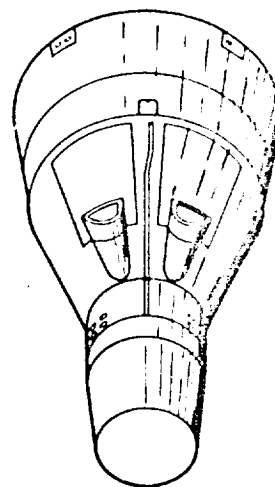


GEMINI



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GUIDANCE AND CONTROL PERFORMANCE SUMMARY

MCDONNELL

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(CODE) _____
(PAGES) 24
(CATEGORY) _____
(NASA CR OR TXR OR AD NUMBER) _____
FACILITY ROOM _____

C68-4257

① G E M I N I

GUIDANCE AND CONTROL PERFORMANCE SUMMARY

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MCDONNELL

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15 May 1964	Title pg. 2-6/ 8-VI 16, 17 21-23 26-28 30, 32 35-38 40	ii		Update Report	<i>R.C. Wensler</i>	<i>C. Johnson</i>

**GUIDANCE AND CONTROL
PERFORMANCE SUMMARY**

This document is intended to provide a quick reference to the guidance and control elements of the spacecraft. The material is designed to provide the basic information on equipment and function along with references to more complete documents.

GUIDANCE AND CONTROL SYSTEM

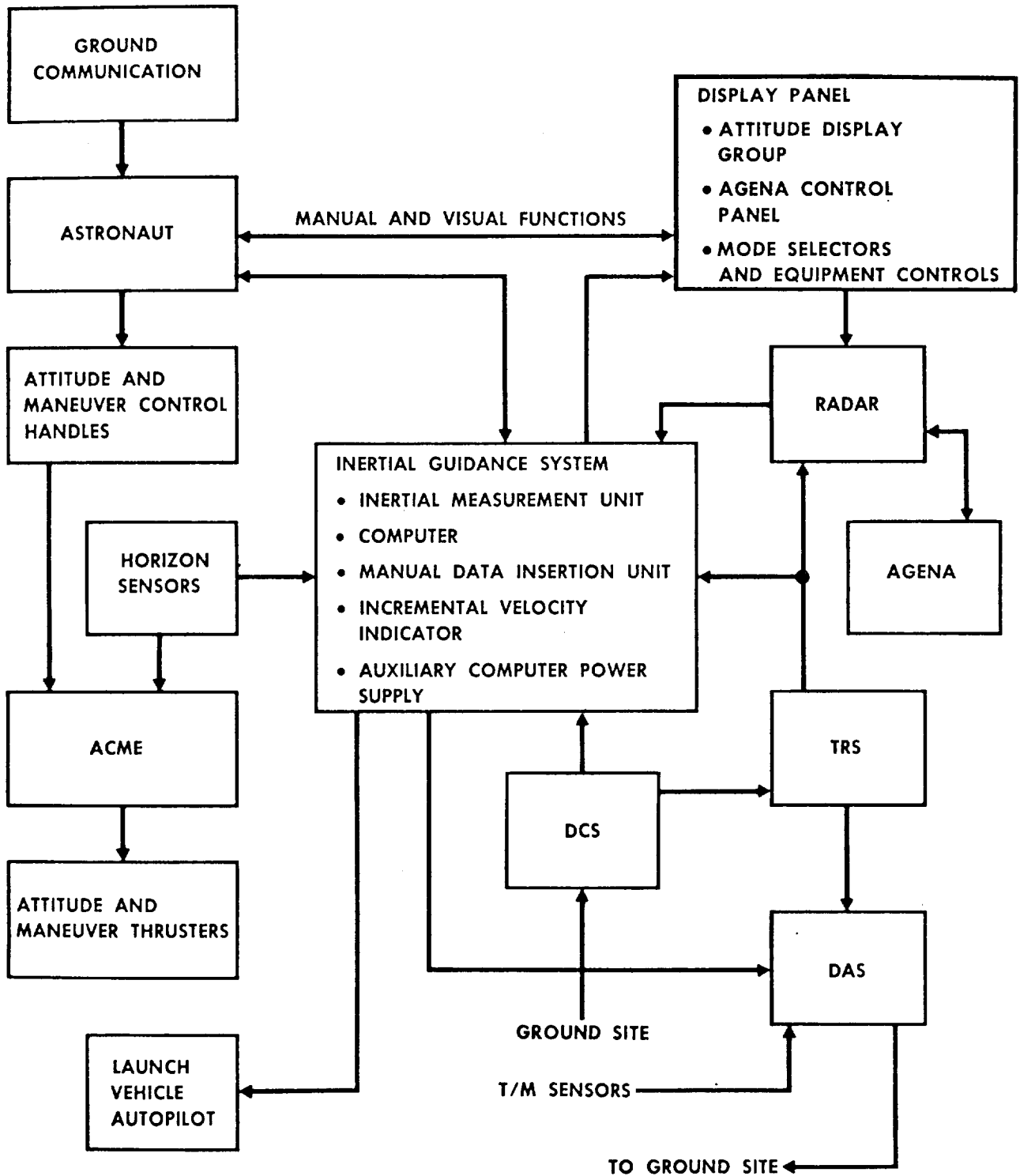


Figure 1

1. Summary - The discussion of the guidance and control system is logically divided so the functions of the guidance system and the control system are presented separately. Guidance has functional responsibility for trajectory management, and control has functional responsibility for spacecraft orientation. The guidance and control system performs required functions during each phase of the mission profile which consists of pre-launch, ascent, catch-up, rendezvous, touchdown prediction, and re-entry. A block diagram of the guidance and control system is presented in Figure 1.

The crew station display panel is presented on page 30 which shows the arrangement of the guidance and control equipment selectors. The individual equipment selectors are discussed separately in the appropriate equipment descriptions.

- 2.0 Guidance - The Gemini guidance system is composed of the rendezvous radar, the Digital Command System, the Time Reference System, and the Inertial Guidance System. The Inertial Guidance System is the heart of the system, providing the coordinate references and computation means for processing sensed or stored information. The radar provides information on the relative geometry to the target for use during rendezvous. The Digital Command System (DCS) provides access to the digital computer for ground transmissions. Accurate timing of critical events is obtained from the Time Reference System (TRS). The following sections provide additional information on the various elements of the guidance system.

GUIDANCE PERFORMANCE CHARACTERISTICS

● ASCENT

INSERTION ACCURACY (FOR PERFECT IMU AZIMUTH ALIGNMENT AND NO VELOCITY UPDATES)

ΔV , VELOCITY MAGNITUDE ERROR.....	35 FT/SEC
$\Delta \Gamma$, FLIGHT PATH ANGLE ERROR.....	.16 DEGREES
Δh , ALTITUDE ERROR.....	1.3 N.M.

● CATCHUP AND RENDEZVOUS

ATTITUDE COMMAND ACCURACY 1 ARC MIN
VELOCITY CHANGE COMMAND ACCURACY .1 FOOT/SECOND

● RENDEZVOUS

FUNCTIONAL SEQUENCE COMMAND TIMING ACCURACY .5 SECOND

● TOUCHDOWN PREDICTION

REENTRY FROM 161 N.M. CIRCULAR ORBIT	MISS DISTANCES
a) INITIAL CONDITION DATA BASED ON MULTI-STATION TRACKING.....	(3 σ) 6.33 N.M.
b) INITIAL CONDITION DATA BASED ON SINGLE STATION TRACKING.....	21.6 N.M.
REENTRY FROM 87-161 N.M. ELLIPICAL ORBIT	
a) INITIAL CONDITION DATA BASED ON MULTI-STATION TRACKING	
1) PERIGEE RETROGRADE.....	9.75 N.M.
2) APOGEE RETROGRADE.....	6.27 N.M.
b) INITIAL CONDITION DATA BASED ON SINGLE STATION TRACKING	
1) PERIGEE RETROGRADE.....	16.6 N.M.
2) APOGEE RETROGRADE.....	16.8 N.M.

● REENTRY

REENTRY FROM 161 N.M. CIRCULAR ORBIT	
a) INITIAL CONDITION DATA BASED ON MULTI-STATION TRACKING.....	(3 σ) 4.74 N.M.
b) INITIAL CONDITION DATA BASED ON SINGLE STATION TRACKING.....	7.23 N.M.
REENTRY FROM 87-161 N.M. ELLIPTICAL ORBIT	
a) INITIAL CONDITION DATA BASED ON MULTI-STATION TRACKING	
1) PERIGEE RETROGRADE.....	5.10 N.M.
2) APOGEE RETROGRADE.....	3.95 N.M.
b) INITIAL CONDITION DATA BASED ON SINGLE STATION TRACKING	
1) PERIGEE RETROGRADE.....	5.85 N.M.
2) APOGEE RETROGRADE.....	6.48 N.M.

2.1 Guidance Functional Modes - The guidance system operates in the following functional modes as controlled by the computer mode selector.

Pre-launch - The pre-launch functions are associated with the flight readiness preparation of the guidance system which includes astronaut functions, platform torquing by AGE, and computer data insertion using the DCS. Guidance system status is established through the combined functions of AGE, the data acquisition system (DAS) and by astronaut monitoring.

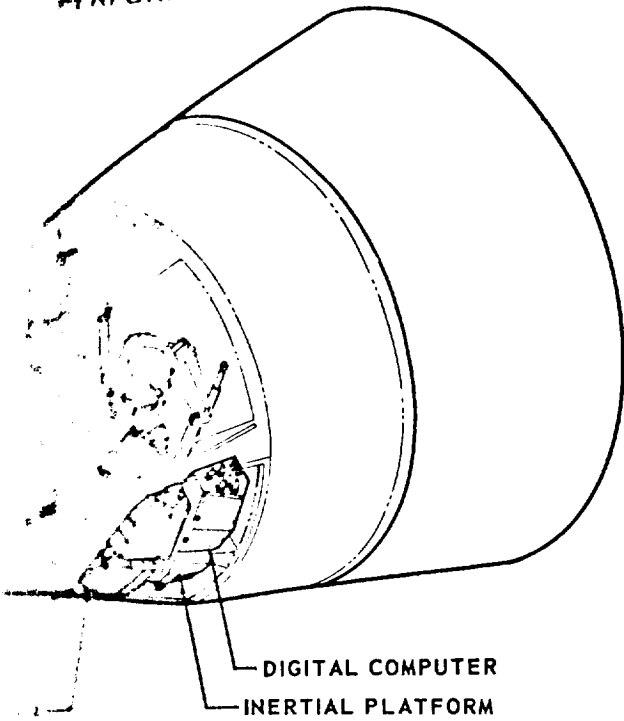
Ascent - The guidance system provides backup guidance in the event of a radio guidance system (RGS) malfunction, capability to correct errors in orbit insertion conditions, display information for astronaut evaluation of mission status, and navigational data for re-entry guidance in the event of an abort during boost. The guidance system effectively duplicates the computations of the RGS in an inertial frame defined by the launch pad vertical and the desired azimuth at insertion. Insertion guidance is initiated at SSECO. The required velocity increment to be added for proper insertion is determined based on the computer navigational data and the desired insertion conditions stored in the computer.

Catch-up - In the catch-up mode, the computer has the capability of accepting ground computed incremental velocity components (platform frame) either through the DCS or MDIU. The computer determines the desired attitude commands such that the required velocity increment will be along the X body axis. During thrusting, the measured platform accelerations are transformed to a body axis frame to null the IVI display.

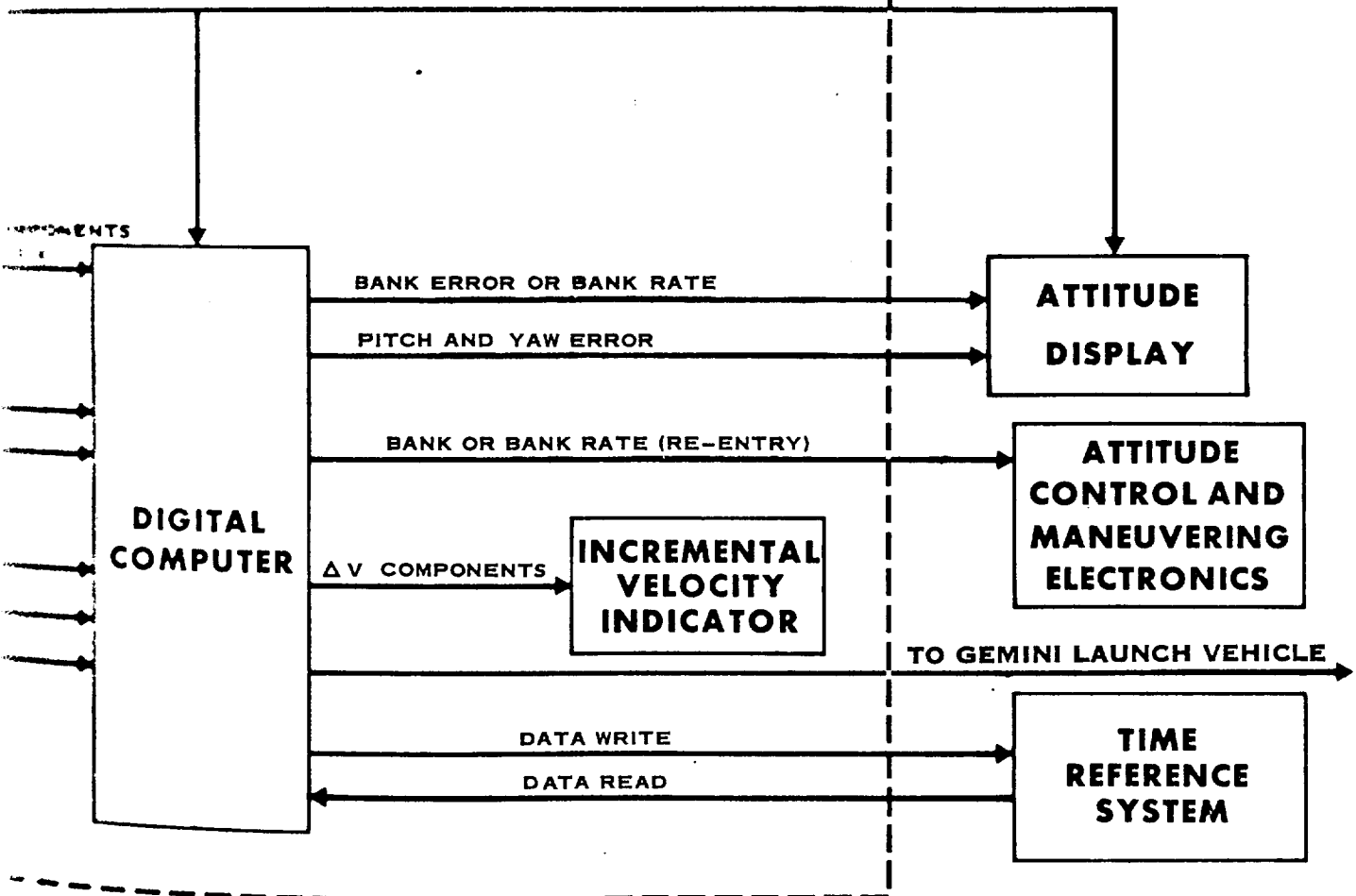
Rendezvous (Orbital Mechanics) - For orbital mechanics rendezvous, the computer processes radar observations of the target, properly resolves the measurement into the coordinate system maintained by the inertial platform, and computes the proper maneuver to cause a future intersection of the orbits. The velocity increment commands are automatically registered on the IVI at the proper times and the astronaut applies thrust until the increment is added.

Touchdown Prediction - The touchdown prediction mode is designed to enable the astronaut to select a landing site in the event of an orbital abort situation. A set of navigational initial conditions are required from the ground prior to astronaut insertion of the first trial retrograde time via the MDIU. The longitude and latitude of the landing site corresponding to the trial retrograde time are displayed on the MDIU and upon astronaut acceptance the time-to-go to retrograde, is automatically displayed and transferred to the TRS.

Re-entry - Spacecraft navigational data and predictions of the impact site are computed to provide sequencing and logic to command spacecraft control action to achieve the desired impact point. The subsystem accepts significant accelerations (retrograde, impulse, drag) sensed by the inertial platform for navigation computations and ignores insignificant accelerations (noise, bias) during periods where real accelerations are non-existent. Bank angle commands are derived to control horizontal and vertical components of lift as a function of down and cross range errors based on differences between predicted and desired touchdown point. The bank angle errors are displayed on the FDN for manual attitude control operation and also serve as the commands for the re-entry attitude control mode.



