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REPORT NO. B743

GEMINI LARGE EARTH ORBIT

VOLUME I

TECHNICAL

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MAC CONTROL NO. C-100858
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REPORT B743
19 JUNE 1965

VOLUME I TECHNICAL

GEMINI LARGE EARTH ORBIT_(U)

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INTRODUCTION

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SUMMARY

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1. SUMMARY

Results of the studies performed indicate that Gemini with modest revisions can be utilized to accomplish a large earth orbit or circumlunar mission.

Two versions of Gemini are presented for consideration.

Configuration I - A rendezvous Gemini spacecraft put into orbit by the GLV and rendezvous with a Titan IIIC with a modified second transtage.

Configuration II - A direct ascent Gemini spacecraft launched by a modified Titan IIIC.

Both of the spacecraft configurations presented use Gemini qualified hardware and new components only where new capabilities are essential for a large orbit mission.

1.1 Configuration Selection

1.1.1 Configuration I - The re-entry module is the same basic configuration as GT-7 off loaded for a seven-day mission. A range-range rate radar is used during rendezvous to insure minimum propellant consumption. The basic adapter structure is the same as present Gemini, additional hardpoints are required for mounting equipment.

1.1.1.1 Guidance and Control - The Guidance and Control system is modified by the addition of two sun sensors, two star sensors, inertial reference package, and a manual sextant. The manual sextant is used in event of system failure. Redundant wiring for the OAMS thrusters is added. It is assumed that the auxiliary tape memory is spacecraft equipment. The horizon sensor is deleted.

1.1.1.2 Environmental Control System - The ECS primary oxygen system uses four 45 lb. bottles in conjunction with the fuel cell reactant supply system. The coolant system has an additional heat exchanger so that each fuel cell system is completely independent from the other and the main coolant loops. With the exception of the cabin fan and cabin heat exchanger that are removed, the remainder of the environmental control system is the same as in the present Gemini.

1.1.1.3 Electrical Power - Electrical power is provided by three fuel cell sections. The reduced load on each stack as the result of using three sections greatly increases the reliability of a fuel cell system. The re-entry and post-landing electrical system is similar to present Gemini.

1.1.1.4 Communication and Data Acquisition Systems - The communication system is the same as present Gemini with the deletion of systems that are no longer required as the result of adding redundant communications for the larger orbit. A Unified S-band System is added for communicating over ranges up to approximately 225,000 n.m. and is compatible with the Deep Space Instrumentation Facilities. The operation of the Unified S-band System has a duty cycle of 15 minutes on out of every two hours, plus five hours for ΔV maneuvers.

Data acquisition is the same as Gemini.

1.1.1.5 Structural Design Criteria - The structural design criteria is the same for Gemini (LEO) as it is for the present Gemini.

1.1.1.6 Heat Protection - The afterbody shingles and heat shield are revised to accommodate higher re-entry heating. A more efficient ablator is used for the shingles than is used on the present Gemini heat shield.

1.1.1.7 Thermal Control - The spacecraft is oriented with its Z axis at 45 degrees to 70 degrees with respect to the sun and rolled .6 degrees/sec. All heaters for fuel, oxidizer, water, and urine lines, and thrusters are removed.

1.1.1.8 Propulsion - The Re-entry Control System (RCS) for Gemini LEO is a blow-down system. The motor-operated shutoff valves are replaced by pyrotechnic valves. The blow-down system is simpler and lighter than the present system.

The OAMS has a redundant set of two aft firing and eight attitude thrusters. Three propellant supply groups provide the desired reliability to insure a ΔV

1.1.1.8 Continued)

capability for a safe return should a failure occur at any time during the mission.

1.1.1.9 Reliability - The overall spacecraft reliability for a safe mission is 0.98308.

1.1.1.10 Abort - The ejections seats are the same as present Gemini. The retro rockets have been removed.

1.1.1.11 Experiments - A 100-pound allowance has been made for Gemini LEO experiment. The configuration as shown uses 58 lbs. for photographic and television experiments. The balance of 42 lbs. is not assigned.

1.1.1.12 AGE - AGE requirements and test plans follow as much as possible the existing Gemini AGE plan. Using the GLV keeps the AGE changes to a minimum.

1.1.1.13 Weight - Launch weight for a rendezvous Gemini is 8,374 pounds. A weight derivation is shown in Table 1.1-1.

1.1.2 Configuration II - Configuration II is capable of a direct ascent into a circumlunar trajectory. The re-entry module is the same basic configuration as GT-7 off loaded for a seven-day mission.

1.1.2.1 Guidance and Control - Same as Configuration I.

1.1.2.2 Environmental Control System - Same as Configuration I.

1.1.2.3 Electrical Power - Same as Configuration I.

1.1.2.4 Communication and Data Acquisition System - The communication and data acquisition systems are the same as Configuration I except the rendezvous radar is removed.

1.1.2.5 Structural Design Criteria - Same as Configuration I.

1.1.2.6 Heat Protection - Same as Configuration I.

1.1.2.7 Thermal Control - Same as Configuration I.

TABLE 1.1-1
ESTIMATED WEIGHT STATEMENT

CONFIGURATION I

	WEIGHT (POUNDS)
LAUNCH GROSS WEIGHT	8374
ORBIT INSERTION WEIGHT	8303
PRE-RENDEZVOUS WEIGHT	8252
DOCKED WEIGHT	7975
CIRCUMLUNAR INJECTION WEIGHT	7969
PRE-RE-ENTRY WEIGHT	7246
RE-ENTRY WEIGHT	4759
PILOT CHUTE DEPLOYMENT WEIGHT	4609
PARACHUTE DEPLOYMENT WEIGHT	4374
TOUCHDOWN WEIGHT - NO PARACHUTE	4263

TABLE OF EXPENDABLES

	LOADED WEIGHT (POUNDS)
RCS PROPELLANT	75
FUEL CELL REACTANTS	139
PRIMARY OXYGEN	42
OAMS PROPELLANT	1141
DRINKING WATER - PRIMARY	30

1.1.2.8 Propulsion - The RCS is the same as Configuration I.

The OAMS system uses the same components as the present Gemini. For this version, the forward firing and lateral firing thrusters are removed (no rendezvous requirement). A redundant set of two aft firing and eight attitude thrusters are added. As in Configuration I, three propellant supply groups are provided to give the desired reliability to insure a ΔV capability for midcourse corrections should a failure occur.

1.1.2.9 Reliability - The over-all spacecraft reliability for a safe mission is 0.98363.

1.1.2.10 Abort - Ejection seats are the same as Configuration I. The Transtage is used for escape at altitudes between 100,000 and 250,000 ft. (not available in Configuration I).

1.1.2.11 Experiments - Same as Configuration I.

1.1.2.12 AGE - AGE requirements and test plans follow as much as possible the existing Gemini AGE test plan. A direct launch mission requires modification to the AGE as well as moving it to a new pad.

1.1.2.13 Weight - Launch weight for Configuration II is 7,764 pounds. A weight derivation is shown in Table 1.1-2.

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TABLE 1.1-2
ESTIMATED WEIGHT STATEMENT

CONFIGURATION II

	WEIGHT (POUNDS)
LAUNCH GROSS WEIGHT	7764
CIRCUMLUNAR INJECTION WEIGHT	7719
PRE-RE-ENTRY WEIGHT	7043
RE-ENTRY WEIGHT	4713
PILOT CHUTE DEPLOYMENT WEIGHT	4563
PARACHUTE DEPLOYMENT WEIGHT	4363
TOUCHDOWN WEIGHT - NO PARACHUTE	4252

TABLE OF EXPENDABLES

	LOADED WEIGHT (POUNDS)
RCS PROPELLANT	75
FUEL CELL REACTANTS	139
PRIMARY OXYGEN	42
OAMS PROPELLANT	728
DRINKING WATER - PRIMARY	30

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1.2 Design Changes for Weight Consideration - For Configurations I and II, it is necessary to make changes to the present Gemini design and system concepts to make the spacecraft weight compatible with the launch vehicle capability. A summary of these weights is given in Table 1.2-1. Those changes applicable to both configurations as weight saving items are:

- A. The paraglider provisions are eliminated in the rendezvous and recovery section, and the can is shortened twenty inches; no change to the RCS section.
- B. The retrograde and adapter equipment sections separation provision is removed. Separation provisions are not needed.
- C. Reduce the density of insulation used on the RCS and R & R sections since the ablative layer provides bond line temperatures which are lower than Gemini re-entry shingle temperatures.
- D. One of three squib batteries is removed from the re-entry module.
- E. Convert RCS to "Blow Down" system, remove heaters and motor shutoff valves, add pyrotechnic valves which improve system reliability.
- F. One of the two UHF voice transmitter receivers is removed. Backup is provided by the unified S band system.
- G. The parachute riser ring guard and the drogue chute are deleted.
- H. The rendezvous radar is modified to include only the range and range-rate components.
- I. The fuel cell water is considered potable. The drinking water supply is reduced accordingly.
- J. Reduce heat protection by changing L/D from .25 to .15 and range from 5,300 n.m. to 3,838 n.m. (Nominal footprint 900 n.m., maximum dispersed range 2,600 n.m. - heat protection margin achieved by designing for 3,838 n.m.)
- K. S-6 ablative material is used in lieu of S-3 for afterbody heat protection.

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TABLE 1.2-1
SUMMARY OF WEIGHT SAVING ITEMS THAT
ARE INCLUDED IN CONFIGURATIONS I AND II

ITEM	WEIGHT EFFECT (POUNDS)
S-6 ABLATIVE MATERIAL IN LIEU OF S-3 FOR AFTERBODY HEAT PROTECTION	-57
RENDEZVOUS AND RECOVERY SECTION REDESIGN	-50
REMOVAL OF THE RETROGRADE SYSTEM AND ADAPTER EQUIPMENT SECTION SEPARATION PROVISIONS	-411
REMOVAL OF (2) 85-POUND FWD FIRING AND (4) 100-POUND RADIAL FIRING THRUSTERS	-100*
REDUCE THE DENSITY OF INSULATION USED ON THE RCS AND R & R SECTIONS DUE TO BOND LINE TEMPERATURES WHICH ARE LOWER THAN GEMINI RE-ENTRY SHINGLE TEMPERATURES	-15
REMOVAL OF ONE SQUIB BATTERY FROM THE RE-ENTRY MODULE	-9
CONVERT TO RCS "BLOW DOWN" SYSTEM, REMOVE HEATERS AND MOTOR SHUTOFF VALVES, ADD SQUIB VALVES	-20
USE OF A 7-DAY INSULATED L ₂ OH CARTRIDGE IN LIEU OF 14-DAY L ₂ OH CARTRIDGE	-22
REMOVAL OF UNNECESSARY RENDEZVOUS AND DOCKING PROVISIONS INCLUDING REMOVAL OF THE NOSE FAIRING AND EJECTOR AND REPLACEMENT WITH A NON-JETTISONED NOSE COVER	-31*
REDUCTION IN GEMINI COMMUNICATIONS COMPONENTS (REF. PARA. 3.4)	-44
THE HORIZON SENSORS AND INSTALLATION ARE DELETED	-41
THE DROGUE CHUTE AND PARACHUTE RISER RING GUARD ARE DELETED	-28
THE RENDEZVOUS RADAR IS MODIFIED TO INCLUDE ONLY THE RANGE AND RANGE-RATE COMPONENTS	-30**
THE FUEL CELL WATER IS CONSIDERED POTABLE. THE DRINKING WATER SUPPLY IS REDUCED ACCORDINGLY	-86
REDUCE HEAT PROTECTION BY CHANGING L/D FROM .25 TO .15 AND RANGE FROM 5,300 N.M. TO 3,838 N.M.	-236
TOTAL CONFIGURATION I	-1,049
TOTAL CONFIGURATION II	-1,150

*CONFIGURATION II ONLY

**CONFIGURATION I ONLY

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1.3 Flights

1.3.1 General - Two flights are required to qualify the spacecraft and a third performs the program objective. Refurbished re-entry modules are utilized in flights 1 and 2. Table 1.3-1 lists launch vehicle/spacecraft combinations by configuration and flight number. These flights are defined in the following paragraphs.

1.3.2 Flight 1 - Heat Protection Qualification - The purpose of this unmanned flight is to qualify the spacecraft heat protection using a modified adapter to mate the re-entry module to the Titan IIIC launch vehicle. The Titan IIIC is required to accelerate the spacecraft to escape velocity on a particular shaped trajectory. The flight sequence is shown in Figure 1.3-1. The same launch vehicle and AGE as provided for MOL HSQ could be used.

1.3.3 Flight 2 - Spacecraft Qualification - The purpose of this flight is to qualify the spacecraft systems. The spacecraft is launched by GLV, or Titan IIIC if direct is selected, into an earth orbit and all systems checked by instrumentation and utilization. It could accomplish medium earth orbit (one day) rendezvous with other satellites or make a major plane change.

1.3.4 Flight 3 - Circumlunar Orbit - The program objective of employing the Gemini spacecraft in a manned circumlunar flight is accomplished by Flight No. 3. The Gemini spacecraft for this mission are defined in Paragraphs 2.1 and 2.2

1.4 Study Ground Rules

1.4.1 Mission duration is seven days.

1.4.2 For a rendezvous configuration, the earth parking orbits are limited to four orbits.

1.4.3 Equipment and arrangements are the same as present Gemini except special equipment needed for this mission.

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TABLE 1.3-1
LAUNCH VEHICLE REQUIREMENTS BY CONFIGURATION AND FLIGHT

CONFIGURATION 1 RENDEZVOUS	CONFIGURATION 2 DIRECT LAUNCH-TO-ORBIT
FLIGHT 1	FLIGHT 1
SPACECRAFT - TITAN IIIC	SPACECRAFT - TITAN IIIC
FLIGHT 2	FLIGHT 2
SPACECRAFT - GLV	SPACECRAFT - TITAN IIIC
RENDEZVOUS - TITAN IIIC VEHICLE PLUS TRANSTAGE	
FLIGHT 3	FLIGHT 3
SPACECRAFT - GLV	SPACECRAFT - TITAN IIIC
RENDEZVOUS - TITAN IIIC VEHICLE PLUS TRANSTAGE	

FLIGHT I GEMINI HEAT PROTECTION QUALIFICATION FLIGHT SEQUENCE DIAGRAM

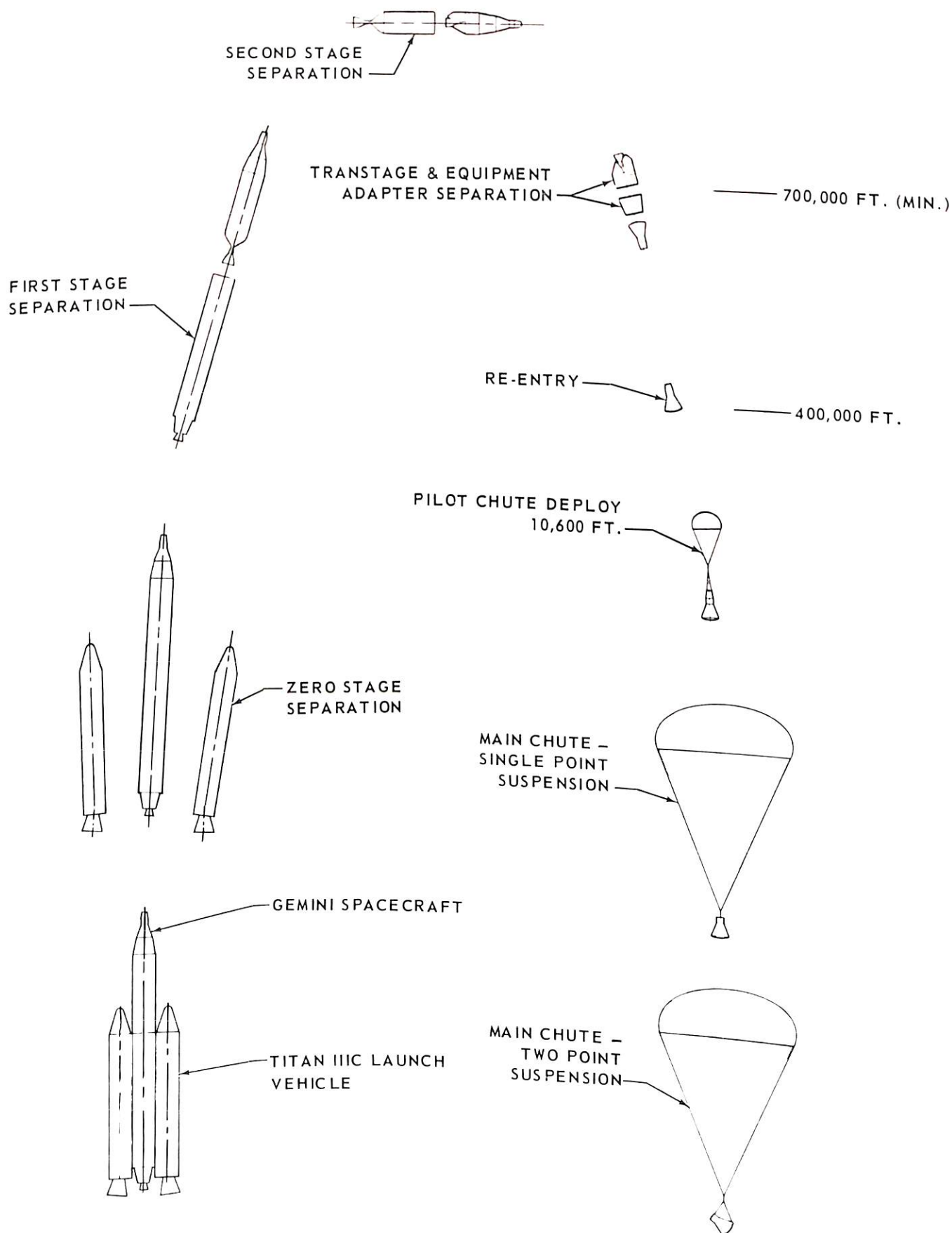


FIGURE 1.3-1

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1.4.4 Redundant systems are added as required due to inability to re-enter from a large orbit at will.

1.4.5 Systems not essential for the mission are removed to reduce weight.

1.4.6 Three spacecraft are required for the program.

1.4.6.1 Unmanned heat protection qualification spacecraft.

1.4.6.2 Manned spacecraft with all systems operational. Mission is an earth orbit.

1.4.6.3 Manned circumlunar spacecraft.

1.4.7 All mission parameters and design requirements are the same as Gemini or Apollo.

1.4.8 The first two spacecraft are refurbished previously flown Gemini. The third spacecraft is new or unflown from the present series.

2. SPACECRAFT CONFIGURATIONS

2.1 Configuration I - This spacecraft is a rendezvous version of Gemini launched by a GLV. The post-docking assembly is a modified Transtage mated to the Gemini vehicle utilizing the Target Docking Adapter (TDA) developed during the Gemini program. The Transtage is inserted into orbit by a Titan IIIC.

The modified Transtage provides the impulse necessary to insert Gemini into a large earth orbit. The attitude control during insertion is provided by the Transtage Gimballed Engine while the guidance is supplied by the Gemini Inertial Guidance System (IGS). At the completion of the insertion, the Transtage is jettisoned and the Gemini vehicle, plus its own adapter, is the circumlunar vehicle. Much of the equipment necessary has been developed during the present Gemini program, and, in this configuration, components already developed are rearranged to satisfy the new mission requirements. No development in the areas of pyrotechnics, environmental control, propulsion, and structures is required. The radar system utilized for rendezvous provides only range and range rate, and is a modification of the present system to take advantage of a significant weight saving. The general arrangement of this configuration is shown in Figure 2.1-1. The interior arrangement of the Gemini spacecraft, depicting the propulsion, environmental control, electrical, communication, guidance and attitude control major components, is shown in Figure 2.1-2.

The heat protection consists of a heat shield utilizing the present material with the maximum thickness increased by approximately 0.3 inch, columbium shingles on the leeward side of the conical section and ablative shingles on the remainder of the afterbody. Some of the possible methods of accomplishing the detail design are shown in Figure 2.1-3.

GENERAL ARRANGEMENT - LAUNCH VEHICLES FOR
RE-ENTRY MODULE AND DOCKING ADAPTER
CONFIGURATION I

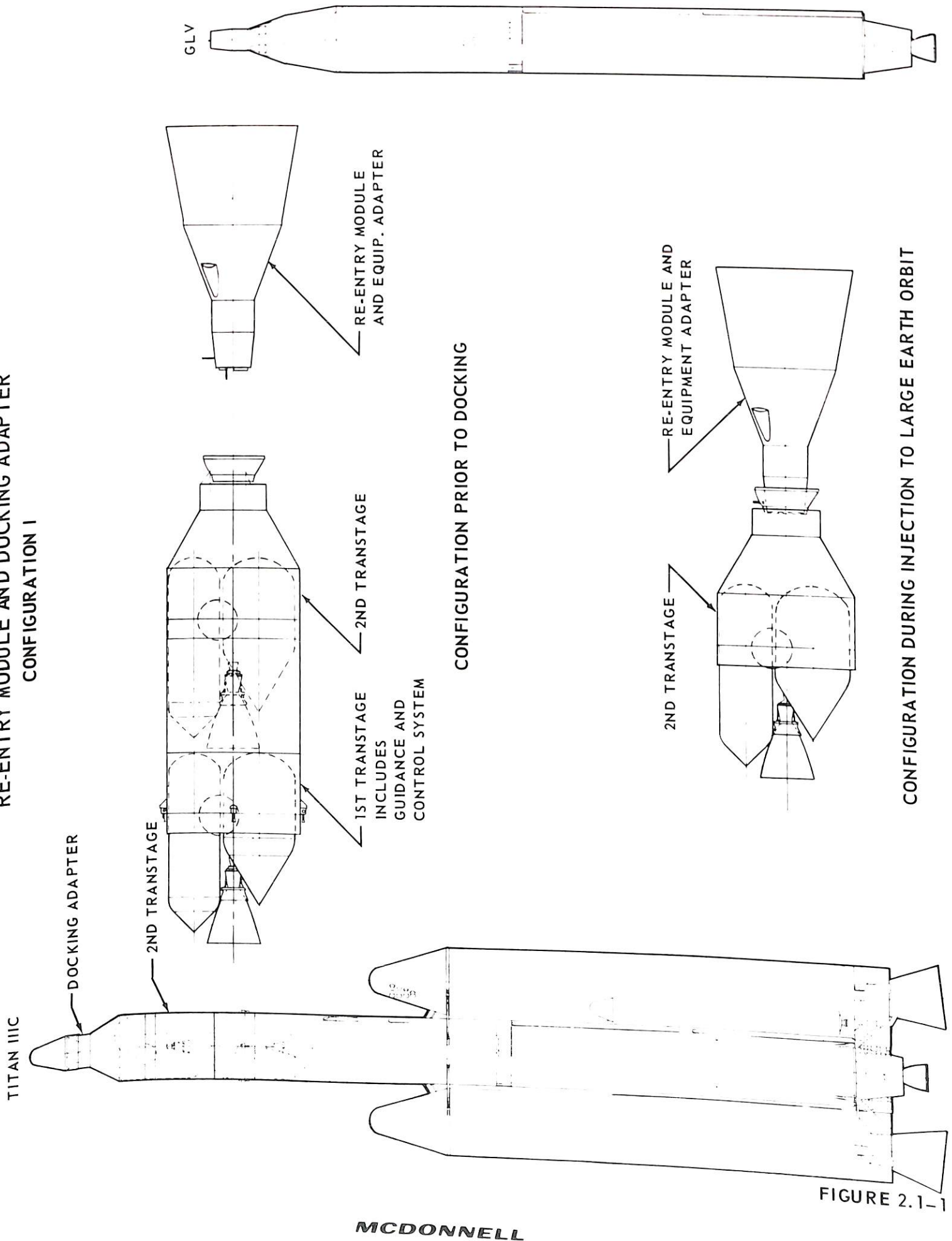


FIGURE 2.1-1

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RENDEZVOUS CONFIGURATION NUMBER 1

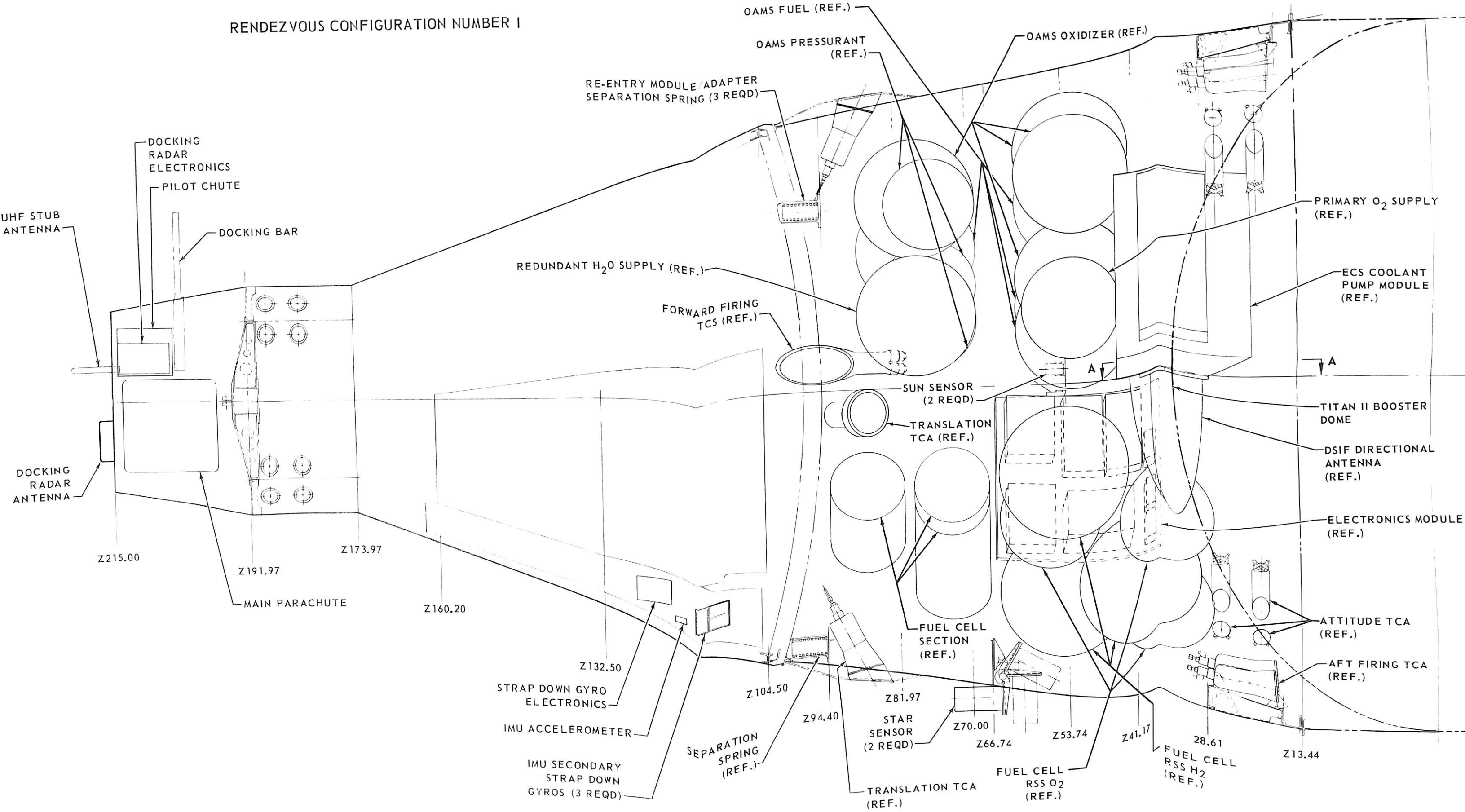


FIGURE 2.1-2

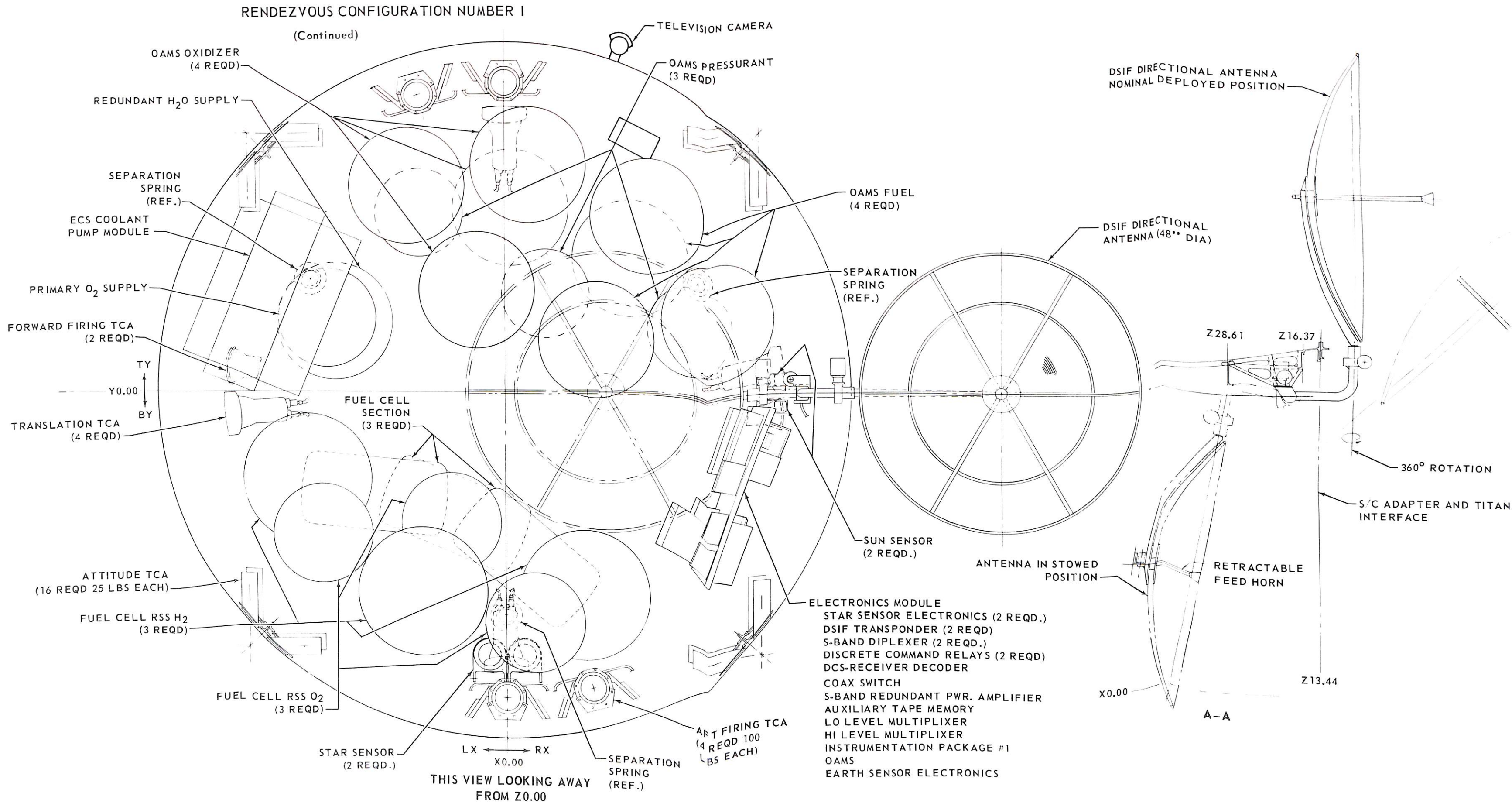


FIGURE 2.1-2

EXTERNAL HEAT PROTECTION - FOREBODY AND AFTERBODY

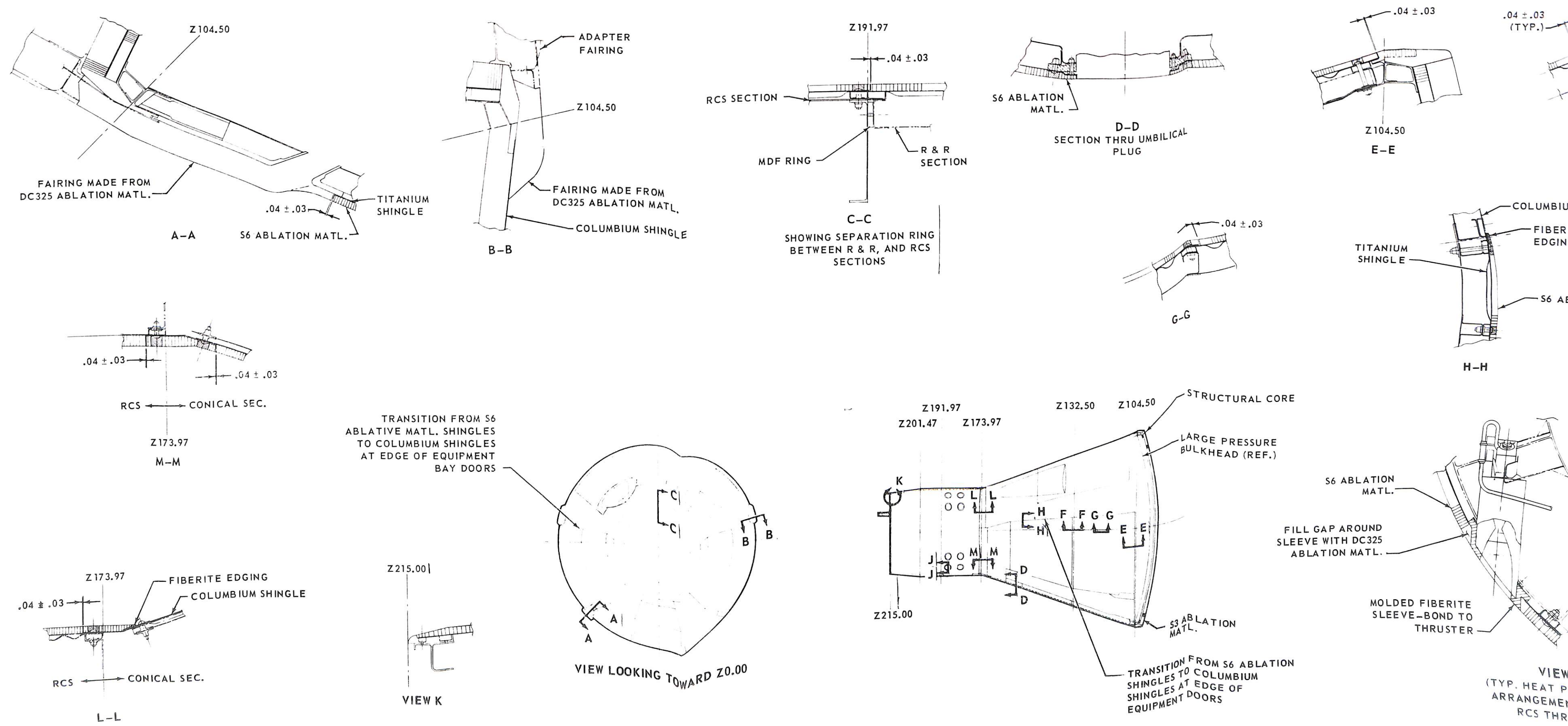


FIGURE 2.1-3

2.1 (Continued)

The Transtage for the circumlunar injection utilizes developed hardware and at present is being qualified for flight. The modification to the Transtage to mount the TDA requires only a structural adapter.

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2.2 Configuration II - This spacecraft is essentially the same as Configuration I with the exception that the rendezvous capability is removed. The docking provisions, range and range rate radar, forward firing and radial firing OAMS thrusters with the necessary propellant are removed. Gemini, mounted on the modified Titan IIIC, is injected into a direct circumlunar orbit. The advantage of requiring only a single booster for circumlunar flight is, in part, offset by the requirement for an AGE installation on a new launch pad.

The general arrangement is shown in Figure 2.2-1.

The detailed interior arrangement is shown in Figure 2.2-2.

