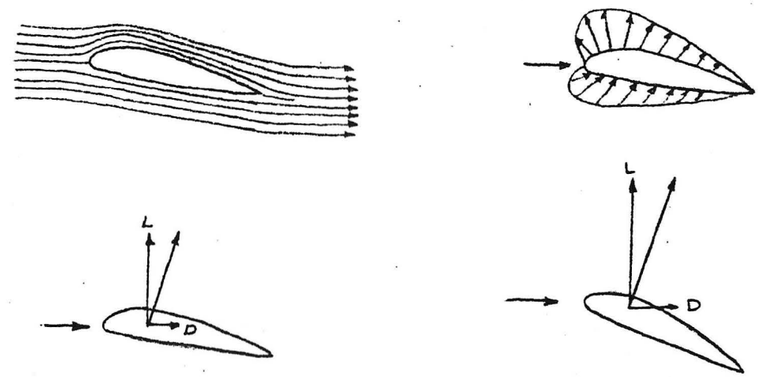
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TL. 13-58/06

BASIC AERODYNAMICS

A. DEVELOPMENT OF LIFT



Lift is developed by a wing acting on the relative wind.

1. Change in direction of the wind velocity vector by the lower surface of the wing produces a reaction force acting in an upward direction.

2. Increase of velocity of the air over the upper surface produces a low pressure area on this surface. Since the pressure acting on the lower surface of the wing is higher than that acting on the upper surface, a force in the upward direction is produced.

Figure (1a) shows air flow around a wing cross section, and Figure (1b) shows the distribution of pressure resulting from this flow.

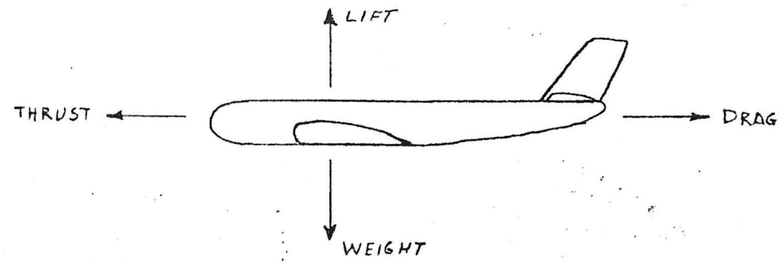
The resultant force due to all these pressures can be represented by one vector acting at the centre of lift of the wing. Figure (1c) shows this resultant force R and its two components, lift and drag.

Lift is always the component which is at right angles to the relative wind and drag is the component which acts in the same direction as the relative wind.

Figure (1d) shows that by increasing the angle of attack of the wing, the lift can be increased. However, drag will also increase when this is done.

A point to remember is that lift acts in the vertical direction only when the relative wind is in the horizontal direction. If the aircraft climbs or dives, lift does not act in the vertical direction.

B. AIRCRAFT IN LEVEL FLIGHT



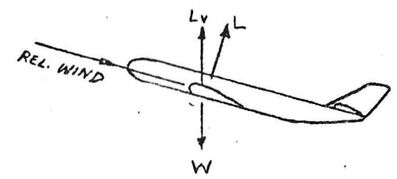
The above diagram illustrates the balance of forces acting on an aircraft in level flight.

If the aircraft is to maintain level flight, the force due to gravity must be balanced by the lift developed by the wing. If the aircraft is to continue its motion at a constant velocity, the thrust produced by the engine must balance the drag forces acting on the aircraft. Drag as used here includes not only the drag due to the horizontal component of lift, but all other frictional and aerodynamic drag effects.

If the pilot wishes to climb he can do so by either increasing his angle of attack or by increasing thrust or both. Increased thrust will be necessary if the climb angle is quite high.

C. CHANGE OF ALTITUDE

If the pilot wishes to increase his altitude, he must move the stick forward to move the elevator surfaces upward. This will move the tail down, and change the flight path of the aircraft. The pilot will then return the elevator surface to the neutral position.



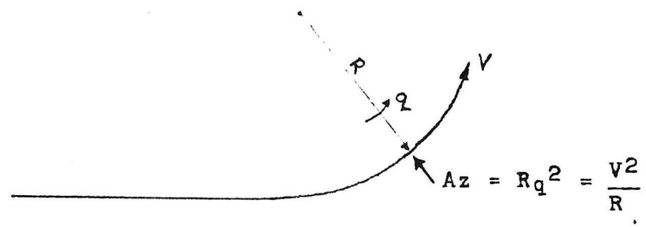
2.

As shown in diagram 2 above, the relative wind will no longer be horizontal. The lift developed by the wing will not act in a vertical direction. In order to continue the on flight path, the vertical component of lift must be equal to the weight of the aircraft. In order to increase lift to obtain the balance between the weight of the aircraft and the vertical component of lift the pilot must either increase his velocity or increase his angle of attack by trimming the aircraft or both if the climb angle is high.

In summary, gradual or minor changes of altitude may be made by moving control surfaces only, that is, changing angle of attack. High rates of altitude change require not only change of angle of attack, but engine power changes as well.

In automatic flight control work, the problem is to prevent altitude changes by stabilizing the aircraft attitude. This stabilizing action should require no changes of engine power.

D. NORMAL ACCELERATION

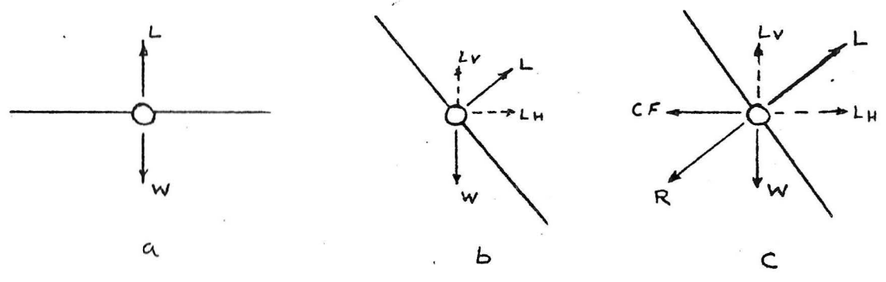


Normal acceleration is any acceleration which acts perpendicular to both the longitudinal and lateral axes of the aircraft.

If an aircraft changes its flight path as shown in the diagram above due to either pilot command or external air conditions, a normal acceleration will be experienced by the aircraft due to the centrepedal force acting on it. As shown by the equations, the normal acceleration Az is proportional to the square of the velocity of the aircraft. This indicates that in high speed aircraft, high normal accelerations, and consequently high structural stresses are set up in the wing during relatively small flight path changes. An automatic flight control system must then modify its stabilizing action so that high normal accelerations are not set up during corrective control action.

E. TURNS

Contrary to general opinion held by laymen, turns are not produced primarily by use of the rudder. In an ideal aircraft, no rudder would be required during a turn. As shown in the diagram below, turns are produced by the horizontal component of lift when the aircraft is in a bank.

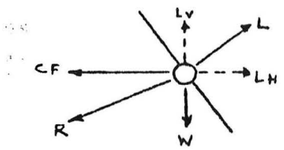


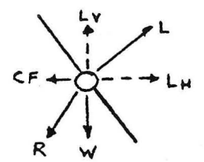
When the aircraft is flying straight and level (a) above, lift force balances weight and no forces are acting in the lateral direction. When the aircraft is in a bank, the lift vector, which is always perpendicular to the lateral axis, is inclined to the right. In (b) above, two components of this lift force are shown, one acting vertically the other horizontally. The horizontal component of lift will then start the aircraft turning. In (c) above the aircraft is shown in a coordinated turn, with all forces balanced. To do this, the lift must be increased by changing the angle of attack using up-elevator so that its vertical component will still balance gravity. The reaction to the horizontal component of lift will be a centrifugal force produced by the curved flight path of the aircraft. The resultant of the centrifugal force and the force of gravity will exactly balance lift and the turn will be coordinated.

F. SLIPPING AND SKIDDING

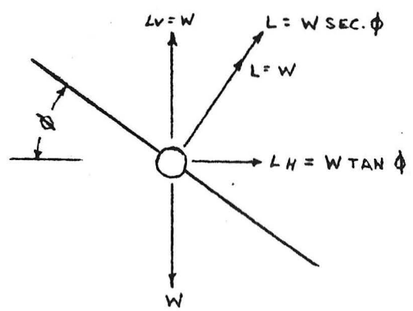
If during a turn, aiding rudder is added, the radius of the turn will decrease. Since centrifugal force is inversely proportional to the radius of the turn, it will increase. The increase of centrifugal force will cause the aircraft to skid outward.

If during a turn, opposing rudder is used, the radius of turning will be increased. This then reduces the centrifugal force, causing the aircraft to slip inward. Force diagrams of slipping and skidding turns are shown below.





Maintaining Altitude During A Turn

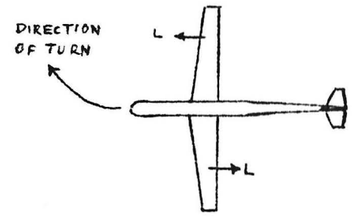


As shown in the diagram, the lift produced by the aircraft in a turn must be W sec. ϕ so that its vertical component will be W and balance the force of gravity. The increase in lift required during the turn will be:

$$W \sec \phi - W = W(\sec \phi - 1)$$

F. ADVERSE YAW DUE TO ROLL RATE

When an aircraft is rolling from wings level to some bank angle to start a turn, an adverse yaw (yaw in a direction opposite to the direction of the turn) may be set up due to the rolling rate of the aircraft. On the down going wing the relative wind will change its direction from directly rearward to rearward and upward. On the up going wing, the relative wind will change to rearward and downward. Since the lift developed by a wing is always perpendicular to the relative wind, the lift vector on the down going wing will be inclined forward, and the lift vector on the up going wing will be inclined rearward.



As shown in the diagram, the turning moment set up by the horizontal components of lift will cause a yawing action in a direction opposite to the direction of turn. The use of aiding rudder during turn entry can counteract this adverse yaw effect.

G. ADVERSE OR PROVERSE YAW DUE TO AILERON POSITION.

To attain a desired bank angle, the pilot must deflect his ailerons. On the upgoing wing, the aileron must be displaced downward. On the downgoing wing, the aileron is displaced upward. The wings then are no longer symmetrical. Drag effects on one wing will be different than the drag effects on the other. This may result in either an adverse yaw action or a proverse yaw (yaw in the same direction as the direction of turning). In addition, this yawing effect may change from an adverse to a proverse yaw or from a proverse to an adverse yaw as airspeed increases.

H. RADIUS OF TURN

As described above, the horizontal component of lift causes a turn to be set up. This lift component is then balanced by a centrifugal force. The relation between C.F. and radius is as follows:

$$CF = M \frac{V^2}{R}$$ where M is the mass of the aircraft and V is the velocity of the aircraft.

Radius of the turn is then equal to:

$$R = \frac{MV^2}{CF}$$

but $CF = L_H = W \tan \phi$ in a coordinated turn

$$\therefore \quad R = \frac{MV^2}{W \tan \phi}$$

$$M = \frac{W}{g}$$

$$\therefore \quad R = \frac{W}{g} \frac{V^2}{W \tan \phi} = \frac{V^2}{g \tan \phi}$$

This equation shows three things:

1. The radius of turn is independent of the weight of the aircraft.

2. The radius of turn will increase as the velocity of the aircraft increases.

3. The radius of turn will decrease as the bank angle increases.

Number 2 above can be interpreted to mean that the rate of heading change will decrease as the velocity of the aircraft increases.

I. DEFINITIONS OF TERMS

E - Elevation Angle: Angle between a true horizontal plane and the longitudinal axis of the aircraft.

θ - Pitch displacement: Angle between the longitudinal axis of the aircraft and a plane containing the lateral axis of the aircraft and a true horizontal line perpendicular to the lateral axis.

q - Pitch Rate: Pitch Angular Velocity

$\dot{q}$ - Rate of Change of Pitch Rate: Pitch Angular Acceleration

Φ - Bank Angle or Roll Angle: Angle between the lateral axis of the aircraft and a horizontal plane.

p - Roll Rate: Rolling Angular Velocity.

β - Crab Angle: Angle between the direction of the relative wind and the longitudinal axes of the aircraft measured in a plane containing the lateral and longitudinal axes of the aircraft.

r - Yaw Rate: Yawing Angular Velocity

$\dot{r}$ - Rate of Change of Yaw Rate: Yaw Angular Acceleration

Az - Normal Acceleration: Linear acceleration measured along a line perpendicular to both the lateral and longitudinal axes of the aircraft.

Ay - Lateral Acceleration: Linear acceleration measured along the lateral axis of the aircraft.

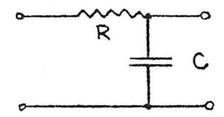
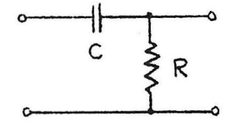
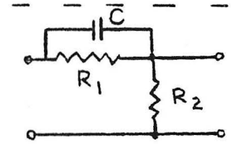
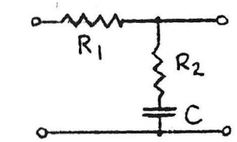
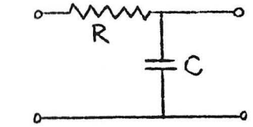
qc - Differential Pressure: Difference between Pitot and Static Pressure.

h - Altitude in Feet.

P_T - Ram pressure plus static pressure.

P_S - Static air pressure.

TRANSFER FUNCTIONS

$\frac{1}{1 + TS}$		$T = RC$	Low Pass
$\frac{TS}{1 + TS}$		$T = RC$	High Pass
$\frac{1 + T_1S}{1 + T_2S}$	$T_1 > T_2$ 	Lead $T_1 = R_1C$ $T_2 = \frac{(R_1R_2)}{R_1+R_2} C$	
$\frac{1 + T_1S}{1 + T_2S}$	$T_1 < T_2$ 	Lag $T_1 = R_2C$ $T_2 = (R_1 + R_2)C$	
$\frac{1}{TS}$	$T \gg 1$ 	Integrator $T = RC$	

The above diagram shows blocks containing transfer functions, and to the right of each, the equivalent electrical network which it represents. The purpose of the following discussion is to familiarize the student with the development of and the use of transfer functions and to make it possible for him to recognize the action of a section of a block diagram by identifying its transfer functions with one of the networks above.

All of the above transfer functions contain the term 'S' which is known as the Laplace Operator. The transfer functions themselves, are derived by using what is known as the Laplace transform of a function.

THE LAPLACE TRANSFORM

The response of a dynamic system such as an aircraft in flight to pilot control action or external air conditions can be expressed approximately, by mathematical equations which are functions of time. Some of the terms which enter these equations are linear displacement, velocity and acceleration and angular displacement, velocity and acceleration. It can be seen then that these equations describing dynamic response will be complex high order differential equations. By solving these equations for many different control

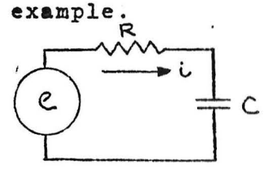
commands and air conditions, the aircraft designers can determine if the aircraft is basically stable or if it will require augmentation to improve its stability.

If the aircraft does require stability augmentation, it is obvious that the equations describing the response of the augmentation will also be differential equations of high order.

The manipulation of the aircraft and augmentations (Damper) differential equations is a tedious and complex job. By using the Laplace transform, differential equations can be reduced to algebraic form which make the equations much simpler to solve.

For the purposes of this discussion, we will not go into mathematical proofs of the validity of the Laplace transform method. We will merely state the definitions required and show the use in practical examples. It should be noted that the Laplace transform method has certain limitations and cannot be applied as a means of solving all differential equation problems. In our work here we will not encounter these limitations, however it should be recognized that limits exist.

To show the use of the Laplace transforms and to state its definitions we will use the simple R-C circuit shown in the diagram as an example.



The differential equations describing this circuit is:

$$e = iR + \frac{1}{C}\int i\,dt$$

It is important to note that e, the driving function can be a sine wave, a ramp function, a square wave or any time function. As a result, i will also be some time function. We can rewrite the equation as:

$$f_1(t) = f_2(t)R + \frac{1}{C}\int f_2(t)dt$$

where $f_1(t)$ is the function of time e

and $f_2(t)$ is function of time i

Laplace transform of a function of time.

The symbol "$\mathcal{L}$" indicates that a Laplace transform has been taken.

$$\mathcal{L}\left[f(t)\right] = \int_0^\infty \left[f(t)\right] e^{-st}dt = F(S)$$

This is the basic definition. The Laplace transform of a function of time is the integral of that function multiplied by e to the power minus st. This whole operation can be represented by F(S). Using this as a fasic definition, several other relations can be stated.

1. Laplace transform of a derivative

$$\mathcal{L}\left[\frac{df(t)}{dt}\right] = \int_0^\infty \left[\frac{df(t)}{dt}\right] e^{-st}dt = SF(S)$$

2. Laplace transform of a second derivative

$$\mathcal{L}\left[\frac{d^2f(t)}{dt^2}\right] = \int_0^\infty \frac{d^2}{dt^2}\left[f(t)\right] e^{-st}dt = S^2F(S)$$

3. Laplace transform of an integral

$$\mathcal{L}\left[\int f(t)dt\right] = \int_0^\infty \left[\int f(t)dt\right] e^{-st}dt = \frac{F(S)}{S}$$

Going back to our original equation,

$$e = iR + \frac{1}{C}\int idt$$

Let $\mathcal{L}(e) = E(S)$

and $\mathcal{L}(i) = I(S)$

We will drop the subscript (s) for simplicity

Using the definitions above

$$\mathcal{L}(e) = \mathcal{L}\left[i\ R\right] + \mathcal{L}\left[\frac{1}{C}\int idt\right]$$

$$E = IR + \frac{I}{S}\left(\frac{1}{C}\right)$$

$$E = I\left(R + \frac{1}{CS}\right)$$

If we wish to find the relation between voltage and current in the circuit we must find $\frac{i}{e}$. Using the transformed equations above

$$\frac{I}{E} = \frac{1}{R + \frac{1}{CS}} = \frac{CS}{1 + RCS} = \frac{1}{R}\left(\frac{RCS}{1 + RCS}\right) = \frac{1}{R}\left(\frac{TS}{1 + TS}\right)$$

where T = RC = time constant

We now have the transfer functions of this network for the relation of i to e.

In order to use the transfer functions we must rewrite it.

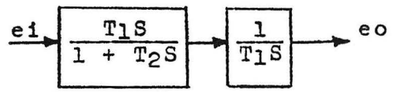
$$\frac{I}{E} = \frac{I}{R}\left(\frac{TS}{1 + TS}\right)$$

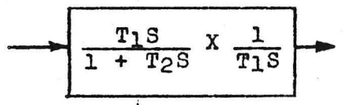
$$I = \left(\frac{TS}{1 + TS}\right)\frac{E}{R}$$

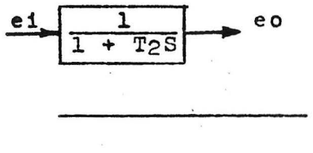
If we wish to find the current in the circuit due to a ramp voltage applied, it is only necessary to find the Laplace transform of a ramp input from tables, multiply it by the transfer function, then find the inverse transform of this product from tables.

Since most common time functions are tabulated with their Laplace transforms, it becomes relatively simple to solve differential equations of higher orders using this method.

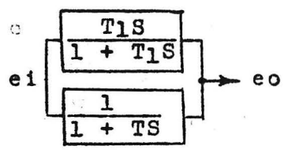
In our work, it will not be necessary to solve differential equations, but we will be required to manipulate transfer functions.

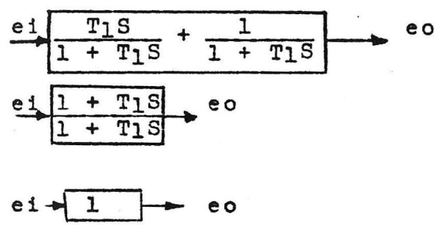






In the diagram, two blocks connected in series have two transfer functions acting on the input ei. By simply multiplying the transfer functions we can obtain a new transfer function which represents the effect of both the blocks. To find the output eo when an input ei is applied, it is only necessary to do one set of Laplace transformations rather than two as would be required in the first case.

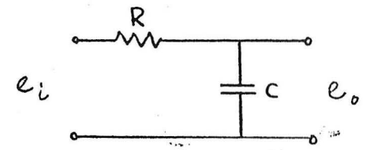




The above diagram of parallel connected blocks is self explanatory The addition of transfer functions is used in this case.

DEVELOPMENT OF TRANSFER FUNCTIONS

1. Low Pass Filter



The differential equation for this circuit is:

$$ei = iR + \frac{1}{C}\int idt$$

Let $\mathcal{L}e_i = E_1$ and $\mathcal{L}i = I$

$$E_i = IR + \frac{I}{CS} = I(R + \frac{1}{CS})$$

$$e_o = \frac{1}{C}\int i\,dt$$

Let $\mathcal{L}e_o = E_o = \frac{I}{CS}$

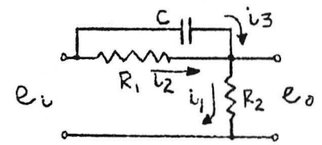
$$\frac{E_o}{E_i} = \frac{\frac{I}{CS}}{I(R+\frac{1}{CS})} = \frac{\frac{1}{CS}}{\frac{1+RCS}{CS}} = \frac{1}{1+RCS}$$

if T = RC

$$\boxed{\frac{E_o}{E_i} = \frac{1}{1 + TS}}$$

if TS is largely compared to 1 $\frac{E_o}{E_i} = \frac{1}{TS}$ an integrator

2. Lead Network



The differential equation describing this network is:

$$e_i = \frac{1}{C}\int i_3 dt + i_1 R_2$$

$$i_1 = i_2 + i_3$$

$$i_2 = \frac{\frac{1}{C}\int i_3 dt}{R_1} = \frac{1}{R_1 C}\int i_3 dt$$

$$\therefore \quad i_1 = i_3 + \frac{1}{R_1 C}\int i_3 dt$$

$$\text{and} \quad e_i = \frac{1}{C}\int i_3 dt + i_3 R_2 + \frac{R_2}{R_1 C}\int i_3 dt$$

$$= \int i_3 dt\,(\frac{1}{C} + \frac{R_2}{R_1 C}) + i_3 R_2$$

$$e_o = i_1 R_2 = i_3 R_2 + \frac{R_2}{R_1 C}\int i_3 dt$$

Let $\mathcal{L}e_i = E_1$, $\mathcal{L}i_3 = I$, $\mathcal{L}e_o = E_o$

$$Ei = \frac{I}{S}\left(\frac{1}{C} + \frac{R_2}{R_1C}\right) + IR_2 = I\left(\frac{R_1+R_2 + R_2R_1CS}{R_1CS}\right)$$

$$Eo = IR_2 + \frac{R_2}{R_1C}\frac{I}{S} = I\left(\frac{R_2R_1CS+R_2}{R_1CS}\right)$$

$$\frac{Eo}{Ei} = \frac{I\left(\frac{R_1R_2CS+R_2}{R_1CS}\right)}{I\left(\frac{R_1+R_2+R_1R_2CS}{R_1CS}\right)} = \frac{R_2+ R_1R_2CS}{(R_1+R_2)+R_1R_2CS}$$

$$\frac{Eo}{Ei} = \left(\frac{R_2}{R_1+R_2}\right)\frac{1 + R_1CS}{1 + \frac{R_1R_2}{R_1+R_2}CS}$$

let $T_1 = R_1C$
and $T_2 = \frac{R_1R_2}{R_1+R_2}C$ and $\frac{R_2}{R_1+R_2} = A$

$$\frac{Eo}{Ei} = A\left(\frac{1 + T_1S}{1 + T_2S}\right) \text{ where } T_1 > T_2$$

Usually, the gain factor A is omitted when writing the transfer function of a lead network, but if it is included, the transfer functions would be written:

$$\frac{1 + T_1S}{K + T_2S} \text{ where } T_1 = R_1C$$

$$T_2 = \frac{R_1R_2}{R_1R_2}KC$$

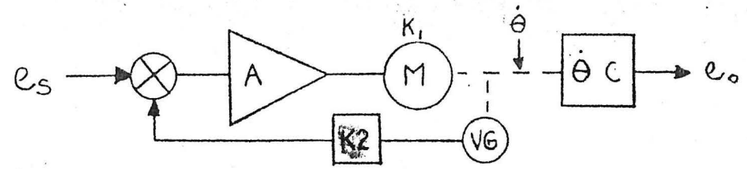
$$K = \frac{1}{A}$$

It is left to the student to develop the transfer function of

1. Lag Network
2. High Pass Filter

It is important to note that a transfer function contained in a block will indicate only the type of action that will be performed on the signal. The transfer function of a high pass filter does not mean that a resistor and capacitor will be found connected in series in the actual circuit. However, the mechanization of the circuit will produce the effect of a high pass filter. In the system which we will study, signal information is produced as amplitude modulation on a 400 cycle carrier frequency. It is obvious that such a signal applied to a series R-C circuit would <u>not</u> produce high pass action on the signal information carried as modulation.

Mechanization of Transfer Functions in a Carrier Type System



1. Integration

In the above diagram, an AC carrier type signal is applied to an AC motor through an amplifier. The motor speed $\dot{\theta}$ will be proportional to the amplitude of the input signal. If the output shaft drives a potentiometer wiper or other signal pickoff, the output signal e_o will be proportional to the angle through which the shaft has driven. In the diagram above, a velocity generator has been added, feeding its voltage back to the input.

K_1 is the scale factor of the motor with units of rad/sec/volt
K_2 is the scale factor of the velocity generator with units of volts/rad/sec.

The equation for the system is:

$$(e_s - e_f)\, AK_1 = \dot{\theta}$$

but $e_f = K_2\dot{\theta}$

then $(e_s - K_2\dot{\theta})AK_1 = \dot{\theta}$

$$\theta = \frac{d\theta}{dt}$$

$$(e_s - K_2 \frac{d\theta}{dt})A_1K_1 = \frac{d\theta}{dt}$$

$$e_sAK_1 = \frac{d\theta}{dt}(1 + AK_1K_2)$$

let $\mathcal{L}e_s = E_s$ and $\mathcal{L}\theta = \theta$

$$E_sAK_1 = S\theta(1 + AK_1K_2)$$

$\frac{\theta}{E_s}$ is the transfer function of this system

$$\frac{\theta}{E_s} = \frac{AK_1}{S(1 + AK_1K_2)}$$

$$= \frac{1}{\frac{(1 + AK_1K_2)S}{AK_1}}$$

$$= \frac{1}{(\frac{1}{AK_1} + K_2)S}$$

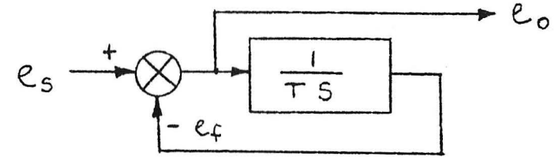
let $(\frac{1}{AK_1} + K_2) = T$

then $\frac{\theta}{E_s} = \frac{1}{TS}$, an integrator

Since $\frac{1}{AK1}$ will usually be small compared to K2, T ≐ K2

Thus it can be seen that the time constant of the integrator can be varied simply by varying the amount of velocity generator feedback.

2. High Pass Filter



If an integrator is connected in a circuit as shown above it will produce the action of a high pass filter.

Let $\mathcal{L}e_s = E_s$, $\mathcal{L}e_o = E_o$, $\mathcal{L}e_f = E_F$

We wish to find $\frac{E_o}{E_s}$

$$e_s - e_f = e_o$$

$$E_S - E_F = E_O$$

but $E_F = E_O \left(\frac{1}{TS}\right)$

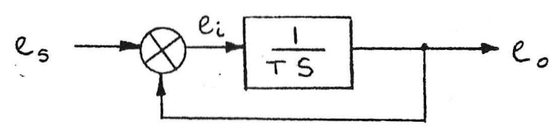
$$E_s - \frac{E_o}{TS} = E_o$$

$$E_s = E_o\left(1 + \frac{1}{TS}\right) = E_o\frac{(1 + TS)}{TS}$$

$$\boxed{\frac{E_o}{E_s} = \frac{TS}{1 + TS}}$$

This is the transfer function of a high pass filter.

3. Low Pass Filter



An integrator connected as shown will produce the action of a low pass filter. It is left to the student to derive the transfer function $\frac{E_o}{E_s}$ for this network.

AIRCRAFT CONTROLS AND HYDRAULICS

A. General

1. During both manual and automatic flight, the aircraft control surfaces are moved by hydraulic jacks. During manual flight, the control valves which port hydraulic fluid to the jacks are actuated through direct mechanical linkages from the pilot's controls. During automatic flight these control valves are actuated by small hydraulic rams which are electrically controlled by the Honeywell damper system. These small rams and their associated electrical controls are known as hydraulic servos.

2. Two types of servos are used during damper operation:

   (a) Differential servos can move the control surfaces without causing reflected motion of the pilot's controls.

   (b) Parallel servos replace the action of the pilot in moving the cockpit controls.

   In general it can be stated that the differential servos actuate in response to damper signals which augment the aircraft stability, and parallel servos actuate in response to damper command signals which produce a change of aircraft attitude. The distinction is not as clear cut as stated above, since command signals will produce some differential servo movement and damping signals will produce some parallel servo movement under certain conditions.

3. Hydraulic jacks and hydraulic servos are used rather than electrical actuators because of their faster response, smaller size and lower weight. System hydraulic pressure has a nominal value of 4000psi. Two sources of hydraulic pressure designated A and B are available.

B. Rudder Control Systems

1. Diagram one shows the rudder control system. It should be noted that no parallel servo is used in the rudder control system. This indicates that damper action on the rudder produces improved stability only. No provision to produce rudder commands during damper operation is made.

2. Typical Control Action (Manual)

   If the pilot produces a force on the rudder pedals in the direction shown, the same force will be applied to the front quadrant and through control cables to the rear quadrant. The force applied to the rear quadrant will be transmitted to the hydraulic jack linkages at point E. Initially, point A on the jack shaft is fixed, and the linkage will pivot about A so that B moves in the direction of the arrow. During manual flight,

the differential servo is spring loaded to its centre position, so that point C will be fixed. The linkage will pivot about point C so that D moves in the direction of the arrow. When D moves (.05 inch max.) the control valve ports hydraulic fluid to the main ram (or jack) so that the ram shaft will extend and begin to move the rudder. As the ram extends, point A will move in a direction to return point B to its original position by pivoting the linkage about E. If the pilot is maintaining pressure on the rudder pedals however, point B cannot return, the control valve will remain open, and the jack will continue to extend. As the ram extends, A continues to move and the linkage pivots about point B, moving E forward and permitting the rudder pedals, to move. When the pilot releases pressure on the pedals, movement of point A pivots the linkage at point E and returns point B to its original position, thus closing the control valve. If no wind loading exists on the rudder, (as would be the case during a ground check out of the rudder action) the ram will remain extended, the control valve will be closed, and the rudder pedals will remain deflected. It is important to remember that a control valve movement of only 0.05 inch will port maximum hydraulic flow to the main ram. This means that the movement of rudder pedals required to open the control valve will be so small that it will not be noticed by the pilot. Rudder pedal movement will be produced when the main ram begins to move.

3. Damper Control Action

When the damper system is engaged, the rudder control diagram will remain as shown in diagram one, however, the differential servo will no longer be spring loaded to centre. The differential servo ram will extend and retract in response to electrical signals from the damper. If the differential servo attempts to extend to the left in response to a damper signal, a force will be applied to point C. The linkage will pivot about point F so that point D will move in the direction of the arrow and open the control valve. The main ram will begin to move, and point A will pivot the linkage about point E so that B will move in the same direction as the servo ram. Movement of point B in this direction will tend to remove the force applied by the differential servo from the control valve. If the differential servo continues to extend, however, the force will not be removed from the control valve and the main ram will continue to extend. When movement of the differential servo ram ceases, movement of B to the left will remove the force applied to the control valve, the valve will close, and main ram travel will cease. During all the above action, point E has not moved. This indicates that there can be no movement of the rudder pedals due to differential servo action.

It will still be possible for the pilot to deflect the rudder when the damper is engaged by the same method as described above

system, no movement of the control stick will result due to operation of the differential servos.

When the damper system is engaged, hydraulic fluid will be ported to the parallel servo by its Moog valve in response to electrical command signals from the damper. Since the parallel servo applies a force to the front quadrant, the control action produced will be identical to the action produced when the pilot applies a force to the front quadrant during manual flight.

4. Emergency System

If a power failure or malfunction causes the system to change over to the emergency damper, the rudder only will be controlled. The elevator differential servos and parallel servo will be inactive and in the condition described for manual flight. In the emergency condition, the pilot must control the elevators manually.

D. Aileron Control System

1. Diagram three illustrates the aileron control system. This diagram is basically the same as the elevator control diagram. Two differential servos, control valves and main rams are used, and a parallel servo acting on the front quadrant. Control action in both damper and manual flight is identical to that described in the description of the elevator control system. As in the elevator system, ailerons must be controlled manually during emergency operation of the damper.

E. The Moog Valve

1. The Moog valve, sometimes referred to as the hydraulic amplifier, is a low power device that has the job of determining how much of and in which direction the servo actuator should allow hydraulic force to actuate the main hydraulic ram.

In Diagram 4 we have a simplified cross section view of the Moog valve. In the CF-105, we have a hydraulic pressure of 4,000 psi applied at "P". This pressure encircles the Moog ram plunger at the dotted lines shown and is felt at the orifices noted. The size of these orifices determines the speed of response, or "gain", of the Moog valve.

When the currents through both coil windings of the Moog valve are exactly equal, the reed is "dead center". Thus the leakage at points A and B is exactly the same. This leakage, you will note, drops down and out the return drain after completely encircling the ram plunger at the dotted lines as indicated. Note also, the return drains are internally connected and ported to the drain line through one hole only, not two.

This leakage at A and B, allows the pressure at A and B to be about 300 to 500 psi. This pressure exists in the tube from the orifices previously noted to points A and B. Remember, on the up stream side of the orifices, the pressure is 4,000 psi but down stream of the orifices, it may be 300 to 500 psi due to the positioning of the "reed" and the leakage around it. The pressures on the down stream side of both orifices will virtually be the same.

Ordinarily, in a "dead center" type Moog valve, the tension of the springs on the left and right ends of the "Moog plunger" will hold the "Moog Plunger" dead center. This position is determined by the adjustment of the "tension screw". Therefore, with equal pressures on the down stream side of the "orifices", the position of the "Moog Plunger" should be center.

As long as the two coil currents are equal, the "reed" is centered, pressure at A is equal to B; therefore, the "Moog Plunger" is centered. Now if the damper feeds an unbalanced current to the Moog coils of approximately .5ma or more, the "reed" will be moved off center. --- Let us assume that coil A has the larger current. The "reed" will be deflected "off center" towards A. The pressure at orifice "A" will now build up, while the pressure at orifice "B" will reduce. This means the pressure on the Moog plunger will increase on the left but decrease on the right. Therefore, the "Moog Plunger" will move to the right.

As the "Moog Plunger" moves to the right, the 4000 psi is ported through the "left cavity" of the plunger out the "left" port to the servo actuator. At the same time, the "right" port from the servo actuator is vented through the "right cavity" of the plunger out to the "drain".

This action described will continue until such time that the input currents to the Moog coils balance. When the currents balance, the "reed" will center itself and the Moog plunger will center itself by action of the left and right tension springs.

F. Parallel Servos

1. Diagram DMG55N shows the complete electrical and hydraulic schematic of the pitch parallel servo. The diagram includes a Moog valve, described above, an engage valve which removes hydraulic pressure from the servo when the damper is disengaged, the parallel servo ram, and the several additional electrical and mechanical components. A description of each item follows.

2. Auto Trim, Pressure Trim and Anti-Engage

The output of the Moog valve that passes eventually to the servo actuator must first pass the trim actuator. Any unbalance of hydraulic pressures between the left and right ports, will cause the trim actuator to open the anti-engage switch, close the trim switch and displace the pressure trim sensor.

(NOTE: Prior to engaging the damper the airplane trim switch is not used.)

The pressure trim sensor, while displaced acts to balance the input voltages to the servo amplifier (damper disengaged). Once the input to the servo amplifier falls below the threshold of that amplifier, the Moog currents will be balanced. The reed in the Moog valve will be positioned such that the Moog plunger will be centered. Therefore, the pressures at the left and right ports will be equal. The trim actuator will center, the trim switch will open and the anti-engage switch will close. With the anti-engage switch closed, it is now possible to engage the Damper.

NOTE: With equal pressures at the left and right ports, there will be no bump upon engagement of the Damper.

Once the Damper becomes engaged, the anti-engage switch and the pressure trim sensor are of no further use until such time that the Damper is disengaged. The airplane trim switch, however, is now used during the Damper engage period, to maintain the airplane trim (in the elevator channel only).

NOTE: This same basic servo system is used in the aileron channel as well as the elevator; however, none of the switches (anti-engage, and airplane trim) are used in the aileron channel after engaging.

Once the Damper is engaged, approximately 20 lbs. of force on the servo actuator is sufficient to cause the trim actuator to close the airplane trim switches. This in turn energizes the airplane trim mechanism.

3. Engage Solenoid and Engage Valve

   (i) Disengaged

   As seen in upper right of the diagram, the engage solenoid, is de-energized and therefore the engage valve is de-energized too. With the engage valve de-energized, the pressures from the Moog valve are terminated at the engage valve. The spring in the engage valve holds the engage valve plunger down. The pressure on the servo ram is ported to the return lines so that the servo ram is free to move with the front quadrant while dis-engaged.

   (ii) Engaged

   As shown the diagram, the engage solenoid when energized, allows hydraulic pressure to push the engage valve up. The engage valve thus allows the Moog output to position the servo actuator. Positioning the servo actuator in turn positions the control surface via the main hydraulic ram.

4. Force Limiters

The force limiters shown in the diagram limits the maximum force the servo actuator could ever apply to the hydraulic ram, but just as important is the fact that the pilot himself can, by exerting force on his control stick, overpower the servo actuator. The excess hydraulic forces developed are bled off through the force limiters. The force ratio between the pilot's control stick and the servo actuator is approximately 1 to 5. In other words, the force necessary to override the servo actuator is approximately 1/5th the force limit settings. The maximum force limits are adjustable from approximately 75 to 300 lbs.

5. Electrical Sensors

In the diagram, two electrical sensors are shown, one driven by the trim actuator and one driven by the parallel servo ram. The sensor driven by the ram produces an output signal when the ram moves away from the centre position. This signal is fed back to the damper to indicate that the movement commanded by a damper signal has been completed. The signal, called the servo balance signal has an amplitude proportional to the distance the ram has moved and a phase relative to a reference phase which will indicate the direction of motion.

The second sensor is known as a pressure trim sensor and its action was discussed above. The signal produced is similar to that of the servo balance, having amplitude and phasing to indicate distance and direction.

The aileron parallel servo is similar to the elevator parallel servo. As mentioned above, the airplane trim switches are not used.

G. Aileron and Elevator Differential Servos

1. The aileron and elevator differential servos are identical. Diagram C8751 shows the complete electrical and hydraulic schematic of one of these four servos. The system is shown in the engaged condition, however the servo ram centering mechanism is shown in the disengaged (centered) position. Two centering cans are held apart by a spring and engage shoulders on the ram shaft to hold the ram at a centre position.

   The system consists of a Moog valve, a timing control and engage valve, a hydraulic ram and centering device, a feedback transformer and a direction switch.

2. The Moog valve operating principles have been described above. By following an engaging sequence, the remainder of the system can be described.

When the unit is engaged, hydraulic fluid passes through the engage solenoid valve and on through a small orifice in the centering mechanism timing control applying limited hydraulic pressure to the Moog valve. Thus if a signal is acting on the Moog coils, it can supply reduced hydraulic flow to the servo ram. The high pressure fluid also passes down the centre of the floating piston in the centering mechanism timing control to the centering cams which begin to close together and release the piston ram. The floating piston in the timing control moves across to the right and stays there until the centering cams are held together. Once the centering cams are together, no more fluid flows through the centre orifice in the floating piston, so there is no longer a pressure drop across it. The floating piston then moves toward the left because the pressure acts on a larger area on the right hand side. When it reaches its full travel to the left, the hydraulic pressure can by-pass the series orifice applying full hydraulic pressure to the Moog valve input. This centering control timing is completed in one to four seconds.

3. Electrical Sensor

The feedback transformer is driven by the servo ram and produces an output signal when the ram is displaced from the centre position. This signal is fed back to the damper to indicate that the movement commanded by a damper signal has been completed. The signal, called the parallel servo balance signal, has an amplitude proportional to the distance the ram has moved, and a phase relative to a reference phase which will indicate the direction of motion.

4. Direction Switch

The ram direction indicating switch is a displacement actuated switch that is normally open if the servo ram is at neutral. Only if the servo ram is off neutral, does the switch close and it remains closed until such time as the servo ram is neutralized. The action of the switch is to prevent the automatic "G" limiter device from disengaging the pitch damper, if the differential servo is moving in the direction to oppose the "G" of the aircraft maneuver. That is to say, if the aircraft were pulling maximum allowable "G's" on a pull-up maneuver, the direction switch would not allow the automatic "G" limiter device to disengage the damper if the elevator differential servo is moving in the direction to produce down elevator. Thus you note down elevator movement is in a direction to oppose the "G's" produced by the pull-up maneuver. Conversely, had the differential servo moved in a direction to produce up-elevator, this movement would have increased the "G's" of the pull-up maneuver and under this condition, the direction switch on the differential servo would not prevent the automatic "G" limiter from disengaging the damper. The switch on the right elevator differential servo is used for this purpose.

5. Check Valves

When the differential servos disengage, the Moog valve plunger returns to centre. However, if disengagement occurs while the servo is displaced, action of the centering mechanism forcing the servo piston back to center would be prevented by the hydraulic fluid in the Moog valve lines to it. The check valves will allow the hydraulic fluid to flow out to the valve pressure input port which, in the disengaged condition is connected to a return line. The centering springs will be able to return the ram to the centre position.

6. Restrictors

Two restrictors can be seen in the pressure lines from the Moog valve to the servo ram. These restrictors prevent rapid flow of hydraulic fluid to the ram piston and thereby reduce the rate of response of the servo.

H. Rudder Dual Differential Servo

1. Diagram C8752 shows the electrical and hydraulic schematic of this servo. As mentioned in the description of the Rudder control system, this servo must continue to operate when the damper switches to the emergency mode of operation. To do this, much of the servo system is duplicated. Two engage valves, Moog valves and feedback transformers are used. Only one timing mechanism and only one centering mechanism is used, but two separate piston areas are available on the ram shaft. A shuttle valve and a by-pass valve have been added.

2. Operation

The system is shown in the normal condition in which pressure is supplied from the "B" hydraulic system. Operation of the timing mechanism and the servo in general will be identical to that described for the aileron and elevator differential servo. The student will notice that the normal system pressure is applied to the by-pass valve to insure that any fluid trapped in the emergency piston cavity of the servo will not lock the servo during normal operation.

When the system switches to emergency, the normal engage solenoid de-energizes removing pressure from the by-pass valve, the normal Moog valve and the timing control. At the same time, the emergency engage solenoid will actuate, porting emergency system 'A' pressure to the shuttle valve, the emergency Moog valve, and the by-pass valve. Emergency system pressure applied to the shuttle valve causes its floating piston to move quickly to the left end, thus re-applying pressure to the centering mechanism of the servo to prevent recentering action. Emergency system pressure also drives the by-pass valve to the down position, connecting the pressure lines from emergency Moog valve to the emergency ram piston. When the down position, any hydraulic fluid trapped in the normal piston will be by-passed, as was

emergency piston fluid during the normal operation, to prevent the servo from being locked. The servo will now respond to emergency damper signals applied to the emergency Moog valve coils.

Parallel Servo Frequency Response

In order to stiffen parallel servo with dry torque assist valves require reduction of response and phase.

(a)	Present	Proposed
3 cps	95°	33°
6 cps	55°	70°
Max response	1 dB	1.5 dB

Description and Operation of the XMG 62A-1 Dual Damper Servo for the Rudder Axis of the CF-105 Airplane

Description

The XMG 62A-1 Dual Differential Damper Servo is an electrically controlled, hydraulic powered, rectilinear motion servo-motor which is designed to mount on, receive its fluid power through, and differentially operate the rudder axis power control valve of the CF 105 airplane. It is designed to operate on MIL-O-5606 hydraulic fluid at a normal pressure of 4000 psi and provide a minimum output force of 300 pounds through a total stroke of 1.0 inch. The device is comprised of the following components:

1. Two (2) two-stage electro-hydraulic transfer valves. (Moog Valve Co.)
2. One (1) tandem hydraulic cylinder and ram, each half of which is controlled by one of the transfer valves.
3. Two (2) solenoid operated engage valves.
4. One (1) hydro-mechanical ram centering device.
5. One (1) pressure operated, spring biased shuttle valve.
6. One (1) pressure operated, spring biased dual bypass valve.
7. Two (2) reluctance type positional pickoffs.
8. Two (2) porous metal filters.

Operation

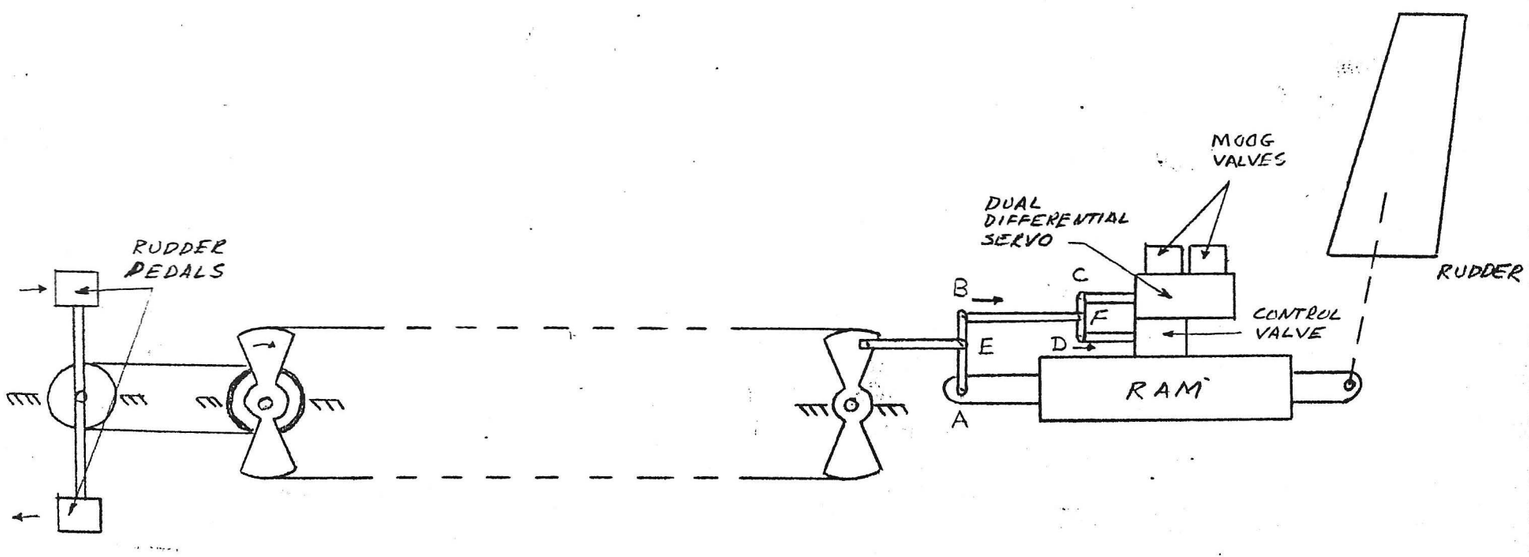
This device is driven by two separate amplifiers and is intended to operate as two independent servo-motors, only one of which will operate at any one time. Referring to drawing AD 443742, the operation is as follows:

Upon energizing the primary system solenoid engage valve, hydraulic fluid at 4000 psi is admitted through the filter and engage valve to the shuttle valve; one end of the dual bypass valve, and to the primary system transfer valve. From the shuttle valve fluid flows to the ram centering device which releases and allows the meterod flow from the primary system transfer valve to operate the ram in response to signals from the primary system amplifier. The dual bypass valve accomplishes two things simultaneously; it blocks the two sides of the primary ram cylinder from each other so that it will operate, and it also connects the two sides of the secondary ram cylinder together to prevent a hydraulic lock.

A positional signal to close the loop on the servo-motor is obtained from the secondary of the primary system feedback transformer and fed back to the primary system amplifier.

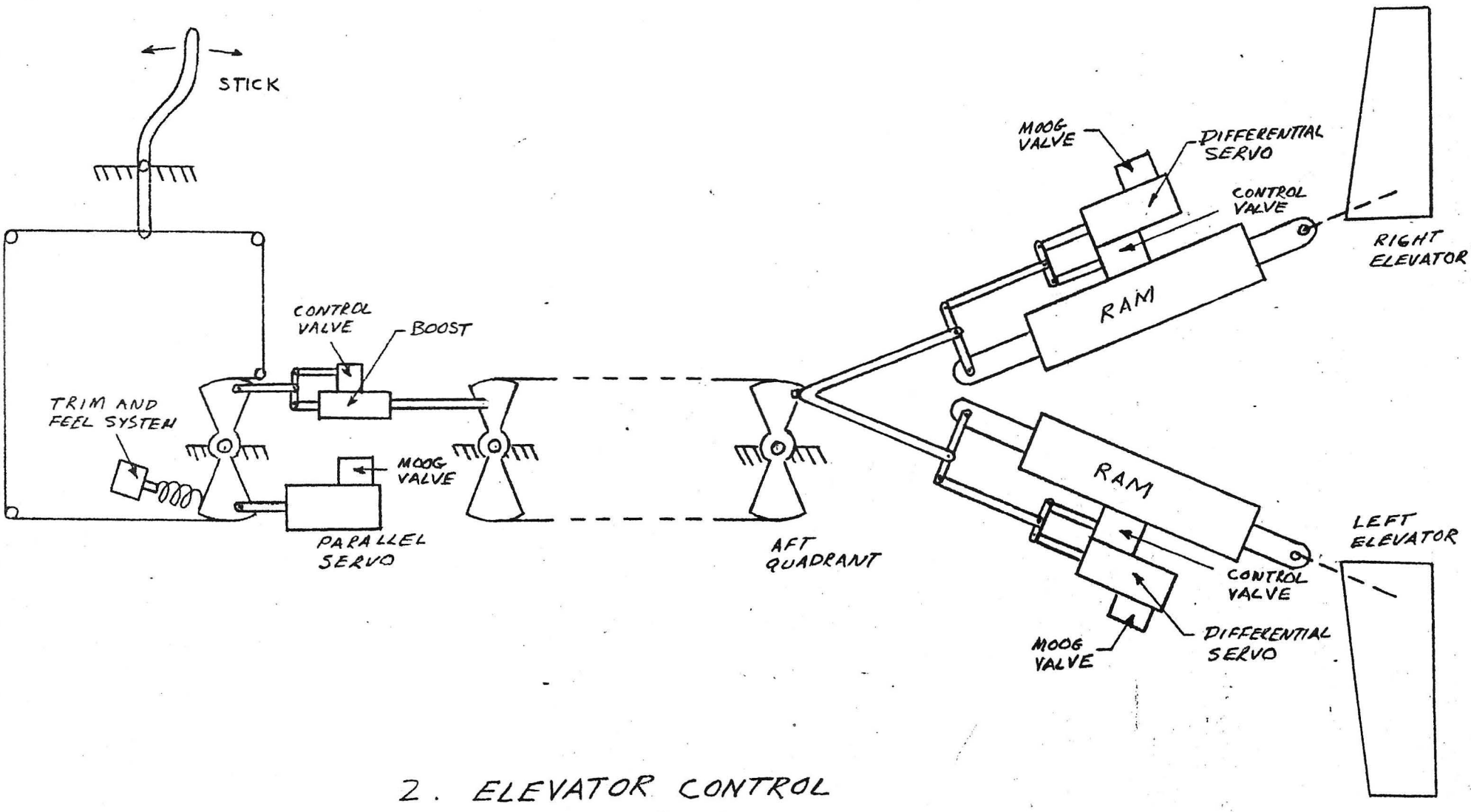
Operation of the secondary system is exactly the same as that for the primary system except that the shuttle valve and dual bypass valves are spring loaded to be normally in position to allow operation of the primary system. On disengagement of the primary system and engagement of the secondary system, the shuttle and bypass valves must move to new positions to allow operation of the secondary system.

Upon accidental or intentional loss of hydraulic pressure to either or both the primary and secondary hydraulic systems, the ram will center itself under the action of the centering spring and remain cetered unless overpowered by an external force in excess of 200 pounds.



