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SPECIFICATION NO. 1417524
DATE 8 January 1965

[redacted]

[redacted]

PROGRAM [redacted] SATELLITE SYSTEM

SPECIFICATION (U)

Downgraded at 12 Year
Intervals Not Automatically Declassified

PRELIMINARY DRAFT

APPROVED BY ____________________ [redacted] Chief Systems Engineer

APPROVED BY ____________________ [redacted] Program Manager

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C., SECTIONS 793 AND 794. THE TRANSMISSION OR REVELATION OF WHICH IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

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1. SCOPE

1.1 Scope. This specification defines the capability and performance requirements for the Program [redacted] Satellite System. The Program [redacted] Satellite System major elements as defined in this specification shall be the Thor booster, the Agena orbital vehicle, the radio guidance system, and the primary payload.

1.2 Function. The functions of the Program [redacted] Satellite System shall be to place and support in orbit primary and secondary payloads. In addition, the system shall provide for the recovery of two capsules from orbit.

1.3 Descriptive Title. The descriptive title of the total system is Program [redacted] Satellite System, herein called the system.

1.4 Contractual Limitations and Variations. This specification shall not be intended to change, and shall not change, the contractual obligations of Lockheed Missiles and Space Company or those of other contractors, except as to the issuance of this system specification, LMSC 1417524, but shall serve only to coordinate into a system specification the Program [redacted] activities of LMSC and other contractors, as directed by [redacted] to Contract [redacted]

2. APPLICABLE DOCUMENTS

2.1 Government Documents. The following government documents, with issue dates as listed below, form a part of this specification to the extent specified herein:

Standards

MIL-STD-442 12 December 1957	Cable (Wire) Two Conductor Parallel (Ripcord)

Publications

ANA Bulletin No. 438b 15 September 1964	Age Controls for Synthetic Rubber Parts
Air Force Missile Test Center Regulation 80-7 25 September 1958	Airborne Flight Termination Systems (Range Safety)

(Copies of specifications, standards, drawings, and publications as required by Contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

2.2 Other Documents. The following non-government documents with issue dates as listed below form a part of this specification to the extent specified herein:

Lockheed Missiles and Space Company (LMSC)

Specifications

1345097B 24 March 1964	Agena D Aft Section, Program Space Allocations
1414691C 17 January 1964	Specification, Satellite Control Subsystem for Program [redacted] Satellite Vehicles 39205, 1601 and Up
1416559A 15 August 1964	Detail Specification, Model 39205, Program [redacted]
T-3-3-001	Design Control Specification Primary Payload
T-3-3-004	Payload Interface Specification
T-3-4-001	Design Requirements Specification, Primary Payload

Drawings

1324216	Vehicle/Payload Mechanical Interface
1345366B	LV-2A, SLV-2/Standard Agena Mechanical Interface
1358140	Secondary Payload Mounting Hole Patterns
1363999	Secondary Payload Mounting Hole Patterns
1365461	Antenna Configuration Drawing, Guidance
1397367	Paint and Marking
1398571	SLV-2A/S-01E Electrical Interface
1399138	Agena/AGE Interface
2P10762B (LMSC, RP)	Standard Research Payload Aft Rack Equipment Panels

Publications

A068386 21 June 1963	Umbilical and Test Plug Pin Assignments
A069558B 7 August 1963	Reliability Program Plan, Program [redacted]
A315106B 30 December 1963	Trajectory Performance, Model LV-2A/O1B
A381275 12 July 1963	Test of Attainment Reliability Requirements and Goals
A705291 1 November 1964	Program [redacted] Quarterly Reliability Estimate and Analysis Report
A386369 SDR-6	Program [redacted] System Design Requirements for Vehicle Payload Interface

Douglas Aircraft Company (DAC)

Specifications

DS-2345A 1 July 1964	Detail Specification DSV-2C (SLV-2A) Space Booster

Drawings

1A18680	Standard Instrumentation Kit First Stage Booster
1B07688	Flight Controller
7837664	Multicoder High Level PDM

Western Electric

Specifications

G.S. 19-900

G.S. 64-250

Inter Range Instrumentation Group

IRIG Document

106-60

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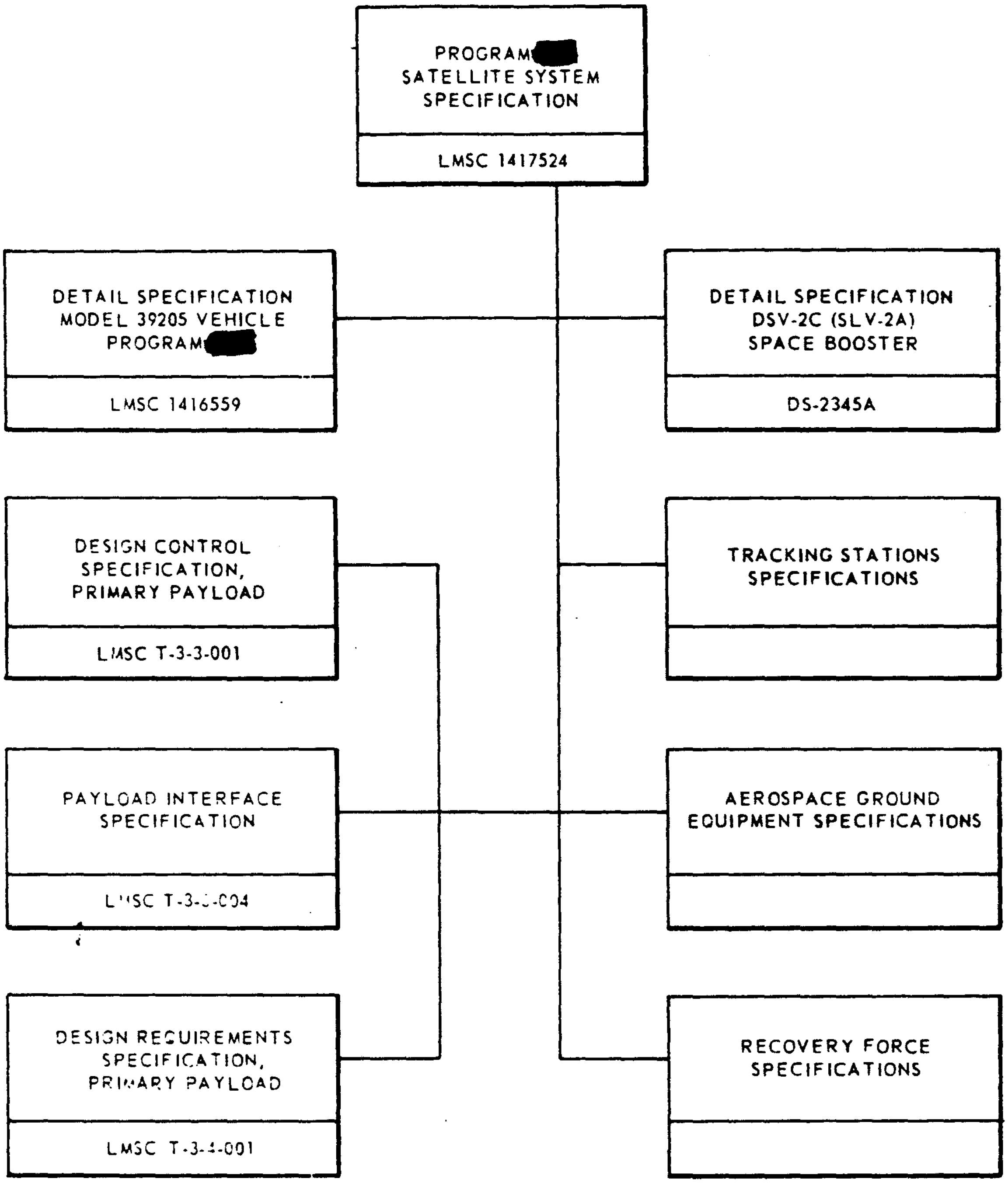
(Applications for copies of non-government documents should be addressed to the appropriate organization at the following addresses:

a. Lockheed Missiles and Space Company
P.O. Box 504
Sunnyvale, California

b. Douglas Aircraft Company, Inc.
Missile and Space Systems Division
Santa Monica, California

c. Western Electric Company, Inc.
Defense Activities Division
North Carolina

2.3 Specification Tree. The specification tree shown in Figure 1 depicts the first sublevel of Program [redacted] Satellite System specifications.



[illegible] . Specification [illegible]

3. REQUIREMENTS

3.1 Operational Performance.

3.1.1 Primary Mission. The primary mission of the system shall be to place into orbit and to support a primary payload with objectives as specified in section 3.1.15 of this specification.

3.1.2 Secondary Mission. The secondary mission of the system shall be to provide accommodations for secondary payloads as specified in section 3.1.16 of this specification. The secondary payloads shall be flown on the basis of non-interference with the primary payload mission.

3.1.3 Mission Capabilities.

3.1.3.1 Orbit Parameters. The system shall be capable of a range of missions with orbit parameters generally within the following limits:

a. Period: 88 minutes to 92 minutes

b. Inclination: 60 degrees to 140 degrees

c. Height of Perigee: 80 nautical miles to 220 nautical miles.

The effects of mission selection on the weight-in-orbit capability shall be as specified in detail in section 3.1.7 of this specification.

3.1.3.2 Mission Selection. The capability of the system to incorporate or change missions within the orbital parameters specified in section 3.1.3.1 shall be limited as follows:

3.1.3.2.1 New Missions. Incorporation of a new mission not previously flown and without complete prelaunch data shall require 22 working days prior to launch.

3.1.3.2.2 Repeat Missions. Repeating a previously flown mission with complete prelaunch data shall require 17 working days prior to launch.

3.1.3.2.3 Alternate Mission. An alternate mission shall be interchangeable with a specified primary mission up to 17 working days prior to launch.

3.1.3.3 Mission Duration. A nominal mission shall consist of a total of eight active days of flight. During the active periods the orbital vehicle shall be attitude stabilized with command links operating and with the orbital programmer controlling the payload and orbital housekeeping functions. The system shall be capable of entering a passive period lasting up to 11 days, either during or immediately following the active flight. The system functional description is included in section 3.1.5.

3.1.3.4 Recovery Capability. The system shall provide both primary and independent backup subsystems for ejecting recoverable capsules. Detail of the recovery system shall be as specified in section 3.1.12 of this specification.

3.1.4 Flight System Elements. For the purposes of this specification, the flight system shall consist of the Thor booster, the radio guidance system, the Agena orbital vehicle, and the primary payload. These flight system elements shall contribute to the overall mission by performing the following functions:

3.1.4.1 Thrust Augmented Thor (TAT) Booster. The function of the Thor booster shall be to deliver the orbital vehicle with payloads to a predetermined point in space with a predetermined velocity vector. The Thor booster's contribution to the satellites' orbital energy and the Thor booster's error contributions to final orbital parameter errors are specified in section 3.1.8 of this specification. The Thor booster shall be used for providing the initial lift-off, thrust, and flight control required for launching a space flight payload. The Thor/Agena/primary payload configuration is shown in Figure 2. The Thor booster shall be a vertically-launched, liquid-fueled space booster powered by a main, gimballed rocket engine and three thrust-augmentation rocket engines. The main engine shall provide pitch and yaw control by gimballing in the pitch and yaw planes during the powered flight phase. Two gimballed vernier rocket engines shall provide roll control and shall augment the main engine to provide pitch and yaw attitude control prior to main engine cutoff (MECO). Three solid propellant rocket engines shall supplement the thrust of the main liquid-propellant engine. They shall be mounted externally on the aft section of the booster and shall be jettisoned after their burnout. The booster shall be controlled by the flight controller (DAC Drawing 1B07688) during powered flight. The flight controller shall maintain vehicle stability and shall direct the vehicle to the guidance initiation point. The operational tolerances at MECO shall be plus or minus four degrees in flight path angle, five nautical miles in position, and 500 feet per second in velocity (without radio guidance). With radio guidance, ground initiated signals may be received in the vehicle to cause the main engine to gimbal pitch and yaw attitude rates of 1.2 degrees per second. An accuracy of within one percent of the computed radio guidance steering commands shall be achieved. Radio guidance steering shall be enabled by the flight controller and shall be terminated by the radio guidance command or by the flight controller at or near MECO. The Thor Booster shall conform to DS-2345A "Detail Specification DSV-2C (DSV-2A) Space Booster."

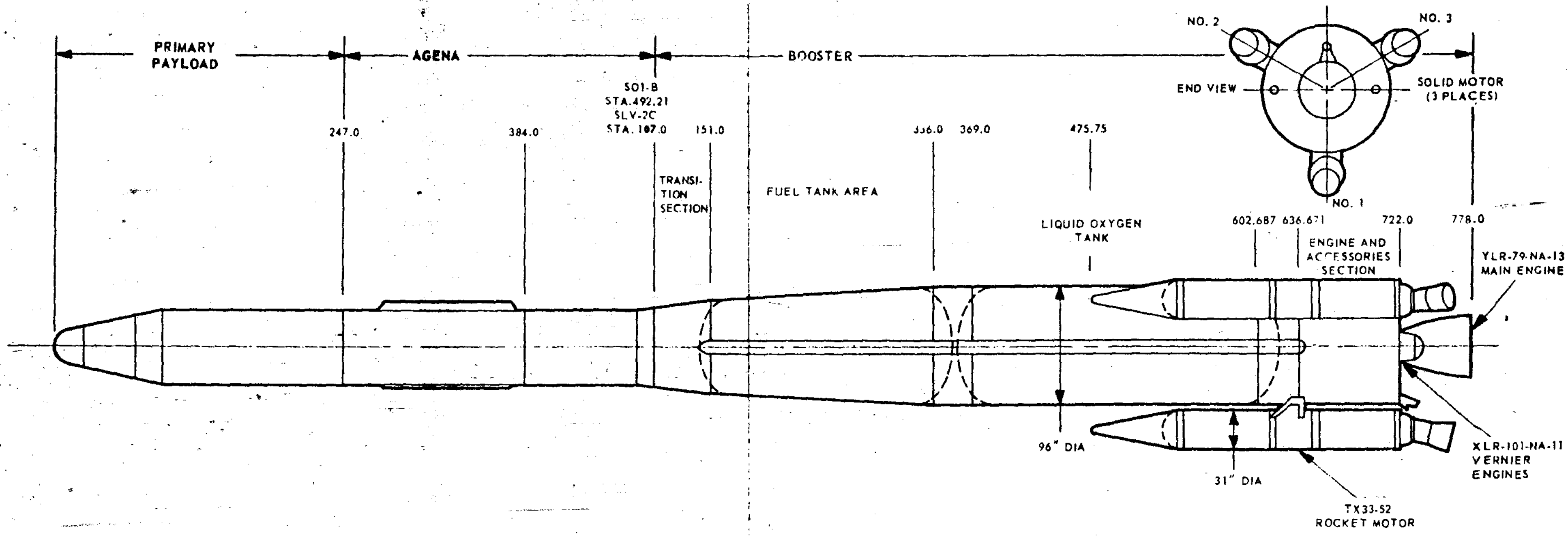


Figure 2 Typical Configuration of Primary Payload/Agena/Thor Booster

3.1.4.2 Agena (Model 39205). The Agena shall perform both ascent and orbital functions. The ascent functions of the Agena shall be to:

a. Provide a means for relaying radio guidance correction commands to the booster during ascent.

b. Provide thrust with an oriented vector for injection of the Agena with payloads.

The on-orbit functions of the Agena shall be to:

a. Provide a stable test platform for both the primary payload and secondary payload.

b. Provide the required electrical power to the payloads.

c. Provide a means for real-time commanding and stored-program commanding of the payloads.

d. Provide a means for sending payload information back to the ground.

e. Provide a means for ejecting two recoverable capsules from orbit.

The Agena vehicle shall be a liquid-fueled second stage booster, launched by a Thor (SLV-2A) booster, and powered by a gimballed rocket engine which shall provide pitch and yaw control during powered flight. The vehicle shall house and support the system components and modules that comprise the Program [redacted] Model 39205. The vehicle shall provide satellite attitude, environmental protection, and a support platform for the payloads and required flight equipment. The four major sections of the vehicle shall be the forward section, the tank section, the aft section, and the booster adapter section. The forward section shall have mounting provisions for the Program [redacted] payloads, and shall accommodate the major part of the basic vehicle system equipment. The tank section shall consist of integrally constructed, dual-chamber propellant tanks which shall supply the vehicle rocket engine. The aft section shall provide support for required system equipment including the rocket engine, structural assembly, gas reaction jets, and hydraulic system. The booster adapter section, which shall attach to the aft part of the tank section, shall be designed to support the entire Agena vehicle with payloads when attached to the Thor booster. At separation of the Agena vehicle from the Thor booster during the ascent phase, the adapter section shall remain attached to the Thor booster.

The propulsion system shall consist of a rocket engine and components required to develop a nominal thrust of 16,000 pounds. The engine shall be designed for a nominal thrust duration of 240 seconds. The propellants used for engine operation shall be unsymmetrical dimethylhydrazine

(UDMH) and inhibited red fuming nitric acid (IRFNA). The propellant tanks shall be pressurized with helium to insure proper propellant pump operation.

An electrical system shall be provided to supply electrical power to operate vehicle electrical equipment.

A guidance and control system shall be provided.

Vehicle electronic equipment shall be provided for command, orbital programming, gathering and transmitting telemetry data, and for tracking functions.

3.1.4.3 Radio Guidance System. The function of the Radio Guidance shall be to increase guidance accuracy by providing real-time sequenced events and real-time steering corrections to the Thor booster and to the Agena vehicle during their boosting phases. Steering corrections implemented by the radio guidance shall be in the nature of vernier corrections, and if none are received, the onboard guidance shall continue to function in the pre-programmed inertial mode. Similarly, the sequenced events, namely, booster MECO and separation, and Agena timer start and velocity meter enable, shall be actuated by the program timers if they are not commanded by the radio guidance system. Hence, failure of the onboard radio guidance equipment, lack of adequate tracking data, failure of the ground-to-vehicle radio command link, or failure in the ground computer facility shall not interfere with the standard inertial guidance mode of operation. The vehicle-borne functional components of the radio guidance system shall include a radar transponder to aid ground tracking, a command receiver, and circuitry for utilizing the RF commands to control various activities of the Agena guidance system components. The ground-based components of the radio guidance system shall include a radar tracking station, which tracks the vehicle and transmits RF commands, and a computer which processes the tracking data, computes trajectory corrections, and issues steering commands and timely discrete commands. The radio guidance system shall have capability for seven command signal formats, but only five shall be used for Program [redacted]. These include the four steering commands (pitch up, pitch down, yaw right, yaw left) plus a discrete signal.

The steering commands shall cause attitude changes of the vehicle during engine burn to implement thrust vector corrections. These corrections shall result from the computer calculation of anticipated cutoff conditions repeatedly predicted from the radar tracking data and continuously compared with the desired velocity state of the vehicle at cutoff. The radio guidance system shall not force the vehicle to fly a nominal flight path, but shall command steering corrections to assure a specified velocity vector at cutoff.

The discrete command shall be a specific pulse group transmitted to the radio guidance receiver each time the Ledex switch is to be triggered to a new sequenced position. Four Ledex switch positions shall be used for the Agena. The "sequence-zero" position shall be an open circuit. The "sequence 1" position shall energize the booster MECO and release the Agena standard D-timer brake. The "sequence 2" position shall cause booster separation. The "sequence 3" position shall enable the velocity meter which, after a preset velocity-to-be-gained is achieved, commands the Agena engine.

3.1.4.4 Primary Payload. The primary payload shall perform the functions specified in section 3.1.15 of this specification.

3.1.4.5 System Exclusions. Although aerospace ground equipment (AGE) and the tracking station network are necessary components of the Program [redacted] Satellite System, these shall be considered ancillary for purposes of this specification and shall not be specified in this specification. However, this specification shall cover the electrical and mechanical interfaces between the system and AGE and also shall cover the RF interface between the system and a typical tracking station. The System's capability to accommodate and support secondary payloads shall be included in this specification. However, the secondary payloads, which vary from vehicle to vehicle, shall not be included as a part of the system as defined for this specification, except for Link 2 telemetry which is considered standard equipment even though it is physically located on the secondary payload side at the interface.

3.1.5 System Functional Description. A nominal ascent program shall be programmed by the booster programmer for the Thor boosting phase of the flight, and by the Standard D-Timer for the Agena boosting phase. Refinements of the trajectory shall be provided in the form of pitch and yaw corrections to the Thor and Agena by the radio guidance system. The radio guidance system also shall command major events such as MECO (main engine cut off), stage separation, and velocity meter enable. These functions shall be as described in section 3.1.11. Following injection into orbit the Agena with payloads shall be yawed into a tail-first attitude and stabilized in this attitude. The orbital programmer shall provide programming for payload events and vehicle housekeeping functions such as turning on telemetry links and the S-Band command and tracking link during station acquisition. The S-Band link shall provide real time commands to the payloads, and to the vehicle for housekeeping functions. Two VHF Zeke command links shall provide real time commands to vehicle functions.

The vehicle shall be capable of entering a passive mode on command. During the passive mode the vehicle shall tumble to maintain temperature stability. All electrical power shall be shut off with the exception of auxiliary power to the secondary payloads, the Zeke Command link, and telemeter Link 1, which can be turned on by command.

Reactivation shall be by command. One complete deactivate-reactivate sequence shall be possible. The reactivation shall be limited geographically to the station at which deactivation occurred.

A primary recovery system shall be provided as well as a backup system. The primary recovery shall be selected and enabled by real time command. Initiation of the primary recovery sequence shall be by the H-Timer. The orbital test vehicle shall eject the recoverable capsule at an optimum retro-firing attitude. The secondary recovery system shall be an independent system capable of ejecting the capsule in the event of primary system failure.

3.1.6 System Interface Requirements. Prior to specifying specific interface requirements, the command, information, and power flows are discussed.

3.1.6.1 Command Flow. The system shall utilize stored program commands and real time commands. During the Thor boost, stored program events shall be programmed by the booster programmer. Real time events shall be commanded through the radio guidance system or through the range safety link. During the Agena boost the Standard D-Timer shall program stored program events. The real time event shall be commanded through the radio guidance system. During the orbit phase stored program commands shall originate in the orbital programmer. Real time commands shall be sent by either the Zeke or S-Band command links. The recovery events shall be programmed by either the recovery timer or the lifeboat timer. Deactivate and reactivate sequences shall be programmed by the Standard D-Timer. During ground testing, umbilicals and test plugs shall handle all commands necessary to assure flight readiness. The general command flow shall be as shown in Figure 3. Detailed requirements for command and tracking are specified in section 3.1.14.

3.1.6.2 Information Flow. Booster diagnostic and status information shall be transmitted via a booster telemetry link. See Figure 4. Details of the booster telemetry link are specified in section 3.1.13. Link 1 shall handle payload information as well as Agena status information. Details of Link 1 are specified in section 3.1.13.

Link 2 shall handle selected Agena status information redundantly, as well as secondary payload information. In addition Link 2 shall be used to record and transmit primary payload data when required by the particular primary payload being flown. Detailed requirements for Link 2 are specified in section 3.1.13.

During the recovery operation, primary payload status data shall be transmitted via a capsule telemetry link. Requirements for this link are specified in section 3.1.13.

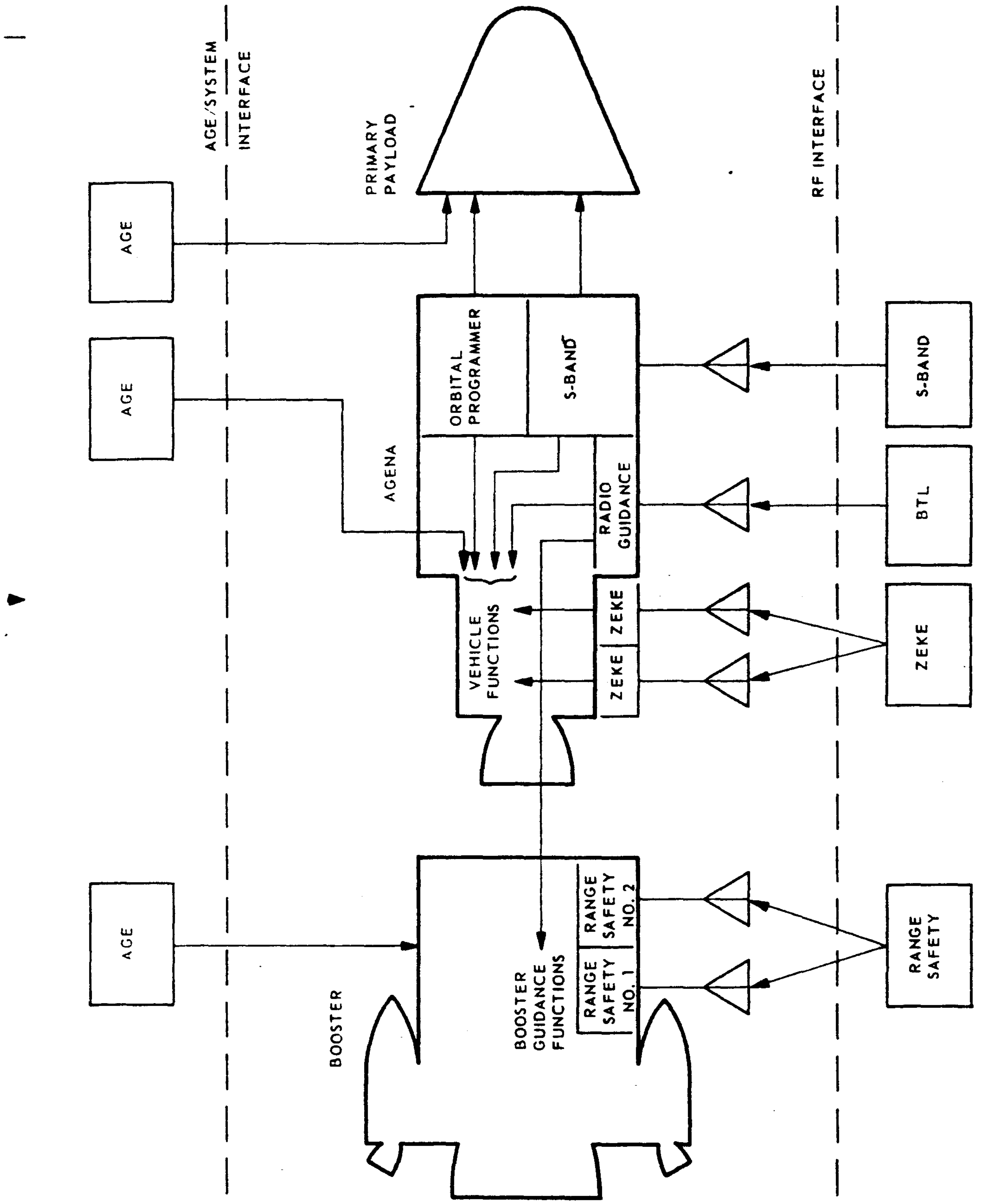


Figure 3. Command Flow

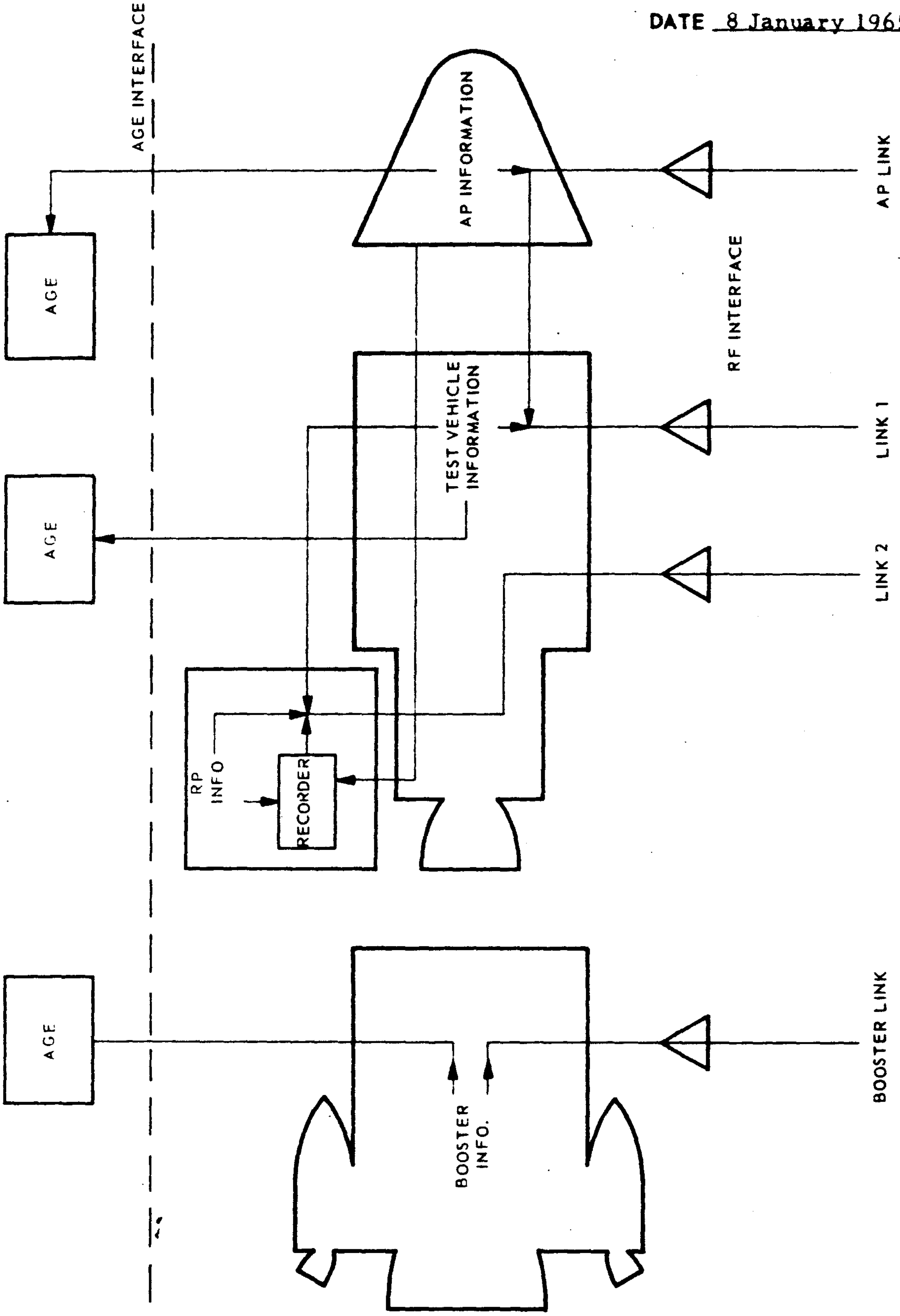


Figure 4. Information Flow

During ground testing, umbilicals shall be provided to handle all status and diagnostic information necessary to assure flight readiness.

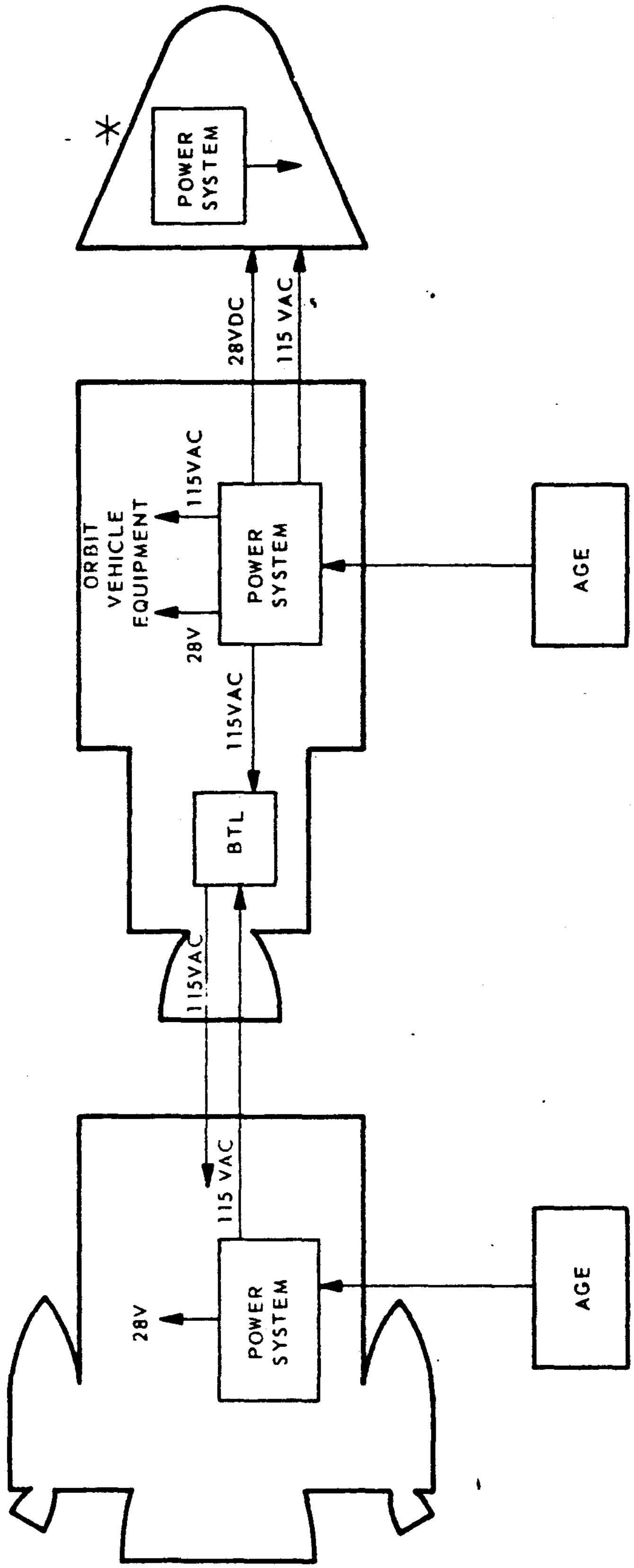
3.1.6.3 Power Flow. The booster shall be provided with an electrical power system as a source of 115 volt ac, 3-phase, 400 cps power, and 28 volt dc power as required by the various booster equipment. The electrical power distribution shall be as shown in Figure 5. During ground operations and check-out, power shall be provided by means of umbilical connectors. The orbital test vehicle shall be provided with an electrical power system as a source of 115 volt ac, 3-phase, 400 cps, power and 28 volt dc regulated and unregulated power as required by the Agena orbital vehicle, the primary payload and the secondary payloads. The primary payload shall provide its own power during recovery only.

3.1.6.4 System Internal Interfaces. The interfaces internal to the system shall be as specified in the following subparagraphs.

3.1.6.4.1 Thor/Agena Interface. Because the Thor adapter shall be used to mate the Thor to the Agena, there shall be two distinct interfaces involved, namely, 1) The Thor/booster adapter and 2) Agena/booster adapter interfaces.

3.1.6.4.1.1 Mechanical Interface. The Agena/adapter mechanical interface shall be located at S-O1B station 384.0. The Thor/booster adapter mechanical interface shall be located at S-O1B station 492.21 (SLV-2A station 107). The booster shall be finally mated to the Agena at the Thor/booster adapter interface, but shall separate at the Agena/adapter interface by detonation of an explosive cord encircling the station. The mechanical interface shall conform with LMSC Drawing 1345366B, "LV-2A, SLV-2/Standard Agena Mechanical Interface."

3.1.6.4.1.2 Electrical Interface. The Thor/booster adapter electrical interface shall consist of one connector, P106, which carries commands. See Figure 6. The Agena/booster adapter electrical interface shall consist of two connectors; P700, which carries commands, and P701, which provides for instrumentation. P701 shall provide for 8 temperature transducer outputs. P106 and P701 shall provide for radio guidance steering commands to the booster. P106 shall provide for two redundant sets of destruct signals. Backup destruct signals shall be provided by a separation switch which shall actuate on premature separation. P106 and P701 shall carry sequence 1 and 2 enable signals from the Thor booster timer. P106 and P701 shall provide for carrying the radio guidance MECO command from the Agena to the Booster, back to the adapter to disarm the destruct system, and back to the Agena to start the standard D-timer. The vernier engine cut-off (VECO) command shall cross P106 to provide a backup disarm to the destruct system. This signal shall cross P701 to actuate appropriate Agena functions. The separation signal shall cross from radio guidance and from the D-timer through P700 to the separation detonator and retro rockets located in the adapter. The electrical interface shall conform to LMSC Drawing 1398571, "SLV-2A/S-O1B Electrical Interface".



* ACTIVATED IMMEDIATELY PRIOR TO RECOVERY

Figure 5. Power Distribution

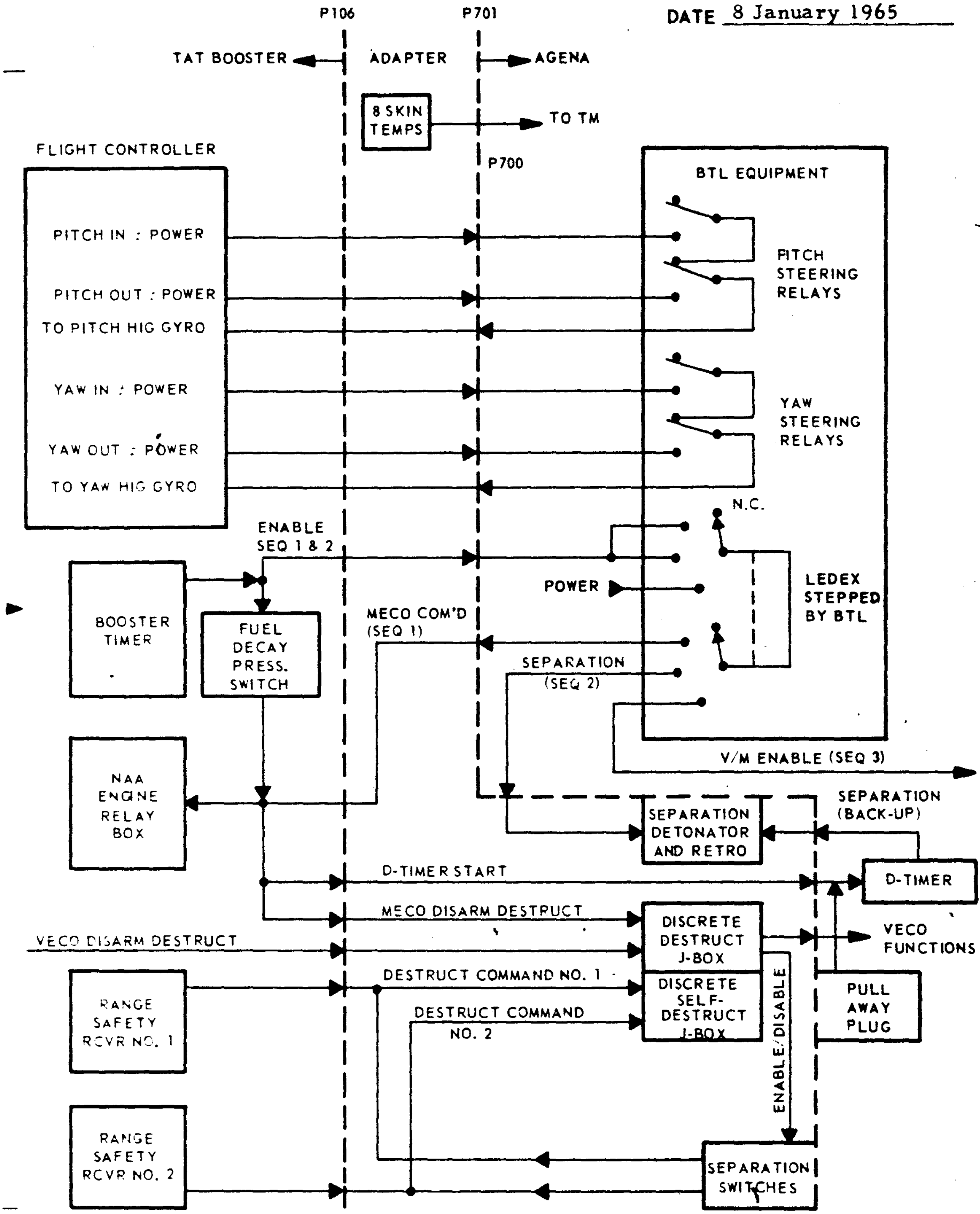


Figure 6. Thor Booster/Adapter/Agena Interface

3.1.6.4.2 Agena/Radio Guidance System Interface. The interface between the Agena and the vehicle borne radio guidance equipment shall be the equipment mounting brackets and one electrical connector.