GENERAL ARRANGEMENT
CONFIGURATION II

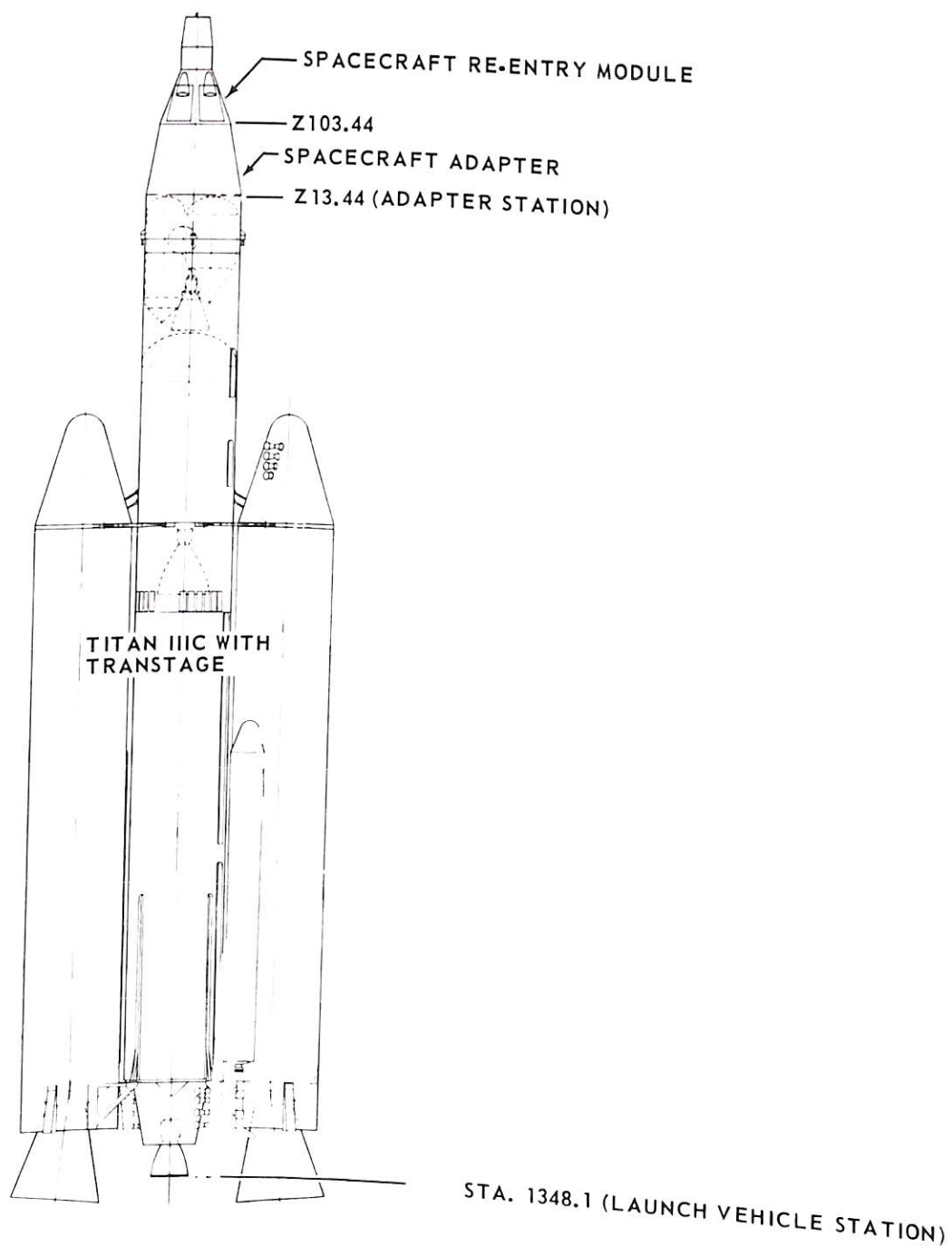


FIGURE 2.2-1

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DIRECT ASCENT CONFIGURATION II

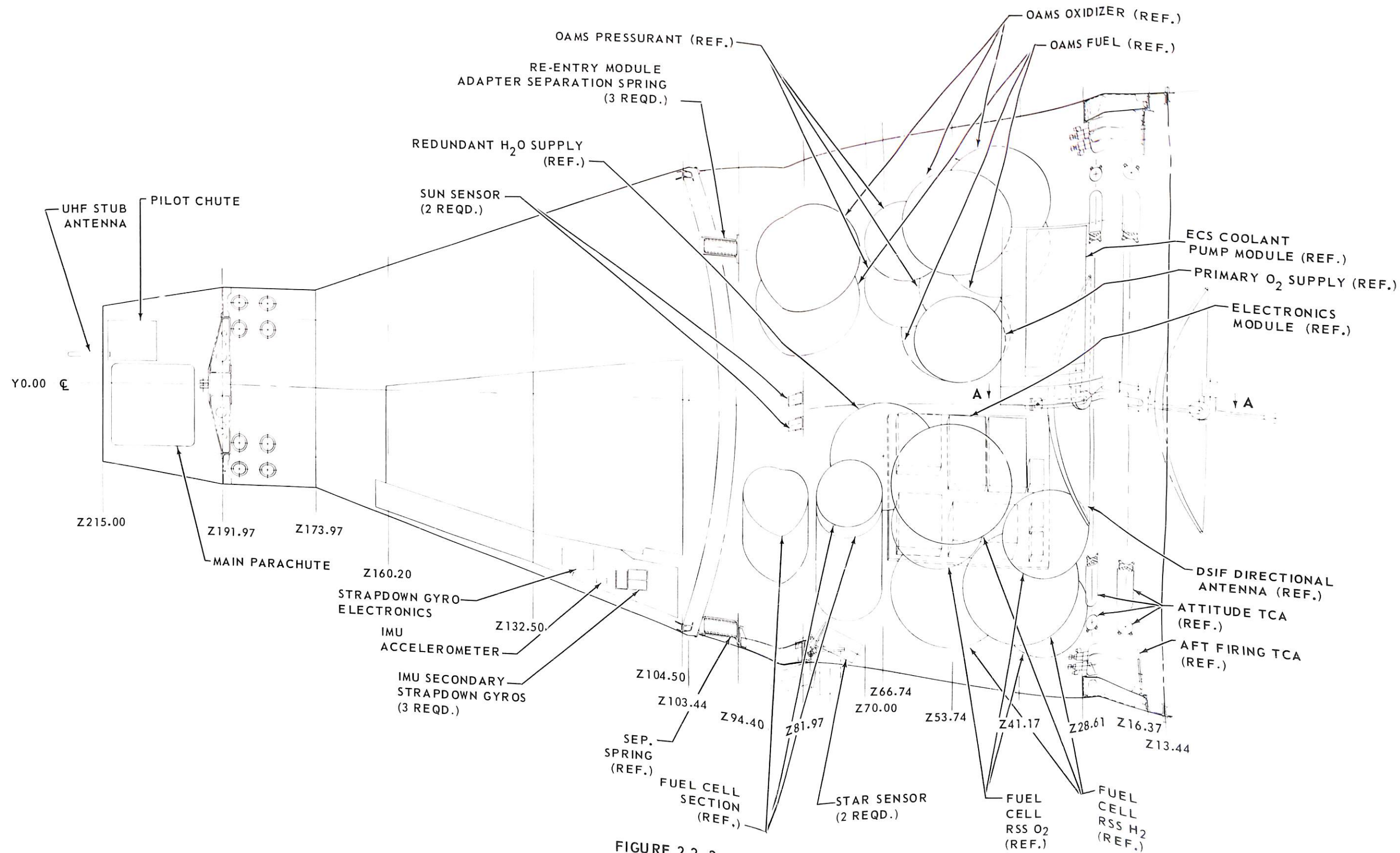
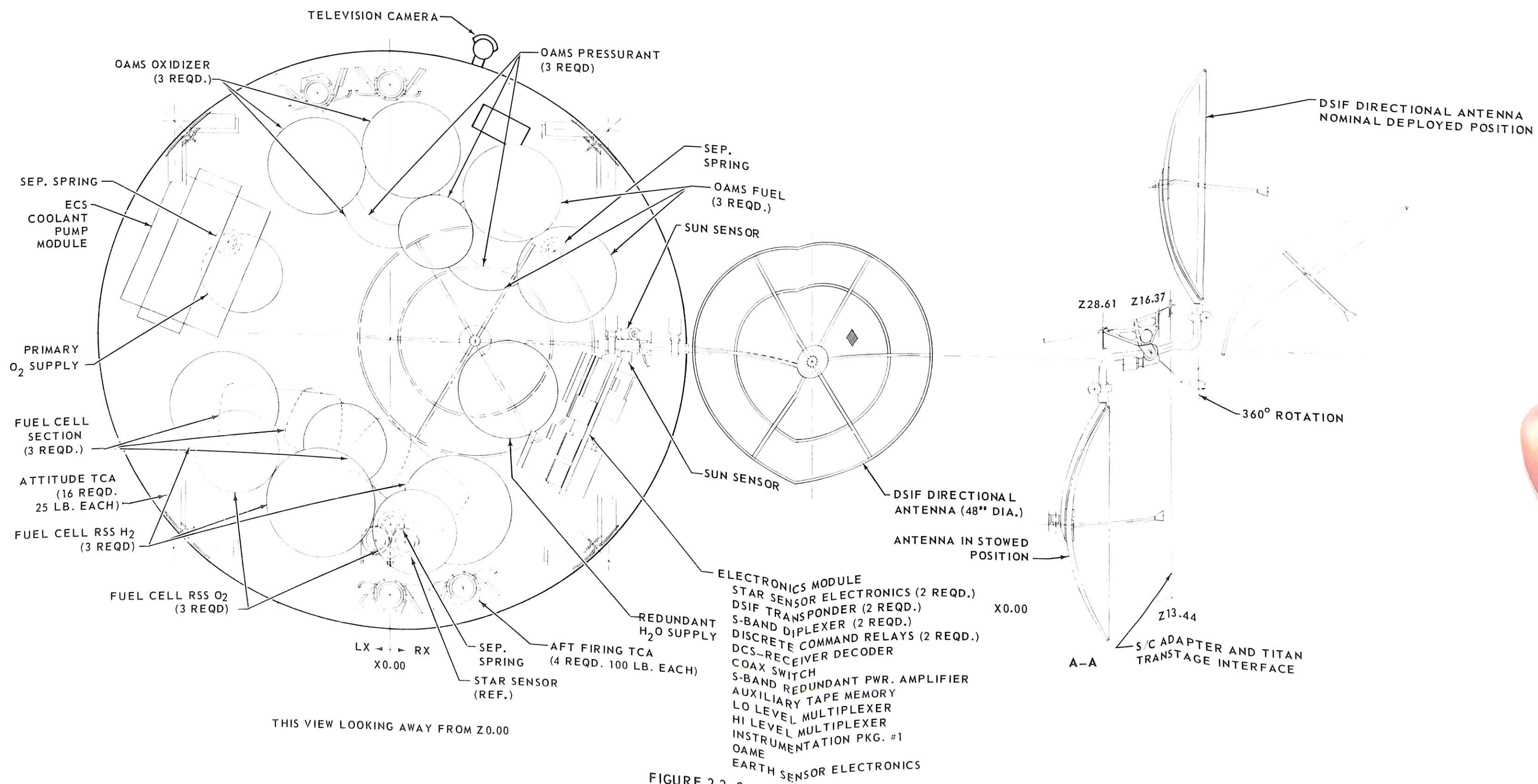


FIGURE 2.2-2

DIRECT ASCENT CONFIGURATION II

(Continued)



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FIGURE 2.2-2

3. SPACECRAFT SYSTEMS

3.1 Guidance and Control - The basic Gemini guidance and control system requires some modification to provide for the special functions peculiar to the large earth orbit mission, and to provide the redundancy for a high degree of reliability for the successful completion of such a mission.

The mission profile and the accuracy requirements are consistent with a rendezvous and docking maneuver velocity increment of 320 fps, a midcourse correction velocity increment of 458 fps, and an uncertainty in the re-entry flight path angle of 0.2 degree.

The basic functions which the guidance and control system provides in the large earth orbit mission are the following:

- A. Backup ascent guidance.
- B. Rendezvous and docking with the target vehicle.
- C. Insertion guidance from the parking orbit (rendezvous orbit) into the large earth orbit.
- D. Attitude control to provide proper orientation with respect to the sun to achieve thermal control and for DSIF antenna orientation.
- E. On-board orbit determination, navigation, and guidance.
- F. Midcourse maneuvers and attitude control during maneuvers.
- G. Re-entry guidance and control.
- H. Redundancy for both guidance and navigation, and attitude control functions.

The guidance and control system configuration is shown in Figure 3.1-1. The equipment required to provide the guidance and control function is listed in Table 3.1-1.

GUIDANCE AND CONTROL BLOCK DIAGRAM

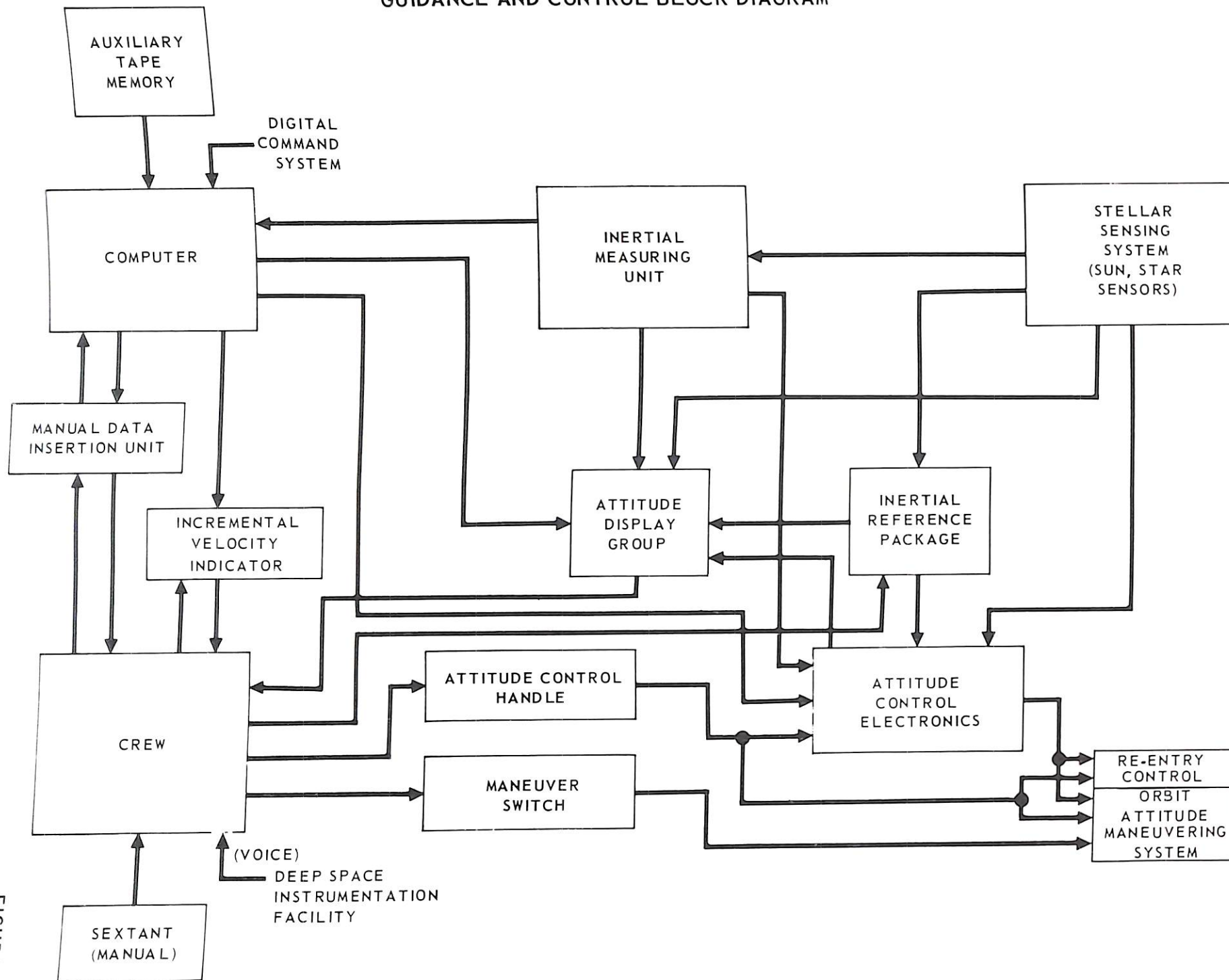


FIGURE 3.1-1

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TABLE 3.1-1
GUIDANCE AND CONTROL EQUIPMENT

EQUIPMENT (QUANTITY)	(A) FUNCTION (B) DESCRIPTION (C) REMARKS	CHARACTERISTICS		
		WT.	POWER	ACCURACY
ATTITUDE CONTROL ELECTRONICS (1)	(A) PROVIDES RATE CORRECTION SIGNALS TO THRUSTERS, MAINTAINS ROLL RATE DURING THERMAL RELIEF. (B) CONSISTS OF RATE GYROS AND ELECTRONICS NECESSARY TO ACCEPT ERROR SIGNALS FROM THE COMPUTER, INERTIAL REFERENCE PACKAGE, HAND CONTROLLER, IMU AND STAR TRACKER. (C) SYSTEM IS COMPLETELY REDUNDANT AND HAS DIRECT MODE WHICH REQUIRES NO ELECTRONICS.	40	12	RATE GYROS 0.04 DEGREES SECOND AT LESS THAN 5 DE- GREES SECOND. 0.4 DEGREES SECOND AT GREATER THAN 5 DEGREES SECOND.
INERTIAL MEASURING UNIT (1)	(A) PROVIDES ATTITUDE REFERENCE AND ACCELERATIONS DURING ENTIRE MISSION. (B) CONSISTS OF 4 GIMBAL, 3 ACCELEROMETER INERTIAL PLATFORM, POWER SUPPLY, SYSTEM ELECTRONICS, AND GIMBAL CONTROL ELECTRONICS.	129.6	300 (AVE)	GYRO DRIFT G-INSEN. $.3^{\circ}$ h ^g G-SENSITIVE $.5^{\circ}$ h ^g ANALOG READOUT 4 MIN. DIGITAL (& COD) 8 MIN. ACCELEROMETER - SAME AS IRP ACCELEROMETER.
COMPUTER (1)	(A) COMPUTES S C TRAJECTORY FOR RE-ENTRY, ORBIT DETERMINATION AND BACK-UP ASCENT, COMPUTES VELOCITY CHANGES FOR DISPLAY, MONITORS DSIF GUIDANCE. (B) 4096 WORD MEMORY, 39 BITS PER WORD, NON-DESTRUCTIVE RANDOM READOUT, 12,288 INSTRUCTION CAPABILITY, FIXED PRINT ARITHMETIC. (C) MEMORY CAPACITY INCREASED BY USE OF AUXILIARY TAPE MEMORY.	60	122	—
INCREMENTAL VELOCITY INDICATOR (1)	(A) DISPLAYS VELOCITY CHANGES OR COMMANDED VELOCITY CHANGES AS DIRECTED BY THE COMPUTER. (B) THREE WINDOWS OF THREE DIGITS EACH INDICATES VELOCITY CHANGES OF ± 999 IN EACH OF THE 3 S C AXIS.	3	8	—
MANUAL DATA INSERTION UNIT (1)	(A) INSERTS DATA INTO THE COMPUTER AND DISPLAYS DATA WORDS REQUESTED BY THE CREW. (B) CONSISTS OF A MANUAL DATA KEYBOARD UNIT AND A MANUAL DATA READOUT UNIT. INSERTS AND READS WORDS CONSISTING OF TWO DECIMAL ADDRESS DIGITS AND FIVE DECIMAL DATA DIGITS.	4	5	—
SUN SENSOR (2)	(A) SENSES SUN PITCH AND YAW POSITION WITH RESPECT TO THE S C BODY AXES. THESE SIGNALS ARE USED TO ALIGN PLATFORM AND INERTIAL REFERENCE PACKAGE. (B) CONSISTS OF OPTICAL SYSTEM, SOLID STATE DETECTOR, AND ASSOCIATED ELECTRONICS. (C) SENSORS ARE COMPLETELY REDUNDANT.	6	2	30 SEC. ALIGNMENT

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TABLE 3.1-1
GUIDANCE AND CONTROL EQUIPMENT
(Continued)

EQUIPMENT (QUANTITY)	(A) FUNCTION (B) DESCRIPTION (C) REMARKS	CHARACTERISTICS		
		WT.	POWER	ACCURACY
STAR SENSOR (2)	(A) SENSES TWO ORTHOGONAL INERTIAL REFERENCE AXES TO ALIGN THE PLATFORM AND INERTIAL REFERENCE PACKAGE, SENSES PITCH AND YAW DURING COAST PORTION TO PROVIDE REFERENCE FOR S C THERMAL RELIEF. (B) CONSISTS OF OPTICAL SYSTEM, PHOTOMULTIPLIER DETECTOR, AND ASSOCIATED ELECTRONICS. (C) SENSORS ARE REDUNDANT IN A DIRECT MISSION.	12	15	0.9 MIN ALIGNMENT
AUXILIARY TAPE MEMORY (1)	(A) EXTENDS COMPUTER MEMORY BY A FACTOR OF SEVEN. USED TO STORE NON-SAFETY OF FLIGHT PROGRAMS. (B) CONSISTS OF TAPE DRIVE MECHANISM AND ASSOCIATED ELECTRONICS. (C) HAS TRIPLE REDUNDANT READOUT OF TAPE AND MAJORITY VOTING.	15	15	—
INERTIAL REFERENCE PACKAGE	(A) PROVIDES ATTITUDE INFORMATION ABOUT THREE ORTHOGONAL AXES AND ACCELERATION INFORMATION ALONG THE S C LONGITUDAL AXIS. (B) CONSISTS OF THREE ± 10 DEGREE GYROS, ONE ACCELEROMETER, AND ASSOCIATED ELECTRONICS. (C) BACKS UP PLATFORM FOR SAFETY OF FLIGHT.	16	20	GYRO ALIGNMENT - 1.0 MIN. GYRO DRIFT - SHORT TERM. G-INSENSITIVE - .02° HR. ACCELEROMETER - BIAS - 1.67×10^{-5} g's - ALIGNMENT 10 SEC. SCALE FACTOR - .33 $\times 10^{-4}$ g g LINEARITY .833 $\times 10^{-5}$.
ATTITUDE DISPLAY GROUP (2)	(A) DISPLAYS IMU GIMBAL POSITIONS ON THE ATTITUDE BALL AND IMU GIMBAL POSITIONS, COMPUTER, OR INERTIAL REFERENCE PACKAGE ERROR SIGNALS ON THE INDICATOR NEEDLES. (B) GROUP CONSISTS OF ONE ATTITUDE BALL AND THREE INDICATOR NEEDLES. (C) ONE GROUP FOR EACH ASTRONAUT.	10	22	+ 30 MIN. ATTITUDE BALL + 9% NEEDLES
RENDEZVOUS RADAR* (1)	(A) DISPLAYS RANGE AND RANGE RATE TO THE TARGET VEHICLE TO THE CREW. (B) CONSISTS OF TRANSMITTER, RECEIVER, AND ASSOCIATED ELECTRONICS. (C) TRANSPONDER IS REQUIRED IN THE TARGET VEHICLE.	51	20	RANGE 10° RANGE RATE 1°
MANUAL SEXTANT	(A) ACCURATELY MEASURES ANGLE BETWEEN VARIOUS CELESTIAL BODIES. (B) CONSISTS OF AN OPTICAL SYSTEM. MICROMETER READOUT DIAL, AND TIMER. (C) BACK-UP FOR NAVIGATIONAL INFORMATION DETERMINATION.	8	NA.	30 SEC. 10 MIN. REPEATABILITY

*THIS EQUIPMENT REQUIRED FOR THE RENDEZVOUS MISSION ONLY

3.1.1 Guidance and Navigation

3.1.1.1 Functions and Requirements (Rendezvous Mission) - The guidance and navigation functions are described below in the framework of the computer modes.

Ascent Mode - The backup ascent guidance is provided by the present Gemini guidance system. A velocity increment of 111 fps is allotted for insertion dispersions. This is consistent with the present accuracy of the Gemini IGS.

Rendezvous and Docking - The spacecraft has the capability to perform the catch-up, braking, go-around and docking maneuvers. Efficient accomplishment of these functions requires a spacecraft to target vehicle range and range rate system during the spacecraft maneuvers which lead up to docking. This requirement is the result of a tradeoff study comparing the weight of a range and range rate system with the weight of the additional OAMS propellant required if an all-optical guidance technique were used.

Injection Guidance - In order to provide the guidance capability necessary to achieve proper insertion into the large earth orbit from the parking orbit, two general requirements are imposed on the guidance and navigation system. The first requirement is one of aligning the platform following docking with the target vehicle. The second is programming appropriate steering and navigation equations for the on-board computer in order that the injection into the large earth orbit may be accomplished. A platform alignment accuracy of 0.3 degrees in all axes is required to meet the insertion accuracy consistent with the allotment of midcourse maneuver velocity increment for insertion dispersions. Of the 458 fps velocity increment provided for midcourse maneuvers, 220 fps are allocated for correction of insertion dispersions.

Orbit Determination and Navigation - The on-board computer must ultimately provide the crew with the magnitude and direction of the velocity increment to

3.1.1.1 (Continued)

be applied. Pointing commands are provided in the form of attitude error signals, both to the Attitude Display Group (ADG) and to the Attitude Control Electronics (ACE). An on-board orbit determination and navigation capability is provided as a backup means for determining the magnitude and direction of the midcourse maneuvers.

Orbit Maneuver - This mode of operation is the same as that which is presently provided by the catchup mode of the Gemini computer. In addition, navigation during the midcourse maneuver is provided. The latter is required in the event of a loss of DSIF information, thereby allowing continuous navigation with a subsequent reduction in the number of manual sextant readings. This mode will supply velocity information to the Incremental Velocity Indicator (IVI) and attitude information to the ADG, and to the ACE. The ultimate accuracy requirement for this mode is a 0.2 degree flight path angle uncertainty at re-entry into the earth's atmosphere.

Re-entry - The re-entry guidance requirement is imposed for the purpose of providing the steering commands necessary to avoid a re-entry "g" level greater than 10 g's and the skipout boundary. In addition, re-entry guidance is provided to reduce touchdown dispersions, thereby decreasing the magnitude of the recovery operation. A steering logic similar to that proposed for the Apollo missions provides the required capabilities. The touchdown dispersion for a re-entry at supercircular velocity, with an update of spacecraft position and velocity prior to 400,000 ft. is presented in Table 3.1-2.

3.1.1.2 Functions and Requirements (Direct Mission) - The guidance and navigation functions required for a direct insertion into the large earth orbit are the same as discussed in Paragraph 3.1.1.1 with the following exceptions:

TABLE 3.1-2
RE-ENTRY TOUCHDOWN ACCURACY

ERROR SOURCE	ERROR MAGNITUDE	DISPERSION (NA.MI.)
GYRO ALIGNMENT	.3 DEG.	16.7
GYRO DRIFT	.3 DEG./HR.	.6
ACCELEROMETER BIAS	3×10^{-4} g	5.0
ACCELEROMETER SCALE FACTOR	3.6×10^{-4} g/g	2.3
ACCELEROMETER MISALIGNMENT	100 SEC.	1.7
RE-ENTRY UPDATE	MULTI-STATION	8.7
COMPUTER		10.0
TOTAL (3σ)		22.2

3.1.1.2 (Continued)

Ascent Mode - Since GLV is not used, a modification to the present ascent guidance equations is required. For this configuration, the accuracy requirements of the ascent mode are established in terms of the maneuver velocity increment required for correction of insertion dispersions. The midcourse velocity requirement for insertion dispersions is 220 fps.

Table 3.1-3 shows a comparison of the midcourse velocity requirements for the present Gemini IGS, the present Gemini IGS with improved accelerometers, and the launch vehicle (Titan IIIC) inertial guidance system. As is indicated in Table 3.1-3, the present IGS provides insertion accuracy within the midcourse maneuver propellant allotment, whereas the insertion errors associated with the present IGS with improved accelerometers would provide added margin.

3.1.1.3 System Operation (Rendezvous Mission) - The navigation and guidance equipment, listed in Table 3.1-1, provides command signals to the spacecraft displays and/or control system for steering the vehicle to the nominal trajectory during the following mission phases:

- A. Ascent
- B. Rendezvous
- C. Large Earth Orbit Insertion
- D. Coast
- E. Midcourse Correction
- F. Re-entry

The guidance and control mission profile is shown in Table 3.1-4.

Ascent - The IGS provides backup guidance for the launch vehicle, identical to the present Gemini, necessary to complete the specified mission should the primary system

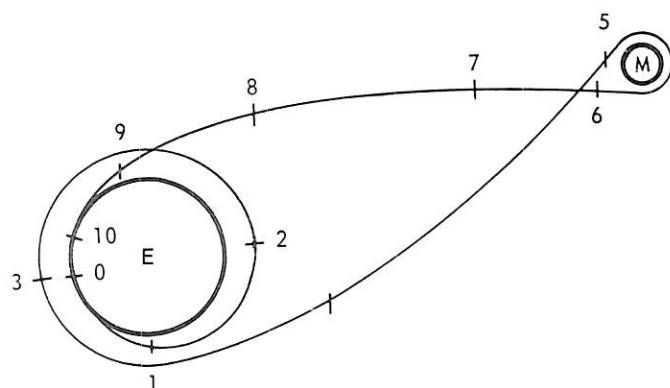
TABLE 3.1-3
MIDCOURSE VELOCITY REQUIREMENT FOR INSERTION ERRORS

ASCENT GUIDANCE PROVIDED BY:	MIDCOURSE MANEUVER VELOCITY (FEET PER SECOND)
PRESENT GEMINI IGS	220.0
PRESENT GEMINI IGS WITH IMPROVED ACCELEROMETERS	180.0
TITAN III-C SYSTEM	30.0

TABLE 3.1-4
GUIDANCE AND CONTROL MISSION PROFILE
RENDEZVOUS MISSION

MISSION PHASES

0-1	ASCENT
2	RENDEZVOUS, DOCKING
3	INSERTION INTO LEO
3-4, 4-5, 6-7, 7-8, 8-9	COAST
4, 5, 6, 7, 8	MIDCOURSE MANEUVERS
5-6	MOON OBSERVATION
9-10	RE-ENTRY



PHASE	TASK	PRIMARY	BACK-UP
ASCENT	● MONITOR PRIMARY GUIDANCE ● PROVIDE BACK-UP STEERING COMMANDS ● PROVIDE INITIAL CONDITIONS FOR CONTROLLED RE-ENTRY FOLLOWING LAUNCH ABORT.	● PILOT MONITORS BOOSTER OPERATION. ● SEPARATION FROM BOOSTER ● INSERTION VELOCITY CORRECTION	● SWITCH-OVER TO BACK-UP GUIDANCE IN THE EVENT OF PRIMARY GUIDANCE FAILURE. ● SAME AS PRIMARY ● SAME AS PRIMARY
RENDEZVOUS & DOCKING	● PERFORM CATCH-UP, BRAKING, GO-AROUND, AND DOCKING MANEUVERS.	● SEMI-OPTICAL USING RANGE AND AND RANGE RATE RADAR.	● ALL - OPTICAL
INSERTION INTO LEO	● PERFORM GUIDANCE NECESSARY TO INSERT INTO LARGE EARTH ORBIT.	● ALIGN INERTIAL PLATFORM. ● PROVIDE STEERING SIGNALS TO TARGET VEHICLE.	● ALIGN STRAP-DOWN GYROS. ● SAME AS PRIMARY.
COAST	● PROVIDE ATTITUDE CONTROL FOR THERMAL CONTROL AND PILOT RELIEF. ● PROVIDE ATTITUDE CONTROL FOR DSIF ANTENNA POINTING. ● MANUAL SEXTANT READINGS FOR ON-BOARD ORBIT DETERMINATION.	● PILOT ORIENTS SPACECRAFT TO ORBIT ATTITUDE. ● MAINTAIN ORBIT ATTITUDE AUTOMATICALLY. ● DSIF DATA RECEIVED FOR: (a) MIDCOURSE MANEUVERS (b) UPDATING COMPUTER WITH SPACECRAFT POSITION AND VELOCITY.	● SAME AS PRIMARY ● MANUAL CONTROL TO ORBIT ATTITUDE ● EARTH-STAR, MOON-STAR READINGS (MANUAL SEXTANT) FOR ORBIT DETERMINATION. ● ON-BOARD CALCULATION OF MID-COURSE MANEUVERS.
MIDCOURSE MANEUVERS	● ALIGN PLATFORM ● PERFORM MIDCOURSE MANEUVER ● ATTITUDE CONTROL DURING MANEUVER.	● FORTY-FIVE MINUTES PRIOR TO MANEUVER, PLATFORM TURNED ON-ALIGNED WHEN WARM. ● FIVE MINUTES PRIOR TO MANEUVER SPACECRAFT IS ORIENTED TO PROPER ATTITUDE. ● PILOT INITIATES MANEUVER. ● ATTITUDE CONTROL AUTOMATIC - COMPUTER GENERATES ATTITUDE ERROR COMMANDS.	● FORTY-FIVE MINUTES PRIOR TO MANEUVER, STRAP-DOWN GYROS TURNED ON - ALIGNED WHEN WARM. ● TEN MINUTES PRIOR TO MANEUVER, SPACECRAFT IS ORIENTED TO PROPER ATTITUDE. ● SAME AS PRIMARY ● ATTITUDE CONTROL AUTOMATIC - STRAPDOWN GYROS GENERATES ATTITUDE ERROR COMMANDS.
MOON OBSERVATION	● ATTITUDE CONTROL FOR MOON TRACKING.	● MANUAL USING RATE COMMAND OR PULSE MODE.	● MANUAL USING DIRECT MODE.
RE-ENTRY	● RE-ENTRY GUIDANCE TO PRE-DETERMINED TOUCHDOWN POINT.	● PLATFORM IS ALIGNED. ● RCS IS TURNED ON - ADAPTER IS JETTISONED. ● AT 400,000 FT., ROLL ATTITUDE DETERMINED BY COMPUTER IS FLOWN - RATE DAMPING IN PITCH AND YAW. ● AUTOMATIC ATTITUDE CONTROL IN ROLL.	● HOLD ATTITUDE USING HORIZON VIEW. ● SAME AS PRIMARY ● AT 400,000 FT., OPEN LOOP ROLL ATTITUDES AS DETERMINED BY GROUND ARE FLOWN USING HORIZON REFERENCE - RATE DAMPING IN PITCH AND YAW. ● MANUAL CONTROL USING RE-ENTRY RATE COMMAND OR DIRECT MODES.

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3.1.1.3 (Continued)

become inoperative. Switchover is either automatic by the malfunction detection system in the booster, or manually by the crew and ground control.

Rendezvous - The guidance and navigation equipment provides the crew with the range and range rate of the target vehicle with respect to the spacecraft. Range and range rate information enable the crew to apply the incremental velocity necessary for efficiently maneuvering the spacecraft to the target. With an initial rendezvous target vehicle to spacecraft position displacement of 20-30 miles, no other guidance equipment is required. The rendezvous is accomplished by first placing the target vehicle into a 120 n.m. circular orbit, and one orbit later placing the spacecraft into a 87-100 n.m. orbit. For the docking function, the target vehicle is stabilized by the Transtage control system and the Gemini spacecraft performs the necessary "go-around" and docking maneuvers.

Large Earth Orbit Insertion - The IGS provides steering command signals to the spacecraft displays and to the Transtage autopilot to guide the spacecraft-Transtage combination into the large earth orbit. Paragraph 3.1.1.1 establishes the requirement that the platform alignment be better than 0.3 degrees with respect to the stellar reference. Alignment of the IMU is accomplished using the star sensors. The inertial reference is that defined by the two stars Canopus and Fomalhaut. In order to minimize alignment time after docking is completed, some alignment is done prior to the completion of the docking phase of the mission.

Coast - During the coast phase of operation, no guidance functions as such are performed. However, during this phase, the on-board computer is periodically updated via the DSIF command link in order to provide information for the on-board orbit determination and navigation functions. Should the DSIF become in-

3.1.1.3 (Continued)

operative, measurements of earth-star, moon-star angles are made using the manual sextant. These angles are inserted into the computer where they are processed by the computer for orbit determination. Midcourse correction maneuvers are based on these parameters.

Midcourse Correction - During the midcourse correction maneuver, the guidance and navigation equipment provides attitude and velocity information necessary to guide the spacecraft. The velocity corrections required to place the spacecraft on the proper trajectory to circumnavigate the lunar target, and to be within the re-entry corridor, are determined by ground-based computers and inserted into the on-board computer via the DSIF link. Prior to performing the midcourse maneuver, the crew aligns the platform to the inertial reference using the sun and star sensors. To accomplish the midcourse maneuver, the crew selects the computer mode of the Attitude Control System (ACS) and maneuver mode of the computer. In this mode, the ACS positions the spacecraft so that the entire velocity increment is applied along the longitudinal axis. This DSIF supplied velocity increment is also displayed on the IVI. At a specified time, the crew actuates the Maneuver System until the IVI indicates zero velocity to be gained.

Alignment of the platform to the inertial reference is required to perform the midcourse corrections with a velocity increment of 458 fps. Error analysis indicates that the platform misalignment is less than 18 minutes of arc. This misalignment is the Root Sum Square of the independent variables associated with platform alignment.

The crew performs the following functions to align the platform:

- A. Cage the platform to the spacecraft.
- B. Position the spacecraft until the sun is in the field of view of the sun sensor.

3.1.1.3 (Continued)

- C. Roll the spacecraft until Canopus is in the field of view of the star sensor while maintaining the sun alignment.
- D. Place the platform mode switch to the "Align" position and the attitude control switch to the "Platform" mode.

The platform then utilizes the sun and star sensor errors to align itself to the stellar reference.

The Inertial Reference Package (IRP) provides backup for the platform. The procedure for alignment of this package is identical to that described for the platform. Should the IVI or computer fail, pointing directions and thrusting times are given the crew via the DSIF.

Re-entry - The IGS controls the spacecraft during re-entry in a manner similar to that of the present Gemini. However, the re-entry guidance logic is modified to include guidance during the supercircular flight regime.

3.1.1.4 System Operation (Direct Mission) - The requirements for this mission are the same as outlined for the rendezvous mission except that mission phases "Rendezvous" and "Large Earth Orbital Insertion" are deleted. The deletion of these phases removes the requirement for the range and range rate system.

3.1.2 Stabilization and Control System3.1.2.1 Functions and Requirements (Rendezvous Mission)

Spacecraft - The stabilization and control system of the spacecraft provides:

- A. Ability to establish and hold spacecraft attitude.
- B. Low spacecraft rates for alignment of attitude reference and tracking.
- C. Attitude orientation and programmed spacecraft roll rate for thermal control.
- D. Backup for all of the above functions.

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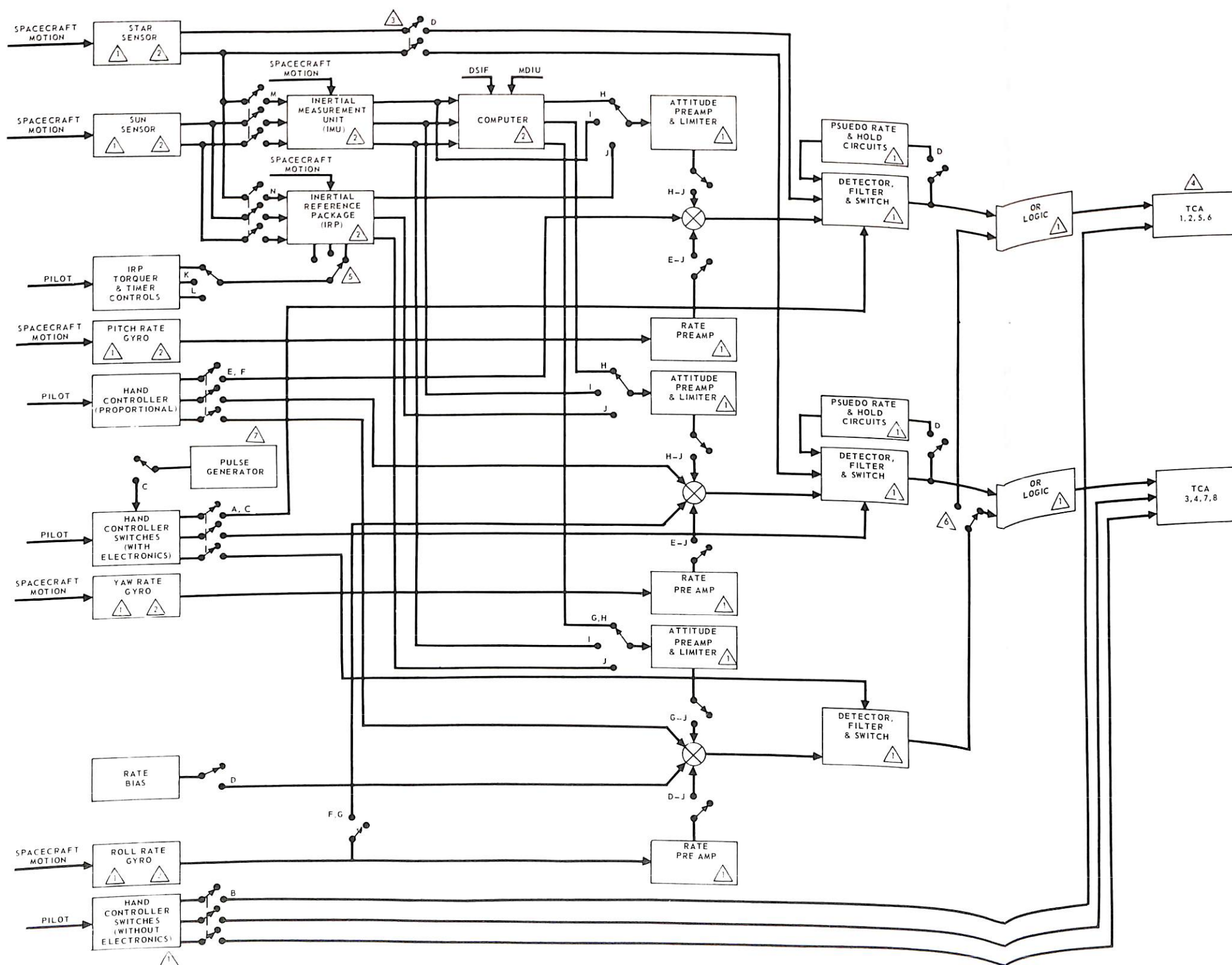
3.1.2.1 (Continued)

The capability of changing spacecraft attitude at a reasonable rate is provided, as well as the capability to maintain specified orientation under all flight conditions for all phases of the mission. Included in this category is attitude control during the rendezvous and docking phase, during insertion into orbit, during midcourse maneuvers, and during re-entry. Low spacecraft rates are required to manually align either of the two attitude references accurately, and to allow precision tracking of celestial bodies. Orientation of the spacecraft to achieve thermal control over long periods of the mission is provided. Because of the long duration of flight, some form of crew relief is also provided. Since these two functions are performed simultaneously, they are combined in one automatic control mode. For all of the functions listed above, various backup modes are available in the event that the primary means of control is inoperative. The crew, in most instances, provides the most efficient means of establishing the orientation of the spacecraft. However, once the spacecraft is properly oriented, an automatic means of maintaining attitude is more efficient. In order to obtain minimum rates, the crew is provided with a pulse mode capability. Wiring between the IRP and attitude control system is provided such that torquing of the IRP can automatically control the attitude one axis at a time. A functional block diagram, illustrating the attitude control modes and the reference alignment to do the tasks outlined above, is shown in Figure 3.1-2. The essential operational details of the attitude control modes are summarized in Table 3.1-5.