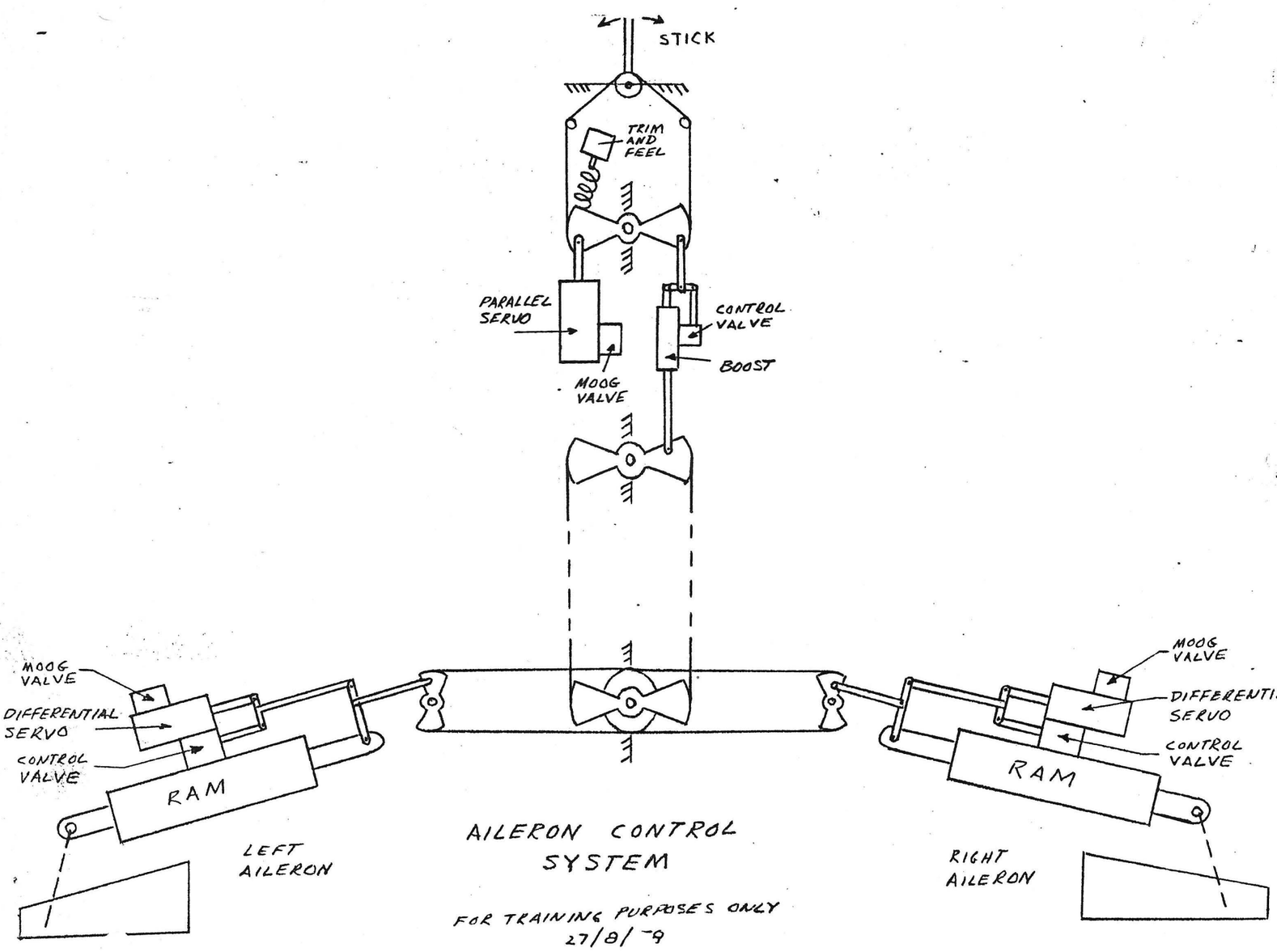
1. RUDDER CONTROL SYSTEM

FOR TRAINING PURPOSES ONLY
27/8/58



2. ELEVATOR CONTROL SYSTEM

FOR TRAINING PURPOSES ONLY
27/8/58

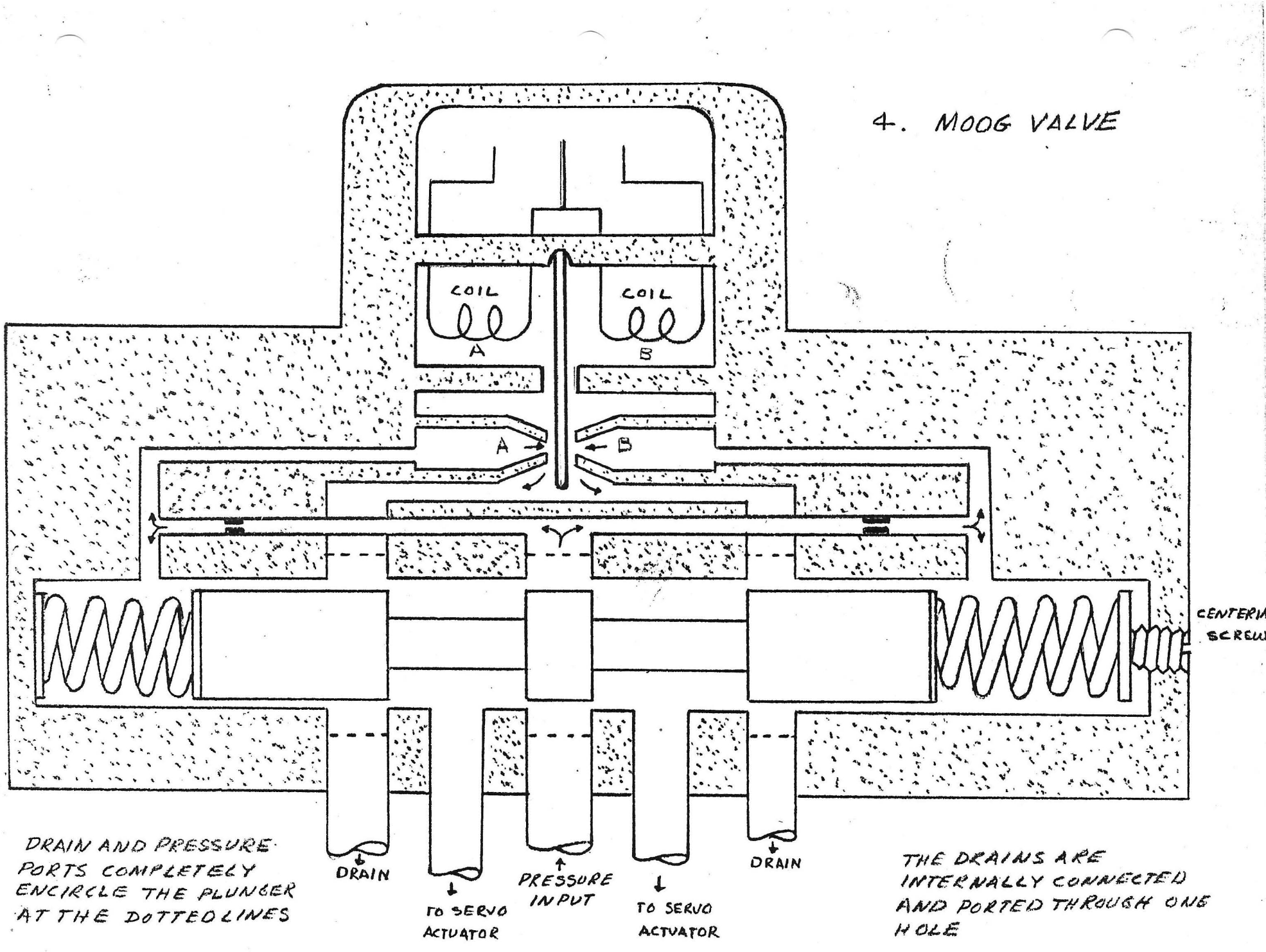


AILERON CONTROL SYSTEM

FOR TRAINING PURPOSES ONLY

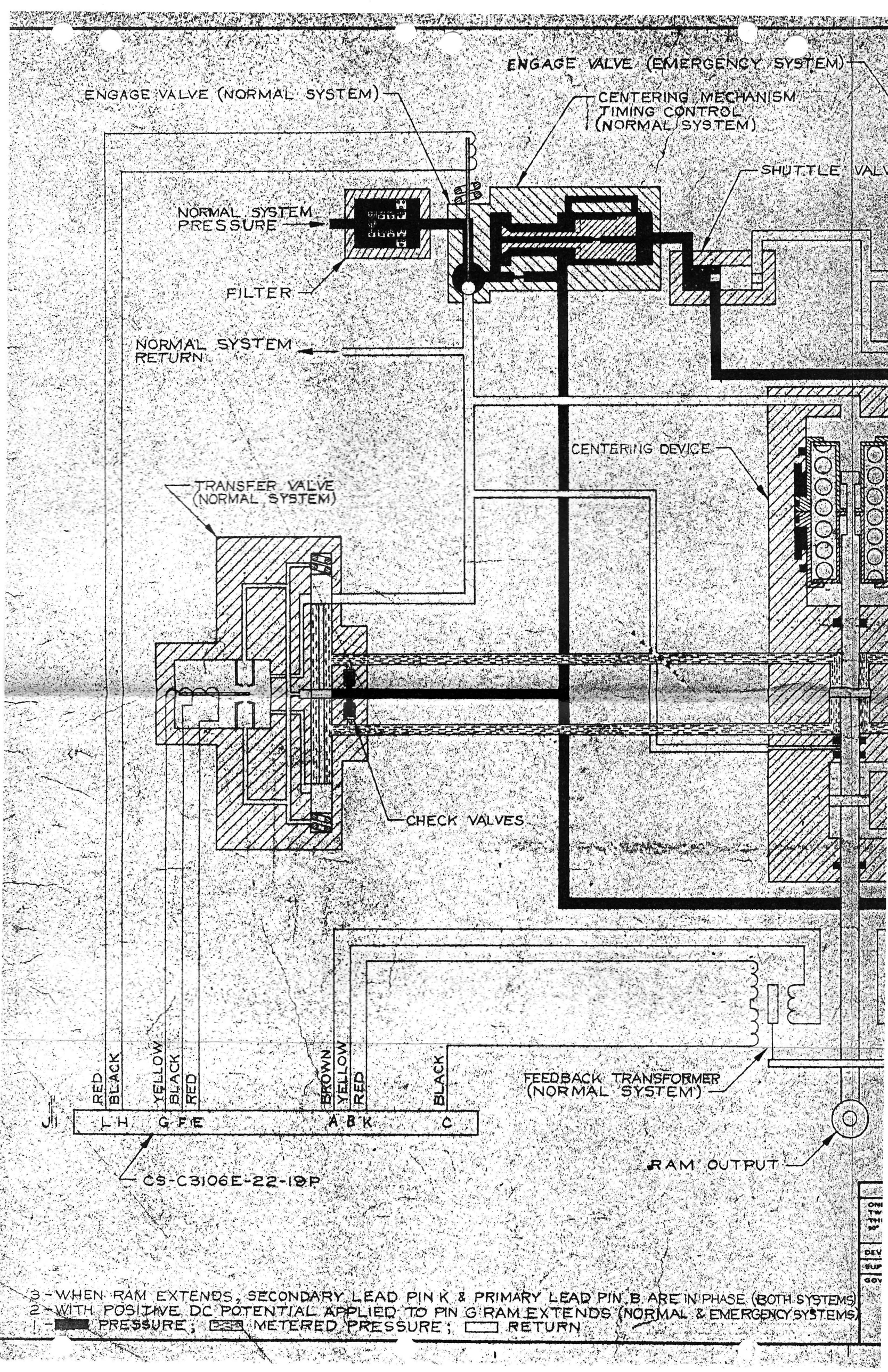
27/8/-9

4. MOOG VALVE



DRAIN AND PRESSURE PORTS COMPLETELY ENCIRCLE THE PLUNGER AT THE DOTTED LINES

THE DRAINS ARE INTERNALLY CONNECTED AND PORTED THROUGH ONE HOLE



3 - WHEN RAM EXTENDS, SECONDARY LEAD PIN K & PRIMARY LEAD PIN B ARE IN PHASE (BOTH SYSTEMS)
2 - WITH POSITIVE DC POTENTIAL APPLIED TO PIN G RAM EXTENDS (NORMAL & EMERGENCY SYSTEMS)
1 - ■ PRESSURE; ▨ METERED PRESSURE; □ RETURN

SUPPLEMENTARY INFORMATION		
FIRST USED D-MG62B-2		
NEXT ASSEMBLY		REFERENCE

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	FOR J1 & J2 PIN H HAD BROWN LEAD	1-27-58	DA-14-30

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.030
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±1°
DEV 01196-38		
SUPSD		
GOVT MATL SPEC		

DRAWN BY	Gonyea	12-10-57
DRAFTING √	Mason	8-20-57
MATL & FIN. √		
DESIGNER √	H. Ogren	9-11-57
DESIGN √	Mason	8-20-57
APPROVED	[illegible]	9-17-57
REF WH MATL (SPEC		
FINISH (SEE NOTE)		

SCHEMATIC DIAGRAM-
ELECTRO-HYDRAULIC
(RUDDER, DUAL DIFFERENTIAL)

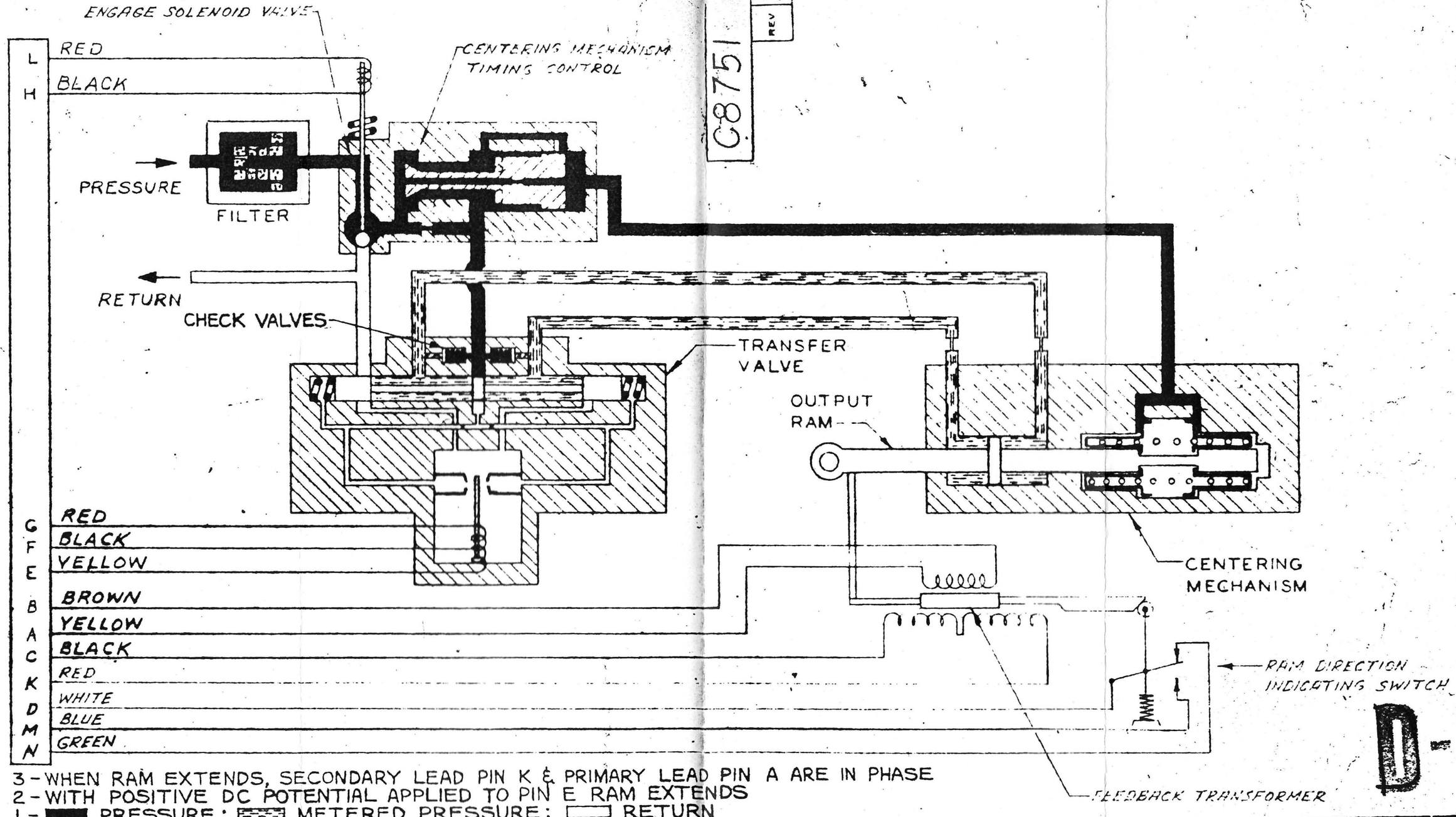
SCALE NONE | WT ACT. CALC LB

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

C8752

C8751 A

CS-C3106E-22-19P



3 - WHEN RAM EXTENDS, SECONDARY LEAD PIN K & PRIMARY LEAD PIN A ARE IN PHASE
2 - WITH POSITIVE DC POTENTIAL APPLIED TO PIN E RAM EXTENDS
1 - ■ PRESSURE; ▤ METERED PRESSURE; ▭ RETURN

REV	REVISIONS	DATE	CLASS	E.O
A	W COLOR FROM CONN PIN H WAS BROWN	1-27-58		9A-1425

NEXT ASSY | USED ON | MFG LICENSE

MAT'L | MIL SPEC ( ) | FINISH

GOVT FOR COMM'L ALTERNATE (IF DIFFERENCE)

TOLERANCE UNLESS NOTED
ONE PLACE (.1) ± .020 | MACHINED C'S
TWO PLACE (.01) ± .010
THREE PLACE (.001) ± .005 | BENT & FORMED C'S ± 1°

SIZE B

DRAWN BY	Dompier	12-10-57
TRACED		
INSTR. ENG.		
DESIGNED	K Green	9-11-57
APPROVED	[illegible]	12-10-57

SCALE ~ | SIZE | DEV. D1196-36
FIRST USED ON D-MG51D-2
BUY | WT. LBS

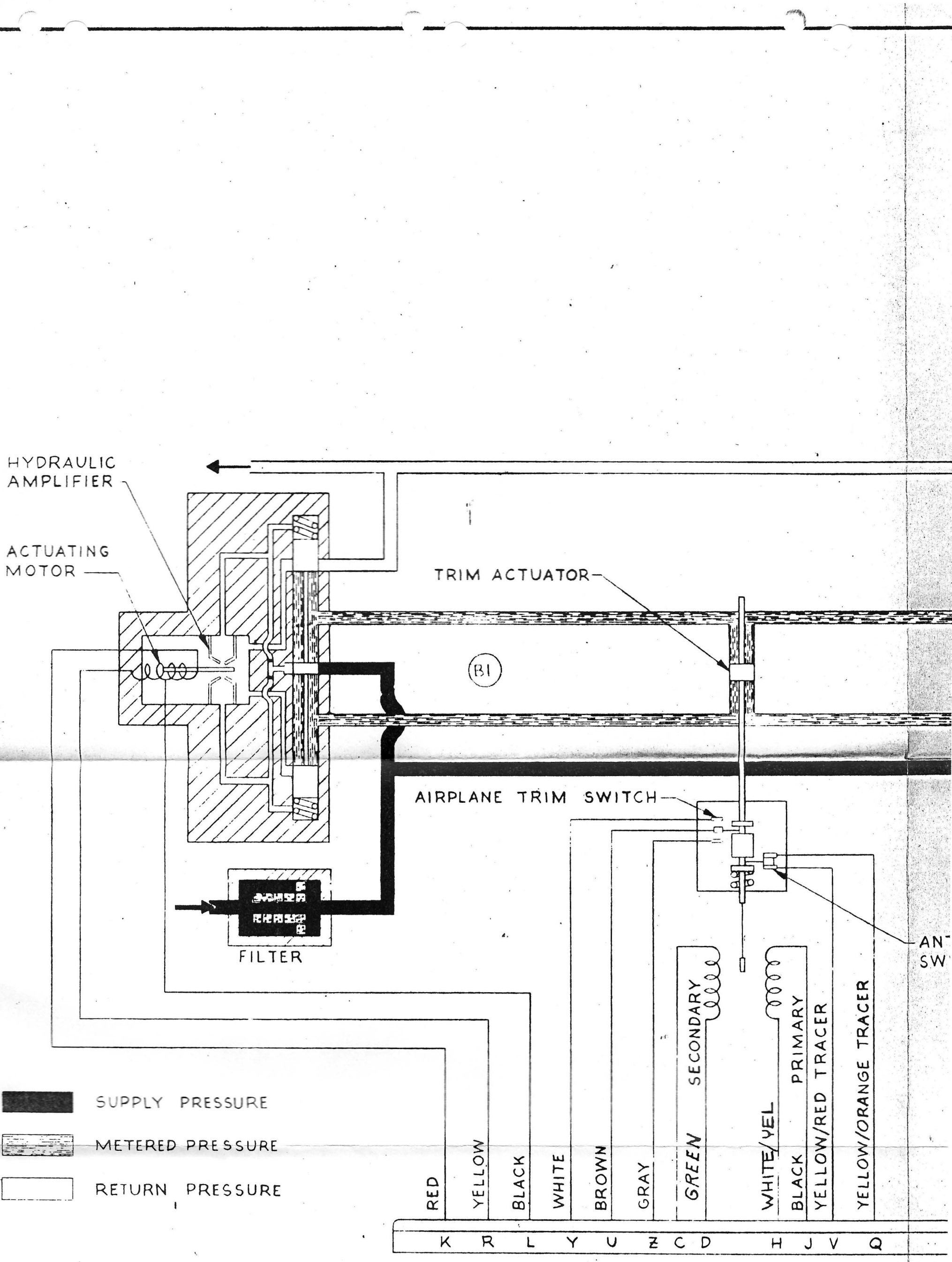
MINNEAPOLIS-HONEYWELL REGULATOR CO.
Aeronautical Division
Minneapolis, Minnesota

HYDRAULIC SCHEMATIC
DIFFERENTIAL ACTUATOR

C8751
NUMBER

D-

87-54



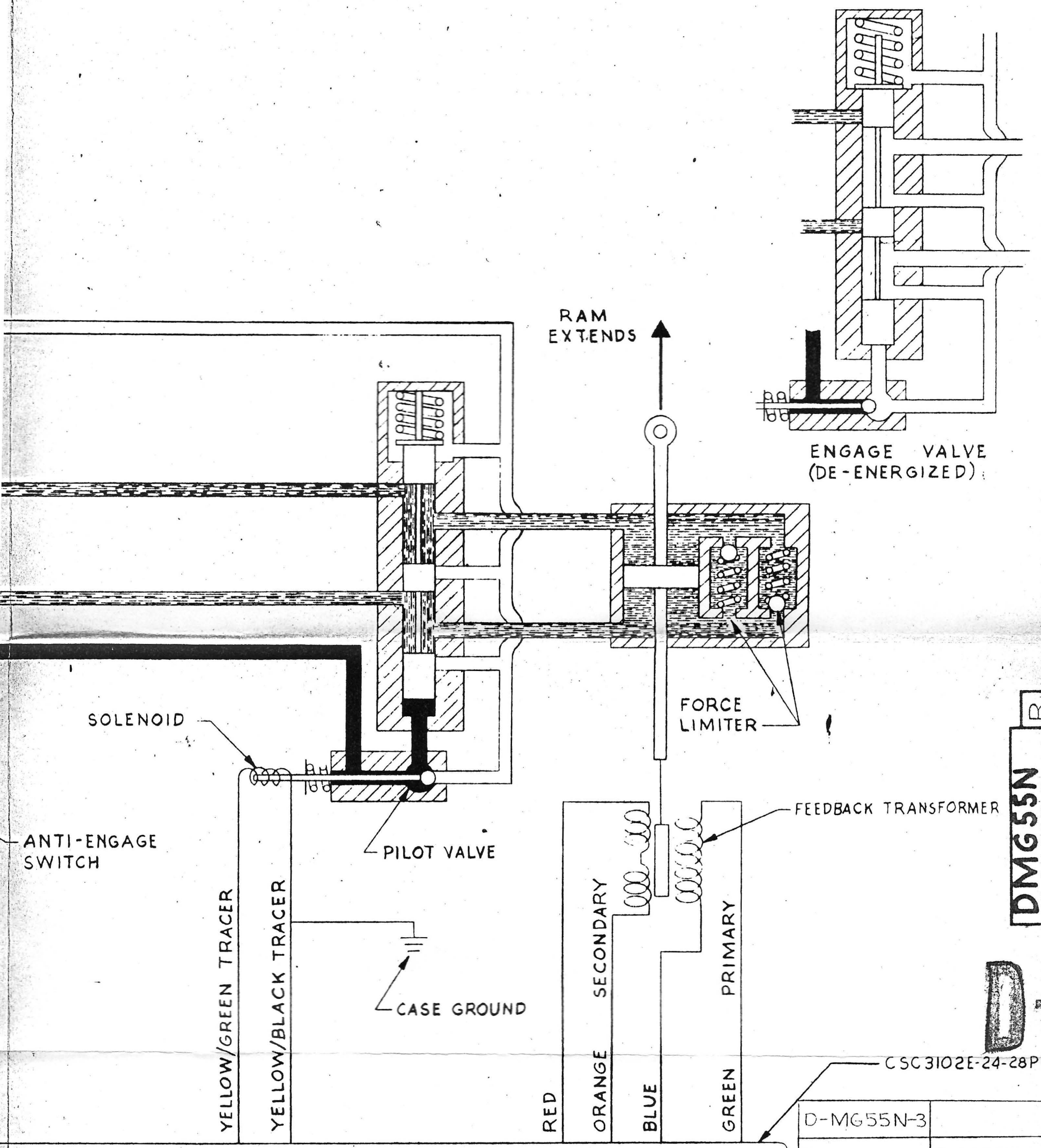
3- ON THE TRIM TRANSFORMER, SECONDARY LEAD (PIN D) AND PRIMARY LEAD (PIN H) ARE IN PHASE WHEN THE SIGNAL TO THE SERVO IS SUCH THAT THE MAIN RAM IS FORCED HARD OUT

2- ON THE MAIN FEEDBACK TRANSFORMER, SECONDARY LEAD (PIN A) AND PRIMARY LEAD (PIN F) ARE IN PHASE WITH THE RAM EXTENDED

1- POSITIVE POTENTIAL APPLIED TO PIN R CAUSES RAM TO MOVE OUT

SUPPLEMENTARY INFORMATION		
FIRST USED		
NEXT ASSEMBLY		REFERENCE

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	IN APPL. TAB. ADD -1 & -2; DWG NO. WAS -1	9-16-57	9A-1227
B	REM(I) ORIFICES (V RESTRICTORS); IN APPL TAB. ADD D-MG55N-3	1-13-58	9A-1392



D-MG55N-3	
D-MG55N-1	D-MG55N-2

THIS DRAWING APPLICABLE TO THE ABOVE UNITS

DMG55N B SHEET 3

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	± .020
TWO PLACE	(00)	± .010
THREE PLACE	(000)	± .005
90° FORMED ANGLES		

DEV 02278-01

SUPSD

GOVT MATL SPEC

DRAWN BY	Bolster	2-15-57
DRAFTING		
MATL & FIN		
DESIGNER	Engel	1-29-57
DESIGN		
APPROVED	[illegible]	4-24-57

REF MH MATL (SPEC)

FINISH (SEE NOTE)

ELECTRO-HYDRAULIC SCHEMATIC

SCALE NONE SIZE WT ACT. CALC LB

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY

AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

DMG55N
SHEET 3 OF 3

GYROS

General Considerations

The action of a spinning top has been a source of fascination and amazement to people of all ages and levels of education for so long a time that no historian can tell who carved out the first one. For centuries, the top was merely a toy. Its chief interest lies in its spinning motion, and in the almost paradoxical fact that those objects having shapes which are most stable when spinning are the very ones which are most unstable when not spinning. The "Tippetop" is one of the latest, and most ingenious of these toys. This top is a sphere with a short cylindrical stem. It is heavy at a point on the sphere opposite the stem, and when not spinning it rests with the stem up as any such well behaved body should. However, when it is spinning, it proceeds to turn itself over and spins on the stem with the heavy portion up. Explanation of this toy is not a trivial problem. W.A. Pliskin in the U.S., C.M.Braams, and N.M. Hugenholz in Belgium, and J.L. Synge in England have all written articles concerning it and disagree on the importance of frictional forces in its action.

During the nineteenth century, dynamicists were able to make quantitative studies of the motions of the top and predict its reactions to various external forces. During the early part of the twentieth century, practical applications of the top began to appear in Germany and in the United States. Probably the first important use of the gyroscope was its application as a north-seeking gyro-compass. Since that time, a multitude of new uses have appeared. Gyroscopes are now used not only to measure and indicate angular motions of all sorts, but also to control such motions in so many applications that it would probably be impossible to list all of them. A short list of the more important applications of the gyroscope would include, besides the gyrocompass, the following: attitude and angular rate indicators for aircraft, attitude and rate measuring devices for aircraft autopilots, measurement of angular rates in fire control problems, angular references in bombardment problems, stabilization of platforms for radar antennae in aircraft and on shipboard, stabilization of gun platforms in a variety of different vehicles, and in navigational computers.

A simple static mass will not exhibit gyroscopic action. The mass must be spinning about some axis. However, any mass which does spin fulfills the only requirements for gyroscopic action, so that the effect is widely apparent, both in man-made devices, and throughout nature in general. All spinning masses from the tiny electron to the earth itself, and, in fact all astronomical bodies, exhibit this effect.

Gyroscopic action may be described, in a qualitative way, as follows: Mount a spinning body so that it may rotate around any axis contained in a plane perpendicular to its spin axis, and with its spin axis initially in a vertical position. Now, apply a couple to the spin axis so that if the wheel were not spinning, the top part of the axis would rotate to the north. With the wheel spinning, however, the axis will <u>not</u> move in the northerly direction it is being pushed, but rather, will move east or west! The particular direction of motion depends upon the direction of wheel rotation. This phenomena is called precession.

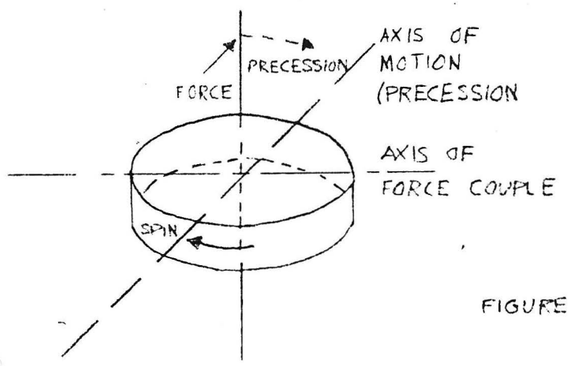
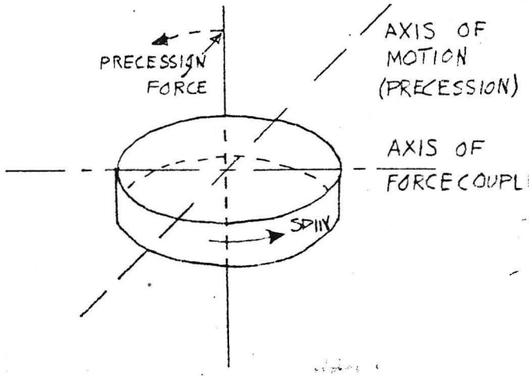


FIGURE 1



While the derivation of the equations which describe this motion is more complex than is consistent with the purposes of this paper, its results may be stated rather simply. The equation which is of most interest in the description of the gyroscopic instruments previbusly mentioned relates the rate of precession; that is, the angular rate of turn of the spin axis around the precession axis, to the size of the torque applied around the force axis. It is most important to remember that these two axes, with the spin reference axis, form a cartesian coordinate system which is dependent on the position of the rotor spin axis only, and is quite independent of the geometry of the supporting system; that is, <u>the basic axis is the spin axis</u>. Any axis in a plane perpendicular to this basic spin reference axis may be chosen as the force or torque axis, but as soon as we choose the force axis, the precession axis is fixed, since this axis must be perpendicular both to the basic spin reference axis and to the chosen force axis. If we wish to remember the information of Figure 1 regarding the relative directions of motion and force, we may write this relation simply:

$$T = \omega(H) \qquad (1)$$

If we wish to have the equation represent directions of motion and force, as well as their magnitudes, we may use the "vectorial" form: (Assuming a "right-hand" system)

$$T = \omega \times H \qquad (2)$$

Now, let us examine the terms of this relation:

T is effective torque applied to the wheel. If torque is applied to a gyroscope about some axis which is not in a plane perpendicular to the spin axis, the component of the torque which appears around the nearest axis in such a plane must first be found before the equation is applicable. A consistent system of units may be established if T is stated in dyne-centimeters. ω is the angular rate of precession, or motion. It is the angular rate of motion which the gyroscope will establish about an axis which is, at once, perpendicular to the spin reference axis, and to the axis about which T is applied. ω will be stated in units of radians per second, if both T and H are dimensioned as noted. H is angular momentum of the wheel around the spin axis. It is the product of the moment of inertia of the wheel and the angular rate of turn about its spin axis. It will be expressed in $GMCM^2\left(\frac{\text{Radians}}{\text{Sec}}\right)$ if the units of T and ω are to be as noted.

In order to make use of this effect, some means of holding the spinning wheel must be found. This means must prevent translational motion, yet must allow a degree of angular motion appropriate to the particular application. Viscous or spring restraint to motion around certain axes is sometimes desired, but static or dynamic frictions always cause an uncertainty in the completed instrument, and must be carefully avoided. Usually, these supports must be accurately balanced both statically and dynamically so that unbalance torques do not appear around any of the gyroscope axes. The particular means of support selected is usually one or more gimbals. These gimbals, or cardans, constitute a universal joint or hooke's joint, in cases where two such structures are used. Gyroscopes have been classified, on a dynamic basis, according to their number of degrees of freedom.

A gyroscope having one degree of freedom may be obtained by mounting the wheel in a single gimbal or cardan.

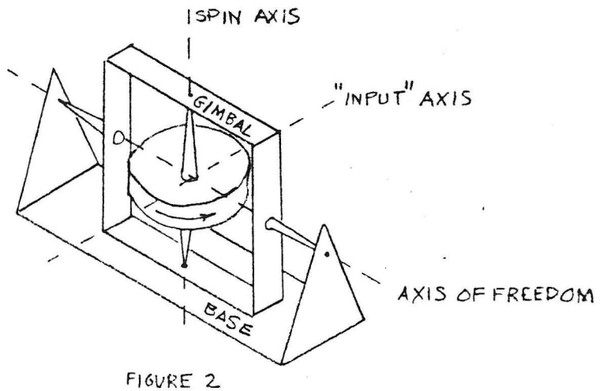


FIGURE 2

This mounting is typical of gyroscopes used as angular rate measuring devices, and is also used in certain applications as a single axis angular displacement measuring device. In connection with this mounting, it is interesting to note that a gyroscope which is locked so that it cannot precess around one axis -- for example, the axis labeled "input axis" of Figure 2-- will not exhibit gyroscopic stability around the other axis. That is, if the single degree of freedom gyroscope of Figure 2 is locked to some fixed object so that the gyroscope cannot turn around the "input" axis, the gimbal and rotor are perfectly free to turn around the "axis of freedom" without apparent reaction from the rotor.

A gyroscope having two degrees of freedom may be obtained by use of two gimbals.

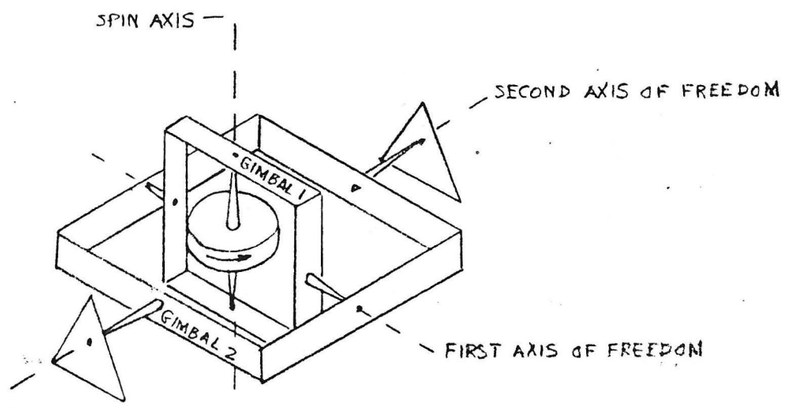


FIGURE 3

This mounting is typical of gyroscopes used as angular displacement measuring devices.

It would be possible to provide a third degree of freedom about the gyroscope spin axis, giving the gyroscope complete angular freedom; this, however, is of no significance with a single rotor, as a gyroscope is not sensitive to rotation or torque about its spin axis and could provide no useful data. Three, and even more gimbals, however, are used for various reasons in some instruments. These instruments are generally stabilized platforms of one type or another which usually consist of three single degree of freedom gyroscopes mounting on a single platform; this, in turn, is mounted in three or more gimbals, so as to provide a reference device capable of indicating motions around all three major axes of a coordinate system.

The gyroscopes to be discussed in the first portion of this paper are of three general types. First - Two-degree of freedom gyrocopes having no intentional restraints about their axes. These are used to indicate angular displacements about their two axes of other than their rate springs, and others which have various amounts of viscous restraint in addition to their spring restraint. A third general type of gyroscope is also a single degree of freedom device which has viscous restraint about its axis, but which has no spring restraint. A torque generator whose torque output is proportional to input current, and independent of gimbal angle, is provided in place of the rate spring. This very flexible and very accurate type of gyroscope may be used either as a single axis rate gyroscope or as a single axis displacement gyroscope. With its gimbal unbalanced in a proper way, it may even be used as linear accelerometer.

Rate Gyroscopes

The rate gyroscopes have only one gimbal, and apply the basic torque--rate relation directly as it has been written. For this reason, they will be considered first. A rate gyroscope, as has been mentioned previously, is an instrument which is used to indicate or to control angular rates of turn around some chosen axis.

A major application for this type of gyroscope is as a sensor for autopilot systems. In this application, from one to three gyroscopes are used to measure rates of turn of the airframe about its major axes. The output signals from these gyroscopes are added, in appropriate proportion, to other control signals from such sources as the roll, pitch, and yaw displacement measuring gyroscopes, which will be discussed later, the flux gate compass system, the fire control devices, and the pilot's control panel. These summed signals are amplified and control the airplanes' surfaces by means of hydraulic or electrical servomotors. The "authority" assigned to the rate gyroscopes - that is, the ratio of their maximum output signal level to maximum output signal levels from other sources, varies with the age of the autopilot, from the earliest autopilots, which had no rate gyroscopes to the yaw dampers, which are essentially single axis autopilots, and which utilize no other signals except those from a rate gyroscope.

Another important use of this type of gyroscope is in gunfire computers used in aircraft, in tanks, and on shipboard. In this application the gyroscope measures rate of turn of the sighting device or radar antenna as it tracks the target. This information is combined with other data concerning range, airspeed, and so on, to compute the point of aim for the guns.

The rate gyroscope again, is a single degree of freedom device. Its rotor is mounted in a single gimbal so that the spin reference axis is at right angles to the gimbal axis. This gimbal axis is called the "output axis", and a third axis, at right angles both to the spin axis and to the output axis, is called the "input axis." Springs attached to the base of the gyroscope and to the gimbal restrain the motion of the gimbal around the output axis. These springs are

centered so that the rest position of the gimbal places the spin reference axis of the gyroscope at right angles to the axis about which rate measurement is desired. The rate springs exert a force on the gimbal which increases in direct proportion to the angle of departure of the gimbal from its rest position. In addition to the spring restraint, varying degrees of viscous restraint are also added to the output axis. The viscous restraint varies, ideally, in direct proportion to the rate of turn of the gimbal around the output axis, and should be completely independent of the particular angle to which the gimbal is deflected. In summary then, the spring restraint varies in proportion to output axis deflection angle, while the viscous restraint varies in proportion to rate of change of output axis deflection angle. An angular position pickoff, either a potentiometer, or some type of electromagnetic device, measures the deflection of the gimbal around the output axis. It signal is used as an indication of rate of turn of the instrument around its input axis.

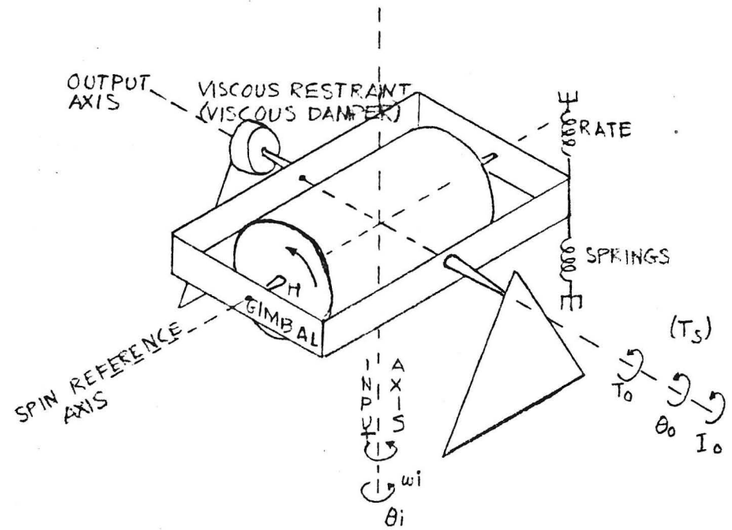


FIGURE 4

The rate gyroscope is a direct application of the basic torque--rate relationship.

$$T = \omega(H) \qquad (1)$$

In Figure 4, the rate is shown as around the input axis, and is designated ω_1. Torque, due to this input rate, and designated above as T_o, will appear around the output axis, according to the basic relation which we may rewrite, using the new designations as:

$$T_o = \omega_1 (H) \tag{3}$$

In recent gyroscropes, H, wheel angular momentum, varies from 1×10^3 $GMCM^2/sec$ in certain of the floated gyroscopes to 2.5×10^6 $GMCM^2/sec$ in certain of the older, non-floated rate gyroscopes. The gimbal will then deflect in response to T_o, to an angle θ_o in Figure 4, at which T_o, the torque due to ω_1, is exactly counterbalanced by some torque T_S, due to the deflection of the "rate" springs. That is, at θ_o;

$$T_o = T_s \tag{4}$$

T_s, which is the torque due to deflection of the springs, is proportional to the angle of deflection of the gimbal about the output axis, θ_o. That is:

$$T_s = K_1 \theta_o \tag{5}$$

Where K_1 is some constant of proportionality. If we substitute (3) and (5) in (4), and solve for θ_o, we find:

$$\omega_1 (H) = K_1 \theta_o \tag{6}$$

$$\theta_o = \frac{H}{K_1} \omega_1 \tag{7}$$

Now, if speed of the wheel and spring rate remain constant, H/K_1 is constant, and ω_1, input turning rate, may be measured in terms of θ_o, output axis deflection. If we have some pickoff whose output E_o/E_1, is in turn, proportional to output axis deflection, we have:

$$E_o/E_1 = K_2 \theta_o \tag{8}$$

and we may write:

$$E_o/E_1 = K\omega_1 \tag{9}$$

Where $\frac{K_2 H}{K_1}$ is written as a single constant K. The need for output axis damping, the viscous restraint mentioned previously, is that T_s of (5), the spring restraint about the output axis, and l_o of Figure 4, the inertia of the gimbal and rotor around the output axis form a second order system, which can oscillate at some frequency F_N. The viscous restraint damps this motion and prevents large overshoot. In terms related to the second order system, this damping varies in various designs from no intentional damping in certain units, to above critical (damping factor 1.0) in other designs. In most of the intentionally damped designs, the damping factor may be set at the factory to satisfy customer requirements.

PICKOFFS

Three general types of pickoffs are common for rate, and incidentally, for most other gyroscopes. These are the various inductive pickoffs; the potentiometer pickoffs; and the on--off, or switch pickoffs. Each of these classes of pickoff has its own advantages and disadvantages. An inductive pickoff is desirable where high accuracy, wide dynamic range, low "generator' or source impedance, and absence of frictional torques is desired.

The inductive pickoff has, however, certain disadvantages. It has such a low voltage gradient, in volts per unit of angular deflection, that it is usually necessary to provide an extra amplifier stage in the servo loop which it controls. When an inductive pickoff is excited and furnishes output current to a load, its armature exhibits a very slight spring restraint which, however, must be taken into account in work of the highest order of accuracy. The inductive pickoff is usually expensive and must be mounted quite accurately. This pickoff exhibits phase shift between excitation and output voltages.

The potentiometer pickoff is usually less expensive than the inductive type. Its mounting requirements are far less critical and there is no phase shift, for moderate frequencies, between input and output voltages and direct current excitation is practical. A potentiometer may be built so that its voltage gradient is quite high. Potentiometer accuracy is poor as compared to the inductive pickoff, as is its resolution. A potentiometer has a very high source impedance so that its accuracy is adversely affected by any loads placed across its output. It is probably not the best choice where series summing is to be used in the input to a servo system, and it exhibits very appreciable frictional torques.

The on--off or switch type pickoff is simple, inexpensive and perfectly satisfactory where the control system requires no more information than that some preset rate has been exceeded. There are two configurations. One is a commutator block across which a wiper slides. This type has a low contact current rating, and causes a frictional uncertainty similar to that caused by a potentiometer. The other is a leaf spring type. This can have a higher contact rating than the wiping type, and contributes little friction but it exhibits some spring torque at input rates exceeding the contact closure rate, and has a "break point" different from its "make" point.

The inductive pickoff, as has been previously mentioned, adds a very slight spring torque to that due to the rate springs. This causes an error in the output axis deflection angle. Apart from errors which the pickoff causes in the output axis angle itself, it exhibits errors in reporting this angle in terms of voltage. An "E", or differential transformer pickoff, is the inductive pickoff most frequently used on rate gyroscopes. This type of pickoff has many advantages over a synchro control transformer where the angular motion which must be reported is small as it is in the rate gyroscope.

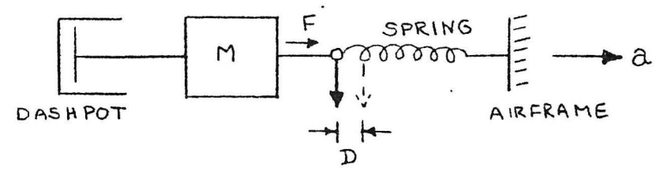
Without filtering and quadrature compensation, a synchro has a dynamic range of about 1000:1, while the differential transformer has only about 500:1. However, the angular range of the synchro is necessarily 90 degrees from maximum voltage to null, while the total travel maximum to null for the "E" pickoff, which is basically a linear motion sensor, may be made as low as 0.005 inch. Now, a rate gyroscope must necessarily have a range of angular freedom from maximum positive to maximum negative rate which is far less than plus and minus 90°. Thus, much of the plus and minus 90° angular range of the synchro, and therefore, much of its dynamic range, must be wasted. By use of an appropriate lever arm, however, the entire range of active travel of an "E" pickoff may be matched to the gyroscope's total angular travel so that all of the dynamic range of the "E" pickoff is usable. If a greater dynamic range is required, advanced systems of null detection, utilizing noise and quadrature rejection and harmonic filtering may be used. Such techniques can extend the dynamic range of such an electromechanical transducer to 5000:1 in certain cases. An additional advantage for the "E" type pickoff is that its armature has no coils, so that no flexible or sliding contact current conductors need be provided for the pickoff. The linearity errors of a differential transformer are quite easily held to plus and minus 1%, and may be held, with only the simplest of additional provisions, to plus and minus .1% of the pickoff's maximum output voltage. In order to attain these accuracies, care must be taken to see that the maximum rated loading run 100:1 lower in impedance than is the case for typical potentiometers. In addition, tendency of a long unshielded signal lead pair to pick up extraneous signals is reduced as its line to line impedance is reduced. The only remaining source of error in the inductive pickoff is its centering. The null of an inductive pickoff may be made to coincide with the rest position of the gyroscope gimbal more easily than is the case with a potentiometer pickoff. This is due to the absence of friction and its attendant hysteresis effects from the inductive sort of pickoff, and to the inductive pickoff's better resolution.

Probably the most advanced sort of inductive pickoff for angles of no greater than plus and minus 12 degrees is the microsyn, it is basically an "E" type pickoff in a circular configuration, and its dynamic range, with proper null detection techniques, is so wide that its threshold has not been fully evaluated.

In contrast to the inductive pickoffs, the potentiometer pickoffs present frictional rather than spring-like torques to the output axis. In some sensitive designed, hysteresis and "stiction" induced threshold must be considered. The linearity error inherent in this sort of pickoff seems to be about 0.5%. The limiting factor in dynamic range of the potentiometer pickoff is number of turns. Dynamic range is sometimes considered to be about the same as the total number of turns. This is due to the fact that while only half the turns appear on either side of center, a change in ratio equivalent to about one-half turn can be detected as the wiper moves across the potentiometer, so that there are about as many steps in ratio between null and either maximum as there are total turns in the whole potentiometer. Since these steps are inherent in a wirewound potentiometer, no amount of care in null detection can improve dynamic range. The error in

centering the potentiometer null (50% ratio point) to the gimbal rest point also contributes to pickoff error. For a typical design, this is 0.75% of maximum output signal. The potentiometer has certain advantages. The mounting concentricity which must be held between armature and stator of an inductive pickoff is most critical, and unless the design of the gyroscope insures that the pickoff is assembled to the gyros cope identically as it was mounted in test, test results cannot be quoted with much confidence as accuracy on the gyroscope. The mounting concentricity which must be held between armature and stator of an inductive pickoff is assembled to the gyroscope identically as it was mounted in test, test results cannot be quoted with much confidence as accuracy on the gyroscope. The mounting concentricity of wiper to pot is not nearly as critical since the accuracy limits are due to other phenomena. End play, also, is frequently critical to inductive pickoffs, but is less so to potentiometers. The potentiometer can, of course, be used with direct current or alternating current control systems, while the inductive pickoffs may be used directly only in alternating current systems. This is of great advantage where a phase lead must be produced in the carrier modulation envelope, since while this operation may be performed directly on a direct current carrier, a demodulator and modulator must be provided in addition if it is to be performed on an alternating current carrier. It should be noted that both types of pickoffs, as they are used on rate gyroscope, are essentially linear motion devices so that the voltage ratio they report is proportional to the tangent of the output axis deflection angle rather than proportional to the angle itself.

ACCELEROMETERS



1. General

An accelerometer is a device which is used to measure linear accelerations. The diagram above illustrates a typical accelerometer in symbolic form. It is a simple device consisting of a mass, a spring and a dashpot. When an acceleration is acting on the aircraft, the spring will extend a distance which is proportional to the acceleration acting on the aircraft.

2. Theory

If an acceleration, a, acts on the aircraft frame, it is applied to the mass, m, through the spring. The spring will extend a distance such that the force applied to the mass will cause it to accelerate at the acceleration, a. The force acting on the mass must be:

$$F = ma$$

The deflection of the spring will be determined by the spring constant, S.

$$D = FS$$

Since F = ma, we may rewrite the above:

$$D = mas$$

Since m and s are constants,

$$D = Ka$$

The deflection of the spring is proportional to the acceleration, a.

3. Damping

If an impulse acceleration is applied to a mass-spring combination, the system will oscillate at its natural frequency for some time after the impulse is removed. The use of a dashpot damps these oscillations to the point where they do not interfere with normal operation. The natural frequency of the spring and mass combination must be made quite high compared to the highest acceleration input frequency which the accelerometer is expected to measure.

4. Temperature Control

In manufacture of accelerometers, the mass is held centered by two springs rather than one. Damping action is provided by immersing the mass in a damping fluid.

Since the viscosity of the damping fluid will vary with temperature, a heater is usually incorporated in the accelerometer design. A thermostat, also incorporated in the design will energize and de-energize the heater to maintain the damping fluid temperature relatively constant.

5. Pickoffs

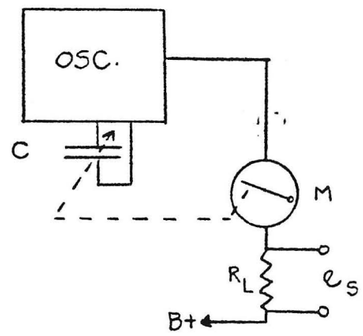
To convert the spring defleqtion into an electrical signal, either a potentiometer or a linear inductive pickoff may be used.

THE FORCE REBALANCE ACCELEROMETER

1. General

The accelerometer described above is subject to certain limitations. Extension of the springs must be kept small. After large spring extensions, the mass may not return exactly to its correct centre position due to hysteresis effects in the springs. In order to produce useable spring deflections, the mass, spring combination required may not have a natural frequency which is as high as desirable. Both of the above limitations can be overcome by using a force reblance accelerometer.

2. Operation



The system includes a superregenerative oscillator, a standard D'Arsonval meter movement M, and a variable capacitor C. The moveable plates of the capacitor can be driven by the meter movement. The oscillator frequency is in the 200 megacycle range, and its quench frequency can be varied from roughly 5Kc to 500Kc by varying C by roughly 5 micro-microfarads.

This can be done by moving the capacitor plates through a distance of 0.002 inches. Since the quench pulses are of nearly constant amplitude and duration (one microsecond) the plate current of the oscillator can be duty factor modulated by varying the capacitor plate displacement.

The moveable capacitor plate is the mass which is acted upon by an external acceleration. As the capacitor plate changes position due to an acceleration acting on it, the oscillator average plate current will change. If this average current is applied to the meter movement it can be used to apply a torque to the capacitor which will oppose any further capacitor plate movement. A point will be reached where the capacitor plate displacement will produce enough change of plate current so that the torque produced by the meter movement will just balance the torque produced by the acceleration acting on the mass of the capacitor plates. This torque will then be proportional to the acceleration. Since torque produced by a D'Arsonval movement is proportional to the current applied to it, the average plate current will be proportional to the acceleration acting on the mass. The voltage developed across resistor RL can be used as the output signal of this accelerometer.

It should be noted that variations of supply voltage and frequency, or oscillator frequency will have little effect on the performance of the force rebalance system. Natural frequency of this device is above 100 cycles per second because of the small mass involved. Hysteresis effects are negligable since movement of the mass is so small. In addition, over the operating range, linearity can be held to 0.1% or better.

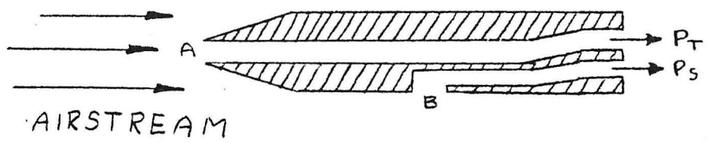
QC SENSOR

1. General

When a pilot wishes to change the attitude of his aircraft he deflects his control surfaces. The impact pressure, qc, of the airstream, acting on the control surfaces determines the rate of response of the aircraft to control surface displacement. As the aircraft increases its velocity through the air, the impact pressure will increase. As the aircraft increases altitude, air density decreases and the impact pressure decreases. If a pilot increases his velocity he knows that he will require less control surface movement to achieve a desired rate of change. He also knows that as he increases altitude he will require more surface deflection to obtain a desired rate of change.

Any automatic control system which actuates the aircraft control surfaces must modify its control action in the same manner as a pilot would, as the impact pressure, qc changes. Therefore, a qc sensor is required in this system.

2. Theory of Operation



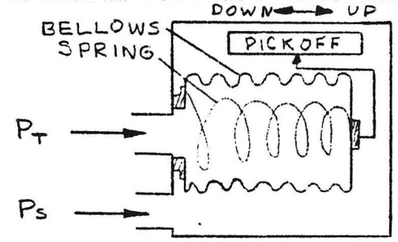
The diagram above shows the pitot head of an aircraft. When in flight, opening A can be used to produce pressure P_T, the total pressure of the airstream and opening B, which is normal to the airstream can be used to produce a pressure P_S which is the static air pressure.

Since P_T is the sum of impact pressure and static pressure, $P_T - P_S$ will be equal to impact pressure. Therefore

$$P_T - P_S = q_c$$

Since impact pressure is used to drive airspeed indicators, impact pressure scheduling of control surface action is often referred to as airspeed scheduling.

3. Differential Pressure Transducer



The diagram shows the manner in which a signal proportional to differential pressure $P_T - P_S$ can be obtained. P_S is applied to the inside of a bellows assembly and P_T is applied to the outside of the bellows by acting in the sealed container around the bellows. As P_T increases, the bellows will contract against the spring, moving the pickoff downward, a distance proportional to the pressure. As P_S increases, the bellows will expand against the spring, moving the pickoff upward a distance proportional to the pressure. The resultant position of the pickoff will be the subtraction of two linear distances, one proportional to P_T and the other proportional to P_S.

The linear motion of the bellows can be used to either drive scheduling potentiometer wipers directly or to move an inductive pickoff to produce a signal proportional to the impact pressure, q_c.

ALTITUDE SENSOR

1. General

In an automatic control system for an aircraft it is sometimes necessary to schedule control action with true airspeed. By modifying a signal proportional to impact pressure, we can obtain

a signal proportional to true airspeed.

The relation below describes qc.,

$$qc \doteq 1/2\ \rho V^2$$

where ρ is air density and V is true airspeed of the aircraft. As the aircraft altitude increases, air density decreases, so that qc cannot be used as a measure of true airspeed. But if we divide qc by :

$$\frac{qc}{\rho} \doteq 1/2\ \frac{\rho V^2}{\rho} = 1/2V^2$$

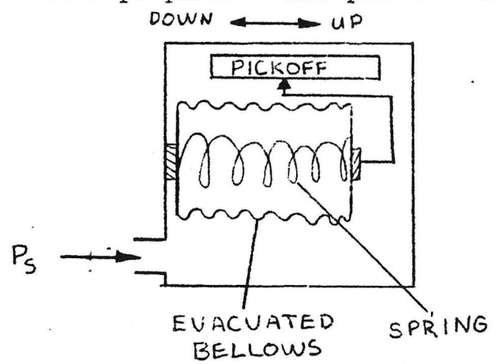
Therefore: $\frac{qc}{\rho} = KV^2$

By scheduling qc inversely with ρ, we obtain an output proportional to the true airspeed of the aircraft.

Since air density decreases as altitude h, increases, we can obtain scheduling proportional to true airspeed by scheduling first with qc and then with altitude, h.

2. Altitude Sensor

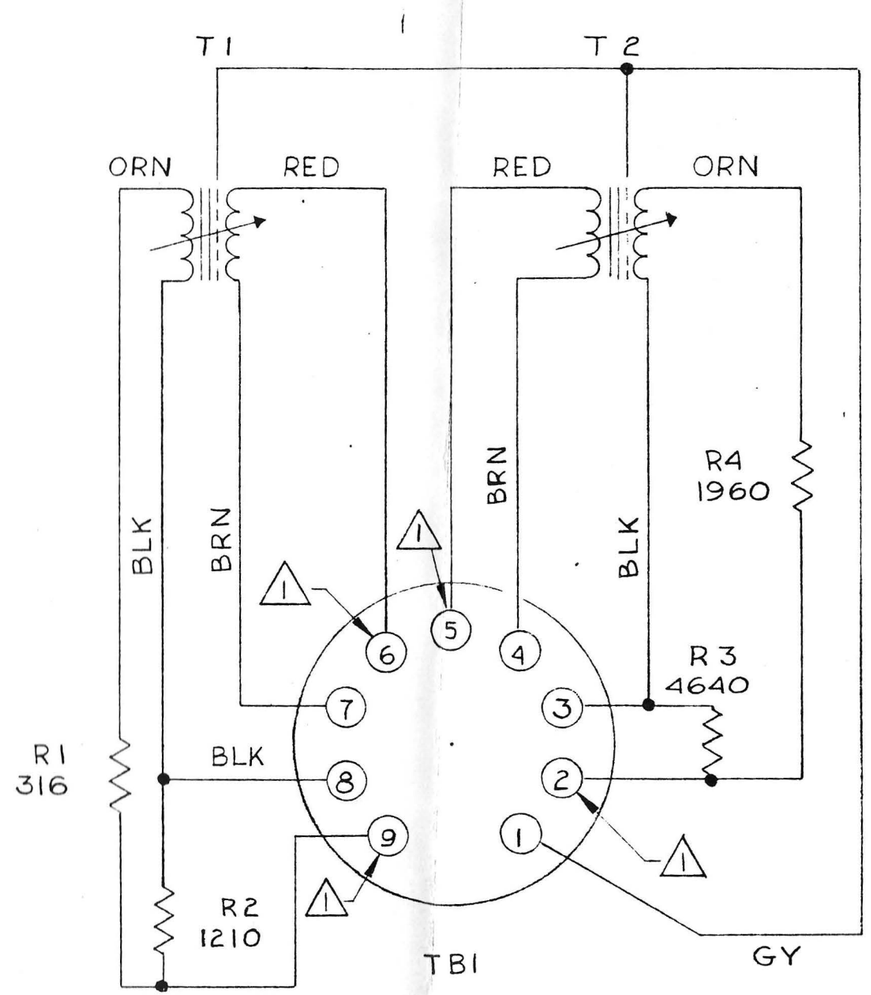
Since static pressure Ps decreases as altitude increases, it may be used as an indication of altitude. By measuring Ps, we can find altitude. The assembly shown can be used for this purpose. When pressure Ps is applied to the input port, the evacuated bellows will begin to contract against the spring, moving the pickoff downward a distance proportional to the pressure. When the pressure decreases (as altitude increases) the bellows will expand and move the pickoff upward. The pickoff position can then be used as an indication of altitude, or to drive scheduling potentiometers with altitude.



