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Electronics Research Program

Satellite Attitude Control System Experiments

SEMIANNUAL TECHNICAL REPORT

(1 July - 31 December 1962)

21 MARCH 1963

Prepared by A. J. SCHIEWE and J. G. ZAREMBA

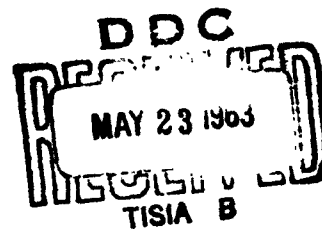
Control Systems Department

Systems Research and Planning Division

Prepared for COMMANDER SPACE SYSTEMS DIVISION

UNITED STATES AIR FORCE

Inglewood, California



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**Report No.
TDR-169(3250-32)TR-1**

**ELECTRONICS RESEARCH PROGRAM
Satellite Attitude Control System Experiments**

**SEMIANNUAL TECHNICAL REPORT
(1 July - 31 December 1962)**

**Prepared by
A. J. Schiewe and J. G. Zaremba
Control Systems Department
System Research and Planning Division**

**AEROSPACE CORPORATION
El Segundo, California**

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Satellite Attitude Control System Experiments

SEMIANNUAL TECHNICAL REPORT

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Prepared

A. J. Schiwe

A. J. Schiwe, Associate Head
Control Systems Department SRPD

J. G. Zarembo

J. G. Zarembo
Control Systems Department SRPD

Approved

C. W. Sarture

C. W. Sarture, Assistant Director
Sensing & Information Sciences SRPD

M. T. Weiss

M. T. Weiss, Director
Electronics Research Laboratory
Laboratories Division

AEROSPACE CORPORATION

El Segundo, California

ABSTRACT

A satellite attitude control system utilizing a digitally controlled gas jet scheme is presently being investigated. Progress during the period 1 July 1962 through 30 December 1962 has been achieved primarily in the completion of the design and development of special purpose test equipment and in hardware implementation of the attitude control system.

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I. INTRODUCTION

The purpose of these experiments is to study the feasibility of advanced satellite attitude control system concepts. One such system is the digitally controlled gas jet system described in Reference 1. This system is presently being investigated; the over-all objectives of the experiment continue to be as outlined in Reference 1. A major part of the laboratory effort to date has been directed toward the development of special purpose equipment necessary in performing the experiment and toward hardware implementation of the first attitude control experiment. This development is essentially complete so that now emphasis may be directed toward experiments on and testing of the attitude control systems themselves.

II. SUMMARY

The design and development of the special purpose devices required to perform the attitude control experiments are now nearly complete. The development of the earth horizon simulator has been completed. It has been calibrated and is ready for use. The space vehicle simulator (wire table) has been moved to a new location to provide more room for both the satellite attitude control system experiments and the inertial guidance system experiments. Nearly all the hardware necessary for performance of the first attitude control experiments has been mounted on the wire table (gas jet system, control logic electronics, horizon sensing system, rate gyro for acquisition, etc.) and checked out in partial simulations. All subsystems are operating properly except for the horizon scanning system. The difficulty appears to be in the sensors themselves (General Electric Model II Horizon Sensor). A replacement set of scanners has been requested.

Initiation of the main experimental portion of this program awaits acquisition of a working set of horizon scanners and construction of an environmental enclosure for the experiments. When these items become available, then experimental verification of the attitude control system described theoretically in Reference 1 can be fully realized and other experiments now being planned can follow in rapid succession.

Reports which have been written during this period are presented as References 2 through 7.

III. SPECIAL PURPOSE EQUIPMENT DESIGN AND DEVELOPMENT

A. EARTH-HORIZON SIMULATOR AND TEMPERATURE CONTROLLED CIRCUITS

The earth-horizon simulator design and development are now at the stage of experimental evaluation and the necessary calibrations. Thus far, satisfactory correlations have been observed between the experimentally obtained data and both the simulator's theoretical heat transfer model and predicted temperature controller capability. Calibrations involving the desired temperature settings referenced to thermocouple sensors embedded .07 inch from the simulator's surface were also completed.

Further experimental evaluation work will commence with the acquisition of a reflection-and-air-currents free laboratory area, which is to house both the earth-horizon and the space vehicle simulator. Such an area is required because it has been determined that due to the long data gathering periods involved with the low limit cycle rates of interest (of the order of .002 deg/sec), the experiments cannot be conducted in a room wherein there is other activity. When the new area has been acquired, it is planned to experimentally determine the surface temperature distribution by thermographic and radiometric techniques. The obtained surface temperatures will be correlated with the already established calibrations.

Figure 1 illustrates the physical configuration of the earth-horizon simulator system. The hemispherical simulator's shell is divided into six thermal zones of equal areas separated from each other by a 1/8-inch wide separation region. Each thermal zone has one temperature control thermistor and three instrumentation thermocouples. The thermistor and one thermocouple are located at the geometric centroid of each zone while the two remaining thermocouples are symmetrically located ± 45 degrees from the centroid.

The heating of each thermal zone is accomplished by an electrically conductive resin sprayed onto the interior surface of each zone. The heater itself was fabricated by Electrofilm, Inc., North Hollywood, California, and consists of a .007-inch thick base and top insulation coatings (Spec. 1097T) sandwiching the .005-inch thick conductor coating (Spec. 1701A).

The composition and thickness of the conductor per thermal zone are such that each zone will dissipate 4 watts/in² of area or approximately 1400 watts. The power input to each zone is provided by a 400-cps laboratory power supply via a variable transformer (0-135 volts). The exterior surface of the sphere is coated with a flat-black resin (Product Techniques, Inc., Los Angeles, Spec. PT401) to provide high emissivity factor.

Figure 2 illustrates the temperature control and the power and the instrumentation circuits.

The temperature control circuit consists of a thermistor temperature detector in a bridge configuration with the temperature command potentiometer. The resulting bridge output signal is amplified (gain 20,000), demodulated, and then introduced to the relay Schmitt trigger circuits. An error signal change of .1 mv (bridge output) will cause the relay to close, thus controlling the power flow to the heater load. The sensitivity of the control system ranges from .15°F in the temperature range from ambient to 150°F upward to 1.5°F in the range from 180°F to 300°F.