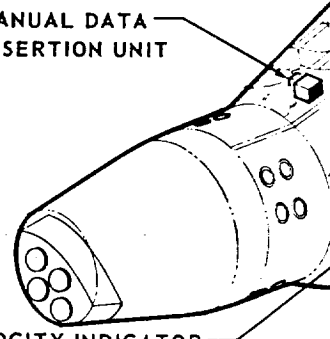
GUIDANCE SYSTEM



~~CONFIDENTIAL~~

GUIDANCE SYSTEM

MANUAL DATA
INSERTION UNIT



INCREMENTAL VELOCITY INDICATOR

POWER SUPPLY
PLATFORM ELECT

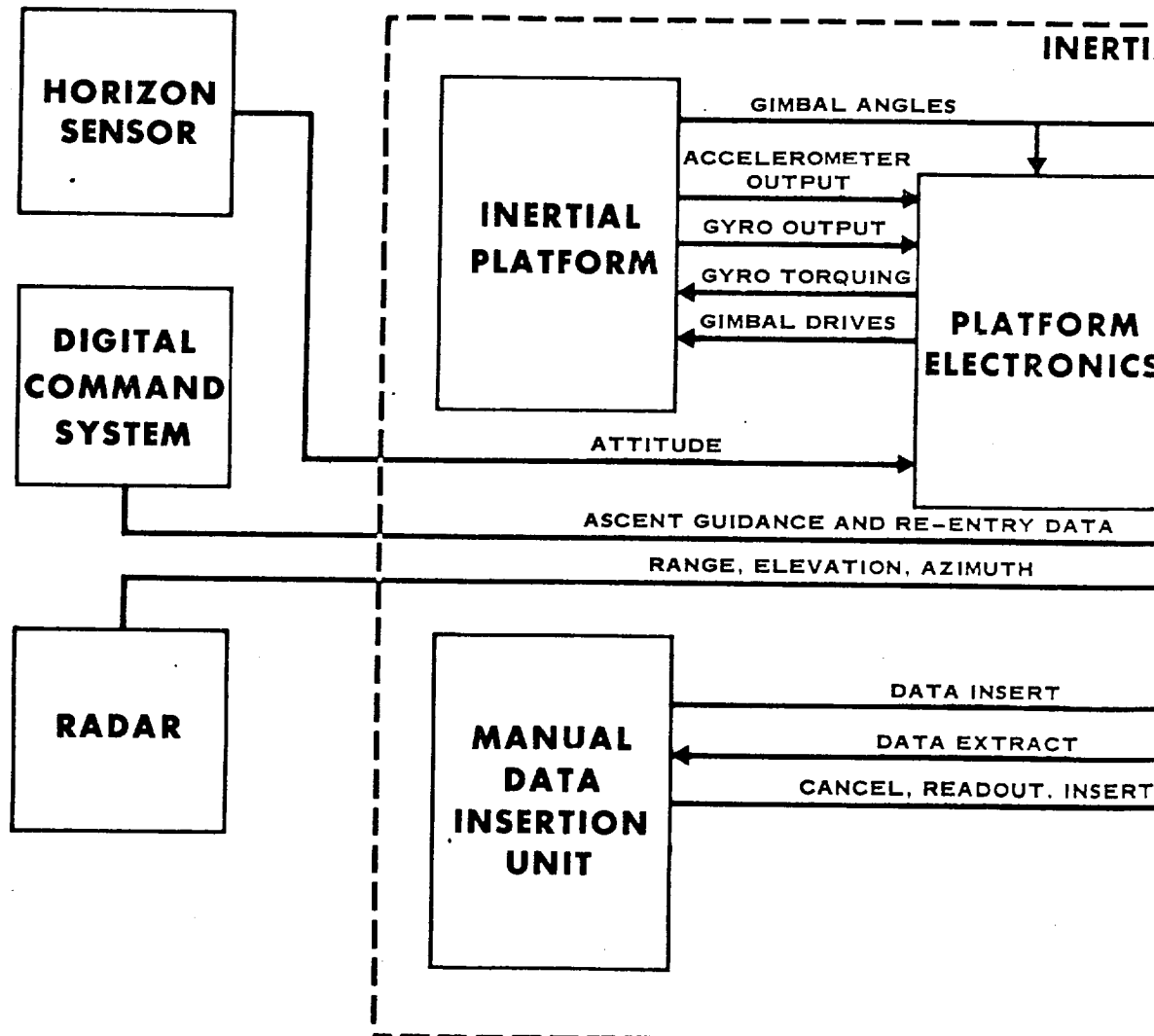


Figure 2

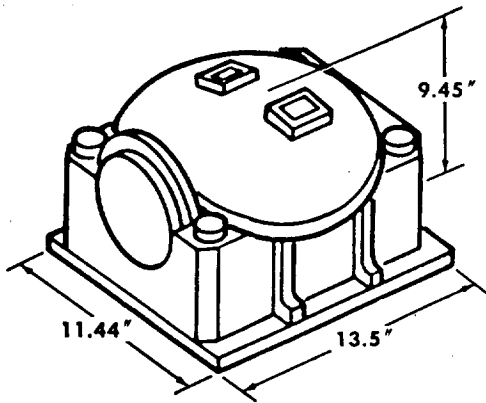
2.2 Equipment - Figure (2) indicates the components that comprise the Gemini Guidance System and shows their functional relationship.

2.2.1 Inertial Guidance System (IGS) - The IGS is outlined in Figure (2), and consists basically of an inertial measurement unit, a digital computer and associated input/output equipment. The Inertial Guidance System power supply operates from spacecraft DC power and provides all power required for operation of the inertial platform, associated platform electronics, the computer, and the attitude control and maneuver electronics (ACME). Honeywell-St. Petersburg for the IMU, and IBM Federal Systems Division for the digital computer and system integration are sub contractors to MAC. Specifications for the IGS are contained in MAC SCD-52-87710.

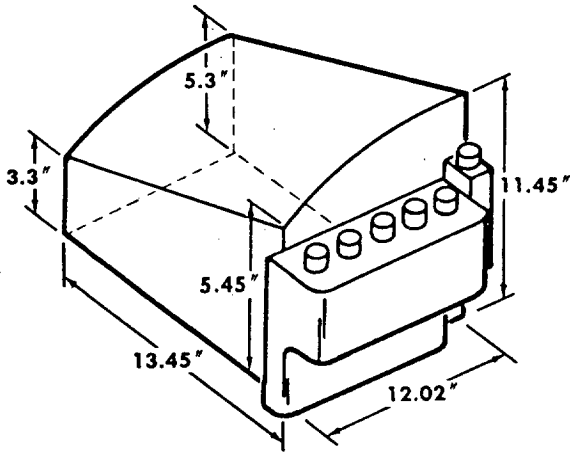
██████████

INERTIAL MEASUREMENT UNIT

MH DGG 8046A1



INERTIAL PLATFORM



IMU SYSTEM ELECTRONICS



IGS

PHYSICAL CHARACTERISTICS

- TOTAL WEIGHT 101 LB.
- TOTAL VOLUME 2 CU. FT.

POWER REQUIREMENTS

- S/C 28VDC
- 300 WATTS AVERAGE
- 400 WATTS MAXIMUM

OPERATING CHARACTERISTICS

- 360 DEGREE OF FREEDOM EXCEPT FOR INNER ROLL ($\pm 15^\circ$)

INPUTS

- HORIZON S
- ERROR SIGN
- AGE TORQU
- VEHICLE AN
- AND LINEAR

ALIGNMENT ACCUR

- PITCH
- YAW
- ROLL

GYRO CHARACTERISTICS

- FLOATED RATE INTEGRATING
- TYPE MH-DGG 8001 B17

POWER REQUIREMENTS

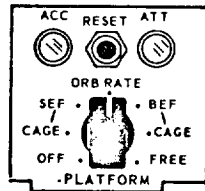
- 26 VOLTS, 3 PHASE, 400 CPS SQUARE WAVE
- 2.5 WATTS

OPERATING CHARACTERISTICS

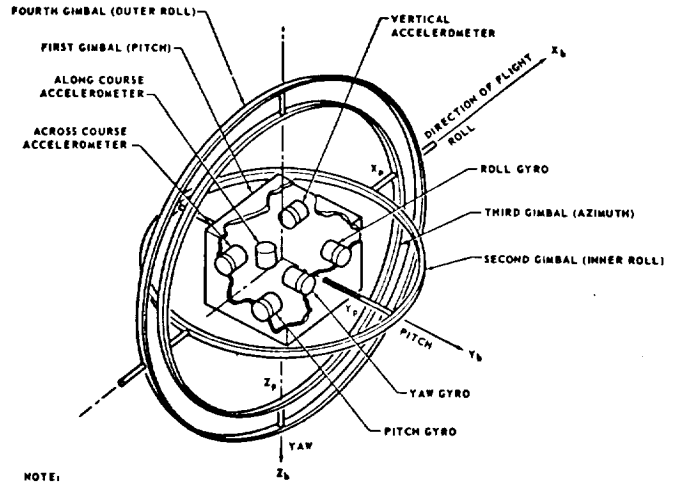
- ANGULAR MOMENTUM 10^5 gm-cm²/SEC AT 24,000 RPM
- TEMPERATURE 185° F
- DRIFT: ACCELERATION INSENSITIVE .30 DEG/HR (UNTRIMMED)
ACCELERATION SENSITIVE 0.5 DEG/HR/g
ANISOELASTIC COEFFICIENT 0.02 DEG/HR/g²
RANDOM: OUTPUT AXIS HORIZONTAL 0.02 DEG/HR
OUTPUT AXIS VERTICAL 0.01 DEG/HR

PHYSICAL CHARACTERISTICS

- INPUT ANGULAR FREEDOM ± 2.5 DEG.
 - WEIGHT 12 OUNCES (EACH GYRO)
 - SIZE 8.8 CUBIC INCHES
- ██████████



**OPERATING CONTROLS
SELECTOR SWITCH**



NOTE:
PLATFORM COORDINATES - X_p, Y_p, Z_p
BODY COORDINATES - X_b, Y_b, Z_b

AXES SYSTEM

OUTPUTS

- GIMBAL ANGLES
- MEASURED PLATFORM ACCELERATIONS
- MALFUNCTION INDICATION

MAC REFERENCE DOCUMENTS

- MAC REPORT 8637
- MAC SCD 52-87717

PERFORMANCE — INFLIGHT

1 MIN.	.4 DEG.
0.1 MIN.	.4 DEG.
1 MIN.	.4 DEG.

ACCELEROMETER CHARACTERISTICS

- PENDULOUS FORCE REBALANCE
- TYPE MH-DGG 116-A9

POWER REQUIREMENTS

- 6.2 VOLT; 14.4 KC; 2.5 VOLT, 100 CPS; 200 MA CONSTANT CURRENT SOURCE
- 8.3 WATTS (EACH)

OPERATING CHARACTERISTICS

- THRESHOLD $< 1 \times 10^{-5}g$
- DYNAMIC RANGE $1 \times 10^{-5}g$ TO $10g$
- BIAS $3 \times 10^{-4}g$
- LINEARITY $1 \times 10^{-4}g/g^2$
- SCALE FACTOR $3.6 \times 10^{-4}g/g$
- OPERATING TEMPERATURE 182° F
- ELASTIC RESTRAINT 0.05 DYNE-CM/MILLIRADIAN

PHYSICAL CHARACTERISTICS

- WEIGHT 9 OUNCES (EACH)
- SIZE 5.7 CU. IN.
- ACCELEROMETER MOUNTING ALIGNMENT WITHIN 100 ARC SEC OF ORTHOGONAL ORIENTATION

2.2.1.1 Inertial Measurement Unit (IMU) - The IMU provides a stable reference system for use in guidance and control tasks and also provides measurements of velocity changes along each of the orthogonal axes to the digital computer and astronauts (see 2.1.1.5). The following operational modes may be selected by positioning the IMU selector switch.

OFF - No power to IMU - Gimbals unrestrained - Gyros fluid thickens as temperature drops and gyros become inoperable. This position used during orbit periods when no guidance activities are pending.

CAGE (SEF) - Fast warm-up power applied until gyros become operable. Torque applied to align gimbals to body axes after warm-up. This position is selected when IMU use is pending. Warm-up may require 20 minutes if IMU has been inoperable for appreciable time. Alignment to body axes may require 10 - 15 minutes depending on gimbal positions at start. Caging will be started about 1 hour prior to expected guidance initiation (rendezvous, orbital maneuver or re-entry).

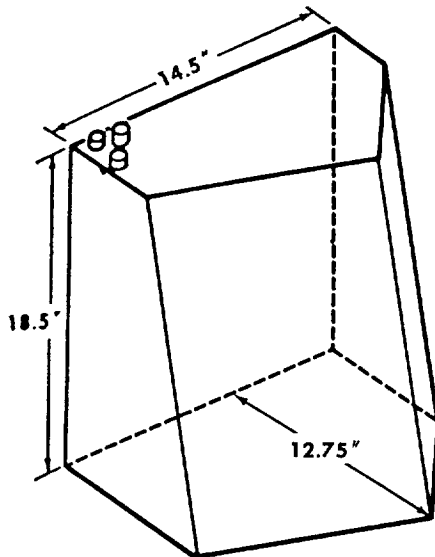
SEF - The spacecraft maintained in precise small end forward (SEF) orientation, and the horizon sensor system roll and pitch outputs are used to establish reference platform alignment orientation. The alignment time is 5 to 15 minutes.

ORBITAL RATE - Inner element torqued at orbital rate during maneuvers (catch-up, rendezvous). The platform must be periodically realigned.

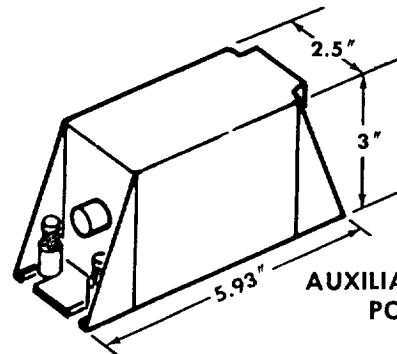
- BEF** - Same as the SEF mode but with switching to reverse horizon sensor phasing. Vehicle is in blunt end forward (BEF).
- CAGE (BEF)** - Similar to CAGE (SEF) operation except that proper electrical phasing is provided so that the platform may be caged with the spacecraft in the blunt end forward orientation.
- FREE** - The FREE position, in which the platform is inertially fixed, is utilized during pre-launch, ascent, and re-entry. The platform is automatically switched to the free condition at retrograde.

A malfunction indication for gyro and accelerometer system is provided by the illumination of the attitude and accelerometer lights on the platform display panel.

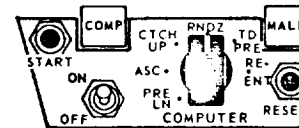
DIGITAL COMPUTER



DIGITAL COMPUTER



AUXILIARY COMPUTER
POWER UNIT
(ACPU)



MODE SELECTOR

CHARACTERISTICS COMPUTER

- BINARY FIXED POINT STORED PROGRAM, GENERAL PURPOSE
- 39 BIT COMPUTER WORDS CONSISTING 13 BIT ADDRESS AND INSTRUCTION, 26 BIT MAGNITUDE AND SIGN
- DIAGNOSTIC SUBROUTINE FOR MALFUNCTION DETECTION

AUXILIARY COMPUTER POWER UNIT

- NICKEL CADMIUM BATTERY
- CONTROL, REGULATOR, SENSING AND CHARGING CIRCUITS

OPERATIONAL PERFORMANCE COMPUTER

- 500 KC ARITHMETIC BIT RATE, 250 KC MEMORY BIT RATE
- RANDOM ACCESS, NON-DESTRUCTIVE READ-OUT MEMORY
- ARITHMETIC TIMES — ADD, SUBTRACT, TRANSFER, 140 μ SEC, MULTIPLICATION — 420 μ SEC, DIVISION — 840 μ SEC
- CONCURRENT ARITHMETIC OPERATION

AUXILIARY COMPUTER POWER UNIT

- .99999 PROBABILITY OF SUCCESSFUL OPERATION
- MAXIMUM RIPPLE CONTENT 1.5 VOLTS

PHYSICAL CHARACTERISTICS COMPUTER

- WEIGHT 55 LB.
- VOLUME 1.1 CU. FT.

AUXILIARY COMPUTER POWER UNIT

- WEIGHT 3.6 LB.
- VOLUME .26 CU. FT.

MAC REFERENCE DOCUMENTS

- MAC SCD 52 — 87710 (COMPUTER)
- MAC SCD 52 — 87723 (ACPU)

SUBCONTRACTORS:

COMPUTER, IBM; ACPU, ENGINEERED MAGNETICS

2.2.1.2 Digital Computer - The digital computer consists of an executor program, operational programs and standard computational sub-routines. The executor program performs the diagnostic checks, determines elapsed time, selects the desired operational program and performs all the data input/output subroutines. The computer memory capacity is equivalent to 12,288 thirteen bit words which can be used as 13 bit instruction words or as 26 bit data words. Limited modification of the stored computer data can be accomplished through interface with the MDIU and DCS. Positioning of the computer Mode Selector switch automatically initializes the selected operational program and pressing the start computation switch initiates program computation. Illumination of the "start computation" switch indicates that the computer is operating in the assigned computational mode and in the rendezvous mode provides signals for periodic platform alignments and target acquisition. Malfunction conditions as determined by the diagnostic subroutine are indicated by illumination of the "Computer Malfunction Light" (computer panel). The computer operational programs are:

Ascent (ASC)

- The launch vehicle boost flight performance is monitored and launch vehicle steering commands generated for secondary guidance. The IGS is in standby operation in the event of launch vehicle primary guidance system malfunction.

Catch-Up

- The catch-up position provides calculation of ΔV required for the catch-up maneuver and provides signals for calculated display on the incremental velocity indicator.

Rendezvous (RNDZ) - The RNDZ position provides for calculation and display of required rendezvous information to achieve intersection with the target vehicle based on platform and radar data.

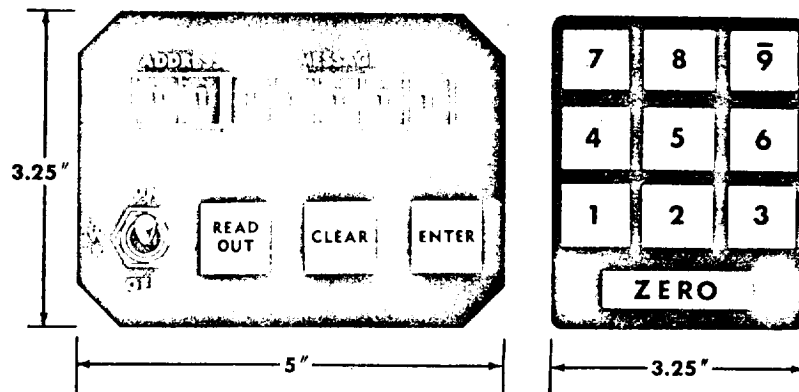
Touchdown

Prediction (TD - PRE) - In this mode the computer calculates and displays a predicted touchdown latitude and longitude based on a manually inserted trial retro-grade time.

Re-entry (RE-ENT) - The RE-ENT position provides calculations to control the flight path in order to achieve the desired touchdown point.

2.2.1.3 Auxiliary Computer Power Unit - The ACPU furnishes power as required to enable the computer to operate during short periods, 100 milliseconds maximum, of low spacecraft power. Also, the ACPU performs controlled shutdown of the computer if power is lost for more than 100 milliseconds.

MANUAL DATA INSERTION UNIT



CHARACTERISTICS

- INSERTION AND READOUT PANELS
- NON-ILLUMINATED

OPERATIONAL PERFORMANCE

- EXACT READOUT AND SIGNAL TRANSMISSION
- MEAN TIME BETWEEN FAILURE 3000 HOURS

PHYSICAL CHARACTERISTICS

- WEIGHT 4 LB.
- VOLUME 80 CU. IN.