Target Vehicle - During the thrusting period of the target vehicle, i.e., insertion into the large earth orbit, complexity is added to the guidance and control system due to the flexibility of the spacecraft target vehicle configuration. Rate gyros and an autopilot in the target vehicle are considered to

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ATTITUDE CONTROL AND STABILIZATION SYSTEM FUNCTIONAL BLOCK DIAGRAM INCLUDING REFERENCE ALIGNMENT



NOTES:

- 1 THESE ITEMS ARE REDUNDANT.
- 2 THESE QUANTITIES ARE DISPLAYED AT PILOT'S OPTION.
- 3 CONTROL MODES:
SWITCH POSITIONS NOT IDENTIFIED ARE FOR ALL MODES NOT OTHERWISE DELINEATED.
A. DIRECT WITH ELECTRONICS
B. DIRECT WITHOUT ELECTRONICS
C. PULSE
D. ORBIT
E. RATE COMMAND
F. RE-ENTRY RATE COMMAND
G. RE-ENTRY
H. COMPUTER
I. IMU
J. IRP
K. IRP ATTITUDE
L. IRP RATE
M. ALIGNMENT OF IMU
N. ALIGNMENT OF IRP
- 4 THE NUMBER SYSTEM IS THAT SHOWN IN FIGURES FOR EITHER OAMS OR RCS TCA'S.
- 5 ONLY ONE CHANNEL AT A TIME MAY BE CONTROLLED. THE POSITIONS ARE FOR PITCH, YAW AND ROLL.
- 6 ROLL COMMANDS CAN USE EITHER PITCH OR YAW JETS.
- 7 THE PULSE GENERATOR PRODUCES A SINGLE PULSE WHEN ACTUATED BY THE HAND CONTROLLER.

FIGURE 3.1-2

TABLE 3.1-5
ATTITUDE STABILIZATION AND CONTROL SYSTEM
MODES OF OPERATION

CODE (FIG. 3.1-2)	MODE NAME	PILOT CONTROL	DEADBAND	OPERATIONAL DETAIL	PRIMARY USAGE
A	DIRECT WITH ELECTRONICS	HAND CONTROLLER	NOT APPLICABLE	PILOT DEFLECTS HAND CONTROLLER MORE THAN 25 PERCENT OF FULL TRAVEL; CONTINUOUS THRUST OCCURS.	PRIMARILY A BACK-UP FOR OTHER MODES OF ATTITUDE CONTROL.
B	DIRECT WITHOUT ELECTRONICS	HAND CONTROLLER	NOT APPLICABLE	SAME AS A.	SAME AS A.
C	PULSE	HAND CONTROLLER	SINGLE MINIMUM IMPULSE PULSE	PILOT DEFLECTS HAND CONTROLLER MORE THAN 25 PERCENT OF FULL TRAVEL. A SINGLE MINIMUM IMPULSE PULSE IS GENERATED. PULSE GENERATOR IS RESET BY REDUCING HAND CONTROLLER DEFLECTOR TO LESS THAN 25 PERCENT OF FULL TRAVEL.	MANUAL ALIGNMENT OF ATTITUDE REFERENCE; ACCURATE TRACKING OF CELESTIAL BODIES; REDUCTION OF BODY RATES TO A MINIMUM.
D	ORBIT	NONE	ROLL-0.2 DEG./SEC. PITCH-3 DEG. YAW-3 DEG.	MODE IS ENGAGED WHEN DESIRED STAR IS WITHIN FIELD OF VIEW OF STAR SENSOR. MAINTAINS SPACECRAFT LONGITUDINAL AXIS BETWEEN 45 AND 70 DEGREES FROM SUN'S RAYS. MINIMUM ROLL RATE OF 0.8 DEG./SEC. IS ESTABLISHED.	PRIMARY MODE OF ATTITUDE CONTROL DURING COAST PHASES. ORIENTATION AND ROLL RATE REQUIRED FROM THERMAL CONSIDERATIONS. ALSO PROVIDES A PILOT RELIEF MODE.
E	RATE COMMAND	HAND CONTROLLER	0.2 DEG./SEC. - ALL AXES.	AUTOMATIC RATE DAMPING IN ALL AXES. FOR ATTITUDE CHANGES, PILOT DEFLECTS HAND CONTROLLER WHICH COMMANDS RATES PROPORTIONAL TO THE DEFLECTION GREATER THAN 10 PERCENT FULL TRAVEL. MAXIMUM RATES ARE 10 DEGREES/SECOND IN PITCH AND YAW, 15 DEGREES/SECOND IN ROLL.	PRIMARY MODE FOR CHANGING ORBIT ATTITUDE. MANUAL BACK-UP TO AUTOMATIC ATTITUDE CONTROL MODE DURING MIDCOURSE MANEUVER.
F	RE-ENTRY RATE COMMAND	HAND CONTROLLER	ROLL-2 DEG./SEC. PITCH-4 DEG./SEC. YAW-4 DEG./SEC.	SAME AS E.	MANUAL RE-ENTRY ATTITUDE CONTROL MODE WITH AUTOMATIC RATE DAMPING WITH MANUAL OVERRIDE.
G	RE-ENTRY	NONE	ROLL-2 DEG./SEC. PITCH-4 DEG./SEC. YAW-4 DEG./SEC.	SPACECRAFT COMPUTER GENERATES ROLL ATTITUDE ERROR SIGNAL. RATE DAMPING IN ALL AXES.	AUTOMATIC RE-ENTRY CONTROL MODE.
H	COMPUTER	NONE	0.5° - ALL CHANNELS	COMPUTER GENERATES PITCH, YAW AND ROLL ATTITUDE ERROR SIGNALS FOR PROPER SPACECRAFT ORIENTATION DURING MIDCOURSE MANEUVER. ATTITUDE ERROR SIGNALS MIXED WITH RATES.	AUTOMATIC ATTITUDE CONTROL DURING MIDCOURSE MANEUVER.
I	IMU (INERTIAL MEASUREMENT UNIT)	NONE	0.5°-ALL CHANNELS	GIMBAL ANGLE OUTPUTS OF IMU USED TO CONTROL SPACECRAFT ATTITUDE TO NULL OF IMU.	AUTOMATIC ATTITUDE CONTROL DURING IMU ALIGNMENT.
J	IRP (INERTIAL REFERENCE PACKAGE)	NONE	0.5° - ALL CHANNELS	ERROR SIGNALS FROM STRAP-DOWN GYROS USED TO CONTROL SPACECRAFT ATTITUDE TO NULL OF IRP.	AUTOMATIC ATTITUDE CONTROL DURING IRP ALIGNMENT; ALTERNATE METHOD FOR CONTROLLING ATTITUDE DURING MIDCOURSE MANEUVER; BACK-UP FOR HAND CONTROLLER.

3.1.2.1 (Continued)

be a minimum requirement to allow adequate attitude stabilization and control during the insertion phase.

3.1.2.2 Functions and Requirements (Direct Mission) - The stabilization and attitude control system for the direct mission configuration is the same as delineated in Paragraph 3.1.2.1 for the spacecraft with the exception that the control during rendezvous and docking are deleted. This, however, does not in any way change the configuration presented in Figure 3.1-2.

3.1.3 Guidance and Control Display Requirements (Rendezvous Mission) - Modification to existing displays and switches on the present Gemini display panel is required due to the addition of the sun and star sensors, ATM, and IRP.

Flight Director Indicator - The Flight Director Indicator (FDI) reference switch has five (5) positions: IMU, IRP, COMP, SENSOR (ALIGN), SENSOR (ORBIT). The FDI mode switch remains unchanged. The scale factor switch for the Flight Director Needles are changed from a two-position switch to a three-position switch. The needle sensitivities are (1,1,1 deg./deg. per second), (5,5,5, deg./deg. per second) and (10,10,15 deg./deg. per second).

Inertial Reference Package Control - Two rotary switches and a dial are provided to control the strapdown gyro unit. One rotary switch controls the mode of operation of the IRP. The switch positions are: Off, Cage, Align, Free, Free-Att. and Free-Rate. There is a dial or register to load the angles through which a selected gyro is to be torqued or the rate at which it is to torqued. In addition to this dial or register, a channel selector (pitch, yaw or roll) is provided.

Sensor Switches - There is an On-Off switch for each star sensor and the sun sensors in addition to a light indicating when each sensor is tracking.

3.1.3 (Continued)

The positioning of the star sensor is accomplished with a rotary switch with the following positions: Stow, Align (Rendezvous), Align (Orbit) and Coast.

Auxiliary Tape Memory (ATM) Switches - Operation of the ATM requires a rotary switch to select the mode of operation, an error light, a running light and a reset button. The positions of the mode switch are Off, On, Wind, Rewind, Verify, and Store.

The display devices currently in Gemini, IVI, MDIU, and Flight Director are utilized wherever possible. No additional displays are required.

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3.2 Environmental Control System

3.2.1 Suit and Cabin Systems - The suit circuit environmental control system for Configurations I and II is the same as the present Gemini with a 7-day LiOH canister. The cabin environmental control system is also the same as the present Gemini, except the cabin fan and heat exchanger are removed. This removal is substantiated by cabin temperature data from early Gemini flights.

The modes of operation of the cabin and suit environmental control systems during the various phases of flight are the same as Gemini.

3.2.2 Coolant System - Each fuel cell has its own coolant loop and is separate from the main coolant loops. The heat from the fuel cells is transferred in a heat exchanger to the spacecraft's main cooling system. This allows for better fuel cell temperature control, a very low pressure fuel cell cooling system, and, if one fuel cell fails or leaks, it does not invalidate the spacecraft's main cooling system or the remaining fuel cells. Each fuel cell pump is operated continuously unless a fuel cell fails. The spacecraft's cooling system has a primary loop and a secondary loop, and each loop contains two pumps as does the present fuel cell powered Gemini. During the high heat load periods (all electronic equipment operating), one pump in each loop is operating unless one loop had failed in which case both pumps in the remaining loop are required. The schematic of the system is shown in Figure 3.2-1.

3.2.3 Cryogenic System - This system must supply oxygen in the gaseous state for metabolic purposes, as well as oxygen and hydrogen in the gaseous state for fuel cell operation. The system consists of four 2-day RSS O_2 containers and three 2-day hydrogen containers. Each container is serviced so that if one oxygen and one hydrogen container fails, the mission can be successfully completed with the amount of cryogen remaining. The systems are configured such that the failed container is isolated from the rest of the system and will not impair normal operation. There are

ADAPTER COOLANT SYSTEMS

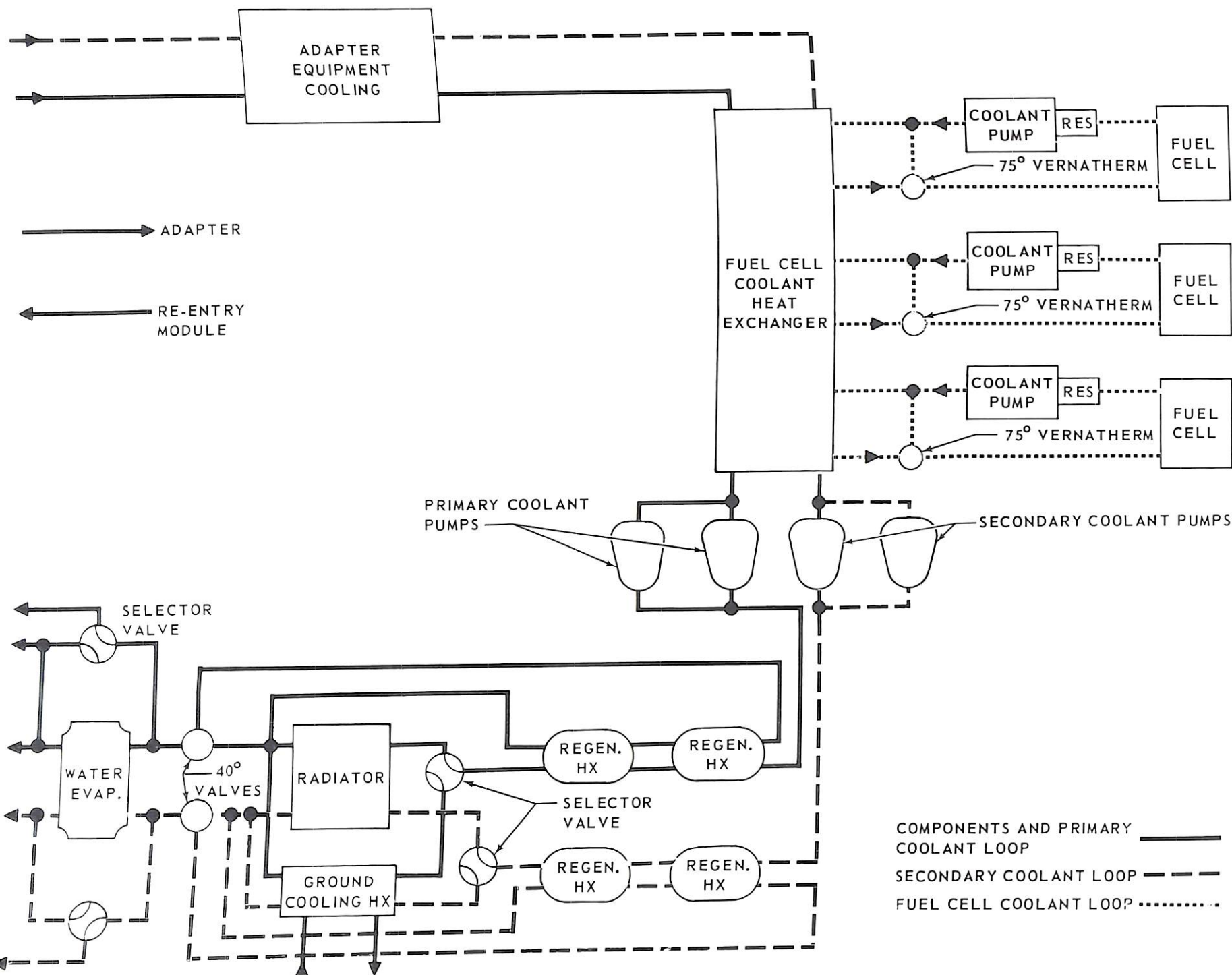


FIGURE 3.2-1

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3.2.3 (Continued)

two pressure regulators in parallel in the metabolic oxygen system so that if one fails, the crew can select the other regulator by means of solenoid valves. A similar situation exists in the fuel cell oxygen-hydrogen delivery systems with three dual pressure regulators in parallel such that if one regulator fails, the crew can isolate it with a solenoid valve and continue to operate all three fuel cells by means of dual O₂-H₂ crossover valves. The schematic is shown in Figure 3.2-2.

3.2.4 Water Management System - The water management system in the re-entry module is the same as the present Gemini. The water management system in the adapter is shown schematically in Figure 3.2-2. Water is formed in the fuel cells as a by-product. This water has a high acid content and is neutralized and purified by the ion exchange columns before it is potable. Fuel cell pressure is controlled by the water pressure relief valve which supplies a reference pressure for the dual O₂-H₂ pressure regulators. The water is then collected in the bladder of two 20-inch OAMS tanks which are being used as water storage tanks. The re-entry module water tank is then replenished as required from the storage tanks by manipulation of a manual shut-off valve in the cabin. Prior to launch, the tanks are serviced with 29.70 lbs. of water, the remainder required for the mission, 76.30 lbs., is generated by the fuel cells during the mission. Water transfer from the storage tanks is accommodated by oxygen pressure on the gas side of the bladder. This is serviced with 100% gaseous oxygen at 14.7 psia. The fuel cells can be operated for approximately 40% of the mission without water being drawn from the tanks before the pressure in the tank will affect the fuel cells. When the last pound of water is delivered from the tanks, the pressure is 11.6 psia which is sufficient since the maximum pressure in the cabin during orbit is 6.0 psia.

CRYOGENIC AND WATER MANAGEMENT SYSTEMS

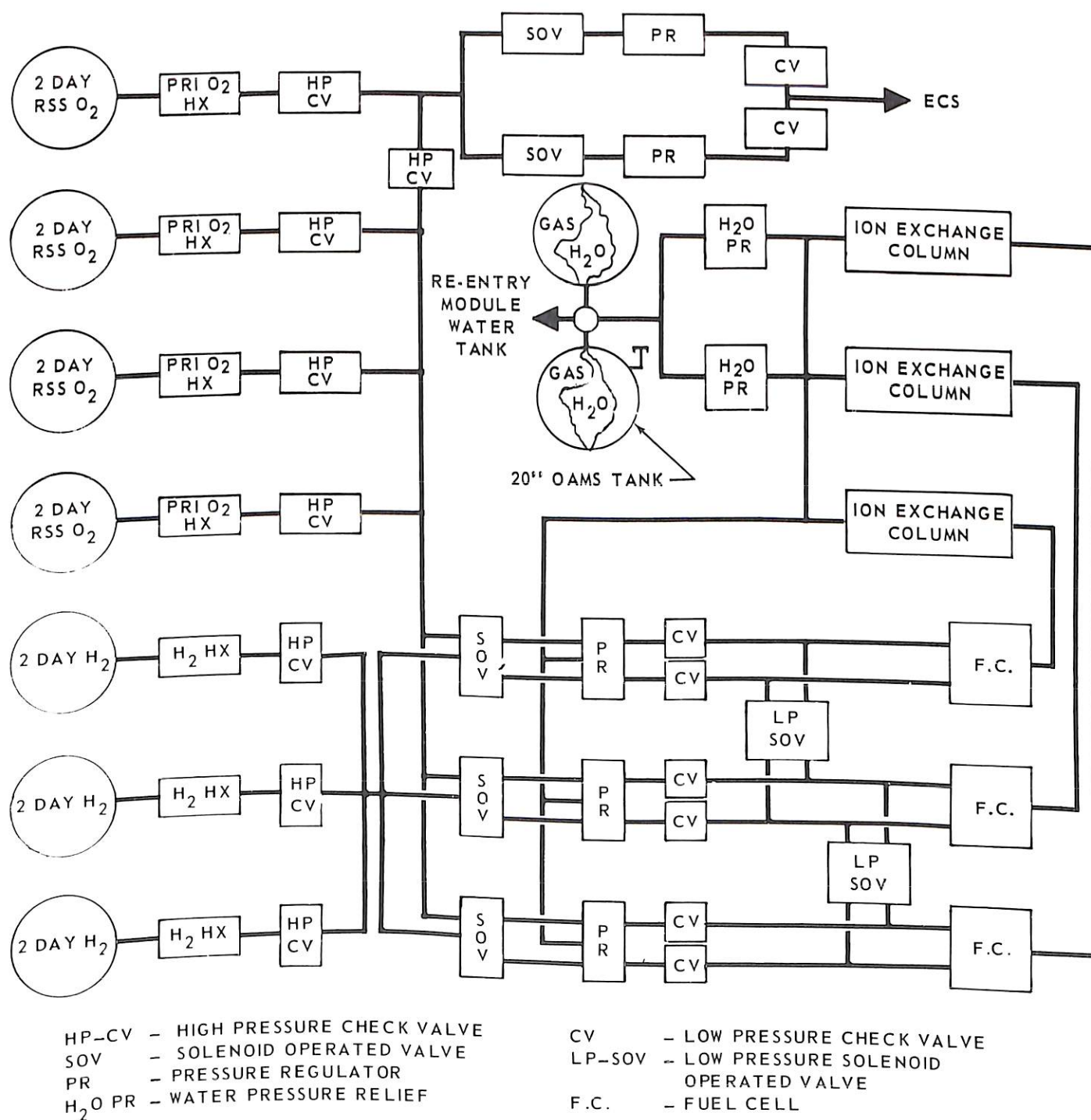


FIGURE 3.2-2

3.2.4 (Continued)

The re-entry module water tank is serviced with 13.0 lbs. of water for post-landing purposes. Seven lbs. of water are stored in the water boiler for launch cooling.

In case of a water tank (OAMS tank) failure the moisture from respiration and perspiration, which is condensed in the suit circuit heat exchanger, can be collected in the cabin water tank and used for drinking. Otherwise, it is dumped overboard through the launch cooling heat exchanger.

The urine disposal system is the same as the present Gemini.

3.3 Electrical Power

3.3.1 Configuration I

3.3.1.1 Description - The electrical power system consists of three (3) Fuel Cell sections and six (6) Silver-Zinc batteries which supply three (3) independent busses, consisting of the Main Bus and two (2) separate and independent Squib busses.

The Fuel Cell Main Bus battery system supply will provide the mission power, and any two (2) of the three (3) sections will supply the normal mission load requirements. The emergency mission load requirements can be provided by any one (1) of the three (3) sections. The emergency mission is assumed to begin ten (10) hours after injection into a large earth orbit at a load level of approximately 71 per cent of normal load.

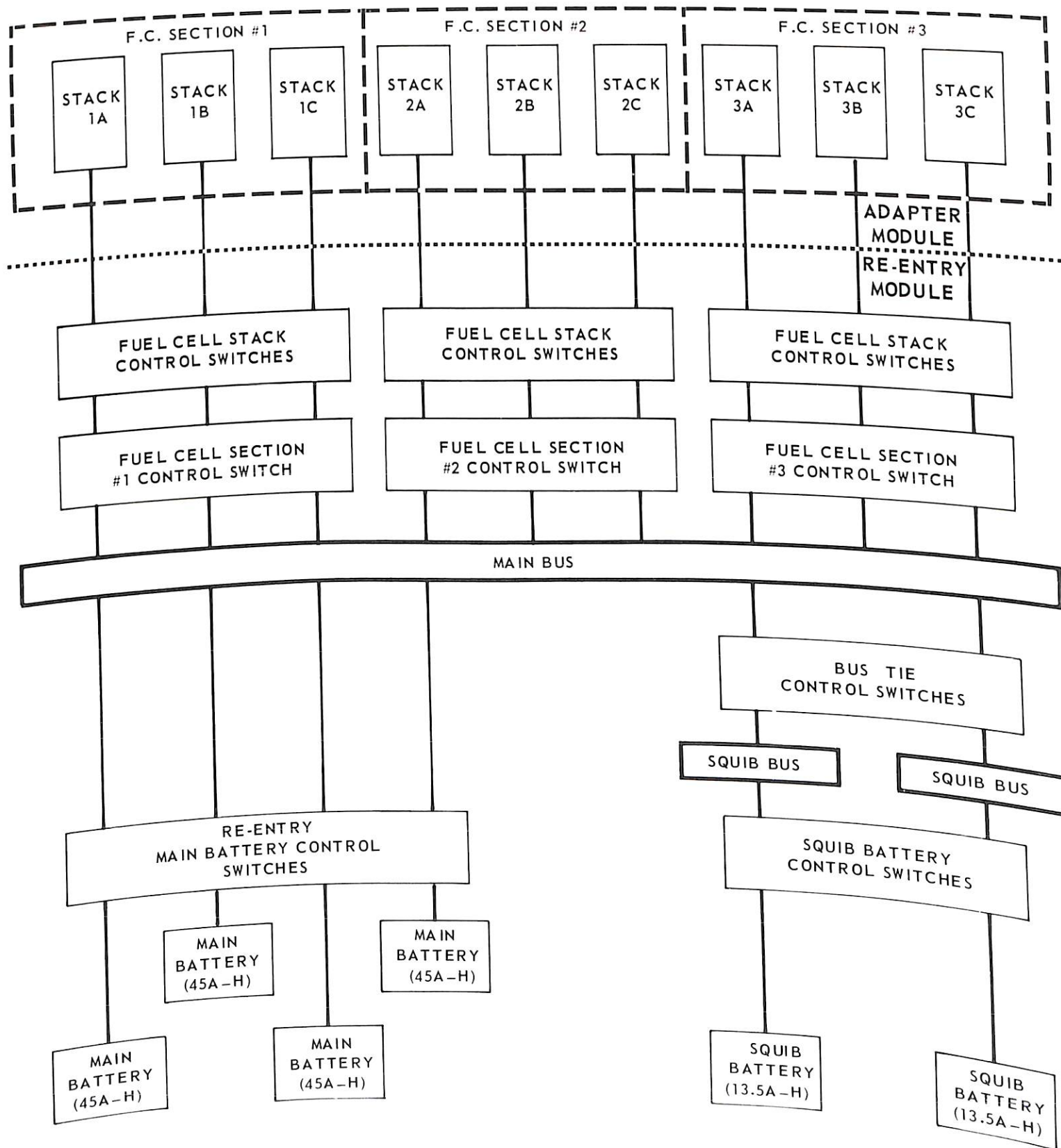
Re-entry, landing and post-landing power is supplied by the four (4) forty-five (45) ampere-hour Re-entry Main Bus batteries. Under normal conditions, the load can be provided by any three (3) of the four (4) batteries.

Redundant thirteen and one-half (13.5) ampere-hour Squib batteries and busses are provided to supply transient loads such as solenoids and pyrotechnics. Isolation of these loads from the Main Bus system tends to minimize the effects on the operation of sensitive electronic equipment.

Maximum in-flight control flexibility is provided by individual Fuel Cell stack and section switches, Silver-Zinc battery control switches, and Bus Tie provisions. This electrical power system is shown in block diagram form in Figure 3.3-1.

3.3.1.2 Utilization - The detailed system normal load requirements are tabulated in Table 3.3-1 and the associated load profile is presented in Figure 3.3-2. Emergency loads and profile appear in Table 3.3-3 and Figure 3.3-4, respectively.

ELECTRICAL POWER SYSTEM - BLOCK DIAGRAM



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FIGURE 3.3-1

TABLE 3.3-1
ELECTRICAL LOAD SUMMARY
CONFIGURATION I

EQUIPMENT	SYSTEM TOTAL	PRE-LAUNCH	LAUNCH & INJECTION	ORBIT	ORBIT CORRECTION	APOGEE CONTROL	ORBIT	ORBIT CORRECTION	RE-ENTRY	POST LANDING
	AMP.-HRS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.
COMMUNICATIONS	258.59	.80	3.69	.94	6.26	6.26	.94	6.26	2.78	1.36
ENVIRONMENTAL	1948.15	14.34	17.54	10.83	15.34	15.34	10.83	15.34	3.12	.85
GUIDANCE	505.70	22.96	23.77	1.09	23.01	23.01	1.09	23.01	17.80	
INSTRUMENTATION	410.66	3.68	3.68	2.32	2.93	2.93	2.32	2.93	3.48	
LIGHTING	65.83	.85	.37	.38	.38	.38	.38	.38	1.08	.02
MISCELLANEOUS	139.23	3.45	3.51	.57	3.48	3.48	.57	3.48	.44	
EXPERIMENTS	7.32		.12	.03		1.46	.03			
TOTAL	3335.48	46.08	52.68	16.16	51.40	52.86	16.16	51.40	28.70	2.23

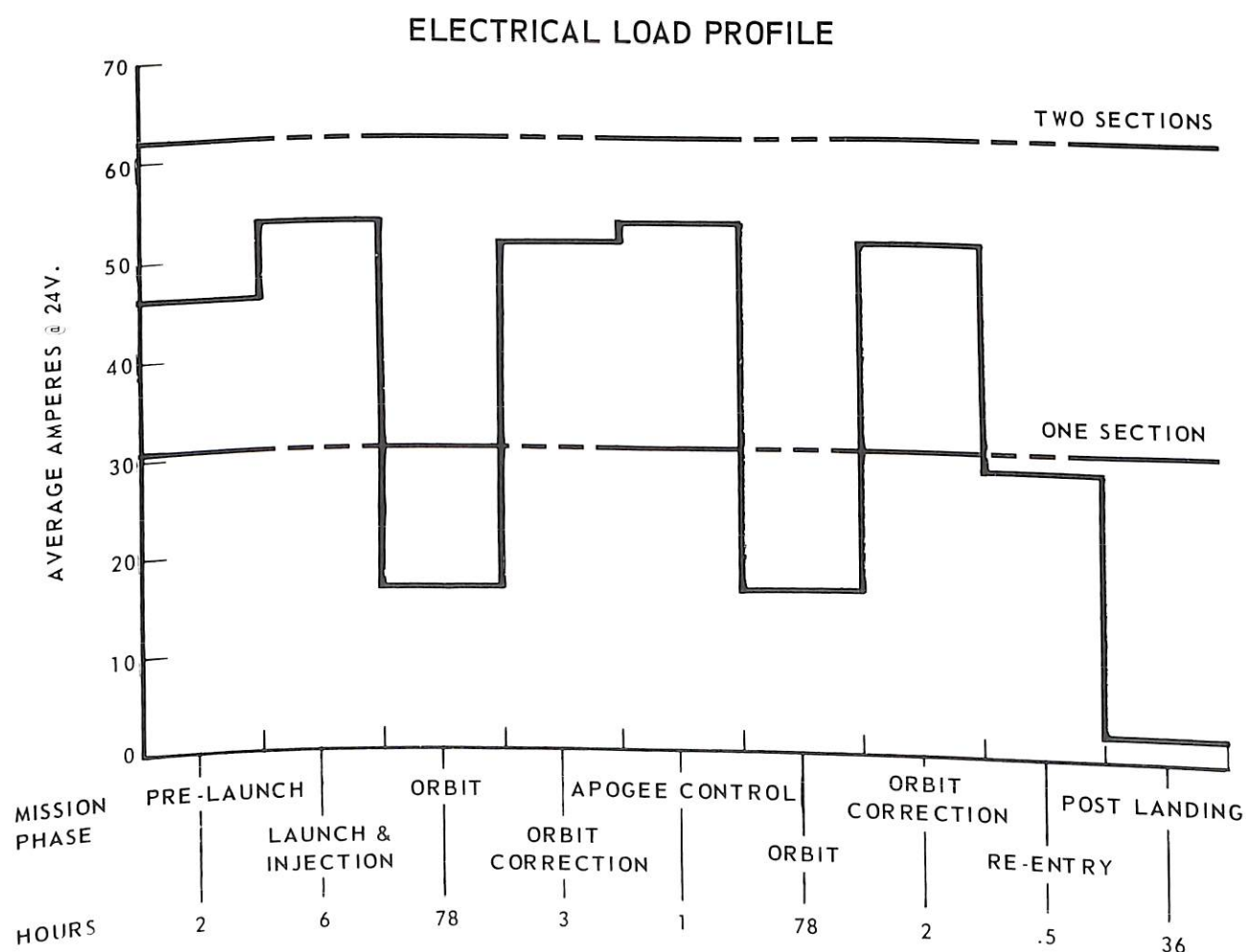


FIGURE 3.3-2

CONFIGURATION II

TABLE 3.3-2

ELECTRICAL LOAD SUMMARY

EQUIPMENT	SYSTEM TOTAL	PRELAUNCH	LAUNCH & INJECTION	ORBIT	ORBIT CORRECTION	APOGEE CONTROL	ORBIT	ORBIT CORRECTION		POST LANDING
	AMP.-HRS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.
COMMUNICATIONS	248.92	.80	5.02	.94	6.26	6.26	.94	6.26	2.78	1.36
ENVIRONMENTAL	1915.68	14.34	15.34	10.83	15.34	15.34	10.83	15.34	3.12	.85
GUIDANCE	413.21	22.96	22.96	1.09	23.01	23.01	1.09	23.01	17.80	
INSTRUMENTATION	404.77	3.68	3.49	2.32	2.93	2.93	2.32	2.93	3.48	
LIGHTING	66.90	.85	.82	.38	.38	.38	.38	.38	1.08	.02
MISCELLANEOUS	126.40	3.45	3.45	.57	3.48	3.48	.57	3.48	.44	
EXPERIMENTS	7.30			.03		1.46	.03			
TOTAL	3183.18	46.08	51.08	16.16	51.40	52.86	16.16	51.40	28.70	2.23

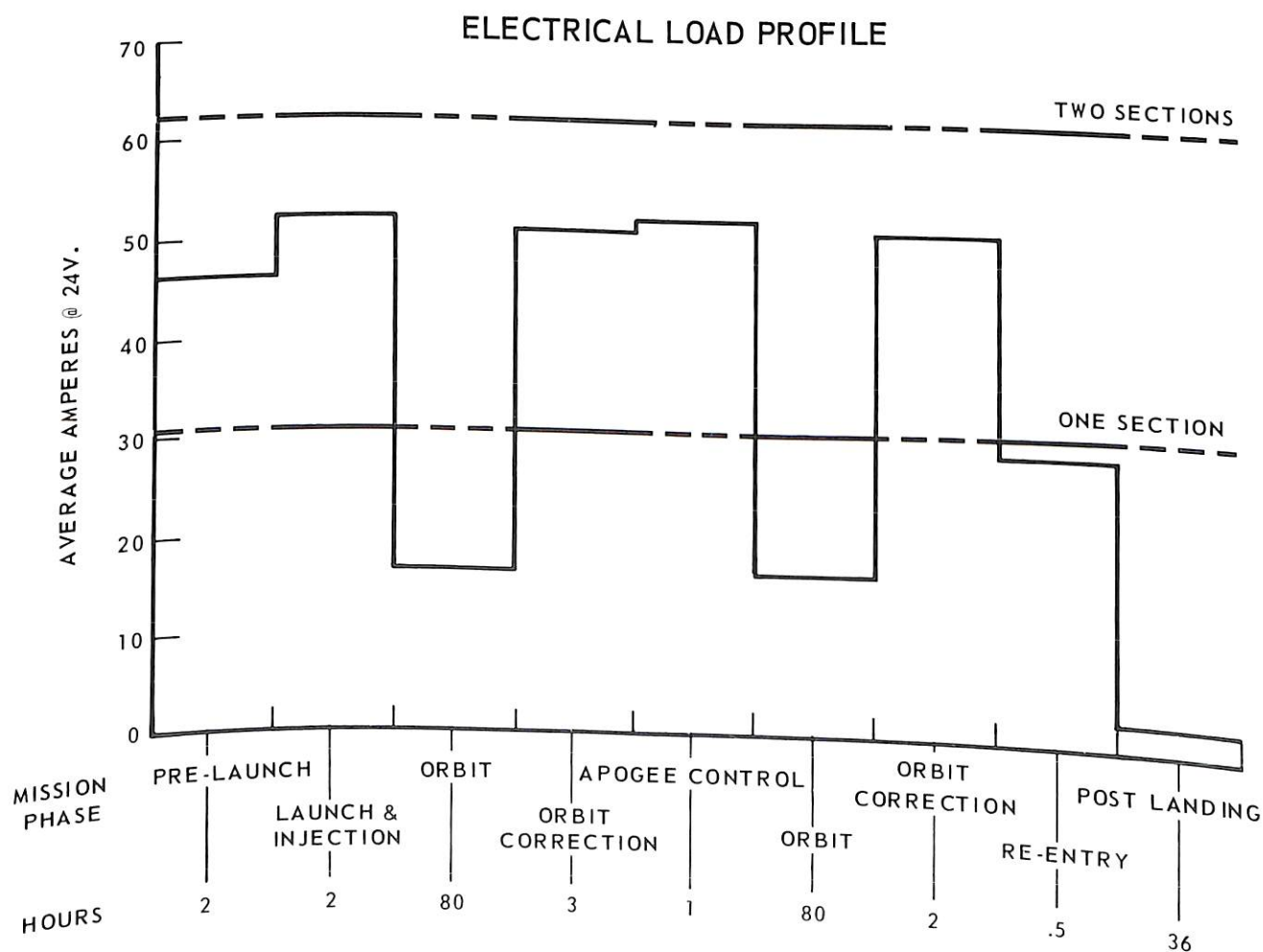


FIGURE 3.3-3

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3.3.1.2 (Continued)

The following tabulation is a comparison of normal Main Bus and Squib Bus load versus Fuel Cell (RSS) and Squib Battery capacity.

BUS	CAPACITY	LOAD	MARGIN	
	AMP.-HRS.	AMP.-HRS.	AMP.-HRS.	PER CENT
Main	3805	3335	470	14.1
Squib	27	20.2	6.9	34.3

Although no detailed load analysis is presented here, the Squib Bus load requirements are essentially the same as a seven (7) day Gemini "A" configuration.

3.3.1.3 Power Management - Control and monitoring of the power system is provided by switches and displays located on the instrument panel(s). The Bus Tie switches shown in Figure 3.3-1 allows the Main Bus system to be connected to either Squib Bus in the event of a Squib Bus failure.

Although the Re-entry Main Bus batteries are primarily used during the re-entry, landing and post-landing periods of the mission, the possibility of a powered flight abort requires that they be "on" during this time in order to assure Main Bus power continuity.

3.3.2 Configuration II

3.3.2.1 Description - The system is the same as described in Paragraph 3.3.1.1 and illustrated in Figure 3.3-1.

3.3.2.2 Utilization - The detailed system normal load requirements are tabulated in Table 3.3-2 and the associated load profile is presented in Figure 3.3-3. Emergency loads and profile appear in Table 3.3-4 and Figure 3.3-5, respectively.

The following tabulation is a comparison of normal Main Bus and Squib Bus load versus Fuel Cell (RSS) and Squib Battery capacity.

