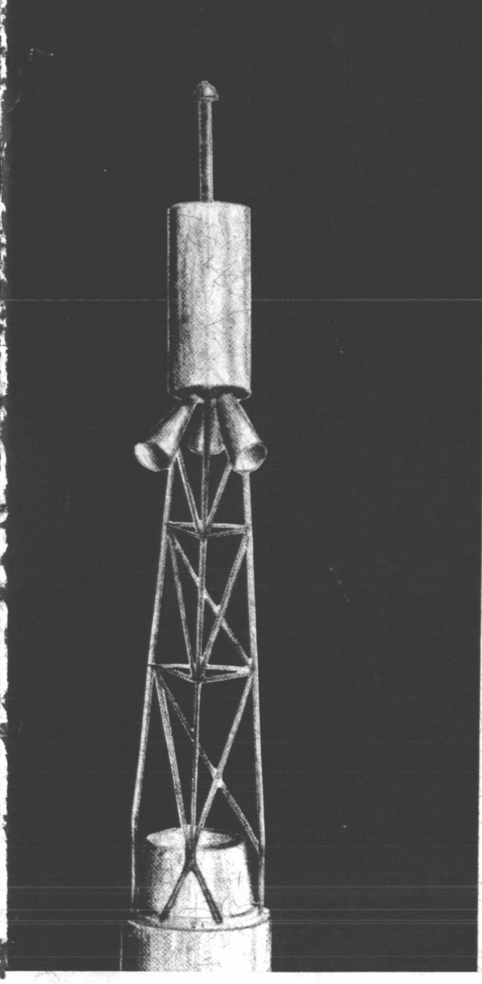
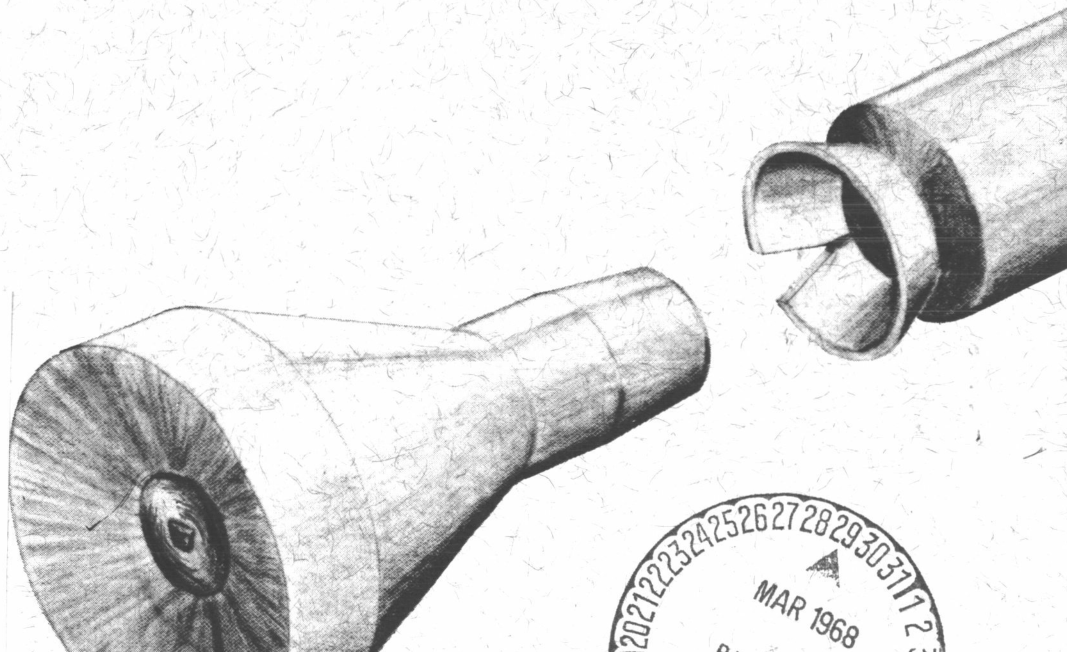
REPORT F917



MERCURY/GEMINI PROGRAM DESIGN SURVEY

NASA/ERC
DESIGN CRITERIA PROGRAM
STABILITY, GUIDANCE & CONTROL



MCDONNELL DOUGLAS CORPORATION

REPORT F917

31 JANUARY 1968

MERCURY/GEMINI PROGRAM DESIGN SURVEY

NASA/ERC Design Criteria Program
Stability, Guidance & Control

SUBMITTED UNDER CONTRACT NO. NAS 12–586

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MCDONNELL ASTRONAUTICS COMPANY

DIVISION OF MCDONNELL DOUGLAS CORPORATION

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FOREWORD

This Design Survey Report was prepared in accordance with the Statement of Work of NASA/ERC Contract Number NAS 12-586 by the McDonnell Astronautics Company, division of the McDonnell Douglas Corporation.

The report summarizes McDonnell's design experience in developing Stability, Guidance and Control Systems for the Mercury and Gemini programs.

Preparation of this report was by direction of C. V. Wolfers and H. L. Motchan. The effort was supervised and coordinated by E. L. Nieman, D. C. Hilty, and R. G. Malchow. Major sections were prepared by: D. L. Bradt, R. C. Brunnert, S. T. Dozier, P. H. Fultz, F. D. Hockett, C. E. Moyers, and J. W. Whiteside. Additional contributors of technical content were: J. E. Hallemann, F. P. Hercules, E. H. Johnson, D. J. Leemann, R. F. Pannett, H. H. Routberg, C. J. Scherrer, L. M. Warren, A. J. Wiegand, and F. H. Zengel. The above named engineers were either assigned to or closely associated with the engineering activity of the Mercury or NASA Gemini project.

The design survey was prepared for the NASA/ERC Design Criteria Office, directed by F. J. Carroll. Guidance on contents, format, and organization was provided by Curtis H. Spenny of ERC, with technical assistance from Ronald Madigan.

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1.0 Introduction

1.1 General Description - This report was prepared for the National Aeronautics and Space Administration (NASA), Electronics Research Center (ERC), Cambridge, Massachusetts. It relates the Mercury and Gemini program design experience of McDonnell Astronautics Company (MCASTRO), division of McDonnell Douglas Corporation, to the NASA/ERC Design Criteria Program for Stability, Guidance, and Control (SG&C). Technical data has been assembled for Mercury and Gemini guidance and control equipments and their system integration, to assist in the production of design criteria monographs that will facilitate unification of design approaches by NASA/ERC.

The success of the Mercury and Gemini programs has been well documented. However, problems had to be overcome to achieve this success. This report documents significant problems, particularly those related to stability, guidance and control, and the evolution of the equipment design.

The report has the following objectives:

a. To identify important SG&C design considerations and problems encountered in the design, development, test, and flight of space vehicles.

b. To discuss techniques of corrective action for problems encountered.

c. To note performance and reliability constraints imposed by mass property limitations.

d. To reveal limitations of specific design techniques.

e. To relate special factors required for a final design that complies with performance, reliability, and environmental requirements.

f. To demonstrate the type of comprehensive pre-flight testing necessary to ensure flightworthiness.

g. To present special design considerations applicable to space environments.

1.1 (Continued)

h. To provide information on research and development approaches taken where techniques were not readily available.

i. To illustrate the potential of contributions resulting from related past experience.

j. To yield basic information required for equipment design criteria checklists.

k. To reflect the effects of state-of-art constraints on program scheduling.

In formulating the design survey, McDonnell reviewed reports, design notes, technical notes, in-house correspondence, and subcontractor control documents to identify significant SG&C design considerations and problems. This data was supplemented by discussions with project personnel. Emphasis was placed on gathering data on the following specific items:

a. Environmental and performance requirements as related to hardware design.

b. Trade-offs and design decisions.

c. Failures, malfunctions, and near-miss incidents.

d. Unique problems encountered, and how they were solved or circumvented.

e. New concepts and hardware that evolved.

f. Major design changes and design evolution.

g. Suggested changes, "if one had it to do over again."

h. Unforeseen performance restrictions.

This survey documents the nature of each problem considered, the circumstances under which it occurred, the technology required to solve it, the design considerations and data involved in decisions, and the final solution, whether it be redesign, institution of procedures to circumvent the problem, or, in some instances, lessons learned or solutions that were not implemented.

1.1 (Continued)

The survey has been primarily directed toward the Gemini program because of the complexity of its equipment as compared to the Mercury equipment; however, related Mercury experience is reflected in the Gemini design.

All design survey material is cataloged chronologically to reflect the design evolution process through design, development, test, and flight. In some cases explicit design criteria have been extracted and are presented.

Some of these criteria may appear to be too well known to bear repeating in this type of document. However, we have included them here as a convenient, unified reference for future designers.

Sections 2 through 5 are devoted to equipment design. In Section 6, design criteria associated with equipment interface or integration aspects are reported, together with design data on SG&C systems. Section 7 collects conclusions, based on our experience, applicable to all SG&C and spacecraft design. The references listed in Section 8 contain further detailed information on the operation and requirements of the equipment discussed in this report.

1.2 System Descriptions

1.2.1 Mercury Spacecraft Stabilization Control System - The Mercury Spacecraft Stabilization Control System was designed to provide automatic stabilization and orientation of the capsule continuously from booster separation until landing parachute deployment. These requirements were accomplished by the Automatic Stabilization Control System (ASCS) in conjunction with two subsystems, the Horizon Scanners and the Reaction Control System. In addition, manual capability was provided to increase reliability and system flexibility, and an independent Rate Stabilization and Control System (RSCS) was provided for backup rate damping. The system is best understood by considering the operational requirements demanded of it. These requirements were:

(a) Provide rate damping in early abort situations.

(b) Provide rate damping and vehicle orientation for high altitude and orbital aborts.

(c) Maintain orbit orientation for a 28-hour period.

(d) Align and maintain vehicle attitude during retrofire.

(e) Place the vehicle in the reentry attitude following retrorocket firing.

(f) Provide rate damping during reentry, after sensing 0.05 g longitudinal acceleration, and provide a steady 10°/sec roll rate thereafter until parachute deployment.

Automatic Stabilization Control System - The Automatic Stabilization Control System (ASCS), shown in block diagram form in Figure 1.2-1, employed the following components:

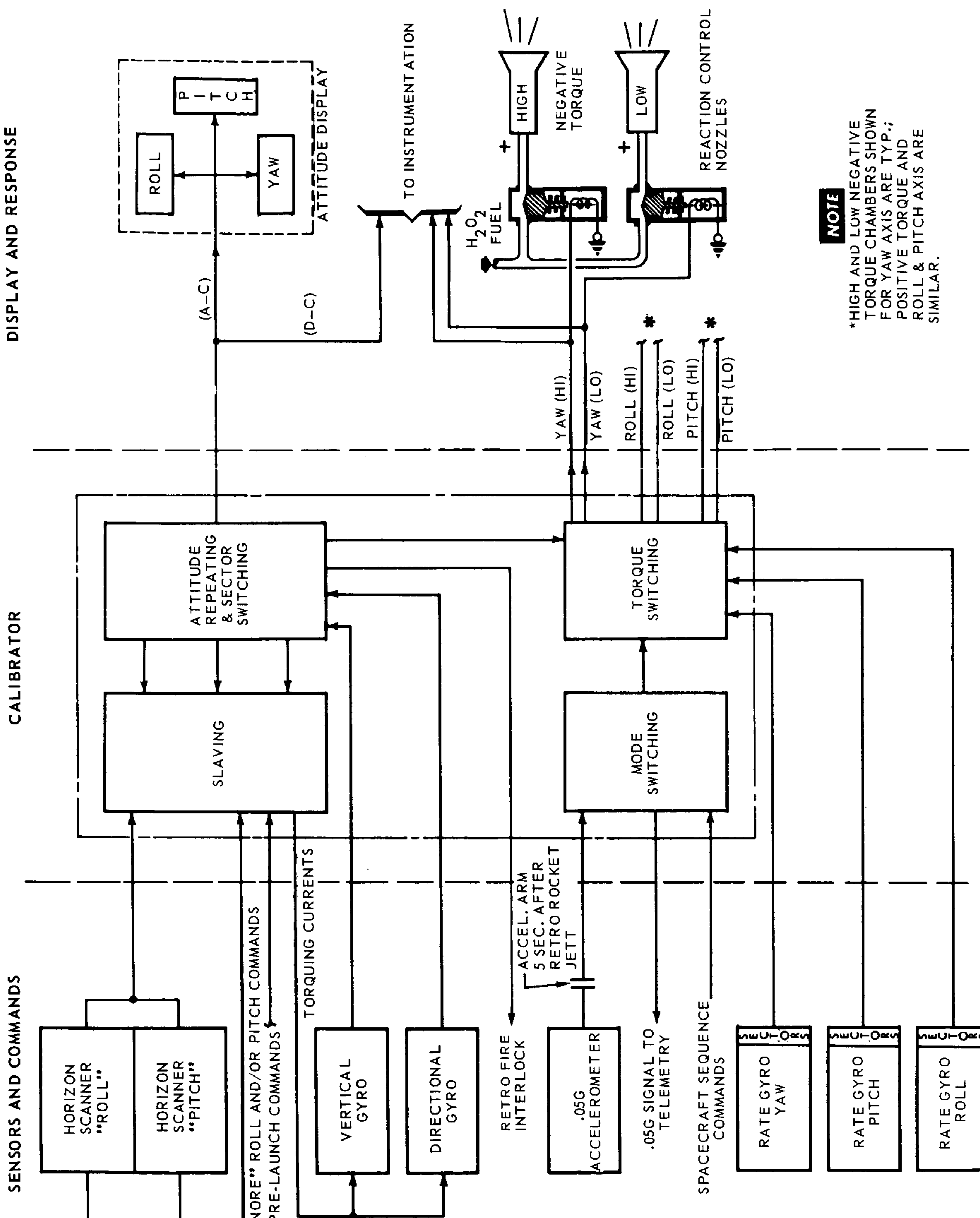

FIGURE 1.2-1 MERCURY ATTITUDE STABILIZATION CONTROL FUNCTIONAL BLOCK DIAGRAM

1.2.1 (Continued)

(a) Three single degree of freedom rate gyros sensed angular rates about the pitch, roll, and yaw axis of the spacecraft and switched electrical circuits at specific angular rates.

(b) Two attitude gyros (vertical and directional) were utilized to determine attitude angles between a set of fixed axes in the spacecraft and the local vertical reference. Both attitude gyros were "free" gyroscopes with slaving capabilities.

(c) An acceleration switch capable of sensing a 0.05 g acceleration force initialized the reentry mode.

(d) The calibrator unit contained the switching logic, attitude repeaters, summing and erection circuitry, relays, and power supply to integrate all elements of the system. The output signals of this unit included thrust control solenoid signals and attitude signals for visual display and/or telemetry.

(e) The attitude and rate indicator visually indicated spacecraft rate and attitude for all three axes.

Horizon Scanner System - Operation of the Horizon Scanner was based upon the infrared radiation received from the earth as compared to the essentially zero radiation from space. The system incorporated two identical scanning units. One unit was used as a roll sensor, the other as a pitch sensor. Each unit was capable of a 110° conical scan. The system provided a roll and pitch reference during the orbital phase of the normal mission. The scanners produced an output signal that slaved the ASCS attitude gyros to the proper angles upon command from an external programmer.

1.2.1 (Continued)

Reaction Control System - The Reaction Control System (RCS) was used for spacecraft yaw, pitch, and roll control. The system utilized 90% hydrogen peroxide as an energy source. Nitrogen gas was used to expel the propellant from a bladder into the thrust chamber catalyst beds. The system was divided into two independent portions, one for automatic control, and one for manual control. In addition, a reserve fuel system could be connected to the automatic portion.

Fly-By-Wire System - The fly-by-wire (FBW) system disabled the automatic feature and permitted the RCS solenoid control valves to be electrically operated by movement of the control stick. Stabilization was accomplished by moving the control stick in the desired plane. Two thrust level selections were provided to save fuel.

Manual Proportional System - The Manual Proportional (MP) system regulated fuel flow to a set of thrusters, independent of those used for other control modes, by deflection of the control stick. This was accomplished by mechanical linkages which transmitted the control stick movement to proportional control valves. Thrust output was directly proportional to hand controller displacement.

Rate Stabilization Control System - The Rate Stabilization Control System (RSCS) included a separate control electronics package and utilized independent rate gyros. Summing of stick positions and rate gyro outputs provided rate control.

1.2.2 Gemini Guidance and Control System - The mission requirements of the Gemini program dictated that a Guidance and Control System provide the spacecraft with the following capabilities:

(a) Backup launch vehicle guidance

(b) Three-axis attitude control

(c) Orbit adjust and rendezvous

(d) Vehicle rate and attitude display

(e) Retrograde time determination and update

(f) Reentry steering

The Guidance and Control System, shown in block diagram form in Figure 1.2-2, consisted of two major subsystems: an Inertial Guidance System and an Attitude and Maneuver Control System. These subsystems in turn interfaced with the additional spacecraft systems defined in Section 1.2.3.

The spacecraft mounting locations of the guidance and control equipment are shown in Figures 1.2-3 through 1.2-6. Figure 1.2-3 is a view of a reentry module mockup, showing the location of the IMU (3 packages), the digital computer, the horizon sensor electronics (2 packages), and the auxiliary computer power unit, mounted in the module's left-hand equipment bay. Also shown in this view is the location of the horizon sensor heads on the side of the module. Figure 1.2-4 shows components of the Attitude Control and Maneuver Electronics (ACME) in the reentry module center equipment bay. Figure 1.2-5 is a view of the adapter module mockup, showing the locations of the Orbit Attitude and Maneuver Electronics and Auxiliary Tape Memory. Figure 1.2-6 shows the rendezvous radar installation on the front of the spacecraft. The mounting of the transponder and its associated antennas in the Target Docking Adapter is shown in Figure 1.2-7.

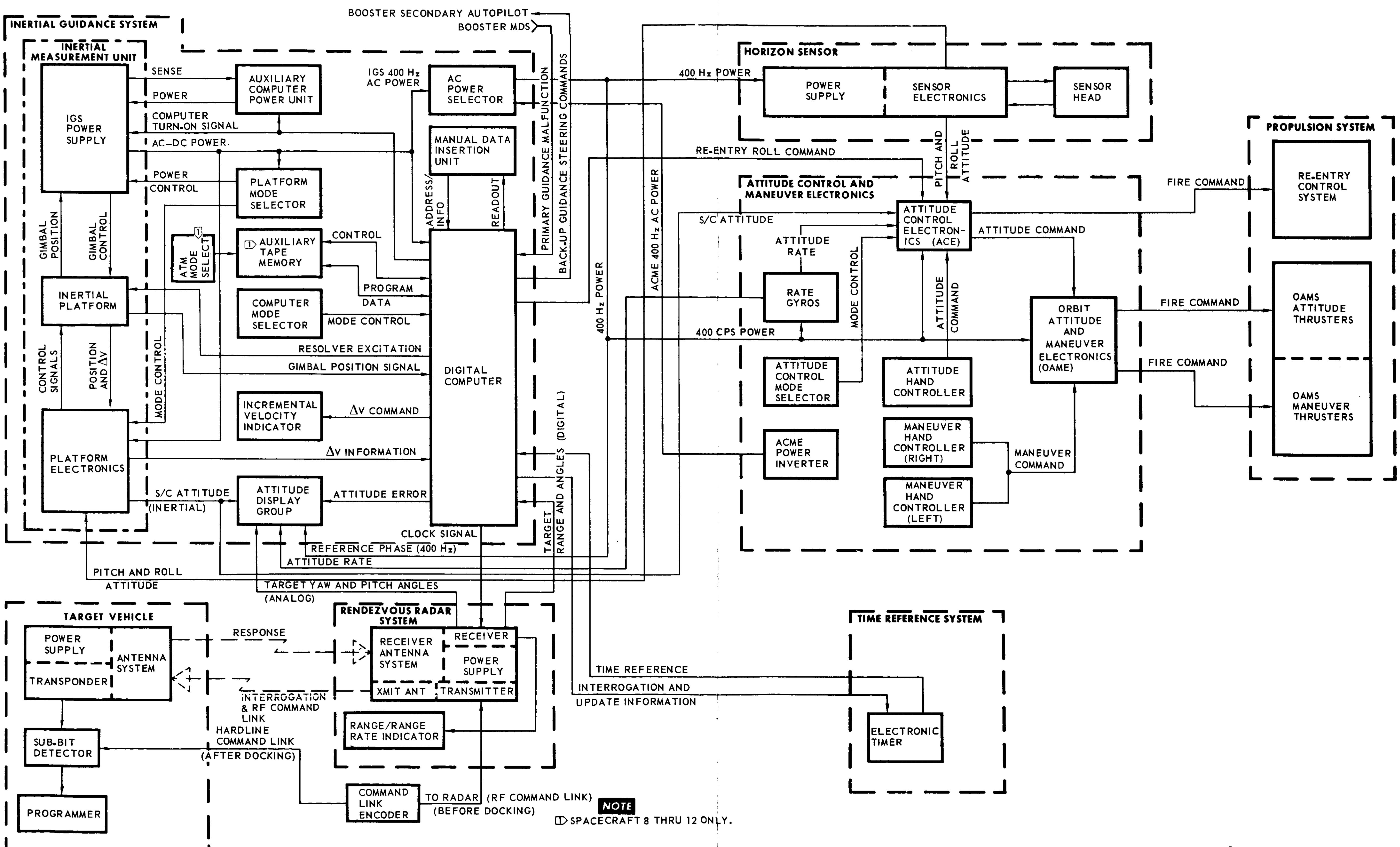

FIGURE 1.2-2 GEMINI GUIDANCE AND CONTROL FUNCTIONAL BLOCK DIAGRAM

FOLDOUT FRAME

9-A

FOLDOUT FRAME

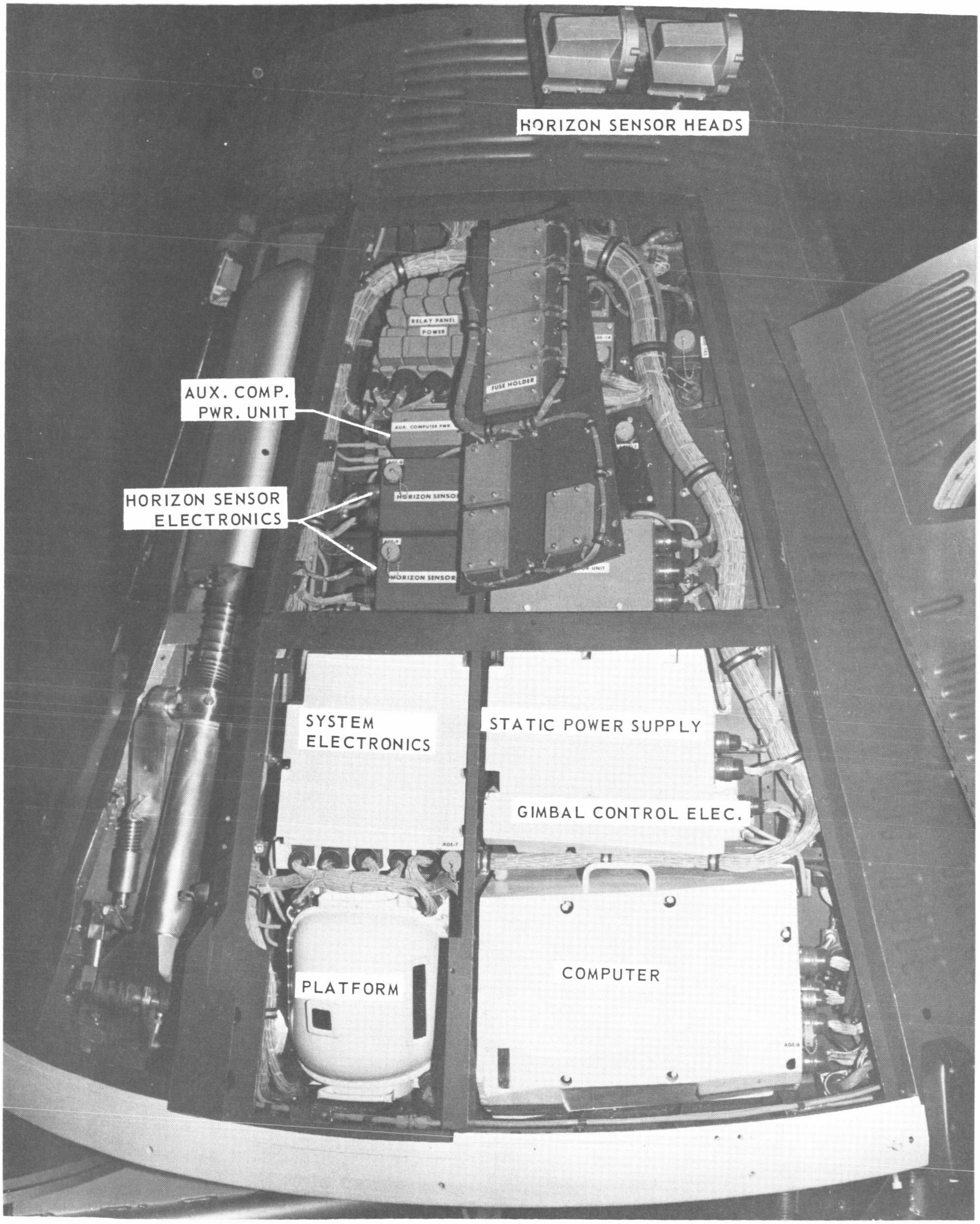

FIGURE 1.2-3 REENTRY MODULE LEFT HAND EQUIPMENT BAY (MOCKUP)

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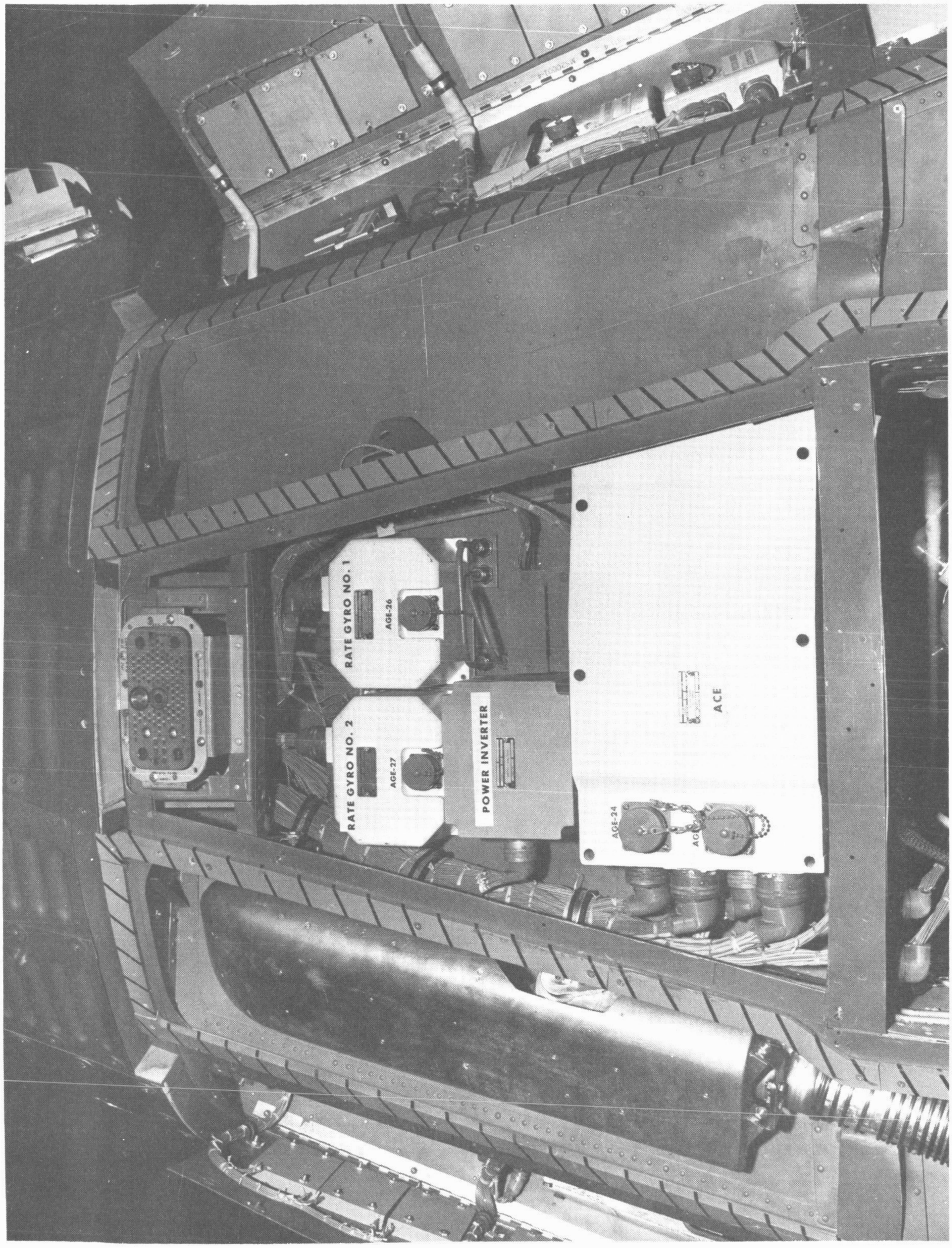


FIGURE 1.2-4 REENTRY MODULE CENTER BAY (MOCKUP)

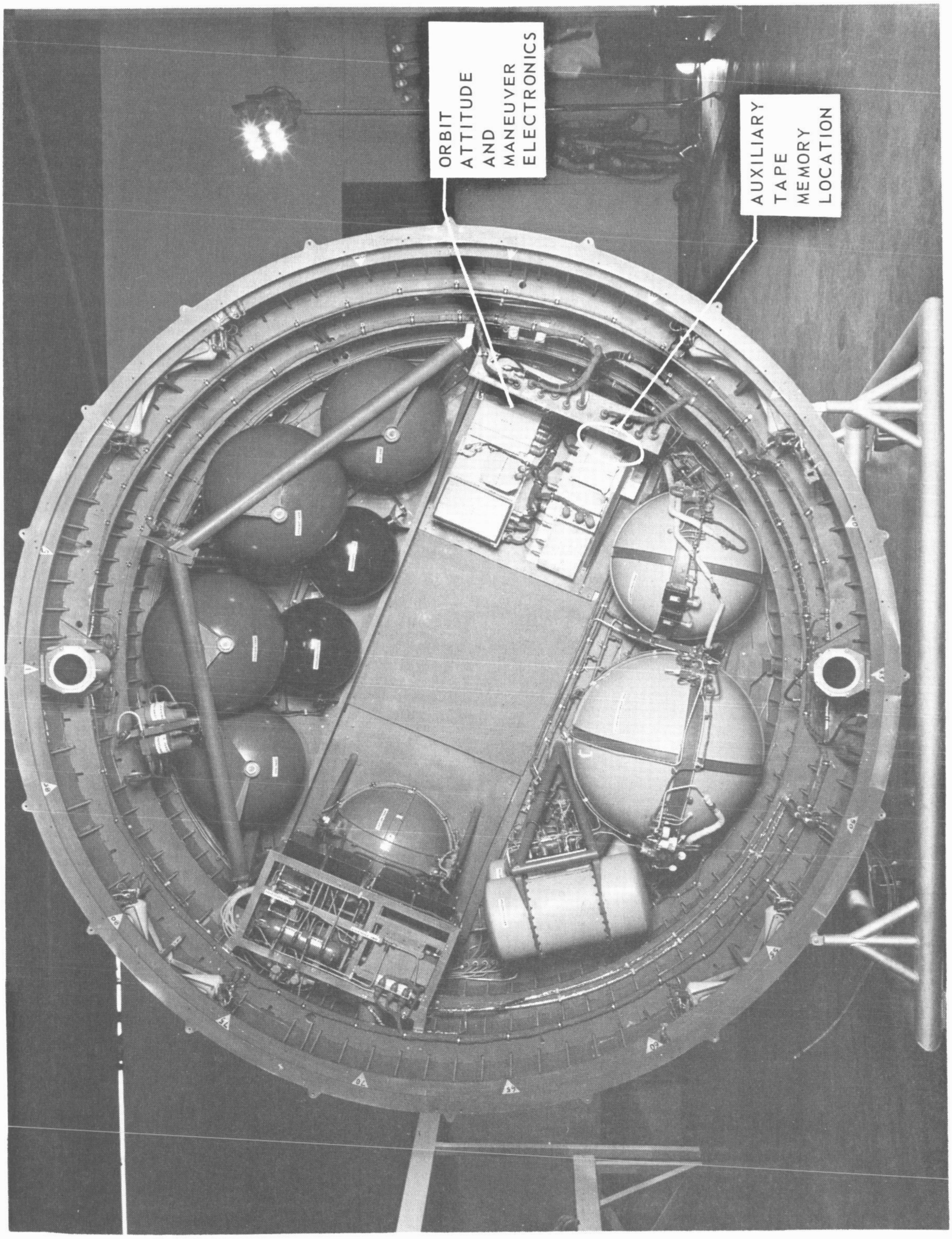


FIGURE 1.2-5 ADAPTER MODULE (MOCKUP)

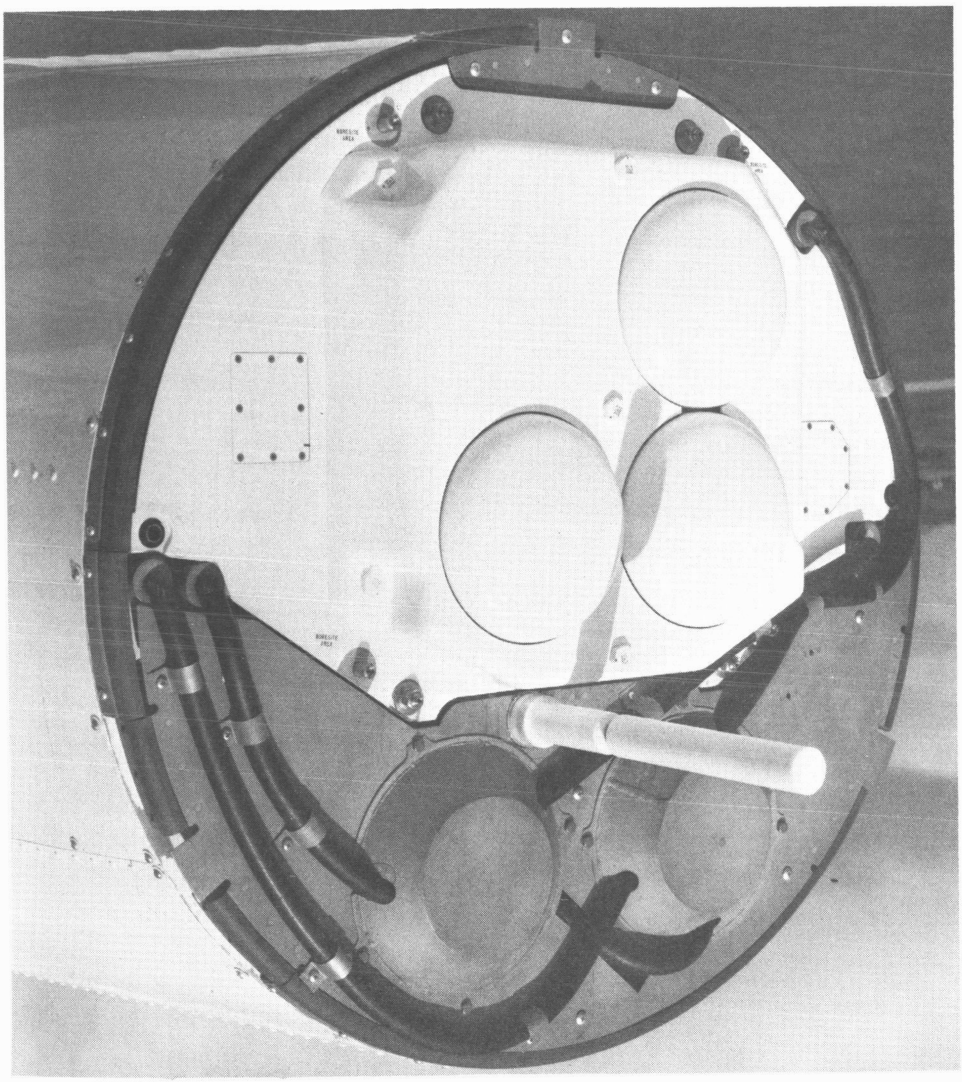


FIGURE 1.2–6 RENDEZVOUS RADAR INSTALLATION (MOCKUP)

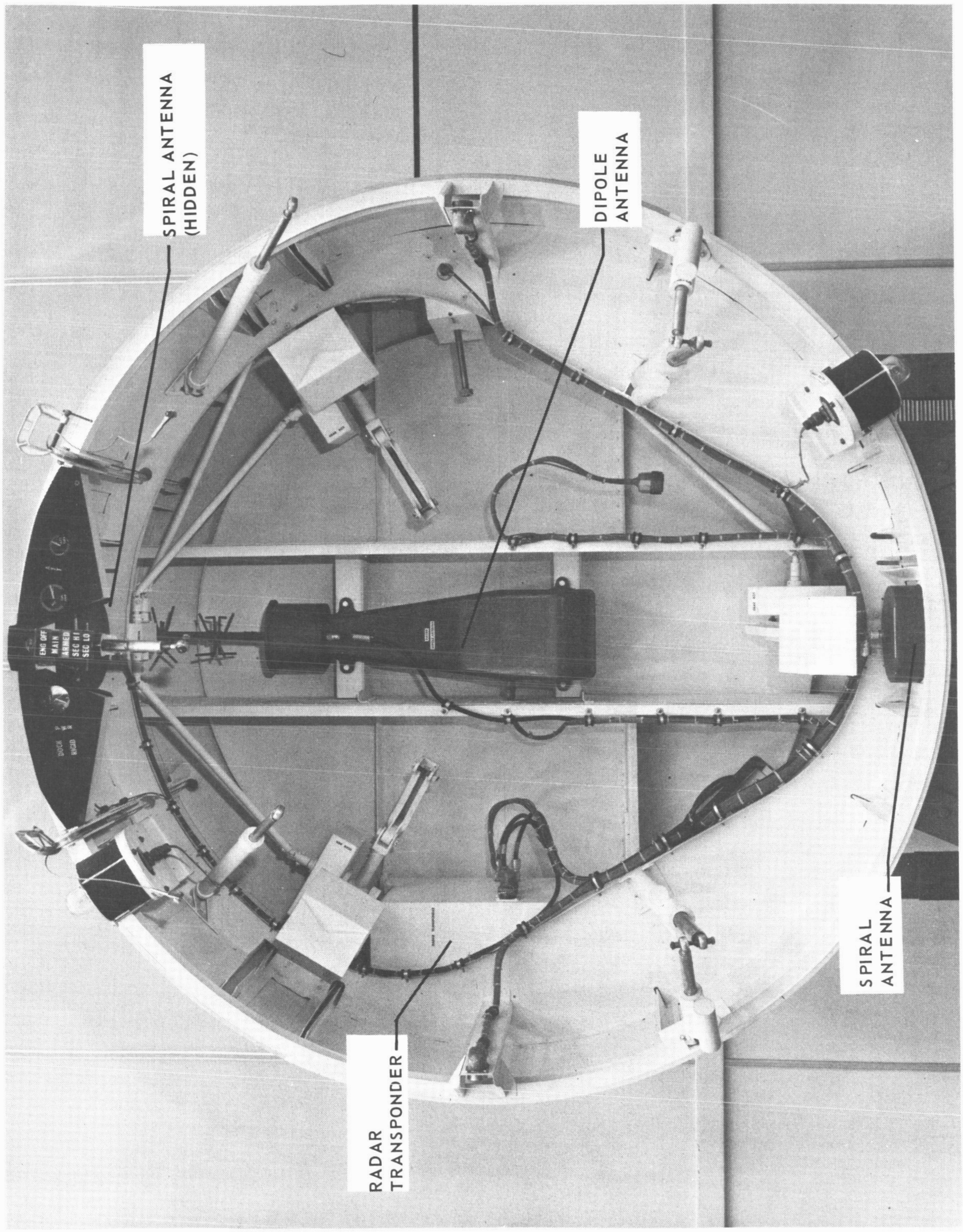


FIGURE 1.2-7 TARGET DOCKING ADAPTER (MOCKUP)

1.2.2 (Continued)

The major components of the guidance and control system and associated systems are described below:

1.2.2.1 Inertial Guidance System - The Inertial Guidance System (IGS) contained the following components: the Inertial Measurement Unit (IMU), the Digital Computer, the Auxiliary Computer Power Unit (ACPU), the Manual Data Insertion Unit (MDIU), the Incremental Velocity Indicator (IVI), and the Auxiliary Tape Memory (ATM).