C 7655

SUPPLEMENTARY INFORMATION	
FIRST USED DP629A-1	
NEXT ASSEMBLY	REFERENCE

SYM	DESCRIPTION



3—ALL RESISTANCES ARE IN OHMS UNLESS OTHERWISE SPECIFIED

2—TERMINALS VIEWED FROM INSIDE UNIT

⚠1—WITH INCREASED ALTITUDE AND DIFFERENTIAL PRESSURE (ABOVE 0 PSF FOR NO. 1 TRANSFORMER AND ABOVE 1000 PSF FOR NO. 2 TRANSFORMER) NO. 2 TERMINAL IN PHASE WITH NO. 5, AND NO. 9 TERMINAL IN PHASE WITH NO. 6

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	.020
TWO PLACE	(00)	.010
THREE PLACE	(000)	.005
90° FORMED ANGLES		±1°

REV 01196-08

SUPSD

GOVT MATL SPEC

DRAWN BY	Falink 9-5-57
DRAFTING	O Powers 9-6-57
MATL & FIN	
DESIGNER	RWJ 9-12-57
DESIGN	O Powers 9-6-57
APPROVED	RB 9-12-57
REF MH MATL (SPEC	)
FINISH	(SEE NOTE)

SCHEMATIC DIAGRAM—DIFFERENTIAL PRESSURE TRANSMITTER

SCALE NONE | WT ACT CALC LB

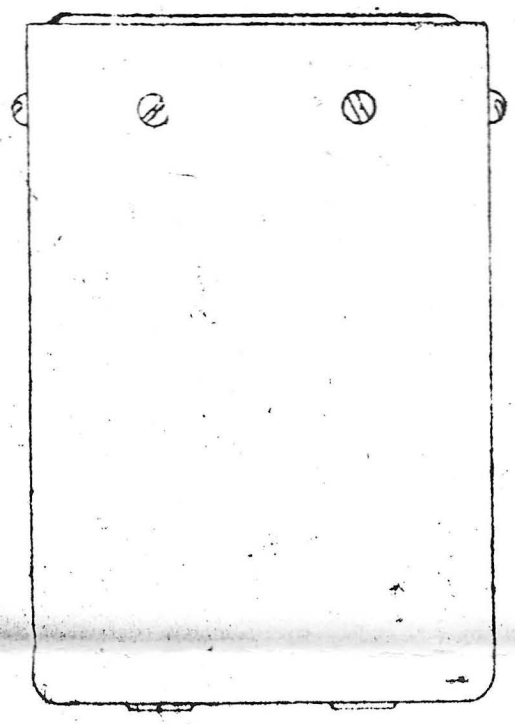
MINNEAPOLIS-HONEYWELL REGULATOR COMPANY

AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

C 7655

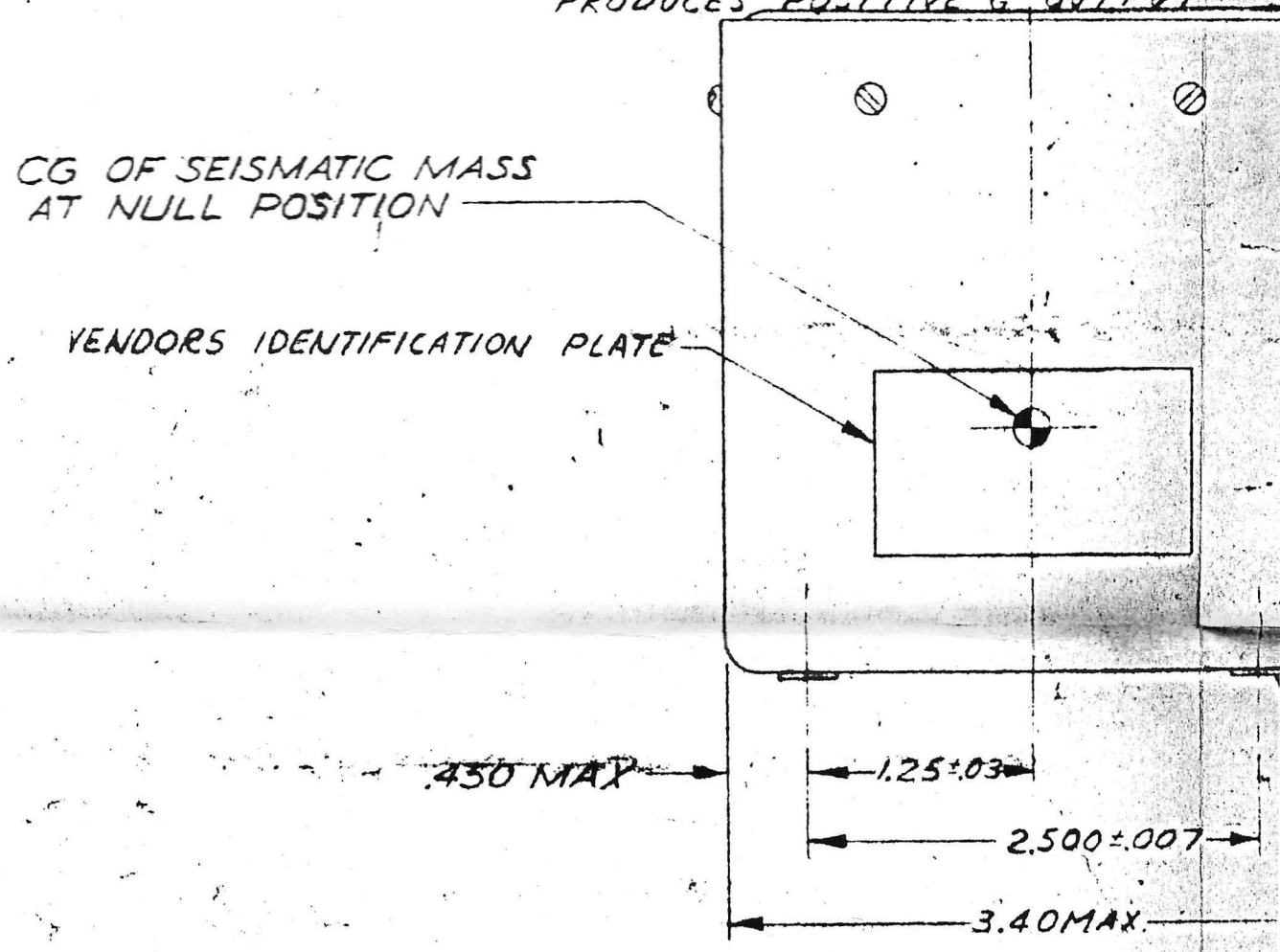
0185-69

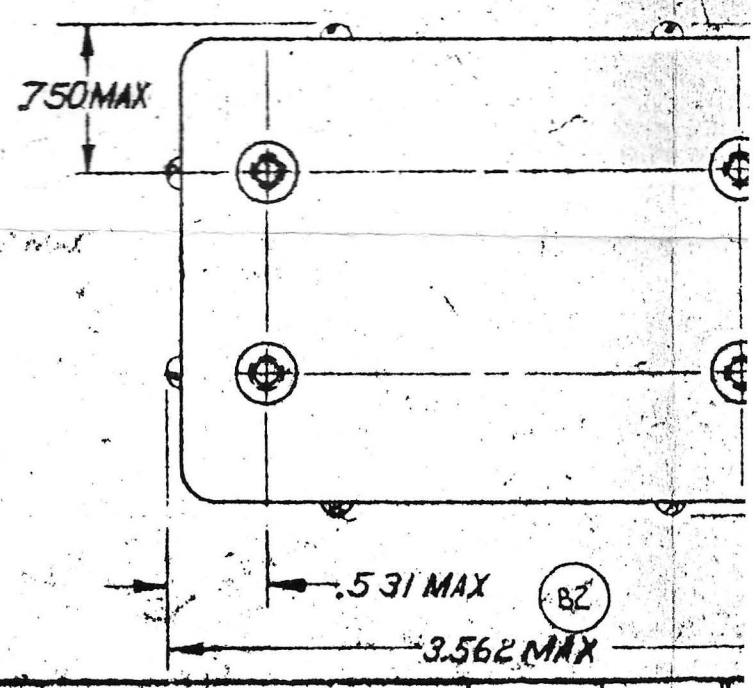
MH PART NO.	CONNECTOR	SCHEM DIAG REF
8-GG58B+1	NO. GS02 18-1PW-'11 CANNON	C6995
-2	NO. GS02-18-1PW-111 CANNON	C8870



SENSITIVE AXIS

ACCELERATION OF CASE IN THIS DIRECTIO
PRODUCES POSITIVE "G" OUTPUT





5 - DO NOT EXCEED A TORQUE OF 14 IN. LBS ON MOUNTING SCREWS WHEN INSTALLING ACCELEROMETER IN AIRCRAFT

4 - APPLY METAL-CAL PER MH SPEC 4010 ~~EXCEPT THAT AFTER SECTION 11-A-2 APPLY ADHESIVE MH MATL SPEC 6263 OVER AREA COVERED BY IDENT PLATE. MH MATL SPEC 6263 SHALL BE APPLIED IN ACCORDANCE WITH MH SPEC 4095 PAR. 2.1.5 AND 2.1.6 (FIRST SENTENCE ONLY)~~

3 - IN USAGE ~~AND STORAGE~~ THE MIN. DISTANCE BET. INDIVIDUAL ACCELEROMETERS SHALL BE $\frac{3}{8}$ ~~IN. SUCH THAT THE INTERACTION OF MAGNETIC FIELDS SHALL NOT EXCEED 50 GAUSSES~~

2 - THIS UNIT SHALL BE MOUNTED AT LEAST ONE (1) FOOT AWAY FROM ANY DEVICE SENSITIVE TO A STEADY MAGNETIC FIELD

1 - VENDOR GENISCO INC LOS ANGELES, CALIF. OR APPD EQUIV

REVISIONS	DATE	E

REV	REVISIONS		
C	IN NOTE 3 ADD "MIN"	6-9-58	9A-1552
B	(1).218 WAS .38; (2) RELOC; ADD NOTE 5 & IN TAB. -2; REV NOTE 3 & 4	4-25-58	9A-1523
A	ADD (1-2) & NOTES 1-4; NAME WAS "ACCELERO-METER"; REV LH VIEW	8-13-57	9A-1170

CONNECTOR

CROSS - DRILLED FILLISTER HD

DIRECTION OUT

CG OF SEISMIC MASS AT NULL POSITION

3.62 MAX.

.72 +.06 -.00

2.63 ±.06

1.37 MAX (A1)

A2 MH IDENT PLATE 435100 (4)

1.38 ±.03

.032 MAX.

.53 ±.03

.669 MAX

1.062 ±.007

2.40 MAX.

.007

MTG. PADS COPLANAR WITHIN .005 (5)

MTG. HOLES 4 PLACES #8-32 THDS .218 DEEP MIN. SYMMETRICAL IN BASE (B1)

2.562 MAX

D-GG58B

TABULATED

D

			TOLERANCE UNLESS NOTED		
NEXT ASSY	NEXT ASSY	REFERENCE	SIZE C	DRAWN BY	
MATL		MH DRG.	FINISH	DRAFTING	
COVT OR COML ALTERNATE (FOR REFERENCE)				CHKD	5/15/63
				APPROVED	

SCALE FULL SIZE DEV 01196-15

FIRST USED ON GG58B-1

WT LBS

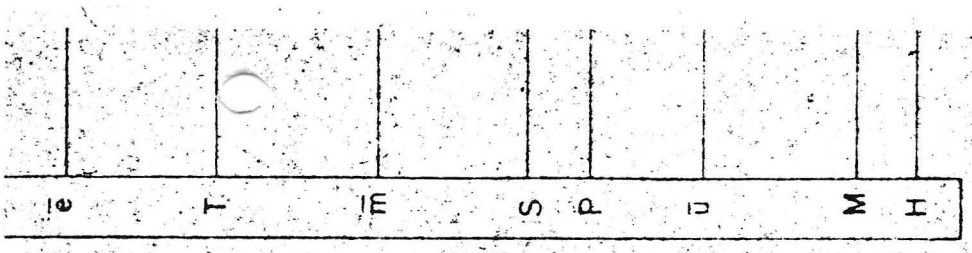
MINNEAPOLIS-HONEYWELL REGULATOR CO.
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

NAME ACCELEROMETER, LATERAL

D-GG58B

SHEET 1 OF 1

TABULATED



C6951 A

ADVANCED COPY

SUBJECT TO CHANGE

D

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	± 020
TWO PLACE	(00)	± 010
THREE PLACE	(000)	± 005
30° FORMED ANGLES		

REV 01294-C3

(UPS0)

GOVT MATL SPEC

DRAWN BY	5-21-57
DRAFTING	
MATL & FIN	
DESIGNER	RJB 5/31/57
DESIGN	
APPROVED	
REF MH MATL (SPEC	)
FINISH (SEE NOTE	)

SCHEMATIC DIAGRAM —

WITHOUT END RESISTORS

D-DG-33A-7

SCALE NONE

WT: ACT. CALC LB

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY

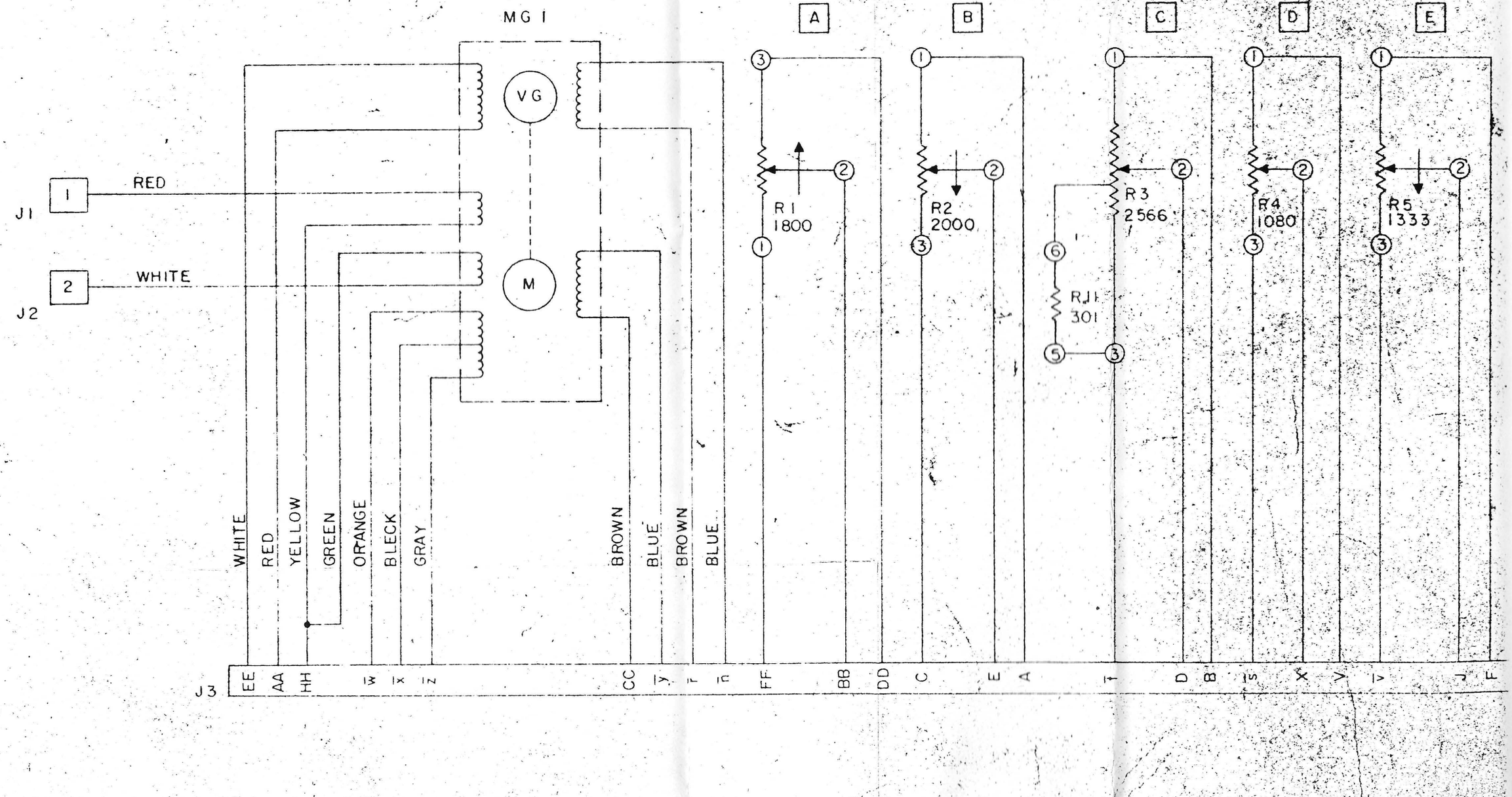
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

C6951

C#

POT STACK
NO. 1

POT STACK
NO.



ON
DDER

LATERAL ACCEL TO RUDDER

C

R3
2566

R11
301

NORMAL ACCEL

D

R4
1080

Sag

E

R5
1333

INTEGRATED NORMAL ACCEL

G

R14
549

R6
15060

YAW RATE TO RUDDER

H

R7
1956

R17
499

PITCH RATE GAIN

J

R8
4145

R19
787

ROLL RATE

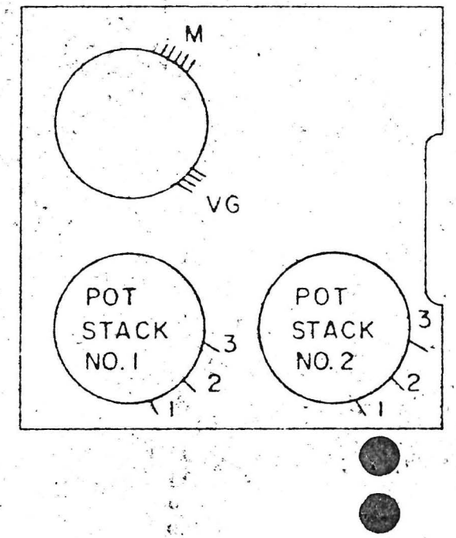
K

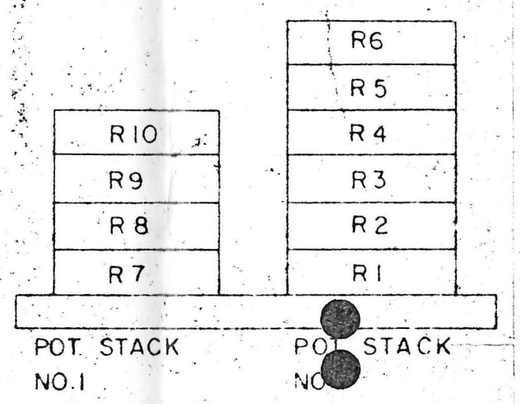
R9
880

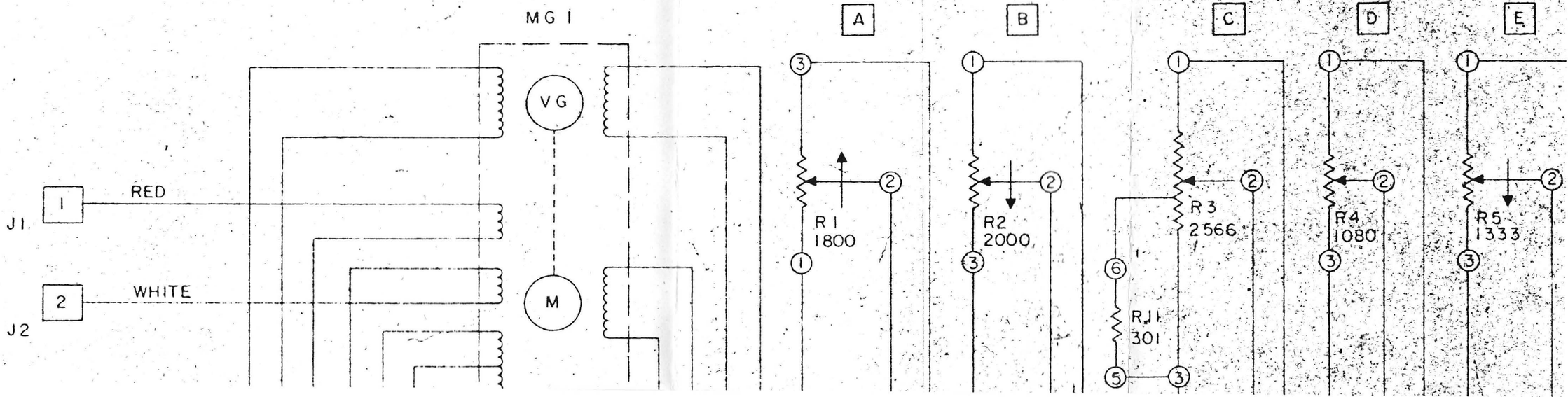
ROLL CONTROL STICK

F

R10
2000


M
VG
POT STACK NO. 1
POT STACK NO. 2
3
2
1
3
2
1


R6
R5
R4
R3
R2
R1
R10
R9
R8
R7
POT. STACK NO. 1
POT. STACK NO.


MG 1
VG
M
J1
1
RED
J2
2
WHITE
FEEDBACK
A
R1
1800
AILERON TO RUDDER
B
R2
2000
LATERAL ACCEL TO RUDDER
C
R3
2566
RJ1
301
NORMAL ACCEL
D
R4
1080
Sag
E
R5
1333

THE YAW AXIS

1. GENERAL

   (a) Since the yaw axis of the damper actuates a differential servo, there will be no indication of damper action in the rudder pedals.

   (b) Since the damper automatically coordinates turn entry and exits, no provision has been made to introduce command signals into the yaw channel and no means of introducing electrical trim is incorporated. For the same reason, no parallel servo is required.

   (c) Since the yaw axis of the damper will be engaged under normal conditions when the aircraft is on the ground before take-off, no elaborate pre-engage synchronizing action is required. In addition, the yaw axis signals are transient in nature and no steady state signals requiring synchronization will be present.

2. CONTROLLED FUNCTIONS

   In order to stabilize the yaw axis of the aircraft, the damper must sense the accelerations and rates acting in the yaw axis. In straight and level flight:

   (a) Yaw Rate
   (b) Rate of change of yaw rate.
   (c) Lateral Acceleration

   In turns:

   (a) Adverse yaw due to aileron position.
   (b) Steady state yaw rate.

   In climbing or diving turns:

   (a) Lateral acceleration due to combined pitch rate and roll rate.

3. The attached block diagrams indicate the build up of the yaw axis of the damper.

   (a) Diagram one shows a signal proportional to yaw rate from the yaw rate gyro being fed through an amplifier to the differential servo. When the servo moves, it will produce a feedback signal which will tend to balance out the signal from the yaw rate gyro. When the servo has moved a distance such that the rate signal is exactly cancelled by the feedback, the amplifier input will be zero and the servo will stop moving.

   When the error yaw rate has been decreased to zero by the action of the servo, the feedback voltage will appear at the amplifier input and cause the servo to return to the neutral position.

(b) Diagram two shows the addition of a lateral accelerometer which will produce servo action as described above if lateral accelerations act on the aircraft. This will occur if the aircraft is not flying directly into the relative wind.

Since the lateral accelerometer is not mounted at the centre of gravity of the aircraft, it will sense rate of change of yaw rate as well as lateral accelerations.

(c) Diagram three shows the addition of an aileron position signal. When the ailerons are displaced, an adverse yawing action will be produced while the ailerons are moving. The signal produced by the aileron position source will aide the yaw rate gyro in damping out this effect.

(d) Diagram four shows the addition of a signal derived from the product of pitch rate and aileron position. When the ailerons are deflected, the aircraft must have a rolling rate. When it also has a pitching rate, a lateral acceleration is sensed by the accelerometer due to the fact that the aircraft is not rolling about its centre of gravity. The lateral accelerometer signal would cause rudder deflection in a direction which would introduce an undesirable yaw rate. The aileron position times pitch rate signal counteracts this undesirable accelerometer output. When the aircraft is entering a climb and rolling, the aileron position times pitch rate output will produce aiding rudder into the turn. When entering a dive and rolling it will produce opposing rudder to the turn.

Diagram four also includes an integrator connected to act as a high pass filter.

When a turn has been established, the yaw rate gyro will sense a steady state yaw rate due to the rate of heading change of the aircraft. It is not desirable to oppose this steady yaw rate. By applying the yaw rate signal to the servo through a high pass filter, only transient yaw rates will be opposed. Steady state yaw rate signals will not pass through the filter and no opposing action will be produced. Similarly, adverse yaw due to aileron position occurs only as the ailerons are moving. By passing the aileron position signal through the high pass filter, the steady state signal produced when the ailerons are held displaced will be blocked and will produce no servo action.

(e) Diagram five shows the block diagram with airspeed compensation added. As airspeed changes, the effectiveness of the aircraft control surfaces changes. To maintain correct control action, each of the signals acting in the yaw channel must be scheduled with airspeed.

(f) Diagram six shows the yaw damper in the gear down configuration. It must be assumed that when the pilot extends his landing gear he will be at a low airspeed and is preparing to land. During and landing, it is often necessary to side slip the aircraft onto the runway. This means that a large steady lateral acceleration will be produced. The normal damper would oppose the pilot's side slip rudder command. To allow the pilot to make a satisfactory landing without fighting the action of the damper, ninety percent of the lateral accelerometer signal is fed to the servo through the high pass filter when the gear is down so that steady accelerations will not be opposed by the damper. Ten percent of the signal is fed to the servo as in the gear up mode to maintain stability of the system.

(g) Diagram seven shows the emergency yaw damper in block diagram form. The student will notice that the diagram is the same as that of the normal yaw axis except that no aileron position times pitch rate signal is present.

The subscript 'E' on all of the blocks in the diagram is to point out the fact that these are <u>not</u> the same devices used in the normal yaw channel. The differential servo ram is the only device common to both the normal and emergency system.

The emergency gear down mode is shown in dotted lines. The complete block diagram of the yaw axis (SK66500) shows additional blocks labelled with the letter K. Those are ratio adjustments. They are simply adjustable voltage dividers which may be set to select the optimum amount of each signal for correct control action. The settings of these adjustments are specified when the system is designed and are modified during flight testing to optimize performance. In general, these adjustments should not require changes in a normal system. Changing these settings should never be used as a means of correcting a system malfunction.

The two additional block diagrams of the normal and emergency yaw axisare functional diagrams which show how the block diagrams previously discussed are mechanized.

The series summing method is used to introduce the yaw axis stabilizing signals to the servo amplifier.

Two additional blocks are shown on these diagrams.

(i) Quadrature Filter: This unit is used to prevent stray signals and undesirable phase shifts from hampering system performance.

(ii) S.D.C. - Signal Data Converter: This unit was added to improve system stability in the eight cycle per second signal range.

Both of these units will be described in detail later in these notes.

YAW AXIS COMPONENTS

1. BG92B-1 Amplifier Calibrator (1)
2. GG72A-1 Yaw Rate Gyro (2)
3. GG58B-2 Lateral Accelerometer (2)
4. MG62B-2 Dual Differential Servo (1)

AVRO SUPPLIED COMPONENTS

1. Aileron Position Transformer (2)
2. Aileron Position Potentiometer (1)

GENERAL INFORMATION

A. Settings (Tentative)

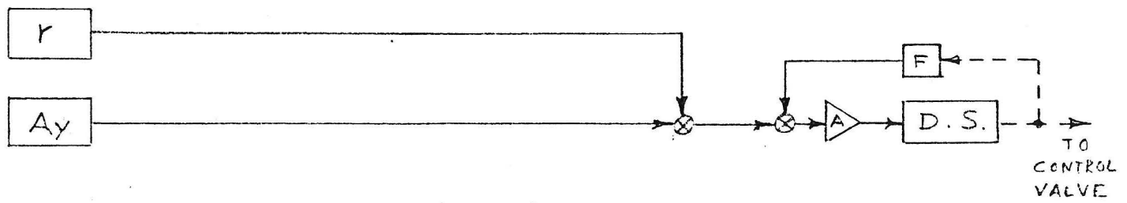
	NORMAL	EMERGENCY	
1. Yaw Rate	R146	R148	5.8
2. Servo Balance	R20	R42	6.9 (8.6ft.
3. Aileron Position	R151	R149	3.4
4. Lateral Acceleration	R152	R155	8.25
5. Aileron Position Times Pitch Rate	R153	-	7.8

B. Scale Factors

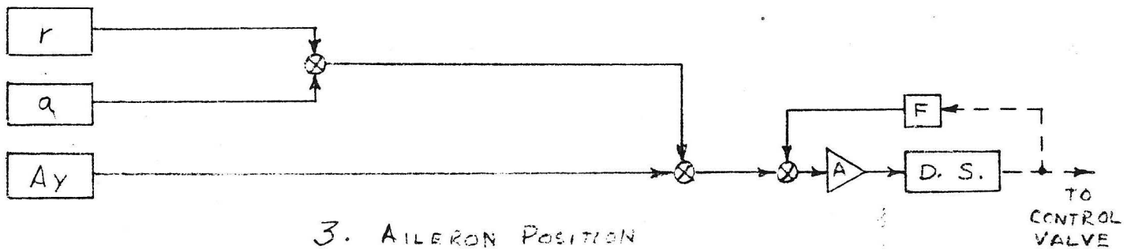
1. Differential Servo Ram Travel	±0.5 inch
2. Servo Balance Output	5.5V/Inch
3. Rudder Movement	±13 degrees
4. Yaw Rate Gyro Scale Factor	4V/15°/sec.
5. Lateral Accelerometer Scale Factor	5V/0.5 g
6. Aileron Position Output	6V/19 degrees
7. Aileron Position x Pitch Rate Output	3.4V/19°Ail/20°/sec Pitch
8. High Pass Time Constant	2 Seconds (Rate
9. Rudder Servo Amplifier Threshold	3.1 mv



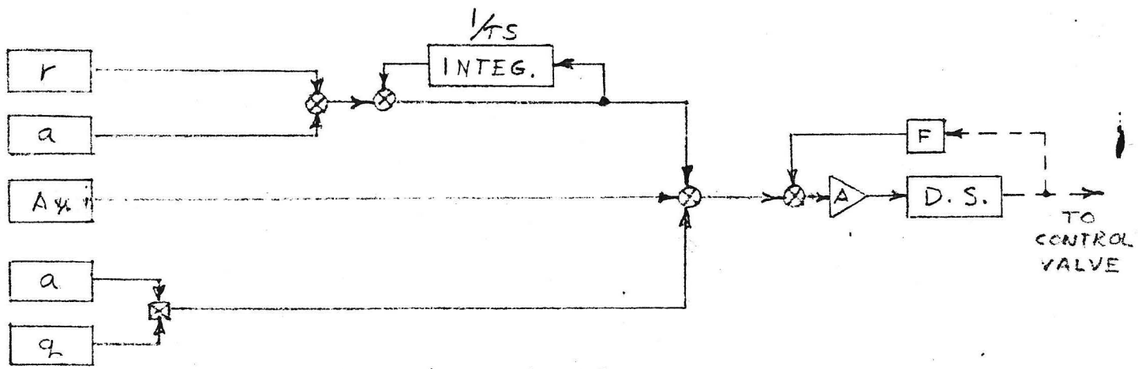
1. YAW RATE



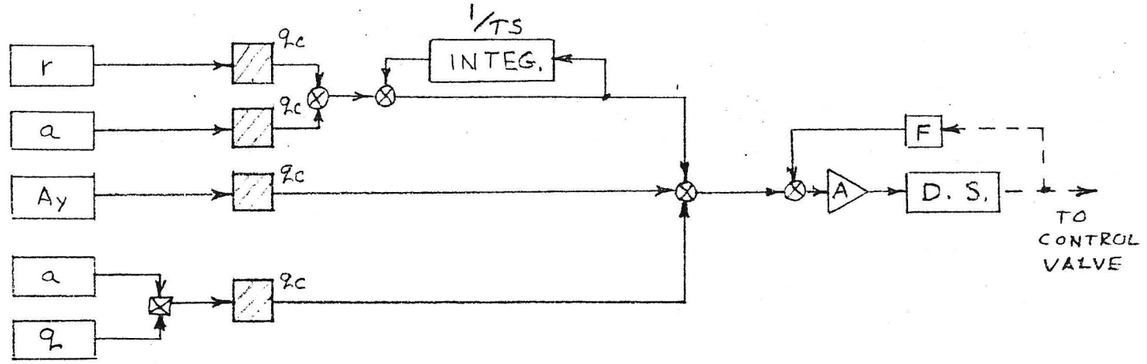
2. LATERAL ACCELERATION



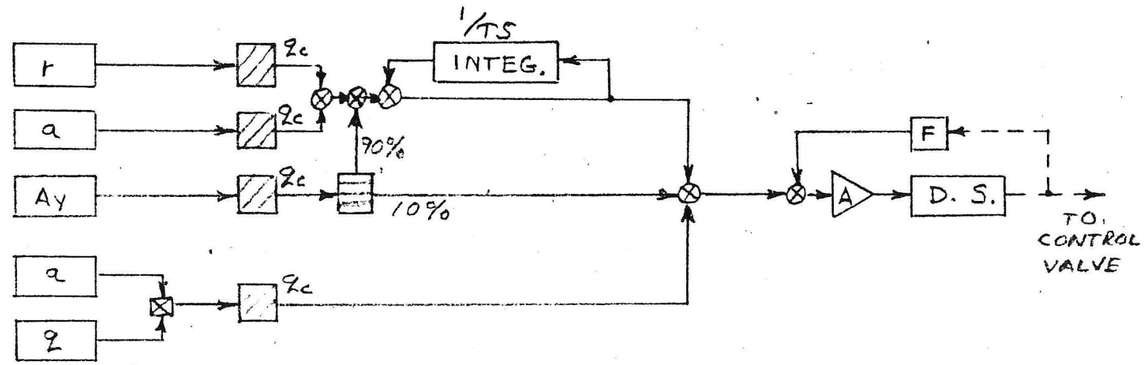
3. AILERON POSITION



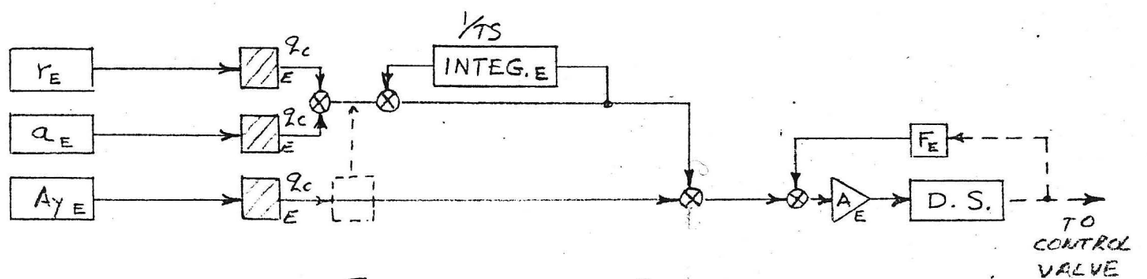
4. INTEGRATOR,
PITCH RATE X AILERON POSITION



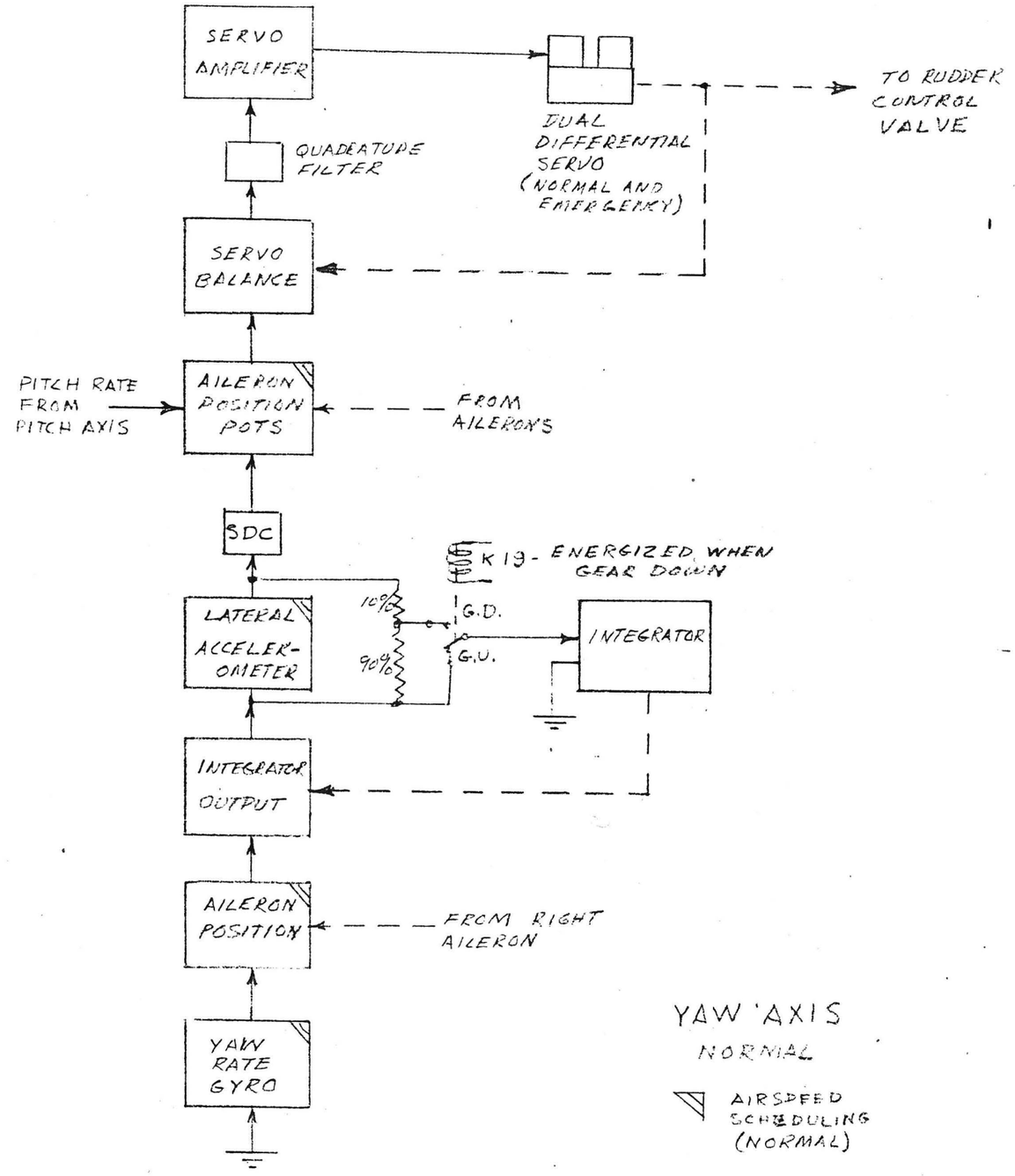
5. AIRSPEED COMPENSATION

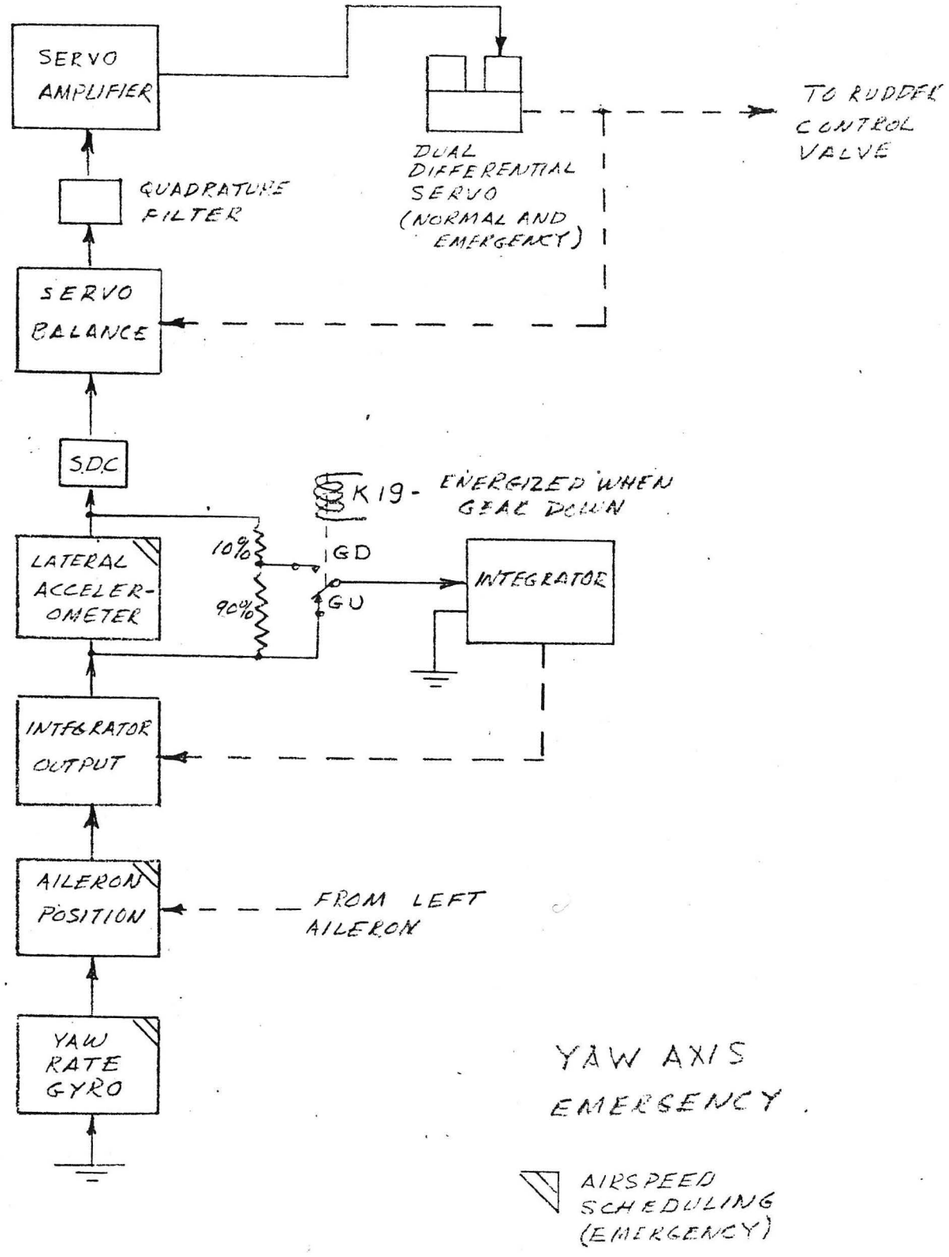


6. GEAR DOWN, PRE ENGAGE

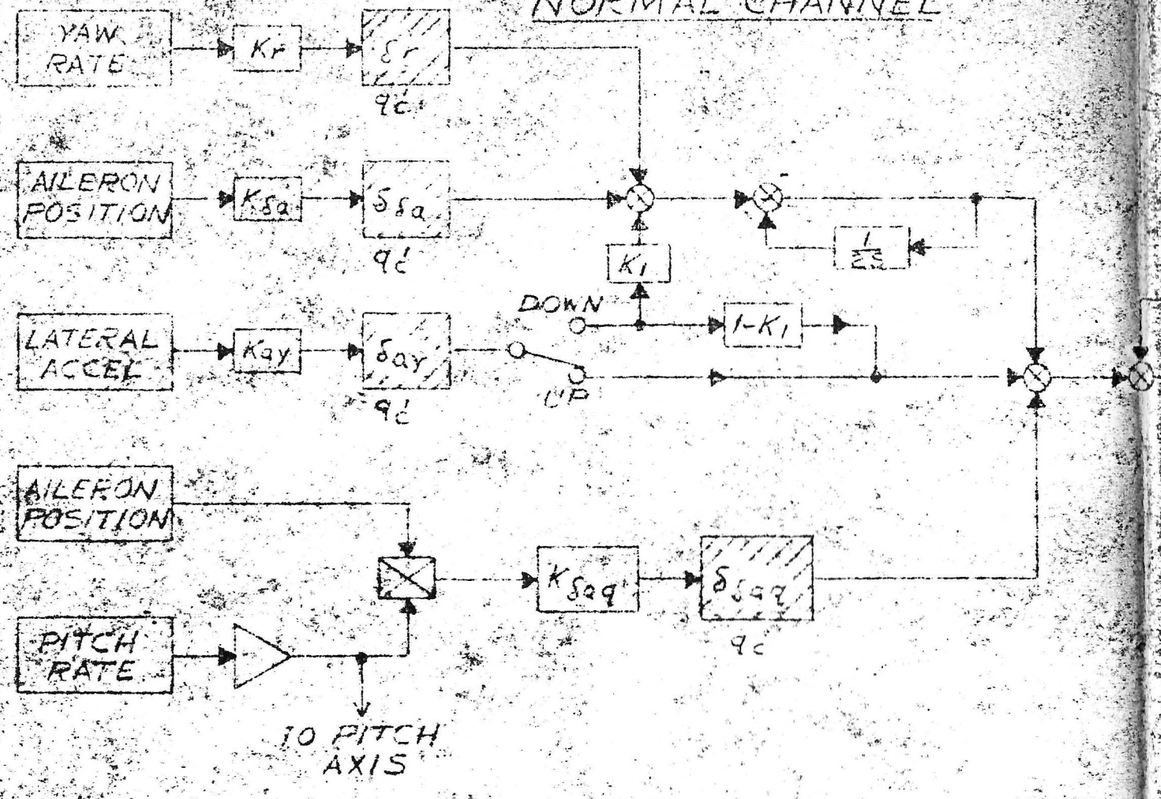


7. EMERGENCY SYSTEM

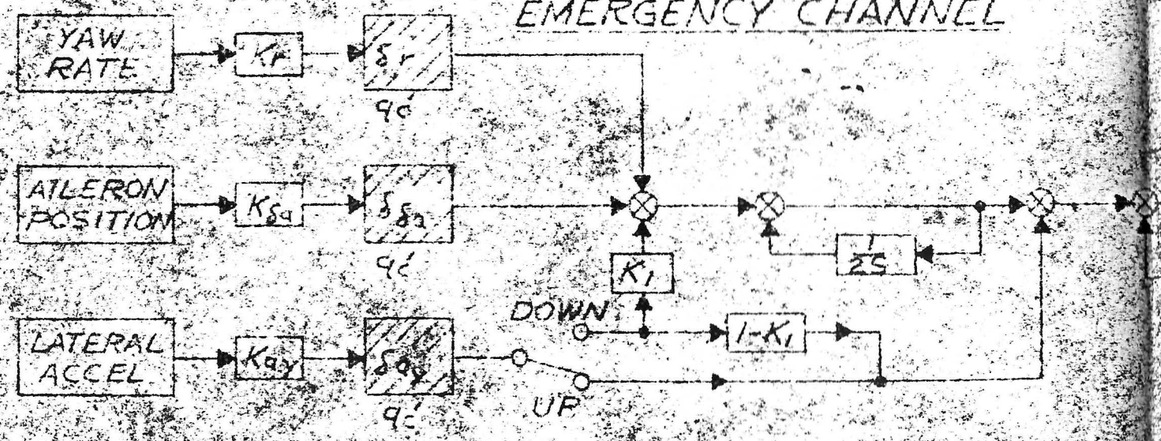

SERVO AMPLIFIER
QUADRATURE FILTER
DUAL DIFFERENTIAL SERVO (NORMAL AND EMERGENCY)
TO RUDDER CONTROL VALVE
SERVO BALANCE
PITCH RATE FROM PITCH AXIS
AILERON POSITION POTS
FROM AILERON'S
SDC
K 19 - ENERGIZED WHEN GEAR DOWN
LATERAL ACCELER-OMETER
10%
90%
G.D.
G.U.
INTEGRATOR
INTEGRATOR OUTPUT
AILERON POSITION
FROM RIGHT AILERON
YAW RATE GYRO
YAW AXIS
NORMAL
AIRSPEED SCHEDULING (NORMAL)

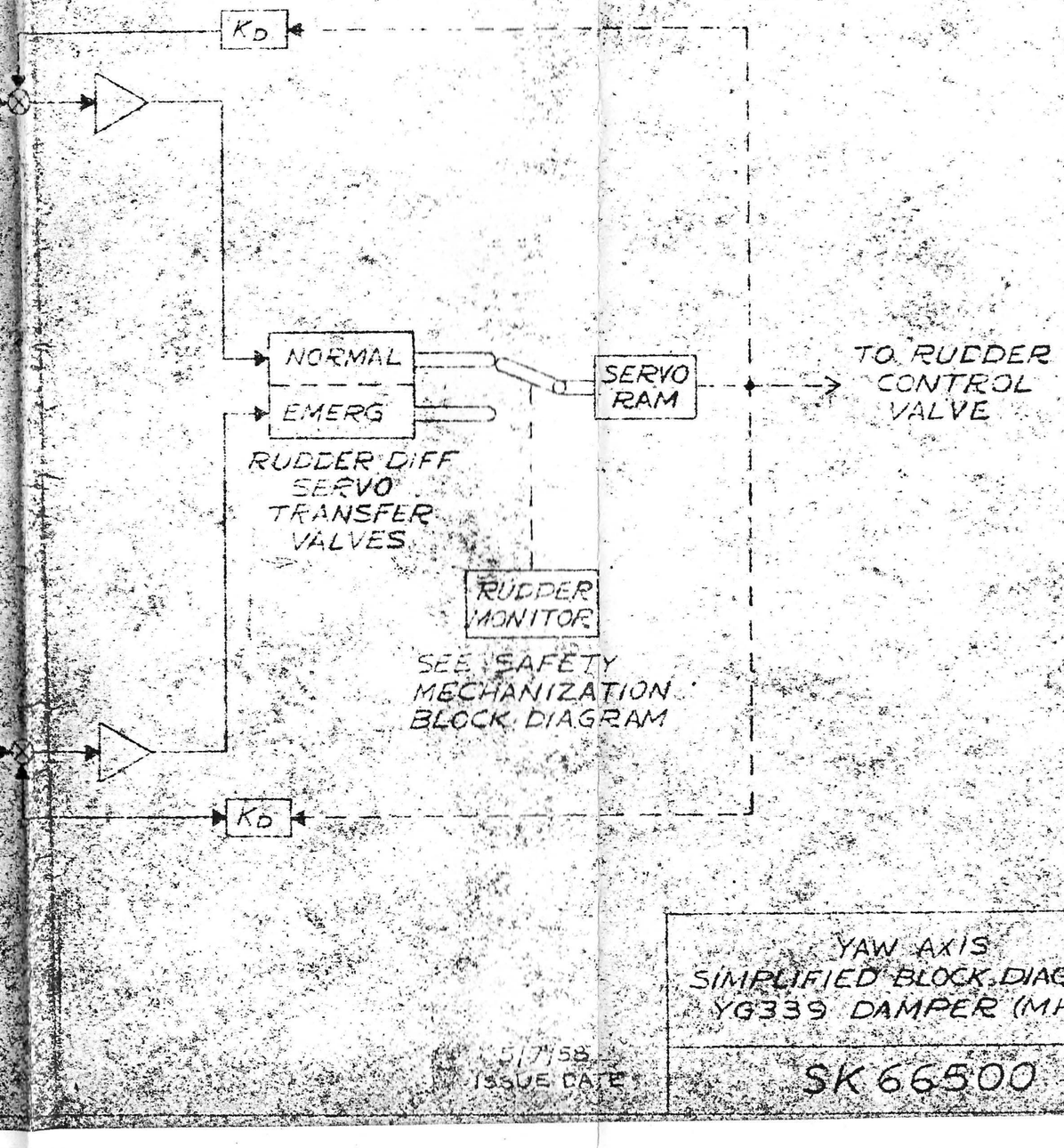

SERVO AMPLIFIER
QUADRATURE FILTER
DUAL DIFFERENTIAL SERVO (NORMAL AND EMERGENCY)
TO RUDDER CONTROL VALVE
SERVO BALANCE
S.D.C
K19 - ENERGIZED WHEN GEAR DOWN
LATERAL ACCELER-OMETER
10%
90%
GD
GU
INTEGRATOR
INTEGRATOR OUTPUT
AILERON POSITION
FROM LEFT AILERON
YAW RATE GYRO
YAW AXIS EMERGENCY
AIRSPEED SCHEDULING (EMERGENCY)

NORMAL CHANNEL



EMERGENCY CHANNEL





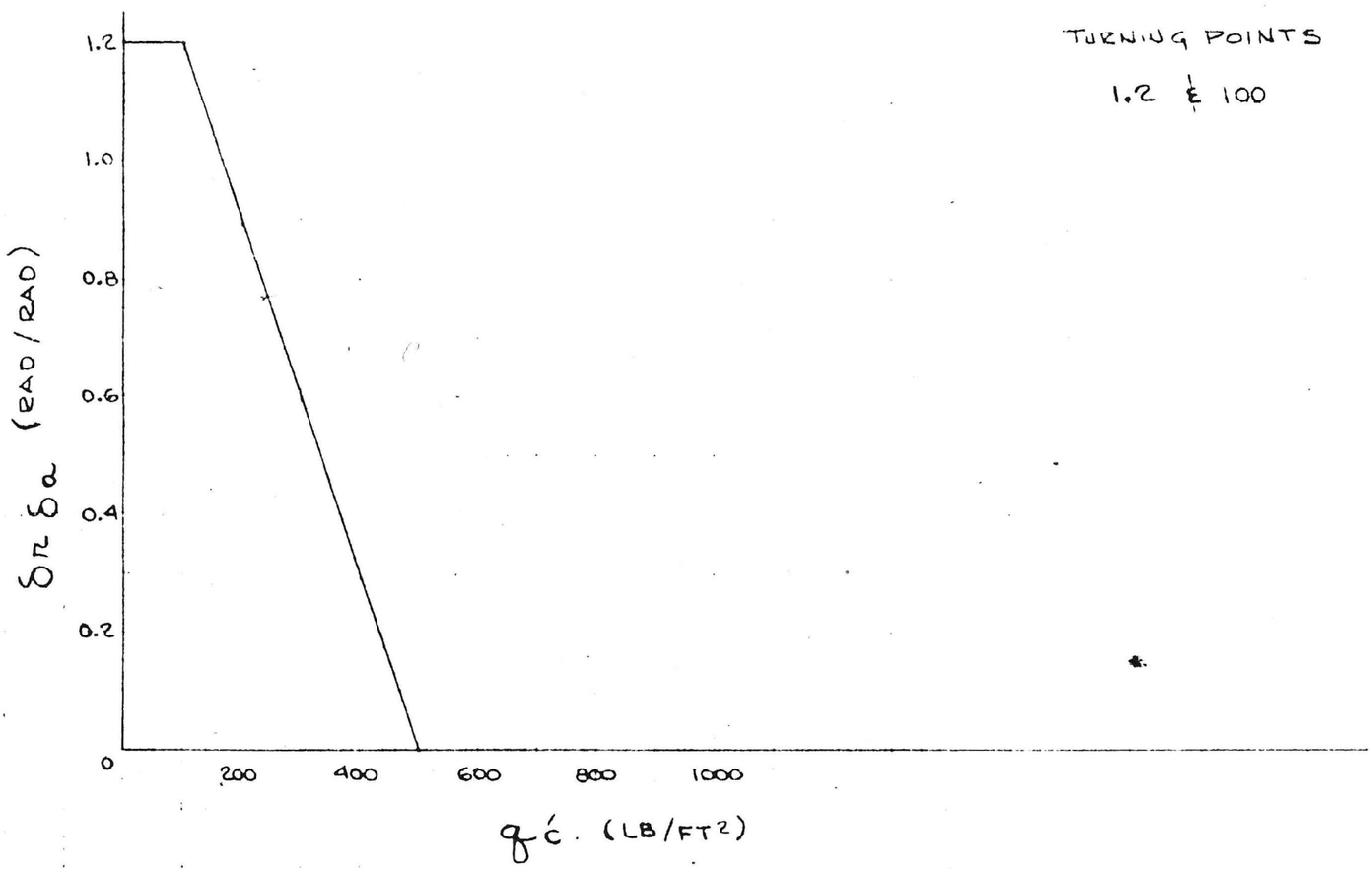
1 - RELAYS SHOWN IN THE GEAR UP POSITION

YAW AXIS
SIMPLIFIED BLOCK DIAGRAM
YG339 DAMPER (MH64)

