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FLIGHT CONTROL COMPUTER FOR SATURN SPACE VEHICLES

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ABSTRACT

The flight control computer for Saturn receives attitude signals from the stable platform, rate signals from rate gyros or lead networks, and angle-of-attack information from body-fixed accelerometers or other sensors.

These signals are a result of several forces acting on the vehicle such as engine thrust, wind, gravity, and internal vehicle flexing and bending. The signals are shaped, given a weighting function, and combined with program data in several servo amplifiers. The resultant amplified outputs drive servo actuators which gimbal the engines to provide thrust vector control for the S-I and S-IV stages. Feedback signals from the actuators close the servo loop. Provisions are made to program or switch gains in any channel for the varying needs of the flight program as it progresses.

INTRODUCTION

By controlling the attitude, a vehicle can be made to follow a prescribed trajectory. The system that determines the desired attitude is often referred to as the "guidance system", whereas the "control system" performs the vehicle orientation. This discussion concerns the attitude control system and the resulting electronic hardware.

The use of a very large multi-engine, multi-stage space vehicle such as the Saturn requires special design techniques if a single unified control system is to be used. Signals of proper polarity must be routed to each control engine and must be switched from stage to stage as the flight progresses. This is in additon to the proper mixing, shaping, and gain programming of signals from the various sensing devices.

This flight control computer was designed for the Saturn I, Block II vehicle (S-I & S-IV stages) shown in Figures 1 & 2.

ENGINE CONFIGURATIONS

The first stage of Saturn 1 has four fixed engines, 5, 6, 7, and 8, arranged in a square pattern and canted at an angle of 3° to the vehicle axis, and four outer gimballed engines, 1, 2, 3, and 4, in a similar square and canted at an angle of 6° as shown in Figure 1. The cant angle directs the individual engine thrust approximately through the nominal center of gravity of the vehicle. The four gimballed engines provide control about the pitch, yaw, and roll axes. The six gimballed engines in the S-IV stage of Saturn I are grouped in a pattern shown in Figure 2. Engines one through four provide control in pitch, yaw, and roll, while engines five and six provide control in pitch and yaw only.

COMPUTER INPUTS

As shown in Figure 3, the vehicle attitude may not coincide with the relative air flow because of winds or changes in heading. The resultant angle-of-attack produces structural loading that must be reduced.

Three alternate methods of sensing angle-of-attack are provided on the early test flights; a nose cone differential pressure transducer, a probe extending from the side of the nose sensing differential pressure (used on Jupiter), and body-fixed accelerometers. The outputs of the accelerometers are shaped into angle-of-attack information in the control computer. All of the angle-of-attack signals are brought into the control computer to allow a choice; however, the accelerometer information has been tentatively selected for use. Also, angular rate information from rate gyros is supplied to the computer for use if needed; but differentiation of the attitude signal is preferred as being more reliable. The attitude rate information will be obtained by differentiation, if the bending mode transmission is not too severe at the stabilized platform location. Figure 4 is a simplified diagram of the control system and rigid body dynamics.

In addition to the rigid body dynamics, the vehicle is elastic and is subject to bending at various resonant frequencies. Some of these frequencies are only slightly above the control frequency and, since the sensing elements cannot differentiate between bending and error signals, elaborate filtering and shaping circuits must be provided in the control computer.

Changes in vehicle characteristics occur because of staging, changing mass, and varying aerodynamic environment. However, these changes can be predicted as a function of flight time and compensated for in the control computer. The computer has a small program timer and receives a number of external timing signals to accomplish these changes.

COMPUTER BLOCK DIAGRAM

Figure 5 is a block diagram showing the major modules of the computer. Each module is constructed in such a manner as to permit bench testing and verification before assembly into the main assembly. Some measuring resistors and switching relays not shown on the block diagram are grouped into modules so that all electronic components can be pre-assembled and plugged into the main chassis.

MODULE DESCRIPTION

Servo Amplifier The servo amplifier used in the computer provides a high degree of flexibility in that signal mixing and summing is accomplished in a magnetic pre-amplifier, shown in Figure 6. The sensitivity of each input channel can be set by selecting the proper number of turns in the signal winding, by connecting two or more windings in series, or by selecting the value of a series resistor. Close matching of the core parameters B_{max}, H_c, and Δh to within $\pm 3\%$ is required to minimize any drift in the output. The magnetic amplifier is followed by a two-stage direct-coupled transistor amplifier, and again the transistor pairs Q_1 -Q_2 and Q_3 -Q_4 have closely matched beta characteristics. The transistor amplifier drives a differential servo valve which controls the actuators used to gimbal the engines. Further stability in the overall amplifier is achieved by providing approximately 15 db of negative feedback to the magnetic amplifier. The gain of the amplifier is adjusted to provide a ΔI gain of 2.83 ua per ua turn. For example, with a 424 turn input winding, 10 ua is required for a ΔI of 12 mA which will provide full flow through the S-I stage actuator valve.

Filter Modules The filter modules used as part of the networks in the computer have been designed as individual channel plug-in modules for ease in testing before installation into the computer. In addition, rapid changes in the shaping network program can be made if required prior to a particular flight.

Passive elements only are used in the shaping networks. While the values of reactive elements required for the frequencies are quite large, careful component selection and design keep the overall size reasonable. For instance, the largest inductor used (2000 henries) will weigh less than 1/2 kg (1 lb.). It is expected that the total weight of the shaping networks will be 7.3 kg (16 lbs.) in the "as flown" version. This is compared to a total computer weight of 16 kg (35 lbs.).

A typical filter configuration used in an attitude signal channel with resultant phase and amplitude characteristics is shown in Figure 7. For the control frequency, this filter acts as an RC network which provices a 40° phase lead with respect to the output signal. For the first bending mode, it acts as a shaping network which provides approximately 60° to 80° of phase lag; amplitude attenuation is provided for the second bending mode. Higher frequencies are suppressed by the servo loop. Similar type filters and shaping networks are used in the $\dot{\gamma}$ channels.

Control Attenuator Timer A variable gain program in the a_0 and g_2 channels is required to provide optimum control of the vehicle during portions of the flight when high wind forces may be encountered. A typical gain program is shown in Figure 8. During the initial portion of the flight, primary control is obtained from the a_0 channel (stable platform). After a period of time, gain in this channel decreases, while the gain in the g_2 channel (angle-of-attack) increases so the g_2 gain is at a maximum at 90 seconds. After 100 seconds, the a_0 gain again increases while g_2 decreases to zero. A further change in a_0 occurs after 125 seconds since less gain in this channel is then needed because of a reduction in vehicle moment of inertia caused by fuel depletion. A step reduction in a_1 gain (rate information) occurs after 125 seconds flight time. The variable gain programs are provided by a control attenuator timer, shown in Figure 9, which consists of a cam driven through a gear train by a synchronous motor. A sector gear riding in the cam groove rotates a two-gang precision linear potentiometer. The cam also has grooves about the circumference which activate microswitches to provide a zero set indication, end of program, and an active channel interval.

Complete Computer The computer housing was fabricated from sheet metal aluminum and made rigid by a dip-brazing process. Since this process requires that the housing be heated in a flux bath to a temperature close to the melting point of the aluminum, a pair of stainless steel plates was required to support the inside chassis during dip-brazing. Thorough washing after brazing is needed to remove all traces of the flux; testing with a silver nitrate solution will show the presence of any flux residue. The housing was then given an Iridite finish and the exterior was painted. Figure 10 shows the complete computer while Figures 11, 12, and 13 are photographs of individual modules. Computer wiring was accomplished by a harness running along the center of the chassis.

Reliability of the control computer is obtained by simplifying the design, by careful selection and testing of components, and by rigorous testing of the completed design.