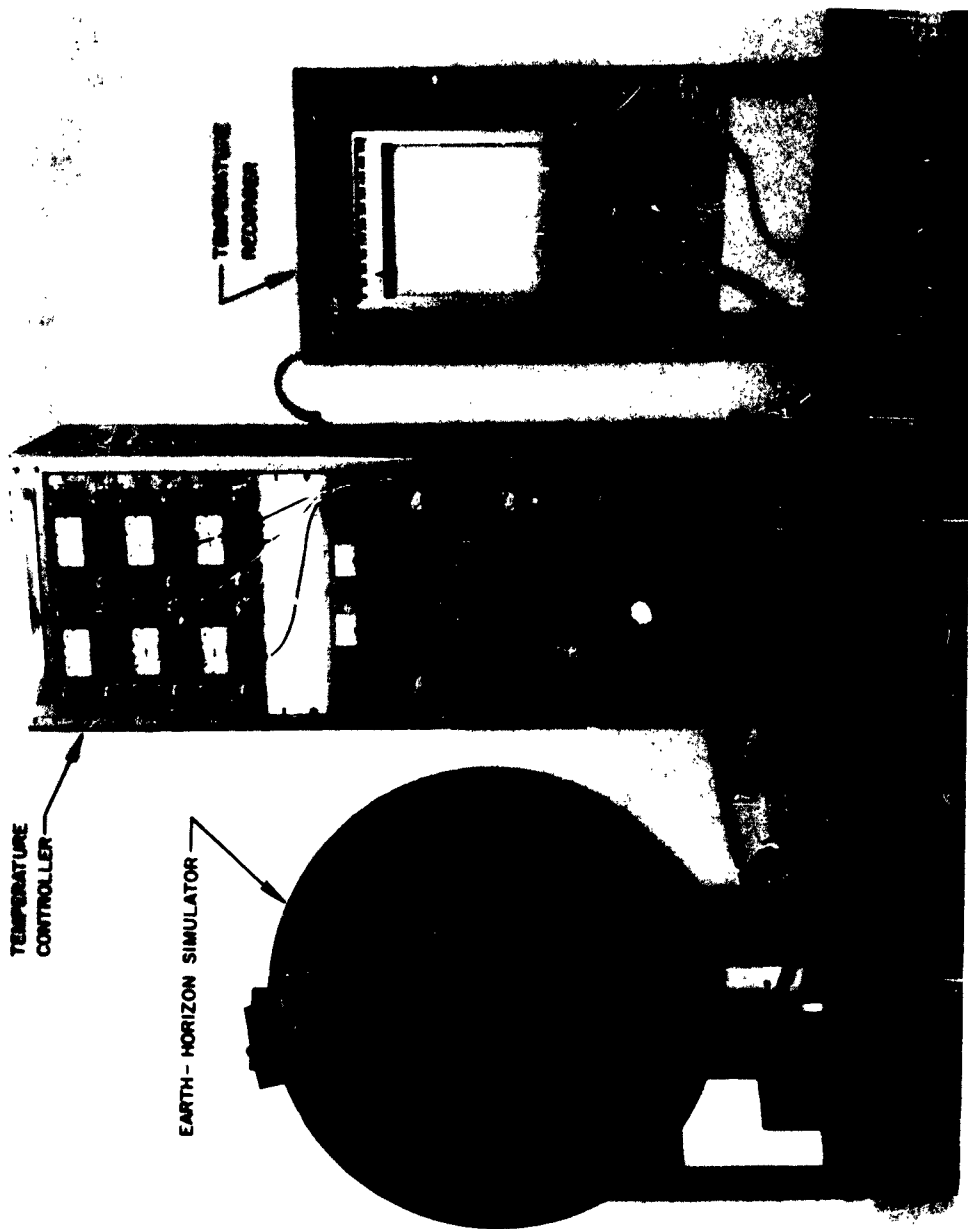


Figure 1. Earth Horizon Simulation System

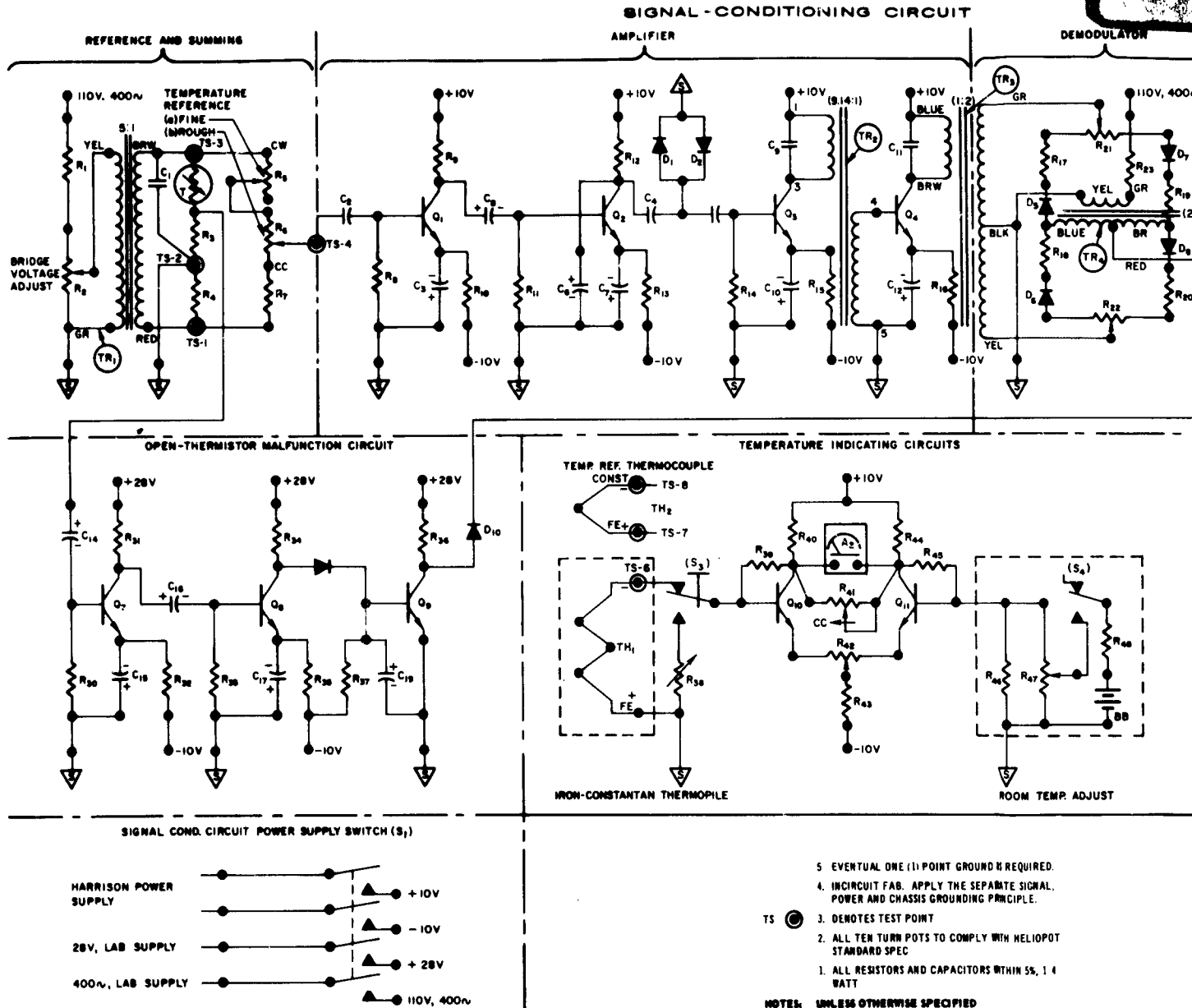
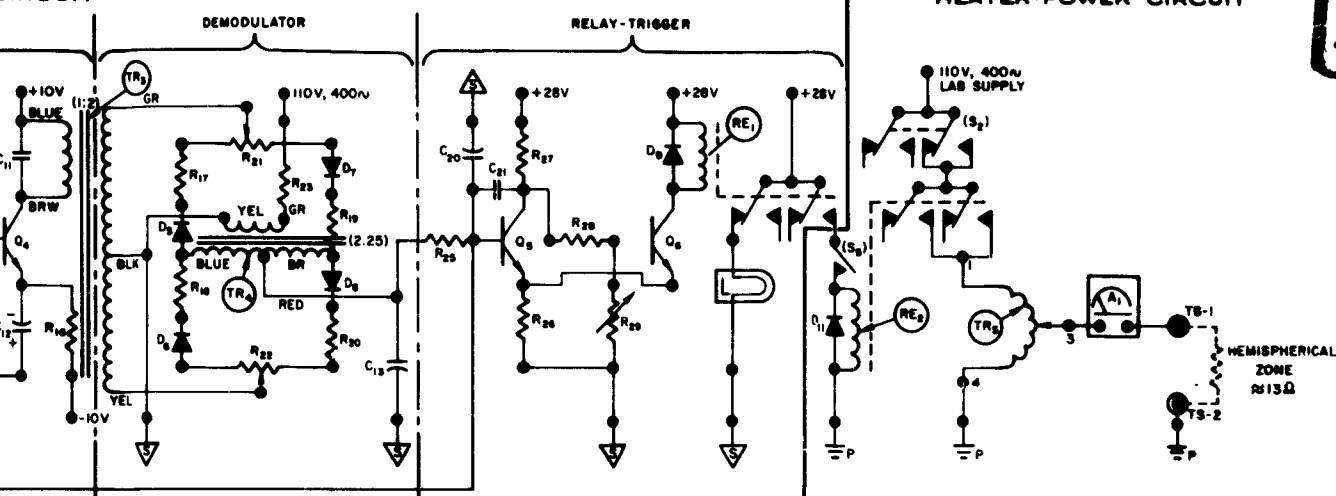
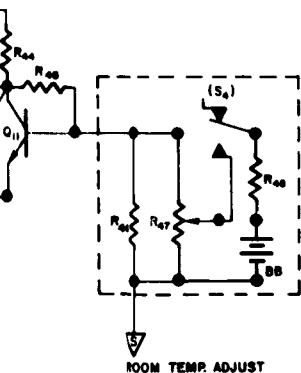


Figure 2. Temperature Control Circuit for One Zone of a Hemisp

CIRCUIT



S CIRCUITS



NE (1) POINT GROUND REQUIRED.
 AD. APPLY THE SEPARATE SIGNAL,
 CHASSIS GROUNDING PRINCIPLE.
 ST POINT
 IN POTS TO COMPLY WITH HELIOPOT
 PEC.
 RS AND CAPACITORS WITHIN 5%, 1/4
 UNLESS SPECIFIED