TABLE 3.3-3
EMERGENCY
ELECTRICAL LOAD SUMMARY
CONFIGURATION I

EQUIPMENT	SYSTEM TOTAL	ORBIT	ORBIT CORRECTION	ORBIT	ORBIT CORRECTION	RE-ENTRY	POST LANDING
	AMP-HRS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.
COMMUNICATIONS	145.37	.55	6.26	.55	6.26	2.78	1.36
ENVIRONMENTAL	1444.51	9.23	13.74	9.23	13.74	3.12	.85
GUIDANCE	167.01	1.09	1.59	1.09	1.59	.66	
INSTRUMENTATION	120.18	.75	3.02	.75	3.02	3.48	
LIGHTING	2.21	.02		.01		1.08	.02
MISCELLANEOUS	84.49	.52	3.45	.52	3.45	.44	
EXPERIMENTS							
TOTAL	1963.77	12.15	28.06	12.15	28.06	11.56	2.23

EMERGENCY
ELECTRICAL LOAD PROFILE
CONFIGURATION I

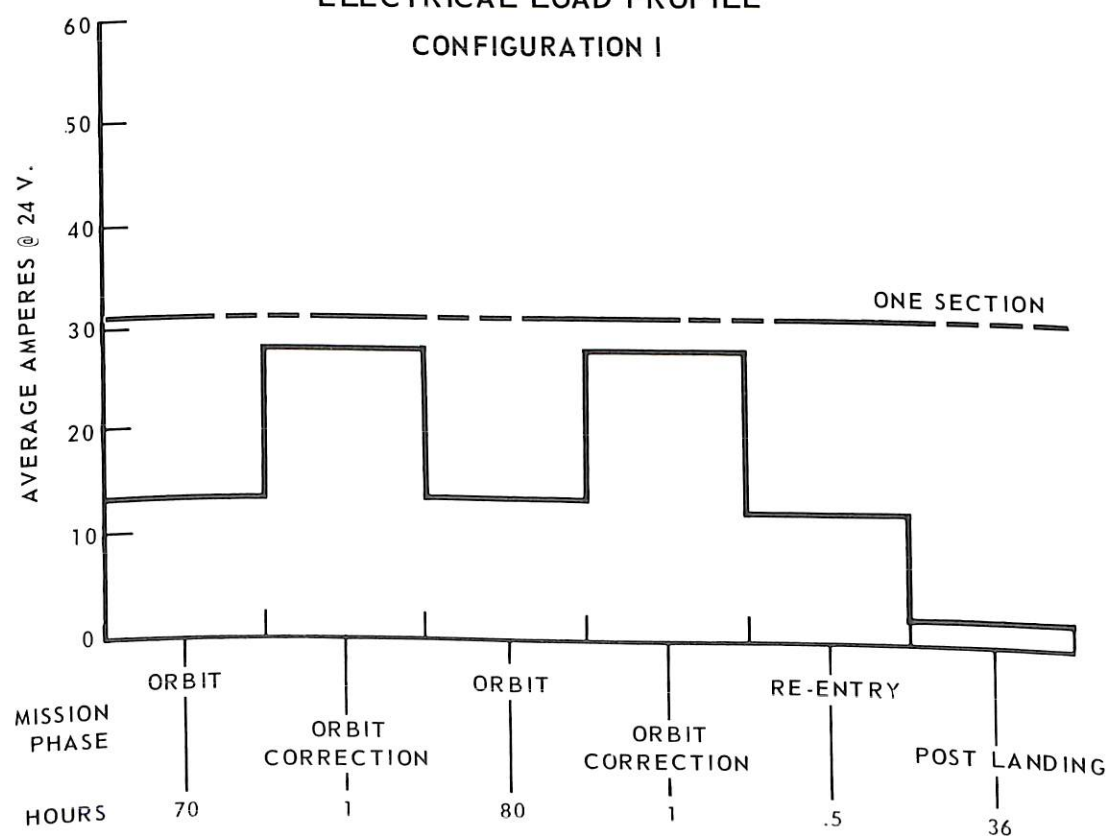


FIGURE 3.3-4

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TABLE 3.3-4
EMERGENCY
ELECTRICAL LOAD SUMMARY
CONFIGURATION II

EQUIPMENT	SYSTEM TOTAL	ORBIT	ORBIT CORRECTION	ORBIT	ORBIT CORRECTION	RE-ENTRY	POST LANDING
	AMP.-HRS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.	AMPS.
COMMUNICATIONS	148.01	.55	6.26	.55	6.26	2.78	1.36
ENVIRONMENTAL	1480.67	9.23	13.74	9.23	13.74	3.12	.85
GUIDANCE	171.37	1.09	1.59	1.09	1.59	.66	
INSTRUMENTATION	123.38	.75	3.02	.75	3.02	3.48	
LIGHTING	2.23	.01		.01		1.08	.02
MISCELLANEOUS EXPERIMENTS	86.59	.52	3.45	.52	3.45	.44	
	2012.25	12.15	28.06	12.15	28.06	11.56	2.23

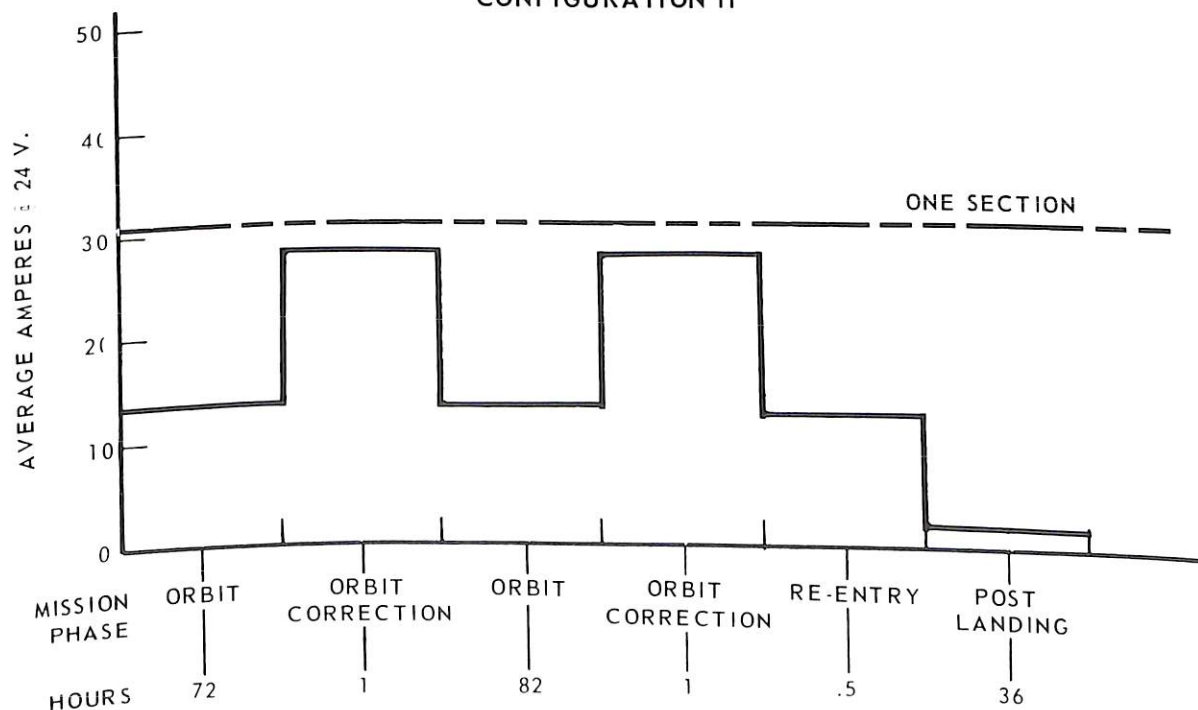
EMERGENCY
ELECTRICAL LOAD PROFILE
CONFIGURATION II

FIGURE 3.3-5

3.3.2.2 (Continued)

BUS	CAPACITY	LOAD	MARGIN	
	AMP.-HRS.	AMP.-HRS.	AMP.-HRS.	PER CENT
Main	3805	3183	622	19.6
Squib	27	20.2	6.9	34.3

Although no detailed load analysis is presented here, the Squib Bus load requirements are assumed to be essentially the same as a seven (7) day Gemini "A" configuration.

3.3.2.3 Power Management - The power management is the same as described in Paragraph 3.3.1.3 and illustrated in Figure 3.3-1.

3.3.3 System Selection - The recommendation to utilize three fuel cells instead of batteries, or batteries plus a fuel cell, is influenced strongly by the weight advantage. The comparative weights are presented in Table 3.3-5.

TABLE 3.3-5
WEIGHT COMPARISON OF ELECTRICAL POWER, PRIMARY OXYGEN AND
DRINKING WATER SYSTEMS

ITEM	WEIGHT - POUNDS		
	BATTERIES (6)	FUEL CELL (1) BATTERY BACKUP (5)	FUEL CELLS (3)
FUEL CELLS	-	69	207
FUEL CELL REACTANTS	-	50	136
REACTANTS TANKAGE	-	48	144
FUEL CELL PLUMBING, VALVES, CIRCUITRY & INSTALLATION	-	54	91
FUEL CELL MODULE STRUCTURE	-	36	56
FUEL CELL COOLANT LOOPS	-	-	23
400 AH BATTERIES & INSTALLATION	847	696	-
PRIMARY OXYGEN	90	45	45
PRIMARY OXYGEN SYSTEM & INSTALLATION	91	60	70
DRINKING WATER	106	56	30
DRINKING WATER SYSTEM & INSTALLATION	24	19	29
TOTAL	1,158	1,133	831

3.4 Communication System - The present Gemini Communications System is supplemented by a Unified S-Band System (USB) for ranges to 225,000 n.m. (all distances are approximate). This USB system is compatible with the Deep Space Instrumentation Facilities (DSIF) presently being implemented for the Apollo program. The USB system provides all spacecraft to earth communications (voice, data, commands, and television) for ranges in excess of 10,000 n.m. A circuit margin analysis in Table 3.4-1 shows that the carrier-to-noise ratio more than satisfies the minimum limit of eight db for both the up and down links. A block diagram of the complete communications system is depicted in Figure 3.4-1.

3.4.1 System Description - System utilization for the present Gemini system and the USB system for all phases of the mission is shown in Table 3.4-2.

The present Gemini system is modified for the near earth configuration. The HF and adapter C-band Beacon systems are deleted, the telemetry system is reduced to one transmitter, the VHF voice system is reduced to one transceiver and the Digital Command System is changed to accept inputs from either the UHF command receiver or the USB system. This reduction in the present Gemini system is considered reasonable in view of the short duration for the near earth phase and the USB system utilization as a backup. Post impact voice backup is provided by the GFE Recovery Voice T/R Beacon.

3.4.1.1 Unified S-Band System - The USB provides duplex voice and data links up and down utilizing the Apollo subcarrier frequencies, and is functionally identical with the one utilized in Apollo and described by NASA TN D-2208. System components are redundant transponders, power amplifiers, diplexers, control box, and antennas consisting of OMNI-directional flush mounted slots and an erectable 48-inch high gain paraboloid.

Transponder - The transponder, which is the heart of the system, consists of a solid state transmitter, receiver and power supply. The subcarrier discriminators and the down link subcarrier oscillators are not included in existing transponders;

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TABLE 3.4-1

UNIFIED S BAND CIRCUIT ANALYSIS

SPACECRAFT TO GROUND

FREQUENCY = 2295 MC RANGE = 225,000 N.M. RECEIVER BANDWIDTH = 8 MC

ITEMS	VALUE	TOLERANCE	SOURCE
1. TOTAL TRANSMITTED POWER (20 WATTS)	+13.0 dbw	± 1.0 db	ASSUMED
2. SPACECRAFT CIRCUIT LOSS	-3.0 db	± 0.5 db	ESTIMATED
3. SPACECRAFT ANTENNA GAIN (4 FT. DIA. PARABOLOID)	+25.5 db	± 0.5 db	CALCULATED
4. SPACECRAFT ANTENNA POINTING LOSS	-1.0 db	± 0.5 db	ESTIMATED
5. FREE SPACE LOSS AT 2295 MC	-211.9 db	-	ITT HANDBOOK
6. GROUND ANTENNA GAIN (85 FT. DIA.)	52.8 db	± 1.0 db	JPL MEMO 33-83
7. GROUND RECEIVER CIRCUIT LOSS	-	-	CONTAINED IN ANTENNA GAIN
8. NET LOSSES	-137.6 db	± 1.32 db (RMS)	CALCULATED
9. TOTAL RECEIVED POWER	-124.6 dbw	± 1.66 db (RMS)	CALCULATED
10. RECEIVER NOISE POWER EFFECTIVE SYSTEM TEMP. = 330°K $P_N = K T_e \Delta f = 1.38 \times 10^{-23} (330^\circ K) 8 \times 10^6$	-134.4 dbw	± 1.0 db	JPL EPD-29
11. RF SIGNAL TO NOISE IN A 8 MC RF BANDWIDTH	+9.8 db	± 1.94 db (RMS)	CALCULATED
12. SIGNAL TO NOISE OUT OF THE FIRST DE- TECTOR. USING THE BROAD BAND FM ANALYSIS WITH LIMITING.	+19.5 db		TECH. REPORT 38 HARVARD UNIV. D. MIDDLETON
13. SIGNAL TO NOISE INPUT TO THE FM VOICE DISCRIMINATOR	30 db	± 3.0 db	CALCULATED
14. SIGNAL TO NOISE INPUT TO THE DATA DETECTOR	28.7 db	± 3.0 db	CALCULATED
15. SIGNAL TO NOISE TO THE RANGING FILTER INPUT	25 db	± 3.0 db	CALCULATED

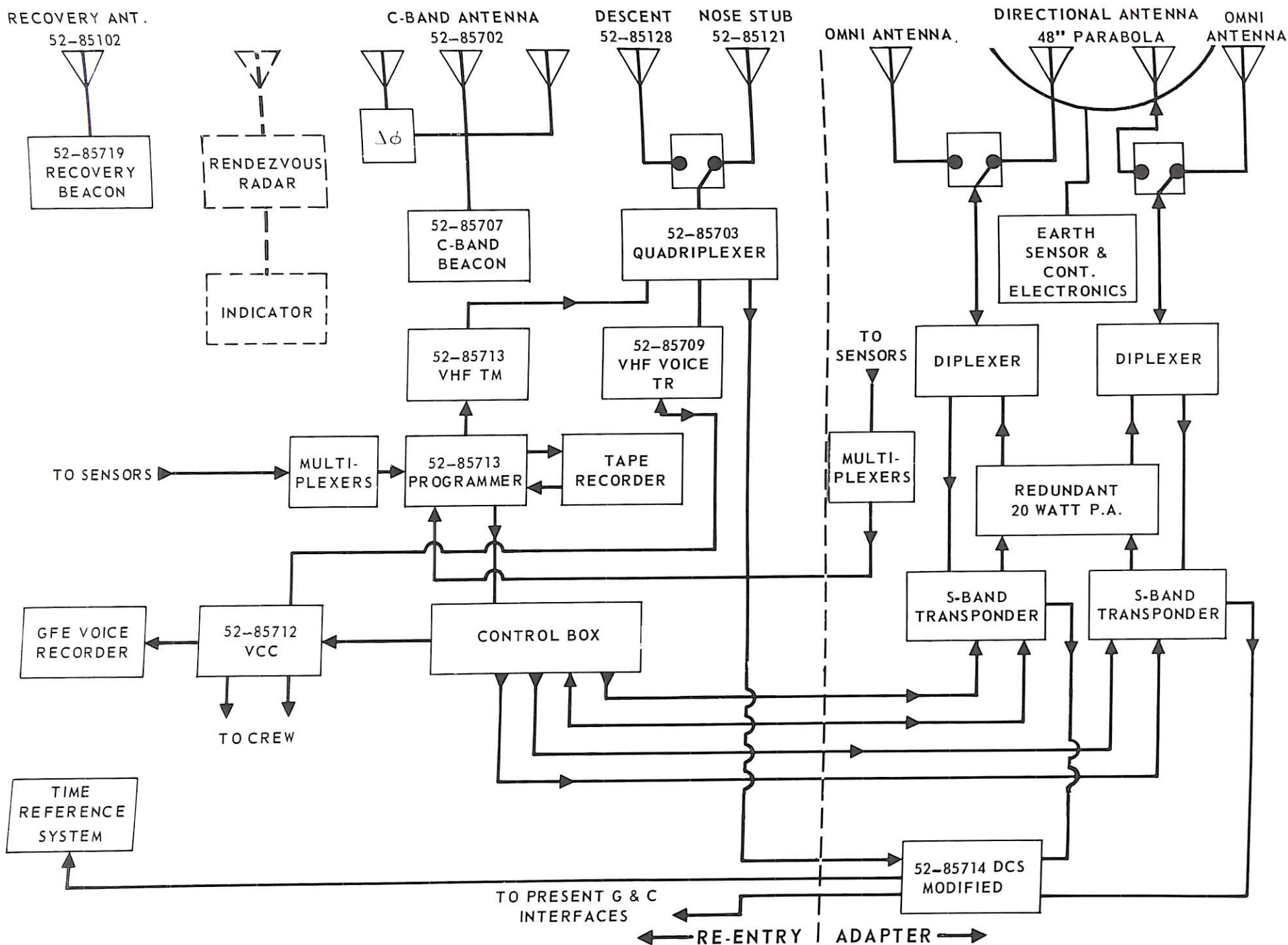
GROUND TO SPACECRAFT

RECEIVED FREQUENCY = 2106 MC RECEIVER BANDWIDTH = 6 MC

ITEMS	VALUE	TOLERANCE	SOURCE
1. TOTAL TRANSMITTED POWER (10 KW)	40 dbw	± 1.0 db	JPL EPD-29
2. GROUND ANTENNA CIRCUIT LOSS (INCLUDED IN ANTENNA GAIN)	-		JPL EPD-29
3. GROUND ANTENNA GAIN (85 FT. DIA.)	52 db	± 1.0 db	JPL MEMO 33-83
4. GROUND ANTENNA POINTING LOSS	-1.0 db	± 0.5 db	ESTIMATED
5. FREE SPACE LOSS AT 2106 MC	-211.3 db	-	ITT HANDBOOK
6. SPACECRAFT ANTENNA GAIN (4 FT. DIA. PARABOLOID)	+25.5 db	± 0.5 db	CALCULATED
7. SPACECRAFT RECEIVER CIRCUIT LOSSES	-3.0 db	± 0.5 db	ESTIMATED
8. NET LOSSES	-137.8 db	± 1.58 db (RMS)	CALCULATED
9. TOTAL RECEIVED POWER	-97.8 dbw	± 1.66 db (RMS)	CALCULATED
10. RECEIVER NOISE POWER EFFECTIVE SYSTEM TEMP. = 3935°K	-125 dbw	± 1.0 db	MISHA SCHWARTZ "INFORMATION TRANSMISSION, MODULATION AND NOISE"
11. RF SIGNAL TO NOISE IN A 6 MC BANDWIDTH	+27.2 db	± 1.94 db (RMS)	
12. SIGNAL TO NOISE OUT OF THE FIRST DETECTOR	30 db		TECH. REPORT 38 HARVARD UNIV. D. MIDDLETON

BLOCK DIAGRAM TELECOMMUNICATIONS

GEMINI - LARGE EARTH ORBIT



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FIGURE 3.4-1

3-35

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VOLUME 1
19 JUNE 1965

TABLE 3.4-2
TELECOMMUNICATIONS SUBSYSTEM UTILIZATION

GEMINI - LARGE EARTH ORBIT

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	PRE-LAUNCH	LAUNCH	NEAR EARTH	DEEP SPACE	RE-ENTRY	RECOVERY	REMARKS
FUNCTION		0 TO 100 MILES	100 TO 10K MILES	2K MILES TO APOGEE AND BACK TO 2K MI.	80 MI. TO IMPACT		
I. VOICE a) INTERPHONE b) VHF c) DSIF	←→	←→	←→	←→	←→	←→	GND. INT. DURING PRELAUNCH 1. DUPLEX VOICE 2. REDUNDANT CAPABILITY
II. DOWN-DATA a) VHF TELEMETRY b) DSIF	←→	←→	←→	←→	←→		REDUNDANT CAPABILITY
III. UP-DATA a) UHF COMMAND b) DSIF	←→	←→	←→	←→	←→		REDUNDANT CAPABILITY
IV. TRACKING C-BAND DSIF RANGING	←→	←→	←→	←→	←→		REDUNDANT CAPABILITY
V. RECOVERY VHF BEACON	←→	←→				←→	
VI. TIME REFERENCE SYSTEM	←→	←→	←→	←→	←→	←→	
MISSION TIME		0 → 6 MIN.	6 → 60 MIN 147 → 148 HRS.	.5 → 147.5 HR.	148 TO IMPACT	IMPACT + 36 HR.	

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3.4.1.1 (Continued)

therefore, they are incorporated in the control box.

The receiver can simultaneously receive up data (commands), voice and the ranging code. The receiver is phase locked to the received carrier and provides the frequency and phase reference for the transmitter exciter. Phase lock is required for accurate tracking during deep space operations. The phase modulated up link data is fed to the appropriate subcarrier discriminators and filters.

The ranging information, which is a code developed by Jet Propulsion Laboratory, is then filtered out and applied to the transmitter phase modulator.

The voice and command subcarriers are sent to the control box where they are detected, amplified and fed to the VCC and DCS, respectively.

The transmitter will simultaneously transmit down voice, PCM data and either the ranging code or television. The transmitter consists of a wide band phase modulator and frequency multipliers with a power output of 250 milliwatts. The composite signal of the FM voice and PM down-link PCM data subcarriers is summed with the received ranging signal and then applied to the wide band phase modulator. The transmitter output is then applied to the power amplifier.

The Apollo transponders include power supplies that convert 115-volt, 3-phase, 400-cycle into the necessary D.C. voltages for the receiver and transmitter. The Mariner transponder has a supply which converts 50-volt, 2400-cycle into the desired D.C. voltages for proper operation of the transmitter and receiver.

Presently available transponders used on Apollo and the Mariner programs are considered acceptable for this program. Electrical interfacing is performed outside the transponder, thereby retaining the demonstrated capability of the existing off the shelf transponders.

3.4.1.1 (Continued)

Power Amplifier - The power amplifier provides approximately 20 db gain using an amplatron tube which has been developed for the Apollo Lunar Excursion Module. The amplatron was selected over the Traveling Wave Tube because of its higher efficiency and feed-through characteristic.

Control Box - The Control Box consists of Voltage Controlled Oscillators (VCO), discriminators, mixing networks and speech clippers. The received subcarriers from the transponder are discriminated and the information is adjusted for level and impedance to be compatible with the Voice Control Center and Digital Command System. The audio for transmission is clipped and frequency modulated on a 1.25 Mc carrier and the PCM data is bi-phase modulated on a 1.024 Mc carrier. Both signals are then linearly mixed prior to being applied to the wide band phase modulator in the transponder. The Control Box also includes the switching circuitry for selection of transponders and power amplifiers along with the necessary indicators for phase lock, antenna earth alignment and antenna gimbal angles.

Diplexer - The redundant diplexers provide isolation of the transmit and receive frequencies for each transponder. The diplexer is a passive resonant filter. This diplexer is currently being utilized in the Apollo Block I equipment.

Antennas - The antennas employed with the USB system are the OMNI directional elements for coverage to approximately 35,000 n.m. and the 48-inch diameter paraboloid for coverage to 225,000 n.m. The paraboloid includes two feeds each, driven by an independent source. This antenna provides approximately 25 db gain and a beam width of eight degrees. The antenna includes an earth sensor whose outputs provide error signals to the antenna control electronics which, in turn, excite the antenna drive motors. The OMNI antenna is identical to the Gemini S-band slot antenna except the impedance matching transformer is changed. The 48-inch paraboloid is identical to

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3.4.1.1 (Continued)

the Ranger 48-inch diameter dish except the L-band feed assembly is replaced with the S-band crossed dipoles and the gimbals modified to be compatible with the Gemini installation.

3.4.1.2 Radar - The Configuration I spacecraft includes a modified Gemini rendezvous radar. This modification consists of eliminating the angle track feature to reduce the weight approximately 25 pounds.

3.5 Data Acquisition System

3.5.1 Summary - The Gemini "A" Data Acquisition System (DAS) requires only sensor and signal conditioner selection for use on the large earth orbit mission. The system is shown in Figure 3.5-1 and provides the following functions:

- A. Sensors and signal conditioners for display to the crew of those parameters not provided by other systems.
- B. Transmission to the ground stations of those parameters required for monitoring mission progress, assessing spacecraft status, and for making decisions assigned to the ground station personnel.
- C. Documentation of those portions of the mission not monitored in real time through on-board recording and later playback to the ground stations.
- D. Continuous records of biomedical parameters through on-board recording.
- E. Sensors for AGE readout and prelaunch verification of systems when not provided as part of other systems.
- F. Monitoring of parameters required for post-mission evaluation of systems operation.
- G. Recording and transmission of parameters from scientific experiments.
- H. Photographic equipment for lunar observation and mission documentation.

3.5.2 Operation and Control - The DAS operates continuously from prelaunch through landing. Signals from sensors or signal conditioners, or directly from other systems, are applied to the PCM system. The PCM master frame contains 338 channels which are combined as a 51.2K bits per second (bps) non-return-to-zero (NRZ) PCM pulse train provided as an output to the transmitters. The prime subframe, containing 317 of these data channels, is also assembled as a 5.12K bps return-to-zero (RZ) PCM pulse train and recorded continuously. The recorded

DATA ACQUISITION SYSTEM

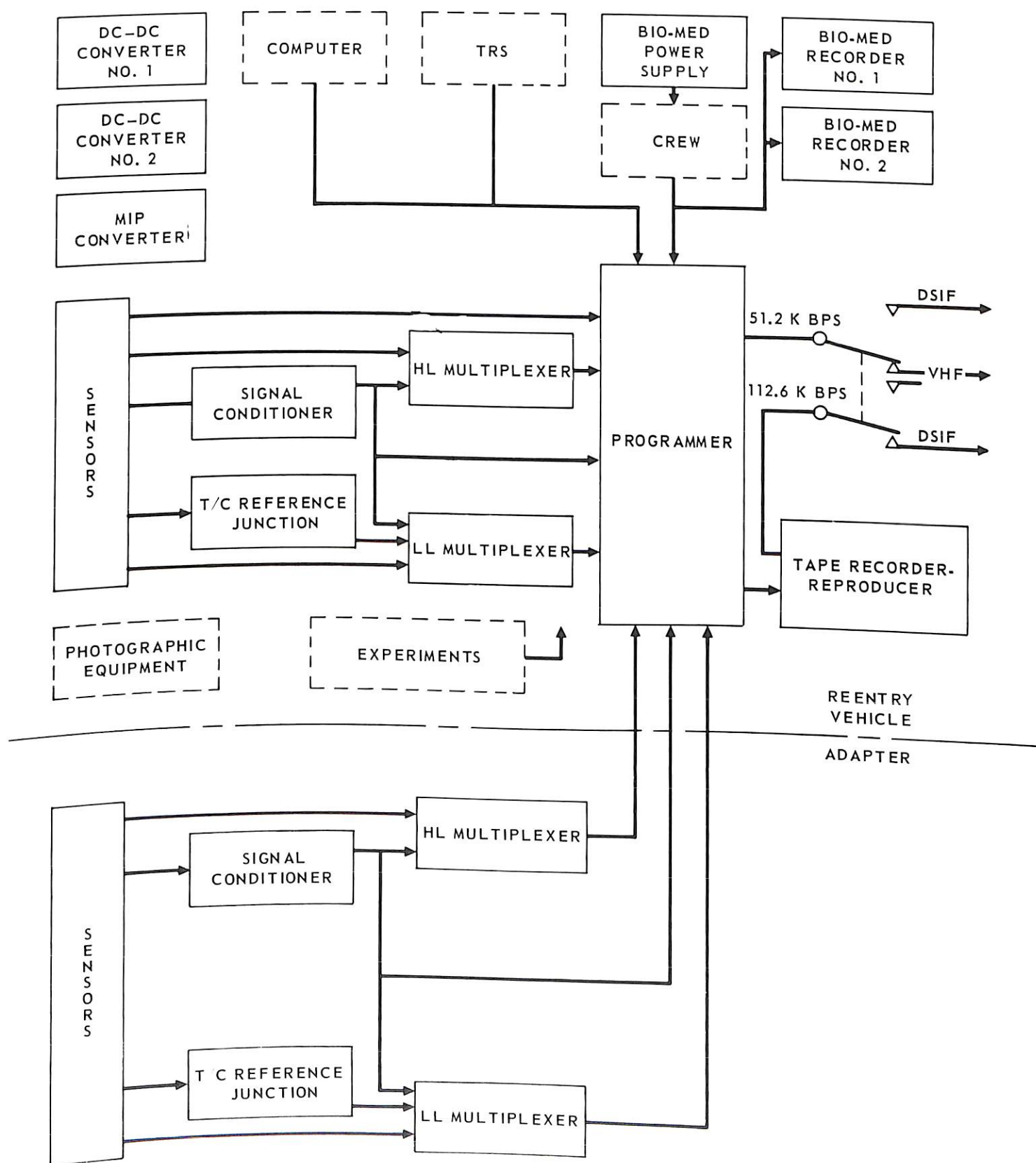


FIGURE 3.5-1

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3.5.2 (Continued)

data is played back on command at 22 times record speed providing a 112.6K bps NRZ PCM pulse train output to the transmitters.

Two data modes are used. In Mode 1 the master frame PCM pulse train is transmitted over VHF during periods of contact with ground stations, while the prime subframe is continuously recorded on board or played back on command over DSIF communications during periods of contact. The second mode provides transmission of the master frame over DSIF during periods of contact with continuous recording of the prime subframe. Playback is accomplished by substituting the recorder output for the master frame and operating DSIF in a data-only mode.

3.6 Structure

3.6.1 Structural Design Criteria

- A. Objective - These criteria define the assumptions used in the design of a minimum change spacecraft, capable of a large earth orbit mission with a re-entry inertial velocity of 36,400 fps. Modifications to the Gemini design are minimized to utilize, to the fullest extent, existing qualified hardware. In pursuance of this objective, no changes are required for the primary loads from the launch, abort and re-entry flight phases, with appropriate limitations being applied to attain this goal. The primary structure is designed for the loads given in Reference 3.6-1, and the structural integrity is verified in the Structural Analysis Reports, References 3.6-2, 3.6-3, and 3.6-4, using the safety factors defined in the Structural Design Criteria, Reference 3.6-5.

Where structural changes are required, the structural design criteria for Configurations I and II adhere, if applicable, to the criteria of Reference 3.6-5.

- B. Configuration Description - The spacecraft structure is essentially the same as Gemini "A" except for modifications in the following areas:
1. Thermal protection on both forebody and afterbody.
 2. Rendezvous and Recovery Section.
 3. Adapter equipment mounting.
 4. Target Docking Adapter dampers (Target Vehicle).
- C. General Environment - The natural and vehicle induced environments for the spacecraft associated with wind velocity, wind shear, gusts, acoustic noise, vibration, shock and acceleration are unchanged from the Gemini environments and are defined in References 3.6-5 and 3.6-6. A preliminary investigation

3.6.1 (Continued)

of the combination of all environments associated with a Titan IIIC launch indicates that they are no more severe than the design environment of the Gemini "A".

- D. Design Weights - The spacecraft structural design weights are given in Table 3.6-1.

TABLE 3.6-1
STRUCTURAL DESIGN WEIGHTS

MISSION PHASE	DESIGN WEIGHT POUNDS
LAUNCH	7500-8500
RENDEZVOUS	8300
RE-ENTRY	5050
PILOT CHUTE DEPLOYMENT	4930
MAIN CHUTE DEPLOYMENT	4500
LANDING	4400
RECOVERY	6600

3.6.1 (Continued)

E. Flight Phases

1. Launch - The combination of all launch loadings, separately and in combination with launch heating, does not result in strength requirements which are greater than the requirements used for the original design of the Gemini spacecraft.
2. Orbit - The docking criteria is defined in Reference 3.6-5. The combination of all loadings during maneuvering in orbit does not result in strength requirements which are greater than the requirements used for the original design of the Gemini spacecraft.
3. Re-entry - The re-entry corridor for the normal mission is bounded by a load factor of 10 g's and a skip limit of 400,000 feet. (Two of the most severe re-entry trajectories on these boundaries were studied in sufficient detail to select the external thermal protection and to verify the strength of the substructure).

Re-entries from aborted missions up to low earth orbit inertial velocities (approximately 25,600 fps) do not result in load factors exceeding 16 1/2 g's. No special thermal limitations exist for these aborted missions.

Re-entries from aborted missions between low earth orbit velocity and escape velocity do not result in conditions which are more severe than either of the aforementioned re-entry cases.

4. Abort and Ejection - The combination of all loadings during abort and ejection does not result in strength requirements which are greater than the requirements used for the original design of the Gemini spacecraft.

5. Parachute Deployment and Landing - The parachute deployment and inversion loading does not result in strength requirements which are greater than the requirements used for the original design of the Gemini spacecraft.

The spacecraft is designed for a water landing only, followed by a maximum flotation period of 36 hours. The criteria for determining water landing loads are based on a rational combination of all landing parameters associated with parachute swing, local wind velocity, wave slope, and sinking speed.

3.6.2 Description of Changes

- A. Thermal Protection - The afterbody thermal protection for the spacecraft consists of a combination of columbium radiation shingles and titanium supporting ablation shingles. Columbium (Cb-752) is utilized on the upper half of the conical section and ablation protection on the remainder of the afterbody. The basis for selection of these materials is graphically shown in Figure 3.6-1.

The columbium shingles are formed with the same dies that are used for the Rene' 41 shingles of the present design. The thickness is .016 inch, coated for oxidation protection to a thickness of .019 inch. The thermal coefficient of expansion for columbium is about one-half of that for Rene' 41; therefore, it is satisfactory to maintain the same expansion allowance and shingle size as the Gemini Rene' 41 shingles. Columbium washers and L-605 bolts are used to attach the shingles to the substructure.

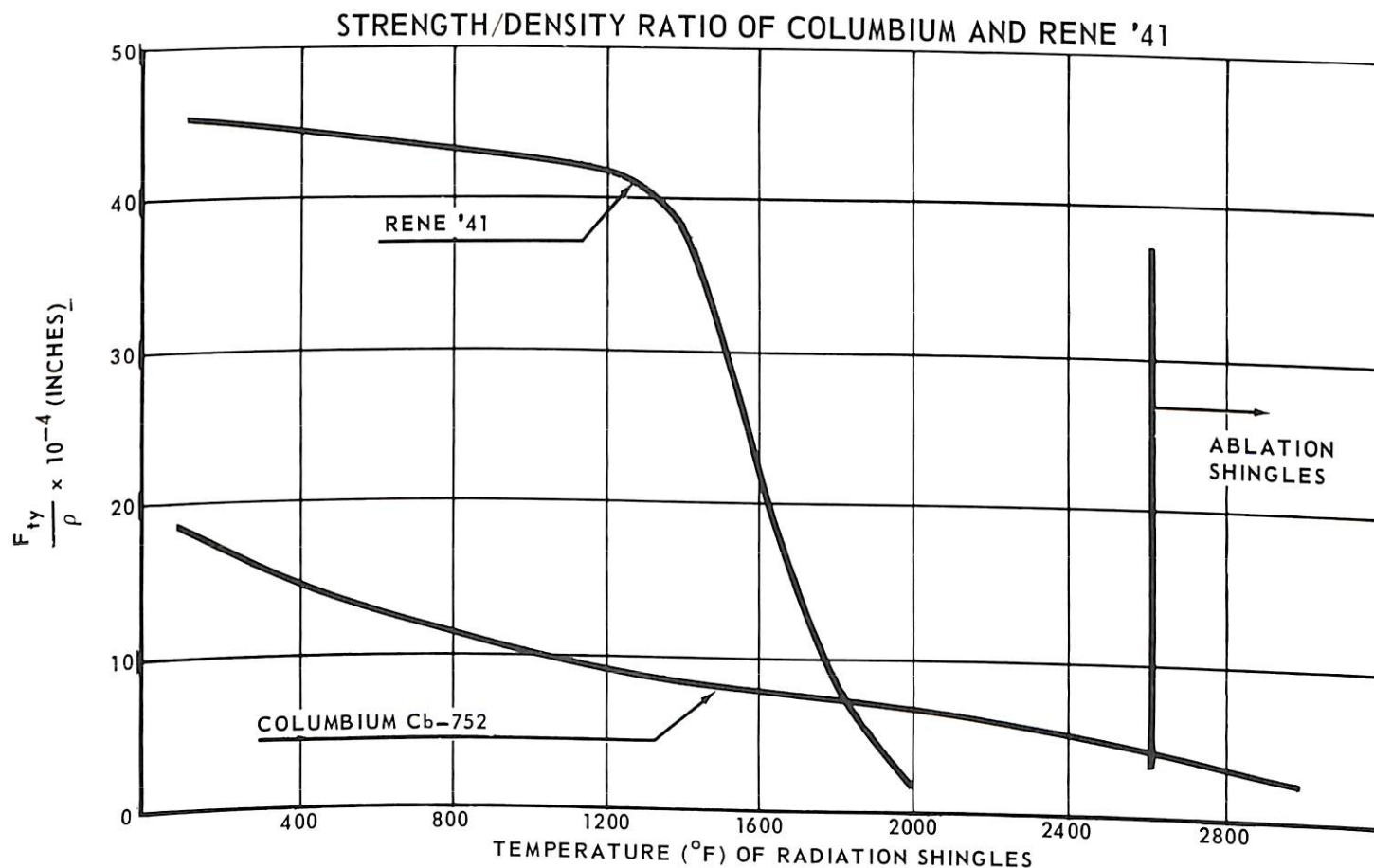


FIGURE 3.6-1

Extensive experience has been gained in the fabrication of columbium structure on the Refractory Metals Structural Development, ASSET and BGRV programs. The coated columbium oxidation protection was successfully tested with heat lamps on the Refractory Metals program to temperatures above 2000°F for over six minutes. The requirement for this mission is approximately one minute above 2000°F. A columbium heat protection system was successfully utilized on the ASSET program.

3.6.2 (Continued)

Columbium (Cb-752) is selected because of its superior strength and because it is easily formed, riveted and welded. Procurement and processing is accomplished by the following steps:

1. Purchase per M.A.C. Material Specification 183.
2. Cold form to shingle configuration.
3. Pickle per M.A.C. P.S. 13154.
4. Spot weld, if required, per M.A.C. P.S. 22142.
5. Lb2 coat for oxidation protection per M.A.C. P.S. 13156.

The ablation shingles consist of a titanium structure with the outside surface covered with an ablative material. The ablator is non-structural and consists of a layer of fiberglass honeycomb filled with a silicone elastomer. The honeycomb is used to contain the ablation material and retain the char layer during the re-entry phase of the mission. The supporting structure for the ablation material consists of two .010 inch thick commercially pure titanium panels, one beaded and one flat, spot welded together. The beaded panel is formed with the same dies that are used for the Rene' 41 shingles of the present Gemini design. To attach the shingles to the substructure, titanium washers and L-605 bolts are installed through counterbored holes in the ablation material. The holes are later filled with a similar ablator after bolt installation and inspection.

3.6.2 (Continued)

Two re-entry trajectories on the boundary, including shingle temperatures in the most severely heated areas, are presented in Figures 3.6-2 and 3.6-3.

The forebody thermal protection is similar to the present Gemini design. The bondline temperature at landing is 580°F on the lower half of the ablation shield graded to 700°F on the top centerline.

- B. Rendezvous and Recovery Section - The Rendezvous and Recovery Section has a length which is in excess of the requirements for this mission. The redesign of the radar and repackaging of the main parachute permits shortening of this section about 20 inches with a significant weight saving.

The structure is similar to the Gemini design, being of semi-monocoque titanium construction, covered, in this case, with ablative shingles. The design loads are identical to those shown in Reference 3.6-4.

- C. Adapter Equipment Mounting - Mission requirements dictate considerable rearrangement of equipment within the Adapter Equipment Section. The existing support points are utilized, where possible, and new supports added where necessary. The Gemini blast shield is removed to effect a weight saving. The design loadings are the natural and vehicle induced environments of References 3.6-5 and 3.6-6.
- D. TDA Dampers - Docking is accomplished with a heavier target vehicle; therefore, the damper loads increase during the docking maneuver. The energy transfer across the interface increases about 50 per cent. The internal damper orificing requires redevelopment, but no structural changes to the dampers are anticipated. In addition, there are some minor increases in the strength of the backup structure for the dampers.

MAXIMUM RANGE RE-ENTRY TRAJECTORY
(3838 N.M.)