COMPUTER TESTING

The computer is subjected to a complete acceptance test at the manufacturer's plant, where both static and dynamic signals are applied to each input from a test console to insure that computer performance is within required limits. The test is repeated as a part of incoming inspection at Marshall Space Flight Center. When the vehicle characteristics are changed, or when a change in computer design is required, the computer is integrated into the complete control system for flight simulation tests. The tests are performed to verify the design and performance of the control computer. Actual flight hardware is used, where feasible, during the performance of these tests.

The Quality Assurance Division at MSFC performs final qualification testing on the complete vehicle. These tests, in addition to pre-launch tests at Cape Canaveral, insure that the computer operates properly when placed in the vehicle system.

RELIABILITY ASPECTS

The parts for the control computer were selected with extreme care. Basis for the selection was the MSFC Preferred Parts List and the Design Guide Lines. A thorough analysis of all test data from a nation-wide parts testing effort was and is being made to assure maximum reliability. Practically all IDEP$_{(1)}$ and ECRC$_{(2)}$ parts test reports as well as test and failure information from all NASA organizations are used for this purpose. Expensive qualification tests have been run only when this information was inconclusive or contradicting. Reliance on information from nation-wide data centers saved considerable time and funds in establishing the qualification of the parts.

All parts are recorded in the computer-operated PRINCE$_{(3)}$ system. This fact insures that the designer of the control computer will be notified automatically when new information on these parts becomes available from the nation-wide parts testing effort, including failure flag-out service. This service enables the designer to keep abreast of any degradation or improvement of the parts with little or no effort.

Even more important than component selection is the training of engineers, technicians, and production workers associated with the program. Each assembly worker is uniquely trained and qualified to perform this particular task. Workers are requalified periodically and are constantly informed of the importance of their work and its relation to the success of the national space effort.

(1) Interservice Data Exchange Program

(2) Electronic Component Reliability Center

(3) Parts Reliability Information Center

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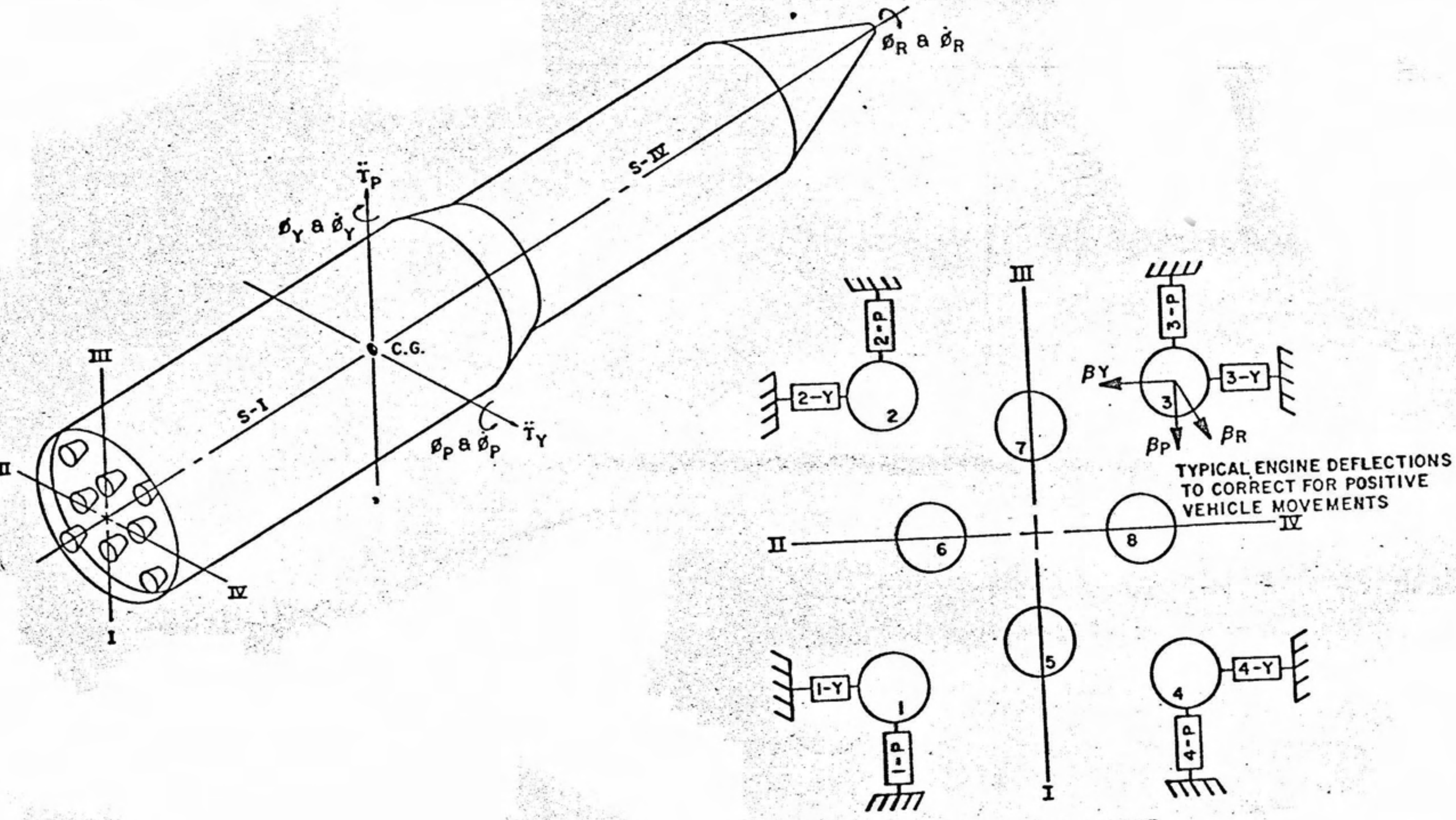
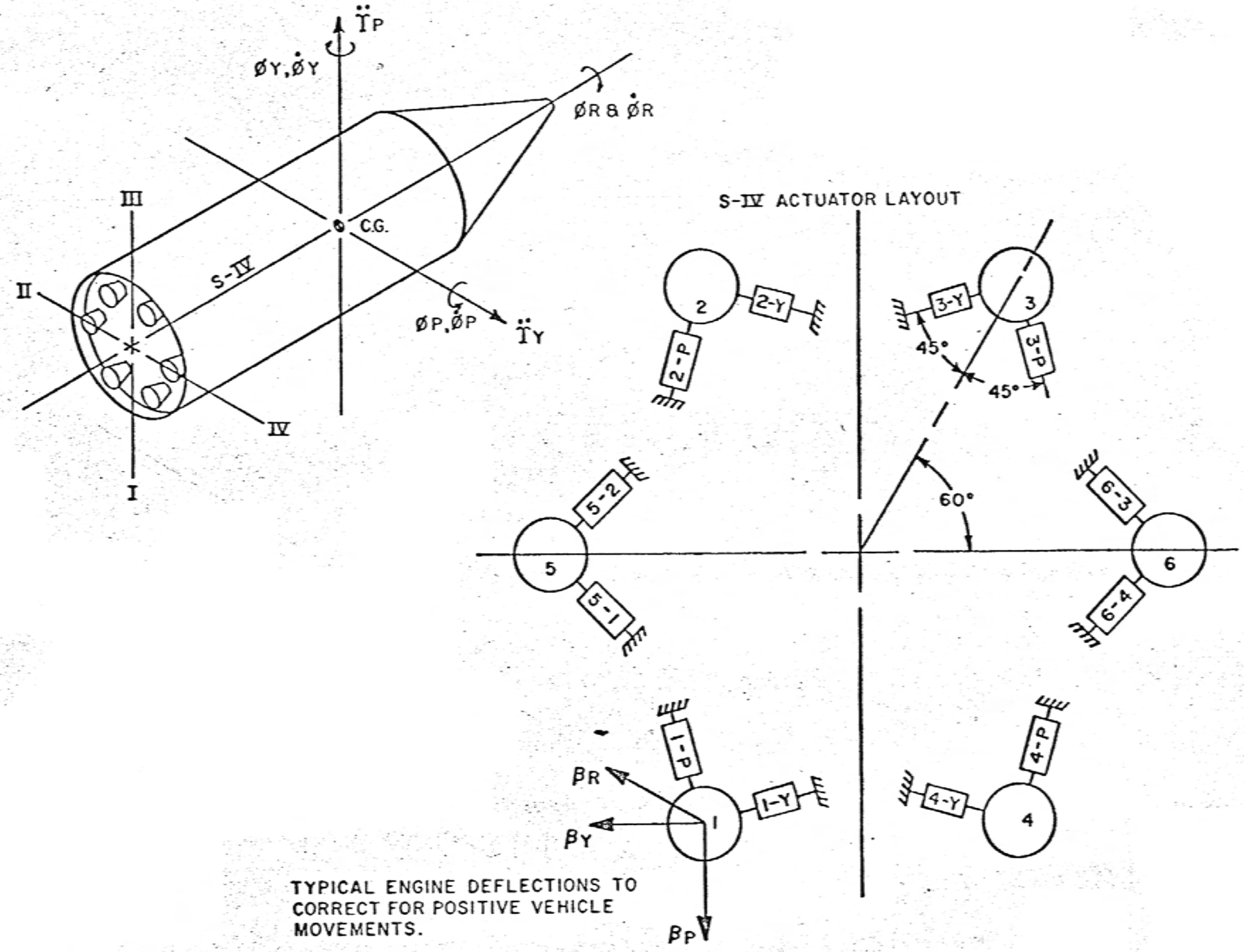