Inertial Measurement Unit - The Inertial Measurement Unit (IMU) consisted of three separate packages: the inertial platform, the system electronics, and the IGS power supply including the gimbal control electronics. All three packages functioned together to provide inertial attitude and acceleration information. Attitude measurements were utilized for automatic control, guidance computations, and visual display. Acceleration measurements were utilized for insertion, rendezvous, and retrograde computations and displays. In addition to attitude and acceleration references, the IMU provided AC and DC power to additional spacecraft equipment. A brief description of the components comprising the IMU follows:

(a) The inertial platform utilized three floated miniature integrating gyros and three floated pendulous accelerometers supported by a four-gimbal assembly. Major components of the platform were: a housing, gimbal structure, torque motors, gimbal angle synchros, resolvers, gyros, and accelerometers.

(b) The system electronics package contained the circuitry necessary to operate the IMU. Circuits were provided for gyro torque control,

1.2.2.1 (Continued)

timing logic, spin motor power, accelerometer logic, accelerometer rebalance, and malfunction detection.

(c) The IGS power supply contained gimbal control electronics and the static power supply unit. Gimbal control electronics drove torquer motors in the platform while the static power supply provided the electrical power (AC and DC) for the IMU, Computer, ACPU, MDIU, IVI, ACME and Horizon Sensors.

The seven modes of operation that could be selected were:

(a) OFF - Platform not energized.

(b) SEF Cage - The IMU had to be caged before it could be aligned. Two cage positions were provided so that the spacecraft could be positioned either small end forward (SEF) or blunt end forward (BEF) for platform alignment. The SEF cage position was used for warmup of platform gyros and for electrical caging of the platform by aligning the platform at 0°, 0°, 0° or 180°, 180°, 180° with respect to the spacecraft coordinates (small end of spacecraft forward). The electrical caging rate was 0.83°/second.

(c) BEF Cage - This mode was used to align the platform gimbals at 0°, 180°, 0°, or 180°, 0°, 180° with respect to the spacecraft coordinates (blunt end of spacecraft forward). Other functions and characteristics were identical to the SEF cage position.

(d) SEF - This mode provided platform fine alignment. The platform was slaved to the horizon sensor output signals for pitch and roll alignment. In addition, a gyrocompassing logic system converted roll angular error signals into yaw gimbal alignment signals. The signal

1.2.2.1 (Continued)

phases were set for correct alignment in the spacecraft small end forward position. An orbital rate signal (nominally 4°/min) was summed with the horizon sensor error signal to the pitch gyro torquer. A sensor ignore signal would automatically cause the platform to function as in the orbit rate mode. A loss of sensor attitude signals without an ignore signal placed the platform in a pseudo cage mode. (Pitch would not cage properly.)

(e) BEF - This mode, which also provided platform fine alignment, was identical to SEF, except that phases were set for correct alignment in the spacecraft blunt end forward condition. An orbital rate signal (nominally 4°/min) was summed with the error signal to the pitch gyro torques. Horizon scanner functions were identical to SEF.

(f) Orbit Rate - This mode was used to maintain attitude reference during spacecraft maneuvers. In this mode the platform was inertially free in roll and yaw, but pitch was torqued at a preset orbital rate for constant platform attitude with respect to the local vertical. The nominal rate was approximately 4°/min pitch down.

(g) FREE - The platform was gyro-stabilized in this mode and thus inertially fixed except for gyro drift. This mode was selected automatically at lift-off, during the ascent phase, and at retrofire.

Digital Computer - The Gemini Digital Computer was a binary, fixed-point, stored program, general purpose computer that operated at a 500 KC bit rate. It interfaced with the Inertial Platform, Platform Electronics, IGS Power Supply, Auxiliary Computer Power Unit, Manual Data Readout, Manual Data Keyboard (via Manual Data Readout), Incremental Velocity

1.2.2.1 (Continued)

Indicator, Radar, Time Reference System, Digital Command System, Data Acquisition System, Attitude Control and Maneuver Electronics, Launch Vehicle Autopilot, Aerospace Ground Equipment, and Pilot Controls and Displays (e.g. Attitude Group Display, Computer Mode Switch, etc.).

From the inputs and its stored program, the computer provided the guidance and control outputs required for the Gemini spacecraft during the prelaunch, catch-up, rendezvous, and reentry phases. In addition, the computer provided backup guidance for the launch vehicle during ascent.

The computer memory was capable of random access and nondestructive readout. The memory array provided for 4096 memory words, each with 39 bits, or a total storage of 159,744 bits. All memory words were divided into three syllables of 13 bits each. The syllables were designated 0, 1, and 2. Syllable 2 could not be altered after being loaded by ground equipment. Data words (25 bits plus sign) were normally coded in the first two syllables, and instruction words (13 bits) were intermixed in all three syllables. Limited modification of the stored data was possible at the launch site through interface with the Manual Data Insertion Unit or the Digital Command System.

The operational computer program contained interleaved diagnostic subroutines that permitted malfunction detection during operation. When a fault was detected, a discrete was issued by the computer to light a computer malfunction indicator lamp on the control panel. The circuit had a manual reset capability to insure that it was not set by a transient type malfunction.

1.2.2.1 (Continued)

The computer was controlled by means of four switches located on the astronaut's control panel - a seven-position computer mode switch, a two-position computer off/on switch, a pushbutton start computation switch, and a pushbutton malfunction reset switch. The computer was turned on or off by the computer switch on the control panel.

The computer received the AC and DC power required for its operation from the IGS power supply. The regulated DC power supplied to the computer was buffered in the IGS power supply to eliminate any loss in regulation due to transients that might have occurred in the prime spacecraft power source. Actual power interruptions and depressions lasting as long as 100 milliseconds were buffered by the IGS power supply and Auxiliary Computer Power Unit.

Auxiliary Computer Power Unit - The Auxiliary Computer Power Unit (ACPU) was used in conjunction with the IGS power supply to maintain the correct DC voltage at the computer. The ACPU furnished power which enabled the computer to operate during low voltage transients of as long as 100 milliseconds. If power was interrupted for a longer period, the ACPU shut down the computer in a controlled manner. The ACPU consisted of the required sense circuitry, a battery, and a trickle charger.

Manual Data Insertion Unit - The Manual Data Insertion Unit (MDIU) consisted of the Manual Data Keyboard (MDK) and the Manual Data Readout (MDR) units. The MDIU permitted the astronauts to insert data into the computer memory and read data from it. The unit accepted and displayed decimal data and converted the data to and from computer binary language.

1.2.2.1 (Continued)

Incremental Velocity Indicator - The Incremental Velocity Indicator (IVI) visually indicated incremental velocities for the longitudinal (forward - aft), lateral (left - right) and vertical (up - down) axes of the spacecraft. These incremental velocities corresponded to the amount and direction of additional thrust necessary to achieve correct orbit. In addition, the IVI displayed the tape position words and module words from the Auxiliary Tape Memory.

Auxiliary Tape Memory - The Auxiliary Tape Memory (ATM) was a self-contained magnetic tape system which stored an additional 85,000 13-bit words for the digital computer. The ATM was used to store operational programs for in-flight loading in the spacecraft digital computer memory. Automatic and manual modes of loading the computer via the ATM were available.

1.2.2.2 Attitude and Maneuver Control System - The Attitude and Maneuver Control System contained the Attitude Control and Maneuver Electronics, the Attitude Display Group, and the Horizon Sensor System.

Attitude Control and Maneuver Electronics - The Attitude Control and Maneuver Electronics (ACME) provided closed loop rate stabilization and attitude control and/or manual control, when utilized with the horizon sensors, computer, platform and attitude control handle. The ACME system converted the input signals into thruster firing commands necessary to attain and maintain an attitude or velocity. For the orbital phase, the Orbit Attitude and Maneuvering System (OAMS) thrusters were used for attitude and maneuvering, while during the reentry phase, attitude control

1.2.2.2 (Continued)

was provided by the Reentry Control System (RCS) thrusters. The ACME system consisted of the Attitude Control Electronics (ACE), Orbit Attitude and Maneuver Electronics (OAME), ACME Power Inverter, and two identical Rate Gyro packages. The attitude control system had both automatic and manual modes of operation while the maneuvering control system had only a manual mode.

(1) Attitude Control Electronics - The Attitude Control Electronics (ACE) subsystem consisted of power supplies, mode logic circuits, proportional circuits, control torque logic circuits, RCS relay drivers, and a pulse generator. These circuits converted the commands or error signals from the computer, platform, horizon sensors, rate gyros, and attitude hand controller into thruster firing commands. The firing commands were routed by a valve driver select system to the RCS or the OAMS attitude valve drivers.

(2) Orbit Attitude and Maneuver Electronics - This package consisted of the attitude solenoid valve drivers, maneuver solenoid valve drivers, spike suppression circuits, and single pulse generator. The OAME accepted signals from the ACE and maneuver controllers for conversion to drive commands to the OAMS solenoids.

(3) ACME Power Inverter - The power inverter converted spacecraft DC power into 26V, 400 Hz (the IGS power supply provided the primary source of AC excitation).

(4) Rate Gyros - The two rate gyro packages (RGP) contained three single-degree-of-freedom, spring restrained rate gyros. The gyros were orthogonally mounted for rate sensing in all three axes. The rate gyro package provided AC analog outputs proportional to attitude

1.2.2.2 (Continued)

rate inputs to (1) the ACE when used in the rate command mode, (2) the attitude display group, and (3) telemetry.

(5) Attitude Hand Controller - Spacecraft attitude was manually controlled by use of an attitude hand controller and a visual reference. Controller outputs were rate, pulse, and direct command signals, depending upon the control mode selected. Output signals were produced by positive or negative handle movements from the center position. Rate signals were proportional to the amount of handle displacement from a center deadband. Direct and/or pulse signals were produced when the hand controller was displaced past a preset threshold or deadband. Pulse signals triggered a pulse generator in ACE to produce a fixed increment firing signal. In this mode, the control handle had to be returned to a neutral position before another single pulse could be commanded.

(6) Maneuver Hand Controller - Translational maneuvers of the spacecraft in the horizontal, longitudinal, and vertical planes were commanded by either of the two maneuver hand controllers that were provided. Displacement of a hand controller, from the centered or neutral position in any of the six translational directions, produced a firing command to the respective solenoid valves.

(7) Attitude Control Selector - The attitude control selector provided the following operational modes for attitude control:

(a) The direct mode, a backup control mode, controlled spacecraft angular acceleration. Switches activated by attitude control handle displacements provided discrete signals to the ACME which were converted into on-off commands to the OAMS thrusters.

1.2.2.2 (Continued)

When used with the RCS, the direct mode was instrumented by two independent methods. In the first method, switches on the attitude hand controller provided on-off commands directly to the RCS solenoid valves. In the second method, switches on the attitude hand controller provided discrete signals to the ACME which were converted into on-off commands to the RCS solenoid valves.

(b) In the pulse mode, the spacecraft angular rate could be changed in incremental steps. Switches activated by attitude control handle displacements triggered a pulse generator in the ACME. The pulses were converted to firing signals commanding thrust for 20 ms (nominal) for each handle displacement. The pulse mode was effective in each of the three major spacecraft axes and was used for fine attitude control, e.g. during platform alignment.

(c) The horizon scan mode provided automatic control of the spacecraft about its pitch and roll axes during the orbital phase and was used in establishing an intermediate reference during platform alignment. The capabilities of the pulse mode were maintained to provide manual yaw attitude control and pitch and roll manual override control. The horizon sensors provided a reference in pitch and roll. A nose down pitch bias was applied so that the nominal pitch attitude variation maintained the earth's horizon with the astronauts' field of view. The IGS and rate gyros were not used in this mode.

1.2.2.2 (Continued)

(d) The rate command mode maintained attitude control during manual attitude control operations and during velocity change maneuvers. This mode was effective about each of the three major spacecraft axes. Vehicle angular rates were controlled such that they were proportional to the attitude control handle displacements. Rate gyro outputs were compared in the attitude control electronics (ACE) with the signals generated by control handle displacements. When the difference between the two signals exceeded the damping deadband, the proper reaction control jets were fired.

(e) In the reentry mode, the spacecraft pitch and yaw rates were automatically maintained within the damping dead zone by the ACME. In addition, the ACME utilized inputs from the computer to control the spacecraft about the roll axis. The computer input was 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 maintained the commanded roll attitude or rate within a deadband about the command value. The reentry mode was utilized with RCS thrusters.

(f) The reentry rate command mode was utilized for manual reentry attitude control and had the same operational characteristics as the rate command mode except that the damping deadbands corresponded to the reentry mode values.

(g) The platform mode was used to maintain spacecraft attitude, in all three axes, with respect to the inertial platform. A horizontal attitude, with respect to earth, could be maintained

1.2.2.2 (Continued)

with the inertial platform in either the orbit rate or alignment mode of operation.

Attitude Display Group - The Attitude Display Group (ADG) consisted of two Flight Director Indicators (FDI) and two Flight Director Controllers (FDC). The attitude display group provided the required reference or rate information to achieve the desired spacecraft orientation.

(1) The flight director indicator consisted of a three-axis sphere with 360° of freedom in each axis with superimposed flight director needles. The sphere was slaved to the inertial platform gimbals and represented the spacecraft body angles relative to platform orientation. Information displayed on the needles was provided by the computer, platform, radar, or rate gyros via the FDC. A scale selector on the FDI allowed the selection of high or low scale indications on the needles.

(2) The flight director controller contained the electronics to provide needle drive signals and switching to permit selection of needle display information. The FDC reference switches selected the source of information supplied to the indicators; i.e., radar, platform or computer, while the mode switch selected the type of information to be supplied; i.e., attitude rates, attitude, or attitude plus rate (mix).

Horizon Sensor System - The horizon sensor system consisted of a sensor head, an electronics package, and the required controls and indicators. The system was used to establish a spacecraft attitude reference to earth

1.2.2.2 (Continued)

local vertical and to generate error signals proportional to the difference between spacecraft attitude and horizontal attitude.

The horizon sensors used an azimuth scanning method (160° scan) to track the infrared (IR) gradient between earth and space at the horizon. The horizon sensor system converted the horizon tracking information to analog signals proportional to spacecraft pitch and roll attitude errors with respect to the earth's local vertical. The roll and pitch output signals were used by ACME for attitude control (horizon scan mode) and by the IMU for platform alignment (SEF, BEF modes). Two complete and separate horizon sensors were provided.

1.2.3 Interfacing Spacecraft Systems - In performing in-flight navigation and control, the guidance and control system operated in conjunction with the following spacecraft systems: Rendezvous Radar System, Command Link, Propulsion System, Time Reference System, Digital Command System, Electrical System, Coolant System, Instrumentation System, and Launch Vehicle.

Rendezvous Radar System - The rendezvous radar system was comprised of two units, an L-band radar system located in the Gemini Spacecraft and a transponder located in the target vehicle. Cooperative operation of the two units enabled the Gemini spacecraft to detect the target vehicle and to determine the range, range rate, and angular relationship of the two vehicles. An interferometer system was used to derive target elevation and azimuth angles. The radar transmission was also used to carry the command link messages. Visual indications of radar lock-on and command link message acceptance were provided. Analog indications of the target vehicle range

1.2.3 (Continued)

and relative velocity were displayed on a range and range rate indicator. Analog indications of the elevation and azimuth position of the target vehicle, with respect to the Gemini spacecraft, were presented on the flight director indicators. Digital indications of range, elevation, and azimuth were available to the computer for calculating the corrective thrusts required for the rendezvous maneuver.

Command Link - The command link provided the capability to control the target vehicle via the spacecraft. Command link control was used to position the target vehicle in the desired attitude and orbital path and to activate target vehicle equipment. Commands were sent via the radar RF link.

Time Reference System - The Time Reference System (TRS) performed all timing functions aboard the spacecraft. The system was comprised of an electronic timer, an event timer, a time correlation buffer, a mission elapsed time digital clock, an Accutron clock, and a mechanical clock.

The electronic timer provided: (1) an accurate countdown of time-to-go to retrofire, (2) time-to-go to equipment reset, (3) a record of elapsed time from lift-off, and (4) time correlation for the PCM data system and the bio-med tape recorder. It was the basic time reference for the computer guidance calculations.

The event timer provided a display for timing of various short-term functions. It was started at lift-off, in conjunction with the electronic timer, and displayed elapsed time during the ascent phase of the mission.

1.2.3 (Continued)

The additional clocks provided back-up capability, while the time correlation buffer conditioned certain output signals from the electronic timer.

Propulsion System - Spacecraft control was accomplished by three separate propulsion systems: the Orbit Attitude and Maneuver System (OAMS), the Reentry Control System (RCS), and the Retrograde Rocket System. The OAMS controlled the spacecraft's attitude and provided maneuver capability from the time of launch vehicle separation until the initiation of retrograde. The RCS provided attitude control for the reentry module during the reentry phase of the mission. The retrograde rocket system imparted the impulse to the reentry module to permit safe reentry into the earth's atmosphere. The OAMS and RCS responded to electrical commands from the ACME and the hand controllers.

Digital Command System - The Digital Command System, the command link between the ground and the spacecraft, enabled the ground to update the computer and the time reference system, in addition to performing other spacecraft functions.

Electrical System - The electrical system was divided into two subsystems, the power system and the sequential system. The electrical power system supplied unregulated direct current for all systems of the spacecraft. The sequential system activated displays and performed switching functions to provide the proper sequence of events.

Cooling System - The spacecraft cooling system consisted of two identical temperature control fluid systems which functioned independently to cool or

1.2.3 (Continued)

heat the spacecraft equipment. The Guidance and Control System components which were coldplate mounted were: the IMU, Computer, ATM, ACE, OAME, Power Inverter, and Rate Gyros.

Instrumentation System - The instrumentation system acquired data on the performance and operation of all spacecraft systems, including the Guidance and Control System, throughout the mission. The instrumentation system documented significant events and data throughout the entire mission by three methods: real-time transmission, delayed transmission, or onboard recording.

Launch Vehicle - During ascent, the spacecraft computer performed guidance computations in parallel with the launch vehicle guidance and control system. Automatic and manual guidance switchover capabilities were provided. In back-up ascent guidance the spacecraft computer provided the steering and booster cut-off commands to the secondary booster auto pilot.

1.2.4 References - More complete descriptions of the Mercury and Gemini Guidance and Control equipment and also their detailed performance and environmental requirements may be found in the references included in Section 8.

2.0 Sensors

2.1 Rate Gyros

2.1.1 Requirements - The requirements for the Gemini rate sensors were more stringent than those for the Mercury spacecraft, and therefore a better quality rate gyro was necessary. The rate gyro requirements are shown in Table 2.1-1. These requirements were considered as being within the state-of-the-art, although they did dictate use of high precision parts and techniques.

The differences in the original requirements and final specifications are primarily due to the following factors:

(a) Further analysis on the control system requirements indicated that the performance requirements could be relaxed for certain conditions (e.g. at large rates). Asking for such performance would have pushed the state of the art and driven costs up unnecessarily.

(b) Requirements for operation at temperature extremes could be relaxed, since, under most conditions, coldplates would maintain equipment at between $+40^{\circ}$ and $+90^{\circ}$F.

The addition of the modulation specification is discussed in the following paragraphs.

2.1.2 Early Design Tradeoffs and Decisions - Considerable trouble had been experienced with the Mercury rate gyro pickoffs. The Mercury employed a digital control system, and therefore the sensors, including the rate gyros, used electromechanical encoders as pickoffs. The rate gyros used sector switches which switched in at ± 0.5 degrees per second, ± 2 degrees per second, and ± 10 degrees per second. However, noise created by the

TABLE 2.1-1

RATE GYRO PACKAGE

FUNCTIONAL REQUIREMENTS

	Final Capability	Initial Requirement
Threshold	$< \pm 0.01$ deg/sec.	Same
Resolution	<0.01 deg/sec. for rates from 0 - 0.5 deg/sec. continuous output from 0.5 deg/sec. to full scale.	Same
Linearity	<0.04 deg/sec. deviation from straight line for input rates between $\pm$ 5 deg/sec.	<0.04 deg/sec. ($\pm$ 5 deg/sec.)
	<0.4 deg/sec. for rates from 5 - 35 deg/sec. monotonically increasing from 35 deg/sec. to stop.	<0.08 deg/sec. ($\pm$ 5 - $\pm$ 40 deg/sec.)
Hysteresis	<0.02 deg/sec. for input changes between $\pm$ 5 deg/sec. <0.04 deg/sec. for input changes between 5 & 40 deg/sec.	<0.04 deg/sec. for full scale defl.
Null (in phase)	<7.2 mv at temperature between $40^{\circ}F$ and $120^{\circ}F$ <33 mv over temperature range $-60^{\circ}F$ to $+120^{\circ}F$	<5.6 mv at $80^{\circ}F$; <6.2 mv at 40° to $120^{\circ}F$; <15.6 mv at $-60^{\circ}F$ to $+200^{\circ}F$.
Null (Total)	<16.5 mv at temperature between $+40^{\circ}F$ and $+120^{\circ}F$ <40 mv over temperature range $-60^{\circ}F$ to $+120^{\circ}F$	<13 mv at $80^{\circ}F$; <46 mv at $+40^{\circ}F$ to $+120^{\circ}F$; <28 mv at $-60^{\circ}F$ to $+160^{\circ}F$.
Linear Acceleration Sensitivity	<0.06 deg/sec/g.	<0.04 deg/sec/g.
Warm Up Time	<30 seconds for temperature $> +40^{\circ}F$ <60 seconds for temperatures $< +40^{\circ}F$	<30 sec. - all temperatures
Modulation	<18.4 mv (design goal)	No Requirement

2.1.2 (Continued)

sliding of the wiper arm caused a significant widening of the attitude control deadbands.

Therefore, a new rate gyro design was selected for Gemini. An analog pick-off device was employed, in keeping with the Gemini control system philosophy of using analog pickoff devices for its sensors. The time history of the rate gyro development is as shown for the Attitude Control and Maneuver System, in Figure 4.1-1.

Two three-axis rate gyro packages were installed. The gyros were switched to allow selection by individual axis, rather than selection by package, resulting in greater reliability. Figure 2.1-1 shows the mounting of the three gyros in the package.

2.1.3 <u>Rate Gyro Development and Design Problems</u>

<u>Rate Gyro Contamination</u> - A problem was encountered early in the manufacture of the rate gyros in that gimbal hang-up caused high outputs with no input rate. The problem was caused by contaminants within the gyros, which interfered with the gimbal movement. Although this problem was caused primarily by a combination of poor quality control and manufacturing procedures, the gyro design also contributed. The Gemini gyro contained a self-test torquer. The subassembly containing this torquer and the pickoff was extremely difficult to clean because of its geometric complexity. Particles not removed during final cleaning could easily cause a gyro hang-up after final assembly. The problem was compounded by the extremely tight clearance between the gimbal and gyro case.

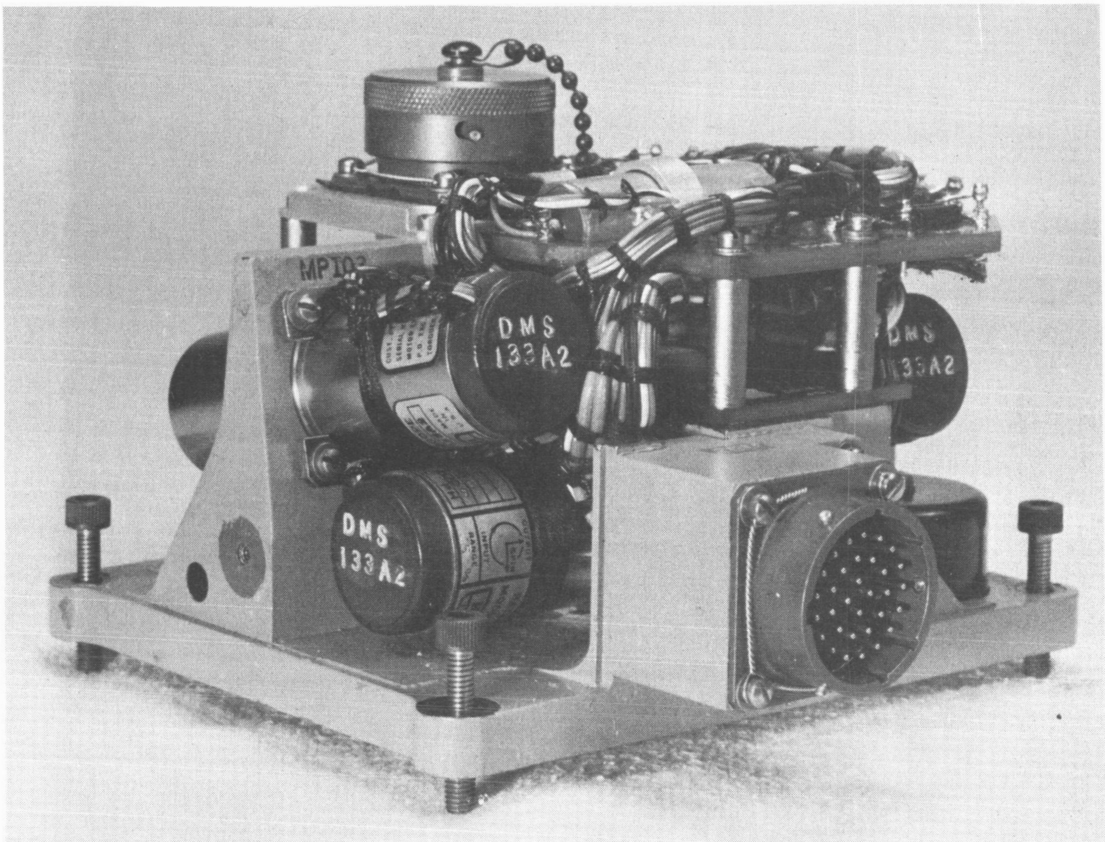


FIGURE 2.1-1 RATE GYRO PACKAGE (COVER REMOVED)

2.1.3 (Continued)

The solution to this problem consisted mainly of establishing new cleaning procedures, including a continual flow of flushing material through the pickoff and torquer assembly during its final cleaning. The flushing was continued until the contamination was reduced to a satisfactory level as determined by microscopic inspection.

The previously high rejection rate at the gyro manufacturer was greatly reduced by the new cleaning procedures, and no further concrete evidence of contamination was seen. Some isolated hang-ups were experienced during testing at the control system vendor, and one time during McDonnell spacecraft system testing, but the cause was never exactly determined. No problems were encountered during any of the flights, however.

Poor Service Life - Early in 1963, several gyro motors were found to be wearing out before the required service life of 670 hours. This problem was traced to the motor bearings' metal retainers, which did not hold sufficient lubricant. The problem was solved by substituting phenolic bearing retainers, which acted as lubricant reservoirs. After this change, a significant increase in motor life resulted, far exceeding service life requirements, and no further problems were encountered.

Modulation - The outputs of the rate gyros contained a time varying component at zero or constant input rates. This phenomenon was apparent only at very low input rates. As input rates built up, the amplitude would diminish or become insignificant. The modulation was an oscillation in the range of 29 to 37 Hz with, in some cases, a beat frequency of approximately 2 Hz and a long-term amplitude modulation at approximately 0.2 Hz. The

2.1.3 (Continued)

amplitude of these oscillations was not repeatable, and appeared to increase or decrease at random. The maximum observed amplitude of the modulation was a voltage equivalent to about 0.1 degrees per second. Extensive testing was conducted at the vendor, utilizing an isolated rate table, from which it was concluded that the spin motor was a contributing factor.

The cause of this modulation was never determined; however, several theories were advanced. One hypothesis advanced was that the gyro gimbal was acting as a cantilever beam whose natural frequency was thought to be approximately that which was measured. The outputs would have been produced by variations in the gap between the pickoff rotor and stator.

The concern about the modulation was due to the requirements of the high gain rate command mode used with the OAMS propulsion system. This mode originally controlled the spacecraft rate within ± 0.1 degrees per second of the commanded rate. It was feared that a high value of modulation would cause instability in the rate loop. The OAMS rate command mode deadband was subsequently raised to ± 0.2 degrees per second, partially as a result of this concern, and the problem never materialized in systems tests or flights.

2.2 Inertial Measurement Unit

2.2.1 Requirements - The performance requirements of the Inertial Measurement Unit (IMU) are classified, but can be obtained from the McDonnell Specification Control Drawing 52-87717.

2.2.2 Initial Design Tradeoffs and Decisions - The Gemini mission required a reference that could provide stabilization about the three spacecraft body axes and measurement of spacecraft velocity changes. A platform was selected over a strapdown system because the gimballed system reduced spacecraft digital computer operations for coordinate transformations and because the platform represented the lesser technical and schedule risk.

Figure 2.2-1 provides a time history for IMU design, development, and flight operation. This figure establishes significant milestones in the design and development of the equipment.

Platform Modifications - The platform chosen was a modified version of a miniature four-gimballed unit used on the Centaur program. These Gemini modifications were included:

(1) An improved optical access and larger porro prism were incorporated.

(2) The GG 49 gyros were replaced with GG 8001's for increased reliability and gyro torquing capability.

(3) Mechanical changes were made to permit the platform to be hard-mounted. Vibration tests on a hard-mounted Centaur platform indicated that added gimbal 3 and 4 torque capability was desirable. This was incorporated. To reduce vibration transmission, a high damping magnesium alloy was used in making the gimbals and case.

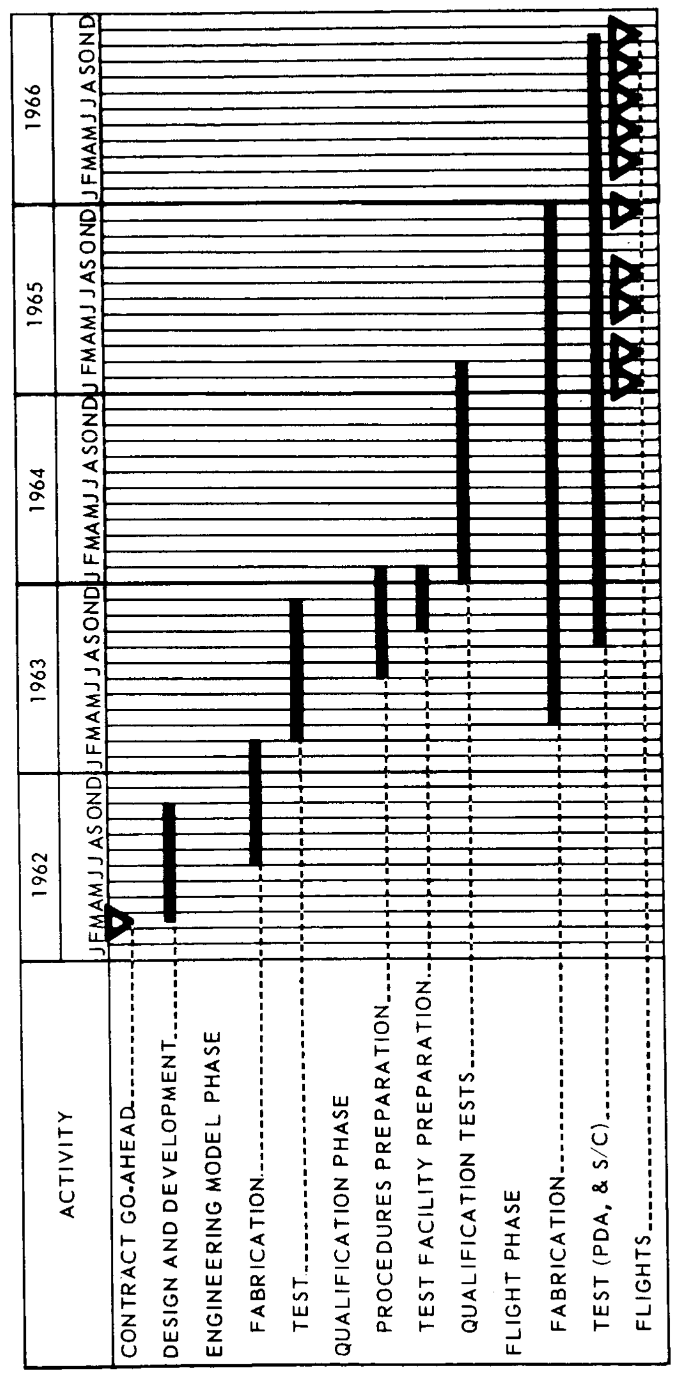


FIGURE 2.2-1
IMU PROGRAM TIME HISTORY

2.2.2 (Continued)

(4) The accelerometer pulse rebalance circuitry was redesigned for improved reliability and performance. The new design for Gemini reduced the number of parts to about two-thirds of those of the old Centaur design. The new design added new rate compensation networks and external phase adjustments to compensate for lead lengths between the accelerometer and electronics.

(5) Fast heat block heaters were removed. The change to the GG 8001 gyro made this possible because it had dual heating elements. One heating element was used for fast warmup and the other for fine temperature stabilization when operating temperature was reached. The accelerometer also had this dual heater arrangement.

(6) The following slip ring changes were implemented:

(a) Ring and brush alloys were changed to reduce noise and increase operating life.

(b) "V" groove rings were used instead of "U" grooves for better brush alignment and tracking.

(c) The number of circuits through the slip rings was increased from 32 to 38.

(d) New brush forms and forming processes were incorporated to avoid brush "memory" distortion due to temperature.

(e) Different plastics were used for insulation barriers to minimize outgassing.

(f) Stiffer and more rigid ring assemblies were used to reduce ring deflections in vibration and acceleration environments.

2.2.2 (Continued)

(7) Phase shift resolvers were added to the pitch, yaw and roll gimbals to provide attitude information to the digital computer. Centaur had used servo repeaters with digital pickoffs which were located in the computer. Further discussion of this method will appear under Paragraph 6.1.4.

(8) A malfunction detection system was added to denote gross failures in the IMU accelerometer and gimbal control loops. Two malfunction lights marked "ACCEL" and "GCA" were located on the instrument panel to inform the astronauts of gross errors. The following is a list of parameters monitored by the malfunction system:

(a) Presence of gyro signal generator excitation

(b) Absence of gimbal stabilization signals for loops 1, 2, and 3

(c) Presence of 57.6 KHz timing reference

(d) Presence of accelerometer rebalance pulses

(e) Presence of critical power supply voltages

(9) The platform gimbal sequence on Gemini was changed from Centaur by interchanging the pitch and yaw gimbals. This resulted in the third gimbal in the Gemini configuration providing yaw axis rather than pitch axis freedom. This change was desirable because, with a standard gimbal sequence, a 90° displacement of the third gimbal would have required an outer roll flip (180° rotation) or an equivalent outer roll torque motor reversal to maintain inertial stabilization. A gimbal flip requires better servo response and more torque if performed under high linear acceleration, such as in re-entry. During re-entry, the platform was left inertially fixed, and pitch angles

2.2.2 (Continued)

exceeded 90° due to earth curvature and vehicle pitch. Therefore, gimbal flip was avoided during re-entry with reorientation of the gimbals.

(10) Incorporation of the spin motor interruption technique (SMIT) was planned to improve the drift rate repeatability of the gyro. One factor which affects gyro drift is spin motor magnetic pole location. The gyro drift rate will vary with the changes in the point of rotor "lock on" with the synchronous rotating field. Each time the spin motor is energized it may lock on a different point on the rotor giving a different drift rate. With SMIT, the spin motor excitation is interrupted periodically causing the lock-on point to vary. The drift rate will visit all points within the drift band and thereby have a more predictable drift rate. Phase angle shift technique (PAST) was later incorporated in place of SMIT because of its higher reliability. This technique accomplished the same purpose as SMIT, but was more reliable because it rotated each phase of the gyro three-phase excitation 30° at predetermined intervals instead of interrupting the spin motor excitation.

(11) An active conductive cooling system was used on Gemini instead of the radiation cooling system of Centaur. This necessitated the use of a platform mount which contained the coldplate for the platform.

<u>IMU Packaging</u> - The IMU was divided into four basic packages: platform, system electronics, gimbal control electronics (GCE), and static power supply (SPS). The SPS and GCE were bolted together to form one package. The SPS also contained the DC power supplies for the computer and a 400 Hz

2.2.2 (Continued)

source for the entire G&C system. This common power supply design was used to minimize weight and volume. Figures 2.2-2 through 2.2-4 show the construction of these packages.

Abort From Orbit Study - An abort from orbit would require fast alignment of the platform assuming worst case condition with the platform off. The requirement was that alignment be such that yaw axis would be gyro compassed to within 1° of 180° azimuth prior to initiating retrograde. The sequence of events was (1) platform fast heat warmup, (2) platform cage BEF (blunt end forward) with the spacecraft manually oriented to local horizontal BEF using the ACME system, and (3) platform alignment BEF. The time required to perform this sequence is about 44 minutes.

Since over half the time required to align the platform is consumed in the fast heat warmup cycle, trickle heating of the platform was considered. However, too much power would have been required to trickle heat the platform to operating temperature and would have reduced the time in orbit approximately one day in a 14-day mission.

Another modification considered was to increase the gimbal torque amplifier (GTA) and torquer capability to 100 ma. With this modification and trickle heating of the platform, alignment time would have been reduced to approximately 10 minutes. This modification would have required a major redesign of the platform.