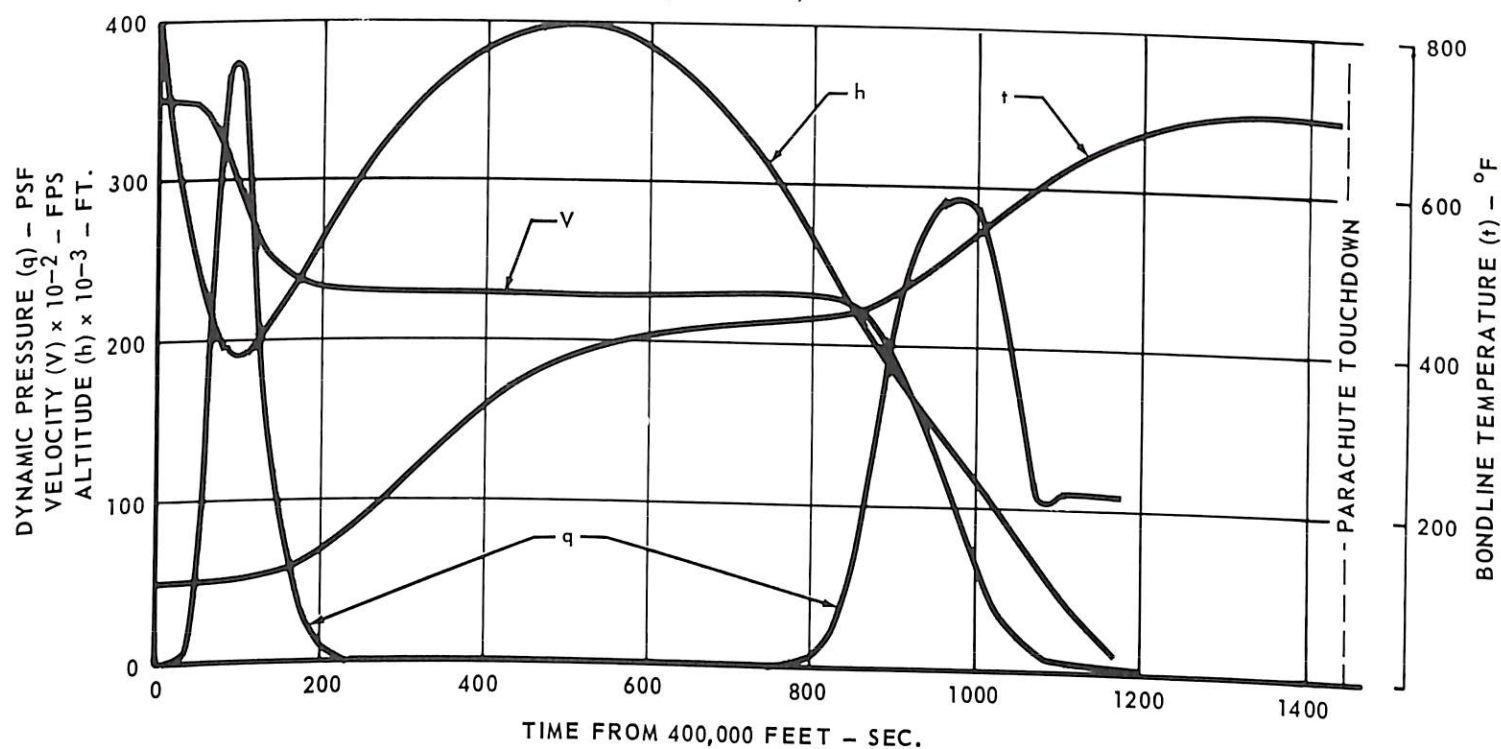


FIGURE 3.6-2

MINIMUM RANGE RE-ENTRY TRAJECTORY
(700 N.M.)

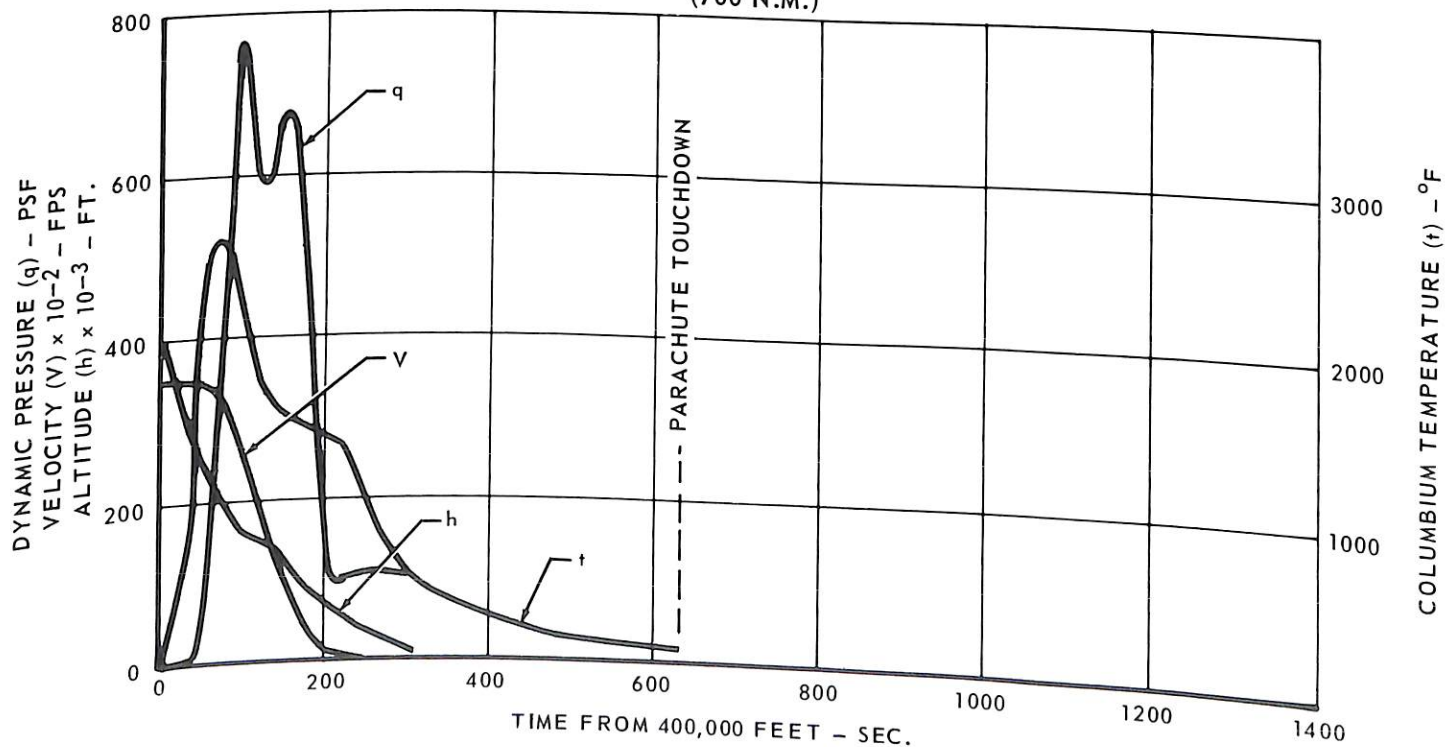


FIGURE 3.6-3

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SUPPORTING STUDIES AND INFORMATION

3.6.3 Additional Structure - An adapter is necessary to mate the MAC designed TDA to the modified Transtage for Configuration I. Some equipment is located in this adapter and an interface connection is accomplished at docking. The adapter is comprised of a semi-monocoque structure and is designed to be equivalent in strength to the TDA. The natural and vehicle induced environments associated with a Titan IIIC launch are also used in the design of this adapter.

3.7 Heat Protection

3.7.1 Launch - Structural temperatures are determined for critical areas using the Eckert reference temperature method of Reference 3.7-1. Transition Reynolds numbers are obtained by correlations obtained from GT-1 flight data. Since the basic shape of the spacecraft is similar to the Gemini shape, the heating rate calculation methods and constants developed during the Gemini project are used to compute local design temperatures.

3.7.2 Space Flight - External heat loads are determined from heat inputs based on solar, earth and moon reflected and emitted radiation, spacecraft orientation, surface emissivity and solar absorptivity. In the near earth orbit (110 n.m. or less), free molecular heating is applied. The type of heating used during the applicable phase of the mission is shown in Table 3.7-1.

TABLE 3.7-1
TYPE OF HEATING VS MISSION PHASE

TYPE OF HEATING	PARKING ORBIT OR RENDEZVOUS	MIDCOURSE	LUNAR VICINITY
DIRECT SOLAR	X	X	X
EARTH REFLECTED	X		
EARTH EMITTED	X		
MOON REFLECTED			X
MOON EMITTED			X
FREE MOLECULAR	X		

3.7.2 (Continued)

To minimize the temperature differences on the surface, the spacecraft is rotated about its Z axis at a rate of approximately 0.6 deg./sec. The spacecraft is oriented such that the Z axis makes an angle of between 45 and 70 degrees with the sun's rays in such a way that the adapter thermal curtain receives solar heat. This spacecraft orientation, combined with the rotation about its axis and proper surface coatings, reduces the need for active temperature control devices (heaters, cooling loops and evaporation heat exchangers) to a minimum. Cooling for the crew and electronic equipment is provided by a space radiator. The cooling fluid is circulated by pumps through suit heat exchangers, electronic equipment and battery cold plates, fuel cell and space radiator as discussed in Paragraph 3.2. Typical radiator skin and fluid temperatures during the parking orbit or rendezvous phase of the mission are shown in Figure 3.7-1. Corresponding temperatures during the midcourse phase of the mission with the spacecraft rotating are shown in Figure 3.7-2.

RADIATOR TEMPERATURES VS TIME

- SPACECRAFT FLYING SMALL END FORWARD PITCHED 10° DOWN
- 100 N.M. CIRCULAR ORBIT
- 0° ORBIT INCLINATION
- NET HEAT REJECTION = 4220 BTU/HR.

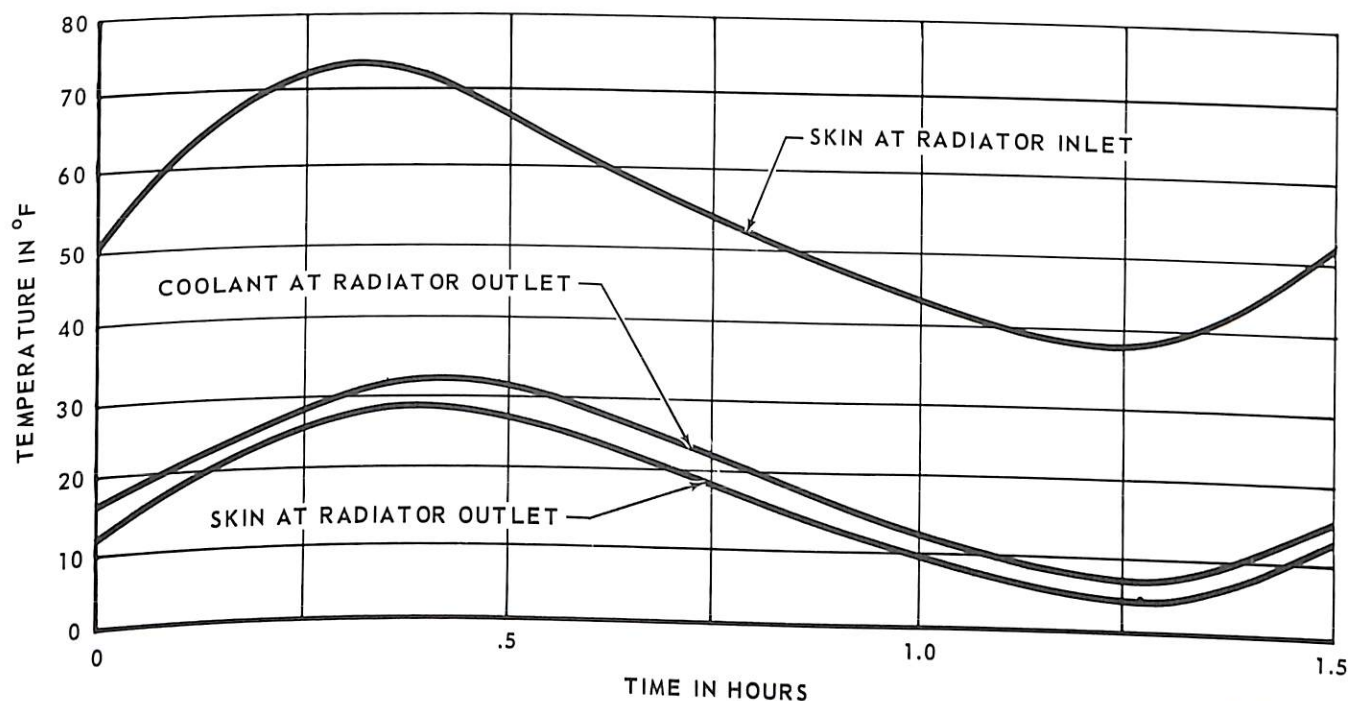


FIGURE 3.7-1

- 0.6 DEG. PER SECOND ROLL ABOUT SPACECRAFT
Z AXIS AT 45° TO VEHICLE - SUN LINE
(AFT END EXPOSED TO SUN)
- NET HEAT REJECTION = 2150 BTU/HR.

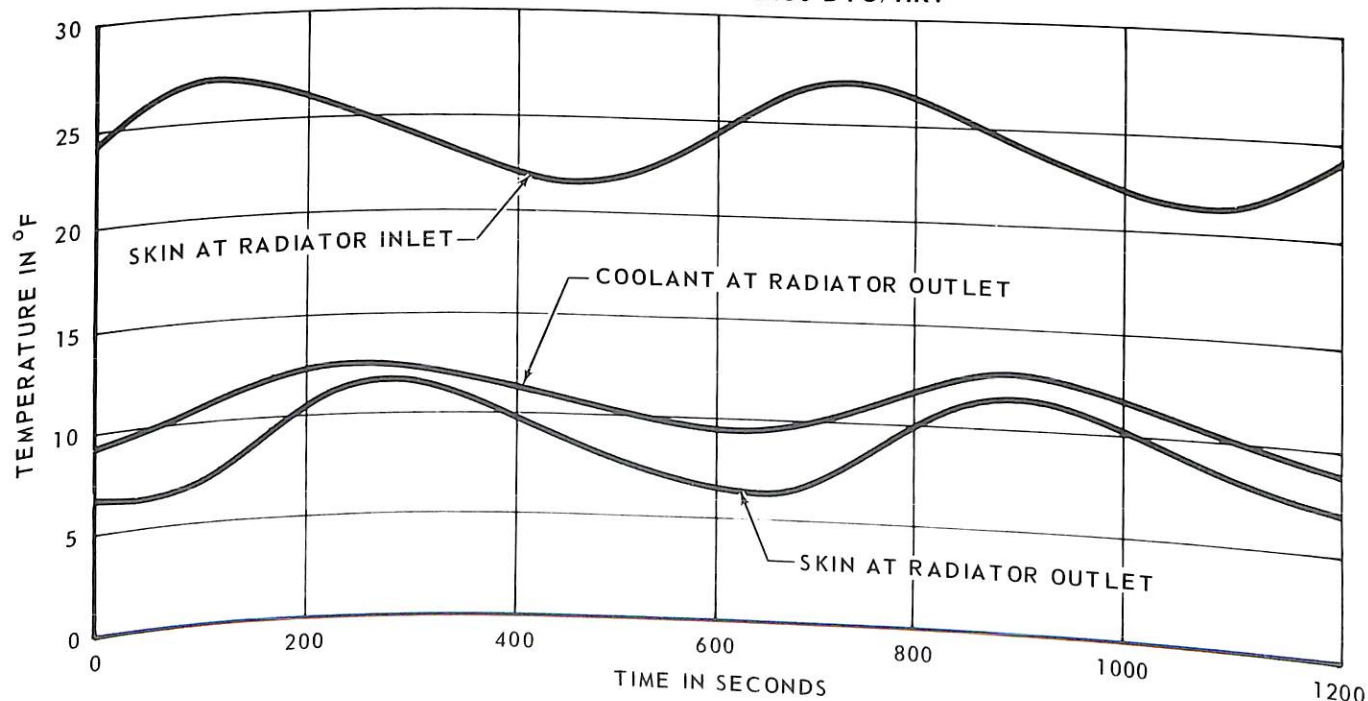


FIGURE 3.7-2

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3.7.3 Re-entry Heat Protection

3.7.3.1 Introduction - A study of ablation heat protection requirements considered two increments of maximum range; namely, 5,300 n.m. and 3,838 n.m. These studies indicate that reducing the maximum range from 5,300 to 3,838 n.m. reduces the heat shield ablation material weight by 25 per cent, and the afterbody ablation material weight by 60 per cent. An additional weight saving is achieved by changing the ablation material on the afterbody from the Gemini heat shield ablator (S-3) to a lower density ablator (S-6). Characteristics of S-6 are shown in Figure 3.7-3. The effects of range on heat protection requirements are shown in Figure 3.7-4. The spacecraft are designed for a maximum re-entry range of 3,838 n.m. and a L/D of 0.15. Note, however, that for the re-entry conditions defined for these configurations, the dispersed maximum range will not exceed 2,600 n.m. This design approach then provides heat protection margin. The front face heat shield ablation material is S-3. The windward half of the cabin conical section, re-entry control section, and the R & R section are covered with S-6. The leeward half of the cabin conical section is covered with columbium shingles. Detailed studies of the window, the MDF ring, and hat section structure indicate that the proposed heat protection maintains acceptable temperatures for internal structural components.

3.7.3.2 Front Face Heating - Convection heat fluxes are computed for the hemispherical stagnation point using Reference 3.7-2, and corrected for the Gemini shape with Reference 3.7-3. Convective heating from a dissociated and partly ionized gas was considered (Reference 3-4), however, the numerical values were similar to Reference 3.7-2 for velocities up to 36,000 fps.

The convective heating distribution, shown in Figure 3.7-5 for the Y-axis at X-0, was developed considering the latest evaluation of wind tunnel data for

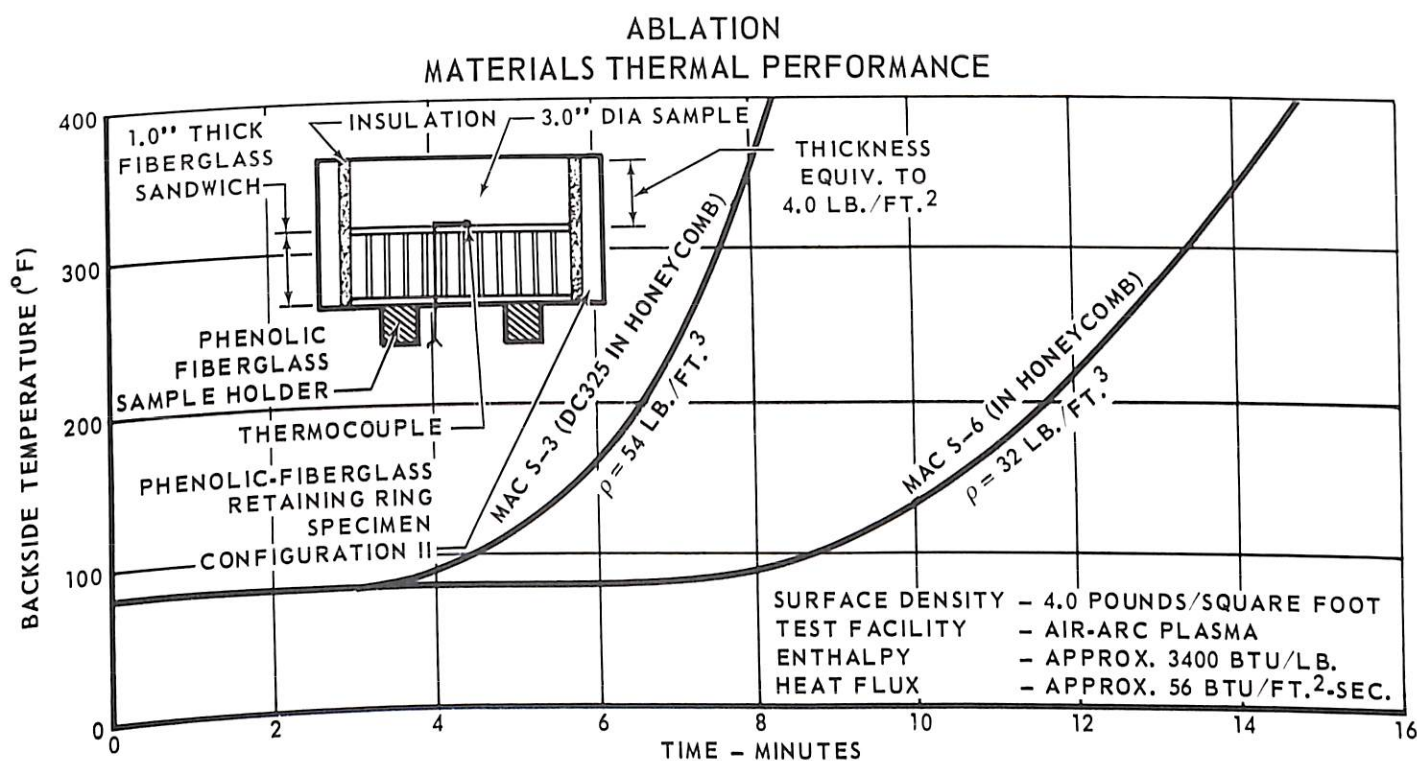


FIGURE 3.7-3

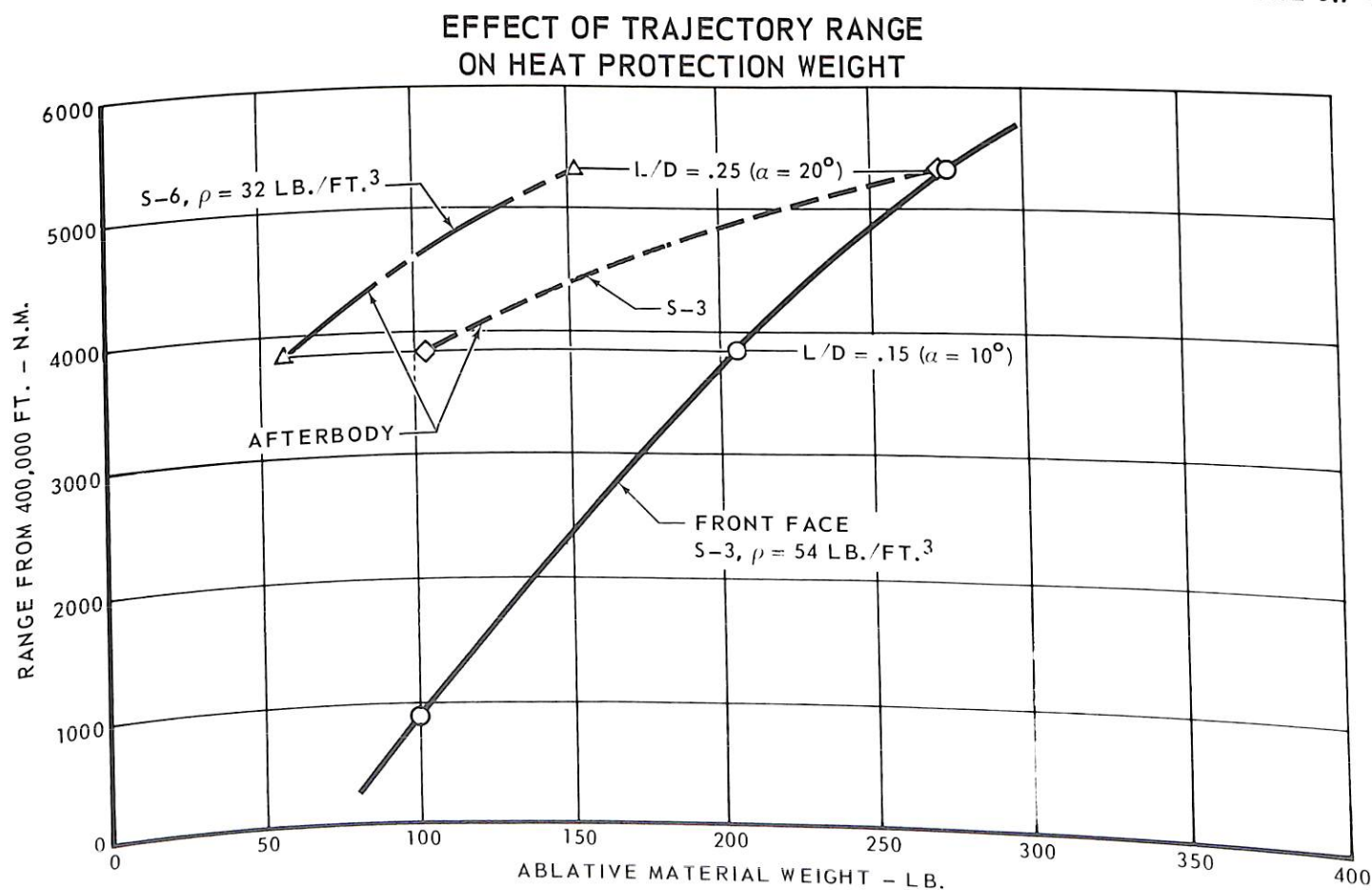


FIGURE 3.7-4

3.7.3.2 (Continued)

the Gemini shape (Reference 3.7-5). For lines parallel to the Y-axis, the convective distribution is assumed identical with Figure 3.7-5 at any given S/D location. Laminar flow is assumed for the majority of the front face surface. Transpiration of gases from the charring and receding ablator thickens the boundary layer and increases the likelihood of turbulent flow for the leeward region. For ablative design purposes, the leeward region is assumed to be turbulent throughout re-entry.

Gaseous radiation emitted from the shock region is computed at the hemispherical stagnation point using Reference 3.7-6, and is corrected for the Gemini shape with Reference 3.7-7. A correlation of Project Fire data (Reference 3.7-8) and data from Reference 3.7-9, which considered radiation from ablative products, indicates the selected method is within the band of data scatter. A reduction of the convective flux due to radiant heating of the air in the boundary layer (without ablative products) is trivial (Reference 3.7-10).

The radiation distribution for the Y-axis is illustrated in Figure 3.7-5. The distribution is computed by assuming an isentropic gas expansion behind the shock using Reference 3.7-11, and by using the Gemini shape shock standoff distances at angle of attack (Reference 3.7-12). The radiation distribution for lines parallel to the Y-axis is obtained by assuming Figure 3.7-5 is a cross section of a circular body.

Nonequilibrium gaseous radiation is computed using Reference 3.7-13, and the distribution, shown in Figure 3.7-5, is obtained by considering a local velocity vector normal to the shock.

Free molecule heating is neglected for re-entry since the heating time prior to 400,000 ft. is approximately 34 seconds where the flux level exceeds a

FRONT FACE HEATING DISTRIBUTION

- q LOCAL HEATING RATE
- CONVECTION ($q_{REF} =$ THEORETICAL STAG. POINT CONVECTION, $\alpha = 0$).
- NONEQUILIBRIUM RADIATION ($q_{REF} =$ STAG. POINT NONEQUILIBRIUM RADIATION).
- - - EQUILIBRIUM RADIATION ($q_{REF} =$ STAG. POINT EQUILIBRIUM RADIATION TO A HEMISPHERE WITH GEMINI HEAT SHIELD RADIUS OF CURVATURE).

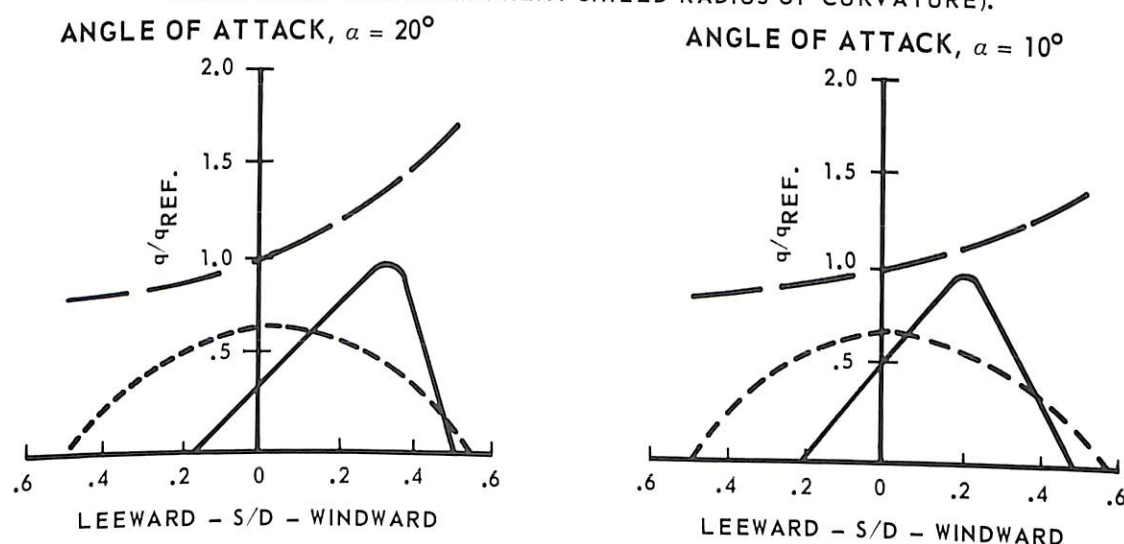


FIGURE 3.7-5

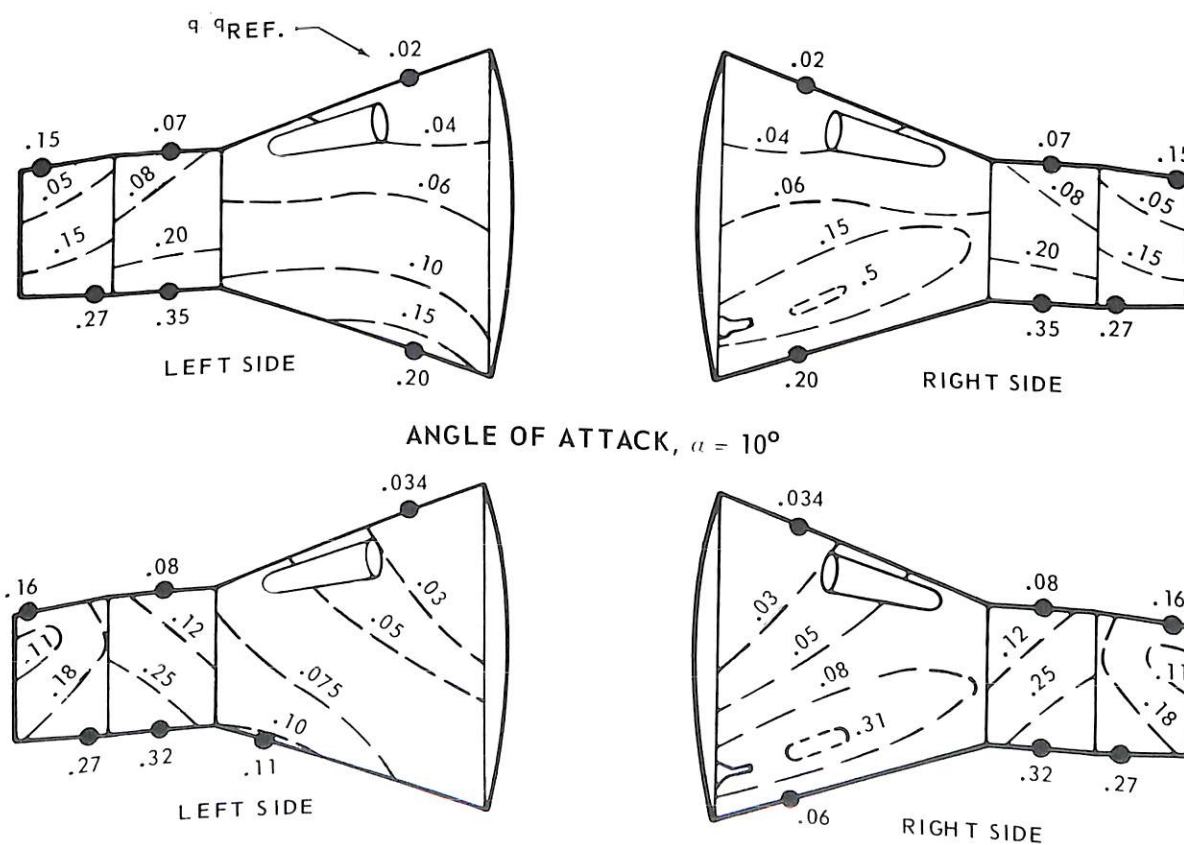
AFTERBODY HEATING DISTRIBUTION
ANGLE OF ATTACK, $\alpha = 20^\circ$ 

FIGURE 3.7-6

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3.7.3.2 (Continued)

100°F equilibrium temperature. At 400,000 ft., the convective flux exceeds free molecule heating by a ratio of three.

3.7.3.3 Afterbody Heating - Only convective heating is considered. The gaseous radiation distribution in Figure 3.7-3 indicates only slight radiant heating at the windward edge. After flow expansion around the heat shield corner, radiation is trivial when compared to the convective flux.

Figure 3.7-6 presents the predicted levels of afterbody heating. Prediction of the afterbody heating distribution is consistent with previous Gemini design procedure (Reference 3.7-14).

Heating aft of the windward wire bundle is based on a correlation used for the GT-2 post-flight evaluation. The relative protrusion of the wire bundle is less than the present Gemini because the surrounding shingles are covered with ablator; therefore, the predicted local heating is expected to be conservative.

3.7.3.4 Trajectory Heating Profiles - Two trajectories were studied in detail to determine the required amount of ablator for the front face and afterbody. The minimum range re-entry trajectory of 700 n.m. (10 g's) provides the largest convection and gaseous radiation heat fluxes; however, for both the front face and afterbody, the ablative thickness requirement is controlled by the maximum range trajectory of 3,838 n.m. which has the longer heating duration. The heating rate profiles and descriptive details for both trajectories are presented in Figure 3.7-7.

3.7.3.5 Ablative Material Behavior

Analytical Procedure - The Gemini shield analysis for the S-3 ablator was based on sea-level plasma jet tests. The analytical method is modified to simulate the swelling of the char layer after initial heating in order to

HEATING RATE PROFILES

$$W/A = 115 \text{ LB./FT.}^2$$

- - - - - CONVECTIVE HEATING
 EQUILIBRIUM GASEOUS RADIATION
 ——— NONEQUILIBRIUM GASEOUS RADIATION

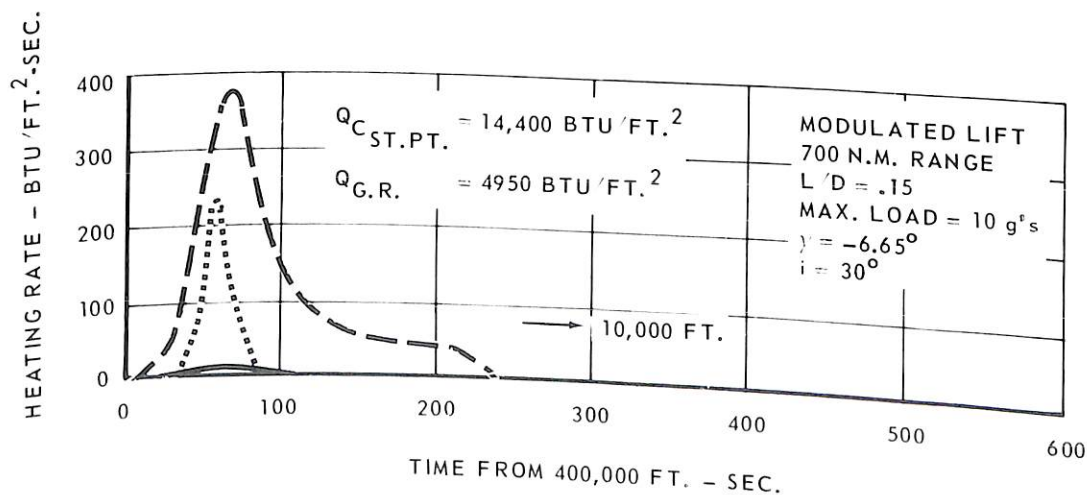
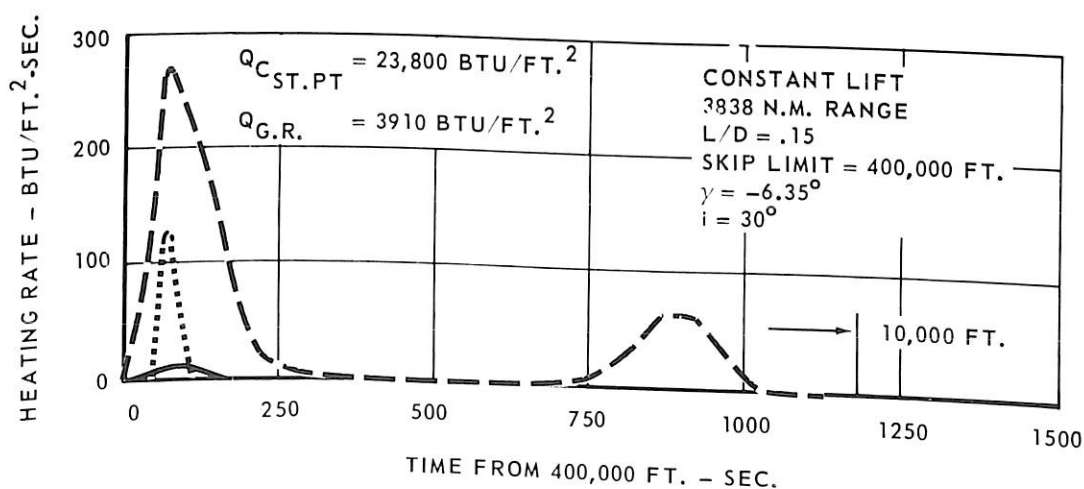
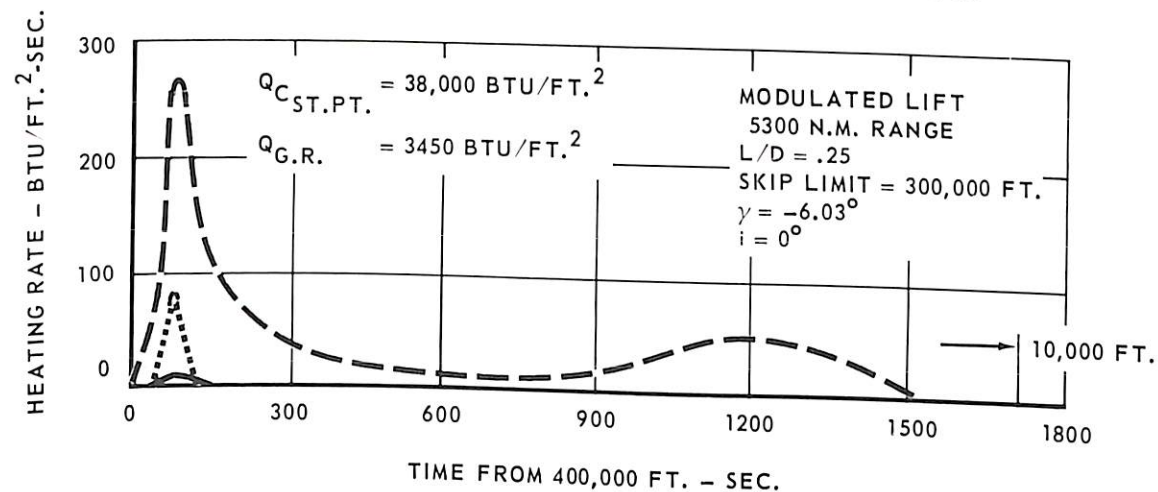


FIGURE 3.7-7

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3.7.3.5 (Continued)

correlate with the performance experienced during the GT-2 re-entry. Since S-6 ablator is considered on afterbody surfaces, the ablative requirement is determined for S-3 and corrected for the S-6 thermal effectiveness. S-6 provides 80 per cent more heat protection than S-3 for the same weight (lb./ft.²) as indicated in Figure 3.7-3. The S-6 ablator will not be used on the forebody because of insufficient test data to define flow shear stress resistance. On the afterbody the shear stresses are not significant. Plasma jet tests conducted by Aerospace and MAC for heat fluxes from 20 to 150 BTU/ft.²-sec. indicate no S-6 surface recession at fluxes of 60 BTU/ft.²-sec. or less.