POWER REQUIREMENTS

- 4 WATTS AVERAGE
- 8.15 WATTS MAXIMUM

MAC REFERENCE DOCUMENTS

- MAC REPORT 8637
- MAC SCD 52-87713
- MAC REPORT 8635

SUB-CONTRACTOR

- IBM

2.2.1.4 Manual Data Insertion Unit (MDIU) - The MDIU provides a means of manually inserting data into the computer and for a readout from the computer of addressed quantities. The unit accepts and displays decimal data and performs the required conversion to computer binary language. Display of stored computer data is accomplished by keying the desired address and subsequently depressing the READOUT button. Verification of manually inserted data is provided on the display unit prior to entry into the computer memory. If an error is noted, the CLEAR button is depressed and the keying process repeated. Negative data are inserted by the selection of the digit "9" as the first digit of the data. If the astronaut inserts or requests readout of an invalid address, inserts more than seven digits, inserts less than seven digits or fails to insert a two digit address prior to depressing the insert or readout button, the seven digit display becomes all zeros to indicate a procedural error. The readout information is displayed one digit at a time in .50 second intervals with the most significant digit of the address first. Insertion of MDIU data requires an estimated 1.45 seconds per digit (including human reaction time). The most significant address digit is entered first. The MDIU is located on the second astronaut display panel and serves the following functions:

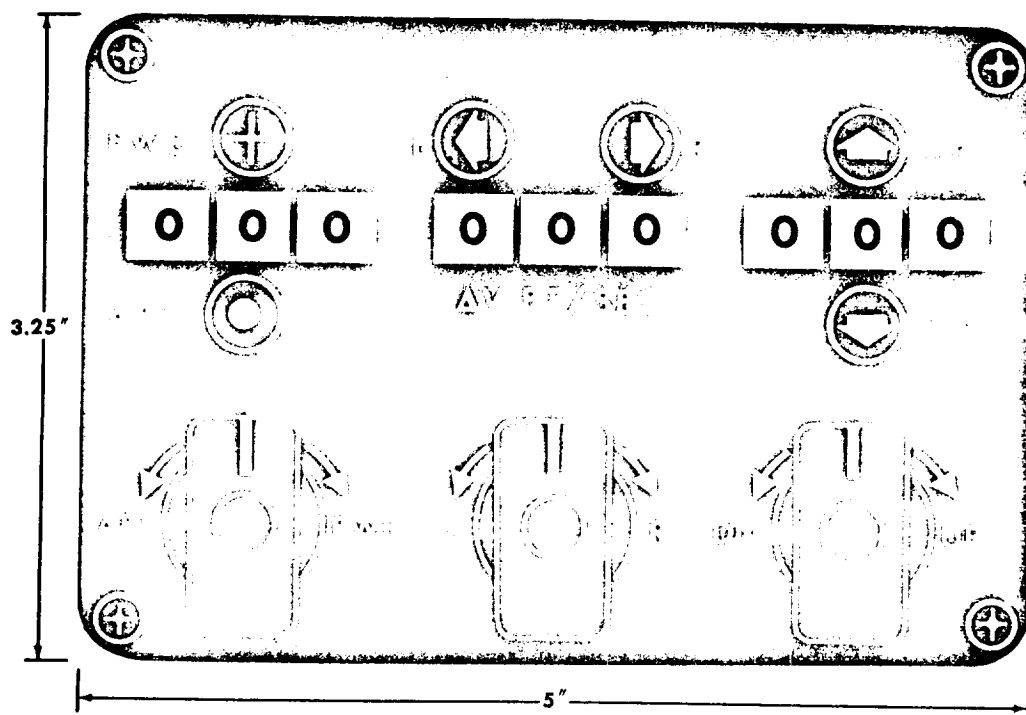
- | | |
|----------------|---|
| PRELAUNCH MODE | - Verification of DCS inserted data. |
| CATCH-UP | - Verification or insertion of the ground computed velocity maneuver. Back-up aid in accomplishing catch-up maneuver. |
| RENDEZVOUS | - Monitor selected quantities in the rendezvous program. |

TOUCHDOWN PREDICTION - Verification or insertion of initial condition data. Insertion of trial retrograde time. Readout of longitude and latitude of predicted touchdown point.

RE-ENTRY

- Verification or insertion of initial condition data for re-entry calculation. Insertion and display of time to go to retrograde.

INCREMENTAL VELOCITY INDICATOR



CHARACTERISTICS

- VELOCITY INDICATOR FOR EACH AXIS
- GREEN INDICATOR LIGHTS FOR THRUST DIRECTION

OPERATIONAL PERFORMANCE

- .5 FOOT/SECOND ACCURACY
- 000 TO 999 FEET/SECOND RANGE

PHYSICAL CHARACTERISTICS

- WEIGHT 3.75 LB.
- VOLUME 82 CUBIC INCHES

POWER REQUIREMENTS

- 7 WATTS AVERAGE
- 8 WATTS MAXIMUM

MAC REFERENCE DOCUMENTS

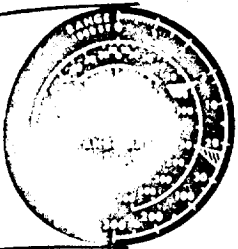
- MAC SCD 52-87712

SUB-CONTRACTOR

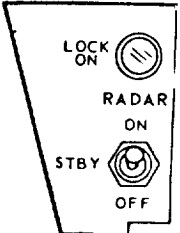
- LEAR/SIEGLER

2.2.1.5 Incremental Velocity Indicator (IVI) - The incremental velocity indicator located on the command astronaut display panel has three incremental velocity counters that display velocity increments along the body axes. The velocity components are displayed in a decimal format with a quantum size of 1 foot per second. The counters may be set manually by means of control knobs or set automatically by the computer. Applied velocity increments can only be obtained from the computer. The computer sets the counters to zero individually prior to the insertion and display of a computed velocity increment. When power is initially applied to the incremental velocity indicator it automatically references to zero. Power is applied to the indicator whenever the computer is turned on but IVI will not recognize computer signals for approximately 30 seconds until it has properly referenced its counters. The computer can reset and count-up the IVI at a maximum rate of 50 units per second while the manual control has four graduated rates up to the maximum rate. The function of the increment velocity indicators in conjunction with computer operational modes is:

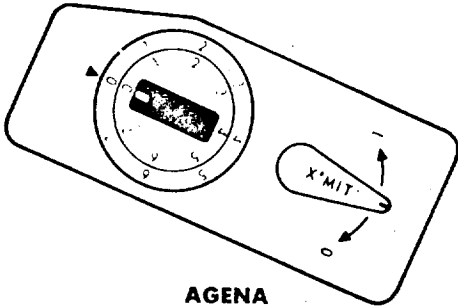
- ASCENT - Display and countdown of insertion velocity correction determined by the computer.
- CATCH-UP - Display and countdown of the catch-up velocity maneuver inserted via the DCS or MDIU.
- RENDEZVOUS - Display and countdown of velocity increment required to rendezvous as determined by the computer.
- RE-ENTRY - Measure of the velocity change experienced during retrograde maneuver.



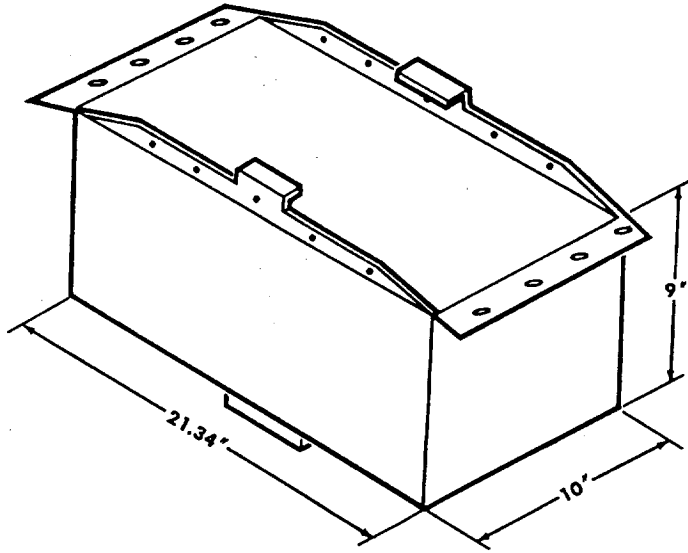
**RANGE-RANGE RATE
METER**



**RADAR CONTROL
PANEL**



**AGENA
CONTROL
PANEL**



TRANSPONDER

DISPLAYS

TRANSPONDER

CHARACTERISTICS

- TRANSPONDER AND COMMAND LINK
- ANTENNA SYSTEM
- BOOST REGULATOR

PERFORMANCE

- RESPONSE TIME SO THAT RADAR RANGE IS NOT COMPROMISED

PHYSICAL CHARACTERISTICS

- TRANSPONDER WEIGHT 22 LB.
- ANTENNA SYSTEM WEIGHT 4.0 LB.
- BOOST REGULATOR WEIGHT 2.5 LB.

POWER REQUIREMENTS

- MAXIMUM POWER, 41 WATTS DURING RECEIVE ONLY, 55 WATTS DURING TRANSMITTER OPERATION
- 28 VDC SOURCE

MAC REFERENCE DOCUMENTS

- MAC SCD 52-87715
- MAC REPORT 8637

SUB-CONTRACTOR

- WESTINGHOUSE

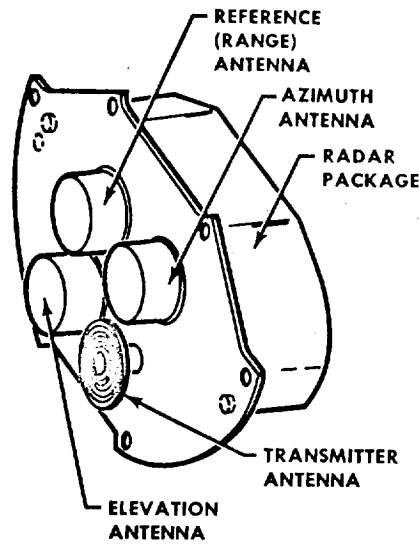
AND RANGE-RATE METER
LOCK ON LIGHT
CONTROL SWITCH
RANGE 300,000 FT
RATE FROM -100 TO +500 FT/SEC
LOCK ON INDICATED BY LIGHT

CHARACTERISTICS
RANGE-RATE INDICATOR
1.9 LB.

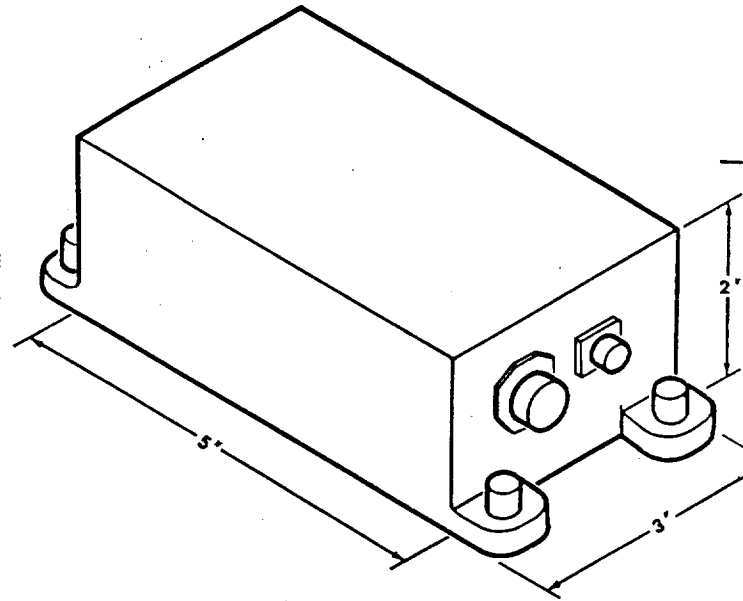
DOCUMENTS
52-87701
PORT 8635

OR
HOUSE

RADAR SYSTEM



RADAR



COMMAND LINK ENCODER

RADAR

CHARACTERISTICS

- L BAND INTERFEROMETER
- FOUR SPIRAL CIRCULARLY POLARIZED ANTENNAS
- 25 DEGREE (HALF ANGLE) BEAM WIDTH
- MAGNET ENCODER (FOR COMMAND LINK)

PERFORMANCE

- ANGLE TRACKING 250 N.M. TO <100 FT (± 3 MILLIRAD.)
- RANGE TRACKING 250 N.M. TO <20 FT (0.1% OR 50 FT WHICHEVER IS LARGER)

PHYSICAL CHARACTERISTICS

- WEIGHT 50 LB (ENCODER WT. 1.42 LB.)
- VOLUME 1.95 CU. FT.

POWER REQUIREMENTS

- MAXIMUM TOTAL POWER FOR RADAR 100 WATTS
- MAXIMUM TOTAL POWER FOR ENCODER 100 MILLIWATTS
- 1 WATT FROM 26 VOLT ($\pm 5\%$), 400 CPS ($\pm 1.5\%$)
- 7 VOLT 500 KCPS SQUARE WAVE CLOCK PULSE FOR DIGITAL OPERATIONS

MAC REFERENCE DOCUMENTS

- MAC SCD 52-87701
- MAC REPORT 8637

SUB-CONTRACTOR

- WESTINGHOUSE

CHARACTERISTICS

- RANGE
- RADAR
- RADAR

PERFORMANCE

- MAXIMUM
- RANGE
- RADAR

PHYSICAL CHARACTERISTICS

- RANGE
- WEIGHT

MAC REFERENCE DOCUMENTS

- MAC SCD
- MAC REPORT

SUB-CONTRACTOR

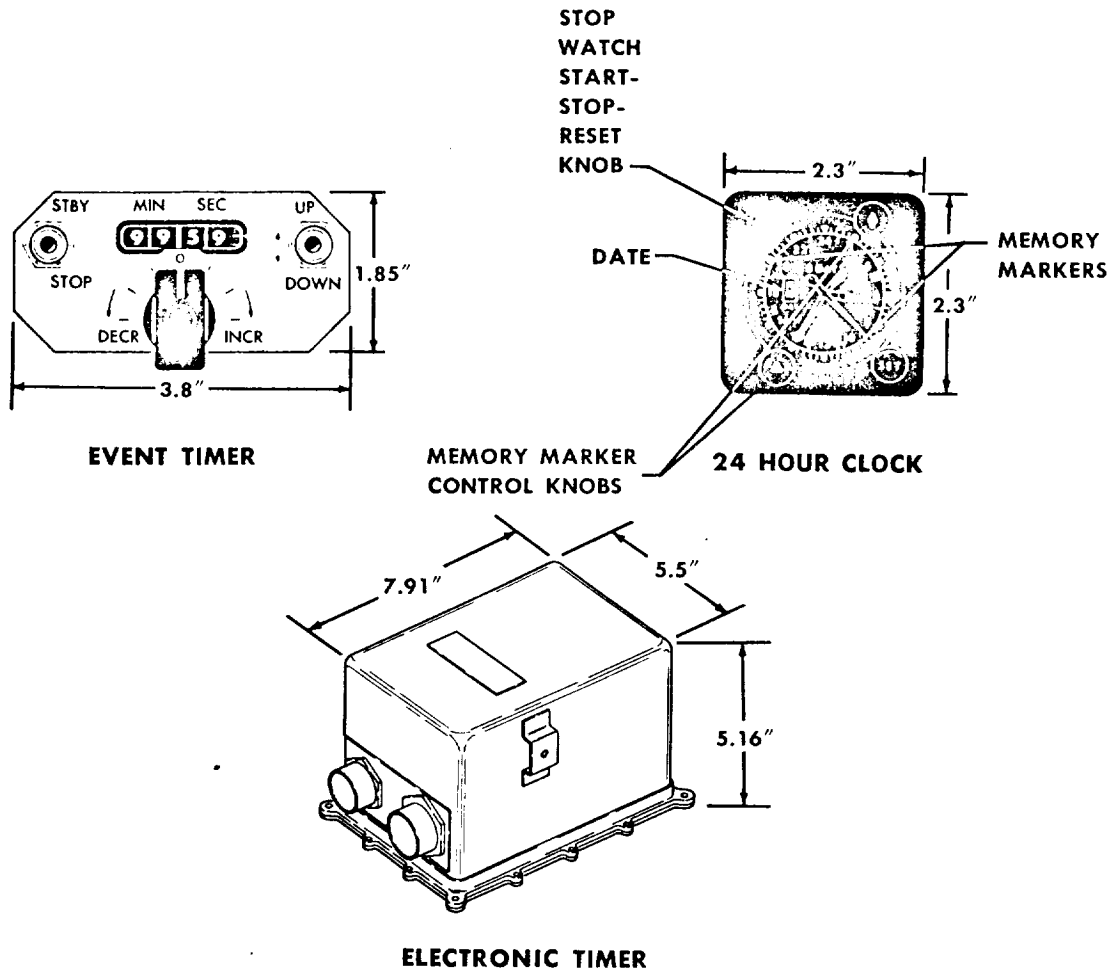
- WESTINGHOUSE

2.2.2 RENDEZVOUS RADAR - The radar supplies the IGS computer with digital range and angle tracking data in addition to servicing as a Command Link Encoder, and providing analog signals for the display of range, derived range rate, and angle tracking data. Four spiral circularly polarized antennas, one transmitting and three receiving, are used. The three receiving antennas consist of a common reference antenna and one each for the pitch and yaw planes. The phase difference between the reference antenna and pitch (or yaw) is nulled by an electrical phase shift produced by mechanical rotation of the corresponding antenna. The angle of antenna rotation is proportional to the sine of the pitch or yaw angle. Radar acquisition of the target vehicle is indicated by illumination of the lock-on light on the radar panel.

The command link transforms the Agena control switch positions into the proper format such that a waveform coder can modulate the radar to provide a "zero" or "one" signal. The target vehicle transponder returns a message verification signal that illuminates a "message received" light. Commands are sent via the radar link until the range separation is less than 25 feet, at which time the radar is turned off. After docking, the hardline (umbilical) is used.

Range and Range-Rate are displayed on separate circular scale type meter movements with both movements encased in the same cover and bezel. The range display is a logarithmic approximation, consisting of three linear segments of range while the range rate scale is approximately equal to the square root of range. A third meter movement is incorporated to indicate range rates of ± 5 ft/sec. The angle track data is displayed on the FDI when the reference switch is set to Radar position.