SI ACTUATOR LAYOUT

FIGURE I



TYPICAL ENGINE DEFLECTIONS TO CORRECT FOR POSITIVE VEHICLE MOVEMENTS.

FIGURE 2

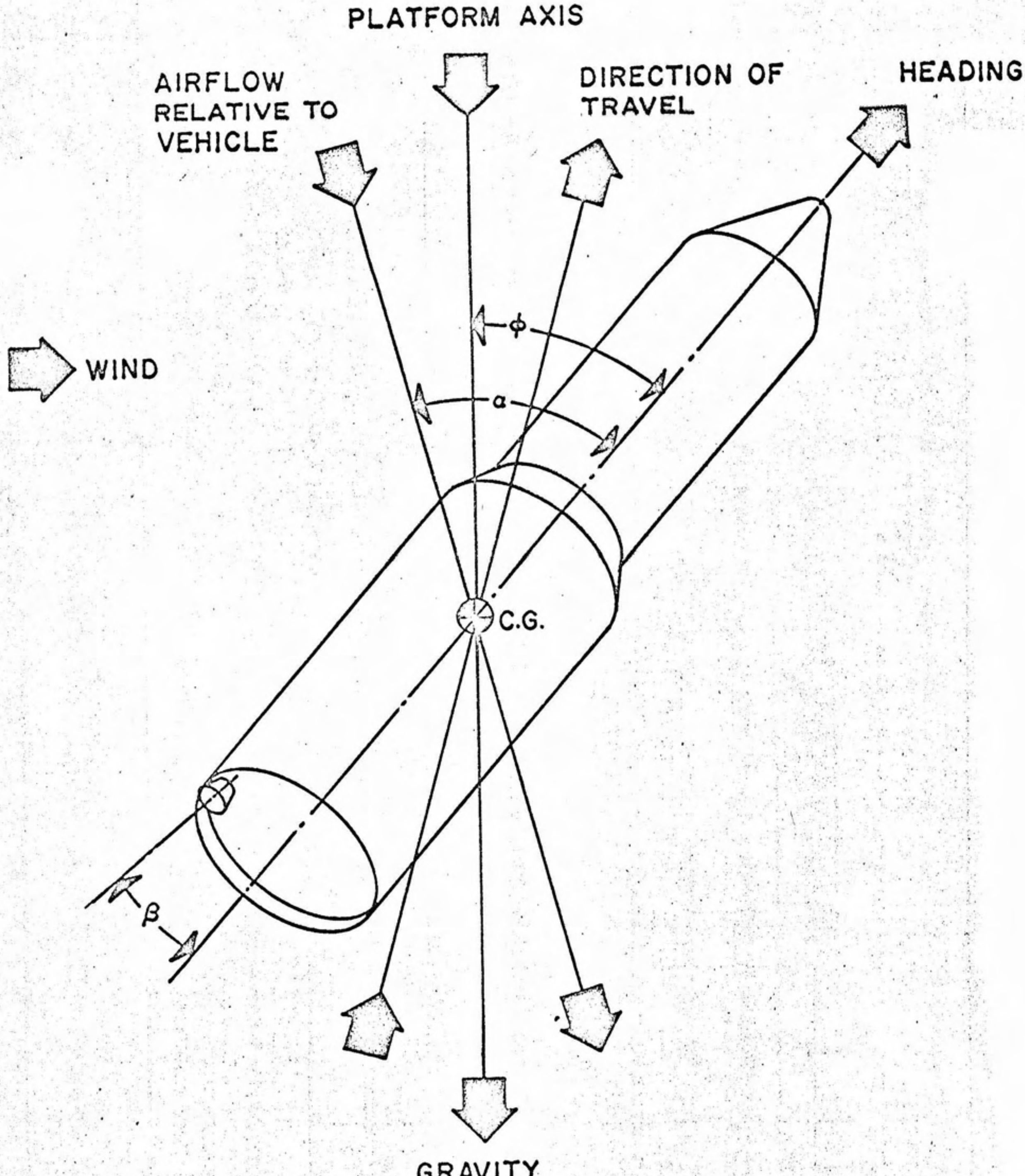
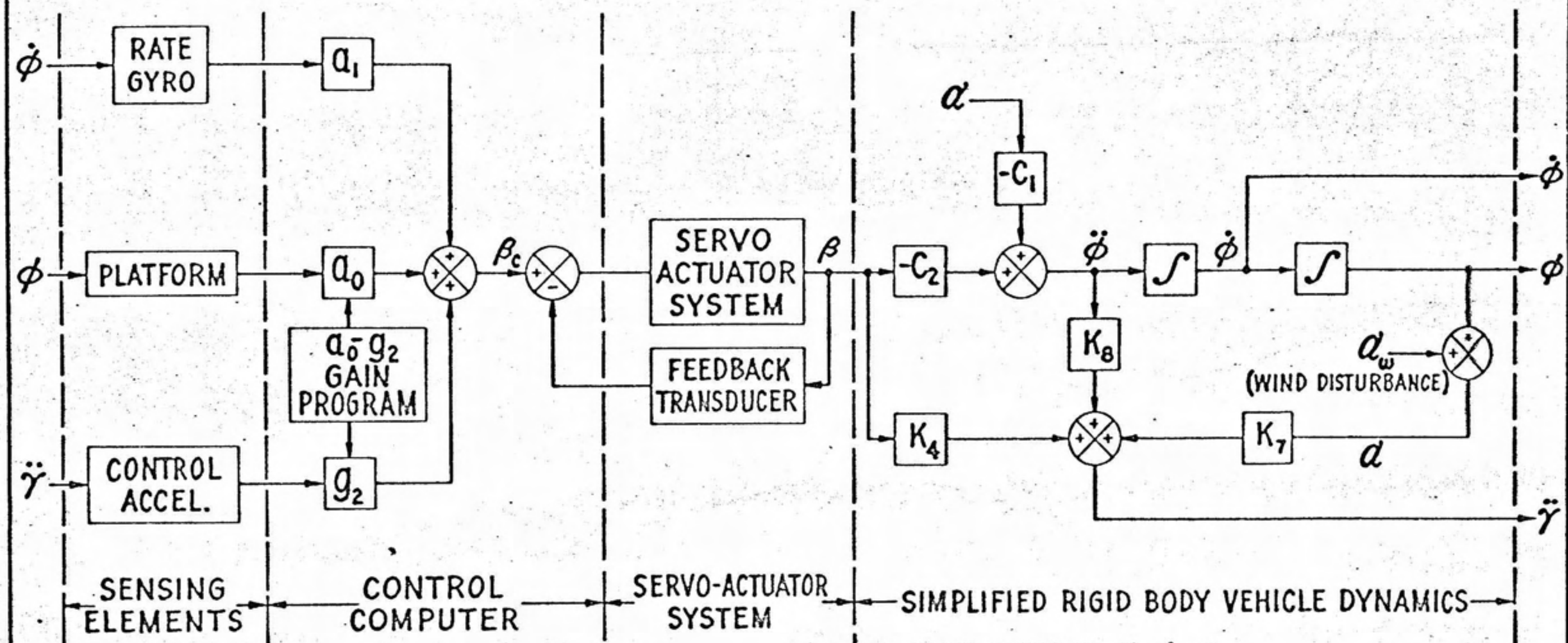