R-17	1.0KΩ	R-48	12KΩ	Q-11	2N 336 TRANS	T	THERMISTOR, VECO A-60
R-16	2.0KΩ	R-47	500Ω, 10 TURN	Q-10	2N 336 TRANS	TH ₂	THERMOCOUPLE (1)
R-15	2.7KΩ	R-46	470Ω	Q-9	2N 697 TRANS	TH ₁	THERMOCOUPLE PILE (2)
P-14	36KΩ	R-45	100KΩ	Q-8	2N 336 TRANS	BB	HG BATTERY RM-425RT2
R-13	4.3KΩ	R-44	4.0KΩ	Q-7	2N 336 TRANS	A ₂	AMMETER 0-150 μA, BACKMAN, 074-4265-0
R-12	5.1KΩ	R-43	4.0KΩ	Q-6	2N 697 TRANS	A ₁	AMMETER, 0-10 AMPS, STERLING 75-889
R-11	27KΩ	R-42	10Ω TRIM	Q-5	2N 697 TRANS	S ₅	CUTLER-HAMMER (SPST), 7500 K-13
R-10	7.5KΩ	R-41	10KΩ (TRIM)	Q-4	2N 335 TRANS	S ₄	MICRO SWITCH (SPST)
R-9	7.5KΩ	R-40	4.0KΩ	Q-3	2N 335 TRANS	S ₃	SWITCH, PUSH BUTTON (SPDT)
R-8	100KΩ	R-39	100KΩ	Q-2	2N 335 TRANS	S ₂	CUTLER-HAMMER 8821-K6 (2PDT) POWER SWITCH
R-7	18KΩ, 1%	R-38	500Ω, 10 TURN	Q-1	2N 335 TRANS	S ₁	CUTLER HAMMER (4PST), 7660-K13
R-6	4KΩ, 10 TURN	R-37	20KΩ			L	24, V. dc. INDICATOR LAMP
R-5	100Ω, 10 TURN	R-36	10KΩ	D-11	1N 91 DIODE		
R-4	20KΩ, 1%	R-35	7.5KΩ	D-10	1N 191 DIODE	RE-2	POTTER-BROWNFIELD, RELAY, KRP110G, 28 V DC, 500Ω
R-3	51.1Ω, 1%	R-34	18KΩ	D-9	1N 91 DIODE	RE-1	HIG, RELAY, HG2BC-1C-113, 15VDC, 1000Ω
R-2	10KΩ, 10 TURN	R-33	36KΩ	D-8	1N 191 DIODE		
R-1	10KΩ, 1/2 W	R-32	3.0KΩ	D-7	1N 191 DIODE		
		R-31	9.1KΩ	D-6	1N 191 DIODE		
		R-30	20KΩ	D-5	1N 191 DIODE		
		R-29	2.0KΩ (1 TURN)	D-4	1N 191 DIODE		
		R-28	3.9KΩ			TR-5	G-R, VARIAC, 0-10, 000~
		R-27	3.9KΩ, 1/2 W			TR-4	TRIAD, TRANSFORMER, 000~TY-34X
		R-26	240Ω			TR-3	TRIAD, TRANSFORMER, 000~TY-37X
		R-25	20KΩ			TR-2	TRIAD, TRANSFORMER, 000~SP-46
		R-24	30KΩ			TR-1	TRIAD, TRANSFORMER, 000~TY-34X
		R-23	11KΩ, 1/2 W				
		R-22	1.0KΩ, 1 TURN				
		R-21	1.0KΩ, 1 TURN				
		R-20	1.0KΩ				
		R-19	1.0KΩ				
		R-18	1.0KΩ				
C-21	1.0 μf						
C-20	0.20 μf						
C-19	0.20 μf						
C-18	1.00 μf						
C-17	0.20 μf						
C-16	1.00 μf						
C-15	0.20 μf						
C-14	1.00 μf						
C-13	3.30 μf						
C-12	3.90 μf						
C-11	0.30 μf						
C-10	3.90 μf						
C-9	0.30 μf						
C-8	1.00 μf						
C-7	1.00 μf						
C-6	0.05 μf						
C-5	2.20 μf						
C-4	2.20 μf						
C-3	1.00 μf						
C-2	1.00 μf						
C-1	0.0047 μf						

LIST OF MATERIALS

at for One Zone of a Hemispherical Earth Horizon "IR" Simulator

The signal conditioning circuits also incorporate an "open" thermistor malfunction circuit. This circuit is mechanized such that a loss of signal from the thermistor due to an "open" type malfunction causes the input signal to transistor Q_7 (Fig. 2) to approach zero. This in turn causes transistor Q_9 and Q_5 to change states from cut-off to saturation, thus disconnecting power flow to the heating element. A thermistor "short" type malfunction is inherently protected by the thermistor bridge configuration, which in this particular case causes a phase shift in the bridge output signal, thus forcing transistor Q_5 and Q_6 into their saturation and cut-off states, respectively.

The instrumentation of each thermal zone consists of a centroidally located temperature reference thermocouple and two other thermocouples connected as a thermopile.

The function of the reference thermocouple is to provide the capability of correlating the hemisphere's surface temperature with the command temperature potentiometer (R_6) settings. The thermopile temperature indicating circuit, appropriately scaled (R_{41}), visually displays (meter A_2) the surface temperature of each thermal zone, thus providing a continuous monitoring of the earth horizon simulator.

Further details involving the horizon simulator and its control circuitry will be given in a report presently being prepared.

B. HORIZON SENSING SYSTEM HOLDING FIXTURE

Procurement (loan) of General Electric Model II Horizon Sensors necessitated design and fabrication of a fixture to mount the sensors on the space vehicle simulator and to calibrate their signal output. This fixture (Fig. 3, 4, and 5) was designed to provide sufficient flexibility to allow its use with a variety of horizon sensing configurations. Some of the design features of this fixture are:

- 1) Adjustment capability of separation angle between two sensor heads: ± 10 to ± 70 deg