3.1.6.4.2.1 Mechanical Interface. The vehicle-borne radio guidance group with antennas shall be mounted in the Agena vehicle in accordance with the requirements of LMSC Drawings 1397367 (S-01B) and 1365461 (Program [redacted]).

3.1.6.4.2.2 Electrical Interface. The signal interface between the radio guidance system and the Agena shall be essentially passive and shall be fail-safe. In addition to DC power, the operating interface shall consist of four gyro torquing reference signals from the guidance units, two gyro torquing signal channels from the radio guidance system, plus three discrete signal sequenced circuits. The gyro torquing signals shall originate within each stage and the appropriate pitch and yaw signals (up or down, right or left) shall be switched into the two return lines as a result of RF signal actuation of the steering relays. The MECO and booster separation signal voltages shall originate within the booster, and the velocity meter enable signal voltage shall originate within the Agena guidance system. These circuits shall be closed by response of the Ledex switch to the radio guidance discrete signal. Failure of the radio guidance system to supply steering or discrete commands shall not interfere with normal operation of the onboard systems in their normal inertial modes. The remainder of the interface shall be telemetry monitors and test and checkout circuits. Telemetry outputs shall monitor the transponder magnetron current, the transfer relays which switch the radio guidance system from the booster to the Agena after first stage separation, the command receiver AGC, and a combined function monitor whose output voltage level indicates various combinations of sequence stepper, steering relays, and discrete signal actuated circuits functioning at the time. Also, there shall be several circuits which shall be used only during installation, systems test, and ground checkout, and which shall permit operation, cycling, or lockout of various components.

3.1.6.4.3 Agena/Primary Payload Interface.

3.1.6.4.3.1 Mechanical Interface. The Agena/primary payload mechanical interface shall conform with LMSC Drawing 1324216, "Vehicle/Payload Mechanical Interface".

3.1.6.4.3.2 Electrical Interface. The Agena/primary payload electrical interface shall consist of plugs designated AP21, 22, 23 and 24. See Figure 7. These plugs are designated as the Pyro, Power, Command, and Telemetry connectors, respectively.

a. AP21, the Pyro Connector, shall provide for the following:

1) Two complete pyro circuits

2) The following events signals:

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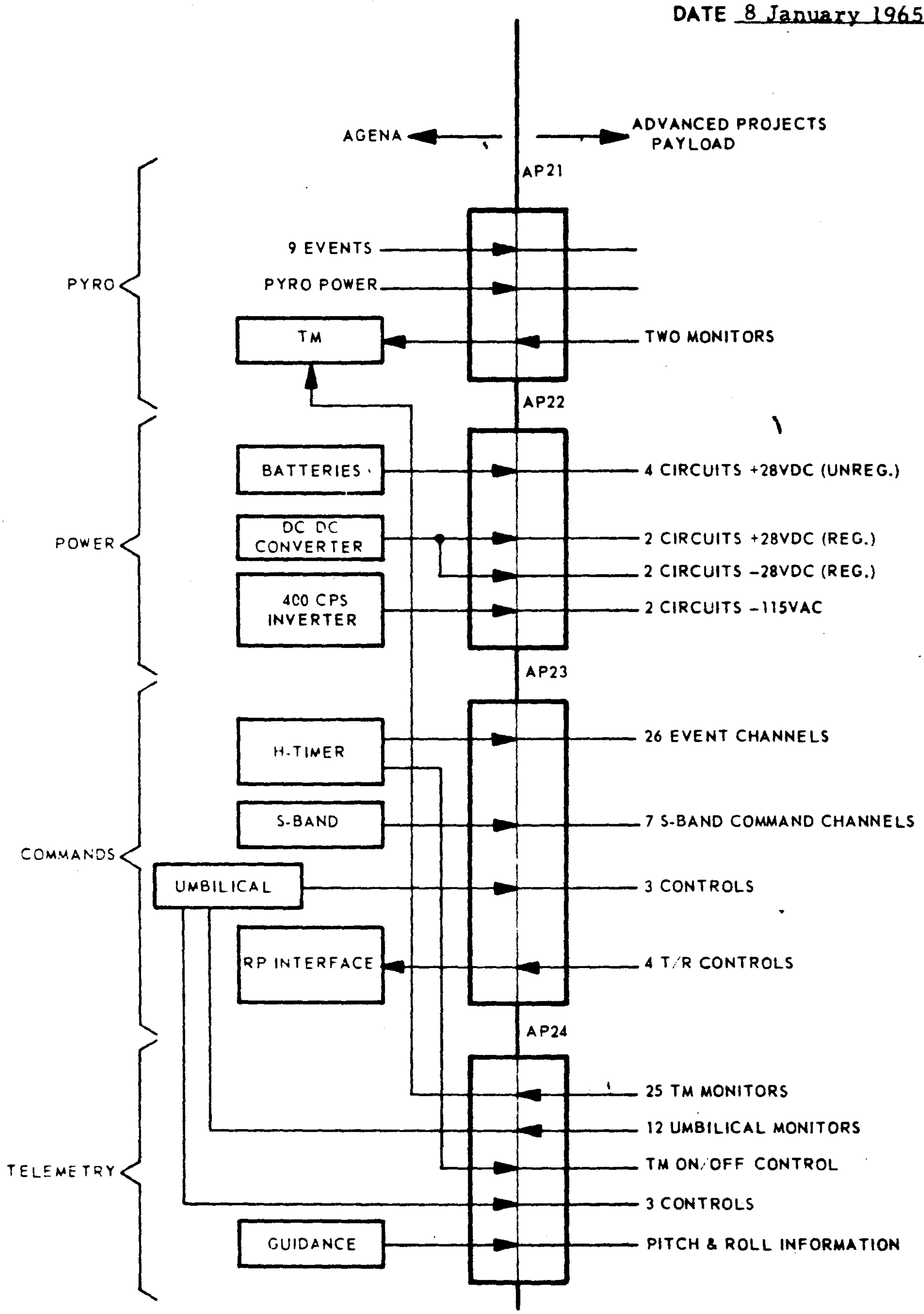

Figure 7. Agena/Primary Payload Interface

a) Agena engine burnout

b) Capsule Arm

c) Capsule Transfer

d) Capsule Separate

e) Transfer to Experiment #2

f) Deactivate Command

g) In Flight Reset (Booster Separation)

3) The following monitors

a) Pyro-Mode Monitor (C-6)

b) Advanced Payload Monitor (AP-21)

b. AP22, the Power Connector, shall provide for the following:

1) Four circuits of unregulated plus 28 VDC

2) Two circuits of regulated plus 28 VDC

3) Two circuits of regulated minus 28 VDC

4) Two circuits of single phase, 400 cps, 115 VAC

c. AP23, the Command Connector, shall provide for the following:

1) Twenty-six orbital programmer channels

2) Seven S-Band commands

3) Three primary payload controls from the Agena umbilical

4) Four tape recorder control commands from the primary payload to the secondary payload interface.

d. AP24, the Telemetry Connector, shall consist of the following:

1) Twenty-five telemetry monitors from the primary payload

2) Twelve umbilical monitors

3) Primary payload telemetry on/off control

4) Three primary payload controls from the Agena Umbilical

5) Pitch and roll channels from the Agena guidance to the primary payload.

3.1.6.4.4 Agena/Secondary Payload Interface.

3.1.6.4.4.1 Mechanical Interface. The secondary payloads shall be mounted on the aft racks of the Agena. LMSC Drawings 1363999 and 1358140 specify the hole patterns for mounting secondary payload panels on the Agena. Space envelopes shall be as specified in section 3.1.16 (Secondary Payloads).

3.1.6.4.4.2 Electrical Interface. The electrical interface for the Agena/secondary payload is shown in Figure 45 of section 3.1.16.

a. C1J317, Telemetry Connector. This connector shall have provisions for carrying commutated vehicle status information from the Agena to the secondary payload to be transmitted over Link 2. Power for the vehicle commutator shall be supplied through this connector to be applied whenever Link 2 is functioning. Primary payload information to be recorded on two tracks shall cross through their connector to the secondary payload. Six individual vehicle status measurements shall cross this connector for commutation and transmission by Link 2. Fifteen secondary payload measurements shall cross to the Agena for commutation and transmission by Link 1.

b. C1J318X, RF Connector. This connector shall carry the output of the Link 2 transmitter located on the secondary payload side of the interface to the Agena umbilical for monitoring Link 2 status prior to liftoff.

c. C60P2X, Command Connector. This connector shall carry power in the form of 28 VDC unregulated and 28 VDC auxiliary to the secondary payloads. Ten channels of programmed events shall be provided from the Agena orbital programmer to the secondary payload for experimental events. Two additional channels shall be provided to control Link 2. A tape recorder "read-in" and "stop" command shall be provided to the Link 2 tape recorder. Other commands from the Agena to the secondary payload shall include "TLM Disable", "TLM Switch to Orbit Mode", "Deactivate," and "Reactivate".

d. J350X, Pyro Connector. The pyro connector shall provide pyro power to the secondary payload.

e. J-333X, System F Connector. This connector shall provide the interface between the Agena and System F, an optional payload. This connector shall provide to System F four stored program commands, a real time command and plus 28 VDC unregulated power. The system shall provide, through this connector, four commands to the control and guidance system as well as a monitor to the umbilical.

f. J-337X, System A Connector. This connector shall interface the Agena with System A. A real time command and two stored commands as well as power shall be provided to the system by means of this connector.

3.1.6.5 System External Interfaces.

3.1.6.5.1 Thor/AGE Interface. The Thor shall mate electrically and mechanically with Aerospace Ground Equipment (AGE) by means of umbilicals and test connectors. These umbilicals and test connectors shall be capable of handling all the signals and information necessary to assure flight readiness as well as handling the propellant and gases necessary for servicing the Thor for launch.

3.1.6.5.1.1 Mechanical Interface. The mechanical interfaces of umbilicals shall conform to DAC Drawing ____________ (Drawing to be provided by DAC when available)

3.1.6.5.1.2 Electrical Interface. The electrical interfaces shall conform to DAC Drawing____________. (Drawing to be provided by DAC when available.)

3.1.6.5.2 Agena/AGE Interfaces. The Agena shall mate electrically and mechanically with Aerospace Ground Equipment (AGE) by means of umbilicals and test connectors.

3.1.6.5.2.1 Mechanical Interface. The mechanical umbilicals shall provide for propellant and gas loading as well as vehicle thermal control. The mechanical interfaces for mechanical as well as electrical connectors shall be as specified in LMSC Drawing 1399138.

3.1.6.5.2.2 Electrical Interface. The electrical umbilical and test plugs shall be as specified in LMSC Drawing 1399138 and shall provide access to system functions for purposes of system-test fault location down to a major module or component, and determination of flight readiness. Pin assignments on test plugs and umbilicals shall be as specified in LMSC/A068386.

3.1.6.5.3 Primary Payload/AGE Interface. The primary payload/AGE interface shall be as specified in section 3.1.15.

3.1.6.5.4 Thor/Tracking Station Interface. The Thor/tracking station interface shall consist of a Range Safety command link and a telemetry link. The Range Safety signal shall conform to existing range safety requirements. The booster telemetry shall be of the pulse duration modulation/frequency modulation/frequency modulation type. The RF signal shall meet the requirements of DAC Drawing 1A18680. Performance requirements for the booster telemetry system are specified in section 3.1.13 of this specification.

3.1.6.5.5 Airborne Radio Guidance/Guidance Radar Interface. The airborne radio guidance equipment shall interface the radar station via an X-band radar link. The RF signal from the tracking station to the airborne equipment shall consist of a series of pulses which are spaced in such a way as to provide an address and any one of five commands. On acceptance of a proper address the airborne transponder shall transmit a pulse for tracking. The RF signal requirements are specified in Western Electric Specification G.S. 10-900. The airborne equipment requirements are specified in Western Electric Specification G.S. 64-250. Performance requirements for the radio guidance system are specified in section 3.1.14 of this specification.

3.1.6.5.6 Agena/Tracking Station Interface. The Agena/tracking station interface shall consist of two FM/FM telemetry links, a VHF amplitude modulated command link and an S-Band radar command and tracking link. The requirements for the above links are specified in LMSC 1414691. Performance requirements for the telemetry are specified in section 3.1.13 of this specification. Performance requirements for the command system are specified in section 3.1.14 of this specification.

3.1.6.5.7 Primary Payload/Recovery Force Interface. The primary payload/recovery force interface shall consist of an FM/FM telemetry link and a pulse coded beacon. Requirements for these signals are specified in section 3.1.12.

3.1.7 System Weight Capability. The capability of the system to place a given weight in orbit shall be a function of the required orbit and the ascent system guidance and propulsion parameters. Nominal weight capabilities of the Thor/Agena ascent system are presented in this section. Curves showing typical orbital lifetime versus perigee altitudes are shown in Figure 8. The Thor booster shall establish initial velocity vectors for the Agena stage. The Thor booster payload is 13,805 pounds (nominal) greater than the Empty Weight On Orbit (EWO) of the orbiting Agena, representing the weight loss (including the propellant dump) of the Agena stage as it increases the Agena energy to meet injection requirements. Figure 9, shows the required booster capability based on 99.75 percent propellant utilization by the booster and injection into a 90 degree inclination orbit. The constant period lines

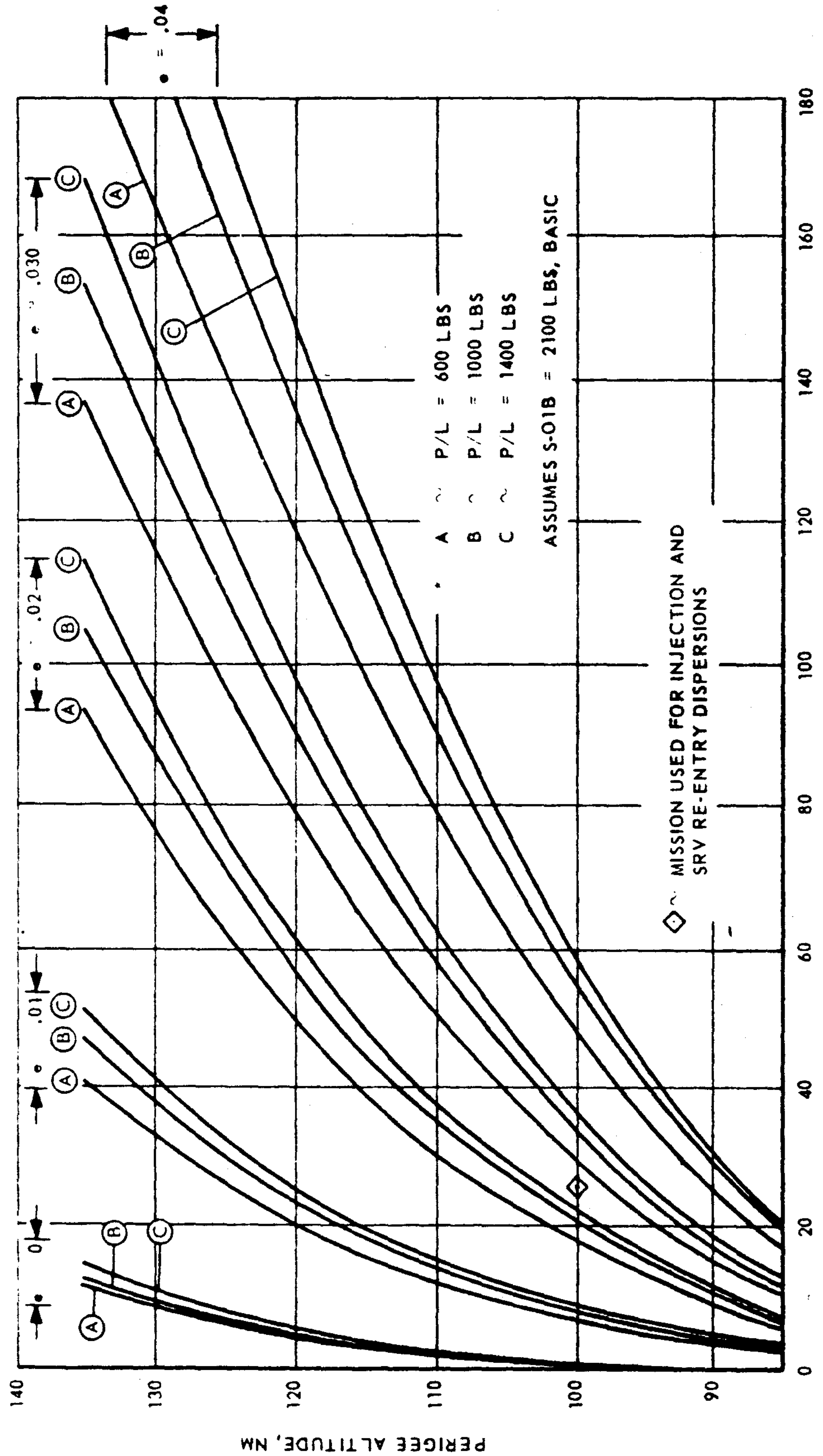

Figure 8. Lifetime, f (Perigee Altitude, Payload Weight, Eccentricity), 4 to 10 Days Stable Flight

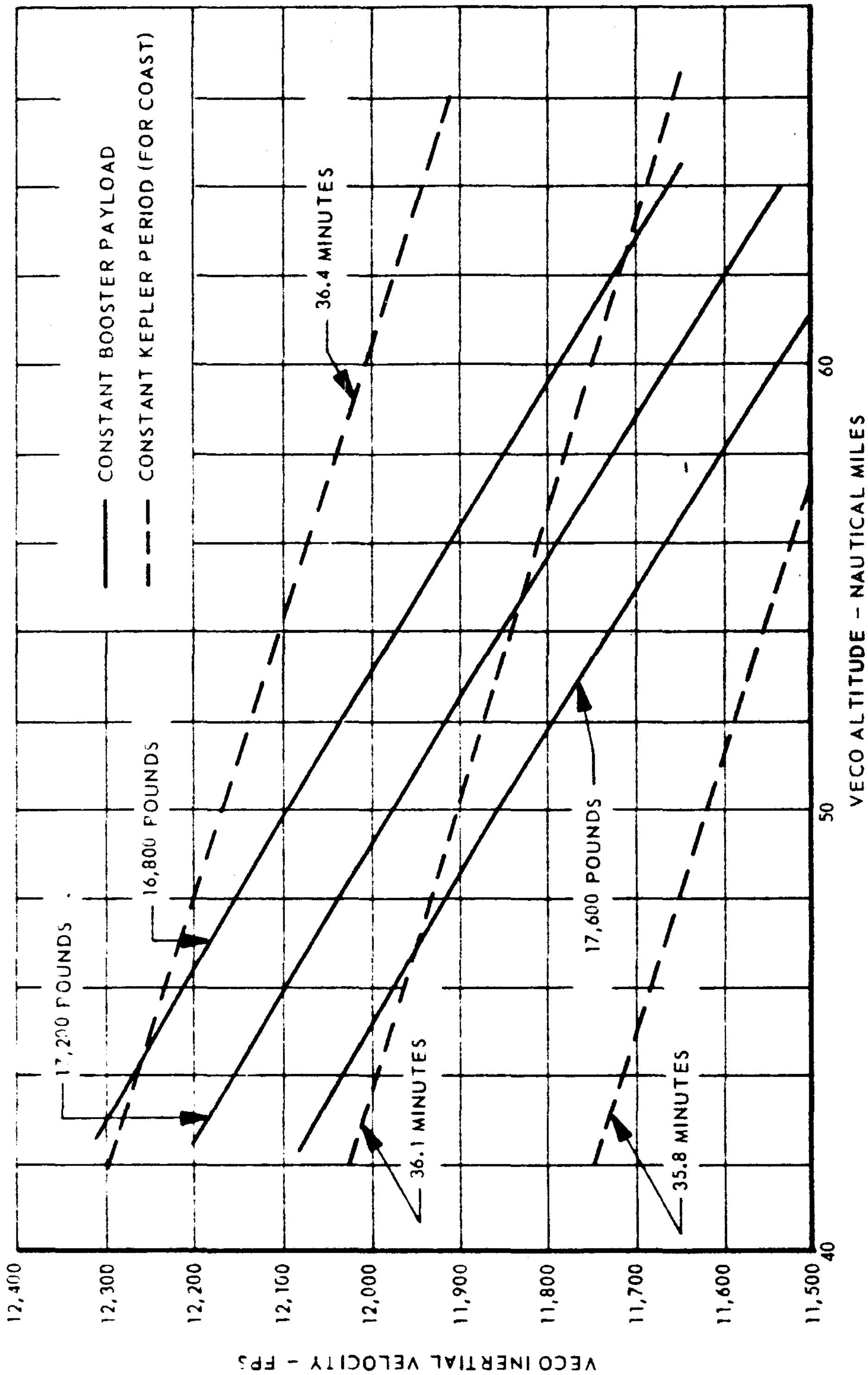

Figure 9. Booster VECO State (Expected) 90° Inclination Mission

of Figure 9 represent constant energy per slug mass. Since the period range is 90 to 92 minutes, it may be seen that the booster provides approximately 40 percent of the energy per slug mass of the satellite.

3.1.7.1 Empty Weight On Orbit (EWO). The EWO capability of the ascent system shall be 3595 pounds for injection to an orbit with:

a. 90.1 minute period

b. 90 degree inclination

c. 100 nautical mile injection altitude (perigee at injection)

d. Launch from VAFB

e. 65 second bottle drop

The EWO shall include the dry weight of the basic vehicle (1587 pounds), batteries, control gas at injection, and payloads of the Agena vehicle. The EWO capability shall be based on specified values of performance parameters. A contingency allowance of 60 pounds shall be provided to protect against permissible random variation about specified performance values. The 60 pound allowance shall provide 90 percent probability of establishing a desired accuracy when the EWO capability is based on propulsion system calibration (TAG) data. Figure 10 shows variations of this contingency as a function of probability of success.

3.1.7.2 Weight Capability Variations. The system weight capability shall vary with changes in orbital parameters and operating conditions. Figure 11 shows the variations in weight capability for variations in orbit period and injection altitude for 90 degree inclination orbits. The constant eccentricity lines of Figure 11 shall apply when injection is at the orbit perigee. When injection is at the perigee, the latitude of the perigee shall be relatively fixed by the pad location. When the perigee latitude is rotated, the injection radius must be greater than the perigee radius. Figure 12 shows the radius increment required (as altitude variation over a spherical earth) for variations in inclination and eccentricity. Eccentricity may be obtained from Figure 11, reading the injection altitude as perigee altitude. The 215 degree inclination implies northerly launches and is equivalent to 145 degrees based on conventional inclination angle definitions. The radius of the (oblate) earth varies with latitude. This must be considered in using Figure 11 to get the weight capability for rotated perigee cases. Figure 13 shows the earth radius variations with latitude. When synchronous orbits are specified, the corresponding period must be established to enter Figure 11. Figures 14 and 15 show periods required for various "days synchronous", and variations in orbit inclination and perigee altitude. A synchronous orbit is defined as an orbit with the descending node of the recovery pass of the required "synchronous" day within one-half of one degree of longitude of the descending node of the first recovery pass after launch.

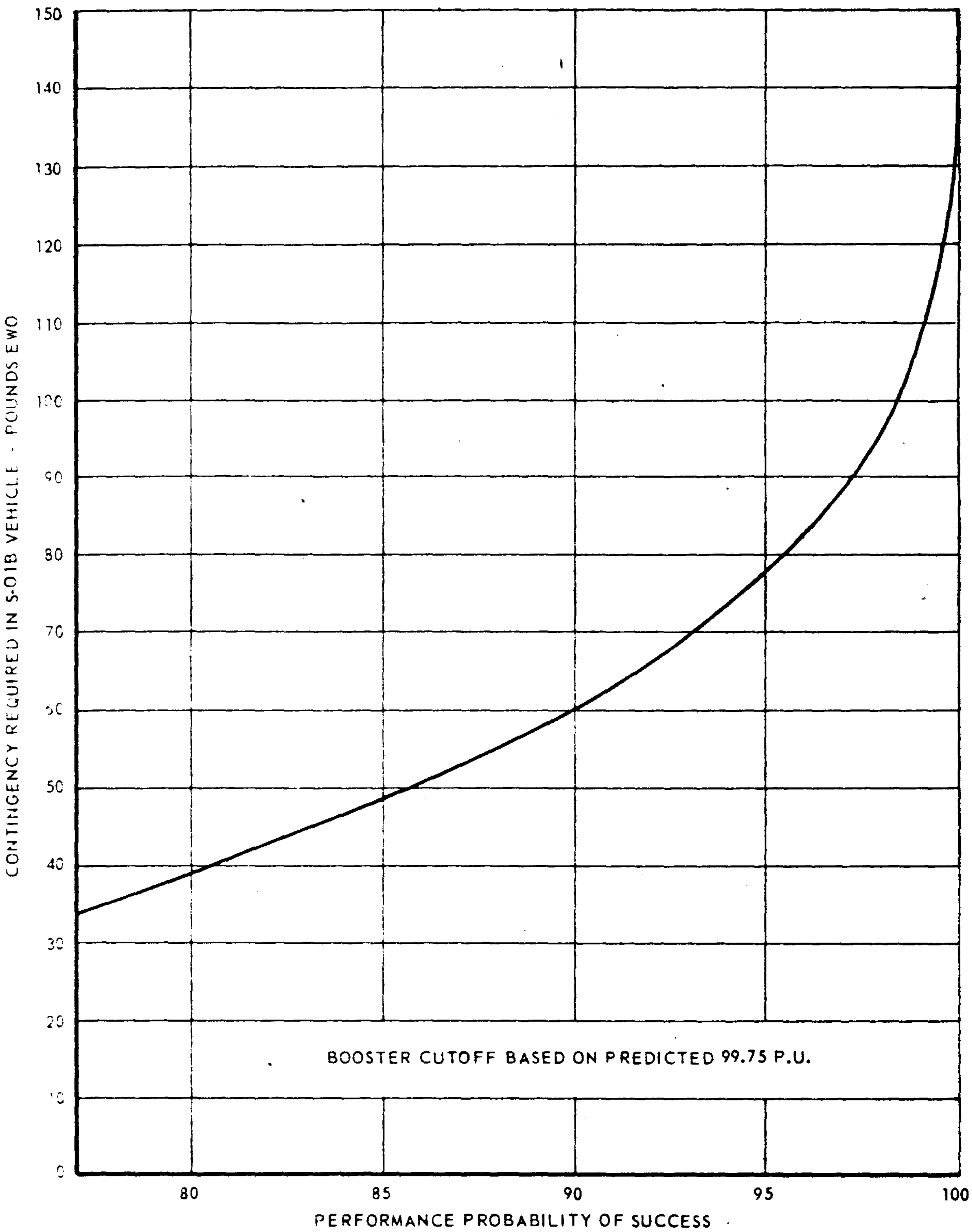

Figure 10. Vehicle System Contingency 90% Confidence

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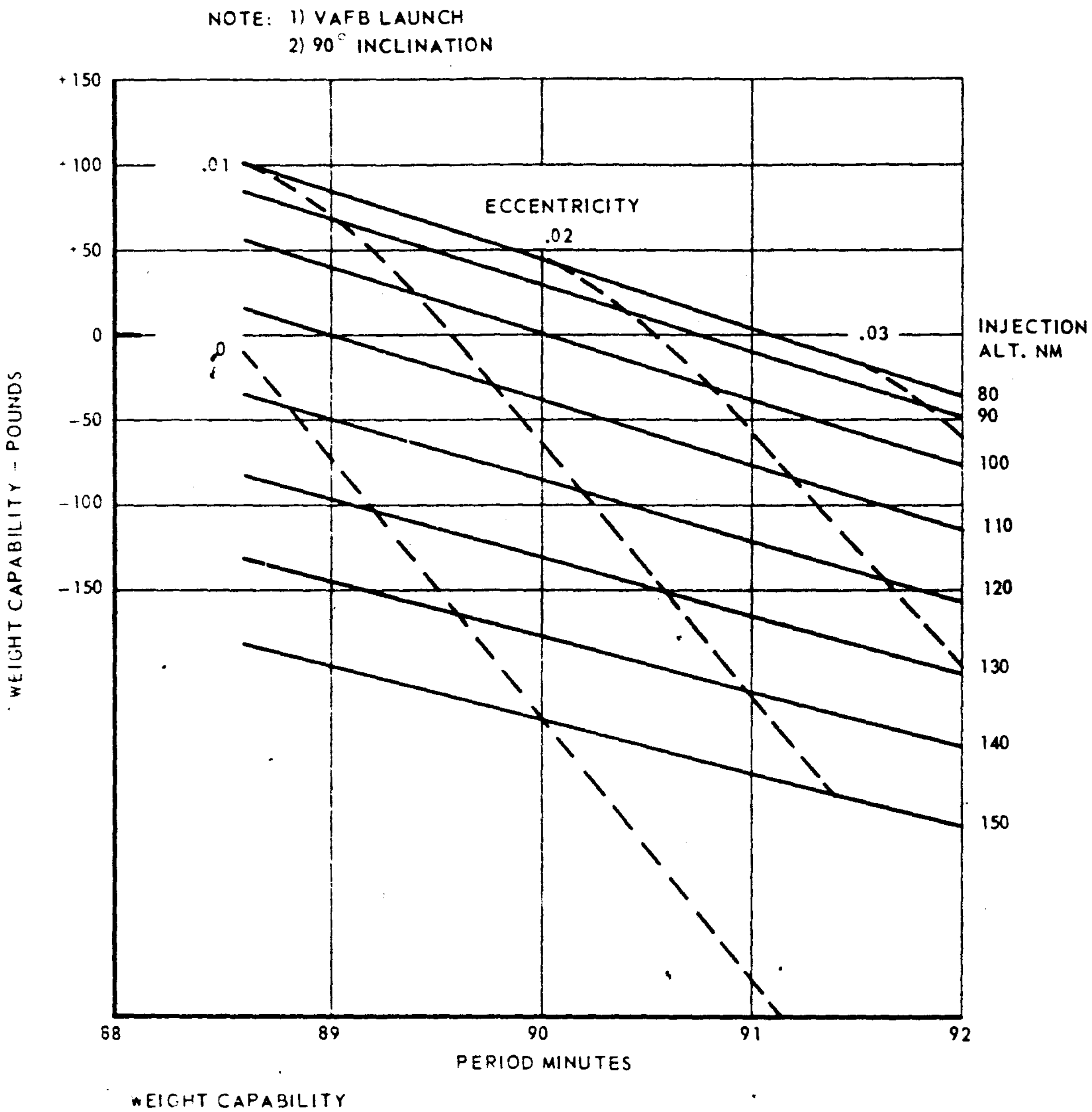

