows:

• EHF subsystem/payload	160
• Power control addition and structure	148
• Antenna	35
• Spacecraft structure	72
	415 pounds

FLTSATCOM 7 & 8 CHANGES



FLTSATCOM Subsystems	Required Modifications	MGO	LGO
STRUCTURE	MODIFY ~~CENTER COLUMN~~, ~~SEPARATION RINGS~~, SOLAR ARRAY BOOMS, ~~AND ADD INERTIA WEIGHTS~~	X	X
THERMAL	TURN ON HEATERS DURING TRANSFER ORBIT	X	X
APOGEE KICK MOTOR	USE THIOKOL STAR 37FM	X	
SOLAR ARRAY	USE MORE EFFICIENT SOLAR CELLS (ALSO ON F6). MINOR REWIRING OF CELL CHARGE STRINGS	X	X
ELECTRICAL POWER	MODIFY POWER CONTROL UNIT ~~TO INTERFACE WITH EHF PCU~~	X	X
TT & C	MODIFY TO INTERFACE ~~WITH EHF PAYLOAD INTEGRATION ASSEMBLY~~	X	X
RCS	~~MOVE FOUR SETS OF DUAL THRUSTER MODULES. ADJUST CANT ANGLE OF ΔV THRUSTERS~~		

/ ⇒ NOT NEEDED FOR MGO/LGO

SYSTEMS ENGINEERING

CHANGES FROM FLTSATCOM 1-5 to 6, TO 7 AND 8

In addition to adding the longer STAR 37FM motor to units 7 and 8, the most significant physical changes are the additions of an extra (EHF) module "aft", the EFH antenna assembly and minor solar panel arm changes (which are necessary for c.g. control in the folded, spinning cruise-out configuration) to the spacecraft.

The modifications to FLTSATCOM discussed in this report use Units 7 and 8, sans the EHF module, as the baseline configuration since this is the version that will be in production. The extra length of the motor can be accommodated in the adapter.

Of all the changes, the final selection of the orbit insertion motor (OIM) will have the most impact on the MGO mission. If the longer STAR 37 FM is selected, then an additional performance margin is available that can be used to reduce drift time into the final Martian sun-synchronous orbit and/or open up launch windows and cruise time, if it is used fully loaded.

In the above, the crossed out portions show that, in the structures area, MGO/LGO will use the beefed-up structure, but probably will not need to carry inertia weights, but may require solar array boom shifts. The RCS thrusters should remain as on Units 1-6.

FLTSATCOM 7 & 8
KEY PERFORMANCE PARAMETERS

Weight in Transfer Orbit*	5125 pounds (including spacecraft adapter)	
Transfer Orbit	90 nmi x 19,323 nmi, 27.1 degree inclination	
Apogee ΔV Required	5770 ft/s	
AKM Specific Impulse	292.6 seconds effective	
AKM Expendables	2320 pounds	
AKM Burnout Weight	167 pounds (including S&A and ETA)	
Geosynchronous Orbit Inclination	2.4 degrees	
Service Lifetime	7 years	
Initial Correction ΔV	70.9 ft/s	Total 252 Pounds Hydrazine
Initial Positioning	3 deg/d (56 ft/s)	
Repositioning	7.5 deg/d (140 ft/s)	
East/West Stationkeeping	47 ft/s	

* Atlas - Centaur G Performance Guarantee

COMMENTS ON PERFORMANCE FLTSATCOM 7 AND 8

As shown above, the total ΔV accommodation is 314 ft/sec for a spacecraft whose initial cruise weight is 5125 pounds (compared to MGO's $\approx$ 4000 pounds). Moreover, the full hydrazine capability of 268 pounds is not taken advantage of. MGO's ΔV requirements (discussed later) should be met by this tankage load. Additional MGO propellant could be carried by going to a constant pressure system by adding a pressurant bottle.

Moment of Inertia (MOIR)

The MOIR of primary concern is the ratio of spacecraft spin axis inertia to maximum transverse axis inertia. A design criteria of MOIR $\geq$ 1.05 was used to ensure stability after AKM burn. This was achieved by:

- Relocation aft of the stowed solar arrays closer to the spacecraft center of gravity
- Partial deployment of the solar arrays prior to spin-up (later dropped)
- Addition of inertia weights in spacecraft X-Y plane

The inertia weights were placed at four locations in the spacecraft module so that the spacecraft transverse moments of inertia were equalized and so that the increase in spin inertia was twice the increase in transverse inertia.

If inertia weights turn out to be needed for MGO/LGO, there is no performance problem in adding them.

FLTSATCOM 7 & 8 WEIGHT

Condition	Weight (lbs) 37E
Weight in Transfer Orbit	5125.0
Less Adapter Weight	50.0
Less Nozzle Cover	0.8
Less Spinup and Precession	9.9
Weight at AKM Ignition	5064.3
Less AKM Expendables	2306.0
Weight at AKM Burnout	2758.3
Less Despin and Precession	18.8
Less Acquisition	2.2
Less ΔV Makeup	83.8
Less Initial Position and Correction	47.2
Weight at Beginning of Life	2606.3
Less East/West Stationkeeping (7 years)	17.8
Less Repositioning (7.5 deg/d)	51.7
Less Normal AVCS (7 years)	7.5
Less Reacquisition	2.1
Less Repositioning AVCS	2.3
Less Residuals, Leakage, Pressurant	8.4
	2516.5
Less AKM at Burnout	167.0
Available Dry Weight	2349.5
Less FSC F-4 Actual	1834.0
Available for EHF Modification	515.5
Total Hydrazine Weight	251.7

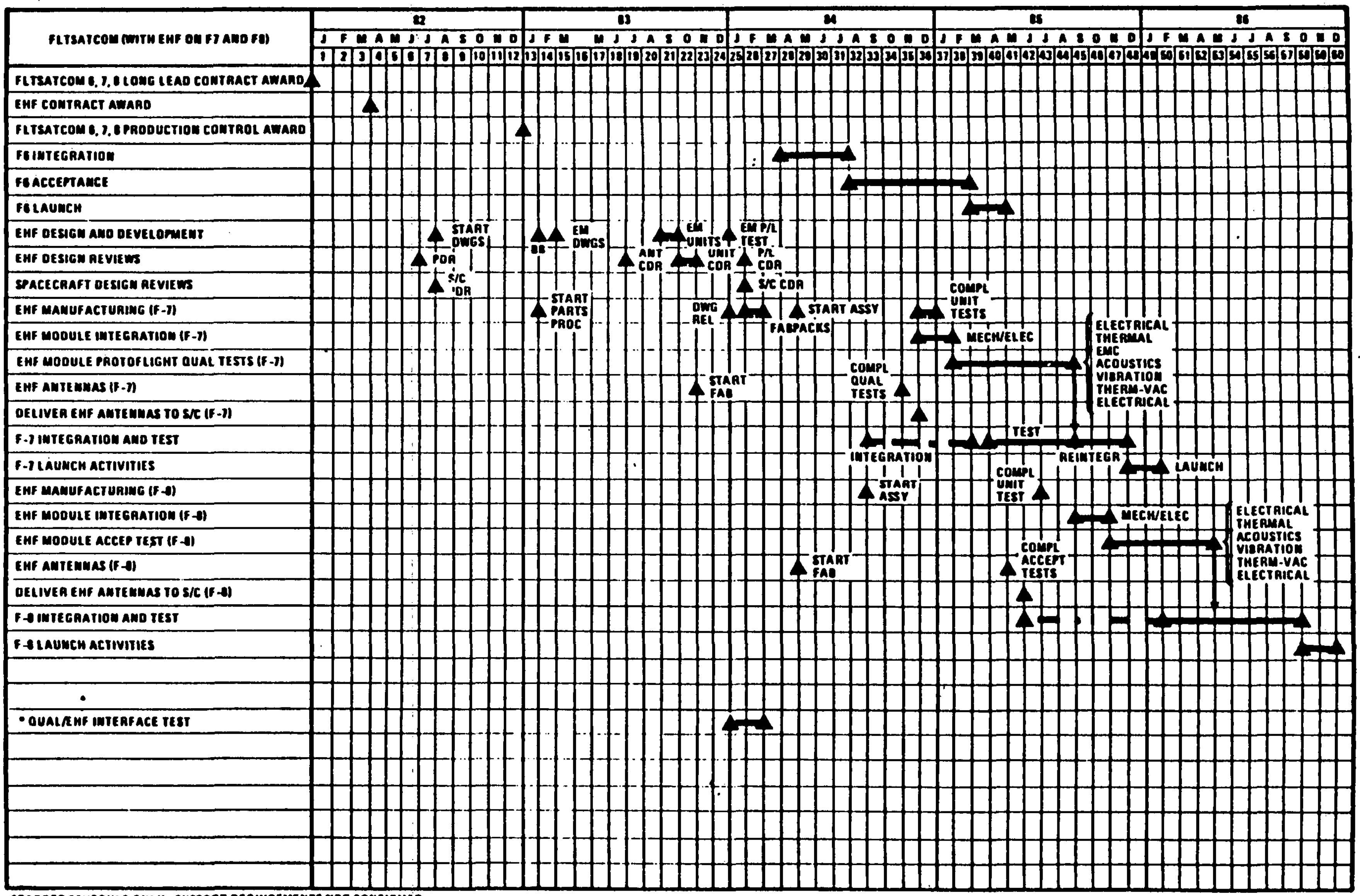
FLTSATCOM 7 AND 8 WEIGHT DIFFERENCES

The above table indicates that the added EHF payload and its support components could weigh up to 515 pounds. About 400 pounds of additional AKM solid propellant is carried to inject the extra payload into GEO orbit.

MGO/LGO will utilize the structural beef-up to accommodate to larger STS lateral loads, and can off-load the Star 37 FM up to ~800 pounds (if higher I_{sp} is desired), or could revert back to the less efficient Star 37 F of Unit 6.

FLTSATCOM 6, 7, & 8 MILESTONES

3 MARCH 1982



*TARGET SCHEDULE ONLY. SUPPORT REQUIREMENTS NOT CONFIRMED.

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THE NEW DELIVERY SCHEDULE

The FLTSATCOM 6, which MGO/LGO will best resemble, will be delivered 3 years and 2 months after long lead item award (01 January 1982). FLTSATCOM 7 is delivered 9 months later. Thus, it is anticipated that if the MGO is accepted $\sim$1 May, 1988 for a July launch, 3½ years will be available to complete the unit, given a 01 January 1985 long lead item start. The FLTSATCOM Program Office, from whom the basic bus would be purchased, believes this schedule could be met. The MGO/LGO units would be interleaved with FLTSATCOM 6, 7, and 8 production.

- THE PROPOSAL, CITING NO CHANGE TO FSC CONFIGURATION, SUGGESTED A 16-MONTH PROGRAM, WITH TWO 2-MONTH SHUT-DOWNS DURING THE PERIOD (WHEN POWER WAS NOT AVAILABLE), WHEN THE SUN WAS NORMAL TO THE LGO ORBIT PLANE ($\Longrightarrow$ NO POWER).

- A SUITABLE ALTERNATIVE, PERMITTING YEAR-ROUND OPERATION, WOULD BE TO UNFOLD THE SOLAR PANELS TO WITHIN 45° OF THEIR NORMAL DEPLOYED POSITION. THIS WOULD REQUIRE TWO 180° FLIPS OF THE SPACECRAFT DURING THE YEAR; THE WAY WE WOULD OPERATE THE EXPERIMENT (i.e., START AT NEAR "FULL" MOON).

- A LESS DESIRABLE ALTERNATIVE (REQUIRING ONLY A 90° ROTATION OF THE MOMENTUM WHEEL CASING) WOULD REQUIRE UTILIZING ALMOST THE COMPLETE FSC SOLAR PANEL AREA (RATHER THAN ~1/2 OF THIS AREA AS REQUIRED BY BOTH MGO AND LGO BASELINES). THIS WOULD ALSO MAKE IT DIFFICULT (WITHOUT MAKING A CUT-OUT IN THE PANELS) TO CALIBRATE THE γ-RAY INSTRUMENT DURING LUNAR CRUISE OUT. HOWEVER, THIS COULD BE AVOIDED BY GOING INTO A HIGHLY ELIPTICAL ORBIT AND CALIBRATING AROUND APOAPSIS. THE LGO COULD CARRY SUFFICIENT HYDRAZINE TO ROUND OUT THE ORBIT.

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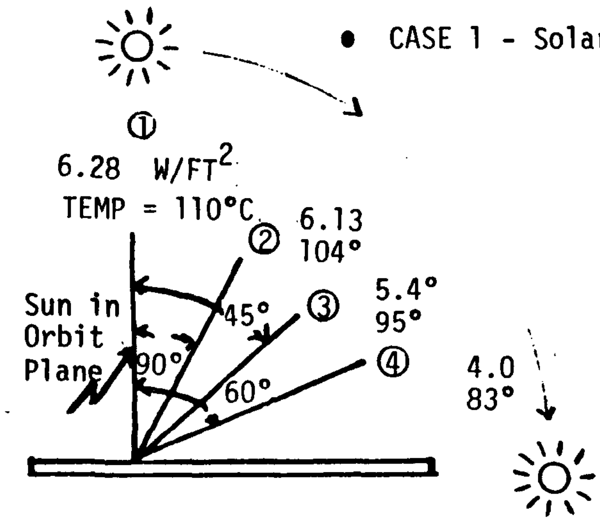
WHY CASE 2 WAS SELECTED

Cases 1 and 2 below, show that power requirements can be met with solar panels ~½ FLTSATCOM area for equivalent results.

- CASE 1 - Solar Panels As Proposed (Same Panel Area As MGO)



	Power Available With 117 Ft.²	Load + 20% Uncertainty For Solar Heating Effects	Excess Power
①	735 W	358 + 70 = 428W	Δ + 307W
②	717	358 + 70 = 428	Δ + 289
③	613	358 + 70 = 428	Δ + 185
④	460	358 + 70 = 428	Δ + 22W

⟵ Sun 90° to Orbit Plane ⟹ No Solar Panel Output

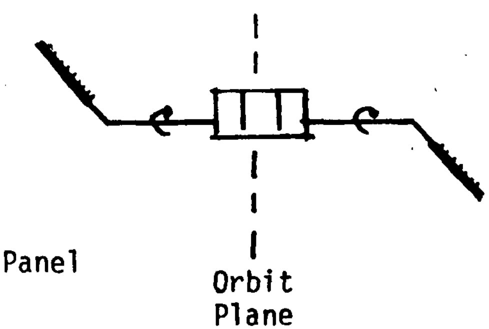
CONCLUSION - Shut Down Total Of $(\frac{120°}{300°})$ = 1/3 of Year ⟹ 16 Month Experiment

- CASE 2 - Bend Solar Panels At 45°

  Then, Power Required = 428 W

  Power Available = 612 X .707 = 430W

  CONCLUSION - This Permits Full Year Operation



- CASE 3 - Rotate Spacecraft 90°

  CONCLUSION - Can Be Done, But Requires 233 Ft.² Solar Panel Same As FLTSATCOM

The case 2 design was selected for two reasons:

1) Several LGO instruments require operation at ~80°K; thus requiring a continuous radiator view of deep space. Because the Case 2 design requires a 180° flip of the spacecraft to maintain power, the radiator side will never "see" the sun.

2) JPL preferred to complete the experiment in one year and not run 16 months.

LGO ATTITUDE SENSOR OPTION

PROBLEM - WHEN SPACECRAFT IS IN LUNAR ECLIPSE, HORIZON SCANNERS ARE INOPERATIVE

- FOR OVER TWO WEEKS PER MONTH IN THIS SITUATION STELLAR/SOLAR OR EARTH/SOLAR SENSING SYSTEMS CAN TRANSMIT REAL TIME POINTING DATA TO EARTH WHILE SPACECRAFT IS IN ECLIPSE

SOLUTIONS

(1) UPON ENTRANCE INTO ECLIPSE, HOLD CONSTANT MOMENTUM WHEEL SPEED UNTIL HORIZON SCANNER(S) PUT OUT PROPER SIGNAL. PERTURBATIONS WILL BE FROM RESIDUAL RCS ACTION PRIOR TO ECLIPSE AND GRAVITY GRADIENT. MAXIMUM NADIR POINTING ERROR WILL BE ABOUT 1^o. <u>THIS IS BASELINE APPROACH</u>. AS NOTED ABOVE, ACTUAL POINTING CAN BE DETERMINED IN REAL TIME AT LEAST 2 WEEKS/ MONTHS, AND STORED POINTING ERROR DATA COULD BE AVAILABLE ALL THE TIME. 45^o SOLAR PANEL BEND COMPLICATES GRAVITY GRADIENT PERTURBATION

(2) IF IT IS DESIRABLE TO MAINTAIN POINTING WITHIN DAYLIGHT LIMITS, IT WOULD BE NECESSARY TO ADD A GYRO PACKAGE AND ASSOCIATED LOGIC. COST ESTIMATES FOR THIS ADDITION HAVE BEEN MADE.

LOSS OF HORIZON SCANNER SIGNAL

The LGO mission poses two AVCS associated problems beyond those encountered by the MGO mission. The first is that existing horizon sensors will not work on the cold dark side of the moon or in the dark side of the terminator. The second is that a sun synchronous polar orbit is not possible on the moon because of its lack of an equatorial bulge (J_2 term of the gravity field). This latter problem has led to the solution of the previous page and the 45° solar panel approach.

The first problem could be solved by adding a gyro package to provide an attitude reference when a horizon is not visible. This solution is expensive (see Volume II) as it makes a major impact on AVCS electronic design. The solution preferred is to "coast" in the absence of a pitch/roll error signal, by maintaining constant momentum wheel speed. The analysis of Appendix A2 (by John Stavlo) shows that unless gravity gradient disturbances are overriding, only a small pointing error should accrue. The asymmetry of the 45° solar panel bend can be precluded by maintaining the panels edge-on to the lunar nadir.

Finally, note that once calibrated, a single horizon scanner can provide the necessary attitude error signals. Thus, the ground, knowing aforehand that one of the scanners will pass through a terminator, can command shut down of that scanner output, and have the AVCS only use the signal from the "sunny side" scanner. This is a normal FLTSATCOM command, thus requiring no system change.

IMPACT OF FLTSATCOM FIXED DESIGN ON MGO/LGO MISSIONS

- THE FSC DESIGN PARTITIONS THE AVAILABLE PROPULSIVE ENERGY BETWEEN THE HIGHLY EFFICIENT (I_{SP} 290$^+$ SECONDS) SOLID ORBITAL INSERTION MOTOR (THE FSC AKM) AND THE LESS EFFICIENT SUPPLY OF HYDRAZINE (I_{SP} 215 SECONDS). THE MGO MISSION DESIGN IS BASED ON TAKING MAXIMUM ADVANTAGE OF THE SOLID MOTOR CAPABILITY.

- THE LGO MISSION IS NON-CRITICAL IN THAT IT REQUIRES BOTH A LESS ENERGETIC INSERTION MOTOR (STAR 37N) AND HAS FEWER DEMANDS ON THE HYDRAZINE SUPPLY, SINCE NEITHER AN ORBIT INCLINATION CHANGE NOR A TERMINAL QUARANTINE BURN IS REQUIRED.

- THE LGO MISSION HAS NOT BEEN STUDIED IN THE SAME DETAIL AS THE MGO, SINCE, FOR THE FLTSATCOM BUS CAPABILITIES, THE MARGINS ARE TOTALLY HEARTWARMING.

- THE MGO MISSION CAN BE ACCOMPLISHED BY THE PRESENT HYDRAZINE SUPPLY. AT ADDITIONAL COST, THE HYDRAZINE SUPPLY CAN BE INCREASED, PERMITTING START OF DATA TAKING VERY SOON AFTER INSERTION INTO MARTIAN ORBIT AND AFFORDING GREATER MARGIN.

CONSERVATION OF HYDRAZINE IN MGO MISSION

Hydrazine may be needed for:

- Spin-up at launch from STS* (solid motors will be used where possible) active nutation control (ANC)*
- Precession of spin axis prior to orbital transfer burn*
- Despin for interplanetary cruise* (yo-yo's will be used where possible)
- Mid-course corrections
- Precession of spin axis to OIM position
- Despin after OIM burn (yo-yo would contaminate)
- Orbit rounding periapsis/apoapsis burns
- Orbit inclination burn(s)*
- On-orbit maintenance
- Final orbit raising/rounding quarantine maneuver burns*

The FLTSATCOM tanks can hold up to 268 pounds of hydrazine without modification while meeting STS safety requirements.

The propellant load can be increased to 342 pounds by adding a pressurant bottle and operating in a constant pressure mode for all but end-of-life operation. Structural beef-up of tank fittings and support structure is required. (See Appendix A4, by Sid Zafran.)

This addition, although minor, represents a cost increase, but enhances STS safety (tanks can be essentially unpressurized during STS launch).

Analyses, to be presented later, indicate that the MGO mission can be accomplished with the FLTSATCOM 7 and 8 propellant load, although margins are not overwhelming.

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* Not needed for Atlas Centaur LGO launch.

MOI/LOI MOTOR TRADE-OFFS

- FOR MGO, - FSC 6 STAR 37F (60", 2056 LBS. $I_{SP} \approx 287.5$) OR FSC STAR 37 FM (66", 2481 LBS. $I_{SP} \approx 292$) COULD BE USED. TENTATIVE CHOICE IS TO USE MORE EFFICIENT MOTOR, SINCE QUALIFICATION IS ASSURED, AND EXTRA LENGTH CAN BE ACCOMODATED IN INTERSTAGE STRUCTURE

  - FULLY LOADED, THE STAR 37FM PROVIDES OVER 1500 FT/SEC EXCESS ΔV, WHICH CAN BE USED TO "CRANK" AROUND THE LINE OF NODES AWAY FROM THE INITIAL ZAP ANGLE AND THIS HASTEN BEGINNING OF DATA TAKING

  - IF THIS IS NOT DONE, THE MOTOR CAN BE OFF-LOADED ~34%, AND THE 800 POUND SAVINGS CAN GO INTO MARGIN, OR TO INCREASE NON-JETTISONED BALLAST WEIGHT, THUS REDUCING DEPLOYABLE OUTRIGGER ARM LENGTH

- FOR LGO, - THE STAR 37N STILL APPEARS TO BE THE BEST CHOICE, ALTHOUGH IT IS OVER-CAPABLE. A SMALLER, LIGHTER MOTOR MIGHT BE FOUND, BUT A NEW FSC ADAPTER WOULD HAVE TO BE DESIGNED

MOTOR CHOICES

The MGO motor choice (Star 37 F or Star 37 FM) may well be dictated by motor availability/qualification.

The Star 37 FM will be qualified by the FLTSATCOM 7 and 8 program. At this point, it is not known if off-loaded versions will also be qualified. If so, it might be advantageous to employ this motor even if its excess energy is not required, as its I_{sp} is several points better than the "old" propellant formulation of the Star 37 F.

On the other hand, there is a dwindling number of old Star 37F motors available - and there have been problems with new motors due to throat cracking. If MGO had to pay for requalification of "new" Star 37 F's, then the choice of the FM version might be more effective. The timing of the procurement will probably settle this issue.

MGO SEPARATION FROM SRM-1 OPTIONS

- AT SRM-1 BURNOUT, DESPIN (ROLL JETS), REACTIVATE ANC AND PERFORM CONVENTIONAL SEPARATION AFTER TAIL-OFF, TRIM MGO ROLL RATE TO 5 RPM AND TURN NORMAL TO ORBIT PLANE
- *DESPIN WITH YO-YO'S ON SRM-1, THEN AS ABOVE**
- DESPIN WITH ONE YO-YO ON SRM-1 AND SIMULTANEOUSLY SEPARATE
- DESPIN AND FIRE SMALL SOLID MOTORS ON SRM-1 TO CLEAR DEPLETED SRM-1 CASE
- PERMIT NUTATION ANGLE TO BUILD UP AND THEN SEPARATE SRM-1 CASE (A LA INSAT), THEN DESPIN MGO AND LET WOBBLE DAMPER REDUCE INHERITED NUTATION

* PREFERRED AS OF NOW, AS MOST CONSERVATIVE APPROACH

MGO SEPARATION FROM UPPER STAGE BOOSTER

The FLTSATCOM has no problem separating from the 3-axis controlled Centaur upper stage, which can release and back away. However, as proposed in this report, MGO could be boosted by a spinning upper stage booster, such as the SRM-1, where the spin rate is substantial (40-60 rpm) during booster motor firing to preclude dispersion due to thrust misalignment, etc.

We wish to avoid approaches that use hydrazine to spin up or down through a large range of rpm's. To avoid dynamic problems of separating two substantial bodies (the spacecraft and the empty motor case), "yo-yo's" or a single "yo" will be used.

Consideration must be given to delaying separation until all residual burning is complete, versus the build-up of nutation angle if ANC is not employed. This study and the ultimate choice is part of a launch vehicle study and thus beyond the scope of the present work.

FLTSATCOM INHERITED POWER S/S COMPONENTS COMPARED TO MGO AND LGO REQTS.

BATTERY SELECTION OPTIONS

SOLAR ARRAY	FLT SAT COM	MGO	LGO	OPTIONS	
AREA	233.5FT^2	116.4FT^2	117FT^2		
POWER AVAIL BOL	2400W	300W (SPINNER)	300W (SPINNER)		
POWER AVAIL 1.67 AU	1084W	508W (DEPLOYED)	470W (DEPLOYED)		
SPIN. PWR REQD		150W	150W		
ORBIT POWER REQD 1.67AU		414 to 450W	390W		
WEIGHT	58 LBS	29.1 LBS.	29.2 LBS.		
BATTERIES	(3) 15AH 24 CELL FLT SAT 6, 7, 8	(3) 15AH 22 CELL (DSCS II)	(3) 15AH 22 CELL (DSCS II)	(2) 24AH 22 CELL (DSP) 5,6	(2) 34AH 22CELL FLTSAT 6,7,8
WEIGHT	240 LBS.	114.4 LBS.	114.4 LBS.	106 LBS.	160 LBS.
AH OUT	~6AH	~6AH	~6AH	~6AH	6AH
DEPTH OF DISCH	6%	13%	13%	12%	9%
1 FAILURE D OF D	9%	20%	20%	25%	18%
CYCLES REQUIRED		8,725	3,515		

BATTERY SELECTION

Once it is determined that the highly over capable batteries of FLTSATCOM 7 and 8 are not needed for "safe haven" mode operation (and, see later, this is a valid utilization), then battery weight can be reduced by a factor of two. Choices are between the types used in our DSCS II program and those that will continue to be delivered to the on-going (through the 1980's) DSP (Defense Support Program) spacecraft. Final selection will depend on cost, availability, conditioning requirements, etc.

LGO CISLUNAR TRANSFER OPTIONS

- ATLAS CENTAUR LAUNCH IS AN OBVIOUS CHOICE, SINCE
  - IT IS NORMAL FLTSATCOM LV - NO CHANGES NECESSARY
  - LV'S WILL REMAIN IN PRODUCTION FOR FLSTATCOM 6, 7, 8, AT MINIMUM
- STS/SRM-1 (MINIMUM PROPELLANT LOAD AND BALLASTED) WOULD BE COST EFFECTIVE IF THERE WERE ALSO AN MGO STS/SRM-1 SPACECRAFT PROGRAM
- STS/PAM-A, ALTHOUGH POSSIBLE, NOT A SERIOUS CONTENDER BECAUSE
  - UNCERTAINTY OF PAM-A AVAILABILITY
  - COST OF STS ADAPTATION ON SPACECRAFT
- OTHER POSSIBILITY FOR COST EFFECTIVENESS MIGHT BE SINGLE STS LAUNCH OF BOTH MGO AND LGO; THEN STS/PAM-A OR STS/SRM-1 WOULD CLEARLY BE A CONTENDER FOR LGO

LAUNCHING LGO

Adaptation of the FLTSATCOM to STS/upper-stage-booster launch presents no apparent technical problems, but will be time consuming and expensive. Either an off-loaded SRM-1 or a PAM-A system could provide upper stage energy.

The FLTSTACOM bus for the LGO mission offers the opportunity for significant launch vehicle cost savings since 1) consumables are not required for an orbit plane change maneuver; 2) midcourse correction ΔV requirements are smaller, and 3) orbit insertion ΔV requirements are much smaller. The LGO FLTSATCOM becomes considerably lighter than the MGO version. The Atlas-Centaur that will be available in the mid 1980s to launch FLTSATCOM 6, 7, and 8 could easily provide sufficient boost for the LGO mission and would avoid many new operational and qualification costs. This approach may be cost effective even if some commonality were sacrificed.

The FLTSATCOM LGO spacecraft, with the still over-powered Star 37 N insertion motor, weighs around 2,900 pounds at booster separation with 150 pounds of hydrazine. Thus, Atlas Centaur launch is no problem (up to ~3,800 pounds would be acceptable), even if all Category I and II science was carried. On the other hand, reducing weight to meet PAM-A capability (~2,600 pounds) appears possible, if some commonality is sacrificed and an off-loaded Star 37 N motor is employed. The fully loaded Star 37 N motor provides considerably more impulse than required for orbit insertion with the basic science payload, but represents the least powerful Star 37-type motor. All Star 37's, having common diameters, have interchangeable mounting capability and do not require spacecraft redesign.

The allowance of 150 pounds for hydrazine expendables is more than adequate. For the LGO configuration, approximately 61 pounds would be needed for a total of 150 ft/sec ΔV for midcourse correction; and around 57 pounds to trim the orbit from an insertion altitude of 250 km (circular) to 100 km (circular). FLTSATCOM experience on-orbit indicates that less than 1 pound of propellants per year have been required to date.

LAUNCH VEHICLE SUMMARY

MGO

- STS/SPINNING SRM-1, UTILIZING
  - AN AS-YET DEFINED NASA SYSTEM
  - RENTAL OF HUGHES/INTELSAT VI CRADLE, WITH BRITISH AEROSPACE RINGS*
  - A TRW SUSPENSION SYSTEM ADDED TO RINGS
- ATLAS-CENTAUR, WITH ADDED SOLID STAGE BURN FOLLOWING CENTAUR FINAL BURN

LGO

- ATLAS-CENTAUR*, OR
  - STS/SRM-1, IF DEVELOPED FOR MGO. THE SRM-1 WOULD BE MAXIMUM OFF-LOADED AND BALLASTED IF NECESSARY
  - STS/PAM-A IS A VIABLE ALTERNATIVE TO ABOVE. A COST/RISK TRADE-OFF WITH SRM-1 APPROACH IS NECESSARY

* PREFERRED, AS PROBABLY MOST ECONOMICAL AND MINIMUM RISK. FOR MGO, OTHER OTV'S MAY BE OR MAY BECOME AVAILABLE, BUT ARE FORESEEN TO BE VERY EXPENSIVE

LAUNCH SUMMARY

Launching on the STS has static and dynamic structural load impacts in both launch and abort situations. Our FLTSATCOM-derived spacecraft will probably require some redesign to meet STS safety standards and for validation to the STS acoustic, vibration, and launch/abort loads. Some primary and interstage structure may require strengthening to accommodate higher lateral loads than FLTSATCOM is designed for. The safety issues mostly affect the propulsion subsystem and are, at best, minor. Other safety factors include assurance of surviving two successive series failures in pyrotechnic systems, such as those which ignite the solid orbit injection motor, and those which provide separation from the booster, and initiate deployment sequences. These two represent minor changes, if they are required.

We believe that use of the STS will not have a serious impact on FLTSATCOM bus design or cost. However, adaptation of boosters, such as the IUS or SRM-1 may have more serious implications.

The expected axial and yaw accelerations on the MGO/LGO spacecraft are within the FLTSATCOM capability, but the 3 g pitch acceleration capability is exceeded for either an STS-IUS or PAM-A upper stage. A pitch acceleration factor of 4.5 g seems to be realistic. In follow-on work, we will review the FLTSATCOM structural safety margins; however, it is expected that very few elements will require strengthening. The biggest impact is on the spacecraft upper stage adapters. A new structure will be required for all but Atlas-Centaur launches and will be designed using the higher pitch loads.

MGO LAUNCH VIA THE SPINNING SRM-1 MOTOR

- MOTOR UNDER DEVELOPMENT FOR INTELSAT VI
- BRITISH AEROSPACE MAKING "CARRIER" SETS (3) AND CRADLE (1) FOR HAC - NOT DELIVERABLE TO INTELSAT CORP.
- NASA HEADQUARTERS (DR. JACK WILD) INQUIRING ON RENTAL OF EQUIPMENT FROM HAC FOR MGO MISSION
- TRW HAS PRELIMINARY DESIGN OF AN ALTERNATE STS ADAPTER UTILIZING ADAPTED "CARRIERS"
- BOTH HAC AND TRW STS RELEASE METHODS CAN BE SUPPLEMENTED BY OUTRIGGER BALLAST TO PRECLUDE NEED FOR SPACECRAFT ANC
- BOTH METHODS REQUIRE EVENTUAL SPIN UP TO HIGH RPM PRIOR TO SRM-1 BURN

THE SPINNING SRM-1 MOTOR

During the study, we investigated the feasibility of renting, borrowing, or copying the Intelsat/Hughes cradle and supporting system to launch the MGO in the same manner as Intelsat VI will be launched from the STS. Pertinent memos and letters of correspondence are retained by JPL. At the time of the final report, the issue is not clear, but fall back positions, all, alas, requiring development funding are available and clearly present no technical problems.

If the spinning SRM-1 is employed for MGO/LGO, the moment of inertia ratio about the spin axis will no longer be favorable (as it is with the spacecraft alone) and, pending further investigation (see Appendix A3 by Stan Rieb), may require active nutation control (ANC) to avoid excessive loss of energy in the injection velocity direction and deleterious dynamic effects. The ANC system could be added to the SRM-1, but it seems obvious that the more cost effective approach would be to add it to the spacecraft. This addition would impact AVCS avionics, hitherto untouched.

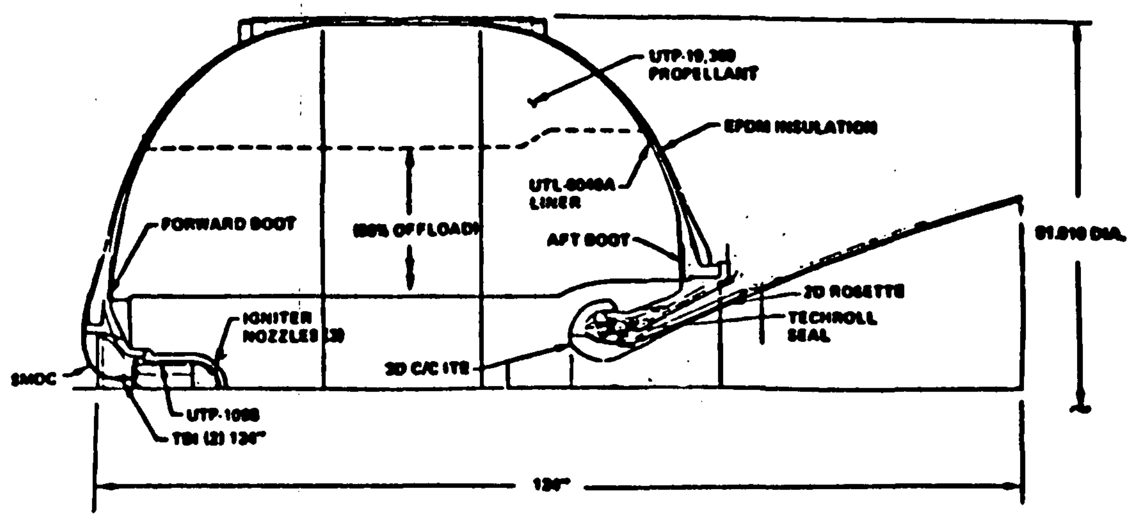
Two other approaches involving the SRM-1 can be explored in an effort to reduce cost within incurring technical risk:

1) Use the ASE developed by Boeing for the IUS stage. Presumably, the aft end of the IUS which adapts to the ASE could be used, but a bridge structure would be needed adapting to the IUS forward ASE.

2) Use a TRW-designed release system, akin to that being designed for the GRO (Gamma Ray Observatory), wherein direct mounting of the spacecraft/booster to the STS sill eliminates the need for heavy airborne support equipment (ASE).

In either case, since the three-axis controlled IUS is not spun up, we would have to provide for attitude acquisition and spin up after separation from the Orbiter. The booster/spacecraft combination may have an unstable moment of inertia ratio, and could therefore require active nutation control before and after burning.

SPINNING SRM-1 DESIGN CHARACTERISTICS



- NOMINAL PROPELLANT LOAD : 9727 KG (21400 LB)
- MINIMUM PROPELLANT LOAD : 4864 KG (10700 LB)
- BURNTIME (SCC) 0% OFFLOAD: 138 SEC
- AVERAGE THRUST : 45,600 LB
- AVERAGE PRESSURE : 640 PSI
- MAX EXPECTED PRESSURE : 870 PSI
- $\frac{\text{PROPELLANT WEIGHT}}{\text{TOTAL MOTOR WEIGHT}}$: .9382
- BURNOUT WEIGHT : 1404 LBS.

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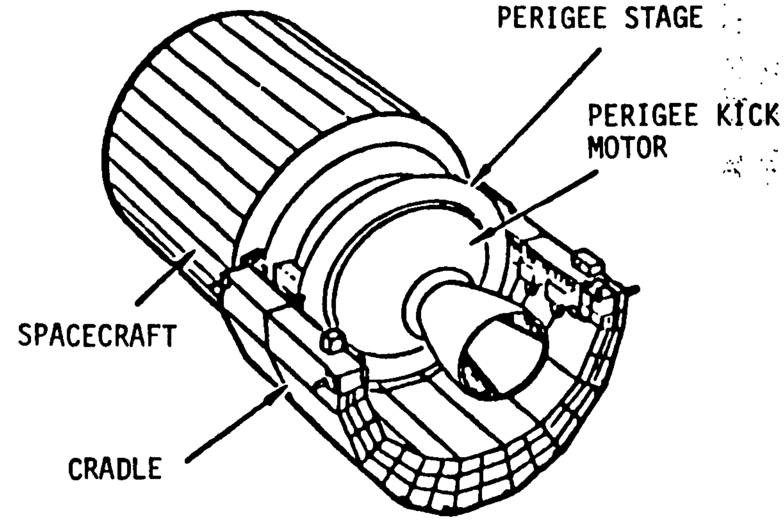
THE SPINNING SRM-1

The major difference between the spinning SRM-1 motor which will be qualified for Intelsat VI and the IUS first stage motor is that the nozzle will be fixed. Key features are noted below. The Intelsat VI cradle and carrier rings are being made by British Aerospace.

- Full Contract Go-ahead
- Program Duration 42 months
- First Test Firing 21 months
- Design and Test Program
  - AEDC Spin Test Stand
  - Fixed Nozzle
  - Added Insulation
  - Offloaded Propellant - Standard SRM-1 Case
  - 30 rpm Spin
  - 4 Motor Test Program
- Production Program
  - 3 Motor Deliveries Beginning 1985
    (Hughes has 2 Firm STS Intelsat VI Launches)
- Option for 9 more Motor Deliveries

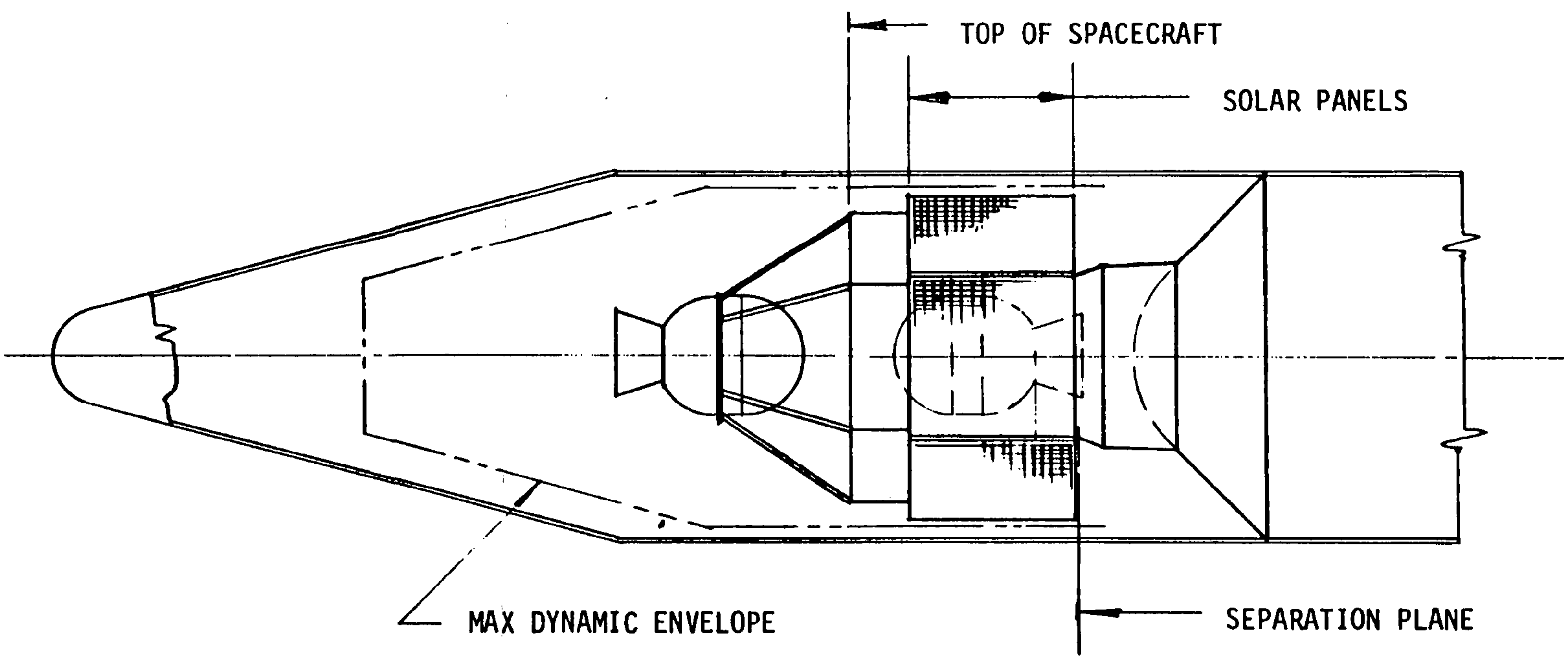


MGO LAUNCH BY ATLAS-CENTAUR

- BY EXERCISING MAXIMUM WEIGHT CONTROL, AND UTILIZING THE LIGHTEST POSSIBLE OIM MOTOR, IT MIGHT BE POSSIBLE TO REDUCE INSERTION WEIGHT TO ~3200 POUNDS

- IN THIS CASE, AN ATLAS-CENTAUR MGO LAUNCH MIGHT BE POSSIBLE BY ADDING A SECOND STAR 37 FM (~2000 POUNDS PROPELLANT) MOTOR, TO BURN AFTER CENTAUR BURN-OUT

- THIS CONFIGURATIONAL STUDY CAN BE PURSUED IN THE NEXT PHASE, IF THERE IS INTEREST



ANOTHER LAUNCHER APPROACH FOR MGO

By exercising rigid control on spacecraft weight, another MGO launch alternative appears feasible on a preliminary design level. As shown above, a second Star 37 FM could be added within the Centaur shroud by mounting it on a new support structure. At Centaur burnout, the 3-axis controlled Centaur would line up the new motor to fire in the correct direction, and would move itself forward and then out of the way. The new stage would spin-up (using solid motors) and fire. At burnout, the empty case and its support structure would separate following yo-yo despin.