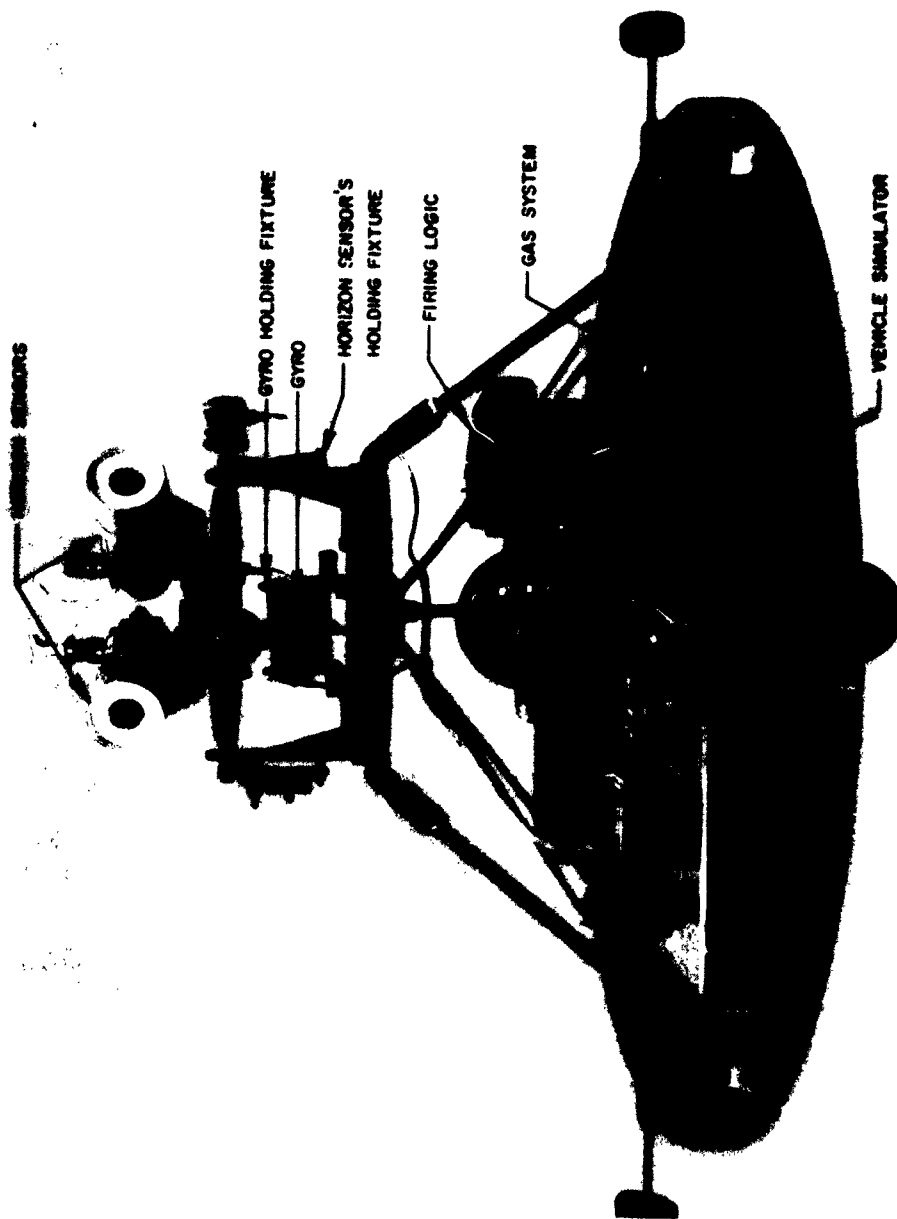


Figure 3. Vehicle Simulator, View 1

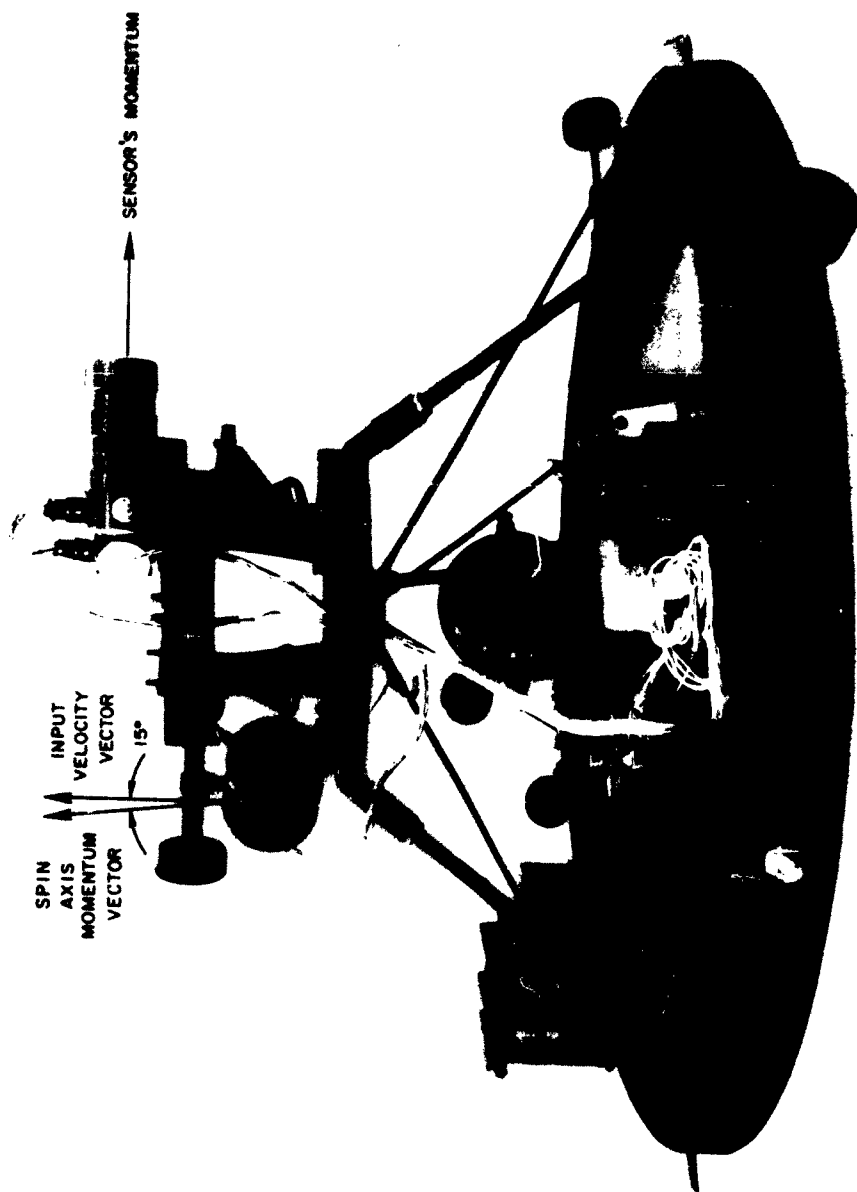


Figure 4. Vehicle Simulator, View 2

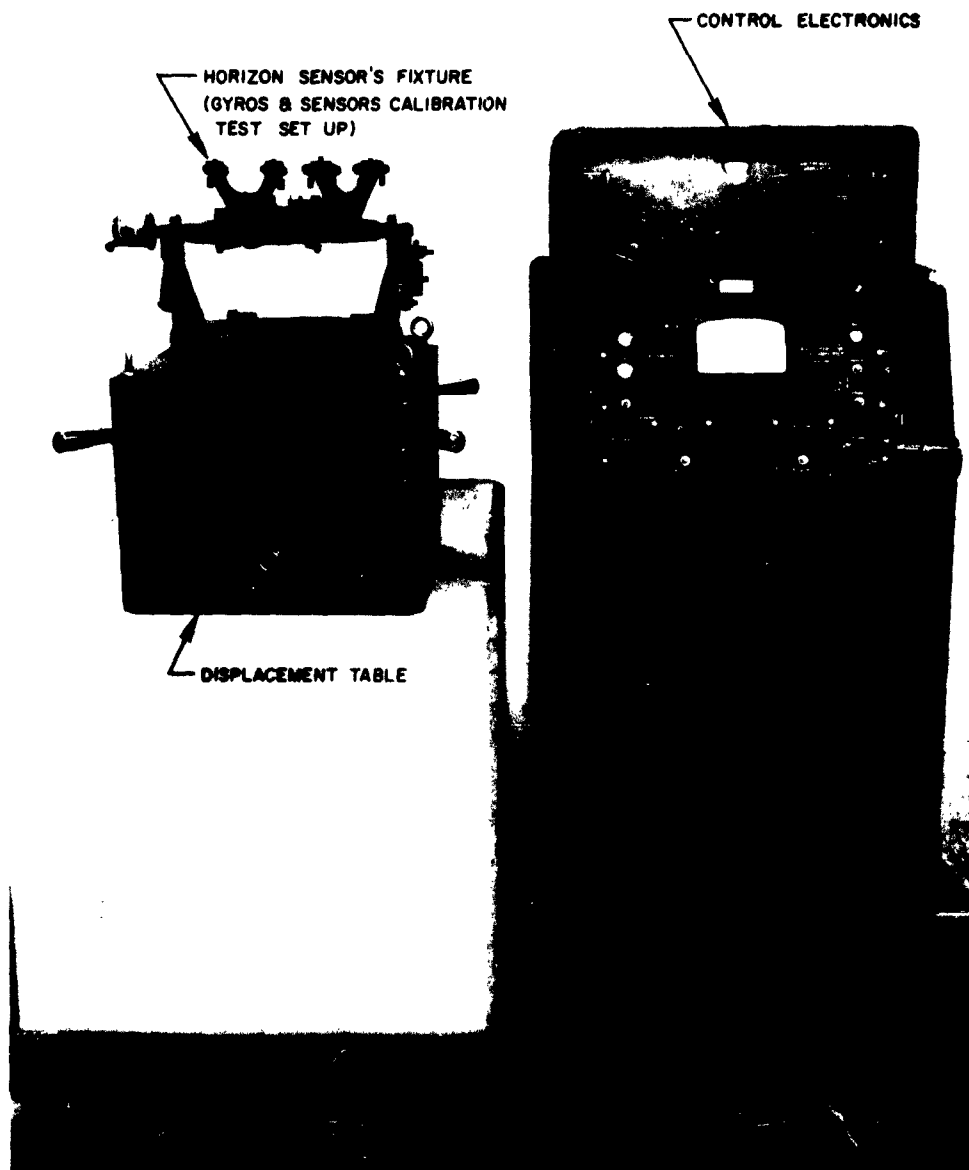


Figure 5. Oscillating Servo Displacement Table System

- 2) Yaw adjustment: complete freedom
- 3) Pitch variation: ± 20 deg
- 4) Elevation adjustment: 1-1/2 in.
- 5) Variation of sensors' focus plane distance from the fixture's center of rotation in yaw: 0 to 20 in.
- 6) Capability to static balance pitch axis

In addition, for the presently conducted satellite attitude control system experiment, it is possible to achieve dynamic equilibrium by mounting the system's gyro as indicated on Fig. 3 and 4, which allows for mutual cancellation of the horizon sensors and the gyro motor momentums. In this particular case the gyro is mounted such that its spin axis is rotated to within 15 degrees of the rate input vector.