Figure 11. Weight Capability

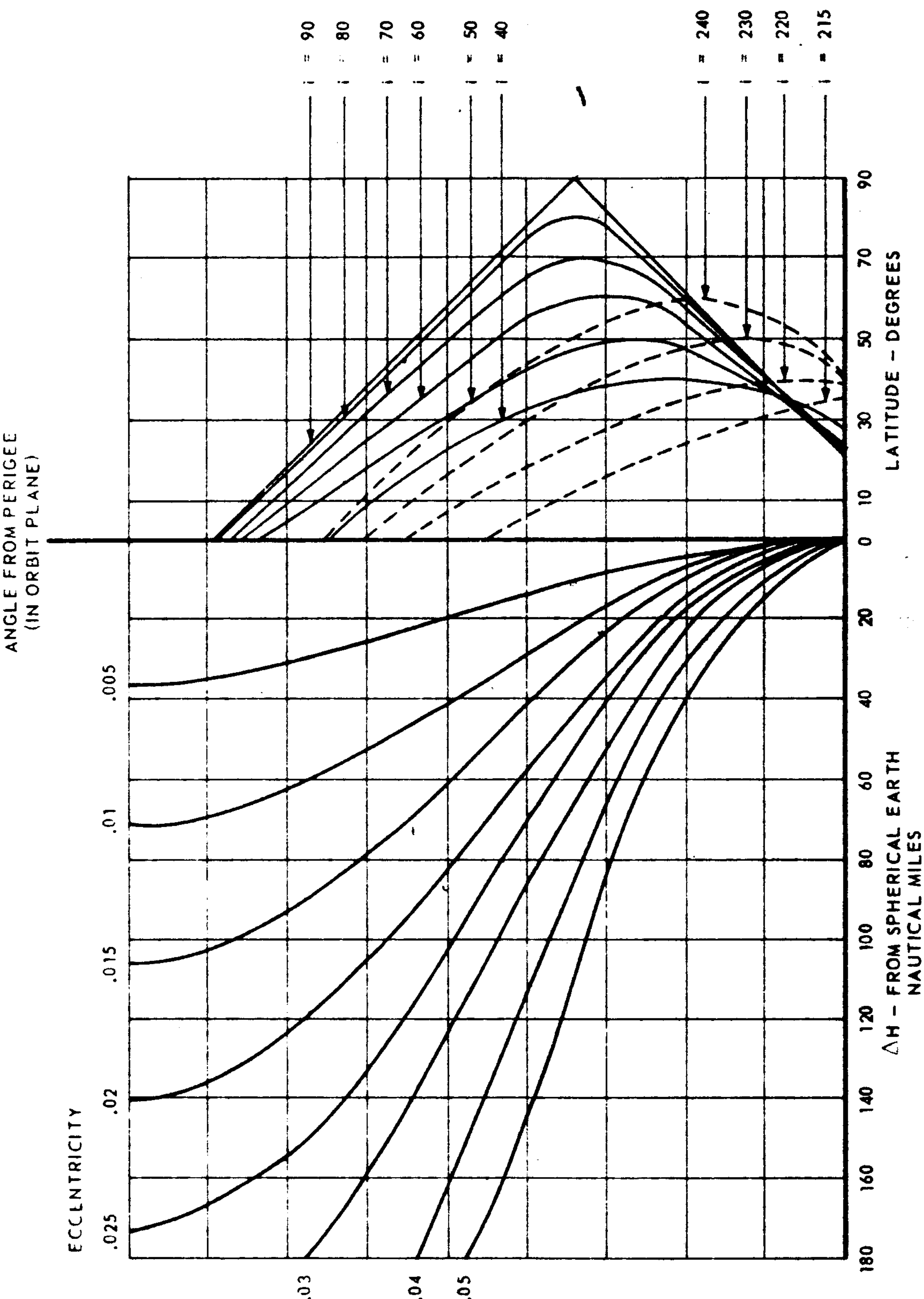

Figure 12. Orbital Altitude Variation

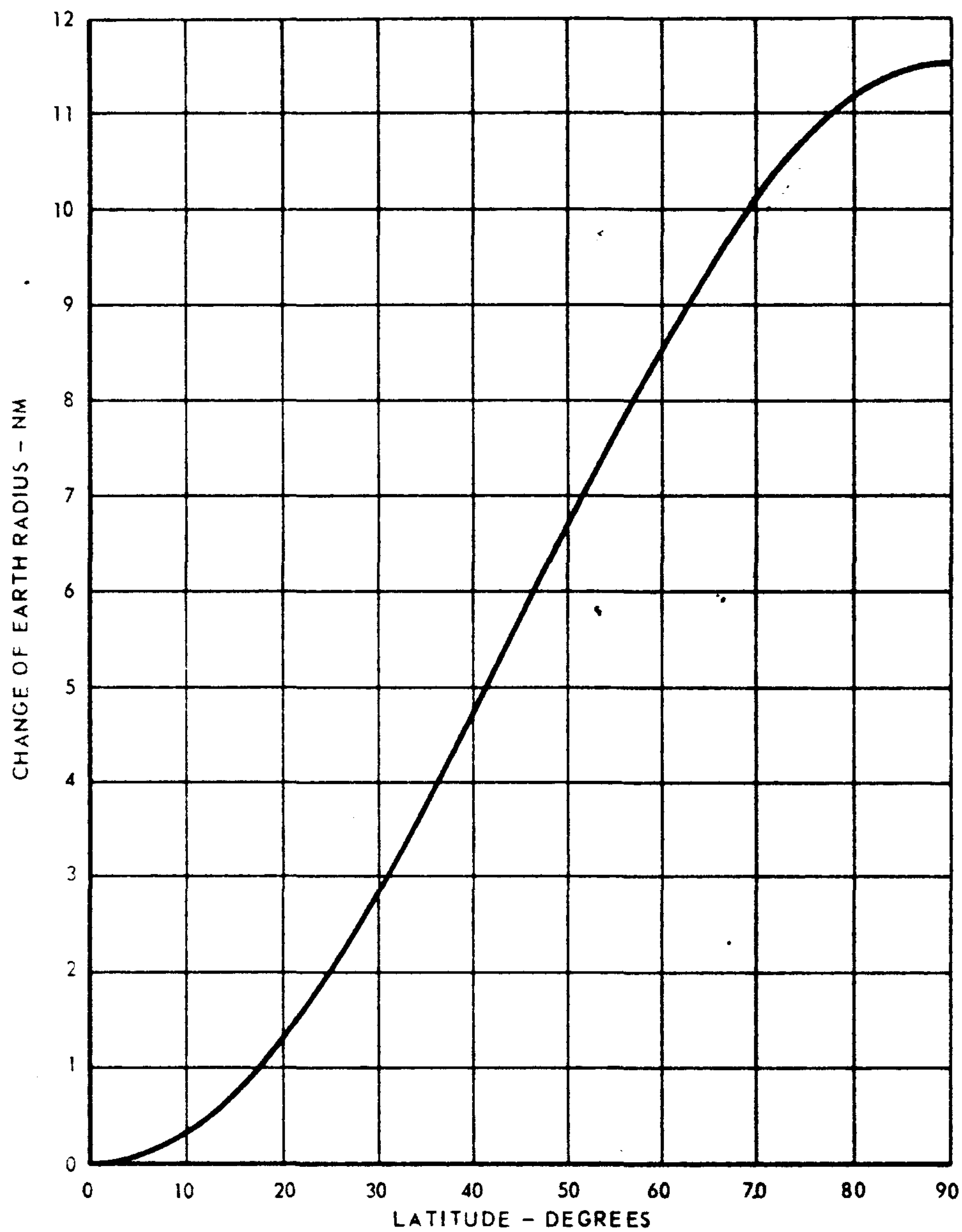

Figure 13. Change of Earth Radius As a Function of Latitude

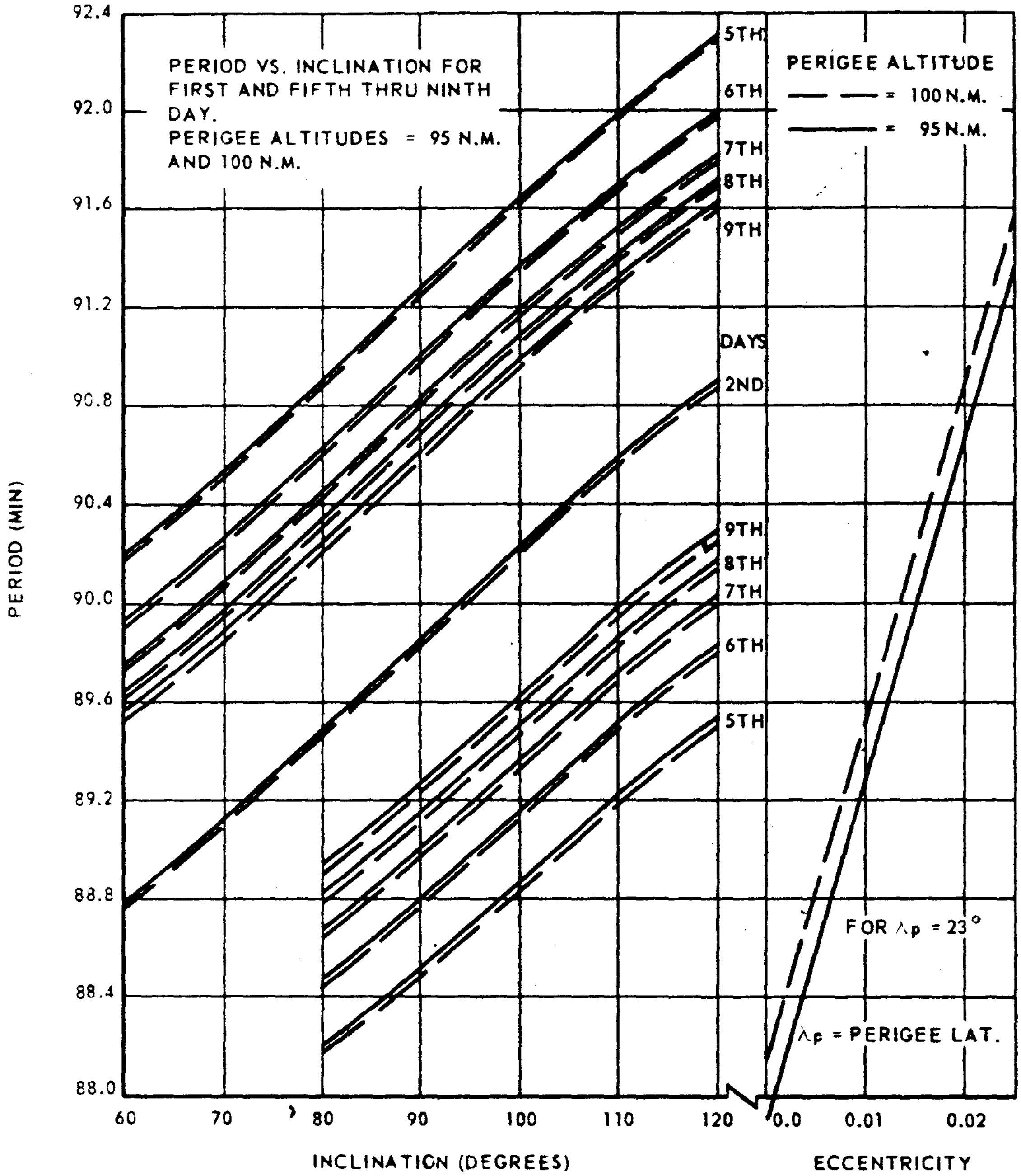

Figure 14. Synchronous Period

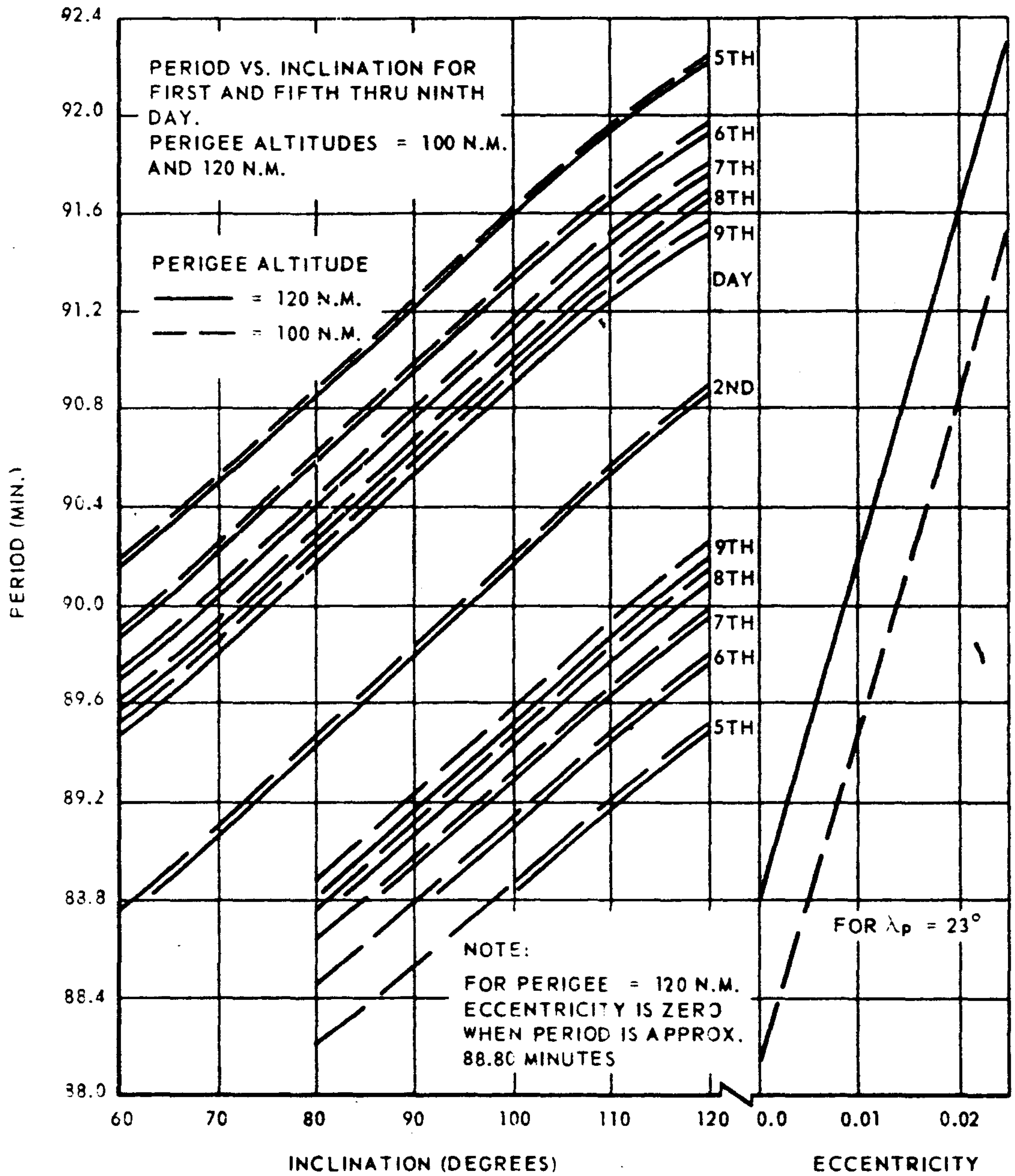

Figure 15. Synchronous Period

The required period shall be established based on nominal vehicle system performance. The system weight capability shall vary with the orbit inclination and pad location (due to earth rotational effects) as shown in Figure 16. For low inclinations (below approximately 85 degrees) dog-leg maneuvers shall be required due to range safety limits on launch azimuth. The launch azimuth limit shall be 170 degrees at PALC and 175 degrees at the VAFB complex. This allows lower inclinations from PALC without the dog-leg maneuver and allows better earth rotation velocity recovery from PALC. Relative pad locations allow a five-second earlier bottle drop from PALC giving approximately 15 pounds weight capability advantage to PALC launches relative to VAFB launches.

3.1.8 Injection Accuracy. Detailed analyses of the interaction of error sources and orbit selected shall be performed to determine expected accuracies as each orbit is selected. The error sources used for these analyses shall be those defined in the Trajectory Performance Model, LMSC/A315106B, in effect at the time of orbit selection.

3.1.8.1 Error Sources. Table I provides a list of error sources and their effect on selected orbit parameters for a typical mission. Table II indicates the percentage of error attributable to the three major contributors: booster, guidance, and second stage.

3.1.8.2 Accuracy Limits. For missions within the altitude inclination and period ranges indicated in Figures 11 and 16, accuracy limits shall be within the following values:

a.	Period:	Plus 0.25 Minutes Minus 0.25 Minutes
b.	Perigee Altitude:	Plus 17 nautical miles Minus 19 nautical miles
c.	Argument of Perigee:	Plus 18 degrees Minus 18 degrees
d.	Inclination:	Plus 0.18 degrees Minus 0.18 degrees

3.1.9 Launch Reaction.

3.1.9.1 Preflight Data. Generation of certain preflight data shall be necessary to define required hardware settings. The procedure for preparation of this data as defined by [redacted] shall be as follows: [redacted]

3.1.9.1.1 New Mission Preparation. New mission preparation shall be based on nominal propulsion system performance of a series and type

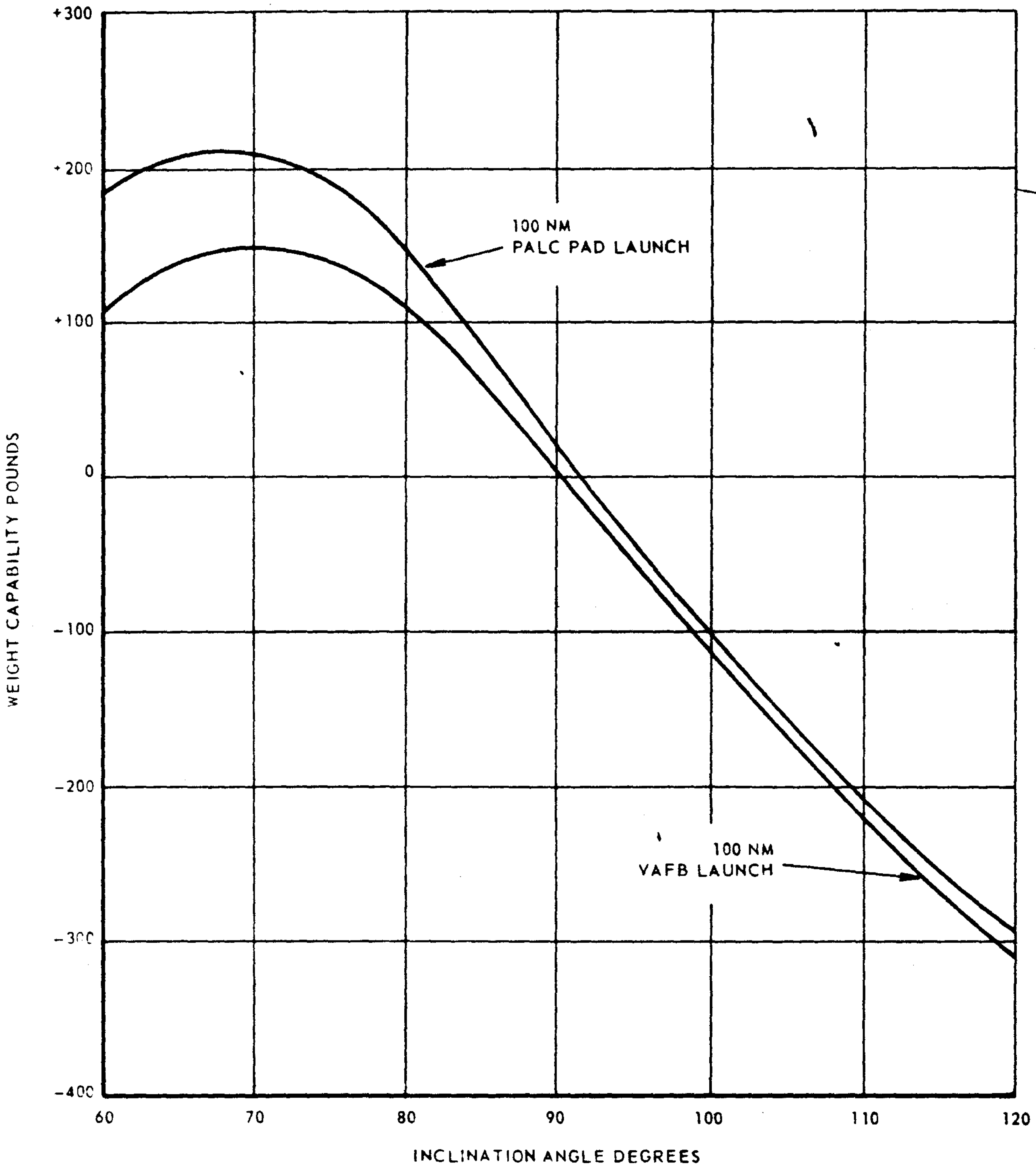

Figure 16. Weight Capability vs Inclination Angle

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σ	[illegible]	[illegible]	[illegible]	Δi	Δh_t	ΔT	$\Delta \lambda_p$	$\Delta \Omega_p$	Δh_a	Δh_p	Δe
						min				nm	
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-.[illegible]	-.[illegible]	+.1[illegible]	-.13[illegible]	+.[illegible]	-.[illegible]	+.000[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-.[illegible]	-.[illegible]	[illegible]	-.[illegible]	[illegible]	-.00021
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	+.[illegible]55	-.[illegible]	[illegible]	[illegible]	-.[illegible]	+.112	-.000[illegible]
[illegible]	[illegible]	[illegible]	-.[illegible]	[illegible]	+.439	-.[illegible]	-1.[illegible]	+1.[illegible]	-.[illegible]	+.[illegible]	-.000[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	+.[illegible]14	-.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]	[illegible]	-.000[illegible]
[illegible]	[illegible]	[illegible]	-.[illegible]	[illegible]	+.103	-.0[illegible]	-.[illegible]	+.[illegible]	-.[illegible]	+.[illegible]	-.000[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-.103	-.0[illegible]	+.[illegible]	-.408	+.230	-.194	-.00006
[illegible]	[illegible]	-.8[illegible]	-.[illegible]		+.119	0	-.1[illegible]	-.160	-.231	+.158	-.00005
Weight [illegible] lb	+450.[illegible]	-2.[illegible]	-.[illegible]3	0	+.076	-.0[illegible]	-.[illegible]	+.080	-.5[illegible]	+.10[illegible]	-.0000[illegible]
-3[illegible]	+[illegible].2	-.[illegible]	+.0[illegible]3	+.[illegible]	+.[illegible]10	-.00[illegible]	+.1[illegible]	-.1[illegible]8	-.[illegible]71	-.015	-.00001
Drag +10%	+[illegible]	-[illegible]	-.[illegible]41	0	+.270	-.09[illegible]	-1.[illegible]	+1.606	-2.55[illegible]	-.[illegible]	-.00043
-10%	+15[illegible]	-.[illegible]64	-.010	-.001	+.247	-.010	-.[illegible]36	+.444	-1.[illegible]24	+.356	-.00019
Pitch +3.4%	+.58[illegible]	-10.06[illegible]	-.0[illegible]	+.[illegible]	+.902	-.00[illegible]	-1.[illegible]	+1.821	-4.[illegible]	[illegible]	-.[illegible]73
Program -3.4%	-4[illegible]	+6.7[illegible]	+.[illegible]8	+.0[illegible]	-.702	+.010	+.[illegible]	-.085	+2.[illegible]	-.0[illegible]	+.00045
Pitch Gyro +80°/hr	+521[illegible]	-3.7[illegible]	-.[illegible]40	+.0[illegible]4	+.863	-.038	-1.[illegible]	+1.5[illegible]1	-3.[illegible]4	+1.206	-.[illegible]060
Drift -80°/hr	-4052.[illegible]	+5.703	+.[illegible]9	+.002	-.669	+.010	+1.[illegible]	-1.066	+1.84[illegible]	-.054	+.00039
Roll Gyro +80°/hr	+181.8	-.25[illegible]	-.003	0	+.030	0	-.12[illegible]	+.129	-.1[illegible]7	+.059	-.00003
Drift -80°/hr	+17[illegible].3	-.[illegible]39	+.0[illegible]1	+.001	+.029	-.006	+.[illegible]	-.0[illegible]9	-.310	+.022	-.00004

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σ / [illegible]	Δ [illegible]	[illegible]	[illegible]	[illegible]	Δh_1	ΔT	$\Delta\lambda_p$	$\Delta\Omega_p$	Δh_a	Δh_p	Δe
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- .[illegible]	[illegible]	-.[illegible]	-.[illegible]	+.[illegible]	+.038	-.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]	+.[illegible]	-.[illegible]
[illegible]	[illegible]	-.[illegible]	-.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	[illegible]	-.[illegible]	+.03[illegible]	-.[illegible]
[illegible]	[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	-.[illegible]	[illegible]	-.[illegible]
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- .[illegible]	-1[illegible]	+.[illegible]			-.30	[illegible]	-.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]	[illegible]
[illegible]	[illegible]	-.[illegible]	-.[illegible]		-.[illegible]	-.[illegible]	-.[illegible]	+.[illegible]	-.[illegible]	+.101	-.[illegible]
Impulse - .[illegible]	[illegible]	-1.[illegible]	-.[illegible]	+.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]	+.240	-1.[illegible]	+.17[illegible]	-.[illegible]
[illegible] - .[illegible]	+73.[illegible]	-.[illegible]	+.[illegible]	+.[illegible]	+.12	-.[illegible]	+.[illegible]	-.[illegible]	-.[illegible]		-.[illegible]3
Propellant Weight - [illegible]	+113.[illegible]	-.[illegible]1	+.[illegible]	+.[illegible]	+.[illegible]19	-.[illegible]3	+.1[illegible]	-.107	-.[illegible]3		-.[illegible]5
[illegible] Inert +1.[illegible]	+141.3	-.[illegible]	+.[illegible]2	+.[illegible]	-.023	-.0[illegible]5	+.133	-.130	-.213	+.00[illegible]	-.[illegible]3
Weight-1[illegible]	+157.[illegible]	-.[illegible]31	-.[illegible]1	+.[illegible]	+.[illegible]26	-.001	-.[illegible]	+.041	-.[illegible]1	+.03[illegible]	-.[illegible]
[illegible] -.[illegible]°	+1[illegible].5	-.101	+.[illegible]2	+.[illegible]	+.029	0	+.107	-.106	+.[illegible]75	+.0[illegible]	+.00001
Alignment -.5°	+131.8	-.12[illegible]	-.[illegible]1	0	+.020	0	-.[illegible]	+.062	-.[illegible]36	+.03[illegible]	-.00001
Shutdown +[illegible] lb sec.	-[illegible].3	+.17[illegible]	+.[illegible]	+.[illegible]	-.036	-.001	+.[illegible]5	-.296	+.148	-.101	+.00003
Impulse -5000 lb sec.	+315.8	-1.15[illegible]	-.00[illegible]	0	+.052	-.009	+.06[illegible]	-.00[illegible]	-.518	+.063	-.00008

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[illegible]	Δr	[illegible]	[illegible]	[illegible]	Δh	ΔT	$\Delta\lambda_p$	$\Delta\Omega_p$	Δh_a	Δh_p	Δe
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[illegible] σ [illegible]											
Pitch [illegible]	-[illegible]	[illegible]	[illegible]	[illegible]	-[illegible]	+[illegible]	[illegible]	-[illegible]	[illegible]	-1.[illegible]	+.000[illegible]
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Yaw [illegible] °	[illegible]	-[illegible]	-[illegible]	+[illegible]	[illegible]	-[illegible]	-[illegible]	+.64	-.175	+.036	-.000[illegible]
Azimuth -1.[illegible]°	-4.8	-[illegible]	+[illegible]	-.1[illegible]	-.04	-[illegible]	+.216	-.316	+.110	-.071	+.00003
Pitch Gyro +1°/hr	[illegible]	-[illegible]	+[illegible]	[illegible]	+.492	-[illegible]	+4.18[illegible]	-4.178	-.300	-.108	+.000[illegible]
Drift -1°/hr	-[illegible]	+[illegible]	-[illegible]	+.009	-.487	+.051	-4.140	+4.102	+1.255	-.126	+.00014
Yaw Gyro +1°/hr	-36.5	-1.[illegible]	-[illegible]	+[illegible]	-.006	-.010	+.040	-.036	-1.024	-.006	-.00014
Drift -1°/hr	+33.0	+1.3[illegible]	+[illegible]	-[illegible]	+.006	+.018	-.001	+.028	+.051	-.066	+.00013
Pitch Program +5%	+4469.3	-1[illegible]	+.112	[illegible]	+.736	-.077	+5.540	-5.532	-1.606	-.653	-.00011
-5%	-4388.0	+11.829	-.137	+[illegible]	-.723	+.067	-5.234	+5.165	+1.022	-.010	+.00025
Roll Bias +.35°	+4.3	+.4[illegible]	[illegible]	-.01[illegible]	+.001	+.0[illegible]3	-.010	+.008	+.145		+.00002
-.35°	-10.8	-.3[illegible]7	0	+.013	-.002	-.004	+.002	0	-.188	-.002	-.00003
Inert Weight +21 Lb	+2254.[illegible]	-[illegible]	-.008	+.001	+.354	-.087	-.189	+.206	-1.02[illegible]	-.401	-.00082
-21 Lb	+1400.5	-3.45[illegible]	-.003	+.001	+.226	-.020	-.004	+.010	-1.311	+.242	-.00021

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	ΔK_1	[illegible]	$\Delta \gamma_1$	[illegible]	Δh_L	$\Delta \tau$	$\Delta \lambda$	$\Delta \Omega_p$	Δh_s	Δh_p	Δe
[illegible]	[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible].463	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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[illegible]-2A [illegible]	[illegible]	[illegible]	[illegible]	[illegible]	+1.605	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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[illegible]-2A [illegible] -[illegible]#	[illegible]	-11.[illegible]	-.[illegible]	[illegible]	+.968	[illegible]	[illegible]	[illegible]	-5.[illegible]	[illegible].520	-.00[illegible]
[illegible] Noise	[illegible]	+13.5[illegible]	+.111	+.[illegible]	+.833	+.143	+4.9[illegible]	+4.0[illegible]	+8.0[illegible]	+1.3[illegible]	+.00095
	-5079.4	-1[illegible]	-.111	-.0[illegible]	-.833	-.143	-4.946	-4.0[illegible]	-8.[illegible]	-1.3[illegible]	-.00095
[illegible]IB [illegible]	+[illegible]34.1	+1[illegible]	+.15[illegible]	+.0[illegible]	+1.463	+.121	+7.[illegible]	+7.[illegible]	+5.33[illegible]	+1.458	+.00087
	-8534.0	[illegible]	-.159	-.071	-1.385	-.13[illegible]	-7.55[illegible]	-7.905	-5.73[illegible]	-1.[illegible]	-.00087
System RSS Total	+14318.[illegible]	+25.[illegible]	+.194	+.[illegible]	+2.483	+.189	+9.48[illegible]	+10.44[illegible]	+10.18[illegible]	+3.50[illegible]	+.00137
	-11[illegible]	-33.[illegible]	-.[illegible]	-.[illegible]	-1.921	-.218	-10.463	-[illegible].504	-14.[illegible]	-[illegible].58[illegible]	-.00[illegible]

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[illegible]

[illegible]	[illegible] σ)			[illegible] - σ		
	[illegible]			Percent [illegible]		
	[illegible]	[illegible]	Sec. Stg.	Booster	Guidance	Sec. Stg.
Perigee [illegible]	*	[illegible]	41	[illegible]	13	38
Perigee Alt. [illegible]	[illegible]	16	17	32	26	41
Argument of Perigee [illegible]	3	27	70	36	22	52
Inclination [illegible]	3	34	63	8	36	56

* [illegible] eccentricity above [illegible] deviation is indeterminant for lower eccentricity.

of vehicle system. A new mission shall be assigned no later than launch minus 22 days (M-Day) and be prepared in accordance with the following schedule to bring the new-mission to repeat mission status:

a. M-day: [redacted] shall assign two missions, designated "Primary" and "Alternate", to be flown. One mission shall be a repeat mission. One may be a new mission. If two repeat missions are assigned the schedule outlined for repeat missions shall apply.

b. M + 1: LMSC shall transmit a Range Safety Supplemental Flight Data TWX to VAFB and PMR.

c. M + 3: LMSC shall transmit a nominal ascent trajectory for the new mission to Douglas Aircraft Corporation (DAC) and Bell Telephone Laboratories (BTL).

d. M + 4: LMSC shall complete the new mission ephemeris for internal use.

e. M + 4: LMSC shall transmit Range Safety nominal trajectory (magnetic tape) for the new mission to VAFB.

f. M + 6: DAC shall complete ascent program settings for the new mission.

g. M + 6: BTL shall transmit guidance coefficients (IBM card deck) for the new mission to LMSC and Univac.

3.1.9.1.2 Repeat Mission Preparation. Repeat mission preparation shall be based on propulsion system calibration (TAG) data of a specific vehicle system. A repeat mission shall be assigned no later than launch minus 17 days (L-17) and shall be prepared in accordance with the following schedule:

a. L - 17: [redacted] shall assign a repeat mission as the flight mission.

b. L - 12: DAC shall transmit the booster TAG performance data for the flight mission to LMSC.

c. L - 10: LMSC shall transmit the orbital timer program for flight mission to VAFB.

d. L - 8: LMSC shall transmit the final TAG ascent trajectory for flight mission to DAC and BTL.

3.1.9.1.3 [redacted] Options. [redacted] may assign a primary and alternate mission of a specific vehicle at any time consistent with the schedules specified in paragraphs 3.1.9.1.1 and 3.1.9.1.2. One of these missions may be a new mission. The other mission shall be a repeat mission. However, selection between the primary and alternate missions shall be in accordance with paragraph 3.1.9.1.2.

3.1.9.1.4 Lead Time Requirements. Lead time requirements for mission selection have been established to conform with Range Safety requirements and hardware preparation and checkout schedules necessary to support the scheduled launch date.

3.1.9.2 Items Requiring Setting or Other Action. Items requiring setting or other action shall include but shall not be limited to:

a. Agena vehicle and payload painting for thermal control as determined from sun angle history.

b. Agena recovery timer shorting connector patch

c. Agena standard timer, velocity meter, and guidance antenna

d. Satellite orbital program timer

e. Booster flight controllers

f. Ground control guidance computer controls

g. Satellite payload module(s) installed

h. Batteries and control gas

3.1.10 Orbital Lifetime.

3.1.10.1 Draglife. The draglife of an orbiting vehicle, based on limited flight data and theoretical considerations, is estimated to be within 25 percent of the lifetime shown in Figure 8 (section 3.1.7).

The existence of many physical characteristics which directly influence orbital lifetime have been observed. However, to date these variables are not well understood and their magnitudes have not been accurately determined. For these reasons absolutely accurate predictions of orbital lifetime are not possible. The physical variables which influence orbital lifetime are:

a. Atmospheric physical phenomena:

(1) Diurnal bulge due to heating (primarily significant above 130 nautical miles)

(2) Daily and seasonal density variations

(3) Solar activity effects (11-year cycle)

(4) Atmospheric oblateness

(5) Atmospheric tides

b. Earth/orbit relationship:

(1) Orbital plane inclination

(2) Argument of perigee

(3) Orbital plane/sun relationship

(4) Earth's oblateness

c. Vehicle contribution:

(1) Stable versus tumbling flight modes

(2) Tumble rate

(3) In-orbit weight changes

3.1.10.2 Battery Loading. Battery loading of the Agena shall be dependent on battery installation capability, performance capability, and mission power requirement. The maximum battery power output capability shall be five type IC batteries and the minimum battery power output capability shall be one type VI battery. However, the use of one type VI battery alone is impractical. The power capability and weight of various practical battery combinations are shown in Table III. The battery power capability shown in Table III is the 3 sigma low (99.87 percent probability of achieving or exceeding the power capability shown). The power capability of the batteries varies with temperature. Battery power provided shall exceed the minimum requirements by five percent. The battery weight capability is equal to the vehicle system weight-into-orbit capability less the vehicle and payload weight.

The mission power requirement shall be dependent on mission selection, such as active days, deactivated days, altitude and inclination. The nominal power usage and expected tolerances of the vehicle and payloads are shown in Table IV. The altitude and inclination shall affect the number of tracking stations acquired and the length of acquisition, therefore varying the power requirement.

3.1.10.3 Control Gas. The predicted control gas expenditures for the primary and lifeboat systems shall be as shown in Tables V and VI. These expenditures are based on computer analysis and post flight data. The control gas loading for the primary and lifeboat system to assure

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TABLE III

		BATTERY POWER 50°F	
[illegible]		POWER WATT - HRS	WEIGHT LBS
[illegible]	[illegible]		
1		[illegible],500	118
1	1	12,000	145
1	2	13,718	172
1	3	15,470	199
2	0	21,000	236
2	1	22,500	263
[illegible]	2	24,215	290
[illegible]	[illegible]	[illegible],270	317
		[illegible]	354
3	1	33,645	381
3	2	35,363	408
3	3	37,115	435
[illegible]	[illegible]	[illegible]	472
[illegible]	1	[illegible]	499
[illegible]	[illegible]	[illegible]	526
[illegible]	3	[illegible],370	553
		[illegible]	590

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TABLE IV*

NOMINAL POWER REQUIREMENT		
Item	Nominal Power Req/Day (Including Losses) Watt Hours	Expected Tolerance Watt Hours
[illegible] Radio Guidance	5[illegible] (ascent only)	± 10
[illegible]	1450	± 100
[illegible]	1250	± 120
[illegible] Payload	[illegible]00	+ 190 - 100
[illegible] Payload [illegible] [redacted]	[illegible]	± 10
[illegible]	[illegible]	± 60
[illegible] Vehicle w/TM [illegible] Twice Per Day Battery Capacity at [illegible]% used)	75	± 7
[illegible] Vehicle w/TM [illegible] Twice Per Day [illegible]	[illegible] (Including battery capacity [illegible])	± 25

[illegible] than 5 percent power margin between [illegible] power available.

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TABLE V

[illegible] GAS USAGE - PRIMARY SYSTEM

[illegible]	Minimum Impulse (lb-sec)	Nominal Impulse (lb-sec)	Maximum Impulse (lb-sec)
[illegible]	[illegible]	255	5?
[illegible]	[illegible]	280	255
[illegible]	[illegible]	490	600
[illegible]	[illegible]	80	100
[illegible]	[illegible]	130	200
[illegible]	[illegible]	130	255
[illegible]	[illegible]	4?0	520
[illegible]	[illegible]	80	100
[illegible]	[illegible]	255	255
[illegible]	[illegible]	[illegible]	36??

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TABLE VI

[illegible] GAS USAGE - LIFEBOAT SYSTEM

[illegible]	[illegible]	*Nominal Impulse	[illegible]
A[illegible]	[illegible]	([illegible])	([illegible])
[illegible]	[illegible]	1[illegible]0	[illegible]
[illegible]	[illegible]	2[illegible]	± [illegible]
[illegible]	[illegible]	250	[illegible]
[illegible]	[illegible]	2[illegible]	± [illegible]
[illegible]	[illegible]	3[illegible]	± [illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]		[illegible]	[illegible]

*[illegible] thrust valve [illegible]rque arm ratio (Iy/Ry) = 500

that the required impulse is available for mission completion shall be as shown in Table VII. Control gas loading has been standardized to include all missions, configurations, and lifeboat requirements. The lifeboat system requires 555 lb-sec available impulse. Therefore, the lifeboat sphere volume dictates the control gas mixture.

3.1.11 Guidance and Control System. The guidance and control system shall:

a. Provide attitude, time, and velocity references sufficient to control the vehicle along the specified trajectory to attain the prescribed orbit;

b. Provide attitude reference and control of the vehicle in orbit;

c. Provide the proper attitude and commands for the recovery capsule.

A block diagram of the guidance and control system is shown in Figure 17.

3.1.11.1 Guidance System.

3.1.11.1.1 Horizon Sensor. The horizon sensor (H/S) shall provide an earth reference for the vehicle by detecting infrared radiation contrasts between earth and space. The horizon sensor shall generate a corresponding output which shall be transformed into pitch and roll error signals. These signals shall be fed to the inertial reference package (IRP) in the form of torquing signals to the pitch, yaw, and roll gyros for vehicle attitude control.

3.1.11.1.2 Inertial Reference Package. The inertial reference package shall contain three single-degree-of-freedom rate-integrating gyros, each individually oriented so that it senses the angular displacement of the vehicle about one of the three major vehicle axes. The primary function of the three gyros shall be to maintain the vehicle in a fixed attitude with respect to inertial space. The gyros shall detect the difference between the attitude of the vehicle and the IRP reference attitude and shall generate an error signal with an amplitude proportional to the difference in attitude. This signal shall be processed through the flight control (F/C) electronics to the pneumatic and/or hydraulic components of the system.

3.1.11.1.3 Velocity Meter. The velocity meter shall sense vehicle change in velocity over a specified period of the engine burn time and shall send a signal for engine shutdown after the desired velocity has been achieved. The velocity meter shall consist of an accelerometer, electronics package, and a counter. Acceleration of the vehicle shall be sensed by the accelerometer and shall be processed in the electronics package. The electronics package output shall be proportional to

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[illegible] VII

[illegible] - [illegible]

[illegible]	[illegible]	[illegible]	[illegible]	ISP (lbs-sec / lb)	[illegible]	[illegible]	Pressure-Temperature [illegible] Limits
[illegible]	-	[illegible]	[illegible]	51	[illegible]	[illegible]	[illegible]
[illegible]	-		[illegible]	55	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	65	[illegible]	±13	[illegible]
[illegible] - LIFE [illegible]							
Volume (in³)	Gas Mixture	[illegible] Weight ([illegible])	Tolerance ([illegible])	ISP (lbs-sec / lb)	Impulse ([illegible]-sec)	Tolerance ([illegible]-sec)	Pressure-Temperature [illegible] Limits
[illegible]	-[illegible]	11	± 1	51	561	±51	Same as primary
[illegible]	-1	11.5	± [illegible]	55	[illegible]	±28	Same as primary
[illegible]	$[illegible]_2$	[illegible]	± [illegible]	65	[illegible]	±29	Same as primary

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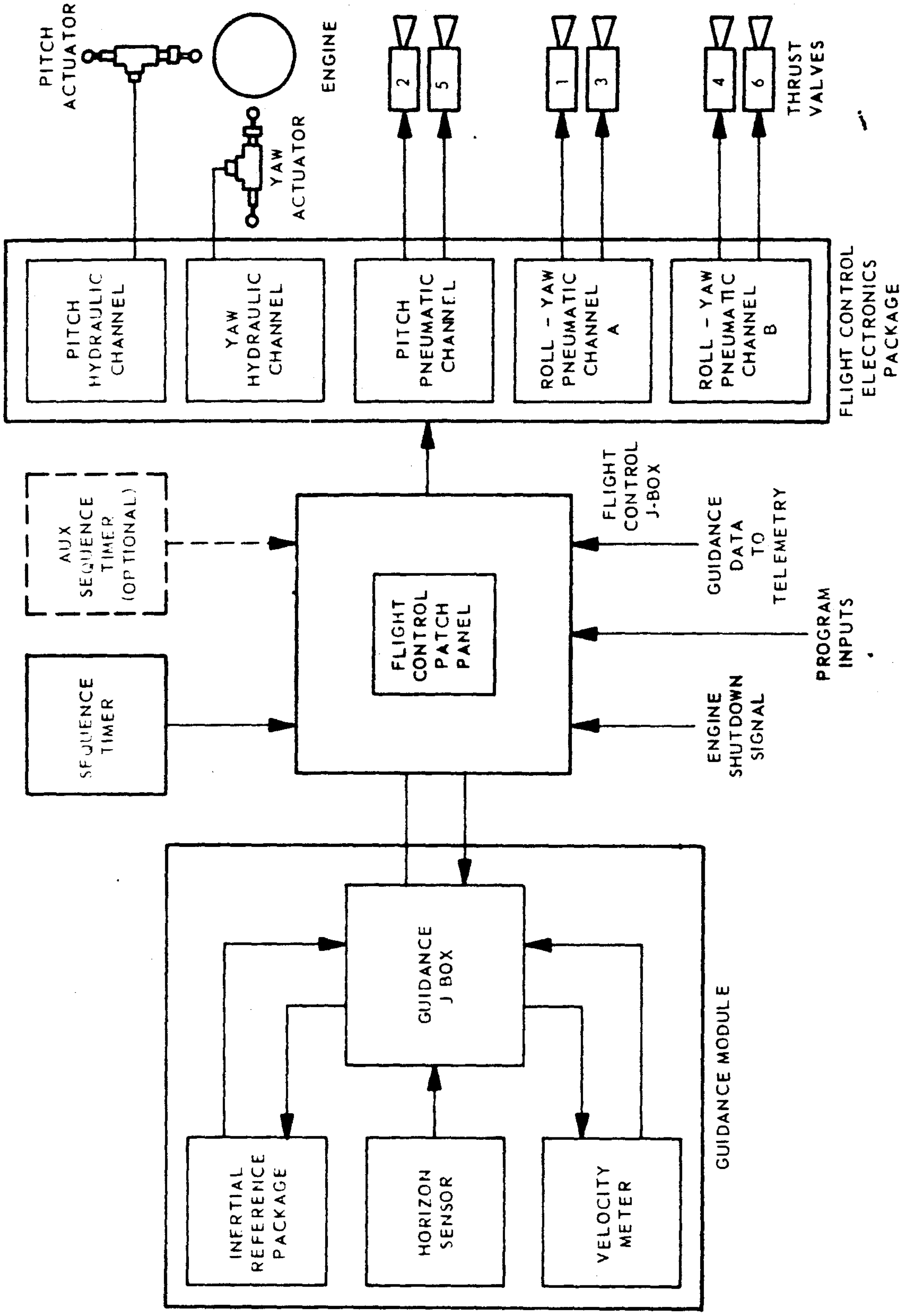

Figure 17. Guidance and Control System Block Diagram

vehicle acceleration. A pulse counter shall be used to count down the output of the electronics package and provide a switch closure when the required velocity has been achieved.

3.1.11.1.4 Junction Boxes. The guidance and flight control junction boxes shall provide the electrical means of interconnecting the guidance and F/C components, gain change logic, telemetry (T/M) conditioning circuitry, and fixed torquing program circuits. A patch panel, which is part of the F/C J-Box, shall be utilized for making program peculiar modifications.

3.1.11.1.5 Standard Timer. The standard timer shall dictate the sequence of guidance and control system ascent, deactivate, and re-activate functions as well as switching for other vehicle functions. The standard timer shall be an electro-mechanical device consisting of 72 cam-actuated switches and a three-phase motor. The timer setting resolution shall be 1.0 second with a repeatability of 0.2 second. It shall be capable of a 4000 second maximum operation and of providing 24 discrete events.

3.1.11.1.6 Recovery Timer. The recovery timer shall be a solid state device utilizing core logic and latching relays. Its function shall be to provide the required recovery sequence. The recovery timer shall provide the capability for thirteen switching events. The recovery timer shall be capable of being programmed for a maximum timing range of 0 to 98,304 seconds. The timer accuracy shall be 0.5 second or 0.1 percent between events, whichever is greater.

3.1.11.1.7 Vehicle-Borne Radio Guidance Equipment. The vehicle-borne radio guidance equipment shall provide steering corrections to the Thor booster and the Agena guidance system (pitch and yaw gyros) during ascent. Discrete commands shall be provided by the radio guidance system for booster main engine cut-off, booster separation, and velocity meter enable (start). The main engine cutoff (MECO) command also shall start the standard timer. The radio guidance system shall utilize a tracking station and ground computer to provide the flight trajectory corrections.

3.1.11.2 Flight Control System. The principal function of the flight control system shall be to provide control of the vehicle attitude in response to signals from the guidance system. Both the hydraulics and pneumatics shall be controlled by the IRP through the F/C electronics. During the engine sequence of operation, control of the thrust vector (pitch and yaw) shall be attained by the use of hydraulic actuators. Roll attitude shall be controlled by pneumatics. During all other phases of operation the pneumatics control system shall dictate vehicle attitude through the use of six thrust controllers for pitch, yaw, and roll control.

3.1.11.2.1 Flight Control System Performance. Performance parameters for the flight control system during the various phases of flight shall be as defined in the subsequent subparagraphs. The sequence of events and the times at which these events occur shall be in accordance with the applicable flight sequence of events for each vehicle.

3.1.11.2.1.1 Pre-Launch Phase. The guidance and control system shall perform no function during this phase. The system shall be conditioned for launch by application of power to the required equipment and loading of control gas.

3.1.11.2.1.2 Boost Phase. The boost phase shall begin at launch and terminate at vernier engine shutdown. The standard timer brake shall be released by the radio guidance system MECO discrete at which time the standard timer shall begin operation. Subsequently, by the VECO command, the IRP gyros shall be uncaged providing the IRP reference for the vehicle, and the horizon sensor fairings shall be ejected.

3.1.11.2.1.3 Coast Phase. The coast phase shall be initiated with the command separation discrete by radio guidance. Immediately following physical separation, control of the vehicle attitude and rates about all three axes shall be initiated, utilizing the vehicle reference attained at VECO. In addition, by utilizing the horizon sensor reference to the roll gyro, the roll axis shall be referenced to earth.