At first blush, such a system might prove to be cost effective by precluding STS adaptation costs. It is suspected that development costs might be substantial and thus obviate any possible advantage.

Yet another launch scheme can be envisaged. General Dynamics/Convair has a preliminary design to strap-on four Castor solid motors to the Atlas G-Centaur and thereby achieve the capability of boosting a 3,500 to 3,800 pound payload to Mars. With the STAR 48 upper stage, this might be raised to ~4,000 pounds. Again, however, development would be required. Thus, <u>with the facts we now know</u>, the present day launch vehicle possibilities for the MGO mission using FLTSATCOM are the Titan/IUS and the STS/IUS.

USE OF SPINNING SRM-1 MOTOR FOR MGO

- THE CHOICES - ACTIVE NUTATION CONTROL (ANC) OR BALLAST
- IF BALLAST

<u>BALLAST RETAINED THROUGH SRM-1 BURNOUT</u>

- BALLAST LOAD CAN BE INCREASED BY OFF LOADING STAR 37FM → LONGER DRIFT TIME
- BALLAST MUST BE CARRIED ON DEPLOYED OUTRIGGER
- CONFIGURATION WILL BE STABLE AFTER SRM-1 BURNOUT
- LESS WEIGHT CHARGE AGAINST MGO IF SHARED STS RIDE (WEIGHT ON OUTRIGGER)

<u>BALLAST JETTISONED PRIOR TO SRM-1 BURN</u>

- NO LIMIT TO BALLAST WEIGHT, EXCEPT FOR STS CARGO PENALTY IF SHARED RIDE
- DISPOSAL OF BALLAST IS A PROBLEM-BUT POSSIBLY SOLVABLE IF WATER IS USED AS BALLAST AND WATER HOLDER RETAINED DURING BOOST.
- MAY REQUIRE QUICK SEPARATION OF BURNED OUT SRM-1 CASE IF CONFIGURATION IS UNSTABLE AFTER BURNOUT
- SRM-1 DOES NOT REQUIRE FULL PROPELLANT LOAD.

NUTATION CONTROL WITH SPINNING SRM-1

If the spinning SRM-1 motor is used to launch MGO/LGO, it is probable that since up to 45 minutes might elapse between release from the STS and motor firing, the nutation angle might have built up to an unacceptably high value. If so, active nutation control would have to be added to the system. The implications are discussed on the next page.

There are alternatives to ANC: Elimination of active nutation control by adding ballast weight to SRM-1 motor/carrier. The trade involves the cost of adding ANC to the spacecraft versus cost of adding ballast to the SRM-1 motor, which must then carry a 100% propellant load (added STS launch cost), versus cost of a 3-axis stabilized STS or Titan III compatible transfer vehicle (e.g., one or two stage IUS).

TRW has shown the cost of adding ANC to the spacecraft. Ballast, however, would have to be added by others. Thus, this trade might be better made by JPL.

Ballasting schemes are discussed in more detail later.

IMPLICATIONS OF ANC

- ANALYSIS IS DIFFICULT WITH A TANK/BLADDER SYSTEM $\Longrightarrow$ GO TO CONSTANT PRESSURE TANKS WITH LOW ULLAGE TO PERMIT GOOD ANALYSIS

- PRELIMINARY CONCLUSIONS:
  - IF FULL TANKS, τ IS ABOUT 1 HOUR (PERHAPS PRECLUDING NEED FOR ANC)
  - τ FOR BASIC CONFIGURATION AT 268 POUNDS OF HYDRAZINE IS ~6 MINUTES
  - 2 POUNDS OF THRUST PROVIDES ADEQUATE ANC CONTROL AUTHORITY FOR LOW RPM RATES (~ 5 RPM)
  - AT 50 RPM, CONTROL AUTHORITY IS LOST IF NUTATION ANGLE EXCEEDS 6°
  - AT 5 RPM, MAXIMUM NUTATION THAT CAN BE REMOVED IN ONE FIRING IS ~0.7° (.007° @ 50 RPM)
  - AT 5 RPM, ANC REQUIREMENTS FOR AN HOUR PRIOR TO SRM-1 BURN ARE ~1.2 POUNDS OF HYDRAZINE WITH A FULL TANK AND ~1.5 POUNDS FOR THE BASIC CONFIGURATION

ADDITION OF ANC TO FLTSATCOM AVIONICS

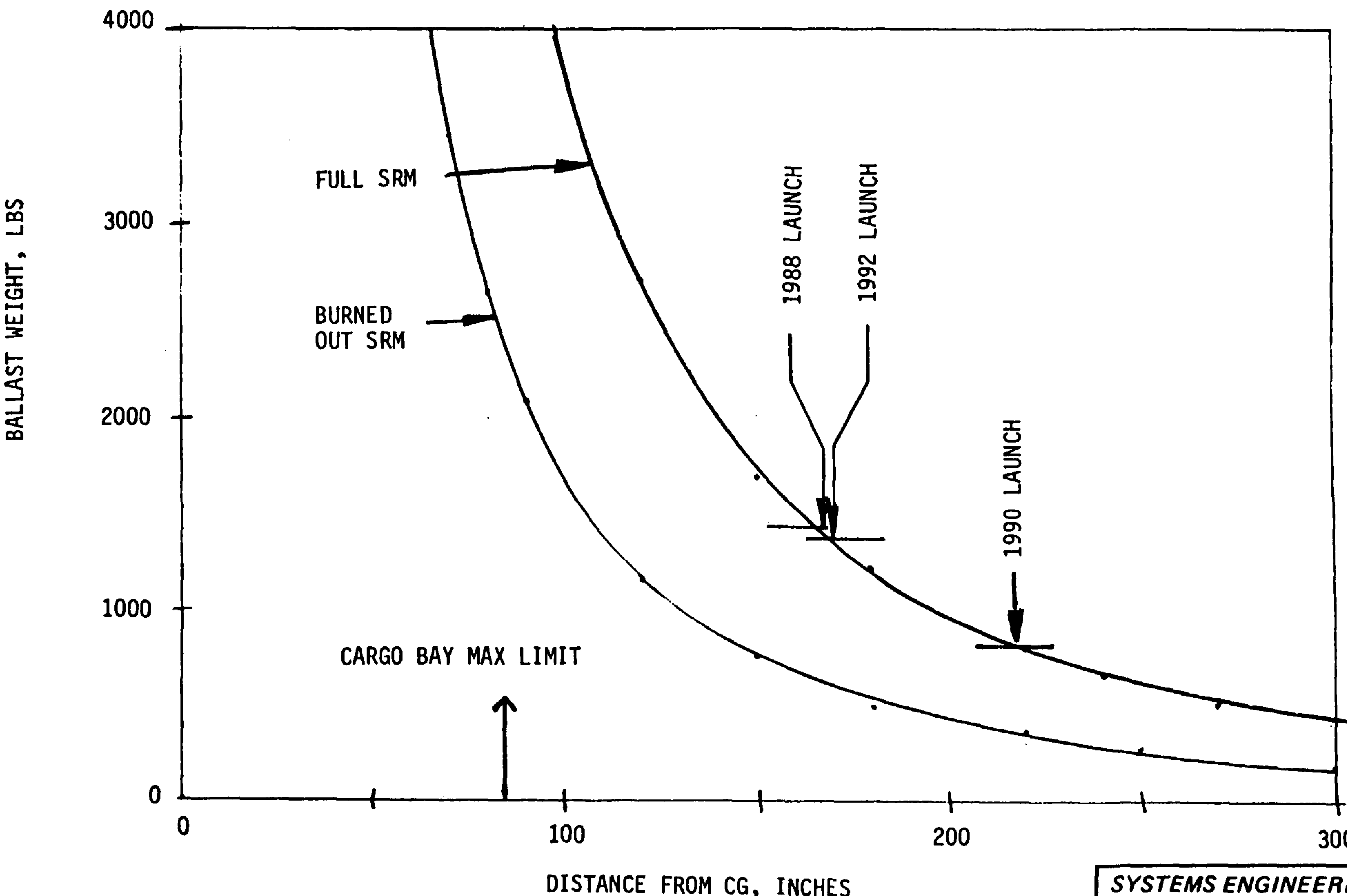
Nutation control is discussed in Appendix A3.* For the MGO configuration most likely to fly, the time constant might be as short as 6 minutes. Thus, if SRM-1 firing is not done at the first opportunity (15 - 45 minutes after STS release) initial nutations could grow to unacceptably high values, and ANC would be necessary.

The spacecraft's one pound thrusters available for control are sufficient to provide the necessary moments for nutation angles less than 6°. Changes in the AVCS avionics will be necessary to process (new) accelerometer outputs in order to provide thrust for the correct time at the correct roll position. Although the technical task is well defined, AVCS costs are obviously increased, as shown later in the report.

The alternatives to ANC (namely, ballasting) are discussed in the next two page sets.

* By Stan Rieb

BALLAST REQUIRED TO ACHIEVE 10% FAVORABLE MOMENT OF INERTIA RATIO AT MGO LAUNCH WITH SRM-1

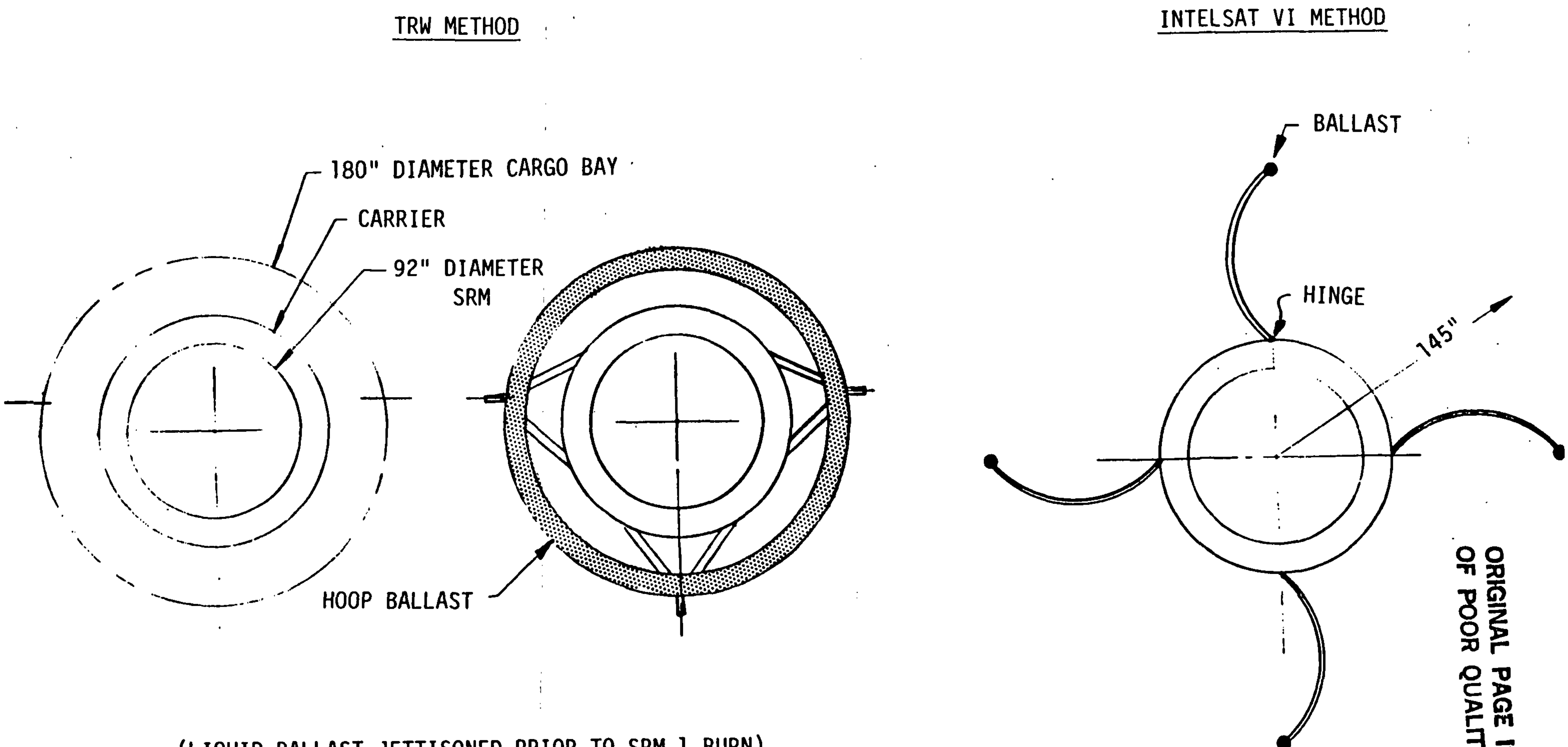


PERFORMANCE WITH BALLAST

Since the fully loaded SRM-1 motor affords excess energy to launch the MGO towards Mars, one can ask if sufficient ballast can be added to adjust the moment of inertia ratio sufficiently to preclude ANC. The above curve shows that, depending on the launch date, anywhere from ~800 pounds to ~1200 pounds can be devoted to strategically placed ballast which will be retained by the SRM-1 booster during burning and through jettison following burnout.

It may also be seen that this ballast must be located at distances from the longitudinal center line that exceed the STS cargo bay radius, and thus must be erectable. The support structure for ballast weights must be sufficiently strong and stiff to withstand the maximum "g" loads during SRM-1 burn. A conceptual design, and yet another ballast alternative is shown on the next page.

TWO METHODS OF RETAINED BALLAST ADDITION FOR MGO LAUNCH



(LIQUID BALLAST JETTISONED PRIOR TO SRM-1 BURN)

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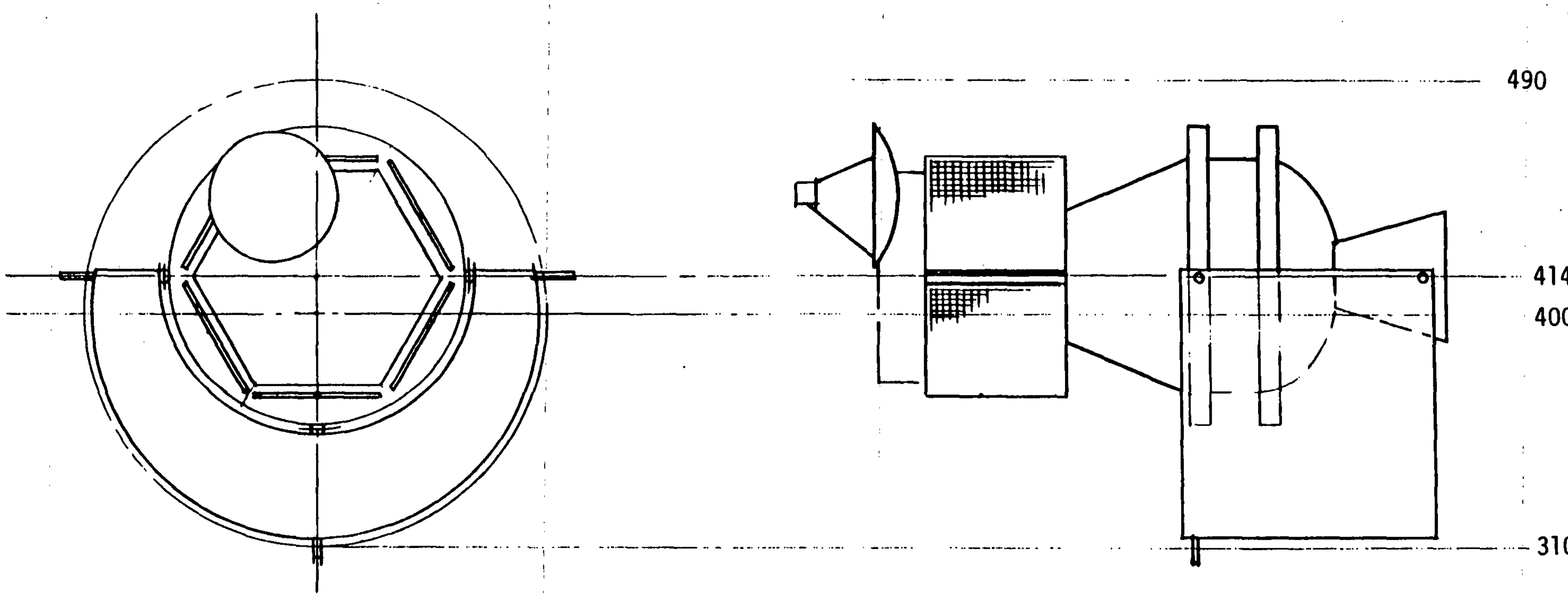
RELEASED AND RETAINED BALLAST

On the right, above, is shown a retainable ballast scheme, wherein the ballast is moved into position and locked as the spacecraft/SRM-1 combination is rolled up to speed. Fortuitously, the ballast is stored in its folded position forward of the forward carrier ring and does not interfere with the Intelsat VI cradle or release system.

Since the combined weight of the MGO/SRM-1 is far short of STS cargo bay capability, another ballasting scheme which fits within the STS cargo bay might be possible. The left hand drawing above shows a toroidal tank, which weighs less than 800 pounds and thus, when empty- can be taken along during SRM-1 burn. The tank is filled with some jettisonable (and non-freezing) liquid which provides the necessary ballast weight to provide a favorable MOIR up to jettison (which occurs just prior to SRM-1 burn). Jet damping during SRM-1 burns limits the nutation angle, and the empty case and ballast are jettisoned shortly after SRM-1 burnout.

These ballasting schemes were not pursued further, since the ANC solution appears to be the better part of valor.

MGO IN INTELSAT VI FRISBEE CRADLE



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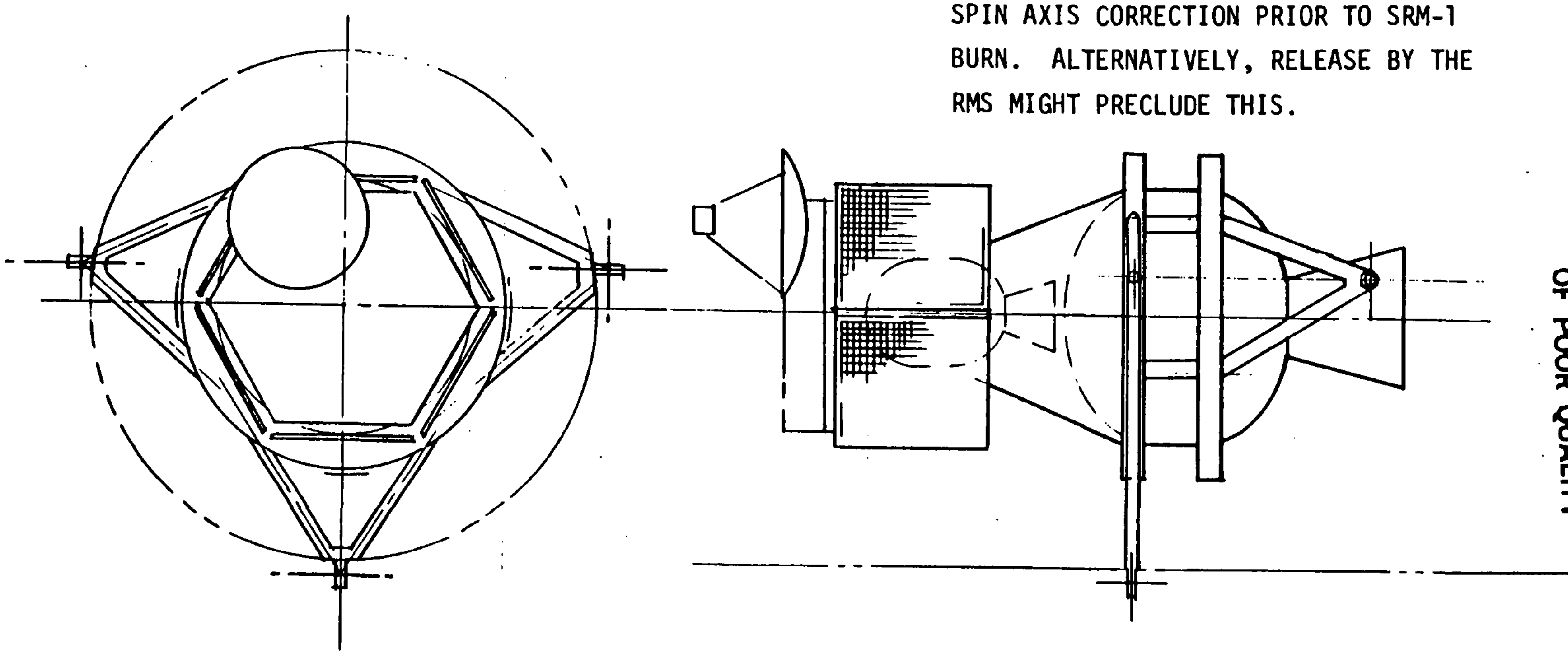
THE MOST ECONOMICAL STS RETENTION AND RELEASE METHOD FOR MGO

If suitable arrangements can be made with Intelsat and Hughes, an available, and by then, tried and true STS cradle and release method (frisbie roll-out) will be directly adaptable to MGO. Indeed, Intelsat VI and MGO spacecraft weights are similar, and MGO, having considerably less propellant aboard than I-VI, should present fewer dynamic problems.

The sketch above shows MGO in the I-VI cradle, which mates to the motor mounted carrier rings. At release, springs act on the rings on one side of the STS, while a partial hinge holds the other side down (and prevents pitch) until the hinge is disengaged. The frisbie roll rate is less than 2 rpm, and must be increased somewhat as soon as feasible to retain the release spin axis orientation.

MGO IN NEW DESIGN "TRANSLATE-OUT" RETENTION SYSTEM

NOTE: TRANSLATING OUT WILL LEAD TO TIP-OFF RATES WHICH WOULD PROBABLY REQUIRE SPIN AXIS CORRECTION PRIOR TO SRM-1 BURN. ALTERNATIVELY, RELEASE BY THE RMS MIGHT PRECLUDE THIS.



NOTE: IF BALLAST IS JETTISONED PRIOR TO SRM-1 BURN, A FOURTH (DUMMY) OUTRIGGER WOULD BE ATTACHED AND BALLAST WOULD BE LOADED OUTBOARD ON ALL FOUR OUTRIGGERS.

ALTERNATE STS RELEASE SCHEMES

It is possible that a lighter (than the I-VI cradle) STS retention system could be designed for MGO, utilizing the I-VI carrier rings as intermediate structure, as shown above. Two release schemes could be utilized:

1) Utilizing the RMS to remove the MGO/SRM-1 by holding at the standard RMS fitting attached to the carrier ring system. The actual release, at the correct attitude would be accomplished by the RMS.

2) By a four point spring system, translate the MGO/SRM-1 out of the cargo bay, and spinning up as soon as feasible, with ANC limiting the nutation angle.

Either scheme appears feasible and reasonably economical. However, development analyses and testing may be in order.

The on-going FLTSATCOM Program offers an attractive spacecraft bus for both the MGO and LGO missions. Its mass is low enough to allow these missions to be performed using relatively low cost launch vehicles. The LGO can even be launched with the Atlas-Centaur or the STS/PAM-A. The operational mode, which involves nadir pointing of the experiment mounting face and orienting the array axis perpendicular to the orbit plane, is retained.

The solar panels provide more than enough mounting area for solar cells and the attitude control pointing accuracy appears adequate for the science payload. The structure is unchanged except for details involved with the various candidate launch vehicle interfaces. The thermal control concept is retained although the ratio of insulation area to radiator area must be changed and heaters sized and located for the new payload. The communication tracking, telemetry, and command (TT&C), and data management subsystems are the only ones not applicable to the MGO and LGO missions.

MGO MISSION SEQUENCE

(STS-SRM-1 LAUNCH)

1. ROLL OR TRANSLATE OUT OF STS WITH LONGITUDINAL AXIS NEAR CORRECT
2. AT 200' FROM STS, SPIN UP TO 5 RPM BY RCS JETS AND ADJUST SPIN AXIS DIRECTION BY COMMAND (1) (FROM GROUND VIA STS). ACTIVATE ANC, IF $\sigma_f > 1$
3. PRIOR TO SRM-1 BURN, ROLL UP TO 30 - 60 RPM. BURN AT COMMANDED TIME AFTER RELEASING BALLAST*
4. AT END OF SRM-1 BURN, SEPARATE AND DESPIN (SEE EARLIER TRADE-OFF)
5. ERECT MGO WITH SPIN AXIS NORMAL TO ECLIPTIC, AT 5 RPM
6. MAKE MID COURSE PRECESSION TURNS AND BURN CORRECTIONS, AS REQUIRED. CALIBRATE γ-RAY INSTRUMENT
7. PRECESS TO OIM ATTITUDE AND SPIN UP TO 50 - 60 RPM BY RCS ROLL JETS
8. DESPIN TO 5 RPM USING RCS ROLL JETS
9. TRIM ORBIT TO ROUND IT OUT AND ADJUST INCLINATION, IF NECESSARY
10. COMPLETE MARS NADIR CAPTURE MANEUVER SAME AS FSC EARTH NADIR CAPTURE
11. MAINTAIN ON ORBIT WHILE DRIFTING TOWARD NOON ORBIT NODAL LINE
12. ADJUST ORBIT INCLINATION**
13. MAINTAIN ON-ORBIT, (INCLUDING DRAG MAKE UP, IF NECESSARY)
14. PERFORM TERMINAL MANEUVER, VIA PRECESSIONS AND TWO BURNS

* IF NO ANC

** UNLESS CONTINUOUS DRIFT MODE

(1) IF TIME AND SPINNING EARTH SENSOR OUTPUT PERMITS: IF NOT, ERRORS WILL BE CORRECTED DURING MID - COURSE

There are three "open" possibilities that are subject to further study:

(1) Can the spin axis be maintained to within $\sim 1^{\circ}$ of the desirable angle at SRM-1 booster firing, particularly if the first opportunity to fire is not used, and another orbit must be completed. The MGO has spinning sensor instruments to define the spin axis precisely, and, via the STS, can be commanded to precess to the desired position - albeit this may be a stressful process for ground operations.

(2) There is a best time to arrive at final pre-SRM-1 burn roll rate. On one hand, one would like to spin slowly ($\sim$ 5 rpm) until just before burn (to minimize ANC hydrazine usage). On the other hand, the rapid solid spin motor acceleration from 5 to $\sim$30 or more rpm might result in a too large a nutation angle. This deserves further study.

(3) Spin rates for both SRM-1 and the OIM motor will have to be studied. Motor qualification results could well determine the rate.

In addition, following step 6 above, a battery reconditioning cycle will be considered. This would require about two days.

LGO MISSION SEQUENCE
(ATLAS-CENTAUR* LAUNCH)

1. LAUNCH EXACTLY LIKE FSC-CENTAUR TO ERECT PROPER CRUISE-OUT LONGITUDINAL (SPIN) AXIS AND SEPARATE
2. SPIN UP TO 5 RPM (JUST LIKE FSC)
3. COURSE CORRECTION PRECESSIONS AND BURNS, CALIBRATE γ-RAY INSTRUMENT
4. PRECESS TO ORBIT INSERTION BURN ATTITUDE, AND SPIN UP TO 50 - 60 RPM
5. BURN ORBIT INSERTION MOTOR
6. TRIM ORBIT TO CIRCULARIZE
7. DESPIN TO 5 RPM
8. PERFORM NORMAL FSC LUNAR NADIR CAPTURE MANEUVERS
9. MAINTAIN IN ORBIT
10. ROTATE SPACECRAFT 180°, AS REQUIRED

* PAM-A LAUNCH WOULD BE LIKE MGO LAUNCH, WITH ANC

LGO MISSION

The combination of spinning earth sensors (if their field of view is adjusted) and sun sensors which determine FLTSATCOM spin axis orientation on its way out to GEO will also serve the same function in cislunar space.

Because there is the capacity to carry considerably more hydrazine than is needed, two orbit insertion options exist (and would be decided in the next program phase):

1. Calibrate the γ-ray instrument during transit (as for MGO) and insert into a near circular orbit. If the STAR 37N motor is still too energetic, insertion orbit "cranking" can be used.
2. By using a smaller motor then the STAR 37N, an elliptical orbit could be achieved, thus permitting γ-ray instrument calibration at apoapsis. Successive periapsis hydrazine burns would then be used to round out the orbit.

ΔV/HYDRAZINE REQUIREMENTS

MGO

EVENT	BASELINE DESIGN (268 $\#N_2H_4$) ΔV (FT/SEC)	$\#N_2H_4$	PRESSURE BOTTLE ADDED (342 $\#N_2H_4$) ΔV (FT/SEC)	$\#N_2H_4$
SPIN UP TO 5 RPM		1.0		1.0
ANC		1.5		1.2
PRECESS TO CORRECT SPIN AXIS		~0		~0
DESPIN TO 5 RPM (S/C ONLY)		7.0		7.0
MIDCOURSE CORRECTIONS- PRECESS AND ADD ΔV	150	75.9	250	140.5
PRECESS TO OIM ATTITUDE		0.7		0.7
SPIN UP TO OIM RPM		7.0		7.0
DESPIN IN MARTIAN ORBIT		7.0		7.0
ROUND OUT ORBIT AT 300 KM	70	13.1	70	13.9
DEPLOYMENT AND ERECTION ON MARS ORBIT		~0		~0
CHANGE ORBIT INCLINATION (^{o})	(2.5^{o})	94.0	(3^{o})	113.0
MAINTENANCE ON ORBIT		15.0		15.0
TERMINAL QUARANTINE MANEUVER	148	26.8	148	26.8
MARGIN		13.6		13.3

ΔV REQUIREMENTS SUMMARY

The above summarizes estimated ΔV requirements for the MGO and LGO missions from Earth departure through mission completion. These data will require updating and augmentation after a more detailed study of dispersion following burn of the selected upper stage booster.

Areas that need to be clarified in the MGO mission include the amount of orbit trim required under realistic dispersions of solid motor total impulse delivered, the orbital plane change requirement from the drift orbit to the sun synchronous orbit inclination, sensitivity of science observations to accurate orbit maintenance, and the issue of an acceptably safe the end-of-mission quarantine boost. The 150 ft/sec mid course correction allotment is a 2σ figure according to the earlier sample calculation. It is noted that the hydrazine supply for these maneuvers could be almost doubled by making the feed system a constant pressure one. Moreover, the 2.5° orbital plane inclination change allotment is arbitrary, and could be reduced to near zero for a nonsynchronized orbit.

In the LGO mission the ΔV requirements are less critical to FLTSATCOM capabilities, primarily because there will be no need for a plane change maneuver and there is no quarantine. However, orbit sustenance at 100 km altitude may require more than the allocated 50 m/sec and will require further analysis.

DESCRIPTION OF MGO/LGO BASIC SPACECRAFT

	MGO	LGO
● LAUNCH SYSTEM	STS/SRM-1	A-CENTAUR
● DESIGNED FOR STS SAFETY/ LOADS	YES	NO
● ADDED PROPELLANT CAPACITY	NO	NO
● 2 AXIS HGA	YES	NO
● S & X BAND TT&C	YES	S ONLY
● CONICAL HORIZON SCANNERS	YES (2)	YES (4)
● CATEGORY 1 P/L ONLY	YES (3 + A)	YES (3 + B)
● SPINNING STELLAR SENSOR	YES	NO (FSC SPINNING EARTH SENSOR)
● ACTIVE NUTATION CONTROL	NO	NO
● STEM DEVICES FOR INSTRUMENT MOUNTING AND EXTENSION	YES	YES
● OPTIMUM STAR 37 MOTOR	YES (FM)	YES (N)
● 'STRAIGHT' SOLAR ARRAY	YES (OPTIMUM SIZE)	NO (SAME SIZE, AT 45°)
● MOMENTUM WHEEL MOUNTED AS IN FSC	YES	YES
● POWER SYSTEM	OPTIMIZED FOR MGO	OPTIMIZED FOR LGO
● MAGNETIC CLEANLINESS PROGRAM	YES	NO

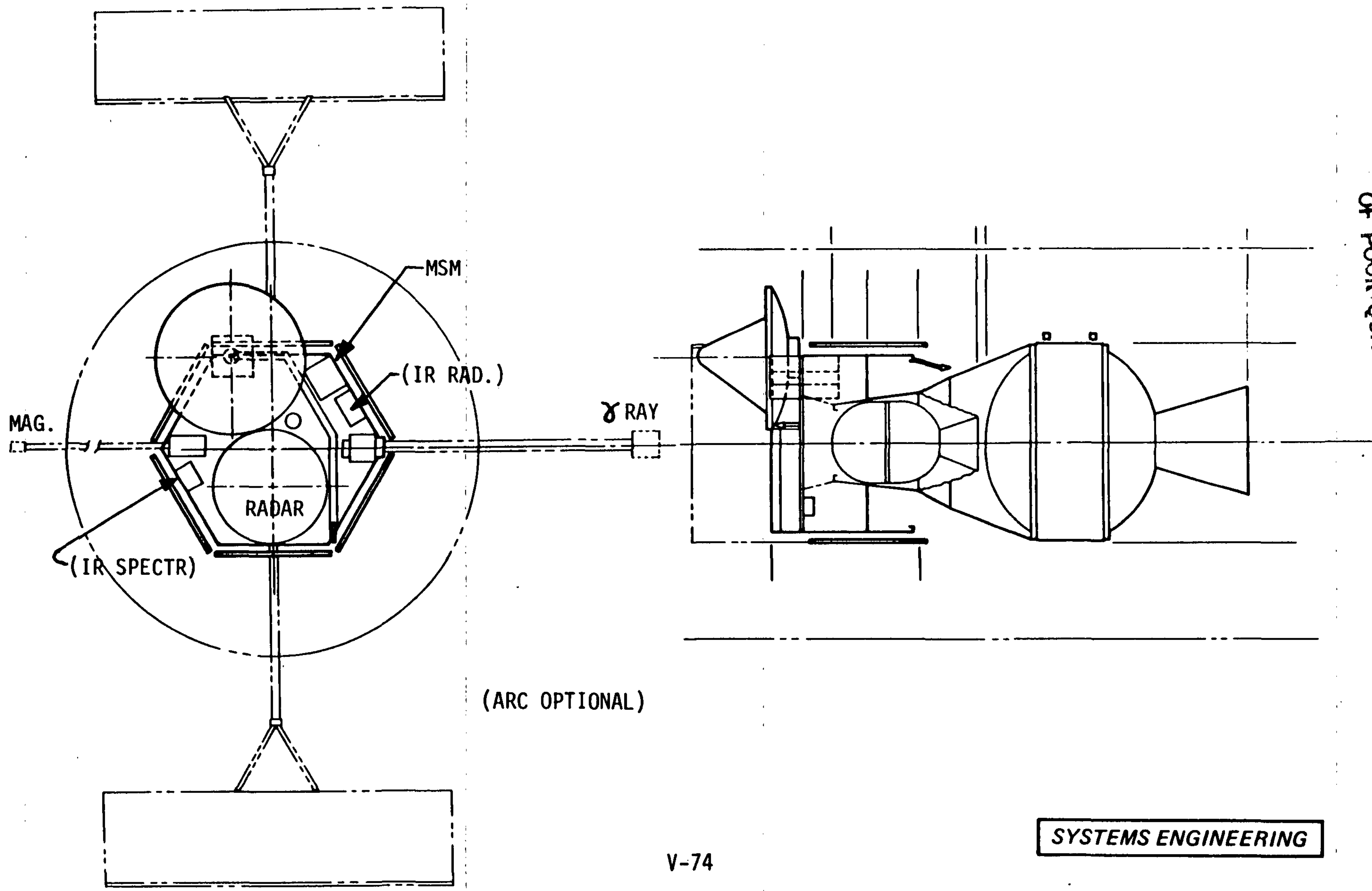
CHANGING THE FLTSATCOM INTO MGO/LGO

In this section, VE, with subsequent subsections VE 1-6, we will describe the changes required to convert FSC into an MGO or an LGO. First, the two orbiting spacecraft will be described, including the structural subsystems, and their performance characterized. Then, the other major subsystems will be discussed, as indicated below:

VE1 - Communications
VE2 - Command and Data Handling
VE3 - Power and Distribution
VE4 - Thermal Control
VE5 - Attitude and Velocity Control
VE6 - Propulsion

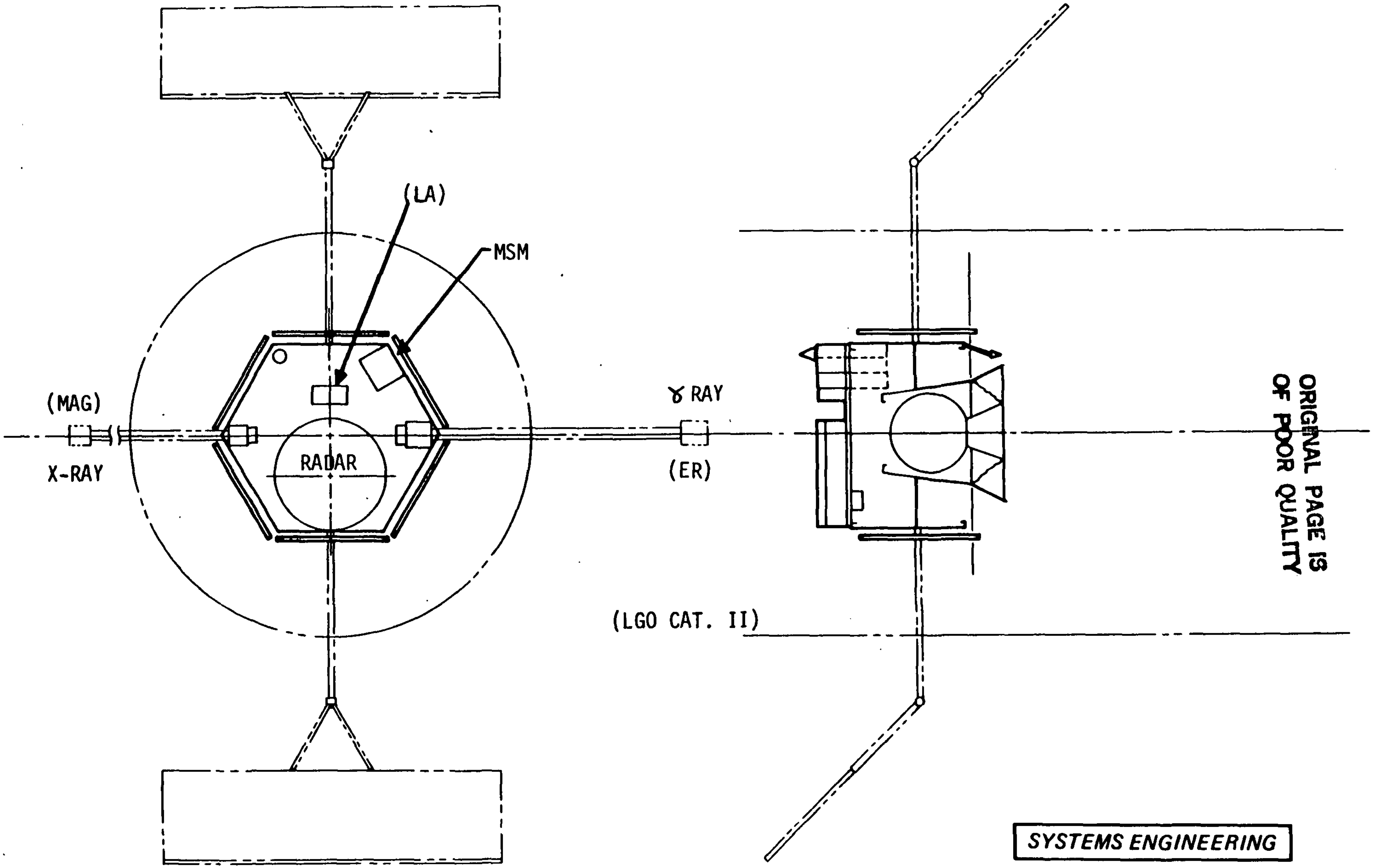
As noted above, there are significant differences between MGO and LGO. It would not be cost effective to maintain absolute commonality. The major subsystem changes to FLTSATCOM 7 and 8 configurations are to:

- Replace military network-compatible TT&C and omni antenna system with components taken from ongoing programs to form a system compatible with DSN operations, including a data storage/dump system.
- Add a bi-axial Earth tracking antenna to facilitate the downlink data stream for MGO mission.
- Replace spinning Earth sensor needed to assist attitude determination during transfer to GEO with a spinning stellar mapper, to provide attitude determination during interplanetary cruise (at least for MGO mission).
- Replace linear scan on-orbit Earth horizon scanners with conical scan horizon scanners to establish nadir on Mars and moon.