A fast heat cycle change was also considered. It would have reduced the normal warm-up time from 23 minutes to 17 minutes by adding 50 more watts

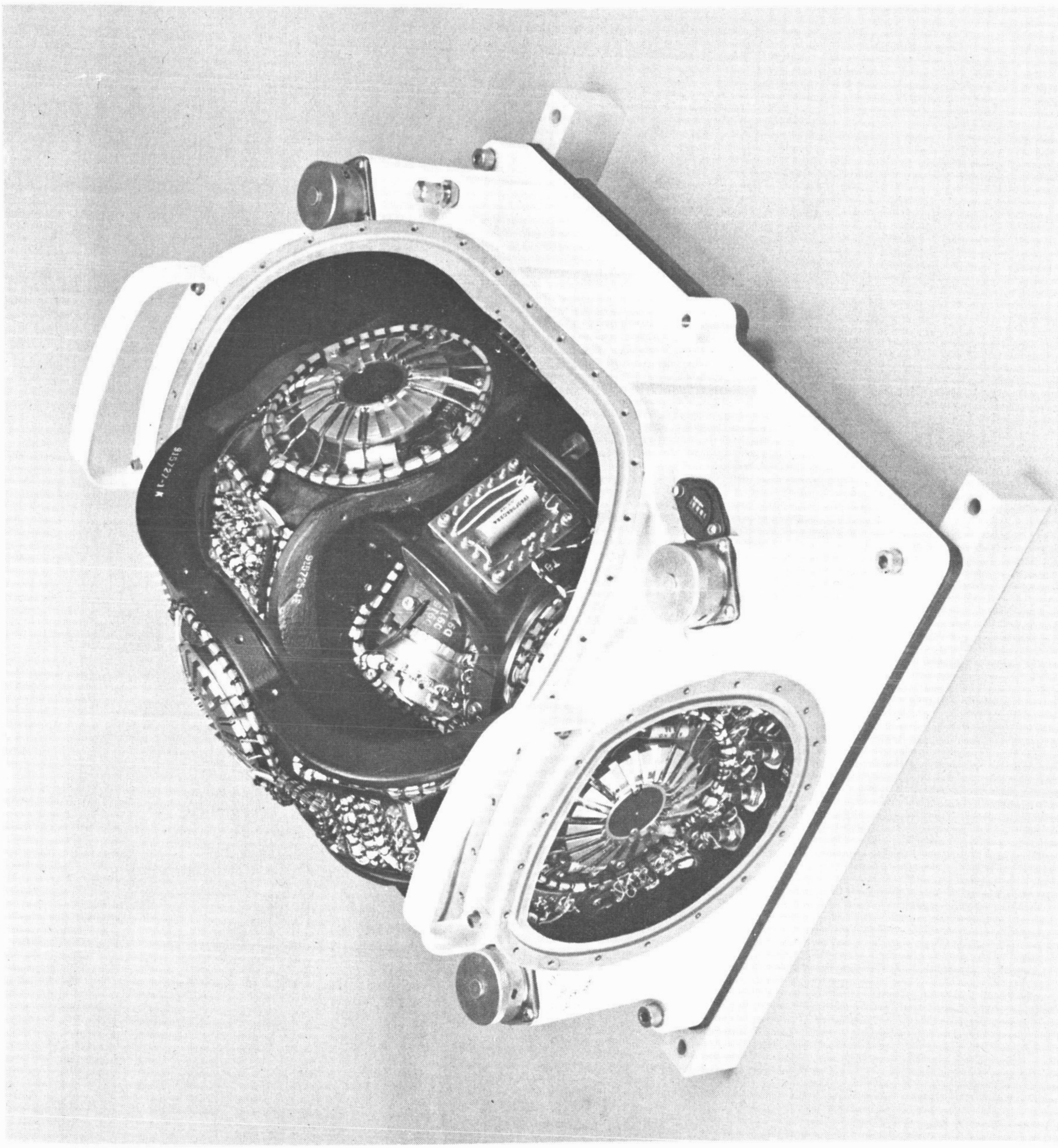


FIGURE 2.2-2 IMU PLATFORM (COVER REMOVED)

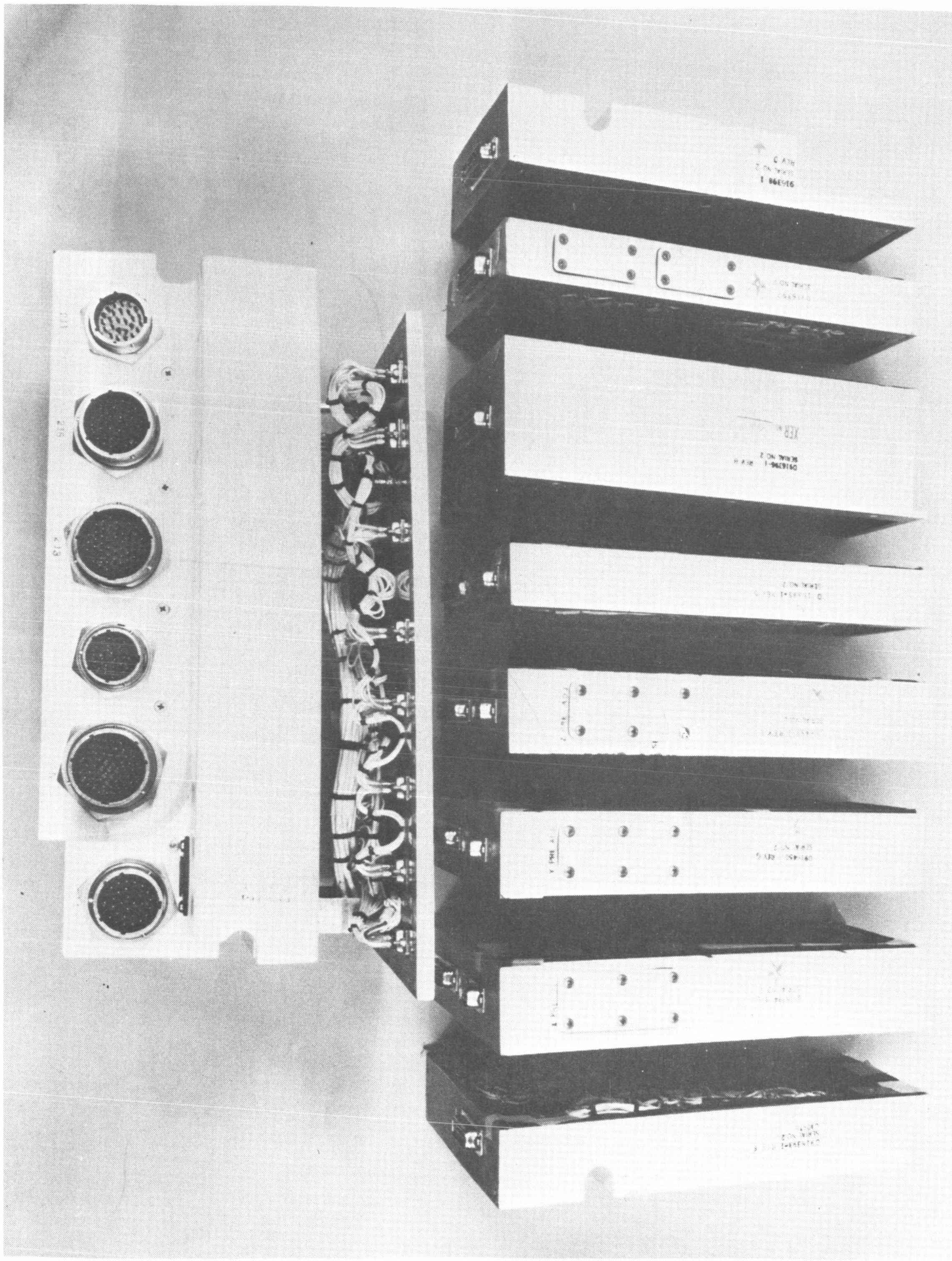


FIGURE 2.2-3 IMU SYSTEM ELECTRONICS (EXPLODED VIEW)

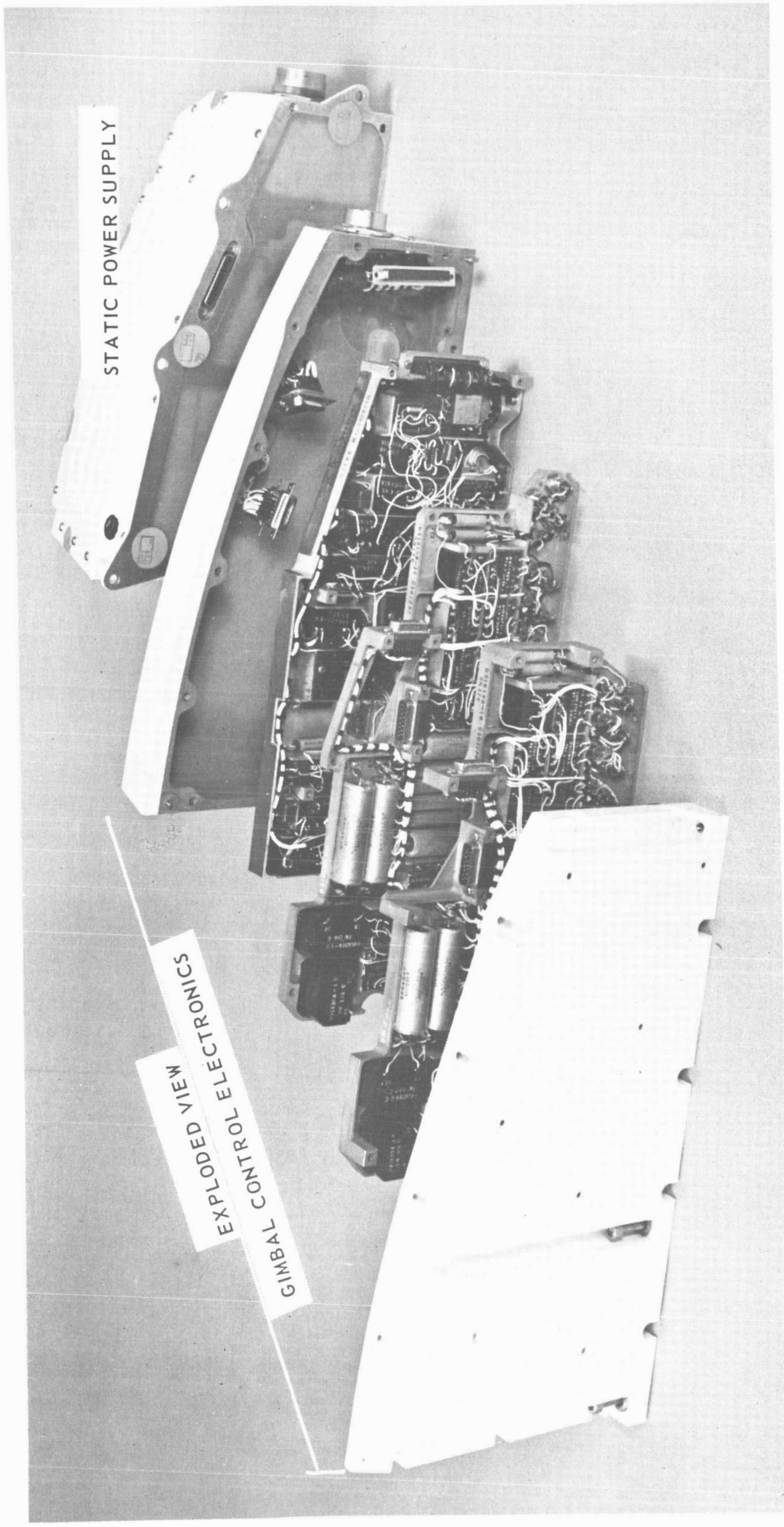


FIGURE 2.2-4 IGS POWER SUPPLY

2.2.2 (Continued)

to fast heat heaters. This change would have required an increase in the capacity of the ±40 VDC power supplies and additional slip rings on the platform to carry the additional heater current.

A fast heat override switch was also considered for use with the fast heat cycle modification. The astronauts could override the fast heat cycle at approximately 11 minutes and start alignment of the platform with degraded accuracy.

It was concluded at the end of this study in December 1962, that no design changes were required. If an abort from orbit in less than 60 minutes was necessary, a manual re-entry would have to be performed.

2.2.3 Development Problems, Tradeoffs, and Design Decisions -

Gyro and Accelerometer 100 Hour Run-In - A 100-hour run-in was initiated on gyros and accelerometers to eliminate infant mortality failures and to provide stability data over run-up, cool-down, run-up periods. When accelerometers or gyros indicated stable characteristics, they were accepted for platform installation. Acceptance percentages using this method were approximately 75 for the gyros and 85 for the accelerometers. This run-in program was initiated in early 1963 and applied to all production hardware.

Conclusion - We recommend that inertial components be operated periodically while in storage. If inertial parameter changes occur after non-operating periods, further operation of the device prior to rechecking characteristics may be advisable.

2.2.3 (Continued)

Gyro Eutectic Melt Temperature Instability - A eutectic capsule was mounted on the gyro gimbal for balance trimming. A problem caused by this capsule, appearing as a large mass unbalance drift, was detected by the subcontractor on two gyros. The cause was traced to the tin lining of the capsule. Ternary alloying had occurred due to the migration of the tin into the eutectic alloy, lowering the eutectic melt temperature. When the gyro was at operational temperature, creeping of the contaminated eutectic alloy occurred and caused mass unbalance of the gyro. This problem was corrected, on gyros built after the Gemini gyros, by gold plating the trim capsule to prevent ternary alloying. The change was not incorporated during the Gemini program because the infrequent occurrence of the problem did not warrant the added expense of modifying existing gyros.

Fluorolube Leakage (Gyro Main Header Seal Failure) - Fluorolube was used as a gyro floatation fluid. Leakage of the Fluorolube ultimately resulted in gas bubbles forming in the gyro's fluid cavity. These bubbles introduced a transient mass instability, causing erratic performance, and excessive drift rates. Two gyros exhibited this type of failure. The first, the pitch gyro on the Spacecraft 3 platform, caused excessive gyro drift during launch, accentuated by the motion of the bubbles due to the launch acceleration. The second was discovered after removal of the header dust cover when dismantling a gyro to look for another failure. The subcontractor stated that the leakage was caused by seal failures due to inadvertent handling abuse or to the method by which the epoxy sealent was applied. As corrective action, three pressure leak checks were made prior to assembly

2.2.3 (Continued)

of the dust cover at the gyro level. A dynamic tilt test was instituted at the IMU level to determine if bubbles were present in the fluid cavity.

Conclusion - Gyros and accelerometers should be designed so that the Fluorolube seal can be visually observed. An output axis tumble test should be performed at the gyro level, along with the dynamic tilt test at IMU level. Repetitive thermal cycling tests should also be performed, with visual checks for fluid leakage at the component level.

Oscillatory Drift - During acceptance testing on the first production unit, drift trim difficulty was encountered because the gyro drift changed continuously. The amplitude of this oscillation was approximately 1 arc minute with a period of 2 to 35 minutes. The amplitude and frequency of this cyclic drift was dependent on gimbal orientation. This problem was caused by a 400 Hz voltage from the spin motor coupling into the gyro signal generators. This set up a mechanical oscillation in the gimbals via the gimbal torquers. This oscillation beat with the rotation of the other two spin motors to produce a drift whose period was determined by the beat frequency, and an amplitude that was a function of the magnitude of the gimbal oscillation and spin-motor imbalance. The oscillating drift observed was the result of the difference in the rotational speed of the gyros. Variation in synchronous speed appeared to be caused by the phase angle shift technique (PAST), which varied the synchronous speed slightly from gyro to gyro.

2.2.3 (Continued)

This cyclic drift was eliminated by placing a 400 Hz notch filter in parallel with the sum amplifier for gimbal loops 1, 2, and 3. The best approach would have been to eliminate the spin motor noise pick-up in the gyro; however, schedule pressure did not allow this approach.

Slip Ring Problems - Numerous slip ring problems were observed on the platform during the Gemini Program. The original slip rings, used on engineering models and the first four production models, exhibited high brush-to-ring resistance. Construction of the brushes and brush blocks made it difficult to control alignment of the brushes during assembly. Contamination of the rings caused by brush block chips and bearing lubricant also contributed to the problem.

In August 1963, a new slip ring assembly, employing some concepts from Saturn slip rings, was incorporated in Gemini platforms. However, intermittently open and noisy signals were still observed. Intermittently and continuously open slip rings were caused by the polymerization and deposition of organic materials on the slip ring contact surfaces. To prevent organic contamination, the following changes were incorporated:

(1) The slip ring assembly material was changed to reduce epoxy outgassing.

(2) The amount of lubricant in rotary component and slip ring bearings was reduced.

(3) Procedures were initiated in late 1964 to prevent build-up of organic material on the slip rings by wiping the rings periodically (10 rotations of platform CW and CCW around each axis).

2.2.3 (Continued)

(4) The platform pressurizing gas mixture was changed from 30 percent helium and 70 percent nitrogen to 30 percent helium, 50 percent nitrogen, and 20 percent oxygen. The added oxygen apparently kept "Loctite" (organic staking material for platform screws) from hardening and adhering to the rings. The "Loctite" was an outgassing product deposited on the slip rings. This change, incorporated in March 1965, noticeably reduced noise on slip rings.

Another anomaly involved shorted slip rings due to dielectric breakdown of insulation material on the leads leading into the slip ring assembly. The primary cause of this problem was the elevated temperature in which the rings operated in the platform. The design of internal leads was changed to reduce breakdowns, and temperature cycling and soak tests were initiated to check dielectric characteristics of each slip ring assembly prior to installation.

A hard gold plate finish was incorporated on the slip rings in April 1965, to reduce galling. A run-in test was added with a subsequent visual inspection of the slip rings for galling prior to installation.

<u>Conclusion</u> - The use of organic materials whose outgassed residues are nonconductive should be minimized. In addition, the initial design and usage of slip rings should be closely monitored.

<u>Addition of Horizon Sensor Ignore Light</u> - In June 1963, a spacecraft display light was added to inform the astronauts that the platform was not aligning. This light was controlled by the ignore relay in the system electronics package which was activated by the ignore signal from the horizon sensor.

2.2.3 (Continued)

The ignore relay's function was to open the pitch and roll horizon sensor signals to the X and Y gyro torquer loops. This light also informed the astronauts of horizon sensor failure if the light failed to extinguish after a reasonable time period.

Horizon Sensor Ignore Delay Problem - During Electronic System Test Unit (ESTU) testing, an incompatibility was observed between the IMU and horizon sensor following a horizon sensor loss of track. It was observed that the horizon sensor had to track the horizon approximately 3 to 6 seconds before its pitch and roll error signals were stabilized. This anomaly was corrected by adding a 6-second time delay in system electronics which delayed the dropout of the ignore relay.

Synchro Phase Shift Problem - During ESTU testing, it was noted that excessive phase shift in the platform synchro outputs widened the ACME control deadband. The problem was corrected by adding an inductance-capacitance network to the synchro rotor circuit, which reduced the phase shift from approximately 15° to 9° or less.

Conclusion - Interface characteristics should be closely coordinated between subcontractors during the initial design phase and interface tests should begin as early in the program as possible.

2.2.4 Problems and Design Changes During Development and Qualification Testing - Platform Vibration Problems - In April 1963, exploratory tests on the test bed platform showed the platform's performance to be degraded when exposed to a vibration environment. These tests consisted of low level

2.2.4 (Continued)

(±2 g peak) sinusoidal and random (to 6.3 g RMS) vibration. Limited performance information was obtained because of excessive electrical lead breakage on the inner gimbal. The platform resonance was determined to be approximately 100 Hz with a corresponding transmissibility to the inner gimbal of 17 g's per g input.

The corrective action taken after this Phase I vibration testing was as follows:

(1) Inner gimbal wiring was improved by rerouting, incorporation of stress loops, and addition of stainless steel wire bundle stiffeners.

(2) Various capacitors and resistors were attached more securely.

(3) Rotary component screw torques were increased and staked with cement.

Phase II testing, started after incorporation of the above modifications, consisted of low level sinusoidal vibration and random vibration up to 12.6 g RMS. Fewer mechanical problems were encountered than during the Phase I tests, but wire breaks occurred after some high level vibration exposure. In addition, a phase shift resolver capacitor broke loose and there was loss of mechanical integrity in the gimbal rotary components. Platform performance indicated excessive gyro drift rates, and gimbal 3 lost its inertial reference at approximately 9-10 g's RMS. The loss of reference condition was partially due to the fact that the higher torque capability gimbal drive motors mentioned previously were not incorporated in the test bed platform.

Phase I and Phase II testing indicated that the majority of the platform vibration problems could be attributed to the rotary components.

2.2.4 (Continued)

Phase III and Phase IV (mock-up) tests were initiated at the subcontractor's facility immediately after the Phase II tests. These tests revealed the rotary components were introducing the equivalent of relatively weak spring constants into the gimbal support system. In addition, unequal spring rates across a gimbal were observed. These conditions indicated extensive inner gimbal motion with cross-talk and rocking. The low rotary component spring rates indicated little stressing of the high damping capability of the "KIA" magnesium gimbals. Thus, the gimbals were not absorbing energy as the design intended.

A parallel study by the McDonnell structural dynamics group confirmed the hypothesis that the rotary components were the major problem, and the following platform modifications were incorporated:

(1) Rotary components were stiffened and their spring rates matched by adhesive bonding of stainless steel discs.

(2) Rotary component bearings were changed, incorporating higher contact angles and preloading was increased.

Phase V testing on the test bed platform with stiffened rotary components consisted of sinusoidal vibration up to ± 2 g's peak and random vibration up to 12.6 g's RMS.

Results showed losses of reference by gimbal 3, high gyro drift rates, and non-repeatability of data. The platform did not meet specification of 12.6 g's RMS, but the Phase V testing was considered to have verified the platform's structural integrity through 6.3 g's RMS, and performance through this vibration level was considered acceptable.

2.2.4 (Continued)

Phase VI tests were performed to evaluate possible platform vibration isolation. Annular isolators were installed between the platform case and gimbal 4. Vibration tests indicated this approach would permit satisfactory performance up to 12.6 g's RMS. However, cost and schedule considerations did not permit incorporation into the production units.

The original vibration specification (12.6 g's RMS) for the Gemini was lowered to 8.8 g's RMS; the IMU still did not exhibit specified performance at this new level. Acceptance tests on each unit were performed to 6.2 g's RMS with the requirement that the platform would not lose reference at this vibration level. In June 1965, the acceptance vibration spectrum was notched at platform resonance frequencies, as flight data indicated g levels at these frequencies could be reduced.

> Conclusion - Exploratory vibration tests should be performed as early in a platform's development phase as possible. Specified test vibration levels must be realistic. The addition of vibration isolators now available would permit the Gemini platform to be qualified at 12.6 g's RMS.

2.2.5 Problems and Design Changes Associated With Flights

GT-2 Flight Failure - Comparison of tracking data with the IGS outputs during the ascent phase indicated errors in these outputs. The anomaly was evidenced as an out-of-tolerance value for downrange velocity. This velocity increased rather than decreased at booster and second stage engine cutoffs.

2.2.5 (Continued)

The cause of the error was determined to be due to an accelerometer loop problem, later defined as "pulse lockup." The problem was evidenced by saturation effects, such that the loop introduced a significant lag prior to responding to an accelerometer polarity reversal. This lag was sufficient to cause the accelerometer pendulum to be torqued in the wrong direction. Post-flight centrifuge testing confirmed the saturation problem encountered during the GT-2 flight. The accelerometer rate compensation network was modified to correct this situation.

Conclusion - Dynamic tests should be initiated as early as possible on new designs using realistic acceleration inputs.

GT-3 Flight Problem - An excessive drift rate in the pitch gyro during ascent caused saturation of the pitch steering signal from 290 seconds after liftoff through second stage engine cutoff. This failure was caused by bubble migration in the Fluorolube of the pitch gyro, which is discussed in Section 2.2.3.

2.3 Horizon Sensor System

2.3.1 Requirements - The initial performance requirements for the Gemini horizon sensor system and its final capability are shown in Table 2.3-1.

2.3.2 Mercury Horizon Sensor Experience - Mercury horizon sensor experience greatly influenced Gemini design. On Mercury, two sensors were employed, one each for the pitch and roll axes. A conical scan technique was used to detect the infrared discontinuity between earth and space. The scan's intersection of the earth's radiation level produced a square wave output from the IR detector for each rotation of the scanner. Processing the output signal by "slicing" reduced the effects of the higher amplitude IR intensity variations and lower amplitude noise. The relation of the portion of the waveform produced by earth's radiation and a fixed spacecraft reference was used to produce a positive or negative DC output corresponding to the misalignment of the spacecraft axis with respect to local vertical.

The IR detector was a germanium immersed thermistor bolometer with a passband of 5 to 20 microns. The original design called for a 2 to 20 micron passband (produced by germanium filtering). The assumption that this wide band was satisfactory was based on a lack of knowledge, at that time, of the earth's IR characteristics. It was found later that variations were present at 2 microns, because solar reflection was greater than expected. Therefore, the passband was reduced to 5 to 20 microns.

The output of the thermistor bridge circuit was highly noise sensitive, since its output was only 50 to 100 microvolts. Shielding and filtering were added to the spacecraft to protect the sensors from transients on power lines, caused primarily by relay actuation, and from interference from the

TABLE 2.3-1

HORIZON SENSOR SYSTEM FUNCTIONAL REQUIREMENTS

	Final Capability	Initial Requirement
Null	$\pm$ 0.3° with IR anomaly $\pm$ 0.1° without IR anomaly, 50-150 nmi $\pm$ 0.3° without IR anomaly, 900 nmi	$\pm$ .3° 3σ
Scale Factor	Non-linear: $\sim$0.4 volts dc/deg. 0-14°	.4 volts dc/deg. @ $\pm$ 2°
Linearity	$\pm$ 10% of nominal curve 50-150 nmi $\pm$ 15% of nominal curve 900 nmi Repeatable, monotonically increasing $>$ $\pm$ 14°	0.1° between 0 and $\pm$ 2° 0.5° between $\pm$ 2° and $\pm$ 5° Repeatable, monotonically increasing $>$ $\pm$ 5°
Altitude	50-900 nmi Optimum operation @ 150 nmi	50-900 nmi Optimum operation @ 150 nmi
Noise Content	20 mv rms @ null 4 mv rms/deg + 20 mv overall	5 mv rms @ null 4 mv rms/deg + 5 mv overall
Cross Coupling	$<$.1° output in one axis for 2° tilt in other axis	Same
Warm Up Time	120 seconds	60 seconds

2.3.2 (Continued)

high frequency transmitter. One of the causes of this latter problem was the mounting of the sensors on and near the front of the spacecraft, which was the ground plane of the antennas.

Two major horizon sensor problem areas were encountered on Mercury flights - errors due to cold clouds and errors due to the sun. Cold clouds have a radiant energy level (in the IR bandpass used on Mercury) which is only about 10% to 30% of the radiant energy of the earth. The cloud temperatures were found to be as low as 150°K, compared to a normal horizon temperature of 240° to 280°K. This lower radiation produced notches in the bolometer output which could cause attitude errors of 2° to 35° in the sensor output. The exact magnitude of the error was dependent upon size and location of the cloud and its temperature.

These errors were reduced by lowering the slicing, or limiting, level of the bolometer output to 20% of the peak output. This level corresponded to a temperature of 190°K. Further reduction of the slicing level was not feasible due to limitations in the state of the art of bolometers, to the Johnson noise generated in the bolometers, and to the noise generated in the pre-amplifiers.

The sun-caused errors were of two types. The first occurred when the sun was near the edge of the horizon at the point intersected by the sensor's scan. The sensors contained an "ignore" circuit which was triggered by normal sun intensity in the field of view. However, with the sun near the horizon, its intensity was attenuated by the atmosphere and would not trigger the ignore circuit. The sun, therefore, gave the effect of "moving" the

2.3.2 (Continued)

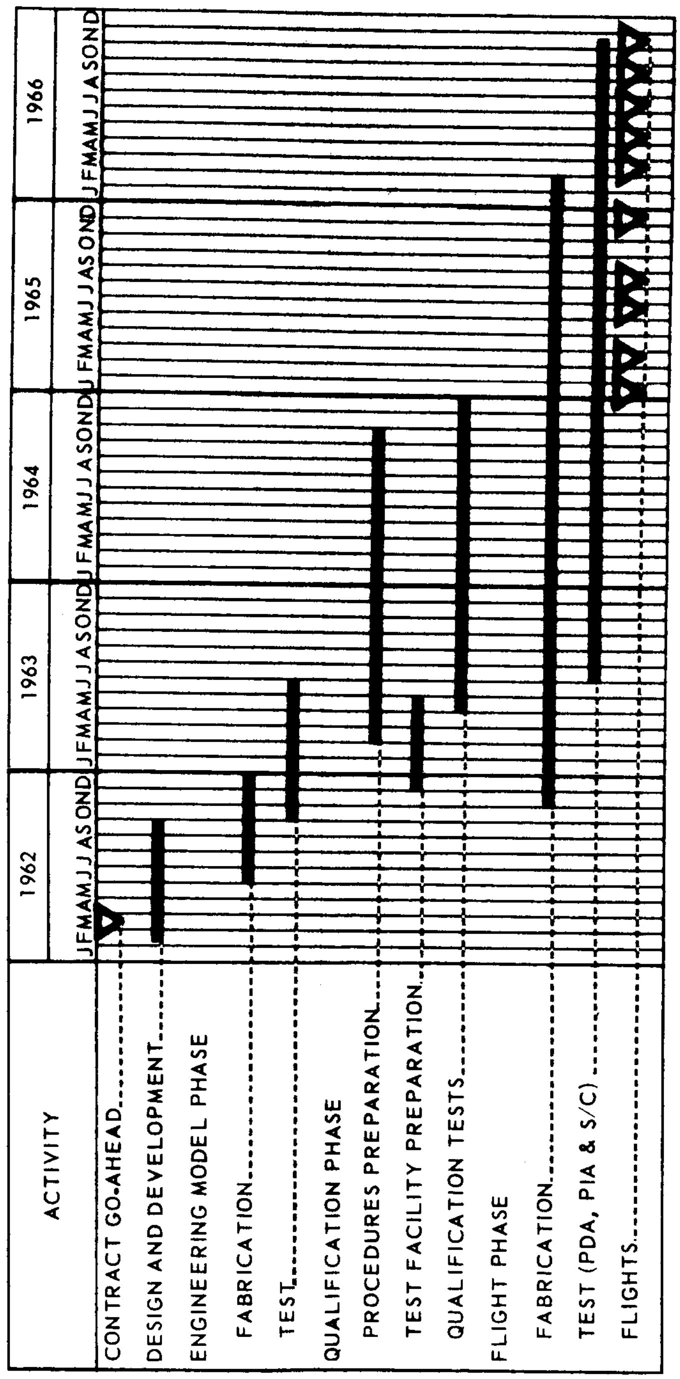
horizon edge out by about 3° to 6°, giving an effective 1.5° to 3° output error.

The second sun-caused error resulted when the sun, although not directly in the field of view, was near enough for sun-radiated energy to reach the bolometer and cause an output error. This error was referred to as "sun-tail-off" error and lasted for several minutes intermittently throughout the daylight portion of the Mercury orbits.

2.3.3 Gemini Horizon Sensor Early Design Decisions and Trade-Offs - The time history of the Gemini horizon sensor design is shown in Figure 2.3-1. In addition to supplying attitude control reference, the Gemini horizon sensor was required to provide an accurate reference for alignment of the inertial platform for on-board guidance. The original concept utilized both a coarse and a precision horizon sensor. The precision sensor would have been employed for alignment of the inertial measurement unit, and the coarse one for attitude control in orbit (Section 4.1). However, the system, as mechanized, employed a single precision sensor to fulfill all mission requirements. Two identical systems were included for redundancy. Each system consisted of two packages, a head and an electronics package, shown in Figure 2.3-2.

In order to avoid some of the pitfalls of the Mercury sensors, a system employing edge tracking was chosen rather than the conical scan method employed on Mercury. With the "across-the-earth" type scanning inherent in a conical scanner, there was no method of eliminating anomalies being caused by the sun near the horizon. The edge tracking concept employed a

FIGURE 2.3-1
HORIZON SENSOR PROGRAM TIME HISTORY



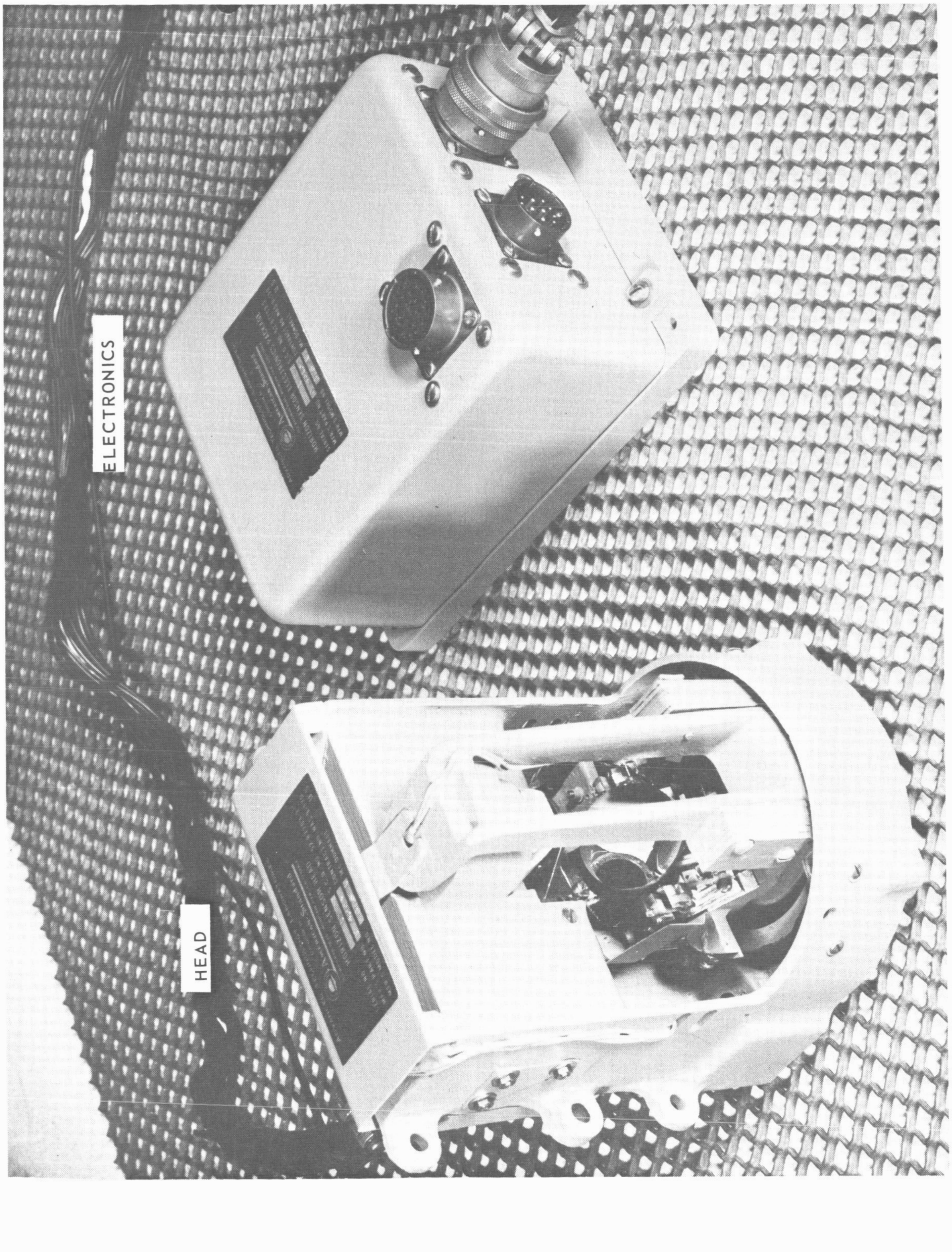


FIGURE 2.3-2 HORIZON SENSOR SYSTEM

2.3.3 (Continued)

tracker which locked onto the earth-space IR gradient. The output signal therefore was not a direct function of the magnitude of the IR gradient. Also, cold clouds on the face of the earth would not affect the edge tracker as much as they would a conical scan device. Section 2.3.5 discusses some of the sun and cold cloud problems that were still evidenced with the Gemini sensors. These proved not as large an error source as that on Mercury.

Azimuth scanning was used to derive pitch and roll information from a single sensor. This method derived its outputs from a 160° azimuth scan rather than from single points on the horizon, and thus provided higher accuracy.

Selection of the optimum IR spectral bandpass was important in designing the horizon sensors. The IR horizon varies less at certain wavelengths than at others. The most consistent bandpass is known to be 14-16 microns, but, at the time of initial design, no filters were available for this bandpass. Filters were available for the next best choice - 13.5 to 22 microns - but these filters were subject to deterioration when subjected to high humidity. Since the Gemini environmental requirements dictated that the sensor must not be affected by humidity, this second-best bandpass was rejected. The bandpass of 8 to 22 microns was finally selected as the best compromise. This bandpass eliminated the effects of reflected solar energy (wavelengths less than 4 microns) and the vibrational absorption bands of water (6.33 to 6.85 microns). Later in the program, a 13.5 - 22 micron system was developed and flown. It is discussed in Section 2.3.5.

2.3.3 (Continued)

Mounting Location - The following considerations influenced the ultimate location of the horizon sensors on the spacecraft re-entry module (shown in Figure 1.2-3):

(a) Platform orbital alignment accuracy requirements.

(b) Astronaut and sensor field of view requirements during various phases of the missions.

(c) Aerodynamic heating effects from horizon sensor head projection.

(d) Crosscoupling and non-linearity errors introduced by offsetting the horizon sensor axes from the spacecraft axes.

Due to configuration and field of view requirements, the heads containing the optics were mounted so as to project beyond the side of the spacecraft. A fairing was therefore required during launch, to prevent aerodynamic heating. (The fairing was jettisoned after launch.) The heads were jettisoned just prior to re-entry, for the same reason.

The location of the sensors was changed twice during the early stages of the program. Since the heads could not re-enter with the re-entry module, the most desirable location was on the adapter, which was jettisoned prior to re-entry. However, in order for the sensors to align the inertial platform with the required accuracy, the physical alignment error between the sensors and the platform had to be as small as possible. A maximum misalignment of 0.05° was hoped for. In this location, the alignment uncertainties in mating the adapter and re-entry module were too great, so the sensors were moved to the side of the re-entry module, near the IMU. However, a thermodynamic analysis indicated that air flow around the

2.3.3 (Continued)

horizon sensor fairing might cause severe aerodynamic heating of the adapter during launch. Therefore, in mid 1962, the horizon sensors were moved to the re-entry module, near the RCS thrusters, with a slight sacrifice in platform alignment accuracy.

At this time, it was planned to mount the sensors with a 5^{o} offset in pitch and a 10^{o} offset in yaw. The pitch offset was through to provide the astronauts with the best view of the horizon during various phases of the mission, while allowing the sensors to be operated in a linear portion of their field of view, even though the offset necessitated a bias network in the IMU. Studies conducted in late 1962, however, indicated that better performance would be obtained with the horizon sensor pitch axis aligned with the spacecraft pitch axis, so that alignment would be performed at horizon sensor null. This orientation was still compatible with the astronaut and radar field of view requirements for rendezvous, pre-retro fire, and attitude control, since the astronauts would still have a good view of the horizon with the spacecraft horizontal.

The yaw offset was introduced to minimize the fairing size. In 1962, the yaw offset was increased to 14^{o} to further reduce the fairing size required. This necessitated a compensation network in the horizon sensors to transform the angles detected by the horizon sensor into spacecraft angles.

2.3.3 (Continued)

Input Power for Horizon Sensors - The sensors were originally planned to be supplied with spacecraft DC power. Early in the program (mid-1962), the power source was changed to 26 volt 400 Hz AC from the IGS power supply or ACME inverter, to improve overall spacecraft power efficiency.

The use of 400 Hz power caused some difficulties when the horizon sensor's azimuth drive "kick" pulses caused distortion on the IGS 400 Hz power. These, however, did not affect other systems seriously enough to warrant modification.

2.3.4 Problems and Design Changes During Development and Qualification Testing - Output Circuit Changes - In late 1962, it was found that the 750 ohm output impedance of the sensor was incompatible with the interfacing circuitry in the IMU. Changes in the horizon sensor scale factor occurred as a result of mode changes of the IMU. A new specification of less than 100 ohms was placed on the output impedance, and the design was modified incorporating operational amplifiers instead of the previous emitter follower circuits. The first operational amplifier design proved to be unstable when capacitively loaded by spacecraft cabling, so the circuit was redesigned to yield a larger phase margin.

Horizon Sensor Electromagnetic Interference Problems - The horizon sensor was highly susceptible to radiated and conducted electromagnetic interference (EMI), due largely to its requirement for handling the small signals from the IR detector. The system originally did not have adequate protection from external interference sources.