3.1.11.2.1.4 Orbital Boost Phase. The orbital boost phase shall begin with Agena engine ignition and shall terminate with Agena engine shutdown. During this phase the control about the pitch and yaw axes shall be provided by the hydraulic system, while roll shall be controlled by pneumatics. The guidance system shall be capable of maintaining the vehicle attitude within the following limits, assuming no disturbing torques:

a.	*Pitch:	Within plus or minus 1.3 degrees of the required pitch attitude
b.	Roll:	Within plus or minus 0.8 degrees of the roll reference
c.	*Yaw:	Within plus or minus 0.7 degrees of the required yaw attitude.

*The above values are based on guidance performance excluding radio guidance. Refer to Table VIII for contributing guidance and control error sources.

At a predetermined "remaining velocity-to-be-gained" point after ignition the radio guidance system shall enable or start the velocity meter (V/M). In turn, after the V/M has counted through the preset velocity-to-be-gained it shall command engine shutdown. The three sigma error in velocity-to-be-gained due to the velocity meter shall be less than 0.1 percent of the preset velocity.

TABLE VIII

ORBITAL BOOST PHASE (Radio Guidance Excluded) ERROR ANALYSIS

No.	Type	Error Source	Limit of Error	LIMITS OF ERROR Yaw	Roll	Pitch	Distribution
1	Random	Horizon Uncertainty	0.1°	--	0.10	--	Normal
2		Horizon Sensor Noise	0.3°	--	0.30	--	
3	Random	Limit-cycle		0.2	0.6	0.2	Uniform
4		Horizon Sensor Mis-align		--	0.15	--	
5		Horizon Sensor Acc.		--	0.30	--	
6		Pitch Program Acc.		--	--	0.2 1.09	
7		Deadband Tolerance		.1	.3	.1	
8		Pitch Gyro Drift	8°/hr	--	--	0.58	
9	BIAS	Yaw Gyro Drift	8°/hr	0.58	--	--	Unknown
10		Roll Gyro Drift	1.5°/hr	--	0.08	--	
		Yaw Error due to (4), (5)		0.16	--	--	
11		Yaw Gyro Mis-align	0.15	0.15	--	--	
		Pitch Gyro Mis-align	0.15	--	--	0.15	
TOTAL RSS				0.65	0.82	1.29	

3.1.11.2.1.5 Orbital Conditioning/Reorientation Phase. The orbital conditioning/reorientation phase shall start with engine shutdown and shall terminate with yaw-around. The primary guidance function shall be to condition the vehicle for the orbit mode, i.e., activate pitch and yaw pneumatics, initiate oxidizer dump and fuel dump, initiate utilization of the horizon sensor pitch reference, and reorient the vehicle 180 degrees about the yaw axis. Errors in the yaw-around attitude due to oxidizer and fuel dump torques, command program tolerance, and standard timer tolerance shall be corrected by the horizon sensor. The three sigma roll or yaw error at yaw-around termination shall be within plus or minus nine degrees. The horizon sensor shall correct the offset encountered within approximately 30 minutes.

3.1.11.2.1.6 Orbital Phase. In the orbital configuration the vehicle roll axis (X axis) is nominally along the local horizontal and in the plane of the orbit. The guidance and control system RSS limits of angular deviations of the vehicle from the required attitude shall be less than plus or minus 0.76 degree in pitch, plus or minus 0.73 degree in roll, and plus or minus 1.09 degrees in yaw. Refer to Table IX for contributing guidance and control error sources. The limit cycle characteristics shall be as follows for all three axes:

Maximum peak-to-peak amplitude:	1.2 degrees
Rates:	60 degrees per hour
Period:	<200 seconds

NOTE: Unperturbed, except for steady-state torques of 0.01 ft.-lb.

3.1.11.2.1.7 Recovery Phase. The guidance recovery phase shall be initiated with the start of the recovery timer. After timer start the guidance gains shall be switched to the ascent mode and the horizon sensor reference disconnected. In addition, the vehicle shall begin the required pitch-over maneuver. Following the pitch down reorientation of nominally 60 degrees, the control system error contribution to the angle between the vehicle longitudinal axis and the local horizontal shall be within plus or minus 3.06 degrees, while the contribution to the yaw deviations shall be within plus or minus 1.44 degrees, and the contributions to the roll deviations shall be within plus or minus 1.[illegible] degrees. Subsequently the timer shall dictate the capsule eject sequence after which the orbit gains are again invoked, the sensors reconnected, and the timer reset. Contributing to guidance errors during this phase are the initial conditions, the recovery timer, and pitch program inaccuracies. Refer to Table X for contributing guidance and control error sources.

3.1.11.2.1.8 Deactivate. The deactivate sequence shall be initiated with a real-time command which starts the standard timer deactivate

TABLE IX

ORBITAL CONTROL SYSTEM ERROR ANALYSIS

No.	Type	Error Source	Limit of Error	*LIMITS OF ERROR Yaw	Roll	Pitch	Distribution
1	Random	Horizon Uncertainty	0.1°	--	0.10	0.10	
2	Normal	Horizon Sensor Noise	0.3°	--	0.30	0.30	Normal
3	Random	Arc Sin Orbital Eccentricity	0.1°	--	--	0.10	Arc Sin
4	Random	Uniform Limit Cycle --SS Torque	0.5°	0.5	0.50	0.50	Uniform
	RSS						
5		Horizon Sensor Mis-Align.	0.15	0.15	0.15	0.15	
6	Fixed	Horison Sensor Accuracy	0.30	--	0.30	0.30	
7	BIAS	Yaw Error Due To (5)	--	0.07	--	--	Unknown
8		Yaw Error Due To (6)	--	0.14	--	--	
9		T/M & Data Proc.	0.1°	--	0.10	0.10	
10		A/P Mis-alignment	0.15°	0.15	0.15	0.15	
11		Attitude Deadband Tol.	20%	0.10	0.10	0.10	
12		Pitch Program Accuracy	5%	--	--	0.20	
[illegible]		Pitch Gyro Drift	8°/hr	--	--	0.13	
[illegible]		Yaw Gyro Drift	8°/hr	0.86	0.18	--	
[illegible]		Roll Gyro Drift	1.5°/hr	0.36	--	--	
	RSS TOTAL RSS			[illegible]	0.73	0.76	

* Unperturbed, except for steady-state torques of [illegible] .01 ft-lb.

TABLE X

ORBITAL CONTROL SYSTEM CONTRIBUTION TO RECOVERY ATTITUDE ERROR

RECOVERY PHASE

Error Source	LIMITS OF ERROR		
	Yaw	Roll	Pitch
Attitude Control Initial Condition with Ascent Deadbands	0.99	0.80	0.60
Pitch Program Error			3.00
Roll-Yaw Coupling w/$\dot{e}_c$	1.05	1.05	
Recovery Times Accuracy ±0.5 sec. At Retro Ignition			
RSS	[illegible]	1.32	3.06

NOTE: The effect of residual vehicle rates is negligible.

sequence. The main functions of the sequence shall be to arm the guidance system for reactivate, to spin-stabilize the vehicle into a three-degree-per-second plus or minus 0.3 degree-per-second tumble rate about the pitch axis for temperature stabilization, and to switch vehicle power to external (28 unregulated bus OFF). Initially the vehicle shall attain the desired tumble rate but it may be affected by torques caused by leaks, aerodynamic drag, and other factors, which result in a rate change (increase or decrease). Guidance limitations relative to the length of the deactivate period are:

a. The IRP gyro temperature shall always be above zero degrees F.

b. The H/S head temperature shall always be above minus 20 degrees F.

3.1.11.2.1.9 Reactivate. The reactivate sequence of the vehicle shall be initiated by a real time command which switches the vehicle back to internal power and starts the standard timer. The guidance system IRP and horizon sensor head thermostat controlled heaters, armed during the deactivate sequence, shall be repowered for approximately 3060 seconds. This shall allow the IRP temperature to attain the minimum spin-motor-start temperature of plus 120 degrees F. In addition, it shall allow the horizon sensor heads to heat up above the minimum start temperature for the motors of plus 20 degrees F. Subsequently, the IRP gyro spin motors and horizon sensor motors, turned off during the deactivate sequence, shall be repowered and the F/C pneumatics shall be turned on. This shall facilitate the beginning of the restabilization process. The guidance system shall restabilize the vehicle to the required orbital parameters within 30 minutes (50 percent confidence level) to 90 minutes (99 percent confidence level).

3.1.11.3 Lifeboat System. The lifeboat (L/B) system shall be an auxiliary system for backing up the primary recovery system. It utilizes:

a. An independent auxiliary RF command link between the ground station and the orbiting vehicle, and

b. An independent attitude control system.

Availability of these back-up functions provides redundancy of all significant aspects of recovery, except batteries.

3.1.11.3.1 Major Components. The major components of the lifeboat system and their relationship to each other are shown in Figure 18 and described in the subsequent paragraphs.

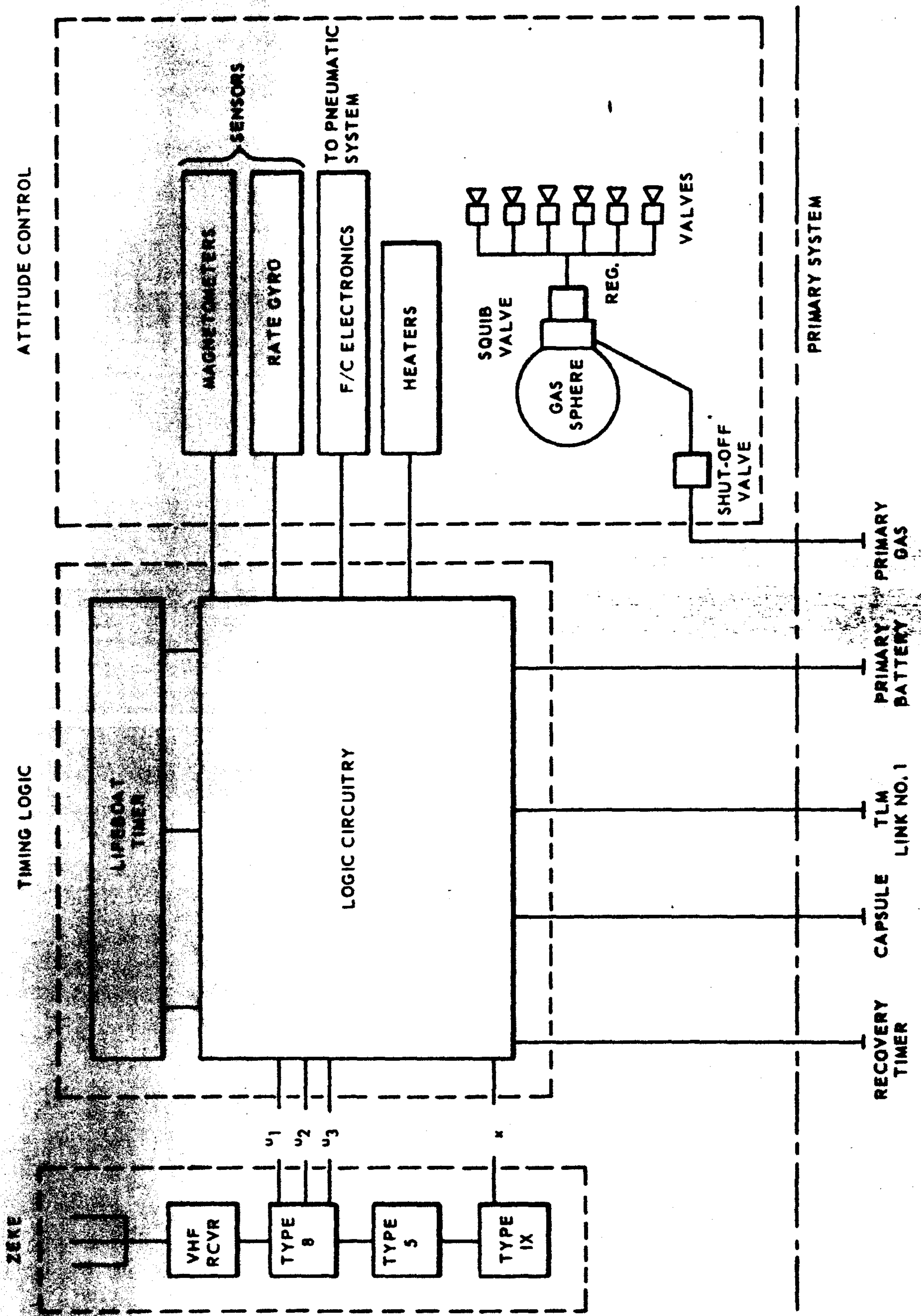

Figure 16. Lifeboat System Block Diagram

3.1.11.3.1.1 Zeke Communications Equipment. The Zeke equipment shall provide the communications link from the ground to the vehicle for the lifeboat system. The Zeke system shall meet the requirements of section 3.1.14.2.3.3.

3.1.11.3.1.2 Control System Equipment. The function of the attitude control system shall be to align the vehicle with the earth's magnetic field vector. The attitude control equipment is specified in two general categories, namely, electronics and pneumatics.

3.1.11.3.1.2.1 Electronics

3.1.11.3.1.2.1.1 Magnetometer. The magnetometer shall be the attitude sensing device for the lifeboat system. It shall be comprised of an electronics unit and a sensor unit. In the sensor unit there shall be three orthogonally arranged probes containing a highly permeable magnetic core surrounded by an excitation and a signal pickup winding. These probes shall sense magnetic field intensity along their sensitive axis and shall supply a signal, proportional to the intensity, to the electronics unit. The output signal shall be proportional to attitude offset about the lifeboat axes.

3.1.11.3.1.2.1.2 Roll Rate Gyro. The function of the roll rate gyro shall be to control the rate of change of vehicle motion about the roll axis. It shall be designed to limit the motion to plus or minus one degree per second for proper operation of the lifeboat system. The gyro shall sense the rate and shall supply a signal proportional to the rate into the F/C electronics roll channel.

3.1.11.3.1.2.1.3 Flight Control Electronics. The flight control electronics shall contain three electronic channels and telemetry monitoring circuitry. Its function shall be to accept signals from the magnetometers and rate gyro and to convert them into appropriate commands to the pneumatic system for control.

3.1.11.3.1.2.1.4 Lifeboat Timer. The lifeboat timer shall be a solid state device utilizing core logic and latching relays. Its function shall be to initiate the required lifeboat sequence. The timer shall provide the capability of thirteen switching events. It shall be capable of being programmed for a maximum timing range of 0 to 98,304 seconds. The timer accuracy shall be 0.5 second or 0.1 percent between events, whichever is greater.

3.1.11.3.1.2.1.5 Lifeboat Junction Box. The lifeboat junction box shall provide the electrical means of interconnecting the lifeboat components. It also shall incorporate telemetry monitoring circuitry for the system functions, and landline control monitor points.

3.1.11.3.1.2.2 Pneumatics. The pneumatics portion of the lifeboat system shall provide the means for changing the vehicle attitude through the use of six thrust controllers, supplied by a storage sphere through a pneumatic regulator. The valves shall operate with an ON/OFF type operation at a 10 pound nominal thrust level. The regulated pressure shall range from 110 to 130 psig. Utilization of squib valves in the primary and lifeboat systems allows turn ON/OFF control of the gas supply to the valves.

3.1.11.3.2 System Performance. The lifeboat system shall be capable of orienting an unstable vehicle, having rates of up to 20 degrees per second along any axis, within 1.5 minutes, and holding the orientation for more than 30 seconds. The lifeboat system shall be capable of overcoming a guidance and control failure, a primary command system failure, and a combination of primary failures requiring immediate action. Three basic modes of operation for the lifeboat system shall consist of two complete lifeboat modes using the lifeboat control and pneumatic system, and one mode using only the lifeboat timer and Zeke system. The two complete lifeboat modes are designated U_1 and U_3. The other mode is designated U_2. The Lifeboat system is designed for a one-time per flight operation of the pneumatics. The usage of U_1 or U_3 is therefor controlled by this requirement. The vehicle conditioning of the various lifeboat modes shall be accomplished through the Zeke link Type 8 decoder. Each of the modes shall have a Zeke unsecure Type 8 command assigned for it. The function of the command shall be to condition relays in the lifeboat junction box so as to enable the required mode functions. The timer start command shall be a Zeke secure command specifically assigned for this purpose. Two such commands shall be available, enabling the start of the lifeboat timer twice per flight. For security purposes, the command used to start the lifeboat timer shall thereafter be permanently locked out by relay logic in the lifeboat junction box. In addition, the timer start event in each mode shall lock out the Zeke unsecure mode commands so that no inadvertent mode change command can enter while the sequence is in progress. The operational modes are described in the typical sequence of Table XI. The contributing recovery error sources from the Lifeboat system shall be as shown in Table XII.

3.1.11.3.2.1 U_1 Mode (Lifeboat Next-Orbit). The U_1 mode shall be designed for vehicle reorientation and capsule recovery one orbit after the start of the lifeboat timer to overcome a failure of the primary guidance and control system where time is not a factor or immediate recovery is not ideal. This mode shall use the lifeboat control and pneumatic system for reorientation. Initially the system shall be conditioned to the mode by the applicable Zeke unsecure command. Because U_1 is a complete lifeboat mode, using lifeboat pneumatics, the primary pneumatics shall be turned off by activation of the normally-open squib valve at lifeboat timer start. Subsequently, approximately one orbit after timer start, the reorientation maneuver shall begin by the arming of the lifeboat pneumatics (activation of the normally closed squib valve) and the turning on of control system power. The total time from start of reorientation to capsule ejection

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TABLE XI

RELAY	TIME	L/B NEXT [illegible] (T_1)	T-TIMER NEXT [illegible] (T_2)	L/B REAL TIME MODE (T_3)
	Zeke "A"	Turn on Link I T/M, F/C, Mag., Gyro, & Htrs.	Turn on Link I T/M, F/C, Mag., Turn Gyro & Htrs. Off. Reset L/B Timer "B" Sequence	Turn on Link I T/M, F/C, Mag., Gyro, & Htrs.
	Zeke Secure Command (L/B Execute)	Prim. Pneu. Off, Disable Type 8 Decoder Output, L/B Timer "B" Sequence Start	Rec. Timer Pwr On, Disable Type 8 Decoder Output, Arm. Rec. Timer Initiate	Disable DRP & Type 8 Decoder Output, Prim. Pneu. Off, L/B Pneu. On, Valve Pwr On, T/M Off Lockout Pwr On, L/B Pwr to Turn On F/C & Mag., L/B Timer "B" Sequence Start
K502(T[illegible])	X+0	Lockout L/B Timer Restart, Disable Start Command Utilized	Lockout L/B Timer Restart, Disable Start Command Utilized	Lockout L/B Timer Restart, Disable Start Command Utilized
K503(T_1)	X+25	Remove Pwr for Prim. Pneu. Off	No Effect	Arm, Remove Pwr for Pneu. Off
K504(T_2)	X+100	No effect	No effect	Transfer
K505(T_3)	X+102.5	No effect	No effect	Separate, Remove Valve Pwr, Remove T/M Off Lockout Pwr, Remove L/B Pwr for Mag & F/C Turn On, Remove L/B Pneu On & DRP Disable Pwr
K501	$U+T_A$ or $(U+600) \pm 30$	Turn Link I T/M, F/C, & Mag. Off	Turn Link I T/M, F/C, & Mag. Off	Turn Link I T/M, F/C, & Mag. Off

FORM LMSC [illegible]

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TABLE XI (Continued)

[illegible]	[illegible]	[illegible] MODE (U_1)	[illegible]-TIMER [illegible] (U_2)	L/B REAL TIME MODE (U_3)
K5[illegible]	X+[illegible]	[illegible] Link [illegible] T/M, [illegible] Mag. [illegible] Initiate "A"	Turn Link [illegible] T/M, F/C, & Mag. On, Initiate "A"	Turn Link I T/M, F/C, & Mag. On, Initiate "A"
K[illegible]	X+[illegible]	[illegible] Backup	K5[illegible] Backup	K5[illegible] Backup
K5[illegible]	X+[illegible]	L/B Pneu. [illegible] Disable DRP. Pwr. [illegible] L/B Valves, T/M Off Lockout Pwr On. L/B Pwr [illegible] Turn On [illegible] & Mag	No Effect	No Effect
K5[illegible](T_7)	X+5[illegible].5	K5[illegible] Backup	Initiate Rec. Timer "B" Sequence, T/M Off Lock-out Pwr On	No Effect
K510(T_8)	X+5495	Arm	No Effect	No Effect
K511(T_9)	X+55[illegible]	Transfer	No Effect	No Effect
K512(T_{10})	X+5562.5	Separate	No Effect	No Effect
K513(T_{11})	X+5590	Remove Valve Pwr, Remove DRP Disable Pwr, Remove Gyro & Htr. Pwr, Remove T/M Off Lockout Pwr; Remove Arm, Transfer, & Separate Pwr, Remove L/B Pwr for F/C and Mag. Turn On, Remove L/B Pneu. On [illegible]	Remove Rec. Timer Initiate Pwr, Remove T/M Off Lockout Pwr	Remove Arm, Transfer and Separate Pwr., Remove Gyro & Htr Pwr.

TABLE VI (continued)

[illegible]	[illegible]	[illegible] MODE (U_1)	-TIMER NEXT [illegible] (U_2)	L/B REAL TIME MODE (U_3)
K[illegible] (T_1)	[illegible]	[illegible] Type [illegible] Output, Arm Reset "B", Remove [illegible] Timer Start Pwr, K[illegible] Backup	Remove Rec. [illegible] Pwr on Sig., Enable Type [illegible] Decoder Output, Arm Reset "B", Remove L/B Timer Start Pwr, K513 Backup	Enable Type B Decoder Output, Arm Reset "B" Remove L/B Timer Start Pwr. K513 Backup
	T_L+T_A ([illegible])	Turn Off Link I T/M, F/C Mag.: Reset "B" Sequence, Enable Timer Restart	Turn Off Link I T/M, F/C Mag.: Reset "B" Sequence, Enable Timer Restart	Turn Off Link 1 T/M, Mag.: Reset "B" Sequence, Enable Timer Restart

Tolerance on timer between events is 1.0 sec. or 0.11%, whichever is greater except as specified in sequence. Zero time reference shall be the instant the execute command reaches the output of the decoder.

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TABLE XII

LINE [illegible] ATTITUDE CONTROL ERRORS

NO.	[illegible] SOURCE	IN RE-ENTRY PLANE	OUT OF RE-ENTRY PLANE
1	Accuracy of Magnetic Vector Predictions	2.0°	2.0°
2	Deadband tolerance	1.0°	1.0°
3	Nominal Deadband Setting	6.0°	6.0°
4	L/B T[illegible] accuracy at capsule separation point = ±6.0 sec		
5	Magnetometer Misalignment	±0° 30'	±0° 30'

NOTE: The effect of residual vehicle rates is negligible.

	RSS L/B ERROR contribution	± 6.4°	± 6.4°

shall be 102.5 plus or minus 2 seconds. The lifeboat junction box relays and subsequently the timer itself shall then be reconditioned to the reset position.

3.1.11.3.2.2 U_2 Mode (Recovery-Timer Next-Orbit). The U_2 mode is also designed for capsule recovery one orbit after lifeboat timer start. In the U_2 mode the recovery reorientation and capsule functions shall be initiated by the recovery timer which shall be started by the lifeboat timer. As described for the U_1 mode, the U_2 mode shall also be initiated by a Zeke unsecure command. The U_2 mode is the launch configuration of the lifeboat system. Therefore, in addition to the Zeke command conditioning, a landline reset shall also condition the system to a U_2 mode. In the U_2 mode the Zeke start signal for the lifeboat timer shall also turn on recovery timer power. Approximately one orbit later, the lifeboat timer shall start the recovery timer sequence for reorientation and recovery. Subsequently, the lifeboat system shall be reconditioned and the lifeboat timer reset.

3.1.11.3.2.3 U_3 Mode (Lifeboat Real-Time). In the event of a primary system control failure or power degradation when immediate recovery is required or desired, the lifeboat real-time U_3 mode shall be available. The initiation of the reorientation and recovery sequences in this mode shall be coincident with the lifeboat timer start event, facilitating a present orbit or real-time usage of the system. Again the start of reorientation through capsule separation sequence shall be 102.5 plus or minus 2 seconds, after which the system logic and timer shall reset.

3.1.12 Recovery System.

3.1.12.1 System Description. The recovery system shall use a government-furnished reentry subsystem, an LMSC parachute subsystem, and a government-furnished recovery force. The recovery system shall be capable of accommodating either one or two reentry vehicles on the Agena. The basic configuration and system breakdown shall be as shown in Figure 19. The interface between the Agena and the reentry subsystem shall be as shown in Figures 20 and 21 for a single and a double subsystem configuration, respectively. The primary mission for the recovery system shall be the return of payload from orbit. This shall be accomplished by separation of the reentry subsystem from the Agena, de-orbiting, reentry, and subsequent parachute deployment and ablative shield separation. Recovery shall be effected by locating the descending capsule by means of recovery aids, and accomplishing aerial recovery with a JC-130B aircraft. As a backup in the event air recovery is not attempted, or successful, the capsule shall float and shall be acquired by a surface force.

Reentry system ejection shall be achieved on either north to south or south to north passes over the Hawaiian recovery area by use of the

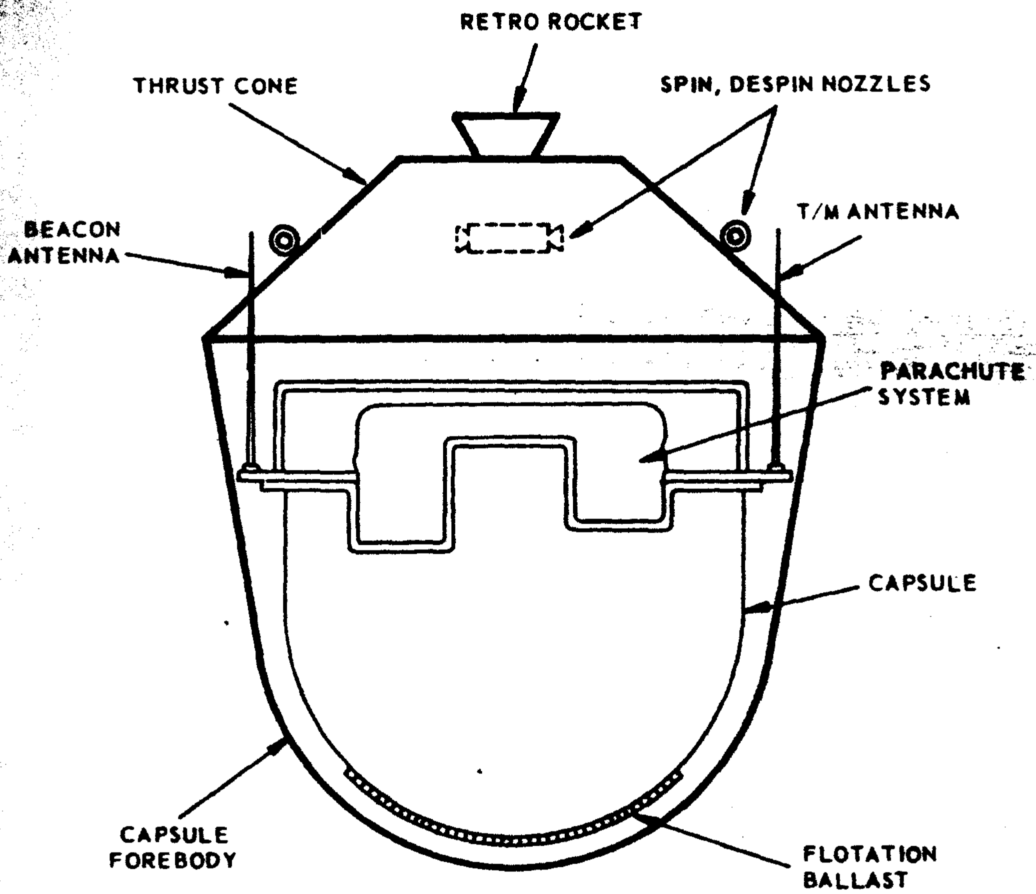

Figure 19. Recovery System

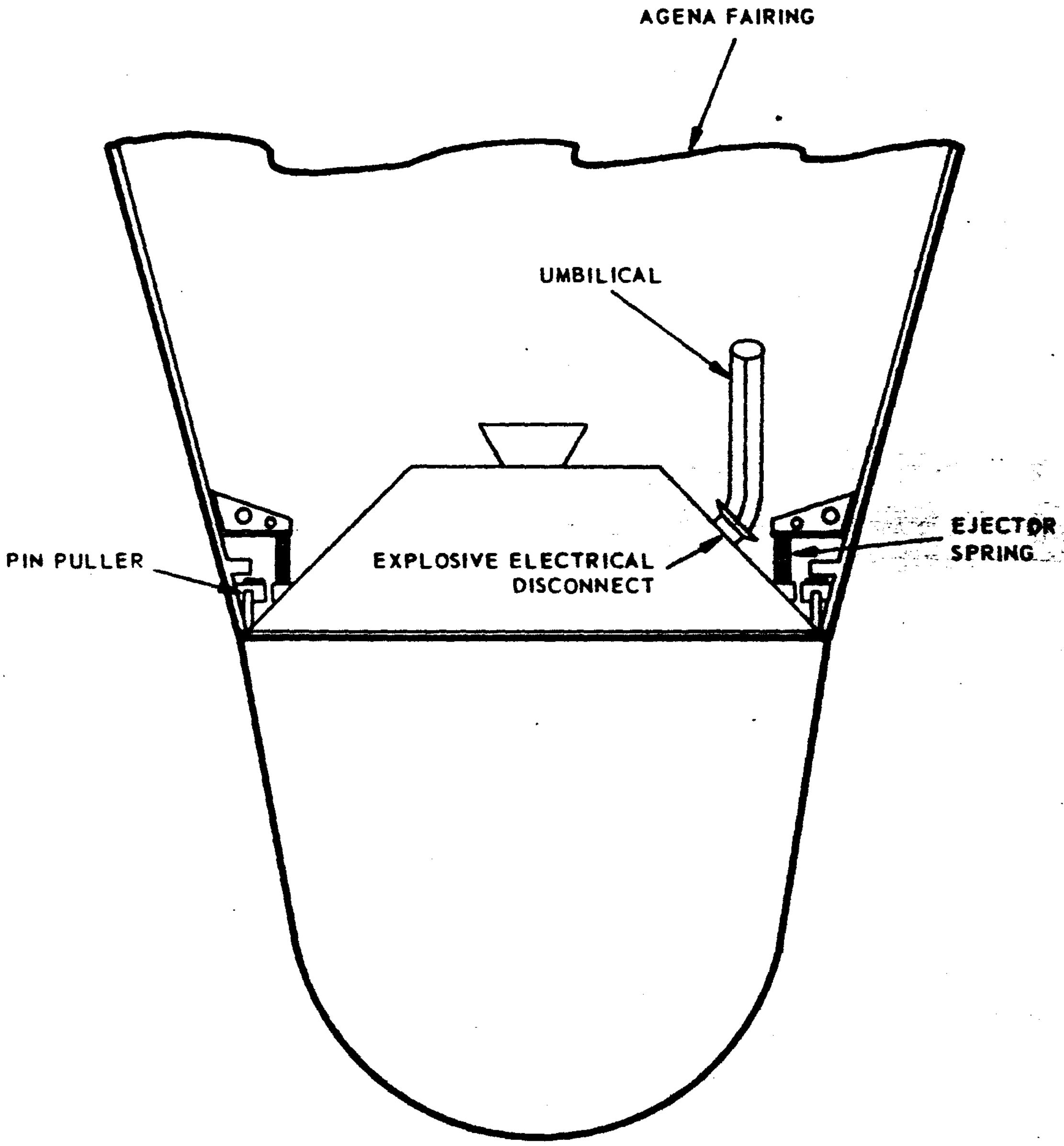

Figure 20. Single Reentry System

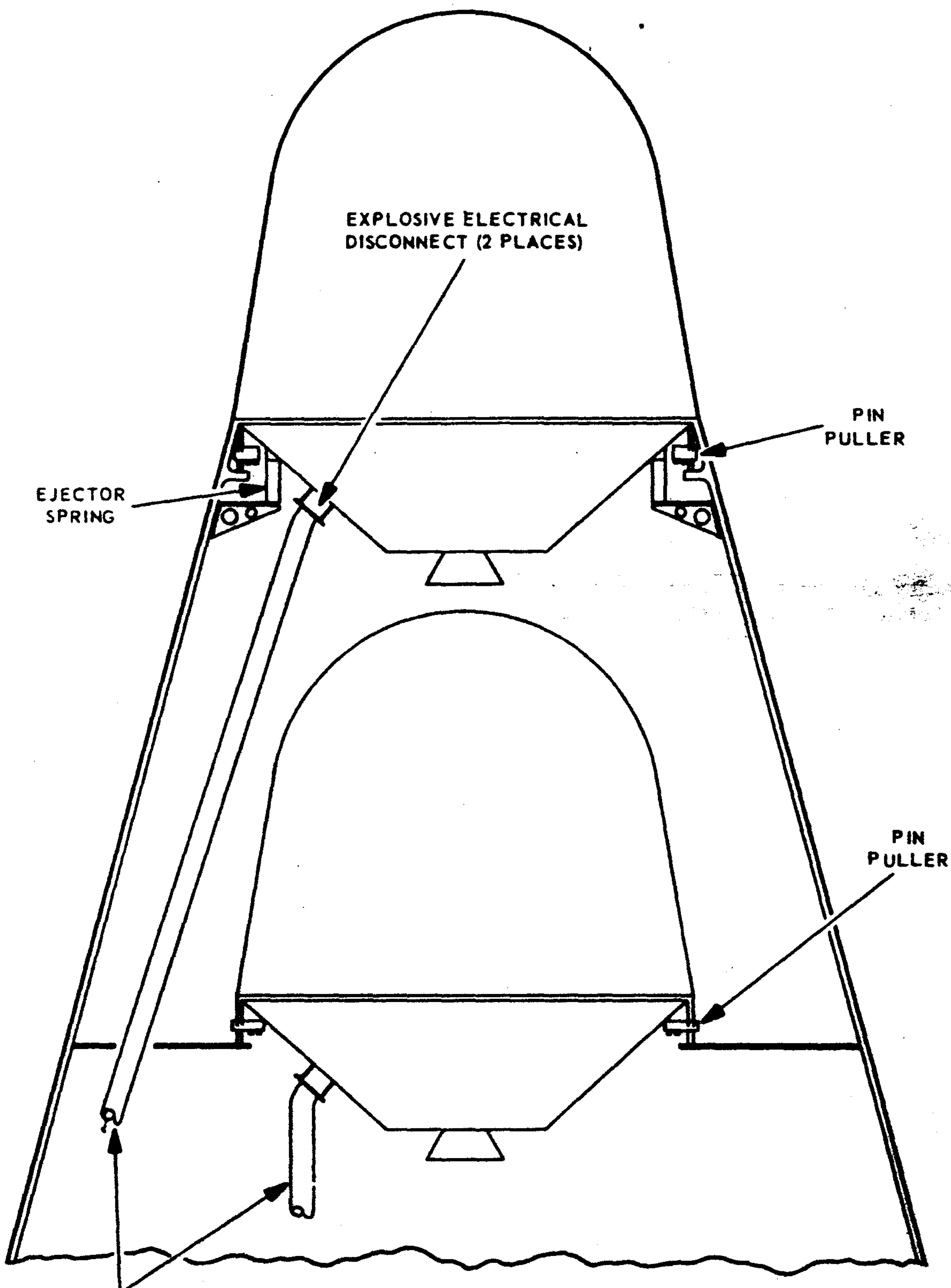

Figure 21. Double Reentry System

Agena primary guidance and control (G&C) system. Only north to south recovery shall be achieved by use of a backup guidance and control (G&C) system (lifeboat). If a south to north recovery is utilized, air recovery cannot be programmed as de-orbit would take place during the period of darkness and the JC-130B aircraft are not equipped for same. Recovery thus must be by means of the surface force.

Primary impact latitude for Hawaii recovery zone shall be 24 degrees north on north to south and 16 degrees north for south to north passes. See section 3.1.12.2 for operational limits and constraints that the reentry subsystem places on overall recovery system performance, correlative to the initial conditions resulting from the Agena orbit parameters and the G&C subsystem accuracies effects upon reenty system dispersion.

The nominal sequence of events for the recovery system is shown in Table XIII. Note that in the event the attitude control of the Agena and the reentry system separation commands are supplied by the backup system (lifeboat), a change in the initial phases of the sequence shall occur. See Table XIV.

3.1.12.2 System Characteristics.

3.1.12.2.1 Operational Capabilities.

3.1.12.2.1.1 Primary Recovery System. The operational capability of the primary recovery system shall be as shown in Figure 22 for a re-entry vehicle weight range of 230 to 308 pounds. Figure 22 indicates that re-entry vehicles within the assumed weight range can be successfully placed into a north to south (NTS) or south to north (STN) reentry trajectory for impact at 24 degrees north latitude (NTS) and 16 degrees north latitude (STN) from orbits having any combination of perigee altitude and period described by the envelope presented. These data are based on the assumption that perigee was located between [illegible] degrees north latitude and the most northerly latitude attainable with an orbit inclination range of 60 degrees to 120 degrees.