5/7/58
ISSUE DATE

SK 66500

AILERON GAIN



TURNING POINTS

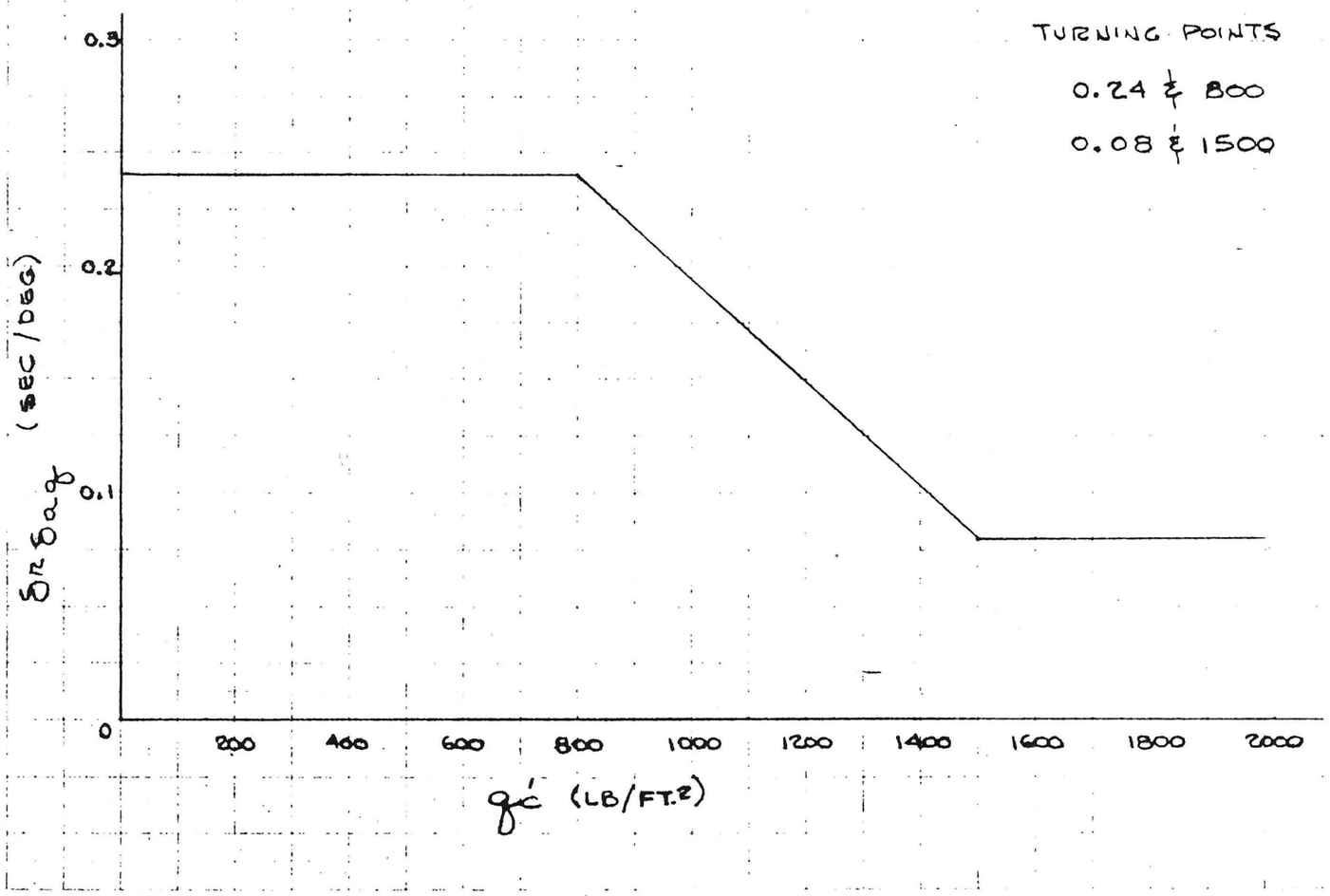
1.2 & 100

PITCH RATE x AILERON POSITION GAIN

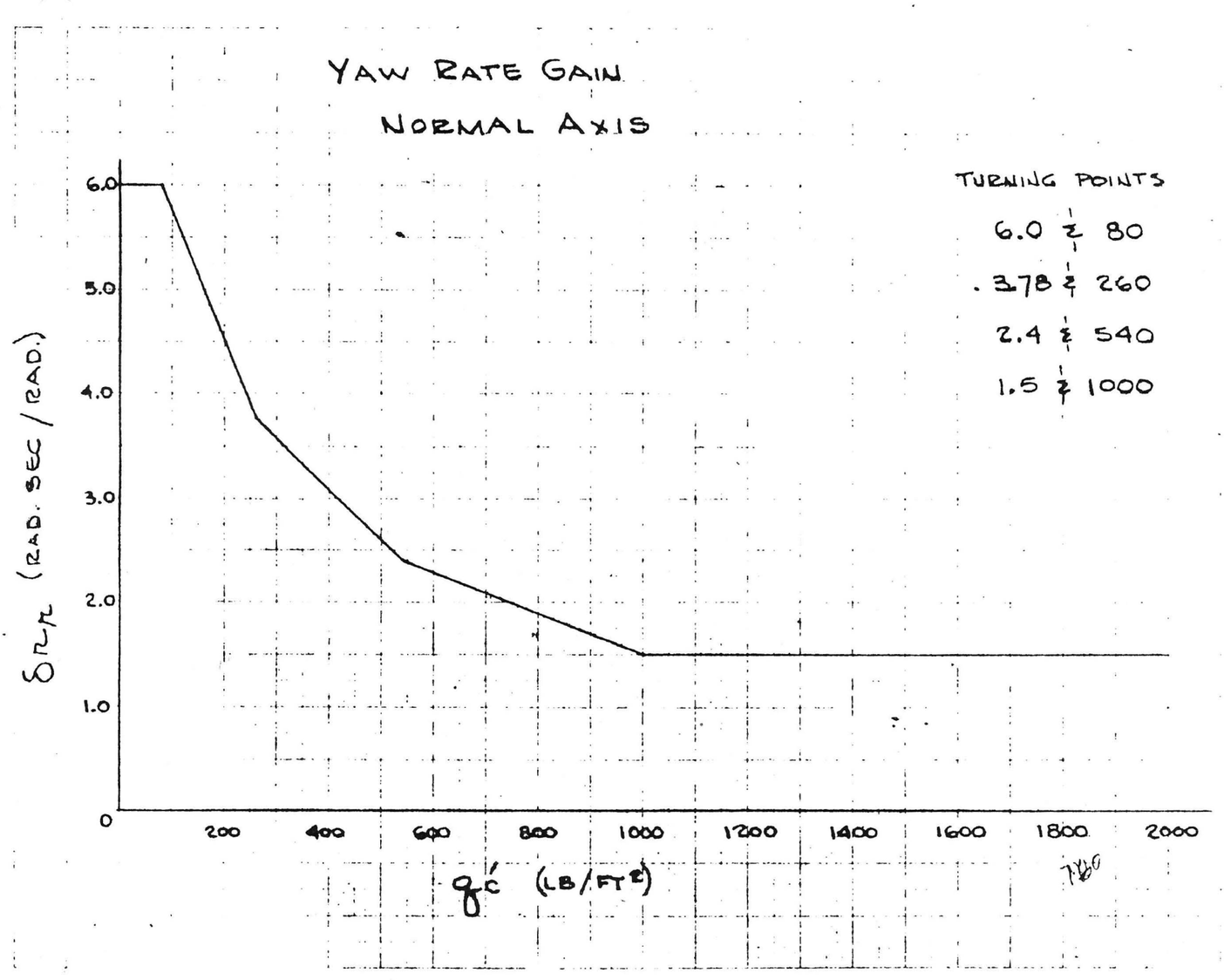
TURNING POINTS

0.24 & 800

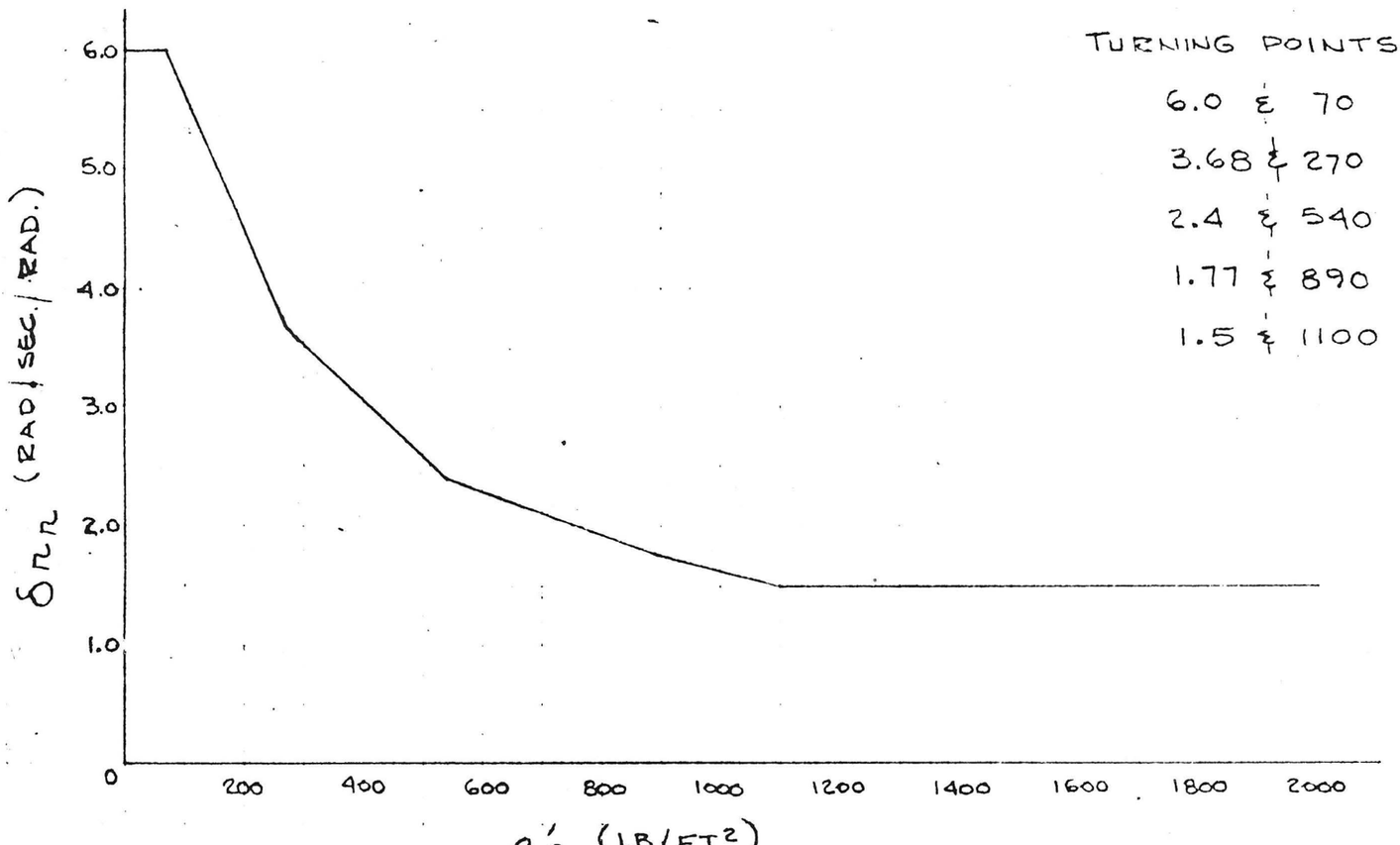
0.08 & 1500



GRAPH NO 6 CEL # 161


YAW RATE GAIN
NORMAL AXIS
TURNING POINTS
6.0 ± 80
3.78 ± 260
2.4 ± 540
1.5 ± 1000
δRUR (RAD. SEC / RAD.)
6.0
5.0
4.0
3.0
2.0
1.0
0
200
400
600
800
1000
1200
1400
1600
1800
2000
qc (LB/FT²)
760

YAW RATE GAIN
EMERGENCY AXIS



TURNING POINTS

6.0 @ 70

3.68 @ 270

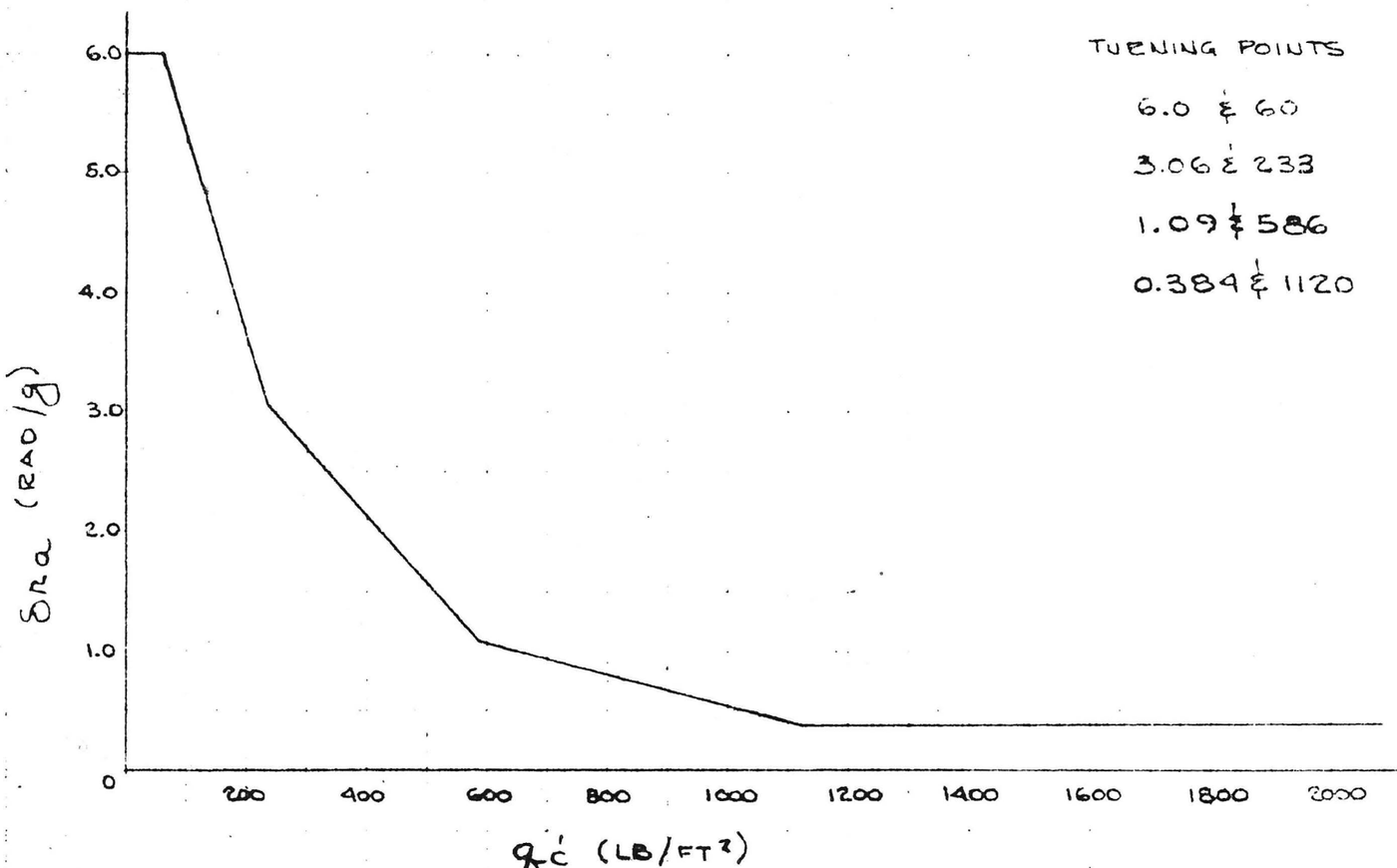
2.4 @ 540

1.77 @ 890

1.5 @ 1100

LATERAL ACCELERATION GAIN

NORMAL AXIS



TUENING POINTS

6.0 & 60

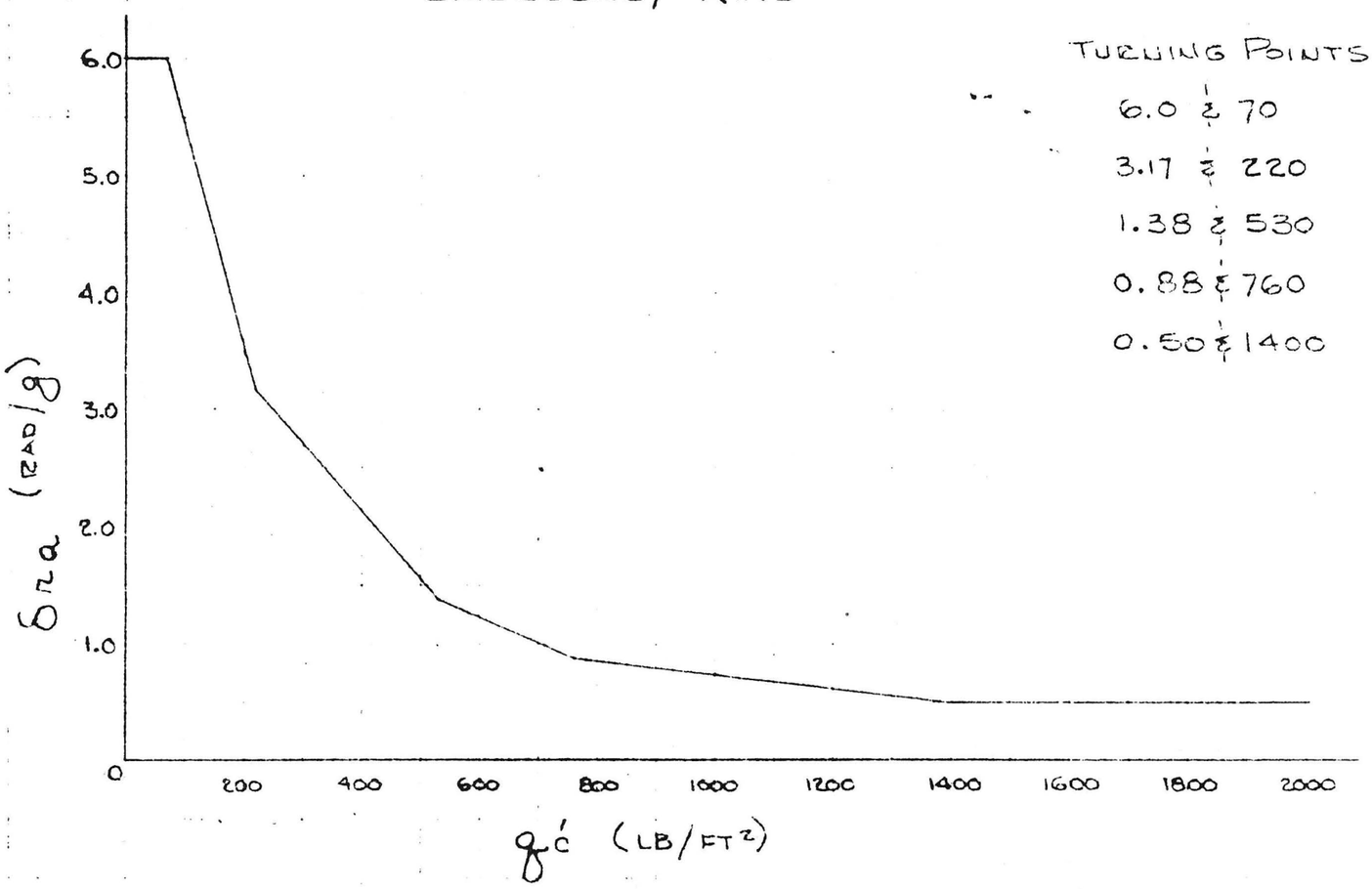
3.06 & 233

1.09 & 586

0.384 & 1120

LATERAL ACCELERATION GAIN

EMERGENCY AXIS



TURNING POINTS

6.0 ↓ 70

3.17 ↓ 220

1.38 ↓ 530

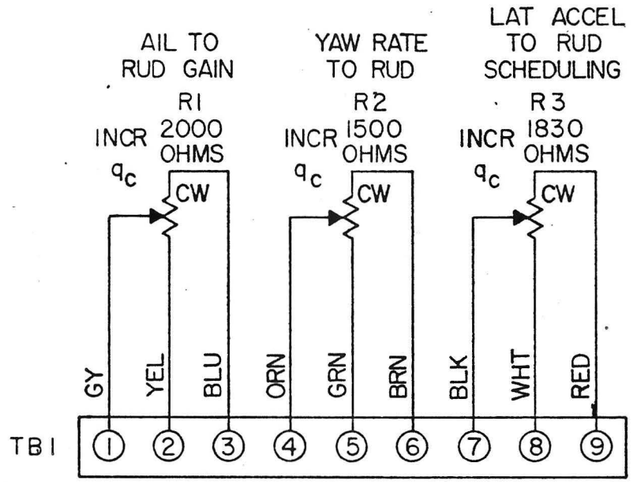
0.88 ↓ 760

0.50 ↓ 1400

SUPPLEMENTARY INFORMATION		
FIRST USED DPG32A-1		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL

C767



424590-19

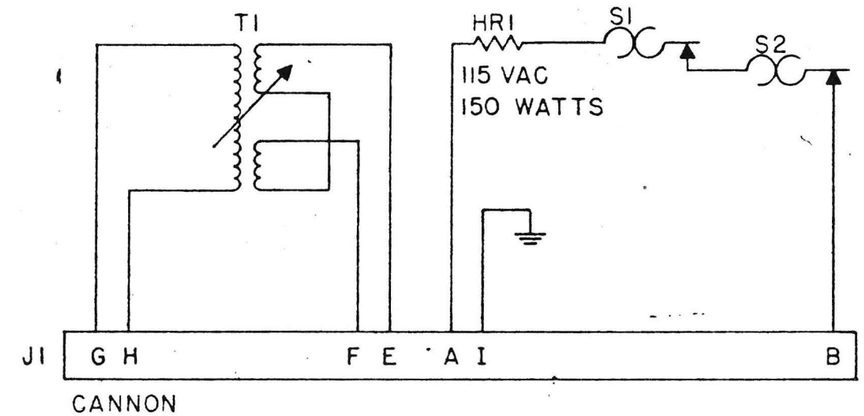
TOLERANCE UNLESS NOTED					
ONE PLACE (.0) ±.020	DRAWN BY	Dickard	8-23-57	SCHEMATIC DIAGRAM – COMPENSATOR	MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
TWO PLACE (.00) ±.010	DRAFTING √	Buttyfield	8-29-57		
THREE PLACE (.000) ±.005	MATL & FIN. √				AERONAUTICAL DIVISION
90° FORMFD ANGLES ±2° −1°	DESIGNER √	RWJ	9-12-57		MINNEAPOLIS, MINNESOTA
01196-33	DESIGN	Buttyfield	8-29-57		
S)	APPROVED	[illegible]	9-13-57		
GOVT MATL SPEC	REF MH MATL (SPEC)			SCALE NONE	C7678
	FINISH (SEE NOTE)			WT ACT. CALC LB	

8-2799

D-C887

SUPPLEMENTARY INFORMATION		
FIRST USED D-GG58B-2		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL



CANNON
GS02-18-1PW-111

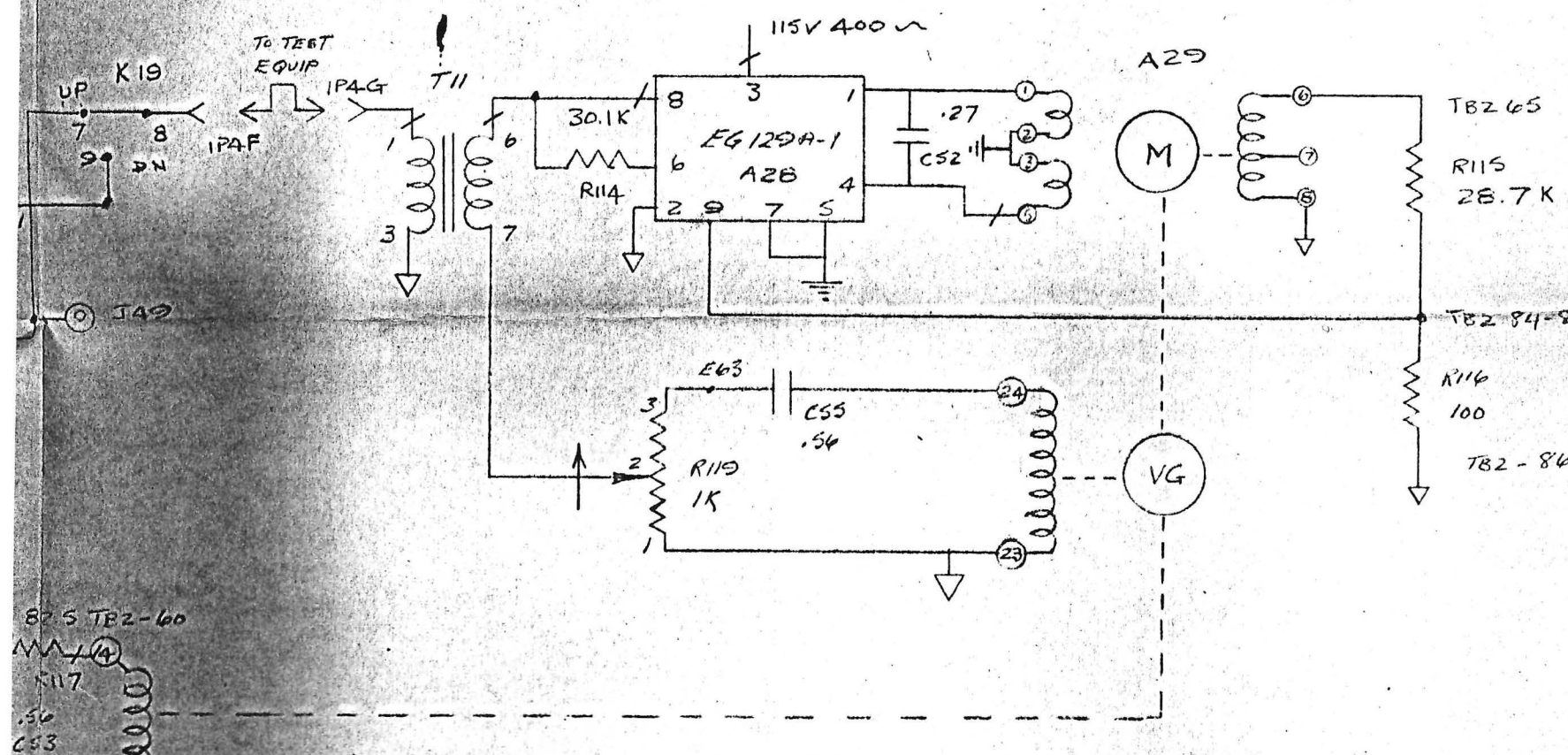
D-

2 - FOR +G OUTPUT, VOLTAGE GH & EF ARE IN PHASE
1 - CASE CONNECTED TO PIN I FOR GROUNDING OF CASE

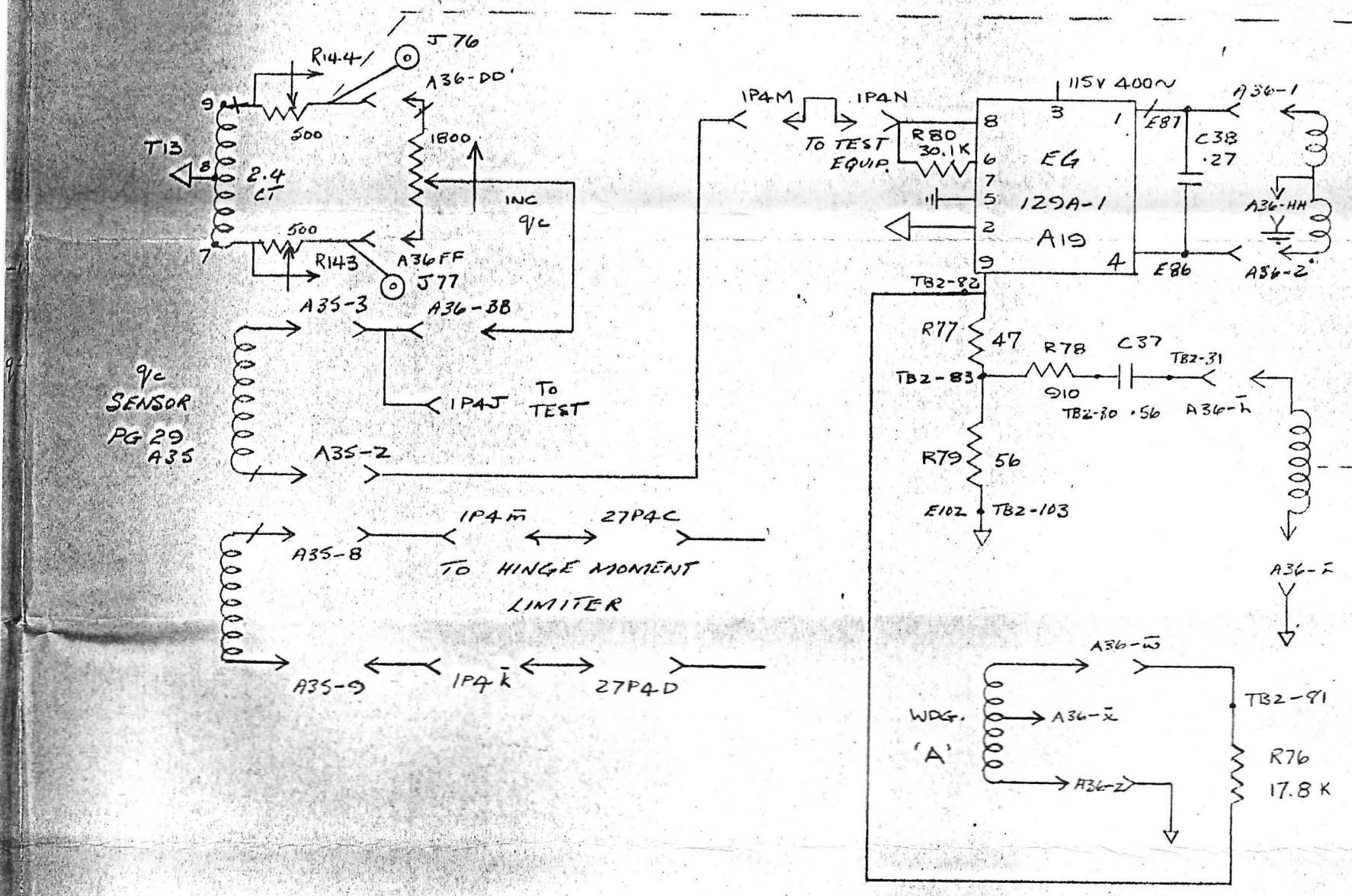
TOLERANCE UNLESS NOTED ONE PLACE (0) ± .020 TWO PLACE (.00) ± .010 THREE PLACE (.000) ± .005 90° FORMED ANGLES ± 1°	DRAWN BY L. Johnson 4-14-58 DRAFTING ANDRUSKY 4-17-58 MATL & FIN. DESIGNER gd7 4/18/58	SCHEMATIC DIAGRAM— ACCELEROMETER	MINNEAPOLIS-HONEYWELL REGULATOR COMPANY AERONAUTICAL DIVISION MINNEAPOLIS, MINNESOTA
2021-05-01 50	DESIGN ANDRUSKY 4-17-58 APPROVED R.H. 4-22-58		
GOVT MATL SPEC	REF MH MATL (SPEC)	SCALE NONE	C8870
	FINISH (SEE NOTE)	WT ACT. CALC LB	

AA 2788

15 ✓ SCHED. POTS, 1 Vc SENSOR.



NORMAL ϕc SCHEDULER DG 33 A-7



CSK 1778

TOLERANCE UNLESS NOTED	DRAWN BY	J. LAW	10/15/58
ONE PLACE (.0) ±.020	DRAFTING √		
TWO PLACE (.00) ±.010	MATL. & FIN. √		
THREE PLACE (.000) ±.005	DESIGNER √		
90° FORMED ANGLES ±2°			
OTHER ANGLES ±1°			
DEV. 1-23-57	DESIGN √		
SUPSD.	APPROVED		
GOVT. MATL. SPEC.	REF. MH. MATL. (SPEC.)		
	FINISH (SEE NOTE)		

MH-64-DAMPER
YAW AXIS
DEVELOPMENT
YG 3398-1

SCALE SIZE WT. ACT. CALC. LB.

HONEYWELL CONTROLS LIMITED
AERONAUTICAL DIVISION
TORONTO, CANADA

CSK 1778

SUPPLEMENTARY INFORMATION			FIRST CLASS.
FIRST USED			
NEXT ASSEMBLY		REFERENCE	
		SK 66127	
		REV B	

REVISIONS			
SYM.	DESCRIPTION	DATE	APPROVAL

NSFER VALVE
18P2F 18P2E
26TB4-A
1P2B
15
C27
G113B
A13
8
6 — 115V 400~
3 4 7
J46
J47
115V (E)

CONNECTORS

1P AMP CAL
2P1 RATE GYRO YAW (N)
30P TEST PANEL
8P1 LATERAL ACCEL EMERG.
2P2 RATE GYRO YAW (E)
18P SERVO 18P1 NORMAL
18P2 EMERG.
27P1, 27P2 } AVRO FURNISHED DEVICES
7P1 LATERAL ACCEL NORMAL
15 ¹/₆ SCHED. POTS, ¹/₆ SENSOR.

87.5 TB2-60
R117
.56
C53
TB2-61

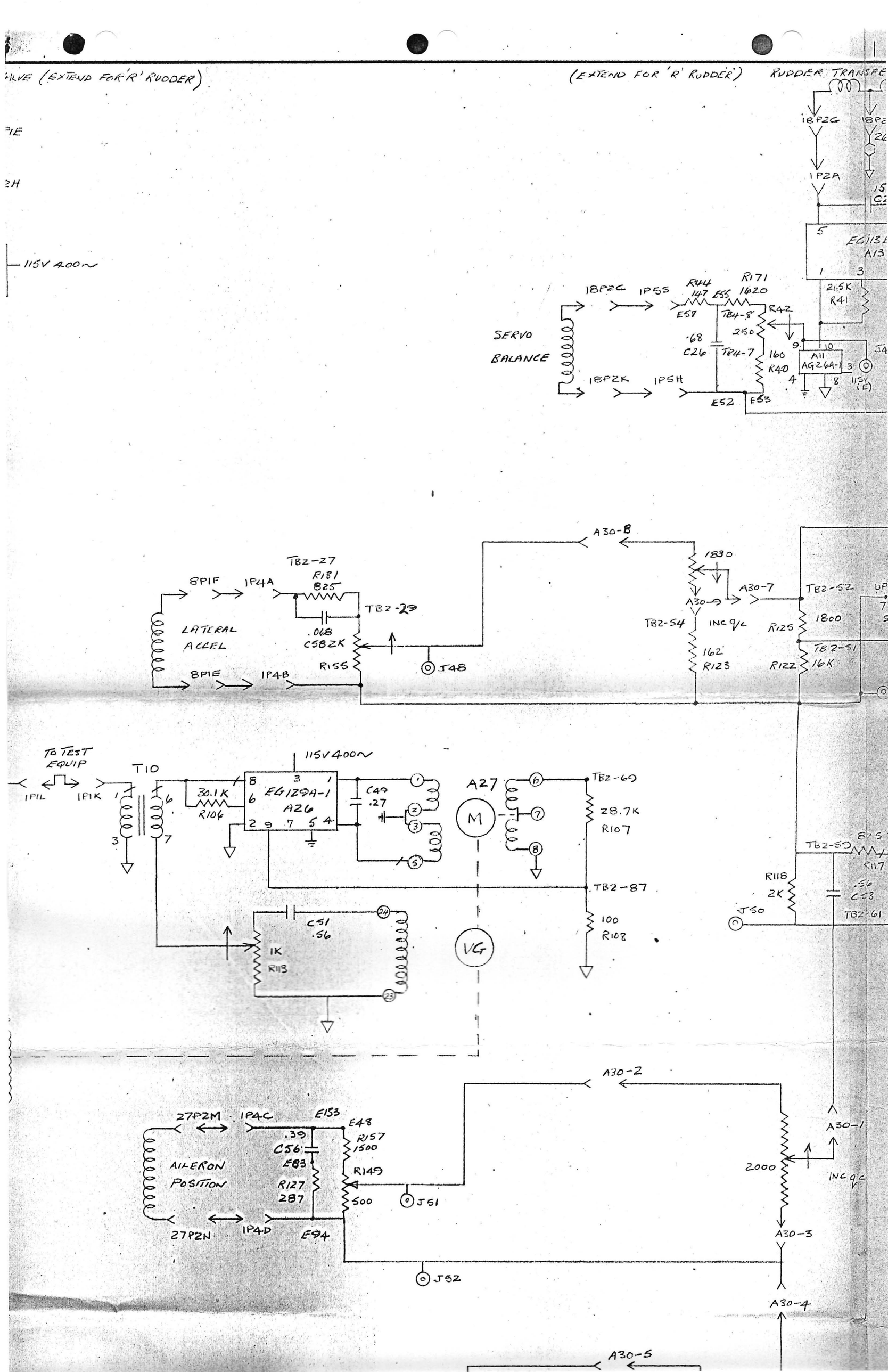
TO TEST EQUIP
1P4G
T11
K19
UP
7 8
9
DN
1P4F
1
3
6
7
30.1K
R114
8 3 1
6
2 9 7 5
EG 129A-1
A28
.27
C52
4
115V 400~
A29
M
TB2 65
R115
28.7K
TB2 84-85
R116
100
TB2 - 86
E63
C55
.56
R119
1K
VG
J49

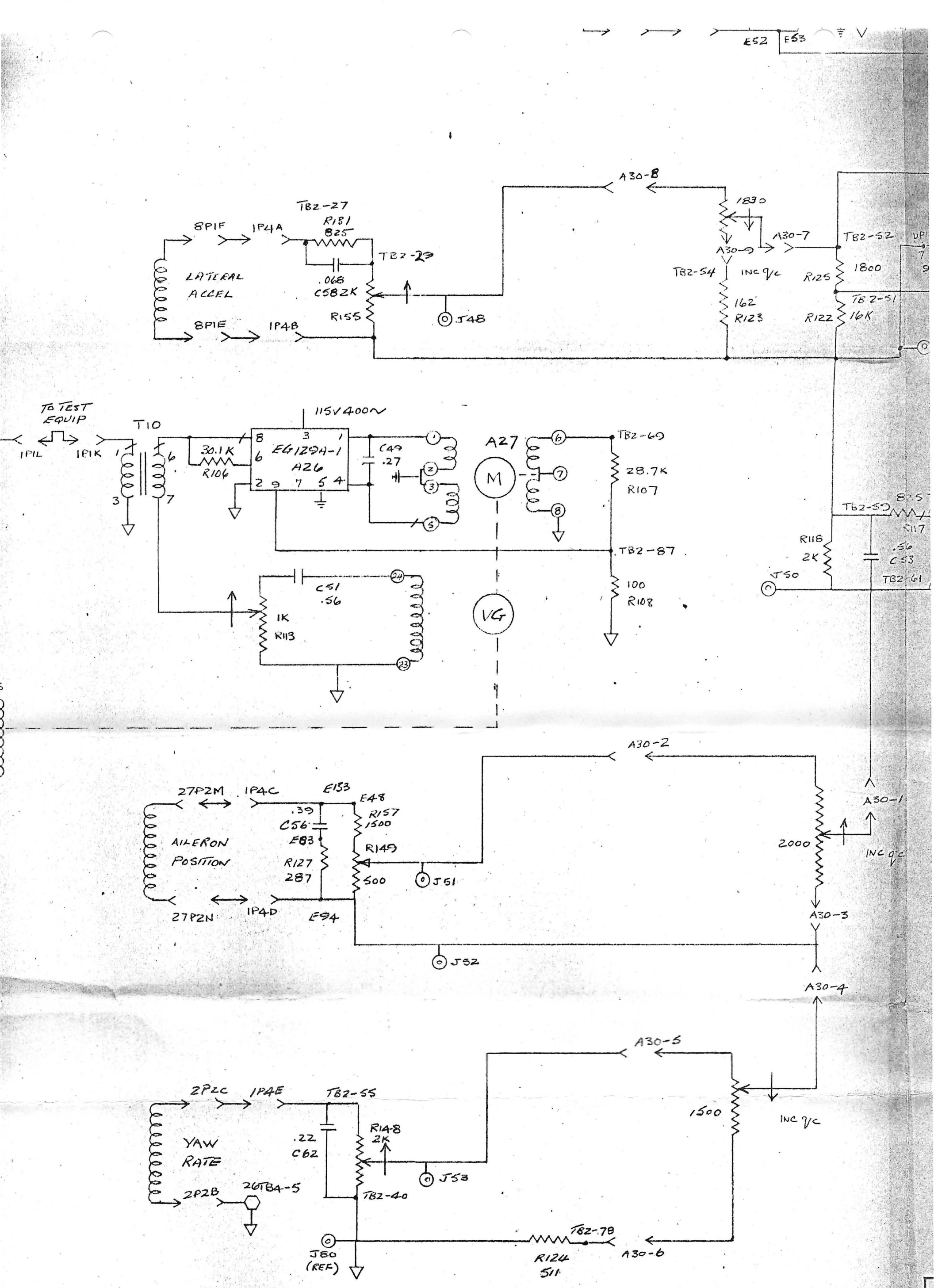
TO ALL NORM
¹/₆ SCHED. POTS
A36
M
VG
WDG
'A'

1P4M 1P4N
TO TEST EQUIP.
R80
30.1K
8 3 1
6 7
5
2
9 4
EG
129A-1
A19
E87
C38
.27
E86
A36-1
A36-HH
A36-2
TB2-82
R77
47
R78
910
C37
.56
TB2-83
TB2-80
TB2-31
A36-I
R79
56
E102
TB2-103
A36-I
A36-ω
TB2-91

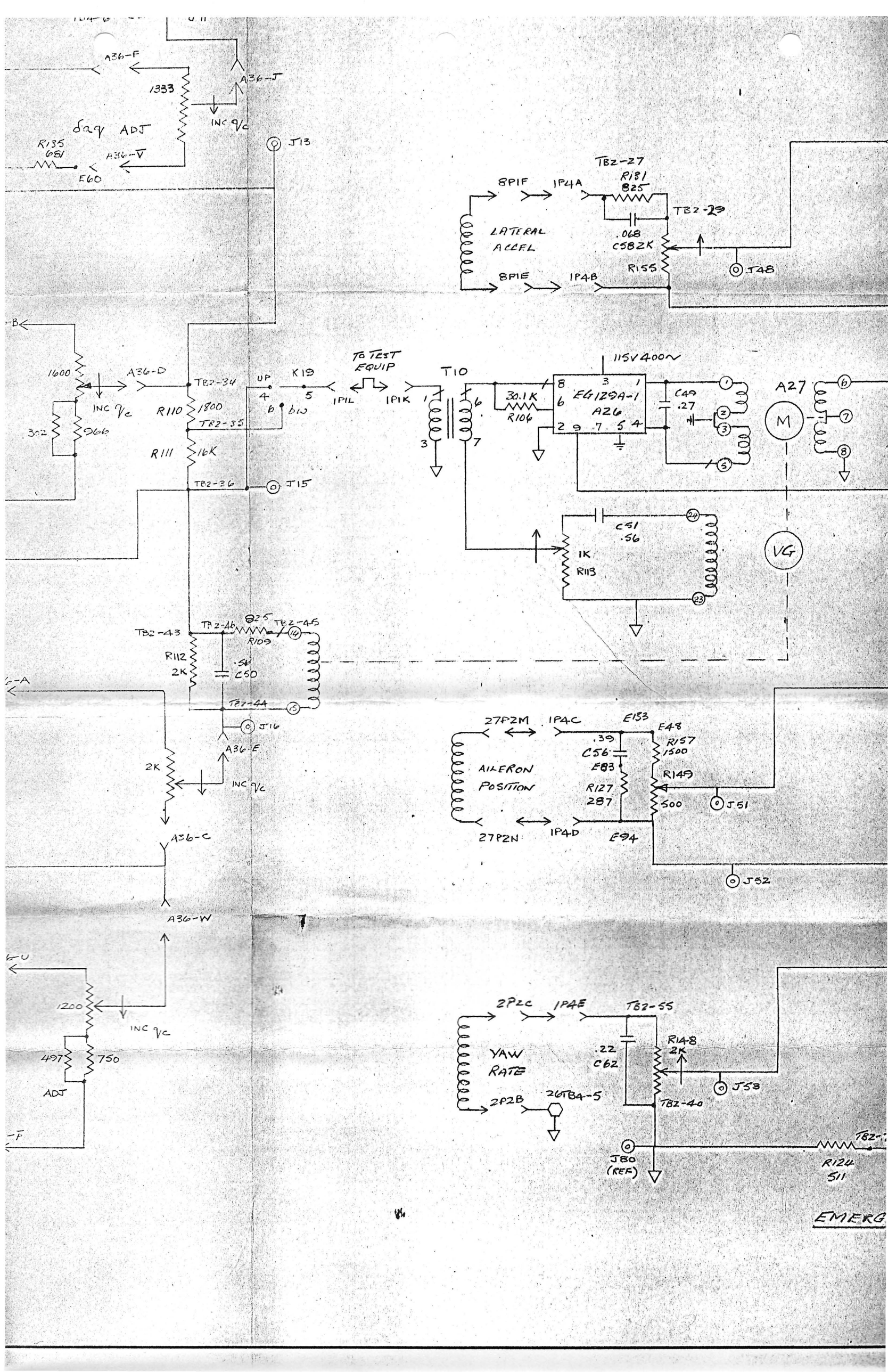
R144
J76
A36-DD
9
500
1800
T13
8
2.4 CT
INC ¹/₆
7
500
R143
A36FF
J77
A35-3
A36-BB
¹/₆ SENSOR PG 29 A35
1P4J To TEST
A35-2
A35-8
1P4m
27P4C
TO HINGE MOMENT LIMITER
A35-9
1P4k
27P4D

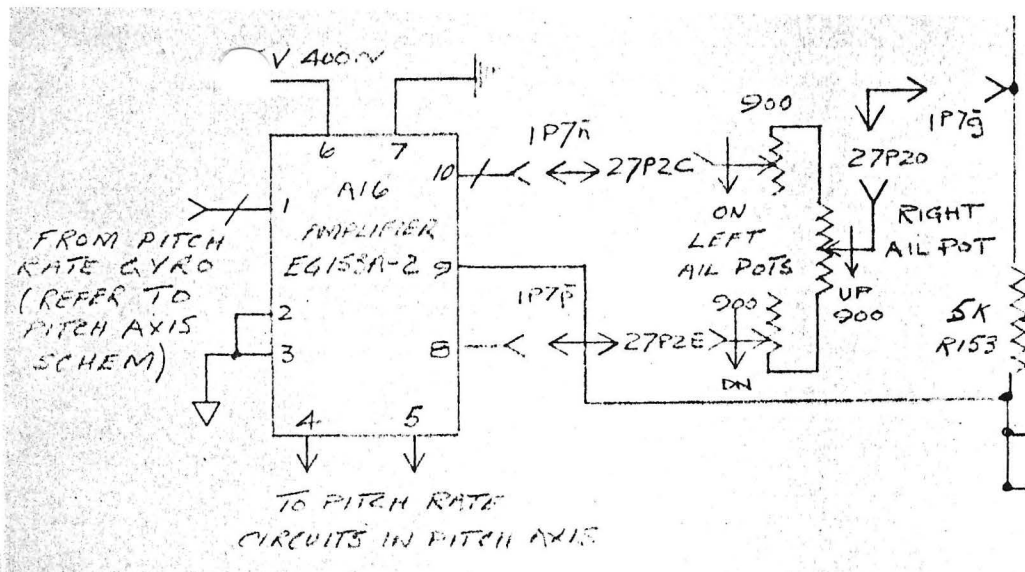
CSK 1778

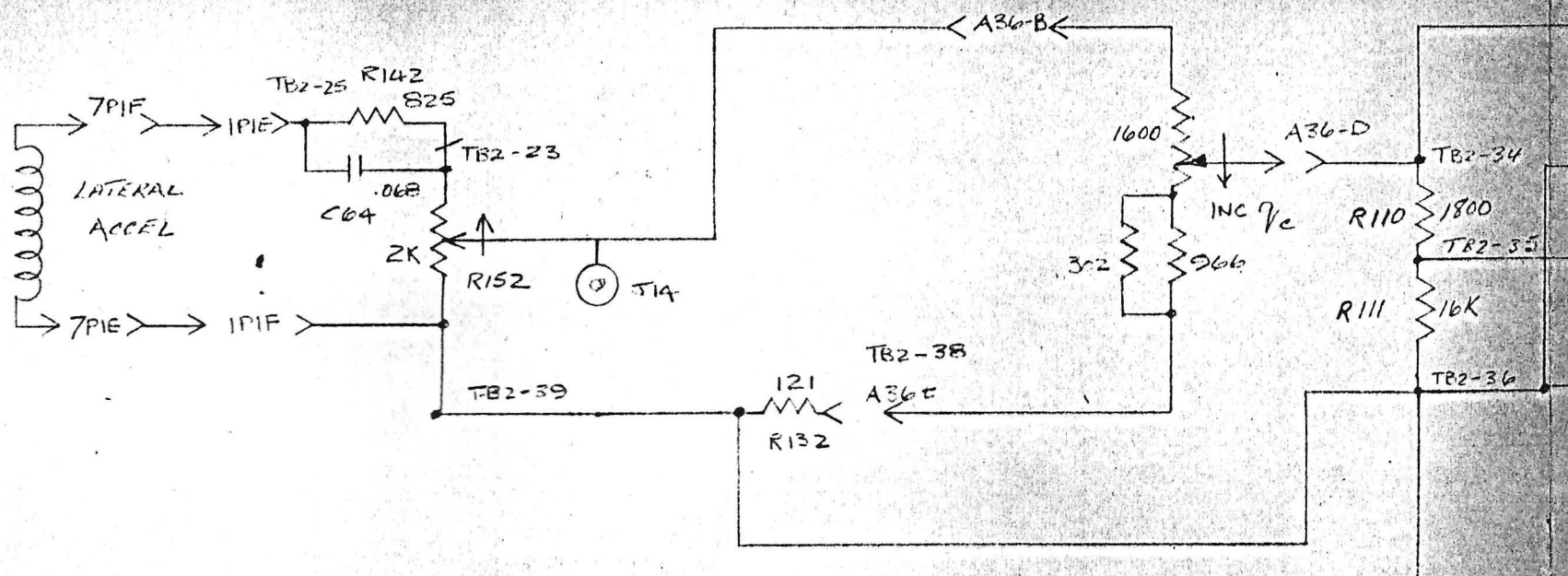

(EXTEND FOR 'R' RUDDER)
(EXTEND FOR 'R' RUDDER)
RUDDER TRANSFE
115V 400~
18P2G
1P2A
SERVO BALANCE
18P2C
1P55
R44 147
E55
E54
R171 1620
TB4-8
R42 250
.68 C26
TB4-7
160 R40
E52
E53
18P2K
1P54
21.5K R41
A11 AG26A-1
115V (E)
A30-8
1830
A30-7
A30-9
TB2-54
INC CW
162 R123
TB2-52
1800 R125
TB2-51
16K R122
TBZ-27
R181 825
8P1F
1P4A
.068 C582K
TBZ-29
R155
8P1E
1P4B
J48
TO TEST EQUIP
1P1L
1P1K
T10
115V 400~
30.1K R106
EG129A-1 A26
C49 .27
A27
M
TBZ-69
28.7K R107
TBZ-87
100 R108
C51 .56
1K R113
VG
TB2-50
87.5 R117
R118 2K
.56 C53
TB2-61
J50
A30-2
27P2M
1P4C
E133
E48
.39 C56
E83
R127 287
R157 1500
R149 500
J51
27P2N
1P4D
E94
J52
A30-1
2000
INC CW
A30-3
A30-4
A30-5


E52
E53
A30-8
1830
TB2-27
R131
825
8P1F
1P4A
TB2-23
LATERAL
ACCEL
.068
C582K
R155
8P1E
1P4B
J48
A30-7
TB2-52
A30
TB2-54
INC Vc
R125
1800
162
R123
TB2-51
R122
16K
TO TEST
EQUIP
1P1L
1P1K
T10
115V 400~
30.1K
R106
EG1294-1
A26
C49
.27
A27
M
TB2-69
28.7K
R107
TB2-87
100
R108
C51
.56
1K
R113
VG
TB2-53
825
R117
R118
2K
C53
.56
TB2-61
J50
27P2M
1P4C
E153
E48
.39
C56
R157
1500
E83
R149
AILERON
POSITION
R127
287
500
J51
27P2N
1P4D
E94
J52
A30-2
A30-1
2000
INC Vc
A30-3
A30-4
A30-5
1500
INC Vc
2P2C
1P4E
TB2-55
.22
C62
R148
2K
YAW
RATE
J53
2P2B
26TB4-5
TB2-40
TB2-78
J80
(REF)
R124
511
A30-6

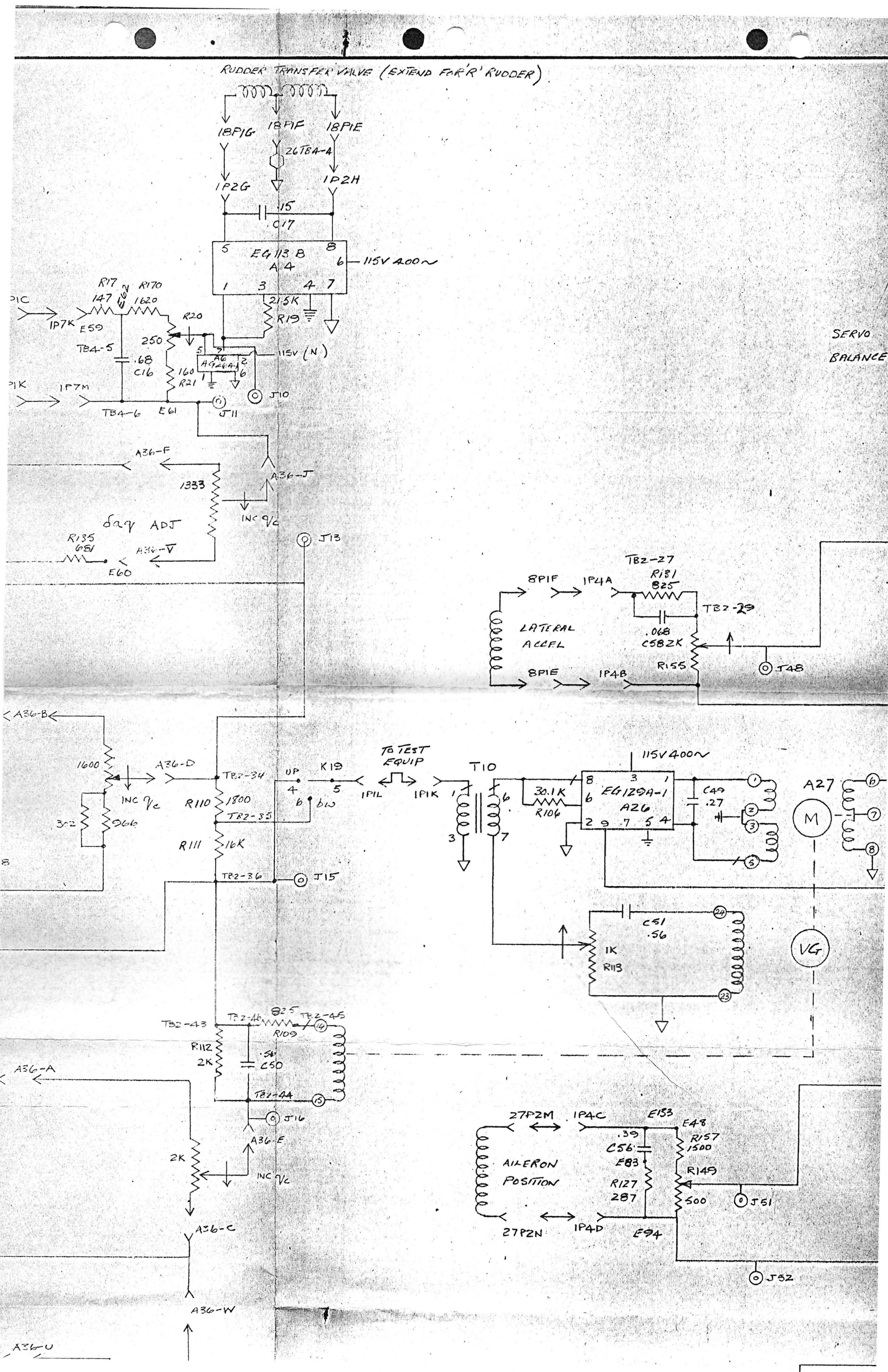
EMERGENCY


A36-F
1333
A36-J
INC Vc
δaγ ADJ
R135
681
A36-V
E60
J13
TB2-27
R181
825
8P1F
1P4A
TB2-29
LATERAL
ACCEL
.068
C582K
R155
J48
8P1E
1P4B
1600
A36-D
TB2-34
UP
K19
TO TEST
EQUIP
T10
115V 400~
INC Vc
R110
1800
TB2-35
4
5
1P1L
1P1K
30.1K
R106
EG129A-1
A26
C49
.27
A27
M
362
966
R111
16K
TB2-36
J15
C51
.56
1K
R113
VG
TB2-43
TB2-46
825
TB2-45
R109
R112
2K
.5
C50
TB2-44
J16
A36-E
2K
INC Vc
A36-C
A36-W
27P2M
1P4C
E153
E48
.39
C56
E83
R157
1500
AILERON
POSITION
R127
287
R149
500
J51
27P2N
1P4D
E94
J52
1200
INC Vc
497
750
ADJ
2P2C
1P4E
TB2-55
YAW
RATE
.22
C62
R148
2K
J53
2P2B
26TB4-5
TB2-40
J80
(REF)
R124
511
EMERG


V 400~
A16
AMPLIFIER
EG158A-2
6
7
10
1
9
2
3
8
4
5
FROM PITCH RATE GYRO (REFER TO PITCH AXIS SCHEM)
TO PITCH RATE CIRCUITS IN PITCH AXIS
1P7R
27P2C
900
ON
LEFT AIL POTS
1P7P
27P2E
900
DN
1P7S
27P20
RIGHT AIL POT
UP 900
5K
R153
J12
A36-F
1333
δaγ ADJ
R135
68Ω
A36-V
E60
TB4-6
E61
J11
INC γc


7P1F
1P1E
TB2-25
R142
825
TB2-23
.068
C64
LATERAL ACCEL
2K
R152
J14
7P1E
1P1F
A36-B
1600
INC γc
A36-D
TB2-34
R110
1800
TB2-35
3K2
966
R111
16K
TB2-36
TB2-38
121
R132
A36-E
TB2-39
TB2-43
TB2-46
R112
2K
C50
A36-A
27P2H
1P1H
TB2-9
E57
AILERON POSITION
R141
287
C63
.39
R158
1500
R151
500
27P2J
1P1G
TB2-21
J17
J18
2K
A36-C
A36-W
A36-U
1200
INC γc
497
750
ADJ
2P1C
1P1J
TB2-37
.22
C59
YAW RATE
R146
2K
2P1B
26TB4-5
J45
TB2-40
J80
(REF)
TB2-79
TB2-80
A36-P
R136
511

NORMAL


RUDDER TRANSFER VALVE (EXTEND FOR 'R' RUDDER)
18P1G
18P1F
18P1E
26TB4-4
1P2G
1P2H
.15
C17
E4113 B
A4
115V 400~
R17
147
E62
R170
1620
R20
250
TB4-5
.68
C16
160
R21
21.5K
R19
115V (N)
A6
A6A6A1
J10
1P7K
E59
1P7M
TB4-6
E61
J11
A36-F
1333
A36-J
INC Vc
δaγ ADJ
R135
681
A36-V
E60
J13
SERVO
BALANCE
TB2-27
8P1F
1P4A
R181
825
TB2-29
LATERAL
ACCEL
.068
C582K
R155
8P1E
1P4B
J48
A36-B
1600
A36-D
INC Vc
TB2-34
R110
1800
TB2-35
R111
16K
TB2-36
J15
UP
K19
4
5
6
bw
TO TEST
EQUIP
1P1L
1P1K
T10
115V 400~
30.1K
R106
EG1294-1
A26
C49
.27
A27
M
C51
.56
1K
R113
VG
TB2-43
TB2-46
825
TB2-45
R109
R112
2K
C50
.56
TB2-44
J16
A36-A
A36-E
2K
INC Vc
A36-C
A36-W
A36-U
27P2M
1P4C
E153
E48
.39
C56
E83
R157
1500
R149
R127
287
500
J51
AILERON
POSITION
27P2N
1P4D
E94
J52

RUDDER

18P1G

1P2G

5 EG

1 3

SERVO BALANCE

18P1C

1P7K

R17 147 E62

R170 1620

E59

TB4-5

.68 C16

R20 250

5 7 AG A6 AG2A-1 2 6 1

160 R21

18P1K

1P7M

TB4-6

E61

J11

J9

115V 400~

6 7 A16 AMPLIFIER EG155A-2 1 10 9 2 3 8 4 5

1P7R

27P2C

900

ON LEFT AIL POTS

27P20

RIGHT AIL POT

UP 900

DN

1P7P

27P2E

900

5K R153

J12

FROM PITCH RATE GYRO (REFER TO PITCH AXIS SCHEM)

TO PITCH RATE CIRCUITS IN PITCH AXIS

A36-F

1333

INC Vc

δaq ADJ

R135 681

A36-V

E60

A36-B

1600

A36-D

INC Vc

TB2-34

R110 1800

TB2-35

3.2

966

R111 16K

TB2-36

TB2-38

A36-E

121 R132

TB2-39

7P1F

1P1E

TB2-25 R142 825

TB2-23

.068 C64

LATERAL ACCEL

2K R152

J14

7P1E

1P1F

TB2-43

TB2-46

R112 2K

C5

A36-A

27P2H

1P1H

TB2-9 E57

R141 287

R158 1500

AILERON POSITION

C63 .39

27P2J

1P1G

TB2-21

R151 500

J17

J18

2K

A36-C

A36-W

A36-U

2P1C

1P1J

TB2-37

PITCH AXIS

A. General

1. The damper pitch axis actuates two differential servos, one for each elevator surface. Action of the differential servos does not produce any stick movement in the cockpit.