MSM
(IR RAD.)
MAG.
RADAR
(IR SPECTR)
γ RAY
(ARC OPTIONAL)

The above depicts the "short" (like FSC 6) body MGO with the STAR 37FM motor and a two-meter HGA. Note that the boom mounted category 1 instruments (mag. and γ-ray) can be extended and retracted without interference with the folded solar panels which are about one-half the size of normal FSC panels. Two Mars I.R. climatology mission (NASA Ames Research Center mission) instruments are also shown. The upper stage booster depicted is the SRM-1. For reference, the outline of the STS cargo bay is shown.

Compatibility of the spacecraft structure with launch vehicles other than Atlas-Centaur needs continued investigation. Utilization of STS, Titan, IUS or SRM-1 vehicles and combinations thereof will result in new acoustic, vibration, and loading environments for which the spacecraft will have to be qualified. Integration with these vehicles may also require new hard points and special handling fixtures and GSE. Utilization of the STS brings another dimension into the integration problem because of man rating, and special umbilical features. The impact of STS safety requirements primarily affect the propulsion subsystem and require a review and possible revision of pyrotechnics and latches, etc., to meet two fault tolerance requirements. Issues involved with compatibility with the STS were previously discussed. The more difficult problem of loads induced into the spacecraft via the STS (or Titan) through the orbital transfer vehicle (OTV) requires detailed analyses that were beyond the scope of this study.



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SYSTEMS ENGINEERING

The main obvious differences are the lack of a not-needed HGA, and the 45° bent solar panels - necessitated for instrument thermal control reasons.

A number of alternative solar array concepts were examined for the lunar orbiter, primarily because the FLTSATCOM operational mode (body nadir pointing and array axis normal to the orbit plane) leads to inadequate power when the sunline is near the orbit plane. Thus when the sun was within 30-degrees of the orbit plane (2 months out of 6), a sun-pointing mode would be used and no science data gathered. This increases the mission time to 1-1/3 years for 1-year's worth of data gathering. The first alternative concept involves flying with the array axis in the orbital plane. In this aspect, the minimum array exposure is when the orbit plane contains the sunline. This concept therefore demands a large array area since power is available only through a $\sim 90^{\circ}$ sector. It also demands a 90° rotation of the momentum wheels and thrusters and poses array shadowing problems.

The second concept uses bent solar array arms and gives 70 percent of full power at the two worst sun orientations. However, it requires a 180-degree spacecraft flip twice a year and may require a minor modification to the wheel electronics to command spin in either direction.

As in MGO, two STEM-type deployable booms are utilized to carry scientific instruments, and are ballasted as necessary for balance, particularly when being calibrated in the spinning mode.

MGO/LGO WEIGHT BREAKDOWN
BASELINE CONFIGURATIONS
(POUNDS MASS)

SUBSYSTEM	LGO	MGO (NO BALLAST)
STRUCTURES	330.4	336.5
INTEGRATION HARDWARE/BALANCE	18.5	18.5
THERMAL CONTROL	35.8	35.8
ELECTRICAL POWER AND DISTRIBUTION	300.0	300.0
ATTITUDE VELOCITY AND CONTROL	130.3	130.3
HIGH GAIN ANTENNA	-----	76.0
RESIDUAL HYDRAZINE	16.0	16.0
COMMUNICATIONS P/L	-----	-----
STEM DEVICES (2)	24.0	24.0
BALANCE	10.0	10.0
TT&C/DATA HANDLING	98.8	145.2
REACTION CONTROL SYSTEM (DRY)	64.7	64.7
SCIENCE PAYLOAD	127.6	92.4
SPACECRAFT DRY WEIGHT	1161.1	1244.4
RCS FLUIDS + PRESSURANT	134.5	250.0
FIRED OIM (AKM)	140.0	161.7
SPACECRAFT AT BOL	1435.6	1656.1
SOLID EXPENDABLES - OIM	1232.0	2320.2
SPACECRAFT AT UPPER STAGE SEPARATION	2672.6	3981.2
UPPER STAGE ADAPTER	41.0	41.0
UPPER STAGE OTV + CARRIER	-(*)	2376.0
PROPELLANT (UPPER STAGE)	-----	21400.0
FLIGHT WEIGHT	2708.6	27793.3

* ATLAS-CENTAUR

SPACECRAFT WEIGHT

The weight breakdown shown indicates a slightly higher residual hydrazine margin than the sample ΔV calculation. The insertion motors employed are the STAR 37FM (MGO) and the STAR 37N (LGO). No spin balancing ballast is shown; nor motor propellant off-loading. The MGO presumes an SRM-1 upper stage, and the LGO is presumed launched by Atlas-Centaur. Note that the flight weight of the MGO-upper stage is such that two similar vehicles could be carried in a single STS, in the case that both LGO and MGO missions were launched from the same flight.

Structure and antenna boom modifications are due to Shuttle (STS) launch loads and payload accommodations from substituting the scientific payload and communication subsystem for the present communication payload. The OIM loads are more severe than STS induced loads, but since FLTSATCOMs 7 and 8 are being upgraded to the STAR 37FM OIM, we conclude that static load conditions will not require any additional strengthening of the structure for the MGO/LGO missions.

SEQUENCED WEIGHT SUMMARY FOR MGO BASELINE CONFIGURATION

MISSION PHASE	ITEM	ITEM WT	SEQUENCED WT
• END OF LIFE	• SCIENCE PAYLOAD		
	• SPACECRAFT (DRY)		
	• STAR 37FM MOTOR (BURNOUT)		
	• MARGIN + RESIDUAL FUEL		1244
	• EMPTY AKM	161.7	1406
• END OF MISSION	• PROPELLANT - SAFE ORBIT INSERTION	27	1433
• POST-MOI	• PROPELLANT - MANEUVERS	165	1599
• PRE-MOI	• PROPELLANT - MOI	2320	3919
• POST-SRM SEPARATION	• PROPELLANT - MIDCOURSE ADJUSTMENTS	76	3995
• PRE-SRM SEPARATION	• SRM (BURNOUT WITH CARRIER)	2376	6371
	• 100% PROPELLANT LOAD, ETC.*	21422	27793
• STS CARGO	• CRADLE (HAC DESIGN)	2928	30699

* LEAVES ~1400 LBS. AVAILABLE FOR BALLAST FOR 1988 LAUNCH

EFFECT ON ENVIRONMENT ON WEIGHT

The FLTSATCOM is launched on the relatively benign Atlas-Centaur. However, fortuitously, it is designed to sustain the larger loads of the apogee kick motor (AKM). The buses will have to be validated for new launch situations; but, at first blush, it appears that of all launch vehicles under consideration, only STS induced lateral loads might exceed present design capabilities. The FLTSATCOM 7 and 8 versions will be altered to be compatible with the more powerful, higher thrust STAR 37FM motor. The above presents MGO sequenced spacecraft weight summaries.

The FLTSATCOM was originally designed to be compatible with the launch loads associated with the Atlas-Centaur vehicle and with its STAR 37F AKM. The Atlas-Centaur imparts a ΔV of $\sim$8000 ft/sec to the FLTSATCOM, and has the capability to spin up to about 5 rpm before separation. The Atlas-G Centaur, which will be first available in 1984 and will later launch FLTSATCOMs 7 and 8, will have the capability to place a 3,800 pound payload on a lunar intersect trajectory. With a STAR 48 upper stage on the Centaur, this figure might be raised to $\sim$4,200 pounds. However, this launch system would require development dollars. Since the LGO spacecraft could be significantly lighter than the MGO spacecraft (propellant loading differences alone could amount to over 200 pounds), it is possible that the LGO could be launched by an Atlas-Centaur and still retain essential commonality with the MGO bus. The capabilities of STS/PAM-A for the lunar mission are about 2,500 pounds; thus making this launch combination marginally possible. Other STS launch combinations that could accomplish the LGO injection are STS/IUS, STS/Centaur, and, if available, STS/SRM-1 (off-loaded and ballasted). Titan/IUS and Titan/SRM-1 (if available) are other expendable launch vehicles capable of lunar injection of the FLTSATCOM bus LGO spacecraft.

CHANGES TO FSC

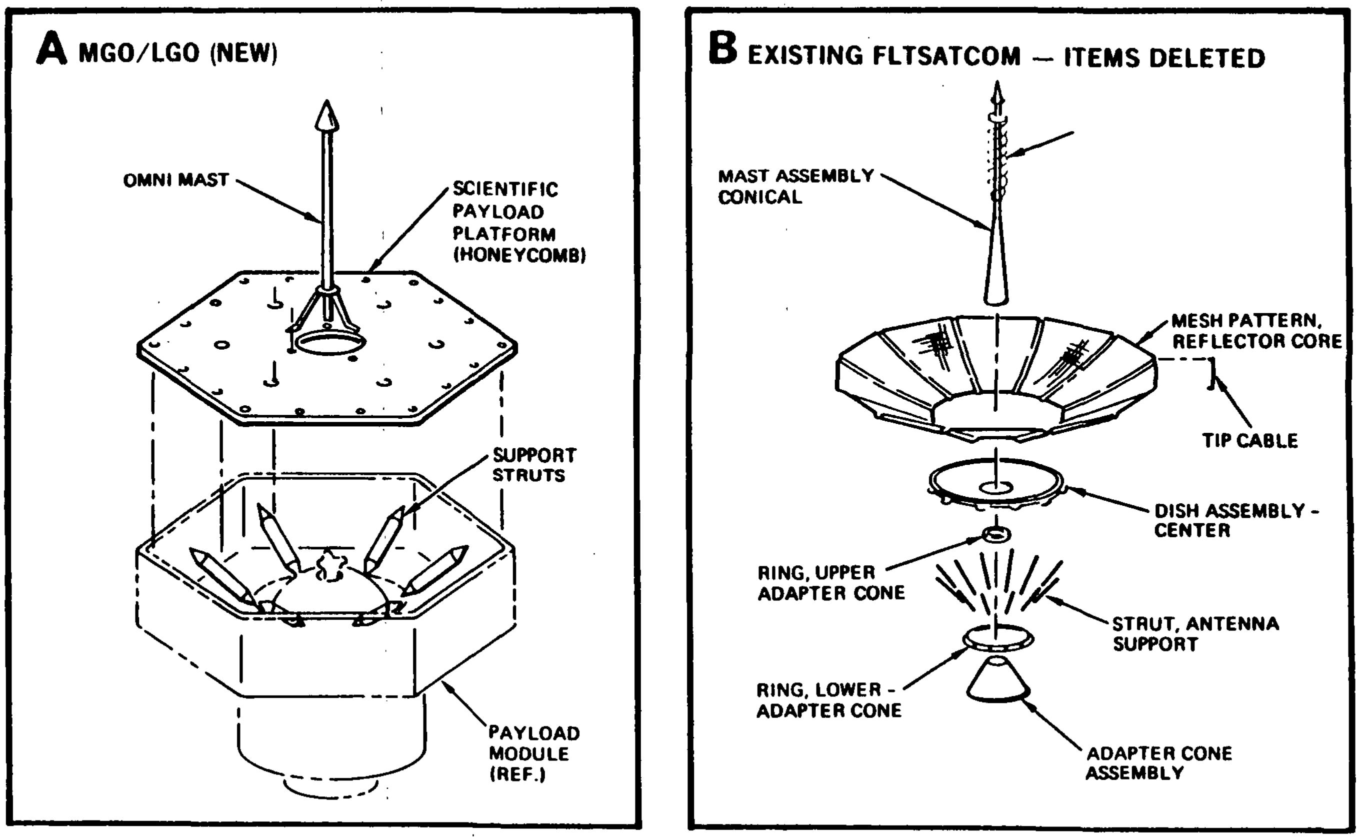
	For Basic	MGO	LGO
• Adapter to L.V.		X	
• Umbilical		X	X
• Payload/Structure (Remove Comm., Add Science Instruments)		X	X
• TT & C (Remove Military System, Mount NASA System + OMNI Antennas)		X	X
• Add Data Handling System		X	X
• Add High Gain Antenna and Support/Drive System		X	
• Replace Linear Horizon Scanners With Conical Horizon Scanners		X(2)	X(4)
• Replace Spinning Earth Sensor With Spinning Stellar Sensor		X	
• Remount Spinning Earth Sensor			X
• Revise Harnesses		X	X
• Reduce Height of Solar Panels		X	X
• Exchange FSC Power Control and Battery Units		X	X
• Changes in Thermal Control Treatment		X	X
• Changes Required for Safety for STS Adaptation		X	
• Changes in Tests and Procedures		X	X
• Solar Panels Extended at 45°			X

ALTERATIONS TO FLTSATCOM BUS

A significant effort has been devoted to developing operational modes that avoid the need for major modifications and lead to nearly identical MGO and LGO spacecraft. The Mars mission is particularly suitable since the sun synchronous polar orbit is operationally equivalent to an earth synchronous equatorial orbit.

A very detailed account of FSC modifications; the reasons for the mods; and the impact of the changes is listed in Appendix A6. The only significant item not covered in A6 is the question of enabling the momentum wheel to stop and turn in the opposite direction when the LGO is rotated 180°. It is known that the wheel can turn in either direction. The change, if needed, is deemed to be minor and will be commanded as part of the "flip" sequence.

The major structural configuration change is to remove the FLTSATCOM communications module which is replaced by a structurally identical payload module for the MGO and LGO missions. This payload module also supports the high gain antenna (HGA) for the MGO and a broad coverage antenna for both missions. The FLTSATCOM initial (folded) deployment configuration is retained. The LGO configuration has no high gain antenna. Another pervasive modification is associated with adaptation to the STS from the Atlas-Centaur and can not be specified until an upper stage system is selected.


A MGO/LGO (NEW)
OMNI MAST
SCIENTIFIC PAYLOAD PLATFORM (HONEYCOMB)
SUPPORT STRUTS
PAYLOAD MODULE (REF.)
B EXISTING FLTSATCOM — ITEMS DELETED
MAST ASSEMBLY CONICAL
MESH PATTERN, REFLECTOR CORE
TIP CABLE
DISH ASSEMBLY - CENTER
RING, UPPER ADAPTER CONE
STRUT, ANTENNA SUPPORT
RING, LOWER - ADAPTER CONE
ADAPTER CONE ASSEMBLY

PAYLOAD MODULE CHANGES

The Scientific payload is supported on a new platform made from fiberglass honeycomb and 0.006 aluminum face sheets as shown above. Six support struts provide inboard supports to the platform for added stiffness. The forward omni mast is also shown and is a new item. The aft omni mast is supported from the equipment module aft frame and poses no particular concerns. The solar array structural/mechanical elements remain unchanged.

The high gain antenna (HGA) used for the MGO mission, uses the DSCS-II/GRO derived dish and dual-axis drive. To preclude RF transfer across rotating joints, the HGA RF system is mounted in a box that mounts to the existing antenna back structure. During launch and ascent loads, latches at the RF box and at the two-axis drive are used to rigidly support the whole assembly to the forward platform. After despin, the latches are released and a damped, spring pivot assembly deploys the HGA to its trailing position. The trailing arm from the pivot to the two-axis drive ends up paralleling the spacecraft side panels, allowing a deployed position latch to make the assembly rigid during operations.

Another alternative to mounting the HGA should be investigated in later studies. Mounting on the rotating solar panel arm is possible, since, on-orbit, the Earth is always within a relatively small solid angle from the sun, and azimuth and elevation off-sets change very slowly with time.

ITHACO CONICAL EARTH SENSOR

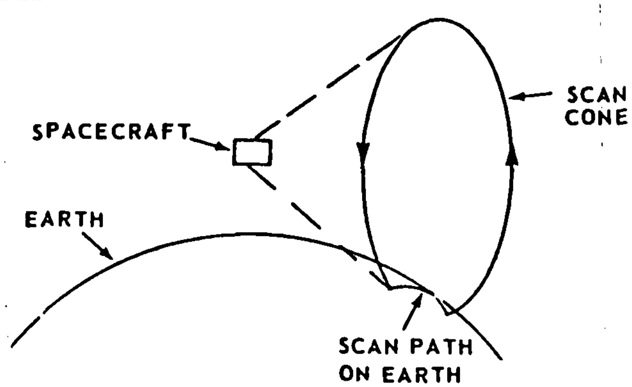
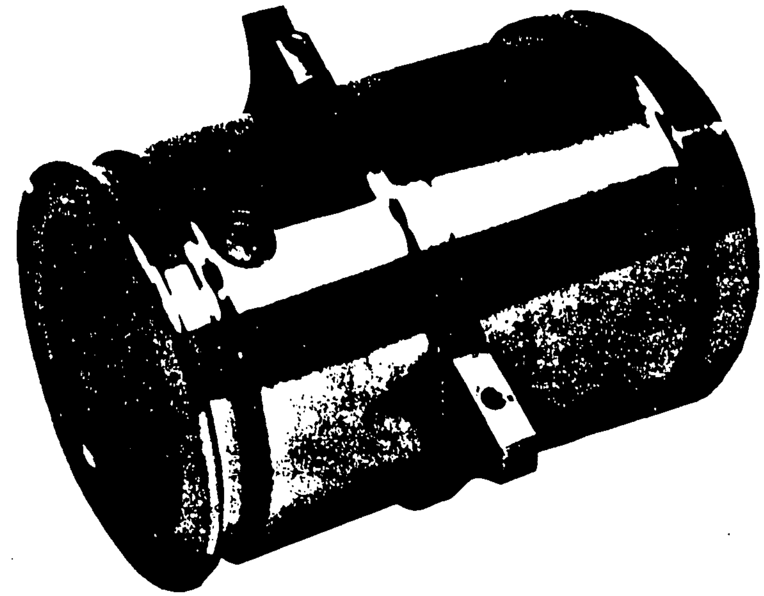
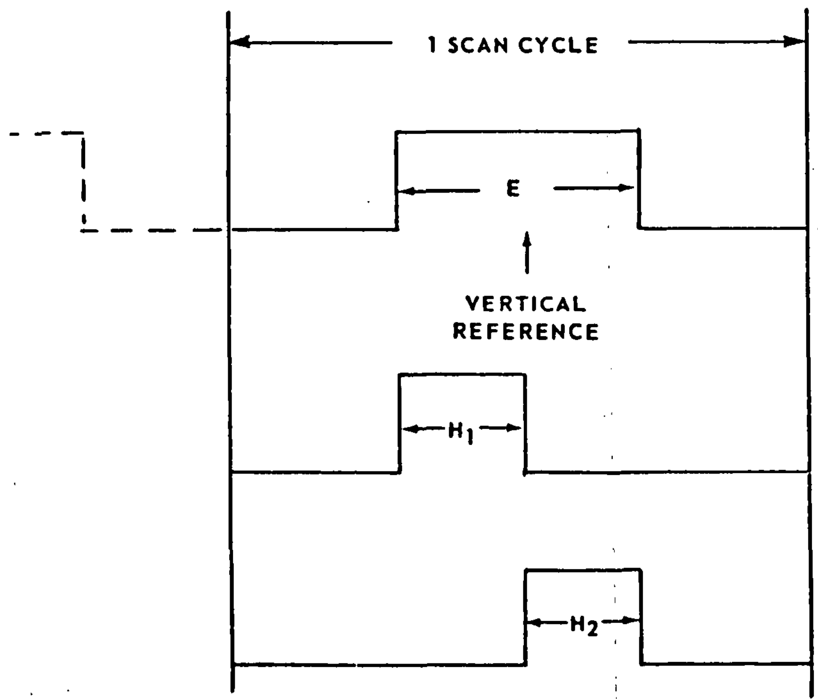


FIGURE 1





AFTER CALIBRATION, SCANNER(S) SHOULD YIELD ALTITUDE DATA TO WITHIN ~ 100 METERS

Specification Summary

APPLICATION	Two axis attitude determination for spacecraft		
ATTITUDE RANGE	150 Km to super-synchronous		
ACCURACY	Geosynchronous < .05°		
	Low Orbits < .10°		
	SENSOR	ELECTRONICS	TOTAL
SIZE	4.0" x 3.0" dia.	6" x 6" x 1.7"	
WEIGHT	1.9 lbs	2.1 lbs.	4.0 lbs.
POWER	4 Watts	2 Watts	6 Watts

SYSTEMS ENGINEERING

CONICAL EARTH SCANNER

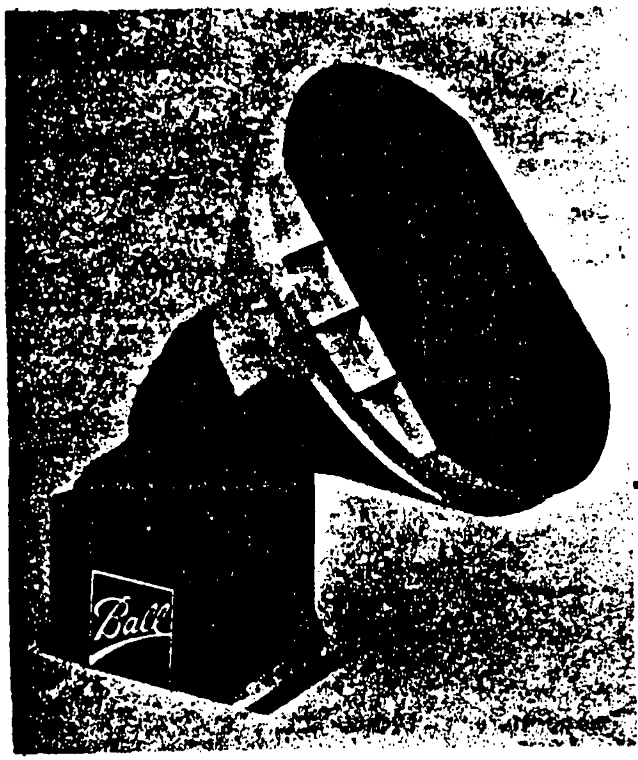
The Conical Earth Sensor can be used without modification for attitude determination in virtually any orbit from 100 km to super-GEO (Earth), whether circular or highly elliptical. Its attitude coverage is unique - 90° roll maneuvers are permitted, for example. The sensor described is being used on three current programs, LANDSAT D, P80-1 and SPACELAB, and is a modification of a horizon sensor flying on the successful P78-2 spacecraft. It replaces the linear limb scanning FSC horizon scanner, but since its output is essentially the same (time from limb-to-limb and halfway pip), only a signal conditioner change may be necessary in the electronics.

The scanner consists of a sensor head and an electronics box. The sensor head is illustrated above. A hollow shaft motor contains within its shaft an optical barrel that includes a germanium lens and a germanium immersed bolometer (detector). A wedge-shaped prism rotates in front of the lens to deflect the image of the detector by 45°. A coated germanium window not only establishes the 14 - 16 micron passband, but also hermetically seals the unit.

The field of view of the sensor is deflected by ~45° and rotated continuously, which results in the scan pattern shown above, left. When the scan path intersects the earth, the infrared detector senses the difference in temperature between earth and space. The resulting pulse can be processed to determine both pitch and roll attitude. A vertical reference pulse on the sensor defines where the center of the earth pulse out to be, and the width of the pulse is a function of the attitude error in the other axis.

Error analyses made for Earth application indicate that the scanner, after on-orbit calibration confirmation of pure theoretical calculations, can yield orbital altitude with great accuracy. This attribute relaxes ground requirements for tracking, which is now needed mostly to confirm the plane of the orbit in relation to the sunline.

SOLID STATE STAR SCANNER CS-201



ELECTRO-OPTICS

Detector: Silicon diode array, two element

Lens: 50 mm, f/0.95

PERFORMANCE

Field-of-view: 24° elevation

Star Sensitivity: +1.7 to −1.4 silicon magnitude (8 commandable thresholds)

Probability of Detection: 0.99 (+1.7 silicon magnitude)

Spacecraft Spin Rate: 4 to 60 rpm

Sensor Optical Axis: 56° from spin axis

Accuracy
- **Position:** ±10 arc minutes (1σ) *
- **Star Intensity:** ±0.25 magnitude

Sun Shade Protection: up to ±55° vertical axis, ±46° horizontal axis

Power Consumption: Less than 1.9 Watts from 28 Vdc ±10%.

Reliability: .9998 for 1 year

MECHANICAL CONFIGURATION
(including two-stage sun shade)

Size: 19.5" x 18" x 14" (495mm x 457mm x 356mm)

Weight: 6.5 lbs. (2.95kg)

* At 60 rpm; accuracy improves as rpm decreases

STELLAR SCANNER

The Scanner will replace the FSC Spinning Earth Sensor on the MGO mission as the second attitude reference (in addition to the fine spinning sun sensor) to determine spin axis orientation. Its output is directly entered into the T/M stream for ground calculation, thus in no way affecting AVCS electronics.

The CS-201 is a solid state star scanner which can provide a spin stabilized spacecraft with star position measurements for both on-board and ground based attitude determination. The scanner's large field-of-view, coupled with its high probability of detection for stars as dim as +1.7 silicon magnitude, make it also suitable for closed loop attitude control systems. The scanner, built for the Pioneer-Venus program, has fully redundant electronics channels, resulting in very high reliability for long missions. Two flight units and a qualification model have been delivered on this program, and the unit has performed successfully.

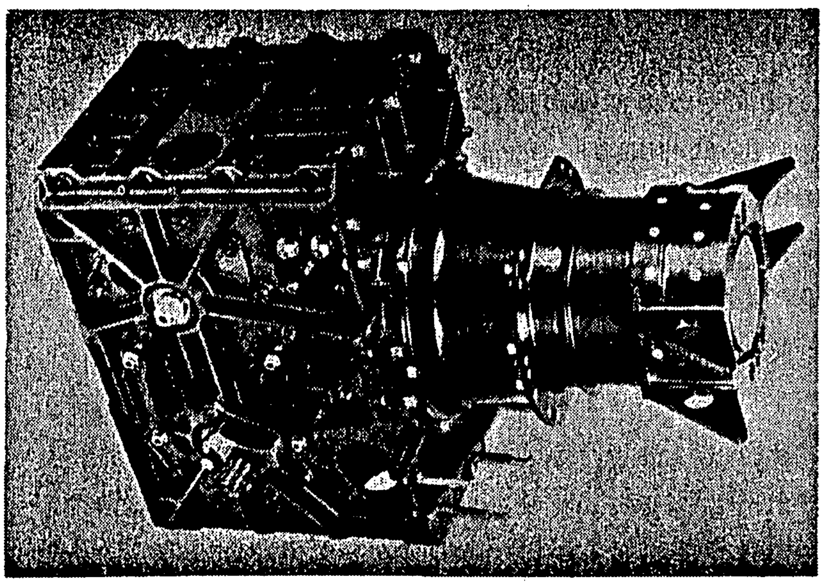
The heart of the CS-201 is a silicon detector with two active photodiodes, one for each channel. Each dual scanner channel has narrow, fan shaped fields-of-view; one aligned parallel to the plane containing the spin axis, the other inclined 20°.

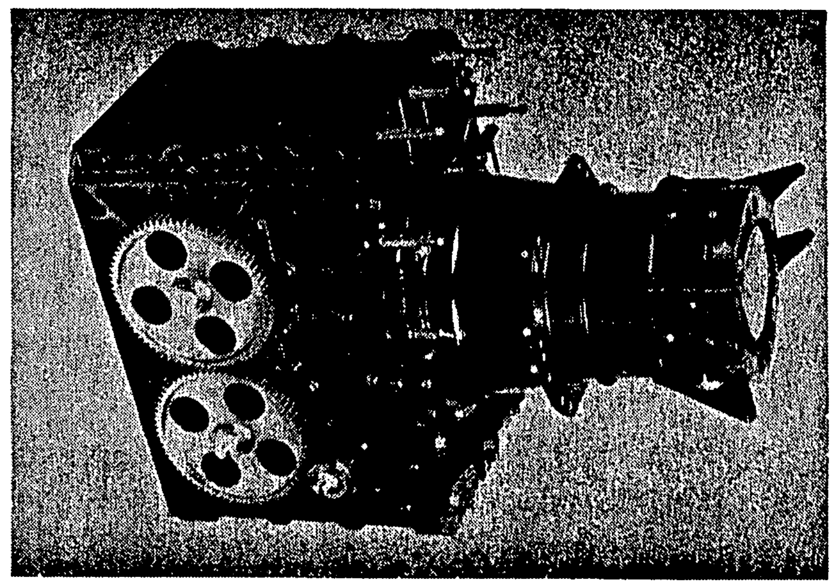
Field uniformity over the 24° field-of-view is achieved by stopping down a 50mm, f/0.95 lens to f/1.2. The signal processing electronics consumes less than 1.9 Watts in generating a star position pulse for star crossings detected by each channel, and a corresponding star intensity signal. Eight commandable thresholds permit selection of the stars to be detected within the range of -1.4 to +1.7 silicon magnitude. Probability of detection of a +1.7 magnitude star is a high 99%. The range can be extended to +2.5 magnitude with a reduced probability of detection.

The CS-201's accuracy of 10 arc minutes (1 sigma) was dictated by mission specifications. Other BASD solid state scanners are designed for sub-arc minute accuracy.

BOOM EXTENSION DEVICES

- WILL EMPLOY BI-STEM UNITS OF SPAR DESIGN SUPPLIED BY ASTRO RESEARCH CORP EXTENSION/RETRACTION SYSTEM WILL BE CONTROLLABLE BY STEPS AND SYNCHRONIZED TO MAINTAIN BALANCE
- UNITS WERE QUALIFIED FOR CTS (CANADIAN COMMUNICATIONS TECHNOLOGY SATELLITE) AS SOLAR PANEL EXTENDER DEVICES. NEW BRUSH-LESS DC MOTORS WITH ELECTRONIC DRIVE CIRCUITS WILL BE UTILIZED
- BOOMS WILL BE 1.38" DIAMETER (~ 0.1 LB/FT) WITH "ZIPPERED" JOINTS CAPABLE OF 26' EXTENSION (i.e., 3+ SPACECRAFT DIAMETERS)
- REQUALIFICATION WILL BE PERFORMED AS REQUIRED. ENVIRONMENTAL LIMITATIONS ARE READILY ACHIEVABLE





INTERLOCKED BI-STEM

The STEM (Storable Tubular Extendible Member) is essentially an element of thin metallic material which assumes a tubular shape of high strength when extended. It can be stored in a minimum of space when coiled in the flattened condition on a spool.

The BI-STEM consists of two elements formed into an open circular section, such that each circumscribes an angle of approximately 330 degrees. One element is then placed inside the other, diametrically opposed, thus forming the basic BI-STEM element. A further development of the BI-STEM is the Interlocked BI-STEM, which provides much greater torsional stiffness and strength by connecting the edges of the elements with a tab/slot arrangement. This is the unit which will be employed on MGO/LGO.

New drive motors (brushless D.C.) with electronic drive circuits would be qualified and utilized. Commanded circuitry will be developed to drive the STEM units out and back to maintain spacecraft dynamic balance.

STS SAFETY REQUIREMENTS

- CRITICAL HAZARDS - NO SINGLE FAILURE SHALL RESULT IN DAMAGE TO STS EQUIPMENT OR IN THE USE OF CONTINGENCY OR EMERGENCY PROCEDURES

- CATASTROPHIC HAZARDS - NO COMBINATION OF TWO FAILURES OR RF SIGNALS SHALL RESULT IN THE POTENTIAL FOR PERSONAL INJURY, LOSS OF THE ORBITER, GROUND FACILITIES, OR STS EQUIPMENT

- SRM-1 FIRING - P/L MUST COAST FOR AT LEAST 45 MINUTES WITH MINIMUM SEPARATION VELOCITY OF 1 FT/SEC BEFORE MOTOR IGNITION

- RCS FIRING - FOR THRUSTERS 10 LBS. OR LESS, THE MINIMUM SAFE FIRING DISTANCE AFTER STS SEPARATION IS 200 FT. FROM THE ORBITER

- OTHER
  - PREMATURE P/L DEPLOYMENT AND/OR SEPARATION
  - ADIABATIC DETONATION OF HYDRAZINE
  - STRUCTURAL/ELECTRICAL DESIGN
  - EMC

STS ADAPTATION

In addition to meeting STS safety requirements - as noted above - the adaption of the spacecraft to the STS and upper stage environment will require detailed investigations for load and other environmental verifications. In themselves, the safety requirement impose minimal constraints and may require no-to-minor changes in the RCS and ordnance systems.

The acoustic environment spectra for payloads launched on an Atlas-Centaur are somewhat more benign than on the STS. The STS-2 spectrum represents a single measurement from the second Space Shuttle flight test. This is the maximum environment measured to date. Due to the limited STS acoustic data there has not been an updated payload bay environment established. This measurement is therefore representative of a maximum local response which may increase after additional measurements have been made. The STS-2 environment generally exceeds the Atlas-Centaur environment below 100 Hz. This will result in increased response of large surfaces, such as solar arrays which have major reasonances below 100 Hz. Electronic components are generally sensitive to excitation between 100 and 1000 Hz where circuit boards and other internal parts resonate. In this intermediate frequency interval, the STS and Atlas-Centaur environments are essentially equivalent. With the exception of the discrete frequency excitation at 315 Hz on STS, the FLTSATCOM equipment is qualified for both launch vehicles. Launch on the Shuttle would require an evaluation of solar array for the increased low frequency environment. Further study will be needed to define existing qualification levels of the new components and determine the margins available.

DESCRIPTION OF MAXIMUM COMMONALITY* SPACECRAFT

	MGO	LGO
• LAUNCH SYSTEM	STS/SRM-1	STS/PAM-A
• ADDED PROPELLANT CAPACITY	NO	NO
• 2 AXIS HGA	YES	NO*
• TT&C (BANDS)	S+X	S* ONLY
• CONICAL HORIZON SCANNERS	YES	YES
• CATEGORY 1 PAYLOAD ONLY	YES	YES
• SPINNING STELLAR SENSOR	YES	NO*
• ACTIVE NUTATION CONTROL	NO	NO
• STEM DEVICES	YES	YES
• STAR 37 MOTOR	FM	N*
• SIMILAR POWER SUBSYSTEM (SIZE OF ARRAYS, BATTERIES, PCU, ETC)	YES	YES (CELLS WIRED DIFFERENTLY)
• MOMENTUM WHEEL MOUNTING SAME AS FSC	YES	YES

* NON-COMMON ITEMS

THE QUESTION OF COMMONALITY

There is a subtle question of how much commonality should be strived for if both MGO and LGO come into existence simultaneously. The above shows what might be a reasonable compromise, but still questions remain. For example, since the MGO must fly on the STS, would it not be more efficient to also fly LGO on the STS, instead of verifying Atlas-Centaur for compatibility with the new design?

It does seem clear that commonality for commonality's sake would not pay - particularly if equipping LGO with an unneeded HGA or an overkill insertion motor.

AVAILABLE AUTOMATIC FAIL SAFE MODE
(WHICH CAN BE DISABLED ON COMMAND)

- INHIBIT OF PITCH THRUSTER REP. RATE (1000 SEC. INTERVAL)
- SHUT DOWN OF HYDRAZINE SUPPLY, WHEN
  - HORIZON SCANNER SIGNAL LOST FOR > 4 SECS (POWER DOWN ⟹ SLOW TUMBLE)
  - THRUSTER FIRING COMMAND > 4 SECS
  - THRUSTER FIRING AT > 20% DUTY CYCLE
- SWITCH BATTERY CHARGE CHANNELS, IF
  - BATTERY OVERTEMP
  - CHARGE CURRENT EXCEEDS PRESET VALUE
- TURN OFF PAYLOAD IF
  - VOLTAGE DROPS BELOW PRESET VALUE

OTHER THAN ABOVE, FSC HAS NO AUTONOMOUS OR SAFE HAVEN FEATURES

FAULT TOLERANCE AND CONTINGENCY ANALYSIS

Here we begin a series of pages dealing with chief concerns in the conversion of Earth orbiting spacecraft to planetary orbiters. The concerns are twofold:

1. Will the spacecraft die before a problem is detected and corrected?

2. Are there enough controls available to defeat the "worst" contingencies?

The latter question is the more difficult. A detailed contingency analysis was performed for FLTSATCOM. It will have to be repeated for MGO (and LGO). FLTSATCOM may not pass the test of the first question without some modifications, as will now be treated:

- The FLTSATCOM-based MGO/LGO spacecraft have some fault tolerant features built into their TT&C (switchover if lock lost), power (overheat, overload, overvoltage), thermal control (temp. limits), and ACS subsystems (loss of propellant).

- A deficiency, especially for MGO, is prevention of the lack of an automatic "safe haven" mode that could maintain spacecraft health for one to two days. An example would be an automatic power-down, with solar panel rotation stopped and panels pointing towards the sun. The mode would be keyed by any one of several criteria.

- Implementation of such a system will be accomplished. The FSC already incorporates a sun lock-on mode as part of its nadir capture sequence. As suggested above, a solar search will be activated by proper cues. Communications will be maintained via omni. In the next program stage, this is a vital area of investigation.

BUILT-IN "BACK-UP" SYSTEMS

- SPINNING SENSORS PERMIT ADJUSTMENT OF SPIN AXIS PRIOR TO MGO SRM-1 BURN⟶MINIMIZES MID-COURSE HYDRAZINE.
- CONICAL SCANNER (MGO + LGO) YIELDS HIGHLY ACCURATE ORBITAL ALTITUDE DATA (MORE SO, AFTER CALIBRATION)⟶OBVIATES NEED FOR TRACKING FOR LONG PERIODS.
- SPINNING STELLAR SCANNER (MGO) USED FOR LAUNCH AND CRUISE-OUT ORIENTATION CAN ACT AS BACK-UP FOR ON-ORBIT OPERATION. COULD BE ADDED TO LGO TO PROVIDE ATTITUDE DATA DURING ECLIPSES.
- INSTRUMENTS AND LOGIC ARE AVAILABLE (MGO, LGO) TO PERMIT AUTONOMOUS SAFE HAVEN (LOCK-ON-SUN) MODE UPON PROPER CUES.
- PERFORMANCE MARGINS PERMIT WIDE VERSATILITY OF MISSION OPERATION.

POLICY ON RELIABILITY, REDUNDANCY, AND SAFING

Reliability - No changes to basic FSC policy. 100% high reliability parts will continue to be employed. Failure mode analyses would be checked for key mission milestones. No spacecraft modification envisioned.

Redundancy - No changes to fully redundant FSC design. Redundant units will normally (see "safing" below) be commanded. No modification forseen, although HGA on MGO will require special study.

Safing - Policy will be subject of intensive study in next phase of program. The main "trouble" triggers will be (1) loss of horizon, (2) failure to acknowledge receipt of "NO-OP" command signal transmitted by 34m DSN dish at beginning of each DSN shift,* (3) loss or abnormal decline in solar power, and (4) the other propulsion-allied triggers already built-in. Other problems (under/over heat, e.g.) will be normally corrected and reported on via housekeeping data. The triggers will in every case power down the spacecraft to minimum, latch propellant valves, and unpower the momentum wheel. Subsequent "safe havens" that will be studied are:

- Enabling "new" redundant batteries to extend spacecraft battery-only life in powered-down state to 30-40 hours.

- By "simple" programmable sequencer, direct spacecraft to point at sun with solar panels normal to sun. "Roll" around sun line will be permitted. This modification would be considered "moderate-to-major". (See details next page.)

- A new ground based automatic system called "Auto Safe" is now in operation at remote sites to support FSC. The software can examine any housekeeping signal and take pre-planned action if an error signal is out of limits. In a highly sophisticated modification, the "Auto Safe" software and computer could be placed aboard the spacecraft. This would be a major modification, and its impact on intrinsic reliability would have to be studied.

* or, TBD, at each eclipse emergence (plus 5 minutes).

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

CONTINGENCIES

- Procedures have been written for major contingency operation
    - Spacecraft equipment converter dump, high tranverse rate, no spin-up at separation, etc.
- Procedures captured in computer software (block commands) for time critical reactions
- Non-time critical contingencies captured in pass plans
- Contingency procedures updated based on spacecraft configuration/status
- Possible upgrades
    - Place ground based "auto safe" software in on-board computer
- On-board software would allow automatic "safe haven" for spacecraft (sun acquisition)
- On-board software could facilitate Mars acquisition
- On-board software could be used for array updates

DOWNLINK DATA RATE REQUIREMENTS FOR MGO MISSION

- OPERATIONAL ASSUMPTIONS FOR WORST-CASE DOWNLINK RATES:
  - 8 HOUR DSN SHIFT PER 24 HOURS
  - 63% ORBIT TIME IN EARTH VIEW
  - 5 MINUTE LOCKUP TIME PER CONTACT, 4 CONTACTS PER SHIFT
  - MSM AVERAGE DATA RATE IS 1 KBS FOR EACH 24 HR PERIOD
- DATA DUMP TIME FOR 24 HOUR PERIOD IS 288 MINUTES (WORST CASE)
- TOTAL DOWNLINK RATE INCLUDES OVERHEAD FOR ENGINEERING DATA (1/8 OF EACH FRAME)

MGO MISSION	24 HR DATA RATE (KBS)	DOWNLINK SCIENCE RATE (KBS)	DOWNLINK TOTAL RATE (KBS)
4 INSTRUMENTS - PRIMARY MGO	3.5	17.5	19.7
6 INSTRUMENTS - 4 PRIMARY PLUS IR RADIOMETER & IR SPECTROMETER (FROM AMES MARS CLIMATOLOGY MISSION)	3.8	19.0	21.4

DOWNLINK DATA RATE REQUIREMENTS FOR MGO MISSION

The downlink data rate requirements drive the selection of downlink antenna size and on-board tape recorder storage capabilities. To economize on this equipment selection, the facing page downlink sizing assumes an <u>average</u> MSM data collection rate of 1 KBS per 24 hour period.