3.1.12.2.1.2 Backup Recovery System. The operational [illegible] of the back-up recovery system shall be as shown in Figures [illegible] for re-entry vehicle weights of 230 and 308 pounds, respect[illegible] data presented indicate the various combinations of perigee [illegible] and orbital period for which the system would permit the [illegible] of reentry trajectories (north to south) terminating in impact [illegible] degrees north latitude or less. Impact at 24 degrees north [illegible] becomes less likely as the perigee altitude is increased and [illegible] location is rotated north beyond 30 degrees latitude. The [illegible] approximated limiting zones present the systems capability [illegible] recovery and indicate the most northerly nominal impact loca[illegible]

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TABLE XIII

PRIMARY RECOVERY SEQUENCE OF EVENTS

FUNCTION	Time (Seconds)	Tolerance (Seconds)
"D" Timer Start (Fairing Ejects for 2nd Capsule)	- 81.0	± .5
Arm Signal	- 75.0	± .5
Transfer	0	
Electrical Disconnect	.90	+ .33 - .5
Separation	2.0	± .25
Spin	4.3	± .30
[illegible]	11.85	± .45
De-Spin	22.60	± .54
Thrust Cone Separation	24.12	± .15
Voltage Monitor Closed	96.90	± 40.0
Voltage Monitor Open	106.90	± 40.0
"G" Switch Open	0	-
[illegible] Cover Off	34.0	± 1.5
Deceleration Chute Deployed	34.6	± .05
Main Deployment Bag Separation	44.56	± 1.5
Main Chute Deployed (Reefed)	45.36	± .25
Main Chute Full Open	50.28	± .74
Arm Capsule Ballast System	50.60	+ 10.0 - 4.0
[illegible] Recovery - 15,000 Feet or Lower		
Water Impact If not aerially recovered - Ballast Releases	-	

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TABLE XIV

BACK-UP RECOVERY SEQUENCE OF EVENTS

FUNCTION	Time (Seconds)	Tolerance (Seconds)
[illegible] Timer Start (Fairing Separate 2nd Cap)	- 100	± .5
Arm Signal	- 75	± .5
Transfer	0	-
Electrical Disconnect	.9	+ .43 - .40
Separation	2.5	± .25
Spin	4.3	± .30
Retro	11.35	± .45
De-Spin	22.60	± .54
Thrust Cone Separation	24.12	± .15
Voltage Monitor Closed	104.9	44.0
Voltage Monitor Open	214.9	25.0
"G" Switch Open	0	-
Parachute Cover Off	34.0	± 1.5
Deceleration Chute Deployed	34.63	+ .03 - .06
Main Deployment Bag Separation	44.77	+ .43 - .40
Main Chute Deployed (Reefed)	45.29	+ .49 - .29
Main Chute Disreef	50.28	± .75
Arm Capsule Ballast System	50.60	+10.0 - 4.0
Aerial Recovery - 15,000 Feet or Lower		
Water Impact If not aerially recovered Ballast Releases		

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CONFIDENTIAL

REENTRY VEHICLE WEIGHT = 230 & 308 LBS
(RETRO VELOCITY = 1150 & 909 FPS, RESPECTIVELY)

i = 60° TO 120°

Λ_p = 22°N TO MAXIMUM °N

REENTRY = NORTH TO SOUTH OR SOUTH TO NORTH

i = ORBITAL INCLINATION

Λ_p = LATITUDE OF PERIGEE AT $90° < \bar{\omega}_p < 180°$

$\bar{\omega}_p$ = ARGUMENT OF PERIGEE

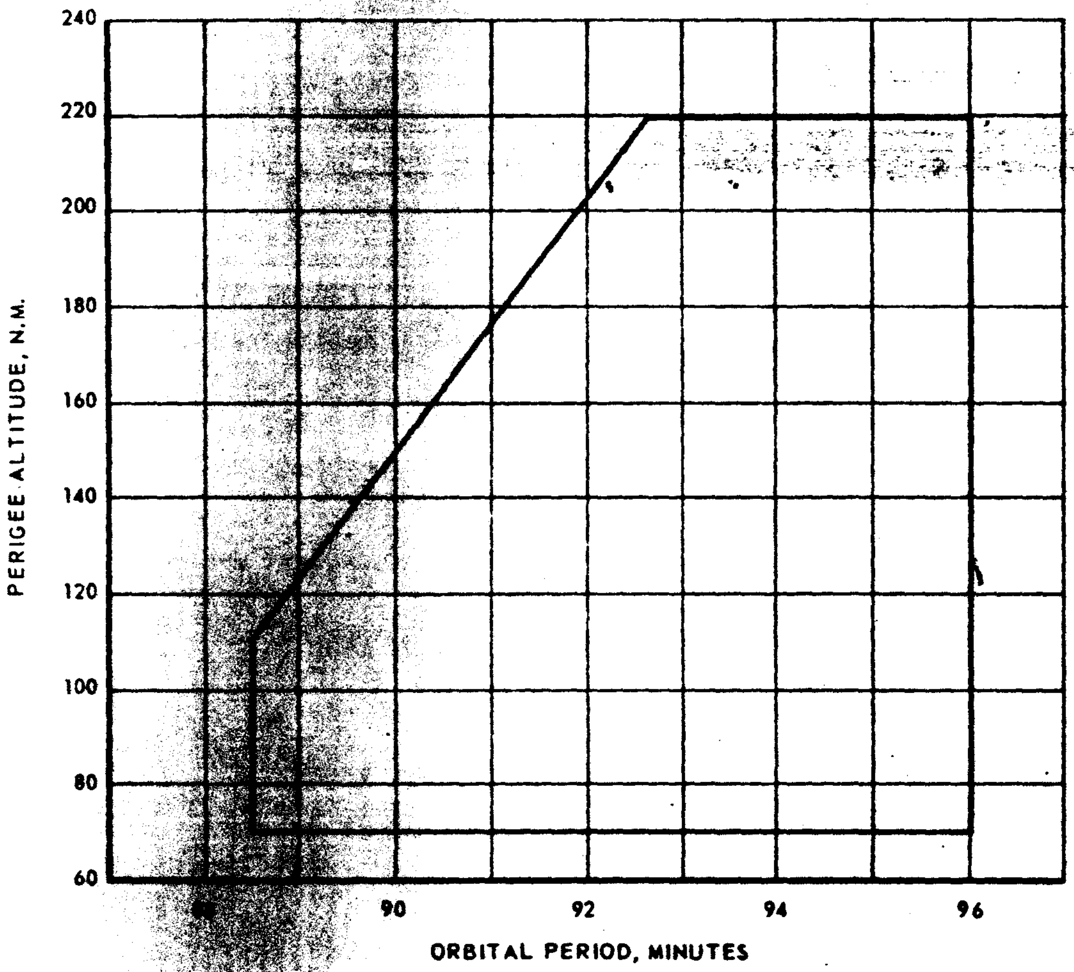

Figure 22. Primary Reentry System Operational Capability

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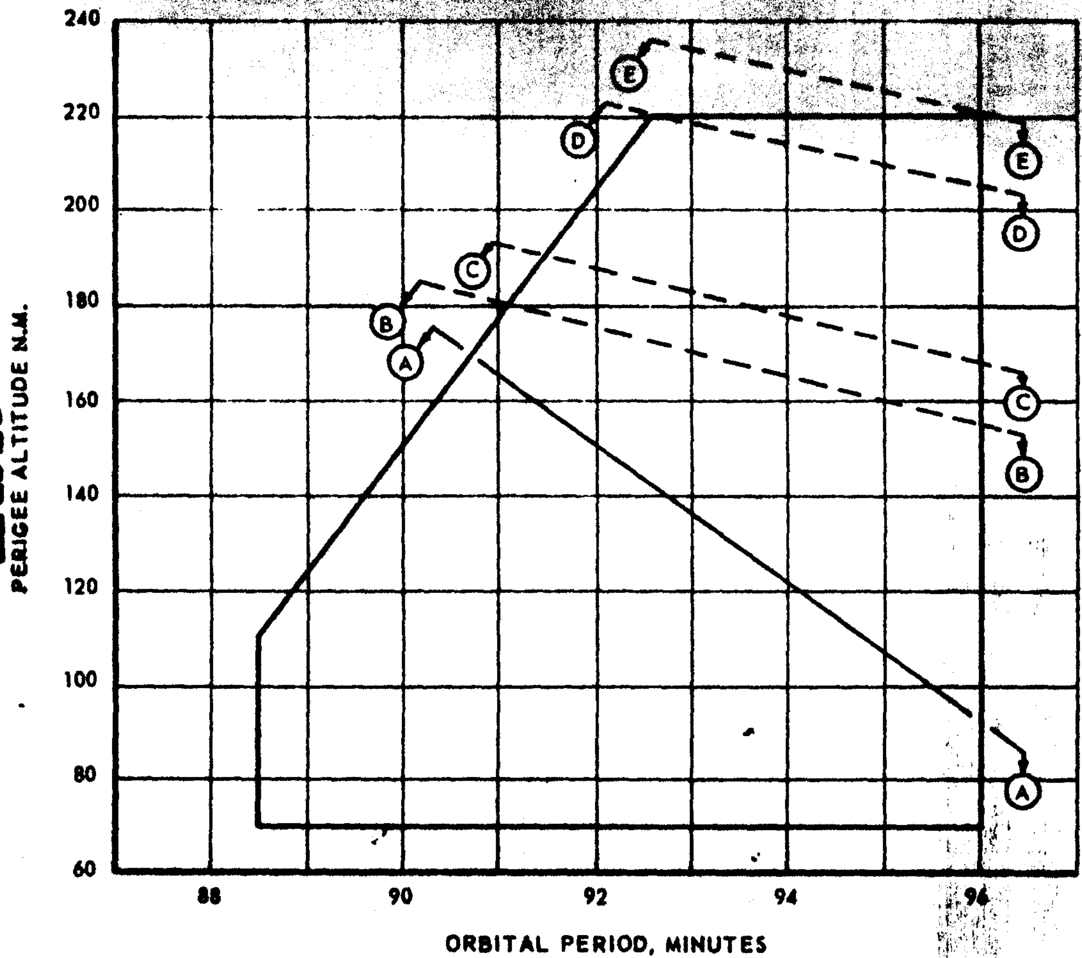
REENTRY WEIGHT = 230 LBS
RETRO VELOCITY = 1150 FPS
REENTRY - NORTH TO SOUTH



ZONE	λIMP	i, DEG.	λ_p	
A	24°N	60° - 81.8°	30°N TO MAX.	**
B	24°N	60° - 81.8°	30°N	*
C	10°N	60° - 81.8°	30°N	*
D	0°	60°	30°N	*
	3°S	81.8°	30°N	*
E	10°S	81.8°	30°N	*
	12°S	60°	30°N	*

* RECOVERY GENERALLY NOT POSSIBLE AT $\lambda_p > 30°N$ EXCEPT AT $e \doteq 0$

** MAX. = f (i) $\doteq$ i °N

λIMP = IMPACT LATITUDE

i = ORBITAL INCLINATION

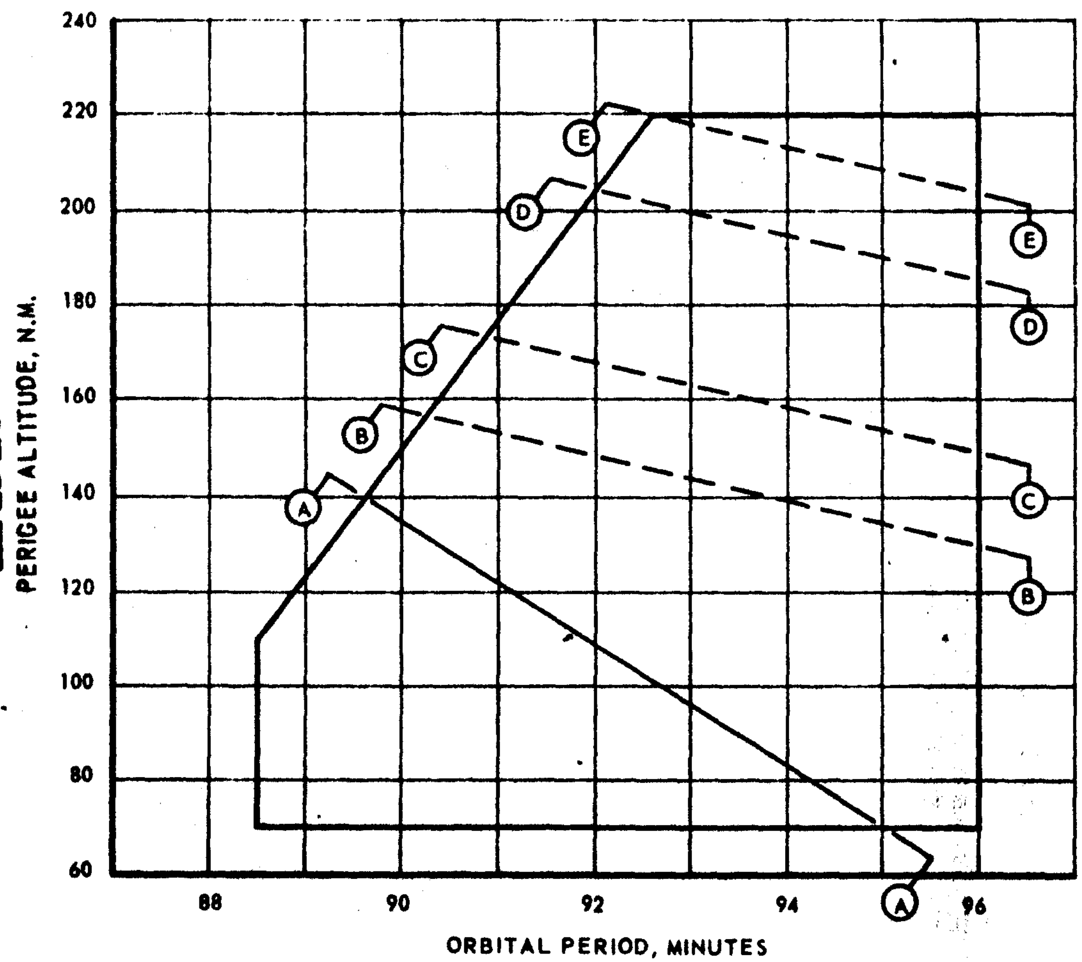
λ_p = LATITUDE OF PERIGEE AT $90° < \bar{\omega}_p < 180°$

$\bar{\omega}_p$ = ARGUMENT OF PERIGEE

— — ~ DENOTES ESTIMATED LIMITS

Figure 23. Backup Reentry System Operational Capability

REENTRY WEIGHT = 308 LBS
RETRO VELOCITY = 909 FPS
REENTRY - NORTH TO SOUTH



ZONE	λ_{IMP}	i	λ_p
A	24°N	60° - 81.8°	30°N TO MAX. **
B	24°N	60° - 81.8°	30°N *
C	10°N	60° - 81.8°	30°N *
D	0° 3°S	60° 81.8°	30°N * 30°N *
E	10°S 12°S	81.8° 60°	30°N * 30°N *

*RECOVERY GENERALLY NOT POSSIBLE AT $\lambda_p > 30°$N EXCEPT AT • ± •

**MAX. = f(i) ± i °N

i = ORBITAL INCLINATION

λ_p = LATITUDE OF PERIGEE AT $90° < \bar{\omega}_p < 180°$

$\bar{\omega}_p$ = ARGUMENT OF PERIGEE

λ_{IMP} = IMPACT LATITUDE

— — — DENOTES ESTIMATED LIMITS

Figure 24. Backup Reentry System Operational Capability

3.1.12.2.2 Reentry Limitations.

3.1.12.2.2.1 Thermal-Structural Capability.

3.1.12.2.2.1.1 Primary Recovery System. The existence of operational capability to successfully initiate a reentry trajectory from orbit (Figure 22) shall not guarantee that the reentry vehicle will survive the associated thermal-structural environment. In fact, the operational capability of the system shall be significantly reduced when thermal-structural limitations are imposed. The system capability limitations corresponding to a high probability of survival were obtained from a limited thermal-structural analysis effort. The results of the analysis are presented in Figures 25, 26, 27, and 28. The reduction in the system capability is obvious when Figures 25, 26, 27 and 28 are compared with Figure 22. The data indicate that the system capability is very sensitive to reentry vehicle weight and to the location of perigee, but rather insensitive to orbit inclination.

3.1.12.2.2.1.2 Backup System. The operational capability of the backup system was discussed in section 3.1.12.2.1.2 and the data presented in Figures 23 and 24. These data were presented without consideration of the structural capability of the reentry vehicle to survive the resulting environment as characterized by extreme temperature histories and high structural loads.

Consequently, the back-up system capabilities (Figures 23 and 24) were evaluated employing a criteria of a high probability of reentry survival. The results of this evaluation are presented in Figure 29 for the 230 pound reentry vehicle. Comparison of Figures 23 and 29 dramatically indicates the drastic reduction in system capability, especially the high sensitivity to perigee location.

With perigee located at its most northerly latitude ($i = 60^{o}$, 81.8^{o}) the 208 pound reentry vehicle does not exhibit an ability to structurally survive the reentry environment for the range of parameters of Figure 24. However, it is intuitively felt that some capability does exist for orbits having perigees located at 30 degrees north latitude or less, but these cases were not evaluated.

3.1.12.2.2.2 Terminal Deceleration.

3.1.12.2.2.2.1 Parachute Recovery System. The parachute recovery system shall be defined as a two parachute, three stage system. The system shall consist of a 5.4 diameter FIST ribbon deceleration parachute and a 29.6 foot diameter, 25 degree conical, ring slot main parachute. The system shall include all associated GFE hardware and pyrotechnic devices.

REENTRY WEIGHT = 230 LBS
RETRO VELOCITY = 1150 FPS
REENTRY = NORTH TO SOUTH

i = ORBITAL INCLINATION
λ_p = LATITUDE OF PERIGEE AT $90° < \bar{\omega}_p < 180°$
$\bar{\omega}_p$ = ARGUMENT OF PERIGEE

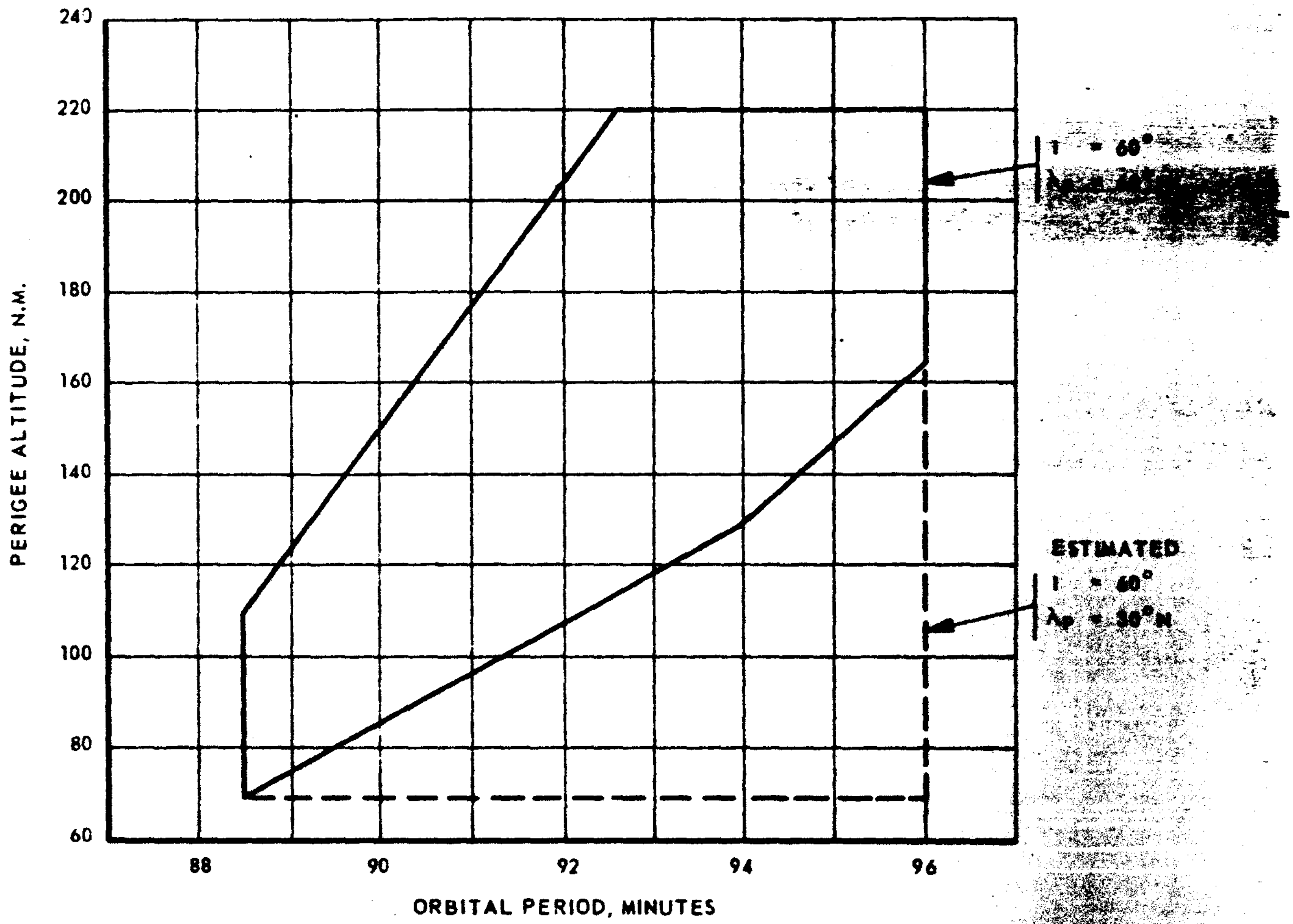


Figure 25. Primary Reentry System Thermal-Structural Capability

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REENTRY VEHICLE WEIGHT = 230 LBS
RETRO VELOCITY = 1150 FPS
REENTRY= NORTH TO SOUTH

i = ORBITAL INCLINATION
λ_p = LATITUDE OF PERIGEE AT $90^\circ < \omega_p < 180^\circ$
ω_p ARGUMENT OF PERIGEE

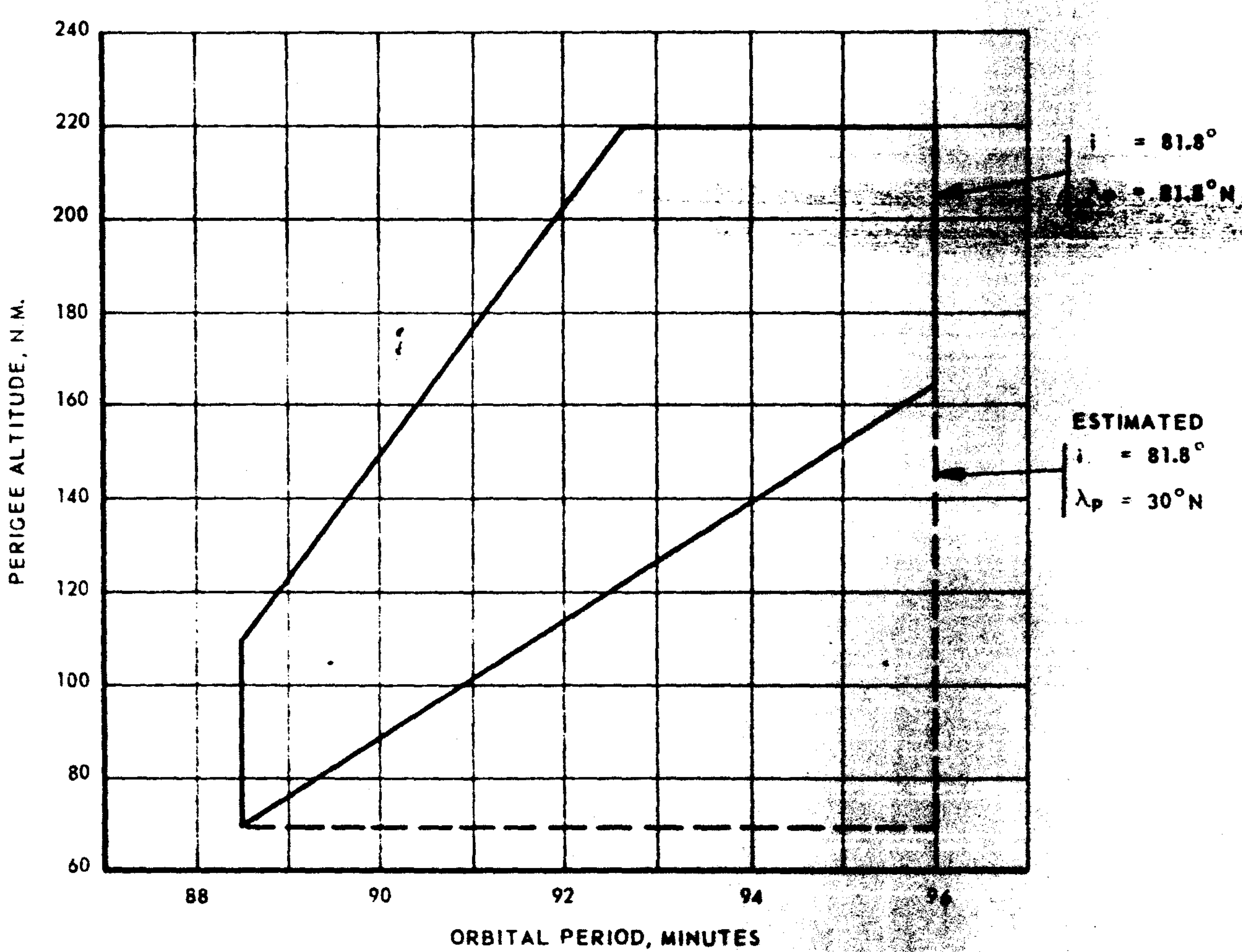

Figure 26. Primary Reentry System Thermal-Structural Capability

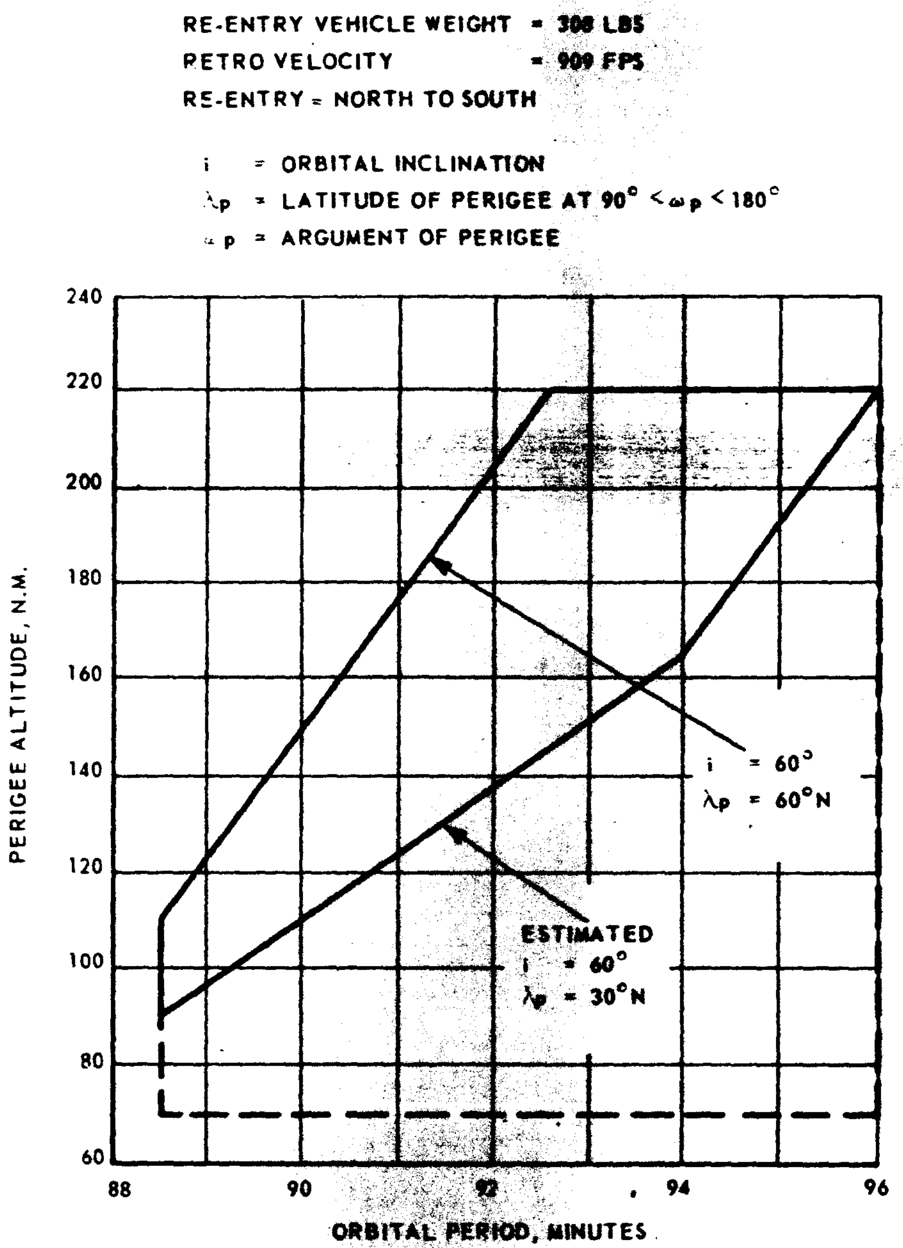

Figure 27. Primary Reentry System Thermal-Structural Capability

RE-ENTRY VEHICLE WEIGHT = 308 LBS
RETRO VELOCITY = 909 FPS
RE-ENTRY - NORTH TO SOUTH

i = ORBITAL PERIOD
λ_p = LATITUDE OF PERIGEE AT $90° < \omega_p < 180°$
ω_p = ARGUMENT OF PERIGEE

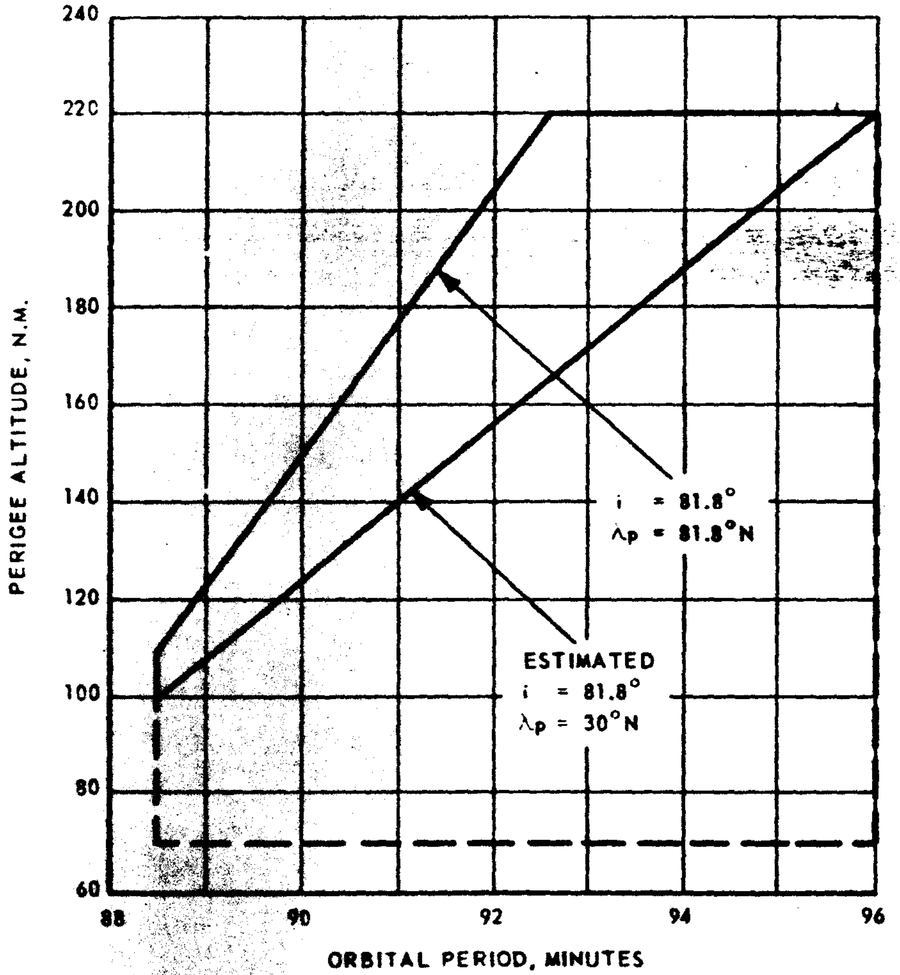

Figure 28. Primary Reentry System Thermal-Structural Capability

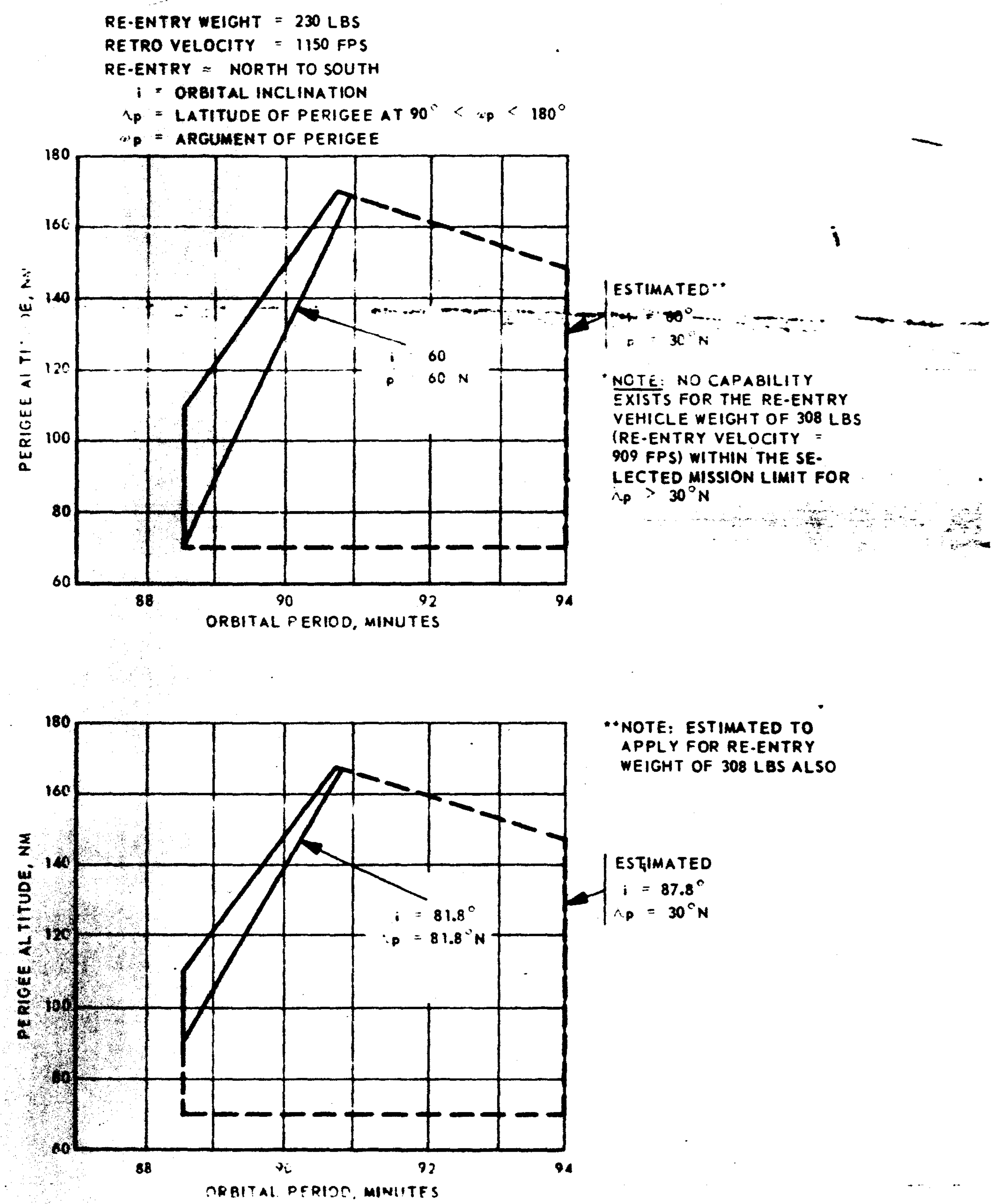

Figure 29. Backup Reentry System Thermal-Structural Capability

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The MK5B parachute recovery system shall be capable of effecting the necessary deceleration and stabilization of a reentering vehicle. The suspended weight of the vehicle shall range from 90 pounds to 180 pounds excluding the parachute system. The following deployment conditions represent the two extremes the system should experience.

a. 90 Pounds Suspended Weight

(1) Deceleration Parachute

Maximum deployment altitude	-	68,011 ft MSL
Maximum deployment velocity	-	916 ft/sec
Maximum deployment dynamic pressure	-	68.1 lbs/ft^2

(2) Main Parachute

Maximum deployment altitude	-	62,250 ft MSL
Maximum deployment velocity	-	345 ft/sec
Maximum deployment dynamic pressure	-	12.5 lbs/ft^2

b. 180 Pounds Suspended Weight

(1) Deceleration Parachute

Maximum deployment altitude	-	55,739 ft MSL
Maximum deployment velocity	-	900.0 ft/sec
Maximum deployment dynamic pressure	-	124.0 lbs/ft^2

(2) Main Parachute

Maximum deployment altitude	-	48,000 ft MSL
Maximum deployment velocity	-	300 ft/sec
Maximum deployment dynamic pressure	-	19.0 lbs/ft^2

At time of deployment the vehicle may oscillate plus or minus 20 degrees and may be rotating at a rate of 15 rpm.

The desired rate of descent at 10,000 ft MSL is 25 to 26 feet per second under standard atmospheric conditions.

The MK5B main canopy shall be designed for aerial recovery, with 90 to 180 pounds suspended weight, by a JC-130B aircraft. Maximum aerial recovery altitude shall be 15,000 ft. Maximum aircraft speed shall be 135 KIAS.

3.1.12.2.2.2.2 Recovery Aids. Three recovery aids shall be provided. Two of these shall be RF types and one, a visual type. The primary RF aid shall be a pulse beacon and the backup RF aid shall be a TM system. The visual recovery aid shall be a flashing light.

3.1.12.2.2.2.2.1 Recovery Beacon. The beacon shall have a stabilized frequency of 235.0 mc. Deviation from this frequency shall not be more than 0.01 percent. The beacon shall have a pulse rate of 1000 cps with a pulse duration of 30 microseconds. The peak power output shall be 7.5 watts minimum. A signature shall be attached to the beacon which modifies the pulse rate so that audio identification can be achieved. The system minimum life shall be 10 hours after water impact.

3.1.12.2.2.2.2.2 Recovery Telemetry System. The recovery telemetry shall consist of a 3 channel FM/FM system. Two channels, IRIG 7 and 9, shall be used to monitor reentry system events. The third channel, IRIG 11, shall read out acceleration along the vehicle longitudinal axis with an accelerometer whose range is minus five g to plus five g. The system shall operate at 228.2 mc, with a minimum power output of 2.0 watts. The maximum frequency deviation shall be plus or minus 0.01 percent. The operating life of the system shall be 40 minutes minimum after separation from the Agena.

3.1.12.2.2.2.2.3 Flashing Light. The flashing light shall have an output of 15 lumen-seconds per flash with a minimum flash rate of 60 per minute. Minimum operating time after water impact shall be 10 hours.

3.1.12.2.2.2.3 Water Flotation Period.

3.1.12.2.2.2.3.1 Sea Conditions. The capsule shall be capable of sustaining water impact under a sea state of 3 with 18 knots surface winds. After water impact the capsule shall float in an upright position and shall not capsize in sea states of 3 or less, as defined by the U.S. Navy Hydrographic Office.

3.1.12.2.2.2.3.2 Flotation Period. A sink system shall scuttle the capsule after a flotation period of 55 hours minimum. Flotation shall not exceed 95 hours maximum.

3.1.12.3 System Accuracy.

3.1.12.3.1 Error Sources. The primary error sources that influence the reentry vehicle impact dispersions can be generalized as follows:

a. Agena attitude and attitude change rates at reentry vehicle separation,

b. Reentry vehicle attitude and attitude change rates during separation, spin-up, and retro-rocket impulse,

c. Reentry vehicle static and dynamic balance characteristics,

d. Retro-rocket impulse tolerance,

e. Uncertainty of orbit parameters at reentry body separation,

f. Event timing errors,

g. Uncertainties in actual ballistic parameter, atmospheric density, and surface winds.

The same general error sources are applicable to both the primary and back-up recovery systems alike, although the discrete values of these error sources differ. A tabulation of the error sources is presented in Tables XV, XVI and XVII.