TIME REFERENCE SYSTEM



OPERATIONAL PERFORMANCE

- ELECTRONIC TIMER ACCURATE TO 35 PARTS IN 1,000,000 FOR A 24 HOUR PERIOD
- EVENT TIMER ACCURATE TO ± 1 SEC/HOUR
- CLOCK ACCURATE TO ± 2 SEC/HOUR

PHYSICAL CHARACTERISTICS

- ELECTRONIC TIMER WEIGHT, 10 LB; VOLUME 224.5 CU. IN.
- EVENT TIMER WEIGHT, 2 LB, VOLUME 30. CU. IN.
- 24 HOUR CLOCK WEIGHT, 1 LB, VOLUME 22.4 CU. IN.

POWER REQUIREMENTS

- INPUT POWER 18 TO 30.5 VDC
- MAXIMUM POWER 13 WATTS

MAC REFERENCE DOCUMENTS

- MAC REPORT 8635
- MAC REPORT 8664

CONTRACTOR

- MAC

2.2.3 Time Reference System (TRS) - The time reference system consists of an event timer, a clock, and an electronic timer (binary counter). The electronic timer provides the basic time reference for the computer guidance calculations. The electronic timer performs the function of counting up time or counting down time-to-go by an add/subtract program which is repeated every 1/8 second. Elapsed time and time-to-go data are stored as binary-coded words in magnetic shift registers. Elapsed time and time-to-go-to-retrograde, T_R , are stored in 24 Bit registers and have a 24 day capacity. The time-to-go-to equipment reset (Tx) register has an 18 bit capacity corresponding to approximately a 2 hour interval. The T_R and Tx can be updated by either the DCS or MDIU. Data forwarded via the DCS are binary coded 24 bit messages which are accepted directly by the electronic timer. The countdown data inserted via the MDIU must be ground/crew coordinated i.e. the ground station forwards time-to-go as a decimal number which is set on the MDIU and then entered into the computer on the "mark signal" (voice) from the ground. The decimal data are converted to binary coded message in the computer and forwarded to the electronic timer. To prevent inadvertent, premature countdown to zero of T_R as a result of personnel or equipment error during update, the timer will reject any new T_R which is less than 512 seconds. The timing functions provided by the electronic timer are summarized below:

COMPUTER

- Electronic timer provides the lift-off discrete for computer initiation of Ascent computations. Provides continuous countdown of time-to-go-to-retrograde.

SPACECRAFT SEQUENTIAL SYSTEM - The electronic timer provides the T_R -5 minute, T_R -30 second and the T_R discretizes to the sequential system in the retrograde sequence.

DATA ACQUISITION SYSTEM

- The electronic timer provides requested time information on elapsed time, and T_R .

COMMAND LINK ENCODER

- Timing pulses for message coding are provided by the electronic timer.

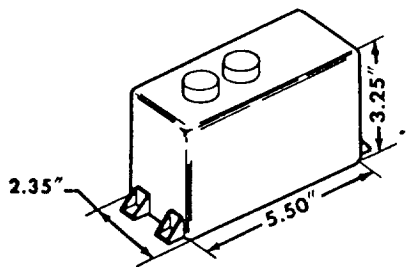
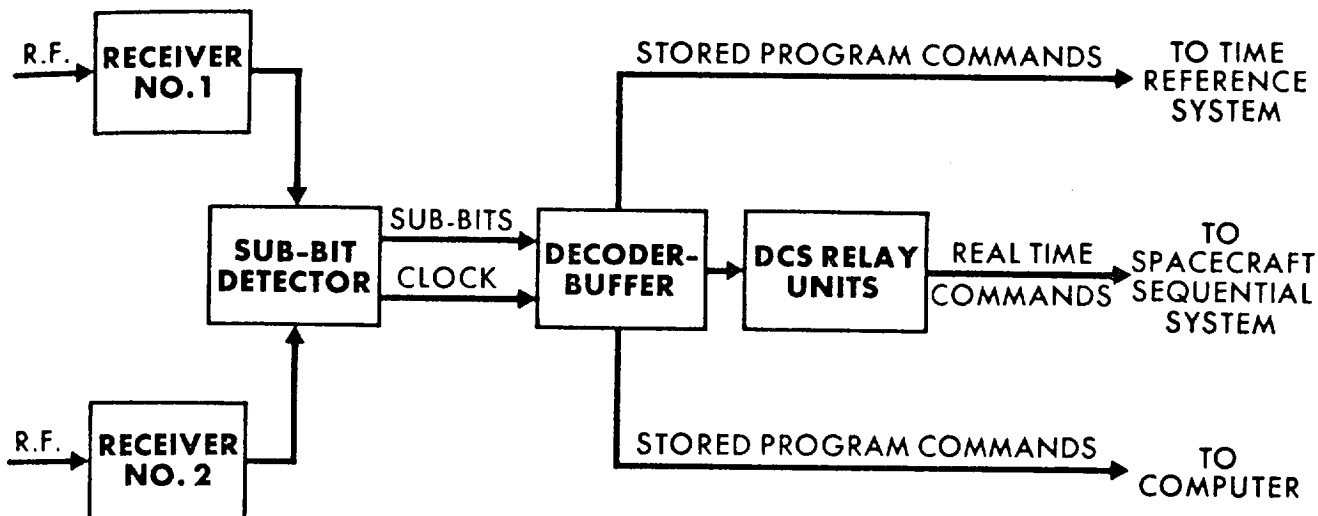
EQUIPMENT RESET SYSTEM

- Countdown of time-to-go-to-equipment reset is performed.

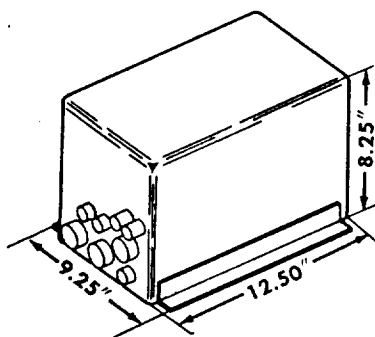
The event timer is located on the command astronaut panel and provides a time display in minutes and seconds for short-term timing functions, e.g. back-up countdown of retrograde time. Manual time insertion, countup and countdown controls are provided for the event timer.

The clock is a standard 24 hour display of Greenwich Mean Time which is modified to include a stopwatch movement and two separate settable index markers on the minutes portion of the dial face. Short term elapsed time can be obtained by direct reading of the stopwatch hands. Time-to-go countdown can be obtained by setting the index markers and observation of the stopwatch hands.

DIGITAL COMMAND SYSTEM



4 SECTION RELAY
UNIT (2 REQUIRED)



8 SECTION RELAY UNIT



DCS LIGHT

CHARACTERISTICS

- UHF TRANSMISSION LINK
- TRANSMISSION RATE 200 BIT/SEC
- CONSISTS OF: 1 RECEIVER-DECODER UNIT
3 EIGHT-SECTION RELAY UNITS
2 FOUR-SECTION RELAY UNITS

OPERATIONAL PERFORMANCE

- MEAN TIME BETWEEN FAILURES 16,800 HOURS

PHYSICAL CHARACTERISTICS

- WEIGHT 25 LB.
- VOLUME 1148 CU. IN.

POWER REQUIREMENTS

- 15 WATTS AVERAGE
- 20 TO 30.5 VDC

MAC REFERENCE DOCUMENTS

- MAC SCD 52-85714

SUB-CONTRACTOR: MOTOROLA

2.2.4 Digital Command System (DCS): The digital command system performs functions necessary to transfer the commands received from surface stations to the appropriate spacecraft systems. DCS commands are classified as real-time commands and stored program commands which are applicable for the computer and time reference system. The real time commands trigger relays which then cause a variety of spacecraft occurrences (displays, start-stop sequences, etc.) and a total of 64 commands are possible. The stored program commands to the TRS and computer are temporarily stored in a buffer and after decoding and identification, the appropriate register is signalled that new data exist. When the TRS or computer replies (by emitting a string of clock pulses to the DCS) the new word is inserted. Verification of receipt of message, proper message sub-bit coding, and request of message by the appropriate system is provided on real-time-telemetry and by illumination of the DCS light on the display panel. Each message consists of 175 sub-bits, the last 25 of which are rejected at the DCS. The sub-bits are arranged in a Barker-type code resulting in a decoded message of 30 bits: 3 bit vehicle address, 3 bit system address, and 24 bit data word. In the case of computer data words, a 6 bit address is included so that a total of 18 bits is available for sign and parameter magnitude. Transmission of DCS data to the computer can be made without regard to a particular computer mode except in those instances where the DCS data represents information required in the current computations. The nominal DCS functions in the various computer modes are:

- | | |
|------------|--|
| PRE-LAUNCH | - Loading of computer data to correct measured hardware errors. |
| ASCENT | - Loading of pre-launch targeting information, velocity updates, and abort landing site longitude and latitude for guided re-entry aborts. |
| CATCH-UP | - Velocity maneuver commands (magnitude) in navigational coordinates. |

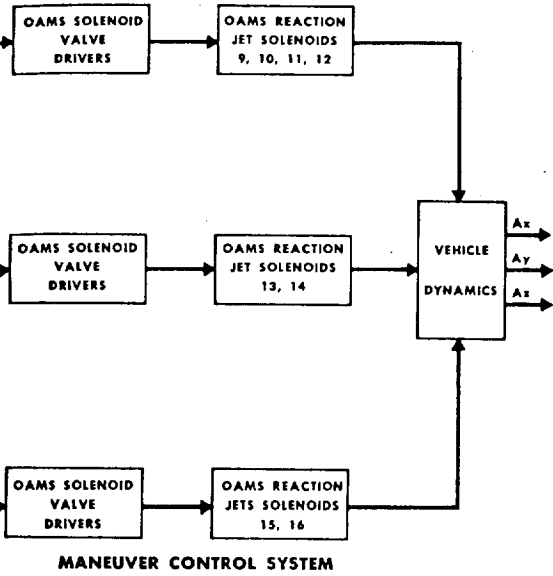
TOUCHDOWN PREDICT - Position, velocity, initial longitude, and data time tag information.

RE-ENTRY - Initial condition navigational data corresponding to the retrograde time and landing site longitude and latitude.

The stored program commands for the time reference system are:

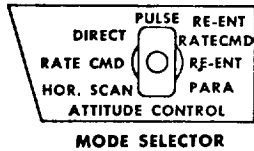
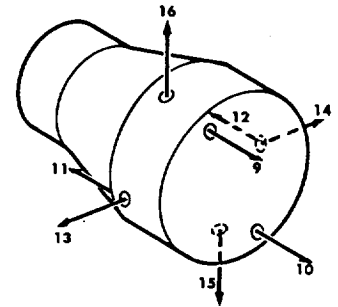
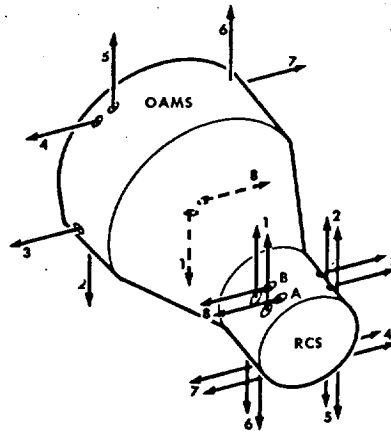
T_R - Time-to-go-to-retrograde with the limitation that $T_R > 512$ seconds.

T_X - Time-to-go-to-equipment-re-set.



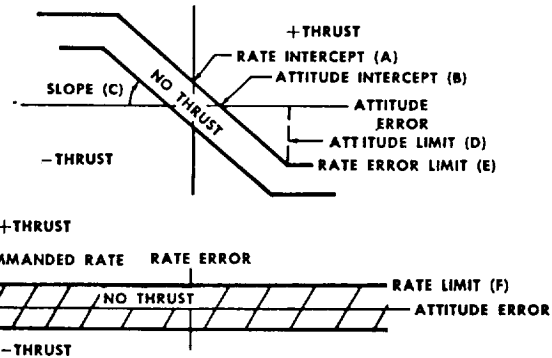
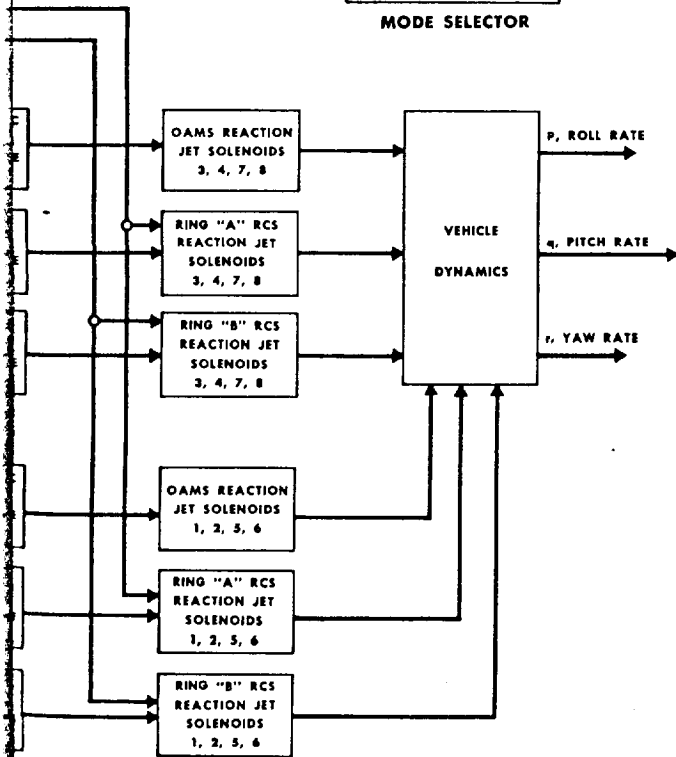
THRUSTER FIRING LOGIC

COMMAND	OAMS JETS FIRED		RCS JETS FIRED
	ROLL CONTROL	ROLL CONTROL	
	(YAW)	(PITCH)	
PITCH UP	5 AND 6	5 AND 6	5 AND 6
PITCH DOWN	1 AND 2	1 AND 2	1 AND 2
YAW RIGHT	3 AND 4	3 AND 4	3 AND 4
YAW LEFT	7 AND 8	7 AND 8	7 AND 8
ROLL RIGHT	3 AND 7	1 AND 5	3 AND 7
ROLL LEFT	4 AND 8	2 AND 6	4 AND 8
TRANSLATE FORWARD	9 AND 10	9 AND 10	
TRANSLATE AFT	11 AND 12	11 AND 12	
TRANSLATE RIGHT	13	13	
TRANSLATE LEFT	14	14	
TRANSLATE UP	15	15	
TRANSLATE DOWN	16	16	



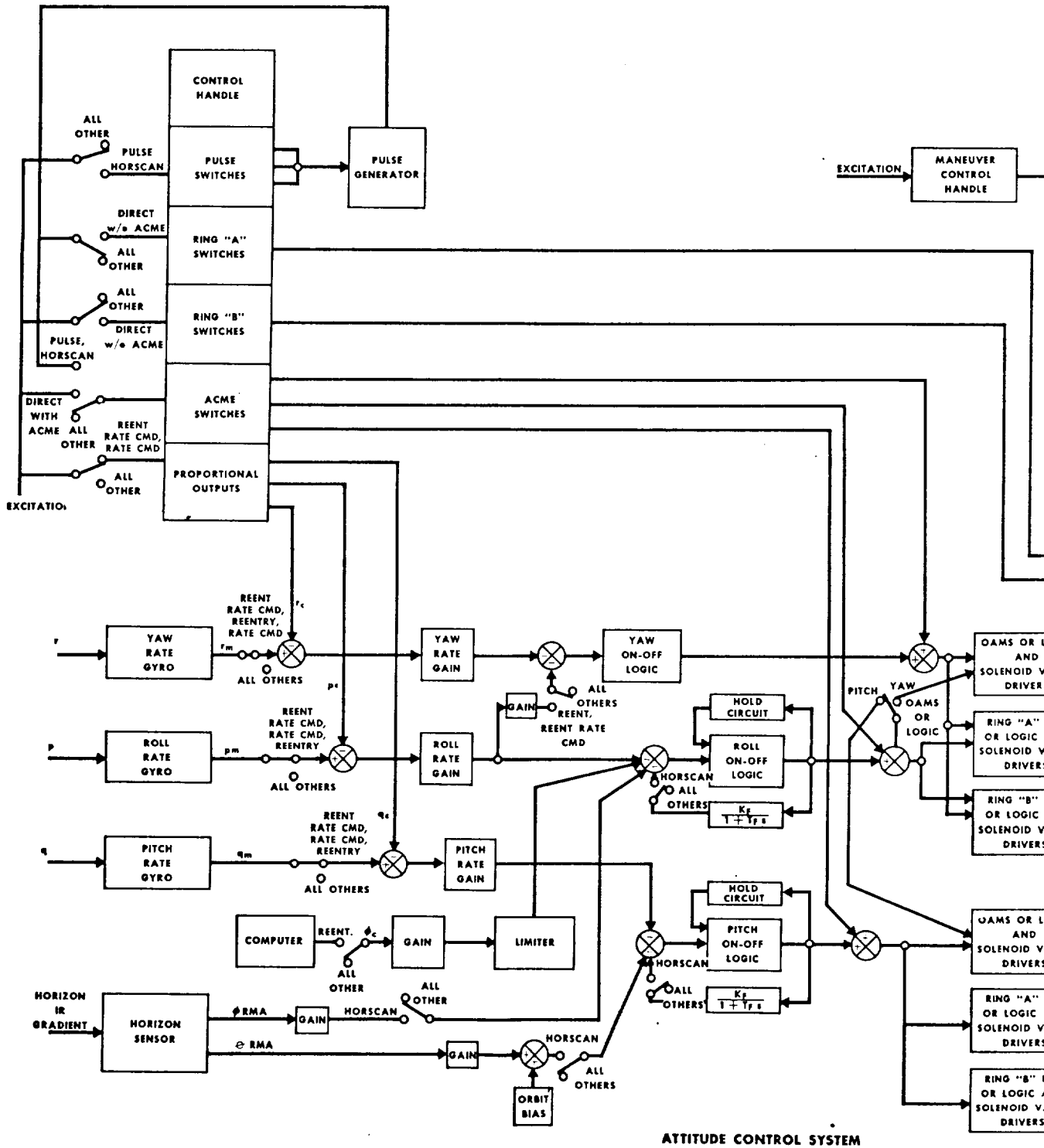
**ATTITUDE CONTROL THRUSTERS
25 LBS THRUST PER UNIT**

**OAMS MANEUVER THRUSTERS
100 LB THRUST PER UNIT
(EXCEPT FOR 85 LB FORWARD FIRING UNITS)**



CHAN- NEL	ATTITUDE MODE	A DEG/SEC	B DEG.	C SEC ⁻¹	D DEG.	E DEG/SEC	F DEG SEC
PITCH AND YAW	RATE CMD-OAMS						0.2
	RATE CMD-RCS						0.5
	REENTRY RATE CMD						4.0
	REENTRY-OAMS, RCS						4.0
ROLL	RATE CMD-OAMS						0.2
	RATE CMD-RCS						0.5
	REENTRY RATE CMD						2.0
	REENTRY-OAMS, RCS	2.0	2.0	1.00	15.0	13	

ATTITUDE AND MANEUVER CONTROL SYSTEMS FUNCTIONAL BLOCK DIAGRAMS



3. Control System - The control system provides for spacecraft orientation about its three major axes and for translational maneuvering. For the orbital phase, the OAMS thrusters are used for attitude and maneuvering, while during retrograde and re-entry, attitude control is provided by the RCS thrusters located on the small end of the spacecraft. The attitude control system has both automatic and manual modes while the maneuvering control system has only a single manual mode.