C. RATE GYRO HOLDING FIXTURE

The design and fabrication of the gyro holding fixture (Fig. 3 and 4) have been completed. The special features of this fixture are the capability of incremental rotation (0 to 75°) of the gyro spin axis with respect to the rate input vector and the capability of mounting it either on the horizon sensor's holding fixture or on a displacement table.

D. PROCUREMENT OF AN OSCILLATING SERVO DISPLACEMENT TABLE SYSTEM

During this report period, Micro Gee Products, Inc. delivered a modified version of its 63A model oscillating displacement table (Fig. 5); some of the performance characteristics thereof are:

- 1) Displacement range: ± 15 deg
- 2) Gain: greater than 1 deg/v
- 3) Threshold: less than .002 deg
- 4) Max load weight: 50 lbs
- 5) Corner Frequency: .2 cps

This table was partially checked out and it appears that all applicable specification requirements have been satisfactorily met except for the system's input signal distortion. This distortion was traced to the system's

low-frequency generator (Krohnkite) which produces a distorted sinusoid in the frequency range from .01 to .1 cps; correction thereof is being presently discussed with Micro-Gee Products, Inc.

This particular inability to meet the harmonic distortion does not render the system inoperative. It was found that use of a Hewlett-Packard function generator (202A) allows full utilization of the displacement table.

In the laboratory, the function of this equipment is to provide a facility for calibration of horizon sensors and flight control system gyros. In addition, this equipment will be used for closed loop tests of the experimental satellite attitude control system. Briefly, the testing technique employed will be to mount the horizon sensors and the system rate gyro on the servo table. The servo table motion will be commanded from electronically simulated vehicle dynamics. A reference for the control system will be established in this test set-up with the earth-horizon simulator. The detector will sense any deviation of the servo table position from the established reference and an error thus generated will be introduced to the firing logic circuit affecting the control circuit activity. In the process of this test, various firing logic adjustments will be made.

IV. ATTITUDE CONTROL SYSTEM IMPLEMENTATION

A. HORIZON SENSING SYSTEM

The horizon sensing system (Ref. 3) has been analyzed with respect to its physical adaptiveness to the geometric configuration of the laboratory space vehicle simulator and earth simulator.

The horizon sensing system was originally intended for a vehicle orbiting earth at altitudes of 1,000 n mi. At that altitude, the system was characterized by a 1 v/deg signal output. The geometric requirement for the laboratory attitude control system experiment is a 5,000 n mi orbit.

Hence, it is necessary to know the amplitude of the sensor's separation angle, the distance between the center of rotation of the laboratory simulated vehicle, and the focus plane of the horizon system optics which will produce a linear ($\pm 10^\circ$ range) signal output of 1 v/deg or more.

The horizon sensing system which is being considered for the initial attitude control experiments consists of two conical scan subsystems placed at a constant angle, $\pm K$, with respect to the vehicle's yaw axis. Each scan subsystem has a relatively large conical field of view of 2γ degrees, accomplished by rotating prismatic components of the system's optics by a hollow shaft motor. A thermistor bolometer detector is used to sense difference between the radiance of earth and space. The detection of attitude errors depends on the length of time the thermistor bolometer receives the earth's radiant energy during one revolution of the scan motor.

For vehicle rotation about the local roll axis, one sensor will see more of the earth's IR field than the other. If then the time that each sensor receives earth's radiation is translated into pulse lengths, the difference of these pulse lengths will constitute a measure of the vehicle roll error.

In terms of the laboratory test geometry indicated on Fig. 6, the roll deviation is measured by first determining the time periods during which the cones, due to scans one and two, intersect the hemisphere. The corresponding angles of intersection are indicated by the angles $2B_1$ and $2B_2$, respectively. These time periods are differenced and the resultant Δt becomes an indication of the desired roll angle measurement.

Based on the configuration illustrated by Fig. 6, analytical expressions were derived that led to the conclusion that for separation angle $K = 37.5^\circ$, for a focus distance of 10.06 inches, for an earth simulator radius $r_e = 15$ inches, and for a distance from center of simulated earth to center of simulated vehicle of $\rho = 37$ inches, the horizon sensing system will produce a signal of 1.4 v/deg.

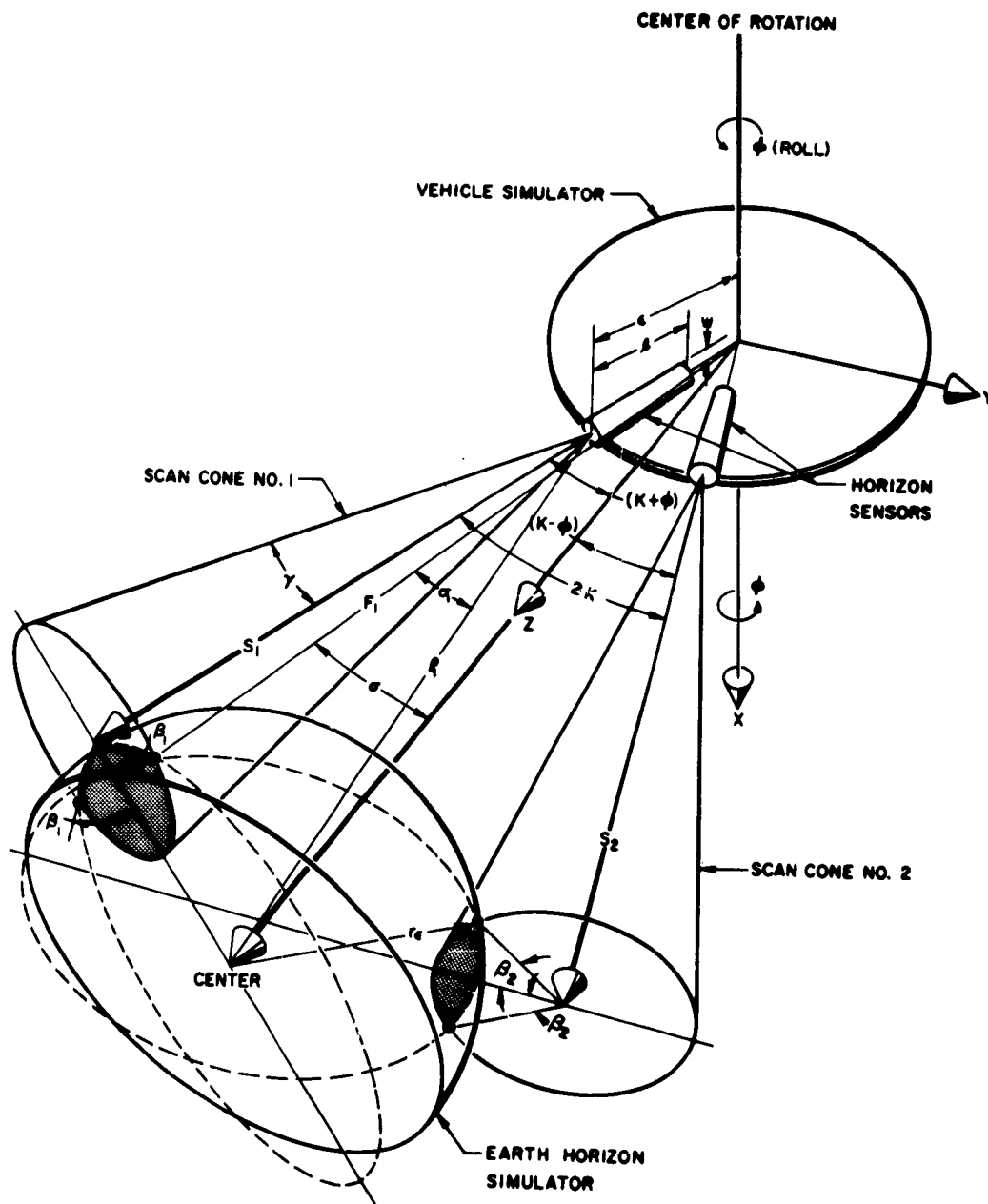


Figure 6. Laboratory Test - Geometric Configuration Earth and Vehicle Simulation (Top View)

The summary of the analytical effort is presented in Fig. 7, 8, and 9.