Data dumps are scheduled to occur during a single 8-hour DSN shift for every 24-hour period of data collection. Under worst-case conditions, only 288 minutes of actual contact time are available due to the worst-case 63% orbit time viewing of the earth. For each shift, the contact time is distributed over four periods of 72 minutes in worst-case, which are reduced to 67 minutes each after allowing 5 minutes lockup time.

If it is required to acquire MSM data, under worst-case conditions at the peak rate of 12 KBS at a 50% duty cycle over a full 24-hour period, the MGO mission downlink data rate requirements must be substantially increased, as shown in the table below.

MGO Mission (MSM at 6 KBS)	24-Hour Data Rate (KBS)	Downlink Science Rate (KBS)	Downlink Total Rate (KBS)
Primary MGO - 4 instruments	8.5	45.7	51.4
6 Instruments - 4 Primary plus 2 IR Ames Instruments	8.8	47.3	53.2

DOWNLINK DATA RATE REQUIREMENTS FOR LGO MISSION

- OPERATIONAL ASSUMPTIONS FOR WORST-CASE DOWNLINK RATES:
  - 8 HOUR DSN SHIFT PER 24 HOURS
  - 39.5% MAXIMUM EARTH ECLIPSE TIME
  - 5 MINUTE LOCKUP TIME PER CONTACT, 4 CONTACTS PER SHIFT
  - MSM AVERAGE RATE IS 1 KBS PER 24 HOUR PERIOD

- DATA DUMP TIME FOR 24 HOUR PERIOD IS 266 MINUTES (WORST CASE)

LGO MISSION	24-HR DATA RATE (KBS)	DOWNLINK DATA RATE (KBS)	
		SCIENCE DATA	TOTAL DATA
4 INSTRUMENTS - PRIMARY LGO	3.4	18.4	21.6
6 INSTRUMENTS, WITH RADAR ALTIMETER REPLACED BY LASER ALTIMETER	13.5	73.1	86.1
6 INSTRUMENTS, WITH RADAR ALTIMETER	4.1	22.1	26.1

- S-BAND LINK CAPABILITY EXCEEDS 1 MBPS FOR 34M DSN ANTENNA
  - AMPLE LINK MARGIN

- LGO MISSION WITH LASER ALTIMETER REQUIRES SPECIAL CONSIDERATION
  - ALL OTHER LGO MISSIONS ACCOMMODATED BY 24 KBS DOWNLINK RATE

DOWNLINK DATA RATE REQUIREMENTS FOR LGO MISSION

The downlink data rate calculations for the LGO mission are similar to the MGO mission calculations, except that the downlink S-band capability is very high — exceeding 1 MBPS for the 34m DSN antenna — so that the limiting constraint is the 24-hour data storage capability for the on-board recorder. To maintain similar tape recorder requirements for primary MGO and LGO missions, the facing page calculations assume an average 24-hour MSM data acquisition rate of 1 KBS.

If it is required to acquire MSM data, under worst-case conditions at the peak rate of 12 KBS at a 50% duty cycle over a full 24-hour period, the LGO mission downlink data rate requirements must be substantially increased, as shown in the table below.

LGO Mission (MSM at 6 KBS)	24-Hour Data Rate (KBS)	Downlink Science Rate (KBS)	Downlink Total Rate (KBS)
Primary LGO - 4 instruments	8.4	45.5	51.2
6 Instruments (with laser altimeter)	18.5	100.2	112.7
6 Instruments (with radar altimeter)	9.1	49.3	55.5

MGO DOWNLINK MISSION DATA ANTENNA SIZING

DOWNLINK RATES FROM X-BAND HGA TO DSN BY ANTENNA SIZE

TO 64M DSN

	HGA ANTENNA SIZE		
ORBITAL RANGE	1.15M	1.55M	2.0M
CLOSEST (0.99 AU)	100 KBS	181 KBS	303 KBS
MAXIMUM (2.66 AU)	14 KBS	25 KBS	42 KBS

TO 34M DSN

	HGA ANTENNA SIZE		
ORBITAL RANGE	1.15M	1.55M	2.0M
CLOSEST (0.99 AU)	28 KBS	51 KBS	85 KBS
MAXIMUM (2.66 AU)	4 KBS	7 KBS	12 KBS

WORST-CASE MGO REQUIREMENTS SATISFIED BY 1.55M HGA TO 64M DSN ANTENNA

- WORST CASE REQUIRED IS 23 KBS FOR MGO (WITH MSM AT 1 KBS AVERAGE RATE)
- SOME DATA LOSS WILL OCCUR NEAR MAXIMUM RANGE USING 34M DSN ANTENNA, MINIMIZED BY SELECTING 2M HGA ANTENNA

FOR 34M DSN ANTENNA, SOME DOWNLINK DATA WILL BE LOST

- AT DISTANCES CLOSE TO MAXIMUM RANGE
- WORST-CASE REQUIRES 24-HOUR DATA DUMP IN 268 MINUTES, ALLOWING FOR 5-MINUTE LOCK-UP TIME

WORST-CASE DATA LOSS FOR DOWNLINK TO 34M DSN ANTENNA ASSUMING MSM AVERAGE RATE OF 1 KBS PER 24-HOUR PERIOD

X-BAND HGA (M)	MAX DISTANCE WITH NO DATA LOSS (AU)
1.15	1.09
1.55	1.49
2.00	1.92

FOR 34M DSN ANTENNA, SOME DOWNLINK DATA WILL BE LOST

- AT DISTANCES CLOSE TO MAXIMUM RANGE
- WORST-CASE REQUIRES 24-HOUR DATA DUMP IN 268 MINUTES, ALLOWING FOR 5-MINUTE LOCK-UP TIME

WORST-CASE DATA LOSS FOR DOWNLINK TO 34M DSN ANTENNA ASSUMING MSM AVERAGE RATE OF 1 KBS PER 24-HOUR PERIOD

X-BAND HGA (M)	MAX DISTANCE WITH NO DATA LOSS (AU)
1.15	1.09
1.55	1.49
2.00	1.92

MGO DOWNLINK MISSION DATA ANTENNA SIZING

Downlink data rates to DSN 64m and 34m antennas are shown in the facing page as a function of X-band HGA size selection, for reasonable available antenna sizes.

Assuming an average MSM data acquisition rate of 1 KBS per 24 hour period, the worst-case downlink conditions can be satisfied by an HGA antenna size of 1.55m, provided the 64m DSN antenna is used. For the 34m DSN antenna, some downlink data loss will occur at large distances from earth. The maximum distance from earth at which no data loss occurs is shown in the following chart, as a function of antenna size.

Due to the repetitive opportunities for data acquisition, it may be reasonable to tolerate some data loss in favor of economizing on HGA antenna size and tape recorder capabilities.

If it were required to support the MSM 6 KBS data rate (over 24 hours), the resulting downlink rate of 51.4 KBS for the primary MGO would require an HGA X-band antenna well in excess of 2m to avoid worst-case data loss to the 64m DSN antenna at maximum distance. Further, the maximum distance with no downlink data loss to the 34m DSN antenna from a 2m HGA would be reduced to 1.29 AU, which is only 48% of the maximum distance. To improve on this would require a substantially larger HGA antenna.

We have arbitrarily decided to limit data rate capability to be compatible with a 2m HGA.

MGO COMMUNICATIONS REQUIREMENTS FOR ORBIT INSERTION

- COMMAND AND TELEMETRY LINKS ARE REQUIRED FOR ORBIT INSERTION MANEUVERS
  - PRIOR TO DEPLOYMENT OF HGA ANTENNA
- COMMAND UPLINK PROVIDED BY S-BAND FORE/AFT OMNI ANTENNA FROM 64 M DSN ANTENNA
  - 125 BPS REQUIRED
  - WORST-CASE LINK OF 260 BPS IS ADEQUATE
- TELEMETRY DOWNLINK REQUIREMENTS DEPEND ON TRANSFER ORBIT TYPE AND
  - YEAR OF LAUNCH: AFFECTS INSERTION DISTANCE, HENCE, SPACE LOSS FOR ANTENNA GAIN
  - ORBIT INSERTION: ZAP ANGLE, HENCE SPACECRAFT EARTH-ASPECT ANGLE, HENCE CHOICE OF ANTENNA GEOMETRY OR EIRP
- TBD DOWNLINK ANTENNA SELECTION DEPENDS ON OPTION TO:
  - MINIMIZE COST BY TARGETING FOR SPECIFIC LAUNCH MISSION, OR
  - OPTIMIZE VERSATILITY WITH MULTIPLE ANTENNAS FOR ALL MISSIONS
- SELECTION MAY AFFECT USE OF DOWNLINK FORE/AFT OMNI ANTENNAS, POWER, AND BAND SELECTION.

ORBIT INSERTION REQUIREMENTS

The factors affecting antenna selection for insertion downlink are shown in the table below:

LAUNCH YEAR	TRANSFER TYPE	INJECTION DISTANCE (AU)	ZAP ANGLE RANGE (DEG)	EARTH ASPECT ANGLE FOR SPACECRAFT
1988	I	1.23	104 - 135	$\lvert ANGLE \rvert < 60°$
1990 *	II	2.49	45 - 60	$\lvert ANGLE \rvert > 100°$
1992	II	2.39	45 - 60	$\lvert ANGLE \rvert > 100°$

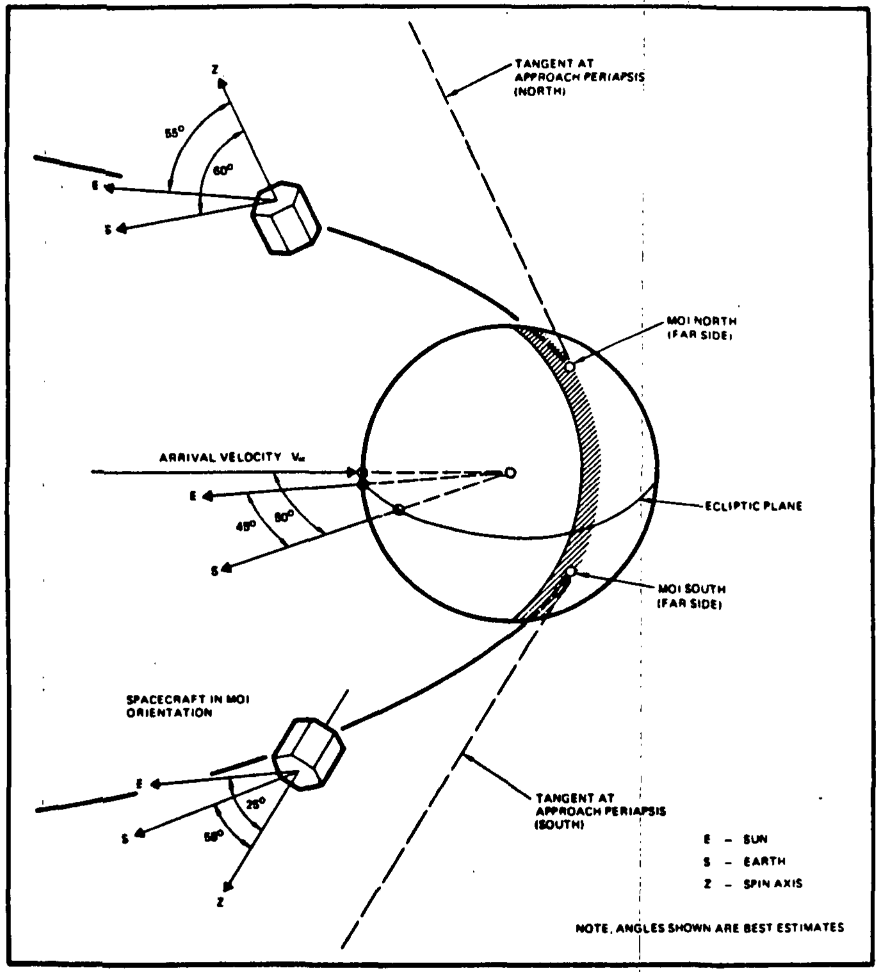
The candidate antennas for 16 bps downlink to 64m DSN at orbit insertion are:

- 1988, 1990 missions: X-band horn (small aspect angle), gain = 3 db
- 1990, 1992 missions: X-band bicone (large aspect angle), gain = 9 db

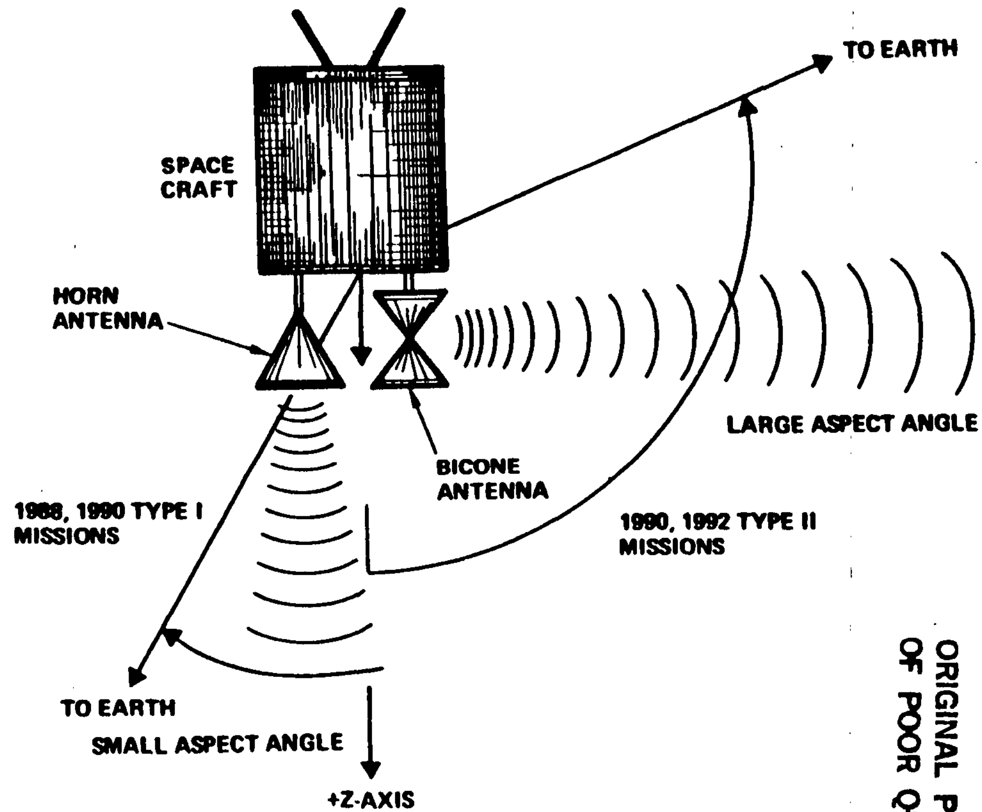
Refer to figures on next page set for relation of aspect angle to antenna geometry.

* Could also be Type I transfer

MGO ANTENNA SELECTION FOR ORBIT INJECTION DOWNLINK



REPRESENTATIVE EARTH/Z-AXIS AND SUN/Z-AXIS ANGLES AT SPACECRAFT MOI ORIENTATION (1988 MGO MISSION, TYPE I TRANSFER, ARRIVAL 21 DECEMBER 1988)



MGO COMMUNICATION SUBSYSTEM ANTENNAS

The normal mission data downlink via HGA X-band antenna includes interleaved engineering data.

The normal engineering data downlink via HGA S-band antenna serves as backup for mission data downlink.

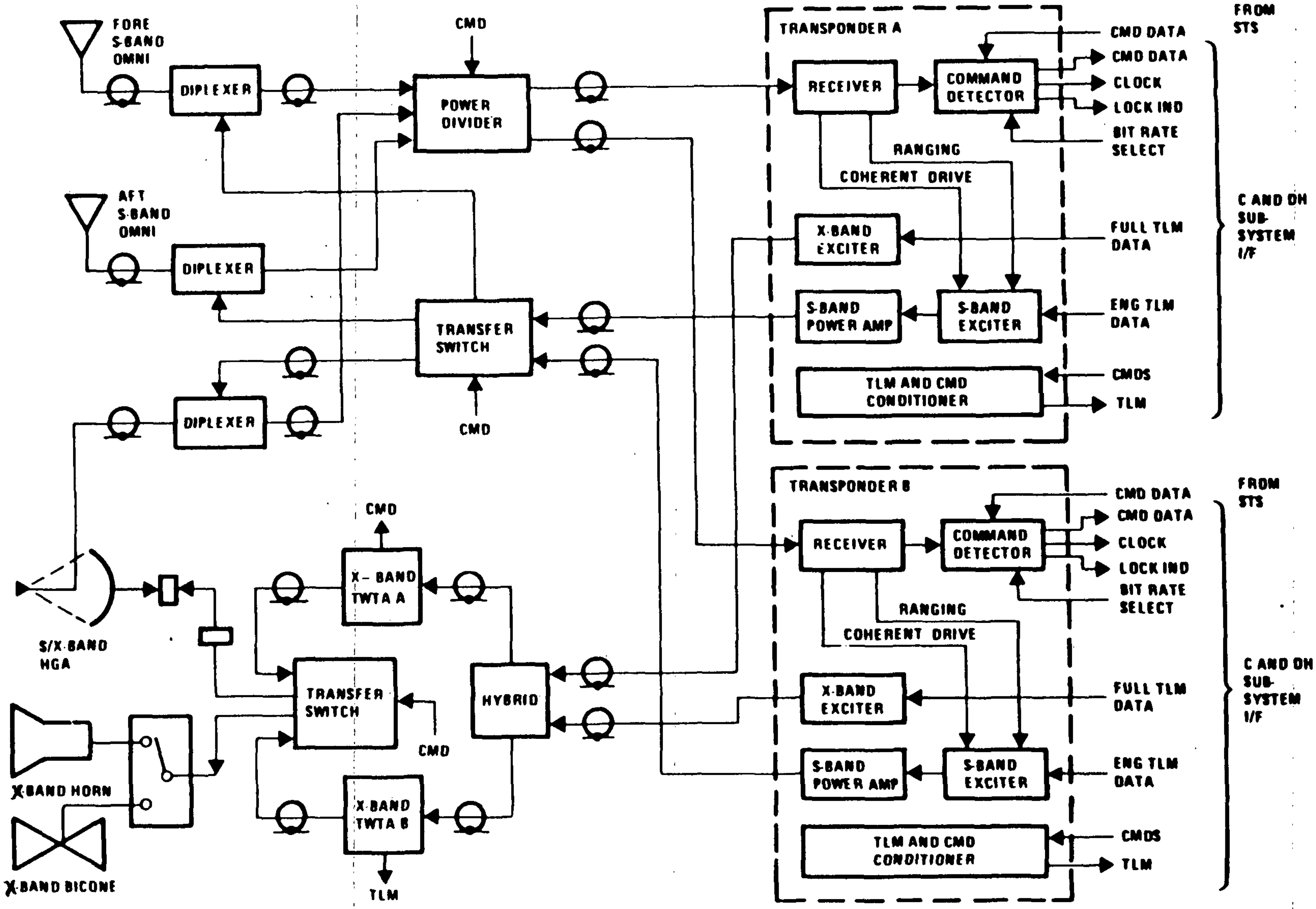
The Mars orbit insertion, maneuvers, while HGA is stowed, uses the DSN 64m antenna, and

- Downlink: S-band omni, fore or aft
- Uplink: X-band horn or bicone
- For 1988, 1990 Type I missions: Fore omni and horn antennas
- For 1990, 1992 Type II missions: Aft omni and bicone antennas

An MGO economy option would delete the S-band HGA downlink:

- Pro: Saves three S-band diplexers, S-band power amp and exciter
- Con:
  - No backup for X-band HGA downlink
  - Loss of commonality with LGO design
  - No spacecraft-to-STS downlink

MGO COMMUNICATIONS SUBSYSTEM BLOCK DIAGRAM



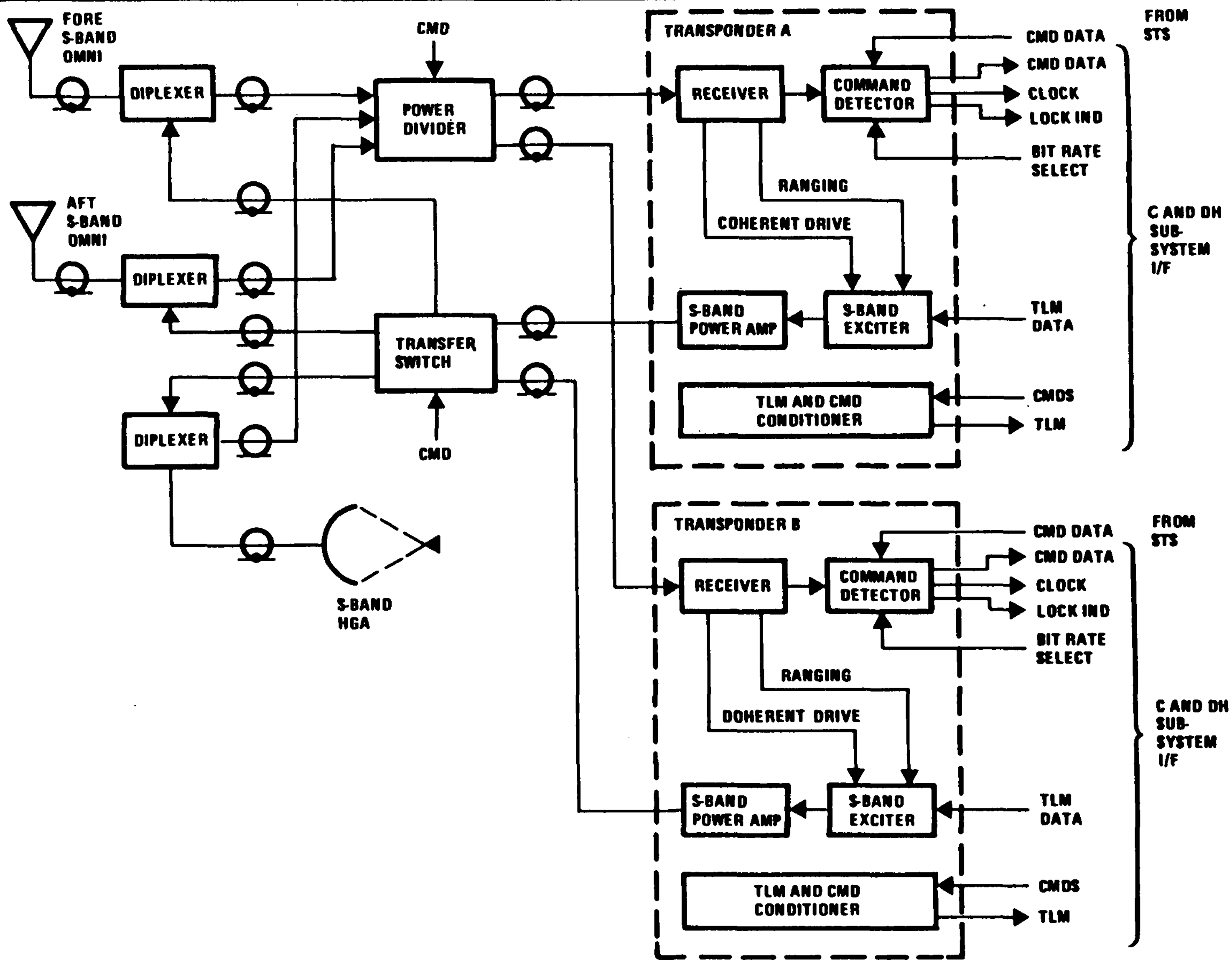
MGO COMMUNICATIONS SUBSYSTEM EQUIPMENT

The definition and heritage of the equipment in the above block diagram is shown in the table below. The downlink antenna(s) for MOI (and cruise) are not shown. They were discussed previously. For LGO, all X-band equipment (as designated by *) is deleted.

EQUIPMENT	NO. PER S/C	UNIT			SUBSYSTEM		PROGRAM DERIVATION	STATUS		
		MASS (KG)	SIZE (CM) LxWxH	POWER (W)	MASS (KG)	POWER (W)		AS IS	MOD	NEW
S/X-BAND TRANSPONDER* (Incl. 5W S-Band P.A.)	2	5.1	28.5x14.0x7.9	36.8	10.2	44.2†	NASA STANDARD (DEEP SPACE)		X	
S-BAND DIPLEXER	3	0.65	20.8x9.9x8 9	--	2.0	--	PIONEER 10&11, DSP	X		
S-BAND TRANSFER SWITCH	2	0.14	4.5x3.3x6.4	--	0.28	--	HEAO, DSP	X		
S-BAND OMNI ANTENNA	2	1.0		--	2.0	--	HEAO, DSP, PIONEER 10&11	X		
X-BAND TWTA (20W)*	2	5.2	36.7x15.2x11.4	72.0	10.4	72.0	DSCS II, LANDSAT D		X	
X-BAND TRANSFER SWITCH*	1	0.17	6.1x5.5x3.5	--	0.17	--	DSCS II	X		
X-BAND HYBRID*	1	0.30	4.5x2.5x1.9	--	0.30	--	DSCS II	X		
S/X-BAND HGA*	1	10.2	2 m Dish	--	8.0	--	GRO		X	
RF CABLES/CONNECTORS	Set	1.0	1 Set	--	1.0	--	--			X
WAVEGUIDE	Set	0.5	1 Set	--	0.5	--	--			X
MGO TOTAL	N/A	N/A	N/A	N/A	34.9	116.2	N/A	N/A	N/A	N/A
LGO TOTAL					24.0	44.2				

† S/X Transmitters In Redundant Transponder Turned Off

LGO COMMUNICATIONS SUBSYSTEM BLOCK DIAGRAM



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SYSTEMS ENGINEERING

MGO/LGO COMMUNICATIONS SUBSYSTEM DIFFERENCES

The differences between the communications subsystems for MGO and LGO missions are due to additional requirements for the MGO mission:

- Downlink requirements over larger distances, for which the X-band high gain antenna (HGA) and its support equipment are provided.
- Mars orbit injection (MOI) downlink requirements while the HGA antennas are stowed, for which X-band horn and bicone antennas are provided; the choice of horn or bicone antenna depends on the aspect angle at MOI for the spacecraft relative to Earth, which is a function of launch year. If it is required for the same spacecraft design to satisfy all the transfer types under consideration, both of these auxiliary antennas would be included.

The S-band HGA is not essential for the MGO mission, but it is the data downlink antenna for the LGO mission. In addition to providing commonality between MGO/LGO missions, using the S-band HGA in the MGO mission provides a backup for data downlink and provides a downlink to the STS. In addition, the S-band HGA may be used to downlink engineering telemetry data (as shown in the MGO block diagram) instead of interleaving it with the science data.

The two S-band omni-antennas provide spherical coverage, as in the TRW HEAO and FLTSATCOM spacecraft, to provide near-earth uplink communications, or critical MOI commanding from the 64m DSN antenna.

DEEP SPACE NETWORK USAGE FOR MGO/LGO MISSIONS

ON-ORBIT OPERATION

- DSN USE: 34 M ANTENNA USED ROUTINELY
  - 4 PERIODS OF 72 MINUTES EACH DURING ONE 8-HOUR SHIFT PER 24 HOURS
  - 64 M ANTENNA USED FOR SPECIAL CONDITIONS OR CONTINGENCIES
  - AMPLE LINK MARGINS FOR LGO MISSION
- UPLINK:
  - COMMANDS AND ORBIT DETERMINATION RANGING FROM 34 M ANTENNA VIA S-BAND
  - COMMAND STORE LOADED DAILY OR 3 TIMES/WEEK FOR SCIENCE OPERATIONS
  - ORBIT DETERMINATION TRACKING: ONCE PER WEEK
- MGO UPLINK:
  - WORST-CASE UPLINK RATE TO S-BAND HGA IS 2.5 KBS AT MAXIMUM RANGE
  - FOR 64 M ANTENNA, WORST-CASE UPLINK IS 260 BPS TO S-BAND OMNI (ADEQUATE FOR CRITICAL COMMANDING)
- DOWNLINK:
  - DATA DUMPS FROM ON-BOARD TAPE STORAGE, 4 PERIODS PER DAY VIA HGA
  - VIA X-BAND FOR MGO, S-BAND FOR LGO
  - REAL TIME ENGINEERING AND HOUSEKEEPING DATA INTERLEAVED WITH MISSION DATA (MAY BE DOWNLINKED VIA S-BAND HGA FOR MGO)
- MGO DOWNLINK: FULL DATA DUMPS AT AVERAGE INSTRUMENT DUTY CYCLES
  - EXCEPT FOR RESTRICTIONS ON MULTISPECTRAL MAPPER NEAR MAXIMUM RANGE
  - NON-COHERENT X-BAND TRANSMISSION

SPACECRAFT MONITORING DURING STS FLIGHT

The shuttle avionics supports communication with attached payloads like the MGO (via orbiter comm and tracking subsystem):

- Reception of ground to STS commands
- Transmission of commands to payload spacecraft
- Reception of payload telemetry
- Transmission of real-time or stored payload data to ground

The attached payload interfaces with orbiter payload distribution panel (PDP)

The STS provides three ground links for transmission of payload telemetry:

- S-band PM to TDRS system or NASA STDN stations
- S-band FM to STDN (limited availability)
- Ku-band to TDRS system (mode 1 QPSK or mode 2 FM) - this access is not planned for MGO

The STS also supports detached payload telemetry transmission via same 3 links

- Payload spacecraft interfaces with STS payload interrogator (PLI)
- Via S-band telemetry link (2200 to 2300 MHz)

The latter link could be used to command precession of MGO spin axis prior to SRM-1 burn, since STS will be in the line of sight.

COMMAND AND DATA HANDLING (C&DH) SUBSYSTEM

COMMAND AND DATA HANDLING (C&DH) SYSTEM REQUIREMENTS

- Receive and execute commands from DSN ground stations in any spacecraft attitude, except during Mars/Lunar occultations
  - HGA provided with adequate pointing capability
  - Link budget is adequate: 2.5 KBS via S-band from 34m DSN for worst on orbit case MGO during on-orbit operation
- Receive commands if HGA is mispointed, except during occultations
  - Fore and aft omni antennas allow for command uplink as required
  - 64m DSN link supports 260 bps in worst-case MGO (11 bps for 34m DSN)
  - Omni links may be automatically invoked after a TBD period without HGA commands receipt
- Provide real-time and stored command capabilities
  - Standard capability for proposed configuration
  - Supports up to 256 64-bit stored commands with up to 96-hour time-tags
  - Buffer for cyclical command sequences can be added
- Provide real-time engineering and housekeeping telemetry capabilities
  - Proposed configuration supports up to 256 telemetry points (analog, bi-level, and digital)
  - 4 stored telemetry formats available
  - Telemetry interleaved with stored data dumps by output multiplexer

COMMAND AND DATA HANDLING (C&DH) SUBSYSTEM COMMENTS

- MGO/LGO PROPOSAL SUGGESTED USE OF NASA STANDARD C&DH SUBSYSTEM
  - INCLUDES ON-BOARD COMPUTER AND STINT INTERFACE UNIT
- CURRENT PROPOSAL: USE GULTON C&DH SUBSYSTEM WITH ODETICS RECORDER
  - LOWER COST, OFF THE SHELF EQUIPMENT
  - MODULAR, EASILY MODIFIED COMPONENTS
  - EXTENSIVE TRW EXPERIENCE WITH GULTON C&DH (TDRSS, DSP PROGRAMS)
  - MGO/LGO MISSION DOES NOT REQUIRE ON-BOARD COMPUTER

EQUIPMENT LIST: CANDIDATE C&DH SUBSYSTEM

ITEM	UNITS	WEIGHT	POWER
COMMAND DECODER/PROCESSOR	2	4.9 LBS EA	5.7 W EA
DATA HANDLING PROCESSOR	2	8.5 LBS EA	10.5 W EA
REMOTE MULTIPLEXER	2	4.0 LBS EA	1.5 W EA
TAPE RECORDER	3	19.0 LBS EA	14/28* W EA
TOTAL	10	93.8 LBS	59.7 W

*ONE RECORDING (14 W), ONE SIMULTANEOUSLY PLAYING BACK (28 W)

COMMAND LOADING AND DUMP

THE COMMAND SOFTWARE FROM FLTSATCOM OPERATIONS CAN BE DIRECTLY TRANSLATED INTO MGO/LGO SOFTWARE. ADDITIONAL COMMANDS FOR THE PAYLOAD WILL HAVE TO BE GENERATED. THE REQUIREMENTS AND ALLOCATIONS ARE DISCUSSED BELOW:

- STORAGE OF COMMANDS AND TIME TAGS.

  CONSIDER 512 COMMAND SLOTS x 32 BITS/COMMAND = 16,384 BITS

- LOAD AND DUMP TIMES:

	NORMAL	EMERGENCY
LOADING	125 B/S 131 S (2.2 MIN)	7.81 B/S 2098 S (35 MIN)
READING OUT	4000 B/S 4.1 S	16 B/S 1024 S (17 MIN)

- TELEMETRY SCIENCE DATA FRAMES (1152 BIT FRAMES)

NON SCIENCE	WHEN RECORDED	WHEN PLAYED BACK
SYNCH	24	
ID	8	
TIME TAG	32	16
ENG'G (SUBCOM)	16	16
HOUSEKEEPING (SUBCOM)	8	8
FILL*	40	
TOTAL NON SCIENCE	128	
SCIENCE	1024	
TOTAL FRAME	1152	

$\frac{\text{TOTAL}}{\text{SCIENCE}} = \frac{9}{8}$; "OVERHEAD" $= \frac{1}{8}$

* FILL BITS ARE REPLACED BY REAL-TIME DATA AS INDICATED, WHEN PLAYED BACK FOR DOWNLINK TRANSMISSION.

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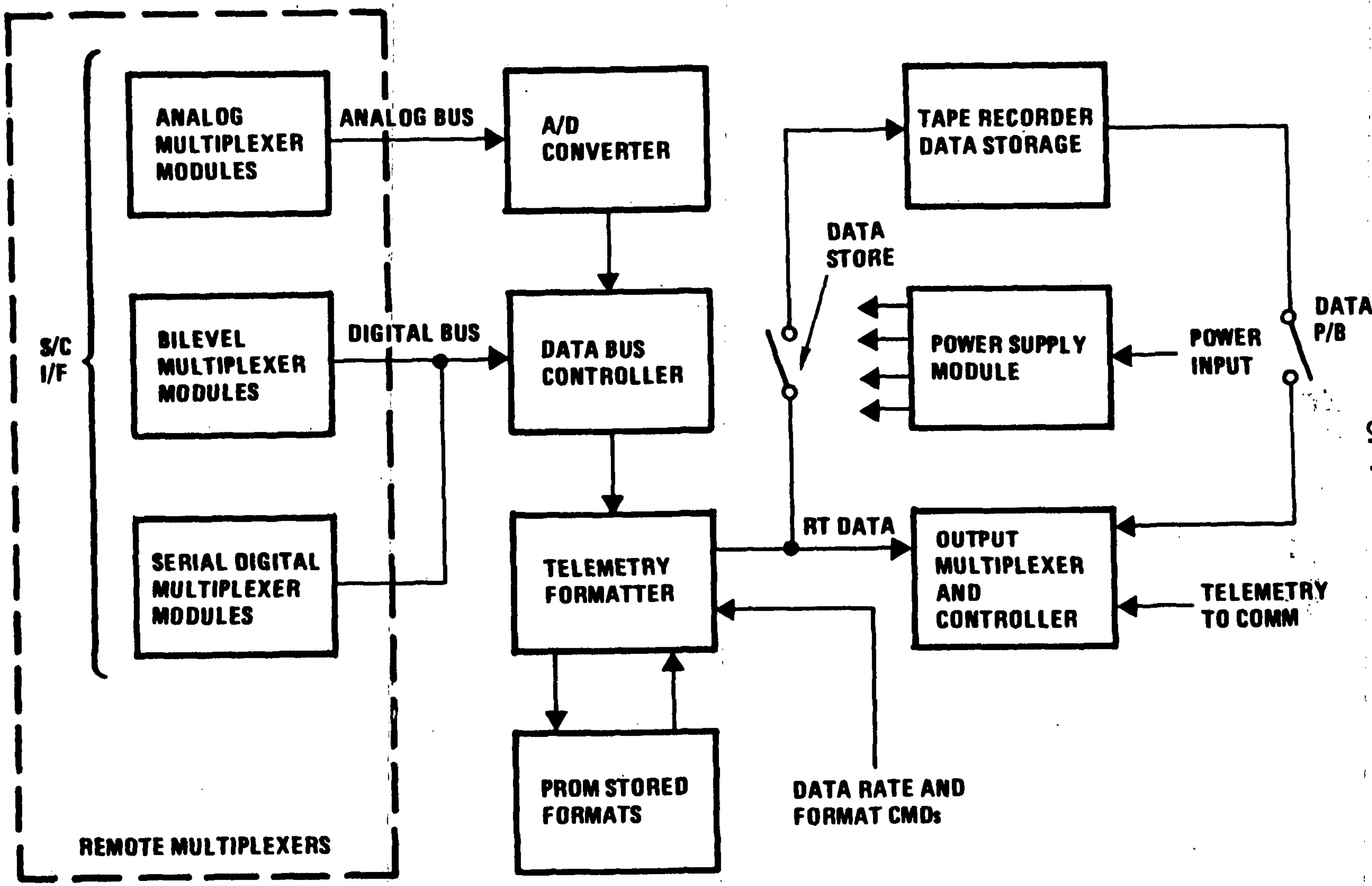
MGO/LGO DATA HANDLING SUBSYSTEM BLOCK DIAGRAM



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MGO/LGO DATA HANDLING SUBSYSTEM BLOCK DIAGRAM

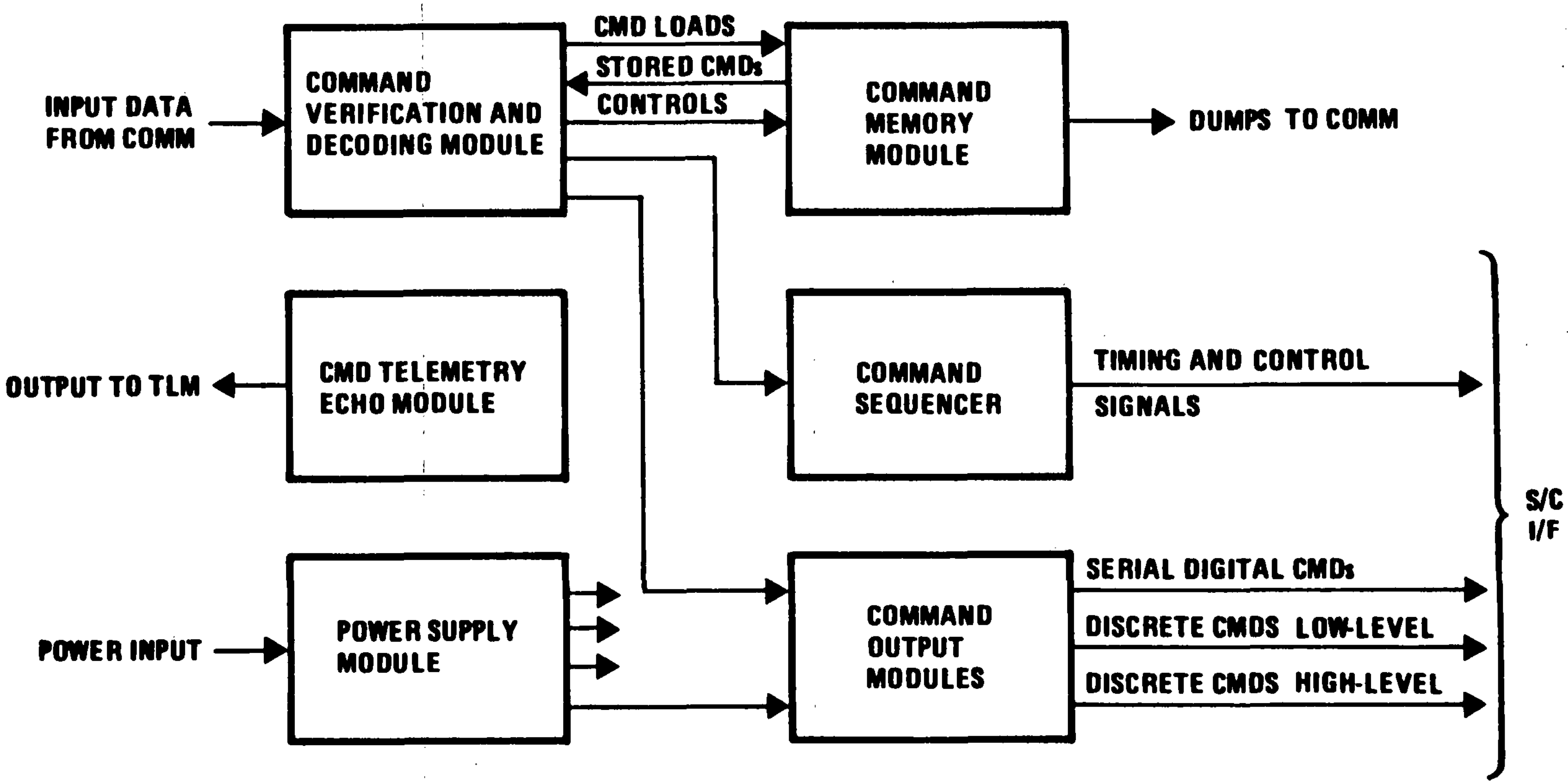
The data handling subsystems for MGO/LGO missions are the same, except possibly for the option of downlinking MGO engineering data via the S-band HGA, instead of interleaving it with the science data.