PLATFORM AXIS
AIRFLOW RELATIVE TO VEHICLE
DIRECTION OF TRAVEL
HEADING
WIND
φ
α
C.G.
β
GRAVITY

FIGURE 3

SIMPLIFIED VEHICLE CONTROL SYSTEM BLOCK DIAGRAM



ANGLE EQUATION: $\alpha = \phi + \alpha_\omega$

MOMENT EQUATION: $\ddot{\phi} = -C_2\beta - C_1\alpha$

CONTROL ACCEL. EQUATION: $\ddot{\gamma} = K_4\beta + K_7\alpha + K_8\ddot{\phi}$

CONTROL EQUATION: $\beta_c = a_0\phi + a_1\dot{\phi} + g_2\ddot{\gamma}$

LIST OF SYMBOLS

- $\phi, \dot{\phi}, \ddot{\phi}$ *ATTITUDE ANGULAR POSITION, RATE, AND ACCELERATION*
- a_0, a_1, g_2 *CONTROL COEFFICIENTS*
- β *CONTROL DEFLECTION*
- C_1 *AERODYNAMIC RESTORING COEFFICIENT*
- C_2 *CONTROL MOMENT COEFFICIENT*
- K_4 *LATERAL ACCEL. PER DEG. β*
- K_7 *LATERAL ACCEL. PER DEG. α*
- K_8 *COEFFICIENT OF COUPLING BETWEEN $\ddot{\phi}$ AND $\ddot{\gamma}$*
- $\ddot{\gamma}$ *ACCEL. NORMAL TO MISSILE LONG AXIS*
- α *AERODYNAMIC ANGLE OF ATTACK*
- α_ω *WIND ANGLE OF ATTACK*

FIGURE 4

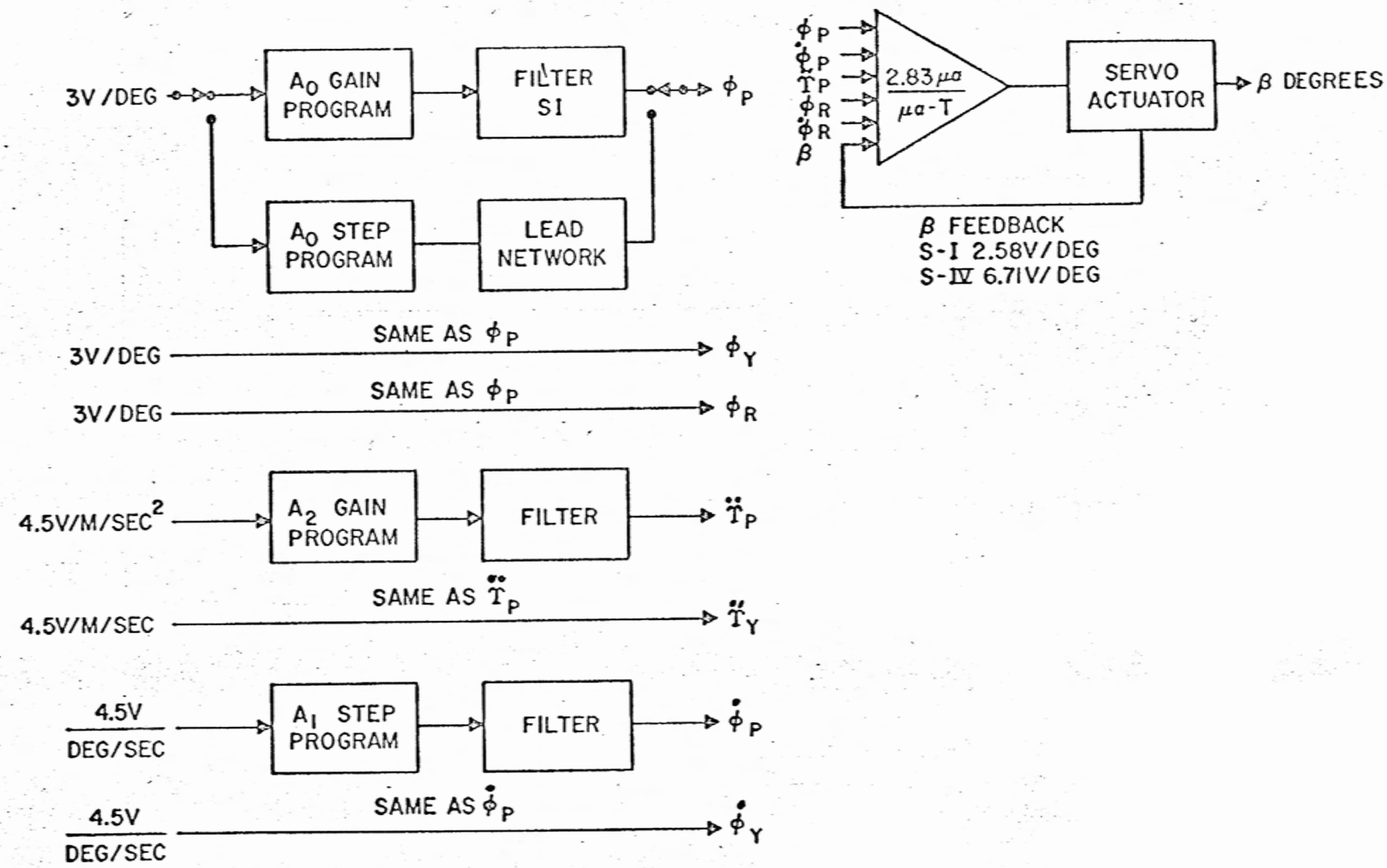

3V/DEG
A_0 GAIN PROGRAM
FILTER SI
ϕ_P
A_0 STEP PROGRAM
LEAD NETWORK
ϕ_P
$\dot{\phi}_P$
$\ddot{T}_P$
ϕ_R
$\dot{\phi}_R$
β
$\frac{2.83\mu a}{\mu a\text{-}T}$
SERVO ACTUATOR
β DEGREES
β FEEDBACK
S-I 2.58V/DEG
S-IV 6.71V/DEG
3V/DEG
SAME AS ϕ_P
ϕ_Y
3V/DEG
SAME AS ϕ_P
ϕ_R
$4.5V/M/SEC^2$
A_2 GAIN PROGRAM
FILTER
$\ddot{T}_P$
4.5V/M/SEC
SAME AS $\ddot{T}_P$
$\ddot{T}_Y$
$\frac{4.5V}{DEG/SEC}$
A_1 STEP PROGRAM
FILTER
$\dot{\phi}_P$
$\frac{4.5V}{DEG/SEC}$
SAME AS $\dot{\phi}_P$
$\dot{\phi}_Y$