2. Provision is made in the pitch axis for the introduction of command signals which may be initiated by the pilot or may be produced by the Automatic Flight Control System (AFCS). One parallel servo is used to move both elevator surfaces in response to command signals or long term accelerations.

3. An electrical trim system which also acts as a synchronizing system prior to engagement is incorporated in this axis.

B. The attached diagrams show the build up of the roll axis in block diagram form.

1. Pitch Rate, Normal Acceleration

For damping action, the following control signals are desirable:

a) Pitch Rate
b) Rate of change of Pitch Rate
c) Normal acceleration

Diagram one shows the pitch axis with no command signal inputs and no trim.

A signal proportional to pitch rate is obtained from the pitch rate gyro. In the diagram this signal is applied to an integrator which is connected to act as a high pass filter. The purpose of this filter is to prevent the gyro signal from producing opposition to steady state pitch rates. In general, a steady state pitch rate will be set up when the pilot or AFCS commands it. Opposition to steady state rates caused by commands is not desirable. Transient pitch rates, will continue to pass through the filter at all times and produce damping action by the differential servos.

Since the normal accelerometer is located some distance forward of the centre of gravity of the aircraft, it will sense rate of change of pitch rate (pitch angular acceleration) as well as normal acceleration. Since rate of change of pitch rate signals are always transient in nature, high pass action is not necessary. Any normal accelerations which are not the result of commands, will be opposed as well by signals produced by the normal accelerometer.

2. Control Stick Steering

Diagram two shows the addition of control stick steering by which the pilot may command normal accelerations to change the pitch attitude of the aircraft. The parallel servo circuits, including an integrator are added.

When the pilot wishes to change pitch attitude, he will apply a longitudinal force to the stick. The stick will not move initially, but the force applied to the stick will produce an electrical command signal proportional to the force applied to the stick which is fed to the differential servo amplifiers and to the integrator input. The differential servos will move immediately to deflect the elevators and initiate a pitch acceleration. The integrator output will build up at a rate proportional to the integrator input signal amplitude and the parallel servo will move in a direction to produce the commanded normal acceleration.

As the aircraft changes pitch attitude, the pitch rate gyro will produce an output signal which will oppose the change of pitch attitude. However, when the pitch rate becomes steady, the high pass filter will prevent this signal from acting.

As the same time, a signal from the normal accelerometer will build up in a direction which will oppose the increase of normal acceleration acting on the aircraft. When the normal accelerometer output is equal to the command signal, the two signals will cancel, and the input to the differential servo amplifiers will be zero. The differential servos will return to the neutral position. The input to the integrator in the parallel servo circuits will also be zero, however, the integrator output does <u>not</u> go to zero. Its rate of increase will go to zero. This means that the parallel servo will maintain the elevator surfaces deflected in the correct amount to maintain the accelerometer output signal equal to the command signal. Since the command signal is proportional to stick force, the resultant normal acceleration must be proportional to stick force.

NOTE: In practice, the pilot will think that he has moved the stick when he applies a force to it. In actual fact, it will be the parallel servo which moves the stick. The greater the force applied, the greater will be the normal acceleration produced. This method then produces electrical "feel" without the use of springs or tim motors.

The student will notice in diagram two that high passed pitch rate is also applied to the integrator input. For flight path control it is desirable to have a lagged pitch rate signal acting during commanded pitch attitude changes.

The transfer function of the high pass is:

$$\frac{TS}{1 + TS}$$

The combined transfer function of both the high pass and integrator is:

$$\frac{TS}{1 + TS} \times \frac{1}{TS} = \frac{1}{1 + TS}$$

This is the transfer function of a low pass filter which will lag the input pitch rate by factor determined by its time constant T. It should be remembered that steady state pitch rates will not pass through this system in spite of the fact that it is a low pass filter.

By following a similar discussion to that above, it can be seen that if the pilot releases the force on the stick, the accelerometer signal which has been established will remain as the only input to the system. This signal will produce differential servo action to oppose the normal acceleration, and at the same time, will start the integrator on its return to the neutral position. The aircraft will then stabilize at a new pitch attitude with the stick returned to the neutral position. In practice, the pilot will hasten the establishment of a stable pitch attitude by applying a force to the stick in the opposite direction to that he used in initiating the attitude change.

During normal damping action, (no command signals acting) the output of the normal accelerometer will be applied to the integrator input as well as to the differential servo amplifiers. Under normal conditions normal acceleration signals will be transient in nature and the average of these signals at the integrator input will be zero. If the aircraft experiences a normal acceleration which is long term in nature, the integrator output will move the parallel servo to produce opposition to this error signal. When the acceleration has reduced to zero a normal velocity may still exist. The integrator input will be zero but the integrator output will not be zero. Since the integrator integrates normal acceleration, its output will be proportional to normal velocity. The parallel servo will be actuated by this output to bring the normal velocity to zero.

3. Trim, Scheduling

Diagram three shows the addition of trim circuits and airspeed and altitude scheduling. The trim system consists of a motor which drives an inductive pickoff such that either positive or negative normal acceleration signals may be produced. When the pilot actuates the stick trim button, power is applied to the motor and an electrical signal is produced. The amplitude of the signal will depend on the length of time the pilot holds the trim switch closed and its phase will depend on the direction he has caused the motor to run. The trim signal will act in the

same manner as a command from the control stick. The pilot can use the trim button to either set up a fixed normal acceleration or to trim out an undesirable normal acceleration due to aircraft mistrim.

Since control surface effectiveness varies with airspeed, both the pitch rate gyro signal and the normal acceleration signal (including C.S.S., trim) which are applied to the differential servos are scheduled with airspeed. As described in an earlier section, normal acceleration during a flight path change is proportional to the square of the velocity of the aircraft. By scheduling commands with both airspeed and altitude, an approximation of true airspeed scheduling is achieved.

4. Gear Down

When the pilot wishes to land, it is desirable that the control action of the stick be changed. Stick commands should now produce surface displacement proportional to stick force rather than normal accelerationsproportional to stick force. Normal rate damping action should remain the same. Diagram four shows the pitch axis in the gear down configuration. Several changes are apparent. The pitch rate signal is no longer high passed since steady state pitch rates should be opposed during landings. Normal acceleration error signals have been removed from the differential servo input by the action of a fader motor potentiometer which drives to the faded-out end when the gear is lowered. Normal acceleration, CSS, trim, and pitch rate signals have all been removed from the parallel servo integrator input. Any control stick steering signal is fed, uncompensated by airspeed or altitude directly to the parallel servo amplifier input. Surface displacement will be proportional to stick force. It will be noted that this CSS signal is introduced to the parallel servo amplifier by the action of a fader motor potentiometer which runs to the faded-in end when the gear is lowered. This is the same fader motor used to fade out the normal acceleration signal. This fading action is used to avoid any bump which might result if signals were switched by relay action.

In the gear down position, the pilot's trim button no longer drives the normal trim system. When the trim button is actuated, a fixed electrical input is applied to the integrator. The integrator output will increase at a fixed rate until the stick button is released. The integrator output will remain at the value it had reached when the stick button was released.

The student should not confuse this trim system or the gear up trim system with the Avro trim system which is used when the pitch damper is not engaged. The Avro trim system, actuated by the same trim button acts directly and mechanically on the elevator control quadrant.

5. Pre-Engage

Diagram five shows the pitch axis prior to engagement. Any output signals from the pitch rate gyro, normal accelerometer,

or CSS, will be amplified and used to operate the trim motor in a direction which will produce a trim output of the correct phase and amplitude to cancel the amplifier input signal. The input to the differential servo amplifiers is held close to zero by this synchronizing action. Hydraulic pressure is removed from the differential servos and the servos are spring loaded to centre during the pre-engage mode. As described previously, the pilot's trim button now operates the Avro pitch trim system.

Hydraulic pressure is removed from the parallel servo, but unlike the differential servos, it is not spring loaded to centre. Since the parallel servo ram is attached to the front quadrant, it must move when the pilot moves his control surfaces. When the pilot deflects his control surfaces the parallel servo will move, producing a feedback balance signal to the servo amplifier input. Since no other signal is present at this point, the balance signal will cause the moog valve to operate the produce a pressure in one of the two hydraulic lines to the parallel servo. Since these lines are not ported to the servo when the system is not engaged, no detrimental action is produced. However, if the system were to be engaged in this condition, a severe bump would be felt as this pressure was applied to the servo. In addition, the servo amplifier would be in a saturated condition.

A pressure trim sensor which produces a signal whenever the pressures in the two moog valve lines are different is used to rectify this situation. When the control surfaces are deflected and the system is in the condition described above, a signal from the pressure trim sensor is fed back to the integrator input. The integrator time constnat is changed in the pre-engage condition so that its action is much faster and it can now be called a synchronizer. The sync output increases until it cancels the feedback balance signal fromthe servo, thus returning the moog valve to centre and reducing the pressure trim signal to zero. The sync output however, does not return to zero. The sync output rate of increase drops to zero and its output maintains the balance at the amplifier input. As shown on the diagram, an additional path from the pressure trim sensor directly to the amplifier input is supplied. This will aid the sync action when the error is first produced but will have no effect after balance has been achieved.

As an additional safeguard, the system cannot be engaged unless the pressure trim sensor is in the centre position (no signal output). DC power cannot be applied to the engaging circuits of this axis unless it can pass through contacts of a sector switch which is closed only when the pressure trim sensor is centered. (Refer to DC engaging sequence and parallel servo schematic.)

6. AFCS - Diagrams six and seven.

When the AFCS system is engaged, (which cannot occur unless the full damper is engaged) the AFCS pitch output is faded into the damper pitch axis and produces command action in the same manner as a control stick steering signal produced by

the pilot. As shown in the diagram, the trim signal has been removed from active control action. Its output is faded out as the AFCS output is faded in by a fader motor located in the AFCS system. Normal acceleration trim will not be lost, however. When the AFCS is in the pre-engage condition, it continuously monitors normal acceleration and produces a trim signal to balance it as part of the AFCS output signal. When the damper trim fades out, the AFCS trim fades in as part of the AFCS output signal.

The damper trim system is not inactive during the AFCS mode. It is used to monitor CSS and normal acceleration outputs and to produce an output equal to but of opposite phase to any signals from these two sources. If the AFCS system should disengage while a command signal from the AFCS system was present, a severe bump could be produced due to the loss of the AFCS signal. However, if the AFCS system disengages, the trim system output again becomes effective. Since its output is equal but opposite to the CSS and normal acceleration outputs, it will effectively replace the AFCS output signal and no bump will result.

If AFCS is used during the damper gear down mode, the AFCS output signal is applied directly to the parallel servo amplifier and will produce proportional surface movement in the same manner as a control stick command from the pilot. The remaining pitch axis circuits are the same as in the gear down mode without AFCS except that the trim system now monitors CSS and normal acceleration.

NOTE: The pilots trim button will be inoperative during the AFCS mode of operation.

C. Function Block Diagram

Four additional diagrams are included to show the means by which the diagrams discussed above are actually mechanized.

D. Automatic Trim

If an aircraft mistrim condition develops during damper operation, the pilot may remove the condition by causing surface displacement by use of his trim button. If the AFCS system is being used, some modes of operation produce trimsignals to remove aircraft mistrim. This means that an electrical signal from the damper is maintaining elevator surface deflection to maintain aircraft trim. If the pitch damper disengages (by pilot selection or by limiter action) the electrical trim is removed immediately and a severe bump may result. To avoid this condition, an automatic trim system is used.

The complete block diagram SK66501 shows the additional trim circuits.

If the elevator surface is held displaced by a damper signal, a continuous pressure will be held in one of the two moog valve lines to the parallel servo to oppose the airstream loading on the surface. The pressure trim sensor (discussed above in the pre-engage mode) will be displaced from the centre position. Whenever this sensor is displaced from the centre position, it closes a sector switch which will apply power to the Avro trim actuator. The actuator changes the tension of a feel spring so that the spring will oppose the surface loading and release the parallel servo trim this function. The parallel servo will remain displaced, but the pressure in the moog valve lines will return to balance and the pressure trim sensor will return to centre. With the automatic trim system operating, the removal of damper trim signals will not produce a violent bump since the Avro trim spring will maintain the trim action.

As a safeguard, a normal accelerometer sector switch is connected in series with the pressure trim sector switch. If normal acceleration on the aircraft exceeds 0.5g incremental, the sector switch opens and the automatic trim system is made inoperative. If the damper cannot hold the aircraft to within .5g limits it indicates that a malfunction may exist in the system or a gross aircraft mistrim condition exists. If the pilot commands greater than .5g however, the accelerometer switch is bypassed, and the trim actuator will continue to operate.

PITCH AXIS COMPONENTS

1. BG92B-1 Amplifier Calibrator (1)
2. GG72A-3 Pitch Rate Gyro (1)
3. GG58A-2 Normal Accelerometer (1)
4. GG62B-1 Normal Switching Accelerometer (1)
5. MG51D-2 Differential Servo (2)
6. MG55N-3 Parallel Servo (1)

AVRO SUPPLIED COMPONENTS

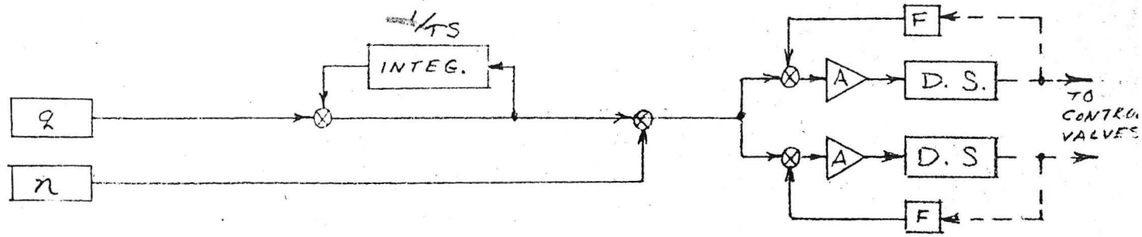
1. Stick Force Transducer
2. Trim Button

SETTINGS (Tentative)

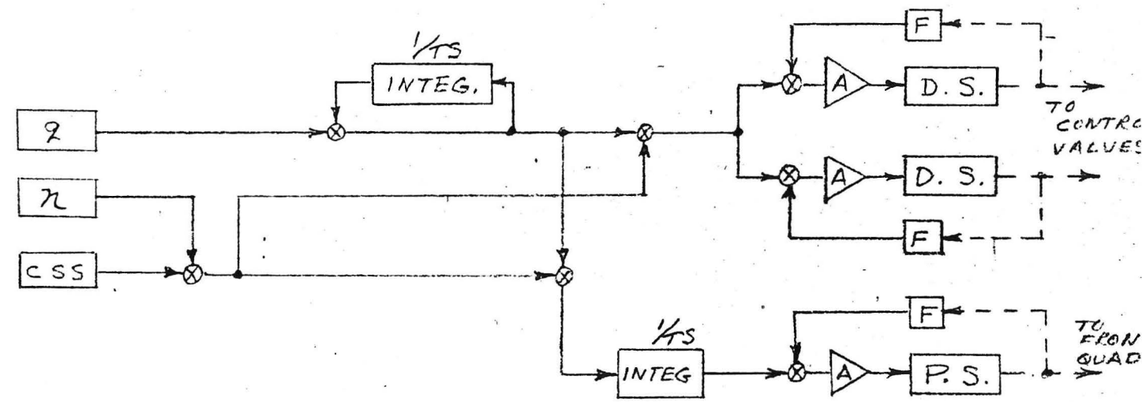
1.	Pitch Rate	R156	6.8
2.	Normal Acceleration	R154	7.0
3.	Control Stick	R145	7.0
4.	Differential Servo Balance	R8, R25	6.8 (4.8ft.)
5.	Parallel Servo Balance	R15	4.0

SCALE FACTORS AND GENERAL INFORMATION

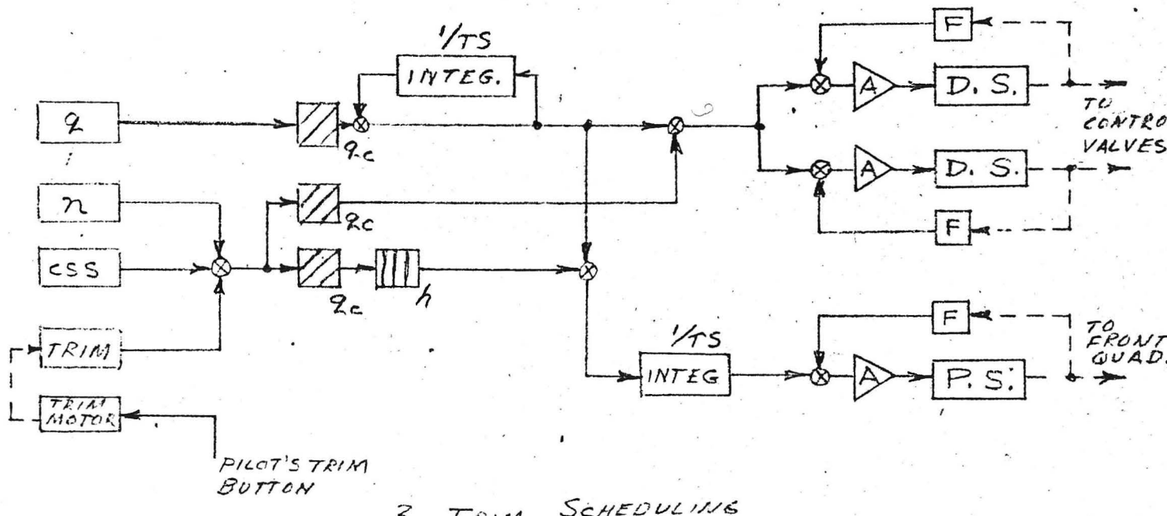
1.	Differential Servo Amplifier Threshold	13 mv
2.	Parallel Servo Amplifier Threshold	4.9 mv
3.	Integrator Rate	0.88 mv/sec/mv
4.	High Pass Time Constant	1.5 sec.
5.	Elevator Movement, Differential	±6 degrees
6.	Elevator Movement, Parallel	+30 - 20 deg.
7.	Differential Servo Ram Travel	±0.375 in.
8.	Parallel Servo Ram Travel Up	1.8 in.
9.	Parallel Servo Ram Travel Down	1.2 in.
10.	Normal Accelerometer (+6 to -4)	1 volt/g
11.	Pitch Rate	4V/20°/sec
12.	Control Stick (30 lbs. max.)	1/6V/lb.
13.	Servo Balance	5.5V/in.
14.	Trim Output Range	1 volt out of phase to 3 volts in phase
15.	Stop to stop trim motor time	57 seconds



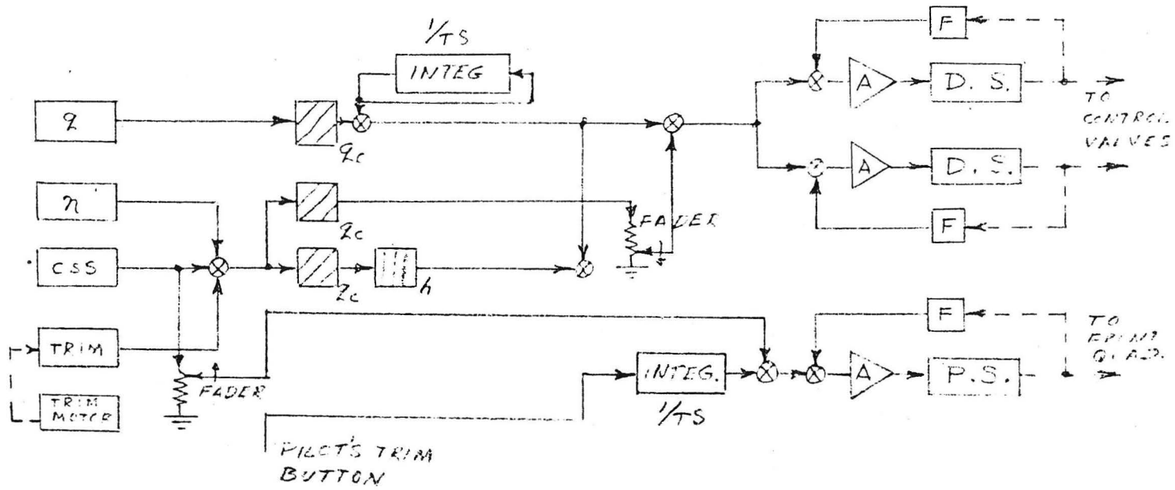
1. PITCH RATE, NORMAL ACCELERATION



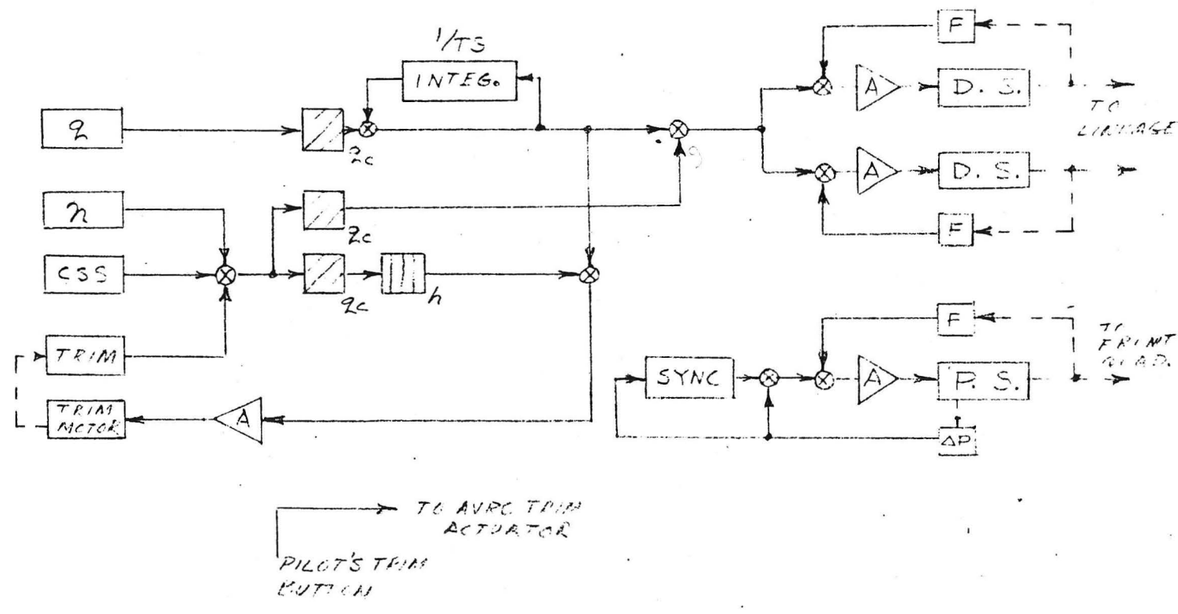
2. CONTROL STICK STEERING, $\int n$.



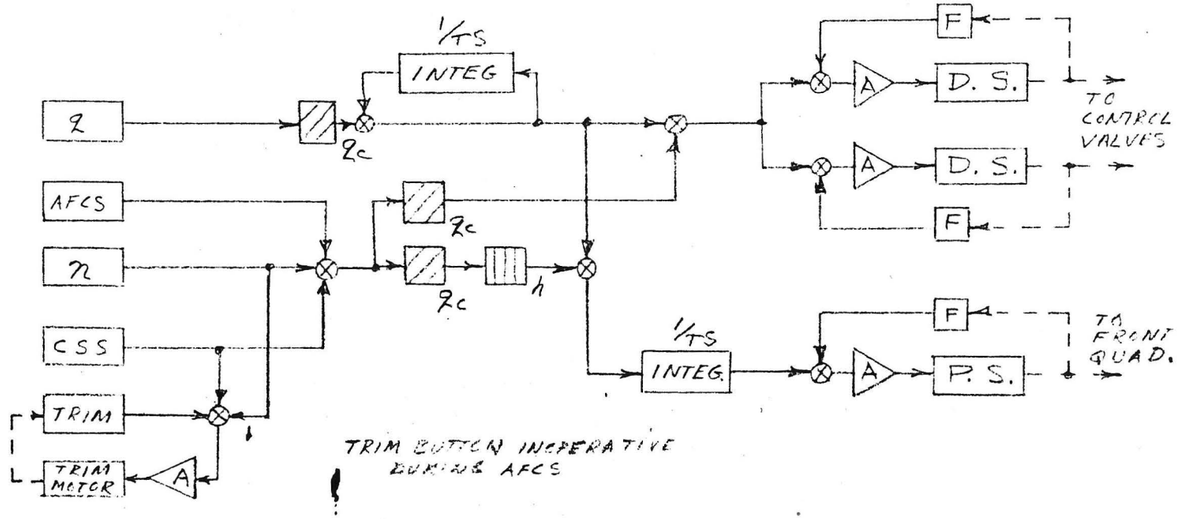
3. TRIM, SCHEDULING



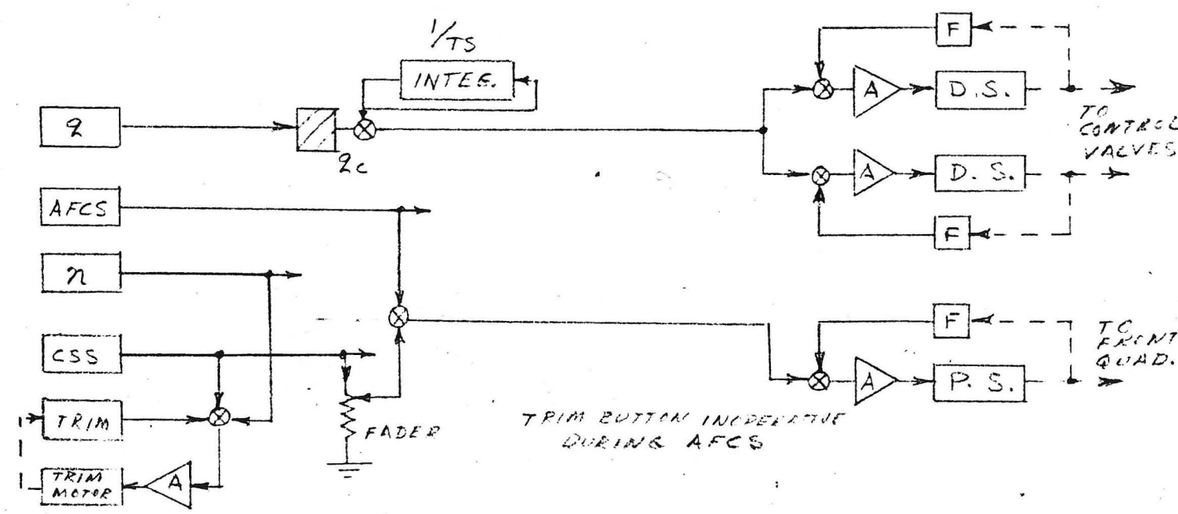
4. GEAR DOWN



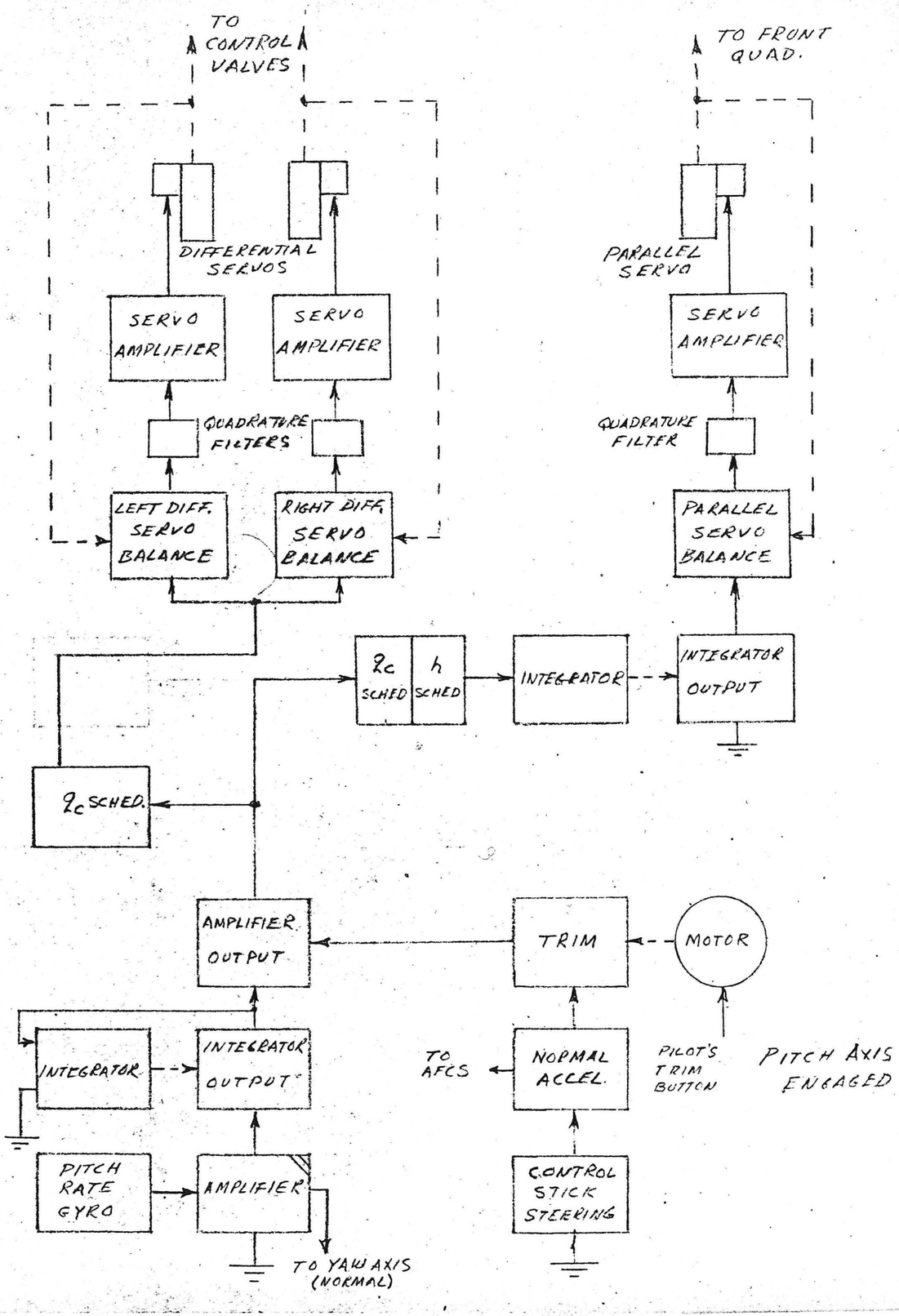
5. PFC FAILURE

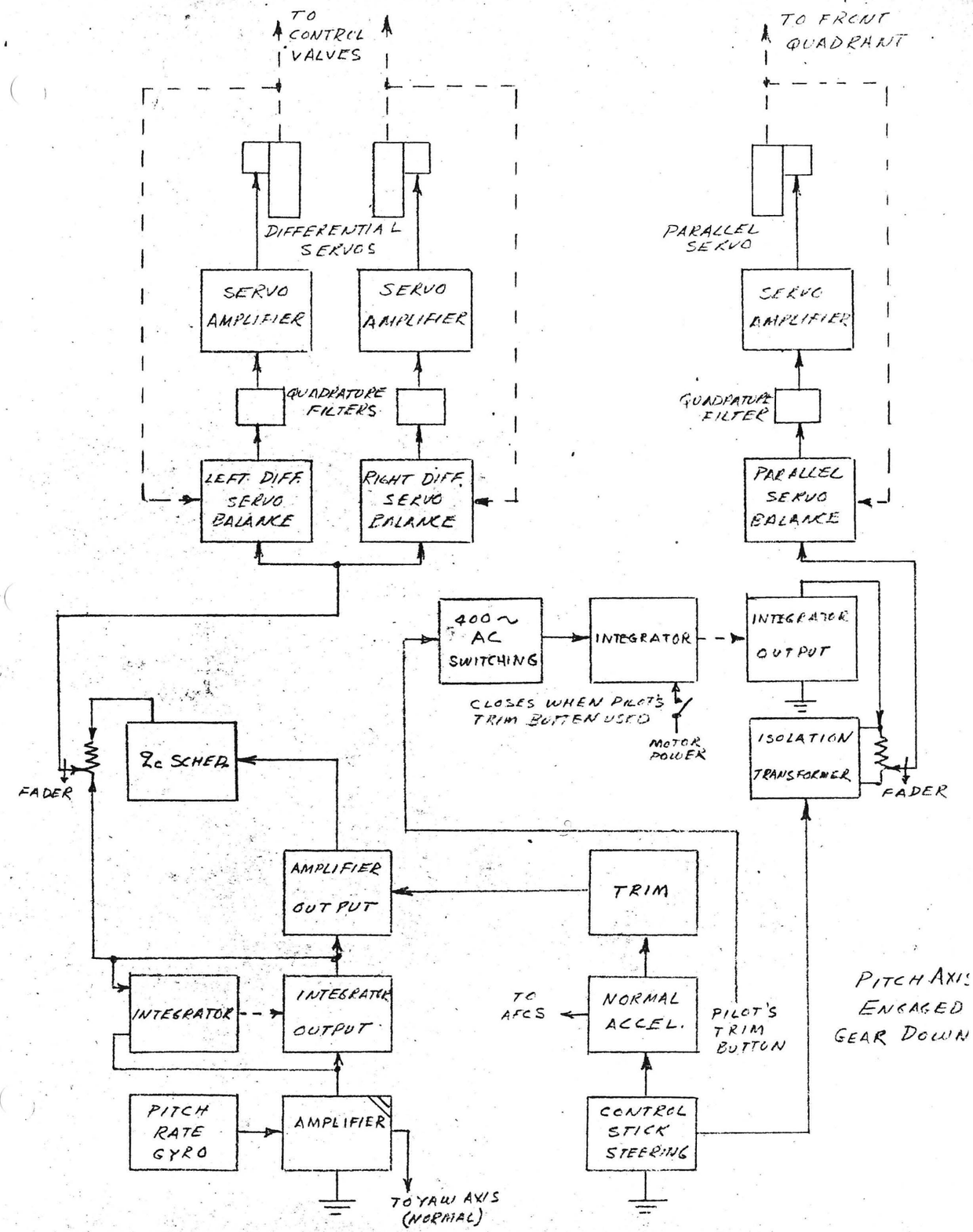


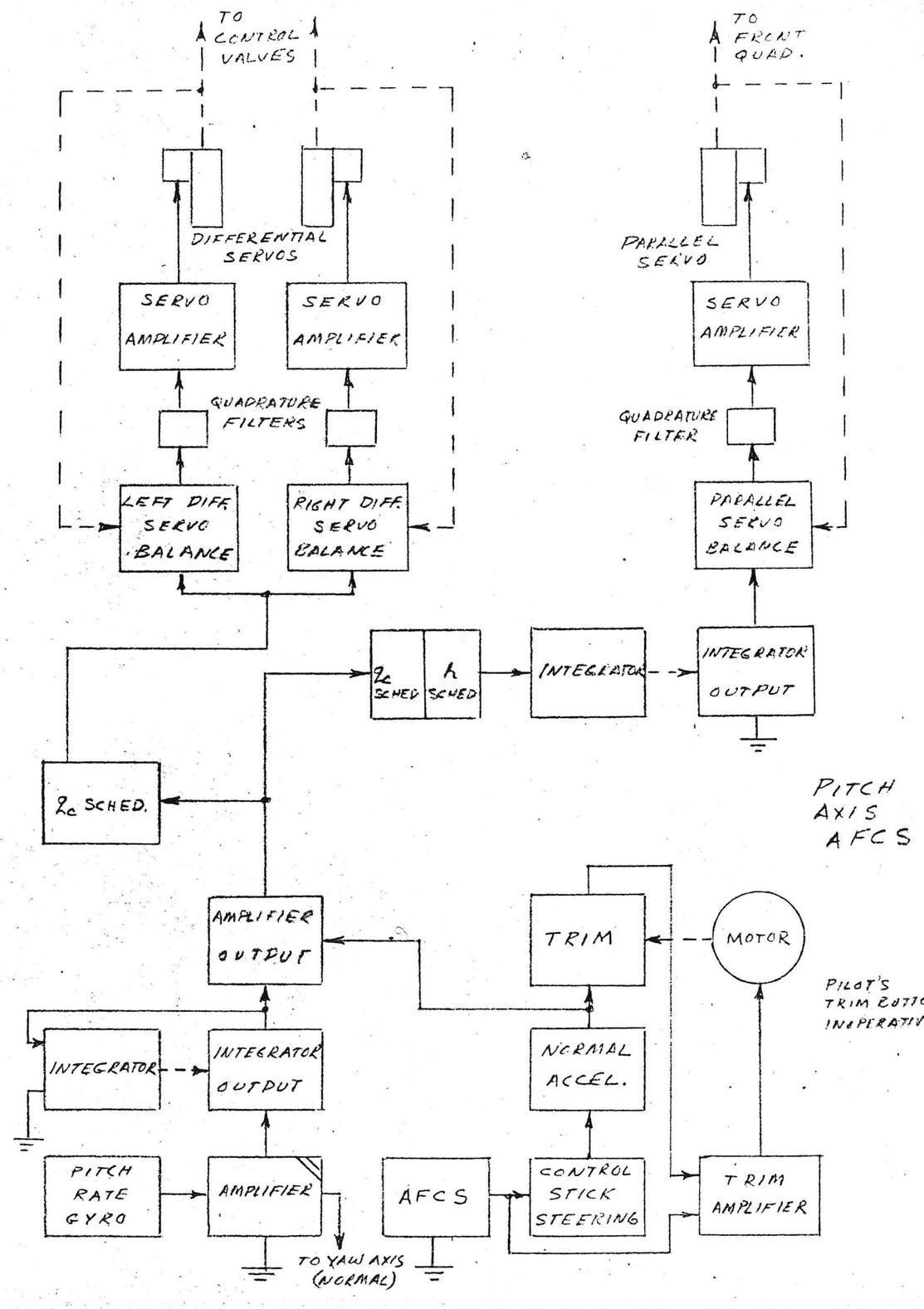
6. AFCS (GEAR UP)

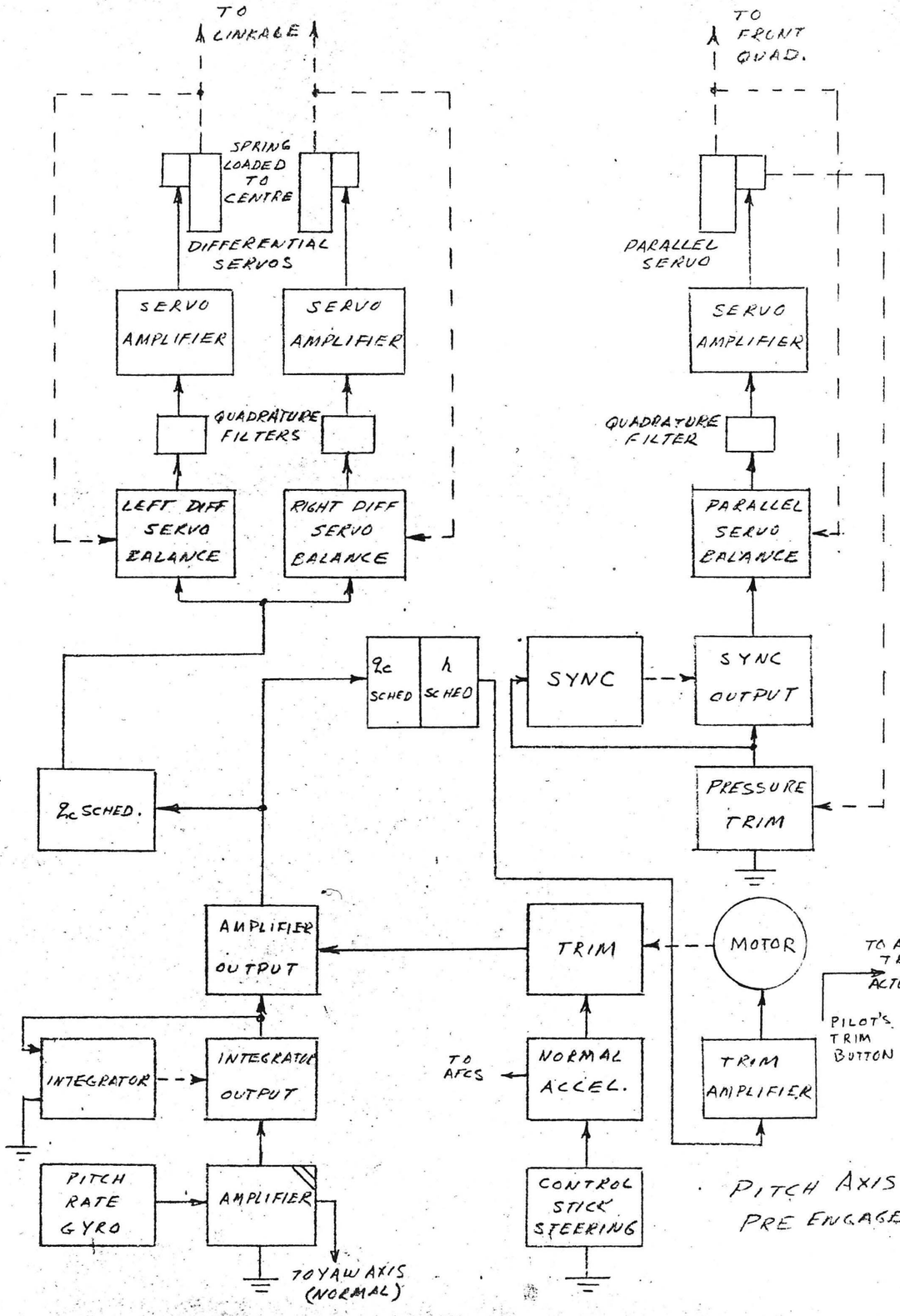


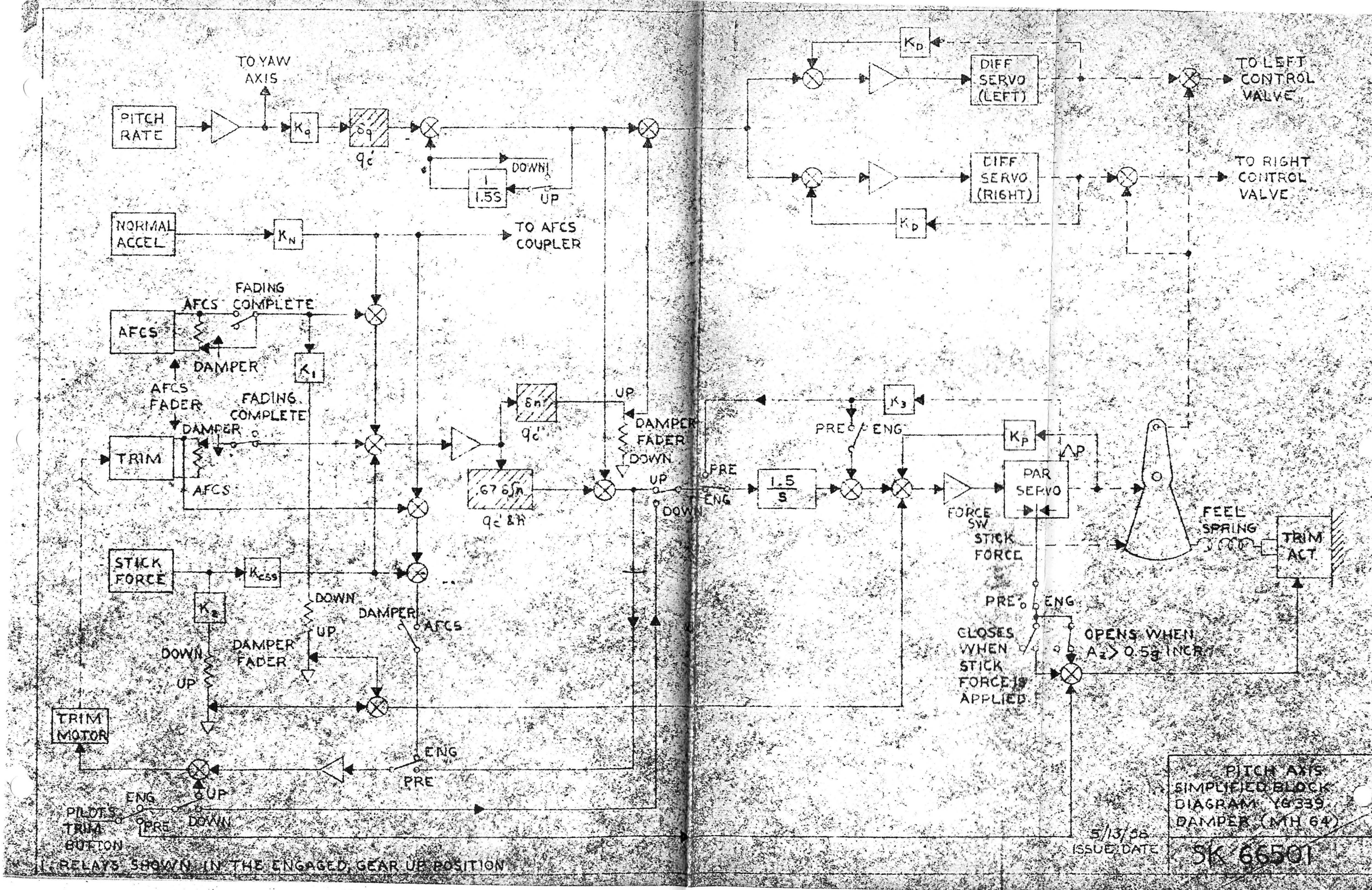
7. AFCS (GEAR DOWN)


TO CONTROL VALVES
TO FRONT QUAD.
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF. SERVO BALANCE
PARALLEL SERVO BALANCE
$\dot{q}_c$ SCHED
h SCHED
INTEGRATOR
INTEGRATOR OUTPUT
$\dot{q}_c$ SCHED.
AMPLIFIER OUTPUT
TRIM
MOTOR
INTEGRATOR
INTEGRATOR OUTPUT
TO AFCS
NORMAL ACCEL.
PILOT'S TRIM BUTTON
PITCH AXIS ENGAGED
PITCH RATE GYRO
AMPLIFIER
CONTROL STICK STEERING
TO YAW AXIS (NORMAL)


TO CONTROL VALVES
TO FRONT QUADRANT
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF. SERVO BALANCE
PARALLEL SERVO BALANCE
400~ AC SWITCHING
INTEGRATOR
INTEGRATOR OUTPUT
CLOSES WHEN PILOT'S TRIM BUTTON USED
MOTOR POWER
ISOLATION TRANSFORMER
FADER
2c SCHED
FADER
AMPLIFIER OUTPUT
TRIM
INTEGRATOR
INTEGRATOR OUTPUT
TO AFCS
NORMAL ACCEL.
PILOT'S TRIM BUTTON
PITCH RATE GYRO
AMPLIFIER
CONTROL STICK STEERING
TO YAW AXIS (NORMAL)
PITCH AXIS ENGAGED GEAR DOWN


TO CONTROL VALVES
TO FRONT QUAD.
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF. SERVO BALANCE
PARALLEL SERVO BALANCE
$\dot{q}_c$ SCHED
h SCHED
INTEGRATOR
INTEGRATOR OUTPUT
q_c SCHED.
PITCH AXIS AFCS
AMPLIFIER OUTPUT
TRIM
MOTOR
PILOT'S TRIM BUTTON INOPERATIVE
INTEGRATOR
INTEGRATOR OUTPUT
NORMAL ACCEL.
PITCH RATE GYRO
AMPLIFIER
AFCS
CONTROL STICK STEERING
TRIM AMPLIFIER
TO YAW AXIS (NORMAL)


TO LINKAGE
SPRING LOADED TO CENTRE
DIFFERENTIAL SERVOS
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
LEFT DIFF SERVO BALANCE
RIGHT DIFF SERVO BALANCE
TO FRONT QUAD.
PARALLEL SERVO
SERVO AMPLIFIER
QUADRATURE FILTER
PARALLEL SERVO BALANCE
q_c SCHED
h SCHED
SYNC
SYNC OUTPUT
q_{rc} SCHED.
PRESSURE TRIM
AMPLIFIER OUTPUT
TRIM
MOTOR
TO AVR TRIM ACTUATO
PILOT'S TRIM BUTTON
INTEGRATOR
INTEGRATOR OUTPUT
TO AFCS
NORMAL ACCEL.
TRIM AMPLIFIER
PITCH RATE GYRO
AMPLIFIER
CONTROL STICK STEERING
TO YAW AXIS (NORMAL)
PITCH AXIS PRE ENGAGE

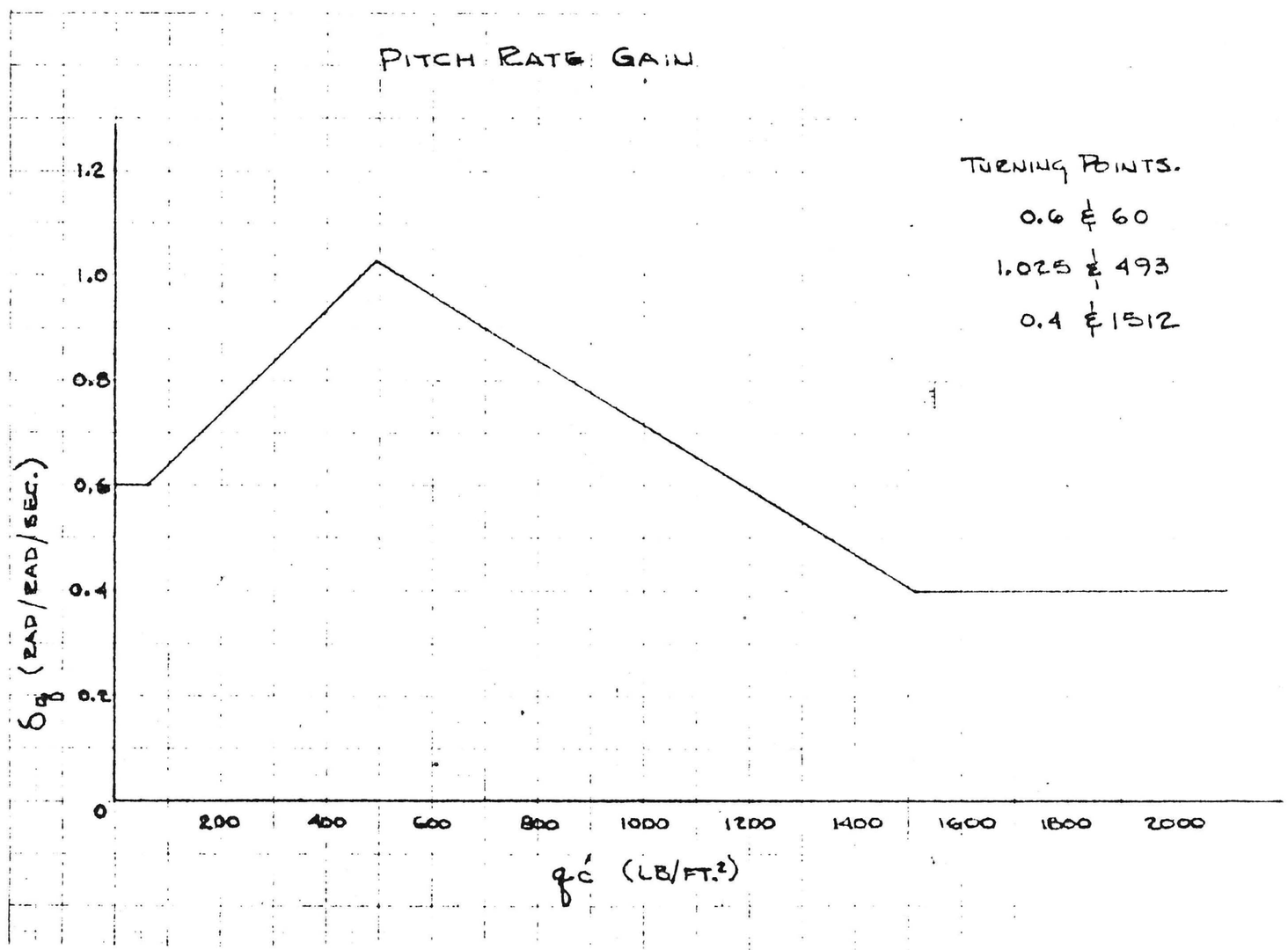


1. RELAYS SHOWN IN THE ENGAGED, GEAR UP POSITION

PITCH AXIS SIMPLIFIED BLOCK DIAGRAM. YG339. DAMPER (MH 64)

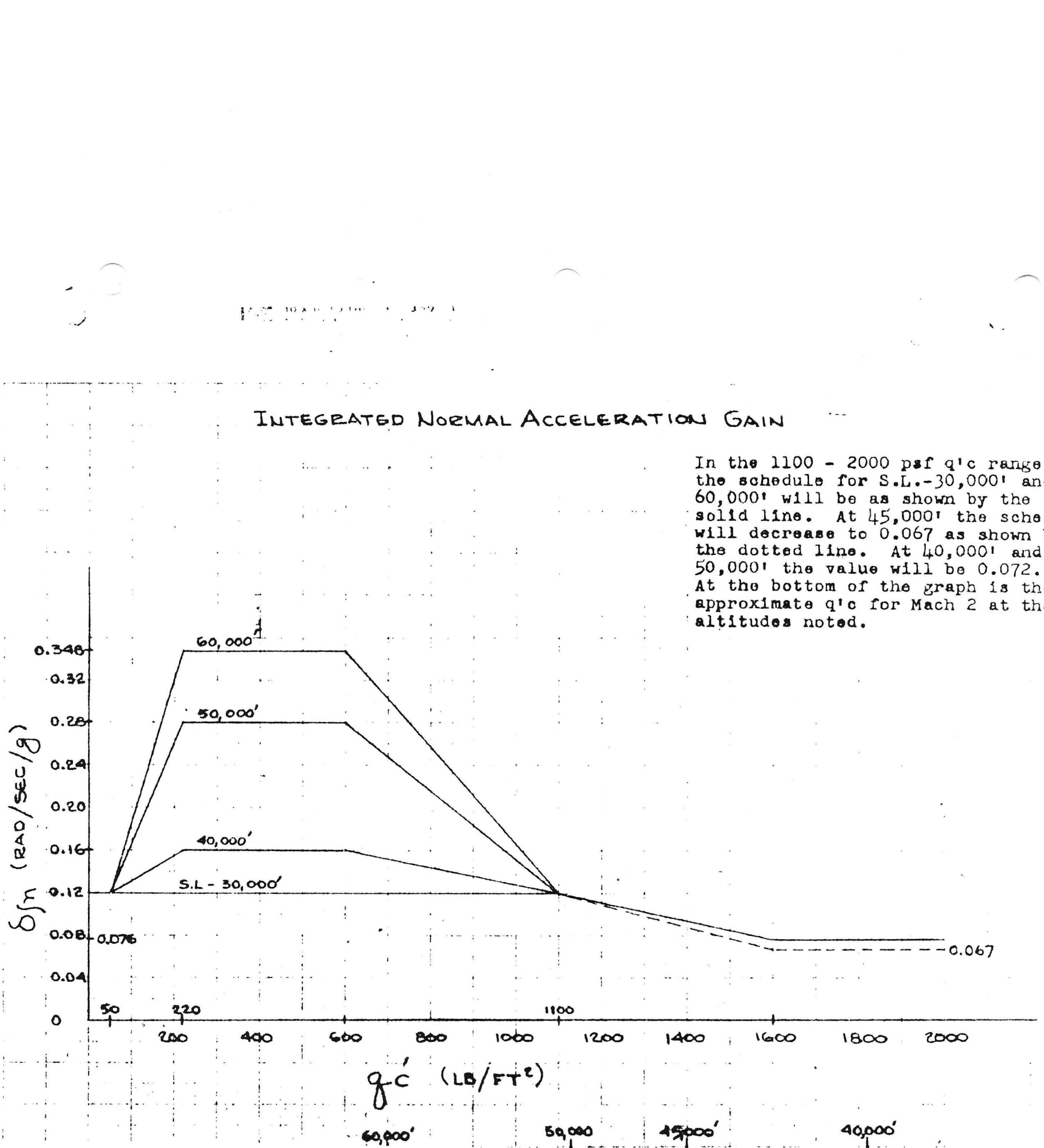
5/13/58
ISSUE DATE

5K 66501


PITCH RATE GAIN
TURNING POINTS.
0.6 & 60
1.025 & 493
0.4 & 1512
1.2
1.0
0.8
0.6
0.4
0.2
0
200
400
600
800
1000
1200
1400
1600
1800
2000
δ_{qc} (RAD/RAD/SEC.)
q_c (LB/FT.^{2})

Integrated Normal Acceleration Gain

In the 1100 - 2000 psf q'c range, the schedule for S.L.-30,000' and 60,000' will be as shown by the solid line. At 45,000' the schedule will decrease to 0.067 as shown by the dotted line. At 40,000' and 50,000' the value will be 0.072. At the bottom of the graph is the approximate q'c for Mach 2 at the altitudes noted.