3.1.12.3.2 <u>Impact Dispersions.</u> An evaluation of the reentry vehicle impact dispersions due to assumed error source deviations has been made for the following assumed set of mission parameters (typical mission):

a.	Perigee altitude	H_p	= 100 N.M.
b.	Latitude of perigee	λ_p	= 40° N
c.	Latitude of injection	λ inj	= 21.6° N
d.	Inclination	i	= 85°
e.	Eccentricity	e	= 0.022
f.	Period	T	= 91.06 minutes
g.	Flight path angle at injection	γ inj	= $+0.362^\circ$

The computed reentry vehicle impact dispersions are itemized in Tables XV, XVI, and XVII, and are summarized below:

TABLE XV   PRIMARY RETRO RECOVERY SYSTEM IMPACT POINT DISPERSIONS
(Generally North to South)

DISPERSION FACTORS	MAGNITUDE OF DEVIATION: RE-ENTRY PLANE	MAGNITUDE OF DEVIATION: NORMAL TO PLANE	UP RANGE (N.M.)	DOWN RANGE (N.M.)	CROSS RANGE (N.M.)
1. Variation of Retro Velocity Direction From Nominal					
a. Agena attitude tolerance at capsule separation	± 3.06 deg.	± 1.44 deg.			
b. Capsule attitude change due to Agena rate at capsule separation	± 1.54 deg.	± 0.05 deg.			
c. Capsule attitude changes due to spring separation of capsule from Agena	± 1.41 deg.	± 0.48 deg.			
d. Capsule attitude changes due to pitch and yaw torques during spin up and initial pitch and yaw rates at spin ignition	± 0.81 deg.	± 0.91 deg.			
e. Capsule attitude changes due to both the pitch and yaw torques during retro-burning and the products of inertia.	± 3.00 deg.	± 3.00 deg.	1.0	(12.0 ; 4.7)	± 5.0
	RSS = ± 4.[illegible] deg.	RSS = ± 3.46 deg.		RSS = 12.9	
2. Variation of Retro-Velocity Magnitude From Nominal					
a. Variations in retro-rocket total impulse (± 3 percent) ($\Delta R / \Delta V_R$ = -2.73)	± 2[illegible] ft/sec		66.0	66.0	
b. Reduction of retro-velocity magnitude due to pitch and yaw rates induced during spin up and pitch and yaw torques and products of inertia present during retro burning.	- [illegible].00			62.1	
				RSS = 95.5	
3. Accuracy of Predicted Orbit Elements at Retro-Ignition					
a. Altitude	± [illegible] ft.		15.2	15.3	
b. Position along orbit patch	± 3 n. m.		3.0	3.0	
c. Position normal to orbit plane		± 1 n. m.			± 1.0
d. Velocity magnitude	± 4 fps.		4.8	4.[illegible]	
e. Flight patch angle of velocity vector (± 1.5 min.)	± 0.025 deg.		21.0	2[illegible].0	
f. Azimuth of velocity vector (±6.0 min.)		± 0.1 deg.			± 3.5
			RSS = 26.[illegible]	RSS = 27.3	
4. Variation in Retro-Ignition Point Due to Timing Instrumentation Errors					
a. H-Timer drift	± 1.0 sec.		4.3	4.3	
b. H-Timer tape punch accuracy	± 1.0 sec.		4.3	4.3	
c. Recovery Timer	± 0.[illegible] sec.		2.1	[illegible].1	
d. Capsule programmer	± [illegible] sec.		3.7	[illegible]	
Orbit Velocity = [illegible] N. M./Sec			RSS = 7.4	RSS = 7.[illegible]	
5. Uncertainty in Ballistic Parameter, W/C_DA	± 5%		5.4	5.9	
6. Uncertainty in Atmospheric Density Model During Reentry	± 10%		10.6	11.[illegible]	
7. Reentry Winds from Orbit Altitude to 50,000 feet			7.0	7.0	± 5.5
ROOT-SUM-SQUARE DISPERSION			72.8	101.5	± 8.3

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TABLE XVI PRIMARY RETRO RECOVERY SYSTEM IMPACT POINT DISPERSIONS
(Re-entry South to North)

DISPERSION FACTORS	MAGNITUDE OF DEVIATION: IN RE-ENTRY PLANE	MAGNITUDE OF DEVIATION: NORMAL TO PLANE	UP RANGE (N.M.)	DOWN RANGE (N.M)	CROSS RANGE (N.M.)
1. Variation of Retro Velocity Direction From Nominal					
a. Agena attitude tolerance at capsule separati[illegible]	± 3.0[illegible] deg.	± 1.44 deg.			
b. Capsule attitude change due to Agena rate at [illegible]le separation	± 1.54 deg.	± 0.05 deg.			
c. Capsule attitude changes due to spring separa[illegible] of capsule from Agena	± 1.41 deg.	± 0.48 deg.			
d. Capsule attitude changes due to pitch and ya[illegible]ues during spin up and initial pitch and yaw rates at spin [illegible]	± 0.81 deg.	± 0.81 deg.			
e. Capsule attitude changes due to both the pit[illegible] yaw torques during retro-burning and the products of inertia	± 3.00 deg.	± 3.00 deg.	95.0	(135.0 ; 2.5)	± 8.0
	RSS = ± 4.9[illegible] deg.	RSS = ± 3.46 deg.		RSS = 135.0	
2. Variation of Retro-Velocity Magnitude From Nominal					
a. Variations in retro-rocket total impulse (± 3[illegible]ent)($\Delta R/\Delta V_R$ = -[illegible])	± [illegible]9.58 ft/sec		86.4	86.4	
b. Reduction of retro-velocity magnitude due to [illegible] and yaw rates induced during spin up and pitch and yaw torques and [illegible]ucts of inertia present during retro burning.	- 31.00			90.5	
				RSS = 1[illegible]5.1	
3. Accuracy of Predicted Orbit Elements at Retro-Igni[illegible]					
a. Altitude	± 3000 ft.		11.[illegible]	11.1	
b. Position along orbit patch	± 3.0 n. m.		3.0	3.0	
c. Positionnormal to orbit plane		± 1 n. m.			± 1.0
d. Velocity magnitude	± 4 fps.		10.5	10.9	
e. Flight patch angle of velocity vector (± 1.5 [illegible])	± 0.0[illegible]5 deg.		19.[illegible]	20.4	
f. Azimuth of velocity vector (± 6.0 min.)		± 0.1 deg.	RSS = 2[illegible]	RSS = [illegible]5.8	± 5.4
4. Variation in Retro-Ignition Point Due to Timing In[illegible]entation Errors					
a. R-Timer drift (Orbit Ve[illegible]y = 4.0[illegible] N.M./Sec)	± 1.0 sec.		4.1	4.1	
b. R-Timer tape punch accuracy	± 1.0 sec.		4.1	4.1	
c. Recovery Timer	± 0.5 sec.		[illegible].0	[illegible].0	
d. Capsule programmer	- 0.5[illegible], + 0.[illegible]9 sec.		3.[illegible]	[illegible].8	
			RSS = 7.[illegible]	RSS = [illegible].7	
5. Uncertainty in Ballistic Parameter, W/C_DA	± 5%		[illegible].9	3.[illegible]	
6. Uncertainty in Atmospheric Density Model During Re[illegible]	± 10%		5.7	[illegible].3	
7. Re-entry Winds from Orbit Altitude to 50,000 feet			7.0	7.0	± 5.5
ROOT-SUM-SQUARE DISPERSION			131.6	1[illegible].3	± 11.2

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TABLE XVII BACKUP RETRO RECOVERY SYSTEM IMPACT POINT DISPERSIONS
(Re-entry North to South)

DISPERSION FACTORS	MAGNITUDE OF DEVIATION IN RE-ENTRY PLANE	MAGNITUDE OF DEVIATION NORMAL TO PLANE	UP RANGE (N. M.)	DOWN RANGE (N. M.)	CROSS-RANGE (N. M.)
1. Variation Of Retro Velocity Direction From Nominal					
a. Agena attitude tolerance at separation due to magnetometer limit cycle	± 6.08 deg.	± 6.08 deg.			
b. Agena attitude tolerance at separation due to uncertainty in magnetic vector.	± 2.00 deg.	± 2.00 deg.			
c. Capsule attitude change due to Agena rates at capsule separation	± 1.54 deg.	± 1.54 deg.			
d. Capsule attitude changes due to spring separation of capsule from Agena	± 1.41 deg.	± 1.41 deg.			
e. Capsule attitude changes due to pitch and yaw torques during spin up and initial pitch and yaw rates at spin ignition	± 0.81 deg.	± 0.81 deg.			
f. Capsule attitude changes due to both the pitch and yaw torques during retro-burning and the products of inertia.	± 3.00 deg.	± 3.00 deg.			
	RSS = ± 7.41 deg.	RSS = ± 7.41 deg.	(35.0 ; 48.0)	(85.0 : 100.0)	(±2.9 ; ±10.5)
			RSS = 59.4	RSS = 131.2	RSS = ± 10.9
2. Variation of Retro-Velocity Magnitude from Nominal					
a. Variations in retro-rocket total impulse (± 3 percent) ($\Delta R / \Delta V_R$ = -2.52)	± 29.5 ft/sec.		74.5	74.5	
b. Reduction of retro-velocity magnitude due to pitch and yaw rates induced during spin up and pitch and yaw torques and products of inertia present during retro burning.	- 31.0			78.1	
				RSS = 107.9	
3. Accuracy of Predicted Orbit Elements at Retro-Ignition					
a. Altitude	± 3000 ft.		17.5	18.1	
b. Position along orbit patch	± [illegible] n. m.		3.0	3.0	
c. Position normal to orbit plane		± [illegible] n. m.			± 1.0
d. Velocity magnitude	± 4 fps.		5.7	5.7	
e. Flight patch angle of velocity vector (± 1.5 min.)	± 0.025 deg.		21.0	26.1	
f. Azimuth of velocity vector (± 6.0 min.)		± 0.1 deg.	RSS = 30.4	RSS = 33.4	±3.7
4. Variation in Retro-Ignition Point Due to Timing Instrumentation Errors					
a. Backup retro system timer } Orbit Velocity = 4.26 N.M./sec	± 6.0 sec.		25.6	25.6	
b. Capsule programmer	-.87, +0.69 sec.		3.7	2.9	
			RSS = 25.9	RSS = 25.8	
5. Uncertainty in Ballistic Parameter, W/C_DA	± 5%		6.2	6.9	
6. Uncertainty in Atmospheric Density Model During Re-entry	± 10%		12.2	13.5	
7. Re-Entry Winds from Orbit Altitude to 50,000 feet			7.0	7.0	± 5.5
	ROOT-SUM-SQUARE DISPERSION		104.4	175.7	±12.8

System	R/E Direction	Up Range N.M.	Down Range N.M.	Cross Range N.M.
Primary	North to South	72.8	101.5	± 8.3
Primary	South to North	131.6	186.3	± 11.2
Back-Up	North to South	104.4	175.7	± 12.8

3.1.13 Telemetry Capabilities: The Agena telemetry shall be comprised of two separate VHF FM systems. These shall consist of several IRIG proportional bandwidth FM subcarriers, as shown in Table XVIII, some of which shall carry commutated data. The basic block diagram is shown in Figure 30.

The primary payload telemetry shall consist of a similar VHF FM link also conforming to IRIG standards. The booster telemetry shall consist of a PCM/FM telemetry link.

3.1.13.1 Link 1. The Link 1 telemeter shall provide a means for transmitting the following:

a. Status and functional quality of the vehicle and payload.

b. Diagnostic data to assist in failure analysis.

c. Environmental data.

3.1.13.1.1 Configuration. The FM/FM telemeter shall provide a minimum transmitter output power of 2 watts (1.5 watts at the antenna terminals). Data shall be carried over continuous subcarrier channels, or commutated subcarrier channels using a 100 percent duty cycle non-return-to-zero format.

The following five tracking stations [redacted] shall be equipped to receive, discriminate, decommutate and store, the telemetry data, and shall have VHF antennas as listed below:

a. [redacted] 60 foot parabolic antenna

b. [redacted] 60 foot parabolic antenna

c. [redacted] 60 foot parabolic antenna

d. [redacted] DISC-ON-ROD antenna

e. [redacted] Tri-Helix antenna

The Tri-Helix antenna has the lowest gain so that the lowest quality receiving capabilities shall exist at [redacted]. Here, the maximum

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TABLE XVIII

PROPORTIONAL BANDWIDTH CHANNELS

		Center Freq Cps	Min. Cps	Max Cps	Freq. MI = 5	Response MI = 2
Standard Channels	[illegible]	[illegible]	[illegible]	430	6.0	15
	[illegible]	560	515	602	8.4	21
	[illegible]	730	675	785	11	28
	4	[illegible]	[illegible]	1032	14	35
	[illegible]	1300	1222	1396	20	50
	[illegible]	[illegible]	1572	1828	25	63
	[illegible]	[illegible]	2127	2473	35	88
	[illegible]	[illegible]	2775	3225	45	113
	[illegible]	[illegible]	3607	4193	59	148
	[illegible]	[illegible]	[illegible]	5905	81	203
	[illegible]	[illegible]	6799	7901	110	275
	[illegible]	[illegible]	[illegible]	11289	160	400
	[illegible]	[illegible]	13412	15588	220	550
	[illegible]	[illegible]	[illegible]	[illegible]	330	875
	[illegible]	[illegible]	[illegible]	[illegible]	450	1130
	[illegible]	[illegible]	[illegible]	43000	600	1500
	[illegible]	[illegible]	48567	56438	790	1980
	[illegible]	[illegible]	64750	75250	1050	2625
[illegible]	A	[illegible]	[illegible]	[illegible]	660	1680
	B	[illegible]	[illegible]	[illegible]	900	2750
	C	[illegible]	[illegible]	[illegible]	1200	3000
	D	[illegible]	[illegible]	[illegible]	1600	3938
	E	[illegible]	[illegible]	[illegible]	1050	2625
	[illegible]	[illegible]	[illegible]	[illegible]	(As used by LMSC)	

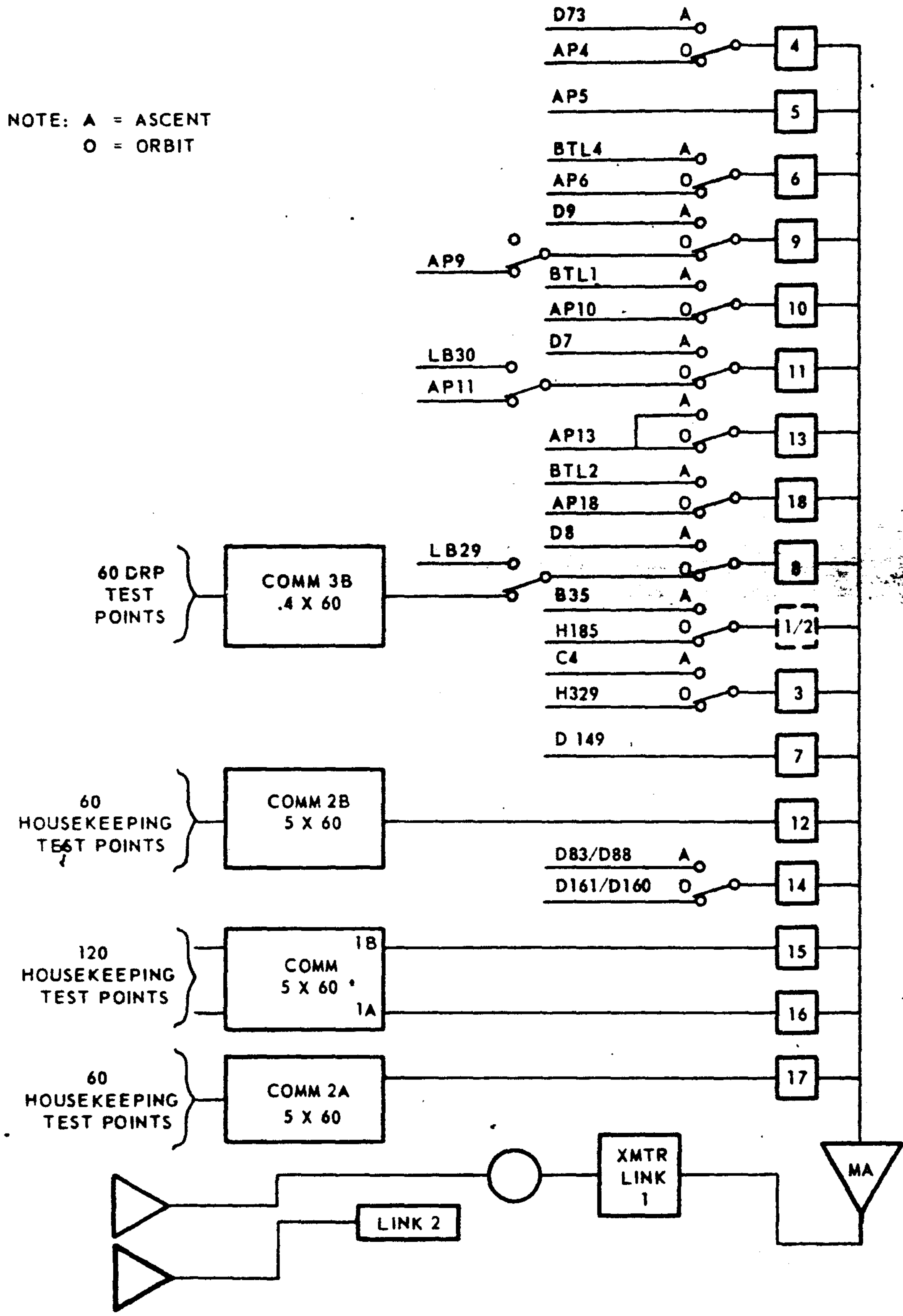


Figure 30. Instrumentation Block Diagram

tracking slant range shall be 1000 nautical miles with no provision for fade margin, giving a predetection signal-to-noise ratio of 10 db, as shown in Table XIX. This is adequate for FM receiver threshold. FM improvement is 7 db (see Table XIX) so that the postdetection signal-to-noise ratio is 17 db. The gains of antennas at all the other tracking stations are higher, providing an improved signal-to-noise ratio.

TABLE XIX

Link Calculations
VHF Telemeter

GAINS

Transmitter Power (2 Watts)		3 dbw
Vehicle Antenna		3 db
Ground Antenna (TRI-HELIX)*		14 db
		20 dbw

LOSS

Vehicle Line Loss		2 db
Space Attenuation (1000 N. M.)		145 db
Propagation Effects		3 db
Polarization Loss		3 db
Ground Antenna Line Loss		2 db
		-155 db
Received Signal		-135 dbw
KTB (300 KC B/W)	= -149.5 dbw	
NF	4.5 db	
Receiver Noise	= -145.0 dbw	-145 dbw
Signal to Noise Ratio		10 db

Which is adequate for FM receiver threshold.

FM Improvement (R):

$$R = \frac{3}{2}\left(\frac{F_D}{fm}\right)^2 \frac{B_{IF}}{fm}$$

F_D = Peak Carrier Deviation

B_{IF} = IF Bandwidth

fm = Highest Modulation Frequency of Interest

* All other stations have higher gain antennas resulting in improved signal to noise.

TABLE XIX (Continued)

$$R = \frac{3}{2}\left(\frac{115KC}{113KC}\right)^2 \frac{300KC}{113KC} = 4.93$$

R = 7.0 db FM Improvement.

$\therefore$ Video So/No = 10 + 7 = 17 db

3.1.13.1.2 Continuous Data Channels. Continuous data channels shall be assigned, as shown in Table XX for a typical case.

TABLE XX

Channel No.	MODE		
	Ascent	Orbit	Life Boat
1-2	Housekeeping	Housekeeping	Housekeeping
3	Housekeeping	Housekeeping	Housekeeping
4	Housekeeping	P/L	P/L
5	P/L	P/L	P/L
6	Housekeeping	P/L	P/L
7	Housekeeping	Housekeeping	Housekeeping
8	Housekeeping	P/L Comm.	Housekeeping
9	Housekeeping	P/L	P/L
10	Housekeeping	P/L	P/L
11	Housekeeping	P/L	Housekeeping
12	Housekeeping Comm.	Housekeeping Comm.	Housekeeping Comm.
13	P/L	P/L	P/L
14	Housekeeping	Housekeeping	Housekeeping
15	Housekeeping Comm.	Housekeeping Comm.	Housekeeping Comm.
16	Housekeeping Comm.	Housekeeping Comm.	Housekeeping Comm.
17	Housekeeping Comm.	Housekeeping Comm.	Housekeeping Comm.
18	Housekeeping	P/L	P/L

NOTES: Housekeeping Comm. = Housekeeping Commutated Data is included here in order to complete the table.

P/L = Payload Data

Comm. = Commutated Data

3.1.13.1.3 Commutated Channels. The commutator assignments as specified in Figure 30 shall be as shown in Table XXI.

TABLE XXI

Mode	Commutator	Ring	Channel Assignment	Sampling Rate or Frame Rate
Orbit	3	B	8*	0.4 rps (e.g.)
Ascent & Orbit	2	B	12	5 rps
Ascent & Orbit	2	A	17	5 rps (e.g.)
Ascent & Orbit	1	B	15	5 rps
Ascent & Orbit	1	A	16	5 rps

3.1.13.1.3.1 Housekeeping Commutated Data Points. Each ring of each commutator shall provide 60 points. Normally, 3 points shall be used for calibration and 3 points shall be used for synchronization, leaving 54 points for transmitting housekeeping data. A total of 4 rings shall be provided by Commutators 1 and 2 giving 216 usable commutator points. Both commutators shall operate at 5 rps, thereby providing a minimum of 5 samples per second per point.

3.1.13.1.3.2 Secondary Payload Commutated Data Points. Ring B of Commutator 3 operating at 0.4 rps shall provide a total of 60 commutated points. Normally 3 points shall be required for calibration and 3 points shall be required for synchronization, leaving 54 points for transmitting secondary payload data.

3.1.13.1.4 System Accuracy. A list of telemetry points and related accuracy requirements shall be contained in Instrumentation Schedules for each vehicle.

3.1.13.2 Link 2. The Link 2 telemetry system shall be as specified in section 3.1.16.

3.1.13.3 Primary Payload Link. The primary payload Link shall be as specified in section 3.1.15.

3.1.13.4 Booster Instrumentation System. The pulse duration modulation/frequency modulation/frequency modulation (PDM/FM/FM)

* During orbit, channel 8 can be switched for sending Lifeboat data.

telemetry system (DAC Drawing 1A18680) shall be installed in the booster center body section. The system shall gather and transmit flight performance data to ground receiving stations via radio frequency (RF) carrier link.

3.1.13.4.1 PDM/FM/FM System. The PDM/FM/FM system shall be essentially a combination of two types of modulation techniques, FM/FM and PDM/FM, which utilize a common RF link, as specified in Figure 31.

3.1.13.4.2 FM/FM Section. The FM/FM section of the PDM/FM/FM system shall provide up to a maximum of nine channels of continuous data. It shall consist of 10 voltage-controlled subcarrier oscillators (VCO), a wideband amplifier, and a frequency modulated (FM) transmitter. Each VCO (with the exception of the 70 kilocycle (kc) carrier) shall be modulated by a single transducer or by a series of sequence signals, to produce an FM output. The FM outputs of the VCO's shall be combined and applied to the wideband amplifier. The composite FM output of the wideband amplifier shall modulate the FM transmitter.

3.1.13.4.3 Multicoder(s). The addition of a multicoder (DAC Drawing 7837664) or multicoders, to the FM/FM section shall produce the complete PDM/FM/FM system. Each multicoder shall provide 43 channels of commutated information to a single VCO. Typical telemetry channel assignments are shown in Table XXII.

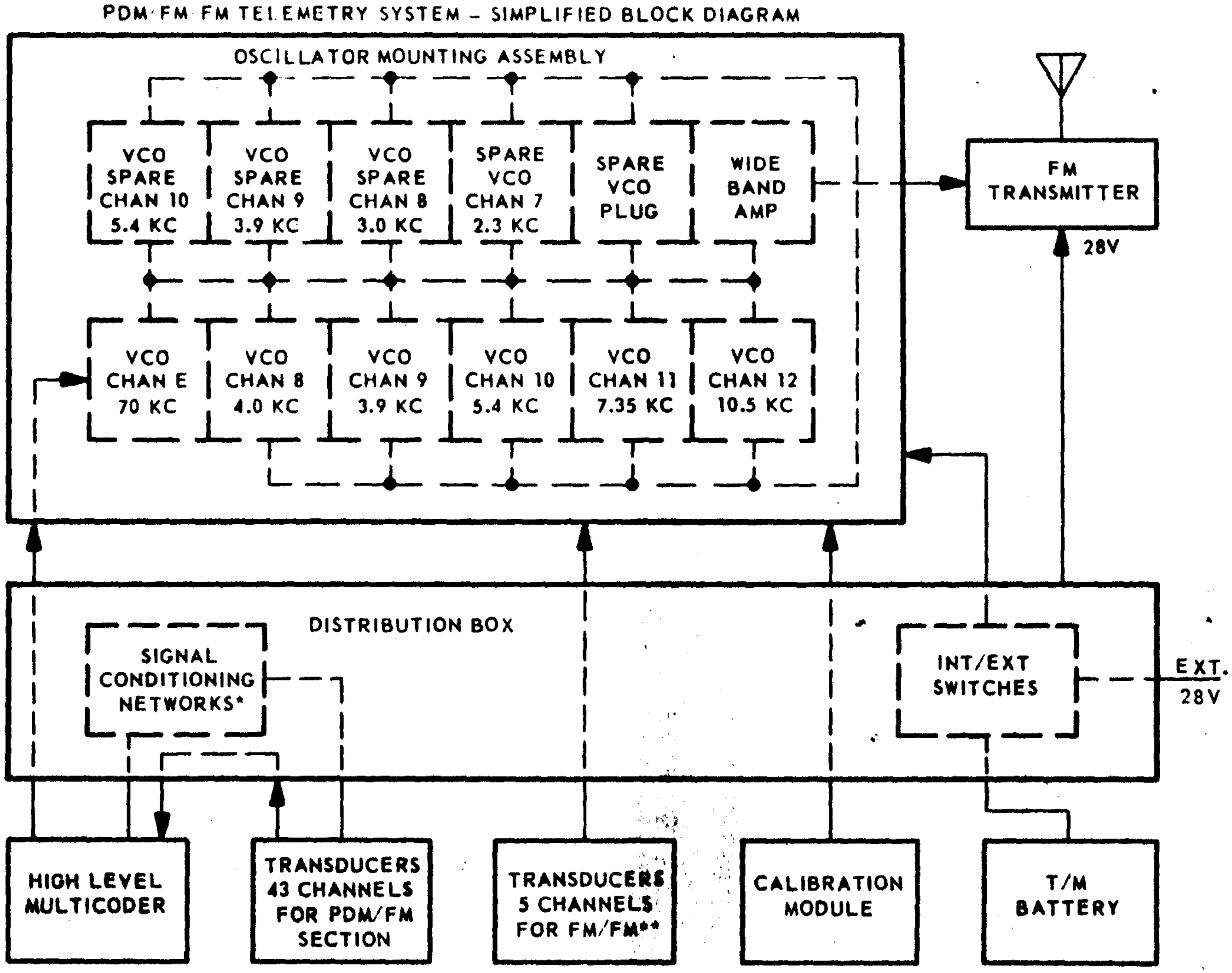
3.1.13.4.4 PDM/FM Section. In the PDM/FM section, transducer data shall be time multiplexed by the multicoder, producing a PDM output which shall be applied to the applicable VCO for conversion into a PDM/FM output. The PDM/FM output of the VCO shall then be combined with the outputs of the FM/FM section VCO's and applied to the wideband amplifier.

3.1.13.4.5 Time Multiplexing of Telemetry Data. The 70-kc channel E VCO shall be used for PDM information so that the standard Inter-Range Instrumentation Group (IRIG) automatic decommutation ground equipment and the playback systems may be adequately utilized. This method of time multiplexing of telemetry data is recommended per IRIG Document 106-60 and is an accepted method of PDM transmission on all missile ranges. System accuracy shall not be degraded due to the non-linearity imposed by narrow-band modulation of the VCO's. The entire system shall be designed to comply with standards established by IRIG and set forth in Standard MIL-STD-442.

3.1.14 Commanding and Tracking.

3.1.14.1 Command Capability.

3.1.14.1.1 Real-Time Command Capability. Both secure and unsecure commands shall be transmitted over the command links. The



NOTES: * INCLUDES 5 VOLT POT. EXCITATION
** SEVEN USED, CAPABILITY NINE CHANNELS, HARDWARE FURNISHED

Figure 31. PDM/FM/FM Telemetry System Simplified Block Diagram

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TABLE XXII

TYPICAL TELEMETRY CHANNEL ASSIGNMENTS
(FM/FM CHANNELS)

CHANNEL	FUNCTION	RANGE OF INTEREST
E 70 KC	PDM/FM/FM	
C [illegible]0 KC	Solid Motor Pc - High Range, Motor	0 - 800 psia
A [illegible]2 KC	Solid Motor Pc - High Range, Motor	0 - 800 psia
13 14.5 KC	Solid Motor Pc - High Range, Motor	0 - 800 psia
12 [illegible] KC	Sequence No. 1	
	A. Main Lox Tank Float Switch	
	B. Main Fuel Tank Float Switch	
	C. Main Engine Cutoff	
	D. Vernier Engine Cutoff	
11 7.35 KC	Main Engine Chamber Pressure	0 - 800 psia
10 [illegible] KC	Sequence No. 2	
	A. Ignition Switch No. 1 Pickup	
	B. Ignition Switch No. 2 Pickup	
	C. Separation Signal, Time Relay	
	D. Separation Signal, Back-up Relay	
[illegible] KC	Vernier Engine No. [illegible] Chamber Pressure	0 - 500 psia
[illegible] KC	Unassigned	
[illegible] KC	Unassigned	

[illegible]

S-band command system shall be capable of handling fifteen unsecure commands (Analog) and two secure commands (Zorro). The Zeke command system shall be capable of handling five secure commands and nominally ten unsecure commands. Of the fifteen S-band unsecure commands, eight commands shall be used to perform the orbital vehicle functions. These functions shall be to control the orbital programmer, to select the recovery pass, to initiate the deactivate sequence, and to select one of two redundant inverters. The remaining seven commands shall be used as primary payload controls. Critical functions shall be backed up in such a manner that reliability will be optimized.

Of the ten permanently assigned Zeke unsecure commands, three shall be used for Lifeboat mode selection. One shall be used to control the beacon transmitter. Six of the Zeke unsecure commands shall be used as backup commands for six selected S-band commands. Two of the backup commands shall provide for H-Timer period adjustments by Zeke in the event of an S-band command link failure. The other backup commands shall be primary payload commands. The utilization of commands is shown in Figure 32.

A system of command interlocks shall be provided to minimize the effects of inadvertent or covert commands.

3.1.14.1.2 Stored Program Command Capability. The orbital programmer shall provide 52 channels of stored program events. Sixteen channels shall be used for controlling vehicle functions. Ten channels shall provide signals to the secondary payload interface for test events. Twenty-six channels shall provide signals to the primary payload.

3.1.14.2 Command System Definition.

3.1.14.2.1 Real-Time Commands. Real-time commands shall be sent to the vehicle by Verlort, Zeke, Radio Guidance and Range Safety command links.

3.1.14.2.1.1 Verlort (S-Band). The S-band link shall use an S-band Verlort or Prelort radar and shall modulate one of its interrogating pulses with the command data. Radar shall interrogate the vehicle transponder by transmitting a succession of pulse groups normally composed of two pulses following one another at some definite time interval. When it is desired to send commands, a third pulse shall be inserted at will between the above-mentioned two pulses, and this third pulse shall be the one that is modulated. Two modulating methods shall be used:

a. Zorro Command System. The center pulse's time-position shall be varied by some fixed amount in relation to the other two pulses in the three-pulse group. Each position shall be given a certain significance which is interpreted in the vehicle decoding devices.

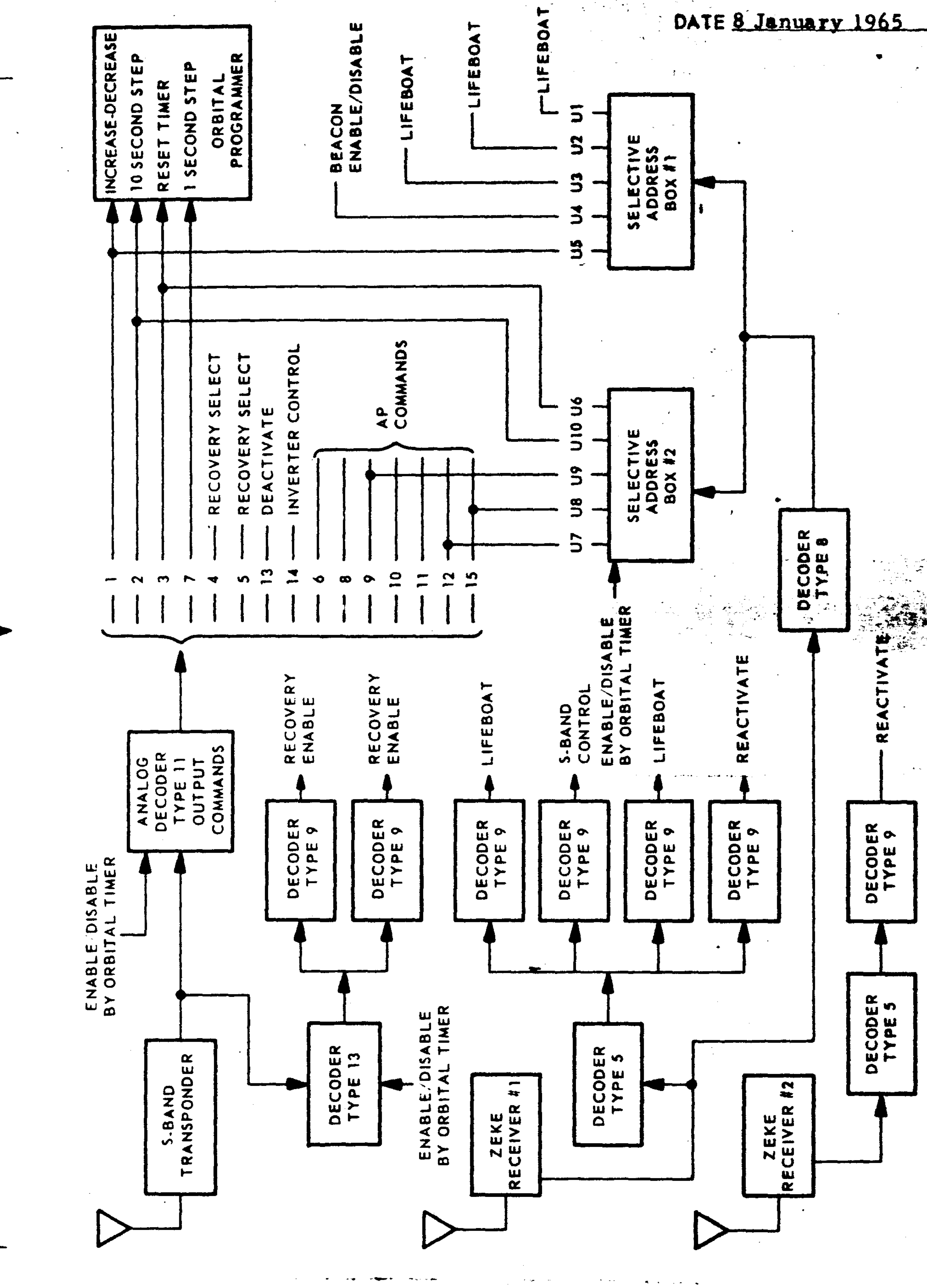


Figure 32. Functional Utilization of Command System

b. Analog Command System. The center pulse's position shall be varied at an audio rate for one second. Six audio tones shall be available, used two at a time, which allows 15 possible combinations.

3.1.14.2.1.2 Zeke (VHF). The VHF link shall consist of a Zeke transmitter and two Zeke receivers where the carrier is amplitude modulated by audio tones.

3.1.14.2.1.3 Radio Guidance Command System (X-Band). The radio guidance command system shall use a continuous X-band radar which shall pulse-position modulate the command spacing between continuous pairs of address pulses.

3.1.14.2.1.4 Command Destruct System (UHF). The system shall be equipped to meet the minimum command destruct requirements of the Pacific Missile Range as specified in AFMTC Regulation 80-7, "Airborne Flight Termination Systems."

3.1.14.2.2 Stored Commands. Stored commands shall be stored in the vehicle prior to launch in the following ways:

a. Fairchild Programmer. Commands shall be stored as perforations in a moving mylar tape. Brushes shall make contact as the programmed perforations pass under them, thus initiating actions in the vehicle.

b. Standard Timer. Commands shall be stored as settings in a mechanical time-interval countdown device which shall initiate actions in the vehicle after programmed time intervals have elapsed.

c. DI/AN Timer. Commands shall be stored as settings in a solid-state, time-interval countdown device which shall initiate actions in the vehicle after programmed time intervals have elapsed.

d. Booster Programmer. Commands shall be stored as perforations on a moving tape and shall initiate actions in the Thor booster in accordance with programmed time intervals.

3.1.14.2.3 Command System Detailed Performance.

3.1.14.2.3.1 Zorro Command System. The Zorro command system shall be a digital command system using an S-Band Verlort or Prelort radar to transmit the commands. The center pulse of the radar's three-pulse group shall be pulse position modulated with the command data. The center pulse shall assume one of three assigned positions in relation to the other two pulses in the group. One position is called a "one", another position is called a "zero", and a third position is called a "reset".

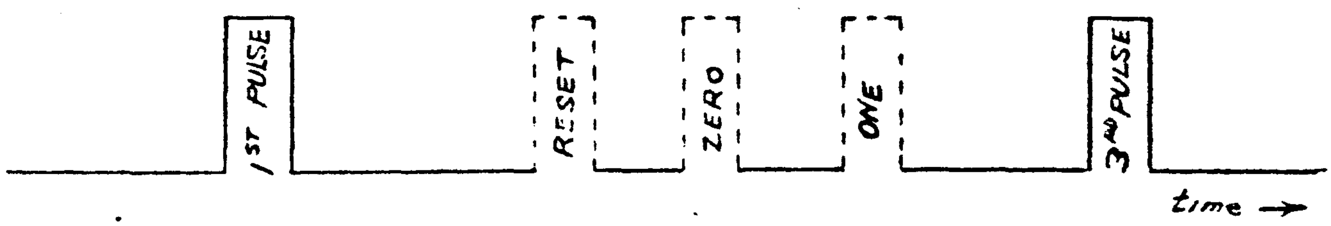


Figure 33.

Figure 33 is a hypothetical illustration of the three possible positions assumed by the center pulse. Only one position at one time shall be possible.

Ones and zeros shall be combined in a digital manner to produce a 40 character command word which also shall include a reset and some blanks. During a blank the center pulse shall not be transmitted.

A Zorro command shall consist of a sequence of 40 equal time intervals, each 40 milliseconds long. The first half of each interval shall carry no information and shall be blank. The last half of each interval from the first through the 36th shall carry command data in a digital form, as shown in Figure 34.

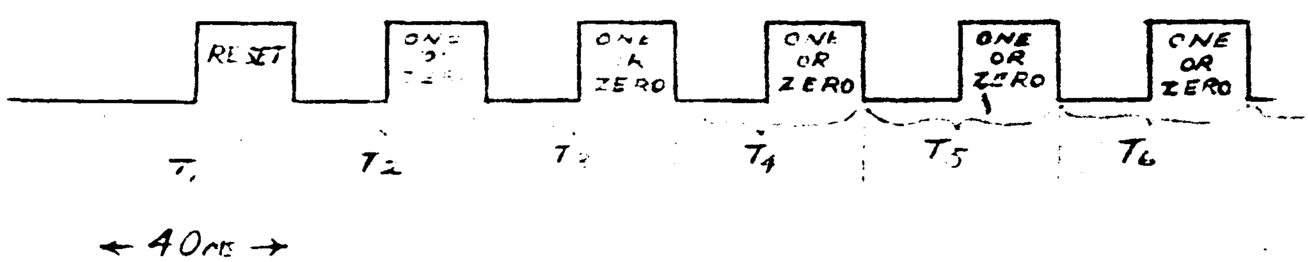


Figure 34.

The first interval shall contain a reset.

Each of the next35 intervals (2 through 36) shall contain either a one or a zero.

Intervals 37 through 40 shall be blank.

Zorro commanding shall be a secure command system. The arrangement of the ones and zeros in each command shall be determined by a classified plug which shall be inserted in the Zeke encoder. Only the one vehicle which contains a duplicate plug shall be capable of decoding that particular command.

Figure 35 is a block diagram of the Zorro System.

Before a Zorro command can be sent by a tracking station, the correct plug shall be inserted in the encoder. When the command button on the command panel is depressed, the encoder shall start to formulate the command word in accordance with the wiring of the plug, starting the command with a reset and putting in the ones and zeros where required. This command word shall be fed into a pulse inserter (RU 654) in the radar which shall translate the resets, ones, and zeros into specific center pulse positions. These center pulse positions shall be transmitted to the vehicle at a 25 bit per second rate. To transmit one command shall require 1.6 seconds.

The radar shall transmit its pulse groups at rates variable between 410 and 630 groups per second. Since each one, zero, or reset has a duration of 20 milliseconds (half of the 40 ms interval previously noted), during that 20 milliseconds, from 8 to 13 pulse groups shall be transmitted. In the event that some of the pulse groups do not reach the vehicle command processing circuits, it shall be sufficient if only 3 pulse groups per bit are received. Figure 36 shows how, with a radar pulse repetition frequency (PRF) of 410 per second, the center pulse is transmitted 8 times during the time it takes for each bit.