2.3.4 (Continued)

Extensive testing was conducted on the horizon sensor in late 1962 to determine means of reducing EMI with the least sacrifice in cost, size, and weight. Results of early integrated system tests in late 1962 and early 1963 were also used to determine particular EMI problem areas. As a result of this testing, more shielding was added to spacecraft wiring, and filtering was incorporated.

The horizon sensor power supply was particularly susceptible to starting transients from the 26 volt 400 Hz power sources, the IGS power supply, and the ACME power inverter. The ACME inverter was modified to reduce its starting transient to 60 volts (see Section 4), and a suppression network was incorporated in the horizon sensor to eliminate the effects of these transients. Most of these changes were added in late 1963 and early 1964.

Design changes were made to improve the horizon sensor signal processing, which reduced the effects of EMI. Early in the program (late 1962) the demodulator in the output signal processing was changed from half wave to full wave. In addition to reducing the effects of noise, this change provided better separation of the one and two Hz signals which corresponded to the pitch and roll signals.

The pre-amplifier was redesigned in early 1964 to incorporate a field effect transistor (FET) in its input stage. The FET's high input impedance was more compatible with the varying low resistance of the bolometer, and provided a better signal-to-noise ratio.

2.3.4 (Continued)

The system's electromagnetic compatibility could have been improved further by incorporating a separate power supply for the bolometer and pre-amplifier. These low level circuits were the most noise sensitive, and isolation of their bias power would have improved their noise tolerance.

Vibration Problem With Positor Springs - The Gemini horizon sensor tracked the horizon by means of a device called a Positor, which was developed by the horizon sensor subcontractor. The Positor consisted of a mirror mounted on the rotor of a permanent magnet type torquer. Rotating with the mirror were two drive coils, moving in a cylindrical air gap. Superimposed on the DC flux from the permanent magnet was a high frequency AC flux, introduced by field coils. The Positor was driven by a current through the drive coils and the high frequency "field coil" EMF induced in these same coils provided an indication of mirror position. The Positor was driven at 2 cps over the entire elevation scan of the sensor while searching for the horizon and was dithered at 30 cps about the edge of the horizon while tracking. When tracking, the Positor, together with the detected IR signal from the bolometer, formed a closed loop so that the mirror would track the edge of the horizon.

The Positor mirror was originally pivoted by means of flexure springs, which gave promise of long operational life since no bearing friction was present. These springs, however, fractured during 12.6 g RMS vibration tests. The Positor was redesigned, using bearing mounts with coil springs to better withstand the vibration. However, during this redesign, the spacecraft vibration test level was reduced first to 9.5 g's RMS and subsequently to

2.3.4 (Continued)

8.8 g's RMS, because of additional knowledge gained about the booster's characteristics. The flexure pivots were never restored to the Gemini sensor, but the feeling was that they could have withstood the lower vibration level. Since the time of the redesign (early 1964), flexure pivots have been designed to withstand much higher vibration levels than the original 12.6 g's, and therefore their use would now be recommended on a similar device, especially for long-term missions.

Lubrication of the Positor pivots posed a unique problem. The proximity of the bearings to the mirror surface dictated a lubricant with extremely low outgassing properties, since any film deposited on the mirror would attenuate the IR input to the sensor. Proper lubricant selection eliminated problems due to this phenomenon.

Conclusions:

(1) Bearingless pivots are preferable for supporting exposed moving parts, contingent on satisfaction of environmental requirements.

(2) Materials with low outgassing properties should be used near exposed optical devices, to avoid optical attenuation by deposition of outgassed products.

Breakage of Azimuth Drive Torsion Rod - The azimuth drive consisted of a yoke assembly which rotated in a 160° arc at a natural frequency of 1 cps. The drive force was applied by "kick" pulses near the center of each swing. A torsion rod was employed to impart the restoring force. The rods were machined from an extremely hard spring material and, due to its brittleness and the difficulty in machining it, breakage was frequent during

2.3.4 (Continued)

development. In an effort to reduce the breakage susceptibility, the torsion rod was lengthened, increased in diameter, and changed from a brazed-on end rod to a barbell configuration. Some of the breakage had been due to "cork-screwing" scratches resulting from grinding the rods to "tune" the oscillatory yoke drive to 1 Hz. A new centerless grinding technique was developed which eliminated this defect.

Bearing Problems - In designing equipment for the space environment, exposed moving parts are avoided where possible. In the case of the Gemini horizon sensor, this was impractical because of the requirement for azimuth-scanning the horizon. Problems were experienced in operating the azimuth yoke bearings at temperature extremes, and also in finding a suitable lubricant.

It was found during testing that after a long time in a vacuum at cold temperatures, the yoke assembly would tighten up and the azimuth drive motor would not drive the yoke to its proper limits. This problem was due to a combination of lubricant aging, outgassing of the lubricant, the tight bearing clearance, and the differential expansion of the stainless steel bearings and the surrounding magnesium casting. A change was incorporated to the system in early 1965 which increased the clearances of the drive bearings.

Conclusions - Recommendations for future systems, regarding bearings, would include:

(1) Isolate moving parts from the space environment where possible.

2.3.4 (Continued)

(2) Provide adequate drive power to overcome any bearing friction due to environmental extremes.

(3) Pay particular heed to outgassing properties of lubricants.

Hermetic Sealing of Horizon Sensor Head - The head of the horizon sensor was exposed to the space environment. The primary reason for not sealing it was the difficulty in designing a cover which would be transparent to the IR wavelengths of the sensor. The problem was complicated by the geometry of the sensor. A window would have to be curved to accommodate the 160° azimuth scan. Germanium would have been optically the best, but machining it would have been difficult. Also it is easily scratched, and oxidation due to fingerprints would have been unrepairable.

"Saran Wrap" was found to be a suitable cover to avoid contamination during ground handling, but would have been difficult to use as a permanent cover. It possessed an IR opaque critical angle of 38°, so anomalies were introduced when its surface was wrinkled. Also, in its conventional form, it was fragile and easily scratched.

Obviously, a suitable sealed cover would go far in solving the bearing problem, and also would eliminate contaminants.

Horizon Sensor Ignore Circuit - The Horizon Sensor would send out an ignore command to interfacing subsystems whenever the sensor would lose track of the horizon and go into its search mode. This command was

2.3.4 (Continued)

extinguished when the sensor regained track, and at this time the pitch and roll signals were again transmitted. A problem developed because the outputs took some time to settle when the sensor regained track. When the ignore command was extinguished, the inaccurate signals transmitted immediately after regaining track caused perturbations in the interfacing subsystems (IMU and ACME). To solve this problem, a time delay was incorporated in the IMU which delayed the receipt of the horizon sensor signals after track was regained.

Conclusion - In a future system, this time delay function could be performed in the horizon sensor by incorporating two track check circuits - a fast acting one to control signals internal to the horizon sensor, and a slower one to ensure valid signals before they are sent out to interfacing subsystems.

2.3.5 Problems and Design Changes Associated With Systems Tests and Flights - Spacecraft 3 Performance - The flight of Spacecraft 3 provided the first opportunity to evaluate horizon sensor performance under actual conditions. The inability to predict exactly the varying nature of the earth's infra-red transmissions precluded an accurate appraisal in ground tests, and made the data from this first orbital flight extremely significant.

The post-flight analysis of Spacecraft 3 data indicated many anomalies. These included 382 losses of track totaling 366 seconds (out of the 4 hour, 28 minute, 3 second duration of the mission) and several instances of wandering outputs, both of which degraded sensor performance. Of the losses of track, 265 were explainable by coincidence with spacecraft pitch or roll

2.3.5 (Continued)

attitude beyond the limits of sensor performance, or by switching between redundant sensors. An extensive analysis was conducted to determine the causes of the remaining losses of track and the wandering outputs. The factors taken into account included:

(a) Thruster plumes falling into the sensor field of view.

(b) The sun and moon in the sensor field of view.

(c) Weather conditions (cold fronts).

(d) Spacecraft switching transients.

(e) RFI from ground tracking stations.

The thruster firing and spacecraft switching transients showed a low probability of causing losses of track. Moon rise and set and activity of ground tracking stations showed no correlation at all.

The analysis of sun in the sensor field of view gave 100% correlation to the occurrence of wandering outputs and losses of track. These both occurred when the sun was between 0^{o} and 10^{o} above the horizon. The possibility of either of these occurrences can be explained by considering the relative position of the sun to the earth/space gradient, and the sensor's field of view. It was shown that the sun being as far as 10^{o} above the sensor's field of view could produce output errors when the sun's IR energy is less than the earth's. When the sun's IR energy appears equal to the earth's, the earth/space gradient disappears and a loss of track occurs. When the sun's energy exceeds the earth's, a reversal of phase occurs, driving the Positor toward space and producing an error, or the detector circuit saturates, giving a wandering sensor output.

2.3.5 (Continued)

During the flight of Spacecraft 4, relatively few losses of track occurred which did not correlate with the excessive spacecraft attitudes or sun conditions. The weather conditions at these occurrences corresponded to the same type of cloud formations that caused problems on Spacecraft 3. The reduction in occurrences was explained by the seasonal weather variations. Spacecraft 4 flew in June, under relatively clear conditions, whereas Spacecraft 3 flew in March, when clouds were frequent.

The data from the flights of Spacecraft 5 through 8 was not analyzed in the same detail as those from Spacecraft 3 and 4.

Spacecraft 9 Narrow Band Optics Modification - As stated previously in Section 2.3.2, the original horizon sensor design utilized optics with a 8 to 22 micron IR spectral bandpass. At the time of the Gemini design, it was known that this bandpass was not ideal, but tradeoffs dictated its use at the time. The performance on Spacecraft 3, discussed above, indicated deficiencies in the sensor's performance under certain conditions, and in order to reduce these effects, a modification was made to the sensor system to use a narrower IR bandpass, 13.5 to 22 microns.

The 8 to 22 micron bandpass contains two frequency bands which are more variable than others in the bandpass. They are the ozone absorber and water absorption bands, between 8.9 to 10.1 microns; and the water continuum window and high altitude dust particles, between 10.75 and 11.75 microns. These bands vary the radiant intensity contour and altitude as a function of the earth's seasonal and climatic conditions. These variations are more erratic in the presence of cloud formations, and are responsible for

2.3.5 (Continued)

The remaining anomalies were compared with weather data, and showed a high correlation to cumulus cloud formations. The distorted spectral radiant energy contour between space and earth will contain two major gradients, cloud/space and earth/cloud, when certain clouds appear near the earth's horizon. The sensor tracks the largest of the two gradients. If the space/cloud gradient is predominant, output variations will be minor, but major output wanderings or losses of track occur when the earth/cloud gradient predominates. At this time, the sensor will track the earth/cloud gradient instead of the earth/space gradient. If the earth/cloud gradient ends, the sensor will view the steady state earth IR radiation, and the system loses track. If the earth/cloud gradient ends with the true horizon in the tracking sensor's field of view, the Positor jumps to the true horizon, producing an output error.

Spacecraft 4 Performance - In order to confirm the post-flight analysis of Spacecraft 3 data, instrumentation was added to the Spacecraft 4 primary sensor system to monitor the amplified bolometer signal, Positor position detector output, and the sensor track check light. Special in-flight tests were conducted to evaluate sensor performance at sunrise and sunset and at moonrise and moonset, and the effect of thruster plumes. Telemetry data and pilot comments confirmed the sun's interference, with output variations and losses of track occurring with the sun between 10° above and $1/2^{\circ}$ below the horizon. The moonrise and set and thruster plume tests indicated no interference from these sources.

2.3.5 (Continued)

the behavior described under Spacecraft 3 performance. Thus, the blocking of these bands by the 13.5 to 22 micron bandpass optics could be expected to improve tracking accuracy of the sensor.

In addition to the improvement while tracking through clouds, the new optics could be expected to improve performance at sunrise and sunset, since the sun's irradiance is reduced by a factor of 4 with this bandpass.

The optimum bandpass would have been 14.1 to 15.6 microns, which contains the very stable carbon dioxide absorption band. However, this bandpass reduces the available IR energy by a factor of 4 over the 8-22 micron bandpass, and thus major modifications would have been required to the sensor optics and electronics to compensate for the low input energy. The modified sensor, with the 13.5 to 22 micron bandpass had approximately the same total IR energy available at the sensor's bolometer as with the 8-22 micron bandpass, so no change to the electronics was necessary. The total IR energy contained in the narrower bandpass was actually less, but transmission through the sensor's optics was improved by depositing the filter directly on the collimating lens, eliminating one element in the optical path. This change was possible because of improvement in deposition techniques from those available at the beginning of the program.

On Spacecraft 9, one wide band (8-22 micron bandpass) and one narrow band (13.5-22 micron bandpass) horizon sensors were flown. No quantitative

2.3.5 (Continued)

analysis of the narrow band versus wide band performance could be made during the flight because both sensors could not be operated simultaneously, but a qualitive analysis, based on astronaut comments, telemetry data, and evaluation of known losses of track, showed that the narrow band optics exhibited superior performance. The narrow band sensor experienced no losses of track or wandering output due to sun or cloud effects, as had occurred with the wide band sensor.

Conclusion - Based on this experience, we recommend employment of narrow band optics on a future sensor design. The very stable 14-16 micron bandpass would be optimum for a new design, where the signal processing could be designed to compensate for the lower input energy.

Failure of Azimuth Drive on Spacecraft 5 Sensor - During the Spacecraft 5 mission, the primary horizon sensor's azimuth drive ceased operating. An exact failure analysis could not be done because the sensor heads were jettisoned before the spacecraft's return to the earth. Based on analysis of available data, the failure was postulated to be caused by either a mechanical failure of part of the azimuth drive, by an obstruction that restricted yoke motion, or by an electrical failure in the drive circuit. Corrective actions were taken on all of these possibilities on systems in production.

2.3.5 (Continued)

A later failure of the same nature was experienced during production testing of Spacecraft 10. This time, the horizon sensor head was available for dismantling, and the cause of the failure was found to be the breakage of the azimuth drive band, which was made of extremely brittle "Elgiloy" spring metal. A small piece of epoxy, used during construction, was found on the drive band, which caused it to fatigue at that point where the band rolled over the drive pulley. This sensor was the only unit built previous to the Spacecraft 5 failure, and had not undergone a cleaning procedure instituted after that failure. Thus, it was felt that the failures of both units were caused by the same thing.

Conclusion - Based on these failures, a recommendation can be made, for future systems, to institute rigorous cleaning procedures for moving parts where metal fatigue may be caused by interference of small particles. Such mechanisms should be designed, where possible, so that foreign materials cannot enter the mechanism.

2.4 Rendezvous Radar System

2.4.1 Requirements - Rendezvous with an orbiting target vehicle dictated a system which measured range, range rate, and angle to the target vehicle, and was operable over the range of environments required. The initial functional performance requirements, together with the final demonstrated capability, are given in Table 2.4-1.

2.4.2 Initial Design Tradeoffs and Decisions - The following design approaches were investigated. Selection of the last, the angle-measuring (interferometer type) radar, was influenced by the tradeoff considerations which follow its description.

a. Hybrid 1 (Radar & Optics) - In this approach a fixed radar would have obtained range and range rate data while a simple optical system determined angular target data. The optical system consisted of an optical sight parallel to the radar boresight (longitudinal axis of spacecraft). Target angles would then be determined by manually aligning the spacecraft along the line of sight to the target and measuring body angles with respect to the inertial reference. Range, range rate, and platform angle data would have been automatically sampled by the computer at fixed intervals, or angular data could have been manually inserted by an astronaut.

b. Hybrid 2 (Radar & Optics) - Again, a fixed radar would have obtained range and range rate data, and an optical system with a first order manual rate correction would have been used to gather angular target data. This more sophisticated optical system was more accurate than that of hybrid 1 and would have included movable cross hairs so that sensed motion of the target could have been added to the gimbal

TABLE 2.4-1

RENDEZVOUS RADAR SYSTEM FUNCTIONAL REQUIREMENTS

Characteristic	Final Capability	Initial Requirement
Range		
Digital	500 ft to 180 NM	100 ft to 250 NM
Analog	20 ft to 300,000 ft	20 ft to 300,000 ft
Range Accuracy (3 Sigma)		
Digital	0.1% or 75 ft whichever is greater	0.1% or 50 ft whichever is greater
Analog	± 3% or 50 ft, whichever is greater, for ranges of 20 to 3000 ft. ± 4% or 400 ft, whichever is greater, for ranges of 3,000 to 30,000 ft. ± 5% or 5500 ft, whichever is greater, for ranges of 30,000 to 300,000 ft.	± 5% or 50 ft whichever is greater
Range Rate	-100 to +500 ft/sec	Same
Range Rate Accuracy (3 Sigma)	± 10 ft/sec plus 3% of range rate for ranges from 20 ft to 30,000 ft. ± 5% + 10 ft/sec for ranges from 30,000 to 300,000 ft.	± 5% or 1 ft/sec, whichever is greater, for ranges of 20 ft to 30,000 ft. ± 5% or 10 ft/sec, whichever is greater, for ranges from 30 to 300,000 ft.
Range Acceleration	± 5 ft/sec^2	Same
Angle (Azimuth and Elevation)	± 25 degrees	Same
Angle Accuracy (3 Sigma)		
Digital	± 9.0 mr on boresight ± 24.0 mr at 25 degrees off boresight	± 3.0 milliradians

TABLE 2.4-1 (Continued)

Characteristic	Final Capability	Initial Requirement
Angle Accuracy (Continued) Analog	$\pm$ 9.0 mr on boresight $\pm$ 24.0 mr at 25 degrees off boresight	0.2 degree for angles from 0 to $\pm$ 5 degrees 1.0 degree or 10% whichever is greater from 5 to 25 degrees
Angle Rate	1.5 deg/sec	Same
Angular Acceleration	$\pm$ 20 deg/sec^2	Same
False Alarm Time	1.0 hour	Same
Probability of Detection	.965	.999

2.4.2 (Continued)

angles. The astronaut was then required to insert angle and angle rate data into the computer manually when the target was centered. Range and range rate were inserted automatically as for Hybrid 1. Gimbaled optics were also considered, with angular data inserted in the computer as stated above.

c. Angle-Measuring Radar - Both gimbaled and interferometer types of radars were considered. The interferometer type provided the following advantages: 1) less volume, weight, and power; and (2) much simpler mechanization, which decreased susceptibility to vibration and facilitated sealing of lubricated parts against vacuum. While these advantages were offset by the higher accuracy capability of a gimbaled radar, the interferometer's accuracy was considered sufficient for the Gemini mission.

When these three systems were considered only in terms of sensors, both hybrid systems appeared to have a definite weight advantage. However, this advantage was offset by the additional fuel required to maintain the tighter attitude control essential to these hybrid systems. We believed the accuracy of Hybrid 2 to be comparable to that of the angle-measuring radar. However, the angle-measuring radar was less subject to human error and appeared to represent a lesser technical and schedule risk. The relatively poor accuracy of the less complex Hybrid 1 would have increased maneuver fuel consumption sufficiently to make this system the heaviest by far.

Since optical system reliability would have been in series with radar reliability for both hybrids, it was questionable if any gain in overall

2.4.2 (Continued)

system reliability would really have been achieved over the angle measuring radar system. Use of a range and range rate only radar would still have required that all the transponder circuitry and much of the radar circuitry function properly; so its reliability was considered only slightly better than that of the range and angle radar.

Development and qualification of a first order correction optical system, in addition to the range and range rate radar, would have cost considerably more than development of a range and angle radar.

Based on past Mercury experience, we questioned whether optical tracking could be maintained in daylight, especially when the sun shone into the cabin. We could not rule out the possibility of optical tracking being lost during a critical portion of rendezvous. The radar did not share this problem and hence had a better proven tracking capability. Also, target acquisition by radar was much easier.

The angle measuring radar offered greater flexibility and growth potential for future missions, and left the astronaut more time to perform other functions and experiments. Rendezvous would have required 2 to 4 hours of tedious, repetitive effort with the long-range optical angle measuring systems.

All radars considered used radomes and pressurized packages to avoid problems associated with lubrication of moving parts and unsealed operation in space.

2.4.2 (Continued)

In summary, the greater mission growth potential, proven tracking capability, improved human factors, and fuel savings of the angle measuring radar outweighed the possible but small advantage of the hybrid systems in reliability and radar system weight. An interferometer device was chosen over a gimbaled radar for angle measurement because of simpler mechanization and a size, weight and power advantage.

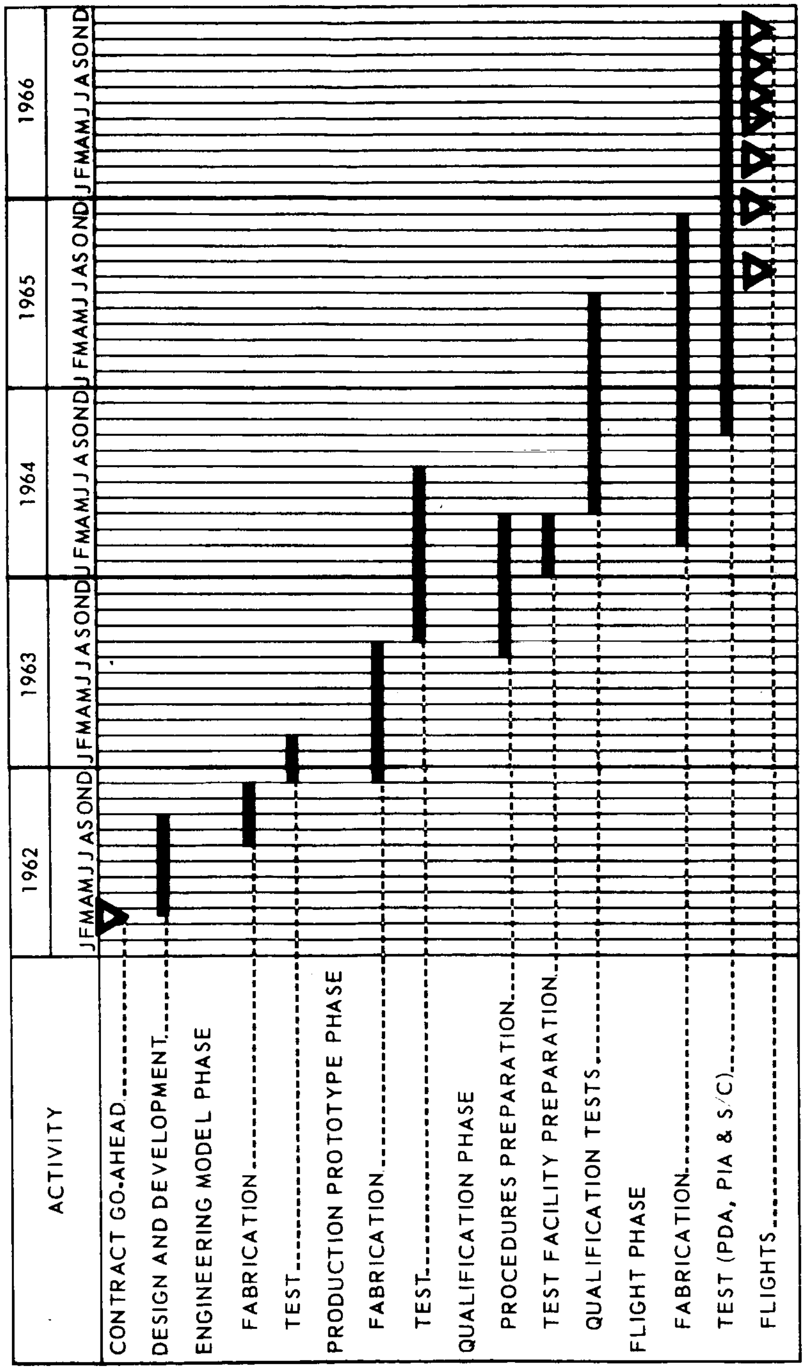
2.4.3 Development Problems, Tradeoffs and Design Decisions - Figure 2.4-1 provides a time history of the radar design, development and flight operations. This figure may be used to establish the occurrence of significant milestones in the design and development of the equipment. The radar is shown in Figure 2.4-2 and the transponder in Figure 2.4-3. The following paragraphs discuss early design tradeoffs and decisions, and any problems resulting from those decisions.

Servo Response - Preliminary studies of angle accuracy and angular tracking rate capability by the radar subcontractor stated that a 2 Hz servo response was required. Although McDonnell studies indicated that a lesser servo response should have been adequate, the initial design was allowed to proceed on this basis. However, angle accuracy measurements on engineering model units revealed that angular error was excessive with a 2 Hz response. The response of the flight units was reduced to 0.5 Hz without degrading the mission tracking capability.

Angle Ambiguity - Any phase-measuring device has an ambiguity if the phase relationship between the two received signals exceeds 180°. For the Gemini interferometer radar, the ambiguity occurred at a line-of-sight angle of

FIGURE 2.4-1

RENDEZVOUS RADAR PROGRAM TIME HISTORY



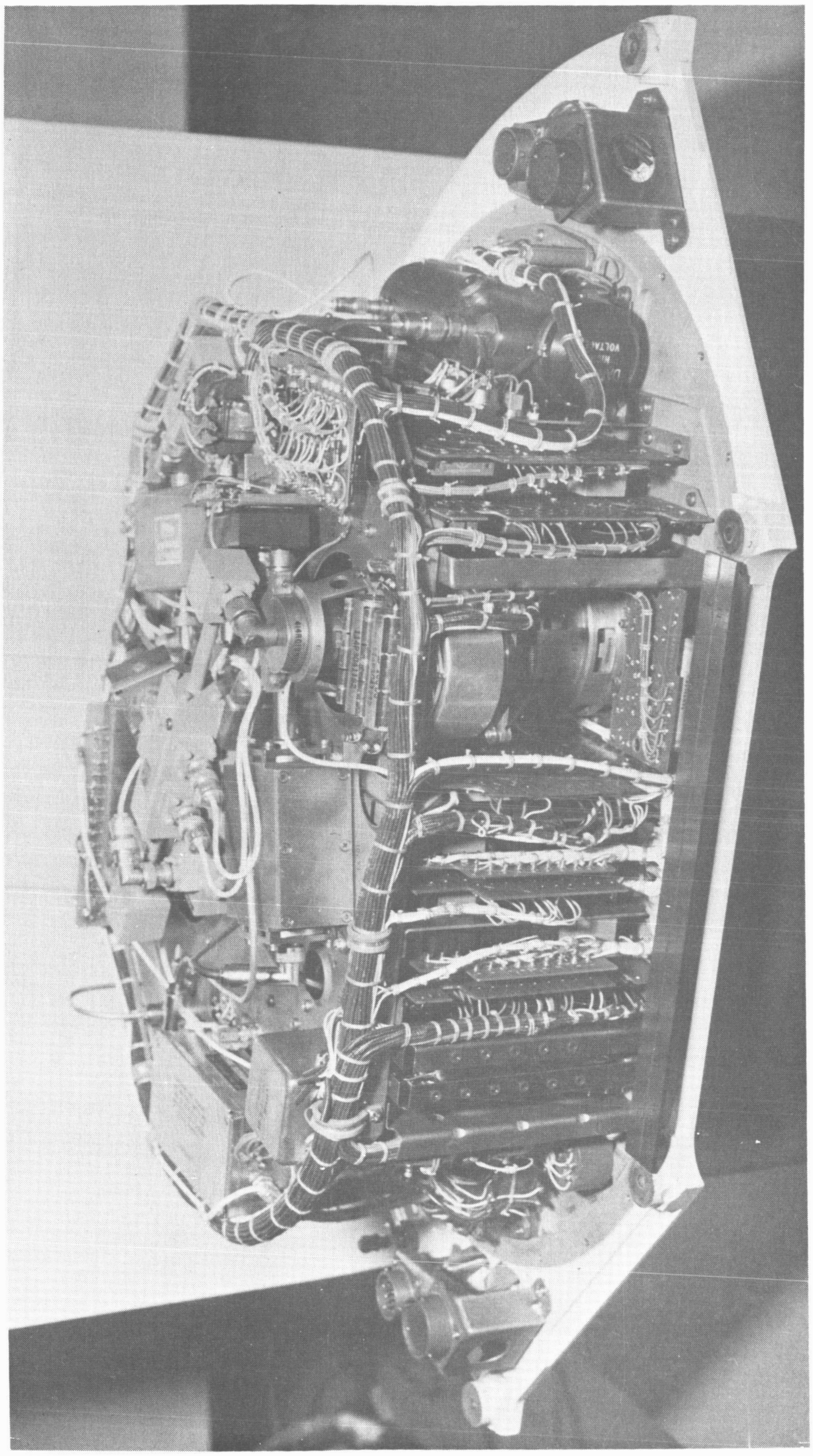


FIGURE 2.4-2 RENDEZVOUS RADAR (COVER REMOVED)

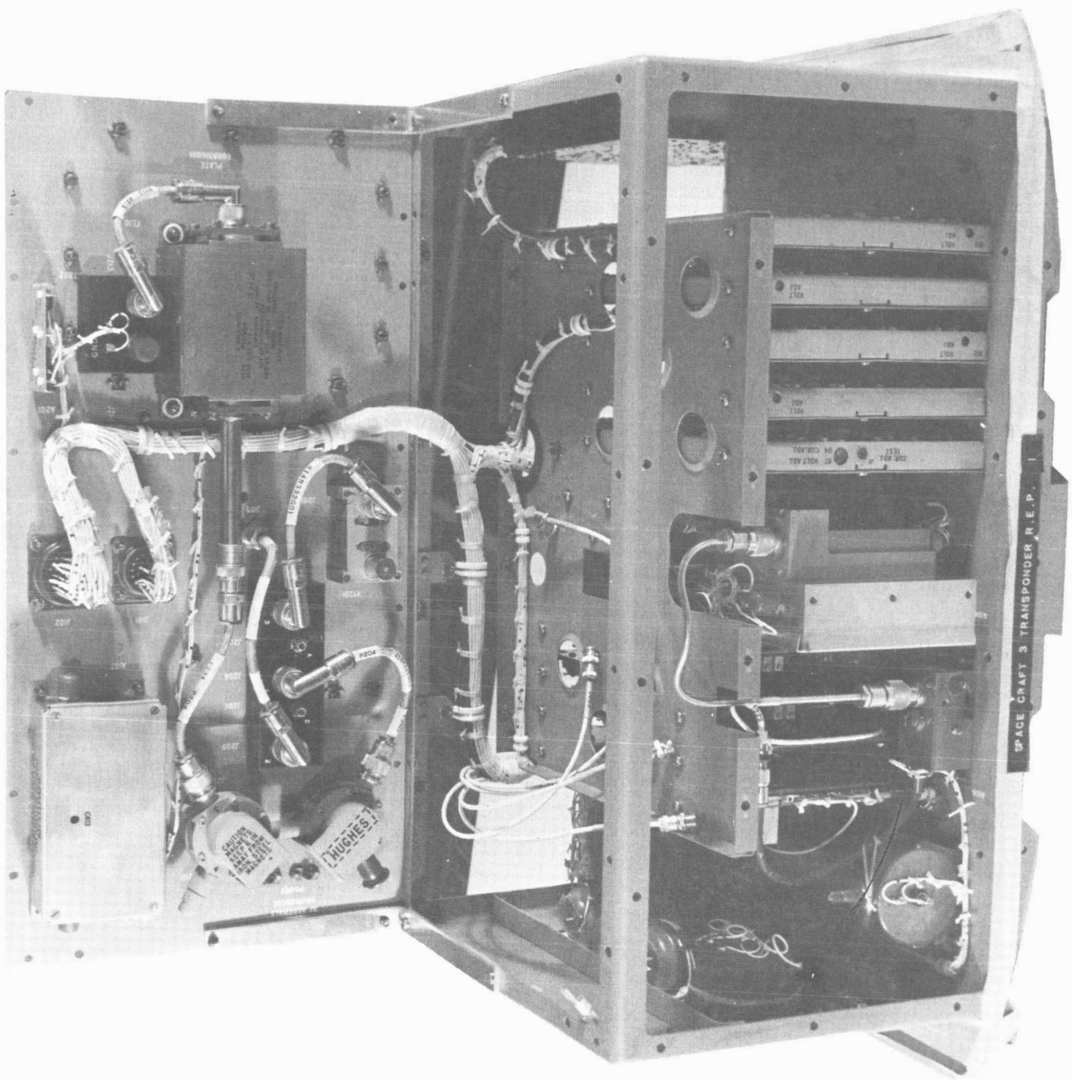


FIGURE 2.4-3 RADAR TRANSPONDER (COVERS REMOVED)

2.4.3 (Continued)

37.6° off the boresight. We recognized that if the radar locked on beyond 37.6°, ambiguous angle data would be furnished to the Gemini computer. To prevent a false null, operational procedures were initiated in which the astronauts would try to null the Flight Director Indicator (FDI) needles while changing the spacecraft attitude in the direction indicated by the needles. However, these procedures were never required in any mission because ground tracking data provided more accurate pointing information than was originally expected.

Conclusion - If a decision is made to use an interferometer type radar where pointing is not expected to be accurate enough to prevent ambiguities, consideration should be given to incorporating electronic circuitry to detect them.

RF Phase Shifters - To track a target located off boresight, an interferometer can be used only in conjunction with a variable phase shifter. When the Gemini rendezvous radar was conceived, high reliability electronic phase shifters were not available. The scheme decided upon was to rotate a spiral antenna to produce the required phase shifting for angle tracking.

Transponder Voltage Regulation - The transponder required a regulated voltage source for efficient power supply operation. Such a source was available on the Agena. But line losses from current drains across the Agena target docking adapter (TDA) interface were greater than originally anticipated. Thus the voltage available at the transponder was less than desired. This resulted in the transponder operating on an unregulated voltage. A boost regulator was subsequently added to the TDA which, in

2.4.3 (Continued)

turn, furnished the transponder with the proper regulated voltage level. This change took place in September 1963.

Conclusion - Determine line losses across vehicle interfaces before evaluating the use of remote voltage sources.

Redundant Design Features - The MTBF requirements on the radar and transponder were 1000 and 10,000 hours, respectively. The design mechanization as originally conceived was solid state, with the exception of the transmitter and the local oscillator. Planar triodes were selected for the local oscillator and transmitter RF sources. To meet the transponder MTBF requirement, redundant local oscillators were originally considered, as well as a solid state local oscillator using a varactor multiplier. Both design concepts were pursued for the first four months of the engineering development. As reliability data was gathered on the local oscillator, a single planar triode type proved to be sufficiently reliable for system MTBF requirements. Further development of the solid-state local oscillator was therefore stopped.

Conclusion - Where possible, pursue more than one design approach to meeting key requirements until the solution is clearly in hand.

Pressurization - Size, weight, and reliability were the primary considerations in the decision to pressurize the radar. Pressurization would obviously add weight. On the other hand, the radar required moving parts for angle tracking, and it was felt that the lubricants available at the time would not survive 14 days in a hard vacuum environment. The decision was therefore made to pressurize the radar in spite of the weight penalty

2.4.3 (Continued)

of approximately six pounds. This decision was rendered prior to contract go-ahead, and the original mechanical design was compatible with this requirement.

A decision was made not to pressurize the transponder on the basis that it contained no moving parts and would survive the long duration exposure to a hard vacuum. This decision resulted in the occurrence of electrical breakdown during qualification testing. Changes required to correct this problem are discussed in Section 2.4.4.

Thermal Control - During the preliminary spacecraft design phase, we determined the optimum location for radar installation to be the front of the spacecraft. This location was far removed from the spacecraft active cooling system, and it was undesirable to provide active cooling for the radar. The radar ground plane area that was available as a radiator was found to be more than adequate to prevent the radar from overheating during the rendezvous exercises. Prior to contract go-ahead, the decision was made that the thermal control of the radar would be passive. A flight failure due to low temperature operation of the radar is discussed in Section 2.4.5.

An active coolant system was not available on the Agena, thus necessitating passive temperature control of the transponder. To conform with this design constraint, the transponder was hard mounted on the mold line of the TDA to allow one side of the unit to function as a radiator.

Frequency Changes - The rendezvous radar system was originally conceived as an L-band system. The early system performance studies were performed with

2.4.3 (Continued)

the frequencies of 1000 MHz for the radar and 1060 MHz for the transponder. This selection of frequencies was thought to be within the interference region of the IFF frequency band, so the original development frequencies chosen for the engineering model radar and transponder were 1428 MHz and 1368 MHz respectively. In August 1962, the FCC approved the frequencies of 1528 MHz for the radar and 1428 MHz for the transponder. This change occurred during the initial deliveries of the microwave components for the engineering model equipments. The microwave components were redesigned for the allocated frequencies, but the system antennas also required redesign. Subsequent analysis and evaluation showed that the RF gain was lower at these frequencies than at those originally planned, and the maximum range capability was reduced to 180 nm. However, this range was still compatible with the requirements of the rendezvous mission, and the radar antennas were not redesigned.

<u>Conclusion</u> - Press for an FCC frequency allocation at the earliest possible date.

<u>Incorporation of Command Link</u> - The decision to ignite the Agena engine subsequent to rendezvous required that the astronauts have command control of the Agena vehicle. Studies of the communications equipment already on board Gemini revealed that multipath problems existed for these lower frequency systems. Further studies showed that the radar could be pulse-position-modulated for the transmission of commands to the Agena and that pulse width modulation of the transponder could be used for command verification.

2.4.3 (Continued)

Go-ahead to incorporate the command link in the radar system was given in September 1962 after the drawing release to build the engineering model equipment. Extensive changes were required to the radar system. The radar modulator was redesigned to accommodate the pulse position modulation by the vehicle time reference system. The command link encoder and the encoder controller were added in the cabin. These units allowed the astronauts to select and generate the appropriate command for transmission by the radar to the Agena. A sub-bit detector was added to the transponder which provided for detection of the sub-bits contained in the transmitted command word. All changes were incorporated in the design, and the production prototypes were delivered with full command capability.

2.4.4 Problems and Design Changes During Development and Qualification Testing - Radome Seal - The original studies of the radar angle tracking performance indicated that use of radomes over both the angle-tracking and the reference antenna assemblies would cancel radome-induced phase errors. The reference antenna assembly was stationary and did not require a pressurized radome as did the azimuth and elevation antenna assemblies. Thus, the radome over the reference antenna was not sealed, resulting in the exposure of the reference antenna to the ambient environment. During humidity qualification tests, a boresight shift was noted in both angle tracking channels. The shift was found to be due to moisture absorption in the epoxy laminate board on which the reference antenna was etched. Design changes were incorporated in the flight units to seal the reference antenna from the humid ground test environment at the Cape.

2.4.4 (Continued)

Range Processing - The original range processing technique utilized by the radar established a composite range value from four range readings. This technique provided smoothing of the range data. During the engineering development phase, we realized that a false alarm that occurred during the four range count interval could result in a gross error in the range reading that was supplied the spacecraft computer. At the conclusion of the engineering development phase, the radar design was modified such that target range was determined with each transmitted pulse instead of with an average of four pulses. The change from four range word accumulation to one range word accumulation caused the bias error to increase from 50 ft to 75 ft, and the minimum range specification was increased to 500 ft. During the rendezvous exercise, each range reading was transferred to the computer, and provisions were incorporated in the math flow to check the validity of each range reading before the computer processed the radar data.

Conclusion - Accomplish data smoothing in a device inherently capable of discriminating against grossly inaccurate single data points.