B. RATE GYRO AND ITS SIGNAL CONDITIONING CIRCUIT

A rate gyro (Kearfott, T2005-1A-A) was procured as required (Ref. 1) for the implementation of the experimental control system gas firing logic. The gyro's function is to sense high angular velocity (acquisition phase) of the vehicle and to provide a signal during transient operation. The rate gyro and its accompanying switching circuit act as a rate switch which provides a plus (+) step signal whenever the angular velocity of the experimental vehicle is greater than + .3 deg/sec and a minus (-) signal for less than - .3 deg/sec. The gyro is de-activated when the vehicle settles into limit cycle operation. The undesirable feature of this gyro is its high steady state momentum (2.86×10^3 gram-cm-sec) which would cause torquing of the simulated vehicle. To minimize this torque, a momentum-cancelling scheme of the gyro momentum with the combined horizon sensors momentums was suggested. This necessitates rotation of the gyro spin axis to within 15 degrees of the simulated vehicle's velocity vector which in turn reduces the gyro signal output by approximately a factor of four (4).

For this reason and because of the characteristics of the associated electronic switching components, it was necessary to design and fabricate the gyro signal conditioning circuit illustrated by Fig. 10.

This circuit is characterized by a gain of 300 and sensitivity (in conjunction with the switching electronics) of approximately .007 deg/sec.

C. PARTIAL SIMULATION EXPERIMENT OF THE SATELLITE ATTITUDE CONTROL SYSTEM

The implementation of the satellite attitude control system progressed to a point where system testing with completed hardware and partial simulation could be conducted. The gas jet system was developed and is reported in Reference 2. The control logic network was packaged in its final "flight"