The equipment proposed for this subsystem is a standard Gulton data handling module, together with a suitable Odetics Series 3100 tape recorder (depending on the types of instruments required, as discussed in later charts).

The full range of MGO/LGO options can be accommodated by standard Odetics 3100 series tape recorders, except for the LGO mission with laser altimeter (if the MSM data rate average of 6 KBS is used), for which an Odetics 5000 series tape recorder would be used.

Use of the NASA standard C&DH subsystem for MGO/LGO was considered but discarded, due to higher costs and the lack of requirements for an on-board computer for MGO/LGO, which the NASA C&DH subsystem is designed to provide.

MGO/LGO COMMAND SUBSYSTEM BLOCK DIAGRAM



MGO/LGO COMMAND SUBSYSTEM BLOCK DIAGRAM

The command subsystem proposed for the MGO/LGO missions is the same for both, a standard Gulton commanding unit. The unit is compatible in capacity and format with the FSC command logic.

For storage of commands with time tags, 512 command slots have been provided, with 32-bit commands. Command loading rates, under normal conditions, are set at 125 BPS, which would require 2.2 minutes for a full load; however, the worst-case uplink from the 64m DSN antenna (to the S-band omni-antennas) supports up to 260 BPS rates, for a full load in 1 minute. From the 34m DSN antenna, the worst-case uplink rate (to an S-band omni) is 11 BPS, which could support an emergency full command load within 26 minutes.

For command dumps, the normal rate of 4 KBS is proposed, which yields a full command dump in 4.1 seconds. This rate is supported by an X-band HGA sized at 1.15m,in worst-case conditions to the 34m DSN antenna. For command dumps without the HGA antennas available, a 16 BPS downlink rate is provided by one of the auxiliary X-band antennas, the horn (with 3 db gain) or the bicone (with 9 db gain).

FLIGHT TAPE RECORDER DATA REQUIREMENTS (WITH AVERAGE MSM DATA RATE AT 1 KBS PER 24 HOUR PERIOD)

- DATA STORAGE REQUIREMENTS

MISSION	24 HR SCIENCE DATA (BITS)	24 HR TOTAL DATA (BITS)
MGO - 4 INSTRUMENTS	3.0×10^8	3.4×10^8
MGO - 6 INSTRUMENTS	3.3×10^8	3.7×10^8
LGO - 4 INSTRUMENTS	2.9×10^8	3.3×10^8
LGO - 6 INSTR (WITH RADAR ALTIMETER)	3.5×10^8	4.0×10^8
LGO - 6 INSTR (WITH LASER ALTIMETER)	11.7×10^8	13.1×10^8

- RECORD/PLAYBACK RATE REQUIREMENTS

MISSION	RECORD PEAK RATE	PLAYBACK RATE
LGO WITH LASER ALTIMETER	24.5 KBS	76.3 KBS
OTHER MGO/LGO (WORST CASE)	15.1 KBS	23.2 KBS

FLIGHT TAPE RECORDER DATA REQUIREMENTS

The facing page shows tape recorder requirements, assuming a 1 KBS average MSM rate over 24 hours, for the various MGO/LGO missions. An allowance of 1/8 per data frame has been included for engineering data in the column showing total data bits. Except for the LGO with laser altimeter, all missions can be satisfied with a tape recorder capacity of 4×10^8 bits.

If the 6 KBS rate over 24 hours is required for the MSM, the corresponding tape recorder requirement is doubled to 8×10^8 bits. The tape recorder requirements under these conditions are summarized below (excluding the 1/8 engineering data allowance per frame).

Mission (MSM at 6 KBS Rate)	24-Hour Data Storage (Bits)	Record Peak Rate (KBS)	Playback Peak Rate (KBS)
LGO with Laser Altimeter	16×10^8	24.5	100.2
Other MGO/LGO (Worst Case)	9×10^8	15.1	49.3

FLIGHT TAPE RECORDER CANDIDATES

- ODETICS DDS-3100 SERIES RECORDERS CAN SATISFY ALL MISSIONS (WITH MSM AVERAGE RATE AT 1 KBS OVER 24 HOUR PERIODS)

MISSION	DDS-3100 CAPACITY (BITS)	POWER (W) REC/PB	HERITAGE
MGO/LGO - 4 INSTRUMENTS	3.5×10^8	9/22	LMSC/SEASAT
ALL MGO/LGO EXCEPT LGO WITH LASER ALTIMETER	4.6×10^8	14/28	JPL/IRAS
LGO WITH LASER ALTIMETER	15.0×10^8	16/30	3 AF PROGRAMS

FLIGHT TAPE RECORDER CANDIDATES

The tape recorder requirements for the various MGO/LGO missions can be satisfied by Odetics DDS-3100 series recorders, as shown in the facing page table.

To support a 6 KBS MSM data rate over 24-hour periods, the requirements of 9×18^8 bits can be satisfied by the Odetics DDS-3100 model to be used for the JPL/GALILEO mission, with power record/playback requirements in the range 11/16 w to 20/10 w. For the LGO with laser altimeter mission requirement of 16×10^8 bits, the DDS-5000 model with storage capacity of 17×10^8 bits used by Air Force programs is applicable, with REC/PB requirements of 10/44 w.

FLIGHT TAPE RECORDER OPERATION

- TWO TAPE RECORDER SYSTEMS USED FOR SIMULTANEOUS DATA DUMP/RECORD
  - AS SHOWN IN TABLE BELOW, SIMULTANEOUS ACTIVITY ALTERNATES BETWEEN RECORDERS EVERY THIRD 8-HOUR PERIOD
  - REAL-TIME ENGINEERING DATA INTERLEAVED WITH SCIENCE DATA DUMP

- TAPE RECORDER OPERATION CYCLES

8-HOUR PERIOD	T/R 1	T/R 2
1	RECORD	DUMP
2, 3	RECORD	STANDBY
4	DUMP	RECORD
5, 6	STANDBY	RECORD

- REDUNDANCY OPTIONS
  (1) USE A THIRD TAPE RECORDER AS BACKUP FOR BOTH ACTIVE RECORDERS

  (2) USE ONLY TWO UNITS, WITH CAPABILITY TO RECONFIGURE TO SINGLE-UNIT OPERATION
  - ACCEPT LOSS OF REAL-TIME MISSION DATA AT EVERY DUMP PERIOD IF ONE UNIT FAILS.

FLIGHT TAPE RECORDER OPERATION

It is proposed to use a system of three tape recorders, with one serving as backup for the other two units. Under normal conditions, the two active tape recorders will alternate functions every 24-hour period, as shown in the facing page table. In particular, while one active tape recorder is dumping data during an 8-hour contact shift, the other active unit is recording real-time data. If a data dump (say, by Recorder 2) is not completed during an 8-hour shift due to ground station or link anomalies, Recorder 2 can be dumped in later passes (in lieu of collecting real-time data during those passes) when downlink conditions are favorable enough to allow longer contact times or a higher downlink rate.

To optimize data dump opportunities during a contact (allowing for signal transit times), ground control will load start/stop commands with time tags in advance of the next contact (e.g. at the last contact in a shift, commands may be loaded for the next 8-hour shift). The start times will make allowance for the expected lockup time.

An economy alternative would use only two tape recorders. If one is used as backup, then no real-time recording can be done during contacts. Since real-time recorder commands cannot be used effectively for short contact periods over long distances, it might actually be preferable to only dump and not record during the 8-hour contact shift, to simplify the start/stop commands to be sent in advance.

If the two tape recorders are used without backup, then a failure in one would result in the above operational restrictions. In particular, there would be no provision for saving data not dumped during an eight-hour shift, except by restricting the MSM data acquisition for the next 24-hour period.

THE MGO/LGO POWER AND DISTRIBUTION SUBSYSTEM

MGO AND LGO POWER SUBSYSTEM

The reduced power requirements and the environmental extreme of the MGO and LGO missions compared to the FLTSATCOM capability require that modifications be made to the FLTSATCOM power system, which was designed specifically for operation in a geosynchronous orbit.

The FLTSATCOM (6, 7, and 8) solar cell panels have a single degree of freedom articulation, an array area of 233.5 ft^2 and provide 2400 watts at 45^{o}C, BOL, and 1585 watts at 57^{o}C, EOL (5 years). The design is based on a 70 VDC bus limit. The solar cell series parallel network (80 by 252, respectively) has been designed specifically for the FLTSATCOM requirements.

Energy requirements for the FLTSATCOM (7 and 8) sun occultation are about 1900-watt hours dictating a need for three 34 AH batteries operating at about a 70 percent depth of discharge. The 5-year mission requires 450 discharge-charge cycles. In addition, there is a 2.1 ampere charge limitation on the array which is compatible to a 24-hour orbit operation.

The FLTSATCOM (1 through 8) power control unit does not have the capability built within to control or limit the maximum bus voltage at 35 VDC, a requirement for the MGO and LGO. It also does not include battery chargers and associated controls for battery charging in 2-hour orbits. Therefore, a change to a different power control unit is required.

Presented herein are brief descriptions of mission characteristics and power requirements, power system design, performance, weight and component inheritance.

COMPARISON OF MISSION CHARACTERISTICS AFFECTING POWER PERFORMANCE AND DESIGN SUBSYSTEM CONFIGURATION

MISSION CHARACTERISTICS	MARS	LUNAR	FLTSATCOM
MISSION DURATION (1988)	375 DAY AVERAGE TRANSIT TIME PLUS 687 EARTH DAYS	485 DAYS * INCLUDING 120 DAYS QUIESCENT PERIOD	1,825 DAYS
ORBITAL PERIOD (HOURS)	1.90	1.96	24
SUNLIGHT TIME	1.20	1.19	24
SUN OCCULTATION TIME	0.70	0.77	1.2
NUMBER OF SUN OCCULTATIONS	8,725	3,515	450
TOTAL NUMBER OF ORBITS	8,725	4,455	1,825
ORBITAL ALTITUDE (km)	300	100	35,786
MAXIMUM MARS DISTANCE FROM SUN (Au)	1.67	–	–
MINIMUM MARS DISTANCE FROM SUN (Au)	1.37	–	–
SUN-MARS MAX DISTANCE (1988 MISSION) (Au)	1.50	–	–
SUN-MARS MAX DISTANCE (1990 MISSION) (Au)	1.60	–	–
SUN-MARS MAX DISTANCE (1992 MISSION) (Au)	1.67	–	–

* SEE COMMENT BELOW

COMPARISON OF MISSION CHARACTERISTICS AFFECTING POWER PERFORMANCE AND DESIGN SUBSYSTEM CONFIGURATION

A comparison of the MGO, LGO and FLTSATCOM mission characteristics are shown here. The combination of increased distance from the sun, the increased number of sun occultations, the decreased orbit period and the power load reduction from about 1,520 watts (FLTSATCOM 6, 7, and 8) to about 450 watts dictates modifications for the MGO. For the LGO application, the combination of the increased number of orbits, the decreased orbit period, the increased number of sun occultations, the significant solar array temperature swing (200°C in about 1.2 hours), and the reduction of power required from about 1,520 watts to about 387 watts also dictates modifications to the FLTSATCOM power system.

The table above indicates the original LGO mission profile, which had two 60-day quiescent periods, at which point LGO went into a sunline lock-on mode. Later, in the study, it was found necessary to bend the solar panels 45°. This change does not impact other design considerations.

PRELIMINARY POWER REQUIREMENTS (CONTINUOUS AVERAGE - WATTS)
MARS ORBITER

SUBSYSTEM	INTERPLANETARY CRUISE	ORBIT INSERTION	SCIENCE ON (TRANSMISSION)	ORBITAL OPERATION: SUN: SCIENCE ON (NO TRANSMISSION)	ORBITAL OPERATION: SUN OCCULTATION: SCIENCE ON
SCIENCE	0	0	42.0	42.0	42.0
PROPULSION	<1.0	<1.0	<1.0	<1.0	<1.0
ATTITUDE CONTROL	12.8	31.5	31.9	31.9	31.9
THERMAL CONTROL	34.3	34.3	10.0	84.2	84.2
REACTION CONTROL	10.0	10.0	10.0	10.0	10.0
TT&C (INCLUDES TAPE RECORDER, TELEMETRY, ETC.)	49.0	49.0	142.9	43.2	43.2
ELECTRICAL POWER CONVERSION LOSS	15.0	15.0	22.0	15.0	15.0
ELECTRICAL POWER DISTRIBUTION LOSS	4.6	5.1	10.4	7.5	3.2
TOTAL	126.7	144.9	270.2	234.8	220.5
BATTERY CHARGE	16	16	179	179	–
TOTAL	142.7	160.9	449.2	413.8	220.5

REFERENCE: FLTSATCOM POWER CAPABILITY: 1585W (1 AU) FOR 5 YEARS, 1084W (1.67 AU) FOR 2 YEARS.
FLTSATCOM POWER LOAD: 1520W

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PRELIMINARY POWER REQUIREMENTS (CONTINUOUS AVERAGE - WATTS)

MARS ORBITER

The preliminary power requirements of the MGO missions are shown here for key example operating modes. Solar array sizing is normally based on providing end of mission power for the science on and transmission mode considered as the maximum requirement. However, for this mission, solar array sizing is based on arrays not deployed, but in a spacecraft spinning mode off-sun prior to encounter. Battery sizing is based on energy consumed during the sun occultation periods.

The power requirements of the MGO missions are about one-fourth that required on the FLTSATCOM 6, 7, and 8 spacecraft.

PRELIMINARY POWER REQUIREMENTS (CONTINUOUS AVERAGE - WATTS)
LUNAR ORBITER

SUBSYSTEM	INTERPLANETARY CRUISE	ORBIT INSERTION	ORBITAL OPERATION	
			SUN	SUN OCCULTATION
			SCIENCE ON (TRANSMISSION)	SCIENCE ON
SCIENCE	0	0	59.0	59.0
PROPULSION	<1.0	<1.0	<1.0	<1.0
ATTITUDE CONTROL	12.8	31.5	31.9	31.9
THERMAL CONTROL	34.4	34.4	10.5	40.1
REACTION CONTROL	10.0	10.0	10.0	10.0
TT&C (INCLUDES TAPE RECORDER, TELEMETRY, ETC.)	49.0	49.0	67.9	43.2
ELECTRICAL POWER CONVERSION LOSS	20.0	20.0	22.0	20.0
ELECTRICAL POWER DISTRIBUTION LOSS	5.3	5.3	6.8	3.9
TOTAL	132.5	151.2	209.1	209.1
BATTERY CHARGE	16	16	178.3	–
TOTAL	148.5	167.2	387.4	209.1

REFERENCE: FLTSATCOM POWER CAPABILITY: 1585W (1 AU) FOR 5 YEARS
FLTSATCOM POWER LOAD: 1520W (1 AU)

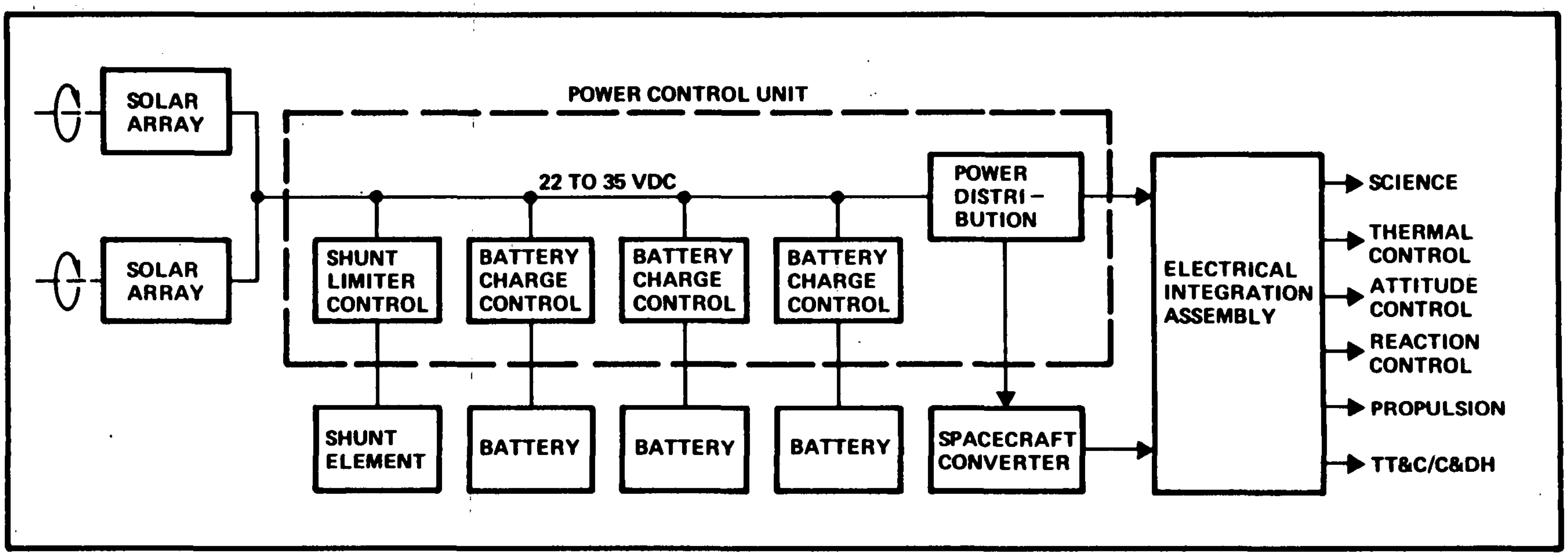
PRELIMINARY POWER REQUIREMENTS (CONTINUOUS AVERAGE - WATTS)

LUNAR ORBITER

The preliminary power requirements of the LGO missions are shown here for key example operating modes. Solar array sizing is normally based on providing end of mission power for the science on and transmission mode considered as the maximum requirement. However, for this mission, solar array sizing is based on arrays not deployed, but in a spacecraft spinning mode off-sun prior to encounter. Battery sizing is based on energy consumed during the sun occultation periods.

The power requirements of the LGO missions are about one-fourth that required on the FLTSATCOM 6, 7, and 8 spacecraft.

MGO/LGO ELECTRICAL POWER SUBSYSTEM



MGO/LGO ELECTRICAL POWER SUBSYSTEM

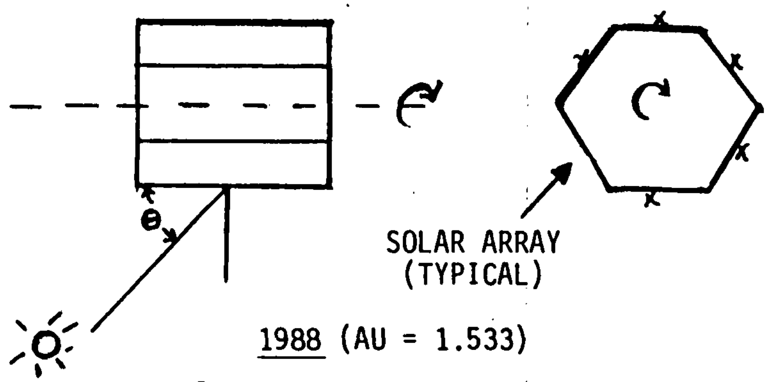
A functional block diagram of the MGO/LGO electrical power and distribution system is shown here. The bus voltage selected is 28 $^{+7}_{-6}$ VDC considered as the NASA standard bus voltage. (Note: the FLTSATCOM inherited bus voltage is 20 to 70 VDC.) Solar arrays which are folded up during the interplanetary period provide power in the spinning configuration. After orbit insertion, the articulated single degree of freedom arrays are deployed. Nickel cadmium batteries provide power (energy) during the sun occultation periods and as a supplement if the load exceeds the solar array power output capability for short periods of time.

The power control unit (PCU) is provided which controls and distributes solar array bus and battery power to all spacecraft users. The PCU also executes power management commands, and provides under-voltage protection to load an independent battery discharge/charge controls. The PCU also accepts commands and conditions telemetry signals. A shunt limiter and associated controls for limiting bus voltage at 35 VDC is also provided within the PCU. Protection of the science payload is also provided in the PCU.

A spacecraft converter is also provided for supplying power at ±5 and ±15 VDC, etc. An electrical integration assembly which contain load on-off controls is also provided.

POWER AVAILABLE

- ARRAY SIZED BY POWER REQUIRED AT MOI ATTITUDE/DISTANCE
- IN UNDEPLOYED CONFIGURATION



EFFECTIVE AREA = 33.7 TO 39 FT^2 = A

TOTAL POWER = A x SPECIFIC POWER

E.G. (1988) 1.53 AU, $\measuredangle\ 50^0 \longrightarrow$

$4.76\ {}^{W}/FT^2\ (33.7) FT^2 = 160.4$ W TO 185.6 W

CRUISE POWER REQUIRED = 143 W

	1988 (AU = 1.533)			1990 (AU = 1.643)			1992 (AU = 1.591)		
θ^0	70	60	50	70	60	50	70	60	50
TEMP	-38^0	-43^0	-50^0	-50	-54	-61	-46	-51	-57
W/FT^2	5.61	5.24	4.76	5.42	5.05	4.58	5.11	4.96	4.62

- DEPLOYED, THE ARRAY IS 116.4 FT^2, AND WITH THE SUN IN THE ORBIT PLANE ($\theta = 90^0$)

TEMP	8^0C	0^0C	-6^0C
W/FT^2	5.00	4.49	4.91
PWR AVAIL.	582 W	523 W	572 W
PWR REQ.	450-414 W	450-414	450-414
MAX ALLOW.	45.8^0	40^0	40^0

$\measuredangle$ OFF-SUN FOR PWR AVAILABLE REQUIRED

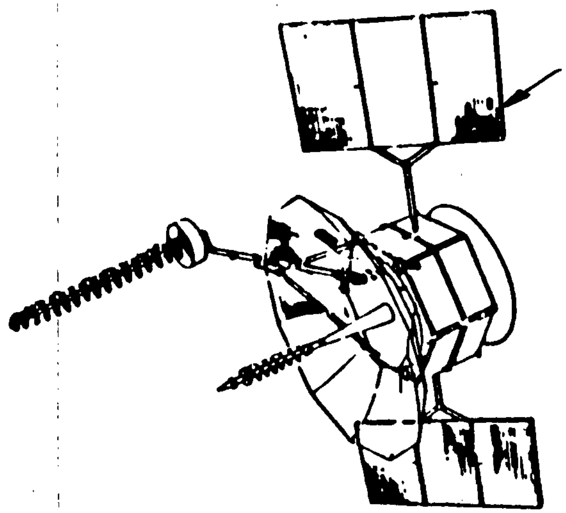
POWER AVAILABLE AT ENCOUNTER

MGO

The solar array power available at Mars just prior to and after encounter for three different launch years, 1988, 1990 and 1992 is shown here with a corresponding Sun-Mars distance. The folded up configuration as shown prior to Mars orbit insertion sizes the solar array. In the example shown, available power during the spinning interplanetary cruise (1988 launch) varies from 160.0 to 186 W at 1.53 AU) which is about 15 to 26% more than that required at this point in time. However, for a 1990 launch and at a Sun-Mars distance of 1.643 AU, with the same array area there is about 7.5% power margin. The available power near earth after a 1988 launch varies from about 270 to 330 W at 34.3 VDC with an off-sun angle of 50 degrees selected to limit solar array temperatures for battery charging voltage requirements.

The solar arrays are deployed at a post Mars orbit insertion. The available power is also shown for three launch dates and corresponding Sun-Mars spacecraft distances. Sufficient power is available to the end of the mission for all the three launch dates.

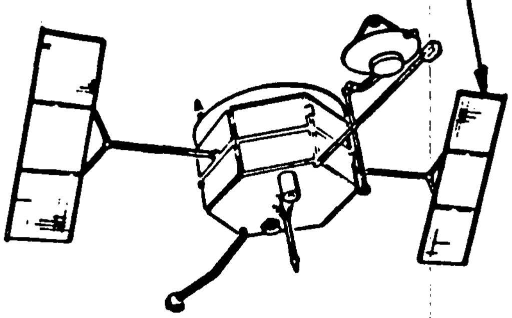
MGO/LGO AND PARENT SPACECRAFT SOLAR ARRAYS



TOTAL AREA: 233.5 FT^2 (DEPLOYED)
BOL POWER: 2400 (1 AU)

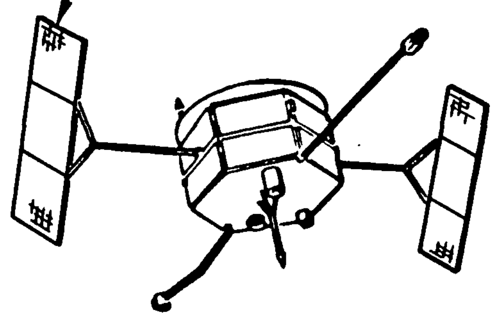
FLTSATCOM

TOTAL AREA: 116.4 FT^2 (DEPLOYED)
POWER (EOL): 523 (1.643 AU)
POWER (EOL): 582 (1.533 AU)



FLTSATCOM MGO ADAPTATION

TOTAL AREA: 116.8 FT^2 (DEPLOYED)
POWER (EOL): 470 (1 AU)



FLTSATCOM LGO ADAPTATION

MGO Solar Array

The deployed solar array configuration as shown here is not the sizing mode. Solar array sizing has been based on the spinning configuration at a Sun-Mars distance of 1.50 AU for a 1988 launch with a 50-degree off Z-axis run angle prior to Mars orbit insertion. The power available after a Mars orbit insertion for a 1988 launch (1.5 AU) at Mars is 606 W at a temperature of 8°C. The worst case power available which occurs during a 1991 launch (1.67 AU) is 514 W at a temperature of -6°C. The number of series solar cells to acquire battery charging voltages is 64 without any significant excess power to be dissipated while limiting bus voltage to 35 VDC. This is a decrease from the FLTSATCOM 80 series solar cells.

LGO Solar Array

Unlike MGO, the LGO array is sized by operations in lunar orbit. When the sun is in the orbit plane, power developed by the array is cos 45° (0.707) times maximum solar capability at that point in time when the spacecraft is not in eclipse. When the sun is normal to the orbit plane, 0.707 time maximum capability is supplied at all times (i.e., there is no solar panel eclipse). Power output varies for sun position in between these examples.

During transit and orbit insertion, the solar array is folded. During these periods, the equipment, particularly the battery, must radiate to the array which then radiates to space. An array temperature of -28°C is desirable to maintain the batteries in the 0 to 10°C range. This results in an off-sun angle of 60 degrees and still leaves a power margin of about 75 percent. Orbit insertion temperatures may rise higher, but this is a transient condition.

FLTSATCOM INHERITED POWER SOURCE COMPONENTS COMPARED TO MGO AND LGO REQTS

SOLAR ARRAY	FLTSATCOM	MGO	LGO	OPTIONS	
AREA	$233.5FT^2$	$116.4FT^2$	$116.8FT^6$		
POWER AVAIL (BOL)	2400W	300W (SPINNER)	300W (SPINNER)		
POWER AVAIL	1084W (1.67 AU)	514W (1.67 AU) (DEPLOYED)	470W (1.0 AU) (DEPLOYED)		
SPIN. PWR REQD		150W	150W		
ORBIT PWR REQD		414 to 450W	390W		
WEIGHT		29.1 LBS.	29.2 LBS.		
BATTERIES	(3) 34AH 24 CELL FLTSATCOM 6, 7, 8	(3) 15AH 22 CELL (DSCS II)	(3) 15AH 22 CELL (DSCS II)	(2) 24AH 22 CELL (DSP) 5, 6	(2) 34AH 22 CELL FLTSAT 6, 7, 8
WEIGHT	240 LBS.	114.4 LBS.	114.4 LBS.	106 LBS.	160 LBS.
AH OUT	6AH	6AH	6AH	6AH	6AH
DEPTH OF DISCH (D OF D)	6%	13%	13%	12%	9%
1 FAILURE (D OF D)	9%	20%	20%	25%	18%
CYCLES REQUIRED & (DISCH/CHARGE)		8,725	3,515		
NEW REQT (LGO)	S/C ROTATED 90^0				
1.5 HR ECLIPSE					
BATT. AH OUT	12.7AH	---	12.7AH	12.7AH	12.7AH
(D OF D)	12.7/102 = 12.4%		12.7/45 = 28%	12.7/48=26.5	12.7/68=18.7%
1 FAIL	18.7%		12.7/30 = 42.3%	12.7/24=53%	12.7/34=37.4%
BATT CHG POWER (.5HCHC)			1095W		
SOLAR ARRAY					
TOTAL POWER REQD			1364W		
AREA REQD			$217.5FT^2$		

FLTSATCOM INHERITED POWER SOURCE COMPONENTS

COMPARED TO MGO AND LGO REQTS

A comparison of the inherited FLTSATCOM power source components and capability with the MGO and LGO requirements and the selected MGO and LGO power sources are shown here. The FLTSATCOM solar array area is over-designed by a factor of 2 for MGO and LGO power requirements, and consequently was reduced about one-half for weight and cost savings. In addition, although not shown, the solar cell series string has been reduced from the FLTSATCOM's 80 series cells to 64 for MGO and increased to 102 for LGO requirements.

In the case of battery usage for the MGO application, the capacity required during the 0.7 hour sun occultation is about 6 AH at a depth of discharge of about 6%. Shown are three options. A tentative selection of three (3) 15 AH twenty-two (22) cell nickel cadmium batteries has been made for about an 8725 discharge/charge cycle operation. Battery capacity required for the LGO application is also about 6 AH, although the number of discharge/charge cycles required is about 60% less or 3515 cycles. On this basis, the same battery configuration was selected for the LGO mission.

Extending the LGO operation, i.e., to reduce the standby period by rotating the S/C 90 degrees, increased the sun occultation (sun eclipse) to 1.5 hours. As a consequence, the solar array area was increased to 217.5FT^2. Battery performance is also shown for the selected baseline and two options. As previously indicated, the LGO top (above chart) line configuration (45° bend) was selected.

POWER SUBSYSTEM AND DISTRIBUTION COMPONENTS

	MGO			LGO		
COMPONENT	QUANTITY	WEIGHT (POUNDS)	INHERITANCE	QUANTITY	WEIGHT (POUNDS)	INHERITANCE
SOLAR CELLS AND COVER SLIDES ONLY	116.4FT^2	29.1	FLTSATCOM	116.8FT^2	29.2	FLTSATCOM
BATTERIES*	3	114.4	DSCS II	3	114.4	DISCS II
POWER CONTROL	1	29.1	HEAO (MODIFIED)	1	29.1	HEAO (MODIFIED)
SHUNT LIMITER (ETC)	2	10.2	DSP-14 (MOD.)	2	10.2	DSP-14 (MOD.)
SPACECRAFT CONVERTER	1	18.1	FLTSATCOM	1	18.1	FLTSATCOM
EIA	1	22.4	FLTSATCOM	1	22.4	FLTSATCOM
HARNESS	1	74.8	FLTSATCOM	1	74.8	FLTSATCOM
TOTAL		298.1			298.2	

* OPTION:
- TWO 22 - CELL, 24 AH DSP BATTERIES: 106 LBS., TOTAL
- INHERITED FLTSATCOM BATTERIES: 24 - CELL, 24 AH

NICKEL CADMIUM BATTERIES: 240 LBS.

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POWER SUBSYSTEM AND DISTRIBUTION COMPONENTS

A weight breakdown and inheritance of the electrical power and distribution components for the MGO and LGO missions are shown here. The total weight of both the MGO and LGO power and distribution system is about 298 lbs. less solar array substrates, hinges, etc. With the exception of the solar cell array series-parallel network, there is a 100 percent commonality of the remaining components of the power and distribution system.

The HEAO power control unit was selected, because of the three battery chargers and associated controls located within are more compatible to MGO and LGO orbits than the FLTSATCOM geosynchronous orbit. The regulate error amplifier, also located within the PCU, can be used with the DSP-14 digital-linear shunt system which does not utilize the resistor shunt radiator system, but utilizes a series of shunt element transistors. The remaining components, electrical integration assembly (EIA) and spacecraft equipment converter, although over-designed, may be retained for both MGO and LGO applications.

MGO/LGO THERMAL CONTROL SUBSYSTEM

REQUIREMENTS:

- MAINTAINS VARIOUS SPACECRAFT EQUIPMENTS AT ACCEPTABLE TEMPERATURES THROUGHOUT THE MGO AND LGO MISSIONS (i.e., PRE-LAUNCH, LAUNCH/SHUTTLE ABORT, INTERPLANETARY CRUISE, AND ORBITAL OPERATION)
- PROVIDES ACCEPTABLE INSTRUMENT THERMAL INTERFACES OR MAINTAINS INSTRUMENTS AT ACCEPTABLE TEMPERATURE THROUGHOUT THE MISSIONS
- MAINTAINS ACCEPTABLE THERMAL INTERFACES WITH STS AND LAUNCH VEHICLE

SYSTEM FUNCTIONS

The function of the thermal control subsystem is to maintain all spacecraft equipment within acceptance temperature limits throughout all mission phases. This is accomplished primarily by passive means, utilizing combinations of insulation, SSMs, and thermal coatings. These elements are supplemented by electrical heaters that are used to maintain the operational environment during cold conditions or to replace the component thermal dissipations where required to provide operations flexibility. The heaters are enabled by ground command. When enabled, thermostats provide control of power to the strip heaters in those circuits used to maintain temperatures in a specific range. The thermostats are located in close proximity to the heater elements being controlled to minimize the time lag in their operation. Heater status and temperature measurements are provided on telemetry to allow monitoring of the thermal control subsystem performance.

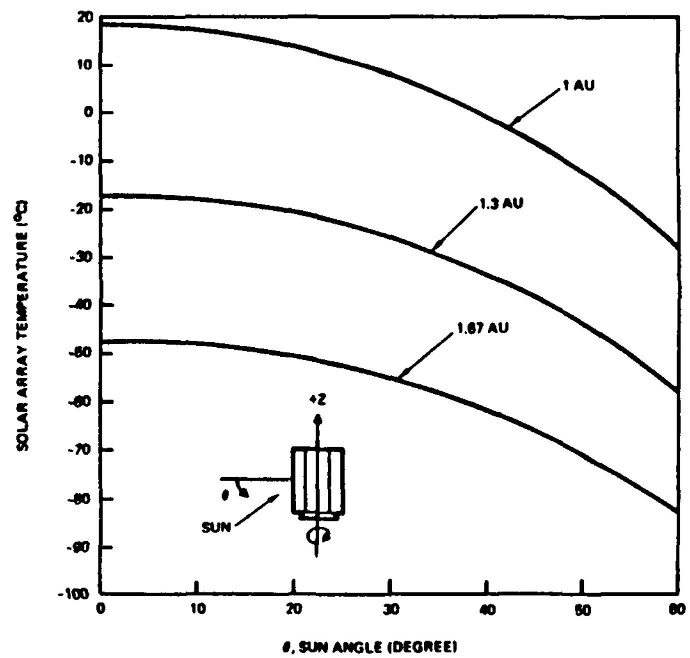
For MGO, consideration will be given for adding autonomous instrument shut-down or heater turn-ons, to supply protection for the 16 plus hour periods when data is not being received on the ground.

MGO/LGO HEATER POWER REQUIREMENTS

MISSION	INTERPLANETARY CRUISE	ORBIT INSERTION		ORBITAL OPERATION, SCIENCE ON	
		ARRAY FOLDED	ARRAY DEPLOYED	WITH TRANSMISSION	NO TRANSMISSION
MGO					
EQUIPMENT COMPARTMENT	10W AT 1.5 AU 25W AT 1.67 AU	0 AT 1.5 AU 6W AT 1.67 AU	81W AT 1.67AU	0	84W AT 1.67 AU
INSTRUMENT	24W AT 1.5 AU 27W AT 1.67 AU	24W AT 1.5 AU 27W AT 1.67 AU	37W AT 1.67 AU	0	0
PROPULSION	44W	44W	26W	26W	26W
LGO					
EQUIPMENT COMPARTMENT	0	0	30W	8W	37W
INSTRUMENT	34W	34W	62W	3W	3W
PROPULSION	44W	44W	26W	26W	26W

Based on the desired maximum average radiator temperature of not higher than 20°C, total radiator area for the equipment compartment approximating 9.2 square feet is required for the LGO mission and 9.3 square feet for the MGO mission. For the various instruments, total radiator area of about 1.7 square feet is required for the MGO mission and about 3.5 square feet for the LGO mission. Based on these chosen radiator areas, and the desired minimum average radiator temperature of not less than 5°C, total heater power required for the two missions was calculated and is presented above.

During interplanetary cruise with the solar array folded about the spacecraft, power dissipated by the various equipment must be rejected to the solar array from the radiators, and the solar array must be kept below -10°C solar array is required to maintain the average spacecraft radiator temperature below 30°C. The figure to the right shows the predicted solar array temperature versus sun angle for three solar distances. The sun angle should be ≥50 degrees for 1 AU.



In addition to the above thermal controls, the reaction control subsystem (RCS) also utilizes thermostatically controlled heaters located on the propellant feed lines, propellant tanks, dual thruster module (DTM) valves, and propellant distribution assembly (PDA) to maintain hydrazine temperatures above freezing.

MODIFICATIONS TO FLTSATCOM THERMAL CONTROL SUBSYSTEM

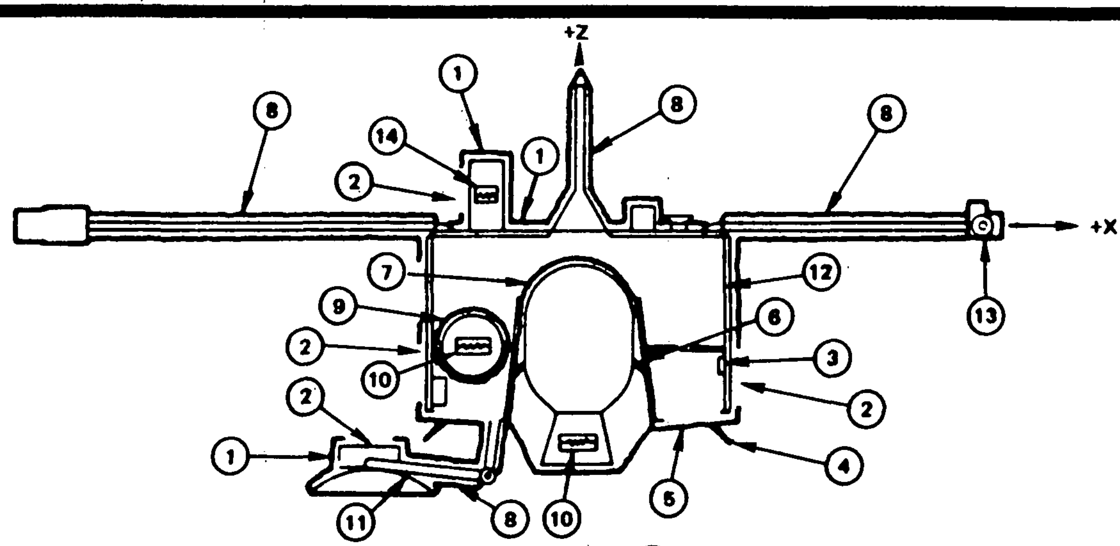
- SECOND SURFACE MIRROR RADIATORS ON THE EQUIPMENT COMPARTMENTS REDUCED TO ABOUT 9.2 SQUARE FEET
- BATTERIES ISOLATED THERMALLY FROM OTHER EQUIPMENTS WITH MLI AND THERMAL ISOLATORS
- THERMOSTAT CONTROLLED HEATER ADDED TO AKM
- VARIOUS BLACK BOX HEATERS REPLACED WITH PANEL HEATERS AT SELECTED AREAS
- NEW RADIATORS, HEATERS, THERMOELECTRIC COOLER, AND RADIANT COOLER FOR INSTRUMENTS AS REQUIRED

MODS TO FSC SUBSYSTEM

The FLTSATCOM thermal control subsystem can be used for the MGO and LGO missions with minimum modifications. The modifications include:

- Second surface mirror (SSM) radiators on the equipment compartment will be reduced to about 9.24 square feet (from about 36 square feet for FLTSATCOM) and redistributed to accommodate the various MGO/LGO equipment.
- The batteries will be isolated thermally from the other electronic components due to their stringent on-orbit operating temperature requirements of 0 to 10°C. The batteries will be located on either the +Y or -Y panel with their separated radiators and heaters and thermally isolated from other equipment with multilayered insulation blankets.
- A thermostat controlled heater will be added to the AKM nozzle to maintain the nozzle above its lower temperature limit of 20°F during interplanetary cruise and before orbit insertion.
- The backside of the solar array will be coated with chemglaze white paint (FLTSATCOM uses 3M black velvet paint).
- The various black box heaters (i.e., transmitter heaters) will be replaced with a new set of thermostat controlled panel heaters located at selected areas of the spacecraft module.
- New radiators, heaters, thermoelectric cooler, and radiant cooler for instruments as required.
- Limiting the sun angle with respect to the solar array such that array temperature does not exceed -10°C during interplanetary cruise.
- Limiting time duration between lunar arrival and solar array deployment may be required to avoid over-heating of equipment compartment components.