Coaxial Switch - A coaxial switch was required in the Agena transponder to switch between the spiral antennas and the dipole antenna to accommodate changing Agena to Gemini look angles during rendezvous. A solid state switch was chosen initially, but it failed during engineering development testing. It was found that the microwave diodes could not handle the transponder peak power while the switch was changing state. Rather than redesign the transponder video section to preclude switching while the transmitter was firing, we decided to incorporate a coaxial relay in the

2.4.4 (Continued)

transponder. This change was incorporated in the production prototype units in the summer of 1963.

Conclusion - If antenna switching is necessary, either preclude transmission during the switching action or specify a switch capable of handling the expected peak power when the switch is in the changing state.

Pressure Sealing of Transponder Coax Switch - As part of the general change to vent outgassed elements from RF components associated with the transponder, the coaxial relay was vented to prevent these entrapped gases from causing glow discharge breakdown. No problem was anticipated from this change; however, when the relay was installed in the transponder and tested in a vacuum chamber, we observed a new phenomena. When the relay was cycled, it would start to switch from one port to the other, and halfway through switching it would return to the initial port. The problem was traced to a large transient which was generated by the switch in the vacuum. This energy was coupled back into the electronics driving the switch, causing the switch to return to the initial state. The phenomena producing the problem was described as "vacuum chop," where the spark generated by the contacts is extinguished much faster in a vacuum than at ambient, causing the induced transient to be 100 times greater. Suppression of the transient by filtering was impractical because the switching mechanism became unreliable when enough filtering was added. Instead, minor modifications were made to the existing humidity seal so that the transponder coaxial switch was not vented, and a satisfactory vacuum seal was obtained.

2.4.4 (Continued)

Radar Vibration - The basic structural member of the radar was the ground plane that mounted flush with the front of the spacecraft. This member served as an RF ground plane and all the system electronics were hard mounted to it. In May 1962, the thickness of this member was reduced during an overall spacecraft weight reduction effort. In the fall of 1963, during engineering development tests, the ground plane was found to "oil can" when subjected to a vibration level of 12.6 g's RMS perpendicular to the plane of the ground plane. This problem was corrected by increasing the thickness of the ground plane and by the addition of vibration isolators to the radar mounts. These changes were incorporated in the production prototype units, and the radar was successfully qualified at an 8.8 g's RMS level.

Transponder Humidity - The original chassis design of the transponder did not incorporate provisions to prevent the electronics from being exposed to the 100% humidity test environment. The electronic assemblies and components were coated and finished to preclude corrosion and moisture absorption. However, moisture that was trapped between heat sinks and printed circuit boards resulted in leakage current paths that rendered portions of the equipment inoperative. This problem was discovered in September 1964, and seals were incorporated in the chassis to prevent the transponder electronics from being exposed to the high humidity test environment.

2.4.4 (Continued)

Transmitter Discharge - A characteristic of a planar triode transmitter tube is that occasionally a discharge occurs between the tube elements at the initial turn on. This discharge may couple energy back into the circuits that can only dissipate small amounts of power. In the rendezvous radar, telemetry and modulator circuits failed as a result of this discharge and the radar was rendered inoperative. This problem was solved with the addition of a diode conductive path between the tube grid and ground in units used in Spacecraft 6 and up.

Conclusion - Carefully isolate low power level circuits from the effects of directly connected or coupled power surges.

Radar Accuracy - The original radar angle accuracy requirement was ± 3 milliradians. Early in the development, it became apparent that this requirement could not be met. Several factors were responsible for the degraded performance. The installation of the radar in the spacecraft required the addition of protrusions to the ground plane that degraded system performance. The effects of ellipticity of the target antenna had not been properly assessed. Also, the phase stability of the microwave components was not as good as preliminary design studies had shown. These factors contributed to the final on-axis performance of ± 9.0 milliradians. For angle off axis, the error increased to ± 24 milliradians at 25°.

To minimize the effects of radar errors on the rendezvous mission, the astronauts flew "boresight" during radar data gathering periods.

2.4.4 (Continued)

Antenna Switching - The design of the radar system was based on the Agena vehicle being stabilized so that initial radar acquisition would be via the transponder's dipole antenna. Later mission definitions, however, called for the spacecraft to approach the target from above, and resulted in initial radar contact via the spiral antennas. Design changes were incorporated in the flight transponders to initiate switching between the spiral and dipole antenna by ground command. Initial radar acquisition could then be accomplished either via the spiral or the dipole antennas.

Flashing Lights - As an aid to rendezvous navigation, an acquisition light was proposed for the Agena. After study of various types of lights, it was decided to procure a flashing Xenon gas discharge light. It was determined during system testing that an excessive amount of interference to the L-Band transponder was being generated due to the characteristics of this light. Both conducted and radiated interference were present. Two steps were taken to minimize the interference: 1) RF filtering was added to the light leads, and 2) a wire screen was placed over the light to minimize radiated interference. The intensity of the lamp was increased to compensate for the reduction in intensity caused by the screen.

Voltage Breakdown - Voltage breakdown occurred in the non-pressurized transponder when qualification tested in a vacuum. This was attributed to materials outgassing under high temperature conditions within the transmitter cavity. The problem was eliminated by pressurizing the transmitter cavity. In addition, all RF components, with the exception of the coaxial relay, were vented to prevent partial pressure buildup. Two

2.4.4 (Continued)

flight failures were attributed to depressurization of the transponder transmitter. During the Spacecraft 11 flight, degradation of the transponder's transmitter output began in the latter stages of rendezvous, with the transmitter failing later in the flight. This failure was attributed to a leak in the transmitter cavity, which allowed the pressure to decay to the critical point where RF arcing occurred. Corrective action was taken on Spacecraft 12 by cementing the screws that seal the transmitter cavity. This procedure was considered a valid corrective action because of observations made when one of the seal screws was removed for a 24-hour leak test on each transponder and later replaced. Although a vacuum test was conducted on the transponder after the leak test, vibration-induced leaks would not be detected.

During the Spacecraft 12 flight, after target acquisition, range and angle information was sporadic. The most probable cause was again considered to be arcing in the transmitter cavity.

Failure analysis of these two cases had to be performed using flight data only, because the transponders were never recovered after the flights.

> Conclusion - Whenever feasible, pressurize the power generating and handling sections of RF equipment. Additionally, it should be recognized that maintenance of an adequate pressure seal may in itself be a problem for long duration space missions.

2.4.4 (Continued)

Analog Range and Range Rate Accuracy - The differences between the analog range and range rate initial requirements and final capabilities were due to the sensitivity of the analog range circuits to temperature extremes. The accuracy attained was adequate for the braking maneuvers, and therefore additional temperature compensation was not incorporated into the unit.

2.4.5 Problems and Design Changes Associated With Systems Test and Flights -

Transmitter Filament Voltage - Frequency drift was encountered in the radar transmitter during the extensive prelaunch test. Frequency drift did not occur in the transponder, and extensive investigation showed the problem to be due to differences in the backheating of the cathode. The filament voltage was the same for the radar and transponder; however, the radar utilized a 1.0 microsecond pulse and the transponder used a 6.0 microsecond pulse. The transponder cathode temperature was higher due to the wider pulse width, resulting in stable frequency operation. The filament voltage was increased on the radar and stable frequency operation resulted. The effectivity of this change was Spacecraft 6.

Conclusion - Excessive component de-rating in this type of application can lead to a transient cathode temperature and subsequent frequency drift.

RF Bonding of Radar in Spacecraft - Several factors influenced the mounting of the radar in the spacecraft. The radar required an optical alignment with respect to the inertial guidance system, dictating a nominal

2.4.5 (Continued)

separation between the spacecraft structure and the radar ground plane to accomplish necessary adjustments. A high temperature rubber gasket was provided between the radar ground plane and adjacent structure to provide the necessary thermal protection.

Prelaunch boresight testing showed that RF leakage around the radar ground plane could produce boresight shifts. RF bonding of the radar ground plane to the adjacent structure was done without compromising the optical alignment or the thermal protection. The effectivity of this change was Spacecraft 6.

Tracking Tumbling Target - The rendezvous radar was designed to track a stabilized target. Stabilized targets were not used with Spacecraft 5 and 9. A characteristic of the tumbling target was missing pulses, which produced excessive range rate transients on the astronauts display. The Spacecraft 5 and 9 radars were modified to filter these transients to an acceptable level.

Digital Interface Problem on Spacecraft 9 - In preparation for the Gemini 9 mission, computer software was changed so that range rate could be determined from the radar digital range information.

To mechanize this routine, data had to be obtained from the radar at a faster rate than that originally specified. In a feasibility study, we found that logic in the radar had to be modified to accommodate the faster

2.4.5 (Continued)

readout rates, and the computer program had to be changed to accommodate the final maximum readout capability of the radar. These changes were made in subsequent flight units.

Digital Range Failure on Spacecraft 5 - The minimum temperature for reliable turn-on of the radar was $0^{\circ}F$, and in-spec performance was required at $20^{\circ}F$. On Spacecraft 5, the temperature of the radar decayed to approximately $5^{\circ}F$ and cycled about this value. On the third day of the mission, the radar would not gather digital range information during an air-to-ground evaluation. The radar unit was not recovered; however, post-flight analysis of possible failure modes indicated a low-temperature-induced failure of a diode in the range logic circuitry. Therefore, Spacecraft 6 and subsequent units were performance tested at $0^{\circ}F$.

3.0 Computers

3.1 Digital Computer

3.1.1 Requirements - The functional requirements for the Gemini Digital Computer are shown in Table 3.1-1.

3.1.2 Early Design Decisions and Tradeoffs - The requirements to perform a rendezvous with an orbiting target and to guide the reentering spacecraft to a predetermined landing point made a computer a necessary component of the Inertial Guidance System (IGS). Eventually, the computer was called upon to perform other tasks, as will be seen in Section 6.1.1 and this section.

The time history of the Gemini Computer Program development is shown in Figure 3.1-1. The internal layout of the computer is shown in the photo of Figure 3.1-2.

Factors which influenced the initial design of the computer were:

(a) Computational speed

(b) Memory type

(c) Memory capacity

Computational Speed - The computational speed of the computer was primarily determined by the reentry guidance requirement. This requirement, however, was not overly stringent because the entering ballistic vehicle response was slow enough that one solution every 2 seconds was considered adequate. The initial specification also required a rendezvous computation in 2 seconds or less. The chosen computer had been under development for the STINGS program; and its speed, as given in Table 3.1-1, was more than adequate to perform these functions. In actual operation, navigation

TABLE 3.1-1

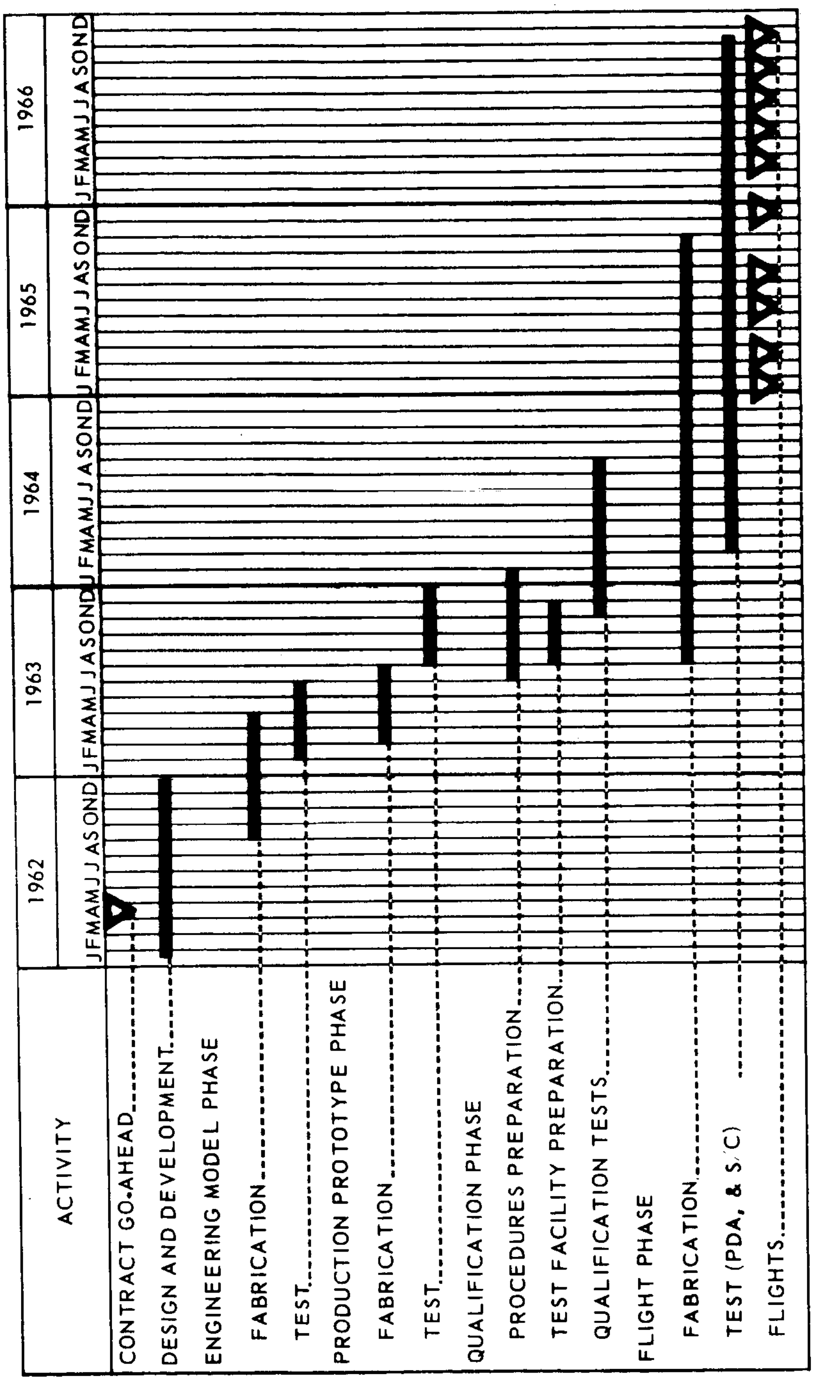
DIGITAL COMPUTER
FUNCTIONAL REQUIREMENTS

	Final Capability	Initial Requirement
Memory Capacity	4096 39-bit words (divided into 3 syllables of 13 bits each) (third syllable read only) - Plus - 85,000 13-bit word Auxiliary Tape Memory	*4096 39-bit words (3 syllables each)
Clock Rate	500 KC	*Same
Arithmetic Operation Times:		
Add	140 μsec.	*Not specified
Subtract	140 μsec.	*Not specified
Transfer	140 μsec.	*Not specified
Multiply	420 μsec. } Mult. or Divide operations may be programmed concurrently with Add, Subtract, or Transfer operations	*Not specified
Divide	840 μsec. }	

* Original specification specified sufficient arithmetic computation capacity and speed to perform re-entry computation and provide output commands at least once every two seconds.

FIGURE 3.1-1

DIGITAL COMPUTER PROGRAM TIME HISTORY



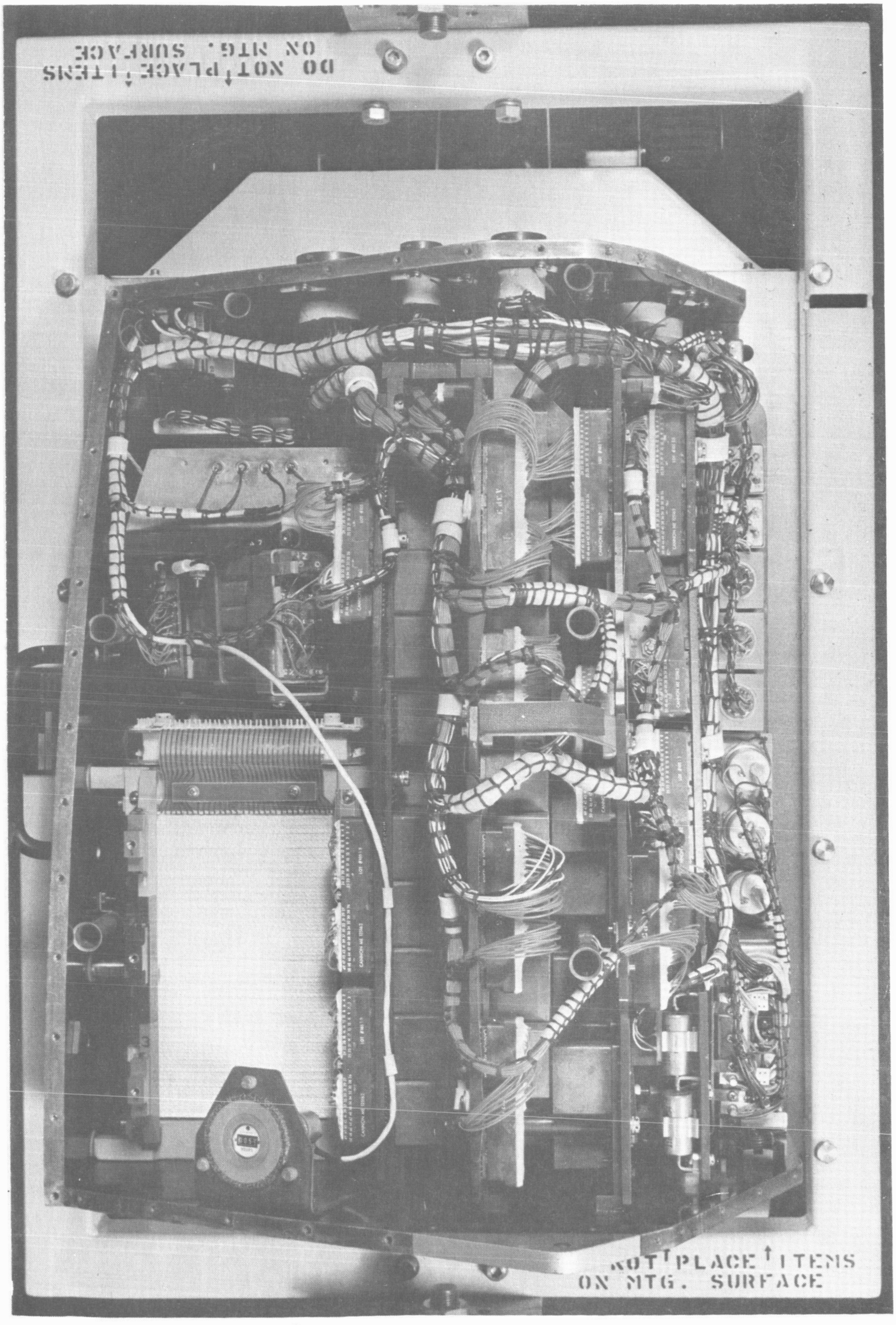


FIGURE 3.1-2 DIGITAL COMPUTER (COVER REMOVED)

3.1.2 (Continued)

solution rates varied. Acceleration pulses had to be collected at a high rate of up to 3600 pps, so as not to lose data, but total navigation solution rates varied from one per 16 seconds during nondynamic periods to between one and two per second during high thrust or drag conditions.

Memory Type - It was desired to have a random-access memory for programming flexibility. Furthermore, a drum type memory was undesirable because of its sensitivity to vibration. The chosen memory used multiaperture cores and was relatively new, having been developed for the STINGS program about a year before the beginning of the Gemini program. At the time of the Gemini design the multiaperture core was the best means of achieving a nondestructive readout, which was desired for added reliability. However, the ferrite cores used had a narrow temperature range (18°F Δt) over which they would operate with a given drive current. A temperature compensation circuit was added which adjusted the drive current to compensate for the changes in the cores' characteristics at temperatures beyond the above range, and coldplate cooling was employed to extend the computer's operating range to 40°F Δt (50°F - 90°F).

Memory Capacity - A very early predesign estimate of memory requirements to perform rendezvous and reentry calculations was 1700 words, including a 25% allowance for "inefficient memory use." However, in mid-1963, the computer, as actually programmed, required 6739 13-bit words to perform these functions. The digital computer had a memory capacity of 4096 39-bit words of three syllables each, or an effective memory of 12,288 13-bit words. This capacity was considered ample, at the time of initial design,

3.1.2 (Continued)

to absorb reasonable program expansions. Memory capacity problems which later arose are discussed in Section 3.1.3.

Other vital factors in computer design were:

(a) Input/output interfaces

(b) Computer malfunction detection

Input/Output Interfaces - The computer had digital interfaces with the Inertial Measurement Unit (IMU), Rendezvous Radar, Time Reference System (TRS), Digital Command System (DCS), and the Data Acquisition System (DAS), in addition to the control/display interfaces with the Manual Data Insertion Unit (MDIU), Incremental Velocity Indicator (IVI), and the computer ground equipment. Analog output interfaces were with the Attitude Control and Maneuver Electronics (ACME), Attitude Display Group (ADG), and the Titan Launch Vehicle. An analog input, representing IMU gimbal angles, was also provided.

Each digital interface design was agreed upon separately, because of the different timing and digital format requirements of the interfacing subsystems, and because of the different stages of development of these subsystems. A more desirable method would be employment of a "standard" interface design for all digital interfaces, but this was not feasible for the above reasons.

Balanced line drivers, balanced line switches, and pulse transformers were among the devices used to achieve ground isolation of the digital interfaces.

3.1.2 (Continued)

The Titan launch vehicle required a DC analog signal, which was achieved by a "ladder" type D/A conversion in the computer.

The ACME and ADG required 400 Hz AC signals which were obtained from magnetic modulators driven by the ladder circuits. The magnetic modulators had advantages over other AC output devices in reliability, size, and weight, but contributed significantly to the harmonics and quadrature problems experienced with the ACME system. As is said in Section 4.0, these problems, although not serious on Gemini, could become so on a system with more stringent control requirements.

Computer Malfunction Detection - Early in the program it became apparent that the astronauts would have to be kept aware of the computer's performance. Therefore, a light was placed on the astronaut's display panel to indicate malfunctions detected by the computer's malfunction detection system. Four self-checks were performed.

(a) A timing check, based on the non-coincidence of certain signals within the computer under proper timing conditions. Any improper functioning of any of the computer's basic timing pulses would cause coincidence of these signals, and trigger the circuit.

(b) GO-NO-GO diagnostic test - This test, the most thorough, exercised all of the computer's arithmetic operations during each computer cycle in all modes.

(c) A "looping-check" to verify that the computer was following a normal program loop. A counter in the output processor was designed to overflow every 2.75 seconds. Each program was written to erase this

3.1.2 (Continued)

counter to zero every 2.7 seconds; and thus, any change in the program flow would cause an overflow and indicate a malfunction.

(d) The pre-launch mode check would verify the contents of memory syllable 2 by summing them. Since this syllable was "read only" (after loading the memory by ground equipment), the sums could be checked against their known values to insure correct data.

3.1.3 Problems Encountered During Development and Qualification Tests

Functions of the Computer Running Light - The need for communications between the astronauts and the computer became apparent early in the program. The Manual Data Insertion Unit (MDIU) and Incremental Velocity Indicator (IVI) provided for data insertion and readout. However, it was also necessary for the computer at various points during the program to signal the astronaut to perform an operation or to verify an operation just performed. A very limited space was available on the control panel, so an array of indicators, each for a different point in the mission, was not feasible.

The indication was provided by the computer running light, driven by the "computer running" discrete, which could be programmed to indicate various events in the computer program. For example, during rendezvous, the operation of the light was controlled by the accumulation of radar data. The light signaled the astronauts when to apply corrective maneuver thrust. In conjunction with operation of the Auxiliary Tape Memory (ATM), it illuminated to show the transfer of data from the ATM to the computer. In addition to specialized functions such as these and others, it performed

3.1.3 (Continued)

its normal function of indicating the start and end of a computer program by illuminating and extinguishing, respectively.

The major disadvantage of providing these indications in this manner was the amount of crew training required so that the astronauts would know the significance of each of the illuminations/extinctions. A more desirable indication would have been provided by a set of indicators, each assigned a particular function. As stated previously, however, this was not feasible on Gemini because of space limitations, and our method of providing these indications is recommended for a future system operating under a similar constraint.

Conclusion - Combine display functions only when space-limited.

Generation of Spurious Output Pulses Upon Power Application - The computer's logic design was such that, during power turn-on, output signals could be generated for a certain time. The outputs were eventually reset by either internal timing or by program steps; but before resetting, output discretes, data, and timing pulses could be sent to interfacing subsystems without being requested. In some cases, this condition could not be tolerated. The situation was discovered in early 1963 during early integrated system testing on the Electronic Systems Test Unit (ESTU). To prevent its reoccurrence, minor wiring changes were made to prevent inadvertent data exchanges, and initialization steps were added early in the computer's "executor" routine to assure proper initial conditions.

Conclusion - Examine the effect on input and output circuits of power up and power down of the computer.

3.1.3 (Continued)

Manual Restart of Computer - The computer was protected from power depressions and outages by the Auxiliary Computer Power Unit (ACPU), discussed in detail in Section 6.1.3 of this report. The ACPU, at the time of its incorporation, was to provide power for computer operation during short power depressions, shutdown of the computer during prolonged depressions, and automatic restart upon return of full power. An automatic restart, however, was potentially more harmful than a power loss, since, at restart, the computer "re-initialized" the problem it had been solving and generated new answers (or control signals) having no relation to the pre-shutdown conditions. This could have resulted in catastrophic error signals being applied to the launch vehicle (in the event the Gemini IGS were being used for backup launch guidance), or the loss of previously accumulated data during rendezvous or reentry. Thus, in its original form, the ACPU was capable of causing an initialization of the computer's program without the astronauts' knowledge. This occurrence was prevented by changing the ACPU circuitry so that the shutdown would be indicated by the computer malfunction lamp, and the astronaut was required to reset the malfunction lamp manually to restart the computer.

Conclusions

1. In a manned operation involving the use of a computer, the astronaut must be made aware of any deviation from the computer's expected operation.
2. The effect of automatic sequencing on computer output behavior must be assessed and precautions taken to avoid application of catastrophic error correction signals to interfacing systems.

3.1.3 (Continued)

Radar Shift Pulse Timing - Because of the asynchronous computer/radar operation, radar data for the computer was buffered in the radar. The data was shifted out of the radar buffer by pulses originating in the computer. For each shift pulse, one data bit was received. During interface testing in late 1963, an excessive round-trip delay between generation of the shift pulse and receipt of the radar data by the computer was discovered.

The delays, occurring in the computer shift pulse logic, spacecraft cabling, radar equipment, and computer serial data logic, were found to be between 0.75 and 0.95 microseconds. Original design allowed for total delays of only 0 to 0.525 microseconds. A computer logic change was made, which moved the train of shift pulses 0.5 microsecond, giving a total allowable delay of up to 1.025 microseconds.

The above change caused another potential radar interface problem. The shift pulses from the computer were sent to the radar in three bursts of 52 pulses each. A balanced line driver, utilizing a pulse transformer, was used as a shift pulse driver. It was found that the amplitudes of pulses near the end of the third burst were degraded, due to energy storage in the pulse transformer. The excess energy storage was due to an increase in width of logic pulses brought about by the above timing change. This could have been corrected by redesigning the balanced line driver to decrease the pulse transformer recovery time, such that saturation would not be reached until the three bursts of pulses were completed. However, in the case of the Gemini system as mechanized, the computer did not use

3.1.3 (Continued)

the last few radar pulses, so this condition did not cause a problem. A specification change was made and the computer was not modified.

Conclusions

1. Conduct a thorough timing analysis of digital interfaces early in the program.
2. Assess the effect of any timing change on other characteristics of the circuits involved.

Electromagnetic Interference (EMI) Problems Encountered During Development Testing - It was discovered early in the program that the computer was highly susceptible to transients which altered the memory and/or caused an indication of a computer malfunction. In early 1963, many problems were encountered with memory alterations, and an intensive analysis was conducted to isolate the areas of transient susceptibility and to minimize the transients. The areas in the computer found to be most susceptible to EMI were the memory strobe circuits and the glass delay lines.

The strobe circuits were used to sample the memory output pulses so that a "0" noise output would not be interpreted as a "1". Originally, this strobe had difficulty in distinguishing between noise and information, but the circuit was changed so that the signal under the strobe gate was integrated. This helped solve the EMI problem.

The glass delay lines were used as temporary storage devices (such as the accumulator and address register). Because of their inherent low signal levels, they were highly susceptible to noise. The wire routing within the computer was changed to make the lines less susceptible to EMI.

3.1.3 (Continued)

Grounding of the computer to its mounting was also reviewed during subcontractor tests, and as a result, short ground straps were used. It was believed that the straps reduced EMI susceptibility. Partially as a result of this experience, a special short ground strap was installed between the computer and the spacecraft structure on Spacecraft 3 through 12.

Redesign of Memory Drive Circuits - A problem occurred in the transition between the engineering model and production configuration computer. The production model utilized multilayer boards to interconnect the circuit modules, whereas the engineering model used hand-wired discrete components. Difficulty in specifying a minimum and maximum tolerance on the thickness of the insulating layers of the multilayer boards led to the capacitance between conducting sides exceeding the worst case design. A major redesign of the memory drive circuits was required to compensate for the added capacitance.

Conclusion - This type problem is typical of those which must be considered when proceeding from a breadboard design to a production model.

Exceeding of Memory Capacity and Incorporation of Auxiliary Tape Memory - Computer memory requirements were increased significantly when the ascent mode was added to the computer in May 1962 to mechanize the Gemini IGS function of backup guidance for the Titan II launch vehicle. Another early addition was the requirement for an orbit navigation mode. These additions, plus numerous small features and improvements to existing programs which required additional memory capacity, used up all the memory space by

3.1.3 (Continued)

January 1963. The programming of the second operational math flow in June 1963 revealed that the memory capacity was exceeded by over 700 words, even after significant savings had been achieved by reprogramming.

The following solutions were available for this problem:

(a) Reduce programming requirements, at the expense of accuracy and/or versatility. This approach could have been accomplished by across-the-board reductions in all modes, or by dropping one of the computer modes (Rendezvous and Orbit Navigation were not required for the early spacecraft).

(b) Provide additional usable memory locations by hard wiring some elementary functions presently programmed. Such an approach would have gained 825 instructions.

(c) Provide an auxiliary tape memory external to the computer, under the control of the astronauts. This unit would contain sufficient storage to handle present requirements, as well as any contemplated future requirements.

The solution chosen was a combination of alternatives (a) and (c). On Spacecraft 2 through 7, the orbit navigation mode was not included. On Spacecraft 8 through 12, an auxiliary tape memory (ATM) was added to the spacecraft.

The development of the ATM was begun in mid-1964. This unit, located in the spacecraft adapter as shown in Figure 1.2-7, provided an additional capacity for storage of 85,000 13-bit words. It permitted a high degree of computer flexibility, as evidenced by the inclusion of nine operational

3.1.3 (Continued)

computer modes in the last four spacecraft. In addition to alleviating the memory capacity problem, the ATM provided a memory reload capability for use in the event of an in-flight memory alteration, such as occurred during the flight of Spacecraft 4. The ATM is pictured in Figure 3.1-3.

Conclusion - Allow ample margin for growth in a computer's memory capacity (at least 200 to 300%). This margin should be a function of how well the computer requirements have been defined. This criterion is particularly important in a developmental type vehicle such as Gemini, where expanded capabilities are added as the program progresses. Consideration should be given to providing an onboard bulk storage device, such as an ATM.

ATM Static Discharge Problem - A problem resulted, during the development of the ATM, from its being filled with dry nitrogen and hermetically sealed. The dry nitrogen caused the tape to build up a static charge and discharge at the tape head, producing erratic outputs. A dry Freon mixture was substituted for the nitrogen, solving the problem. The Freon had a heavy ion which captured electrons, preventing a buildup of static charge.

ATM Vibration Problem - A problem developed during ATM vibration qualification tests, where the flutter output exceeded the specification of 3% under 2 g RMS random vibration. Various methods were tried to increase the ATM's vibration tolerance, including the use of dampers, weights, and flywheels. These methods reduced flutter but did not solve the problem entirely. Finally, the flutter specification was widened to 5% at 1 g RMS and this, in combination with the use of flywheels, enabled the unit to meet the

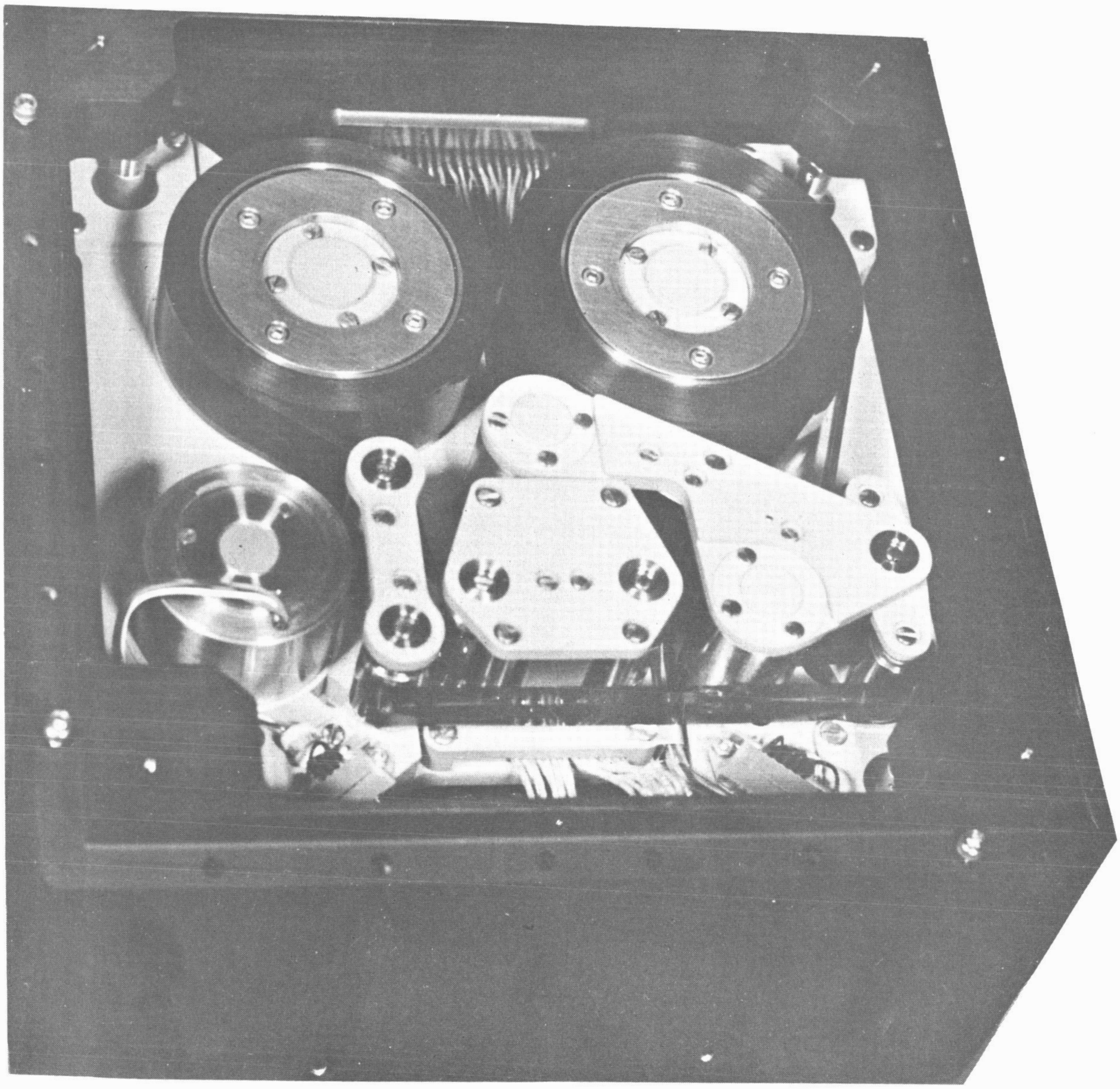


FIGURE 3.1-3 AUXILIARY TAPE MEMORY (COVER REMOVED)

3.1.3 (Continued)

vibration requirements. The original tight specification on flutter was made because of the strict timing requirements of the computer. Widening the specification was justified by analysis which showed that the ATM would not be incompatible with computer timing requirements, but would have a slightly decreased safety margin. When integrated with the computer, no problems were encountered in this regard. The ATM was required to operate only during orbit, and not under the more severe launch vibration environment.

3.1.4 Problems Encountered During Systems Tests and Flights

Electromagnetic Interference - The computer experienced several EMI problems during spacecraft production testing. A considerable number of malfunction indications were experienced during Spacecraft 3 testing; and, in an effort to correct them, the grounding of the computer to the spacecraft was changed. The Gemini grounding philosophy was based on a single point ground (Section 6.1.6). This point on the spacecraft was located approximately 10 feet away from the computer. Although this method worked well for DC and low frequency AC signals, the long wire from the computer to the ground point acted as a high impedance at the frequencies used in the computer. Previous subcontractor tests had indicated that short ground straps reduced EMI susceptibility, so a short ground strap was installed at the computer. However, no significant reduction in the EMI problem was noted, but it is possible that greater problems would have been encountered without it.

The most persistent EMI problem occurred in the computer HALT circuit. HALT was an input discrete from the Aerospace Ground Equipment (AGE),

3.1.4 (Continued)

provided by a manual switch closure, which nominally required a +8 (±2) volt DC signal for activation. With the receipt of a HALT discrete, the computer program was initialized; thus the program variables, such as position, velocity, etc., were set to initial values, disrupting the operation in progress. Although the input circuit was designed to operate at an 8-volt level, tests indicated that a 3-volt, 10 microsecond signal on the HALT line would trigger the HALT circuit. The ambient noise on the unshielded AGE HALT line was found to be as high as 2.0 to 2.5 volts, so a small additional transient on the line could be interpreted by the computer as a HALT command. The HALT circuit was identical to other internal circuits, with a 40-nanosecond response time. Since the HALT was activated by a manual switch closure, this response was unnecessarily fast and, together with the lack of shielding on the AGE line, accounted for the EMI problem with this circuit. The problem was alleviated by adding a decoupling capacitor on the HALT line as closely as possible to the computer, which bypassed induced RF energy to ground.

The turn-on and turn-off of the spacecraft's rate gyros produced noise spikes of up to approximately 150 volts. A problem was encountered during production testing of Spacecraft 6 where these transients caused an excess time counter overflow, inducing a malfunction in the computer. To reduce the switching noise, back-to-back zener diodes were installed across the 400 Hz input lines to the rate gyros. This method of suppression effectively corrected the problem.

Another EMI problem resulted in connection with the computer's interface with the Incremental Velocity Indicator (IVI) and the Auxiliary Tape

3.1.4 (Continued)

Memory (ATM). The "Z" channel set-zero line from the computer to the IVI was routed through a section on the ATM mode switch, because the ATM "write" and the IVI "Z" channel set-zero signals were a shared discrete. The ATM write mode wire was jumpered at the ATM mode switch to the IVI "Z" channel set-zero line. However, the mechanization of the write mode was deleted on the spacecraft, so the ATM side of the write wire was left unterminated at the reentry module/adapter interface. This unterminated wire, approximately 20 feet long, acted as an antenna picking up noise which caused its "Z" channel to drive to zero when the ATM mode switch coupled the wire to the IVI. The problem was left uncorrected only because the IVI "Z" channel was not used with the ATM and no other IVI interface problems resulted from it.

Another EMI problem became evident when operation of the HF whip antennas (part of the communications system) produced a computer malfunction indication. This problem was encountered during Spacecraft 11 production testing. Transients greater than 80 volts and less than 1 microsecond duration were observed on the excitation lines of the antenna's extension motor. The power lines to the motor were in the same wire bundle with a large number of computer signals. The problem was not applicable to flights, since the use of the HF system in orbit was not an operational requirement. However, as a precaution, a flight procedure was instituted to turn off the computer before extending the antennas.