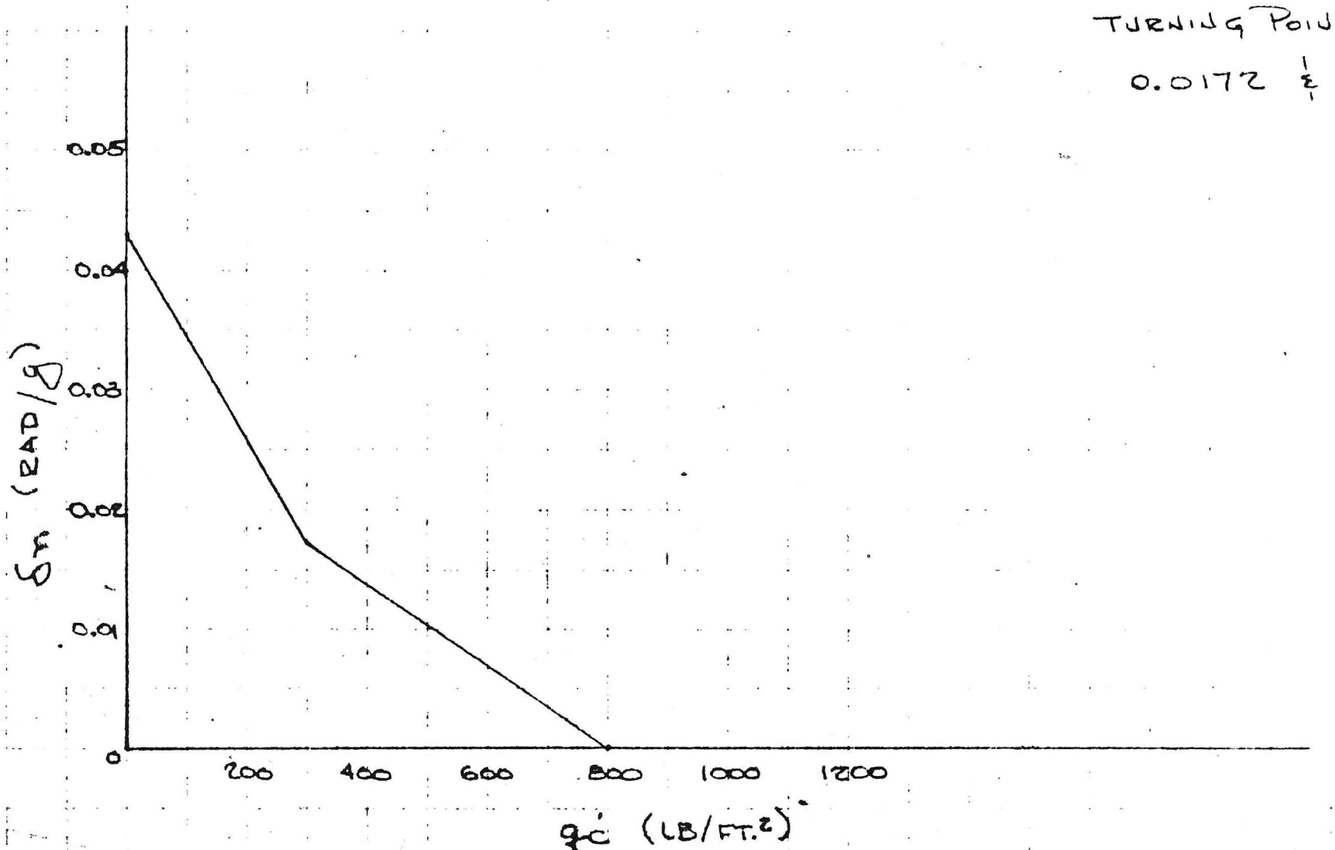


Grafh No 12 CEL 1961

NORMAL ACCELERATION GAIN

TURNING POINTS

0.0172 & 300



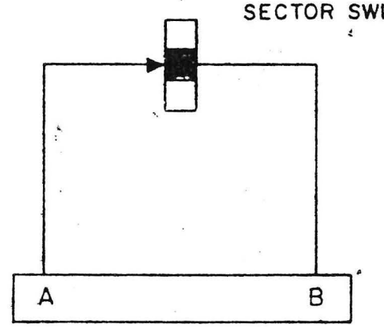
SUPPLEMENTARY INFORMATION		
FIRST USED GG 62 B-1		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	TITLE WAS "SCHEMATC DIAGRAM-SWITCH, ACCELERATION SENSITIVE	11-8-57	9A-1329

A

C7642

SECTOR SWITCH



J1 GS02-14S-5PX
CANNON

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±2°
		±%

DV 01196-32

UPSD

GOVT MATL SPEC

DRAWN BY	W. Lysyj	3-19-57
DRAFTING √		5-31-57
MATL & FIN. √		6-2-57
DESIGNER √		6-14-57
DESIGN √		5-31-57
APPROVED		6-26-57

REF MH MATL (SPEC)

FINISH (SEE NOTE)

SCHEMATIC DIAGRAM - ACCELEROMETER, NORMAL SWITCHING

SCALE NONE

WT ACT. CALC LB

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

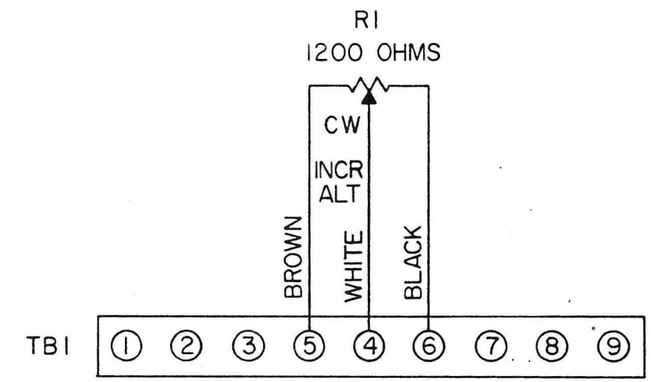
C7645

88-2788

SUPPLEMENTARY INFORMATION	
FIRST USED DPG27A-1	
NEXT ASSEMBLY	REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL

C7679



424590-19

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		+30 −10

DEV 01196-07

SD

T MATL SPEC

DRAWN BY	Bethke	8-23-57
DRAFTING	Sutherland	8-29-57
MATL & FIN.		
DESIGNER	RWJ	9-12-57
DESIGN	Sutherland	8-29-57
APPROVED		9-13-57

REF MH MATL (SPEC)

FINISH (SEE NOTE)

SCHEMATIC DIAGRAM – STATIC PRESSURE TRANSMITTER

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY

AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT ACT. CALC LB

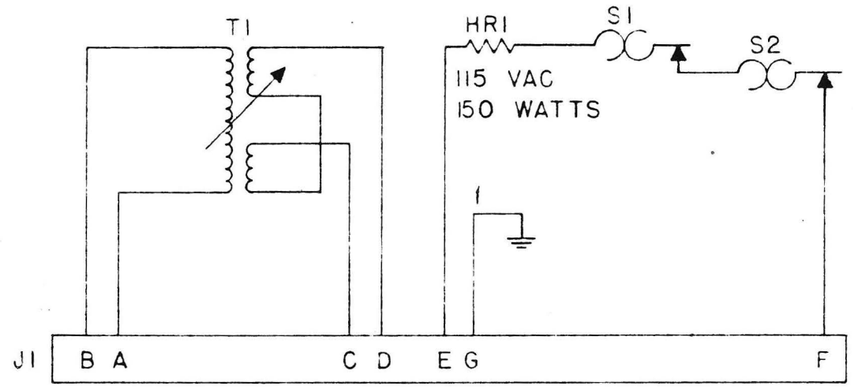
C7679

98-2799

SUPPLEMENTARY INFORMATION	
FIRST USED D-GG58A-2	
NEXT ASSEMBLY	REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL

C8869



CANNON
GS02-16S-1PW-111

2 - FOR +G OUTPUT, VOLTAGE AB & CD SHALL BE IN PHASE
1 - CASE CONNECTED TO PIN G FOR GROUNDING OF CASE

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	± 020
TWO PLACE	(.00)	± 010
THREE PLACE	(.000)	± 005
90° FORMED ANGLES		½°

↓ 2021-05-01

SUPSD

GOVT MATL SPEC

DRAWN BY	L. Johnson	4-14-58
DRAFTING	ANDRUSKY	4-17-58
MATL & FIN.		
DESIGNER	DJH	4/17/58
DESIGN	ANDRUSKY	4-17-58
APPROVED	RJH	4-17-58

REF MH MATL (SPEC)

FINISH (SEE NOTE)

SCHEMATIC DIAGRAM—
ACCELEROMETER

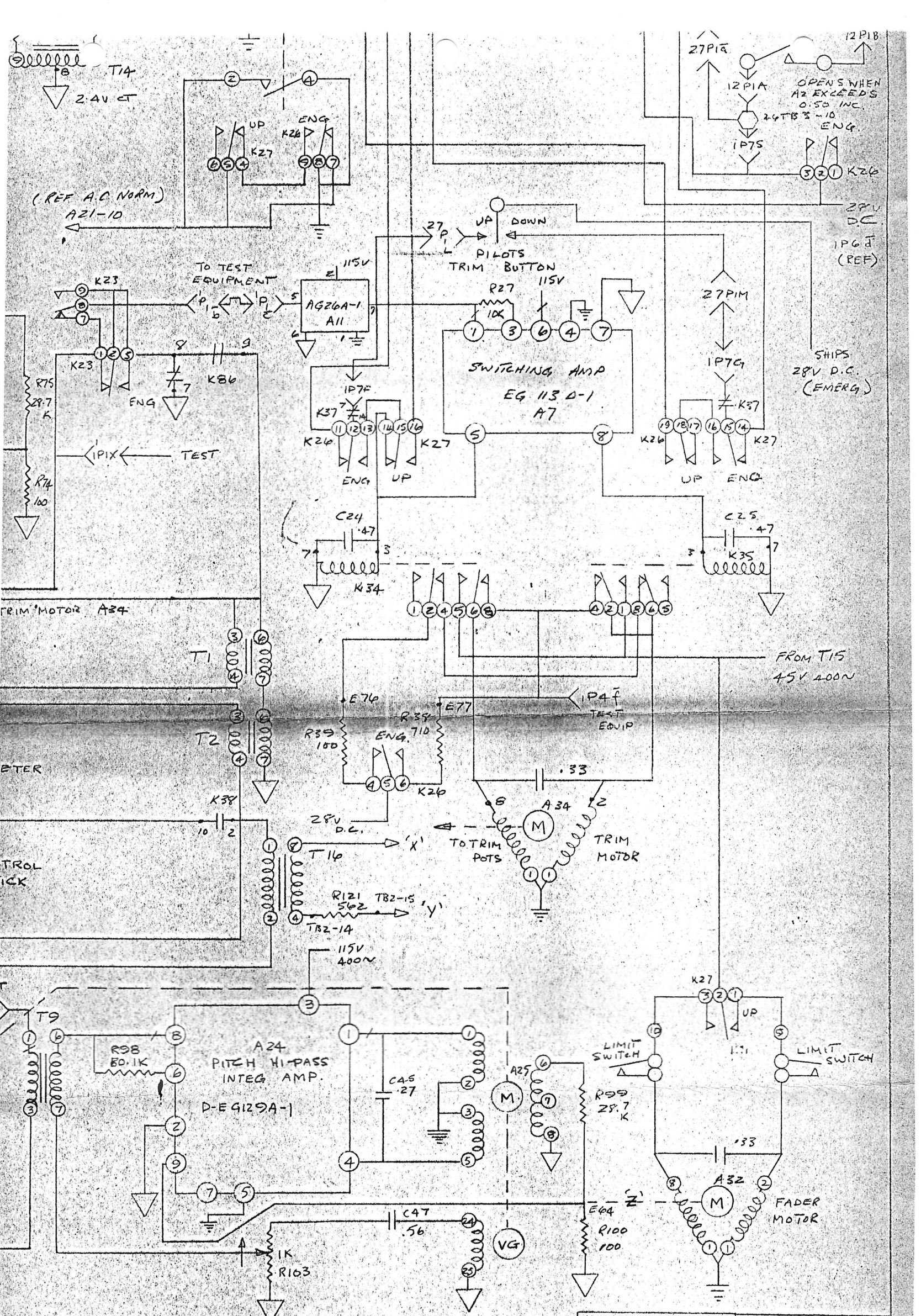
MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT ACT / CALC LB

C8869

4681



DRN BY	J. LAW	10/20/58
REF	M.H. SK66128 REV 0	1/5/57

PITCH DAMPER
MH 64
(YG 539B-1)
DEVELOPMENT

MINNEAPOLIS HONEYWELL
HONEYWELL CONTROLS
AERO DIV
CANADA

CS 1783

P K1 — DOWN K2 — 9 — 12 — 4 11 3 2 10 1 — 3 11 4 1 10 2

27 P2 P — UP — TRIM ACTUATOR — 1P7H — 1P7J — 27 P2 Q — DOWN

6 — 6 — K6 — K7 — 4 — 1 — 3 — 5 — 4 — 1 — 1P1M

PITCH PARALLEL SERVO FORCE SWITCH — 19 P1Z — 1P7N — 19 P1Y

CLOSES WHEN STICK FORCE IS APPLIED — 21P1G — 19P1U — 20TB3-9

27P1Q — 12P1B — 12P1A — OPENS WHEN A2 EXCEEDS 0.50 INC. — 26TB3-10 — ENG. — 1P7S — 3 2 1 K26

R1 121 — E65 — E66 — R2 422 — R3 422 — 9 — 7 — 8 — T14 — 2.4V CT

C4 40μ — 8 — K8

2 — 4 — UP — K27 — 6 5 4 — ENG — K26 — 9 8 7

(REF A.C NORM) AZ1-10

27V D.C. — 1P6d (REF)

27 P1L — UP — DOWN — PILOTS TRIM BUTTON

TO TEST EQUIPMENT — 1P1B — 1P1C — K23 — 2 8 7 — 1 2 5 — 8 — 9 — K86 — ENG — 7

115V — 2 — AG26A-1 A11 — 5 — 7 — 6 — 1

R27 10K — 115V — 1 3 6 4 7 — SWITCHING AMP EG 113 D-1 A7 — 5 — 8

27P1M — 1P7G — K37 — 19 18 17 16 15 14 — K26 — K27 — UP — ENG

SHIPS 28V D.C. (EMERG)

R75 28.7K — 6 — 7 — 8 — R74 100 — 1P1X — TEST

1P7F — K37 — 11 12 13 14 15 16 — K26 — K27 — ENG — UP

C24 .47 — 7 — 3 — K34

C25 .47 — 3 — 7 — K35

1 2 4 5 6 8 — 4 2 1 8 6 5

TO TRIM MOTOR A34

T1 — 3 — 6 — 14 — 7

FROM T15 45V 400N

E76 — E77 — 1P4f TEST EQUIP

T2 — 3 — 6 — 4 — 7

R39 100 — R38 710 — ENG. — 4 5 6 — K26 — .33

OMETER

K38 — 10 — 2

28V D.C.

T16 — 8 — 1 — 2 — 4 — 'X' — 'Y'

TO TRIM POTS — A34 — 6 — 2 — M — TRIM MOTOR — 1 1

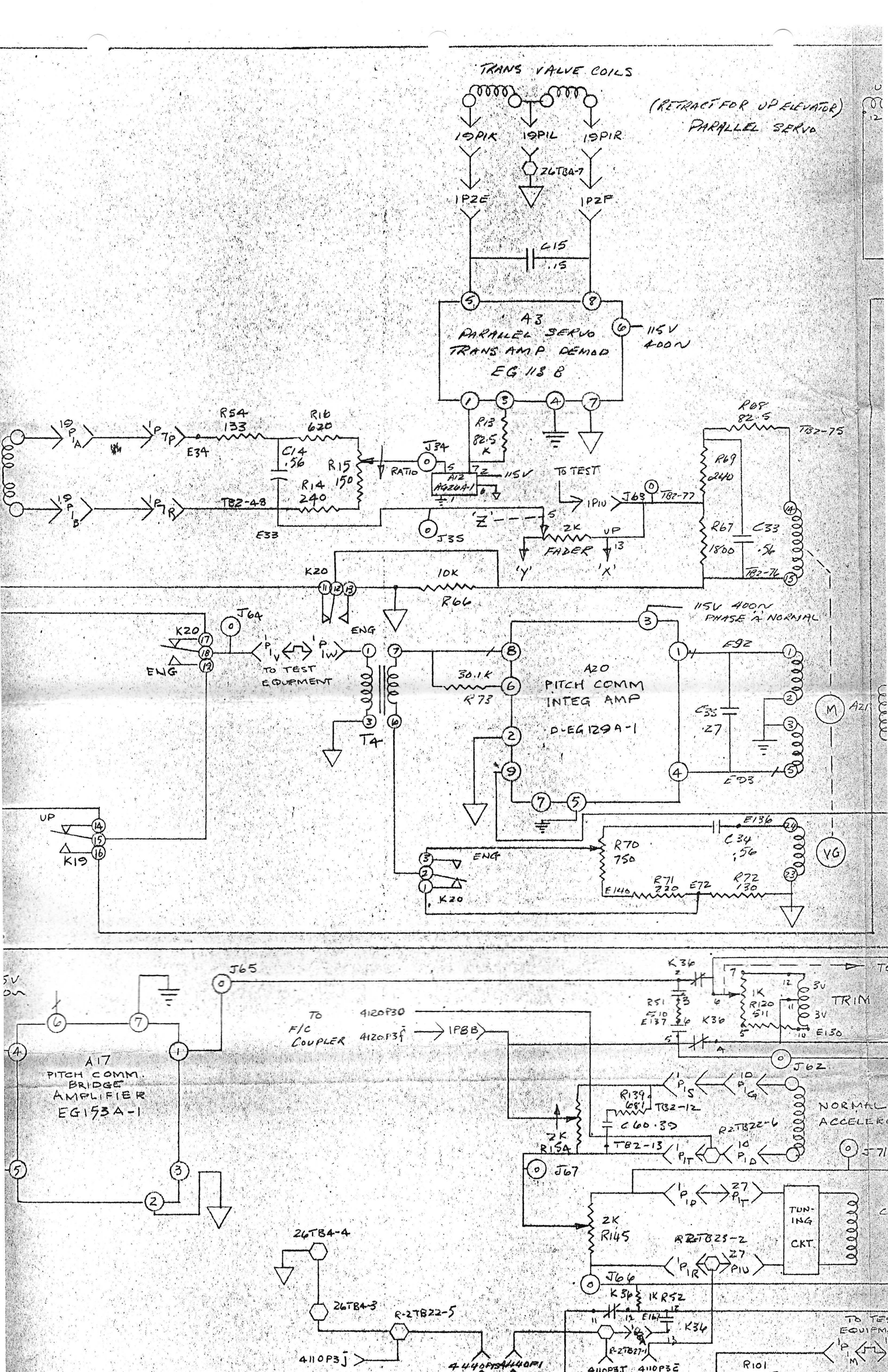
CONTROL STICK

R121 562Ω — TB2-15 — TB2-14

115V 400N

ST MENT IP

K27 — 3 2 1


TRANS VALVE COILS
(RETRACT FOR UP ELEVATOR)
PARALLEL SERVO
19PIK
19PIL
19PIR
26TB4-7
1P2E
1P2F
C15
.15
A3
PARALLEL SERVO
TRANS AMP DEMOD
EG118 8
115V
400N
R54
133
R16
620
R13
82.5
K
R08
82.5
TB2-75
R69
2240
C14
.156
R15
150
R14
240
J34
RATIO
A12
AQZ04-1
115V
TO TEST
1PIU
J63
TB2-77
TB2-48
E34
E33
J35
'Z'
2K
FADER
UP
'Y'
'X'
R67
1800
C33
.56
TB2-74
K20
10K
R66
ENG
J64
TO TEST
EQUIPMENT
T4
30.1K
R73
A20
PITCH COMM
INTEG AMP
D-EG129A-1
115V 400N
PHASE A NORMAL
E92
C35
.27
E03
AZI
UP
K19
E136
C34
.56
R70
750
R71
220
E72
R72
130
F1410
VG
A17
PITCH COMM.
BRIDGE
AMPLIFIER
EG153A-1
J65
TO
F/C
COUPLER
4120P30
4120P31
1PBB
K36
R51
E10
E137
1K
R120
511
3V
TRIM
E130
J62
R139
68.1
TB2-12
C60 .39
TB2-13
R2TB22-6
NORMAL
ACCELER
2K
R154
J67
2K
R145
R2TB23-2
TUN-
ING
CKT
J66
K56
1K R52
E167
K36
26TB4-4
26TB4-3
R-2TB22-5
4110P3J
4110P3E
R-2TB27-1
R101
TO TE
EQUIPM

1P P1A. 1P 7P E34 133 620 C14 .56 R15 150 R14 240 TB2-48 E33 J34 RATIO R13 82.5 K A12 AG26A-1 115V J35 'Z' 5 TO TEST 1PIU J63 TB2-77 2K FADER UP 13 'Y' 'X' 82.5 TB2-75 R69 240 R67 1800 C33 .56 TB2-76 14 15 1P P1B 1P 7R K20 11 12 13 10K R66 ENG J64 K20 17 18 19 ENG 1P P1V 1P P1W TO TEST EQUIPMENT T4 1 7 3 6 30.1K R73 8 6 2 9 A20 PITCH COMM INTEG AMP D-EG129A-1 3 1 E92 C35 .27 4 E93 2 3 5 M A21 115V 400N PHASE A NORMAL UP 14 15 16 K19 7 5 E136 C34 .56 24 23 VG R70 750 ENG 3 2 1 K20 E140 R71 220 E72 R72 130

5V ON J65 6 7 4 1 5 3 2 A17 PITCH COMM. BRIDGE AMPLIFIER EG153A-1 TO F/C COUPLER 4120P30 4120P31 1PBB K36 2 1 7 12 3V R51 3 E10 E137 6 K36 1K R120 511 11 3V 5 10 E150 TRIM A 0 J62 1P P1S 10 P1G R139 68 TB2-12 C60 .39 TB2-13 R2TB22-6 1P P1T 10 P1D NORMAL ACCELER 2K R154 0 J67 1P P1P 27 P1T 2K R145 R2TB23-2 1P P1R 27 P1U TUN-ING CKT 0 J66 K36 1K R52 11 12 E167 13 K36 R-2TB27-1 26TB4-4 26TB4-3 R-2TB22-5 4110P3J 4440P1S 4440P1 1-COS E 4110P3J 4110P3E AFCS TO TE EQUIPM 1P P1M R101 82.5 E66 TB2-71 14 R102 2K C46 .56 15 E69 TB2-91 2470 A36-R 0 J68 1615 787 A36-T INEQ J70 A36-E R133 750 E138 K19 18 17 UP J69 4 0 2K R156 5 J80 (REF) 0 8 1PIP CKT PER

.15

A2
LEFT
PITCH SERVO
TRANS AMP DEMOD
EG 113B

115V
400~

.15

A5
RIGHT
PITCH SERVO
TRANS AMP-DEMOD
EG 113B

115V
400~

619
R62

909
R60

TB2

115V
AG26A-1

A1

J27

115V

A1

AG26A-1

J29

RATIO
250
R8

R7
1100

E39

R26
1100

RATIO
250
R25

R168
620

R169
620

E36

TB4-2

TB4-1

.68
C12

147
R9

E38

TB4-2

TB4-3

.68
C18

E51

R23
147

E50

1P7A

1P7B

1P7C

1P7D

20P1C

20P1K

21P1K

21P1C

A31-5

PG27
INC
ALT.

A31-4

R129
2610

J74

1200

A36-L

A31-6

5K

5K

5K

549

A36-J

A36-N

A36-q

R128
619

R140
825

TB2-6

R130
11K

R127
7.5K

TB2-5

J31

E70

A36-V

J28

UP

7

FADER A32
15K

6

8

A36-X

INC q

1080

A36-S

E68

"Z"
TO FADER
MOTOR

J32

115V
400~

6

J33

PITCH
RATE

2P4B

1P1D

C31
.22

2P4C

26TB4-8

E82

1

2

3

10

1P7N

TO
N
YA

NOTE:

1. PIGTAIL CONNECTIONS TO AMP CAL SHOWN AS —→O
2. IN REGULAR CONFIGURATION, PIGTAIL CONNECTIONS ARE OMITTED AND JUMPERS INSTALLED AS SHOWN

(EXTEND FOR UP ELEVATOR)

DIFFERENTIAL SERVOS

TRANS VALVE COILS

20P1G 20P1F 20P1E

26TB4-9

1P2J 1P2K

C13 .15

A2 LEFT PITCH SERVO TRANS AMP DEMOD EG 113B

115V 400~

TRANS VALVE COILS

21P1G 21P1F 21P1E

26TB4-9

1P2N 1P2P

C19 .15

A5 RIGHT PITCH SERVO TRANS AMP·DEMOD EG 113B

115V 400~

115V AG26A-1

A1

J27

115V AG26A-1

A1

J29

RATIO R7 250 R8 1100

R168 620

E36 TB4-1 .68 C12 TB4-2

147 R9

E38

1P7A 1P7B

20P1C 20P1K

E39 R26 1100 RATIO 250 R25

R169 620

TB4-2 .68 C18 TB4-3 E51

R23 147

E50

1P7C 1P7D

21P1K 21P1C

PARALLEL SERVO TRIM TRANSDUCE

19P1C 19P1D

1P7K 1P7E

R65 619 TB2-106 C32 .47 TB2-

619 R62 909 R60 TB

R129 2610

A31-5 PG27 INC ALT. A31-4

J74

1200

A31-6

A36-L

5K 5K 5K

A36-j 54.9

A36-q

R128 619

R140 825

R175 7.5K

R130 11K

TB2-6

TB2-5

E70

A36-V

J28

UP 7 FADER A32 15K 6 8

A36-X INC q

1080

A36-s

E68

"Z" TO FADER MOTOR

R62 619
R60 909
TB2-109
E34
133
620
C14 .56
R15 150
R14 240
TB2-48
E33
J34
RATIO
R13 82.5 K
A12 AG26A-1
115V
TO TEST
1PIU
'Z'
J35
2K FADER
'Y'
'X'
K20
10K
R66
ENG
J64
TO TEST EQUIPMENT
T4
30.1K R73
A20 PITCH CC INTEG A
D-EG 129
R129 2610
PG27 INC ALT.
J74
A36-L
5K
5K
5K
549
A36-J
A36-K
A36-N
A36-Q
A36-Y
A36-X
INC q
1080
A36-S
UP
K19
J30
TB2-6
R130 11K
R175 7.5K
TB2-5
J31
R140 825
A17 PITCH COMM. BRIDGE AMPLIFIER EG153A-1
J65
115V .400A
E68
TO F/C COUPLER
4120P30
4120P31
1PBB
2K R154
J67
26TB4-4
26TB4-3
R-2TB22-5
4110P31
4440P1S4440P1
1-COS E
J32
115V 400A
J33
C31 .22
E82
A16 PITCH RATE AMPLIFIER EG153A-2
J69
2K R156
J80
(REF)
1P2N
1P1P
TO S&Q CKT NORMAL YAW DAMPER
J68
1615
787
R133 750

PARALLEL SERVO
TRIM TRANSDUCER

19PIC
19PID
1P7K
1P7E
R63
619
C32
.47
TB2-106
TB2-112
R61
K
N
619
R62
909
R60
TB2-109

19P1A
1P7P
E34
R54
133
R16
620
C14
.156
R15
150
R14
240
19P1B
1P7R
TB2-48
E33
J34
RATIO
A12
AG26A-1
115V
TO TEST
1P1U
'Z'
J35
2K
FADER
'Y'
'X'

TRANS VALVE CC
19P1K
19P1L
19P
26TBA-7
1P2E
1P21
C15
.15
A3
PARALLEL SERVO
TRANS AMP DEMOD
EG 118 8
R13
82.5
K

K20
10K
R66
ENG
J64
K20
ENG
1P1V
1P1W
TO TEST
EQUIPMENT
T4
30.1K
R73
A20
PITCH CO
INTEG A
D-EG129

R129
2610
J74
A36-L
5K
5K
5K
549
A36-J
A36-K
A36-N
A36-Q
UP
K19
J30
ENG
K20

TB2-6
R130
11K
R175
7.5K
R140
825
TB2-5
J31
A36-V
115V
400~
J65
A17
PITCH COMM.
BRIDGE
AMPLIFIER
EG153A-1
1080
A36-X
INC 9
A36-S
E68

TO
F/C
COUPLER
4120P30
4120P3J
1PBB
2K
R15A
J67
2K
R14
26TB4-4
26TB4-3
R-2TB22-5
4110P3J
4440P1

ROLL AXIS

A. General

1. The damper roll axis activates two differential servos, one for each aileron. Action of the differential servos does not produce any stick movement in the cockpit.

2. Provision is made in the roll axis for the introduction of command signals which may be initiated by the pilot or may be produced by the Automatic Flight Control System (AFCS). One parallel servo is used to move both aileron surfaces in response to command signals or long term roll rate signals.

3. An electrical trim system, which also acts as a synchronizing system prior to engagement, is incorporated in this axis.

B. The attached diagrams show the build up of the roll axis in block diagram form.

1. Roll Rate

Diagram one shows the system with no command signal inputs and no trim. It indicates that damping action is produced in response to roll rate gyro signals only.

In the other two axes of the system, it is noted that linear accelerations are also sensed and controlled by the damper. In the roll axis, this is not true. The accelerations which could be controlled by action of the ailerons are lateral accelerations, and normal accelerations in turns. However, lateral acceleration is more effectively controlled by the yaw axis of the damper, as is normal accelerations by the pitch axis.

2. Control Stick Steering, Trim

Diagram two shows the addition of control stick steering by which the pilot may command roll rates to change roll attitude, and the addition of trim circuits. The parallel servo circuits, including an integrator are added.

When the pilot wishes to change roll attitude he will apply a lateral force to the stick. The stick will not move, initially, but the force applied will produce an electrical command signal, proportional to the force applied to the stick which is applied to the differential servo amplifiers and to the integrator input.

The differential servos will move immediately to deflect the ailerons and initiate a roll rate.

The integrator output will build up at a rate proportional to the integrator input signal amplitude and the parallel servo will move in a direction to produce the commanded roll rate. The control stick will then begin to move. As the aircraft

rolls, the roll rate gyro will produce an output signal proportional to the roll rate. When the roll rate signal is equal to the command signal, the two signals will cancel, since the roll rate signal will be of a polarity which will oppose the roll rate. When the signals cancel, the differential servo amplifier inputs will be zero and the differential servos will return to the neutral position. The input to the integrator will also be zero, however, the integrator output does not go to zero. Its rate of increase will go to zero. This means that the parallel servo will maintain the aileron surfaces deflected in the correct amount to maintain the roll rate signal equal to the command signal. Since the command signal is proportional to the force applied to the stick, the resultant roll rate must be proportional to stick force.

NOTE: In practice the pilot will think that he has moved the stick when he applies a force to it. In actual fact, it will be the parallel servo which moves the stick. The greater the force applied by the pilot to the stick, the greater will be the roll rate produced. This method then produces electrical "feel" without the use of springs or trim motors.

By following a similar discussion to that above, it can be seen that if the pilot releases the force on the stick, the roll rate signal which has been established will remain as the only signal input into the system. This signal will produce differential servo action to oppose the roll rate and at the same time, will start the parallel servo on its return to the neutral position. The aircraft will then stabilize in a bank with the stick returned to the neutral position. In practice, the pilot will hasten the establishment of a stable bank by applying a force to the stick in the opposite direction to that be used in initiating the bank.

The action described above is similar to that in the pitch axis in which a stick force commands a normal acceleration.

During normal damping action, (no command signals acting) the output of the roll rate gyro will be applied to the integrator input as well as to the differential servo amplifiers. Under normal conditions, roll rate signals will be transient in nature and the average of these signals at the integrator input will be zero. If the aircraft experiences a roll rate that is long term in nature, the integrator output will move the parallel servo to produce opposition to this error signal. When the roll rate has reduced to zero, an undesired bank angle may exist. The integrator input will be zero, but its output will not be zero. Since the integrator integrates roll rate, its output will be proportional to bank angle. The parallel servo will be actuated by this output to return the bank angle to zero.

The electrical trim system consists of a motor which drives an inductive pickoff such that either left or right roll rate command signals may be produced. When the pilot actuates the stick trim button, power is applied to the motor and an electrical signal is produced which acts in the same manner as a command from the control stick. The pilot can use the trim button to either set up a fixed roll rate, or to trim out an undesirable roll rate due to aircraft mistrim. When the pilot releases the trim button, the output of the trim system will remain where it is and will not return to zero unless the pilot returns it by actuating the trim button in the opposite direction.

The student should not confuse this trim system with the Avro trim system which is used when the roll damper is not engaged. The Avro trim system which is actuated by the same trim button acts directly and mechanically on the aileron control quadrant.

3. Scheduling

Diagram three shows the addition of the AFCS command source and the addition of airspeed compensation and lateral acceleration compensation. An additional summing path has been added to the integrator circuit and an amplifier has been added in the signal path.

The AFCS command source, no matter what its control function will act in the same manner as a command from the control stick as described above. It should be noted that the pilot's trim button will be inoperative when AFCS is in use.

Since control surface effectiveness varies with airspeed, both the roll rate signal and the control stick steering signal are scheduled with airspeed.

If the aircraft is being acted upon by a lateral acceleration, a high roll rate could cause excessive structural loading in the rudder. Any pilot's roll rate commands are reduced by the lateral acceleration scheduling if lateral acceleration is present.

AFCS commands are internally airspeed scheduled.

The amplifier shown in diagram three is added to overcome gain losses in the summing operation.

An additional summing path shown on this diagram runs from the input to the parallel servo amplifier, point X, back to the input of the integrator. Under normal operating conditions, the output of the integrator will be continuously balanced by the servo feedback. The voltage of X will be very small, being close to the threshold voltage of the amplifier. If, however, the parallel servo has moved to its limits in response to a command from the integrator output, no additional feedback balance can be produced. If the integrator output continues to rise, a voltage will build up at X, the parallel servo amplifier input. This voltage is fed back to the integrator input in the correct phase to oppose increase in integrator output. In this way, excessive integrator outputs are prevented when the servo has reached the limit of its travel.

4. Gear Down Mode

When the pilot wishes to land, it is desirable that the control action of the stick be changed. Stick commands should now produce surface displacement proportional to the force applied to the stick rather than roll rates proportional to the force applied to the stick. Normal rate damping however, should remain as it was. Diagram four shows the roll axis in the gear down configuration. The student will notice that signal paths to the differential servos have not been changed. The integrator circuit has been changed so that its output is returned to the input in such a way as to cancel the input signal if any exists. Any control stick steering signal is fed, uncompensated by airspeed or lateral acceleration, directly to the parallel servo. Surface displacement will be, proportional to stick force. Trim signals and AFCS signals are also fed directly to the parallel servo and will produce proportional surface deflection in the same manner as control stick commands. The system switches to this configuration automatically when the landing gear is lowered. The pilot will notice a change in the "feel" of his control stick.

5. Pre-Engage

Diagram five shows the roll axis prior to engagement. Any output signals from the rate gyro or control stick are amplified and used to operate the trim motor in a direction which will produce a trim output of the correct phase and amplitude to cancel the amplifier input signal. The input to the differential servo amplifiers is held at zero by this synchronizing action. Since the AFCS system cannot be engaged unless the full damper is engaged, no output signals will be present as inputs to the roll damper.

Hydraulic pressure is removed from the differential servos and the servos are spring loaded to centre during the pre-engage mode.

As described previously, the pilot's trim button now operates the Avro roll trim system.

Hydraulic pressure is removed from the parallel servo, but unlike the differential servos, it is <u>not</u> spring loaded to centre. Since the parallel servo ram is attached to the front quadrant, it must move when the pilot moves his control surfaces. When the pilot deflects his control surfaces the parallel servo will move, producing a feedback balance signal to the servo amplifier input. Since no other signal is present at this point, the balance signal will cause the moog valve to operate and produce a pressure in one of the two hydraulic lines to the parallel servo. Since these lines are not ported to the servo when the system is not engaged, no detrimental action is produced. However, if the system were to be engaged in this condition, a severe bump would be felt as this pressure was applied to the servo. In addition, the servo amplifier would be in a saturated condition.

A pressure trim sensor which produces a signal whenever the pressures in the two moog valve lines are different is used to rectify this situation. When the control surfaces are deflected and the system is in the condition described above, a signal from the pressure trim sensor is fed back to the integrator input. The integrator time constant is changed in the pre-engage condition so that its action is much faster and it can now be called a synchronizer. The sync output increases until it cancels the feedback balance signal from the servo, thus returning the moog valve to centre and reducing the pressure trim signal to zero. The sync output however, does not return to zero. The sync output rate of increase drops to zero and its output maintains the balance at the amplifier input. As shown in the diagram, an additional path from the pressure trim sensor directly to the amplifier input is supplied. This will aide the sync action when the error is first produced but will have no effect after balance has been achieved.

As an additional safeguard, the system cannot be engaged unless the pressure trim sensor is in the centre position (no signal output). DC power cannot be applied to the engaging circuits of this axis unless it can pass through contacts of a sector switch which is closed only when the pressure trim sensor is centered. (Refer to DC engaging sequence and parallel servo schematic.)

C. Functional Block Diagrams

The three additional diagrams attached, show the means by which the diagrams discussed above are actually mechanized. As in yaw and pitch, series summing is used.

ROLL AXIS COMPONENTS

1. BG92B-1 Amplifier Calibrator (1)
2. GG72A-2 Roll Rate Gyro (1)
3. MG51D-2 Differential Servo (2)
4. MG55M-3 Parallel Servo (1)
5. GG47D-2 Accelerometer, Lateral Scheduling (1)

AVRO SUPPLIED COMPONENTS

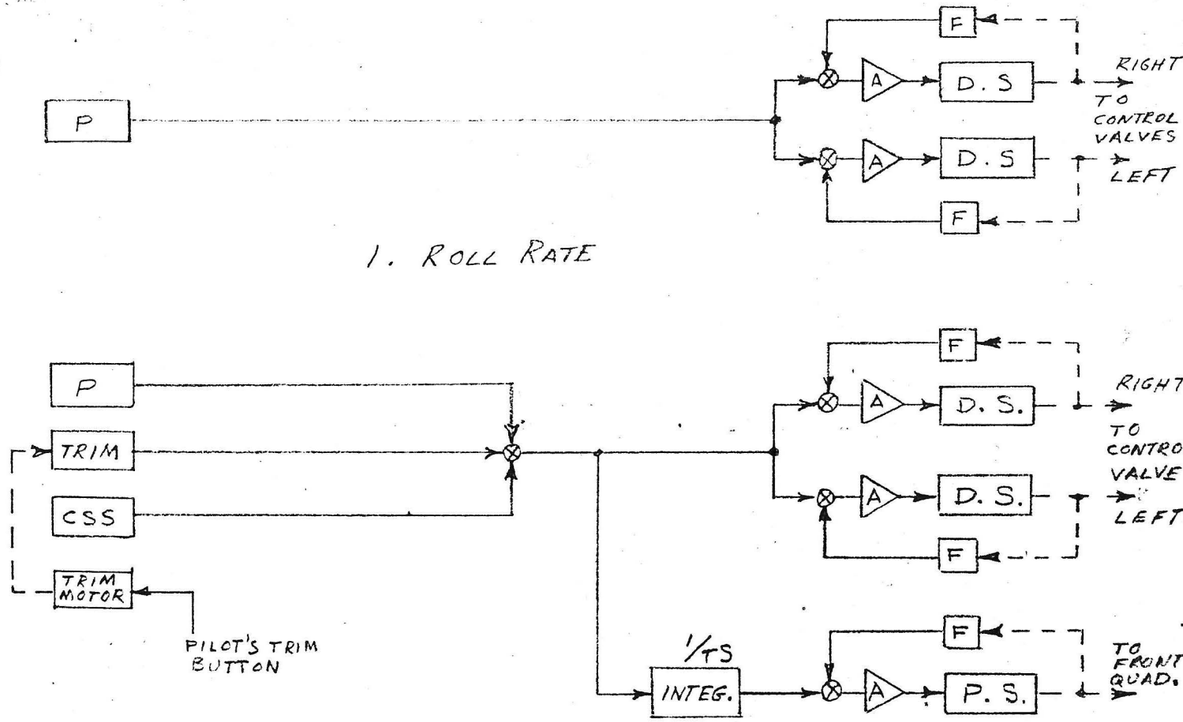
1. Stick Force Transducer
2. Trim Button

SETTINGS (Tentative)

1.	Roll Rate	R147	5.8
2.	Differential Servo Balance	R36, R31	6.3 (6.7ft.)
3.	Parallel Servo Balance	R47	3.6
4.	Control Stick Balance	R150	5.6

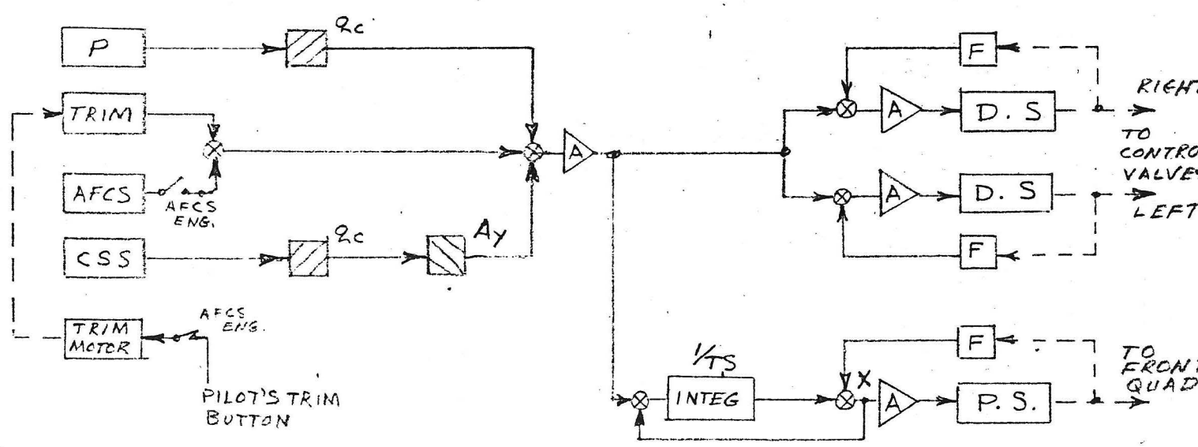
SCALE FACTORS AND GENERAL INFORMATION

1.	Differential Servo Amplifier Threshold	7.3 mv
2.	Parallel Servo Amplifier Threshold	5.1 mv
3.	Stop to Stop Trim Motor Time (.565V)	30 sec.
4.	Integrator Rate	0.97 mv/sec/mv
5.	Single Aileron Movement, Differential	±7.5°
6.	Single Aileron Movement, Parallel	±19°
7.	Differential Servo Ram Travel	±.375 in.
8.	Parallel Servo Ram Travel	±1.5 in.
9.	Roll Rate Gyro	4V/150°/sec
10.	Control Stick	6V/20 lb.
11.	Servo Balance, Diff., Parallel	5.5V/inch

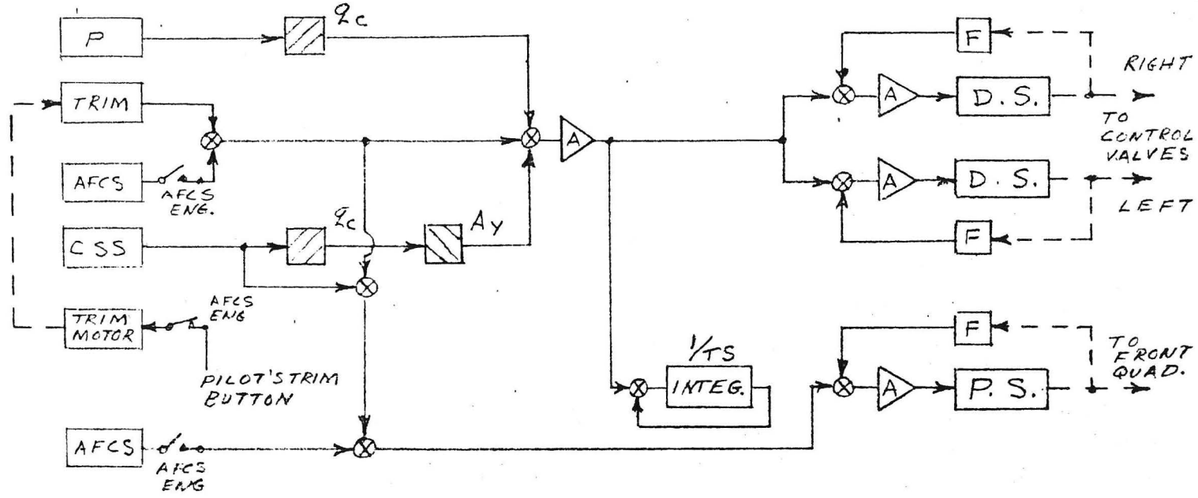


1. ROLL RATE

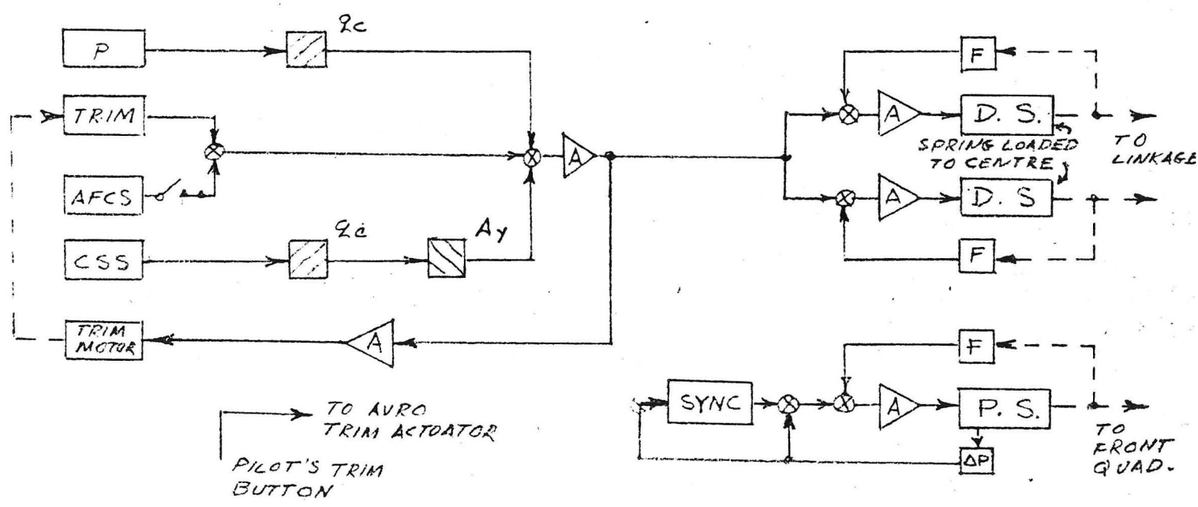
2. CONTROL STICK STEERING, TRIM



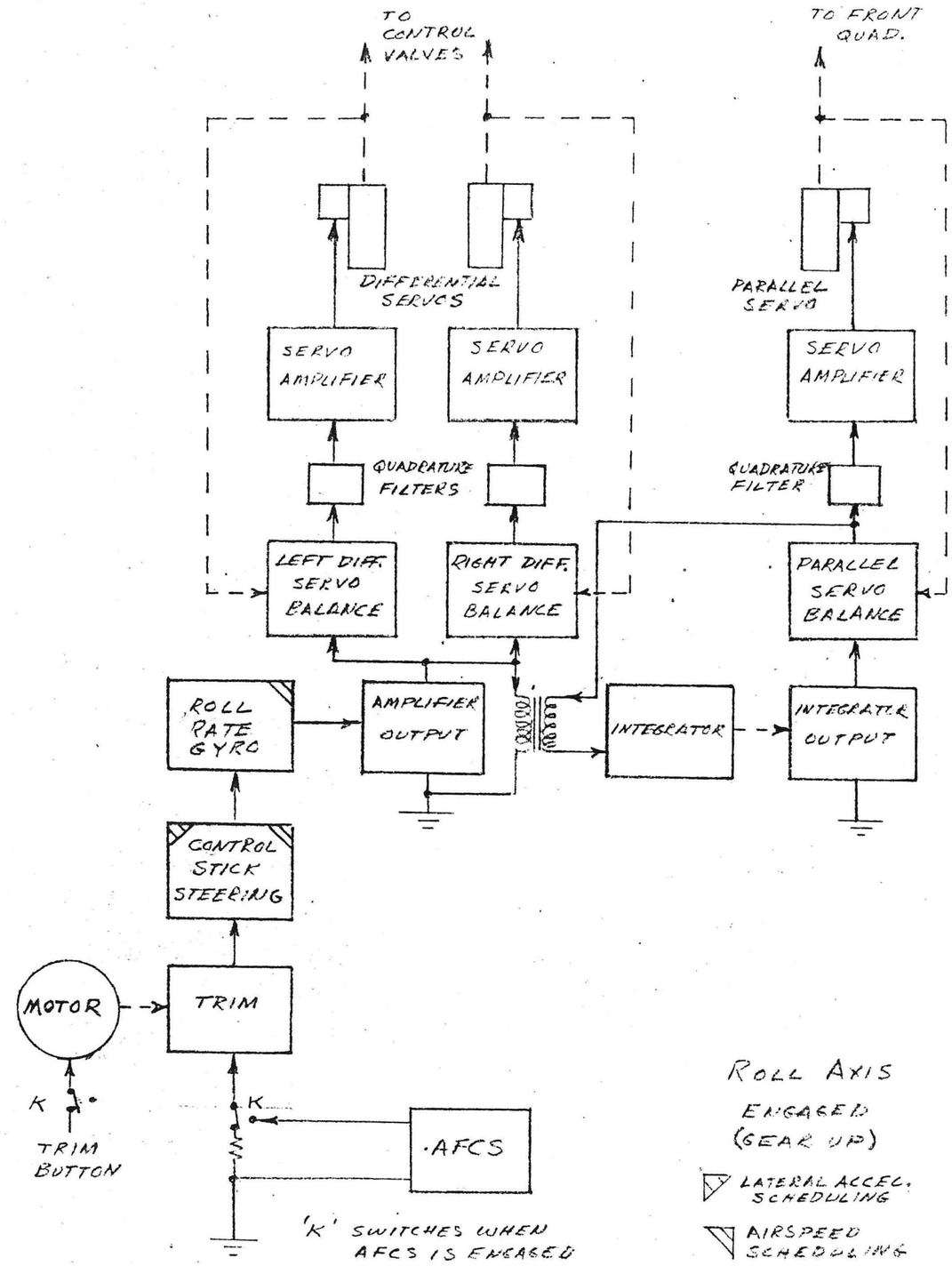
3. AFCS, SCHEDULING

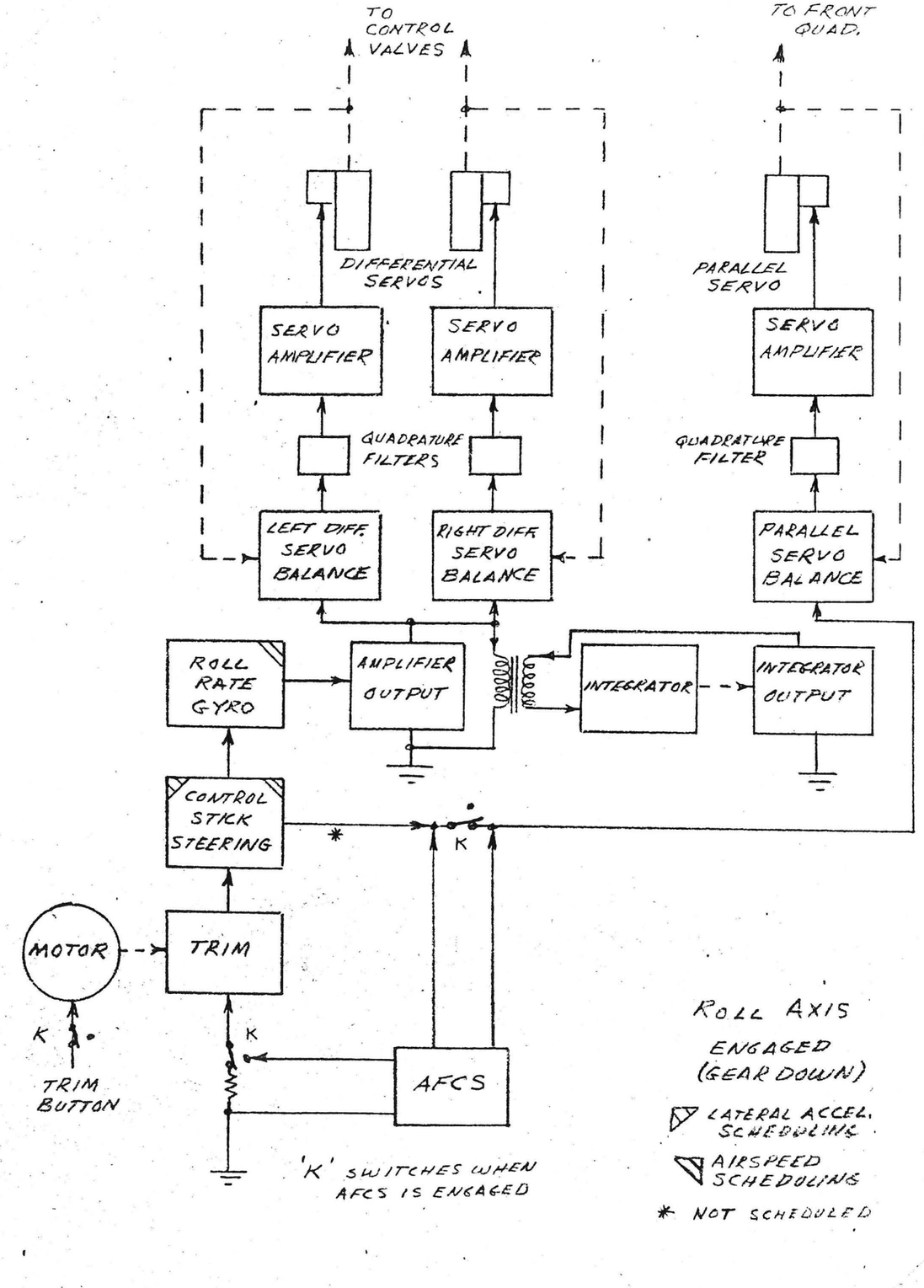


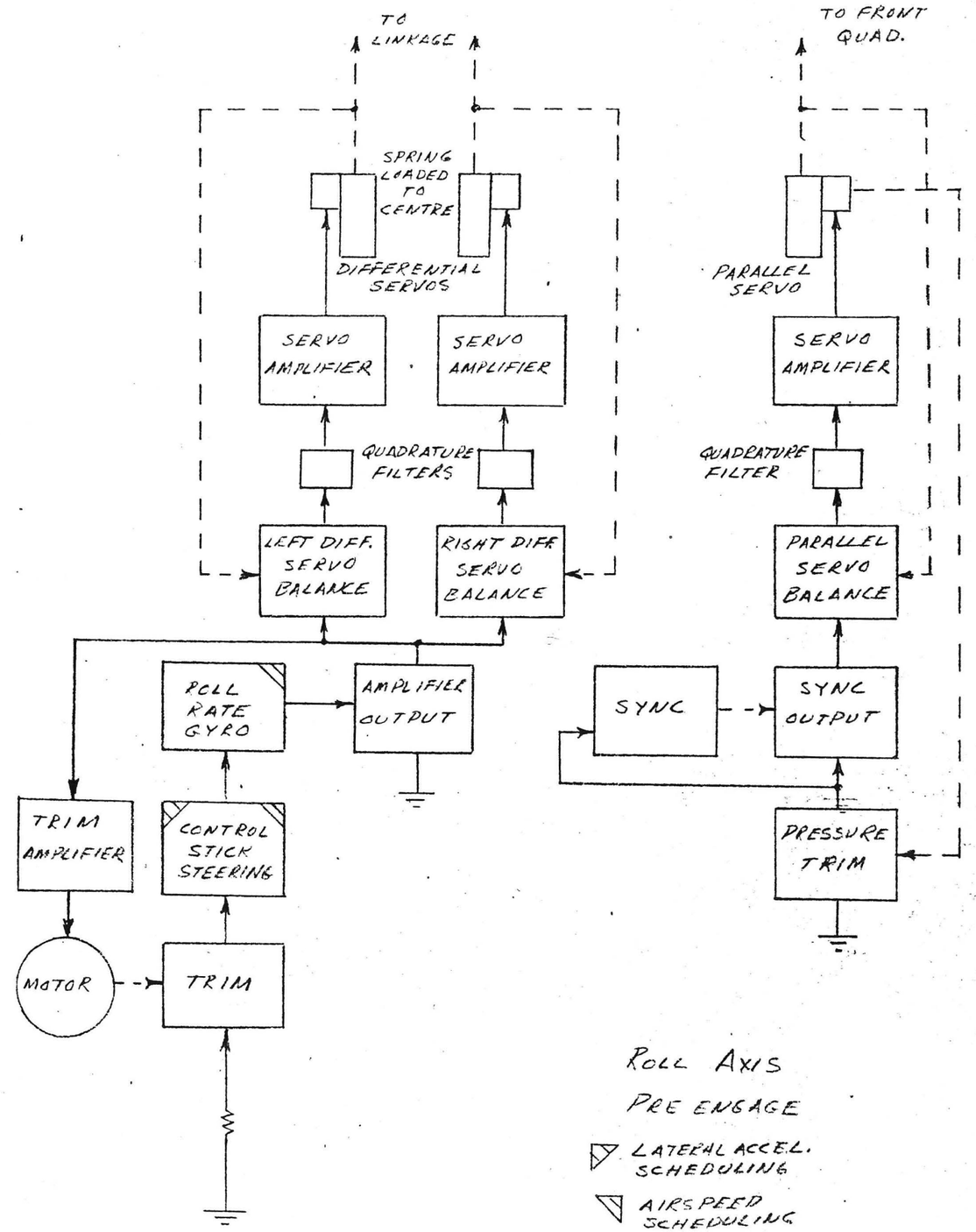
4. GEAR DOWN



5. PRE ENGAGE


TO CONTROL VALVES
TO FRONT QUAD.
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF. SERVO BALANCE
PARALLEL SERVO BALANCE
ROLL RATE GYRO
AMPLIFIER OUTPUT
INTEGRATOR
INTEGRATOR OUTPUT
CONTROL STICK STEERING
MOTOR
TRIM
K
TRIM BUTTON
K
AFCS
'K' SWITCHES WHEN AFCS IS ENGAGED
ROLL AXIS
ENGAGED
(GEAR UP)
LATERAL ACCEL. SCHEDULING
AIRSPEED SCHEDULING