MGO/LGO THERMAL CONTROL SUBSYSTEM DESCRIPTION



(1) DESIGN: 10 LAYERS, 1/4-MIL Al-MYLAR SANDWICHED BY INNER 1-MIL Al-MYLAR AND OUTER 2-MIL Al-KAPTON
FUNCTION: INSULATION TO LIMIT HEAT TRANSFER WITH SUN AND SPACE

(2) DESIGN: TYPICALLY ON EACH PANEL, AND INSTRUMENT, SECOND SURFACE SILVERED 6-MIL THICK FUSED SILICA MIRRORS
FUNCTION: RADIATING AREAS OF STABLE THERMOPHYSICAL PROPERTIES TO REJECT DISSIPATED POWER AT PROPER TEMPERATURE

(3) DESIGN: FLEXIBLE STRIP HEATERS AND BIMETALLIC-THERMOSTATS FOR BATTERIES, SELECTED INSTRUMENTS, AND SELECTED AREAS OF EQUIPMENT COMPARTMENT PANELS
FUNCTION: MAINTAIN SELECTED COMPONENT TEMPERATURE LIMITS

(4) DESIGN: ONE LAYER 3-MIL Al-KAPTON. ALUMINIZED SIDE FACES PLUME
FUNCTION: PLUME SHIELD TO SHADE FOLDED SOLAR ARRAY DURING BURN

(5) DESIGN: 22 LAYERS, 1/4-MIL Al-KAPTON SANDWICHED BY INNER 1-MIL Al-KAPTON AND OUTER 3-MIL Al-KAPTON
FUNCTION: HIGH-TEMPERATURE INSULATION TO LIMIT HEAT TRANSFER WITH SUN AND SPACE, AND WITHSTAND PLUME RADIATION DURING BURN

(6) DESIGN: 6 Al-4V TITANIUM CONE
FUNCTION: LOW CONDUCTIVITY MOTOR ATTACHMENT STRUCTURE TO PREVENT EXCESSIVE CONDUCTION HEAT SOAK INTO SPACECRAFT AFTER BURN AND HEAT LOSS IN ORBIT

(7) DESIGN: 10 LAYERS 1/4-MIL Al-KAPTON SANDWICHED BY INNER 1-MIL Al-KAPTON AND OUTER 2-MIL Al-KAPTON
FUNCTION: HIGH TEMPERATURE INSULATION TO PROTECT COMPONENT INTERIOR FROM AKM PROPELLANT TEMPERATURE EXCURSION DURING BURN

(8) DESIGN: 10 LAYER, 1/4-MIL Al-MYLAR SANDWICHED BY INNER 1-MIL Al-MYLAR AND OUTER 2-MIL Al-KAPTON, SPIRALLY WRAPPED ABOUT BOOMS
FUNCTION: INSULATION TO LIMIT HEAT TRANSFER WITH SUN AND SPACE

(9) DESIGN: 10 LAYER, 1/4-MIL Al-MYLAR ON TANKS AND VALVES. 10 LAYER, 1/4-MIL Al-MYLAR SPIRAL WRAP BLANKETS FOR LINES
FUNCTION: INSULATION TO MAXIMIZE TANKS, LINES, AND VALVE HEATER EFFECTIVENESS

(10) DESIGN: KAPTON TAPE HEATERS AND BIMETALLIC THERMOSTATS FOR TANKS, LINES, VALVES, AND AKM NOZZLE
FUNCTION: MAINTAIN SELECTED COMPONENT TEMPERATURE LIMITS

(11) DESIGN: CHEMGLAZE WHITE PAINT ON REFLECTOR SURFACE
FUNCTION: PROVIDE ACCEPTABLE TEMPERATURE FOR REFLECTOR

(12) DESIGN: CHEMGLAZE BLACK PAINT FOR COMPARTMENT INTERIOR
FUNCTION: HIGH-EMITTANCE COATING TO ENHANCE RADIATION HEAT TRANSFER IN COMPARTMENT INTERIOR

(13) DESIGN: 2-STAGE RADIANT COOLER
FUNCTION: TO MAINTAIN GAMMA RAY SPECTROMETER DETECTOR TEMPERATURE AT $100^{\circ}K$

(14) DESIGN: MULTI-STAGE THERMOELECTRIC COOLER
FUNCTION: TO MAINTAIN INSTRUMENT AT CRYOGENIC TEMPERATURE AS REQUIRED

GUIDELINE FOR MEETING THE INSTRUMENT THERMAL REQUIREMENTS

Next Phase Effort:

1. Perform thermal design to achieve acceptable operating temperature for each instrument

2. Determine thermal gradients and temperature excursions for each instrument

3. Identify instrument accommodation thermal problems as areas for future study

4. Each instrument will contain a thermistor for making power-off temperature measurements

5. Overall thermal design of proposal appears satisfactory

OPERATIONAL CONSTRAINTS

To avoid overheating of equipment compartment components, the following constraints are placed on spacecraft operation:

- Sun angle with solar array normal be limited during interplanetary cruise such that array temperature does not exceed -5°C (i.e., $\geq$ 45 degrees at 1.0 A.U.)

- Time duration between lunar arrival and solar array deployment be limited

IMPACT OF SCIENTIFIC INSTRUMENTS ON THERMAL CONTROL SUBSYSTEM

- IT WILL BE DIFFICULT TO PROVIDE 80°K HEAT SINK TO MSM IF MOUNTED ON BODY. A PLANETARY AND SUN SHADE(S) WILL BE NECESSARY

- THE γ-RAY AND X-RAY EXPERIMENTS WILL REQUIRE SUN, EARTH, AND SPACECRAFT SHADES

- THE ONLY WAY TO MEET EXPERIMENT COOLING REQUIREMENTS ON LGO IS TO <u>ALWAYS</u> HAVE AVAILABLE A RADIATOR FACING AWAY FROM THE SUN. THIS DICTATES ROTATING THE SPACECRAFT 180° AT LEAST ONCE PER YEAR. THIS REQUIREMENT REDUCED THE LUNAR SOLAR PANEL CONFIGURATION OPTIONS FROM 3 TO THE ONE SATISFACTORY SOLUTION - MAINTAIN THE PANELS AT 45°

- MGO/LGO PREDICTED TYPICAL INSTRUMENT TEMPERATURES
  - OPERATING RANGE: -15°C TO 40°C
  - NON-OPERATING RANGE: -30° TO 35°C
  - TEMPERATURE VARIATION DURING ONE ORBIT: 1°C TO 2°C

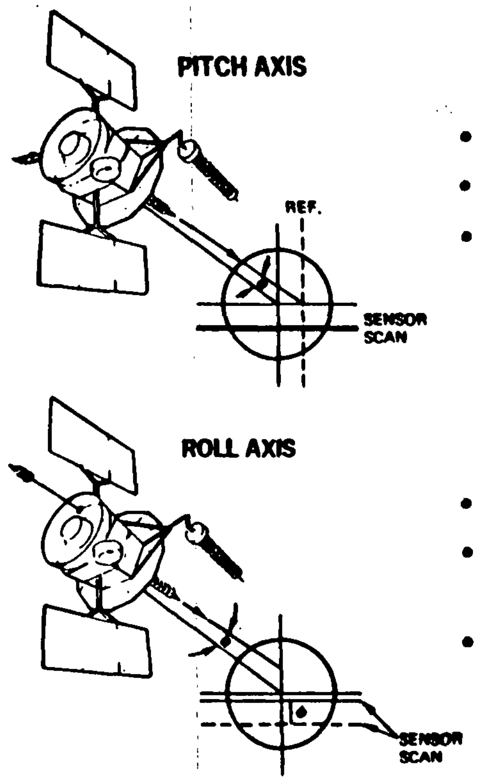
THERMAL CONTROL FOR THE GAMMA RAY SPECTROMETER AND MULTISPECTRAL MAPPER

The multispectral mapper and gamma ray spectrometer pose unique thermal control requirements due to the low detector temperature requirements of about 195°K and 100°K, respectively. The need to achieve these detector temperatures for the LGO mission poses additional challenge due to the large IR heating from the lunar surface.

Thermal control for the multispectral mapper with its 195°K detector can be achieved by combining a simple flat plate radiator with a multistage thermoelectric cooler. TRW has successfully used a combined four-stage thermoelectric cooler and flat plate radiator in the cooling of PbSe detectors to 200°K in the Monitoring of Air Pollution from Satellite (MAP) Program. An alternate design would be to use a single-stage passive radiant cooler such as A.D. Little Inc. supplies that is capable of achieving temperatures in the range of 200 to 100°K.

Thermal control for the gamma ray spectrometer can be achieved by a two-stage radiant cone cooler produced by A.D. Little Inc. Depending on the cooling capacity required, the flight proven, Model 101 (used on the Defense Meteorological Spacecraft Program (DMSP) or a modified extension can be used. However, the use of this type radiant cooler requires careful placement and interface design such that parasitic heat loads from the lunar surface and other spacecraft surfaces are minimized. The gamma ray spectrometer is mounted on a boom extended from the spacecraft body, and the radiant cooler is located on either the +Y or -Y side of the instrument and shielded from the moon and solar array. Since, for LGO, the solar vector could illuminate either the +Y or -Y side depending on the angle between the solar vector and orbit plane, the instrument will be rotated 180 degrees periodically as the sun moves into the cooler's FOV so that the cooler's effectiveness can be maintained. This requirement dictated the LGO solar panel 45° bend design and consequent 180° spacecraft "flip".

AVCS (ATTITUDE AND VELOCITY CONTROL SUBSYSTEM



PITCH AXIS

- PITCH ERROR (ϕ) DETECTED BY EARTH SENSOR
- CHANGES IN WHEEL SPEED PROVIDE CORRECTIVE TORQUE
- PITCH THRUSTERS INFREQUENTLY FIRED TO INDUCE CORRECTION IN WHEEL SPEED

ROLL AXIS

- ROLL ERROR (ϕ) DETECTED BY EARTH SENSOR
- ROLL/YAW 0.1 LB THRUSTERS FIRE PULSES IN RESPONSE TO ROLL ERROR TO MAINTAIN ROLL WITHIN DEADZONE ($0 \pm 0.12^{\circ}$)
- YAW CONTROL ACCOMPLISHED BY PASSIVE GYROSCOPIC COUPLING BETWEEN WHEEL MOMENTUM AND SPACECRAFT ANGULAR MOMENTUM

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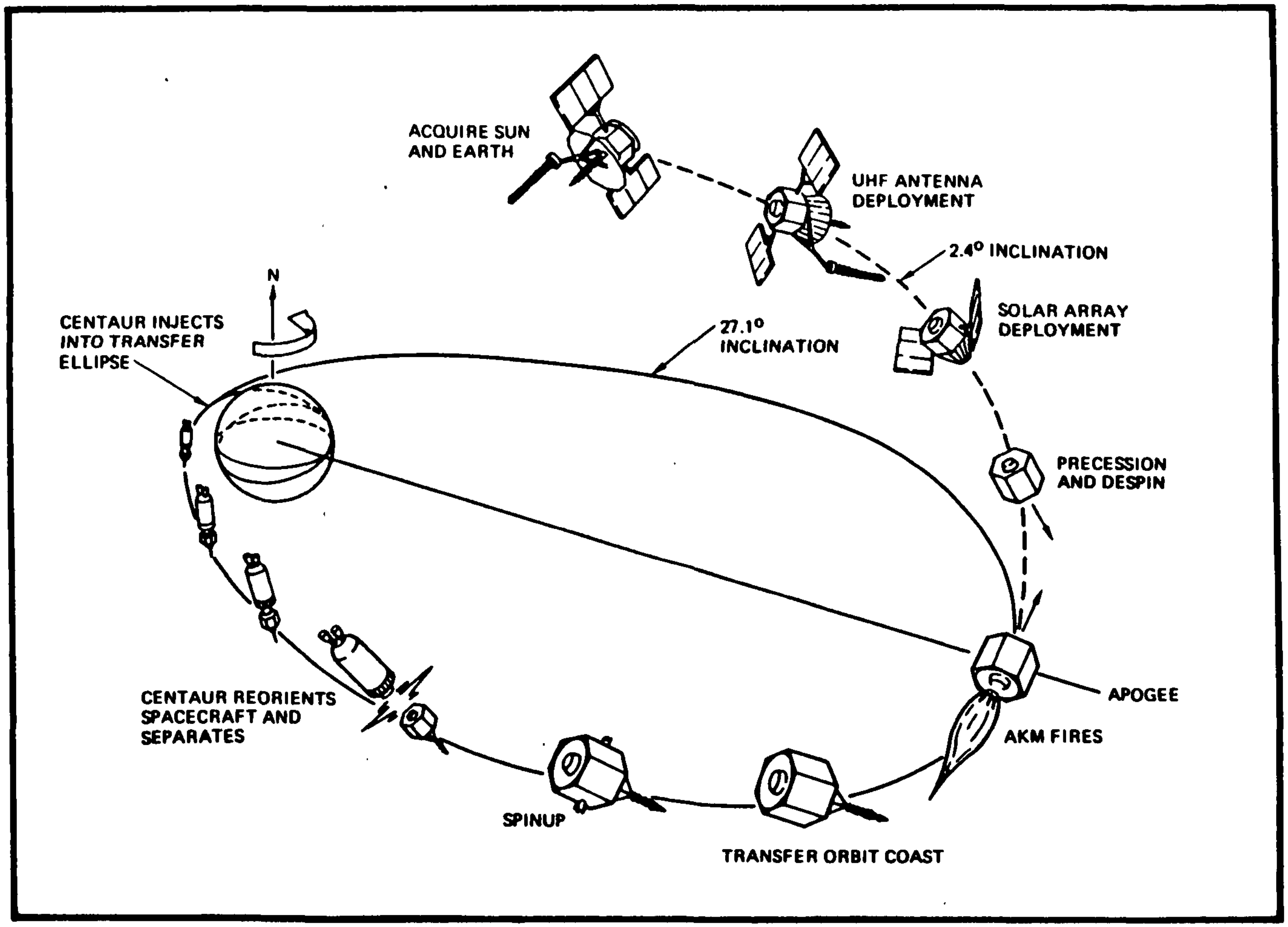
THE BIASED MOMENTUM CONTROL SYSTEM

The above describes the FSC on-orbit attitude control technique, which is directly applicable to MGO/LGO operation. The Earth sensor is the scan-through type and operates in the long infrared (IR) region. The scan plane is offset 5 degrees from the equator so that the Earth pulse is sensitive to both pitch and roll errors. The position of the Earth pulse relative to the center of the scan provides the pitch attitude error and the length of the Earth pulse measures the roll error. Each of the two sensors has two scan planes located $\pm$ 5 degrees from the spacecraft X and Z plane. These scan planes are selectable on command to avoid interference from the sun and moon. The measured pitch error is processed in the control system to modulate the speed of the reaction wheel which is oriented along the Y-body axis. Torques generated by this modulation hold the pitch error close to zero. Secular pitch momentum build-up is corrected by the pitch 1-pound thrusters. The reaction wheel speed is biased so that it operates between 2,000 and 4,000 rpm and always produces a component of momentum along the Y-body axis. The roll control system uses the roll output of the earth sensor to fire a 20 ms pulse from the 0.1-pound thrusters. This pulse is fired when the roll error exceeds its dead-band value of 0.12 degree and imparts a nutation to the spacecraft due to interaction with the spacecraft momentum. The attitude control electronics fires a second pulse 1,000 seconds after the first pulse which cancels the nutation at close to zero roll error. The roll control systems fires approximately 100 pulses of the 0.1-pound thrusters per day. The timing logic prevents nutation build-up and the bias momentum of the pitch wheel, together with a small yaw offset component of the 0.1-pound thruster alignment provides the cross-coupling between roll and yaw axes necessary for yaw stability.

FLTSATCOM MISSION PROFILE



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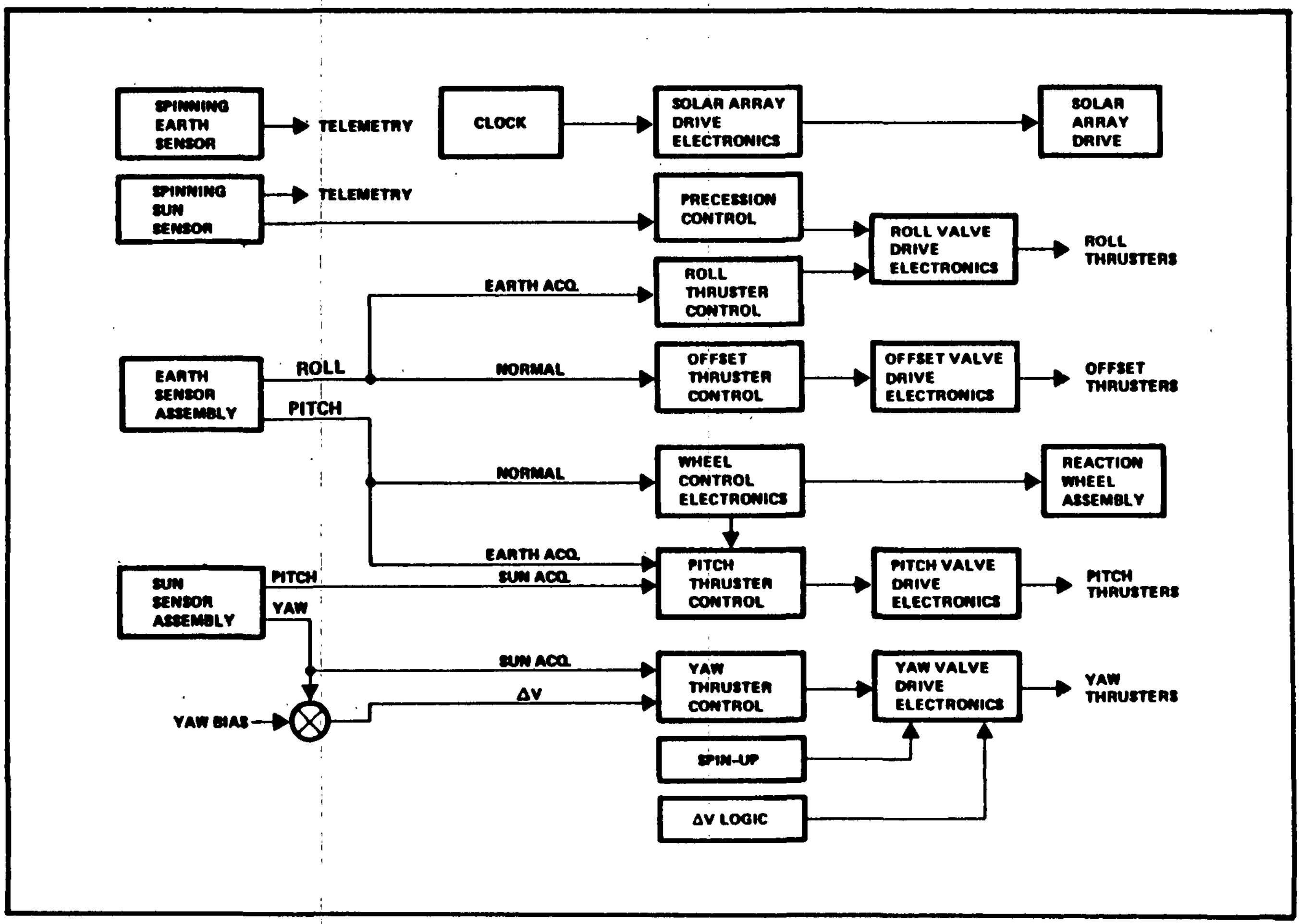
MISSION SEQUENCE

The Attitude and Velocity Control Subsystem provides for error sensing, data storage, signal processing, mode switching, and actuation devices for solar array, transfer orbit, three axis on-orbit attitude control and velocity addition for the FLTSATCOM spacecraft. It is in operation from the time of separation from the launch vehicle until mission termination, operating in two distinct phases. In the first phase during the transfer orbit, the spinning spacecraft is prepared for injection into the geosynchronous orbit. The second phase follows despin and starts with deployment of the solar array and UHF antennas. The above portrays these two phases via the mission profile.

In the first phase, the AVCS provides an automatic spin-up after separation, the ability to increase or decrease the spin speed, sensor data to be used in ground determination of spacecraft attitude, the ability to reorient the spacecraft spin axis before and after the apogee kick motor (AKM) firing, and a ground controlled despin at the termination of the first phase. In the second phase after deployment verification of the solar arrays and antennas, the AVCS provides for acquiring three-axis control with the antennas pointed to nadir and the solar array rotation axis normal to the equatorial plane. Subsequently, fine-pointing control is established for long-term operation.

The AVCS operates in the fine pointing configuration for most of the mission. It self monitors its condition for fault detection, maintains the solar array aligned with the sunline and provides periodic ground commanded velocity changes for either station change or stationkeeping. If some failure causes a loss of three axis attitude reference, the AVCS (with ground intervention) can re-establish the reference and continue the normal mission operation.

ATTITUDE AND VELOCITY CONTROL SUBSYSTEM BLOCK DIAGRAM



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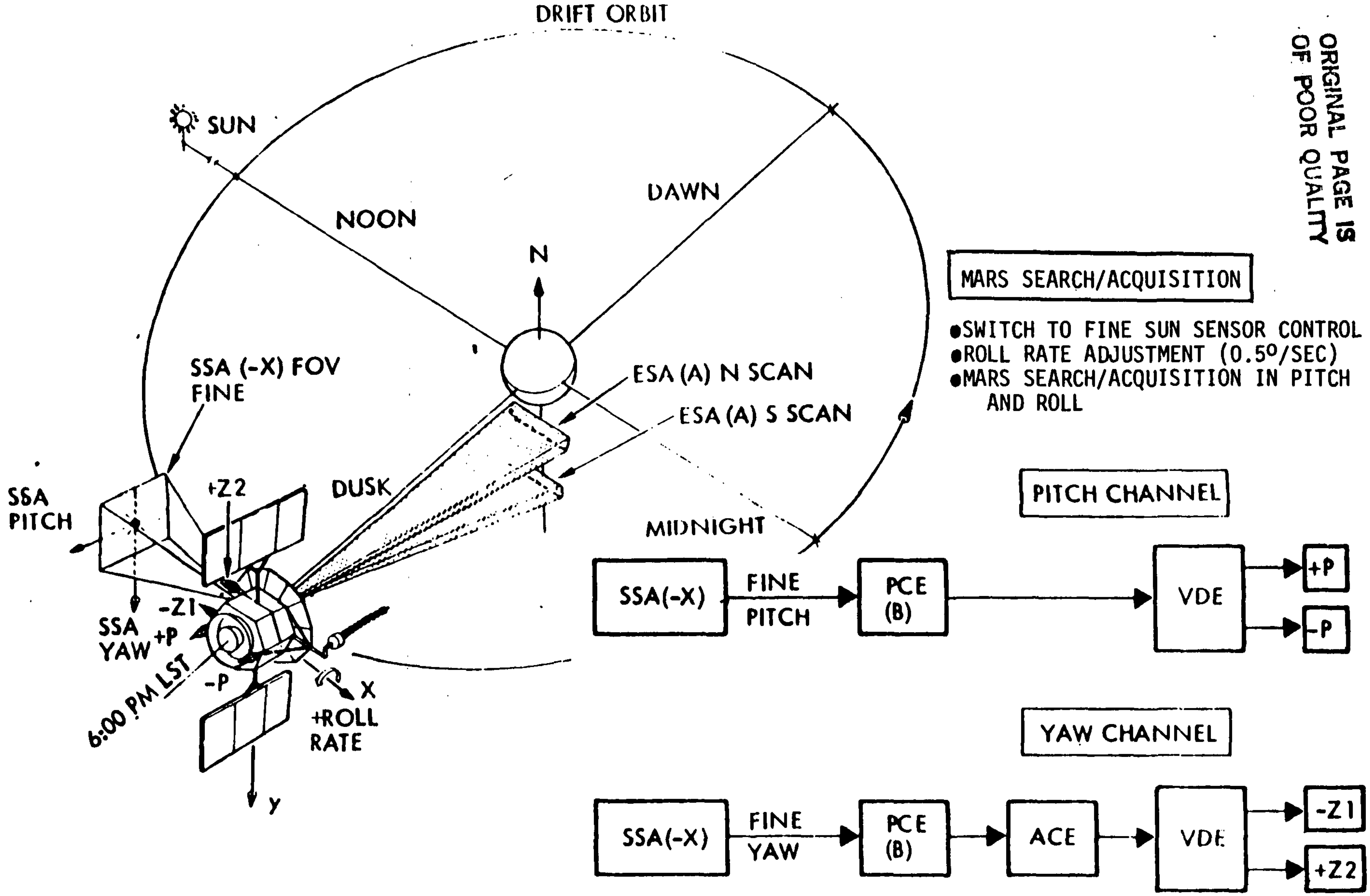
FSC BLOCK DIAGRAM

An overall block diagram of the AVCS appears above. Sensing functions are comprise of the spinning Earth sensor, sun sensor (including spinning and non-spinning functions) and the on-orbit Earth sensor. Torquers and actuation devices are the 1-pound and 0.1-pound level thrusters of the Reaction Control System, the reaction wheel and the two independent solar array drives. The control electronics is divided into the Control Electronics and Auxiliary Electronics with functions as shown. Two of each type of assembly are employed and cross strapped in the spacecraft. The electronics assemblies (CEA and AEA) are subdivided into smaller elements and lower levels of cross strapping to provide a 5 year reliability of 0.89.

Among the major design considerations for the on-orbit controllers were simplicity and accurate pointing using only two axes of error sensing, since yaw error is cumbersome to reliably obtain over an entire orbit. This led to the selection of a momentum bias system consisting of a relatively small, body fixed reaction wheel, low level hydrazine thrusters, an Earth sensor and associated electronics. The reaction wheel has its spin axis along the pitch axis and is speed biased to provide gyroscopic stiffness. The thrusters are positioned (offset) to couple roll and yaw control. Roll control is provided by the wheel momentum and the hydrazine thrusters. The thrusters are driven in response to a derived rate modulated Earth sensor roll error signal. Nutational build-up is prevented by nutation attenuation logic. Yaw control is provided by the wheel momentum with damping from the small yaw torque components of the roll actuated thrusters. Pitch control is conventional; the reaction wheel is driven in response to a shaped, Earth sensor pitch error with accumulated secular disturbance momentum removed by periodically firing a 1-pound thruster (plus or minus) whenever the wheel speed falls outside a thresholded range.

Note that for MGO, replacement of the spinning Earth sensor with a spinning stellar sensor has no impact on the AVCS. Only the addition of ANC or autonomous safe haven modes would.

MARS SEARCH/ACQUISITION ADAPTED FROM FSC



MARS SEARCH/ACQUISITION

- SWITCH TO FINE SUN SENSOR CONTROL
- ROLL RATE ADJUSTMENT (0.5^{o}/SEC)
- MARS SEARCH/ACQUISITION IN PITCH AND ROLL

This three-axis control phase follows the completion of solar panel and HGA/instrument deployment sequence. The first sequence of AVCS functions establishes planet nadir with the yaw axis (antenna axes) earth pointed and the roll axis approximately aligned with the velocity vector. This is accomplished in three steps: sun acquisition, establishing a known rate about the sunline and planetary acquisition. The above depicts the acquisition sequence including deployment.

Prior to enabling sun acquisition the solar arrays are rotated 90 degrees to preclude reflections from the array into the body-mounted sun sensor fields-of-view and to provide maximum power when sun pointed. Sun acquisition aligns the roll axis to the sun. The sun acquisition is closed loop in pitch and yaw employing coarse sun sensors, 1-pound hydrazine thrusters and control logic. Motion about the roll axis is not controlled.

After sun acquisition, the rate about the sunline is estimated and adjusted by ground action. Since the roll axis has the intermediate moment of inertia, spin about the roll axis is gyroscopically unstable in the absence of control. The control system exerts torque about the minimum moment of inertia axis (pitch) to counter the gyroscopic torque and maintain pointing. Rate polarity is determined by activating the reaction wheel and observing the direction of yaw attitude offset. Adjustment of roll rate to approximately 1 deg/sec is accomplished by commanding a single roll thruster of correct polarity for a specific "on" time.

After the correct roll rate has been established, the AVCS remains in this configuration until the planetary acquisition corridor is entered at the optimum spacecraft time. Telemetry is observed for a radiance presence that indicates that the earth sensor has detected a radiance source. The radiance presence immediately switches the pitch channel to the Earth sensor error signal. The pitch attitude begins to converge with the roll rate unchanged as the other earth sensor scan continues to approach the earth disc. After planetary acquisition the next phase of achieving on-orbit mode operation is the performance of the housekeeping functions and the convergence into fine pointing.

MARS INSERTION PROPULSION SUBSYSTEM

MGO OIM

(DETAILS IN APPENDIX A-5)

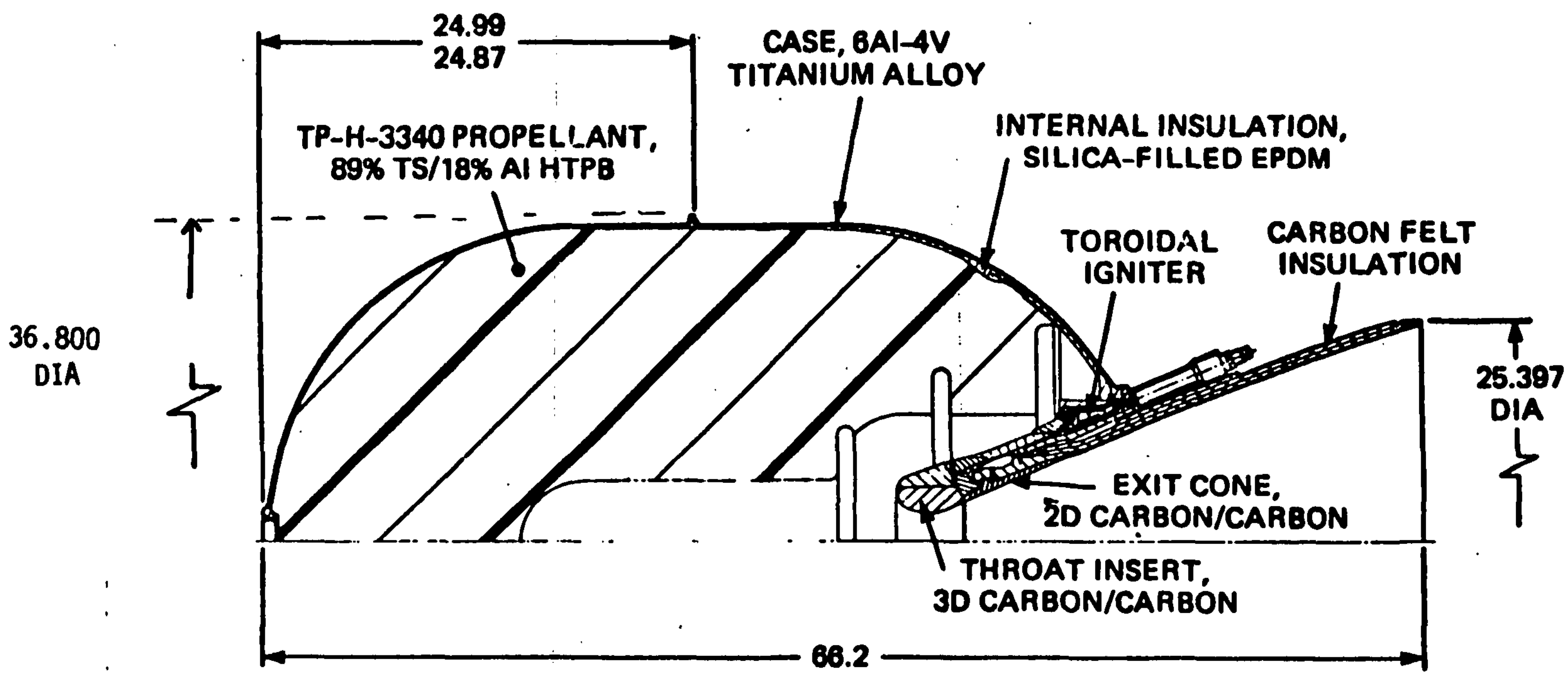
APOGEE KICK MOTOR	SYNCHRONOUS ORBIT INJECTION
	THIOKOL STAR 37F MOTOR
	2017 LBS WT 12500 LBS THRUST
	537025 LB SEC IMPULSE
HYDRAZINE SYSTEM	CAPACITY 350 LBS
	16 1-LB THRUSTERS 4R, 4P, 8Y
	4 0.1-LB THRUSTERS R-Y

FLTSATCOM PROPULSION

The table above shows the FLTSATCOM propulsion equipment characteristics. Injection into synchronous orbit is performed by an AKM mounted along the spacecraft spin axis. The AKM weighs 2,017 pounds (not including igniter, etc.) and imparts 5,736 ft/second to the spacecraft during firing. The spacecraft is spin stabilized during this firing. The velocity increment is applied so as to turn the net velocity vector from 26.7 degrees inclination of the transfer orbit to the final inclination of 2.5 degrees. The total velocity is raised to circularize the orbit. A hydrazine reaction control subsystem (RCS) fulfills the spacecraft low thrust requirements. The RCS consists of two tanks that have the capacity to hold up to 350 pounds of hydrazine if the bladder systems are removed (nominal load is 188 pounds). A system of lines and isolation valves connect the hydrazine tankage to two banks of eight 1-pound thrusters and two banks of two 0.1-pound thrusters. The 1-pound thrusters are arranged to provide plus and minus roll (X-axis) torque, plus and minus pitch (Y-axis) torque, and four thrusters are used in pairs to provide plus and minus yaw (Z-axis) torque. The yaw thrusters also provide four spacecraft spin-up, spacecraft despin, and are used to impart velocity increments to the spacecraft during on-orbit operation. When the spacecraft is spinning, the roll and pitch thrusters precess the spin vector. Redundancy is provided in the propulsion equipment by using two tanks, two sets of distribution lines, and two complete sets of thruster assemblies. Isolation valves allow isolation or cross-connection of the various sections of this system.

STAR 37FM AKM FOR FLTSATCOM

FLIGHTS 7, AND 8



DIMENSION IN INCHES

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MOI/LOI MOTOR TRADE-OFFS

For MGO - Either the FSC 6 STAR 37F (60", 2056 lbs. I_{sp} = 287.5) or the FSC STAR 37FM (66", 2481 lbs. $I_{sp} \approx 292$) could be used. The tentative choice is to use the more efficient motor, since qualification is assured, and its extra length can be accommodated in interstage structure.

- Fully loaded, the STAR 37FM provides over 1500 ft/sec excess V, which can be used to "crank" around the line of nodes away from the initial zap angle and thus hasten beginning of data taking.

- If this is not done, the motor can be off-loaded ~34%, and the 800-pound savings can go into margin, or to increase non-jettisoned ballast weight, thus reducing deployable outrigger arm length.

For LGO - The STAR 37N still appears to be the best choice, even though it is still over-capable. A smaller, lighter motor might be found, but a new FSC adapter would have to be designed.

Note that all STAR 37 motors have the same diameter and attachment fitting patterns.

STAR 37 FM FOR FSC 7, 8

WEIGHT BREAKDOWN, LBM

CASE	69.3
HEAD-END PLUG	0.4
INSULATION, CASE	25.8
LINER	1.3
NOZZLE ASSEMBLY (WITHOUT PROPELLANT)	62.1
INITIATORS (2)	1.2
BOLT, WASHERS, O-RINGS	1.6
EMPTY MOTOR	161.7
IGNITER PROPELLANT	1.0
PROPELLANT GRAIN	2319.2
LOADED MOTOR	2481.9
EMPTY MOTOR	161.7
INERT WEIGHT EXPENDED	-11.6
FIRED MOTOR	150.1
SAFE & ARM	~4LBS

OTHER ORBIT INSERTION MOTOR CHARACTERISTICS

At this writing, there are very few STAR 37Fs in storage, in addition to the one destined for FSC 6 mating. Therefore, if it should be decided to use this motor, rather than the to-be-qualified STAR 37FM, a requalification program might be needed (if a stored motor is not available) because original throat materials are no longer available.

The status of the STAR 37N motor has not been investigated, since there are several other motors that could also do the LGO job.

CHARACTERISTICS	STAR 37F FLTSATCOM	STAR 37N LGO
MANUFACTURER	THIOKOL	THIOKOL
STATUS	QUALIFIED	QUALIFIED
AVERAGE THRUST (lb_f)	12,750	8,900
I_{SP} (SECONDS)	286.3	287.8
BURN TIME (SECONDS)	40	37.5
DIAMETER/LENGTH (INCHES)	36.9/66	36.8/53
TOTAL PREBURN INERTS (lb_m)	148	140
100% PROPELLANT LOAD (lb_m)	1909	1232
LAUNCH WEIGHT (lb_m) (INCLUDING IGNITER)	2057	1372

PROGRAM COST DISCUSSION

PROGRAM COST DISCUSSION

The contract statement of work included:

"Generate cost estimates for the Mars mission as a single entity, for the Lunar mission as a single entity, and for a combined project in which both missions are conducted as part of a single contract. Consider the 1988, 1990, 1992 launch opportunities for Mars and the same time frame for the Lunar launch."

Later guidelines included:

1. All costs should be in FY'82 dollars to prevent any ambiguity as to inflation models.
2. The estimate should be broken into two parts: Start to launch plus 30 days, and launch plus 30 days to end of mission.
3. Assume mission operations conducted from JPL.
4. Assume that the majority of Mission Analysis and Navigation work is done by JPL with some level of participation by your personnel.
5. Do not include launch vehicle costs.
6. Show cost distribution by year. Break down according to a work unit system to be shown later.
7. Use the July 1988 Mars opportunity as a basis for costing the Mars only and combined option.
8. Assume a NASA/JPL Project start at the beginning of FY'85 (October 1984).

SCIENTIFIC INSTRUMENT ASSUMPTIONS
(FOR COSTING PURPOSES)

- PI'S WILL BUILD INSTRUMENTS TO MEET ENVIRONMENT SPECS PROVIDED BY JPL/TRW; AND WILL DELIVER INSTRUMENTS QUALIFIED TO THESE LEVELS

- DELIVERY WILL OCCUR AT LEAST ONE YEAR PRIOR TO S/C ACCEPTANCE

- THE FOLLOWING INFORMATION/HARDWARE WILL ACCOMPANY THE INSTRUMENTS:
  - DRAWINGS AND INTERFACE DOCUMENTS
  - TEST REQUIREMENTS AND PROCEDURES FOR SATELLITE LEVEL TESTING
  - MECHANICAL AND ELECTRICAL TEST EQUIPMENT AND HANDLING FIXTURES, AS APPLICABLE
  - CALCULATED OR TESTED MAGNETIC MOMENT INFORMATION (MGO, AT LEAST)

COSTING ASSUMPTIONS/REPORTING

In this section of Volume I, we will give the background and methodology we used in determining the costs and presenting them in the manner prescribed by JPL. The actual cost figures are presented in Volume II.

The assumptions are made in cost determination were:

- Scientific instruments are GFE; they are designed and tested against contractor provided environmental and interface specs; contractor has role in NASA acceptance; delivery is one year prior to spacecraft acceptance
- We have used 1982 dollars as a base; therefore, the cost of an equal length, equal circumstance program starting later than the baseline July '88 launch program would appear to cost the same. In actuality, various inflationary factors would apply *
- For later start programs, NASA could
  - Avoid a large lump-sum long-lead item purchase at the beginning of a compressed schedule program by starting the program at an earlier (~1 year) than minimum time
  - Minimize inflationary effects by the long-lead item purchase at the beginning of the program

* Statement applies only if program remains locked-in with FLTSATCOM 6, 7, 8 (or follow-on) program.

METHODOLOGY TO DETERMINE COSTS OF BASIC CONFIGURATIONS

- START WITH BASIC BUS PRICE PROVIDED BY FLTSATCOM

MGO

- ADD COST OF STS INTEGRATION
- ADD COST OF INTEGRATING SCIENTIFIC INSTRUMENTS
- ADD MECHANICAL COST OF HGA
- ADD COST OF STEM DEVICES
- ADD Δ COST OF CONICAL SCANNER (2)
- ADD COST OF STELLAR SCANNER
- ADD Δ COST OF DATA HANDLING SYSTEM & DEV.
- ADD Δ COST OF AVCS SYSTEM CHANGES *
- ADD Δ COST OF MAG. CLEANLINESS PROGRAM

LGO

- ADD COST OF INTEGRATING SCIENTIFIC INSTRUMENTS
- ADD COST OF STEM DEVICES
- ADD Δ COST OF CONICAL SCANNER (4)
- ADD Δ COST OF DATA HANDLING SYSTEM & DEV.
- ADD Δ COST OF AVCS SYSTEM CHANGES *
- ADD Δ COST OF 45° SOLAR PANELS

ASSUMPTIONS

MGO:

- POWER SYSTEM EXCHANGE IS EVEN (SOLAR PANELS)

LGO:

- STAR 37N INSTALLATION EXCHANGE IS EVEN
- POWER SYSTEM EXCHANGE IS EVEN (SOLAR PANELS)

SAVINGS

MGO:

- ELIMINATE COST OF HARDNESS TESTING
- ELIMINATE COST OF SHIELDING PLATING

LGO:

- ELIMINATE COST OF HARDNESS TESTING
- ELIMINATE COST OF SHIELDING PLATING

* NONE ANTICIPATED ON BASELINE CONFIGURATIONS

COST SUMMARY (REFER TO PAGE I-16, AND VOLUME II)

As noted previously, detailed costs are presented in Volume II. Qualitatively, it should be noted:

- MGO spacecraft cost is considerably less (~20% ?) than a "new" spacecraft TRW designed to perform the similar NASA ARC Climatology mission.
- The LGO mission, if Atlas-Centaur launched, is about $10M less than MGO.
- If both MGO and LGO are produced concurrently, over $25M can be saved over individual costs.