Conclusions - From the above mentioned problems, we can formulate several design criteria:

3.1.4 (Continued)

(1) Design each circuit response time according to its intended use. The HALT circuit, operated by a manual switch, could have been designed with a much slower response time, making it insensitive to most EMI.

(2) Scrutinize each potential EMI generator to assess its affect on other systems. Early EMI suppression efforts were directed at solenoids, pumps, relays, etc., to see that they were diode-suppressed and isolated to the spacecraft DC control bus. The rate gyros, operating on 26 volts AC, caused a serious problem because their suppression was overlooked early in the program.

(3) Avoid unterminated wires. An unused wire should be cut and tied back at both ends to avoid its acting as an EMI antenna.

(4) Avoid placing signal and power wires in the same wire bundles.

<u>Spacecraft 4 Power Sequencing Problems</u> - During the flight of Gemini 4, the astronauts were unable to shut down the on-board computer in several instances. The failures occurred when the power sequencing circuitry in the computer failed to function. This circuitry powered up and shut down the computer sequentially to avoid alteration of the memory, and was activated automatically when the computer on/off switch was operated or when power was removed from the IGS power supply. The Gemini 4 failures were observed under both conditions. Removal of the computer power, after failure of the sequencing circuit to operate, resulted in alteration of the computer's memory, rendering the computer useless for the rest of the flight. The exact cause of the failure was never determined.

3.1.4 (Continued)

The failure could not be repeated during extensive post-flight testing. However, seven possible failure modes were postulated, in either the computer, auxiliary computer power unit (ACPU), or the spacecraft wiring, which could have produced this type of failure. No evidence of any of these was found during the failure analysis.

Since the failure was never isolated, no specific fix could be applied to avoid the problem in the future. However, the following spacecraft changes and operational procedures were instituted to circumvent it:

(1) All remaining flight computers were tested for 20 on-off cycles at minimum operating temperature (40°F).

(2) Detailed operational procedures were prepared for use in the event of similar or other types of malfunctions in the computer.

(3) A manual sequencing switch was installed on the astronauts' control panel on Spacecraft 5 and 6, to override the automatic sequencing capability in the event of any of the seven postulated malfunctions. This switch, however, was capable of causing an improper power up or power down (with the same possibility of memory alteration) if rotated in the wrong direction. It was not included on subsequent spacecraft.

In addition, telemetry monitoring was added for computer on/off and computer case temperature, to aid in failure analysis of any other malfunction of this type.

An inflight memory reload capability was also considered but not immediately implemented. However, on Spacecraft 8 through 12, the auxiliary tape memory

3.1.4 (Continued)

could perform this function. As a further insurance that the ATM would be able to reload the memory, the "ATM read" subroutine was loaded into "Syllable 2" of the computer memory. This portion of the memory was able to be read only; writing into it was possible only by means of ground equipment. Thus, it was much less susceptible to alteration by transients or computer malfunctions than the other two syllables.

Spacecraft 9 Computer/Radar Interface Problem - The Spacecraft 9 computer/radar interface problem discussed in Section 2.4 had an impact on the computer programming. The computer's Math Flow #7 had been programmed to derive range rate from the digital range; to do so, it was desired to interrogate the radar every 15 milliseconds. However, during production testing of Spacecraft 9, it was found that the radar could not be interrogated at that high a rate, because of a 75-millisecond time delay in the radar's resetting of its logic. Tests indicated that satisfactory radar data could be consistently gathered every 150 milliseconds, so the operational program was altered to interrogate at that rate. Range rate accuracy was not severely degraded by this change.

4.0 Activation Devices

4.1 Attitude Control and Maneuver Electronics (ACME)

4.1.1 Requirements

The functional requirements for the control electronics and power inverter are shown in Table 4.1-1. Additional requirements for control modes are discussed in Section 4.1.2. The evolution of these parameters is discussed in the following paragraphs.

4.1.2 Early Design Considerations, Decisions, and Tradeoffs

The time history of the development of the Gemini ACME system is given in Figure 4.1-1. The Gemini control system had to meet much more demanding requirements then its predecessor on Mercury, including those associated with long term orbital missions, rendezvous and docking with an orbiting target, and controlled reentry. Because of these stringent needs, a new design was decided upon rather than a modification of existing Mercury hardware. The Mercury system had some basic shortcomings, in that it employed electromechanical encoders as sensor pickoffs and did not permit the precise control required on Gemini. Furthermore, the encoders caused significant noise problems, further degrading the precision of the system. Therefore, it was decided to use analog pickoffs on the sensors with switching level detection to be done electronically. An additional reason for choosing this method was the switching required between several different input devices, which originally included the IMU, computer, horizon sensor, radar and rate gyros.

TABLE 4.1-1

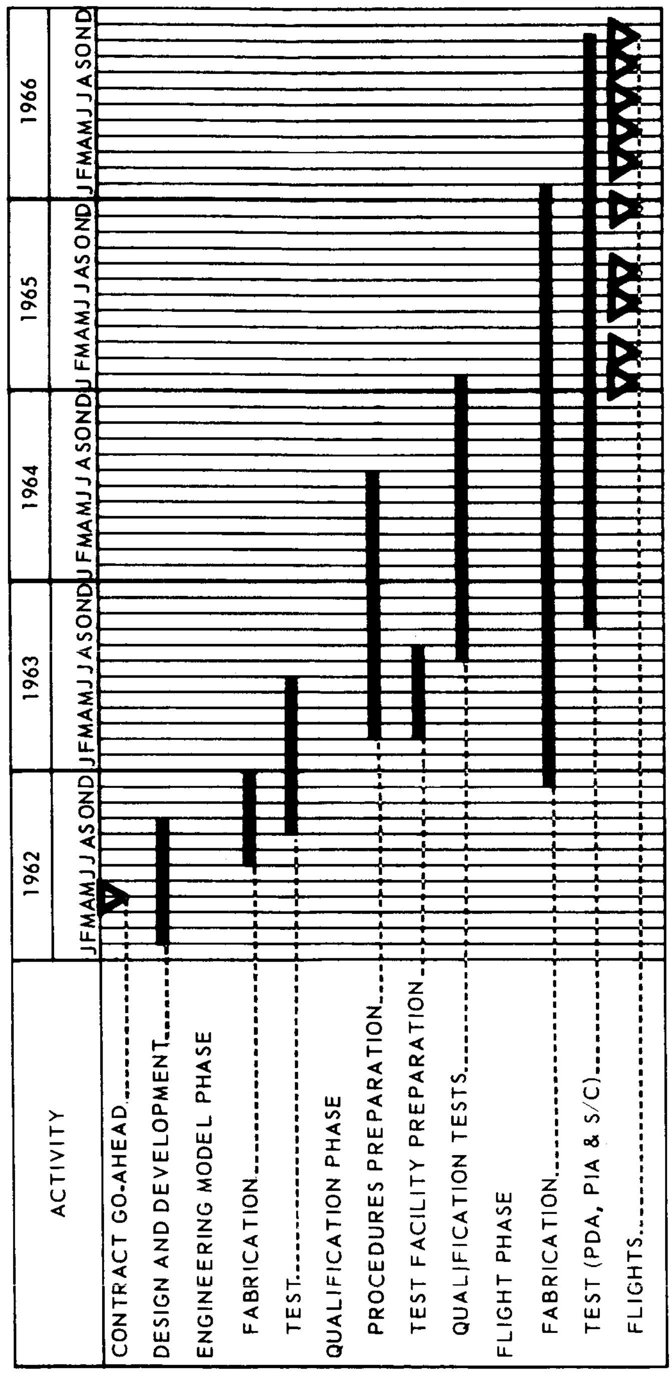
ATTITUDE CONTROL AND MANEUVER ELECTRONICS

FUNCTIONAL REQUIREMENTS

	Final Capability	Initial Requirement
Control Electronics		
On - Off Logic Switching Level	1 ± 0.05 deg/sec.	Same
Time Delay	OAMS - 6 millisec. RCS - 20 millisec.	Same Same
Hysteresis	<1%	Same
Pulse Width (Pulse Mode)	20 millisec. ± 10% OAMS 20 millisec. ± 50% RCS	20 millisec. ± 10%, OAMS and RCS
Minimum Pulse (Pitch and Roll Axes)	18 milliseconds ± 10%	No requirement
Gain Tolerances	±5% attitude; ±7% rate	±5% attitude and rate
Power Inverter		
Output Power	50 volt amperes, 1.0 - .75 pf lagging	10 VA, 1.0 - .8 pf lagging
Output Voltage	26 VAC ± 5% (10 - 50 VA)	26 VAC ± 2%
Frequency	400 Hz ± 1%	400 Hz ± 1%
Harmonic Distortion	10% of fundamental (10 - 50 VA)	10% of fundamental (20% - 80% rated load)
Output Modulation	1.46 volts (10 - 50 VA)	No requirement
Efficiency	31 - 63% (varied with load, input voltage)	60 - 80%

FIGURE 4.1-1

ACME PROGRAM TIME HISTORY



4.1.2 (Continued)

The incorporation of control functions in the digital computer, rather than in a separate package, was considered. The combination would have produced a substantial savings in size and weight, but this mechanization was ruled out because processing the fast changing rate signals would have used a large amount of the computer capacity. Also, any checkout of the control system would have required flight software for the computer.

Overall reliability requirements dictated a high degree of redundancy in the attitude control system. Pilot-selectable, redundant circuitry was included for switching logic, valve drivers, and bias power. The rate gyros (Section 2.1) were redundant, and AC power redundancy was incorporated with the ACME power inverter and the IGS power supply. Additional control redundancy was available through the choice of control modes.

Additional reliability could have been gained by incorporating redundant rate preamplifiers and mode logic. This would have precluded failures in these circuits from preventing mission completion. However, they would have caused a substantial increase in size and weight. It was noted, also, that failure in these circuits would not appreciably affect crew safety.

An early decision was made to electrically isolate, within the packages, all critical parameter outputs which were routed to other systems, such as instrumentation or cockpit displays. Electrical isolation protected the system from external damage, and was eventually incorporated on most of the other spacecraft systems.

4.1.2 (Continued)

The control electronics was split into two packages, Attitude Control Electronics (ACE), located in the reentry module, and the Orbit Attitude and Maneuver Electronics (OAME). The OAME was located in the adapter near the Orbit Attitude and Maneuver System (OAMS) thrusters, as shown in Figure 1.2-5, to minimize the number of wires crossing the reentry module/adapter interface and also to reduce the length of the wires carrying the high currents required for thruster firing. The components of the ACME system are pictured in Figures 4.1-2 through 4.1-4.

The decision to incorporate a separate power inverter for the control system was prompted by the low power requirement for the orbit mode during long flights. It was thought to be more efficient to use a separate inverter rather than the IGS power supply inverter, which was designed for heavy loads. Originally, two power inverters were to be included in the control system, but it was decided to use the IGS power supply as a backup instead. The control system was automatically switched over to the IGS inverter for power whenever the IGS was in operation. The AC frequency was chosen to be 400 Hz because of the availability of components.

Evolution of Control Modes

The attitude control modes were selected to permit maximum control by the astronauts, but yet to incorporate provisions for automatic control. The following modes were originally considered for the control system:

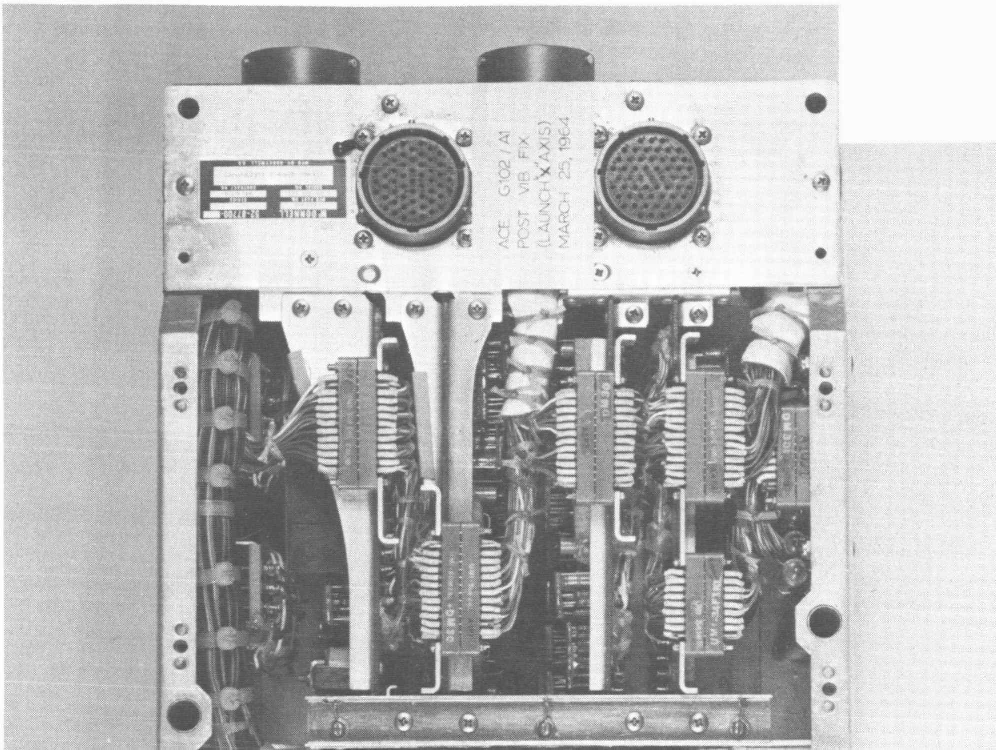


FIGURE 4.1-2 ATTITUDE CONTROL ELECTRONICS (COVER REMOVED)

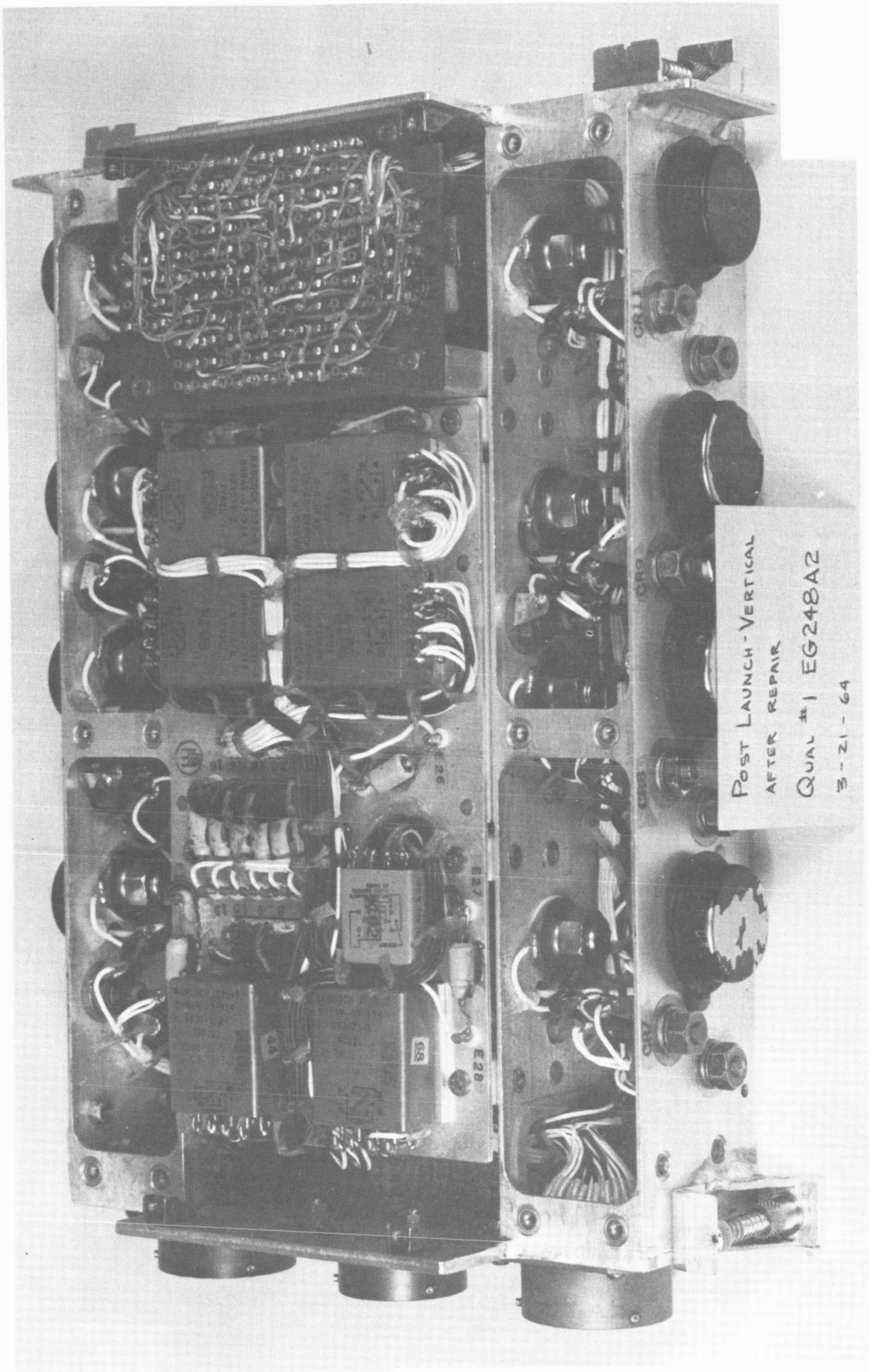


FIGURE 4.1-3 ORBIT ATTITUDE MANEUVER & ELECTRONICS (COVER REMOVED)

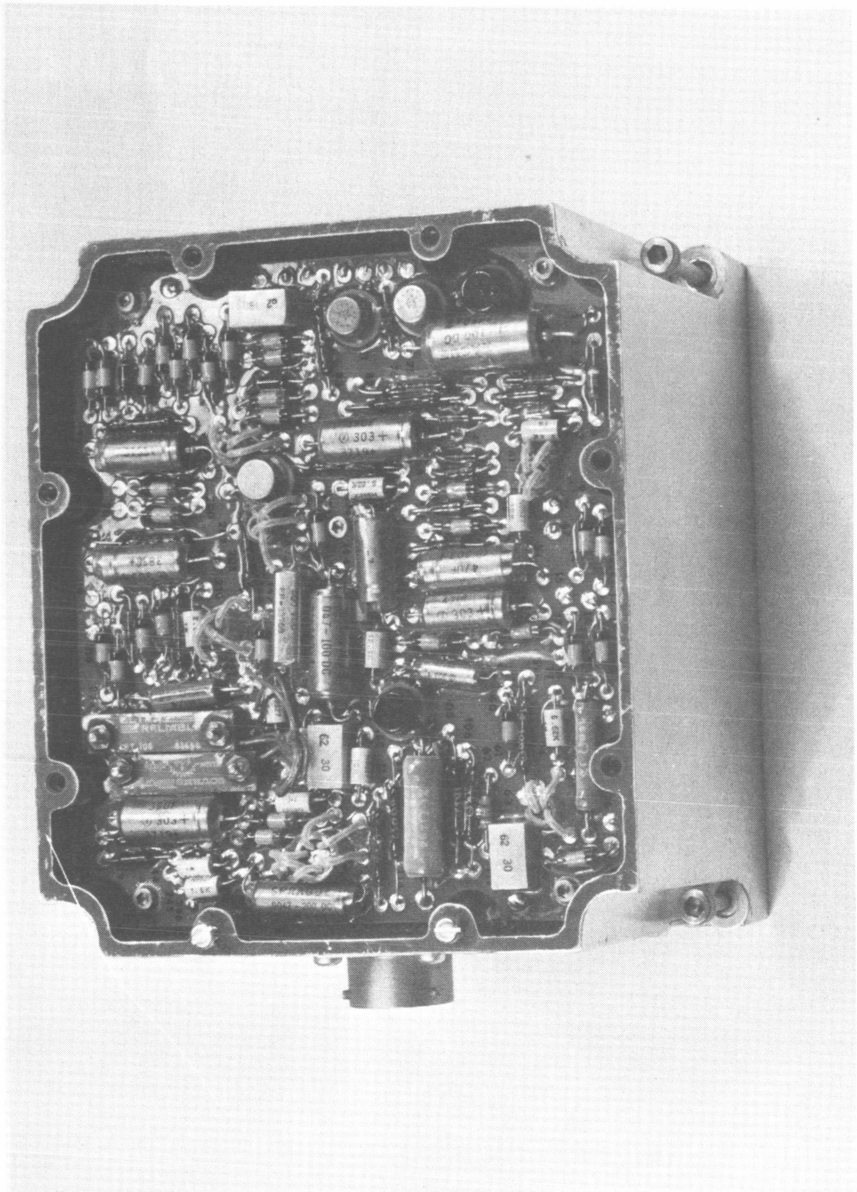


FIGURE 4.1-4 POWER INVERTER (COVER REMOVED)

4.1.2 (Continued)

(a) Rate Command - The astronaut could command a rate about each axis by means of the attitude hand controller. Signals from potentiometers on the hand controller were summed with rate gyro outputs to produce an error signal. With the hand controller at neutral, the spacecraft rates were damped to ±0.2 degrees per second using the Orbit Attitude and Maneuver System (OAMS) thrusters and ±0.5 degrees per second using the Reentry Control System (RCS) thrusters. The original rate command mode deadband was ±0.25°/sec for both OAMS and RCS. When the rate command/attitude hold modes (see (j) below) were deleted, the rate command mode deadband for the OAMS system was reduced to ±0.1°/sec, to permit accurate control to any attitude. This deadband was subsequently increased to ±0.2°/sec when the effects of greater-than-expected jet turnoff time, hysteresis, and rate gyro modulation precluded damping at ±0.1°/sec. The RCS deadband was increased from ±0.25°/sec to ±0.5°/sec because of the slow response of the relays used as solenoid valve drivers for the RCS jets. The ±0.2°/sec and ±0.5°/sec deadbands proved to be satisfactory.

(b) Direct Command - The hand controller could be used to command continuous thrusting about a selected axis when the hand controller was displaced beyond a threshold. Two types of direct command were available with the RCS thrusters; either through the control electronics, or through two sets of additional switches in the hand controller which directly engaged the appropriate thrusters

4.1.2 (Continued)

without going through any electronics. This latter mode was selectable by panel switches and was a highly reliable redundancy option since no electronic circuitry was required for its operation.

(c) Pulse - The hand controller would command a 20 millisecond thrust increment for each displacement of the hand controller beyond the threshold.

(d) Horizon Scan or Orbit - This mode was intended for loose control during long duration flights. Its evolution is discussed in detail in Section 4.1.3.

(e) Platform Attitude Hold (Horizontal) - The spacecraft was automatically controlled to within $\pm 1^{\circ}$ of the platform synchro nulls. This mode was always provided for in the electronics, but was only wired on Spacecraft 5 and up. It was used to control the spacecraft attitude during maneuver thrusting and for pointing the spacecraft for experiments, thus freeing the astronauts from having to control the spacecraft manually.

(f) Radar Attitude Hold - Provisions were incorporated for controlling the spacecraft pitch and yaw attitude within $\pm 1^{\circ}$ of the radar boresight. This mode was never mechanized on the spacecraft.

(g) Computer Attitude Hold - Provisions were incorporated for controlling the spacecraft's pitch and yaw attitude to $\pm 1^{\circ}$ of the commanded attitude from the computer. It also was never mechanized on the spacecraft.

4.1.2 (Continued)

(h) Retro Attitude Hold - Provisions were incorporated to bias the platform pitch signal to orient the spacecraft for retrofire. This mode was mechanized only on the unmanned Spacecraft 2 flight.

(i) Reentry Mode - The spacecraft roll axis was controlled to $\pm 2^{\circ}$ of the roll angle commanded by the digital computer. Pitch and yaw rates were damped to $\pm 4^{\circ}$/sec. Original specifications for this mode were $\pm 1^{\circ}$ deadband in the roll axis and $\pm 2^{\circ}$/sec rate damping, but were changed as a result of control dynamic simulations. This mode was used with great success for the reentry of the later spacecraft.

(j) Rate Command/Attitude Hold - This mode was to have provided a feature in conjunction with the digital computer and inertial platform, in which any attitude commanded by the hand controller would have been held after controller release. Both coarse ($\pm 5^{\circ}$) and fine ($\pm 1^{\circ}$) options were specified. However, this mode was deleted from the spacecraft early in the program (late 1962) because of its complexity and resultant size and weight penalty, and because of a need to conserve display panel space.

In addition to these modes, a reentry rate command mode, discussed in Section 4.1.3, was added. Also a paraglider control mode was incorporated, but was dropped with the deletion of the paraglider from the spacecraft and was never fully developed.

4.1.2 (Continued)

The maneuver thrusters were operated in direct mode only. Displacement of the maneuver hand controllers would command firing of the fore, aft, up, down, right, or left OAMS thruster. Originally, computer controlled maneuvering and a single pulse mode for maneuvering had been envisioned, and relay valve drivers were incorporated in the OAME to provide an interface with the electronics that would have been involved. These drivers were retained in the system and were used on Spacecraft 3 and 4 to drive the maneuver thrusters. On Spacecraft 5 and up, a more reliable direct connection between the hand controller switches and the thruster solenoids was incorporated.

Mode Switching and Redundancy Selection

Latching relays were used to select the redundancy options built into the control system: primary and secondary bias power supplies, logic circuitry, and jet select. Relays were chosen to perform these functions because solid state switching would have required a continuous application of power, whereas latching relays required power for switching only. Since the duty cycle of these relays was extremely low, a continuous application of power would have been inefficient and unreliable.

A problem developed early in the program (late 1962) with the primary/secondary jet select relays. These relays carried large currents, so a high-current relay was chosen. The contacts were designed to be cleansed by the arcing inherent in switching high currents. However, the switching between primary and secondary was invariably performed with no current,

4.1.2 (Continued)

and contaminants built up on the contacts. The problem was solved in early 1963 by changing to a lower current relay which could still handle the higher currents after switching. Circuitry was added to preclude switching while a jet was firing.

Solid state logic was chosen for mode switching because it minimized circuit complexity.

Use of Relays and Solid State Switching for Thruster Solenoid Valve Drivers

The Mercury spacecraft used only relays for driving thruster solenoid valves. The Gemini had tighter attitude control requirements than Mercury, and also a requirement for low fuel consumption, because of the longer flights. Therefore, transistors were used for switching the OAMS attitude thruster solenoid drivers, because of their faster response and their greater consistency in switching time. These thrusters were used during the orbital phase of the Gemini missions, where the tight attitude control, rate damping, and low fuel consumption requirements were critical.

Transistors were susceptible to damage from voltage spikes on the power bus, which were initially estimated to be as high as 130 volts, but this risk could be afforded because the OAMS thrusters were not required in order to bring the astronauts back to earth safely.

The transistors chosen for the OAMS drivers could withstand up to 150-volt peaks, but if one of the bus spikes were to appear simultaneously with a jet turn-off spike, a transistor could have been lost. No attempt

4.1.2 (Continued)

was made at suppressing the bus spikes, however, and as more data was accumulated through integrated systems testing, these spikes were found to be much lower than originally anticipated. No valve driver transistor failures occurred during the program.

Silicon power transistors were chosen for their superior performance over the anticipated temperature range and their low leakage current. However, even the low leakage current that was obtained caused a steady drain on the spacecraft batteries. On a future long term mission, this drain could become significant and should be considered in system design.

Because of the anticipated spikes, more confidence was initially placed in the reliability of relays. Therefore, they were used as valve drivers in life critical operations or where other considerations did not preclude their use, and thus were used as valve drivers for the RCS attitude thrusters. They were initially used also as maneuver thruster drivers, but on later spacecraft these thrusters' solenoids were driven directly by the maneuver hand controller switches. The RCS attitude control dead-bands and switching level tolerances were broader than those for the OAMS because of the use of relays.

Conclusion

Based on our experience with relay and solid state solenoid valve drivers, we recommend the use of solid state drivers in a new design. The only exception may be for an attitude control system on a long duration mission, where transistor leakage could cause a significant power drain.

4.1.2 (Continued)

Hot Side Versus Ground Side Switching

A tradeoff was made between switching the thrusters from the hot side versus switching these from the ground side. Hot side switching provides a fail-safe system, since wires shorted to ground do not turn the thrusters on, whereas, in ground side switching, a short on the ground side of a thruster causes the thruster to "fail" on. However, use of hot-side switching would have required additional switches, so ground side switching was used on Gemini. Another consideration was that the only silicon transistors available at the time of the Gemini design which could handle the current through the OAMS thrusters were NPN, which would have required a separate floating power supply for hot-side switching. Comparable PNP transistors have since been developed, which can be connected for hot-side switching with existing grounded power supplies.

Circuit breakers were provided to isolate a failed thruster. Each OAMS attitude and maneuver thruster was powered through an individual circuit breaker. Each RCS thruster ring was powered through three circuit breakers. A mission could be completed with one failed thruster.

One apparent electrical failure in a thruster circuit occurred during the Gemini Spacecraft 8 flight, where the OAMS Number 8 thruster commenced firing without pilot actuation, was briefly interrupted, then recommenced firing until its circuit breaker was opened. Although the thruster and its associated valve driver circuitry was not returned from orbit, the telemetry data indicated that the failure was probably produced by a short

4.1.2 (Continued)

to ground somewhere in the thruster's electrical circuit. As a result of this problem, a circuit breaker was added to the astronauts' switch panel to remove power from all jets at once.

Spike Suppression

Spike suppression circuitry was incorporated in the control electronics to reduce the amplitude of the inductive spikes generated when the thrusters were turned off. The main reason for this circuitry was to protect the valve drivers, and not to reduce electromagnetic interference. The additional circuitry was located in the Attitude Control Electronics (ACE) and Orbit Attitude and Maneuver Electronics (OAME) for the RCS and OAMS propulsion systems, respectively.

A tradeoff existed as to the location of the circuitry. Placing it within the ACE package caused some problems, since the high currents through the circuitry produced noise which interfered with other circuitry, contributing to switch chattering. Also, during early tests, some double pulsing of the minimum pulse generator was evidenced because of the proximity of the spike suppression circuits. However, this problem never occurred in any integrated spacecraft testing, so no action was taken to eliminate it. The alternative location for the suppression networks would have been near the jets. However, the jets' temperature was thought to be too high for the circuitry to withstand. Also, since the circuitry was primarily to protect other electronic components, it was desirable for them to be part of the same subsystems.

4.1.2 (Continued)

A future system can probably be designed to better withstand the effects of adjacent spike suppression by improving internal grounding techniques and by shielding the suppression circuitry from the more sensitive electronics. The alternative would be to locate the suppression circuitry in a separate package, with an attendant size and weight penalty.

The circuitry for the attitude thrusters was a conventional diode in series with a zener diode, the combination being connected in parallel with the solenoid coils. This particular combination was chosen because of the requirement for a rapid jet turnoff in the attitude control circuits. The solenoid would have turned off rapidly enough by itself, but in doing so would ring, with resulting electromagnetic interference problems and possible damage to control circuitry.

A conventional diode across the coil would have prevented the ringing, but would have extended the turn-off time prohibitively. (With a perfect diode across a perfect inductor, the current would persist indefinitely, with the jet never turning off.) The zener in series with the diode rapidly dissipated the energy stored in the inductor and permitted its turn-off.

The maneuver thrusters originally used the same circuit. However, it was found that, since the inductance of the maneuver solenoids was higher than the attitude solenoids, their energy storage at turn-off was sufficient to damage the junction of the zener diodes. Therefore, only the conventional diodes were retained. This increased the turn-off time of the maneuver jets, but turn-off time was not critical in this case.

4.1.3 Problems and Design Changes

Evolution of the Horizon Scan Mode

The orbit attitude control mode (horizon scan mode) utilized position signals from the horizon sensor to maintain the spacecraft pitch and roll axes relative to the horizon. The original concept for this mode was a carry-over from the Mercury attitude control system. The vehicle was to be stabilized by applying corrective moment impulses at each of several values of attitude error. The advantages of this system would have been a low steady state fuel consumption and insensitivity to switching hysteresis.

However, there were misgivings about its mechanization, because it required close tolerances on the thrust impulses to remain stable, and may have required the astronaut to damp out angular rates before it could be effective. Also, it would have required entirely separate electronics from that used in other control modes. Therefore, very early in the program (early in 1962) another mechanization was sought which would minimize, if not eliminate, these disadvantages and still retain the low fuel consumption required.

A lower fuel consumption could have been achieved by using rate gyros in the loop. However, this mode was intended for use during the long periods of orbital flight, where overall power and fuel economy and reliability were primary considerations. A trade-off study revealed that the use of rate gyros would have caused an increase of approximately 8 lb in the amount of fuel cell fuel required, and also would have reduced control system reliability.

4.1.3 (Continued)

A system was therefore proposed that would have employed the horizon sensor, a lead network, a switching amplifier, a constant pulse repetition frequency pulse generator, and the reaction jets. The lead network, by differentiating the attitude signal, would have provided a measure of attitude rate. The switching amplifier would have turned on the pulse generator and selected the proper jet to be driven by the pulse generator.

The mechanization finally decided upon utilized pseudo rate measurement, or derived rate increment stabilization. This method had advantages over the lead network system, in that it was capable of overcoming higher initial rates (2.5° per second versus 1° per second) and required fewer new components to mechanize. Also, it was thought that, in addition to being used on the orbit mode, this system could also provide a variable-acceleration manual mode in the pitch and roll channels. This manual mode was never mechanized, however, because the yaw channel would also have needed modification and the mode switching would have been complicated by the mechanization.

The pseudo rate system operated by feeding back the acceleration (jet firing) pulses through lag networks, thus deriving the rate. This derived rate was summed with the horizon sensor signals and fed into the switching amplifier. The use of pseudo rate presented a potential problem in that the pseudo rate could turn off the switch before the vehicle reaction jets developed thrust. To avoid this, a minimum pulse generator was provided which prevented the switch from turning off in less than 18 milliseconds from the time of turn-on. By thus assuring a pulse of no less than

4.1.3 (Continued)

18 millisecond duration, the possibility of the pseudo rate turning off the switch prior to development of jet thrust was eliminated. The output of this system was a train of pulses whose PRF was proportional to the input signal. Thus, the steady-state limit cycle consisted of a very low pulse repetition frequency and the system was capable of very low fuel consumption, on the order of .05 lb/hour, exclusive of the yaw axis requirements.

Individual Circuit Breakers

The Gemini employed a bipropellant thruster system, with each jet requiring solenoid valves for fuel and oxidizer. If one of these valves failed closed, the other would operate on each firing command with no resulting thrust, and thus much fuel or oxidizer would have been wasted. To provide for this event, an early design change was made to provide bus power to each OAMS attitude and maneuver jet through an individual circuit breaker, so that power could be removed from both solenoids of a failed jet. At the same time, provisions were added to allow selection of either the pitch or yaw jets for roll control. These changes provided the capability for completing a mission after the failure of one OAMS jet.

Incorporation of Reentry Rate Command Mode

In June 1963, a Reentry Rate Command Mode was added. This mode differed from the orbital Rate Command Mode in that the control was less sensitive and a roll-to-yaw cross-coupling feature was provided to permit the spacecraft to roll about the velocity vector rather than about the spacecraft roll axis. (The difference in the two axes is caused by the offset

4.1.3 (Continued)

center of gravity.) The deadbands, corresponding to the rate deadbands of the reentry mode, were originally $\pm 2^{\circ}$/sec in pitch and yaw and $\pm 1^{\circ}$/sec in roll, but were widened, in mid-1964, to $\pm 4^{\circ}$/sec in pitch and yaw and $\pm 2^{\circ}$/sec in roll, since aerodynamic forces during reentry provided most of the damping required in pitch and yaw.

Vibration Problems

Several changes were required in the control system packages to make them capable of surviving the launch environment. These changes consisted primarily of structural modifications and additional support for wire bundles. The problems were partly due to the complex shapes required for the packages and the resulting difficulty in analytically predicting their behavior under vibration.

Conclusion

In the future it would be advisable to perform vibration tests on a representative mock-up as early in a program as possible, especially where complex packaging requirements exist.

Effects of Harmonics and Phase Shift

The Gemini attitude control electronics utilized AC signal summing, a prerequisite for which is "clean" signals; i.e. pure sinusoids with little harmonic or phase distortion. It was recognized early in the program that the signals supplied to the attitude control electronics from the Inertial Measuring Unit, digital computer, and rate gyros would not be ideal, and therefore steps were taken to minimize the effects of distortion.

4.1.3 (Continued)

The original design employed a half-wave demodulator. Early testing revealed that the sensitivity of this demodulator to harmonics and quadrature caused the switching deadbands to converge when high attitude and rate error signals were summed. A full-wave demodulator was incorporated, which, to a large extent, alleviated the problem. A jet lock-out circuit was also incorporated, to prevent opposing jets from firing simultaneously.

Even after the demodulator change, harmonics and quadrature still were causing some problems. Testing showed that a quadrature component in the input signals would cause widening of the deadbands as the quadrature level increased. This effect was due to saturation of the switching amplifier and demodulator. Harmonics in the input signal could cause the deadband to widen or narrow, depending on the phase relation of the harmonic to the 400 Hz reference. Saturation of the switching amplifier and demodulator prevented the deadband from closing to zero.

A full-wave demodulator with infinite filtering is theoretically a perfect quadrature rejection circuit. However, the system operation must be linear, with no active element saturation, to get theoretical operating characteristics. Thus, to remove the quadrature problem, the switching amplifier would have had to be changed to allow a much greater swing in output voltage. The demodulator would have had to be altered to handle the large signal, and an active filter for the demodulator would have had to be incorporated.

4.1.3 (Continued)

These solutions to the quadrature problem would have increased the harmonic problem, since the linear operation of the system would have allowed the deadbands to converge to zero. Low pass filters would have been required in the signal inputs to reject harmonics. These changes would have been prohibitive in terms of size, weight, and power. Dynamic testing was performed to determine whether the system as mechanized was satisfactory, and the results indicated that the harmonics and quadrature would not present a noticeable problem to astronauts performing a rendezvous. Therefore, no changes were made. These findings were borne out during the actual flights, where nominal performance of the control system was indicated.

Conclusion

In view of the fact that no operational problems existed from these phenomena, it would be unwise to recommend the proposed changes for a future system, without a detailed study of their effects on the total system. Certainly, on a spacecraft with tighter deadband requirements than Gemini, the effects of signal distortion would be more significant.

Part of the difficulty experienced with distortion stemmed from the inadequate knowledge of the input signals by the control system subcontractor. If any recommendation can be made, it would be to learn as much as possible early about interfacing signal characteristics, and preferably to put tight specifications on sensor inputs to control the interface.

4.1.3 (Continued)

Noise Effects on Control System

In each axis of the attitude control system, error signals were amplified, demodulated, and then fed into a low-hysteresis switch. This switch was designed so that the input could turn the switch on but could not turn it off. The first stage of the switch was automatically forced off 400 times each second by a dither signal applied to the switch internally. If the input signal were below the switching level, the switch would remain off. Thus, only an extremely small reduction in input signal was required to allow the switch to remain off and hysteresis was for all practical purposes avoided. However, if the input were above the switching level, the switch would come back on and the process would be repeated. The continual switching resulted in a square wave output of the first stage of the switch which was unsuitable for operating the jet drivers. To obtain a usable signal, an integrator was provided in a later stage of the switch to smooth the square wave and provide a steady-state DC signal to the last stage of the switch (inverter). The inverter then provided a steady ground to operate the valve drivers.

Because of the extremely low hysteresis, noise on the input lines caused the valve drivers to chatter when the input signals were in the region of the switching level. This situation was significant only during system testing because of the high noise environment of the test complex. To evaluate control system performance, a testing specification for a chatter band was formulated, based on results of early integrated system testing.