Maneuver Control Mode - Displacement of the maneuver controller beyond the preset threshold is sensed by switches which provide discrete signals to ACME which provides ON-OFF commands to OAMS thruster solenoid valves.

Attitude Control Mode Selector - The attitude control mode selector provides the following operational modes for attitude control.

Horizon Scan Mode - The horizon scan mode provides automatic control of the spacecraft about its pitch and roll axes during the orbital phase and is used in establishing an intermediate reference prior to the final platform alignment sequence. The horizon sensors provide a reference in pitch and roll to control a limit cycle mode in these axes. The capabilities of the pulse mode are maintained about all axes so that, if required, more precise control can be manually achieved. A nose down pitch bias is applied so that the nominal pitch attitude variation maintains the earth's horizon within the astronaut's field of view. The IGS and rate gyros are not used in the horizon scan mode.

Rate Command Mode - Vehicle angular rates are controlled such that they are proportional to the attitude control handle displacements. Rate gyro outputs are compared in the attitude control electronics (ACE) with the signals generated by control handle displacements and when the difference between the two signals exceeds the damping dead zone the proper reaction control jets are fired. The rate command mode is utilized during manually initiated attitude control operations and will be utilized during velocity change maneuvers to maintain attitude control. The rate command mode is effective about each of the three major spacecraft axes.

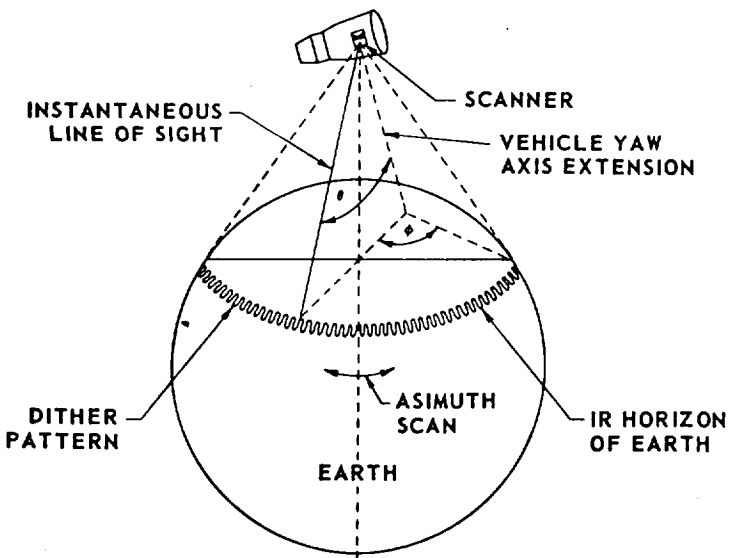
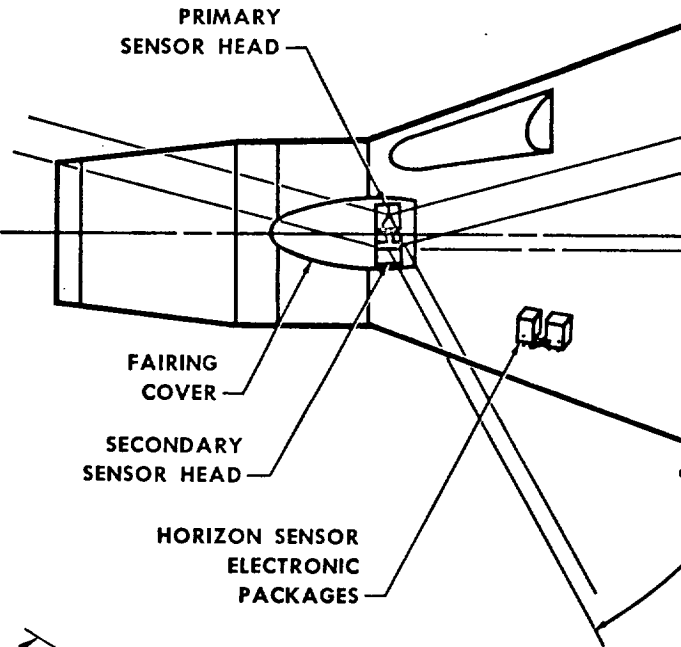
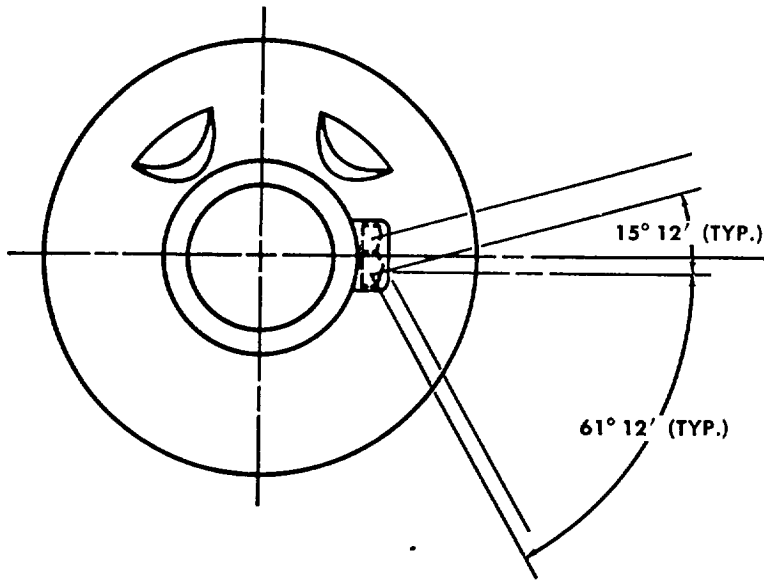
Direct Mode - The direct mode represents a backup control mode and is used to provide spacecraft angular acceleration. Switches activated by attitude control handle displacements provide discrete signals to the ACME which are converted into ON-OFF commands to the OAMS thrusters. The direct mode when used with RCS is instrumented by two independent methods. In the first method, switches on the attitude hand controller provide on-off commands directly to the RCS solenoid valves. In the second method, switches on the attitude hand controller provide discrete signals to the ACME which are converted into on-off commands to the RCS solenoid valves.

Pulse Mode - In the pulse mode spacecraft angular rate can be changed in incremental steps by commanding thrust for a calibrated period of time. After displacement of the control handle, it must be returned to the neutral position before another pulse can be commanded. The pulse mode is effective in each of the three major spacecraft axes and is used for fine attitude control, e.g. during platform alignment.

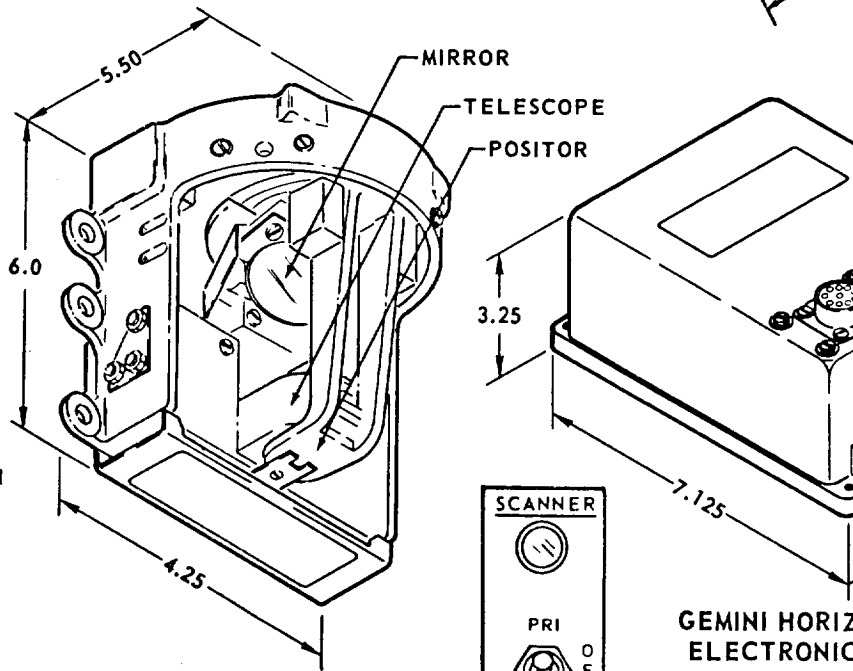
Re-entry Mode - In the re-entry mode, the spacecraft pitch and yaw rates are automatically maintained within the damping dead zone by the ACME. The instrumentation of the pitch and yaw axes control is identical to that utilized in the rate command mode except that commands are not accepted from the hand controller. In addition, the ACME utilizes the inputs from the computer to control the spacecraft about the roll axis. The computer input is a signal corresponding to either a roll attitude error or a fixed roll rate depending upon the relationship between the predicted touchdown point and the desired touchdown point. The ACME maintains the commanded roll attitude or rate within a deadband about the command value. The re-entry mode is normally utilized with the RCS thrusters.

Re-entry Rate Command - The re-entry rate command mode is utilized for manual re-entry attitude control and has identical operational characteristics as the rate command mode except that the damping deadbands correspond to the re-entry mode values and roll rate cross-feed is included in the yaw channel as in the re-entry mode.

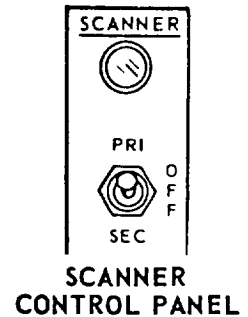
HORIZON SENSOR SYSTEM



HORIZON SENSOR SCAN PATTERN



GEMINI HORIZON SENSOR TRACKER HEAD



SCANNER CONTROL PANEL

GEMINI HORIZON SENSOR ELECTRONIC PACKAGE

OPTICAL CHARACTERISTICS

- A. DETECTOR:
- B. FIELD OF VIEW:
- C. SPECTRAL PASSBAND:
- D. FILTER:

GERMANIUM-IMMERSED THERMISTOR BOLOMETER
~1½° × 1½°
8-18 MICRONS
MULTI-LAYER, INTERFERENCE TYPE

PERFORMANCE CHARACTERISTICS

- A. INSTRUMENT NULL ACCURACY:
- B. SCALE-FACTOR ACCURACY:
- C. ALTITUDE RANGE:
- D. MAXIMUM ALLOWABLE TILT:

±0.1°
±7%
50-2,000 N.M.
20° FOR ALTITUDES FROM 50-850 N.M.

PHYSICAL CHARACTERISTICS

WEIGHT (LB):

- 1) TRACKER HEAD: 4.3 LBS.
- 2) ELECTRONIC PACKAGE: 5.6 LBS.
- 3) SYSTEM TOTAL: 9.9 LBS.

POWER REQUIREMENTS

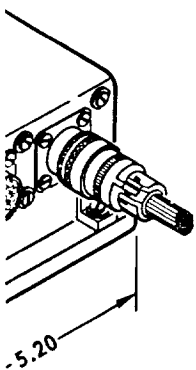
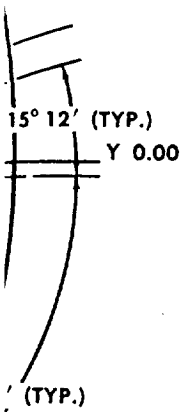
- 26V, 400CPS, MAXIMUM POWER 11 WATTS
- HORIZON SENSOR HEAD HEATER 15 WATTS AT 22VDC

MAC REFERENCE DOCUMENTS

- MAC REPORT D637
- MAC SCD 52-87703
- MAC DRAWING F-4786

SUB-CONTRACTOR

- ADVANCED TECHNOLOGY LABORATORIES

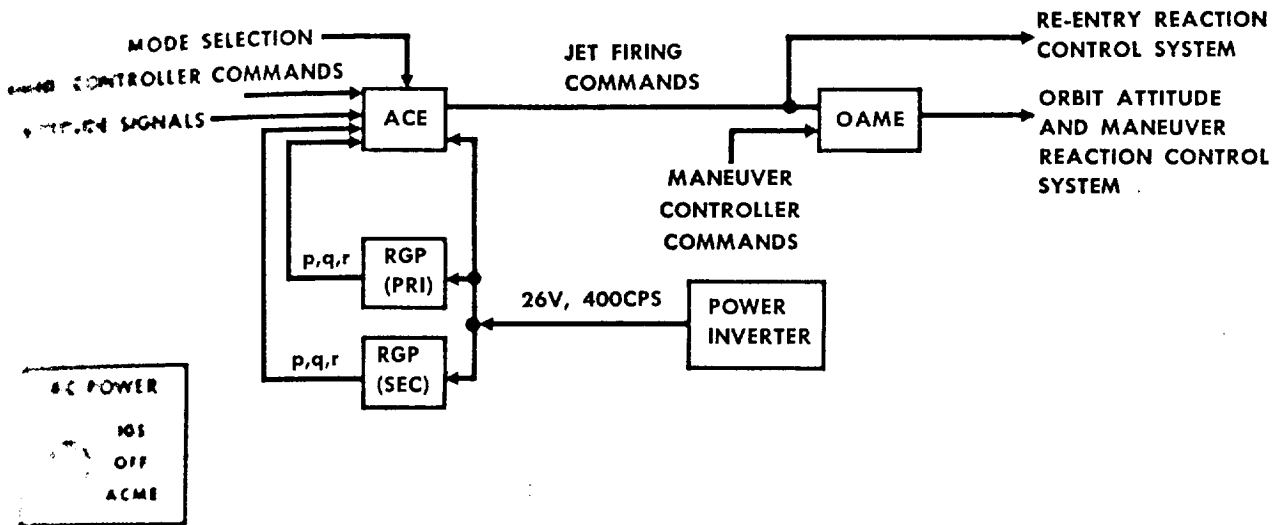


SENSOR
PACKAGE

3.1 Equipment - The control subsystem equipment consists of the horizon sensor system, the attitude control and maneuver electronics, the hand controllers and the displays necessary to pilot the spacecraft.

3.1.1 Horizon Sensor System - The horizon sensors use an azimuth scanning method ($\pm 85^\circ$) to detect the infrared (IR) gradient differential between the earth and atmosphere. The horizon sensor system converts the IR horizon information to analog signals proportional to spacecraft pitch and roll attitude errors with respect to the earth's local vertical. The pitch and roll outputs are electronically cross coupled to compensate for azimuth alignment offset in the horizon sensor head mounting on the spacecraft. The roll and pitch output signals are used by ACME for attitude control (horizon scan mode) and by the IMU for platform alignment (SEF, BEF).

Loss of track or lock-on drives the system into a search mode until lock-on is re-established; automatically switches the platform to an orbital rate mode during platform alignment; and illuminates the scanner light on the display panel. Two complete and separate horizon sensor systems, primary and secondary (standby) are provided and are selected by the scanner selector switch. The horizon sensor heads are jettisoned prior to re-entry.



CHARACTERISTICS

- CONSISTS OF ACE, OAME, POWER INVERTER, AND TWO RGP.
- OPERATIONAL IN 6 ATTITUDE CONTROL MODES, 1 MANEUVER CONTROL MODE
- PROVISION FOR 4 ADDITIONAL ATTITUDE MODES

OPERATIONAL PERFORMANCE

- POWER INVERTER PROVIDES OUTPUT OF 45 VOLT-AMP (0.8 LAG POWER FACTOR) AND 26 V \pm 2% AT 400 CPS
- RATE GYRO LINEARITY \pm 0.8 DEG/SEC, HYSTERESIS < 0.4 DEG/SEC, AND LINEAR ACCELERATION SENSITIVITY LESS THAN 0.04 DEG/SEC/g.
- EACH RATE GYRO HAS A SPIN MOTOR ROTATION DETECTOR

PHYSICAL CHARACTERISTICS

- TOTAL ACME WEIGHT 38.01 LB.
- WEIGHT OF COMPONENTS

ACE	15.93 LB.
OAME	7.20 LB.
RGP (TWO)	8.52 LB.
POWER INVERTER	6.36 LB.
- DIMENSIONS (AS SHOWN ABOVE)
- LOCATION OF RATE GYROS
BODY STATION 150

POWER REQUIREMENTS

- ACE AND OAME, 15 WATTS MAXIMUM
- POWER INVERTER, 55 WATTS MAXIMUM INPUT
- RGP, NORMAL POWER IS 15 WATTS, PEAK POWER IS 20 WATTS.