FIGURE 5

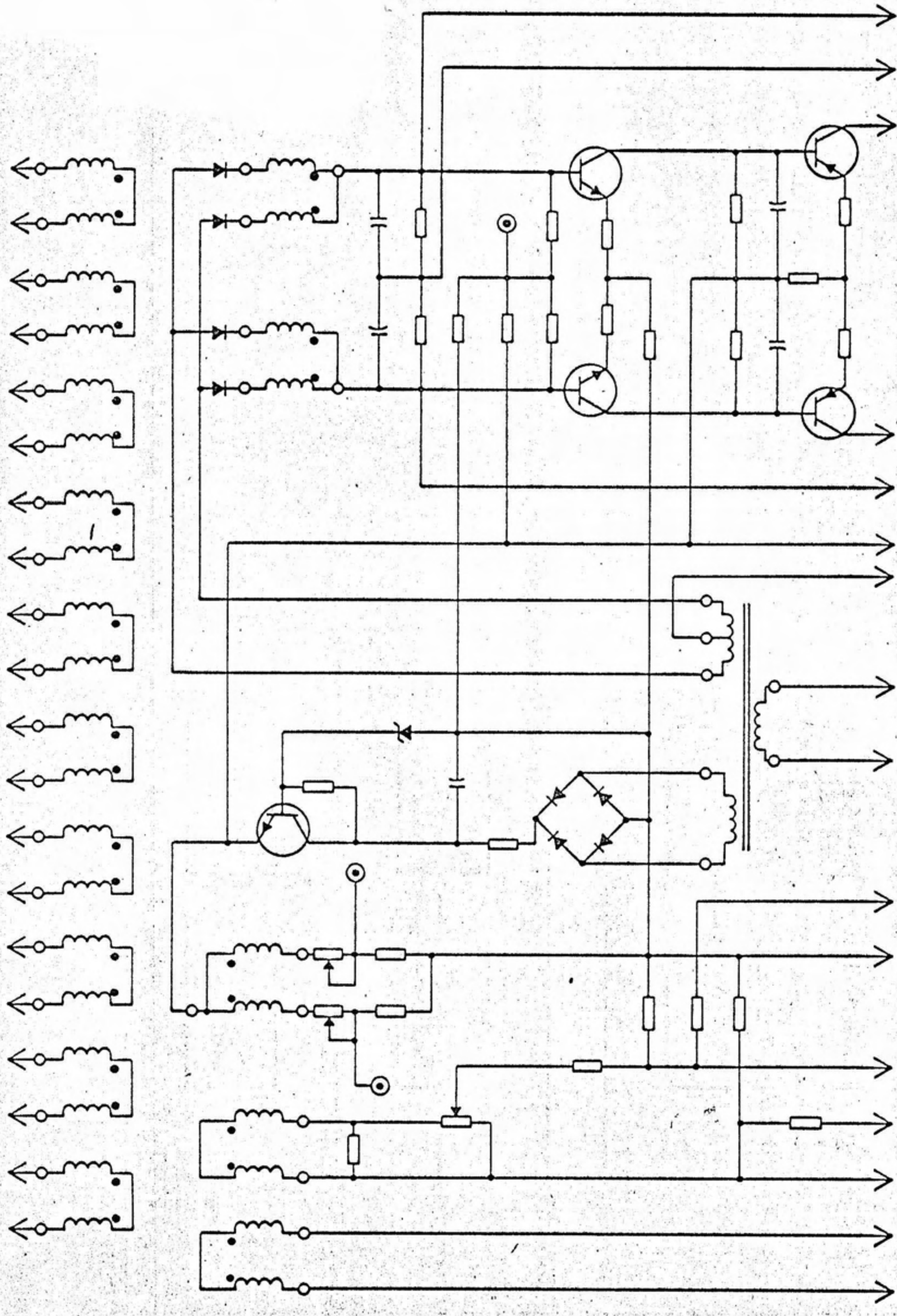
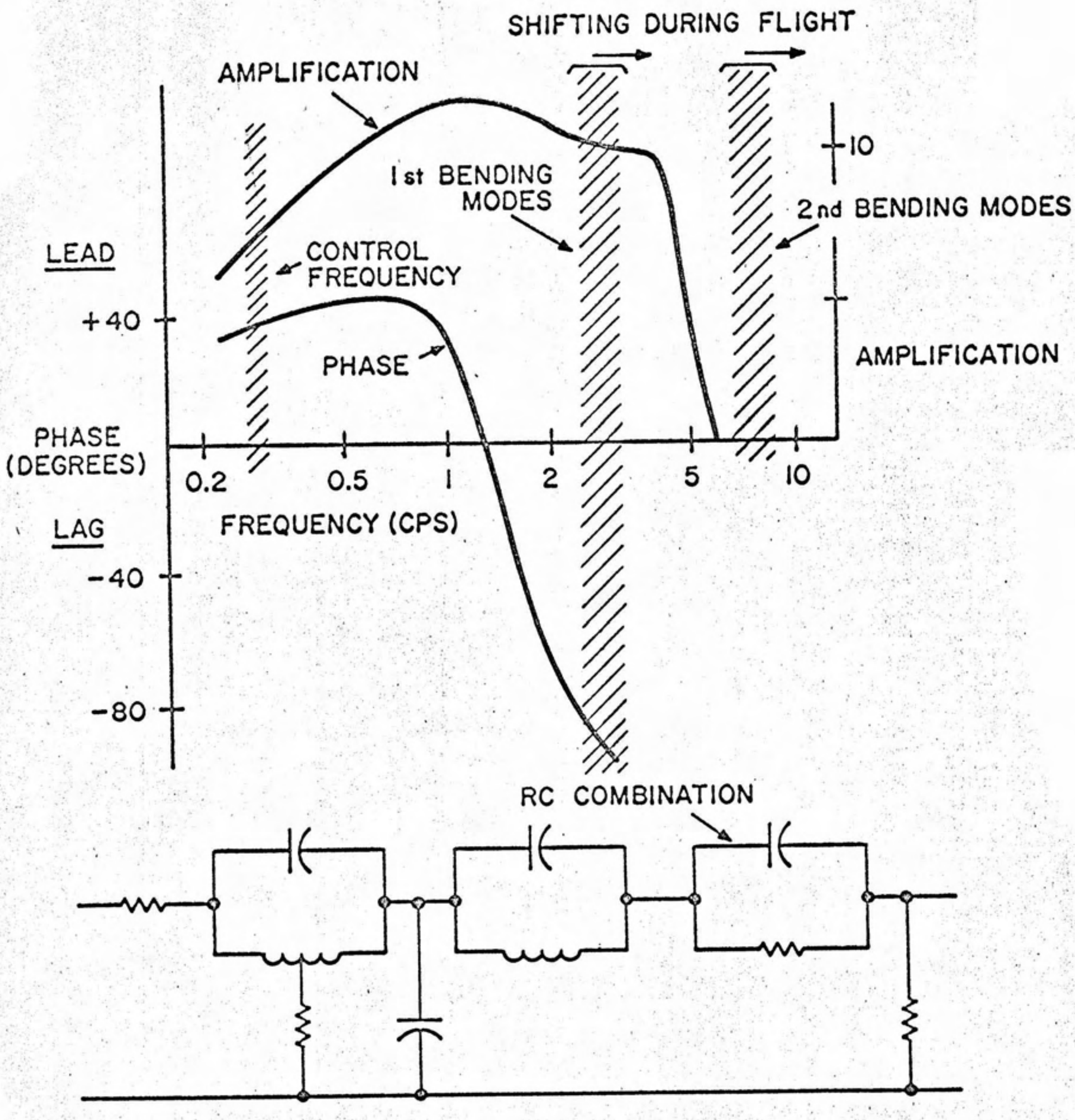


FIGURE 6



SHIFTING DURING FLIGHT
AMPLIFICATION
1st BENDING MODES
10
2nd BENDING MODES
LEAD
CONTROL FREQUENCY
+40
PHASE
AMPLIFICATION
PHASE (DEGREES)
0.2
0.5
1
2
5
10
LAG
FREQUENCY (CPS)
-40
-80
RC COMBINATION