4.1.3 (Continued)

(No evidence of chattering was apparent during flights.) The chatter band was reduced by the following modifications to the ACE package made in late 1964:

(a) The low hysteresis switch was changed so that switching would occur on the positive half-cycle of the dither. This change was based on the subcontractor's empirical data, which showed that, due to differences in loading, the positive half-cycle was less susceptible to noise than the negative half-cycle.

(b) The grounding in the ACE package was changed to separate the ground of the low hysteresis switch from the analog signal grounds. Previously, this ground was coupled into the system, inducing an apparent hysteresis.

Both of the above changes reduced the chatter band significantly. Other improvements would have further reduced it. These included the use of internal shielding (with an attendant weight and volume penalty); the removal of spike suppression circuitry from the ACE package; and an increase in the dither frequency for the low hysteresis switch. The last change would have reduced the time delay uncertainty by turning off the switch more frequently. A trade-off exists between reduced time delay and hysteresis, and so an increase in hysteresis was permitted, with a resulting reduction in chatter.

Valve Driver Retriggering

The original minimum pulse circuitry was found to be susceptible to transients at the input. This original circuit had a pulse transformer

4.1.3 (Continued)

in its input circuit which rang when excited by a transient. The circuit was redesigned in late 1963 to eliminate the pulse transformer.

Later, it was found that the valve drivers could be retriggered by radiated or conducted voltage within the ACE package, induced by large currents produced when the RCS jets were deenergized. These large currents were the result of including the RCS spike suppression in the ACE package. The RCS spikes could cause a refiring of an 18 millisecond minimum pulse in two ways:

(a) The spike could appear at the output of the demodulator, whereby the same jet or a jet in another axis could be triggered.

(b) The spike could appear at the input of the switching amplifier, duplicating the above jet firing or firing the opposite jet in the same axis.

The above problem appeared during qualification testing when the jets were commanded to fire four times per second. Even under these conditions, however, the most frequent retriggering was observed to be once per minute. Extensive retesting during acceptance testing never produced the problem.

It was our opinion that the condition would not exist during actual spacecraft operation and therefore no corrective action was taken.

Relay Solder Ball Problem

Considerable production problems were encountered with the power relays used as solenoid valve drivers. In early 1964, solder particles, formed

4.1.3 (Continued)

while sealing the can, were discovered. Some were large enough to cause pin-to-pin shorts. The problem was resolved either by reworking the affected relays (using an improved process that virtually eliminated the possibility of solder particle contamination) or by using a new relay design, incorporating a welded can.

The new relay design, however, introduced a hang-up problem, caused by galling of the armature and pivot. This problem was solved by a change in the armature material. Both of the above problems had an impact on other spacecraft subsystems, in particular the computer and IGS power supply, which used similar type relays. Although these systems did not experience the same problems, the same precautions were taken.

<u>ACME Power Inverter Design Problems</u>

A power inverter design change was necessitated because other systems, in particular the horizon sensor, were susceptible to transients. The inverter exhibited an output transient of up to 200 volts, peak-to-peak, when first energized. Installation of back-to-back zener diodes limited the transient to 60 volts maximum.

The power inverter was overly susceptible to input ripple of 120 Hz, due to resonance of the input filter network. Several solutions were suggested, including lowering the resonant frequency, mounting an additional filter in a cable or plug, or including an additional filter as a separate component. This problem, however, was not serious enough

4.1.3 (Continued)

to warrant the weight and packaging penalties incurred with any of these solutions. No problems occurred during systems tests or flights as a result of this susceptibility.

The efficiency of the power inverter did not meet the original specification requirements. The inverter was designed for input voltages from 22 to 30.5 volts and was required to operate during intermittent bus voltage extremes of 20 and 32.6 volts. Therefore, there was a trade-off between frequency stability and efficiency, and efficiency was sacrificed for better performance.

The inverter was protected against open circuits and dead shorts, but not half-wave shorts. A future design should include such protection, especially if zener diodes are used to suppress output transients.

5.0 Pilot Controls and Displays

The design evolution of pilot controls and displays for the Gemini Guidance and Control System is discussed in the following paragraphs. These controls and displays are pictured in Figures 5-1 through 5-3. The left hand and right hand instrument panels of a recovered flight spacecraft are shown in Figures 5-1 and 5-2, respectively. The following components are shown in these photos: (1) Attitude Indicators; (2) Flight Director Controllers; (3) Incremental Velocity Indicator; (4) Range and Range Rate Indicator; (5) Manual Data Insertion Unit; and (6) Maneuver Hand Controller. The photos also show the notations marked on the panels by the astronauts. A view of a cabin mock-up in Figure 5-3 shows the center console with the following controls (some of which were discussed in previous sections): (1) Radar Control and IMU Mode Selector; (2) Computer Controls; (3) Attitude Control Mode Selector; (4) Horizon Sensor Switch; (5) Rate Gyro Switches; and (6) the Attitude Hand Controller.

5.1 Attitude and Maneuver Hand Controllers

5.1.1 Requirements - A design requirement was to provide the astronauts with a means of manually controlling the spacecraft's attitude and translational movement. This was accomplished by attitude and maneuver hand controllers which enabled the astronauts to override all electronic control equipment, thus giving them full spacecraft control capabilities.

5.1.2 Initial Design Tradeoffs and Decisions

Mercury Attitude Hand Controller Experience - Initially, a hand controller which was mechanically linked directly to the thruster valves was considered for Mercury. This was abandoned because of excess friction and

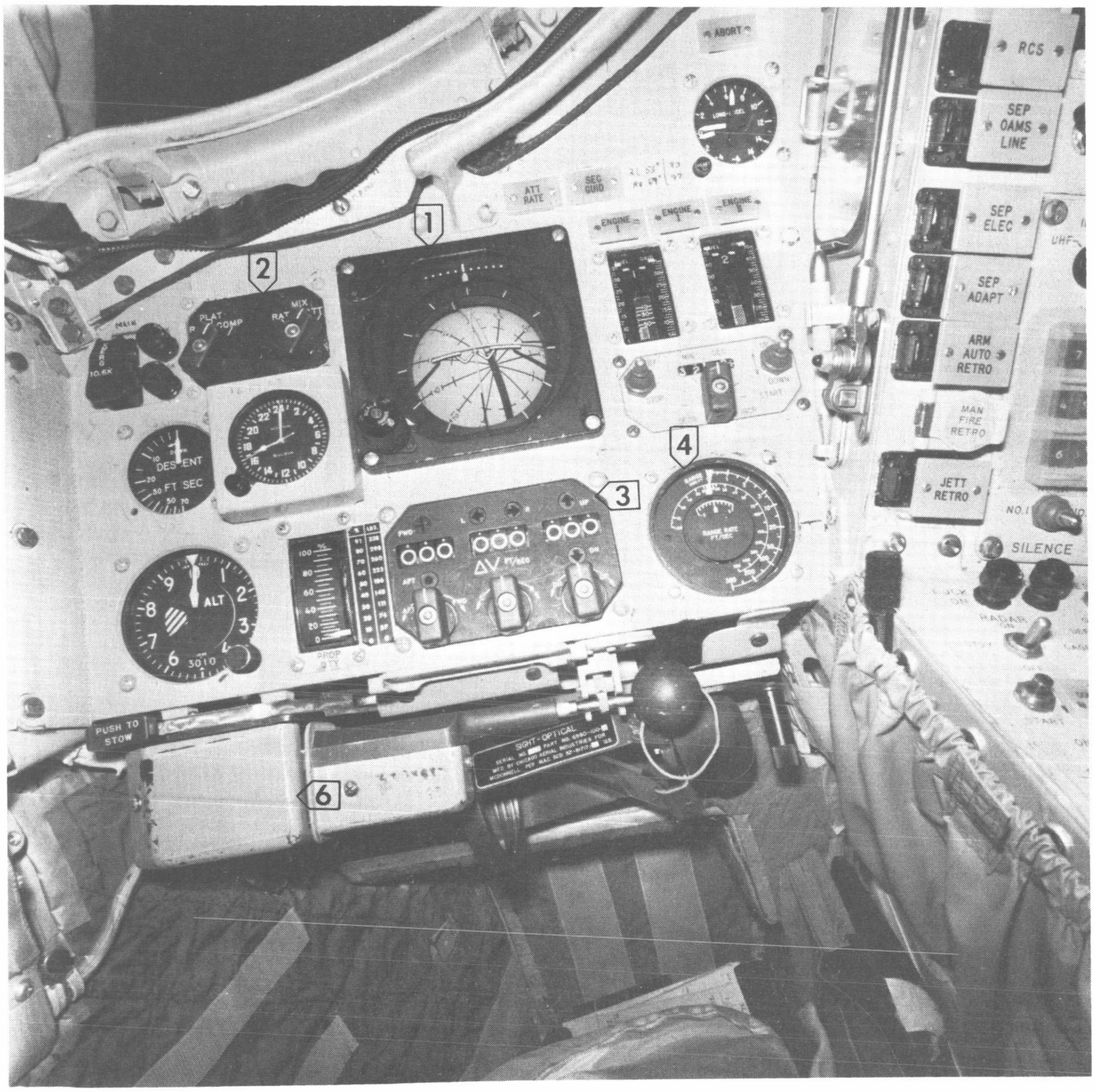


FIGURE 5.0-1 LEFT HAND INSTRUMENT PANEL
(FLIGHT SPACECRAFT)

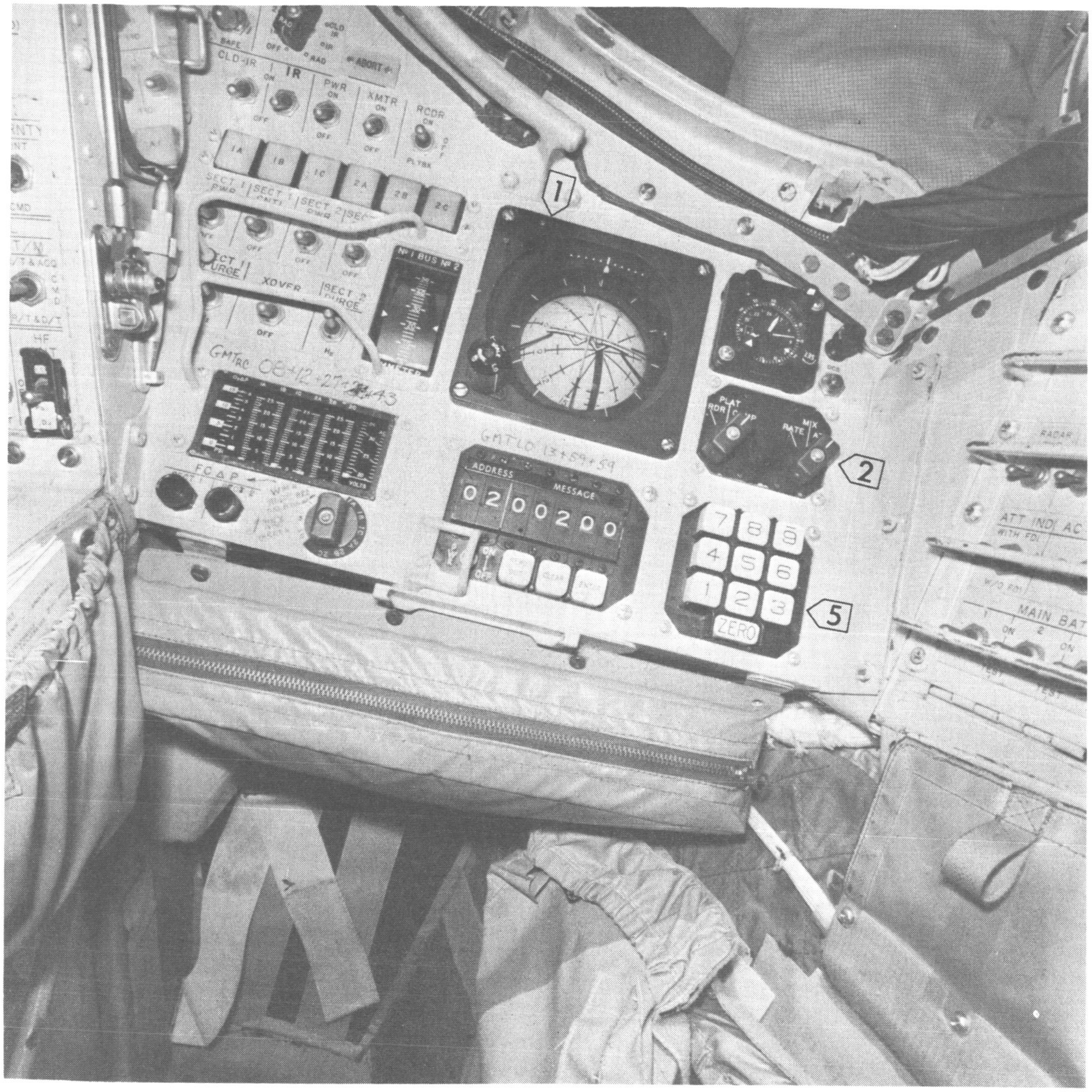


FIGURE 5.0–2 RIGHT HAND INSTRUMENT PANEL (FLIGHT SPACECRAFT)

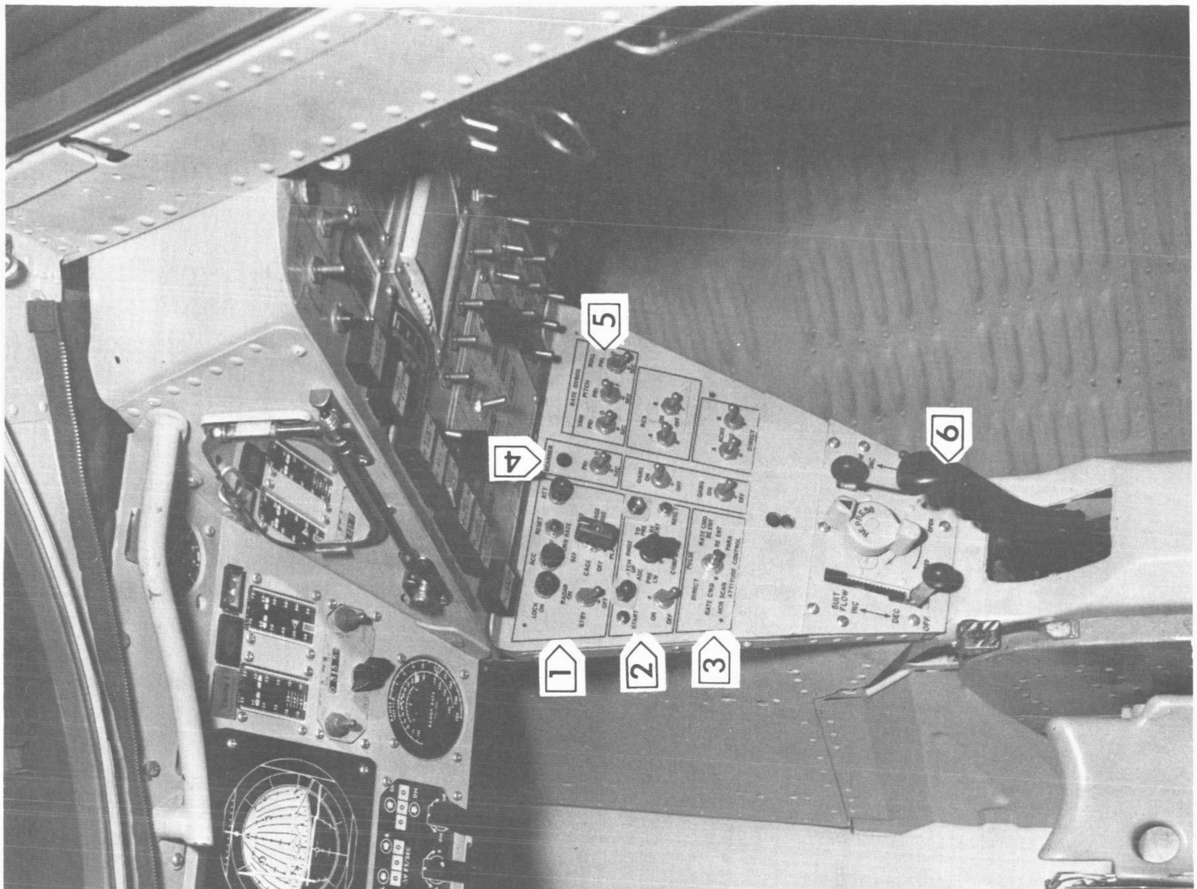


FIGURE 5.0-3 CABIN MOCKUP

5.1.2 (Continued)

hysteresis caused by routing translational linkage through pressurized bulkheads.

The Mercury attitude hand controller had two modes of operation, "fly-by-wire" and manual. For "fly-by-wire", the handle contained four switches for each axis, two each for each direction (i.e. pitch up, pitch down, etc.) and two each for either high or low thrust, and utilized the automatic reaction jets. The low thrust switches were actuated at 30% of full handle travel and the high thrust switches at 75% of full travel. In the manual mode of operation, the hand controller was linked mechanically to throttle valves inside the pressurized area (avoiding the problems mentioned above). The manual thrusters were utilized in this mode, and thrust output was directly proportional to the amount of handle displacement.

The handle pivot point was at the center of the wrist. This location was thought to be best because it utilized the natural human hand movement. However, flight experience showed that mechanical constraints impeded wrist motion and made this pivot point undesirable. The handle was not redesigned for Mercury because of time constraints.

Three new concepts evolved from the Mercury attitude controller experience:

1. The hand controller permitted three-axis control, either individually or simultaneously.

5.1.2 (Continued)

2. The movement of the hand controller handle should be analogous to the desired movement of the spacecraft.

3. The center of the human wrist was not the best place for the handle pivot point because of mechanical differences between the wrist and the controller.

Gemini Attitude Hand Controller - At one time we considered using a hand controller for pitch and roll axis control and rudder pedals for yaw control. This approach was abandoned because of space and weight limitations and because the rudder pedals proved cumbersome. A conventional aircraft hand controller was rejected because of unwanted inputs experienced under high-g loads. We also considered a force field system hand controller which used strain gauges to give a spacecraft control response proportional to the handle motion. This controller was rejected because it depended upon associated electronic equipment which was not reliable enough.

The selected hand controller contained six switches for control about each directional axis (i.e. pitch up, pitch down, etc.), a three-ganged linear potentiometer in each of the three axes, and wiring for Rate Command, Direct-with-ACME, Pulse, and Direct-without-ACME modes of operation. The three-ganged potentiometer contained one potentiometer for rate signals for attitude control, one potentiometer for instrumentation of the handle positions, and one potentiometer for paraglider control, which was not used. The pitch and yaw pivot points were located at the center of the astronaut's hand and, due to the complexity of design, the roll pivot

5.1.2 (Continued)

point was located below the astronaut's hand. This provided a clear distinction of pitch, yaw and roll to prevent inadvertent cross coupling.

This design could be operated in both high and low level g fields with negligible cross coupling.

Conclusions -

1. The attitude hand controller handle should have the pivot points in the center of the hand for ultimate control. However, the roll pivot point below the hand proved to be satisfactory in providing clear distinction to pitch and yaw in preventing cross coupling.
2. The controller should not go through any electronics for direct mode operations. However, this method tends to use more thruster fuel than the automatic modes.
3. A non-linear potentiometer in place of the linear one used would give the astronaut finer control at low rate levels.
4. The implementation of rudder pedals for the yaw axis control in landing the spacecraft by paraglider would be worthwhile.

Maneuver Hand Controllers - Initially, a pulse system was devised for translation control. The handle would be positioned for the desired response and then a button would be pushed for a pulse jet firing. This design was scrapped because it proved cumbersome to operate.

The selected maneuver hand controller required only one movement to actuate a translation thruster or combination of translation thrusters. The

5.1.2 (Continued)

handle contained one switch for each of the six translation directions (i.e., up, down, etc.).

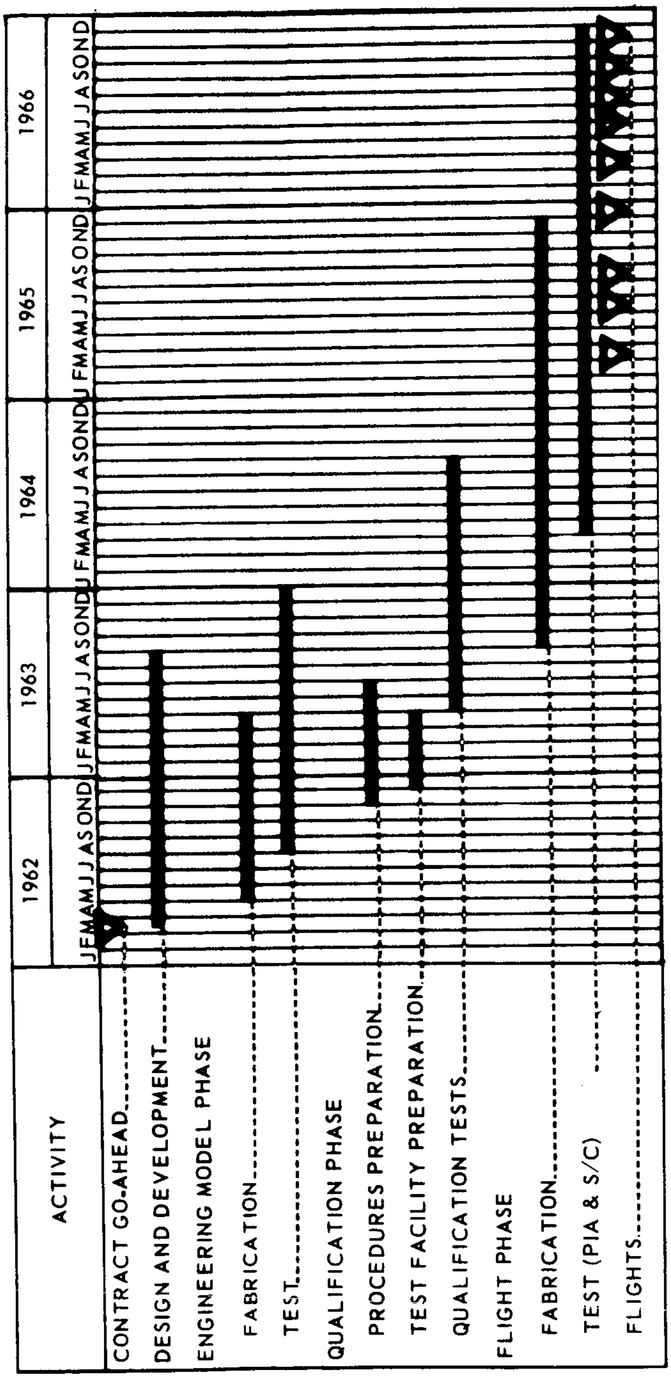
Early in the Gemini program it was decided to have two maneuver hand controllers in the spacecraft, one for each astronaut. This allowed both astronauts to gain experience in docking control, a maneuver which utilized both the attitude and maneuver hand controllers. The second maneuver hand controller was a miniature of the original, but used the same size grip. Two maneuver hand controllers were first installed in Spacecraft No. 5.

5.1.3 Development Problems, Tradeoffs and Design Decisions - The time history of the development of the Gemini hand controllers is shown in Figure 5.1-1.

Gemini Attitude Hand Controller - Shaping the grip to satisfy individual crew members was a major problem. Finally, a removal grip was designed to be used for testing the handle only. It was intended that NASA would replace the removal grips with grips customized for specific crews. However, one grip design was used throughout the entire program without being changed or replaced.

The forces required to position the handle of the attitude hand controller presented a continuing problem throughout the Gemini program. Undesirably high forces were required to actuate the switches and position the potentiometer in each axis. Switch and actuator movements were controlled by springs with very tight tolerances to preclude any damage to the switches. Due to the limitation in the size of the box and the number of switches in each axis, the switch actuation forces were higher than the design goal. These forces made the handle uncomfortable to use in an unpressurized suit,

FIGURE 5.1-1
HAND CONTROLLER PROGRAM TIME HISTORY



5.1.3 (Continued)

but did not degrade the crew's capability. The flight crews were extremely patient and understanding with this problem and accepted the tradeoff of comfort for improved system integration and confidence that the handle positioning would not damage the switches.

The size and stiffness of the wire bundle were reduced to lessen the hysteresis forces that were caused by the limited space inside the package and the flexing of the wire bundle during handle movement. A smaller gauge wire with a more flexible insulation was used. Even then, the routing of the wire bundle was very critical to eliminate added forces and possible binding.

In the last four spacecraft, the handle was changed slightly with good results. The following changes were made:

(a) A "T" configuration was added to the original grip for an overhead grip control. This improved the yaw control when the astronaut's suit was pressurized (mainly during EVA) and the cabin unpressurized. The change was necessary to reduce wrist fatigue experienced during training for extended periods of EVA for these later flights.

(b) The roll rate output of the hand controller was changed to a maximum of 15°/sec for a total handle travel of 9° from neutral. Originally the maximum pitch, yaw, and roll rate output of the hand controller was 10°/sec for a total stick travel of 10° from neutral. This change was made to increase the roll gain capability during the reentry phase of the spacecraft flight.

5.1.3 (Continued)

(c) The deadband of the hand controller potentiometers, used in the rate command mode, was widened from 1° to 1.1° of the handle movement from neutral. This change was made because the potentiometer deadband locations shifted slightly during qualification vibration testing. This change had no measurable effect on the hand controller operation or spacecraft control.

(d) Diodes were incorporated in series with the hand controller switches to rule out any possibility of improper jet firing. In the rate command mode, the closing of the direct command switches could produce "sneak" circuits causing improper jets to fire, in addition to those jets actually commanded by the difference between the commanded and actual rates. This condition could have caused a temporary loss of control and an additional expenditure of fuel, although the commanded rates would eventually have been achieved. Also, an excessive current of up to 10 amperes would have been drawn through the RCS relay contacts, which were rated at 5 amperes, and current of 8 amperes could have been drawn through an RCS thruster circuit breaker rated at 5.4 amperes. The need for this precaution was questioned at the time, because testing which simulated retrograde and normal maneuvers failed to demonstrate "sneak" circuit conditions actually existed in normal operation. Also, an original NASA directive which ruled out any electronics in the RCS "direct-without-ACME" switch circuits had to be violated.

5.1.3 (Continued)

Switch failures were experienced throughout the Gemini program. These failures fell into three major categories: (1) contamination, (2) overstress, and (3) hermetic seal leakage.

Contamination and excessive voltage drops accounted for the majority of the failures. To minimize these problems the switch manufacturer incorporated improved quality control and processing procedures which substantially reduced switch contamination. Then the switches were subjected to a 100% voltage drop screening test to eliminate any contaminated switches.

The mechanical overstress of the switches was attributed to the procedure of setting the maximum allowable over-travel of the switch during the assembly of the hand controller. This problem was eliminated by insuring that the plunger assembly used to depress the switch actuator was adjusted so it was retracted prior to switch assembly and attitude controller grip assembly movement.

The hermetic seal leakage caused unwanted pressure differentials when going into a pressurized condition after being in an unpressurized condition for a period of time. This was overcome by performing a 100% altitude-actuation force decay test during the switch acceptance testing.

Conclusions -

1. The position of the hand controller in the spacecraft is critical because of the sitting position and fatigue encountered during extended periods of stick control.

5.1.3 (Continued)

2. The switches, springs, and general handle assembly tolerances are extremely sensitive. This is due to the limited amount of space available and the excessive forces required to actuate the switches if proper tolerances are not maintained.
3. A 100% screening of the switches must be incorporated to detect hermetic seal leaks and contact contamination.

Maneuver Hand Controllers - The shape of the grip was the obstacle to overcome in the development of the maneuver hand controller. A ball of approximately 1-1/8 inch diameter was accepted to be the best grip. This provided the astronaut with adequate feel control whether his suit was pressurized or unpressurized.

The miniature maneuver hand controller was positioned in the spacecraft so that the astronaut had ready access to it while using the attitude hand controller. However, the miniature maneuver hand controller then became a convenient handle to help those working inside the cabin to move around. Since the handle was not built to take these excessive loads, the stops and mechanical linkage were deformed. A guard was built around the handle to preclude inadvertent handling by anyone in the cabin.

5.2 Manual Data Insertion Unit (MDIU)

5.2.1 Requirement - The basic MDIU requirement was to provide a means for the crew to communicate with the computer.

5.2.2 Initial Design Tradeoffs and Decisions - Initial effort was devoted to determining how data would be displayed and inserted, and to the merits of housing the insertion and display functions in the same or separate packages. We decided to house the insertion and display functions in separate packages because of (1) the physical limitations of the instrument panel (it was easier to mount two small units than one large one), and (2) human factor considerations.

Consideration was given to what type of device would be best for inserting data. Thumbwheels were originally considered, but discarded because of slow insertion speeds and difficulty in sealing the required rotary switches. A telephone-type dial was evaluated and found to be satisfactory from a human factors viewpoint, but a dial was significantly slower to use than a keyboard. Different schemes for increasing the dial rate of return were examined, but a keyboard was eventually selected. After evaluation of several different keyboard configurations, an adding machine configuration was selected because a minimum amount of training was required for a crewman to become proficient in using it. "Egg-crate" guards were placed around the keys to prevent the depression of more than one key at a time. A decimal format for insertion was chosen over binary or octal to minimize crew training requirements.

Numerical displays on electro-mechanical (wheel) devices and binary displays on lamps were judged to be the most promising readout approaches.

5.2.2 (Continued)

Electro-luminescent devices were not considered because the circuitry required for them was too complex, and the display intensity would have been insufficient in bright sunlight. The wheel devices were relatively slow in displaying a data word but were easily interpreted by the operator. Binary devices could display a word more quickly and had less complex circuits. However they were subject to misinterpretation because it was felt that the crews would have insufficient training time to become proficient in binary conversion. Therefore, to minimize the chance for error, decimal wheel devices were selected for the readout.

5.2.3 Problems and Design Changes During Development and Qualification Testing - No significant problems occurred during development and qualification testing. The time history of the design and development of the MDIU is similar to that of the computer.

5.2.4 Problems and Design Changes Associated With Systems Tests and Flights - The keyboard switches provided no positive indication of actuation. While this did not create any problems in the classical sense, it was a source of irritation and concern to the crew. In future programs, switches which provide a positive indication of actuation should be specified.

During certain parts of the Gemini mission the MDIU operation would be required to read out certain parameters over extended periods of time. Each readout required an action on the part of the operator.

Conclusion - In future programs consideration should be given to including a computer program which would automatically update certain parameters periodically.

5.3 Incremental Velocity Indicator (IVI)

5.3.1 Requirement - The IVI provided visual indication of incremental longitudinal, lateral, and vertical velocity components, and the associated signs for each, to be imparted to the spacecraft by maneuver thrusters.

5.3.2 Initial Design Tradeoffs and Decisions - The original philosophy for the IVI was to have a display, driven by means independent of the computer, capable of assisting the crew in the execution of a maneuver in the event of a failure of some other spacecraft component. The system complexity and the resulting cost and size of an independent unit capable of participation in a closed loop maneuver was considered prohibitive for a redundant function. Conversely, it was considered that the advantages of an open loop timing device were not sufficient to justify its size and cost. After considering the various aspects, it was decided to make the IVI a natural extension of the computer for closed loop operation.

5.3.3 Development Problems, Tradeoffs and Design Decisions - The time history of the design and development of the IVI is similar to that of the computer. The IVI consisted of three 3-digit indicators, one for each axis of motion. The indicators were designed to be capable of manual reset. This was provided by three knobs (one knob for each indicator) which drove the indicators at a rate proportional to deflection. This type of slewing control was found to be quite time-consuming in setting in precise quantities. Therefore, different and discrete slewing rates were substituted for the proportional control.

5.3.4 Problems and Design Changes During Development and Qualification Testing - Two minor problems were encountered as a result of the vibration qualification test. Vibration caused some transistors to fail. The mode of failure was shorting between the posts (base and emitter) and the can (collector). Transistors with larger clearances between the posts and can were substituted with satisfactory results.

A leak problem (case sealing) was also experienced as a result of vibration. The problem was corrected by adding a "T" bar across the width of the case to give support.

5.3.5 Problems and Design Changes Associated With Systems Test and Flights - The IVI demonstrated itself to be very satisfactory throughout the Gemini program.

5.4 Attitude Display Group

5.4.1 Requirements - Spacecraft crews required a natural and continuous display of spacecraft attitude and rate information to perform the following control tasks:

Mission Phase	Display Information	Crew Control Task
Launch	Attitude and Rates	Monitor booster attitude control
Spacecraft Separation	Attitude and Rates	Damp rates, then assume small-end-forward attitude.
Orbit Adjust	Attitude	To align spacecraft roll axis along the orbit correction vector for thrust application.
Orbit	Attitude and Rates	To assume and maintain orbit attitude.
Rendezvous	Attitude and Rates	To align spacecraft for the thrust application during slow catch-up maneuver.
Rendezvous - Terminal Phase	Attitude and Rates	Maneuver Thrust Application
Rendezvous Docking	Attitude and Rates	Verify swing-a-round onto docking radial (Primary control is visual)
Rendezvous Docking	Attitude	Attitude control during final closure so that the visual offset defines translation required for alignment with the docking radial.

5.4.1 (Continued)

Mission Phase	Display Information	Crew Control Task
Re-entry	Down Range and Cross Range Errors and Roll Attitude and Rate Commands	Re-entry Control and Splash-down Prediction

5.4.2 Initial Design Tradeoffs and Decisions - Various existing display techniques were investigated for suitability to the Gemini program. These techniques were:

(a) Single Axis Gimbal Position Indicators - This device, the type used on the Mercury program, provides individual gimbal position readout and thus inherently can not account for mutual interaction of the three axes of rotation. An exaggerated example of the lack of mutual interaction of the display is shown by maneuvering 180° in yaw, then 180° in pitch. Now the yaw and pitch gimbal position indicators both read 180° but the roll indicators still read 0° when in effect, this maneuver results in an apparent roll of 180°, with 0° yaw and 0° pitch. From this example it can be seen that this device is not capable of displaying the apparent effect of a complex maneuver involving more than one axis of rotation.

During the Mercury program, when man's abilities in space were unknown, attitude changes were made about one axis at a time in a sequential manner. In order to return to the original attitude, the sequence was reversed. The single axis gimbal position indicator was suited to the Mercury program because its single axis readout only served to enhance the "one axis at a time" maneuvers. However, for the Gemini program, where the astronauts were to fly the spacecraft through the

5.4.2 (Continued)

complex maneuvers of rendezvous, this device was essentially useless.

(b) Moving Airplane - This display is currently being utilized in some limited attitude transport type aircraft. It uses a single display element representing the vehicle, which is capable of being positioned in roll, azimuth and elevation with respect to the face of the indicator. Its primary capability is to provide limited pitch and roll information as well as providing director information for instrument approaches, etc. It cannot be mechanized as an all attitude instrument, and hence could not provide the range of information required for the Gemini spacecraft. General useful range is $\pm 30^\circ$ in pitch, $\pm 60^\circ$ in roll, and in yaw as a director of course error indicator.

(c) Two-Axis Attitude Sphere, with Separate Heading Indicator - This method of presenting attitude is the conventional one utilized in early "century-series" fighters and the majority of the operational aircraft flying today. It provides a true readout of attitude, being normally coupled to the vertical gyro's pitch and roll gimbals, and becomes ineffective only near the zenith. It is suitable for most aircraft installations which provide sensors in the form of a vertical gyro and a compass system. Complications arise when it is utilized with a stable platform reference system. Hence, a third axis gimbal control unit is normally employed to keep track of the interaction effects which the azimuth gimbal imparts.

5.4.2 (Continued)

(d) Three-Axis Attitude Sphere - This device utilizes a sphere which is gimbaled to allow 360^{o} degrees of rotation about its pitch, roll and yaw axes. By the use of synchro transmitters and control transformers to relay gimbal position information, the sphere can be positioned directly and identically to the position of the stable member of the platform. The transformation which occurs in reading out the position from the platform is naturally compensated by reading these angles back into the similar gimbal structure of the sphere. Hence the astronaut, in effect, can see the actual position of the spacecraft in relation to the stable member of the platform. The solution is unique, due to its simplicity in achieving what is normally considered a complex transformation of spherical geometry.

Comparative Evaluation of Displays Considered

	ESTIMATED WEIGHT	MUTUAL INTER-ACTION OF AXES	STATUS OF BASIC INDICATOR	INTERFACE WITH PLATFORM	DISPLAY CAPABILITY
Mercury Type	4.0 lb.	No	Production	1:1 Synchro (ambiguous)	Poor. Predetermined sequential attitude changes only.
Moving Airplane	4.0 lb.	No	*Production	Limited Excursion	Unsatisfactory Not all attitude
2-axis sphere	5.0 lb.	Yes 2-axis only	*Production	1:1 Synchro needs 3rd gimbal control	Poor. 2-axis interaction only, ambiguity at zenith.
3-axis sphere	6.0 lb.	Yes	*Production	Direct 1:1 Synchro	Good. All attitude unlimited control.

*Needed to be ruggedized to Gemini environment.

5.4.2 (Continued)

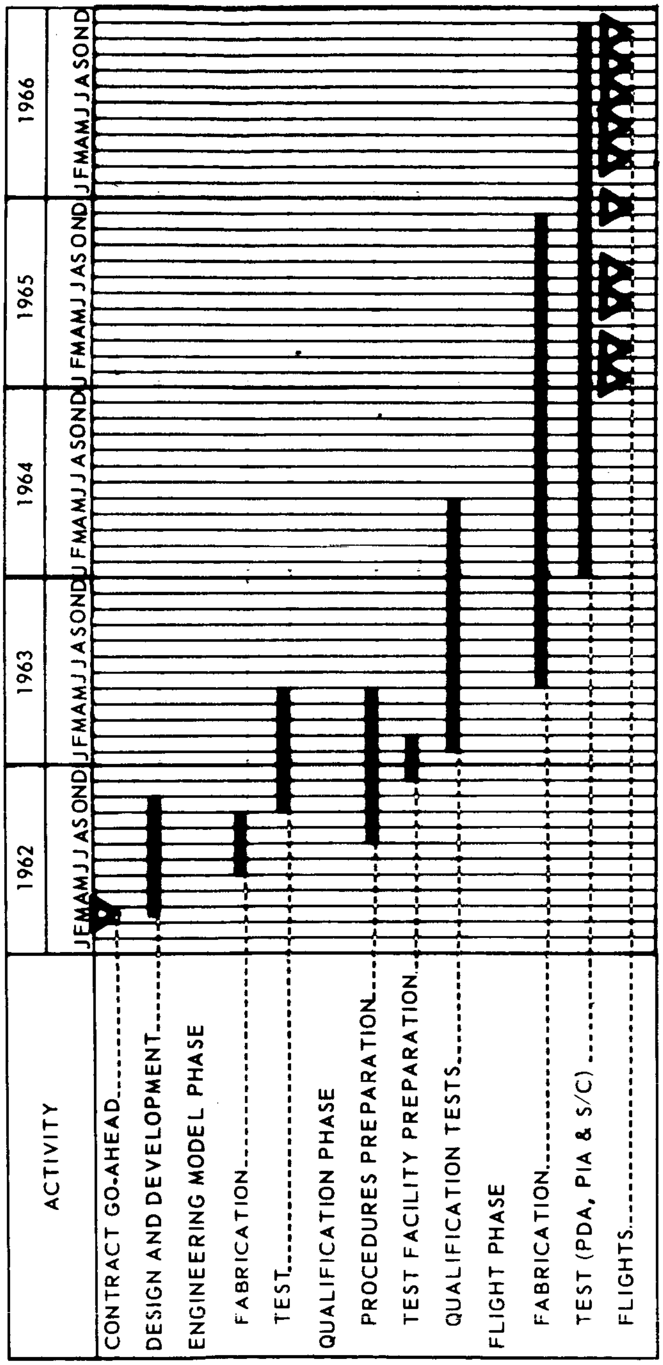
Selection of the Three-Axis Attitude Sphere for Gemini - Previous studies performed on the simulators at McDonnell for the rendezvous and reentry control problems indicated the necessity of an unambiguous all-attitude display for control reference. It was also found that rate and attitude command information is a necessary part of the displays where accurate and efficient control is required. For these reasons, a three-axis attitude sphere with superimposed flight directors was selected for the Gemini program.