TO CONTROL VALVES
TO FRONT QUAD.
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF SERVO BALANCE
PARALLEL SERVO BALANCE
ROLL RATE GYRO
AMPLIFIER OUTPUT
INTEGRATOR
INTEGRATOR OUTPUT
CONTROL STICK STEERING
*
K
MOTOR
TRIM
K
K
TRIM BUTTON
AFCS
'K' SWITCHES WHEN AFCS IS ENGAGED
ROLL AXIS ENGAGED (GEAR DOWN)
LATERAL ACCEL. SCHEDULING
AIRSPEED SCHEDULING
* NOT SCHEDULED


TO LINKAGE
TO FRONT QUAD.
SPRING LOADED TO CENTRE
DIFFERENTIAL SERVOS
PARALLEL SERVO
SERVO AMPLIFIER
SERVO AMPLIFIER
SERVO AMPLIFIER
QUADRATURE FILTERS
QUADRATURE FILTER
LEFT DIFF. SERVO BALANCE
RIGHT DIFF SERVO BALANCE
PARALLEL SERVO BALANCE
ROLL RATE GYRO
AMPLIFIER OUTPUT
SYNC
SYNC OUTPUT
TRIM AMPLIFIER
CONTROL STICK STEERING
PRESSURE TRIM
MOTOR
TRIM
ROLL AXIS
PRE ENGAGE
LATERAL ACCEL. SCHEDULING
AIRSPEED SCHEDULING

ROLL RATE

K_P

δ_P

$\dot{\psi}_c$

STICK FORCE

K_{CSS}

δ_{P_2}

$\dot{\psi}_c$

$f(A_Y)$

K_1

K_2

K_D

DIFF SERVO (LEFT)

TO LEFT CONTROL VALVE

DIFF SERVO (RIGHT)

TO RIGHT CONTROL VALVE

K_D

AFCS

DAMPER AFCS

UP

DOWN

TRIM

K_3

ENG

PRE

PRE

ENG

TRIM MOTOR

ENG

PRE

TO AVRO ROLL TRIM ACTUATOR

PILOT'S TRIM BUTTON

1/S

PRE

ENG

UP

DOWN

ENG

PRE

UP

ENG

DOWN

PRE

ΔP

PAR. SERVO

K_P

1 RELAYS ARE SHOWN IN THE ENGAGED, GEAR UP, POSITION

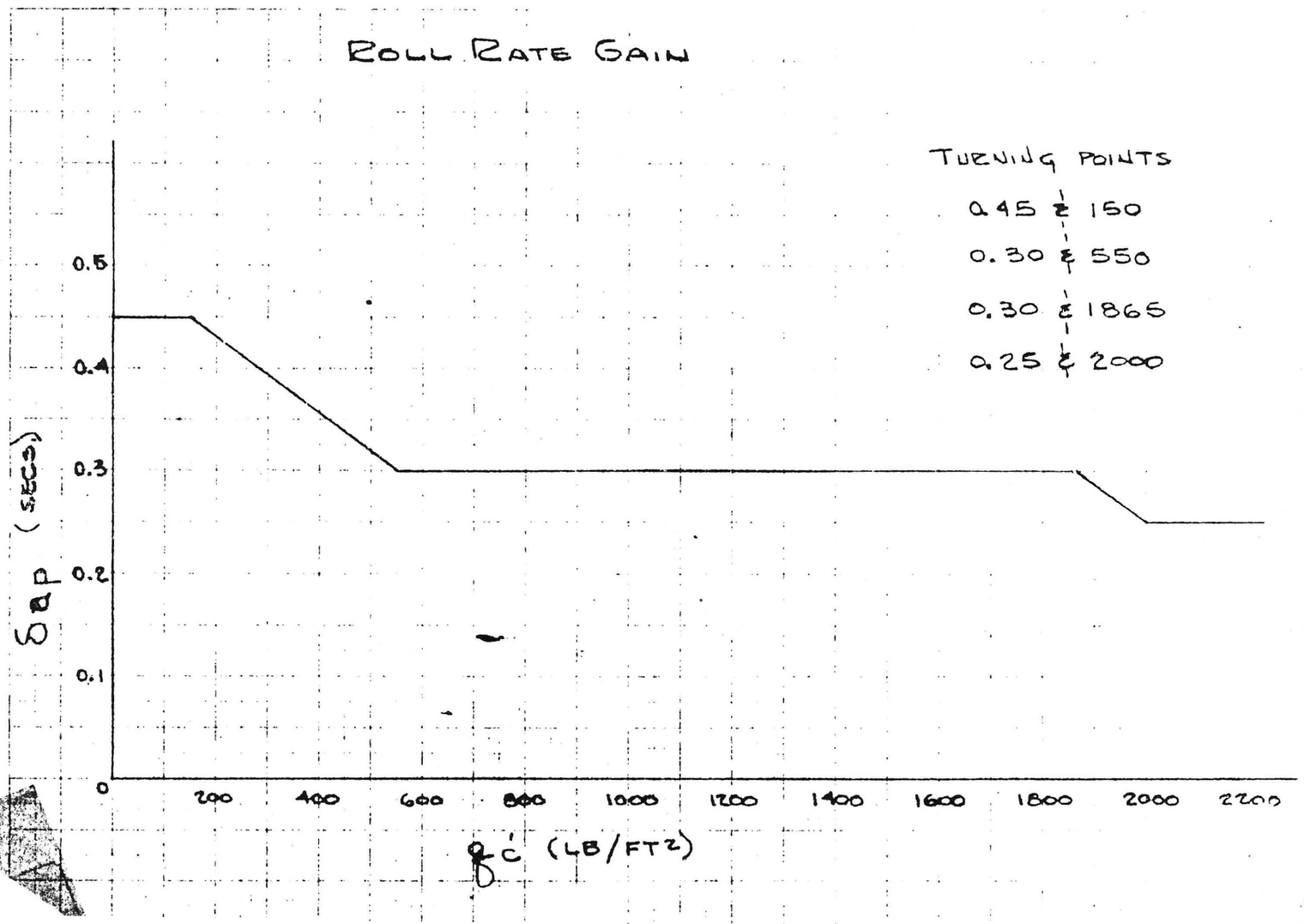
ROLL AXIS SIMPLIFIED BLOCK DIAGRAM YG339 DAMPER (MH-64)

3-7-58 ISSUE DATE

SK66498

KEUFFEL & ESSER CO. 359-14

ROLL RATE GAIN

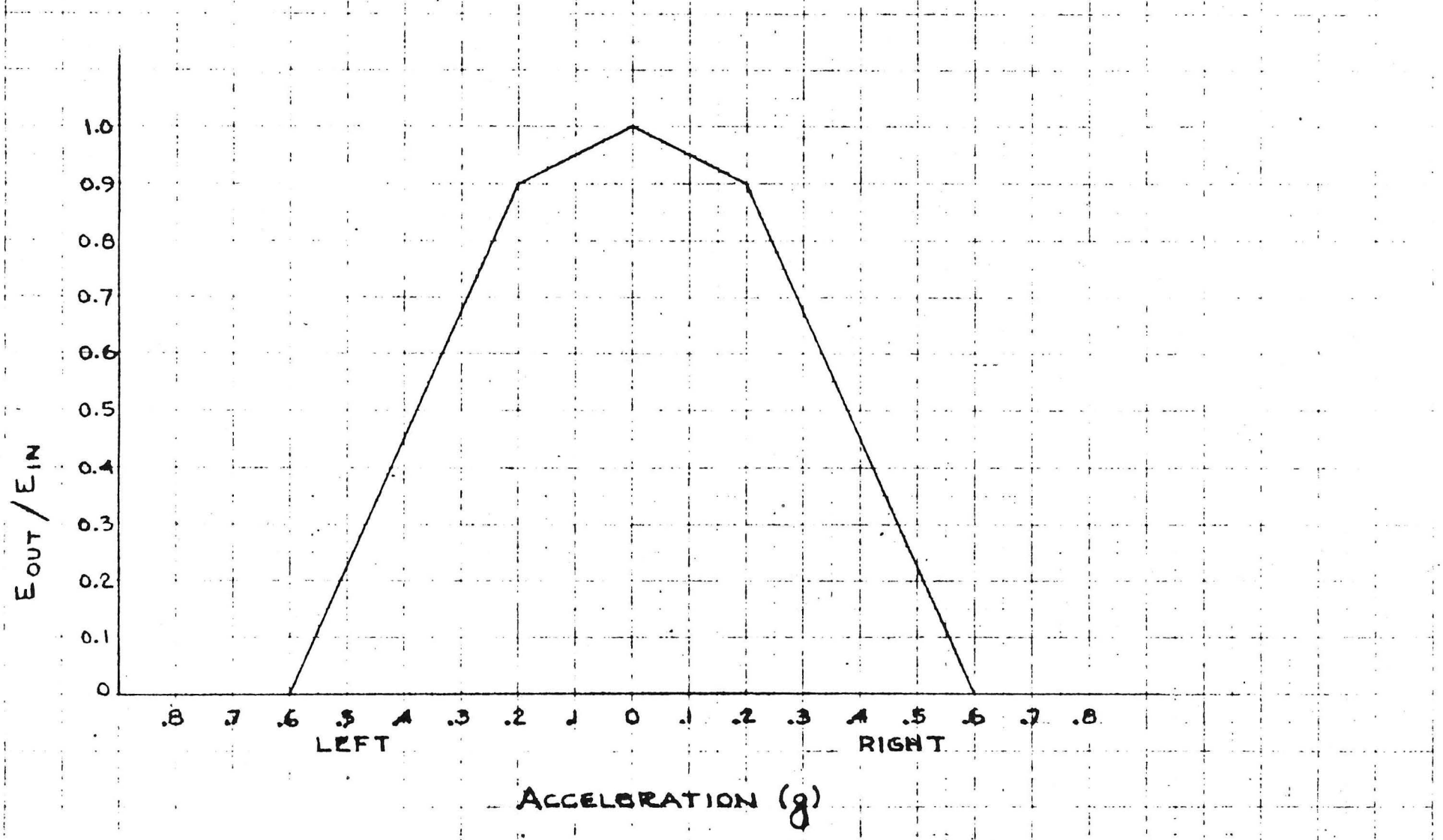


GRAPH NO / SEL 191

359-14

LATERAL ACCELEROMETER

ROLL AXIS



COMMAND ROLL RATE GAIN

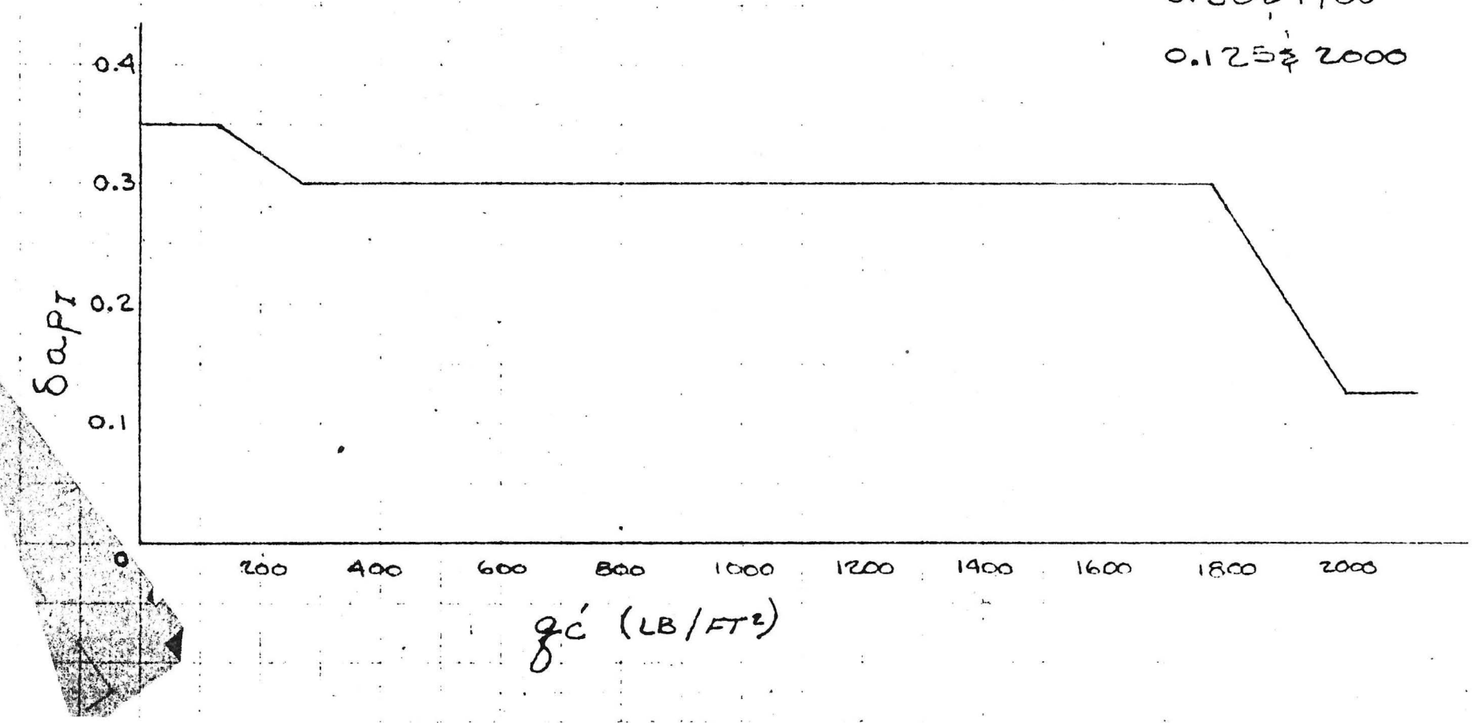
TURNING POINTS

0.35 & 130

0.30 & 270

0.30 & 1780

0.125 & 2000

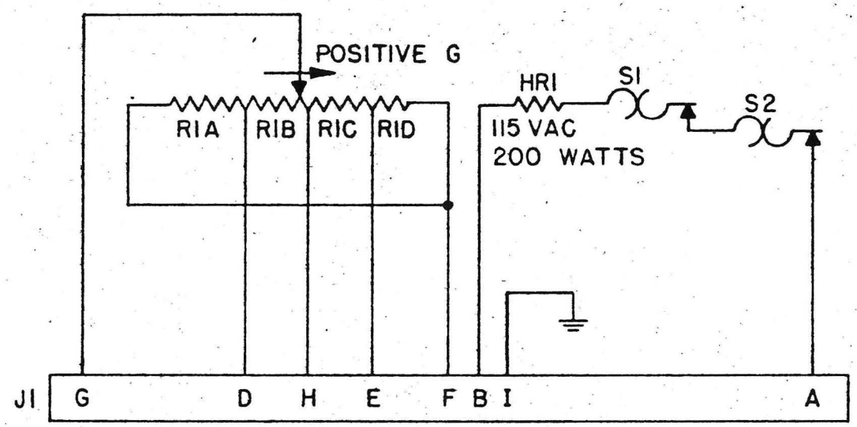


C8866

SUPPLEMENTARY INFORMATION		
FIRST USED GG47 D-2		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL

RESISTANCE NO.	RESISTANCE (OHMS)
RIA	8.1 K
RIB	4.05K
RIC	4.05K
RID	8.1K



CANNON
GS02-18-IP-III

2- ⟶ INDICATES TRAVEL IN DIRECTION INDICATED
1- CASE CONNECTED TO PIN I FOR GROUNDING OF CASE

D-

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±1°

D 2021-13-01
.D
GOVT MATL SPEC

DRAWN B	D. Johnson	4-14-58
DRAFTING √	Andrusky	4-14-58
MATL & FIN. √		
DESIGNER √	GJO9	4/14/58
DESIGN √	Andrusky	4-14-58
APPROVED	VAH	4-16-58

REF MH MATL (SPEC)
FINISH (SEE NOTE)

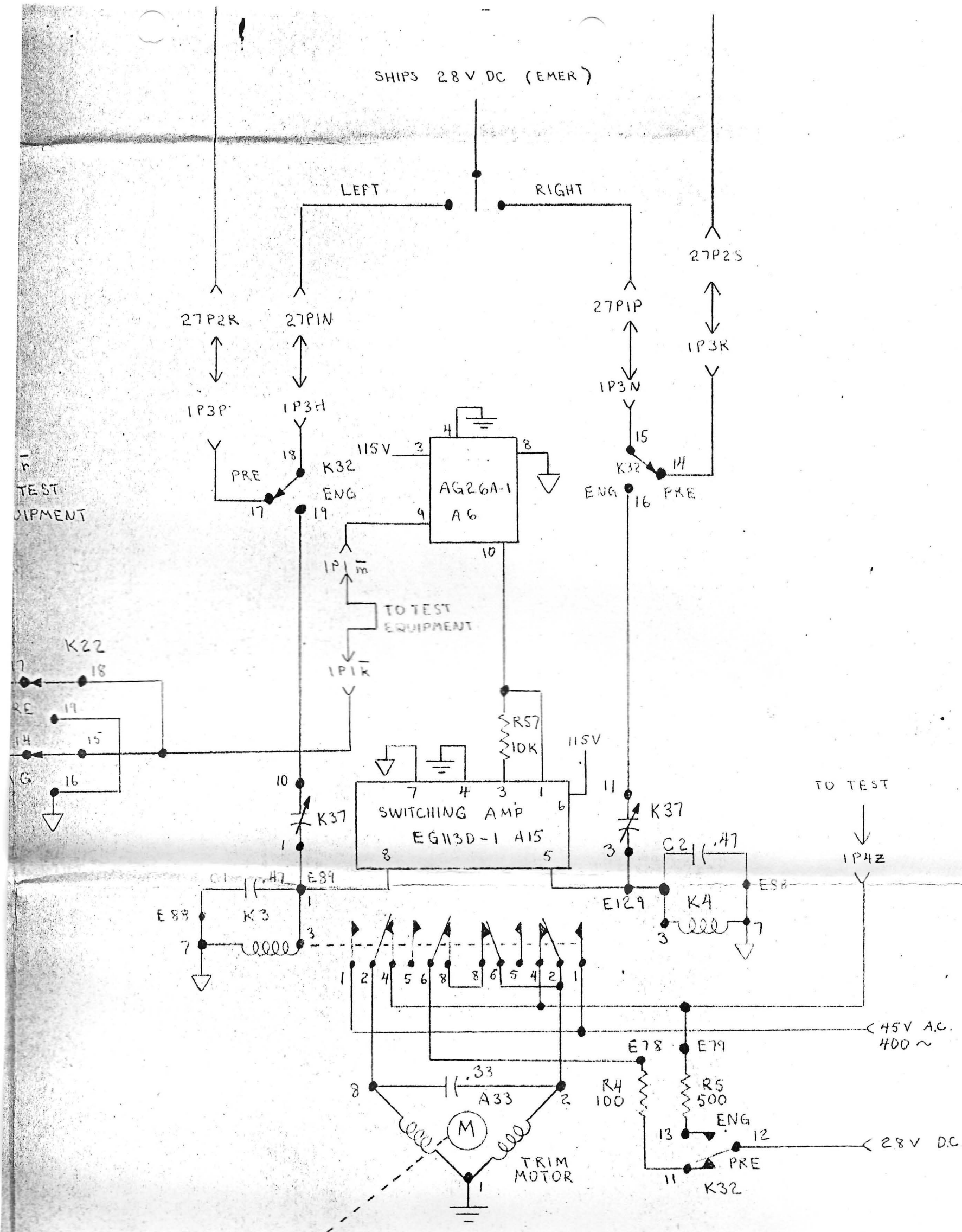
SCHEMATIC DIAGRAM— ACCELEROMETER

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE
WT ACT. CALC LB

C8866

98-2798



CSK 1779

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±1°
OTHER ANGLES		
DEV.		
SUPSD.		
GOVT. MATL. SPEC.		

DRAWN BY	COURTIN	10-16-58
DRAFTING √		
MATL. & FIN. √		
DESIGNER √		
DESIGN √		
APPROVED		

REF. MH. MATL. (SPEC.)

FINISH (SEE NOTE)

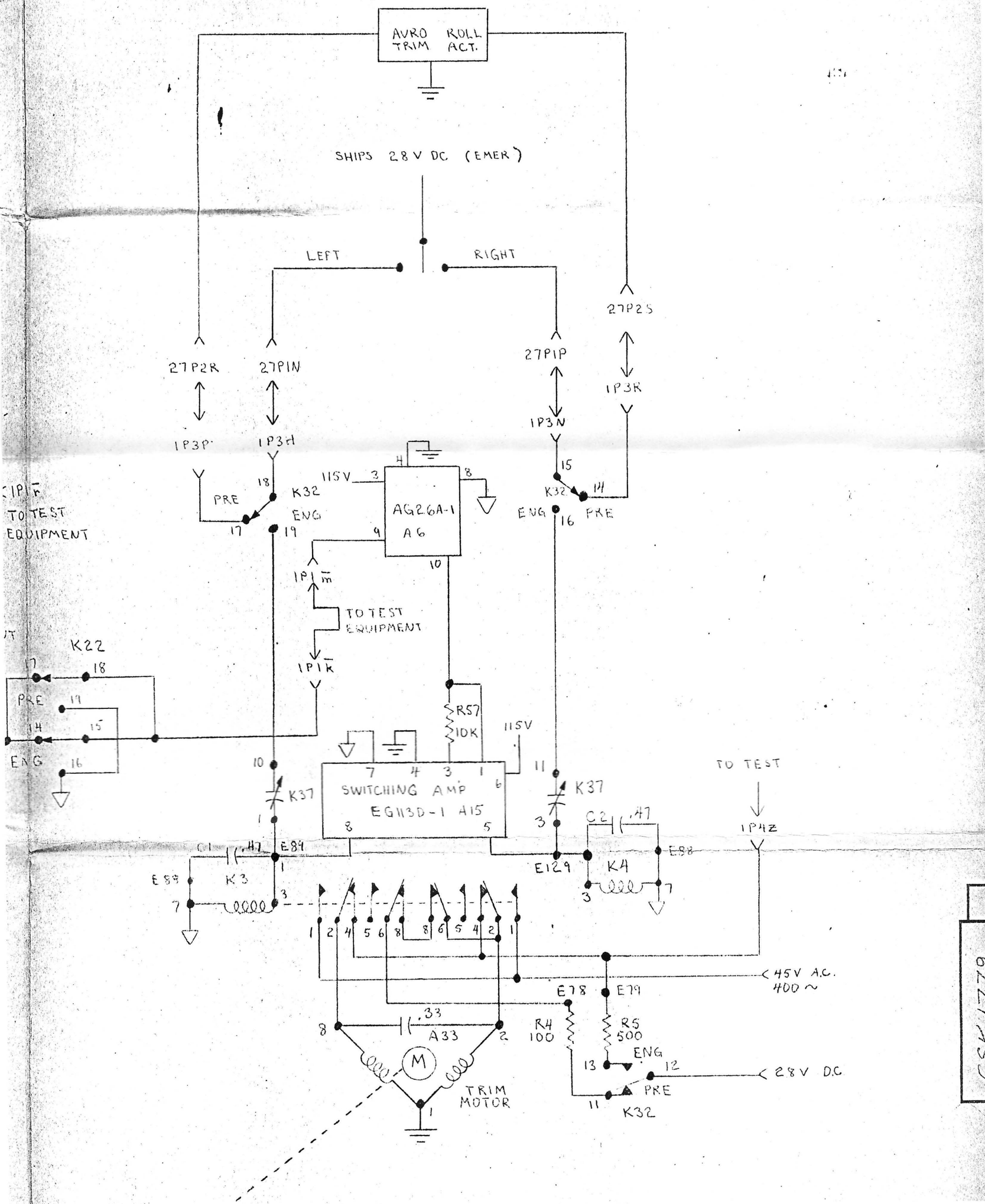
MH-G4 DEVELOPMENT DAMPER
ROLL AXIS
YG339B-1

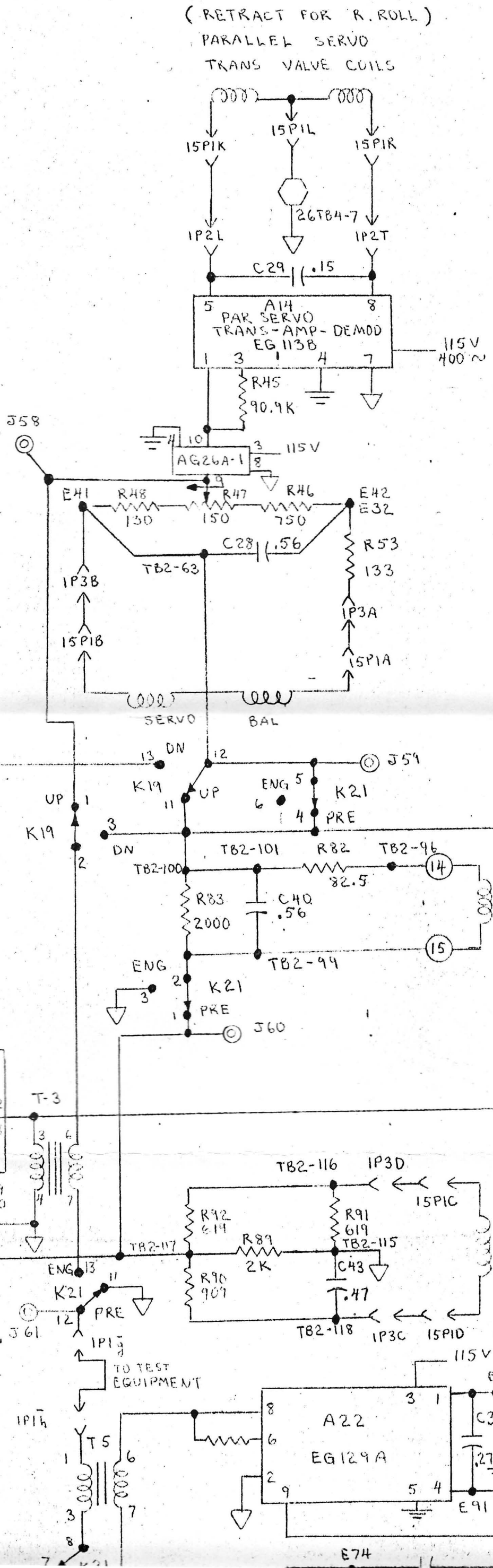
SCALE NONE SIZE | WT. ACT. CALC. LB.

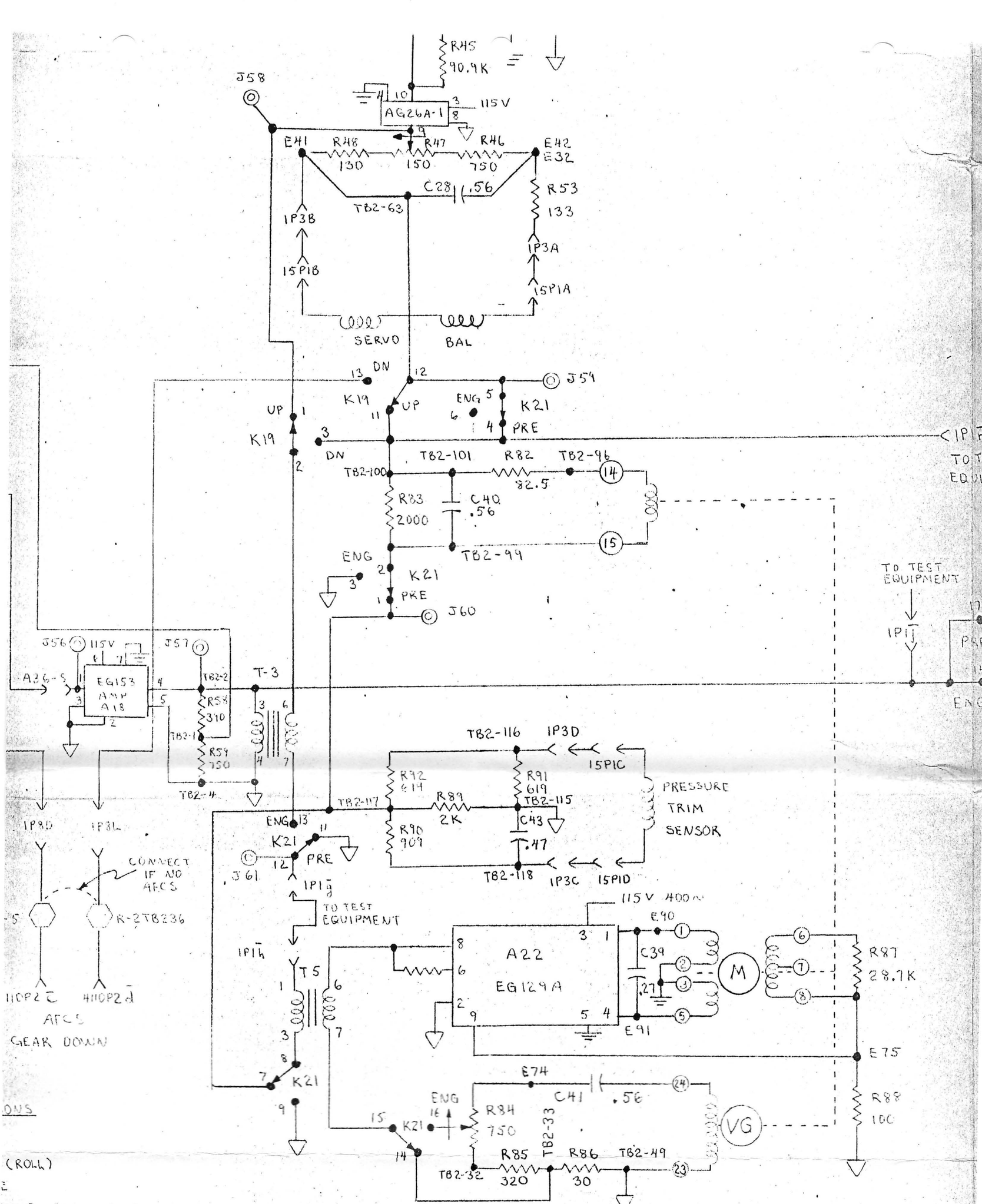
HONEYWELL CONTROLS LIMITED
AERONAUTICAL DIVISION
TORONTO, CANADA

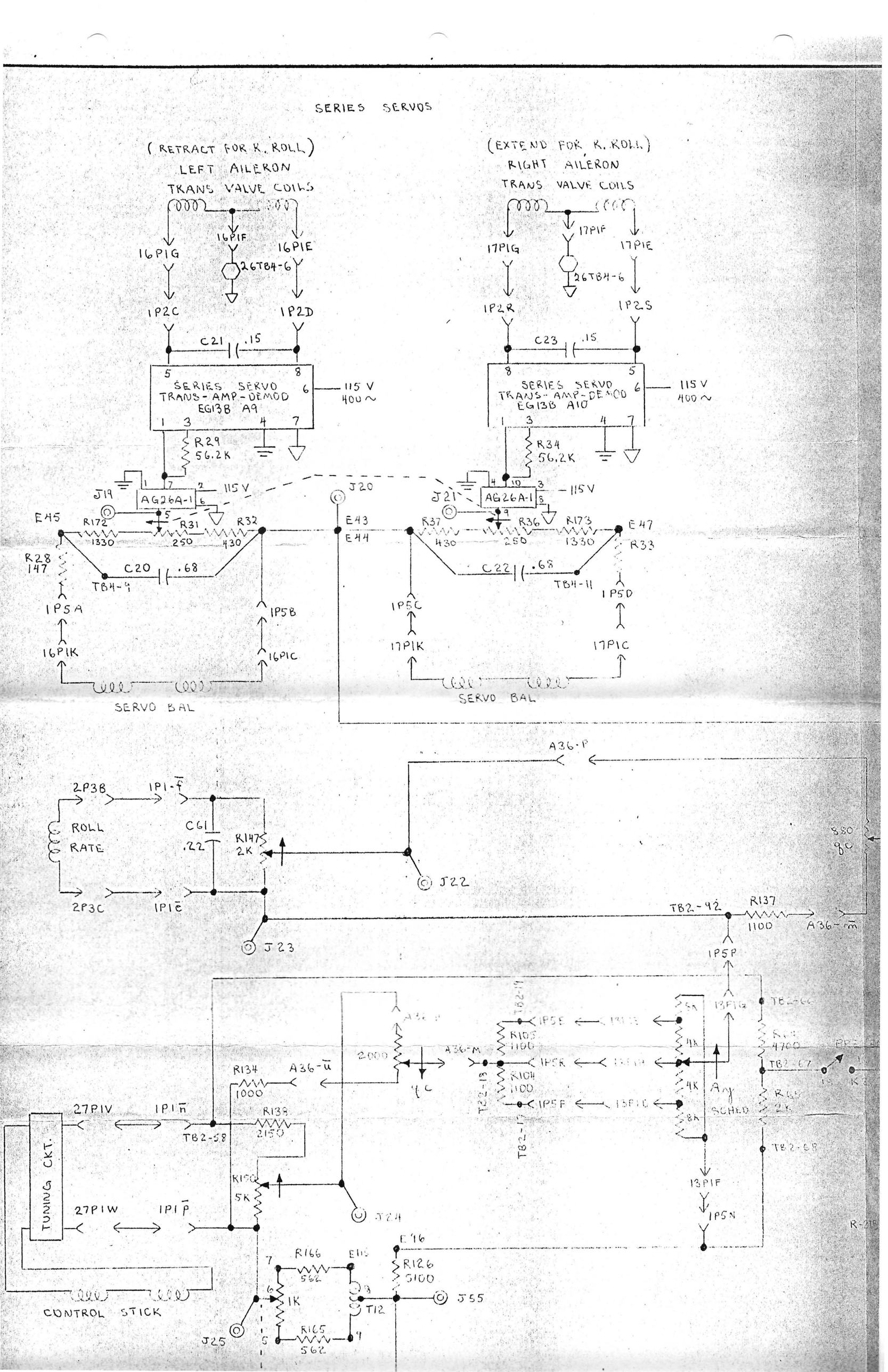
CSK 1779

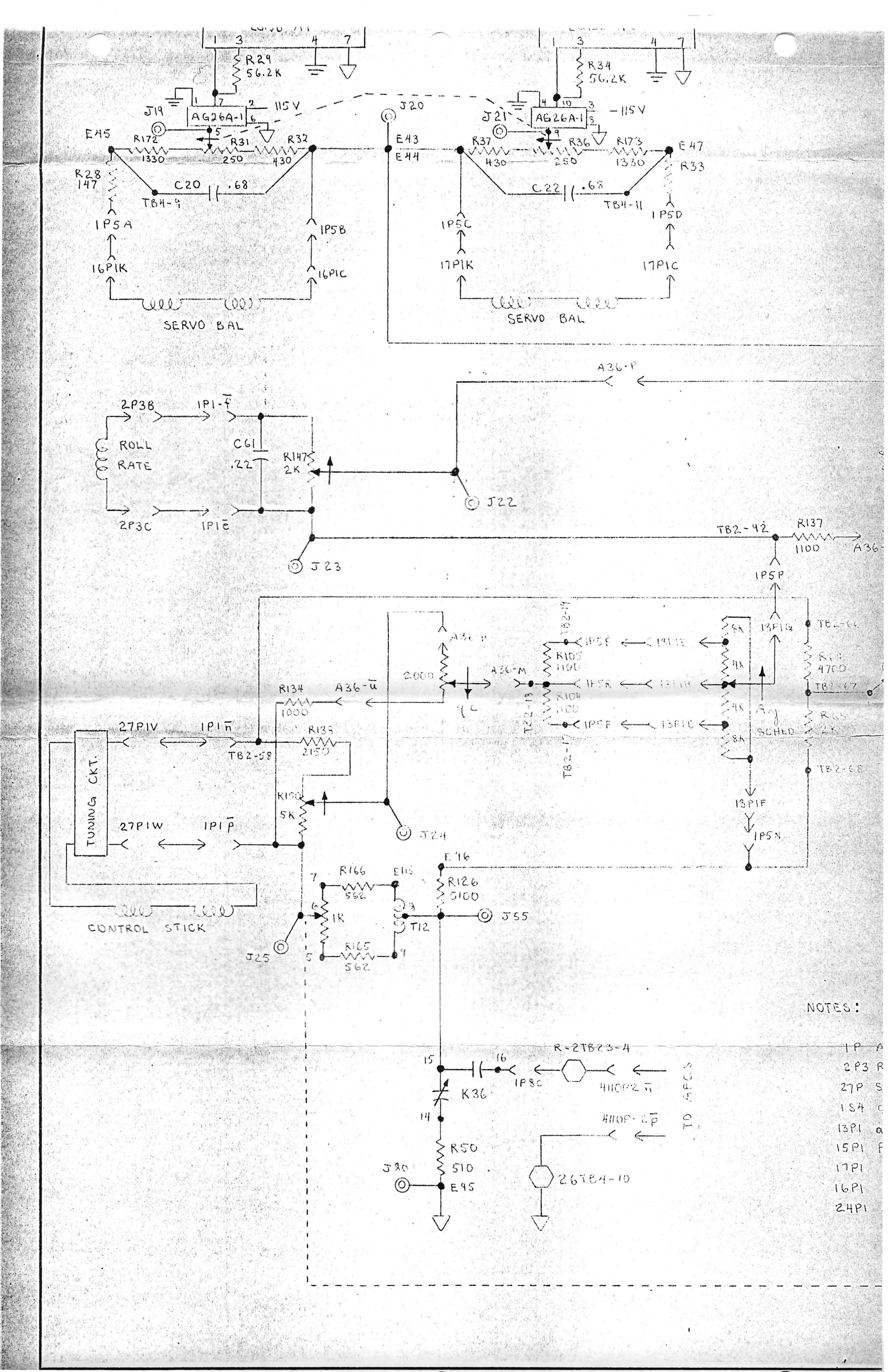
FIRST USED			SYM.	DESCRIPTION		APPROVAL
NEXT ASSEMBLY		REFERENCE				
		SK66126 REV. B				




(RETRACT FOR R. ROLL)
PARALLEL SERVO
TRANS VALVE COILS
15P1K
15P1L
15P1R
26TB4-7
1P2L
1P2T
C29
.15
5
A14
PAR SERVO
TRANS-AMP-DEMOD
EG 113B
8
1
3
4
7
115 V
400 ~
R45
90.9K
J58
10
4
AG26A-1
3
8
9
115 V
E41
R48
130
R47
150
R46
750
E42
E32
C28
.56
R53
133
TB2-63
1P3B
1P3A
15P1B
15P1A
SERVO
BAL
13
DN
12
J59
K19
UP
11
ENG
5
6
4
K21
PRE
UP
1
K19
3
DN
2
TB2-100
TB2-101
R82
82.5
TB2-96
14
R83
2000
C40
.56
15
TB2-99
ENG
2
3
K21
PRE
1
J60
TO TEST
EQUIPMENT
1P1J
J56
115V
J57
A26-S
EG153
AMP
A18
T-3
TB2-2
R58
310
TB2-1
R59
750
TB2-4
TB2-116
1P3D
15P1C
R92
619
R91
619
TB2-115
R89
2K
C43
.47
PRESSURE
TRIM
SENSOR
TB2-117
R90
909
TB2-118
1P3C
15P1D
ENG
13
K21
11
PRE
12
J61
1P8D
1P8L
CONNECT
IF NO
AFCS
R-2TB236
1P1g
TO TEST
EQUIPMENT
1P1h
115 V 400~
E90
A22
EG129A
C39
.27
M
E91
R87
28.7K
E75
T5
E74
4110P2c
4110P2d
AFCS
GEAR DOWN


R45
90.9K
J58
AG26A-1
115V
E41
R48
130
R47
150
R46
750
E42
E32
C28
.56
R53
133
TB2-63
IP3B
IP3A
15P1B
15P1A
SERVO
BAL
13 DN
12
J54
UP 1
K19
11 UP
ENG 5
6
K21
4 PRE
K19
3
DN
2
TB2-100
TB2-101
R82
82.5
TB2-96
14
R83
2000
C40
.56
15
TB2-99
ENG
2
K21
3
1 PRE
J60
TO TEST EQUIPMENT
IP1
J56
115V
6
7
J57
A26-5
1
EG153 AMP A18
4
5
3
2
TB2-2
T-3
R54
310
TB2-1
R59
750
TB2-4
3
6
4
7
TB2-116
IP3D
15P1C
R92
619
R91
619
TB2-115
R89
2K
C43
.47
PRESSURE TRIM SENSOR
R90
909
TB2-117
ENG 13
11
K21
12
PRE
J61
TB2-118
IP3C
15P1D
IP1
TO TEST EQUIPMENT
IP8D
IP8L
CONNECT IF NO AFCS
R-2TB236
-5
110P2 C
410P2 d
AFCS
GEAR DOWN
115V 400~
E40
A22
EG129A
C39
.27
M
R87
28.7K
E91
E75
T5
1
6
3
7
8
K21
9
E74
C41
.56
24
ENG
16
R84
750
TB2-33
15
K21
14
R85
320
R86
30
TB2-49
TB2-32
23
VG
R88
100
ONS
(ROLL)
E
DULE POT
) POT
SERVO
VO RIGHT
VO LEFT
M ACTUATOR


SERIES SERVOS
(RETRACT FOR R. ROLL)
LEFT AILERON
TRANS VALVE COILS
16P1G
16P1F
16P1E
26TB4-6
1P2C
1P2D
C21 .15
SERIES SERVO
TRANS-AMP-DEMOD
EG13B A9
115 V
400 ~
R29
56.2K
(EXTEND FOR R. ROLL)
RIGHT AILERON
TRANS VALVE COILS
17P1G
17P1F
17P1E
26TB4-6
1P2R
1P2S
C23 .15
SERIES SERVO
TRANS-AMP-DEMOD
EG13B A10
115 V
400 ~
R34
56.2K
J19
AG26A-1
115 V
J20
J21
AG26A-1
115 V
E45
R172
1330
R31
250
R32
430
R28
147
C20 .68
TB4-9
1P5A
16P1K
1P5B
16P1C
SERVO BAL
E43
E44
R37
430
R36
250
R173
1330
E47
R33
C22 .68
TB4-11
1P5D
1P5C
17P1K
17P1C
SERVO BAL
A36-P
2P3B
1P1-f
ROLL
RATE
C61
.22
R147
2K
J22
2P3C
1P1-e
J23
TB2-92
R137
1100
A36-m
880
1P5P
A36-P
TB2-11
1P5E
R105
1100
2000
A36-M
R134
1000
A36-u
1P5R
R104
1100
1P5F
TB2-13
TB2-17
13P1G
4K
4K
8K
Any
SCHED
13P1F
1P5N
TB2-67
4700
R66
2K
TB2-69
27P1V
1P1n
R138
2150
TB2-58
TUNING CKT.
27P1W
1P1p
R100
5K
J24
E16
R166
562
E15
R126
5100
7
6
1K
8
T12
9
J55
J25
R165
562
CONTROL STICK


R29
56.2K
R34
56.2K
J19
AG26A-1
115V
J20
J21
AG26A-1
-115V
E45
R172
1330
R31
250
R32
430
E43
E44
R37
430
R36
250
R173
1330
E47
R33
R28
147
C20
.68
TB4-9
C22
.68
TB4-11
1P5A
1P5B
1P5C
1P5D
16P1K
16P1C
17P1K
17P1C
SERVO BAL
SERVO BAL
A36-P
2P3B
1P1-f
ROLL
RATE
C61
.22
R147
2K
J22
2P3C
1P1-e
J23
TB2-42
R137
1100
A36-
1P5P
A36-P
2000
A36-M
R134
1000
A36-u
R105
1100
R104
1100
TB2-18
TB2-17
1P5E
1P5R
1P5F
13P1G
13P1R
13P1E
5K
4K
4K
8K
A
SCHED
TB2-67
TB2-68
13P1F
1P5N
27P1V
1P1n
TB2-58
R139
2150
TUNING CKT.
R130
5K
J24
27P1W
1P1p
E16
R166
562
E15
R126
5100
7
6
1K
5
T12
J55
J25
R165
562
9
CONTROL STICK
15
16
R-2TB23-4
1P8C
K36
14
R50
510
J30
E95
26TB4-10
NOTES:
1P A
2P3 R
27P S
1S4 C
13P1 e
15P1 P
17P1
16P1
24P1

A. DC POWER AND ENGAGING SEQUENCE

1. When the normal bus breaker is closed, 28V DC normal power is applied through the master power switch:

   (a) To one open contact of K30, the engage delay normal auxiliary relay.

   (b) Through a set of closed contacts of the B system pressure switch (if useable hydraulic pressure is present) to the solenoid of K5, the low pressure cut-out relay.

   (c) To one open contact of K15, the engage delay normal relay.

   (d) To the solenoid of the DC normal failure relay.

   (e) Through contacts of three AC normal failure relays (if AC power is present) to K18, the time delay normal thermal relay and to one open contact of K15, the engage delay normal relay.

   (f) Through contacts of a test switch to an open contact of the landing gear lock switch.

   (g) To two open contacts of K25 the yaw normal solenoid engage relay.

   (h) To one open contact of K29, the pitch parallel solenoid engage relay.

2. When the emergency bus breaker is closed, 28V DC is applied.

   (a) Through contacts of three AC emergency failure relays (if emergency AC power is present) to K17, the time delay emergency thermal relay and one open contact of K16, the engage time delay emergency relay.

   (b) To the DC emergency failure relay solenoid.

   (c) Through contacts of K30, the engage delay normal auxiliary relay, to one open contact of the engage button.

3. When 28V DC emergency power is applied to the aircraft, 28V DC is applied through a breaker:

   (a) Through contacts of K26, the pitch parallel solenoid engage auxiliary and/or contacts of K32, the roll parallel solenoid engage auxiliary, to the Roll, Pitch disengaged warning light.

   (b) Through contacts of K31 the damper engage relay to the damper disengage warning light.

   (c) To an open contact of K31 the damper engaged relay.

   (d) Through a second breaker to the stick trim button.

4. About 90 seconds after power is applied to it, K18 the time delay normal thermal relay closes, applying power to the solenoid of K15, the engage delay normal relay.

When K15 energizes, 28V DC is applied through contacts of K15 to:

(a) K15 solenoid as a holding circuit.

(b) Solenoid of K30, the engage delay normal auxiliary relay.

5. When K30 energizes:

(a) 28V DC emergency is removed from one contact of the engage button (see 2(d) above).

(b) 28V DC normal is applied to one contact of the engage button (the same contact mentioned in 5(a) above).

6. About 125 seconds after power is applied to it, K17, the time delay emergency thermal relay closes, applying power to the solenoid of K16, the engage time delay emergency relay. When K16 energizes, 28V DC is applied through contacts of K16 to the solenoid of K16 as a holding circuit.

The damper is now ready to be engaged.

ENGAGING SEQUENCE

B. Engaging Sequence (Normal)

1. When the engage button is depressed, 28V DC normal is applied:

(a) To the solenoid of K28, the lockout relay.

(b) Through contacts of K30 and/or K16 to the solenoid of K31, the damper engage relay.

(c) Through contacts of K30, K5, the emergency button and two sets of contacts of the stick disengage button to the solenoid of K24, the yaw disengage relay and to the solenoid of K25, the yaw normal solenoid engage relay. (If the yaw disengage limiter permits).

2. When power is applied to the solenoid at K31, the damper engage relay (1(b) above) contacts of this relay:

(a) Remove power from the damper disengaged warning light.

(b) Apply a holding circuit on one set of contacts of the engage button.

3. When power is applied to the solenoids of K24 and K25 (1(c) above) contacts of K25 maintain an open circuit to the emergency warning light.

Contacts of K24 and K25:

(a) Supply a holding circuit on a second set of contacts of the engage button.

(b) Apply power to the yaw normal solenoid of the dual differential servo.

The yaw channel is now engaged.

4. NOTE: K28 remains energized only while the engage button is held down.

When K24 and K25 are energized, normal 28V DC power is applied:

(a) Through closed contacts of the pitch disengage limiter, through one set of contacts of K28, the lockout relay to the left and right pitch differential servo engage solenoids, and to the solenoid of K23, the pitch differential solenoid engage relay. Contacts of K23 place a holding circuit across the contacts of K28.

(b) Through contacts of K23, contacts of the pitch disengage limiter, the pitch parallel anti-engage sector switch and contacts of K28 to the pitch parallel servo engage solenoid and to the solenoids of the pitch parallel solenoid engage relays, K29, K26, and K20. Contacts of K29 place holding circuits on the parallel servo anti-engage sector switch and the contacts of K28.

The pitch axis is now engaged.

5. When K24 and K25 are energized, normal 28V DC power is applied:

(a) Through contacts of K28 to the left and right roll differential servo engage solenoids and to the solenoid of K22, the roll differential solenoid engage relay. (If the roll rate disengage limiter permits). Contacts of K22 apply a holding circuit across the contacts of K28.

(b) Through contacts of K22 and the roll parallel servo anti-engage sector switch, and contacts of K28 to the roll parallel servo engaging solenoid and to the solenoids of the roll parallel solenoid engage relays K33, K32, K21. Contacts of K33 apply holding circuits to the anti-engage sector switch and to the contacts of K28.

The roll axis is now engaged

6. When both K26 (4(b) above) and K32 (5(b) above) are energized, contacts of these relays remove power from the Roll, Pitch Disengage Warning Light.

AFCS Engage

When both the pitch and roll axis are engaged, relays K29 and K33 are energized and apply 28V DC normal power through their contacts, and the platform interlock, the AFCS anti-engage switch, the AFCS disengage button to contacts of the AFCS engage switch.

When the AFCS engage switch is closed, 28V DC power is applied to the switch holding solenoid and to the solenoids of two damper relays, K36 and K37.

Gear Down Configuration

When the gear is down, the landing gear lock switch closes, applying power through a second test switch, to the solenoids of the gear down relays K19, K38 and K27.

Contacts of K27 short out one part of the yaw disengage limiter.

C. Roll-Pitch Disengage

1. General

The pilot cannot disengage the roll and pitch axes individually. By selecting the emergency mode he may disengage the roll and pitch axes, but he will not remain on normal yaw damper. The emergency yaw system will come into use.

The pitch and roll axes can disengage individually automatically by the action of the pitch disengage limiter and the roll rate limiter. In these cases, the yaw axis will remain in the normal mode.

2. Pitch Disengage

(a) If any contacts of relays K1G, K2G, K3G, K4G open due to action of the pitch limiter, power will be removed from the pitch parallel servo engaging solenoid and the pitch parallel solenoid engage relays K39, K26 and K20. When K29 de-energizes, hold circuits are removed from K28 contacts and contacts of the parallel servo anti-engage sector switch. Since K28 contacts were closed only while the engage button was held down, the circuit will remain de-energized.

(b) The same set of pitch limiter contacts will remove power from the pitch differential servo solenoids and the solenoid of K23, the pitch differential solenoid engage relay. When K23 de-energizes the hold circuit across the contacts of K28 is removed and the differential servos will remain disengaged. Contacts of K23 also remove power to the parallel servo circuits thus disengaging the parallel servo. If the right pitch differential servo is moving in a direction which will oppose the excessive 'g' load which has caused one of the limiter relays to open, the direction switch of the servo will produce a circuit around the open contact of the limiter and prevent disengagement of the differential servos. However, the parallel servo will still disengage.

In summary, two conditions can exist:

(i) Both pitch differential and parallel servos disengaged.
(ii) Parallel servo only disengaged.

3. Roll Disengage

If the roll rate limiter contacts open, the ground path of K22, the roll differential solenoid engage relay is removed and the relay will de-energize. When K22 de-energizes, power is removed from the roll differential servo engage solenoids and K22. Another set of contacts of this relay, K22 removes power from the roll parallel servo engage solenoid and the roll parallel solenoid engage relays K33, K32 and K21. The differential and parallel servos will remain disengaged.

NOTE: In both of the automatic disengage actions described above, the yaw axis will remain in the normal mode.

Indicator Light

If the pitch parallel servo disengage, relay K26 de-energizes. Contacts of this relay will then apply 28V DC to the Roll, Pitch disengage warning light.

Similarly, if the roll parallel servo disengages, K32 de-energizes which also applies power to the Roll Pitch disengage warning light. The Roll Pitch disengage warning light will come on when:

(a) Roll parallel and differential servos disengage.

(b) Pitch parallel servo disengages.

(c) Pitch parallel and differential servos disengage.

(d) (a) and (b), (a) and (c).

NOTE: It is possible for the roll, pitch disengage warning light to be on when the pitch differential servos are still operating.

AFCS

If the AFCS is engaged it will automatically disengage through the action of contacts of K29, the pitch parallel solenoid engage relay or K33, the roll parallel solenoid engage relay if either the roll or pitch parallel servo disengages.

D. EMERGENCY MODE ENGAGE

1. General

(a) The emergency mode cannot be engaged until after the normal mode has been engaged unless there is some system malfunction or normal power failure. The difference in thermal delays (normal 90 sec., emergency 125 sec.) prevents emergency system engagement before the normal system when power is initially applied to the system.

(b) Unless there has been failure of either or both the normal AC and DC power, the system will continue to use normal power when in the emergency mode.

(c) When in the emergency mode, both the pitch and roll axes will be disengaged.

2. Means By Which Emergency Mode May Be Engaged

(a) Pilot Selection

(i) Function selector emergency button.
(ii) Stick disengage button.
(iii) Normal AC breakers.
(iv) Normal DC breakers.

(b) Automatic Selection

(i) Yaw disengage limiter action.
(ii) B system pressure failure.
(iii) Normal AC power failure.
(iv) Normal DC power failure.