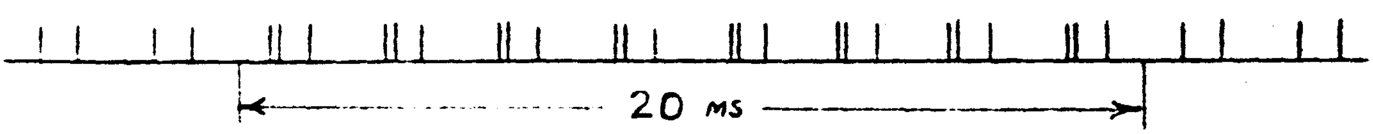


Figure 36

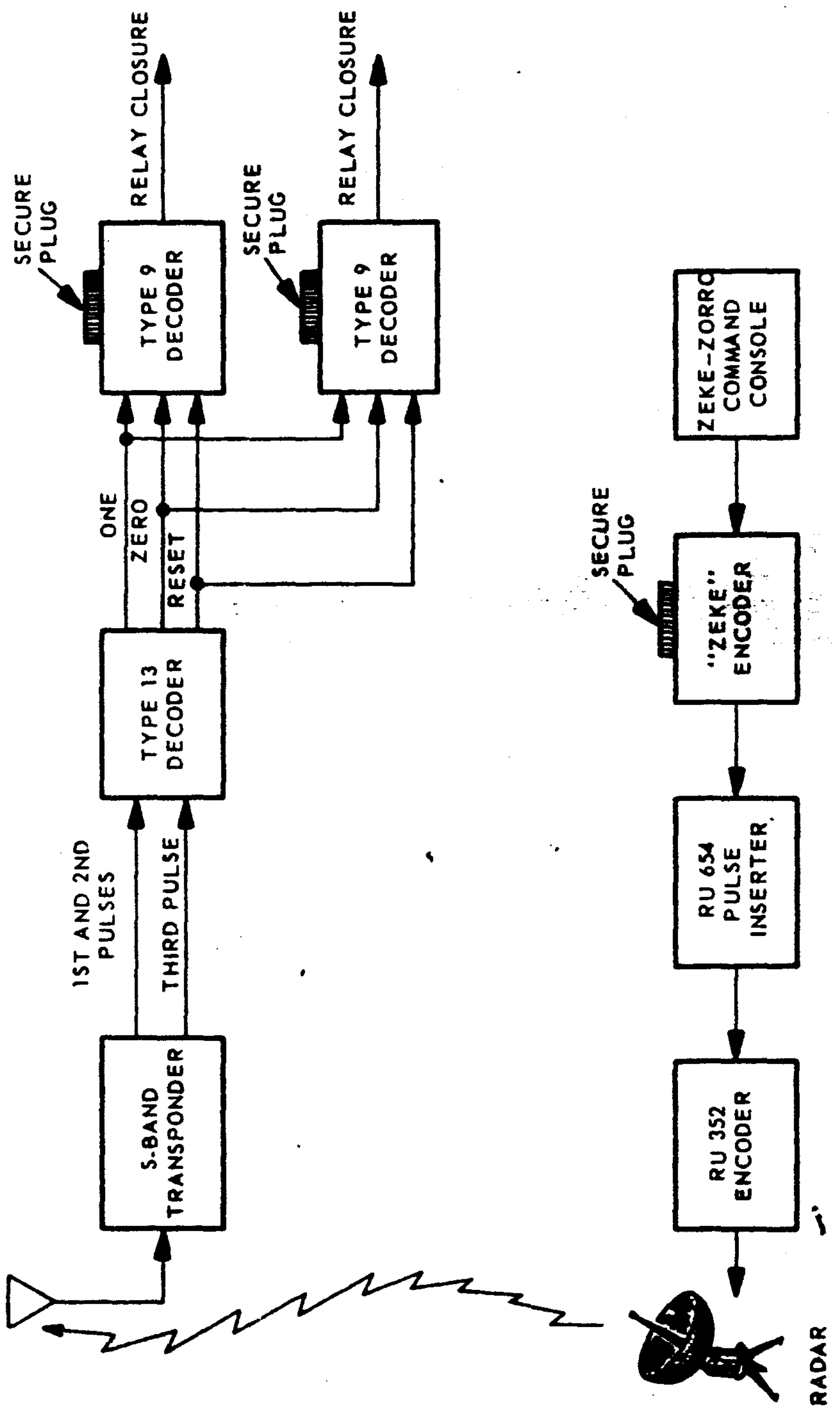


Figure 35. Zorro Command System

In the vehicle the radar pulse groups shall go through the S-band receiver and shall be applied to a Type 13 decoder. In this decoder the one, zero, and reset bits shall be separated and identified on each of three lines as a square pulse. These three lines shall go into one or more Type 9 decoders wired with parallel inputs. Each Type 9 decoder shall decode one command only. A plug shall be inserted in each of these decoders before launch. The plug shall determine the pattern of the command which it will decode. Upon receipt of a valid command a relay shall be actuated to initiate some action in the vehicle.

Vehicle components for the Zorro Command System shall be as follows:

a. S-band Beacon Transponder

b. Type 13 Decoder

c. Type 9 Decoder

3.1.14.2.3.2 Analog Command System. The Analog Command System shall utilize an ~~S-Band Verlort or Prelort~~ radar to transmit the commands. The center pulse's position shall be varied about an initial position at an audio rate for one second. Six audio tones shall be used which are known as tones A, B, C, D, E, and F. For each command two of these tones shall be mixed simultaneously and the center pulse position modulated by this tone pair. Using six different tones in this manner, 15 different commands are possible.

Figure 37 is a block diagram of the Analog Command System.

On the command console shall be fifteen buttons, one for each command. When a button is depressed a logic system shall be caused to send impulses to the radar on two of six wires corresponding to the two tones used to make up this particular command. These impulses, of 1 second duration, shall actuate an encoder (RU 352), causing it to modulate the center pulse with those two tones for one second.

In the vehicle the radar pulse groups shall go through the S-band receiver and shall be applied to a Type 11 decoder where the two tones shall be separated through audio filters and applied to a diode matrix which selects and energizes one of fifteen relays, resulting in some vehicle action.

Vehicle components for the Analog Command System shall be as follows:

a. S-Band Beacon Transponder

b. Type 11 Decoder

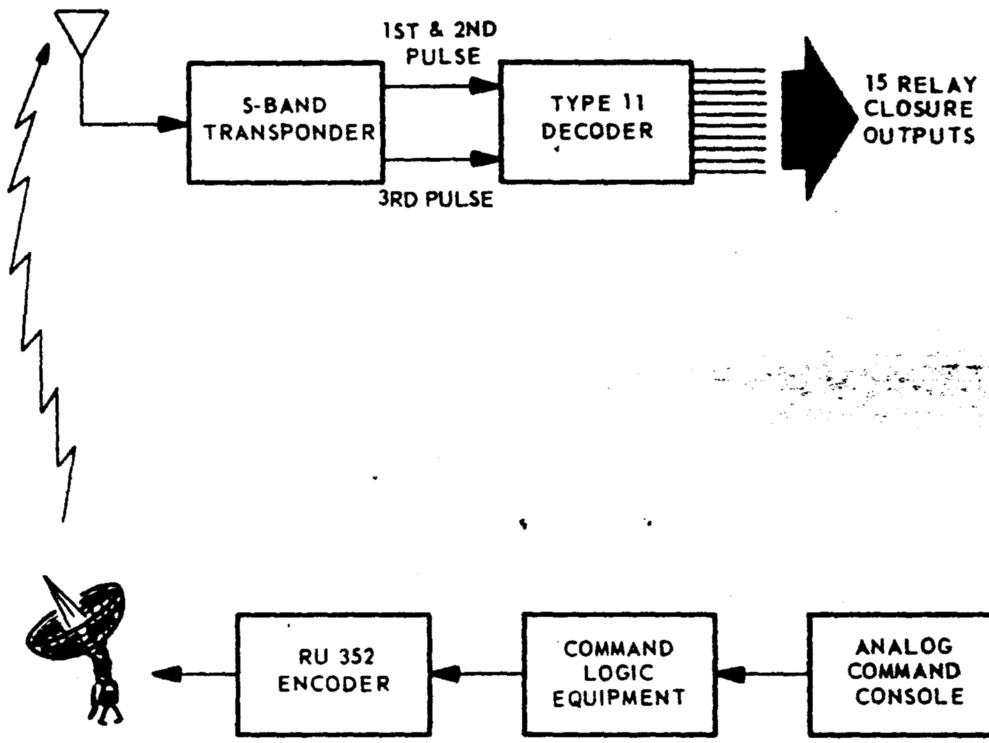


Figure 37. Analog Command System

3.1.14.2.3.3 Zeke Command System. The Zeke Command System shall be a digital system utilizing a VHF transmitter. The carrier shall be amplitude modulated by audio tones to produce the commands. A total of seven tones shall be used. Tones A, B, C, and D shall be for Zeke secure commands. Tones E, F, and G shall be for Zeke Functional Commands. A description of the Secure and of the Functional (unsecure) commands follows:

3.1.14.2.3.3.1 Zeke Secure Commands. Four tones (referred to as A, B, C, and D) shall be used to make up these commands. One tone is a "power" tone, one a "reset", one a "zero", and the fourth, a "one". Each tone shall have a specific audio frequency which never changes.

A Zeke secure command shall consist of a sequence of 40 equal time intervals, each 134 milliseconds long (nominal). During the first half of each interval a "power" tone shall be sent. The last half of each interval from the first through the 40th shall carry command data in a digital form as shown in Figure 38.

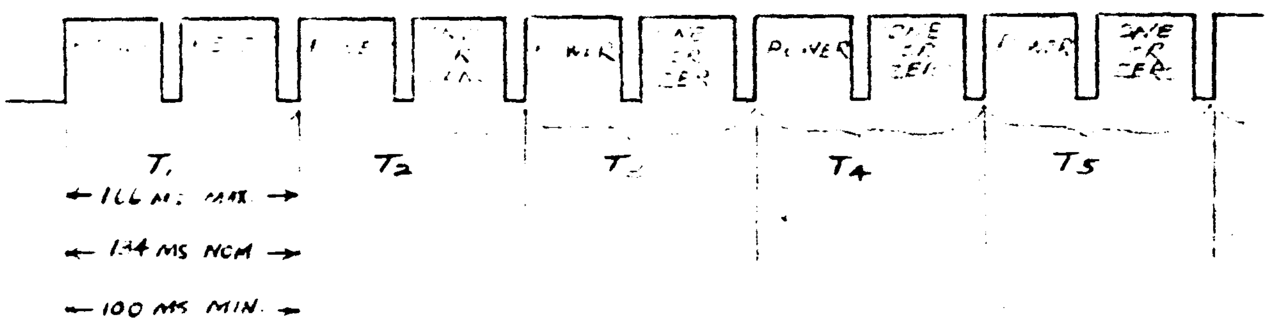


Figure 38.

In the first interval is a reset.

The next 35 intervals (2 through 36) each hold either a one or a zero. Internals 37 through 40 hold power tones.

The arrangement of the ones and zeros in each command shall be determined by a classified plug (similar in kind but not in wiring to the Zorro secure plug) which is inserted in the Zeke encoder. Again, as with Zorro, only the one vehicle which contains a duplicate plug shall decode that particular command.

Figure 39 is a block diagram of the Zeke Secure and Functional Systems.

Before a Zeke Secure command can be sent by a tracking station, the correct plug shall be inserted in the encoder. When the command button on the command panel is depressed, the encoder shall start to formulate the command word in accordance with the wiring of the plug. The output of the encoder, which consists of different voltage levels, shall be fed into a "tone coder" which shall convert the command data into the proper tones which then shall amplitude modulate a VHF carrier.

In the vehicle the VHF signal shall be demodulated by the Zeke receiver which then shall feed the tones to a Type 8 decoder. . This decoder shall have no active function during Zeke Secure commanding, and merely shall pass on the tones to a Type 5 decoder which shall separate them and output ones, zeros, and resets as square pulses on three lines feeding into one or more Type 9 decoders. If the command format matches the secure plug in one of these Type 9's it, shall generate a relay closure to cause some vehicle action. The power tones shall cause the power to be turned on in the vehicle decoders. Each power tone shall actuate a certain relay in the Type 5 decoder which provides the power for 125 milliseconds to the Type 9 decoder as well as to a portion of the Type 5 decoder circuitry. After 125 milliseconds the power relay shall drop out, hence power tones must be repeated throughout the command.

3.1.14.2.3.3.2 Zeke Functional Commands. Zeke functional commands shall not be secure and shall have a different format than Zeke secure commands. Different tones shall be used than for Zeke secure commands. Three tones, designated E, F, and G, shall be arranged as pairs for each functional command, allowing six possibilities, since the order in which they are sent shall be pertinent (e.g. EF, FE, EG, GE, FG, GF).

The command formal shall consist of five consecutive one-second intervals as shown in Figure 40.

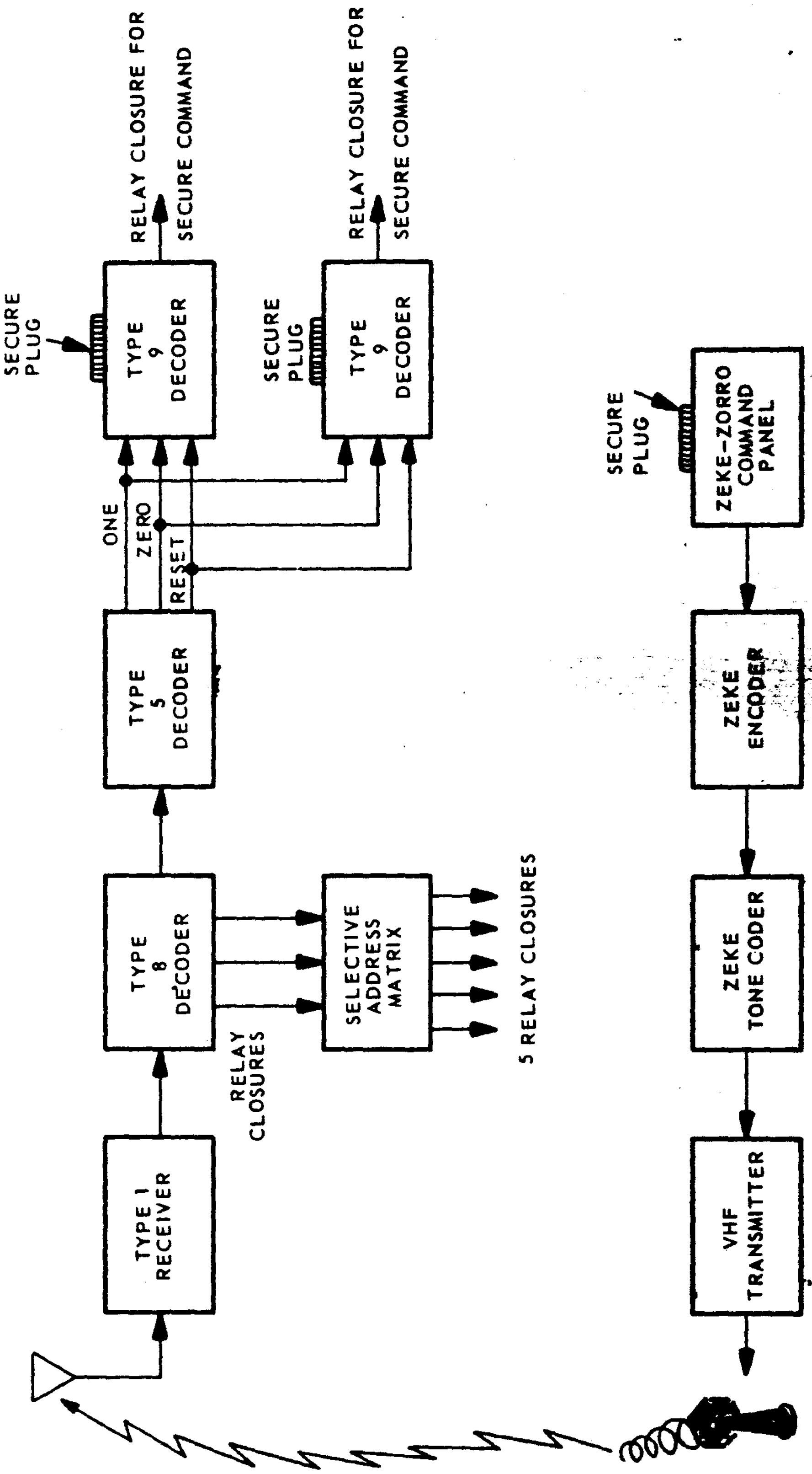


Figure 39. Zeke Secure and Functional System

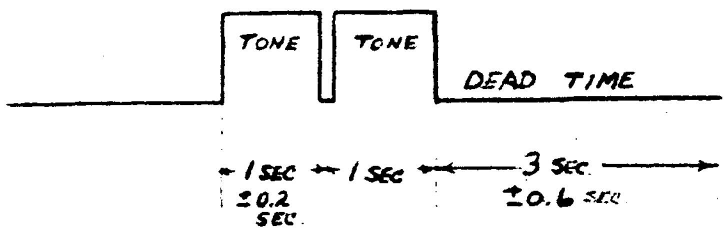


Figure 40

The two tones occupy the first two intervals. The next three intervals are blank. This blank period is to assure that no tones are emitted until certain delayed relay closures have completed their cycle in the Type 8 decoder.

The function of the ground equipment when sending a Zeke functional command shall be essentially the same as for a Zeke secure command, except a secure plug shall not be required in the Zeke encoder. As the command button is depressed on the Zeke-Zorro command panel, the Zeke encoder shall format the command and the tone coder shall send the proper tones to the transmitter modulator. In the vehicle the Zeke receiver shall demodulate the VHF signal and shall feed the tones to a Type 8 decoder and a matrix, where the tones shall be coded and each different pair of tones shall effect a corresponding relay contact closure to control some vehicle function.

Vehicle components for the Zeke Command System shall be as follows:

a. Type 1 Command Receiver

b. Type 8 Decoder

c. Relay Matrix and Selective Address Matrix Boxes

d. Type 5 Decoder

e. Type 9 Decoder

3.1.14.2.3.4 Radio Guidance Command System. The radio guidance command system airborne equipment shall receive pulse coded command signals from the ground radar, shall decode and translate these command signals into steering orders and sequence commands, and shall transmit in each repetition interval a 3 kilowatt pulse to facilitate accurate ground radar tracking. Figure 41 is an illustration of a command interval containing a group of four pulses.

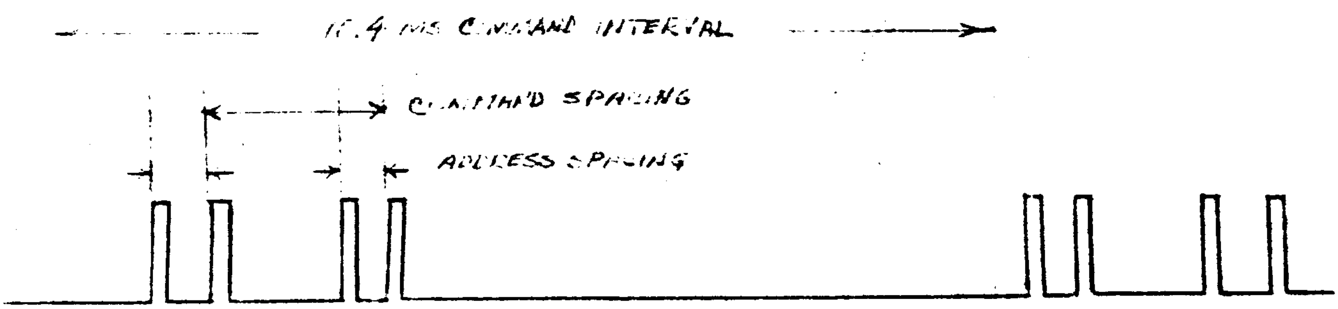


Figure 41

The address spacing must be proper in order for the transponder to operate. The command spacings shall assume any one of eight spacings. Program [redacted] shall use six of the available eight spacings. Each of the four steering commands (pitch up, pitch down, yaw right, and yaw left) shall be transmitted by a single group (4 pulses). A single group shall hold the particular steering command in for a nominal period of 10.4 milliseconds. The command interval shall be approximately 10 milliseconds. When a steering command is repeated continuously, the steering command shall be held in continuously. The ground computer shall be gated to alternate between pitch and yaw steering commands every 200 milliseconds. Consequently, the longest steering command shall be 200 milliseconds in duration. The gyro torquing rates shall be * degrees per second for the Thor booster and 2 degrees per second for the Agena. The maximum correction given in any one axis during a 200 millisecond period shall be * degrees for the Thor booster and 0.4 degrees for the Agena. The contacts of the pitch up and pitch down relays and the yaw left and yaw right relays shall be interlocked to give added assurance that only one command can be given, thus preventing a short circuit of the steering voltage source. Sequence commands used to step the ledex shall be transmitted by sending 30 uninterrupted sequence groups followed by two groups of normal pulses. Functions of the sequence command are specified in section 3.1.4.3.

* This information will be supplied by DAC.

In addition to being sent in conjunction with the sequence command, the normal command shall be sent when none of the other commands is transmitted so that the tracking pulse shall constantly be transmitted to the ground.

3.1.14.2.3.5 Type 8 Fairchild Programmer. The Type 8 Fairchild Programmer shall store commands in the vehicle prior to launch for execution at specific times during flight. The programmer shall consist essentially of four reels of 35 mm mylar tape and a drive mechanism for moving these tapes in unison past a set of brushes at the rate of approximately 6 inches per hour, or 9 inches per orbit. Square holes punched in the tapes prior to launch shall enable each of the brushes to make contact with the grounded drum through the holes as the latter pass under the brushes. These grounding contacts shall be used to initiate vehicle or payload actions, or both.

There shall be 13 brushes for each tape, making a total of 52 brushes. The tapes shall be mylar, 35 mm wide and 0.003 inches (3 mils) thick. Each tape shall be 98 feet long and shall have the capacity to store actions, in the form of holes, for approximately 128 subcycles. A subcycle is defined as that portion of a tape on which the commands are punched for one vehicle orbit. Tape of 1.5 mils thickness may also be used, essentially doubling the programming capacity.

The commands stored in the Type 8 Programmer shall cause vehicle actions or Payload actions or both as the vehicle reaches certain geographical positions. Hence, the tape speed and positioning shall be kept synchronized with the orbital period and overpass plan. The rate at which the tape travels shall be adjustable so that it may be made to match the orbit period. Adjustments shall be accomplished by sending analog commands to speed up or slow down the programmer motor as necessary. The motor shall be a synchronous AC motor, driven by an oscillator at a frequency in the neighborhood of 400 cps. This frequency shall be varied by varying circuit parameters in the oscillator, which is controlled by analog commands. Varying the oscillator frequency shall vary the speed of the motor.

The tapes shall be capable of being advanced or retarded (reset) to a predetermined point within the subcycle. If a predetermined position on the tape does not coincide with the planned latitude during orbit, an Analog command sent to the vehicle when it reaches that latitude shall cause the tape to move rapidly so that the predetermined point is repositioned as planned. The tape then shall continue to move at its former rate.

The controllable rate of travel plus the ability to reset the position of the tapes shall allow the tape programmer to be kept synchronized with the orbit period so that stored commands shall occur at the proper times.

3.1.14.2.3.6 Standard Sequence Timer. The standard sequence timer shall store commands prior to launch for execution during flight. The standard sequence timer is also known as the "D" Timer.

The standard sequence timer shall consist of an assembly containing 24 Veeder-Root counters driven by a constant speed drive motor. The counters shall be set before launch so that after given time intervals, switches shall be actuated to initiate vehicle actions. Thus, 24 events shall be programmable. The Veeder-Root counters shall control from one to five switches each, depending on requirements. The sequence timer shall be started at the desired time by an impulse which engages a clutch and releases a brake on the drive mechanism. The drive motor shall have been started previously.

The maximum time interval possible with the sequence timer shall be 6000 seconds. Should it be desired to program events for longer time intervals, a second sequence timer shall be used and started by the first timer at the end of its run. The accuracy of the timer shall be plus or minus 0.6 seconds.

The sequence timer shall be used by Program [redacted] to time events during the ascent and first orbit phases of vehicle flight, and during the deactivate and reactivate phases.

3.1.14.2.3.7 Recovery Timer. Commands shall also be stored in a recovery timer, known as the Di/An timer. The recovery timer shall be a solid state device based on a two-pulse-per-second magnetic oscillator with a magnetic core counter. At predetermined counts output relays shall close, initiating vehicle actions. A total of 14 such events shall be possible.

A reset function shall be provided to reset the recovery timer counter to the initial count, and to reset the output relays. Normally the reset impulse shall be generated by the recovery timer output at the time of its last event. The recovery timer shall also be capable of being reset externally during ground tests, and by command when in flight.

Two recovery timers shall be used by Program [redacted] during the recovery phase of each flight, one for the normal recovery sequence and the other for the Lifeboat sequence. The normal recovery timer shall be started by a stored command from the Type 8 Programmer. The Lifeboat timer shall be started by a Zeke secure command.

3.1.14.2.3.8 Booster Programmer. The booster programmer shall be capable of applying and terminating controlled sequential commands at predetermined points in the flight. Thirteen program channels shall be available, nine of which shall normally be used for control system functions and the remaining four for backup functions and testing signals. The programmer shall be capable of programming for 250 seconds. The programmer shall be capable of programming events at any preset time with respect to liftoff, within plus or minus 0.1 seconds or plus or minus 1 percent of the program time, whichever is larger. Either clockwise or counterclockwise roll programs may be selected by rewiring within the programmer. Yaw, pitch, and roll rates shall be changed by adjusting respective potentiometers within the programmer.

3.1.15 Primary Payload. This section shall be supplied directly from the LMSC Advanced Projects Payload group to authorized recipients only. **Pages 111 through 124 herein are reserved for the Primary Payload contribution to this system specification.**

Notice of Missing Page(s)

Pages 111 through 124 of the original document were reserved for additional information that was not included.

3.1.16 Secondary Payloads.

3.1.16.1 General. The system shall have a capability for performing research experiments as designated by the AF SSD Project Officer. Research experiments shall be flown on a non-interference basis. That is, the experiments shall have no deleterious effects on the primary mission. LMSC shall design, fabricate, assemble, and systems engineer the selected experiments and the supporting secondary payload equipment to utilize otherwise excess vehicle power and weight capability. LMSC also shall perform checkout tests to determine the payloads flight test readiness. Tests shall be performed to assure that no possible failure mode of the secondary payload will adversely affect the primary mission. The experimental payload modular design shall be adjusted to existing vehicle research payload interface requirements, launch window, vehicle orbit and attitude. The Program [redacted] research payload modular concept shall establish mechanical and electrical vehicle interfaces and a basic telemetry data link interface panel for both research experiment and Program [redacted] support. The accommodations for research payloads are described in the following sections:

3.1.16.2 Mechanical Accommodation. The research payload modular mechanical interface with the Program [redacted] Vehicle shall be as referenced in section 3.1.6.4.4.1. To incorporate selected experiments and support equipment the standard secondary payload aft rack equipment panels shall be as described by LMSC Secondary Payload Drawing 2P10762B. This secondary payload drawing identifies panel mounting locations, loading, and space with respect to each other, to the vehicle aft rack structure, and to the vehicle orientation. The aft rack space available shall be as specified in LMSC 1345097B.

3.1.16.2.1 Aft Rack Equipment Panels. The vehicle aft rack equipment panels shall consist of the four right side panels and two bottom panels as defined in Figure 42, and the left side panels, utilizing the RP-11 panel or the short side panel as defined in Figure 43. The secondary payload panels required shall depend on excess vehicle weight capabilities and experimental equipment weight, size, and view requirements, as well as power availability.

3.1.16.2.2 Aft Rack Loading. The secondary payload panel loading capabilities shall be as defined in Figure 44. All loadings listed are maximum based on total aft rack capabilities. The capabilities also shall depend on distribution and attachment of equipments. A full complement of secondary payload panels (right side, bottom, and [redacted] left side panels) shall have a total payload and support equipment capability of 516 pounds as specified in Figure 44, Item D. When secondary payload weight availability is about 30 pounds, the research payload panel configuration shall consist of the standard panel complement, Figure 45, mounted on the forward bottom panel, a blank aft bottom panel, and diagonal tubes on the aft rack right and left sides, with thermal shields. The diagonal tubes shall maintain the vehicle aft rack structural design.

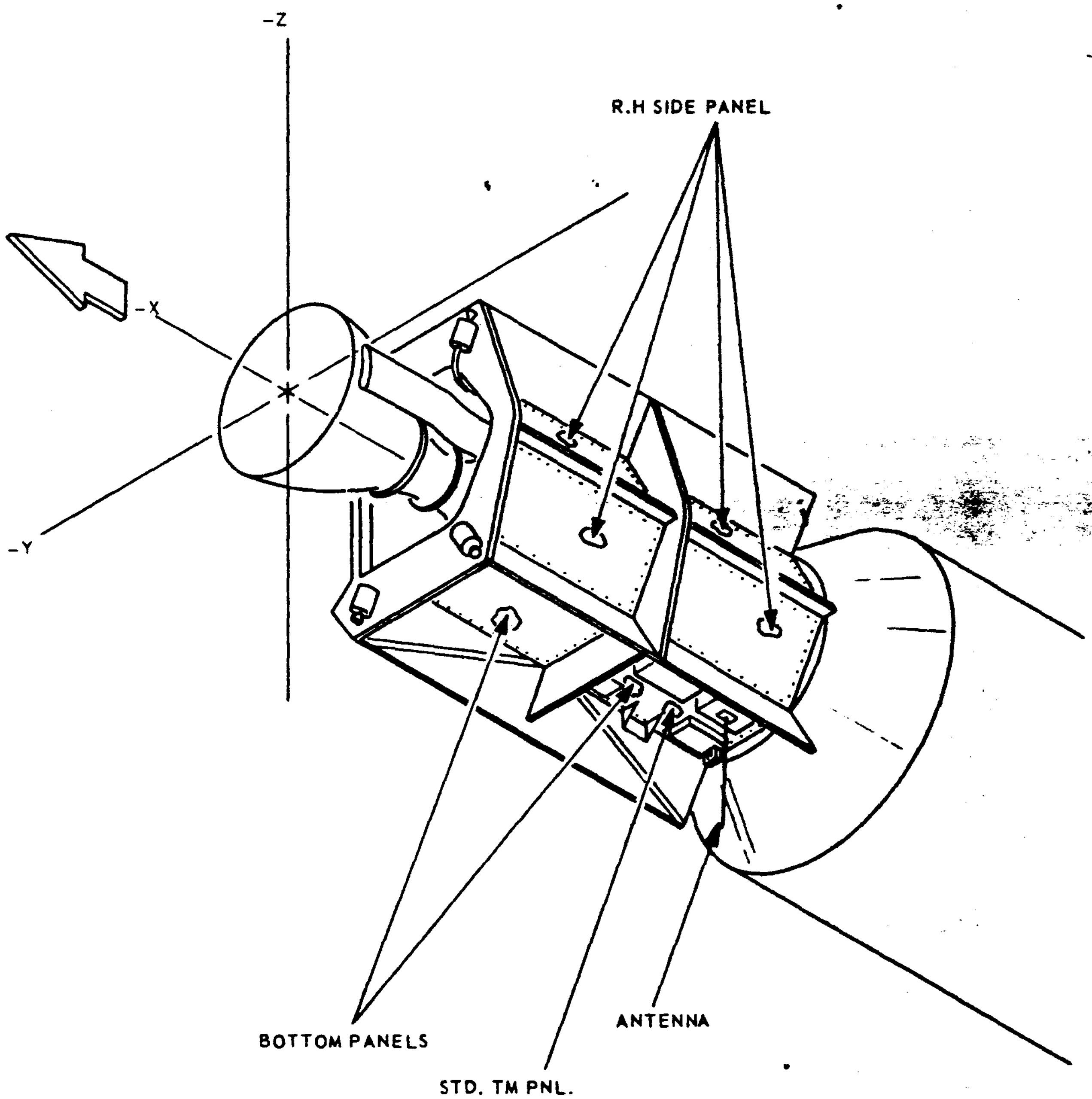


Figure 42. Aft Rack Equipment Panels Right Hand Side and Lower Research Payload

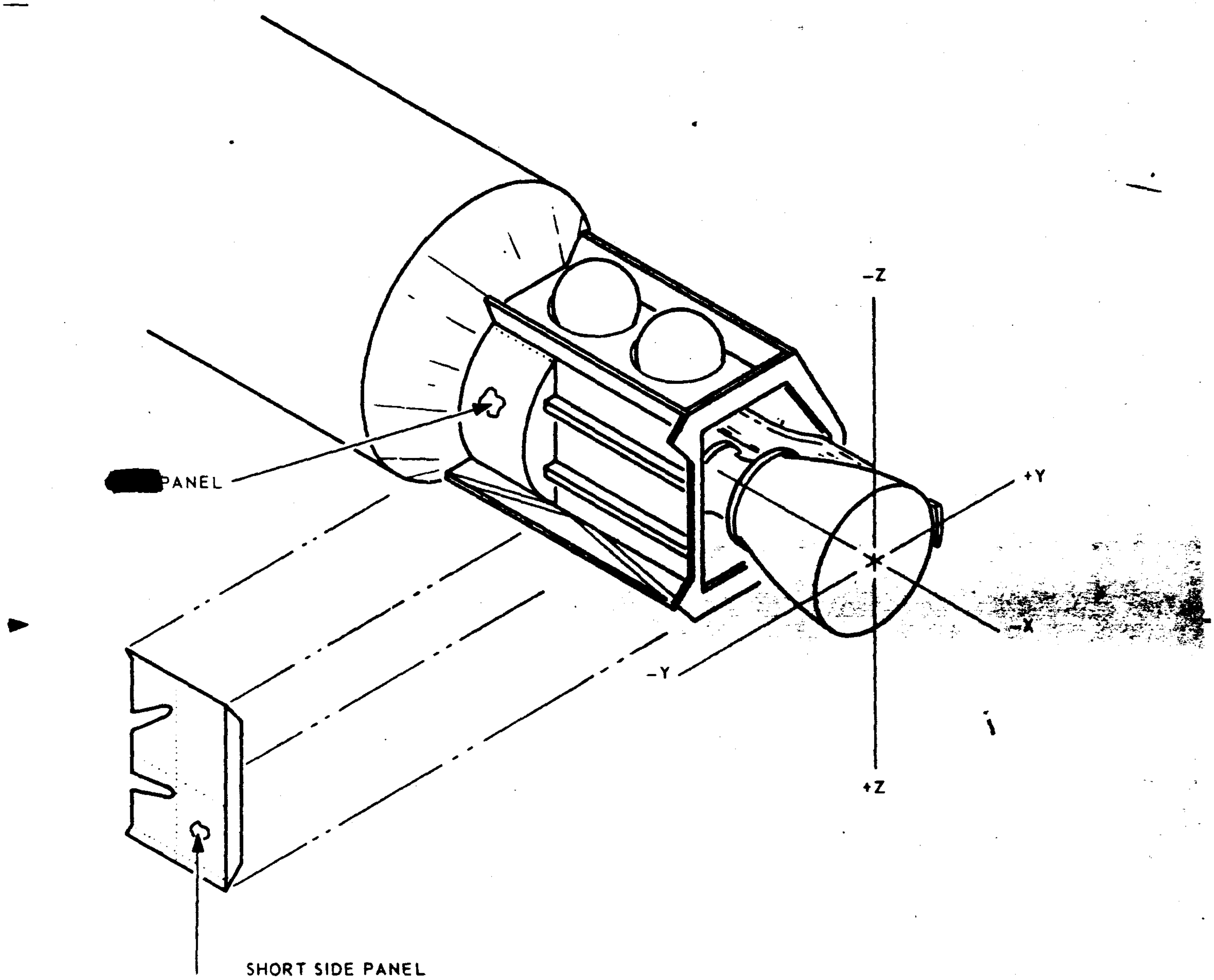
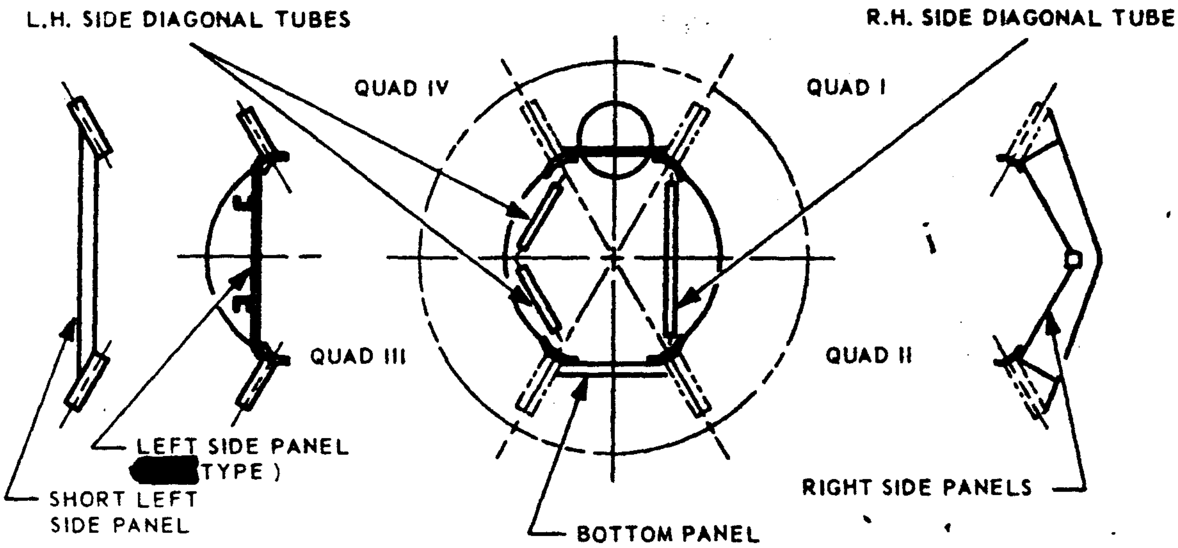


Figure 43. Aft Rack Equipment Panels Left Hand Side Research Payload



		L.H. SIDE	BOTTOM	R.H. SIDE	RACK TOTAL CAPABILITY	WT. INCREASE
A	CONFIG.	DIAGONAL TUBES ONLY	PANEL	DIAGONAL TUBE ONLY	200#	2.3#
	CAPACITY		100#			
B	CONFIG.	DIAGONAL TUBES PLUS SHORT PANEL	PANEL	DIAGONAL TUBE ONLY	200#	9.1#
	CAPACITY	60#	100#			

C	CONFIG.	[redacted] TYPE PANEL (NO DIAGONAL TUBES)	PANEL	DIAGONAL TUBE ONLY	200#	12.9#
	CAPACITY	100#	100#			
D	CONFIG.	[redacted] PANEL AND [redacted] PACK. (NO DIAGONAL TUBES)	PANEL	HEX. RACK WITH INTERM. FRAME(NO DIAGONAL TUBE)	516#	30.6#
	CAPACITY	250#	100#	166#		
E	CONFIG.	DIAGONAL TUBES PLUS LONG, SHORT PANEL	PANEL	DIAGONAL TUBE ONLY	200#	14.2#
	CAPACITY	60#	100#			
F	CONFIG.	DIAGONAL TUBES ONLY	PANEL	HEX. RACK WITH INTERM. FRAME(NO DIAGONAL TUBE)	200#	17.7#
	CAPACITY		100#	100#		

1. ALL CONFIGURATIONS SHOWN ABOVE MUST HAVE AUXILIARY ENGINE CONE PANELS INSTALLED.
2. ALL LOADINGS AS LISTED ABOVE ARE MAXIMUM BASED ON TOTAL AFT RACK CAPABILITIES. CAPABILITIES ALSO DEPEND ON DISTRIBUTION AND ATTACHMENT OF ITEMS.
3. ALL CAPABILITIES SHOWN ARE WEIGHTS OF PROGRAM PECULIAR EQUIPMENT ONLY.