MAC REFERENCE DOCUMENTS

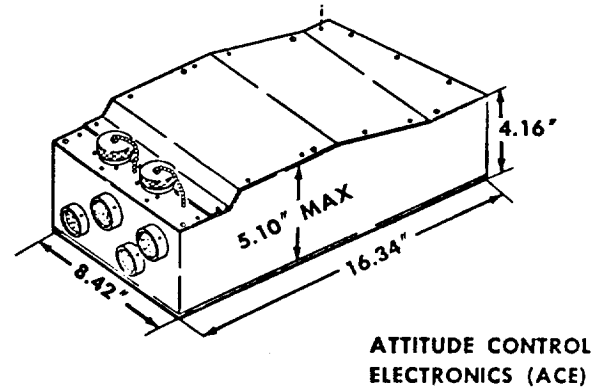
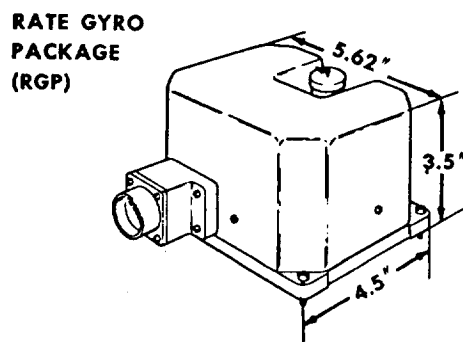
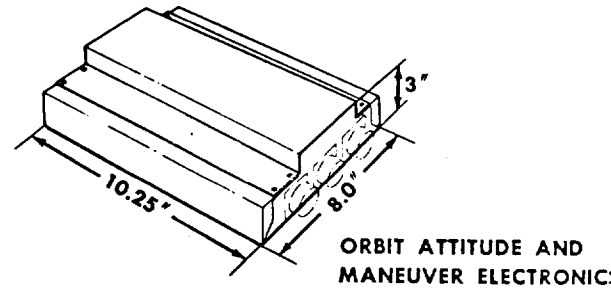
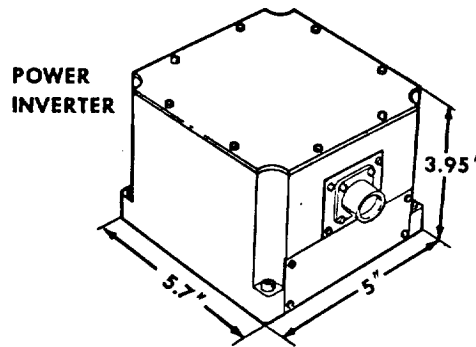
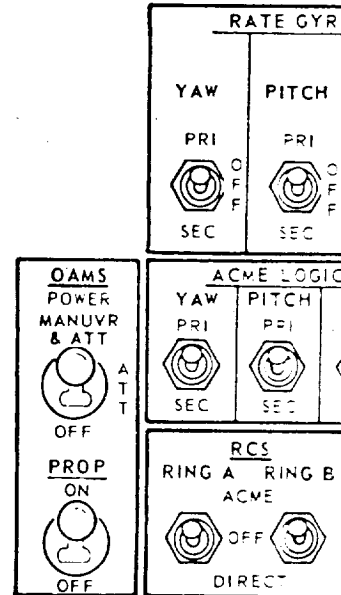
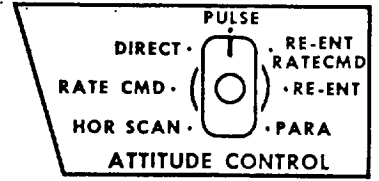
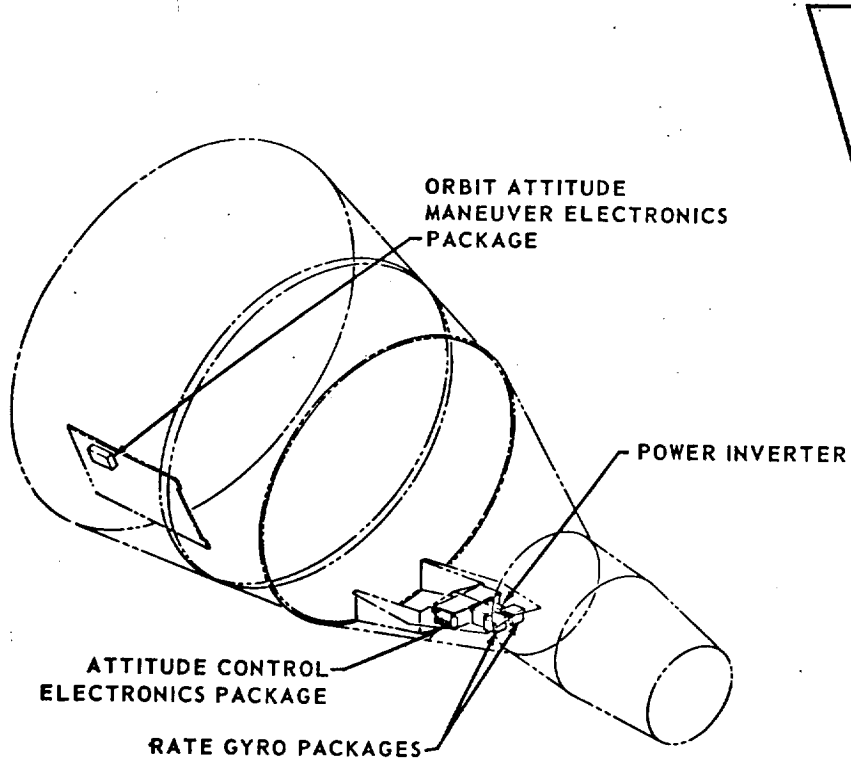
- MAC SCD 52-87700
- MAC REPORT 8637

CONTRACTOR

- MINNEAPOLIS-HONEYWELL

ATTITUDE CONTROL AND MANEUVER ELECTRONICS (ACME)

3.1



3.1.2 ATTITUDE CONTROL AND MANEUVER ELECTRONICS (ACME) - The ACME consists of the electronic circuits that implement the required signal processing in performance of the attitude and maneuver control tasks. Guidance and control inputs to ACME are provided by the digital computer, horizon sensors, and the attitude control and maneuver control handles. The ACME outputs are discrete on-off signals to control the firing of the thrusters. The subassemblies of ACME are defined below. The ACME equipment is located in the lower equipment bay except for the OAME which is located in the equipment section of the adapter.

ATTITUDE CONTROL ELECTRONICS (ACE) - The ACE accepts signals from the guidance subsystems, attitude control handle, rate gyros and horizon sensors, and converts the signals into drive commands to the RCS solenoids and on-off logic commands to the OAME subsystems. The ACE consists of the mode logic circuits, proportional circuits, control torque logic circuits, hold networks, pseudo rate networks, RCS valve drivers, and the pulse generator.

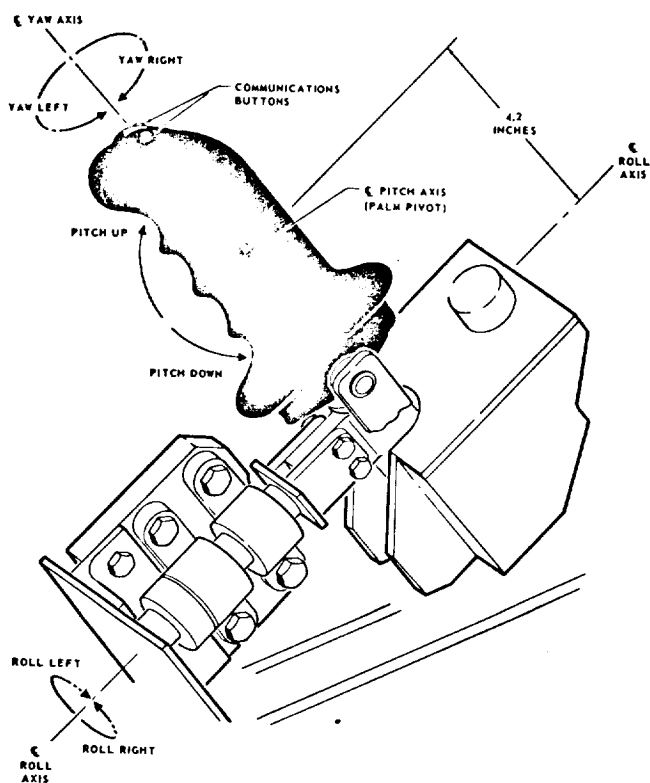
ORBIT ATTITUDE AND MANEUVER ELECTRONICS (OAME) - The OAME accepts signals from the ACE and maneuver controller for conversion to drive commands to the OAMS solenoids. The OAME consists of the attitude solenoid valve drivers, maneuver solenoid valve drivers, and OR logic circuits.

POWER INVERTER - The ACME power inverter converts spacecraft DC voltage into power used by the spacecraft whenever the IMU inverter is not being used. The AC power switch controls which power source that is utilized.

RATE GYROS - The two rate gyro packages (RGP) consists of three orthogonally mounted, single degree of freedom, spring constrained rate gyros which sense angular rates about the pitch, roll, and yaw axes of the spacecraft. All three gyros in each RGP are separately energized by switches on the astronaut control panel, thus providing standby rate sensing about each spacecraft axes. The rate gyros provide signals to the ACE when used in the rate command, re-entry rate command and re-entry modes, to the attitude display group, and to telemetry.

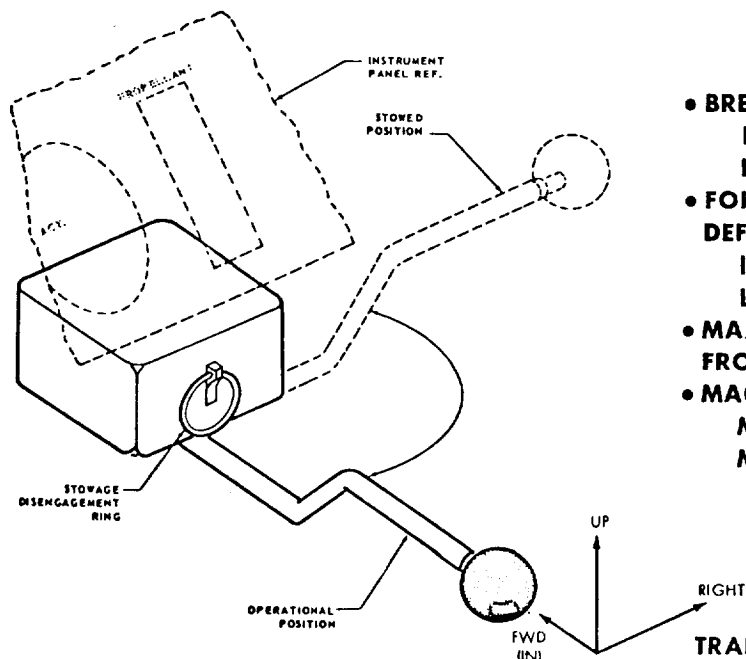
HAND CONTROLLERS

ATTITUDE CONTROL HANDLE
(SHROUD REMOVED FOR CLARITY)



- **BREAK-OUT TORQUE**
YAW 4.0 ± 0.2 IN-LB
PITCH 10.3 ± 1.0 IN-LB
ROLL 3.0 ± 0.2 IN-LB
- **DEFLECTION AT BREAK-OUT**
 0.5 ± 0.25 DEG.
- **TORQUE FOR MAXIMUM DEFLECTION**
YAW 12.7 ± 1.5 IN-LB
PITCH 29 ± 5 IN-LB
ROLL $6.45 \pm .85$ IN-LB
- **MAXIMUM DEFLECTION**
 10.5 ± 0.25 DEG.
- **COMMANDED BODY RATE PER GRIP ROTATION**
PITCH AND YAW, 1 DEG/SEC/DEG
ROLL, 1.5 DEG/SEC/DEG
- **GRIP ROTATION REQUIRED TO MAKE CONTROL HANDLE SWITCHES**
DIRECT MODE, 3 DEG
PULSE MODE, 4 DEG
- **MAC REFERENCE DOCUMENT**
MAC DRAWING 52-61365

MANEUVER CONTROLLER



- **BREAK-OUT FORCES**
IN-OUT AND UP-DOWN, $4 \pm \frac{1}{2}$ LB.
LEFT-RIGHT, $5\frac{1}{2} \pm \frac{1}{2}$ LB.
- **FORCE REQUIRED FOR MAXIMUM DEFLECTION**
IN-OUT AND UP-DOWN, $5\frac{3}{4} \pm \frac{3}{4}$ LB.
LEFT-RIGHT, $7 \pm \frac{3}{4}$ LB.
- **MAXIMUM KNOB DEFLECTION, 0.5 INCHES FROM BREAK-OUT POINT.**
- **MAC REFERENCE DOCUMENTS**
MAC DRAWING 52-61321
MAC DRAWING 52-61371

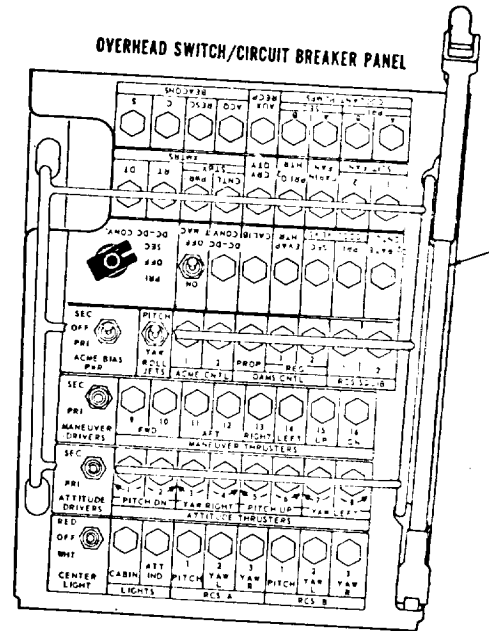
3.1.3 HAND CONTROLLERS - Manual attitude control and maneuver control tasks are accomplished utilizing the attitude control handle and the maneuver controller. Both controllers incorporate self-centering features and require slight forces for displacements thus providing the systems with "feel".

ATTITUDE CONTROL HANDLE - The attitude control handle is a stick grip device and provides control of spacecraft attitude about the pitch, yaw, and roll axes. Output signals to ACME are proportional to handle displacement when attitude control is in the rate command mode and are "ON-OFF" when direct or pulse control modes are employed. The control handle must be displaced past a preset threshold prior to command signal generation and, for the pulse mode, must be returned to the centered position in order to command subsequent pulse. The control handle is located between the astronauts on the crew station pedestal.

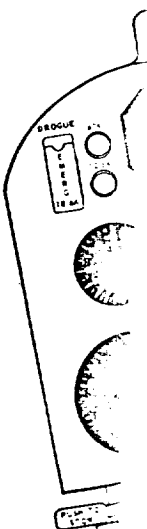
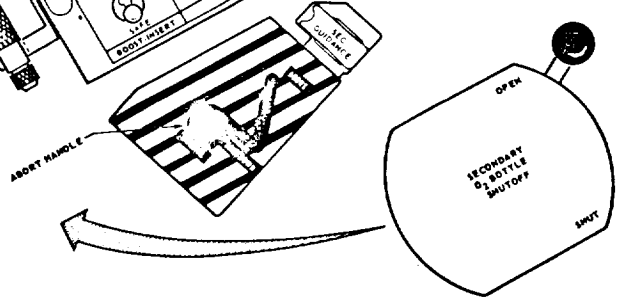
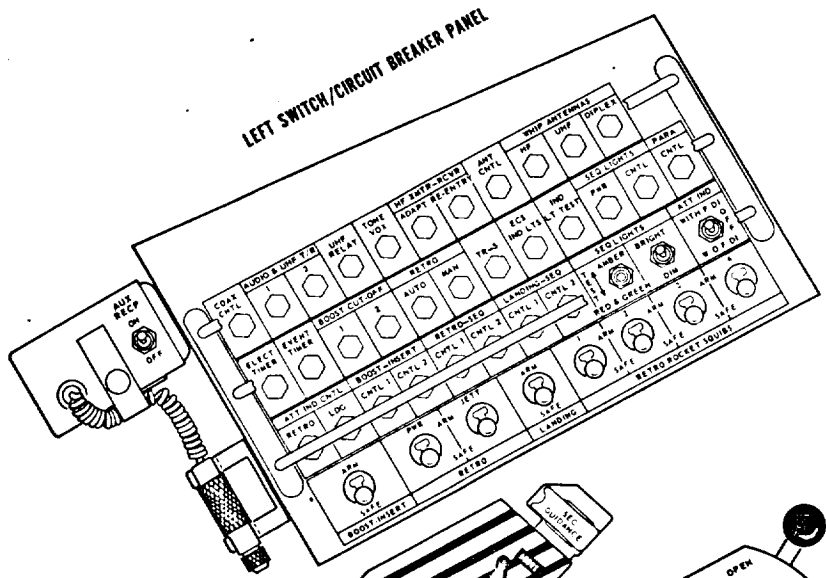
MANEUVER CONTROLLER - The maneuver controller, is a ball grip device and provides the means for manually controlling spacecraft maneuvering in any of six directions along the spacecraft axes. The maneuver controller is stowable and, when in the operating position, displacements in any of the six directions generate signals to ACME for control of thruster firing. Fore or aft, left or right, up and down controller displacements produce corresponding maneuver commands. The maneuver controller is located on the command astronaut display panel.