On the front face, surface recession of S-3 is expected after the initial swelling during the brief re-entry period when velocities are superorbital. For analysis, surface recession of S-3 is based on coefficients recommended in Reference 3.7-15, which correlated two series of high heat flux tests (150 to 400 BTU/ft.² sec.). Surface flow shear stresses are higher than Gemini. Data from Reference 3.7-16 for high-G ICBM trajectories is used to correct the surface recession method, with an adjustment made for the S-3 density. Both ablative transpiration coefficients are reduced to approximate the test results of Reference 3.7-17 for high enthalpy conditions when gaseous radiation exists.

Environmental Effects

- A. Solar Ultra-Violet and Vacuum Exposure - Samples of S-3 were exposed for two weeks with no observed change of specimen dimensions, flexibility or measured hardness. Weight loss was between 0.33 and 0.96 per cent (eight specimens). No decrease in ablative performance was detected considering total heat of ablation and bondline temperature rise (Reference 3.7-18). S-6 is made with the same resin and is expected to provide similar behavior for space exposure.

3.7.3.5 (Continued)

B. Cold Soak to -320°F - Samples of S-3 made with a modified silicone resin were cold soaked at liquid nitrogen temperatures for 12 hours. Sudden exposure of the samples to plasma heating indicated no decrease in ablative performance. Two inch thick panels of S-3 were photographed with X-rays before and after a similar cold soak exposure to illustrate the lack of any shrinkage cracks or voids. S-6 has an increased solar absorptivity over S-3 which avoids cold cabin walls during the flight prior to re-entry.

3.7.3.6 Thermal Design Criteria - Ablative thicknesses were based on the following maximum allowable bondline temperatures:

Region	Maximum Bondline Temperature	Type of Flow
Front Face		
Windward Edge	580°F	Laminar
Center	580°F	Laminar
Leeward Edge	700°F	Turbulent
Afterbody	700°F	As Required (*)

* The region behind the windward wire bundle, the RCS, and R & R sections are computed for turbulent flow. All other afterbody regions are computed for laminar flow.

3.7.3.7 Ablative Thickness Distribution and Weight - Figure 3.7-4 illustrates the effect of trajectory range on heat protection weight for the front face covered with S-3, and for the afterbody covered with S-6 ablative material.

Figure 3.7-8 shows the distribution for the 3,838 n.m. trajectory ($L/D = .15$). The front face is covered with S-3 (density = 54 lb./ft.³), and the afterbody is surfaced with S-6 (density = 32 lb./ft.³). Note that the leeward or upper side

RANGE = 3838 N.M.

10° ANGLE OF ATTACK
S-3 ON FRONT FACE, DENSITY = 54 LB./FT.³
S-6 ON AFTERBODY, DENSITY = 32 LB./FT.³

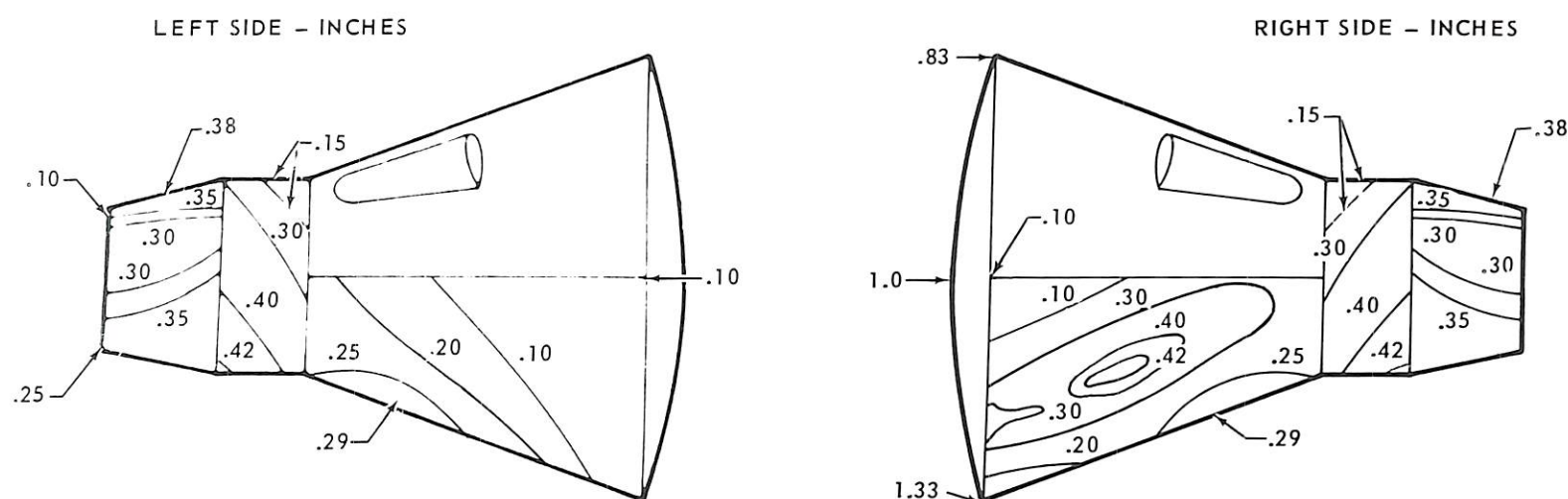


FIGURE 3.7-8

3.7.3.7 (Continued)

of the conical afterbody does not have an ablative covering. Columbium alloy shingles are used for the heat protection surface.

3.8 Propulsion System - The propulsion systems for the Gemini spacecraft consist of the Re-entry Control System (RCS) and the Orbit Attitude and Maneuver System (OAMS). The Gemini "A" retrograde motors (TE-M-385) are not required for this mission and are removed from the spacecraft.

3.8.1 Configuration I

3.8.1.1 RCS - The RCS consists of two identical and redundant hypergolic, bi-propellant rocket engine systems, but differs from the Gemini "A" system in that the pressurization group is modified to a "blow-down" design and the motor shutoff valves and all heaters are removed. The pressurization group is modified by deleting the regulator and Package "B" and adding one pressurant tank, one Package "A", and a low pressure transducer for each propellant of both systems. In the propellant feed system, the motor-operated shutoff valves are replaced with normally open pyrotechnic valves. One of the two redundant systems is shown schematically in Figure 3.8-1. This system yields the same total impulse as the regulated pressure system, but the Thrust Chamber Assembly (TCA) thrust level varies with propellant consumption as shown in Figure 3.8-2.

For the large earth orbit mission, the RCS is not activated until adapter separation. The system is used to provide the impulse for pitch and yaw rate damping, and roll control of the re-entry module until the main parachute is deployed and the spacecraft stabilized. The normally open pyrotechnic valve in each propellant feed line is then fired to prevent any propellant leakage through the TCA's during the final descent phase and the recovery period.

Pertinent characteristics of the RCS are shown in Table 3.8-1. Propellant loading and utilization is shown in Table 3.8-2.

Figure 3.8-3 shows the location of the two RCS rings in the re-entry module and the location and identifying number of the TCA's.

SCHEMATIC OF RE-ENTRY CONTROL SYSTEM
ONE OF TWO REDUNDANT SYSTEMS
CONFIGURATION 1

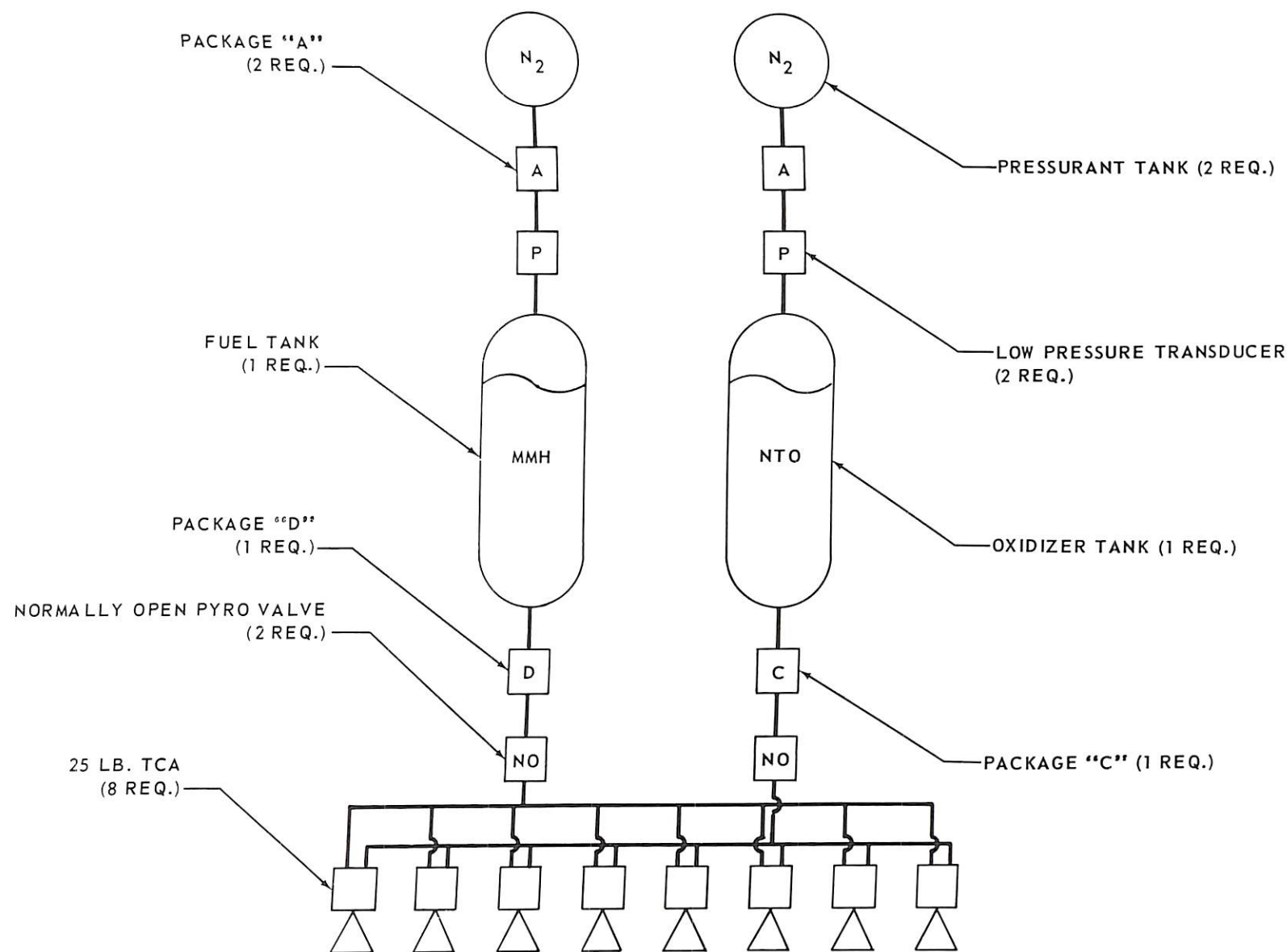


FIGURE 3.8-1

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THRUST VARIATION WITH PROPELLANT CONSUMPTION
RE-ENTRY CONTROL SYSTEM
MAXIMUM SYSTEM PRESSURE = 450 PSIA

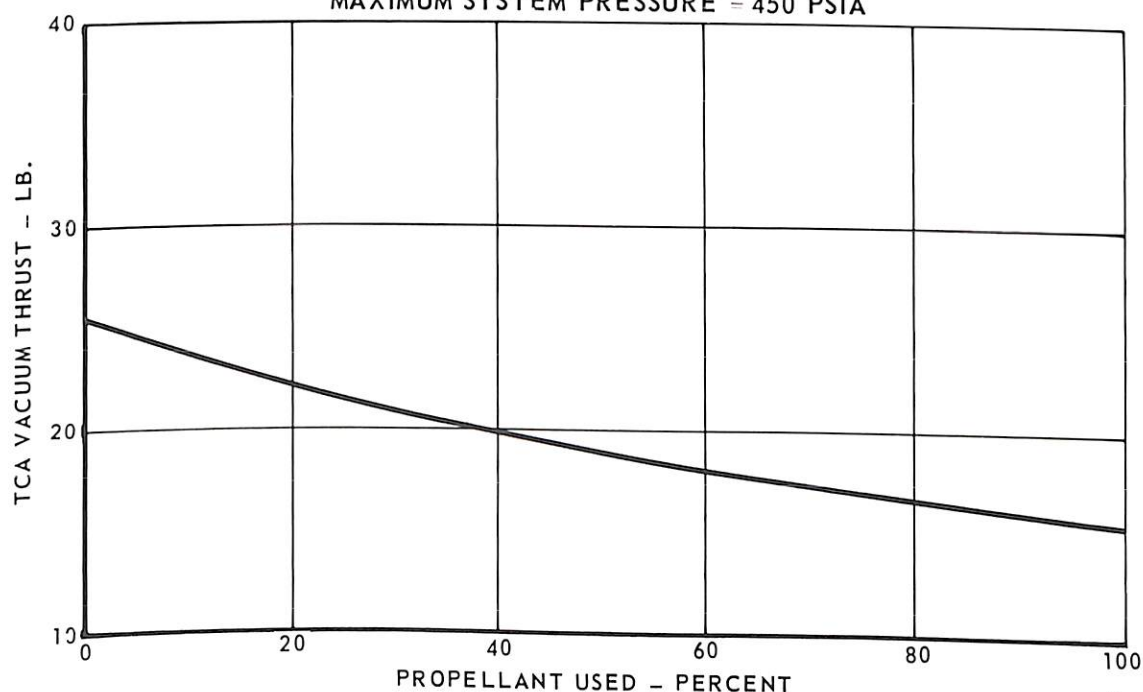


FIGURE 3.8.2

TABLE 3.8.1
RCS CHARACTERISTICS
CONFIGURATION I

PERFORMANCE*		TANKAGE*	
TCA THRUST, LB.	**	OXIDIZER TANK VOLUME, IN. ³	439
SPECIFIC IMPULSE, SEC.	266	FUEL TANK VOLUME, IN. ³	546
TOTAL IMPULSE, LB.-SEC.	9310	PRESSURANT TANK VOLUME, IN. ³	185
OPERATING CONDITIONS		WEIGHTS*	
OXIDIZER TO FUEL RATIO	1.3	PRESSURANT, N ₂ , LB.	.5
SOURCE PRESSURE AT +70°F, PSIG	480	OXIDIZER, NTO, LB.	21
MAXIMUM TEMPERATURE, °F	+ 130	FUEL, MMH, LB.	16
MINIMUM TEMPERATURE, °F	+ 15		
* PER RING, VACUUM, STEADY STATE			
** VARIES FROM 15.5 TO 25.5			

TABLE 3.8-2
PROPELLANT LOADING AND UTILIZATION
RE-ENTRY CONTROL SYSTEM
CONFIGURATION I

PROPELLANT	RING A	RING B
PROPELLANT LOADING, LB.		
LOADED	37	37
TRAPPED	2	2
USABLE	35	35
USABLE REQUIRED, LB.		
ADAPTER SEPARATION TO 400,000 FT.	.5	.5
400,000 FT. THRU RE-ENTRY	27.0	27.0
USABLE PROPELLANT MARGIN		
POUNDS	15	
PERCENT	27	

THRUST CHAMBER LOCATION AND IDENTIFYING NUMBER
RE-ENTRY CONTROL SYSTEM
CONFIGURATION I

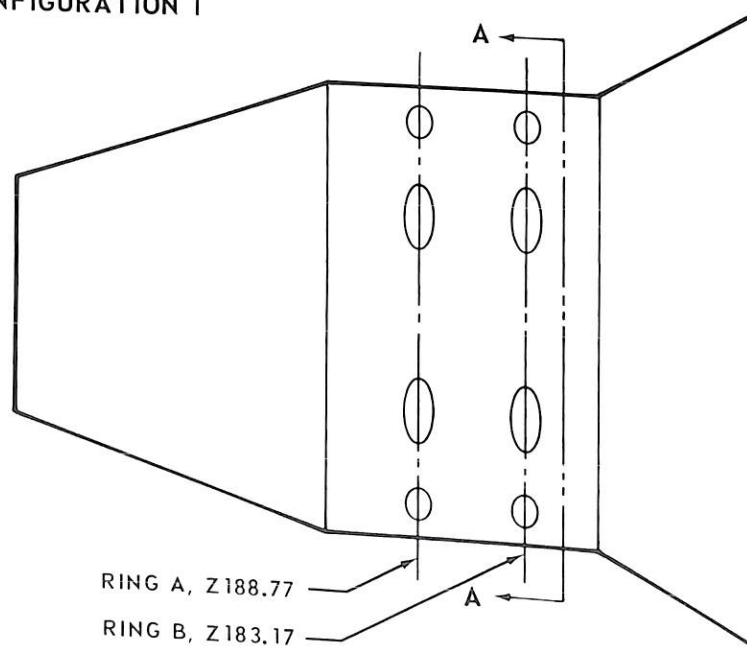
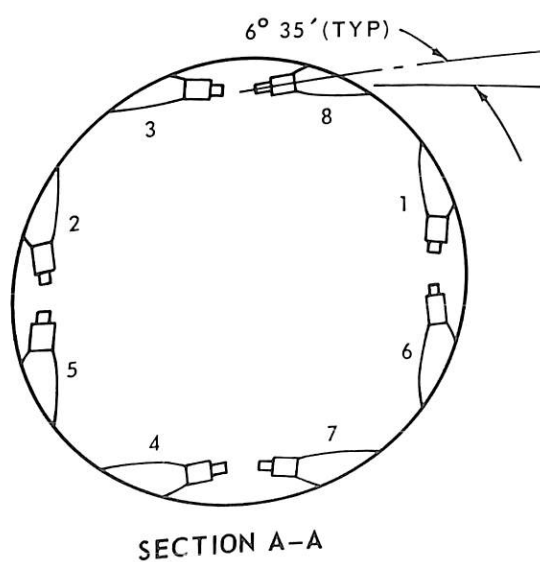


FIGURE 3.8-3

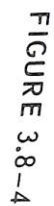
3.8.1.2 OAMS - The OAMS is a hypergolic, bi-propellant rocket engine system that utilizes Gemini "A" hardware design, but differs in configuration from the Gemini "A" OAMS due to the change in mission requirements. This system provides the impulse for separation from the launch vehicle, rendezvous with the target vehicle, propellant settling in the target vehicle, midcourse corrections, attitude control, and retrograde from the parking orbit if the mission is aborted prior to injection into the large earth orbit.

The system consists of three propellant supply groups and two TCA groups shown schematically in Figure 3.8-4. The three groups provide the system redundancy at insertion into the large earth orbit for all the required mission functions after a single failure in any part of the system, and require only a 50 per cent increase in propellant load over the required loading to give 100 per cent redundancy. Interconnection between the pressurant and propellant groups provides maximum propellant utilization in the event of failure(s) in the system. Through the judicious use of the normally closed pyrotechnic valves in the interconnect lines, the check valves in Package "B", the check valves downstream of Packages "C" and "D", and the motor shutoff valves, total propellant utilization exists in the event of either a total loss of either source pressure supply or serious propellant leakage in either TCA group.

Pertinent characteristics of the OAMS are shown in Table 3.8-3. Propellant loading and utilization are given in Table 3.8-4. The propellant is utilized in a manner that, with a single failure, there is always sufficient propellant available to complete the remaining mission functions.

Figure 3.8-5 shows the location of the two TCA groups in the adapter and the identifying number of each TCA.

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TABLE 3.8.-3
OAMS CHARACTERISTICS
CONFIGURATION I

PERFORMANCE*				
	100 LB. TCA AFT FIRING	25 LB. TCA	85 LB. TCA FWD. FIRING	100 LB. TCA RADIAL
THRUST, LB.	94.5	23.0	79.0	94.5
SPECIFIC IMPULSE, SEC.	280	250	280	280
O/F	1.6	0.7	1.6	1.6
LIFE, SEC.	1000+	425	147	220

TANKAGE		OPERATING CONDITIONS	
OXIDIZER TANK VOLUME, IN. ³	16,520	SOURCE PRESSURE AT +70°F, PSIG	3000
FUEL TANK VOLUME, IN. ³	16,520	REGULATED PRESSURE, PSIG	295
PRESSURANT TANK VOLUME, IN. ³	5,085	MAXIMUM TEMPERATURE, °F	+ 120
		MINIMUM TEMPERATURE, °F	+ 15
WEIGHTS			
PRESSURANT, He, LB.	5.7		
OXIDIZER, NTO, LB.	635.2		
FUEL, MMH, LB.	444.1		

TABLE 3.8-4
PROPELLANT LOADING AND UTILIZATION
ORBIT ATTITUDE AND MANEUVER SYSTEM
CONFIGURATION I

PROPELLANT	WEIGHT, LB.
PROPELLANT LOADED	
LOADED	1141.4
TRAPPED	24.9
USABLE	1116.5
USABLE REQUIRED	
SEPARATION FROM LAUNCH VEHICLE (ΔV = 10 FPS)	
INSERTION ERRORS (ΔV = 40 FPS)	9.2
CIRCULARIZATION AND RENDEZVOUS (ΔV = 120 FPS)	36.5
BRAKING, GO-AROUND AND DOCKING (ΔV = 150 FPS)	108.6
ATTITUDE CONTROL DURING RENDEZVOUS	133.7
DISTURBANCE TORQUE	
ESTABLISH ATTITUDE AFTER INSERTION	25.3
HOLD ATTITUDE DURING DOCKING	1.7
TRANSTAGE PROPELLANT SETTLING	12.7
MIDCOURSE CORRECTIONS (ΔV = 458 FPS)	5.6
ATTITUDE CONTROL	393.4
DISTURBANCE TORQUE (DURING MIDCOURSE CORRECTIONS)	
PLATFORM ALIGNMENT	34.6
ORBIT ATTITUDE (ROLL RATE .6 DEGREE SEC., PITCH AND YAW - +3 DEGREES)	2.0
CONTINGENCY	17.5
USABLE PROPELLANT MARGIN (50% OF USABLE AFTER LEO INJECTION)	86.8
	248.9

TCA LOCATION AND IDENTIFYING NUMBER
CONFIGURATION I

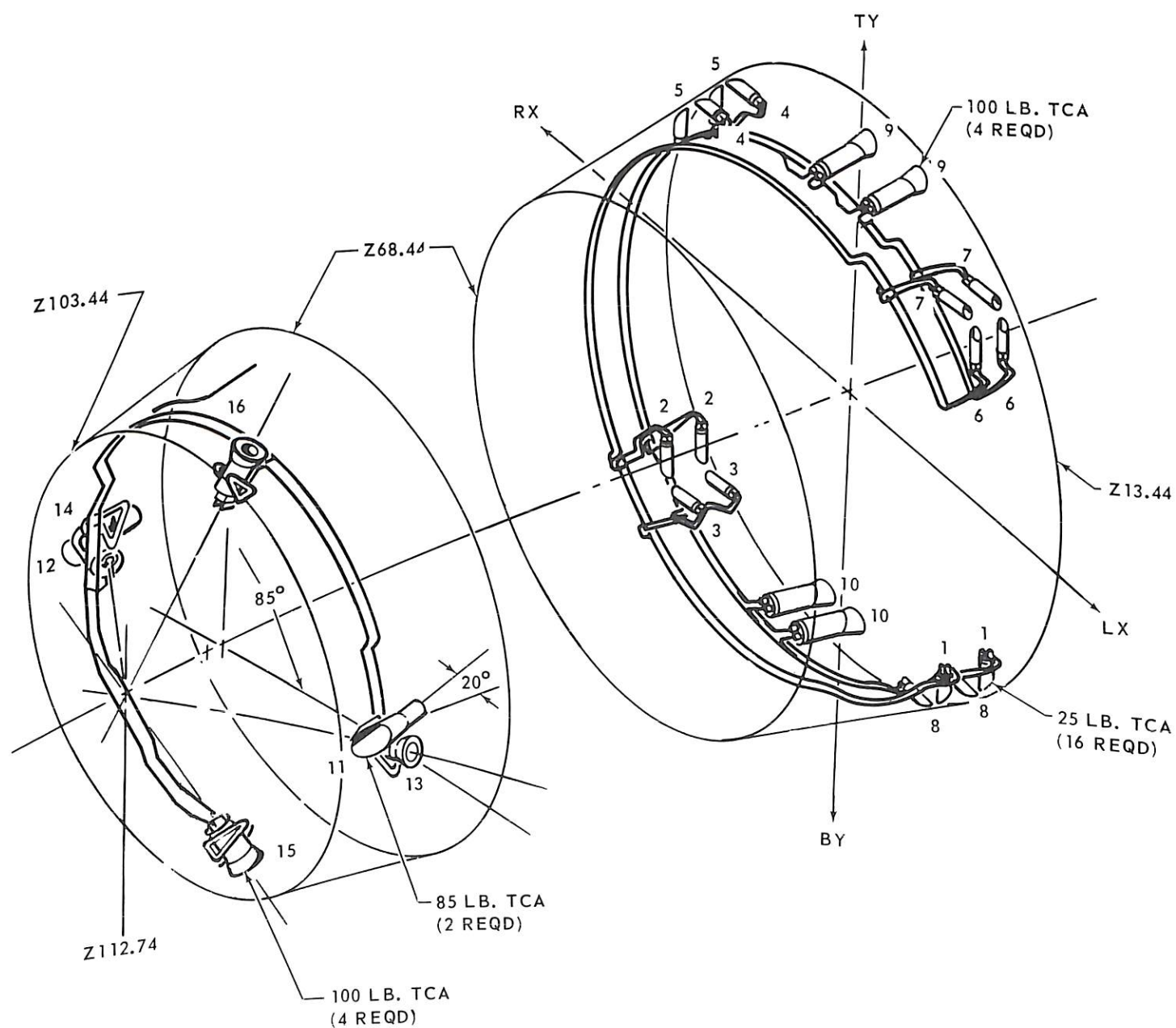


FIGURE 3.8.-5

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3.8.2 Configuration II

3.8.2.1 RCS - The RCS is the same as described in Paragraph 3.8.1.1.

3.8.2.2 OAMS - The OAMS for Configuration II provides the impulse for mid-course corrections and attitude control of the spacecraft from injection to re-entry. The system is basically the same as the system described in Paragraph 3.8.1.2 except for the removal of two propellant tanks and the forward firing and translational TCA's. The system is shown schematically in Figure 3.8-6. Pertinent characteristics of this configuration are given in Table 3.8-5. Propellant loading and utilization are given in Table 3.8-6.

Figure 3.8-7 shows the location and identifying number of each TCA.

SCHEMATIC OF ORBIT ATTITUDE AND MANEUVER SYSTEM CONFIGURATION II

GEMINI - LARGE EARTH ORBIT

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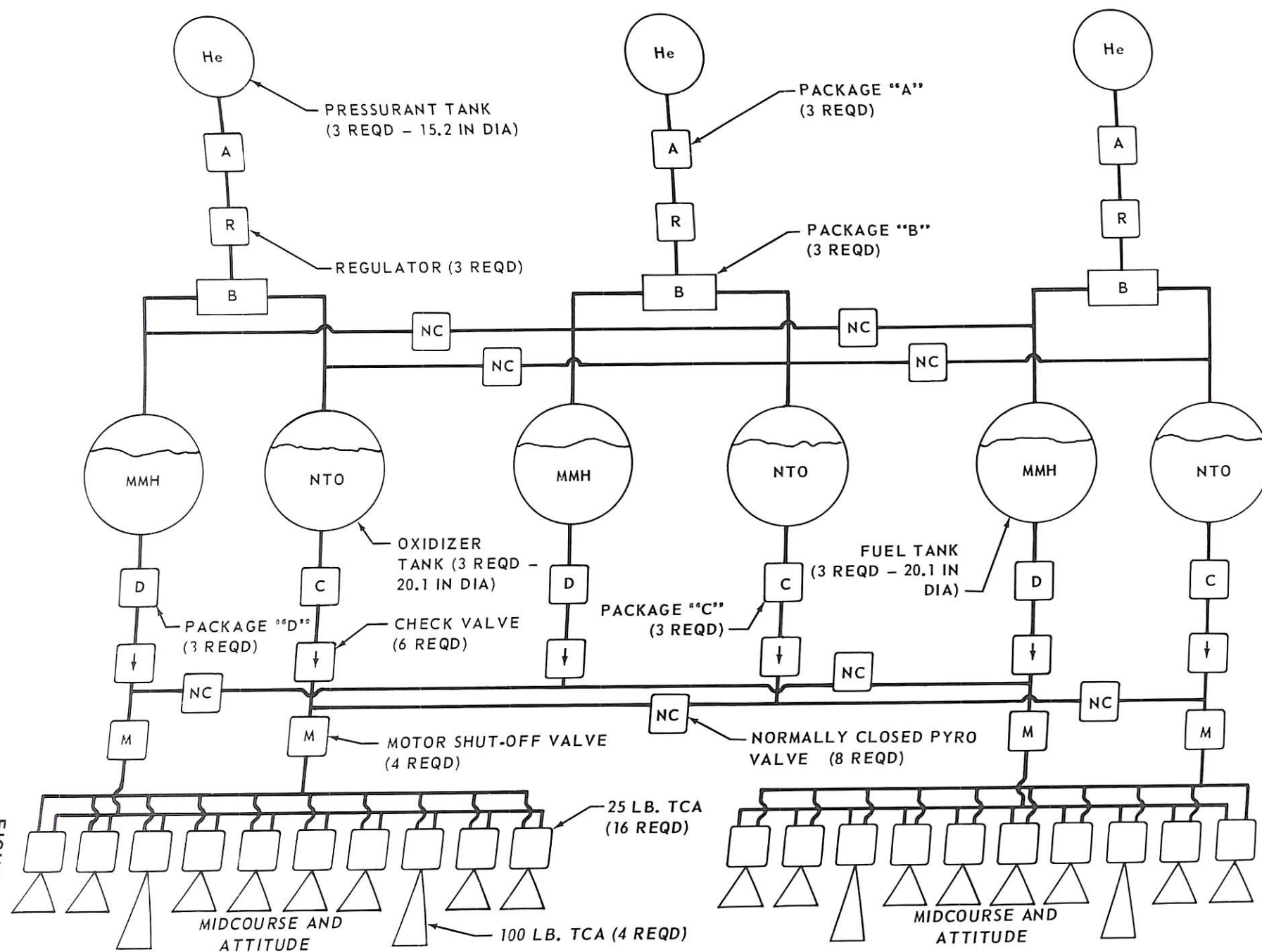


FIGURE 3.8-6

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TABLE 3.8-5
OAMS CHARACTERISTICS
CONFIGURATION II

PERFORMANCE*		TANKAGE	
100 LB. TCA		OXIDIZER TANK VOLUME, IN. ³	12,390
THRUST, LB.	94.5	FUEL TANK VOLUME, IN. ³	12,390
SPECIFIC IMPULSE, SEC.	280	PRESSURANT TANK VOLUME, IN. ³	5,085
O/F	1.6		
LIFE, SEC.	1,000+		
25 LB. TCA			
THRUST, LB.	23.0		
SPECIFIC IMPULSE, SEC.	250		
O/F	0.7		
LIFE, SEC.	425		
		WEIGHTS	
		PRESSURANT, He, LB.	5.7
		OXIDIZER, NTO, LB.	415.8
		FUEL, MMH, LB.	312.1
		OPERATING CONDITIONS	
		SOURCE PRESSURE AT +70°F, PSIG	3,000
		REGULATED PRESSURE, PSIG	295
		MAXIMUM TEMPERATURE, °F	+120
		MINIMUM TEMPERATURE, °F	+15
*VACUUM, STEADY STATE			

TABLE 3.8-6
PROPELLANT LOADING AND UTILIZATION
ORBIT ATTITUDE AND MANEUVER SYSTEM
CONFIGURATION II

PROPELLANT	WEIGHT, LB.
PROPELLANT LOADING	
LOADED	727.9
TRAPPED	17.7
USABLE	710.2
USABLE REQUIRED	
MIDCOURSE CORRECTION ($\Delta V = 458$ FPS)	
ATTITUDE CONTROL	375.0
DISTURBANCE TORQUE (DURING MIDCOURSE CORRECTION)	
PLATFORM ALIGNMENT	33.0
ORBIT ATTITUDE (ROLL RATE .6 DEGREE/SEC., PITCH AND	1.8
YAW - ± 3 DEGREES)	16.0
CONTINGENCY	
	47.7
USABLE PROPELLANT MARGIN (50%) (REDUNDANCY, EARTH	
OBSERVATIONS AND EXPERIMENTS)	236.7

TCA LOCATION AND IDENTIFYING NUMBER
CONFIGURATION II

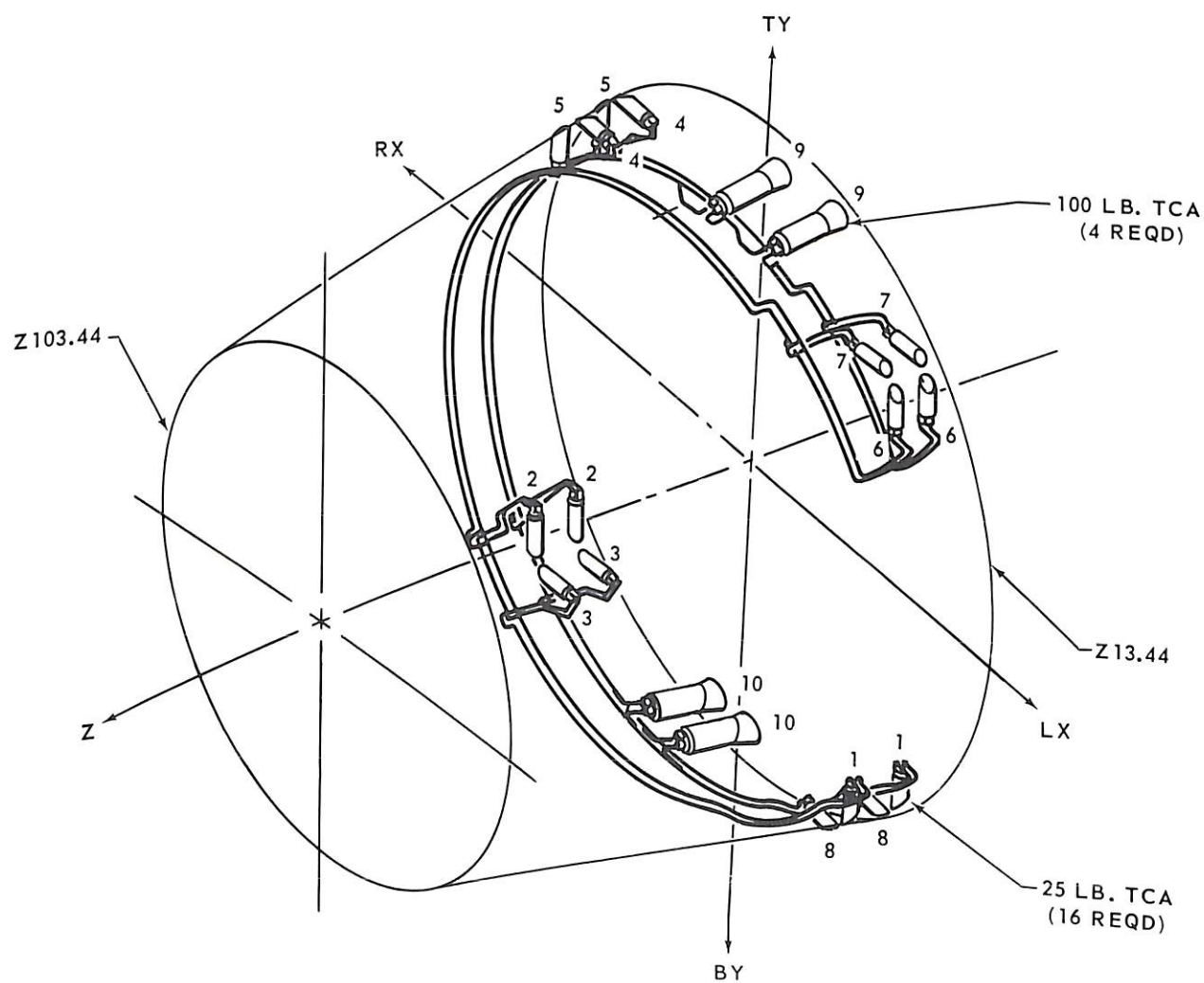


FIGURE 3.8-7

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3.9 Reliability Analysis

3.9.1 General Approach - The reliability predicted for a safe large earth orbit mission is .983 probability of success. The new mission requirement has defined conditions not associated with the present Gemini flights. The reliability analysis was directed towards a reliable resolution to these new situations, and a system analysis for determination of "weak links". The feasibility study for this mission defined launch weight as critical, thus weight became the major reliability tradeoff consideration. New conditions considered were:

- A. O₂ for Astronauts - Considers 500 BTU/hr./man, 1000 cc./min. cabin leakage at sea level, and a 7-day mission duration. The ECS O₂ supply is backed up with an equivalent O₂ supply. Either O₂ supply can be used by the astronauts. Probability for having O₂ available is 0.99798.
- B. H₂O for Astronauts - Considers minimum H₂O required to be 53 pounds. System has capacity for 106 pounds. Probability for minimum H₂O is 0.99958.
- C. Cabin Pressure - Considers astronauts unable to survive long periods in "hard" suits without food and the degradation effects on the ECS system without cabin pressure. Sufficient O₂ is provided to accomplish two cabin repressurizations even if one of the O₂ supplies fails. Assuming cause for depressurization can be corrected, cabin pressure probability is .99660.
- D. OAMS - Considers five maneuvers with attitude control for midcourse corrections, and attitude control to meet thermal requirements during coast. Requirements can be met with a single failure each of the OAMS groups, pressurant, propellant, and thruster. The major midcourse correction will be made in the first ten hours; if successful, the OAMS for the remaining mission is extremely reliable. Probability for meeting the minimum requirements for maneuver and attitude is 0.99700.

3.9.1 (Continued)

- E. Electrical Power - Total power requirements for the mission could be handled by two fuel cell sections. The system has three fuel cell sections giving a probability for adequate power of 0.99858.
- F. IGS, ACME, and Telecommunications - Considers DSIF command and/or voice to be required for correct spacecraft alignment during maneuvers. Three DSIF voice paths, ACME redundancy, redundant sun-star sensors, Inertial Reference Package as backup for the IMU platform, and a manual sextant have been included to achieve the reliability goal. Probability for having adequate knowledge for midcourse corrections is 0.99800.

3.9.2 Probabilities - The over-all probability for completing the mission in its desired modes, and for completing the mission without catastrophic failure by major system, is presented below.