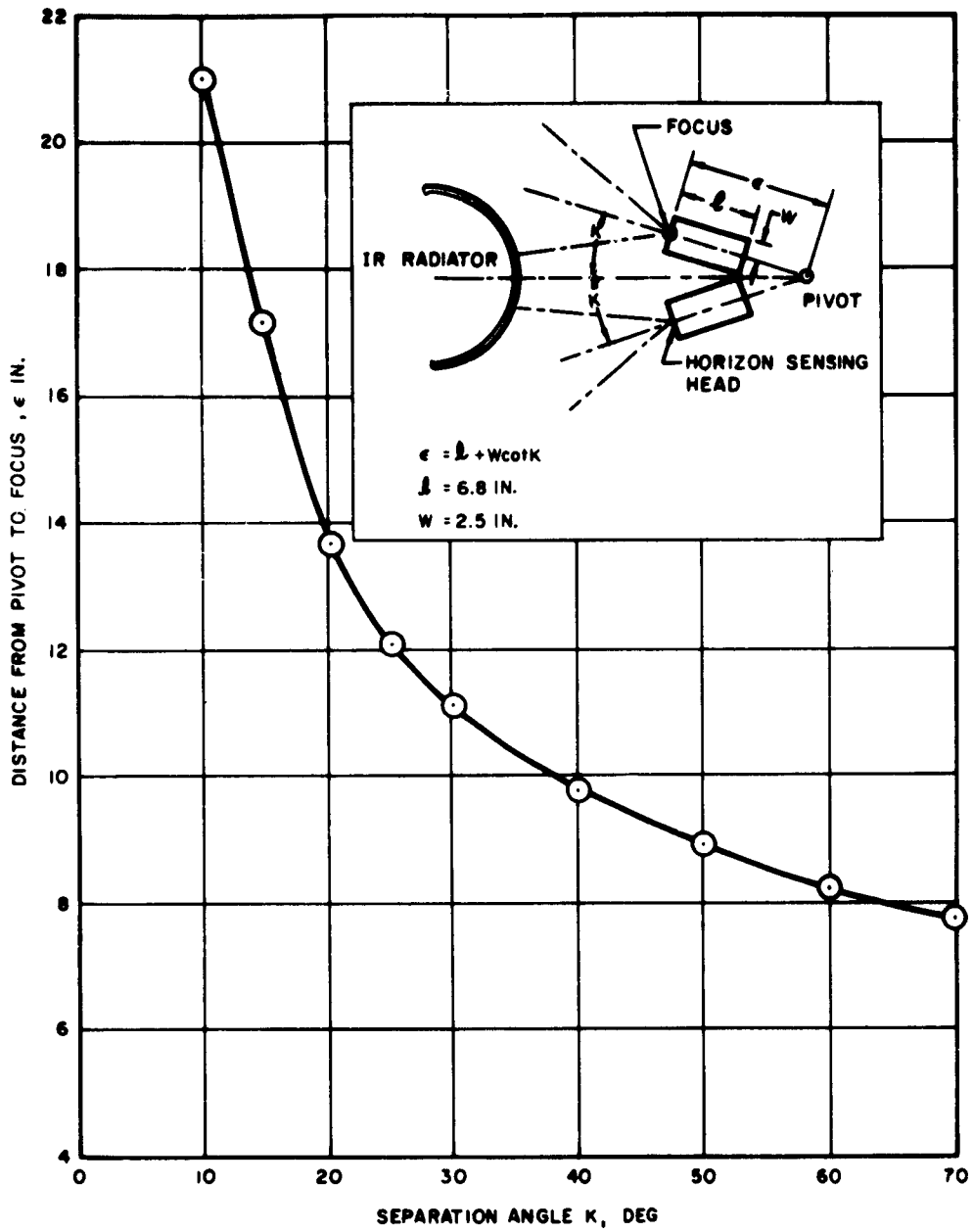


Figure 7. Distance Between Pivot and Focus (ϵ) versus Separation Angle κ

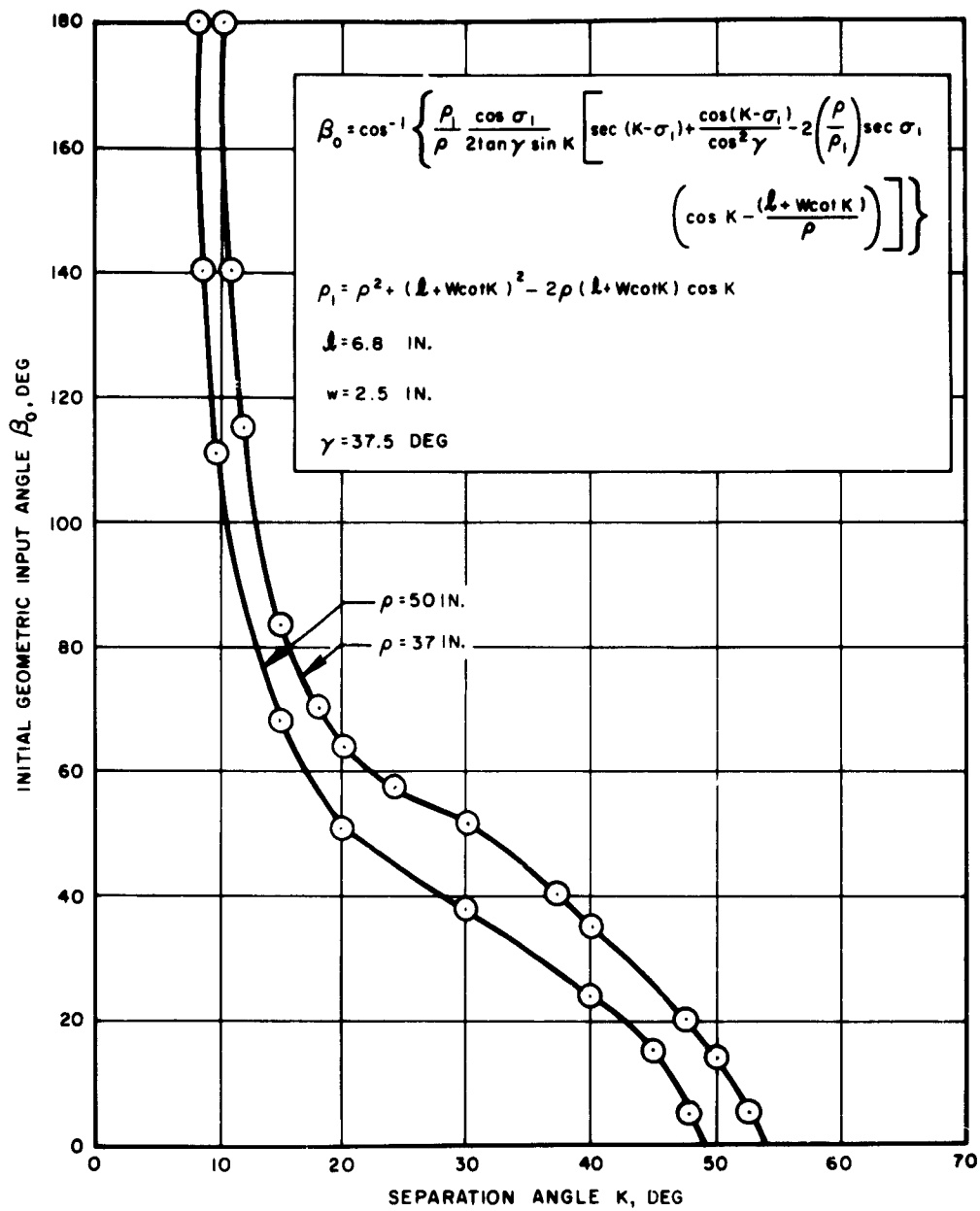


Figure 8. Horizon Sensors' Input Angle β_0 as a Function of Angle (k) for Given ρ (Distance from Center of Simulated Earth to the Center of Rotation of Simulated Vehicle)

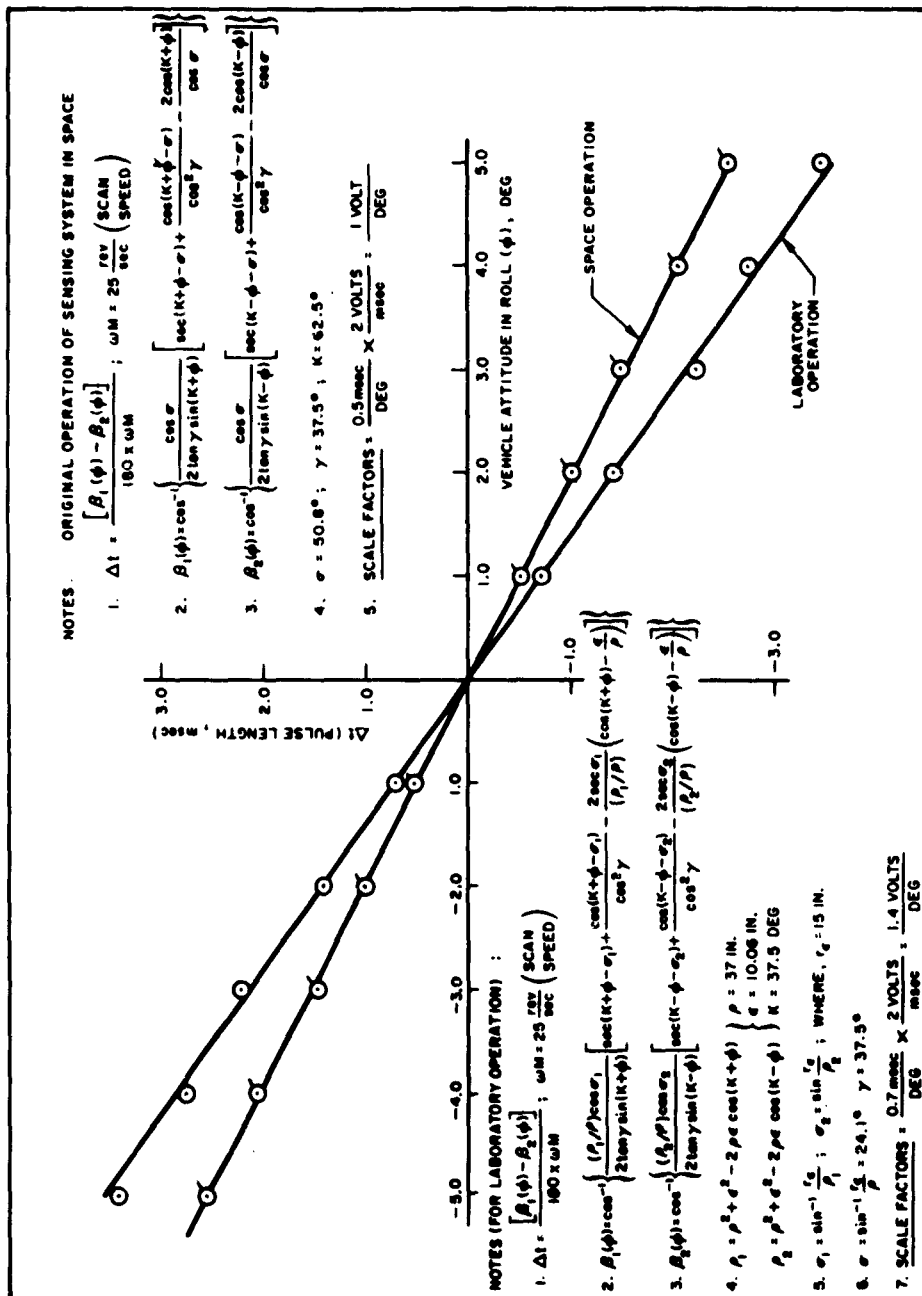


Figure 9. Predicted Performance of Horizon Sensing System for Laboratory Operation

NOTES:

1. ALL TRANSFORMERS — TRIAD — TY-37X
2. ALL CAPACITORS — μf
3. ALL RESISTORS — OHMS
4. DESIGN DATE — NOV 1962 *887*

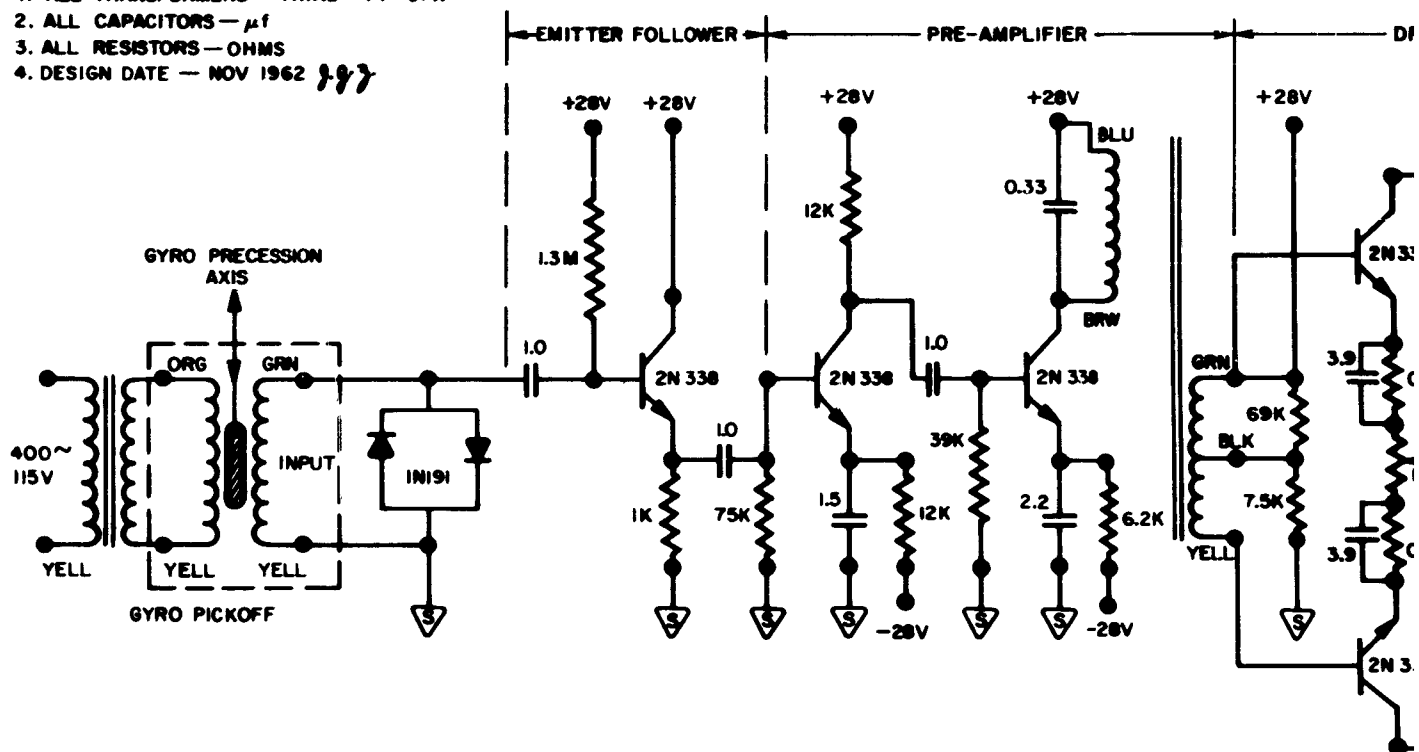
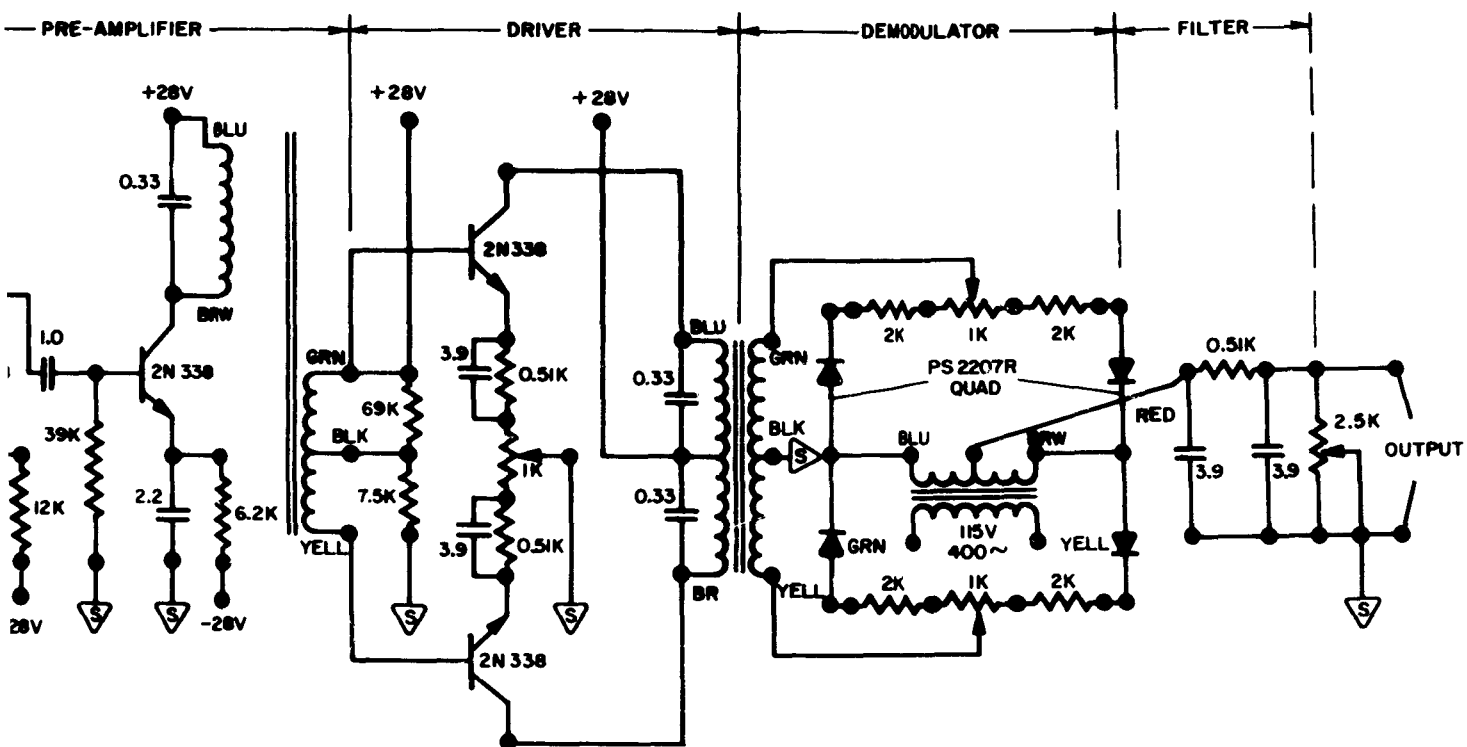


Figure 10. Gyro Signal Conditioning Circ



are 10. Gyro Signal Conditioning Circuit

form and is reported in Reference 4. This gas jet system and the control logic network were integrated with electronically simulated vehicle inertia, horizon scanner, and rate gyro. The primary purpose of the experiment was to check out the control system concept as reported in Reference 1 and to obtain a reference set of test data, for the system operating with an ideal vehicle, to compare with that of subsequent tests on the wire table. Secondly, if required, further debugging of the control logic could be accomplished with varying input signals applied as in a "flight" sequence. The results clearly demonstrated the feasibility and effectiveness of the control system and it was recommended that the control system be mounted on the "actual" vehicle (wire table) for further experimentation. The details of the test set-up and procedure may be found in Reference 5. Only a block diagram, Fig. 11, of the set-up is herein included. The gas jet and control logic hardware, as now mounted on the wire table, may be seen in Fig. 3.

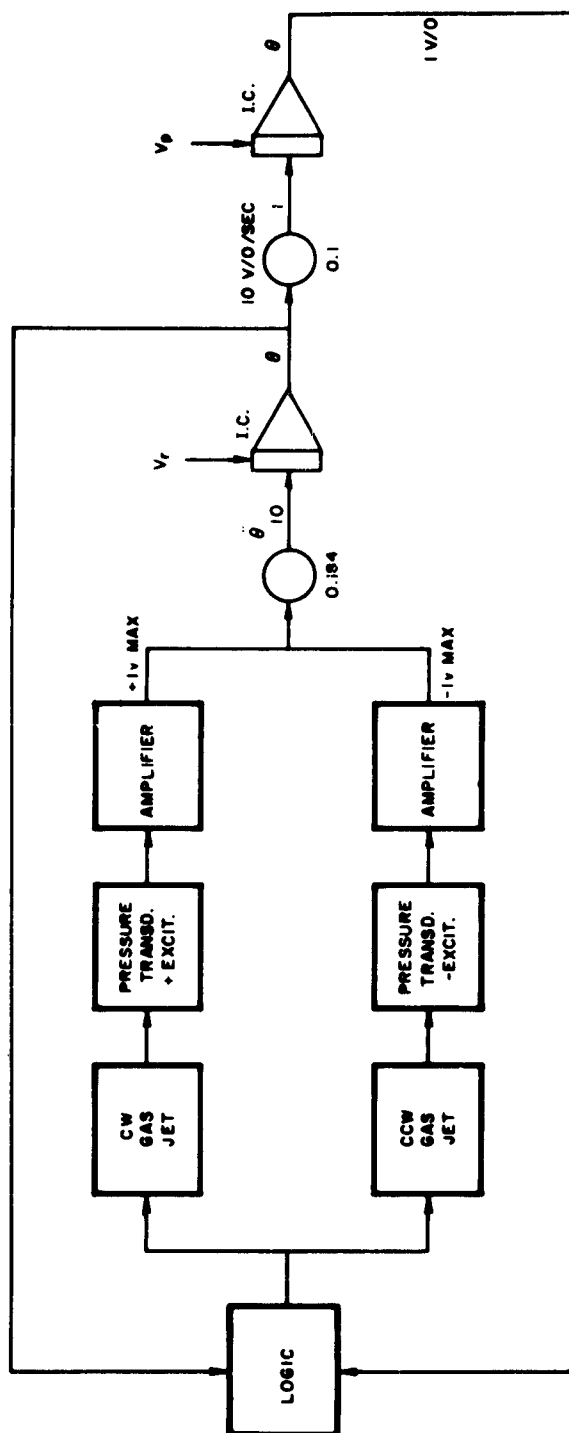


Figure 11. Test Set-Up Block Diagram

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