FIGURE 7

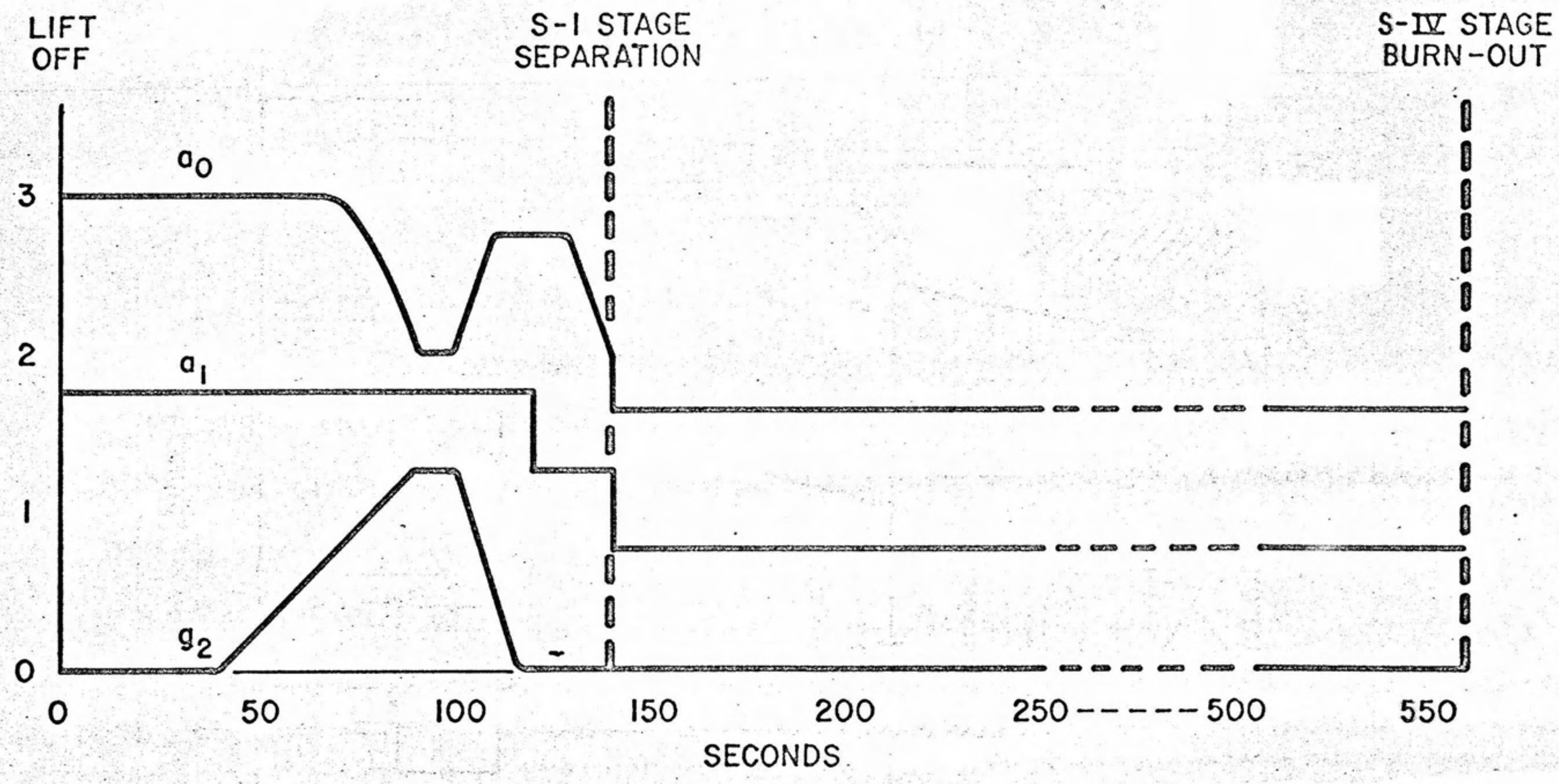
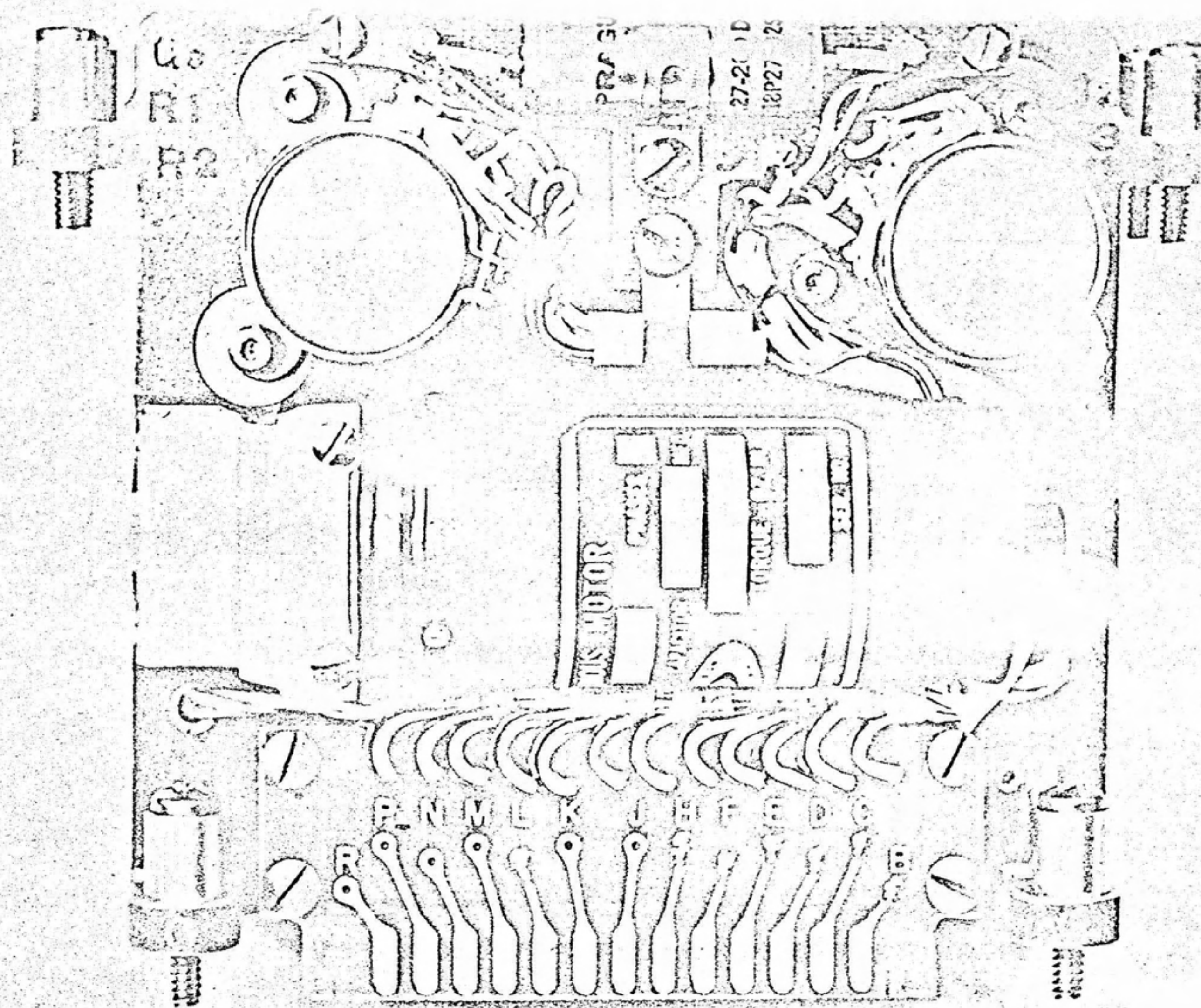
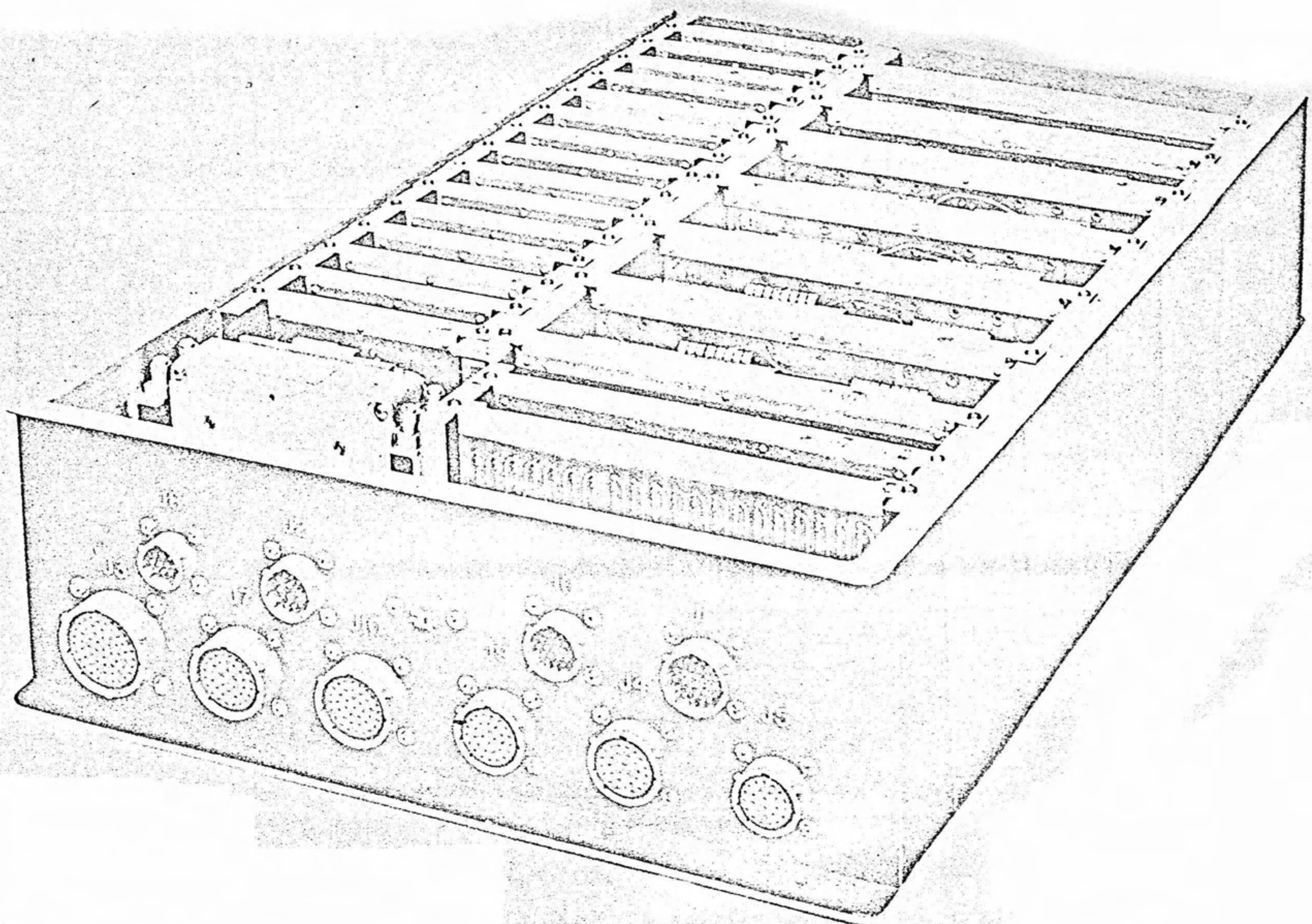