The cost minimizations made possible from adaption of the FSC spacecraft stem from savings due to:

- <u>Direct Utilization or Minor Adaptation of Following Major Subsystems</u>
  - Structure
  - Thermal Control
  - RCS
  - Power
  - Attitude Det. & Control
  - Final Orbit Insertion
- <u>Adaptation of Interfaces</u>
  - Umbilical
  - Mechanical & Separation
- <u>Adaptation of Services</u>
  - Modules of TT&C
  - Command Architecture
  - Propellant Supply
  - Launch Operations Architecture
  - Redundancy/Fault Architecture
  - Operational Sequences
  - Housekeeping Instrumentation
  - Safety, S & A, etc., devices
  - Ground Control Software, etc.
  - Fault Tolerant Features

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1988 MGO LAUNCH COSTS
(BASELINE CONFIGURATION)

	'85	'86	'87	'88	'89	'90	TOTALS
START TO L+30							
MANAGEMENT							
MISSION ANALYSIS AND NAVIGATION							
SPACECRAFT							
SCIENCE INTEGRATION							
ASSEMBLY, TEST AND LAUNCH OPERATIONS							
TOTAL							
L+30 TO EOM (LEVEL OF EFFORT)				495 DAYS			.3
TOTAL							
FEE @ 10%							
GRAND TOTAL							

COST IN MILLIONS OF DOLLARS

SYSTEMS ENGINEERING

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BASELINE CONFIGURATION COSTS

In Volume II, in addition to the completed version of the above JPL-requested-format chart, are two others with the following captions:

1988 LGO Launch Costs
(Baseline Configuration)

1988 Concurrent MGO/LGO Launch Cost
(Baseline Configurations Via Common STS Launch)

The baseline spacecraft are:

	MGO	LGO
• Launch System	STS/SRM-1	A-Centaur
• Designed for STS/safety/loads	Yes	No
• Added propellant capacity	No	No
• Two axis HGA	Yes	No
• S and X band TT&C	Yes	S only
• Conical horizon scanners	Yes (two)	Yes (four)
• Category 1 payload only	Yes (3 + A)*	Yes (3 + B)*
• Spinning stellar sensor	Yes	No (FSC spinning earth sensor)
• Active nutation control	No	No
• Stem device for instrument mounting and extension	Yes	Yes
• Optimum Star 37 motor	Yes (FM)	Yes (N)
• 'Straight' solar array	Yes (optimum size)	No (same size; at 45°)
• Momentum wheel mounted as in FSC	Yes	Yes
• Power system	Optimized for MGO	Optimized for LGO
• Magnetic cleanliness program	Yes	No

* i.e., these instruments are common

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MGO SPACECRAFT COST(1) MATRIX SUMMARY
(X10^6$-1982)*
START TO LAUNCH PLUS 30 DAYS

DESCRIPTION	REMARKS	ROM COST - LAUNCH IN 1988
• MGO BASELINE	SIMILAR TO FSC 7, 8 DESIGN	
• Δ COST TO ADD PRESSURANT BOTTLE AND REGULATOR VALVE	PERMITS ~80 ADDITIONAL POUNDS OF PROPELLANT	
• Δ COST TO (MOUNT ON SRM-1)		
- ADD BALLAST TO FORWARD CARRIER RING	TO MAKE MOMENT OF INERTIA RATIO MARGIN GREATER THAN 5%	
- ADD SOLID SPIN-UP AND YO-YO DESPIN DEVICES	TO SAVE LIQUID PROPELLANT	
• Δ COST TO ADD ANC TO SPACECRAFT	IF MORE COST EFFECTIVE THAN ADDING BALLAST	
• Δ COST TO SUPPORT OPERATIONS AFTER LAUNCH + 30 DAYS	SUPPORT MOSTLY "ON CALL"	
• Δ COST TO ADD NASA ARC INSTRUMENTS	COST Δ SHOWN IS PER INSTRUMENT	
• Δ COST FOR LATER LAUNCH DATE	ASSUME PROGRAM STARTS Δ2 YEARS LATER	
- 1990		
- 1992	FSC PROGRAM OFFICE RESTART	

(1) ADD 10% TO TOTAL COST TO GET PRICE

* COSTS INCLUDE INTEGRATION WITH STS (8M) AND WITH SRM-1 (2M) ASSUMING INTELSAT VI ASE

SYSTEMS ENGINEERING

MGO COST VARIATIONS

The above shows how MGO costs are shown in Volume II, with possible design variations which, in part, may stem from launch vehicle selection, or the addition of propellant supply.

The "ballast" addition of the third bullet (above) would be in lieu of adding ANC (4th bullet). A more detailed design and operational analysis would be necessary to determine the most cost effective way of solving the stability problem accrued by selection of the spinning SRM-1 booster. The addition of solid spin motors and yo yo devices would be indicated if trade studies shows their addition would be more effective than adding propellant (2nd bullet).

The next two pages show similar listings for:

Page VI-10 - LGO

Page VI-II - Concurrent LGO/MGO Programs

LGO SPACECRAFT COST (1) MATRIX SUMMARY

($$X10^6$ - 1982)

START TO LAUNCH PLUS 30 DAYS

DESCRIPTION	REMARKS	ROM COST - LAUNCH IN 1988 CATEGORY I ONLY
• ATLAS-CENTAUR BASELINE	SOLAR PANELS AT 45°; OTHERWISE LIKE FSC 6	
• STS/PAM A BASELINE	- COSTS INCLUDE INTEGRATION WITH STS	
- WITH ANC (Δ COST)	- ANC WILL BE REQUIRED IF PAM A MOTOR CANNOT BE BALLASTED TO ACHIEVE FAVORABLE MOMENT OF INERTIA RATIO	
• Δ COST TO ADD GYRO PACKAGE	IF (1°) NADIR POINTING MAX ERROR CANNOT BE TOLERATED	
• Δ COST IF STELLAR SENSOR ADDED	TO DETERMINE ALTITUDE DURING ECLIPSE	
• Δ COST IF BOTH CATEGORY I AND II INSTRUMENTS CARRIED		
• Δ COST TO SUPPORT OPERATIONS AFTER LAUNCH + 30 DAYS	SUPPORT MOSTLY "ON CALL"	
• Δ COST FOR LAUNCH IN		
- 1990		
- 1992	FSC PROGRAM OFFICE RESTART	

(1) ADD 10% TO TOTAL COST TO GET PRICE

COMBINED MGO/LGO COST(1) MATRIX (X 10^6, 1982)
(SEE DEFINITION OF BASIC MGO/LGO)
TO L + 30 DAYS

	COMMONALITY IMPACTS	ROM COST AT LAUNCH IN 1988 CATEGORY I ONLY
BASIC SYSTEMS	NO STRIVING FOR MAX COMMONALITY - SEQUENTIALLY MANUFACTURED AND DELIVERED	—
Δ COST FOR NEAR SIMULTANEOUS DELIVERY	FOR COMMON STS LAUNCH	—
Δ COST FOR SIMULTANEOUS LAUNCH IN		
1990		—
1992	FSC PROGRAM OFFICE RESTART	—

(1) ADD - 10% TO TOTAL COST TO GET PRICE

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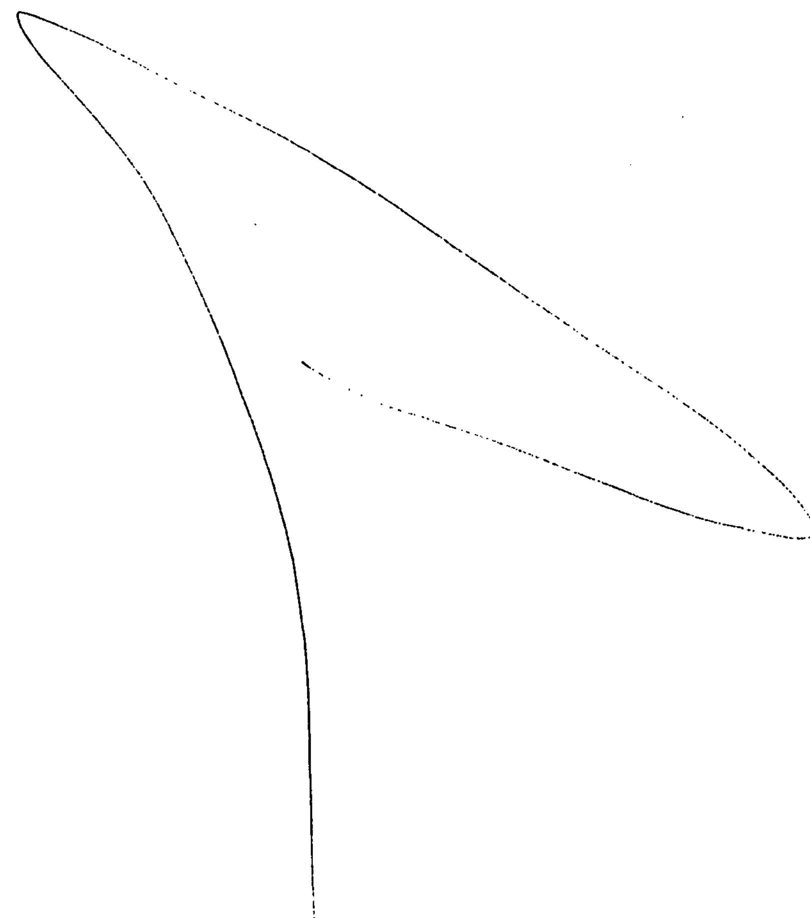

VII. CONCLUSIONS AND

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CONCLUSIONS AND RECOMMENDATIONS

SYSTEMS ENGINEERING

THE ADVANTAGES OF THE FLTSATCOM BUS

The FLTSATCOM Affords:

- Ample performance margins for both interplanetary launch and orbit insertion, thus precluding weight problems, and permitting great flexibility of operation.
- Ready integration of MGO instruments and both Category I and II LGO instruments, all with clear fields of view and necessary radiative surfaces.
- The ability to incorporate the two NASA arc prime MGO climatology mission instruments not carried by MGO - thus performing all (i.e., geoscience and climatology) the scientific measurements desired by both government laboratories.
- Flexibility in the mission plan to compensate for glitches, because of its over-performance ability.

GENERAL CONCLUSIONS FROM THE MGO/LGO STUDY

- IF AN EXISTING SPACECRAFT CAN BE ADAPTED WITHOUT MAJOR MODIFICATION FOR PLANETARY MISSIONS, SIGNIFICANT COST SAVINGS ARE POSSIBLE

- USE OF AN EXISTING SPACECRAFT CAN PROVIDE SIZEABLE WEIGHT, POWER AND VOLUME MARGINS - THUS REDUCING THESE CONSTRAINTS ON SCIENCE PAYLOADS

- FOR THE SPACECRAFT ADAPTATION TO BE ECONOMICALLY SUCCESSFUL, THE SCIENCE PAYLOAD MUST BE WILLING TO RELAX DIFFICULT SPACECRAFT PERFORMANCE DEMANDS

- UNTIL A VARIETY OF STS COMPATIBLE SPACECRAFT BECOME AVAILABLE, AN ADDITIONAL COST WILL BE INCURRED TO ADAPT FROM AN ELV TO THE STS

- AS A RESULT OF THE STUDY, ALL TECHNICAL PROBLEMS HAVE EITHER BEEN SOLVED OR OPTIONS MADE VISIBLE

CONCLUSION COMMENTS

There are no fundamental reasons why an earth orbiter which contains integrated propulsion features and autonomy during cruise-out to final orbit cannot be economically converted to a planetary orbiter for "nearby" planets. The 7 to 10 year lifetimes of many present day earth orbiters can be extended to longer lifetimes, thus assuring success of long term planetary cruises. The main weakness of direct conversions falls into the fault tolerance/autonomous action area. Most designs can take fairly simple steps to make large gains; however, great sophistication cannot be achieved if price is to be kept down.

If an early MGO launch is to be achieved, several programming factors should be observed and/or attained:

Studies/competition(s) to determine a contractor and projected costs in next two years commencing in late 1982 or early 1983.

1988 launch will be difficult to achieve unless tight schedule is maintained in next two years. We proposed, as requisite to a 1988 launch:

- RFP for non-competitive costing in October 1984
- GFE instruments delivered 1+ year prior to launch.

In a hardware program, the TRW MGO/LGO program office would:

- Buy modified FLTSATCOM bus from FSC program office
- Install GFE instruments and other modifications in assembly and test
- Perform system tests to supplement prior testing at bus level.

Costs do not reflect FLTSATCOM 6, 7, 8 negotiations (due to FSC/AF slippage); costs could be considerably more detailed as output of continuation program.

NEXT PHASE PROGRAM

(IN ADDITION TO COST REVIEW BASED ON FSC 6, 7, 8 NEGOTIATIONS)

- ADDITION OF AUTONOMOUS FAULT TOLERANT FEATURES
- ADDITION OF ANC SYSTEM; EFFECT OF PROPELLANT SLOSHING
- INJECTION ACCURACY (INCLUDING SPIN-UP EFFECTS)
  - FROM LEO
  - INTO MARTIAN ORBIT
- EFFECT OF BOOM MOUNTED EXPERIMENTS ON SPACECRAFT DYNAMICS DURING CRUISE-OUT CALIBRATION (WHILE ROLLING) AND ON-ORBIT
- LUNAR PERTURBATIONS AND THEIR AFFECT ON POINTING DURING ECLIPSE
- OPTION SELECTION IMPACTS (SEE BELOW)

STUDY CONTINUATION

In the next phase of the program, the major tasks would be examining the engineering and design implications of the option selection JPL makes as a result of the trade-off choices offered in this report. TRW will provide more "hard" data to enable JPL choices, and will perform more detailed analysis and program planning to permit a review of the costs estimated in the present study. This study will be aided by completion of FSC 6, 7, & 8 pricing and negotiations.

The continuation will be most heavily impacted by two JPL directions:

1. Most probable launch date - as it impacts the schedule and cost of program progress
2. The selection of a launch vehicle or vehicles - as it impacts structural design and the need for active nutation control

Other options, of varying importance, must also be determined:

1. The need for, and degree of autonomous features to be added to the spacecraft, particularly the MGO
2. Detailed error analyses to determine if added MGO propellant is needed
3. Final selection of MGO/LGO orbit injection motors, based on availability, qualification status, and cost as of purchase time
4. Need for improving LGO pointing during eclipse

VIII. APPENDICES

APPENDICES

MGO (MARS ORBITER) ORBIT INJECTION GEOMETRY FOR THREE MISSION YEARS

AN ANALYSIS WAS PERFORMED TO DETERMINE THE LOCATIONS OF MARS ORBIT INJECTION POINTS (MOI) AND THE ORIENTATIONS OF THE SPACECRAFT X-AXIS PRIOR TO INJECTION, RELATIVE TO THE SUN AND EARTH. RESULTS ARE LISTED IN TABLE 1. RELEVANT PARAMETERS OF THE ARRIVAL TRAJECTORY TYPICAL FOR EACH OF THE THREE MISSION YEARS (1988, 90 AND 92) ARE LISTED IN TABLE 2. THESE DATA ARE FROM A LAUNCH AND TRANSFER TRAJECTORY ANALYSIS PERFORMED BY PETE CRESS, USING HARVEY GOODMAN'S ANALYTICAL INTERPLANETARY PROGRAM, AIP.

THE ENCLOSED SPHERICAL PROJECTION CHARTS SHOW ARRIVAL TRAJECTORY TRACKS ON A NON-ROTATING COORDINATE SYSTEM ORIENTED TO THE SUN(S) AND THE MARS ORBIT PLANE, AND ALSO THE ORIENTATIONS OF THE MARTIAN EQUATOR AND THE NORTH AND SOUTH POLES. DATA LISTED IN TABLE 1 WERE OBTAINED BY GRAPHICAL ANALYSIS BY MEANS OF THESE PLOTS. TRAJECTORY PARAMETER SYMBOLS LISTED IN TABLE 2 ARE THOSE USED IN THE TRW TRAJECTORY SIMULATION PROGRAM (SEE ATTACHED DEFINITIONS).

THE RESULTS SHOW THAT, BY CHOOSING THE NORTHERN APPROACH ROUTE, PREFERRED X-AXIS ORIENTATIONS TO THE SUN AND EARTH AND FAVORABLE MOI LOCATIONS ARE OBTAINED. THE FOLLOWING CHART IS A SUMMARY OF SYSTEM AND MISSION DESIGN IMPLICATIONS DERIVED FROM THESE RESULTS.

SUMMARY OF SYSTEM AND MISSION DESIGN IMPLICATIONS

1. MOI X-AXIS ORIENTATION OF ABOUT 70 TO 100 DEG FROM SUN LINE, FAVORABLE FOR ILLUMINATION OF FOLDED SOLAR ARRAY, OBTAINABLE IN ALL THREE MISSION YEARS FOR BOTH NORTHERLY (N) AND SOUTHERLY (S) APPROACHES.

2. MOI X-AXIS ORIENTATIONS OF ABOUT 50 TO 80 DEG FROM EARTH LINE, FAVORABLE FOR FORWARD OMNI-ANTENNA COVERAGE, ARE OBTAINED IN ALL MISSION YEARS IN N APPROACH. S APPROACH UNFAVORABLE IN 1992, RESULTING IN X-AXIS AT 109 DEG FROM EARTH LINE.

3. MOI LOCATIONS AT HIGH, NEAR-POLAR LATITUDES ARE OF INTEREST IF MAJOR CHANGE OF CAPTURE ORBIT ASCENDING NODE IS TO BE ACHIEVED BY UTILIZING LARGE SPARE ΔV CAPACITY OF THE SOLID ORBIT INJECTION MOTOR, AS IN THE FLTSATCOM MISSION CONCEPT. IN 1988 BOTH N AND S APPROACHES RESULT IN SIMILAR MOI LATITUDES ($\sim 50^\circ$N OR 50°S), FAR FROM POLES. IN 1990 AND 1992 $MOI_{(N)}$ LOCATIONS ARE AT 87 AND 63 DEG N LATITUDES, MUCH CLOSER TO THE POLE THAN THE CORRESPONDING $MOI_{(S)}$ LOCATIONS.

4. AVOIDANCE OF EARTH OCCULTATION PRIOR TO MOI: IN 1988 THIS IS ACHIEVABLE ONLY BY S APPROACH. IN TYPE II MISSIONS, 1990 AND 1992, BOTH N AND S APPROACH RESULT IN MOI LOCATIONS WELL WITHIN E VIEW. BOTH $MOI_{(N)}$ AND $MOI_{(S)}$ ARE ON MARS' SUNWARD SIDE, AND E ONLY 17 AND 22 DEG FROM S IN THESE YEARS, COMPARED WITH 40 DEG IN 1988 MISSION YEAR (TYPE I TRANSFER).

5. LARGE DIFFERENCES IN MARS' EQUATORIAL AND POLAR ORIENTATIONS IN THE SPHERICAL PLOTS ARE EXPLAINED BY THE DIFFERENT ARRIVAL DATES WITHIN MARTIAN YEAR.
   - 1988 = $\sim 10^\circ$ PRIOR TO N VERNAL EQUINOX (NVE)
   - 1990 = 102° AFTER NVE
   - 1992 = 129° AFTER NVE

MOI LATITUDES AND X-AXIS ORIENTATIONS* AT MOI (DEGREES)

TABLE 1

YEAR	N - APPROACH		S - APPROACH		MOI LATITUDE	
	X_{MOI} TO SUN	X_{MOI} TO EARTH	X_{MOI} TO SUN	X_{MOI} TO EARTH	MOI_N	MOI_S
1988	69	81	77	38	50	-50
1990	69	52	61	58	87	-17
1992	97	77	100	109	63	-35

* X-AXIS ORIENTATIONS AT MOI DETERMINED BY GRAPHICAL ANALYSIS.

ESTIMATED ERROR ± 2 DEG.

APPROACH TRAJECTORY AND MOI PARAMETERS (DEGREES)(1)

TABLE 2

PARAMETER(2) (PRINTOUT SYMBOL)	1988 ARR. 1/28/89		1990 8/16/91		1992 8/31/93	
LATTP	0.7		1.3		0.5	
LONTP	73.3		185.9		213.2	
GAT	-26.6		23.0		21.8	
LAT	105.9		50.4		71.2	
ZAT	104.2		54.1		72.6	
INCHYP	28.0		23.2		23.3	
PHI	50.9		52.6		49.0	
LTS	0.1		35.3		14.3	
LNS	103.9		62.5		76.5	
	N	S	N	S	N	S
BETA (3)	163.3	14.4	143.9	37.2	156.9	26.9
GAMA (3)	-16.0	-5.3	23.8	13.7	29.8	-3.4
LTRP (3)	51.0	-50.7	86.7	-17.3	63.2	-34.7
LNRP (3)	107.1	107.0	112.6(4)	65.2	81.0	79.0

NOTES:

(1) AS OBTAINED BY PETE CRESS AND JOE VOGL FROM TRW PLANETARY MISSION TRAJECTORY PROGRAM (AIP) FOR CAPTURE ORBIT INCLINATION $i = 92.6^{o}$

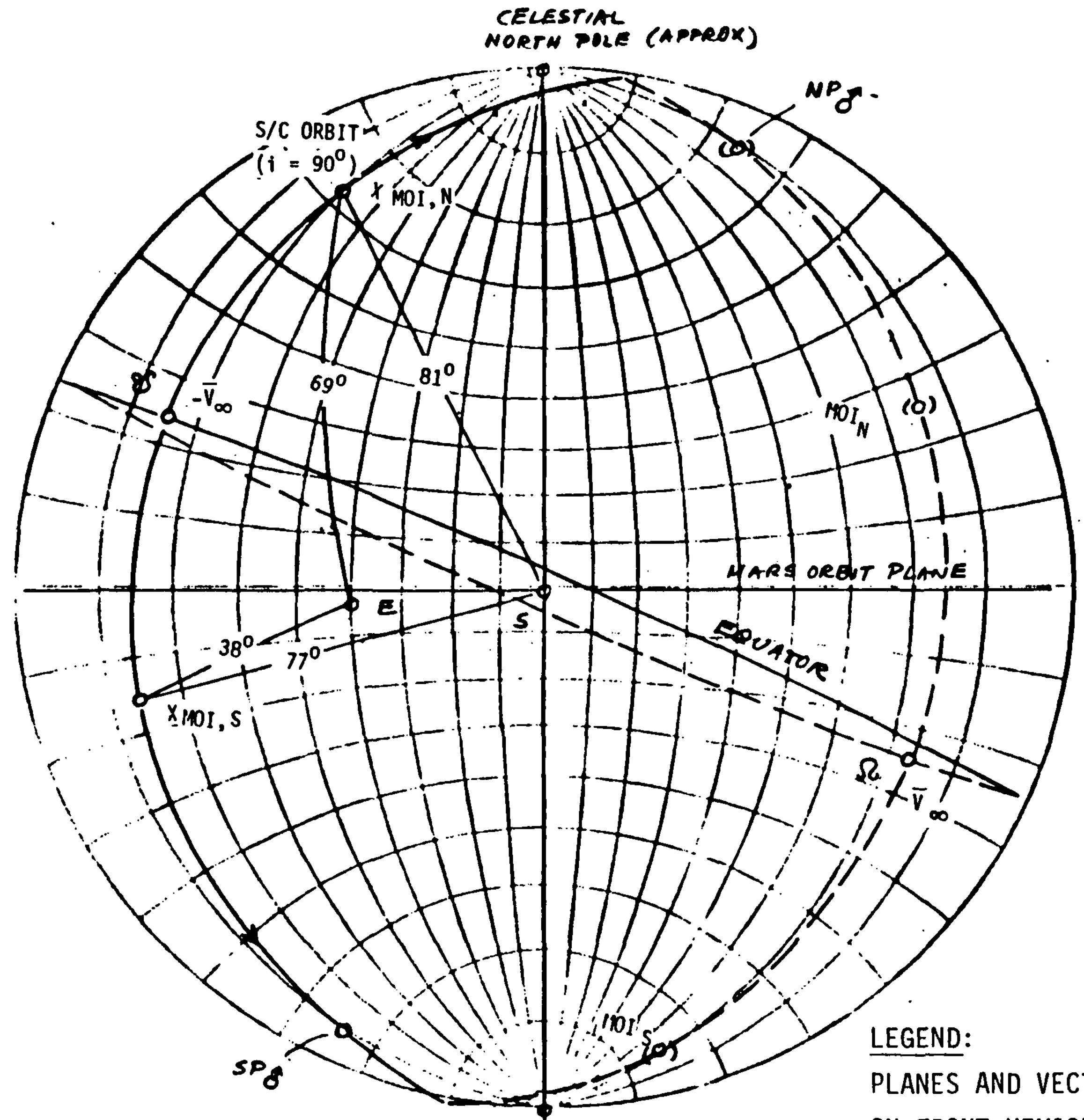
(2) DEFINITION OF PARAMETERS GIVEN IN ENCLOSED NOTES

(3) RESULTS SHOWN FOR NORTHERLY (N) AND SOUTHERLY (S) APPROACH TO MOI

(4) THIS VALUE APPARENTLY INCORRECT, BASED ON GRAPHICAL INTERPRETATION OF APPROACH CONDITIONS

ORBIT GEOMETRY AT MARS ARRIVAL 1/28/89

SOUTH AND NORTH MOI OPTIONS



LEGEND:

PLANES AND VECTORS

——o—— ON FRONT HEMISPHERE

– – (o) – – ON REAR HEMISPHERE

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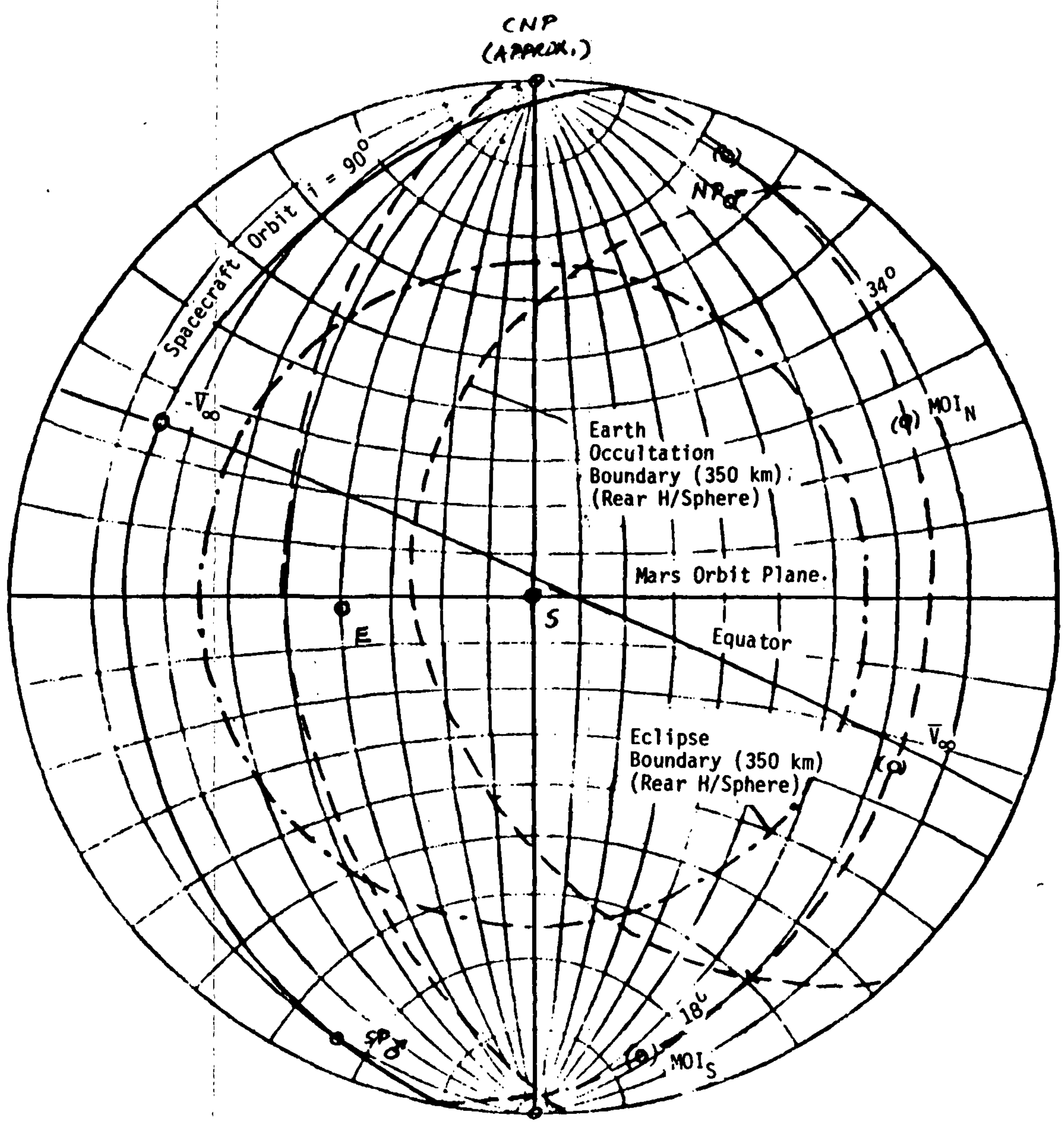
EARTH AND SUN POSITION RELATIVE TO SPACECRAFT ORBIT -
ECLIPSE AND OCCULTATION BOUNDARIES (1/28/89)



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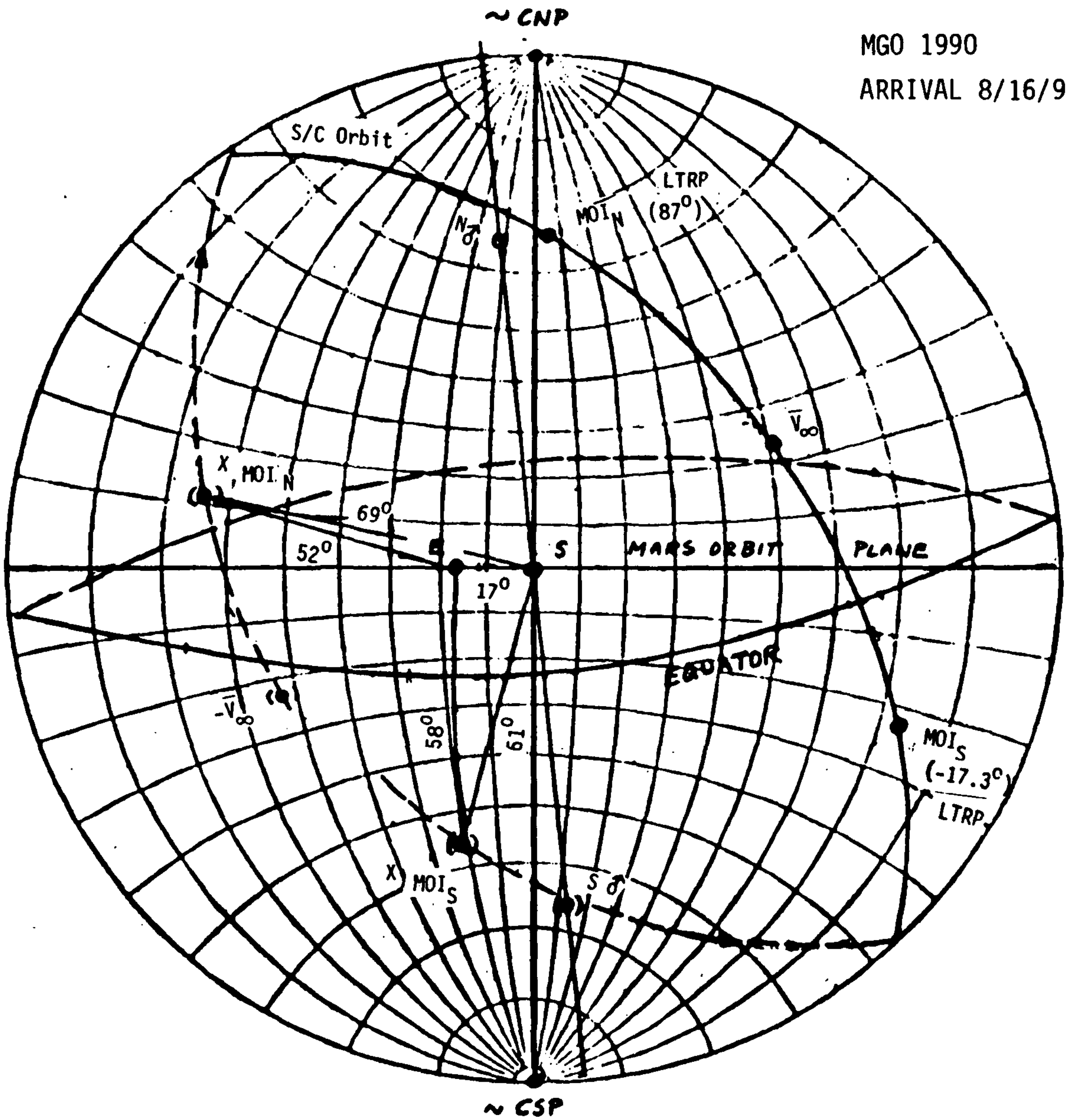
ORBIT GEOMETRY AT MARS ARRIVAL 8/16/91



MGO 1990

ARRIVAL 8/16/91

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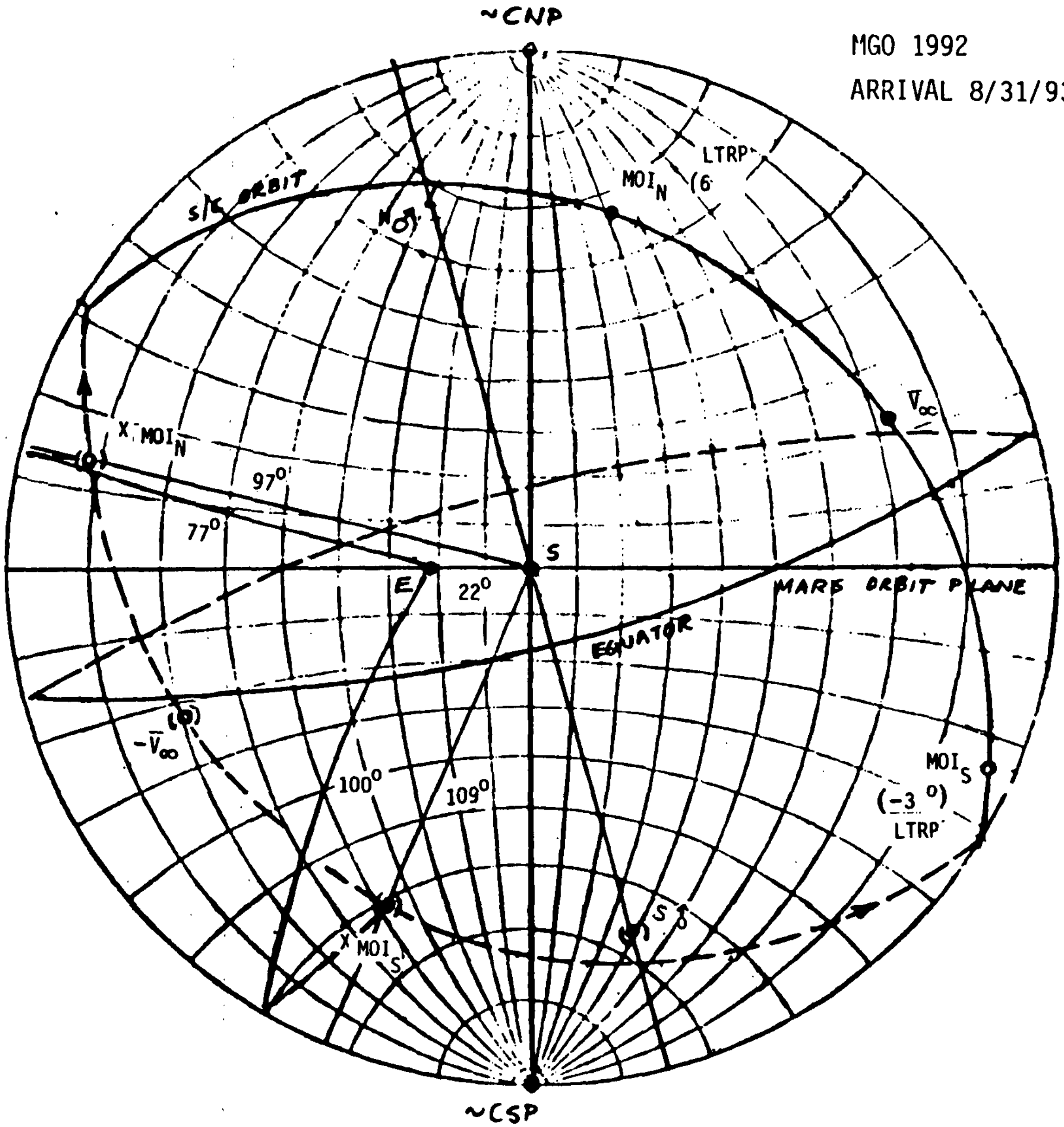
JULY 1982
HANS F. MEISSINGER

ORBIT GEOMETRY AT MARS ARRIVAL 8/31/93

MGO 1992

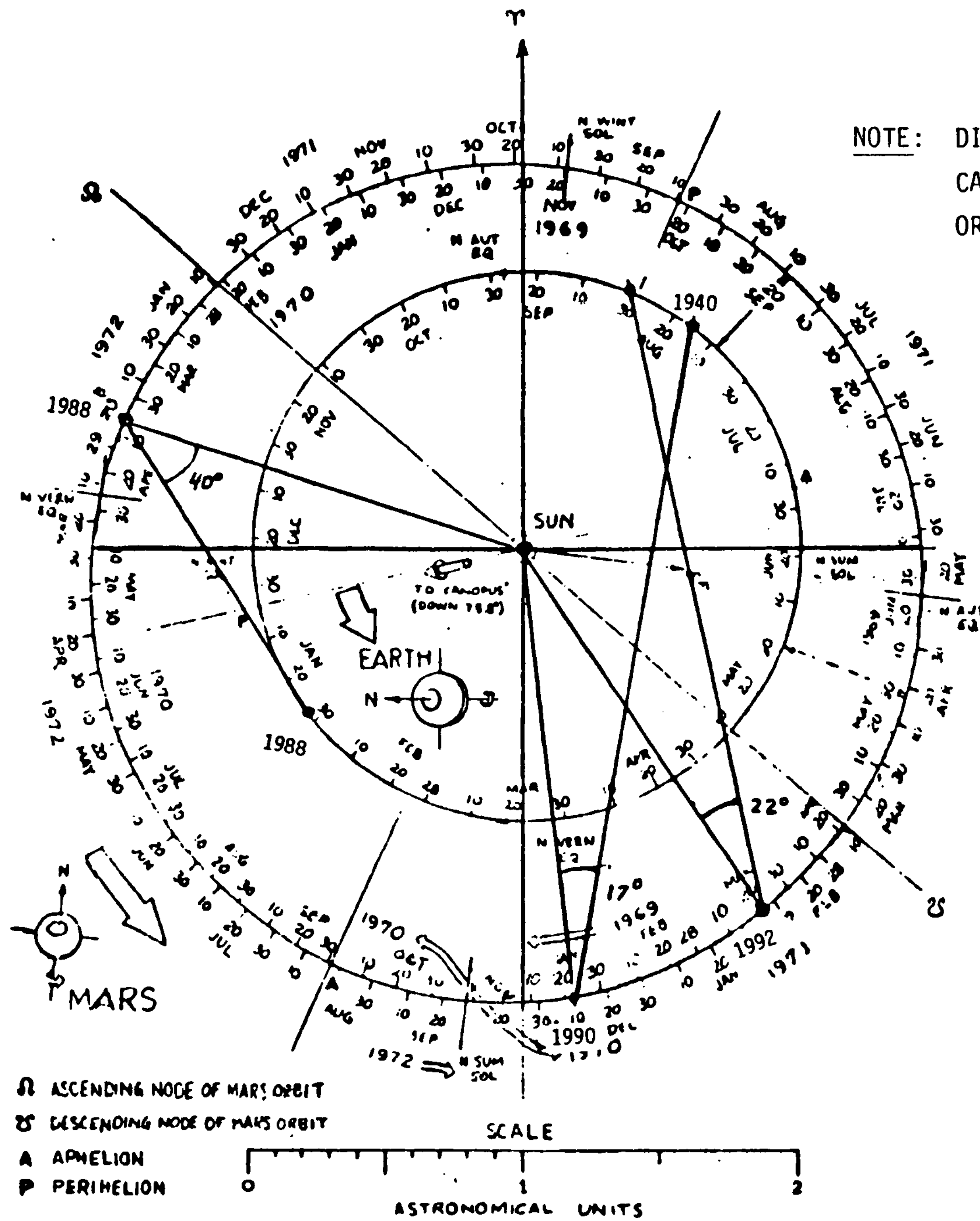
ARRIVAL 8/31/93



JULY 1982
HANS F. MEISSINGER

EARTH, MARS AND SUN RELATIVE POSITIONS
IN 1988, 1990 AND 1992 MISSION YEARS

NOTE: DISREGARD 1969 - 1972 CALENDAR DATES ON MARS' ORBIT



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MARCH 1965
WILLIAM J. DIXON

DEFINITION OF TRAJECTORY PARAMETERS (EXCERPT)

MNEMONIC	DEFINITION
LATTP	ECLIPTIC LATITUDE OF THE TARGET PLANET AT ARRIVAL; DEGREES
LONTP	ECLIPTIC LONGITUDE OF THE TARGET PLANET AT ARRIVAL, MEASURED FROM THE VERNAL EQUINOX; DEGREES
INCHYP	INCLINATION OF THE APPROACH HYPERBOLIC ORBIT PLANE TO THE ECLIPTIC; DEGREES
GAT	ANGLE BETWEEN THE APPROACH V_{∞} ASYMPTOTE AND ITS PROJECTION ON THE TARGET PLANET'S ORBITAL PLANE, MEASURED POSITIVE ABOVE THE TARGET PLANET'S ORBITAL PLANE; DEGREES
LAT	ANGLE BETWEEN THE PROJECTION OF THE APPROACH V_{∞} ASYMPTOTE ON THE TARGET PLANET'S ORBITAL PLANE AND THE TARGET PLANET-SUN DIRECTION, MEASURED COUNTERCLOCKWISE FROM THE TARGET PLANET-SUN DIRECTION RADIUS; DEGREES
ZAT	ANGLE BETWEEN THE APPROACH V_{∞} ASYMPTOTE AND THE TARGET PLANET-SUN DIRECTION, MEASURED COUNTERCLOCKWISE FROM THE TARGET PLANET-SUN DIRECTION RADIUS; DEGREES
PHI	THE ANGLE FROM THE INCOMING S VECTOR TO THE LOCUS OF LOCATIONS OF THE PERIAPSIS VECTOR FOR NO APSIDAL ROTATION (DEGS)
LTS	LATITUDE OF THE INCOMING S VECTOR IN THE PLANET EQUATORIAL SYSTEM (DEGS)

DEFINITION OF TRAJECTORY PARAMETERS (EXCERPT) (CONTINUED)

MNEMONIC	DEFINITION
LNS	LONGITUDE OF THE INCOMING S VECTOR IN THE PLANET EQUATORIAL SYSTEM WHERE 0^{o} LONGITUDE IS DEFINED BY THE MERIDIAN CONTAINING THE PLANET-SUN VECTOR (DEGS)
BETA	INCLINATION OF THE CAPTURE ORBIT PLANE WITH RESPECT TO THE TERMINATOR PLANE (DEGS)
GAMA	ANGLE FROM THE PERIAPSIS VECTOR TO THE TERMINATOR PLANE (DEGS)
LTRP	LATITUDE OF THE PERIAPSIS VECTOR IN EQUATORIAL SYSTEM (DEG)
LNRP	LONGITUDE OF THE PERIAPSIS VECTOR IN THE EQUATORIAL SYSTEM (DEGS)

LGO POINTING ERRORS DURING ECLIPSE OPERATIONS

DURING ECLIPSE OPERATIONS, THE LUNAR MISSION ORBITER LOSES ITS ATTITUDE REFERENCE* AND DRIFTS IN ATTITUDE. THE ATTITUDE DRIFT WILL LAST FOR APPROXIMATELY 0.77 HOUR (LUNAR ECLIPSE PERIOD). THE MAGNITUDE OF THE ATTITUDE DRIFT IS A FUNCTION OF EXTERNAL DISTURBANCES (GRAVITY GRADIENT AND SOLAR PRESSURE), REACTION WHEEL SPEED CHANGES, AND RESIDUAL SPACECRAFT RATES AT THE TIME OF CONTROLLER DEACTIVATION. SOLAR DISTURBANCE TORQUES ARE NOT A DRIFT FACTOR SINCE ECLIPSE OPERATIONS ARE BEING CONSIDERED. IN THE FOLLOWING DISCUSSION, THE EFFECTS OF REACTION WHEEL SPEED DRIFT, RESIDUAL SPACECRAFT RATES AND GRAVITY GRADIENT DISTURBANCES ON SPACECRAFT ATTITUDE DRIFT WILL BE CONSIDERED. AS A FINAL RESULT, A TOTAL ESTIMATE OF THE LGO ATTITUDE DRIFT DURING LUNAR ECLIPSE WILL BE GIVEN.