<u>Configuration I</u>		
<u>System</u>	<u>Desired Mode</u>	<u>Safe Mission</u>
Communications	0.99996	0.99996
ACME	.99906	.99906
OAMS	.99442	.99700
ECS	.99347	.99517
Electric Power	.99858	.99863
IGS	.98668	.99822
Radar	.99963	. -
RCS	.99969	.99969
Landing	.99528	.99528
All the rest	.99900	.99999
For the spacecraft	<u>0.96618</u>	<u>0.98308</u>

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Configuration II

<u>System</u>	<u>Desired Mode</u>	<u>Safe Mission</u>
Communications	0.99996	0.99996
ACME	.99906	.99906
OAMS	.99668	.99700
ECS	.99347	.99517
Electric Power	.99858	.99863
IGS	.98989	.99877
RCS	.99969	.99969
Landing	.99528	.99528
All the rest	.99900	.99999
For the spacecraft	<u>0.97188</u>	<u>0.98363</u>

3.9.3 Single Point Failures - System evaluation for "critical" single point failures has isolated several failures which are under further study for elimination and/or minimizing within the tradeoff criteria for weight, schedule, cost, and reliability. Those are:

- A. Cabin drinking water system.
- B. Common ECS O₂ lines and connections in the high pressure areas.
- C. CO₂ partial pressure sensor.
- D. LiOH Canister.
- E. ECS-O₂ excessive use points - demand regulator, cabin regulator, cabin mounted valves, etc.

4. SUPPORTING STUDIES AND INFORMATION

4.1 Trajectory Analysis

4.1.1 Nominal Trajectory - The circumlunar trajectory used in this study as a nominal is a figure 8 type and is shown projected on the equatorial plane in Figure 4.1-1. This particular trajectory is based on a direct launch. The date of closest approach to the moon is chosen to be consistent with a southern declination of the moon along with the moon being near perigee: the former being a requirement for a direct launch from a northern latitude and the latter being desirable from an energy requirement viewpoint. The close approach distance is consistent with a trip time less than seven days and a landing in the eastern Pacific Ocean.

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NOMINAL TRAJECTORY PROJECTED ONTO EQUATORIAL PLANE

DATE OF CLOSEST APPROACH TO MOON

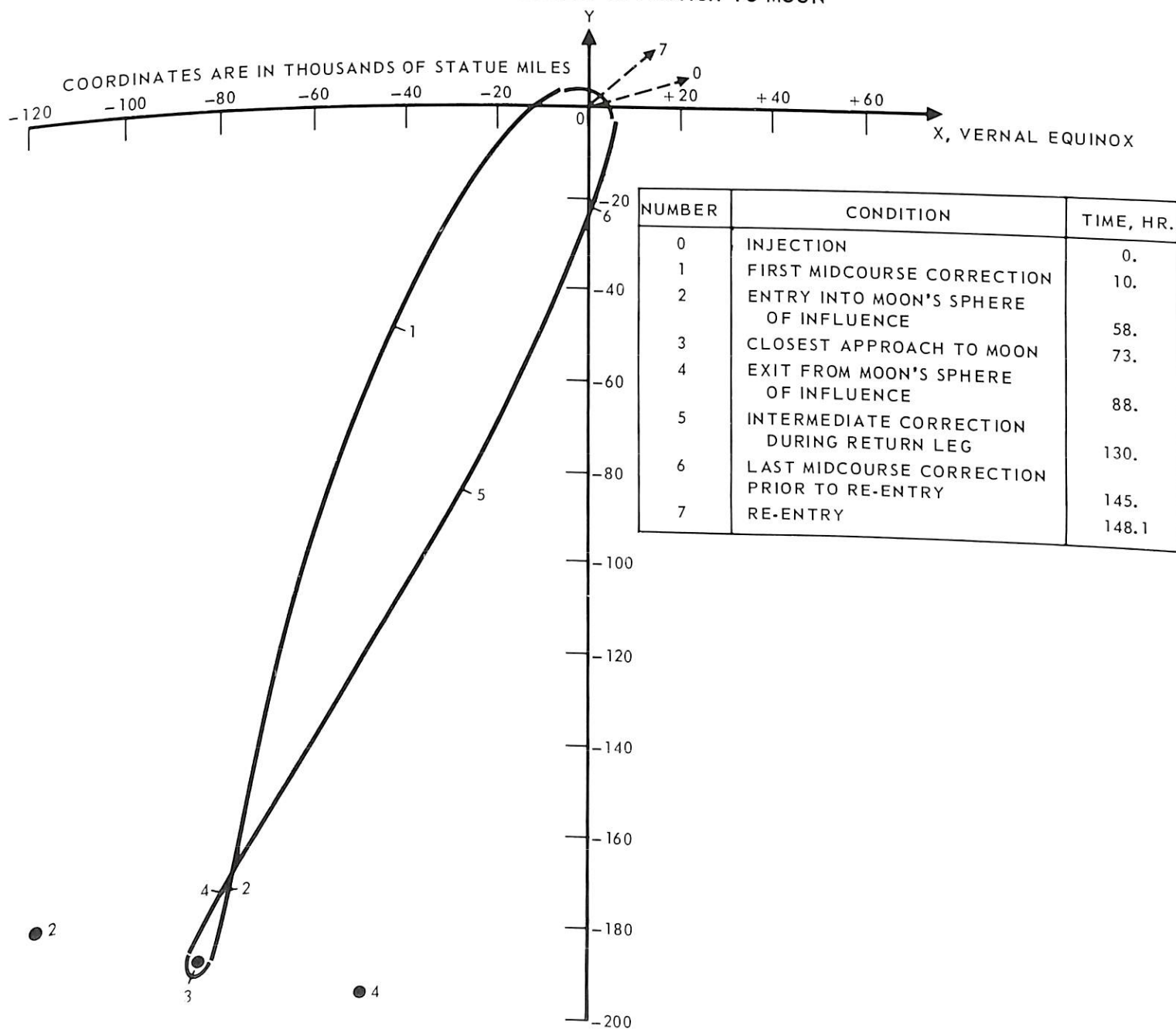


FIGURE 4.1-1

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4.2 Aerodynamics

4.2.1 Re-entry Aerodynamic Characteristics - The re-entry aerodynamic data used in these studies are based upon Gemini wind tunnel data, "modified" to reflect the flight test aerodynamic data acquired from the GT-2 and GT-3 flights. The maximum Mach number tested in the wind tunnel and verified by flight test data was Mach 25. Re-entry following lunar return will result in Mach numbers as high as 40. For the current study, the aerodynamic coefficients above Mach 25 were obtained by extrapolation.

Two hypersonic lift-to-drag ratios were investigated for this re-entry configuration. They are L/D of .25 which corresponds to a trim angle of attack of approximately 15 degrees at Mach 25, and an L/D of .15 with a corresponding trim angle of attack at Mach 25 of approximately 9 degrees. The L/D of .25 provides a wider permissible re-entry corridor when roll modulation is possible, and also results in larger touchdown footprint capability. However, the maximum dispersed ranges which are associated with the higher L/D require greater amounts of spacecraft heat protection. In addition, the larger angle of attack for the higher L/D requires more afterbody heat protection. The L/D of .15 is attractive for several reasons. The lower values of maximum dispersed range and the lower trim angle of attack both reduce the required heat protection. Also, this is the trim condition being employed for the manned missions in the present Gemini program. Thus, considerable flight test data will be available for this L/D to aid in the prediction of the re-entry performance. The final decision on the re-entry L/D will, of course, depend on the results of more detailed studies to determine precisely the amount of touchdown footprint required.

In any event, additional wind tunnel testing of the spacecraft re-entry configuration is required to help define the aerodynamic characteristics above

4.2.1 (Continued)

Mach 25. Configuration changes such as the redesigned R & R section requires additional testing at Mach numbers less than 25. A preliminary investigation of the required range of test Mach numbers and Reynolds numbers indicates that this testing can be done in the McDonnell "Hotshot" wind tunnel facility. Some development work is needed in this respect since the higher Mach numbers require the use of hydrogen gas in the tunnel. This approach is, however, straightforward and can be incorporated into the McDonnell facility with minimum tunnel modification.

4.2.2 Re-entry Considerations

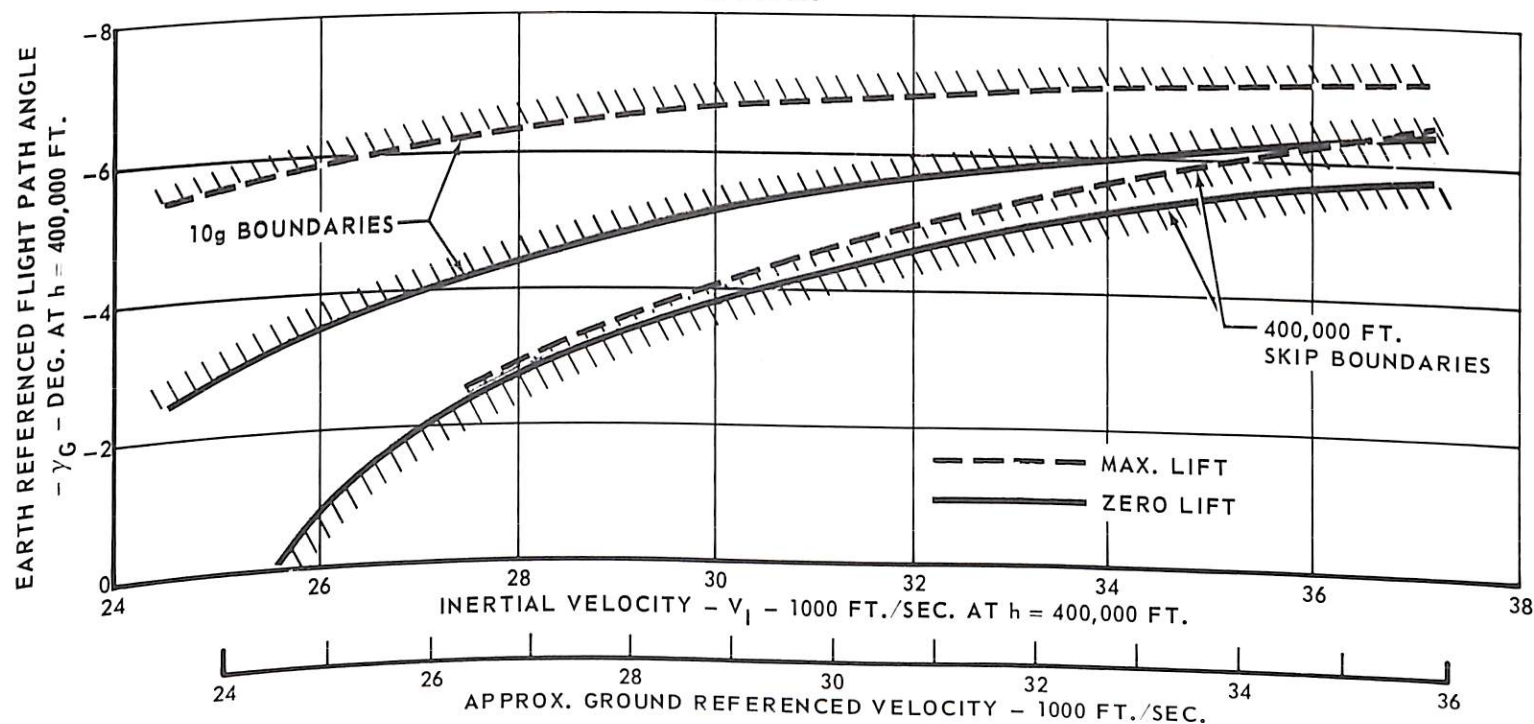
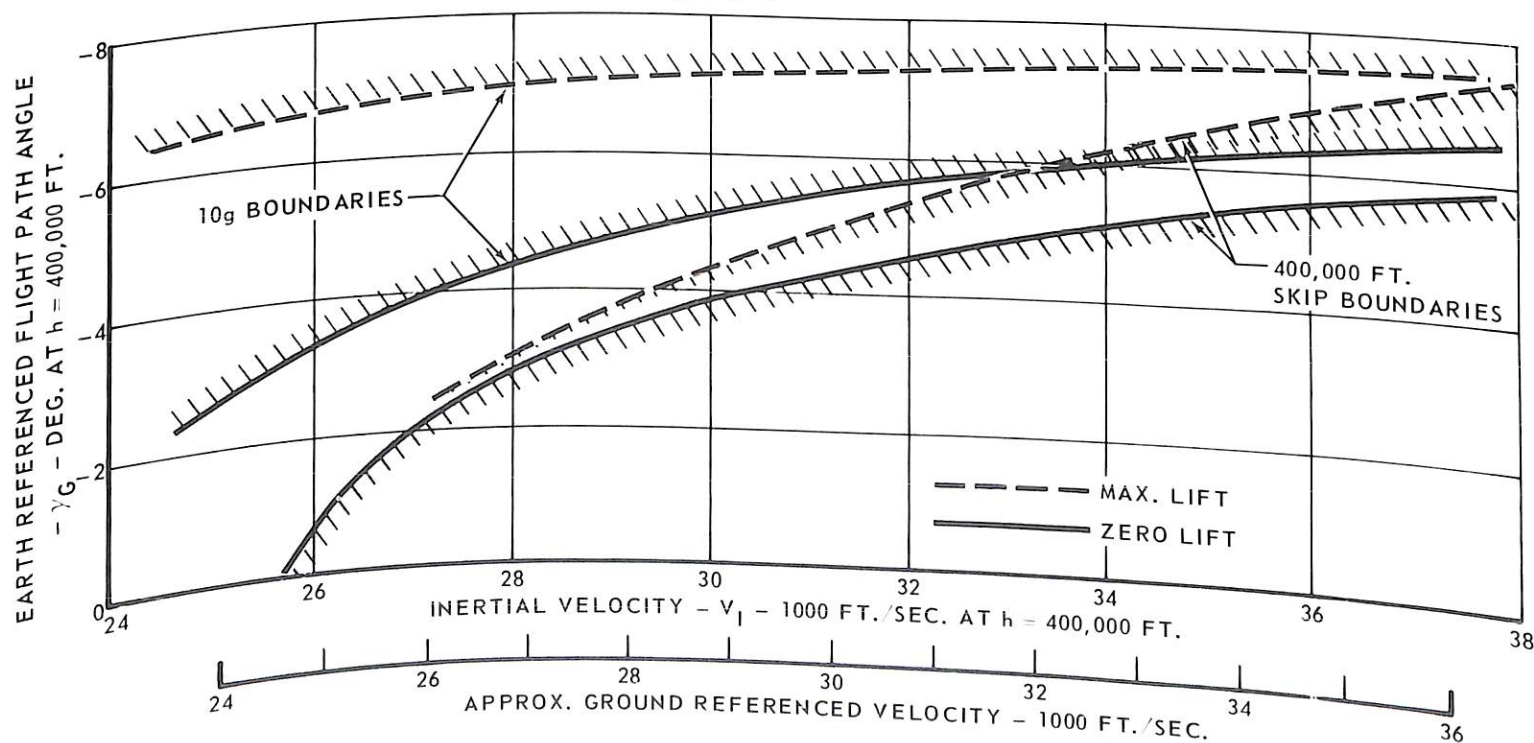
4.2.2.1 Flight Path Angle Boundaries - Re-entry velocity - flight path angle boundaries are based on constant lift (zero or full) re-entries and are shown for L/D's of .15 and .25 in Figure 4.2-1. These boundaries define constants which are: (a) maximum skipout altitude of 400,000 feet, and (b) maximum peak re-entry load factor of 10 g's. The skipout altitude is limited to 400,000 feet since the re-entry flight path angle accuracy required to control skip altitude becomes prohibitive for higher altitudes. The 10 g limit provides a comfortable margin relative to the human acceleration tolerance limit shown in Figure 4.2-2.

SKIP AND LOAD FACTOR BOUNDARIES FOR SUPER ORBITAL RE-ENTRY

CONSTANT LIFT (NO MODULATION)

30° INCLINATION

LARGE EARTH ORBIT GEMINI

 $L/D = .15$  $L/D = .25$ 

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FIGURE 4.2-1

4.2.2.1 (Continued)

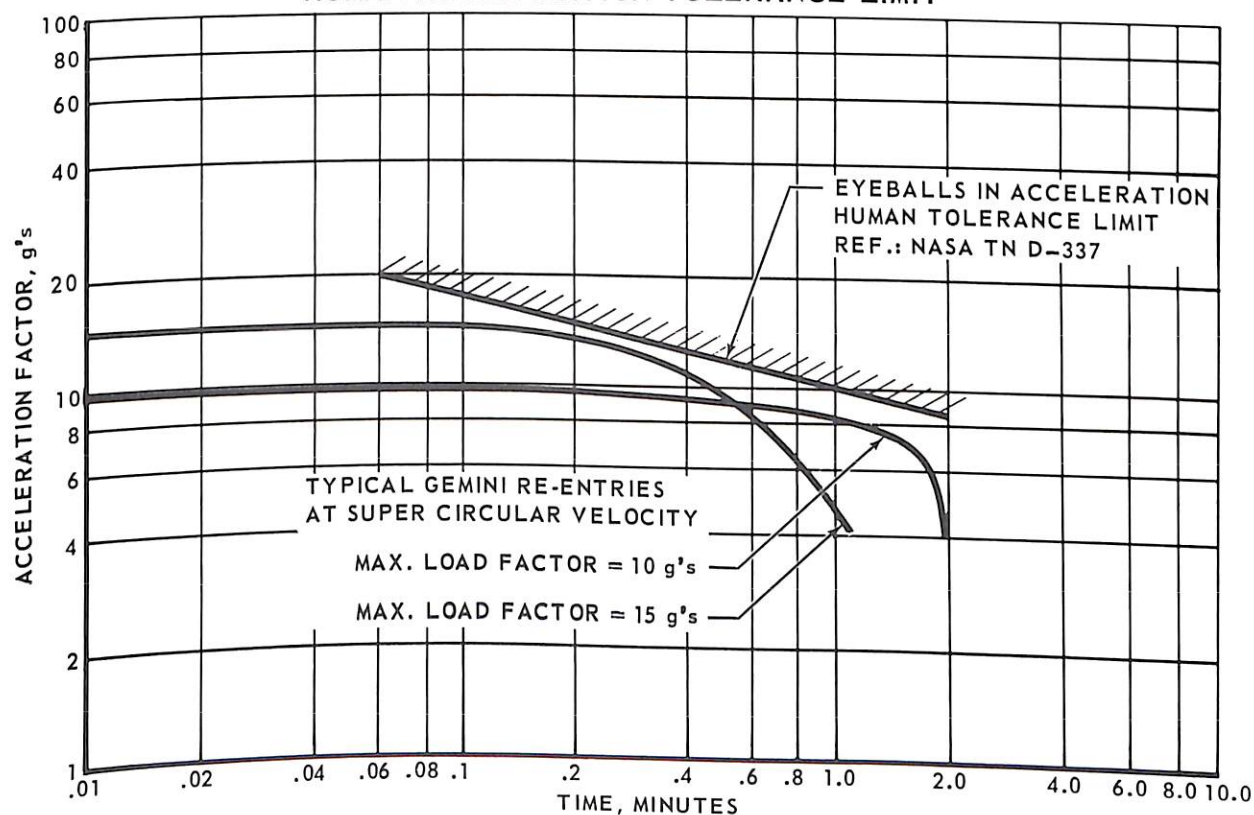
LARGE EARTH ORBIT GEMINI
COMPARISON OF GEMINI RE-ENTRY ACCELERATION AND
HUMAN ACCELERATION TOLERANCE LIMIT

FIGURE 4.2-2

For a re-entry velocity approximating return from a circumlunar trajectory ($V_g = 35,000$ ft./sec.), the usable flight path angle corridors for $L/D = .15$ and $.25$ are 1.48 and 1.88 degrees, respectively.

4.2.2.2 Re-entry Range and Touchdown Footprint Analysis - For a lunar return re-entry within given flight path angle boundaries, considerable variation of range, and the associated touchdown footprint is encountered. In choosing a nominal re-entry flight path angle, several factors must be considered. The anticipated trajectory dispersions, about the nominal, must permit neither a maximum dispersed range greater than that for which the spacecraft heat protection is designed, nor an inadequate minimum dispersed touchdown maneuvering footprint. The nominal re-entry flight path angle should also permit simple

4.2.2.2 (Continued)

manual re-entry control techniques to insure safe return of the spacecraft and crew for touchdown control guidance failures.

The nominal touchdown footprint length available at a given nominal re-entry flight path angle is defined as the difference between the maximum range (from the re-entry point at 400,000 feet to the point where the spacecraft passes through 10,000 feet) and the minimum range; both subject to the skip and load factor constraints discussed above. The minimum dispersed footprint length is equal to the nominal footprint less the sum of the anticipated RSS dispersions on each end of the nominal footprint. The maximum dispersed range is equal to the nominal maximum range plus the anticipated Root Sum Squared dispersions. Within the maximum lift boundaries for 10 g's and 400,000 feet skip altitude, maximum range is attained by flying constant full lift. For shallow re-entry flight path angles between the 400,000 feet skip boundaries for constant full and constant zero lift, maximum touchdown footprint is attained by following an optimized lift variation, i.e., modulated roll attitude. Within these boundaries, for any lift fraction (other than zero), a modulated roll program is mandatory in order that re-entry constraints are not violated. The modulated lift re-entry in the region of more shallow re-entry flight path angles provides more range than that achieved by flying constant lift re-entries with identical re-entry constraints. In a similar manner (modulated roll), the minimum range may be shortened without violating the re-entry constraints.

Figure 4.2-3 shows the variation of the pertinent range and footprint parameters with re-entry flight path angle for a lunar return re-entry. These parameters are presented for L/D of .15 and .25. The assumed errors contributing to range and footprint dispersions are:

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TOUCHDOWN DISPERSION ANALYSIS WITH MODULATED LIFT LARGE EARTH ORBIT GEMINI

DISPERSION ANALYSIS BASED ON USE OF ON-BOARD VISUAL TRACKING DATA
(ASSUMES GROUND BASED TRACKING DATA NOT AVAILABLE-FAILURE MODE)

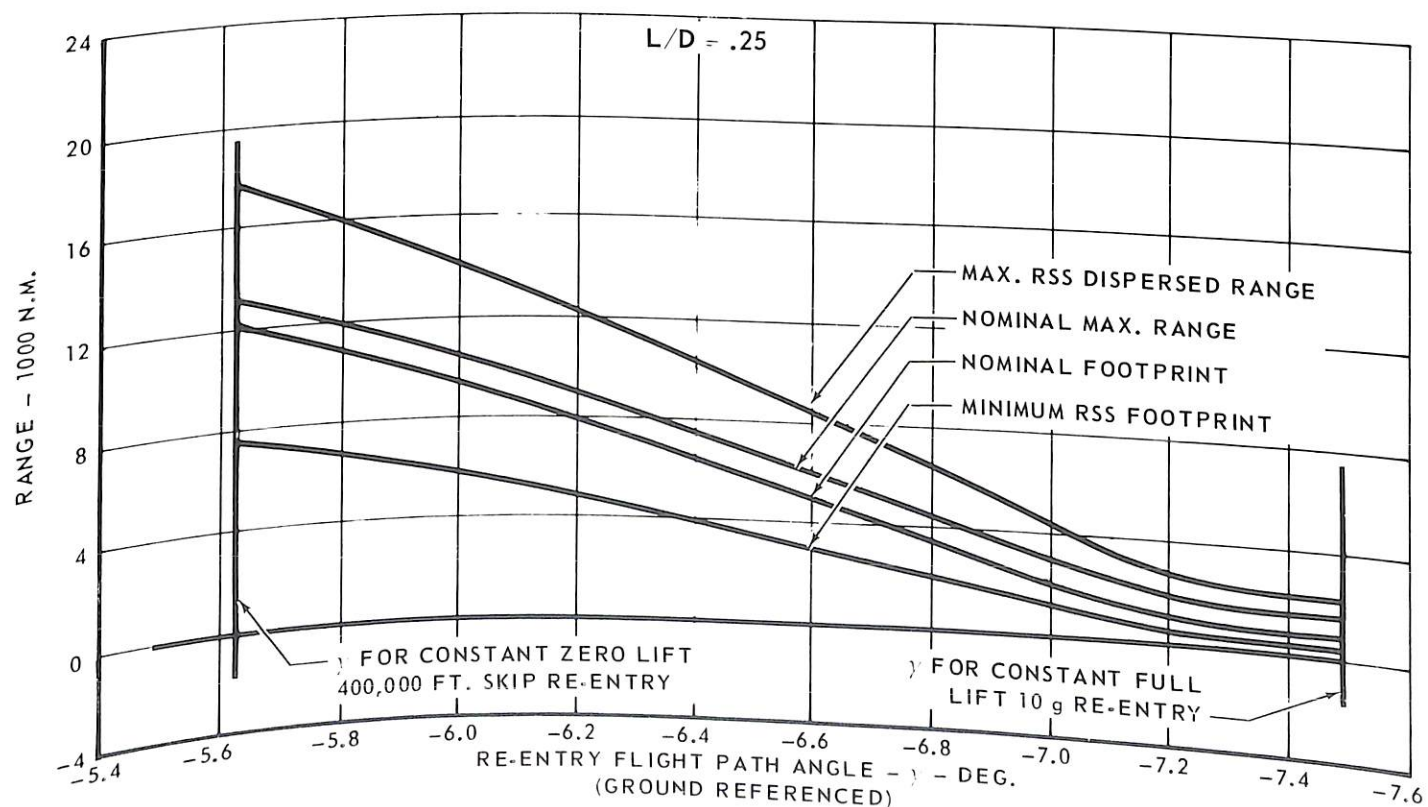
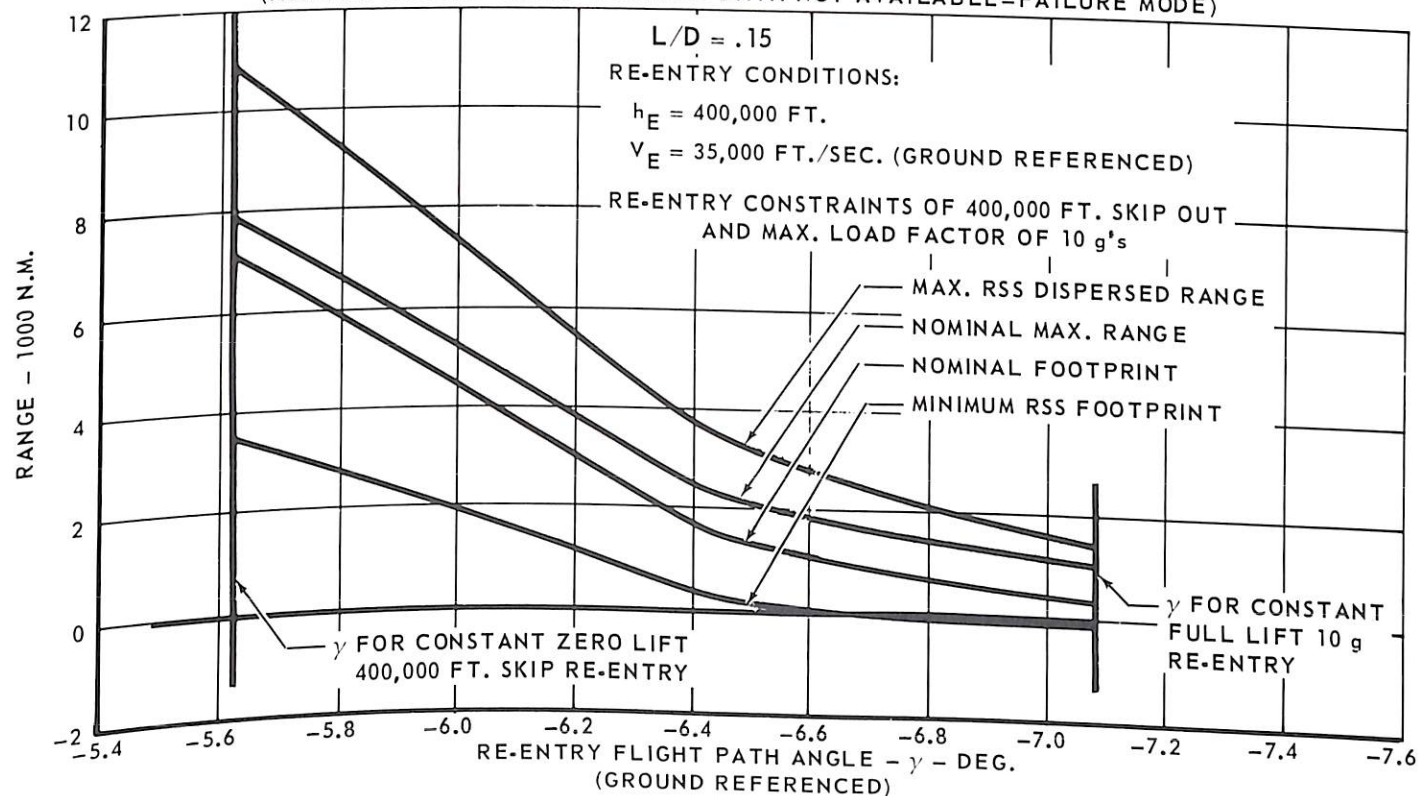


FIGURE 4.2-3

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4.2.2.2 (Continued)

Re-entry flight path angle (Failure condition - no ground tracking information. Midcourse corrections by on-board visual observations)	$\pm .2$ degree
C.G. Uncertainty	$\pm .1$ inch
Aerodynamic Uncertainty Error	± 25 per cent of ideal range
On-board Computer Error	± 10 per cent of ideal range when skip above 400,000 ft. is impossible.
	± 20 per cent of ideal range when skip above 400,000 ft. is possible.
Re-entry Position Error	± 73 n.m.

The range errors resulting from these contributions were Root Sum Squared to obtain the dispersions used in this analysis.

Figure 4.2-3 shows that the larger L/D and more shallow re-entry flight path angles result in greater footprint capability but also have larger maximum dispersed ranges. In order to minimize heat protection weight penalties, an L/D of .15 is preferred. The L/D of .15 results in angles of attack approximately five degrees less than that for L/D = .25 and, consequently, afterbody heating is reduced. In addition, the .15 L/D corresponds to the basic Gemini trim attitude for which aerodynamic flight test results are available. Obviously, within the flight regime where these data are applicable, it is possible to more confidently predict spacecraft performance.

The selection of a nominal re-entry flight path angle of -6.65 degrees for the .15 L/D trim results in the following re-entry characteristics when considering dispersions for the failure condition where ground tracking information is not available (midcourse corrections and navigation accomplished by on-board visual observations).

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4.2.2.2 (Continued)

- . Maximum dispersed range (from 400,000 ft.) = 2,600 n.m.
- . Nominal touchdown footprint length = 900 n.m.
- . Minimum dispersed footprint length = zero n.m.

The normal mission minimum dispersed footprint is increased to approximately 300 n.m. when ground-based tracking is available throughout the entire flight to update the on-board computer. For this case, the range of permissible re-entry flight path angles is still $\pm .2$ degree about the nominal. The actual flight path angle can, however, be determined to within $\pm .01$ degree for computer update.

The -6.65 degree nominal re-entry angle for $.15$ L/D trim provides flexibility in the re-entry control technique. In the event of emergency conditions where guidance displays necessary for modulated lift re-entries are not available to the crew, constant lift (attitude) re-entries are possible. Within the possible $\pm .2$ degree flight path angle band constant, maximum lift will result in re-entries which will not violate the 400,000 ft. skip or the 10 g boundaries. Even for a constant zero lift re-entry at the steepest re-entry flight path angle possible (-6.85 degrees), the maximum load factor should not exceed 15 g's. The load factor-duration profile for the re-entry is still within the acceptable human tolerance region as shown in Figure 4.2-1, and spacecraft structural limits are not violated.

4.3 Abort Considerations

4.3.1 General - The abort considerations for Configurations I and II tentatively identify the areas where existing abort modes may be used, define new abort modes, and identify the apparent problem areas. The abort considerations for Configurations I and II are summarized in Table 4.3-1.

4.3.2 Configuration I - The existing Gemini ejection seats are used for off-the-pad and low altitude escape. While this system retains the original 70,000 ft. design ejection capability, the current GLV ejection criteria are used; that is, a preferred mode upper altitude limit of 15,000 ft. and a permissible 45,000 ft. high altitude limit for some abort conditions.

For the launch phase altitude range of 15,000 ft. to 300,000 ft. no positive escape system is provided, but the launch vehicle reliability does justify consideration of a ride-it-out philosophy. The ride-it-out procedure requires additional study effort but it would likely consist of shutting down the booster and delaying a predetermined length of time before physically separating the spacecraft from the booster. In some of the upper portion of this altitude range, dynamic pressure may decay during the delay to a level where OAMS thrust would be helpful in separating the spacecraft. In the higher dynamic pressure region, the booster will assume large angles of attack and possible structural breakup following thrust termination. The spacecraft aerodynamic lift associated with angle of attack, however, does aid spacecraft lateral separation from the booster.

Above 300,000 ft., the spacecraft OAMS is used as the escape system after launch vehicle thrust is terminated.

For mission aborts that occur near the end of the launch phase, the spacecraft is reoriented after separation from the launch vehicle and a retrograde maneuver performed with the OAMS.

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TABLE 4.3-1
ABORT CONSIDERATION SUMMARY
CONFIGURATION I

METHOD OF ESCAPE	NOMINAL ALTITUDE AND TIME RANGE	COMMENTS
EJECTION SEAT	0 TO 15,000 FT. 0 TO 50 SEC.	SAME SYSTEM NOW USED IN GEMINI PROGRAM. DESIGN PROVIDES EJECTION CAPABILITY UP TO 70,000 FT. 15,000 FT. MAXIMUM ALTITUDE REFLECTS CURRENT UTILIZATION OF EJECTION SEAT IN GEMINI PROGRAM.
	15,000 TO 300,000 FT. 50 TO 187 SEC.	LAUNCH VEHICLE RELIABILITY PERMITS THE CONSIDERATION OF A RIDE IT OUT PHILOSOPHY
SPACECRAFT OAMS	300,000 FT. TO END OF LAUNCH PHASE 187 TO 340 SEC.	DYNAMIC PRESSURE NEGLIGIBLE. OAMS THRUST SUFFICIENT TO SEPARATE SPACECRAFT FOLLOWING LAUNCH VEHICLE THRUST TERMINATION. SPACECRAFT TURNAROUND AND OAMS RETROGRADE NECESSARY FOLLOWING ABORT DURING LATTER PART OF LAUNCH PHASE.
SPACECRAFT OAMS	LOW EARTH ORBIT	ABORT ACHIEVED BY SPACECRAFT SEPARATION (IF DOCKING HAS OCCURRED), AND OAMS RETROGRADE.
SPACECRAFT OAMS	DURING TRANSLUNAR INJECTION	ABORT CONSISTS OF THRUST TERMINATION, SPACECRAFT SEPARATION, OAMS RETROGRADE, AND OAMS ΔV CORRECTION AT APOGEE TO ACHIEVE DESIRED RE-ENTRY CONDITIONS.
SPACECRAFT OAMS	TRANSLUNAR TRAJECTORY	ABORT CAPABILITY IS DEPENDENT ON THE LENGTH OF TIME THE SPACECRAFT HAS BEEN IN A LUNAR TRAJECTORY AND THE MAGNITUDE OF THE AVAILABLE VELOCITY CHANGE. ANALYSIS INDICATES ABORT IS FEASIBLE FOR THE FIRST 30 HOURS OF THE MISSION.
CONFIGURATION II		
EJECTION SEAT	0 TO 15,000 FT. 0 TO 40 SEC.	SAME SYSTEM NOW USED IN GEMINI PROGRAM. DESIGN PROVIDES EJECTION CAPABILITY UP TO 70,000 FT. 15,000 FT. MAXIMUM ALTITUDE REFLECTS CURRENT UTILIZATION OF EJECTION SEAT IN GEMINI PROGRAM. ANALYSIS INDICATES EJECTION SEAT PROVIDES ADEQUATE ESCAPE CAPABILITY FOR TITAN IIIC APPLICATION.
LIQUID CORE STAGES I & II	15,000 TO 250,000 FT. 40 TO 150 SEC.	STAGE I OR II ABORT MAY BE FEASIBLE FOR ACHIEVING ALTITUDE WHERE SEPARATION WITH SPACECRAFT OAMS IS POSSIBLE.
TRANSTAGE	100,000 TO 250,000 FT. 100 TO 150 SEC.	ALTERNATE ABORT TECHNIQUE IN THIS ALTITUDE RANGE. TRANSTAGE PROPULSION TO ATTAIN ALTITUDE WHERE SPACECRAFT OAMS SEPARATION IS POSSIBLE. REQUIRES TRANSTAGE FIRE-IN-THE HOLE CAPABILITY.
SPACECRAFT OAMS	250,000 FT. TO END OF LAUNCH PHASE 150 SEC. TO INJECTION	DYNAMIC PRESSURE NEGLIGIBLE. OAMS THRUST SUFFICIENT TO SEPARATE SPACECRAFT FOLLOWING LAUNCH VEHICLE THRUST TERMINATION. SPACECRAFT TURNAROUND AND OAMS RETROGRADE NECESSARY FOLLOWING ABORT DURING LATTER PART OF LAUNCH PHASE.

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4.3.2 (Continued)

If the mission is aborted prior to the final powered phase for the translunar trajectory, the OAMS is used to perform a retrograde maneuver. To obtain acceptable re-entry conditions from a 100 n.m. circular orbit, a retrograde velocity increment of approximately 100 fps is sufficient.

Abort during the final powered phase of the translunar injection is accomplished by terminating thrust, separating the spacecraft from the propulsion stage with the OAMS, reorienting the spacecraft, and performing a retrograde ΔV and a small velocity change at apogee. This combination provides acceptable re-entry conditions for this type of abort.

Abort after translunar injection is feasible for only a portion of the translunar trajectory as shown in Figure 4.3-1. The technique is similar to that discussed in the previous paragraph. With the anticipated ΔV capability available, aborts out to 30 hours after lunar injection shorten the mission time. After 30 hours, completion of the circumlunar trajectory results in the shorter mission time.

4.3.3 Configuration II - The escape system for the off-the-pad and altitudes to 15,000 ft. is the same as for Configuration I discussed in Paragraph 4.3.2.

For the launch phase between 15,000 ft. to 100,000 ft., an abort procedure is not defined. Additional study effort is required in this area. It is believed that, for some booster malfunctions, abort is feasible with the liquid core Stages I and II; for example, a Stage 0 malfunction and subsequent thrust termination that could be "flight" controlled by ignition of Stage I. This would permit the attainment of launch trajectory altitude (low dynamic pressure) where spacecraft separation with OAMS is possible. Similarly, Stage II propulsion might be used for Stage I malfunctions, but might be somewhat limited by its separation performance under a moderate dynamic pressure environment.