INSTRUMENTS, DISPLAYS, AND CO

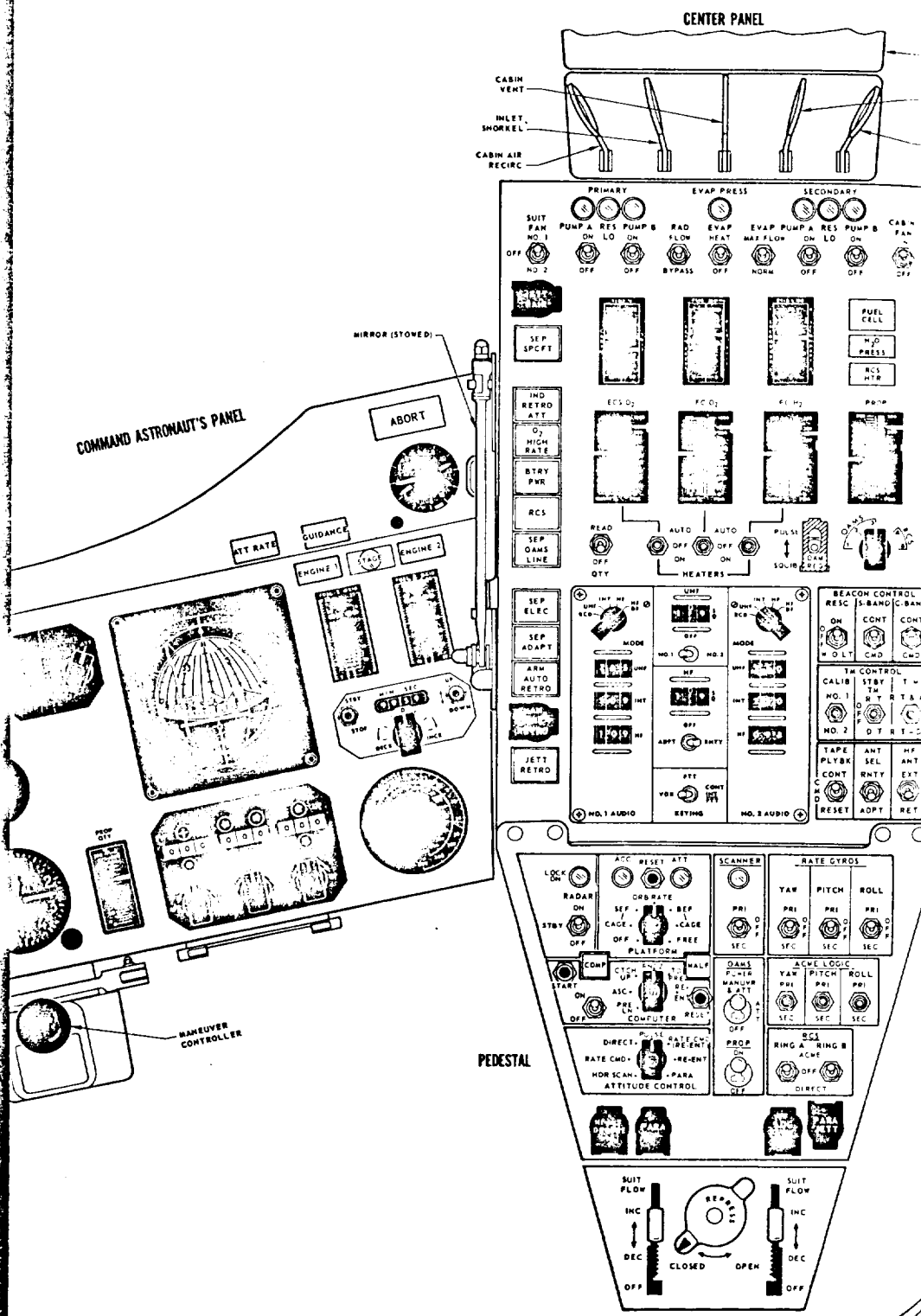
OVERHEAD SWITCH/CIRCUIT BREAKER PANEL



LEFT SWITCH/CIRCUIT BREAKER PANEL



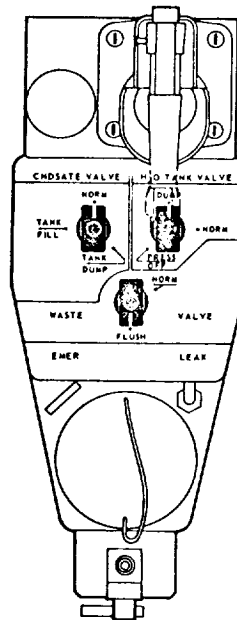
CONTROLS



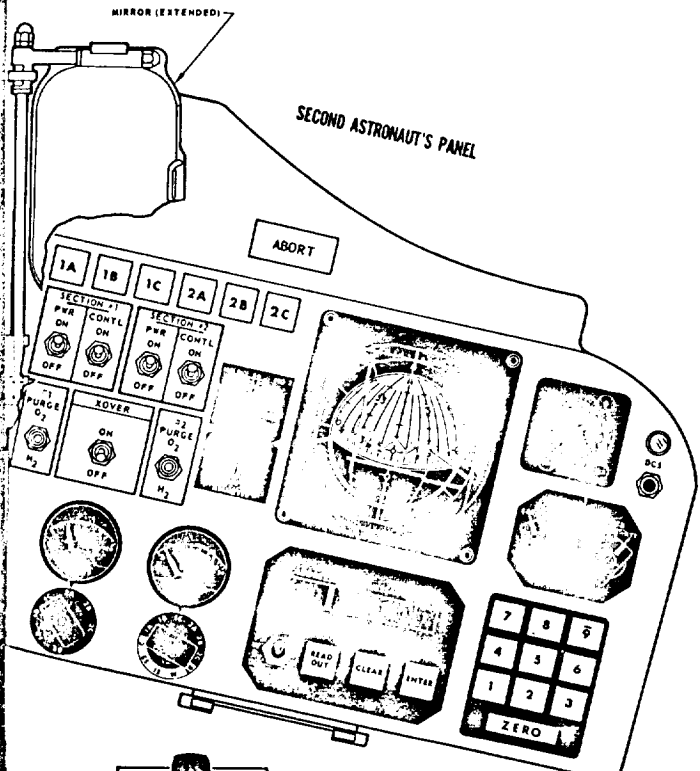
AGENA STATUS DISPLAY



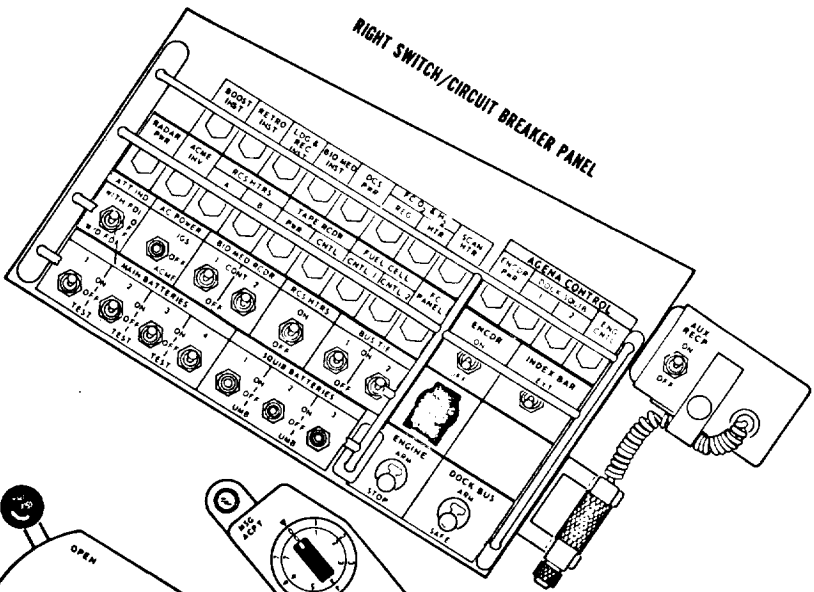
WATER MANAGEMENT PANEL



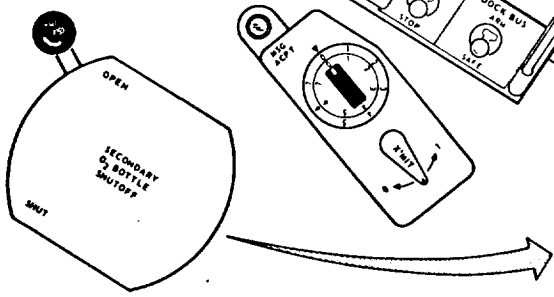
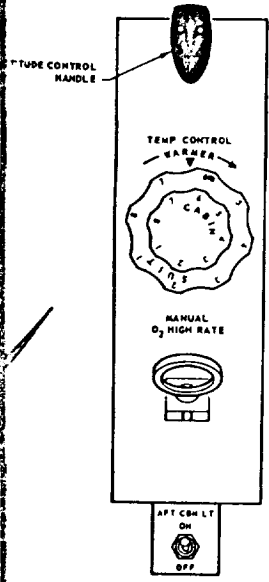
SECOND ASTRONAUT'S PANEL



RIGHT SWITCH/CIRCUIT BREAKER PANEL



CENTER CONSOLE



3.1.4 DISPLAYS - Instruments, displays and controls provided for visual indications, monitoring and control of various spacecraft functions are located on the panels and pedestal so as to be accessible and visible in accordance with recommended duties and responsibilities of crew members. The displays associated with the guidance and control equipments are discussed with the respective systems.

3.1.4.1 Agena Monitor Panel - The target mounted display panel monitors the operation of the Agena systems and will be visible during the docking maneuver.

Dock Light - Illuminated green when conditions are satisfactory for docking. Light is extinguished when rigidization is completed.

Rigid Light - Illuminated green when docking latches are closed and vehicles have been rigidized.

PWR Light - Green illumination indicates acceptable Agena electrical power system conditions.

Engine-Off Light - Red illumination indicates unsafe engine operation.

Main Light - Green illumination indicates acceptable main propulsion system conditions.

Armed Light - Amber illumination indicates primary or secondary Agena engines are armed.

Sec Hi Light - Green illumination indicates acceptable high thrust propulsion system conditions (Secondary System).

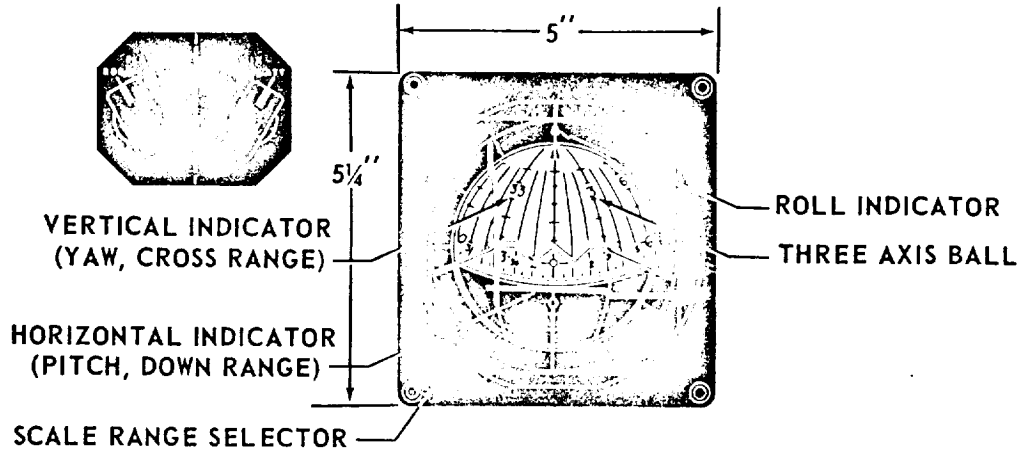
Sec Lo Light - Green illumination indicates acceptable low thrust propulsion system conditions (Secondary System).

Time (Main Engine) - Clock indicates burning time remaining on main engine in minutes and seconds.

Time (Secondary Engine) - Clock indicates burning time remaining on secondary engine in minutes and seconds.

Attitude Gas Meter - Color coded meter indicates attitude control gas supply pressure.

ATTITUDE DISPLAY GROUP

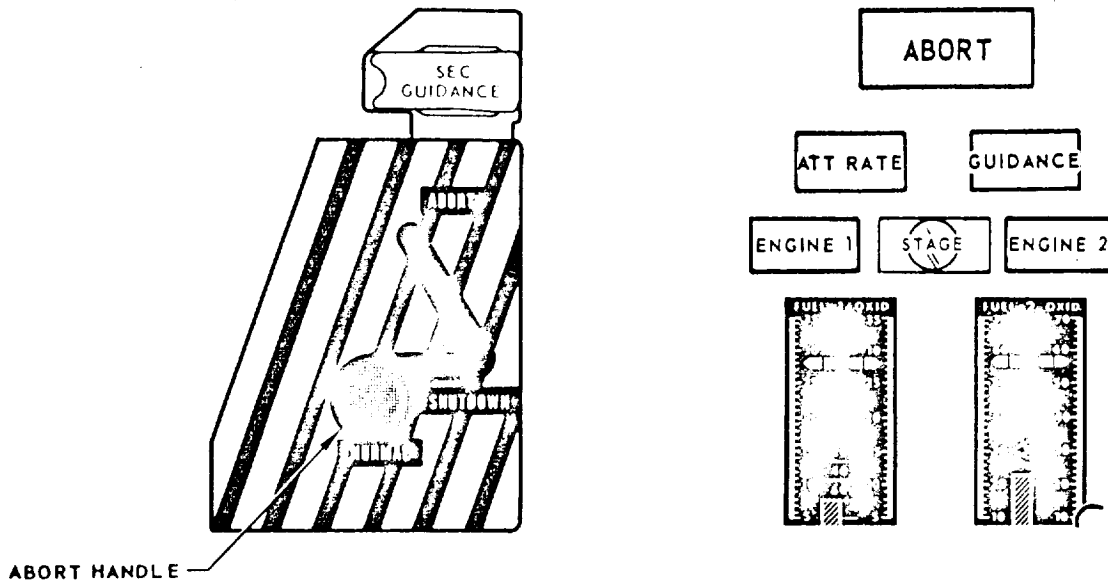


FLIGHT DIRECTOR MODE	FLIGHT DIRECTOR REFERENCE	FLIGHT DIRECTOR INDICATOR					
		ROLL INDICATOR		HORIZONTAL INDICATOR		VERTICAL INDICATOR	
		DISPLAY PARAMETER	RANGE **	DISPLAY PARAMETER	RANGE **	DISPLAY PARAMETER	RANGE **
RATE	RADAR	ROLL RATE	$\pm 5^\circ/S$	PITCH RATE	$\pm 5^\circ/S$	YAW RATE	$\pm 5^\circ/S$
	PLATFORM	ROLL RATE	$\pm 5^\circ/S$	↓	$\pm 5^\circ/S$	↓	$\pm 5^\circ/S$
	COMPUTER	ROLL RATE	$\pm 5^\circ/S$	↓	$\pm 5^\circ/S$	↓	$\pm 5^\circ/S$
	ASCENT CATCH-UP RENDEZVOUS REENTRY	↓ *	$\pm 5^\circ$ $\pm 5^\circ/S$	↓	↓	↓	↓
MIX	RADAR	PLAT ROLL ANGLE AND ROLL RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	ELEVATION ANGLE AND PITCH RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	AZIMUTH ANGLE AND YAW RATE	$\pm 5^\circ$ $\pm 5^\circ/S$
	PLATFORM	ROLL AND ROLL RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	PITCH AND PITCH RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	YAW AND YAW RATE	$\pm 5^\circ$ $\pm 5^\circ/S$
	COMPUTER	ROLL ERROR	$\pm 5^\circ$	PITCH ERROR & PITCH RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	YAW ERROR & YAW RATE	$\pm 5^\circ$ $\pm 5^\circ/S$
	ASCENT	ROLL RATE	$\pm 5^\circ$ $\pm 5^\circ/S$	↓	$\pm 5^\circ$ $\pm 5^\circ/S$	↓	$\pm 5^\circ$ $\pm 5^\circ/S$
	CATCH-UP	↓	$\pm 5^\circ$ $\pm 5^\circ/S$	↓	$\pm 5^\circ$ $\pm 5^\circ/S$	↓	$\pm 5^\circ$ $\pm 5^\circ/S$
	RENDEZVOUS REENTRY	↓ *	$\pm 5^\circ$ $\pm 5^\circ/S$	↓ DN RANGE ERROR AND PITCH RATE	$\pm 5^\circ$ $\pm 125 \text{ N.M.}$ $\pm 5^\circ/S$	↓ CROSS RANGE ERROR & YAW RATE	$\pm 5^\circ$ $\pm 25 \text{ N.M.}$ $\pm 5^\circ/S$
ATTITUDE	RADAR	PLAT. ROLL ANGLE	$\pm 5^\circ$	ELEVATION ANGLE	$\pm 5^\circ$	AZIMUTH ANGLE	$\pm 5^\circ$
	PLATFORM	ROLL ANGLE	$\pm 5^\circ$	PITCH ANGLE	$\pm 5^\circ$	YAW ANGLE	$\pm 5^\circ$
	COMPUTER	ROLL ERROR	$\pm 5^\circ$	PITCH ERROR	$\pm 5^\circ$	YAW ERROR	$\pm 5^\circ$
	ASCENT	ROLL ERROR	$\pm 5^\circ$	PITCH ERROR	$\pm 5^\circ$	YAW ERROR	$\pm 5^\circ$
	CATCH-UP	ROLL ERROR	$\pm 5^\circ$	PITCH ERROR	$\pm 5^\circ$	YAW ERROR	$\pm 5^\circ$
	RENDEZVOUS REENTRY	ROLL ERROR *	$\pm 5^\circ$	DN RANGE ERROR	$\pm 100 \text{ N.M.}$	CROSS RANGE ERROR	$\pm 12.5 \text{ N.M.}$

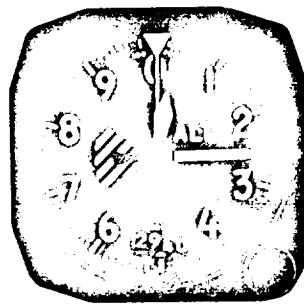
* MIXED S/C ROLL RATE WITH BANK ANGLE OR BANK RATE COMMAND.

** HIGH RANGE PROVIDES 15 UNIT DEFLECTION FOR ROLL QUANTITIES AND DOUBLE RANGE FOR ALL OTHER.

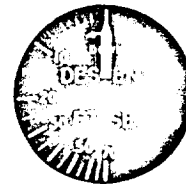
3.1.4.2 ATTITUDE CONTROL DISPLAY - The attitude display group provides the required reference information to enable the astronaut to achieve the desired spacecraft orientation. The attitude display group consists of a three-axis ball with superimposed flight director indicators. An attitude control display group is provided for each astronaut to permit him independent selection of the desired display mode. The three axis ball is driven by the IMU gimbal outputs and represents the spacecraft body angles relative to the platform orientation. The parameters displayed on the flight director indicators are controlled by the reference and mode switches. The reference switch selects the source of information supplied to the indicators. The mode switch selects the type information to be supplied to the indicators, i.e., attitude rates, attitudes, or attitude plus rate (mix).



LAUNCH VEHICLE DISPLAY GROUP



ALTITUDE



**RATE OF
DESCENT**



**AIRSPED
INDICATOR**



**LONGITUDINAL
ACCELERATION**

FLIGHT CONDITION GROUP

3.1.4.3 Launch Vehicle Display Group - These displays monitor the basic operational conditions of the launch vehicle in the areas of propulsion and guidance and control.

Fuel and Oxidizer Pressures - Dual indicators employing dual pointers monitor fuel and oxidizer pressures in stage I and II engines. The critical pressure areas are provided.

Engine Lights - Lights provided for stages I and II engines to indicate under pressure conditions. Lights are illuminated for under pressure condition; stage I red; stage II amber. Engine II light will normally be illuminated during stage I.

Stage Light - Amber illumination indicates that the staging command has been received. Light is de-energized at separation of stages I and II.

Attitude Rate Light - Red illumination is controlled by launch vehicle malfunction detection system and is indicative of an attitude overrate condition. The threshold rates are: stage I roll $20^{\circ}/s$, pitch $+3.5^{\circ}/s$ and $-4.0^{\circ}/s$ and yaw $\pm 3.5^{\circ}/s$; stage II roll $20^{\circ}/s$, pitch and yaw $10^{\circ}/s$.