1) REACTION WHEEL SPEED CHANGES

DISTURBANCES FROM THE REACTION WHEEL PRODUCE ATTITUDE ERRORS IN PITCH (ABOUT Y-AXIS). THE ATTITUDE DISTURBANCE RESULTS FROM THE MOMENTUM EXCHANGE BETWEEN THE REACTION WHEEL AND THE SPACECRAFT. WHEN ECLIPSE OPERATIONS START, THE ATTITUDE CONTROLLER WOULD ACT TO COMMAND THE REACTION WHEEL TO THE LAST VALUE BEFORE ECLIPSE. THE RESULTANT DISTURBANCES WOULD THEN DEPEND ON HOW WELL WHEEL SPEED IS MAINTAINED CONSTANT. BASED ON A CONVERSATION WITH J. GREGORY (CONTROL ELECTRONICS), A WHEEL SPEED CONTROLLER STABILITY OF 0.01% CAN BE OBTAINED WITH THE FLTSATCOM TYPE WHEEL. ASSUMING THE WHEEL SPEED COMMAND IS LATCHED AT THE TIME OF ECLIPSE, THE MAXIMUM MOMENTUM IMPARTED TO THE VEHICLE WOULD BE

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* THE LGO ATTITUDE CONTROL SYSTEM IS SIMILAR TO THE FLTSATCOM SPACECRAFT. IT USES AN "EARTH" SENSOR FOR NADIR REFERENCE, A REACTION WHEEL FOR PITCH CONTROL, AND 0.1 LB THRUSTERS FOR ROLL/YAW CONTROL. UPON LOSS OF NADIR REFERENCE, THE CONTROL SYSTEM IS DEACTIVATED.

$$0.01\% \times 6.77 \text{ FT-LB-SEC}^{*} = 6.8 \times 10^{4} \text{ FT-LB-SEC}$$

THE TOTAL PITCH POINTING ERROR AT THE END OF ECLIPSE OPERATIONS (0.77 HOURS) WOULD BE

$$\theta = \frac{(6.77 \times 10^{-4} \text{ FT-LB-SEC})\ (57.28) \text{ DEG/RAD } 2722 \text{ SEC}}{600 \text{ SLUG-FT}^2}$$

$$\theta = 0.174 \text{ DEGREE}$$

2) RESIDUAL RATES

WHEN THE ATTITUDE CONTROLLER IS DEACTIVATED, THE SPACECRAFT WILL DRIFT AT THE BODY RATES OF DEACTIVATION. THE PITCH BODY RATE IS NEGLIGIBLE SINCE THE REACTION WHEEL IS CONTROLLING POINTING TO WITHIN THE SENSOR QUANTIZATION. THE ROLL/YAW BODY RATE RESULTS FROM THE THRUSTER CONTROLLER LIMIT CYCLE RATES. THE ESTIMATED WORST CASE LIMIT CYCLE RATE** IS

$$0.1 \text{ LB} \times 2 \text{ FT} \times 0.02 \text{ SEC} = 0.004 \text{ FT-LB-SEC}$$

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* ESTIMATED LGO REACTION WHEEL MOMENTUM.

** BASED ON FLTSATCOM EXPERIENCE, THE WORST CASE LIMIT CYCLE IS ONE THRUSTER FIRING PER DEAD ZONE CONTACT.

AT THE END OF THE ECLIPSE PERIOD, THIS WILL PRODUCE A ROLL ATTITUDE ERROR OF

$$\tan \phi = \frac{0.004}{6.77}$$

$$\phi \cong 0.034 \text{ DEGREE}^{***}$$

3) GRAVITY GRADIENT DISTURBANCES

GRAVITY GRADIENT DISTURBANCES ACT TO ALIGN THE SPACECRAFT PRINCIPAL INERTIA AXIS WITH THE LOCAL GRAVITY FIELD. NOMINALLY THE STABLE ALIGNMENT WOULD BE ALONG THE SPACECRAFT TO NADIR VECTOR. THE GRAVITY GRADIENT DISTURBANCE BECOMES AN ATTITUDE DRIFT FACTOR UPON CONTROLLER DEACTIVATION IF THE ATTITUDE SENSOR NULL POINTS ARE NOT ALIGNED WITH THE PRINCIPAL INERTIA AXIS. THE GRAVITY GRADIENT DISTURBANCE WILL ACT TO CAUSE THE SPACECRAFT TO SLOWLY OSCILLATE ABOUT THE NULL GRAVITY GRADIENT TORQUE ATTITUDE. THE AMPLITUDE OF THE OSCILLATION WILL BE THE DIFFERENCE BETWEEN THE NADIR SENSOR NULL ALIGNMENT AND THE PRINCIPAL INERTIA AXIS ALIGNMENT. FOR THE LGO SPACECRAFT, THE SENSOR NULL LOCATION SHOULD BE CONTROLLED TO WITHIN 1.0 DEGREE OF THE PRINCIPAL AXIS LOCATION. THE FREQUENCY OF THIS OSCILLATION IS DETERMINED AS FOLLOWS:

- GRAVITY GRADIENT TORQUES

$$T_x = 3\,\omega_o^2\,(I_z - I_y)\,\phi$$

$$T_y = 3\,\omega_o^2\,(I_x - I_z)\,\theta \cong 0. \qquad \text{AND}$$

$$T_z = 3\,\omega_o^2\,(I_y - I_x)\,\psi$$

*** THIS RESULTS FROM NUTATIONAL MOTION.

LGO POINTING ERRORS DURING ECLIPSE OPERATIONS (CONTINUED)

WHERE

$\omega_o = 8.91 \times 10^{-4}$ RAD/SEC ORBITAL RATE

ϕ, θ, ψ - ROLL, PITCH AND YAW ATTITUDE OFFSETS OF THE PRINCIPAL AXIS OF INERTIA FROM NADIR

I_x, I_y, I_z - PRINCIPAL MOMENTS OF INERTIA

$$\begin{pmatrix} I_x = I_z = 955 \text{ SLUG-FT}^2 \\ I_y = 302 \text{ SLUG-FT}^2 \end{pmatrix}^*$$

- $I_x \ddot{\phi} + 3\,\omega_o^2\,(I_z - I_y) = 0$

$$\omega = \omega_o \sqrt{\frac{3\,(I_z - I_y)}{I_x}}$$

$$= 1.27 \times 10^{-3} \text{ RAD/SEC} \qquad \text{AND}$$

$T_p = 4947$ SEC

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* PROVIDED BY WADE AKLE 7/1/82.

THE ECLIPSE PERIOD LAST $\approx$ 0.77 HOUR OR 2772 SECONDS AND IS AT LEAST ONE-HALF THE GRAVITY GRADIENT INDUCED OSCILLATION. THEREFORE, DURING ECLIPSE, GRAVITY GRADIENT DISTURBANCES WILL ROTATE THE SPACECRAFT FROM THE ATTITUDE SENSOR NULL TO THE PRINCIPAL INERTIA AXIS NULL. BASED ON INITIAL ESTIMATES, THIS COULD BE AT LEAST ONE DEGREE.

CONCLUSIONS

LGO ATTITUDE DRIFT DURING LUNAR ECLIPSE WILL BE DOMINATED BY REACTION WHEEL DRIFT, RESIDUAL SPACECRAFT RATES AND GRAVITY GRADIENT TORQUES. REACTION WHEEL SPEED CHANGES CAN PRODUCE A WORST CASE PITCH ATTITUDE DRIFT (ABOUT y-AXIS) OF 0.17 DEGREE. RESIDUAL SPACECRAFT RATES AT ATTITUDE CONTROLLER DEACTIVATION CAN PRODUCE 0.034 DEGREE ROLL (ABOUT x-AXIS) ATTITUDE CHANGE DURING ECLIPSE. THE MOST SIGNIFICANT ATTITUDE DRIFT SOURCE DURING ECLIPSE RESULTS FROM GRAVITY GRADIENT DISTURBANCES. BASED ON INERTIA ESTIMATES, THE GRAVITY GRADIENT TORQUES ARE SUFFICIENTLY LARGE TO MOVE THE SPACECRAFT FROM THE SENSOR NULL ATTITUDE TO THE PRINCIPAL INERTIA AXIS/GRAVITY GRADIENT NULL ORIENTATION; THEREFORE, IT IS ESSENTIAL TO CONTROL THE ALIGNMENT OF THE ATTITUDE SENSOR NULLS TO THE PRINCIPAL INERTIA AXIS IN ORDER TO MINIMIZE ATTITUDE DRIFT DURING LUNAR ECLIPSE OPERATIONS.

STABILITY CONSIDERATIONS OF SIMPLE SPINNERS

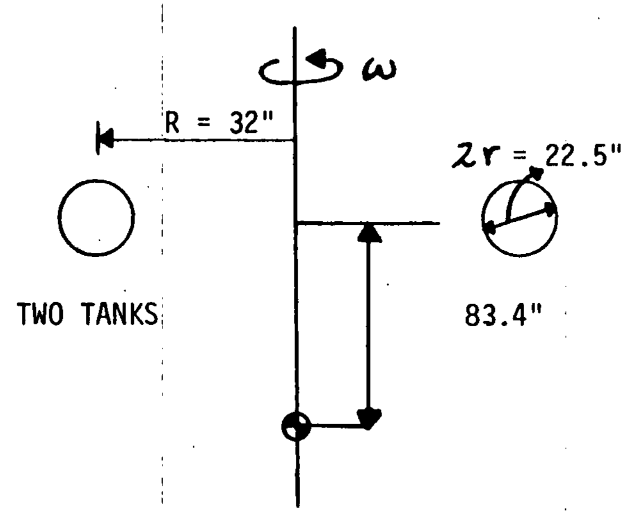
INTRODUCTION

A BODY SPINNING ABOUT AN AXIS OF MINIMUM MOMENT OF INERTIA IS UNSTABLE IN THE PRESENCE OF DAMPING. THE DEGREE OF INSTABILITY OR THE RATE OF DIVERGENCE OF THE SPIN AXIS FROM AN INITIAL POSITION IN INERTIAL SPACE IS A FUNCTION OF THE RATE AT WHICH ENERGY IS DISSIPATED IN THE BODY. THIS IS GENERALLY QUANTIFIED BY A TIME CONSTANT DEFINED AS THE TIME REQUIRED FOR AN INITIAL NUTATION ANGLE TO CHANGE BY A FACTOR OF $e \cong 2.72$. IN THIS APPENDIX, A PRELIMINARY AND HOPEFULLY CONSERVATIVE ESTIMATE OF THE DESTABILIZING TIME CONSTANT IS DERIVED. THIS IS USED TO EXAMINE THE BUS THRUSTER SIZING AND THE PROPELLANT CONSUMPTION.

TIME CONSTANT DERIVATION

THE FIRST TASK IS TO ESTIMATE THE SYSTEM TIME CONSTANT DUE TO ENERGY DISSIPATION IN THE PROPELLANT WHICH IS THE PRIMARY SOURCE OF DISSIPATION IN THE SPACECRAFT.

	LOADED AKM	FIRED AKM
I_{spin}	$= 2.64 \times 10^7$ lb in^2	6.28×10^6 lb in^2
$I_{transverse}$	$= 4.85 \times 10^7$ lb in^2	2×10^7 lb in^2
	$\sigma = \frac{I_s}{I_t} = 0.54$	$\sigma = 0.31$



CONSIDER BOTH A HALF-FULL AND COMPLETELY FULL TANK.

THREE MECHANISMS OF ENERGY DISSIPATION MUST BE CONSIDERED IN THE FLUID:

1. SLOSH
2. VISCOUS SHEAR BETWEEN TANK BOUNDARY AND FLUID
3. CONVECTION WITHIN THE FLUID

FOR CONVENTIONAL SLOSH IN A SPHERICAL CONTAINER, THE NATURAL FREQUENCY IS

$$w_n = \lambda_n \sqrt{g/r}$$

WHILE THE FORCING FUNCTION FREQUENCY IS THE BODY FIXED PRECESSION RATE

$$|(\sigma - 1)\omega|$$

FOR A HALF-FILLED TANK $\quad \frac{h}{r} \approx 1.0, \quad \lambda_n \cong 1.22$

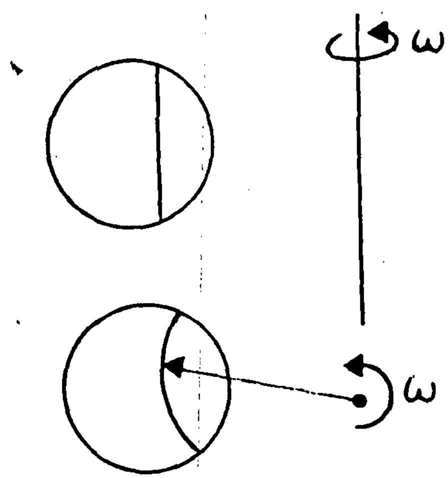
$$r = 11.2'', \quad g \cong R\omega^2 = 32\omega^2$$

$$\omega_n = \omega\,(1.22)\sqrt{\frac{32}{11.2}} = 2.06\omega$$

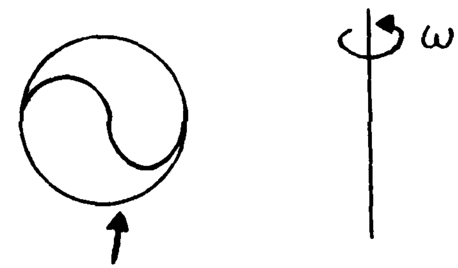
THE BODY FIXED FORCING FREQUENCY

$$|(\sigma - 1)\omega| = .46\omega \text{ (LOADED AKM) OR } 0.69\omega \text{ (FIRED AKM)}$$

IS A FACTOR OF 3 TO 5 REMOVED FROM THE NATURAL SLOSH FREQUENCY OF THE FLUID. THEREFORE, THE FLUID SHOULD NOT BE EXCITED SIGNIFICANTLY IN ITS FIRST OR CONVENTIONAL SLOSH MODE, AND THE ENERGY DISSIPATIONS DUE TO SLOSH SHOULD BE NIL. NOTE THAT THIS IS FOR A SPHERICAL CONTAINER WITHOUT ANY INTERNAL CONSTRAINTS. IN THE REAL CASE, THE TANKS HAVE A FLEXIBLE BUT NON-EXTENSIONAL HEMISPHERICAL DIAPHRAGM ATTACHED AT THE TANK EQUATOR. THIS WILL RAISE THE SLOSH FREQUENCY EVEN HIGHER, BUT COMPLICATES THE FLUID CONFIGURATION DURING SPIN AND MAKES AN ACCURATE PREDICTION OF A TIME CONSTANT VERY DIFFICULT, SINCE CONVECTION WITHIN THE FLUID WILL NOW BECOME THE PREDOMINATE MECHANISM FOR ENERGY DISSIPATION.



TANK WITHOUT INTERNAL CONSTRAINTS



NOTE: SINCE THE DIAPHRAGM HAS SOME BENDING STIFFNESS, THE FLUID SHAPE WILL DEPEND UPON THE SPIN RATE.

TANK WITH EQUATOR-ATTACHED FLEXIBLE DIAPHRAGM

EVEN WITH TEST DATA, THE TIME CONSTANT PREDICTIONS ARE QUESTIONABLE SINCE INTRODUCTION OF DIAPHRAGMS CAN MAKE IT DIFFICULT OR IMPOSSIBLE TO MATCH ALL TEST PARAMETERS TO FLIGHT VALUES. SPECIFICALLY, THREE NEW PARAMETERS ARE INTRODUCED. THEY ARE THE DIAPHRAGM SHAPE FACTOR, MASS AND BENDING STIFFNESS. THE MASS EFFECTS MAY BE NEGLIGIBLE, BUT THE SHAPE FACTOR AND STIFFNESS WILL BE ALTERED BY THE SPIN RATE FOR PARTIALLY FILLED TANKS AS MENTIONED PREVIOUSLY. IF REAL FLIGHT DIAPHRAGMS, TANKS AND FLUID ARE USED IN THE TEST, THEN THE FLIGHT SPIN RATE MUST BE USED, THUS VIOLATING THE FROUDE NUMBER. IF HIGHER SPIN RATES ARE USED TO CORRECT FOR THE FROUDE EFFECTS, THEN EITHER HIGHER DIAPHRAGM STIFFNESS MUST BE USED OR FLUID DENSITY OR TANK DIAMETER MUST BE REDUCED . FOR THE LATTER CASE, THE REYNOLDS NUMBER IS AFFECTED AND VISCOSITY MUST BE ALTERED TO CORRECT IT, ASSUMING A SUITABLE FLUID EXISTS.

FOR PURPOSES OF A BALL-PARK ANALYSIS, IT WILL BE ASSUMED THAT THE TANK WITHOUT A DIAPHRAGM IS HALF-FULL, THEN COMPLETELY FULL, AND THE TIME CONSTANT WILL BE DETERMINED BASED UPON VISCOUS SHEAR DISSIPATION. TO ACCOUNT FOR THE DIAPHRAGM AND CONVECTION EFFECTS, THE TIME CONSTANTS RESULTING FROM VISCOUS SHEAR WILL BE REDUCED BY TWO ORDERS OF MAGNITUDE.*

FOR THE ONE-HALF FULL CONDITION

$$\tau = \frac{\sigma(\sigma - 1)\, I_t}{\eta \rho d^5 \omega \xi^*} = \frac{\sigma(\sigma - 1)\, I_t \omega^2}{\mathring{E}/\theta^2}$$

$$\theta = \theta_0 e^{t/\tau}$$

BASED UPON SOME UNPUBLISHED TEST DATA FOR HALF-FULL SPHERICAL TANKS

$$\frac{\mathring{E}}{\theta^2} \frac{1}{\eta \rho d^5 \omega^3} \approx 1.8 \times 10^{-3}$$

$$\frac{\mathring{E}}{\theta^2} = 1.75\omega^3$$

$$\tau = \frac{0.54\ (.46)\ 4.85 \times 10^7}{386\ (1.75)\omega\ 3600} = \frac{4.95}{\omega} \text{ hrs.}$$

RPM	ω(RAD/SEC)	τ(HRS)	τ/100 X 3600 (SEC)
50	5.23	0.95	34
20	2.09	2.36	85
5	0.52	9.52	343

---

* THE "FULL" TANK ACTUALLY HAS SOME ULLAGE AND DIAPHRAGM EFFECTS.

FOR THE FULL TANK, SEVERAL APPROXIMATIONS EXIST, INCLUDING ONES DEVELOPED BY TRW AND UCLA

$$\tau_{TRW} = \frac{I_t\left(1 - \frac{I_t}{I_s}\right)}{\frac{4\pi}{3}\gamma^4 \rho \eta} \sqrt{\frac{2}{\mu\lambda\omega}}$$

$$= \frac{\frac{4.85 \times 10^7}{386} \quad 1 - \frac{4.85}{2.64}}{(3600)\ \frac{4\pi}{3}\ (11.2)^4\ \frac{62.4\ (.9)}{1728\ (386)}\ 2} \sqrt{\frac{2}{10.4 \times 144 \times 10^{-6}\ (0.46)\omega}}$$

$$= \frac{71.32}{\sqrt{\omega}}$$

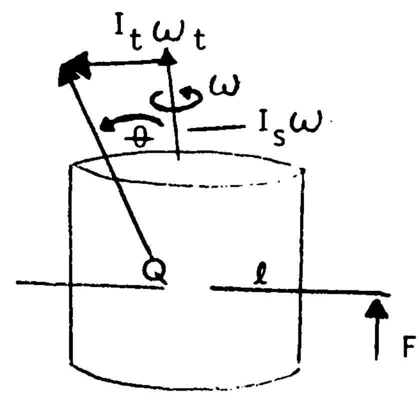
$$\tau_{UCLA} = \frac{I_t}{\frac{4\pi}{3}\ \gamma^4 \rho \eta \sqrt{2\nu\omega}\ \ 3600}$$

$$= \frac{51.5}{\sqrt{\omega}}$$

USING THE FORMER EXPRESSION

RPM	ω (RAD/SEC)	τ (HRS)	τ/100 X 3600 (SEC)
50	5.23	31.1	1120
20	2.09	49.2	1170
5	0.52	99.0	3560

DETERMINE ABILITY TO CORRECT NUTATION BUILDUP USING THE CURRENT PROPULSION SYSTEM CHARACTERISTICS



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$$F\ell dt = I_t \omega_t = \theta I_s \omega$$

$$E I_{sp} W \ell = F\ell dt = \theta I_s \omega$$

$$W = \frac{\theta I_s \omega}{I_{sp} \ell E}$$

WHERE W = WEIGHT OF PROPELLANT

I_{sp} = SPECIFIC IMPULSE OF PROPELLANT

E = EFFICIENCY FACTOR

$$= \frac{\sin \alpha/2}{\alpha/2}$$

α = ANGLE OF THRUSTER FIRING

ASSUME A TWO-POUND THRUSTER AT $\ell \cong 80''$ FIRING OVER A 90° ARC. THE AMOUNT OF NUTATION THAT CAN BE REMOVED BY ONE FIRING IS

$$\Delta\theta = \frac{\text{IMPULSE BIT}}{I_s \omega}$$

$$= \frac{F\ell \frac{2\pi}{4\omega} E \quad 57.3}{I_s \omega}$$

$$= \frac{(80)\ \pi\ (.9)\ (57.3)}{68400\ \omega^2} = \frac{0.189'}{\omega^2} \quad \text{(LOADED AKM)}$$

RPM	ω (RAD/SEC	$\Delta\theta$ (DEG)
50	5.23	0.00693
20	2.09	0.0434
5	0.52	0.701

FROM AN OPERATIONAL STANDPOINT, THE THRUSTING MUST BE TIMED TO REMOVE THE GROWING TRANSVERSE RATE WHICH TRAVELS WITH RESPECT TO THE BODY FIXED REFERENCE AT A RATE OF

$$\dot{\phi} = (\sigma - 1)\ \omega = \lambda\omega = -0.46\,\omega$$

ASSUMING 2 TWO-POUND OPPOSING THRUSTER BANKS 180° APART, EACH BANK SHOULD BE FIRED TO HAVE ITS CENTER POINT OF THE 90° ARC AT THE NUTATION HALF-PERIOD OF $\frac{\pi}{\dot{\phi}}$ FOR OPTIMIZING TRANSVERSE RATE REMOVAL.

RPM	ω(RAD/SEC)	$\dot{\phi}$ (RAD/SEC)	$\pi/\dot{\phi}$ (SEC)
50	5.23	2.40	1.304
20	2.09	0.961	3.269
5	0.52	0.239	13.145

NOW, THE PRECESSION BUILDUP, θ, IS

$$\theta = \theta_0 \, e^{t/\tau}$$

SETTING $\theta = (\theta_0 + \Delta\theta)$, t = HALF-PERIOD OF ONE NUTATION CYCLE, τ = TIME CONSTANT, THE MAXIMUM VALUE OF θ_0 CAN BE FOUND WHICH WILL RESULT IN THE THRUSTERS HAVING INSUFFICIENT IMPULSE CAPABILITY TO REMOVE THE NUTATION BUILDUP DUE TO ENERGY DISSIPATION, i.e., THE SYSTEM GOES UNSTABLE. FOR SMALL ANGLES -

$$\theta_{0\,MAX} = \frac{\Delta\theta}{e^{t/\tau} - 1}$$

RPM	ω	$\Delta\theta$	t SEC	τ HALF-FULL	T-FULL	θ MAX. (DEG) HALF-FULL	θ MAX. (DEG) FULL
50	5.23	0.00693	1.309	34	1120	0.1766	5.926
20	2.09	0.0434	3.269	85	1770	1.107	23.47
5	0.52	0.701	13.145	343	3560	17.9	

ASSUME THAT IT IS REQUIRED TO MAINTAIN THE NUTATION AT ONE-FOURTH OF THE MAXIMUM VALUES TO ASSURE A REASONABLE MARGIN OF SAFETY, THEN THE CORRESPONDING TRANSVERSE BODY RATES BECOME

RPM	ω_t (DEG/SEC) HALF-FULL	ω_t (DEG/SEC) FULL
50	0.12	4.20
20	0.32	6.70
5	1.27	

IT WOULD APPEAR THAT THE FULL TANK CONDITION COULD BE CONTROLLED OVER THE ABOVE RANGE OF RPM RATES WITH THE EXISTING THRUSTERS; HOWEVER, THE HALF-FULL TANK CONDITION IS MARGINAL. EVENTS SUCH AS INJECTION, SEPARATION, MID-COURSE CORRECTIONS OR ATTITUDE MANEUVERS TYPICALLY PRODUCE TRANSVERSE RATES IN THE RANGE OF THE HALF-FULL CONDITION ALLOWABLES. NOTE THAT THE ABOVE OBSERVATIONS APPLY FOR PRE-FIRING OF THE AKM.

POST FIRING CONDITIONS, ACCELEROMETER POSITIONING AND SENSITIVITY REQUIREMENTS MUST BE EXAMINED IN MORE DETAIL BEFORE AN ACTIVE NUTATION CONTROL (ANC) SYSTEM DESIGN IS SELECTED. THE IMPORTANT FACTOR SUGGESTED BY THIS BRIEF STUDY IS THAT ANC IS INDEED FEASIBLE AND THE AMOUNT OF PROPELLANT UTILIZED TO MAINTAIN CONTROL IS MODEST. FOR EXAMPLE, ONE HOUR OF NUTATION CONTROL BETWEEN THE LIMITS OF 0.1 AND 0.5 DEGREE FOR THE FULL TANK CONDITION CAN BE ACCOMPLISHED FOR ONLY 0.5 POUNDS OF HYDRAZINE AT THE 50 RPM SPIN RATE. SIMILARLY, THE HALF-FULL TANK CONFIGURATION COULD BE STABILIZED BETWEEN THE SAME LIMITS FOR 2.4 POUNDS AT THE 20 RPM SPIN RATE.

INCREASED FLTSATCOM PROPELLANT LOADING

INCREASED HYDRAZINE LOADING

THE FLTSATCOM TANKS (IDENTICAL TO HEAO) ARE CAPABLE OF BEING LOADED TO 134 POUNDS OF HYDRAZINE PROPELLANT BY MEANS OF 3 MOUNTINGS LUGS WITH A FACTOR OF SAFETY OF 1.5. STRUCTURAL ANALYSES SHOW THAT THEY CAN BE LOADED WITH 184 POUNDS OF HYDRAZINE BY MEANS OF 4 LUGS WITH A FACTOR OF SAFETY OF 1.03, ASSUMING THAT LOAD DISTRIBUTIONS AND ACCELERATIONS DO NOT CHANGE.

ON THE FLTSATCOM QUALIFICATION TEST, THE 0.1 LBF THRUSTERS WERE QUALIFIED TO A MINIMUM INLET PRESSURE OF 170 PSIA. TEST DATA FOR THESE THRUSTERS ARE AVAILABLE AT 75 PSIA. EXAMINATION OF FIGURE 1 SHOWS THAT 160 POUNDS OF HYDRAZINE LOADED AT 400 PSIA AND 100^{o}F IN EACH TANK BOL (320 POUNDS TOTAL) WILL YIELD 75 PSIA INLET PRESSURE AT 40^{o}F EOL. SIMILAR CALCULATIONS FOR 184 POUNDS OF HYDRAZINE LOADED YIELD SLIGHTLY OVER 20 PSIA INLET PRESSURE EOL. ONE (1) LFB THRUSTERS HAVE BEEN TESTED TO MINIMUM INLET PRESSURES OF 15 PSIA.

THE FLTSATCOM SPECIFICATION REQUIRES $\geq$0.0011 LBF-SEC IMPULSE BIT PER 20 MSEC PULSE. DATA FROM QUALIFICATION TEST - AT BOL, AFTER 100,000 PULSES, AND AFTER 200,000 PULSES - ARE PLOTTED VERSUS INLET PRESSURE FOR THE 0.1 LBF THRUSTERS IN FIGURE 2. ALSO SHOWN ON THIS FIGURE ARE DATA (LIMITED NUMBER OF PULSES) AT 75 PSIA INLET FROM ANOTHER PROGRAM. EXTRAPOLATION OF THE QUALIFICATION TEST DATA INDICATES THAT THE MINIMUM IMPULSE BIT REQUIREMENT WILL BE MET AT 90 PSI INLET AFTER 200,000 PULSES. THE LOW PRESSURE DATA SHOW THAT MINIMUM IMPULSE BIT REQUIREMENTS CAN BE MET AT EVEN LOWER PRESSURE.

FROM THE ABOVE DATA, IT MAY BE SEEN THAT OVER 300 POUNDS OF HYDRAZINE CAN BE LOADED, RESULTING IN 90 PSI EOL INLET PRESSURE TO THE THRUSTERS. AT THIS PRESSURE, THE THRUSTERS SHOULD BE CAPABLE OF MORE THAN 200,000 PULSES OVER THE BLOWDOWN RANGE, WHILE STILL EXCEEDING MINIMUM IMPULSE BIT REQUIREMENTS. IF NECESSARY, THE 0.1 LBF THRUSTERS CAN BE QUALIFIED BY TEST FOR LOW INLET PRESSURES AT A COST IN THE VICINITY OF $200K.

STAR MOTOR PROPELLANT SEPARATION INVESTIGATION

THE INTELSAT V APOGEE KICK MOTOR (THIOKOL/ELKTON STAR 37XF) SHOWED PROPELLANT SEPARATION ON 3 MOTORS, ALL BEING PROCESSED IN THE SAME LOT. SUBSEQUENTLY, A STAR 48 MOTOR ON ANOTHER PROGRAM WAS FOUND TO HAVE THE SAME CONDITION. FACC, INTELSAT, COMSAT, MCDONNELL-DOUGLAS, AND AEROSPACE CORP. ARE ALL PARTICIPATING WITH THIOKOL ON THEIR ONGOING INVESTIGATION OF THIS PROBLEM. IT APPEARS THAT THE PROBLEM IS RELATED TO THE PROPELLANT INGREDIENTS USED IN FORMULATING A PARTICULAR LOT OF MATERIAL WHERE THE POLYMER/CURING AGENT RATIO WAS OUTSIDE ITS PREVIOUS EXPERIENCE RANGE.

FSC 7 AND 8 MGO MOTOR - THE STAR 37FM

THE FOLLOWING DESCRIBES THE MOTOR DESIGN AND GIVES BACK-UP CHARTS. NOTE:

1) QUALIFIED AND DELIVERED BY MID 1985 FOR FSC 7 AND 8

2) COULD HAVE LARGER EXP. RATIO: UP $I_{sp} \sim 2$ SEC

3) 37FM - 7 AND 8 ONLY
FSC 6 - STAR 37F

FIGURE 1. STAR 37FM AKM FOR FLTSATCOM

FLIGHTS 7 AND 8

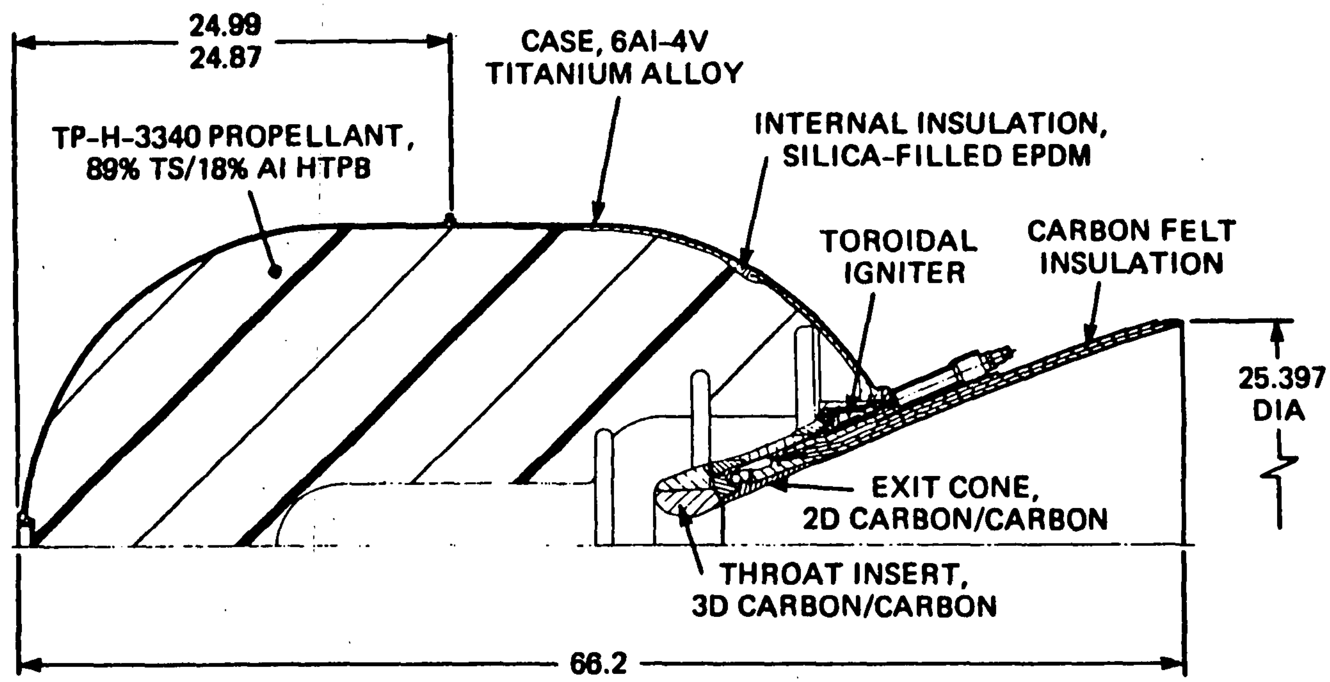
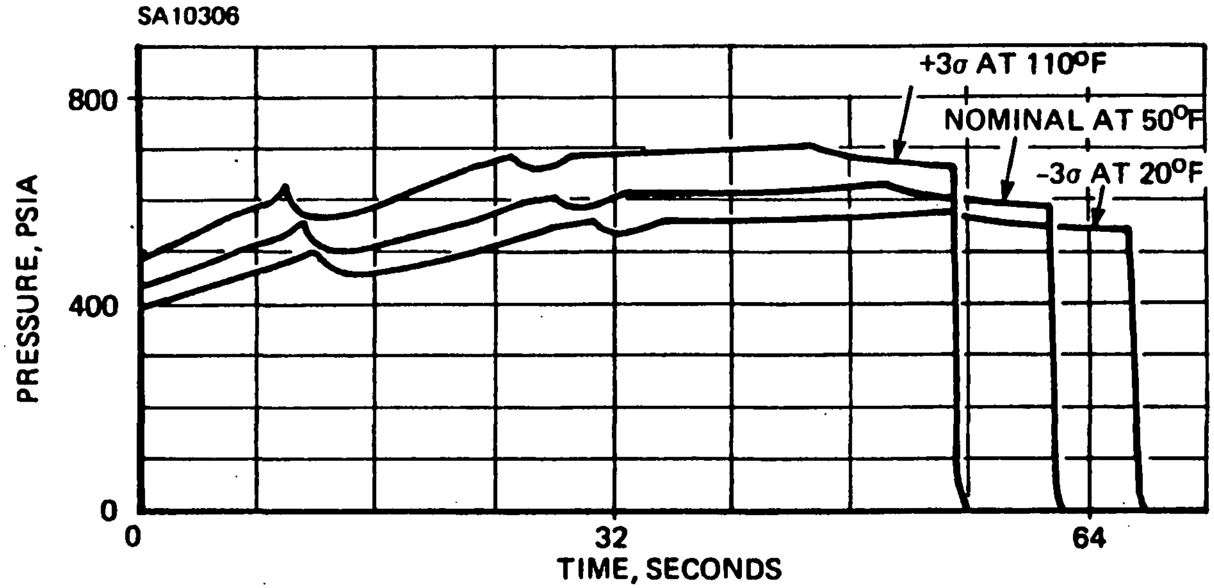


FIGURE 3. BALLISTIC PERFORMANCE - F7/F8 CONFIGURATION



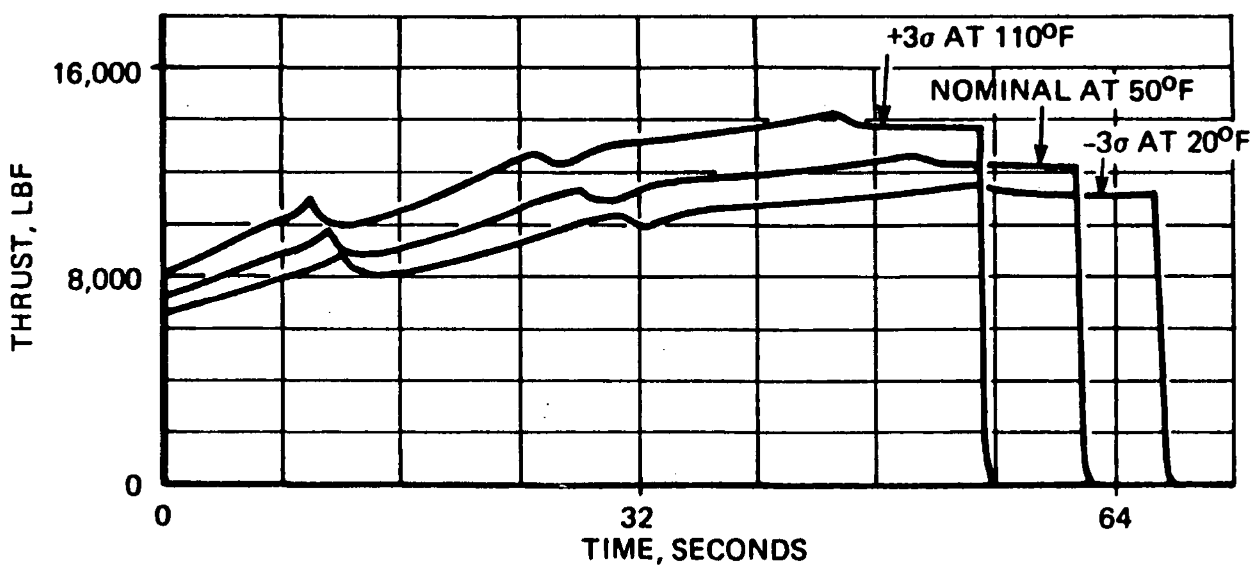


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TABLE I

DESIGN SUMMARY

Design	FSC F7/F8
Propellant, lb	2320.2
A_t, initial, in.^{2} A_t, average, in.^{2} A_{exit}, in.^{2}	9.00 10.143 493.1
Expansion Ratio, avg. Weight Flow Rate	48.6 38.4
Propellant weight, lb	2320.2
P_c, max, psia P_c, avg, psia	628 568
t_b, sec t_a, sec	61.3 62.6
F_{max}, lbf F_{avg}, lbf	12,840 10,930
I_{tot}, lb-sec I_{sp}, prop, lb-sec/lbm I_{sp}, eff, lb-sec/lbm	675,880 291.30 289.85
Inert weight expended, lb	11.6