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LARGE EARTH ORBIT GEMINI
TYPICAL ABORT CAPABILITY AFTER TRANSLUNAR INSERTION

REF: NASA TN D-1655

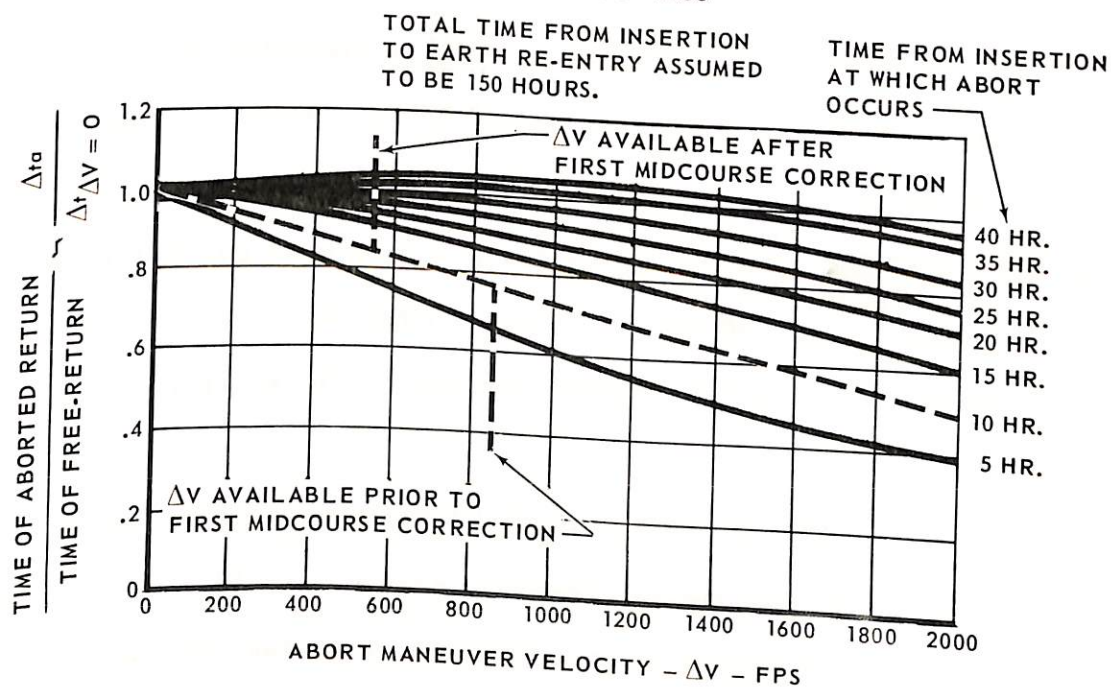


FIGURE 4.3-1

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4.3.3 (Continued)

For nominal launch trajectories, the OAMS propulsion is adequate to accomplish aborts, necessitated by booster malfunctions, which occur later than Stage I burnout.

For the launch phase between 100,000 and 250,000 ft., the Transtage was evaluated as an escape system from a non-thrusting launch vehicle. The separation characteristics afforded by the Transtage propulsion system are shown in Figure 4.3-2 for "flight attitude" malfunctions at three altitudes. The lower three sigma thrust history of the Transtage engines was used and times are referenced to the firing signal to the engine. While separation appears feasible at the 100,000 ft. altitude, safety reliability is believed to be enhanced by riding out failures which occur late in the solid motor burn. This procedure is considered primarily because of the indeterminable loads that may be induced by the exhaust from the thrust cancellation ports of the solid motors as the spacecraft passes near them. Solid motor thrust termination can be initiated any time during solid burn. Certain modifications to the launch vehicle are known to be necessary, and other areas require additional investigation before the feasibility of a Transtage escape configuration can be adequately assessed; these include:

- A. Modifications required to provide Transtage "fire-in-the-hole" capability.
- B. Possible bind-up on the Transtage separation rails under lateral loads.
- C. Propellant settling requirements for proper engine ignition after thrust termination and deceleration due to aerodynamic drag.
- D. Structural adequacy of Stage II tanks under Transtage exhaust plume.
- E. Propellant impact loads in the Stage I, Stage II and Transtage.

Aborts during the final powered flight and after translunar injection are the same as discussed in Paragraph 4.3.2 for Configuration I.

4.3.4 Abort Rockets - Abort rockets could be used to improve the abort capability in the altitude range where the ride-it-out philosophy is proposed. Table 4.3-2 lists the changes and deletions necessary to reduce the weight sufficiently to

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LAUNCH ESCAPE USING TRANSTAGE
CONFIGURATION II

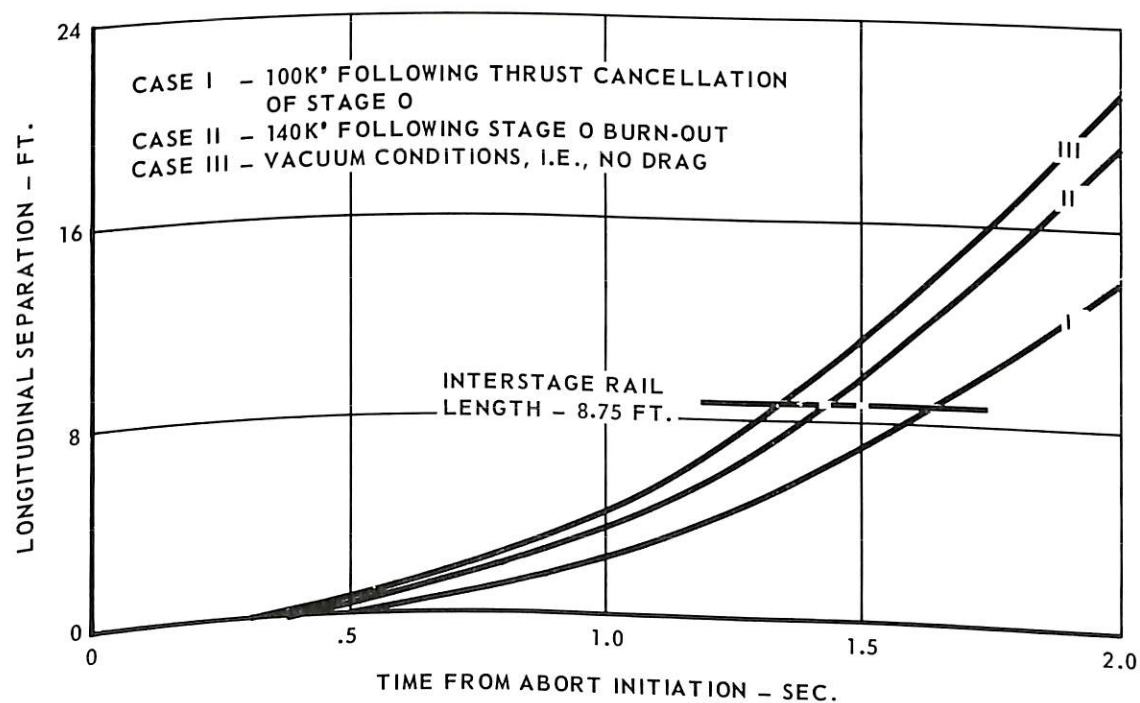


FIGURE 4.3-2

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TABLE 4.3-2

ESTIMATED WEIGHT SAVING FOR POSSIBLE CHANGES TO CONFIGURATION 1

ITEM	WEIGHT (POUNDS)
REDESIGN EGRESS KITS AND OPTIMIZE EJECTION SEAT ROCKET THRUST ANGLE	-40
CHANGE TO "PANCAKE" TYPE ELECTRICAL CONNECTORS	-20
REMOVE LANDING GEAR FITTINGS	-15
REMOVE QUICK ACCESS PROVISIONS TO LANDING GEAR BAYS	-4
REMOVE EPOXY COVERING FROM FLOTATION MATERIAL	-5
ELIMINATE MANUAL SHUT-OFF VALVES FROM COOLANT SYSTEM	-6
ELIMINATE LIQH CANISTER FOR SPECIFIC MISSION LENGTHS	-28
REDESIGN LIQH CANISTER FOR SPECIFIC MISSION LENGTHS	-25
ELIMINATE ALL PARAGLIDER DESIGN REQUIREMENTS AND MOVE EQUIPMENT TO OFFSET C.G. CHANGE	-4
INCORPORATE LIGHT WEIGHT CRYOGENIC DISCONNECTS	-3
REMOVE Z100 SEPARATION SENSING SWITCHES	-150
TOTAL	

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4.3.4 (Continued)

add just two Cherokee rockets. In addition, it would be necessary to remove at least 100 lbs. of contingency propellants. These changes are necessary if the Gemini spacecraft is to remain within the launch capability of the Titan vehicles.

Two TE-M-146 (Cherokee) solid propellant rocket motors submerged in the adapter provide approximately the same escape capability during ascent as is realized in the Gemini "A" program with the four TE-M-385 retrograde motors. A comparison of the escape characteristics of the two Cherokees with the four Gemini "A" retrograde motors is shown in the following tabulation.

<u>Rocket Configuration</u>	<u>Lateral Separation Distance, Ft.*</u>	<u>Longitudinal Separation Distance, Ft.</u>	<u>Time to Achieve Separation, Sec.</u>
2 Cherokees	39	3.4	1.0
4 Gemini Retrograde	11	16	2.8

*Based on time to start "fall-back"

Pertinent characteristics of the Cherokee motor are shown below for operation at 60°F, sea level.

Thrust, lb.	9,580
Total Impulse, lb.-sec.	11,760
Burn Time, sec.	1.1
Loaded Weight, lb.	74.1
Manufacturer	Thiokol

REFERENCES

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4.4 Experiments

4.4.1 Television - A television system on this mission is for observation of earth and lunar surfaces, evaluation of television by comparison with photographs, and monitoring the rendezvous technique for Configuration I.

4.4.1.1 Assumptions and Requirements

- A. The television system must use the slow scan technique employed in the Ranger, Tiros and Nimbus systems to reduce bandwidth and provide high resolution. Investigation has shown the feasibility of employing a slow scan, high resolution television system.
- B. A capability must be provided for storage of 30 to 40 minutes of video information for mission intervals that lack real time coverage.
- C. The system must be compatible with the unified S-band system and a slow-speed video tape recorder to minimize the weight.
- D. The camera lens focal length and the shutter rate must also be fixed to reduce complexity.
- E. This system is mounted on the adapter and folded out once orbit is achieved.
- F. The camera is mounted to "look" forward and make the same observations as the crew.

4.4.1.2 Description - The camera, electronics, and video recorder are mounted in the spacecraft adapter. The system has the capability of operating real time or in the stored video modes. The camera has two scan rates to optimize bandwidth usage during the real time and recorded modes. The reduced scan rate for the record mode allows the video tape recorder to operate at a maximum speed of .313 fps. The real time or playback of tape video transmissions are conducted subsequent to ranging operations since the television and ranging share the same frequency spectrum.

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4.4.1.2 (Continued)

A summary of the system characteristics that are of Ranger quality is given in Table 4.4-1.

TABLE 4.4-1
SUMMARY OF TELEVISION SYSTEM CHARACTERISTICS

WEIGHT	15 LB. (CAMERA AND ELECTRONICS)
POWER	20 WATTS
LINES/FRAME	1150
RESOLUTION ELEMENTS	1000 x 1000
BANDWIDTH	40KC OR 400KC
FRAME RATES	2 SEC. OR 20 SEC.
GRAY SHADES	8

The tape recorder weighs eight lbs., has playback capabilities, and a one hour record capacity. The recorder is mounted in the adapter and provides outputs to the VHF (near earth) or the unified S-band (deep space) systems.

4.4.2 Cameras - Provisions are made for stowing camera on-board suitable for recording pilot activity and obtaining still photographs of the earth and lunar surfaces. Sixteen mm., 35 mm. and 70 mm. cameras with film and lenses selected for optimum use are required.

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4.5 AGE Requirements - The AGE requirements and system test plans follow as much as possible the existing Gemini AGE and system test plan. New concepts, except where launch facilities demand, of AGE and test are not considered.

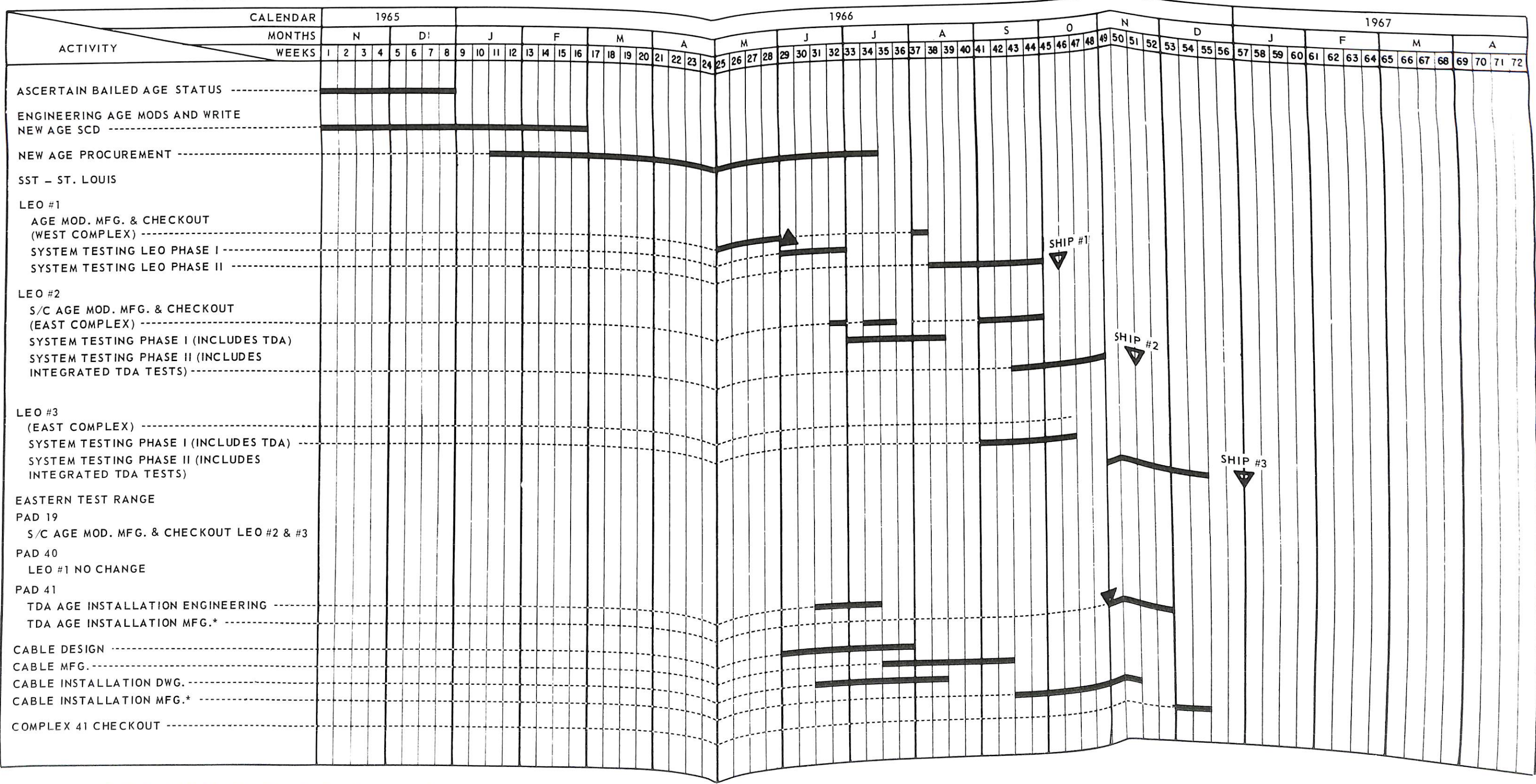
The changes to Spacecraft Systems Tests (SST) (St. Louis) AGE and system test plans are minimal. The majority of SST changes are those introduced by spacecraft system changes and are of the SCD type AGE installation. Cape AGE changes and test plans are influenced by the same spacecraft system changes as influenced SST and the type of launch complex used. Configuration I has approximately 30 per cent of the AGE complexity (amount of redesign and reinstallation) as has Configuration II. Configuration II requires a major redesign and reinstallation of AGE, plus new designed AGE because of the remote monitor/control concepts used during the countdown and the necessity for mobile AGE (Van mounted). Configuration II requires approximately 40 per cent of the effort expended on Gemini "A". Configuration I requires no remote monitor/control techniques and the AGE is fixed (non-Van mounted). Figures 4.5-1 and 4.5-2 depict major AGE tasks for Configurations I and II, respectively. Table 4.5-1 gives a brief description of AGE changes for both configurations. Figures 4.5-3 and 4.5-4 give in bar chart form typical launch complex activities.

4.5.1 General Assumptions and Criteria - Those assumptions and/or ground rules made in order to develop the following AGE/Test Plans are listed below. Assumptions made for a particular configuration are listed with that configuration plan.

- A. All Electronic-Electrical AGE, except that due to spacecraft change or type of launch complex used, requirements are filled from AGE now located in MILA (O/C) building. Minimum modifications to accommodate the mission requirements are planned.

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AGE MODIFICATION/TEST PLAN CONFIGURATION I



*INTEGRATING CONTRACTOR RESPONSIBILITY **LAST TWO WEEKS IN ALTITUDE CHAMBER

- ▲ FABRICATE AN ADDITIONAL SET OF CANNED MAN AGE & CABLES
- ▼ GEMINI "A" TDA AGE AVAILABLE.

FIGURE 4.5-1

AGE MODIFICATION PLAN - CONFIGURATION II - PAD 40

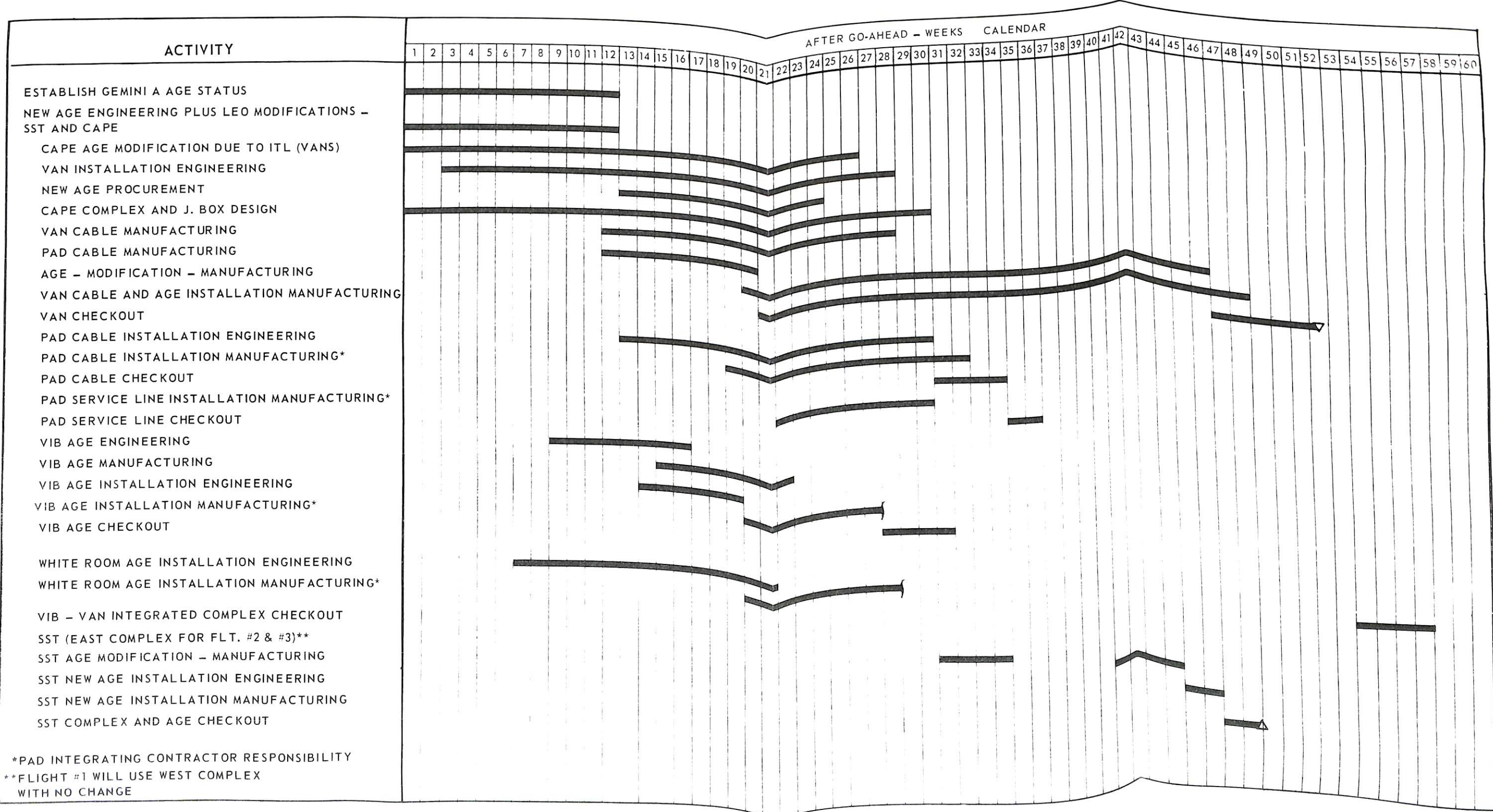


FIGURE 4.5-2

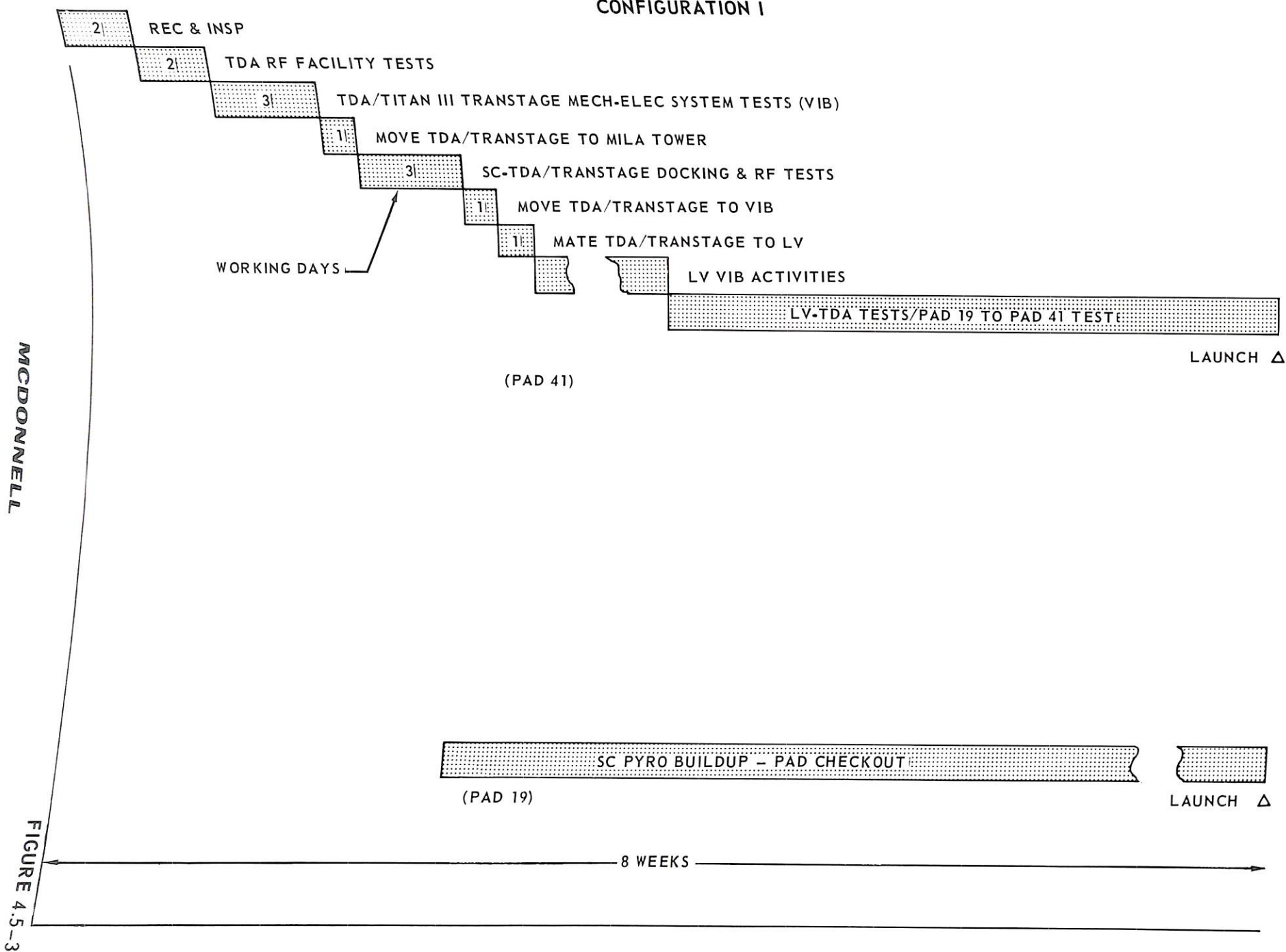
TABLE 4.5-1
AGE REQUIREMENTS VS CONFIGURATION

AGE (BY SYSTEMS)	CONFIGURATION I (A)			CONFIGURATION I' (B)		
	REND.-GLV/SC TITAN III/TDA - 3 F/C			DIRECT - TITAN III 3-F/C		
	SST	CAPE SC-PAD 19 TDA-PAD 41		SST	CAPE PAD 40	
INSTRUMENTATION	NC	NC	-	NC	NC-V ①	
COMMUNICATION	N ₂ + RADAR AGE	N ₂ + RADAR AGE	B14	N ₂ NC	N ₂ -V-NC ①	
GUIDANCE AND CONTROL	M ₅	M ₅	-	M ₅ N ₂	V-R-M ₅ N ₄ -L ₃	
ECS	NC	NC	-	NC	V-R-L ₂	
RCS	M ₂	M ₂	-	M ₂	V-L ₂ -R M ₂	
RSS	M ₂	M ₂	-	M ₂	V-R-L ₂ M ₂	
POWER & SEQUENTIAL	M ₁	M ₁	B14	M ₁	V-R-N ₄ L ₄ -M ₁	
CABLING	NC	NC	N-PAD 14	NC	N?	
SERVICE LINES	NC	NC	-	NC	N ②	
SERVICE TRAILERS	NC	NC	-	NC	NC	
MECHANICAL- TRANSPORT DOLLIES, ETC.	N ₁	NC	B14	NC	NC	
SPECIAL	NC	④	B14	NC	N ₃ -L ₂ -V ③	
VANS	-	-	-	(N.R.)	N ₄	

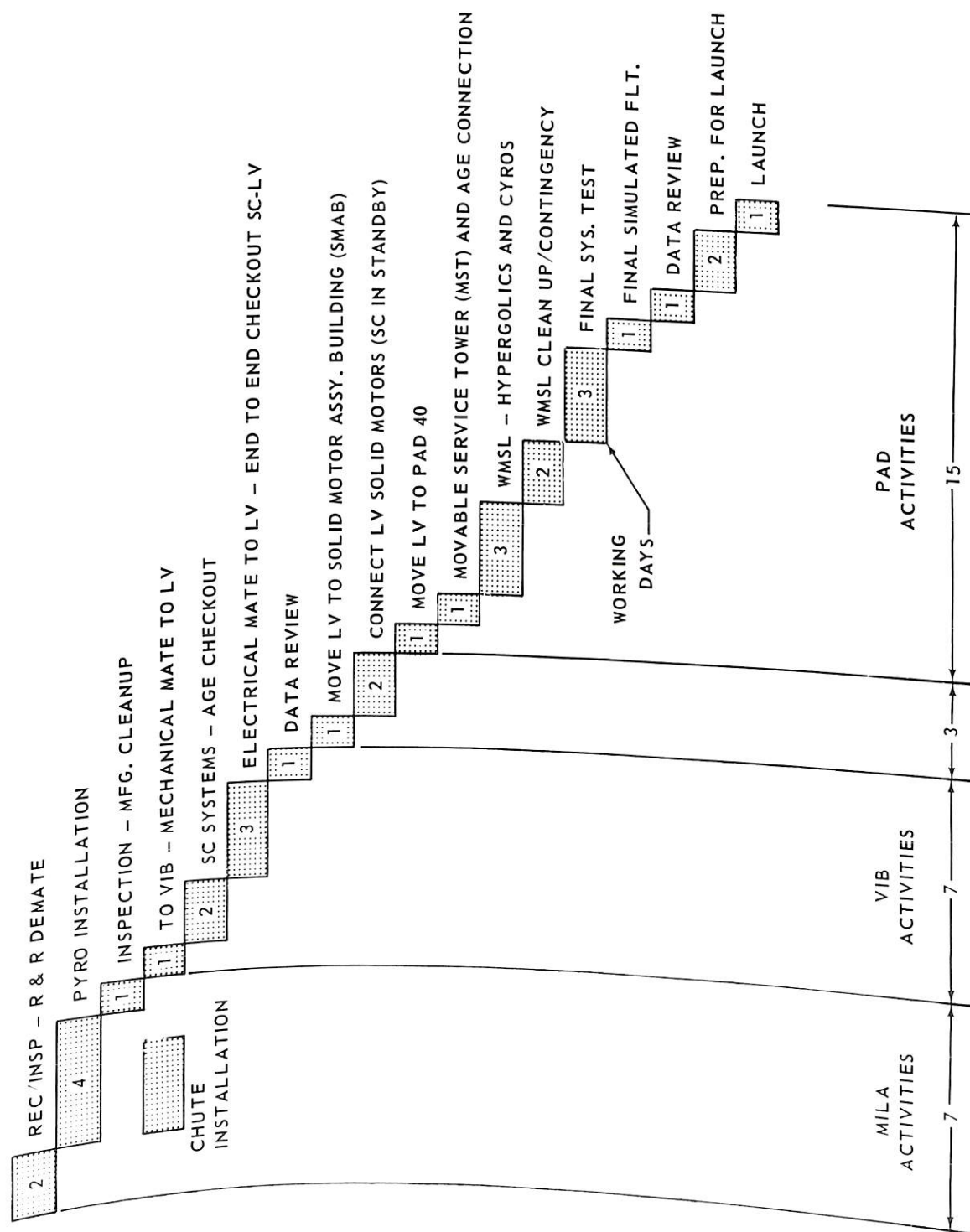
- ① - SOME AGE NOW IN O/C (MILA-NASA) USABLE AS IS IN O/C
 ② - NEW SERVICE LINES AND OTHER FIXED SERVICING AGE REQUIRING
 INSTALLATION BY INTEGRATING CONTRACTOR.
 ③ - REMOTE CONTROL AGE (GFE)
 ④ - POSSIBLE MOD. OF W. RM. DUE TO ELONGATED GLV.
 A-B - RELATIVE DECREASING AGE COST.

- NC - NO CHANGE
 V - VAN MOUNTED
 R - REPACKAGED
 MX - MODIFIED-RACKS
 NX - NEW-RACKS
 L - VIB LOCATION
 B14 - BAILED FROM PAD 14
 (NR) - NOT REQUIRED

LAUNCH PADS ACTIVITIES CONFIGURATION I



LAUNCH PAD ACTIVITIES - PAD 40
SPACECRAFT CONFIGURATION II



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FIGURE 4.5-4

REFERENCES

4.5.1 (Continued)

- B. Any AGE that can be kept in O/C and still support the launch checkout are retained. Some Communications instrumentation, and Aero Med AGE are in this category.
- C. All portable AGE and portable/mobile service equipment now used on Pad 19 are available.
- D. All erector fixed AGE or service equipment, applicable to this mission, will be purchased new if launch is from any pad other than 19. This type of AGE is any that requires bolting, welding, or otherwise permanently attaching to structure that must be installed by the pad integrating contractor.
- E. Complex cabling is assumed to be of a new simplified design. Because of the similarity to Pad 19's AGE end items, the complexity and number of cables for Pad 40 will approximately be the same as for Pad 19. However, because of the change in cable lengths and Van mounting of AGE racks, the cabling requires an extensive redesign effort. Consequently, the majority of the cabling will be new. The Gemini cabling group has partially developed new design and manufacturing techniques for cables and junction boxes. This new technique will be used wherever existing Gemini cables are not directly applicable.
- F. If existing PIA equipment or standard laboratory equipment is not sufficient for PIA, a monitored Acceptance Test at the Vendor will be sufficient. Repair and retest of equipment in these cases will be performed by the Vendor.
- G. Flight No. 1 will be launched from Pad 40 with no changes required to the AGE then existing on Pad 40.

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4.5.1.1 Test Plans - Test plans for SST will be identical to those now practiced or planned for Gemini "A". Cape test plans will be identical to those now used with modifications to accommodate the mobile van and remove control/monitor concepts.

4.5.2 Configurations - Configurations I and II require the use of two launch complexes with separate AGE configurations. The following paragraphs describe each configuration.

4.5.2.1 Configuration I - Configuration I requires the use of Pad 19 and Pad 41. Pad 19 AGE changes are limited to those required to satisfy spacecraft primary system changes associated with the Communications, Guidance, and Fuel Cell systems. Minor changes in other systems will be required. Minimum fixed AGE on Pad 41 is required for the TDA. Remote monitoring/controlling AGE is not required. A new transport dolly for the TDA/adaptor is required.

The criteria used for the above plan is:

- A. The TDA has the same basic electrical-electronic systems as does the Gemini TDA.
- B. The TDA AGE will utilize the present Pad 14 equipment as modified for this mission.

4.5.2.2 Configuration II - Configuration II requires the use of Pad 40. This pad requires mobile AGE and remote launch control techniques which results in repackaging Pad 19 AGE, plus new AGE for remote control and monitor. Radar AGE is not required.

4.5.3 Conclusion - Configuration I is the most economical in both AGE hardware and engineering. The added complexity of dual launches will not be detrimental to the rendezvous mission in that considerable experience will have been obtained from the Gemini-Agena launches.

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4.6 MAC/St. Louis Ground Test Plan - A minimum ground test program is anticipated in order to qualify modified and reconfigured design areas.

The following is a brief test plan by testing categories:

4.6.1 Wind Tunnel Tests - Tests will be run in order to determine aerodynamic drag and stability as well as convective heat transfer in a Mach number regime of 5 to 40, with primary emphasis being placed on the 25 to 40 Mach number range.

4.6.2 Structural Tests - Structural tests of the re-entry module will be primarily directed at qualifying structure which has been modified for thermal protection purposes. Addition of propulsion, ECS, and electric power modules necessitates additional static tests of the adapter. Docking and maneuvering load distribution from the TDA to the Transtage will also be verified by test for the rendezvous configuration.

4.6.3 Structural Dynamic Tests - Dynamic response investigations will be made of the mated re-entry module and equipment adapter with dummy equipment installed. Redesigned equipment modules will also be subjected to launch spectrum vibration in order to substantiate the structural dynamic integrity of the new module design. For the rendezvous version, dynamic response tests in the moored configuration will be performed to verify the new docking adapter structural configuration.

4.6.4 Ablative Material Tests - Plasma jet testing will be performed on samples of thermal protection material to determine ablative effectiveness and other thermal characteristics for a range of expected heat fluxes. In some cases, vibration tests will be combined with thermal input to check bond and char adhesion.

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REFERENCES

4.6.4 (Continued)

Gaseous radiation tests will be performed at NASA/Ames facility to determine influence of ablative products on flux from the shock region ahead of free flight pellets. Tests at Ames will also be run to determine ablative effectiveness when exposed to several levels of combined convection and gaseous radiation. The shear stress influence on S-3 effectiveness will be determined for typical front face heating rates. Tests will be performed on S-6 to determine shear stress resistance for expected afterbody fluxes. An outline of ablation tests is shown in Table 4.6-1.

4.6.5 Environmental Control System Tests - Manned and unmanned 7-day mission ECS tests are anticipated in order to perform component and system checkout. Crewman simulator tests will be performed to check O₂ and H₂O supplies and the LiOH charge. A functional test of the Ion Exchange Column and H₂O management system will be run with an estimated electrical load profile. Water analysis will be made at the beginning and the end of the test runs. Additional ECS testing will be incorporated into Spacecraft 3A thermal qualification testing.

4.6.6 Spacecraft 3A Thermal Qualification Tests - Thermal qualification testing will consist of one unmanned 7-day duration test investigating three different spacecraft orientations. The primary objective of the new testing is to qualify the radiator system for a revised solar heating input due to a change in flight technique. New equipment simulation and line routing changes will also be incorporated.

4.6.7 Functional Tests - Mechanism functional tests will be performed on such systems as the DSIF dish antenna, the sun sensor, the star sensor, etc., to develop and perfect mechanism rigging procedures as well as to demonstrate alignment repeatability. Electronic equipment components will be mock-up simulated.

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TABLE 4.6-1
OUTLINE OF REQUIRED ABLATION TESTS

SUBJECT	FACILITY	PURPOSE
1. GASEOUS RADIATION FLUX	NASA-AMES	a. DETERMINE INFLUENCE OF S-3 ABLATIVE PRODUCTS ON FLUX FROM SHOCK REGION AHEAD OF FREE FLIGHT PELLETS. b. COMPARE WITH PREVIOUSLY PUBLISHED MEASUREMENTS AND DESIGN THEORY.
2. GASEOUS RADIATION INFLUENCE ON S-3 EFFECTIVENESS	NASA-AMES	a. DETERMINE S-3 ABLATIVE EFFECTIVENESS WHEN EXPOSED TO SEVERAL LEVELS OF COMBINED CONVECTION AND GASEOUS RADIATION USING AN ARC IMAGE SUPERIMPOSED WITH VACUUM PLASMA JET HEATING. b. CHECK EFFECTS OF A CATALITIC SURFACE ON MEASURED INPUT FLUX LEVELS TO CALORIMETERS.
3. SHEAR STRESS INFLUENCE ON S-3 EFFECTIVENESS	MAC AND/OR VIDYA	a. DETERMINE S-3 EFFECTIVENESS AT HIGH HEATING RATES WHEN EXPOSED TO A DEFINED HIGH SHEAR STRESS PLASMA JET FLOW. b. MEASURE AMOUNTS OF CONVECTION, GASEOUS RADIATION, AND TOTAL HEAT FLUX. c. CHECK EFFECTS OF A CATALITIC SURFACE ON THE MEASURED INPUT FLUX LEVELS TO CALORIMETERS.
4. S-6 EFFECTIVENESS FOR AFTERBODY HEAT FLUXES.	MAC	a. DETERMINE ABLATIVE EFFECTIVENESS OF S-6 FOR RANGE OF HEAT FLUXES EXPECTED ON AFTERBODY SURFACES. b. DETERMINE THERMAL CONDUCTIVITY VS. TEMPERATURE AND CHAR DENSITY.
5. AFTERBODY ABLATIVE PANELS AND JOINTS	MAC	a. QUARTZ LAMP AND/OR PLASMA JET HEATING OF VARIOUS SIZED AFTERBODY PANELS TO CHECK EXPANSION BEHAVIOR AT JOINTS AND SCREW PLUGS. b. VIBRATION TESTS IN COMBINATION WITH QUARTZ LAMP HEATING TO CHECK BOND ADHESION AND CHAR ADHESION.

4.6.8 Pyrotechnic Tests - Pyrotechnic system demonstration tests will be conducted to qualify all new pyrotechnic devices and requalify separation devices affected by structural design changes. Tests will be performed on both actual spacecraft structure and boiler plate simulators. The purpose of these firings will be to demonstrate system compatibility using actual spacecraft structural components and equipment mounting. Some flight environment test considerations will be included.

4.6.9 Parachute Drop Tests - Modification of the R & R section and the related main parachute repackaging necessitates supplementary parachute system drop tests. Three development bomb drops and five qualification drops using Static No. 7 are anticipated.

4.6.10 Electronic Tests - Testing of the DSIF system (transponder, power amplifier, and diplexer) will be performed to confirm interface compatibility. The directional antenna will be tested to confirm its performance and spacecraft structural compatibility.

4.6.11 Guidance and Control - The three-axis table test will serve two primary purposes: one, to verify the compatibility of the guidance and control equipment unique to this spacecraft and target vehicle; and secondly, to examine the effects of dynamic vehicle motion on the operation of this equipment. The coupling existing in the guidance and control system when functioning dynamically can result in unsatisfactory operation. This test will be the basis for determining and eliminating such problem areas.

In the moored configuration for the rendezvous version, the operations of the guidance and control equipment with the combined elastic vehicle will be determined. If alteration of the control system is required for proper operation, these tests will provide the necessary data to modify the system.

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