Abort Light - Red illumination indicates that an abort has been commanded by flight conductor and/or range safety officer.

Guidance Light - Amber illumination indicates that the IGS has been selected for launch guidance functions.

Abort Handle - Three-position handle controls launch vehicle shutdown and spacecraft separation for Mode II aborts.

Guidance Switch - Two position momentary type switch that transfers guidance authority from the radio guidance system to the spacecraft IGS and also controls reselection of primary guidance system for a guidance switchback situation.

3.1.4.4 Flight Condition Group - A group of displays are provided for monitoring of the flight condition of the spacecraft. This group is applicable during the Ascent and Re-entry phases.

Altimeter - A standard aircraft altimeter with an effective range of -1000 to 80,000 feet is provided for monitoring altitude above sea level.

Airspeed Indicator - The airspeed indicator provides a visual indication of velocity during the landing phase.

Rate of Descent Indicator - The rate-of-descent indicator provides visual indication of vertical velocity during the landing phase.

Longitudinal Accelerometer - A longitudinal accelerometer with a range of -6g to 16g is provided to monitor accelerations experienced during Ascent and Re-entry phases.

SEQUENTIAL TELELIGHT/SWITCHES

	TIME OF ILLUMINATION OR DEPRESSION
JETT FAIRING	BECO +45 SEC
SEP SPCFT	SSECO +20 SEC
IND RETRO ATT	$T_R - 5 \text{ MIN}$
O ₂ HIGH RATE	$T_R - 5 \text{ MIN}$
BTRY PWR	$T_R - 5 \text{ MIN}$
RCS	$T_R - 5 \text{ MIN}$
SEP OAMS LINE	$T_R - 30 \text{ SEC}$
SEP ELEC	$T_R - 30 \text{ SEC}$
SEP ADAPT	$T_R - 30 \text{ SEC}$
ARM AUTO RETRO	$T_R - 30 \text{ SEC}$
MAN FIRE RETRO	T_R
JETT RETRO	$T_R + 45 \text{ SEC}$

BECO — BOOSTER ENGINE CUTOFF
SSECO — SECOND STAGE ENGINE CUTOFF
 T_R — TIME-TO-GO-TO-RETROGRADE

3.1.4.5 Sequential Telelight/Switches - Telelights, telelight/switches, and switches are provided for indication and/or manual initiation of various salient mission events. The units are arranged so that activation of the switches will occur in sequential fashion during a normal mission. The telelights are illuminated amber at the onset of the indicated action and become green upon completion of the task.

Fairing Jettison Switch - A push button switch is provided for activating pyrotechnic devices which eject the nose and scanner fairings.

Separate Spacecraft Telelight/Switch - The switch is pressed at SSECO + 20 seconds and to control squib fire relays which initiate pyrotechnic devices for disconnecting booster-spacecraft wiring, guillotining the booster wires, and removing the structural ties between the booster and spacecraft.

Indicate Retro Attitude Telelight/Switch - Illumination at $T_R - 5$ minutes serves as a signal to the astronaut to achieve the retro attitude. Depression of the switch applies a retro bias (16° nose down) to the FDI as an aid in maintaining the desired retro attitude.

O₂ High Rate Telelight Switch - The light is illuminated at $T_R - 5$ minutes as a signal that the high flow rate should be initiated for the re-entry phase. After $T_R - 5$ minutes, green illumination occurs when the secondary O₂ rate valve is opened.

Battery Power Telelight - Illumination at $T_R - 5$ minutes signals that power should be transferred from fuel-cell to silver-zinc source due to impending separation of the adapter equipment section. Green illumination occurs at any time all five silver-zinc battery switches are in the "ON" position.

RCS Telelight/Switch - Illumination at $T_R - 5$ minutes indicates that the re-entry control system should be activated. The switch

portion controls squib fire relays which initiate pyrotechnic activated isolation valves, permitting flow of RCS fuel and oxidizer into the system.

OAMS Lines Separation Telelight/Switch - Illumination at $T_R - 30$ seconds signals that the OAMS lines should be severed and sealed and the electrical wiring across the adapter separation plane should be disconnected. Pyrotechnic devices are activated by pressing the switch.

Electrical Separation Telelight/Switch - Illumination at $T_R - 30$ seconds serves as a signal that the electrical wiring connecting the adapter equipment and retrograde sections should be severed. Pyrotechnic guillotine devices are employed.

Separate Adapter Telelight/Switch - Illumination at $T_R - 30$ seconds indicates that the adapter equipment section should be separated. Shaped pyrotechnic charges are used to sever the adapter assembly at the equipment/retrograde section separation plane and subsequent separation of these sections is indicated by green illumination.

Arm Automatic Retro Telelight/Switch - Amber illumination at $T_R - 30$ seconds signals the astronaut that the switch must be actuated to provide a direct electrical path for application of the retrograde discrete ($T_R = 0$). The light becomes green when the arming circuit is closed.

Manual Fire Retro - A momentary contact push button switch is provided for manual firing of the retrograde rockets in the proper sequence.

Jettison Retro Telelight/Switch - Illumination at $T_R + 45$ seconds indicates that the retropack should be jettisoned preparatory to re-entry. Green illumination indicates separation of the retro-pack.

INSTRUMENTATION FUNCTIONAL BLOCK

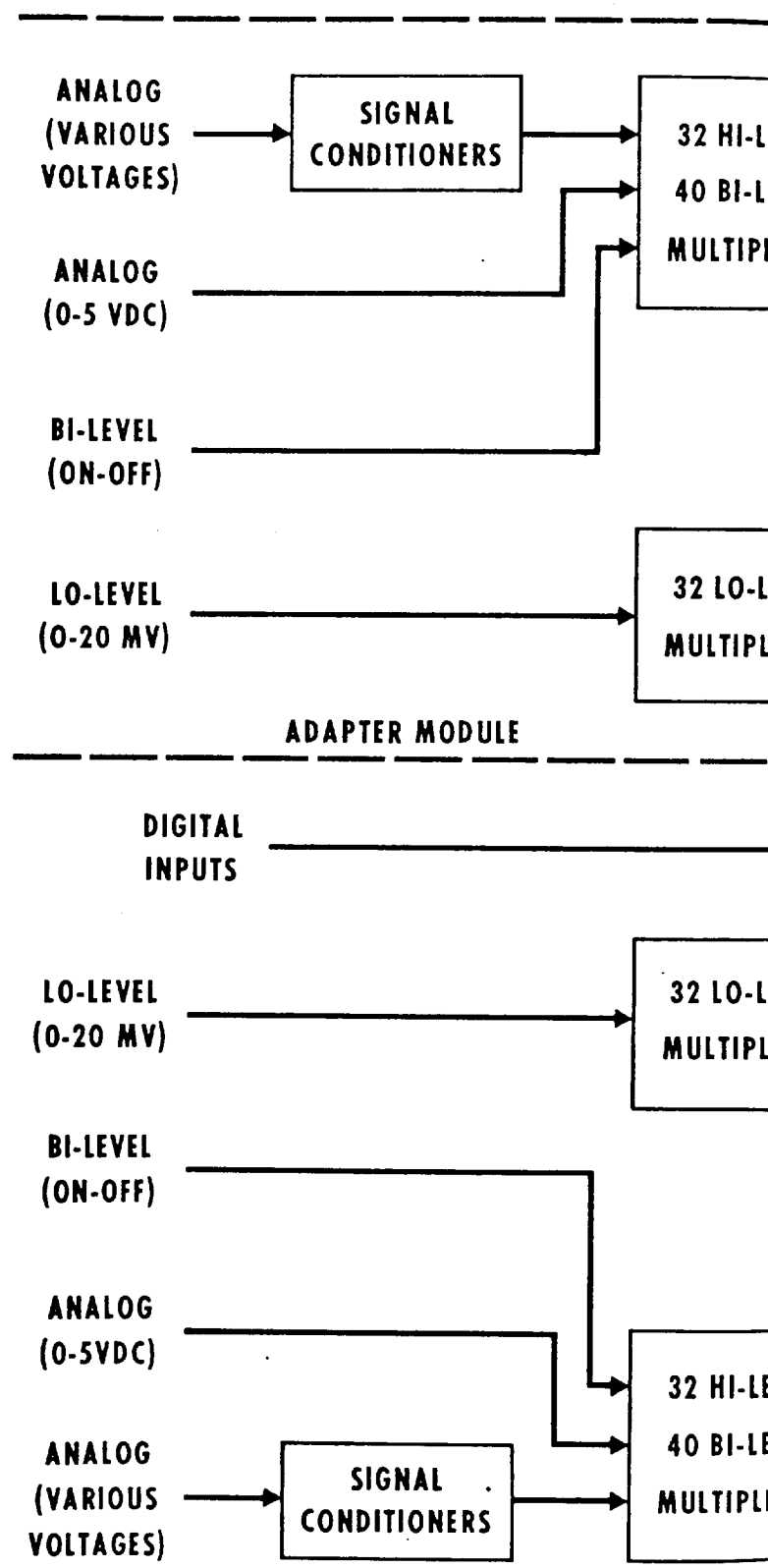
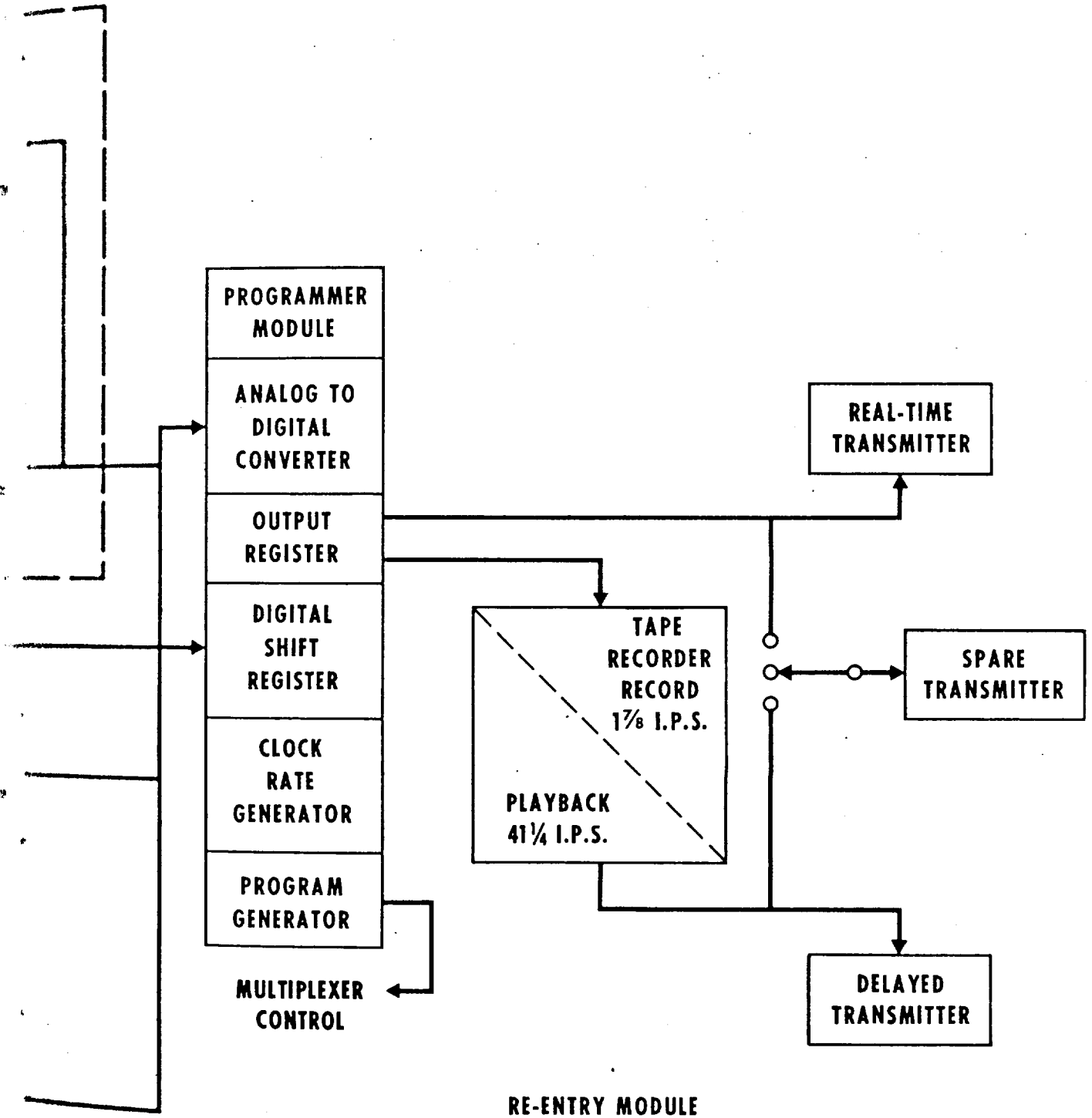


Figure 3

ITEM
ITEM



4.0 GUIDANCE AND CONTROL INSTRUMENTATION - The spacecraft instrumentation, or data acquisition system (DAS) is devoted exclusively to the data acquisition function and provides a means of documenting significant events and data throughout the entire mission by three methods: real-time transmission, data dump, and onboard recording. The real-time and data dump transmissions can be controlled by either the astronaut or the DCS. The real-time transmission system provides for transmission of all pulse code modulation (PCM) data while the spacecraft is in contact with a monitoring site at transmission rate of 51.2K Bit/sec. While the spacecraft is beyond the receiving capabilities of the ground stations, selected data is recorded on tape inside the spacecraft. When RF contact is made with the next ground station, a data dump transmission is performed, i.e. the telemetry data is transmitted from the tape at 22 times the recording rate. A functional block diagram of the instrumentation system is shown in Figure 3.

The guidance and control DAS parameters are provided in three forms: 24 bit words obtained from the digital computer; 8 bit words from T/M monitoring points; and bilevel, i.e. on-off indication, that are represented by specific bits of an assigned 8 bit word. The DAS information is transmitted to the ground sites in 8 bit groups (words) which requires that the ground site must reassemble the 24 bit information.

Parameters

1. $\Delta\theta$, $\Delta\psi$, $\Delta\phi$; Roll, Pitch, and Yaw Attitude Errors

2. ΔV_x , ΔV_y , ΔV_z ; Inertial Velocity Errors in x, y, and z Directions

3. SF_x , SF_y , SF_z ; Sum of x, y, and z Accelerometer Outputs

4. t ; Time in Computer Mode

5. Multiplex of all remaining DAS Computer Quantities 6 Per Sample

Ascent

SF_x ; Sum of x Accelerometer Output

SF_y , SF_z ; Sum of y or z Accelerometer Outputs

V_x ; Inertial Velocity in x Direction

V_y , V_z ; Inertial Velocity in y and z Directions

* $\Delta\theta$, $\Delta\psi$, $\Delta\phi$; Roll, Pitch, Yaw, and Roll Attitude Steering Errors

* t ; Computer Time Since Lift-Off

V_{zg} ; z Velocity Update (Azimuth)

P; Commanded GLV Pitch Rate

W_{yy} ; Yaw Rate Steering Command

x, z positions

y position

T_G ; Time to go to SSECO (IGS)

Catch-Up and Rendezvous

SF_x ; Sum of x Accelerometer Output

SF_y , SF_z ; Sum of y or z Accelerometer Output

* ΔV_{xo} , ΔV_{yo} , ΔV_{zo} ; Desired x, y, and z Body Velocity Change

θ_{bc} , ψ_{bc} ; Pitch and Yaw Attitude Command

0 to 26,000 Ft/Sec

±4000 Ft/Sec

0 to 26,000 Ft/Sec

±4000 Ft/Sec

± 6 Deg

0 to 600 Sec

±1000 Ft/Sec

-

-

+5 x 10⁶ Ft
R_E to 2.15 x 10⁷ Ft

-20 to 200 Sec

±300 Ft/Sec

±150 Ft/Sec

±300 Ft/Sec

+180 Deg

±10 Deg

GUIDANCE AND CONTROL
PERFORMANCE SUMMARY

REPORT NO. A344
30 NOVEMBER 1963
REVISED 15 MAY 1964

NAME: VAM (Contract) A. Y. ...

300 Ft/Sec

1.52 x 10⁶ Ft

+ .41

+ .41

0 to 8000 Sec

Touchdown
Prediction

*R_R, Radar Range

*Sin a₁, Sine Radar Elevation

**Sin r₁, Sine Radar Azimuth

TM, Time of thrust midpoint

CK1, CK2, CK4, CK5, CK7, CK8, Transfer Coefficients

R_{TRE}, Trial Retrograde Time

0-16, 2000 Sec

t, Time in Mode

φ, Computed Spacecraft Latitude

θ, Computed Spacecraft Longitude

ψ_E, Spacecraft Earth Referenced Heading

V_E, Spacecraft Earth Referenced Velocity

0 - 26,000 Ft/Sec

r, Radius to S/C From Earth Center

ṙ, Radius Rate

T_T, Time TAG of Integration Step

a_D, Total Drag Acceleration

0 - 22,000,000 Ft

0 - 10,000 Ft/Sec

0 - 16,200 Sec

0 - 300 Ft/Sec²

Re-Entry

SF_x¹, SF_y¹ Sum of x or y Accelerometer Output

SF_z¹ Sum of z Accelerometer Output

r_s Computed Distance to Earth Center

V_E¹ Spacecraft Relative Velocity

γ_E, Earth Referenced Flight Path Angle

0 to -16,000 Ft./Sec.

+ 3000 Ft./Sec.

0 - 22,000,000 Ft

0 - 26,000 Ft/Sec

0 - π RAD

0 - π/2

*R_N-R_p, Down Range Error

0 - ± 3,000 N.M.

*R_C, Cross Range Error

0 - ± 300 N.M.

B_C, Command Bank Angle

0 - + π RAD

t, Time in Mode

3600 Sec

φ, Computed Spacecraft Latitude

0 ± π/4 RAD

θ, Computed Spacecraft Longitude

0 - 2π RAD

R_p¹, Predicted Zero Lift Range

0 - 6,000 N.M.

D, Density Altitude

0 - 15 None

ψ_T, Heading to Target

0 - π RAD

R_T¹, Range to Target

0 - 6000 N.M.

ψ_E¹, Spacecraft Heading

0 - π RAD

*DENOTES SPACECRAFT INDICATION OR MONITORING CAPABILITY