LIFT OFF
S-I STAGE SEPARATION
S-IV STAGE BURN-OUT
a_0
a_1
g_2
3
2
1
0
0
50
100
150
200
250 - - - - - 500
550
SECONDS

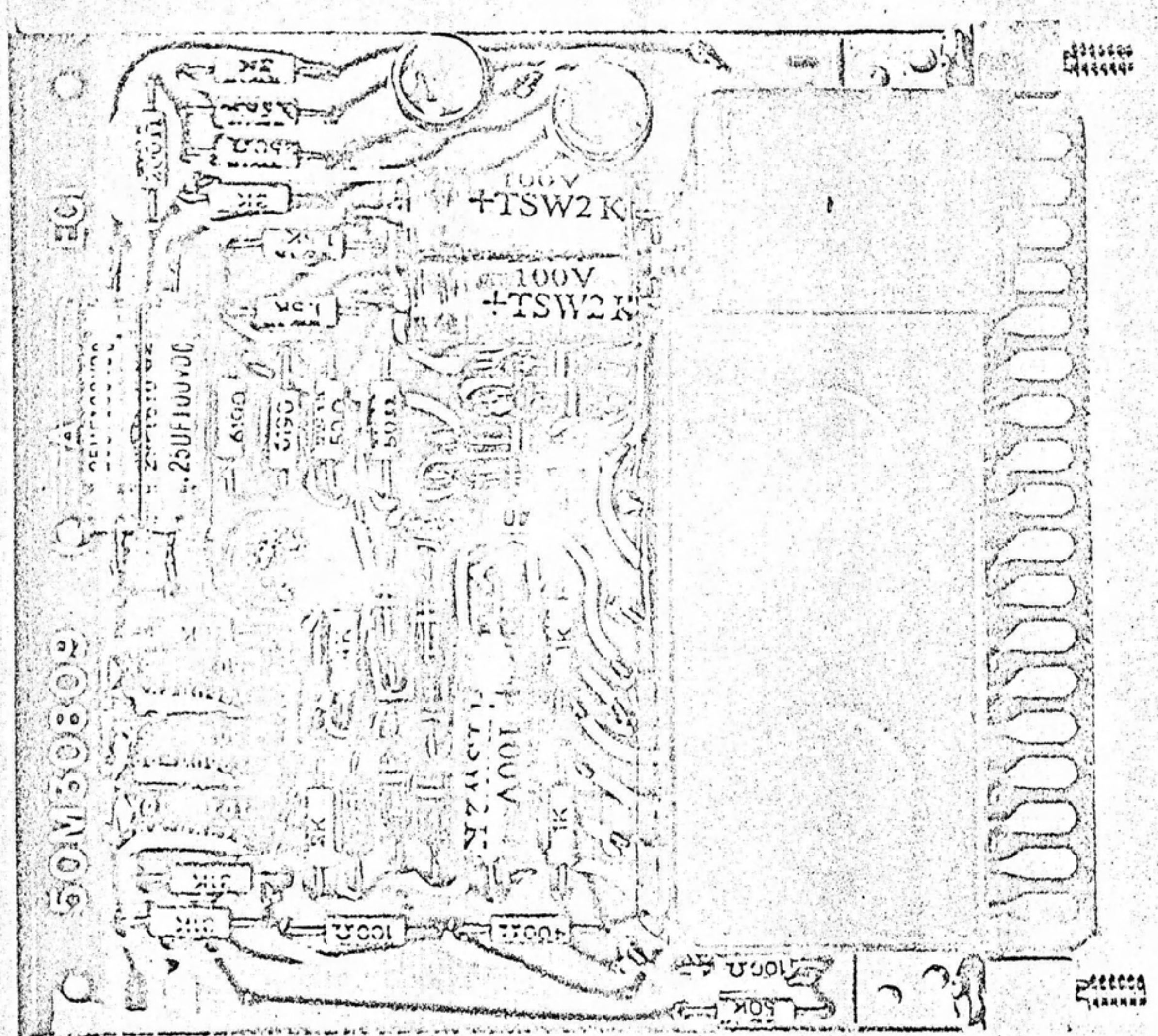
FIGURE 8



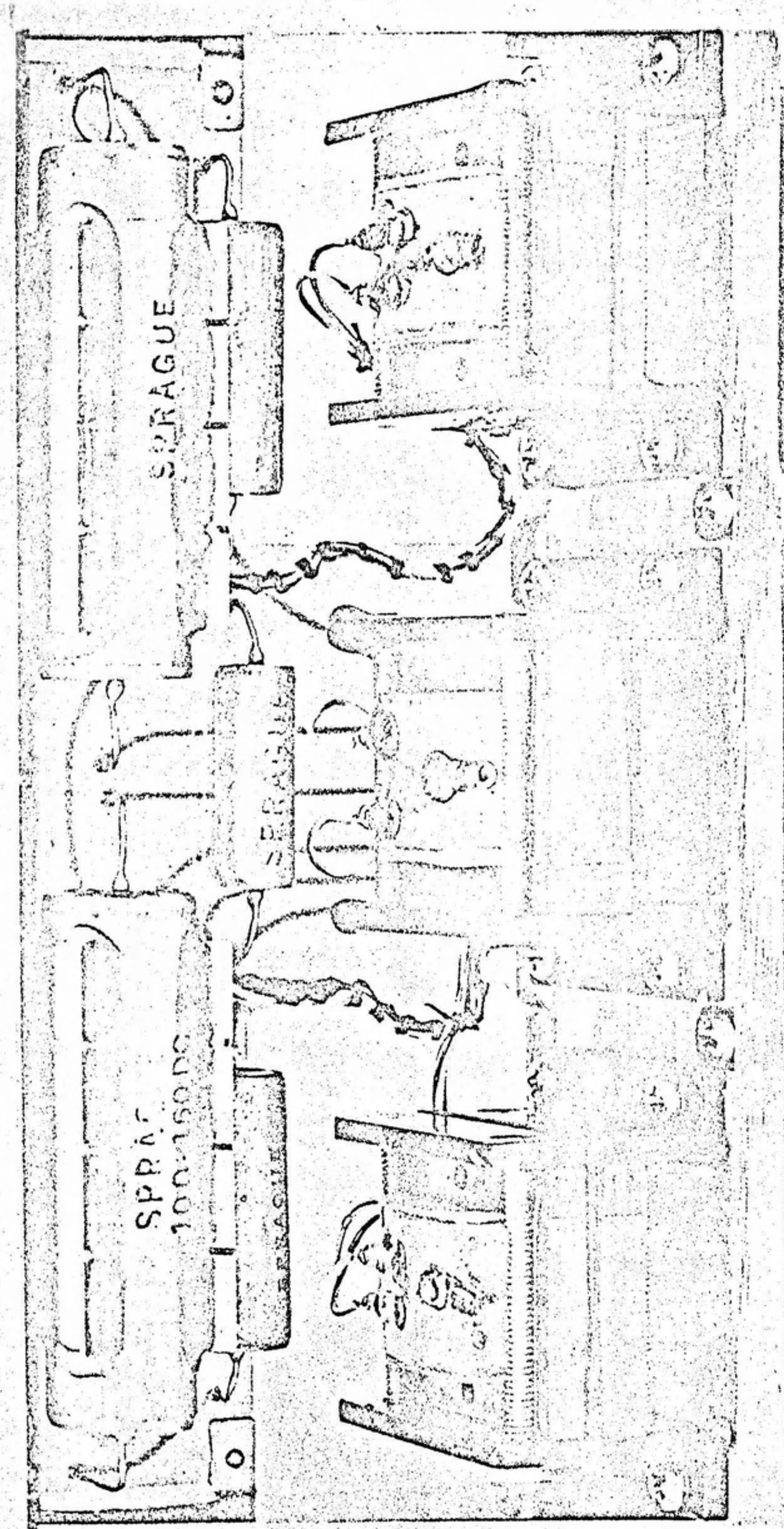