5.4.3 Development Problems, Tradeoffs and Design Decisions - The time history of the development of the Attitude Display Group is shown in Figure 5.4-1.

Adaptation of the Attitude Indicator - A three-axis attitude sphere which had been successfully employed in the F-4 Phantom aircraft was adapted to the Gemini program.

In addition to the electrical and mechanical interface adjustments, much effort was expended to make the sphere and flight director needles look natural to the astronaut and react in an unambiguous manner. The lower half of the sphere was painted darker than the top half in order to create an artificial horizon and simulate visual ground contact when the platform was referenced to local vertical. The polarity of the sphere movements was made such that the miniature airplane could be "flown" on the sphere. Thus, attitude changes could be traversed directly, eliminating the need for sequential attitude changes. This same philosophy was incorporated into the movements of the flight director needles. The astronaut responded to any flight director command by "flying" the miniature aircraft into the command.

FIGURE 5.4-1
ATTITUDE DISPLAY GROUP PROGRAM TIME HISTORY



5.4.3 (Continued)

Development of the Flight Director Controller - Since the flight directors were to display commands from the computer, inertial platform, rendezvous radar, and rate gyros, an intermediate controller was added to allow the astronaut to select commands from each of the command sources and to insure command signal compatibility with the flight directors.

5.4.4 Problems and Design Changes During Development and Qualification Testing - Attitude Indicator - Since the indicator had undergone extensive environmental qualification for the Air Force, the only questionable areas were shock and vibration. In order to find specifically where ruggedization would be needed, a development unit was vibration tested. The results revealed four areas which failed to withstand the vibration levels. These areas, their failure modes, and required design changes are tabulated below.

Area or Component	Failure Mode	Design Change
Roll Gimbal	Aluminum gimbal support ears holding the slip-rings were both completely severed.	Change gimbal material to stainless steel.
Flight Director Needles	Excessive needle movement during vibration and permanent needle deformation.	Change needle material to beryllium and add supporting struts.
Rear Cover	Excessive movement during vibration.	Add screws to attach cover to inner support.
Gear Train	The right sphere half displayed large displacements during vibration and was permanently displaced 1/8" after test.	Gears which move the 2 sphere halves were heat treated for improved stiffness and elasticity.

5.4.4 (Continued)

The reworked prototype unit with the above changes then passed its vibration test requirements and its shock test requirements.

Flight Director Controller - During early development, electrical "ground loops" among the various command signal sources necessitated a design change to improve electrical isolation. Also during development, humidity caused electrical variations in the summing networks and led to a design change to environmently seal the area surrounding the summing networks.

Conclusion - Any new design of a controller should give consideration to both electrical and environmental isolation.

5.4.5 Problems and Design Changes Associated with Systems Tests and Flights -

Attitude Indicator - During spacecraft systems testing, it was noted that a manual range switch should be added to the face of the indicator in order to make effective use of the range and accuracy of the command information to be displayed. The range switch added to the indicator allowed an attenuation of the input signals (Hi range) to the flight directors so that a greater range of information could be displayed. When the attenuation was removed (Lo range), smaller ranges could be read to a higher accuracy. This feature was useful in making small attitude corrections or in maintaining a desired attitude.

5.4.5 (Continued)

During rendezvous, a crew task was to furnish manual rendezvous backup capability to the computer. This was done by aligning the spacecraft with the target using the boresight and periodically reading the angle between the local vertical and the spacecraft; this angle was continuously displayed on the attitude sphere and also could be read with the flight directors. This angle and the rate-of-change of this angle were used to compute thrust vectors required to accomplish rendezvous. As the astronauts became familiar with this operation during simulated rendezvous missions, it became highly desirable to (1) increase the number of divisions along the pitch gimbal scale from one mark every 5° to one mark for every degree and (2) add a scale for the pitch and yaw flight director needles. A change was made to the attitude indicator late in the program, incorporating these additional scales and markings, which provided more accurate and repeatable angular readings and therefore better back-up capability during rendezvous.

5.5 Range and Range Rate Indicator

5.5.1 Requirements - The indicator was required to display that information needed by the crew to permit a qualitative evaluation of proper closing rates and ranges during rendezvous.

5.5.2 Initial Design Tradeoffs and Decisions - Originally, the indicator was to accept and display analog range and range rate data from the radar. Ranges of 0 to 300,000 feet and rates of -200 to +500 feet per second were to be displayed. It was intended that a manual control would provide three scale ranges.

However, human factor considerations indicated that manual scale factor control would detract from other critical crew operations during the rendezvous and docking phases of the mission. Consequently, it was decided to display range data in a logarithmic manner and to add a third "vernier" rate indicator to display rates between ±10 feet per second. The vernier range rate requirement was subsequently changed to ±5 feet per second. The range indicator displayed data in three decades: 0 to 3000 feet, 3000 to 30,000 feet, and 30,000 to 300,000 feet. The indicator movement was linear with range gain changes provided by the radar.

Space restrictions limited the panel area that could be given to this instrument. To take advantage of the maximum possible scale length for a given amount of area, a circular display was selected in preference to a vertical display. The range and range rate scales were mounted concentrically and the range rate scale was designed so that its numeric value was approximately equal to the square root of the range radially opposed from it. This scale selection permitted an easy qualitative evaluation of proper closing conditions.

5.5.3 Development Problems, Tradeoffs, and Design Decisions - There were no significant development problems. The time history of the indicator design and development is similar to that of the radar (depicted in Figure 2.4-1).

5.5.4 Problems and Design Changes During Development and Qualification Testing - The range and range rate indicator failed the 12.6 g RMS vibration test. Structural failures occurred in the meter movements due to high amplifications in the case and movement mounts. Structural changes were incorporated and the indicator passed development tests at the 12.6 g RMS level. Subsequently, the qualification test vibration level was reduced and the indicator was qualified at 8.8 g RMS.

5.5.5 Problems and Design Changes Associated With Systems Test and Flights - No problems were associated with the original requirement to provide the crew with quick, qualitative information related to rendezvous. However, as mission success grew, it was attempted to use the range and range rate indicator in a quantitative capacity. The instrument was unsuited for this function because the indicator pointers were so large that they covered the numerals to be read and they seemed to point to the wrong scale. Furthermore, range readings greater than 6000 feet were difficult for the crew to interpret.

No changes were made to the range and range rate indicator in spite of its inability to function in a quantitative capacity. If this were to be a requirement at the beginning of a new program, consideration should be given to decimal display.

6.0 System Design

6.1 System Integration Design Changes and Problems

This section discusses some specific problems and design changes, involving systems integration, which occurred in the Gemini program.

6.1.1 Addition of Ascent Mode

Ascent back-up guidance for the booster radio guidance system became an Inertial Guidance System (IGS) requirement in May 1962. The back-up guidance capability was obtained with relatively few hardware additions since the inherent accuracy of the IGS was sufficient to perform ascent guidance. However, the computer software requirement was increased considerably with addition of the ascent mode. This addition made pre-launch alignment of the IMU necessary, and added a new interface between the digital computer and the booster autopilot.

Three different methods of providing the necessary accuracy in IMU azimuth alignment were considered:

(1) Gyrocompassing similar to that used on Centaur. This method was ruled out because it required a complex interface with the computer and additional computer capacity.

(2) Optical alignment - This method would have required a line of sight from an optical device on the ground to the platform in the spacecraft. This was not feasible because of interference from the launch complex structure and because optical access through the spacecraft structure would have been required.

6.1.1 (Continued)

(3) Platform synchro electromechanical alignment with ascent update - This method was chosen for azimuth alignment. Launch pad alignment utilized IMU ground test equipment and was performed as follows:

(a) The roll and pitch accelerometers were autoleveled to hold the azimuth axis of the platform vertical within 90 arc-seconds.

(b) The platform was aligned in azimuth by torquing the roll gimbal with the roll synchro on the Alignment Reference Unit (part of the ground test equipment). The azimuth angle (roll gimbal angle) was read out through the computer to provide better alignment accuracy.

(c) After lift-off, the computer utilized azimuth velocity updates, transmitted from the ground, to calculate the azimuth misalignment angle, which was employed to correct navigation and azimuth steering data. The azimuth velocity updates were calculated by the ground system on the basis of GE MOD III tracking data. Update data was automatically transmitted to the computer at 100 and 140 seconds after liftoff.

The basic guidance equations were an outgrowth of the radio guidance equations used during the Mercury program. Modifications to the guidance equations were necessary to make them compatible with the Gemini inertial system.

The additional booster interface requirements caused by the addition of ascent back-up guidance are itemized below:

6.1.1 (Continued)

(a) DC analog steering signals (pitch, roll and yaw) from the computer to the booster autopilot.

(b) A switchover discrete (fade-in) from the booster to the computer.

(c) A gain change discrete from the computer to the booster autopilot.

(d) A second stage engine cutoff command (SSECO) from the computer.

(e) A guidance switchover command to the booster (activated by a cabin switch).

(f) Booster guidance malfunction detection system signals to Gemini displays.

6.1.2 <u>IGS Static Power Supply (SPS) Initial Design Considerations</u>

The SPS was designed as a common power supply for the IMU and the computer in order to minimize weight and volume. Responsibility for the design of the SPS was given to the IMU subcontractor because the IMU interface with the power supply was the most complex.

In June 1962, the capability of the SPS was increased to include:

(1) Separate turn-on capability for the IMU and computer to permit better utilization of the IGS. This allowed the computer to be updated in flight without turning on the IMU, thus conserving spacecraft power. The SPS was divided into three sections, the IMU DC section, the computer DC section and the 400 Hz AC section. Each of the above sections had its own pulse width regulator (PWR) which increased the reliability of the power supply.

6.1.2 (Continued)

(2) An output capacity increase of 10 VA in the 400 Hz inverter to eliminate the horizon sensor inverter.

(3) Use of the IGS 400 Hz inverter as the primary 400 Hz source for the ACME system with capability added to permit the astronauts to switch to the ACME 400 Hz inverter as a back-up power source.

6.1.3 SPS Development Problems, Trade-offs, and Design Decisions

Auxiliary Computer Power Unit (ACPU) Incorporation

The IGS specification required that it operate with a power bus interruption of 30 milliseconds. However, the static power supply could not maintain regulation during this interruption and could cause the computer memory to be altered by improper shutdown. An ACPU was incorporated in December 1962 to prevent alterations during bus interruptions and depressions, or uncontrolled power shutdowns.

The ACPU contained a small battery with sensing and timing circuitry which buffered the computer against a voltage depression sensed at the output of the computer section pulse width regulator (PWR) in the SPS. If the depression exceeded 100 milliseconds, the computer was shut off for 30 milliseconds and then turned on automatically. However, this design caused two additional problems.

A power interruption would initialize the computer program to the point of turn on, and without an indication to the astronauts, the data out of the computer would be meaningless. Interruptions during ascent (calculations in real time) could have caused an abort of the mission if IGS guidance

6.1.3 (Continued)

were being used. Interruptions in the computer modes would have resulted in the loss of all previously accumulated data and affected each mode differently. An interruption during reentry would have resulted in the spacecraft missing the target area. Interruptions during rendezvous phase would have had the least effect, because target data could have been reacquired prior to performing an orbit maneuver.

The second problem involved ACPU oscillation which could be started if the bus voltage (with computer off) was slightly above the ACPU turn-off threshold. If the computer was turned on, the bus would be depressed by the computer load causing the ACPU to turn off the computer and then turn it back on after 30 milliseconds. Turning on the computer again would start an off-cycle which would discharge the ACPU battery in approximately 4 minutes.

The design change which corrected these problems consisted of providing the astronauts with a visual indication of computer turn-off via the computer malfunction light when an ACPU shutdown occurred. Positive action by the astronaut was then required to reset the ACPU prior to computer turn-on.

Conclusion -

Equipment susceptible to power interruptions should have an interruption buffering capability built in during the initial design phase. If undervoltage sense circuits with power shutdown sequencing are used, their design should allow for bus voltage variations due to power unloading.

6.1.3 (Continued)

Computer 28 VDC RFI Filtered Power

The computer required 28 VDC filtered power because spacecraft bus power contained transients which could have caused computer malfunctions. The IGS power supply wiring was modified so that computer 28 VDC power was obtained from a point just after the RFI filter. However, this caused an RFI problem since filters were employed in the hot side and the return for all 28 VDC power. The circuits susceptible to RFI problems were those which utilized the 28 VDC filtered power and were returned directly to spacecraft ground. The circuits in question were the booster gain change relay, computer running light, computer malfunction light, and the SSECO and computer malfunction inputs to telemetry. These circuits were driven by discrete output modules in the computer which utilized the 28 VDC filtered power. The DC returns for these modules were tied to a point after the RFI filter, which put the filter in series (through spacecraft ground) with returns for the above circuits. With this configuration, the possibility of a false booster gain change could exist if noise in the RFI filter (noise was rectified by spike suppression diodes on the relays) reached a level which could pull in the gain change relays. In the case of the telemetry inputs, the noise would appear at the input to the telemetry signal conditioners, causing the signal to appear ragged. The computer display lights were not affected by this configuration. A modification was made to rewire these modules to utilize the 27.2 VDC computer power which was returned directly to ground. This modification was incorporated in December 1962.

6.1.3 (Continued)

Conclusion

Careful analysis of each interface should be made to insure adequate protection from EMI problems.

Incorporation of the Precision 400 Hz Power Supply

During testing on the Electronic Systems Test Unit (ESTU) in early 1963, excessive IMU phase shift resolver errors were caused by distortion and voltage variations in the 400 Hz inverter. This was caused by an impedance mismatch; the SPS was designed for a power factor of 0.8 lagging, whereas the actual loads resulted in a PF near unity.

Another source of distortion for the 400 Hz inverter was cross-coupling between SPS sections. The computer dynamic loading reflected low frequency noise through the computer DC section power supplies onto the spacecraft bus. This low frequency noise modulated the IMU DC section and AC outputs.

The above problems were partially compensated for by the following modifications:

(1) A parallel inductor was incorporated in the 400 Hz section to tune the power supply to the load.

(2) The computer DC section regulation was improved by modifying the feedback resistors, which raised the computer voltages approximately 1 volt.

6.1.3 (Continued)

After these modifications were made, problems were encountered with phase shift resolver readouts of gimbal position, during the predelivery acceptance testing of the first production IMU. The problem was evidenced as a readout instability that was caused by distortion and coupling within the power supply which appeared on the 26 V 400 Hz output. The instability on the 400 Hz output appeared as a jitter, or lack of period stability, on the output. Since the phase shift resolvers operated on a single period basis, the readouts showed a wide variation for a stable gimbal position.

This problem was corrected by incorporation of a separate precision 400 Hz inverter which supplied excitation only to the platform phase shift resolvers. Filters were also incorporated in the computer to reduce the effects of distortion and noise. This precision power supply was flown as a separate unit on Spacecraft 2, and a precision 400 Hz phase shift resolver supply was incorporated into SPS during major modifications which are discussed in a later paragraph.

Static Power Supply Leakage Voltage Problems

During Compatibility Test Unit (CTU) testing, transients appeared on the computer interface power lines when prime DC power was applied to the power supply (with the computer section turned off). The hold circuits in the IGS power supply were unable to completely inhibit the computer section pulse width regulator when the power switch was sequenced on; thus a small leakage current passed through the pulse width regulator sections. To prevent this transient from altering the computer memory, a relay was

6.1.3 (Continued)

installed in the power supply to open the 20.7 VDC line which provided the bias for the computer series regulators. This insured that no leakage voltages would get into the computer.

Conclusion

During development testing, all power supply solid state switches should be checked for passage of transients in the off-state when prime power is applied.

Major Redesign of SPS

A major redesign of the SPS occurred in November 1963 to correct the problems noted earlier in this report. The modifications incorporated were as follows:

(1) The pulse width regulator (PWR) frequency was changed from 2400 Hz to 15 KHz, and magnetic amplifiers on the output of PWR's were replaced with transistorized switching circuits. These changes were made to reduce dynamic loading and cross-coupling.

(2) A separate internal precision 400 Hz power supply was added for phase shift resolver excitation.

(3) During testing, several power supplies were damaged due to inadvertent shorting by test personnel. Therefore, overload protection was added to minimize damage due to excessive current drain on any of the SPS sections. The overload protection during ground testing consisted of (manual - reset) turn-off of AC, IMU, and computer PWR's if current drain became excessive. The 400 Hz output also included sense circuitry to shut down the AC section, and to

6.1.3 (Continued)

automatically reset it after an off time of 7 to 10 seconds. This automatic reset capability remained active during flight. The 400 Hz power was utilized by all G & C equipment, and this reset capability permitted fault isolation of defective units. In addition to shutdown capability, each SPS section was fused. In the event of a short in flight, the fuses were designed to open the affected section; the input stage and the other sections of the SPS would remain operable.

Conclusion

All power interfaces should be carefully examined for regulation, distortion, and noise, especially those which are supplying excitation to sensor circuitry.

Provisions should be made for protection of power supplies during ground testing and during flight.

6.1.4 Gimbal Angle-Computer Interface

Initial Trade-off of Techniques

Two methods were considered for digitizing gimbal angles: (1) a servo repeater with a shaft encoder, and (2) a phase shift resolver used in conjunction with a crossover detector (COD) in the computer.

The servo repeater with a shaft encoder method was not chosen because it was heavier, required more volume, and was less reliable and not as accurate as the phase shift resolver.

6.1.4 (Continued)

Phase Shift Resolver

The phase shift resolver determined gimbal angles in the following manner. The computer measured the time elapsed between the reference)400 Hz phase shift resolver excitation) zero phase crossover and that of the 400 Hz phase shifted signal (shift proportional to gimbal angle displacement) from the resolver. The computer-accumulated count between the two crossovers (1 count equal to 2.16 arc minutes) represented the gimbal angle. This technique was simple in its theory of operation but was very susceptible to phase shift, harmonic distortion and noise. These problems will be discussed in later paragraphs.

Packaging Trade-off

The crossover detectors and 400 Hz filters discussed earlier were installed in the computer because volume was not available in the platform. Therefore, line capacitance and noise pickup became error sources. These error sources could be reduced with an integrated assembly in the platform.

Phase Shift Resolver Modification

One of the major contributors to phase shift, harmonic distortion and noise errors was the phase shift resolver itself. The subcontractor improved the resolver design by incorporating the following modifications:

(1) Reduction of harmonics by lowering the 400 Hz excitation from 20 V to 15 V, and use of an ultra-linear magnetic material for laminations.

6.1.4 (Continued)

(2) Use of selected components to minimize mismatches in the resolver tuning networks. These networks were used in parallel with resolver sine and cosine windings.

(3) Incorporation of improved shielding for the resolvers to minimize crosstalk.

ESTU Gimbal Angle Accuracy Testing Problems

During Electronic Systems Test Unit (ESTU) testing in early 1963, two conditions were encountered which affected the accuracy of the phase shift resolvers. During gimbal angle interface testing, harmonic distortion on the 400 Hz excitation caused errors about 2 to 2.5 times greater than that of the specification. This problem was corrected by adding a 400 Hz precision power supply in the IGS power supply and filters in the computer to filter the 400 Hz excitation.

The second source of noise and harmonic distortion which affected gimbal angle accuracy was the accelerometer temperature control amplifiers (TCA). The TCA output transistor wiring was routed through the same slip rings as the resolver and synchro outputs. Therefore, during the "on" condition of the TCA's, harmonic distortion was twice that of the "off" condition. The gimbal angle error observed was 3 to 4 times that of the specification value.

The TCA wiring was rerouted through the opposite set of slip rings to correct this problem.

6.1.4 (Continued)

Subcontractor Gimbal Angle Accuracy Testing Problem

Subcontractor data on component level testing of phase shift resolvers for gimbal angle accuracy did not correspond to data obtained in platform level tests. The platform level tests were less accurate than the component level tests because of the routing of the wiring for the accelerometer dither bias current. The bias current came in one slip ring assembly and was routed out on the opposite set of slip rings. This effectively caused a one turn magnetic field to be set up which distorted the phase shift resolver magnetic field. The bias wiring was routed through the bore of the resolver which put it in close proximity to the resolver. This problem was corrected by rerouting the bias current wiring in and out on the same slip ring assembly to cancel the one turn magnetic effect.

400 Hz Phase Lock Effect on Gimbal Angle Accuracy

During subcontractor testing on production unit No. 2, the phase shift resolvers exhibited unstable characteristics from run to run. This problem was caused by phase shift between the main 400 Hz power supply and the 400 Hz precision supply. Both supplies were synchronized to the same 2.4 KHz reference, but the supplies could sync in on opposite phases of the 2.4 KHz source. Since the synchro (excited by high power 400 Hz) and phase shift resolver (excited by precision 400 Hz) are "pancake" components, interaction caused by the above phase shift (increments of 30 degrees) caused the gimbal angle instability. The desired repeatability was obtained with incorporation of a phase lock circuit which insured that both 400 Hz supplies would sync to the same phase of the 2.4 KHz reference.

6.1.4 (Continued)

Conclusion

Careful examination should be made of all wiring and packaging techniques used in high density circuits to eliminate electro-magnetic coupling and interaction between these circuits. Precision outputs involving two or more voltages of the same frequency, or incremental ratios of the frequencies, may require more than standard precautions to prevent distortion and heterodyning.

6.1.5 Accelerometer Digital Interface

Accelerometer Digitizing Method Trade-off

Two methods were considered to digitize acceleration data for the computer: (1) an analog rebalanced accelerometer with an analog to digital (A/D) converter, and (2) a pulse rebalanced accelerometer.

Three different types of A/D conversion were considered for the analog rebalanced accelerometer:

(1) Ladder Network A/D Conversion - This network would have operated off the accelerometer rebalance loop and would have used an 11-bit ladder network to digitize the acceleration analog rebalance signal. A three-channel multiplexer would have been used in conjunction with the ladder network. This method of digitizing was not used because:

(a) This circuit had not undergone temperature or reliability analysis by the subcontractor.

(b) The packaging volume was large due to the very large precision resistors required.

6.1.5 (Continued)

(c) It was questionable that this conversion technique was accurate enough to meet the desired accuracy of 1 part in 3000.

(2) Integrator and Discriminator A/D Conversion - In this device the accelerometer rebalance DC voltage would have been first checked for phase, then integrated until a predetermined threshold voltage was exceeded. This generated a phase sensitive pulse of known duration which was used as acceleration data by the computer and also to reset the integrator. The pulse frequency was proportional to the acceleration. At a preset time interval the integrator would be reset if threshold was not exceeded. This was a new device, and since no test data was available to evaluate its design, it was decided not to use this approach.

(3) Voltage Comparison Network A/D Conversion - This method compared a precision voltage reference applied across a precision servo driven potentiometer to the accelerometer analog output. A mechanical digitizer was to provide the digital input to the computer. This method was considered impractical because it required a precision voltage source with 0.005% regulation.

Pulse Rebalance Accelerometer Method

The pulse rebalance accelerometer ultimately used was a modified Centaur approach. This method used a series of accurately timed current pulses (± current) at a 3600 Hz rate to torque the accelerometer to null. Acceleration data was then taken directly from the accelerometer torquing loops where the net of ± pulses monitored by the computer represented the total acceleration.

6.1.5 (Continued)

This technique was chosen to digitize the acceleration data because:

(1) Its accuracy was best and it had no A/D conversion error.

(2) It required fewer parts, and therefore was more reliable, weighed less, and needed less volume.

(3) It required the least power.

Accelerometer Digitizing Problem

The only major problem was the flight anomaly during the ascent of Spacecraft 2. The problem encountered was the "pulse lockup" discussed in Paragraph 2.2.5.

6.1.6 Grounding Philosophy

The grounding philosophy for the Gemini SG&C systems was to have a single point ground for the spacecraft. This philosophy was used to eliminate ground loops for DC and lower frequency AC circuits and to establish adequate grounds for those frequencies. However, suitable regard had not been expended on the HF (EMI) grounding problem; a wire which appears as a short at low frequencies acts like a high impedance at the frequencies used in the computer. The intent to keep the computer signal ground at spacecraft ground potential was defeated because the ground wire was too long. To alleviate this problem a shorter computer ground strap was incorporated.

Conclusion

A workable SG&C grounding philosophy, cognizant of the hardware realities of the equipment used, must be developed early in a program.

6.2 Switching Device Design Considerations

Both solid state and electromagnetic switching devices were used in the Mercury and Gemini SG&C systems. The majority were armature type relays (electromagnetic) as opposed to semi-conductor switches (solid state). The application, including performance, environmental, and operational requirements, governed which type was selected. Most test failures were attributed to misapplication rather than to deficiencies in the switch gear or relays.

Relays were employed in applications that required multiple outputs, double throw action, power off latched control, power switching, noncritical response timing, or design flexibility. Semi-conductor switches were used when the application necessitated minimum size and weight, where response timing was critical, or in low power switching circuits.

Although armature type relays have a reputation for poor reliability, no flight anomalies were attributed to relays during the Mercury and Gemini programs. This phenomenal success was achieved by accepting for use only relays that had been thoroughly evaluated, destructively analyzed, and subjected to special tests. The circuit designer was given a family of relays from which to select, each of which had been previously proven acceptable for the Mercury and Gemini programs. Only approved relays could be used in circuit design. If a new relay was required, it had to first pass the stringent preselection evaluation.

6.2 (Continued)

The family of approved relays was created by establishing performance specifications in excess of those of existing MIL specifications. In particular, both maximum and minimum values were established for each relay performance parameter. Relay candidates were subjected to comprehensive tests to validate conformance with the specifications. The tests were conducted under all applicable environments and performance parameters were measured during the test, in addition to the normal procedure of measurements before and after test. Emphasis was placed on making sure that the procurement specification contained clear requirements to assure quality control and predelivery screening. The following list illustrates the type of performance parameters specified.

(1) Maximum and minimum input bus voltage.

(2) Maximum and minimum pickup voltage.

(3) Maximum and minimum dropout voltage.

(4) Reflected voltage transient.

(5) Maximum release time.

(6) Minimum and maximum operate time.

(7) Contact bounce time.

(8) Contact crossover.

(9) Maximum coil current at specified voltage and temperature.

(10) Contact arrangement.

(11) Contact rating for resistive, motor, inductive, and lamp loads for a minimum of 50,000 cycles.

(12) Minimum current rating.

(13) Time delay characteristics.

(14) Latch and reset requirements.

6.2 (Continued)

Mortality-time characteristics were obtained from life test data. To eliminate relays that fail early in their life cycles, each relay was burned-in as part of the acceptance testing. Critical parameters were frequently monitored during the burn-in cycling. Each relay installation was vibrated as part of the acceptance test. The vibration test was conducted under the critical temperature environments and each relay of the installation was operated during the test. Each relay visibly damaged during the build-test cycle was replaced, regardless of how slightly damaged. To avoid the effects of inductive transients on other susceptible electronic equipment, a separate, independent battery was used to power all relays.

Two major problems were encountered in the design of Mercury and Gemini relay systems. The first was caused by the inductive transient associated with the collapsing field when the relay was deenergized. The Mercury program used brute strength bus filtering for relay-associated transients. The Gemini solution was the addition of diode suppression to each relay. The effects on the response time caused by the added diodes were evaluated in each application. In the design of other equipment, filtering was added to reduce susceptibility to relay transients.

The second problem was failures during installation vibration tests. Vibration transmissability in relay panels caused relay failures, which were corrected in most cases by the addition of panel stiffeners. At times, circuit redesign was necessary to avoid actuation of the critical relays during the high vibration portion of the flight.

6.2 (Continued)

One major problem was found in the solid state time delay switches used. During vibration, the liquid tantalum capacitor used in these devices tended to become slightly charged, thereby causing an out-of-tolerance time-out of the device when subsequently energized. Two approaches were used to eliminate the problem: (1) For time delays less than 5 seconds, solid tantalum capacitors were substituted for the liquid tantalum capacitors. (2) The tolerances of the solid tantalum capacitors were too large to permit their use for time delays greater than 5 seconds. Instead a small relay was added to the device to short out the liquid tantalum capacitors during the high vibration portion of the flight.

Conclusion

Stringent quality control, screening, and tests are necessary to obtain reliable relays for spacecraft designs. Extreme care is necessary to insure that the relay fits the application. More consideration should be given to using semi-conductor switches in lieu of armature type relays as the state-of-art of solid state devices progresses.

7.0 Conclusions and Program Philosophy - Wherever appropriate in this report, we have presented conclusions reached as a result of our experience with specific space vehicle SG&C systems. Some of the major conclusions, pertaining to the design of space vehicle SG&C systems, are restated in Section 7.1. Section 7.2 describes the general program philosophies, including the impact of Mercury experience on Gemini design, and conclusions that apply to spacecraft designs, not to the SG&C systems alone.

Some of the statements given may seem intuitively obvious; however, even with the proven success of the Mercury and Gemini programs, occasionally we found these factors had been overlooked in the early design phase. In these cases, they usually necessitated redesign later in the program.

7.1 The following conclusions are presented elsewhere in this report where the particular SG&C problem is discussed. They are reinterated here because they are applicable to the design of most space vehicle SG&C systems.

(1) When moving parts are used in a system, they should be isolated from the space environment to avoid the effects of large variations in temperature and pressure.

(2) Particular heed should be paid to outgassing of materials in a low pressure environment. The effects of outgassing may appear as (1) a deterioration of exposed lubricants; (2) deposition of contaminants on senstive parts; (3) arcing of vacuum-exposed RF components; (4) deposition of an insulating layer on sliding electrical contacts.

(3) Circuitry in RF equipment which generates or carries high RF power should be pressurized where feasible. The pressure seal design employed for these components must consider the long duration of space missions.

7.1 (Continued)

(4) Positive control of the system thermal environment, by means of an active cooling system, is desirable. With the use of such a system on Gemini, it was possible to avoid the cost of designing equipment to operate over the large temperatures excursions of outer space.

(5) Exploratory environmental tests should be performed early in the system development phase to preclude costly redesign later. Care should be exercised to assure that the specified environments are as realistic as possible and that the interaction associated with the application of multiple environments simultaneously is considered.

(6) All critical input and output signals should be electrically isolated to preclude damage or overstress to components from ancillary loads.

(7) Overrating of components must be employed judiciously to avoid additional problems. For example, a relay overrated to accommodate the high current carried but actually switched under a no-load condition, resulted in contaminants building up on the contacts.

(8) In manned operation, astronaut confidence in GG&C equipment must be maintained by providing a positive visual indication of malfunctions and of critical equipment operation. Operational procedures should be developed and practiced to permit mission completion in the event of malfunctions, and to preclude astronaut-initiated malfunctions. (In general, equipment should be designed to avoid malfunctions caused by erroneous operation but sometimes it is not economically feasible to do so.)

(9) A thorough timing analysis should be conducted on digital interfaces early in the program. The analysis is especially important where delays are introduced by long wire lengths, complex interfacing circuitry, etc.

7.1 (Continued)

(10) Initial system design should take into account the effects of electromagnetic interference (EMI). EMI control plans should be established early in a program by the major contractor and his subcontractors in order to avoid costly later redesign and the addition of filters. The plans should specify such requirements as:

(a) Suppression of potential EMI generators.

(b) Separation of wiring based on signal categorization.

(c) Designing circuits so that they will not be overly susceptible to EMI.

(d) Isolation of power for low signal circuits from the major power source.

(e) Judicious placement of components and use of shielding where necessary to avoid mutual interference, especially in high density packaged equipment.

(f) Adherence to a good grounding philosophy.

7.2 Program Philosophy and Conclusions

7.2.1 Spacecraft Design - The Gemini project organization and procedures were established so as to derive the maximum benefit from Mercury experience. Thus, Mercury experience provided the base from which the Gemini design philosophy evolved. From the first, the Gemini spacecraft was designed:

(1) to be versatile and capable of quick adaptation to different missions;

(2) to be quickly and easily maintained under prelaunch conditions; and

(3) to possess high operational reliability.

Typical elements of the design included:

(a) Provision for equipment accessibility so that last minute changes could be made if necessary.

(b) Provision of sufficient test points to speed checkout and to eliminate the requirement to disconnect equipment to isolate malfunctions.

(c) Emphasis on redundant components and systems, and on multiple alternatives in the case of failures which involved crew safety. Although a major objective was to maximize the probability of mission success, crew safety was of paramount importance to the extent that all design decisions were judged either to affect or not affect crew safety. A special "watch-dog" committee ruled on any changes affecting crew safety.

(d) Provision for the late installation and checkout of pyrotechnic devices used to accomplish critical functions.

(e) Recognition that astronauts could function effectively in space and that their full participation in the mission would increase the spacecraft's flexibility and reliability.

(f) Inclusion of secondary power supplies with the using equipment, rather than using one secondary supply for all spacecraft functions.

7.2.2 Design Implementation and Control - Our experience with spacecraft design indicates that problems can be classified as resulting from deficiencies in component control, design control, and interface control. These problem areas are discussed in the following paragraphs.

Component Control - Component control must be exercised for both singular components (resistors, capacitors, transistors) and the more complex components, such as cavity oscillators, DC motors, printed circuit boards, etc. Component problems are the most prevalent and the most difficult to detect early.

Preparation and use of a Qualified Parts List (QPL) is one solution to this problem. However, preparation, administration, and continual updating of a QPL is a very large task. The QPL removes some design prerogative from the subcontractor, and if not kept up-to-date, can result in an obsolete design.

Another component control technique is lot control with sample testing of each lot. Lot control usually is advisable even if a QPL is used. Lot control requires that components be stored by lot and records kept to show which lots are used in which end items. Satisfactory lot control also entails obtaining a guarantee from the component supplier that all members of the lot are identical. In general, component suppliers make "improvement" changes without changing part numbers or notifying their customers. The contractor can only insure that all components are identical by buying from a single lot. But the component might then spend a long time in storage prior to use, and its reliability may be jeopardized as a result.

7.2.2 (Continued)

An alternate effective approach is to subject each component or component assembly to an acceptance test and to perform destructive analyses on sample "in-service" specimens to verify the adequacy of the quality control practices being employed.

Design Control - Most design changes are required: (1) to improve performance or reliability; (2) to eliminate overstressing of components; and (3) to meet environmental requirements. These changes can be avoided with good communications with the equipment manufacturer and if attention is given to the detail aspects. To minimize changes, we suggest the following control measures:

(a) Early in the program a meeting should be held to insure that the equipment manufacturer fully understands the equipment-defining specification.

(b) The prime contractor should assign an engineer to monitor the detailed design effort. Consideration should be given to permitting the engineer to attend the subcontractor's internal design reviews. Further, all subcontractors should be requested to report their progress during the design stage.

(c) The prime contractor should keep an engineering representative at the qualification facility throughout the qualification test program.

(d) Each equipment supplier should supply and maintain a detail electrical overstress analysis of his design.

(e) Tests performed on engineering models should be extensive, well planned, and documented to insure that the production prototypes can be qualified with a minimum of redesign.

7.2.2 (Continued)

Good system engineering, which is the composite result of the proper application of the experience, judgment, and ability of individuals, can assure design control. No insurmountable problems were encountered on our SG&C systems, and we feel it was because of our close supervision and control of our subcontractors during the design phase. We found it beneficial to issue a report defining the spacecraft system for release to all major subcontractors.

Interface Control - Control of interfaces, including the resolution of interface problems, should be the responsibility of a systems integration group. Our experience indicates that effective interface control is possible if:

(a) Detailed interface specifications are prepared and released as early in the program as possible.

(b) A comprehensive electromagnetic compatibility control plan is implemented.

(c) All interfaces, including those with ground equipment or associate contractors, are considered together with the spacecraft equipment interfaces.

(d) Development test reports include detailed characteristics of each input/output parameter under the various test conditions.

(e) Input/output breadboards are used for early compatibility tests.

(f) A realistic compatibility test unit "spacecraft" is available for test early in the program.

7.2.3 Spacecraft Testing

Test Philosophy - Test design for the Gemini spacecraft began early in the design stage to insure that system operational status would and could be monitored. A compatibility test unit "spacecraft" was manufactured to provide an early opportunity for integrating test equipment and training test personnel. The equipment pre-delivery acceptance and pre-installation acceptance tests were conducted utilizing realistic input and output circuitry so that the results of the tests could be correlated with integrated spacecraft test results.

The integrated spacecraft tests were performed at the St. Louis facility and key tests were repeated at the launch site. Verification and monitoring of critical functions were maintained until launch. The following testing ground rules were implemented:

(a) Uniform test equipment was used for all tests.

(b) Uniform test procedures were used for unit pre-delivery and pre-installation acceptance tests.

(c) Uniform test procedures were used for integrated spacecraft tests at St. Louis and at the launch site.

Operational Philosophy - Early in the Gemini program, key personnel from Mercury launch operations were assigned to McDonnell's St. Louis facility and teamed with the engineers responsible for the development of the major Gemini systems. This insured that the Gemini system design would reflect the practical experience gained from Mercury flight operations, and enabled launch personnel to become familiar with the spacecraft. Additionally, direct lines of communication were established between responsible individuals at St. Louis and Cape Kennedy.

7.2.3 (Continued)

During the latter stages of production and testing, a spacecraft manager was assigned to each vehicle. This engineer was responsible for tests and modifications to his spacecraft prior to launch and evaluation of its launch readiness. He also played a major role in post-flight analysis.

8.0 References

The following documents provide information beyond the scope of this report on the operation and requirements of the equipment discussed herein.

A. NASA Documents

FCH-1 - Project Mercury NASA/MSC Flight Controller Handbook - Spacecraft 20, Revision A, 15 March 1963

SP-45 - Mercury Project Summary, Including Results of the Fourth Manned Orbital Flight, October 1963

- Gemini 12 Operations Handbook - Prepared by Flight Control Division, Manned Spacecraft Center, NASA, Houston, Texas

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A900-12 - Gemini Spacecraft No. 12 Performance - Configuration Specification - Revised 30 November 1966

F169 - Gemini Final Summary Report, 20 February 1967

C. McDonnell Gemini Specification Control Drawings

52-87700 - Attitude Control and Maneuver Electronics - Revision E, 10 December 1965

52-87701 - Rendezvous Radar - Revision E, 10 September 1965

52-87702 - Attitude Display Group - Revision C, 9 June 1965

52-87703 - Horizon Sensor System - Revision E, 19 November 1965

52-87710 - Inertial Guidance System (U) - Revision D, 17 December 1965 (Confidential)

8.0 (Continued)

52-87712 - Incremental Velocity Indicator - Revision D, 12 November 1965

52-87713 - Manual Data Insertion Unit - Revision C, 3 September 1965

52-87715 - Rendezvous Radar Transponder - Revision B, 16 February 1965

52-87717 - Inertial Measurement Unit (U) - Revision E, 23 November 1965 (Confidential)

52-87723 - Auxiliary Computer Power Unit - Revision D, 2 October 1965

52-87727 - Auxiliary Tape Memory - Revision A, 17 December 1965