Figure 44. Aft Rack Loading Configurations

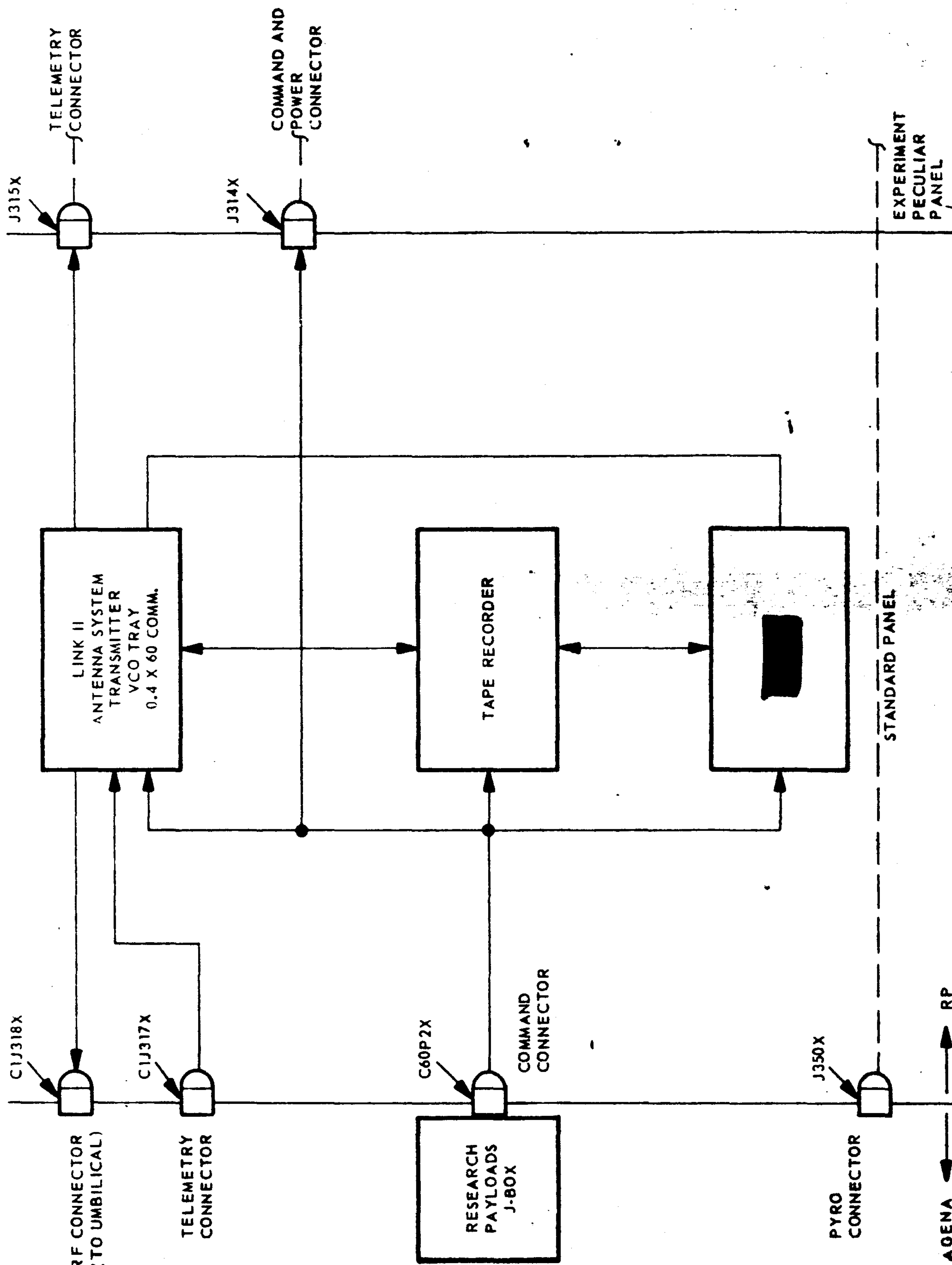


Figure 45. Standard Panel

3.1.16.2.3 Secondary Payload Space. The secondary payload panels shall be designed to maintain aft rack structural integrity. The panel design shall be capable of utilizing the maximum available aft rack space. The space capabilities of the secondary payload panels shall be as defined in Figures 46, 47, 48, and 49 as follows:

a. Left Side [redacted] Panel - Figure 46

b. Short Left Side Panel - Figure 47

c. Bottom Panel - Figure 48

d. Right Side Panels - Figure 49

3.1.16.2.3.1 Qualification. The secondary payload support equipment shall be flight qualified to requirements set forth by Program [redacted].

3.1.16.3 Electrical Accommodation. The secondary payload module electrical interface shall be established through the Program [redacted] secondary payload J-Box, as referenced in 3.1.6.4.4.2. Vehicle unregulated, auxiliary, and plus 28 VDC pyro power, as well as brush commands, real time commands, support and telemetry functions shall be interfaced to the secondary payload through the secondary payload J-Box, as specified by Program [redacted] SDR-6, System Design Requirements for Vehicle/Payload Interface. Within the secondary payload module, the experiment shall be electrically interfaced with the secondary payload standard panel. The telemetry and support equipments, which includes the [redacted] experiment, shall be mounted on the secondary payload forward bottom panel and installed on the vehicle aft rack as shown in Figures 42 and 43.

3.1.16.3.1 Electrical Power. The nominal DC power available to the secondary payload module at the interface shall be:

a. Plus 28 VDC 10 ampere fused (unregulated)

b. Plus 28 VDC 15 ampere fused (unregulated)

c. Plus 28 VDC pyrotechnic power

d. Plus 28 VDC 10 ampere fused (auxiliary)

3.1.16.3.1.1 AC Power. The secondary payload experiment shall provide AC power for its own use. Proper shielding and precautions shall be taken to eliminate radiation or feed back into power lines to prevent interfacing with other vehicle equipment and systems.

3.1.16.3.2 Telemetry System. The secondary payload telemetry link (designated Link 2) shall be flown on all program [redacted] vehicles.

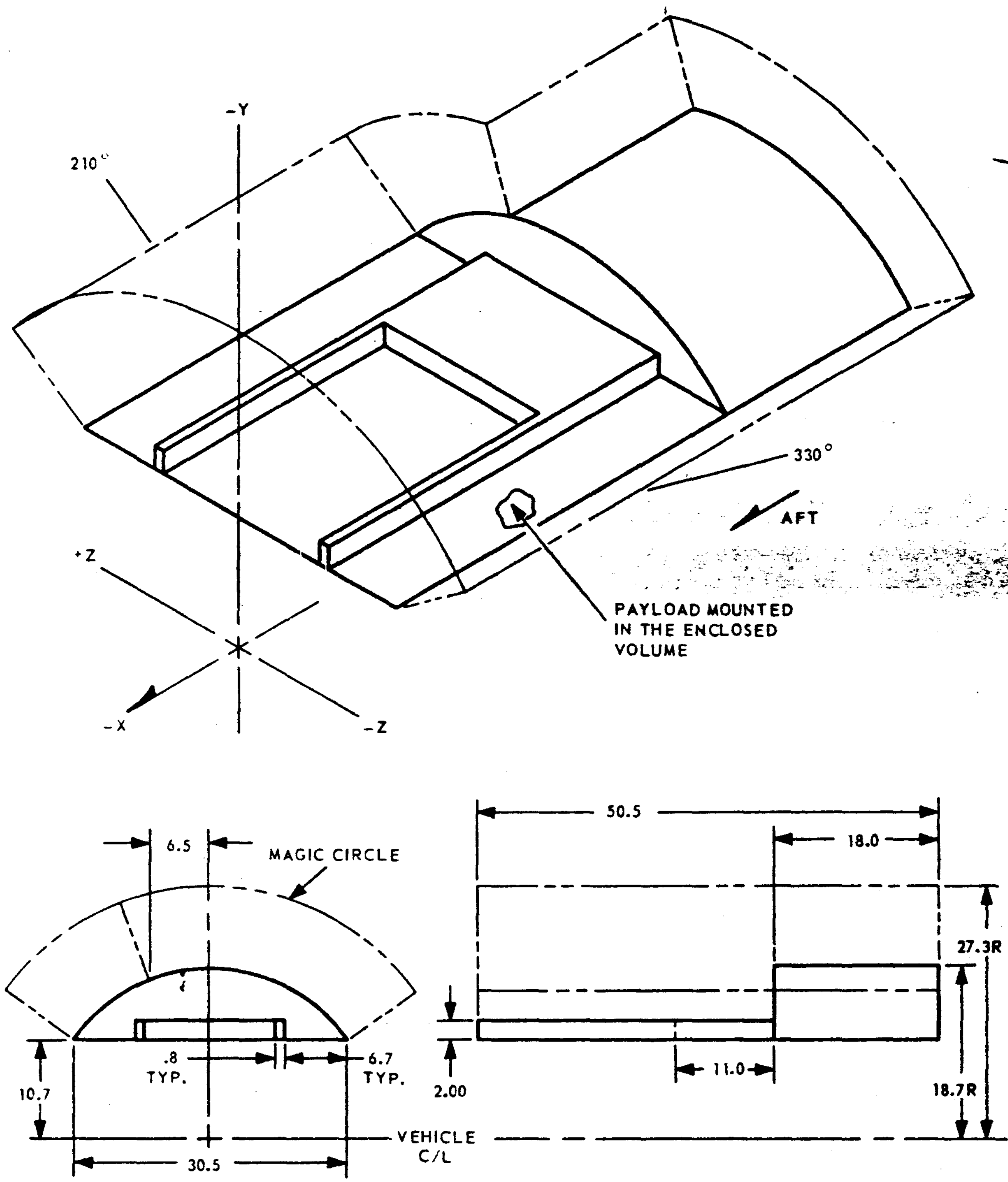


Figure 46. Equipment Volume Diagram Left Side
[redacted] Panel Research Payload

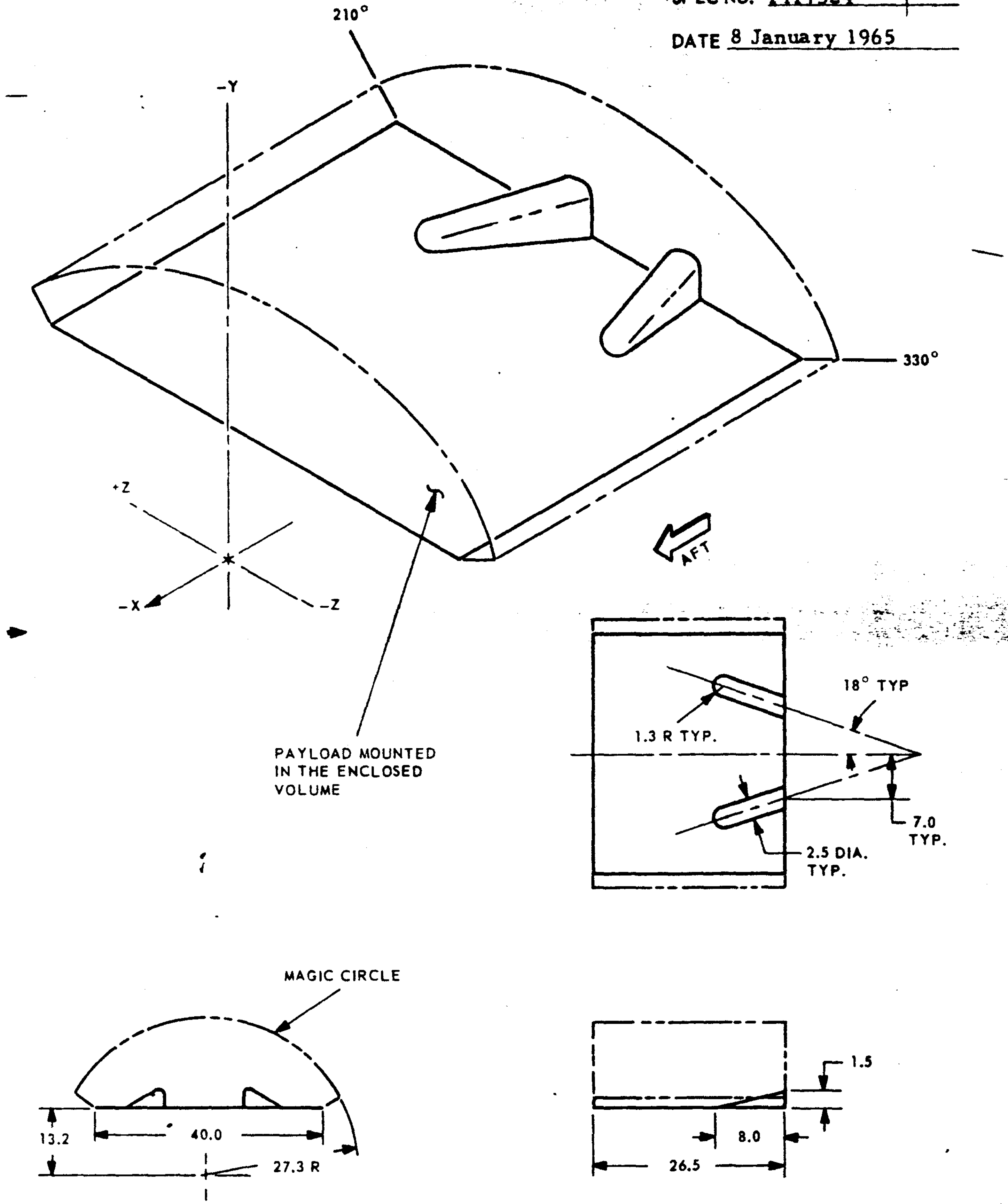


Figure 47. Equipment Volume Diagram Short Left Side Panel Research Payload

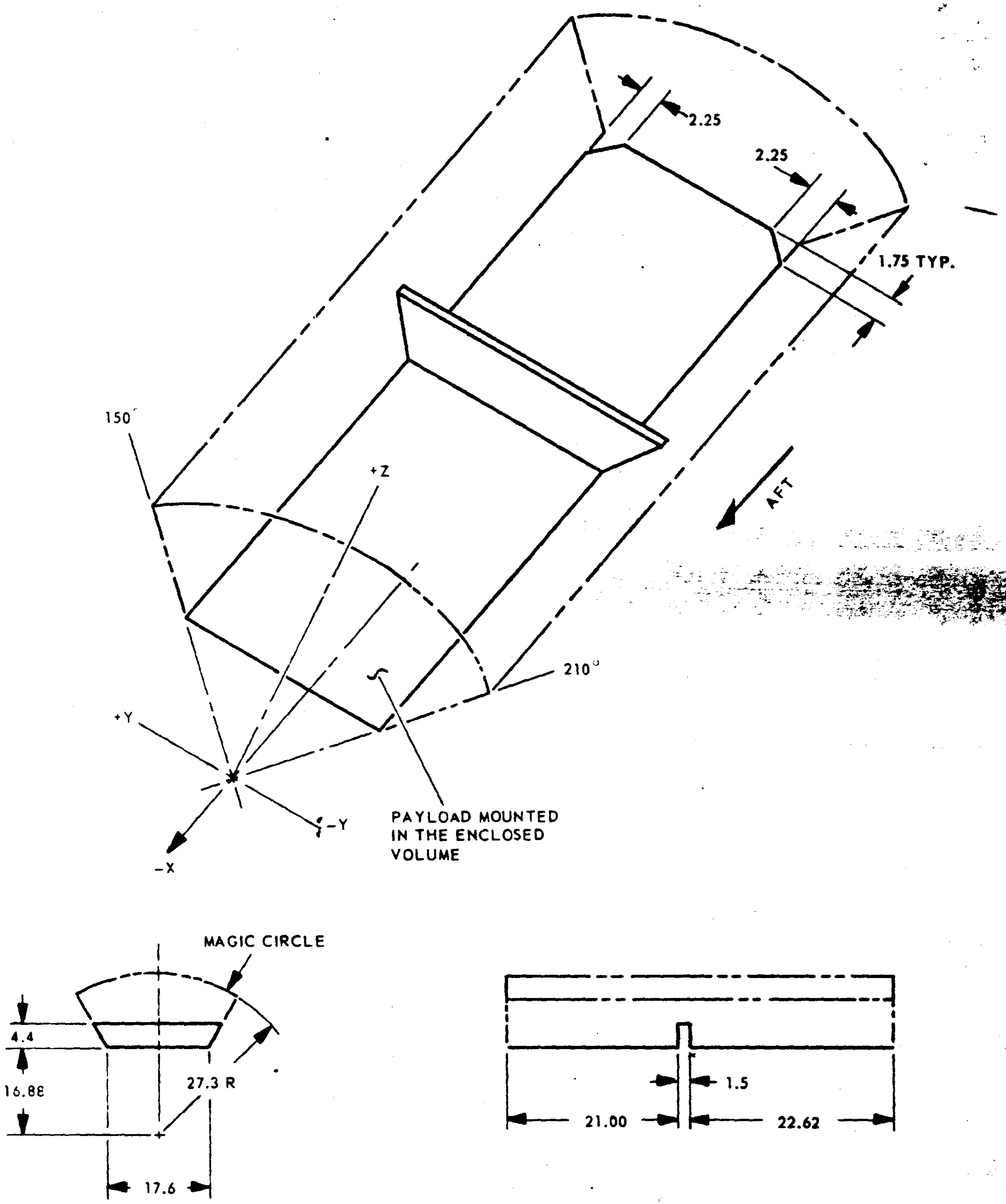


Figure 48. Equipment Volume Diagram Bottom Panel Research Payload

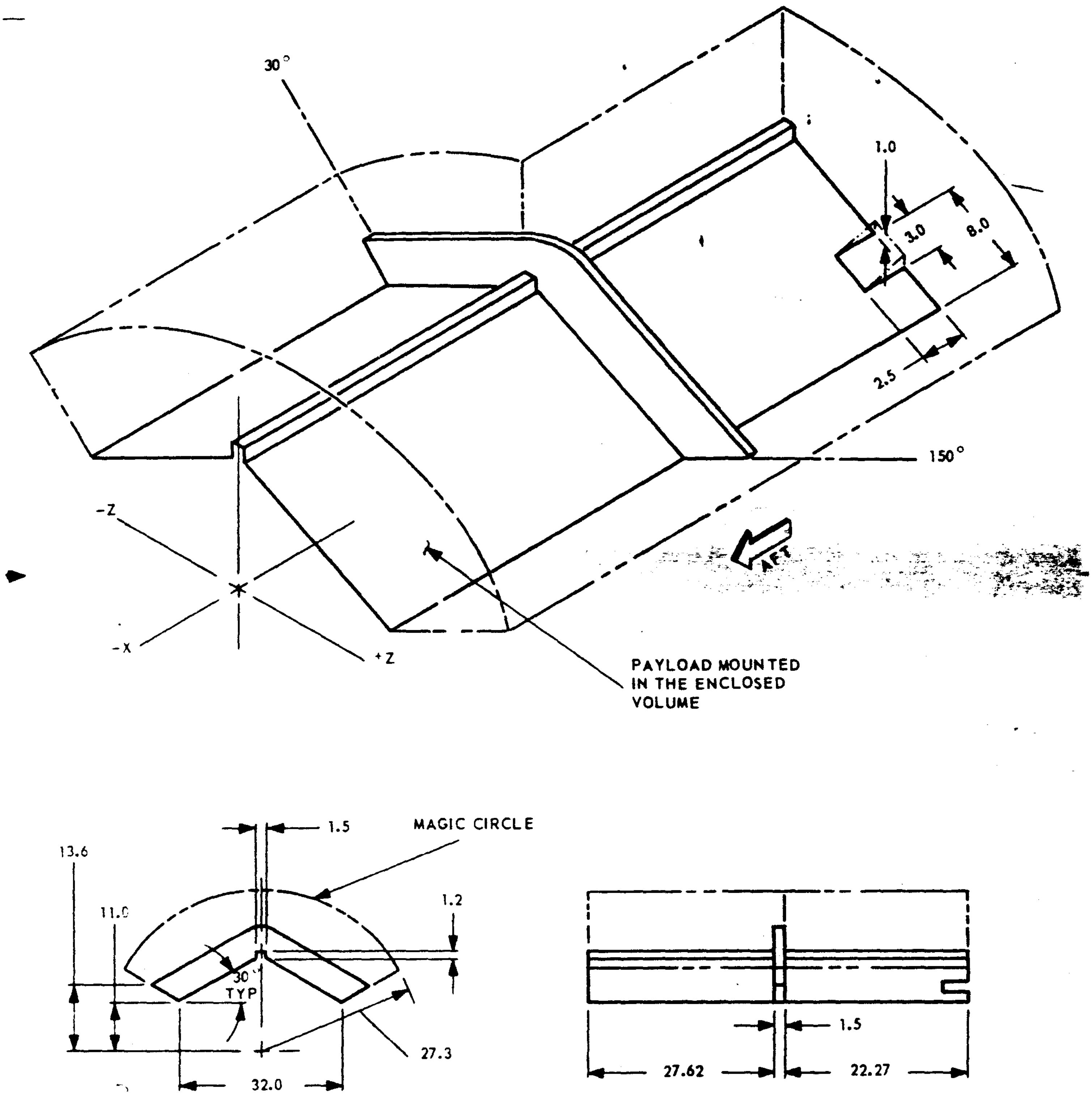


Figure 49. Equipment Volume Diagram 90° Panel System - Right Side Research Payload

The Link 2 performance parameters are the same as specified for Link 1 in section 3.1.13. The secondary payload telemetry system shall consist of an FM/FM VHF data link for transmitting data via a self-contained system interfaced to the vehicle umbilical for Link 2 status during ground operations. The basic components of the system, as shown in Figures 45 and 50 shall be instrumentation sensors, 0.4 x 60 commutator, 26:1 tape recorder, voltage controlled oscillators (IRIG), solid state transmitter (2 watts), and the antenna system for Program [redacted] vehicle mission backup and experiment operational requirements. With increased secondary payload module data requirements and power availability, Link 3 and Link 4 FM/FM VHF data link capabilities can be added.

3.1.16.3.3 Recording Capabilities. The tape recorder shall have dual track data recording capability with a readin-to-readout ratio of 26 to 1. The maximum readin time shall be 182 minutes from a 1 x 60 or 0.4 x 60 commutator with equivalent readout time of approximately 7 minutes. The signal response shall be 300 cps, or DC to 60 pps commutated.

3.1.16.3.4 Commutation. The commutators used as part of the secondary payload module shall be two rings, shorting type, make-before-break, 0.4 or 1 revolution per second, with 60 points per commutator ring. Three points are required for calibration and three points are required for synchronization, leaving 54 points for secondary payload data per commutator ring.

3.1.16.4 Stored Commands. Twelve stored commands shall be available to the secondary payloads for operation of programming equipment during flight. Two of these commands shall be assigned to the secondary payload Link 2 control and ten commands shall be available for the experiment operation. These brush commands shall be plus 28 volts DC unregulated power for nominal 10-second durations.

3.2 System Design and Construction Standards. The design and construction of the system shall be as specified in DAC Specification DS-2345, LMSC Specification 1416559, LMSC Specifications T-3-3-001, T-3-4-001, T-3-3-004, and applicable subsystem specifications.

3.2.1 General Design and Construction Requirements.

3.2.1.1 Selection of Specifications and Standards. Selection of specifications and standards shall be as specified in DAC Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.2.1.2 Materials, Parts, and Processes. Materials, parts, and processes shall be as specified in DAC Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.2.1.3 Moisture and Fungus Resistance. Moisture and fungus resistance shall be as specified in individual subsystem specifications.

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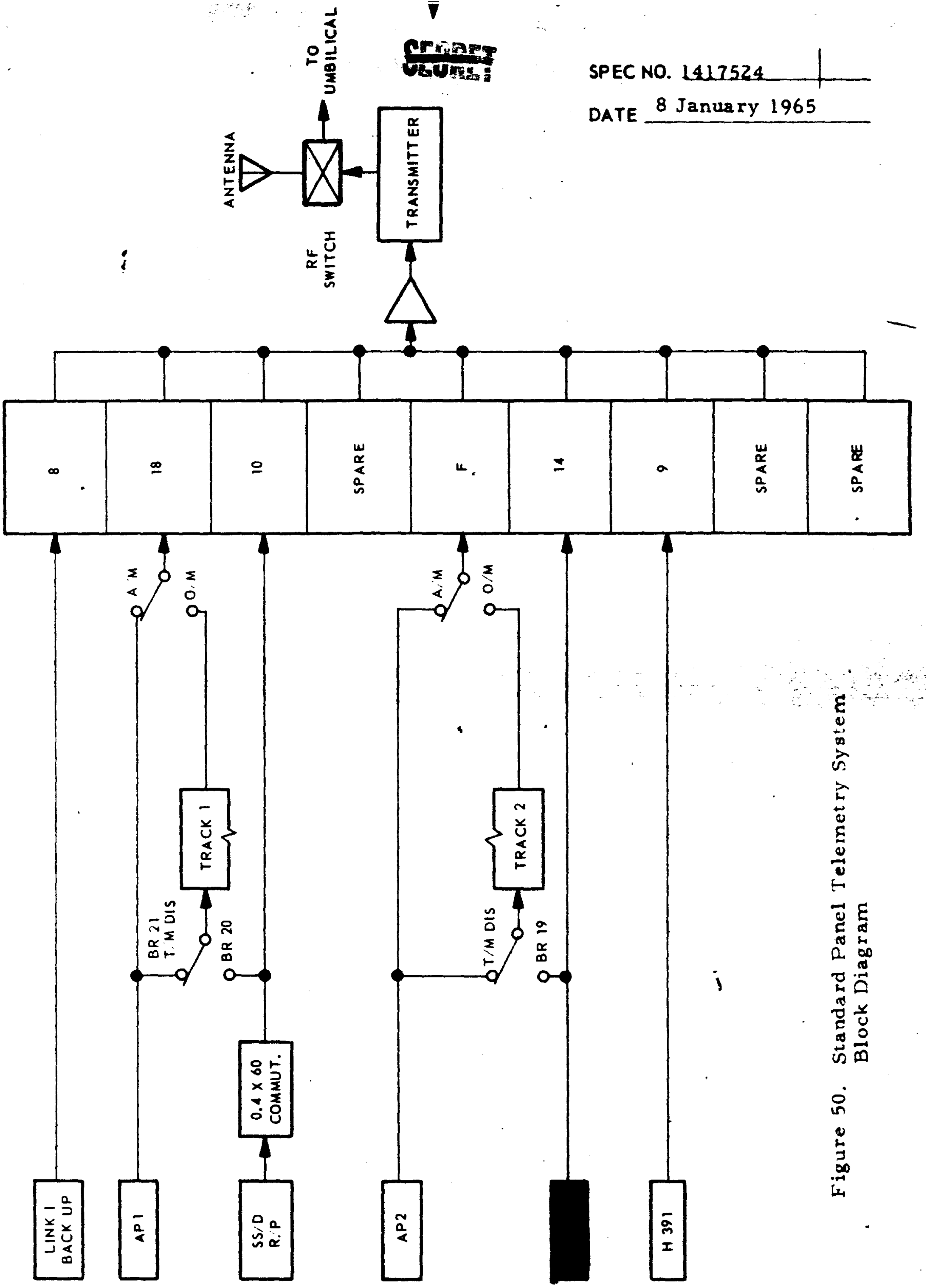

Figure 50. Standard Panel Telemetry System Block Diagram

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3.2.1.4 Corrosion of Metal Parts. Metal parts shall be resistant to corrosion as specified in individual subsystem specifications.

3.2.1.5 Interchangeability and Replaceability. The design of the system shall specify tolerances no more stringent than necessary to achieve the interchangeability and replaceability required throughout its operational life. The extent of establishing and maintaining interchangeability for the system shall be limited to the mechanical interfaces. The interchangeability and replaceability requirements for the system shall be as specified in Douglas Aircraft Company Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.2.1.6 Workmanship. Workmanship shall conform with the standard practices prevalent in the aerospace industry. Uniformity of shapes, dimensions, fit, and performance shall permit replaceability of items as dictated by their operational requirements. There shall be no evidence of poor workmanship in the system.

3.2.1.7 Electromagnetic Interference. Electromagnetic interference requirements shall be as specified in DAC Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.2.1.8 Identification and Marking. The identification and marking requirements for the system shall be as specified in Douglas Aircraft Company Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.2.1.9 Storage. Storage requirements shall be as specified in DAC Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

3.3 Reliability Requirements.

3.3.1 Life Requirements.

3.3.1.1 General. As an objective, all flight equipment shall be designed and assembled of suitable materials so that all Program [redacted] calendar life and operating life requirements can be met under the environments encountered without an increased failure rate. These minimum life requirements are stated below. Where other objectives require that a less durable design be adopted, maintenance, repair, and scrap procedures shall be specified, and the life experience of each limited life item shall be recorded and controlled. The accumulated calendar life and operating life of all limited life items shall be reviewed as part of the final flight certification procedure to insure that sufficient useful life remains to meet the flight requirements without reliability degradation.

3.3.1.2 Limited Calendar Life. As an objective, materials, parts, and assemblies shall be used which are not subject to age or temperature deterioration within a calendar life of three years. Items which do not satisfy the minimum calendar life requirement shall be marked and controlled in general conformance with the procedures established in ANA Bulletin No. 438a, Age Controls for Synthetic Rubber Parts.

3.3.1.3 Limited Operating Life. As an objective, materials parts, and assemblies shall be used which are not subject to an increasing failure rate due to wear during an operating period of at least 2000 hours of nominal operation or equivalent cycles of operation after prime contractor acceptance. In no case shall the non-wear-out cyclical life be less than three times the maximum estimated or experienced Program [redacted] cyclical life. Where the above objectives are not met, the item shall be classified as a limited operating life item with specified maintenance, repair, and scrap procedures. Elapsed time indicators shall be utilized or an operating log shall be maintained to report accumulated operation time or operation cycles for all limited operating life equipment.

3.3.2 Numerical Reliability Requirements and Goals.

3.3.2.1 General. Reliability requirements as stated herein are current contractual requirements with a specified demonstration test. Reliability goals as stated herein are design and production objectives and usually closely approximate the predicted reliability from piece part reliability estimates or historical data.

3.3.2.2 Launch Reliability Goal. A minimum launch probability goal for LMSC aerospace ground equipment at a nominal launch complex shall be 0.95 during the 14-hour terminal countdown phase. Critical and major equipment as defined below shall be assigned reliability objectives which are not less than 0.98 and 0.95 respectively. Critical equipment is defined as equipment whose malfunction could result in vehicle loss or extreme safety hazard. Major equipment is defined as equipment whose malfunction would result in a mission abort, serious program loss, or major safety hazard.

3.3.2.3 Booster Requirements and Goals. The Thor SLV-2A booster shall have a required and demonstrated flight reliability of 0.85. The demonstration shall be a successful performance of a simulated flight system test performed at the contractor's plant. Based on SLV and SLV-2A flight experience with Program [redacted] operations, the reliability goal shall be to maintain a booster launch ascent reliability of at least 0.94.

3.3.2.4 Radio Guidance Requirements and Goals. The Series 600 Radio Guidance System was designed with the objective of exceeding a reliability of 0.985 for a time period consisting of 15 minutes of operation before launch and 5 minutes of operation in the flight environment.

3.3.2.5 Standard Agena Requirements and Goals. The S-01B and SS-01B vehicles shall be designed for maximum reliability commensurate with other applicable criteria to ensure high probability that the vehicle is capable of performing its assigned mission. The design requirement shall be to achieve 90 percent reliability, at 75 percent confidence, in performing the primary flight objectives during the ascent and injection portion of the mission. The reliability goal shall be an ascent flight success ratio of 94 percent. Measurement criteria for conformance to this requirement shall be as outlined in the applicable production contract reliability program plan.

3.3.2.6 Program [redacted] Peculiar Equipment. The Program [redacted] Peculiar Equipment excluding GFE and payloads shall have a reliability requirement beginning at 0.75 for FTV 1601 for a four-day mission through parachute deployment. For subsequent vehicles up to a total of 24 the reliability requirement shall increase so that $R_i = 1 - \frac{1.172}{(T_i + 24)^{0.480}}$ where R_i is the i^{th} vehicle reliability requirement and T_i = accumulated vehicles beginning with 1601. The reliability requirement for the twenty-fourth vehicle shall be 0.817 in accordance with the above formula. The reliability requirement demonstration is established in LMSC A381275, Test of Attainment Reliability Requirements and Goals.

3.3.2.7 Model 39205 Vehicle Reliability. The Model 39205 vehicle shall consist of the standard Agena with Program [redacted] peculiar equipment. The Model 39205 vehicle reliability goal without payload for a 4 day mission from lift-off through capsule reentry shall be 0.860 under Air Force Contract [redacted]

3.3.2.8 Overall Reliability. The over-all reliability goal for the Thor booster, radio guidance, and the Agena Model 39205 satellite by the product rule shall be 0.796 for a four-day mission from launch through capsule reentry, excluding payload equipment. For the requirements and goals for extended missions, refer to the Reliability Program Plan, LMSC A069558.

3.3.3 Estimated System Reliability. A reliability block diagram of the overall Program [redacted] dual mission is included as Figure 51 to provide an indication of the capability of the system in meeting the requirements and goals stated herein. The numerical values shown are based on LMSC flight records of vehicle performance through Vehicle 1178 as presented in the Program [redacted] Quarterly Reliability Estimate and Analysis Report, LMSC A705291, dated 1 November 1964.

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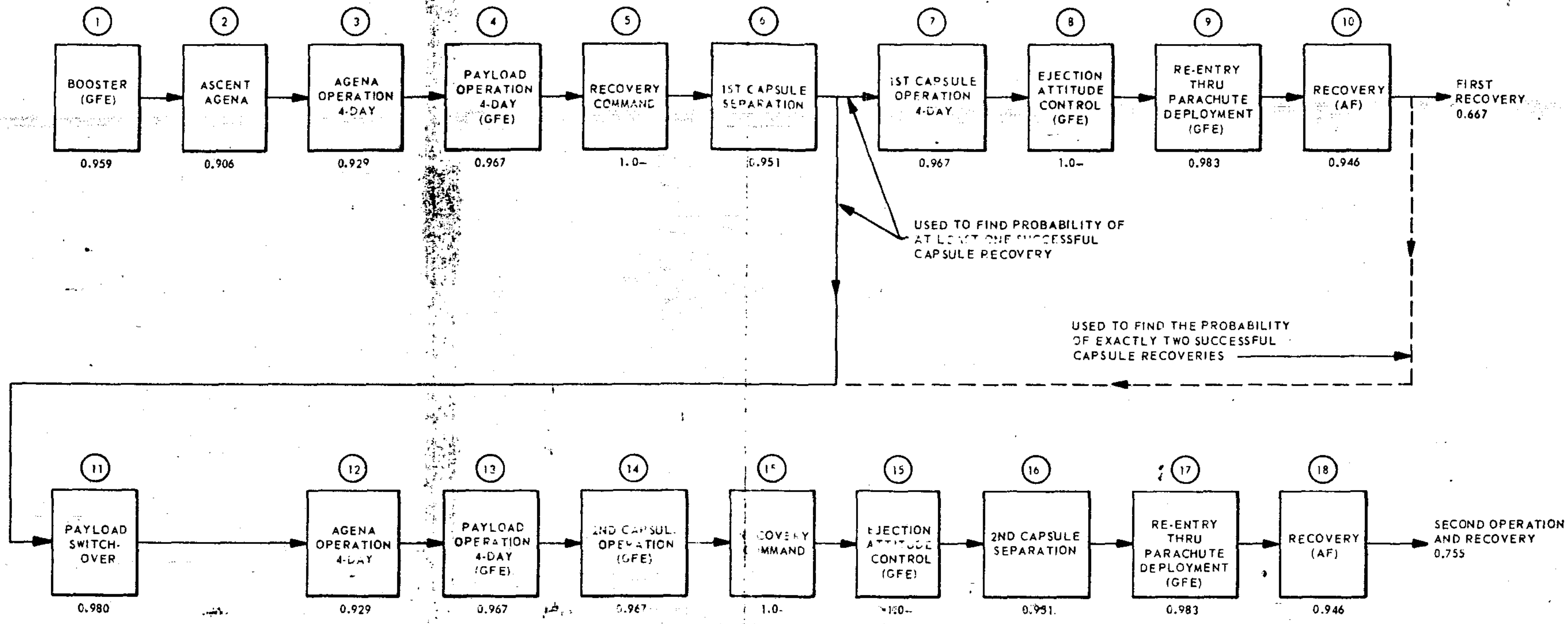

NOTES: (a) BLOCKS 5, 6, 7 AND 8 ARE SHOWN IN THE ABOVE SEC E T T TI Y THE ELIABILITY BLOCK DIAGRAM REQUIREMENTS. THE CORRECT ORDER FOR FIRST RECOVERY SHOULD A AL BE , 8, A 6.

(b) BLOCKS MARKED (GFE OR (AF E NOT ED IN M 39205 ABILIT OBJE TIVES, BUT ARE INCLUDED HERE TO COMPLETE THE OVER-ALL SYSTEM RELIABILITY MODE

PROBABILITY OF RECOVERING BOTH CAPSULES IS 0.503

F 1. -ra. [redacted] Dia
k Dia a

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4. QUALITY ASSURANCE

4.1 Testing Program. The testing program for the satellite system shall consist of manufacturing tests, element system tests, and launch configuration simulated flight tests.

4.1.1 Agena Vehicle Test Plan. The vehicle test plan shall define and limit all testing activities associated with the program vehicle from the receipt of manufactured, purchased, or GFE components to the initiation of terminal launch countdown. A typical vehicle test plan is: "LMSC 1361915, Space Systems Vehicle Test Plan, Model 39205, Vehicle 1613."

4.1.2 Thor Booster Testing. Thor booster testing shall be controlled by DAC (To be supplied by DAC when available).

4.1.3 Primary Payload Testing. The primary payload shall be tested as specified in section 3.1.15.

4.1.4 Radio Guidance Testing. The radio guidance system shall be tested to verify that it meets all the requirements of Western Electric Specifications G.S. 19-900 and G.S. 64-250.

4.1.5 Secondary Payload Testing. The vehicle test plan shall require that the secondary payloads be tested to the extent necessary to assure that they will not in any way degrade or jeopardize the normal functioning of the primary payload or the Agena vehicle. Additional secondary payload testing shall be accomplished as deemed necessary to assure proper operation of the payload.

5. PREPARATION FOR DELIVERY

5.1 Subsystems. Subsystems shall be prepared for delivery as specified in DAC Specification DS-2345, LMSC Specification 1416559, and applicable subsystem specifications.

6. NOTES

6.1 Principal Subsystem Sources. The following principal subsystems shall be supplied by the organizations shown:

a. SLV-2A, Douglas Aircraft Corporation

b. Model 39205, Lockheed Missiles & Space Co.

c. Primary Payload, Lockheed Missiles & Space Co.

d. Secondary Payloads, Lockheed Missiles & Space Co.

e. Radio Guidance, Bell Telephone Laboratories, Western Electric.

6.2 Abbreviations. Abbreviations used in this specifications represent the terms shown opposite them, as follows:

Abbreviation	Full Term
AC, ac	Alternating current
AGC	Automatic Gain Control
AGE	Aerospace ground equipment
AP	Primary payload
BTL	Bell Telephone Laboratories
C&C	Communications and control
DAC	Douglas Aircraft Company
DC, dc	Direct current
deg.	Degrees
F/C	Flight control
FM	Frequency modulated
G&C	Guidance and control
GFE	Government furnished equipment
H/S	Horizon sensor
[redacted]	[redacted] Tracking Station
IRIG	Inter-Range Instrumentation Group
IRP	Inertial Reference Package
KIAS	Knots Indicated Air Speed
[redacted]	[redacted] Tracking Station

Abbreviation	Full Term
LMSC	Lockheed Missiles & Space Company
MSL	Mean Sea Level
[redacted]	[redacted] Station
NM, nm	Nautical miles
NTS	North to south
PAM	Pulse amplitude modulated
PDM	Pulse duration modulated
PMR	Pacific Missile Range
PPM	Pulse position modulated
R/E	Reentry
RSS	Root sum square
[redacted]	[redacted]
STN	South to north
TAG	Tested and guaranteed
T/M	Telemetry
[redacted]	[redacted] Tracking Station
UHF	Ultra high frequency
VAFB	Vandenberg Air Force Base
VCO	Voltage controlled oscillator
VHF	Very high frequency
V/M	Velocity meter
[redacted]	[redacted] Tracking Station

6.3 Glossary of Terms. Principal terms used in this specification are presented below, together with other terms used interchangeably in their place.

Basic Terms	Equivalent Terms
a. Thor booster	SLV-2A First stage Thrust augmented Thor (TAT) Booster

Basic Terms	Equivalent Terms
b. Agena vehicle	Model 39205 Vehicle Orbital vehicle Second stage S-O1B, SS-O1B Standard Agena
c. Primary payload	Advanced payload Advanced Projects payload Payload
d. Secondary payload	Research payload Experimental payload
e. Standard sequence timer	Standard timer Standard D-timer D-timer
f. Recovery timer	Di/An timer Solid state timer

[redacted]