CONTROL ATTENUATOR TIMER
FIGURE 9



FLIGHT CONTROL COMPUTER
FIGURE 10

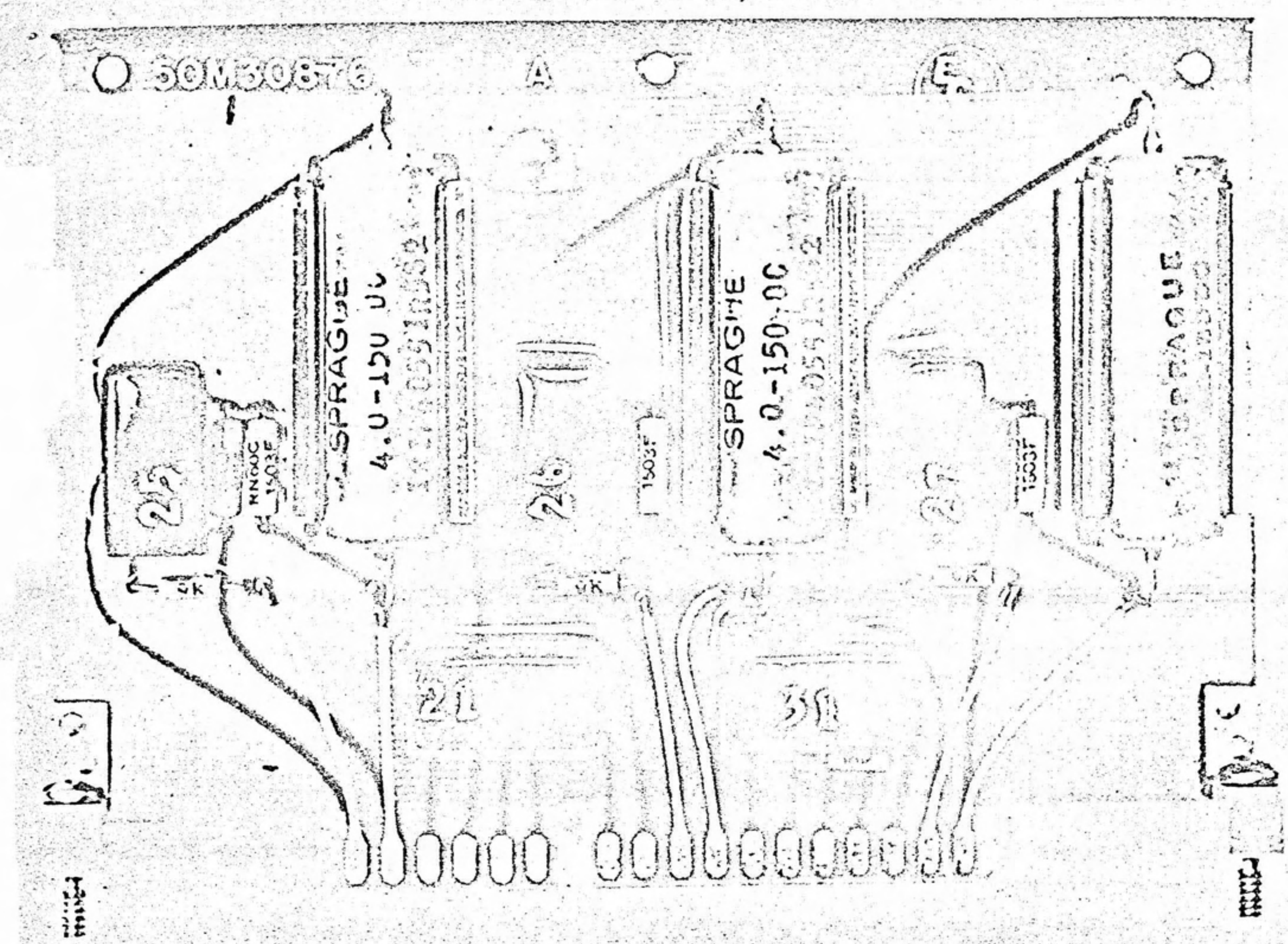


SERVO AMPLIFIER
FIGURE II



FILTER MODULE

FIGURE 12



LEAD NETWORK
FIGURE 13