3. Pilot Selection

(a) Emergency Button or Stick Disengage Button

When the pilot depresses either of these buttons, power is removed from both K24, the yaw disengage relay and K25, the yaw normal solenoid engage relay. When K24 and K25 de-energize, contacts of these relays:

(i) Remove power from the yaw normal engage solenoid of the rudder dual differential servo.
(ii) Remove power from the pitch axis engaging circuits.
(iii) Remove power from the roll axis engaging circuits.
(iv) Apply power to the emergency warning light.
(v) Apply power to the yaw emergency engaging solenoid of the dual differential servo through contacts of K31, the damper engage relay.

(vi) Removes holding circuit across one set of contacts of the engage switch thus ensuring that power will not be re-applied to K24 and K25 when the disengage button (or emergency button) is released.

(b) Normal AC or DC Breakers

If the pilot opens the normal DC breakers the DC normal failure relay will de-energize and contacts of this relay will remove all AC normal power from the system. Removal of AC power will cause all AC normal power failure relay contacts to open. This will also occur if the pilot opens the AC normal breakers.

When the AC failure relay contacts open, power is removed from both K18 the time delay normal thermal relay and the solenoid of K15, the engage delay normal relay.

When K15 de-energizes contacts of this relay remove power from the solenoid of K30, the engage delay normal auxiliary relay. Contacts of K30:

(i) Remove normal power from one set of contacts of the engage button and supply them with emergency power.

NOTE: During this momentary power change, K31, the damper engage relay will not drop out. The discharge current of the resistor capacitor network across the solenoid of K31 holds it energized while the power changes. (40 to 50 milliseconds).

(ii) Open circuit path to K31, the damper engage relay. However, these contacts are by-passed by contacts of K16, the engage time delay emergency relay.

(iii) Removes power from K24 and K25. The subsequent disengaging action is identical to that described when the pilot uses either the emergency button or the stick disengage button. (3,a above)

4. Automatic Selection

(a) Yaw Disengage Limiter Action

When the yaw disengage limiter acts, it opens the path to ground from K25, the yaw normal solenoid engage relay. The subsequent disengage action is similar to that described in 3,a above when the pilot uses the emergency, or stick disengage buttons.

(b) B System Pressure Failure

When the system hydraulic pressure falls below useable limits, power is removed from K5, the B system pressure cut-out relay. Subsequent disengaging action is identical to that described in 3,a above.

(c) Normal AC or DC Power Failure

Disengaging action is identical to that described when the pilot opens either the DC or AC breakers (3,b above).

E. RE-ENGAGING

1. If disengagement of either the pitch axis or the roll axis was caused by limiter action, and the aircraft has returned to within limits, the pilot may re-engage pitch and roll by depressing the engage button.

2. If disengagement of pitch and roll and switch to emergency yaw was caused by action of the yaw disengage limiter, and the aircraft has returned to within yaw limits, the normal system may be re-engaged by depressing the engage button.

3. If emergency mode engagement was caused by normal power failure or normal system malfunction, the normal system cannot be re-engaged by depressing the engage button or by any other means.

F. AC POWER

1. Normal System (SK66129)

When normal AC power is applied to the aircraft and the AC breakers are closed, three phase 115V 400 cycle power is applied to three sets of contacts of the normal DC power failure relay. If normal DC power is present, AC power will be applied through contacts of this relay as follows:

(a) Phase A, B and C through the normal gyro spin motor transformer to the pitch, roll, and yaw normal rate gyro motor windings.

(b) Phase A

(i) Through a rectifier to the solenoid of one of the AC failure relays.

(ii) To all of the normal system amplifier power transformers.

(iii) Through the normal yaw axis transformer to:

(1) Normal qc characterizer pot.
(2) Normal qc scheduler motor and velocity generator.
(3) Yaw rate high pass motor and velocity generator.
(4) To all yaw axes normal signal transducer primaries except gyro.

(iv) Through the roll axis bridge transformer to:

(1) The roll trim pot.
(2) The stick force transducer primaries (roll and pitch).
(3) Roll integrator motor and velocity generator.
(4) All roll axes signal transducer primaries except gyro.

(v) Through the pitch axis bridge transformer to:

(1) The pitch trim circuit.
(2) The gear down pitch trim circuit.
(3) The pitch integrator motor and velocity generator through contacts of K27 and K26 or K8.
(4) To the pitch high-pass motor and velocity generator.
(5) To all pitch axis signal transducers primaries except gyro.

(vi) Through the trim and fader motor transformer to:

(1) Pitch and roll trim motors.
(2) Fader motor.
(3) All gyro signal pickoff primaries.

(vii) To all normal quadrature signal filters.

(viii) To the pitch disengage limiter amp-cal.

(ix) To the roll rate gyro heater switch.

(x) To the roll rate limiter heater switch.

(c) Phase B

(i) To the pitch rate gyro heater switch.
(ii) To the emergency yaw rate gyro heater switch.
(iii) To the emergency lateral accelerometer heater switch.
(iv) To a normal accelerometer heater servo.
(v) Through a rectifier to the solenoid of one of the AC failure relays.

(d) Phase C

(i) To the yaw rate gyro heater switch.
(ii) To the lateral accelerometer heater switch.
(iii) To the lateral scheduling accelerometer heater switch.
(iv) Through a rectifier to the solenoid of one of the AC failure relays.

2. Emergency System (SK66130)

When the emergency AC breakers are closed 115V three phase 400 cycle emergency power is applied to three sets of contacts of the DC emergency power failure relay. If emergency DC power is present, AC power will be applied through contacts of this relay as follows:

(a) Phase A, B and C through the emergency gyro spin motor transformer to the emergency yaw rate gyro motor windings.

(b) Phase A

(i) Through a rectifier to the solenoid of one of the emergency AC failure relays.

(ii) To emergency system amplifiers.

(iii) To the emergency quadrature signal filter.

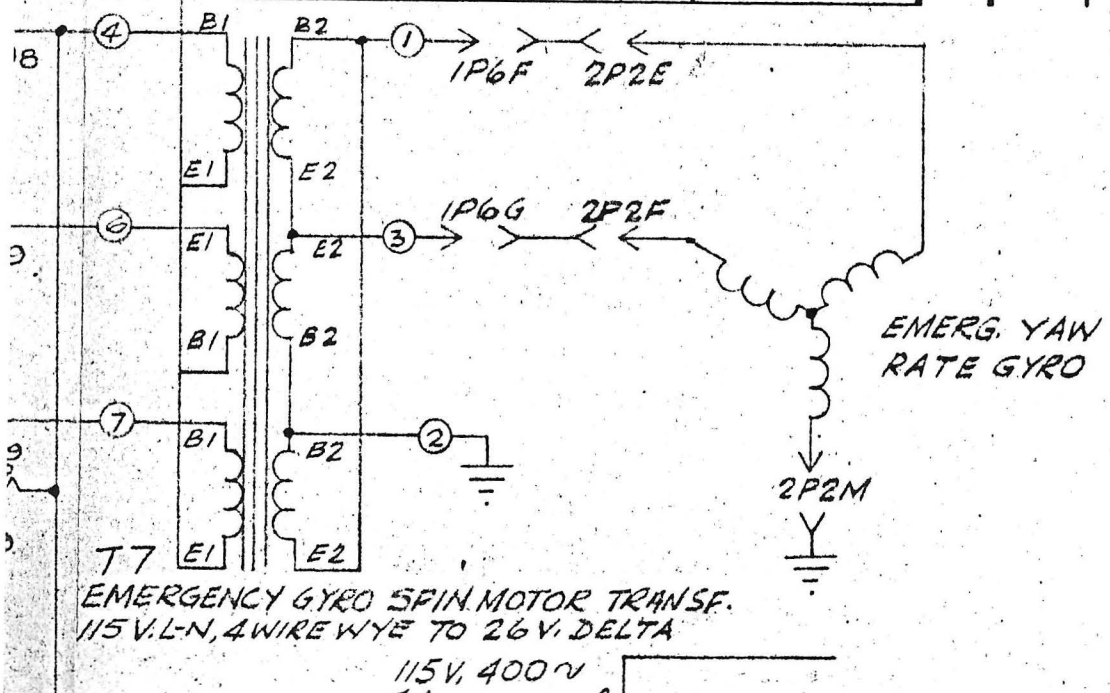
(iv) Through the emergency yaw axis bridge transformer to:

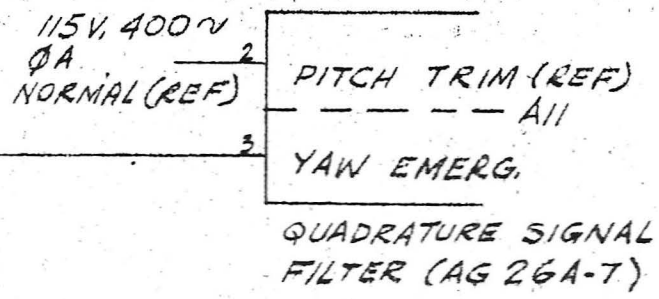
1. Emergency high-pass motor and velocity generator.
2. All signal transducer primaries.

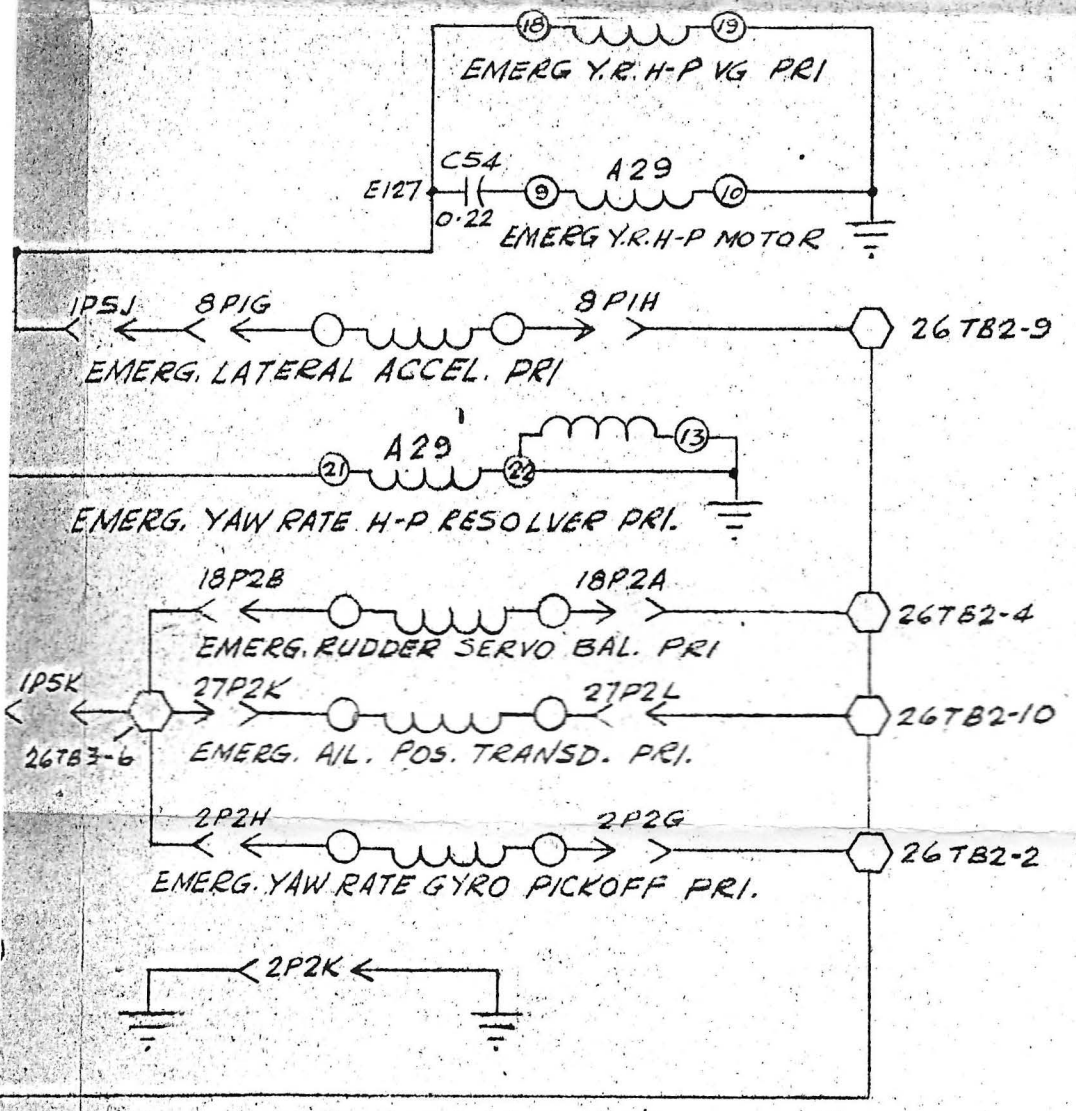
SUPPLEMENTARY INFORMATION		
FIRST USED		
NEXT ASSEMBLY		REFERENCE
		SK66130
		REVA.

FIRST CLASS.

REVISIONS			
SYM.	DESCRIPTION	DATE	APPROVAL







CSK1777

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±2°
OTHER ANGLES		±1°
DEV.		
SUPSD.		
GOVT. MATL. SPEC.		

DRAWN BY		T. MASON	10-15-54
DRAFTING	√		
MATL. & FIN.	√		
DESIGNER	√		
DESIGN	√		
APPROVED			

REF. MH. MATL. (SPEC.)

FINISH (SEE NOTE)

EMERGENCY AC POWER CIRCUIT
SCHEMATIC DIAGRAM
DEVELOPMENT SYSTEM
MH 64 DAMPER (YG 339B-1)

SCALE NONE WT. ACT. CALC. LB.

HONEYWELL CONTROLS LIMITED
AERONAUTICAL DIVISION
TORONTO, CANADA

C-SK1777

(REF)
26TB2-1

GRD

1P66(REF)

A — 14P1E — 14P1K — 1P6Z̄

27P2Z

B — 14P1G — 14P1M — 1P6P̄

27P2ā

C — 14P1N — 14P1P — 1P6ḡ

27P2b̄

K1F

400 ~ AIRCRAFT POWER SOURCE
3 ϕ, 4 WIRE WYE, 115 V. L-N
(FROM EMERG A·C BUS)

TB4-20 — C5 .47

TB4-21 — C8 .47 — J40 — E108

TB4-19 — C6 .15 — E109

E110 — C7 .15 — R161 5100 — R160 5100 — R159 5100

K11 — K10 — K9

AC FAILURE RELAYS

T7
EMERG
115 V. L-

A35-1 — A35-7

A35-6

1P4K — 1P4L

HINGE MOMENT LIM.
TRANSD PRI

27P4B

PG 29
A 35

27P4A

A 13 — E81

EMERG. RUDDER AMP.

A 28 — E122

EMERG. YAW RATE H-P AMP.

T8
EMERG. YAW AXIS
BRIDGE TRANSF.

45V — 1P5J

EMERG.

26V

EMERG.

10V — 1P5K

26TB3-6

1P5L
(SPARE)

EM

TOLE

ONE PLAC
TWO PLAC
THREE PL
90° FORM
OTHER AN

DEV.

SUPSD.

GOVT. MAT

28VDC NORMAL BUSS

MASTER POWER SWITCH

28VDC EMERG. BUSS

TO STICK TRIM BUTTON

A.C. EMERG. FAILURE

D.C. EMERG. FAILURE

K16 ENGAGE TIME DELAY EMERG.

K17 TIME DELAY EMERG.

K30

K31

ENGAGE BUTTON

K26

K32

K31

K25

EMERGENCY

DAMPER DISENGAGED

ROLL, PITCH DISENGAGE

LOCKOUT RELAY K28

K31 DAMPER ENGAGE

K16

K24

K25

YAW EMERG. SOLENOID

EMERG.

K5

STICK DISENGAGE

B SYST. PRESSURE SW.

K5 LOW PRESSURE CUT-OUT SYST. B

K15

K30 ENG. DELAY NORMAL AUX.

K25

YAW DISENGAGE K24

YAW NORM. SOL. ENG.

K24

K25

D.C. NORMAL FAILURE

A.C. NORMAL FAILURE

K15 ENG. DELAY NORMAL

K18 TIME DELAY NORMAL

K27

B LIMITER

LATERAL ACC. LIMITER

YAW NORMAL SOLENOID

TEST SW.

LGLS

TEST SW.

K27 GEAR DOWN

K19 K38 GEAR DOWN AUX.

RIGHT PITCH DIFFERENTIAL SERVO DIRECTION SW.

K28

K23

K36 K16 K26 K46

K23 PITCH DIFF. SOL. ENG.

L R

PITCH DIFF. SOLENOIDS

K23

K26 K16 K46 K36

PITCH PARALLEL ANTI-ENG.

K29

K28

K29

K29 PITCH PAR. SOL. ENG.

K26

K20

PITCH ENG. AUX. PAR. SOL.

PITCH PARALLEL SOLENOID

K25

K24

K28

K22

ROLL DIFF. SOL ENG.

K22

ROLL RATE LIMITER

L R

ROLL DIFF. SOLENOIDS

ROLL PAR. ANTI-ENG.

K22

K33

K28

K33

K33 ROLL PAR. SOL. ENG.

K32

K21

ROLL PAR. SOL. ENG. AUX.

ROLL PARALLEL SOLENOID

AFCS ANTI-ENG

AFCS ENGAGE SW.

K29

K33

PLATFORM INTERLOCK

AFCS DISENGAGE

HOLD SOL.

K36 K37 AFCS ENG.

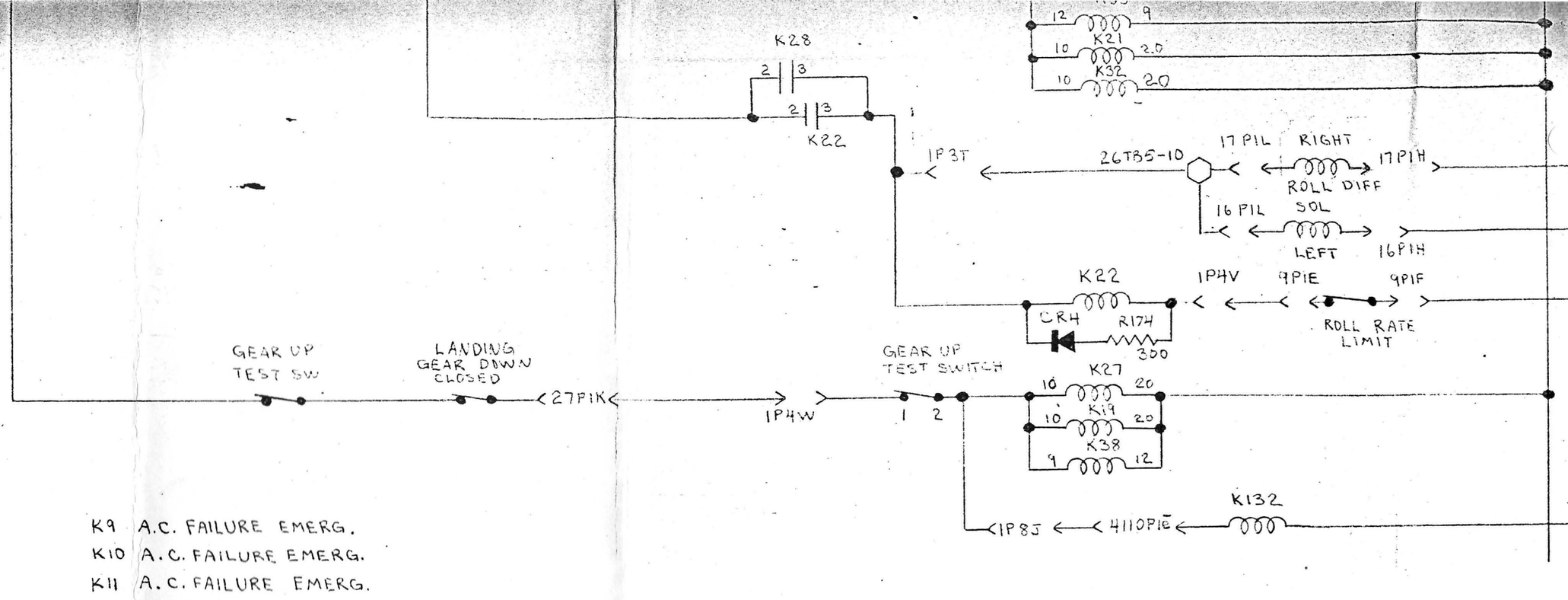
NOTE: K16, K26, K36, K46 ARE NORMALLY CLOSED UNTIL THE 'G' LOAD EXCEEDS SET LIMITS

MOVEABLE CONTACTS AS SHOWN IN THIS DIAGRAM MAY BE STATIONARY – AND STATIONARY CONTACTS MAY BE MOVEABLE IN ACTUAL APPLICATION

D. C. FLOW DIAGRAM
CF-105 DAMPER

FOR TRAINING PURPOSES ONLY

21/8/58



K9 A.C. FAILURE EMERG.
K10 A.C. FAILURE EMERG.
K11 A.C. FAILURE EMERG.
K12 A.C. FAILURE NORMAL
K13 A.C. FAILURE NORMAL
K14 A.C. FAILURE NORMAL
K15 ENGAGE DELAY NORM
K16 ENGAGE DELAY EMERG.
K17 THERMAL TIME DELAY EMERG.
K18 THERMAL TIME DELAY NORMAL
K5 LOW PRESSURE CUT-OUT SYS. B
K25 YAW NORM SOL ENGAGE
K23 PITCH DIFF SOL ENGAGE
K29 PITCH PAR SOL ENGAGE
K20 PITCH PAR SOL ENGAGE AUX.
K22 ROLL DIFF SOL ENGAGE
K33 ROLL PAR SOL ENGAGE
K21 ROLL PAR SOL ENGAGE AUX.
K30 ENGAGE DELAY NORM AUX.

K1 LO-SPEED TRIM UP (ENTIRELY IN PITCH SCHEMATIC)
K31 DAMPER ENGAGE
K2 LO-SPEED TRIM DN (ENTIRELY IN PITCH SCHEMATIC)
K24 REDUNDANT YAW DISENGAGE
K6 TRIM ACT UP
K7 TRIM ACT DN } (ENTIRELY IN PITCH SCHEMATIC)
K3 ROLL TRIM LEFT
K4 ROLL TRIM RIGHT } (ENTIRELY IN ROLL SCHEMATIC)
K34 PITCH TRIM UP
K35 PITCH TRIM DN } (ENTIRELY IN PITCH SCHEMATIC
K27 GEAR DOWN
K19 GEAR DOWN AUX.
K37 AFCS-ENG AUX.
K8 LO-SPEED INTEGRATOR TRIM (ENTIRELY IN PITCH SCHE
K36 AFCS ENGAGE
K26 PITCH PAR ENGAGE AUX
K32 ROLL PAR ENGAGE AUX
K19 GEAR DOWN AUX.
K106 AFCS ENGAGE AUX (IN AFCS AMP CAL)
K28 LOCKOUT RELAY
K132 GEAR DOWN AUX (IN AFCS AMP CAL)

CSK 1780

TOLERANCE UNLESS NOTED				
ONE PLACE	(.0)	±.020	DRAWN BY	Corbett 10-16-58
TWO PLACE	(.00)	±.010	DRAFTING	✓
THREE PLACE	(.000)	±.005	MATL. & FIN.	✓
90° FORMED ANGLES		±°	DESIGNER	✓
OTHER ANGLES		±		
DEV.			DESIGN	✓
SUPSD.			APPROVED	
GOVT. MATL. SPEC.			REF. MH. MATL. (SPEC.)	

FINISH (SEE NOTE)

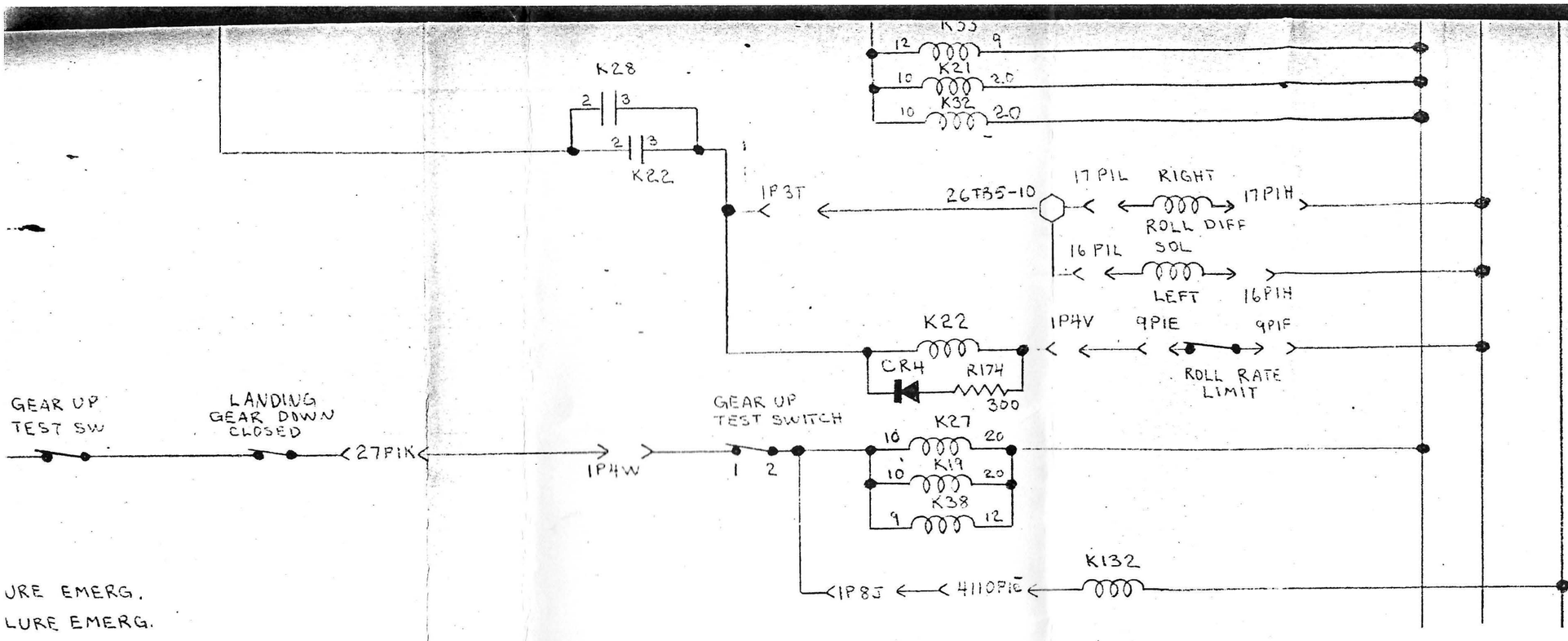
SCALE None SIZE WT. ACT. CALC. LB.

M.H. 64 DAMPER (DEVELOPMENT)
Y@ 3398-1 DC. POWER SCHEMATIC

HONEYWELL CONTROLS LIMITED
AERONAUTICAL DIVISION
TORONTO, CANADA

CSK 1780

CSK 1780



URE EMERG.
LURE EMERG.
URE EMERG.
URE NORMAL
LURE NORMAL
LURE NORMAL
E DELAY NORM
E DELAY EMERG.
L TIME DELAY EMERG.
L TIME DELAY NORMAL
ESSURE CUT-OUT SYS. B
NORM SOL ENGAGE
DIFF SOL ENGAGE
PAR SOL ENGAGE
PAR SOL ENGAGE AUX.
DIFF SOL ENGAGE
AR SOL ENGAGE
AR SOL ENGAGE AUX.
E DELAY NORM AUX.

K1 LO-SPEED TRIM UP (ENTIRELY IN PITCH SCHEMATIC
K31 DAMPER ENGAGE
K2 LO-SPEED TRIM DN (ENTIRELY IN PITCH SCHEMATIC)
K24 REDUNDANT YAW DISENGAGE
K6 TRIM ACT UP
K7 TRIM ACT DN
} (ENTIRELY IN PITCH SCHEMATIC)
K3 ROLL TRIM LEFT
K4 ROLL TRIM RIGHT
} (ENTIRELY IN ROLL SCHEMATIC)
K34 PITCH TRIM UP
K35 PITCH TRIM DN
} (ENTIRELY IN PITCH SCHEMATIC
K27 GEAR DOWN
K11 GEAR DOWN AUX.
K37 AFCS-ENG AUX.
K8 LO-SPEED INTEGRATOR TRIM (ENTIRELY IN PITCH SCHEMATIC)
K36 AFCS ENGAGE
K26 PITCH PAR ENGAGE AUX
K32 ROLL PAR ENGAGE AUX
K19 GEAR DOWN AUX.
K106 AFCS ENGAGE AUX (IN AFCS AMP CAL)
K28 LOCKOUT RELAY
K132 GEAR DOWN AUX (IN AFCS AMP CAL)

CSK1780

SUPPLEMENTARY INFORMATION

FIRST USED	NEXT ASSEMBLY	REFERENCE
		SK66125 Rev B

FIRST CLASS.

REVISIONS

SYM.	DESCRIPTION	DATE	APPROVAL

K1 (REF)

K2 (REF)

K26 (REF)

K32 (REF)

ENGAGE

EMER

STICK DISC.

RIGHT PITCH DIFF SERVO SW

EXTEND

RETRACT

26TB5-2

26TB5-3

26TB5-5

26TB5-4

26TB5-6

26TB5-7

26TB5-8

26TB5-9

J72

J73

J75

YAW NORM SOL

RIGHT PITCH DIFF SOL

LEFT

AFCS ENG

AFCS DISENGAGE

PITCH PAR SOL

ROLL PAR SOL.

PITCH PAR ANTI ENG

ROLL PAR ANTI ENG

ENGAGE
1P4S
14P1U
1P1C̄
1P4R
K24
K24
K30
K5
14P1V
EMER
1P4U
K28
20
10
14P1ē
14P1H̄
1P4C̄
J72
STICK DISC.
27P1X
27P1Y
1P4d̄
12
K24
9
K27
27P1Z
1P4ē
10
K25
20
1P6N
26TB3-7
26TB3-8
Ay
11P1B
26TB2-6
1P4P
27P2A
27P2B
B
11P1A
11P1C
R-2TB21-6
RIGHT
PITCH DIFF
SERVO SW
2P1N
2P1M
EXTEND
RETRACT
21P1D
23P1B
23P1C
23P1A
K25
9
9
1P7F̄
26TB5-6
18P1L
YAW NORM
SOL
18P1H
RIGHT
PITCH
DIFF SOL
26TB5-7
21P1L
21P1H
20P1L
20P1H
LEFT
4110P1F̄
1P7d̄
6P1W
6P1V
K36
K16
K26
K46
6P1b̄
1P7b̄
K23
5
6
K23
10
20
K28
8
3
4110P1D
K100
K37
9
12
K36
20
10
4130P1c̄
1P8H
26TB5-4
26TB5-2
26TB5-3
J38
1P5T
18P2L
18P2H
K29
6
13
K33
8
14
1P8F
4110P1b̄
4110P1a̅
27P1ē
AFCS
ENG
27P1m̄
AFCS
DISENGAGE
4130P1T
4130P1H
4110P1U
2
1
K21
6P1T
6P1S
K46
K16
6P1U
26TB5-8
26TB5-5
K36
K26
1P7ē
23
18
PITCH PAR
ANTI ENG
19P1Q
19P1V
1P7b̄
1P7f̄
14
8
K29
J75
1P7j̄
18P1T
19P1M
PITCH PAR
SOL
K28
K29
K29
12
9
K20
10
20
K20
10
20
K24
14
8
K22
3
3
26TB5-9
ROLL PAR
ANTI ENG
15P1Q
15P1V
1P3L
1P3j̄
K33
1P3K
K28
K33
J73
1P3S
15P1T
15P1M
ROLL PAR
SOL.
K33
12
9

(FROM MAIN BUS) 28V DC NORMAL

(FROM EMER. BUS) 28V DC. NORMAL

27P1E ⟷ 1P4X

K31 13 5 — 1P4J̄ ⟷ 27P1A

DAMPER DISENG

K31 13 6 — K25 5 4 — 1P4ā ⟷ 27P1B

EMER

27P1D

K26 17 18 — K32 1 2 — 1P4b̄ ⟷ 27P1C

R-P DISENG

27P1H 27P1G 27P1F

TO STICK TRIM BUTTON COMMON

14P1F 14P1A 14P1C

MASTER POWER SW.

K1F 11 4 → 14P1S

K2F 11 4

14P1H 14P1B 14P1D

26TB4-1

1P6d̄ 1P6B 1P6C

1P6W

K12 K13 K14

K18 3 2 — 5 — 7

CR1

K15 2 1

K15 3 7

K15 6 5

K30 12 9

1P4T

TO TEST EQUIPMENT

J36 J37

K9 K10 K11

K17 3 2 — 5 — 7

CR2

K16 2 1

K16 3 7

R2TB24-2 → 27P1C̄ — 27P1d̄ ← H110P1n̄

STICK FORCE RELAY

ROLL FORCE SW.

PITCH FORCE SW.

SHIPS GROUND

HYD. PRESS SW

26TB4-2

27P2f̄ — 27P2ḡ ← 1P6U

K5 5 1

R-2TB21-5

ENGAGE

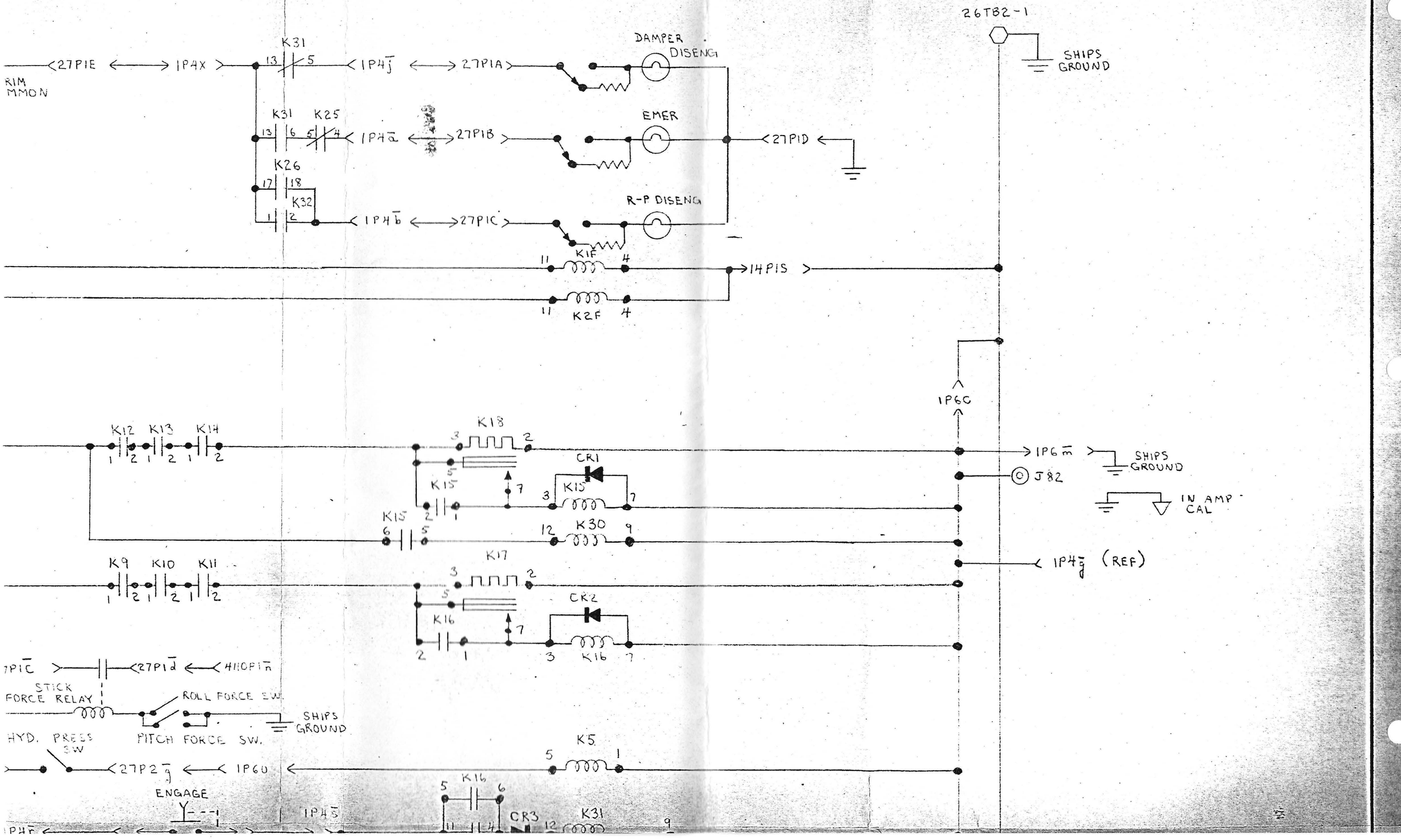
H110P1m̄

K30 2 10 — 1P4F̄ ← 1P4S̄

K16 5 6

CR3

K31 12 9


26TB2-1
SHIPS GROUND
27P1E
1P4X
K31
13
5
1P4J
27P1A
DAMPER DISENG
RIM MMON
K31
K25
13
6
5
4
1P4A
27P1B
EMER
27P1D
K26
17
18
K32
1
2
1P4B
27P1C
R-P DISENG
K1F
11
4
14P1S
K2F
IP6C
K12
K13
K14
K18
3
2
CR1
K15
7
IP6M
SHIPS GROUND
J82
IN AMP CAL
K30
12
9
K9
K10
K11
K17
CR2
K16
IP4J (REF)
STICK FORCE RELAY
ROLL FORCE SW.
PITCH FORCE SW.
SHIPS GROUND
HYD. PRESS SW
27P2G
IP6D
K5
ENGAGE
IP4S
CR3
K31

26TB4-1
1P6D̄
1P6B
1P6C
1P6W
K12 K13 K14
K18
CR1
K15
1P4T
TO TEST EQUIPMENT
J36
J37
K30
K17
CR2
K16
K9 K10 K11
R2TB24-2
27P1C̄
27P1D̄
H110P1H̄
STICK FORCE RELAY
ROLL FORCE SW.
PITCH FORCE SW.
SHIPS GROUND
26TB4-2
HYD. PRESS SW
27P2F̄
27P2Ḡ
1P6U
K5
R-2TB21-5
ENGAGE
K16
CR3
K31
1P4S̄
H110P1H̄
1P4F̄
14P1T
14P1Ū
K25
K24
C66
40μ
R167
51
J38
YAW EMER SOL
1P5T
18P2L
19P2H
14P1U
1P1C̄
1P4R
14P1V
1P4U
K28
EMER
14P1Ē
14P1H̄
1P4C̄
J72
STICK DISC.
27P1X
27P1Y
1P4D̄
27P1Z
1P4Ē
K27
1P6N
26TB3-7
26TB3-8
1P4P
27P2A
27P2B
B
11P1A
11P1B
11P1C
23P1B
23P1C
23P1A
K1 (REF)
K2 (REF)
K26 (REF)
RIGHT PITCH DIFF SERVO SW
2P1N
2P1M
EXTEND
RETRACT
21P1D
6P1W
1P7F̄
26TB5-6
18P1L
YAW NORM SOL
18P1H
26TB5-7
RIGHT PITCH DIFF SOL
21P1L
21P1H
20P1L
20P1H
LEFT
1P7D̄

K12 K13 K14

K18

CR1

K15

K15

K30

IP6C

IP6 $\bar{m}$

SHIPS GROUND

J82

IN AMP CAL

K9 K10 K11

K17

CR2

K16

IP4 $\bar{g}$ (REF)

2 27P1 $\bar{c}$

27P1 $\bar{d}$

41IOP1 $\bar{n}$

STICK FORCE RELAY

ROLL FORCE SW.

PITCH FORCE SW.

SHIPS GROUND

HYD. PRESS SW

$\bar{f}$

27P2 $\bar{g}$

IP6D

K5

ENGAGE

IP4 $\bar{s}$

IP4F

K31

14P1T

14P1 $\bar{a}$

K16

CR3

K31

C66 40μ

R167 51

K25 K31 K30

J38

K25

K24

K24

YAW EMER SOL

19P2H

IP5T

18P2L

ENGAGE

IP4S

14P1U

IP1 $\bar{c}$

IP4R

14P1V

EMER

IP4U

K30 K5

K28

20 10

14P1 $\bar{e}$

14P1 $\bar{k}$

IP4 $\bar{c}$

J72

STICK DISC.

27P1X

27P1Y

IP4 $\bar{d}$

12 K24 9

27P1 $\bar{z}$

IP4 $\bar{e}$

K27

10 K25 20

IP6N

IP4P

27P2A

B

26TB3-7

26TB3-8

27P2B

Ay

11P1A

11P1C

11P1B

26TB2-6

R-2TB21-6

23P1B

23P1C

23P1A

41IOP1 $\bar{r}$

RIGHT PITCH DIFF SERVO SW

2P1N EXTEND

2P1M RETRACT

21P1D

6P1W

K25

9 9

IP7 $\bar{r}$

26TB5-6

18P1L

YAW NORM SOL

18P1H

26TB5-7

21P1L

RIGHT PITCH DIFF SOL

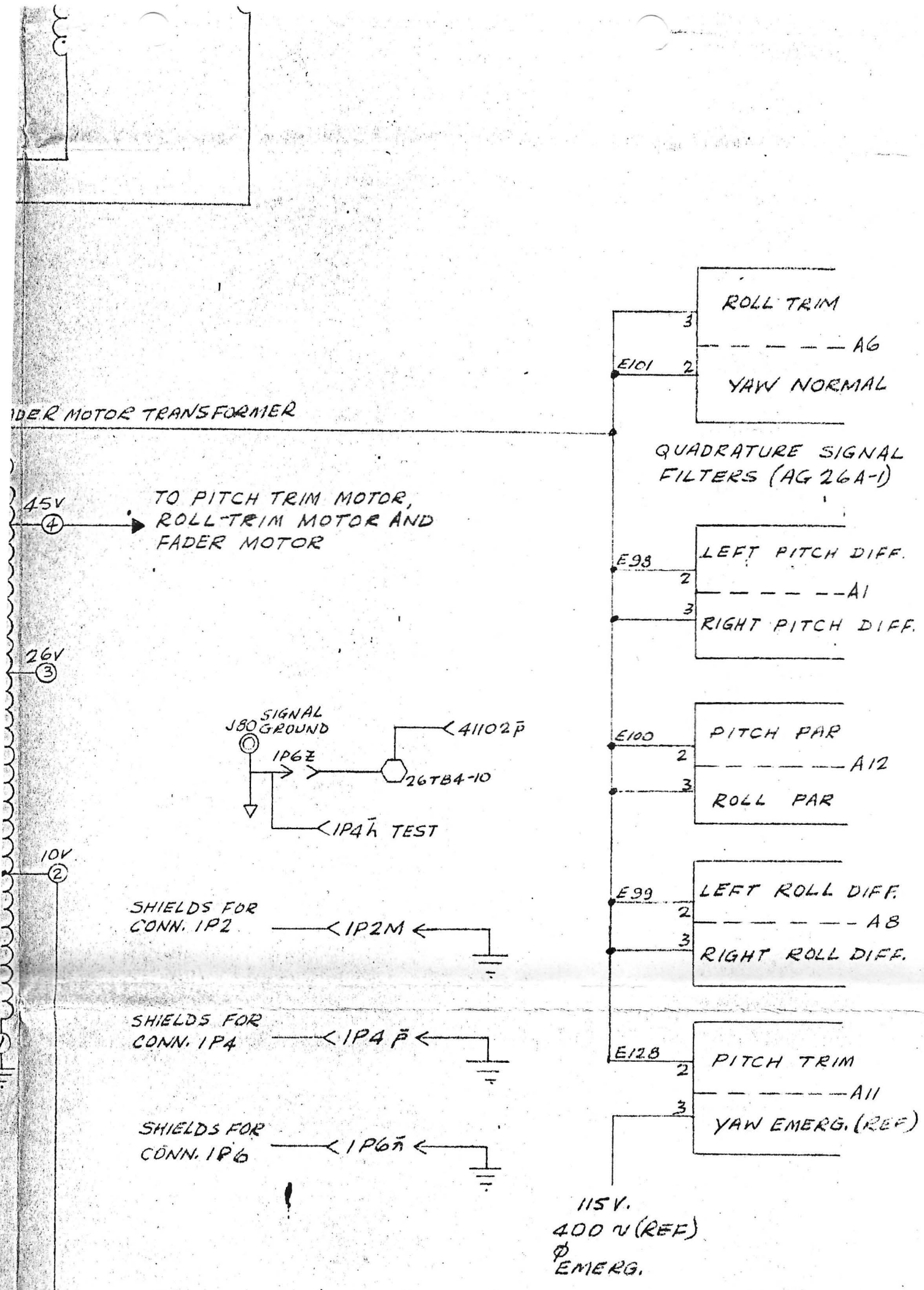
21P1H

20P1L

LEFT

20P1H

IP7 $\bar{d}$



CSK 1781

TOLERANCE UNLESS NOTED					
ONE PLACE	(.0)	±.020	DRAWN BY	T. MASON	10-20-5
TWO PLACE	(.00)	±.010	DRAFTING √		
THREE PLACE	(.000)	±.005	MATL. & FIN. √		
90° FORMED ANGLES		±2° −1°	DESIGNER √		
OTHER ANGLES					
DEV.			DESIGN √		
SUPSD.			APPROVED		
GOVT. MATL. SPEC.			REF. MH. MATL. (SPEC.)		
			FINISH (SEE NOTE)		

NORMAL AC POWER CIRCUIT
SCHEMATIC DIAGRAM
DEVELOPMENT SYSTEM
MH-64 DAMPER (YG338-1)

SCALE NONE WT. ACT. CALC. LB.

HONEYWELL CONTROLS LIMITED
AERONAUTICAL DIVISION
TORONTO, CANADA

CSK 1781

2-10 V

SUPPLEMENTARY INFORMATION			FIRST CLASS.
FIRST USED			
NEXT ASSEMBLY		REFERENCE	
		SK6129	
		REV B.	

REVISIONS			
SYM.	DESCRIPTION	DATE	APPROVAL

PITCH RATE GYRO

ROLL RATE GYRO

ER MOTOR TRANSFORMER

45V ④ → TO PITCH TRIM MOTOR, ROLL-TRIM MOTOR AND FADER MOTOR

26V ③

SIGNAL GROUND J80 — IP6Z — 26TB4-10 — 41102P

IP4H TEST

10V ②

SHIELDS FOR CONN. IP2 — IP2M

SHIELDS FOR CONN. IP4 — IP4P

SHIELDS FOR CONN. IP6 — IP6N

E101
- 3 — ROLL TRIM
- ---- A6
- 2 — YAW NORMAL

QUADRATURE SIGNAL FILTERS (AG 26A-1)

E98
- 2 — LEFT PITCH DIFF.
- ---- A1
- 3 — RIGHT PITCH DIFF.

E100
- 2 — PITCH PAR
- ---- A12
- 3 — ROLL PAR

E99
- 2 — LEFT ROLL DIFF.
- ---- A8
- 3 — RIGHT ROLL DIFF.

E128
- 2 — PITCH TRIM
- ---- A11
- 3 — YAW EMERG. (REF)

115V. 400 ~ (REF) Ø EMERG.

CSK 1781

C11 .47

K14

R164 7500

NORMAL YAW RATE GYRO

2P1M

2P3M

2P4M

1P4 g

TEST

1P4 n

TEST

E104

T15 TRIM & FADER M

T14 PITCH AXIS BRIDGE TRANSFORMER

3V

CT

TO PITCH TRIM CRT.

3V

1.2V

CT

TO LOW SPEED PITCH TRIM

1.2V

A21

PITCH INT. V.G. PRI

C36 0.22

A21

PITCH INT. MOTOR

A25

PITCH V.G. HI-PASS PRI

C48 0.22

A25

PITCH HI-PASS MOTOR

K8

K27

K26

45V

26V

10V

45V E73

1P7T

10P1B

NORM. ACCEL. PRI.

10P1A

26TB2-1

R55 680

R56 820

A21

NORM. ACCEL. INT. RESOLVER PRI.

26V

E131

A25

PITCH HI-PASS RES. PRI.

1P7L

19P1H

19P1J

26TB2-8

20P1A

20P1B

LEFT ELEV. SERVO BAL. PRI.

26TB2-6

21P1A

21P1B

RIGHT ELEV. SERVO BAL. PRI.

10V 1P7U

19P1F

19P1E

26TB3-5

PARALLEL SERVO BAL. PRI.

7V

2P4H

2P4G

26TB2-3

PITCH RATE GYRO PICKOFF PRI.

2P4K

1P7V (SPARE)

26TB3-2

1P3E

E103

1P7Z (SPARE

3n

4110P3N

27P1R

R-2TB23-10

CONNECT IF NO AFCS

4110P2b

27P1f

R-2TB23-8

A23

TRIM COMM. V.G. PRI.

A23

L TRIM COMM. T. MOTOR

27P1S

26TB2-10

15P1J

TRIM PISTON NSD PRI.

26TB2-7

A23

1M. PRI.

16P1B

ERVO BAL. PRI

26TB2-5

17P1B

SERVO BAL. PRI.

15P1E

SERVO BAL. PRI. LERON

2P3G

PICKOFF PRI.

26TB2-3

2P3K

DIE 10PIG

1P6m̄ (REF)

26TB2-1

1P6C (REF)

1P6D

1P6R

1P6f̄

J39

E111

E112

E113

C9 .4

C10 .47

C11 .47

R162 5100

R163 7500

R164 7500

K12

K13

K14

TG NORMAL GYRO SPIN TRANSFORMER

B1 E1 B2 E2

1P6ē

26TB1-9

1P6V

26TB1-10

2P4E

2P3E

2P1E

2P4F

2P3F

2P1F

NORMAL YAW RATE GYRO

2P1M

2P3M

2P4M

1P4ḡ

TEST

1P4h̄

TEST

10P1E

10P1G

E104

T15 TRIM & FADER

T14 PITCH AXIS ERIDGE TRANSFORMER

3V

CT

3V

1.2V

CT

1.2V

TO PITCH TRIM CRT.

TO LOW SPEED PITCH TRIM

A21 PITCH INT. V.G. PRI.

C36 0.22

A21 PITCH INT. MOTOR

A25 PITCH V.G. HI-PASS PRI.

C48 0.22

A25 PITCH HI-PASS MOTOR

K8

K27

K26

45V

26V

E73

1P7T 10P1B NORM. ACCEL. PRI.

R55 680

R56 820

10P1A

A21 NORM. ACCEL. INT. RESOLVER PRI.

26TB2-1

26V

E131

A25 PITCH HI-PASS RES. PRI.

1P7L 19P1H 19P1J

26TB2-8

20P1A 20P1B LEFT ELEV. SERVO BAL. PRI.

26TB2-6

21P1A 21P1B RIGHT ELEV. SERVO BAL. PRI.

10V 1P7U 19P1F 19P1E

26TB3-5

PARALLEL SERVO BAL. PRI.

7V

2P4H

2P4G

26TB2-3

2P3m̄ 4110P3N 27P1R R-2TB23-10

CONNECT IF NO AFCS

2P2ē 4110P2b̄ 27P1f̄ R-2TB23-8

A23 ROLL TRIM COMM. V.G. PRI.

A23 ROLL TRIM COMM. INT. MOTOR

27P1S

26TB2-10

15P1J ROLL TRIM PISTON ANSD PRI.

26TB2-7

A23 COMM. R PRI.

16P1B SERVO BAL. PRI

26TB2-5

17P1B SERVO BAL. PRI.

15P1E SERVO BAL. PRI. AILERON

2P3G

K2F

400∿, 3 Ø, 4 WIRE WYE, 115V L-N
(FROM PRIMARY AC BUS)

ROM VARIABLE 28V. 400∿
ROM PRIMARY SOURCE

27P2E

FUNCTION SELECTOR
EDGE LIGHTS

14P1F

14P1F

9P1B — RR INERTIA G'SN. HTR. — 9P1A

2P3J — ROLL RATE GYRO HTR — 2P3N

2P4J — PITCH RATE GYRO HTR — 2P4N

2P2J — EMERG. YAW RG. HTR. — 2P2N

8P1B — EMERG. LAT. ACCEL HTR — 8P1A

2P1J — YAW RATE GYRO HTR — 2P1N — 8P1J

26TB1-6

7P1B — LAT'L. ACCEL. HTR. — 7P1A

13P1B — LAT'L. SCHED. ACCEL HTR. — 13P1A — 7P1J

T13 NORMAL YAW AXIS
BRIDGE TRANSFORMER

A10 — RIGHT AIL. AMP
A9 — LEFT AIL. AMP
A5 — RIGHT ELEV. AMP
A14 — AIL. PAR SERVO AMP
A2 — LEFT ELEV. AMP
A3 — ELEV. PAR SERVO AMP
A4 — NORM. RUD. SERVO AMP
A15 — ROLL TRIM AMP
A26 — NORM Y.R. H-P AMP
A22 — ROLL COMM. INT. AMP
A24 — PITCH H-P AMP
A19 — NORM g.c SCHED. AMP
A16 — PITCH RATE AMP
A17 — PITCH COMM. BRIDGE AMP
A7 — PITCH TRIM AMP
A18 — ROLL BRIDGE AMP
A20 — PITCH COMM. INT. AMP

12 3V
11 CT
10 3V
9 1.2V — TO NORM. g.c CHAR. POT
8 CT
7 1.2V

A36AA — A36 — A36EE
NORM g.c CHAR VG PRI

C57 0.22 — A36-CC — A36 — A36-Y
NORM g.c CHAR MOTOR

18 A27 19
NORM. YAW RATE HI-PASS V.G. PRI

E126 — C65 0.22 — 9 A27 10
NORM. YAW RATE HI-PASS MOTOR

45V 5

1P7W — 7P1G — 7P1H — 26TB2-1
LATERAL ACCEL. PRI

26V 4 — 21 A23 23 — 13
NORMAL Y.R. HI-PASS RES PRI.

18P1H — 18P1A — 26TB2-4
NORM. RUDDER SERVO BAL. PRI.

10V 3 — 1P7X — 27P2F — 27P2G — 26TB2-10
26TB2-6 AIL. POST TRANSD. PRI.

7V 2 — 2P1H — 2P1G — 26TB2-2
Y.R. GYRO PICKOFF PRI.

2P1K

A35-1
A35-4 — A35-5
PG 29 A35

T12 ROLL AXIS BRI… TRANSFORMER

12 3V
11 CT
10 3V
9 1.2V — TO ROLL TRIMPOT
8 CT
7 1.2V — R-TB23-3

45V 5
R93 680
R94 820
1P3F
E132

26V 4

10V 3 — 1P3G — 26TB3-1

7V 2

GRD.

A — 27P2V — 14P1M — (2)(3)(1) — 14P1F

B — 27P2X — 14P1J — (6)(5)(7) — 14P1J

C — 27P2Y — 14P1G — (9)(10)(8) — 14P1B

K2F

400∿, 3 ϕ, 4 WIRE WYE, 115V L-N
(FROM PRIMARY AC BUS)

R-2TB21-1 — 4110 P1B TO A FCS

26TB1-1

26TB1-2 — 6P2P — 6P2G

26TB1-3

26TB1-4

26TB1-5

26TB1-6

9P1B — RR INERTIA G'SN. HTR. — 9P1A

2P3J — ROLL RATE GYRO HTR — 2P3N

2P4J — PITCH RATE GYRO HTR. — 2P4N

2P2J — EMERG. YAW RG. HTR. — 2P2N

8P1B — EMERG. LAT. ACCEL HTR — 8P1A

2P1J — YAW RATE GYRO HTR — 2P1N

8P1I

7P1B — LAT'L. ACCEL. HTR. — 7P1A

13P1B — LAT'L. SCHED. ACCEL HTR — 13P

7P1I

?OM VARIABLE 28V. 400∿
?OM PRIMARY SOURCE

27P2C — 14P1H — FUNCTION SELECTOR EDGE LIGHTS — 14P1H

T13 NORMAL YAW AXIS BRIDGE TRANSFORMER

A10 — RIGHT AIL. AMP

A9 — LEFT AIL. AMP

A5 — RIGHT ELEV. AMP

A14 — AIL. PAR SERVO AMP

A2 — LEFT ELEV. AMP

A3 — ELEV. PAR SERVO AMP

A4 — NORM. RUD. SERVO AMP

A15 — ROLL TRIM AMP

A26 — NORM Y.R. H-P AMP

A22 — ROLL COMP. INT. AMP

A24 — PITCH H-P AMP

A19

(6) — (12) 3V — (11) CT — (10) 3V — (9) 1.2V — TO NORM. q/c CHAR. POT — (8) CT — (7) 1.2V

A36AA — A36 — A36EE — NORM q/c CHAR VG PRI

C57 0.22 — A36CC — A36 — A36-Y — NORM q/c CHAR MOTOR

A27 — (18) — (19) — NORM. YAW RATE HI-PASS V.G. PRI

E126 — C65 0.22 — (9) A27 (10) — NORM. YAW RATE HI-PASS MOTOR

45V — (5)

1P7W — 7P1G — 7P1H — 26TB2-1 — LATERAL ACCEL. PRI

26V — (4) — (21) A23 (22) (13) — NORMAL Y.R. HI-PASS RES PRI.

18P1H — 18P1A — 26TB2-4 — NORM. RUDDER SERVO BAL. PRI.

10V — (3) — 1P7X — 27P2F — 27P2G — 26TB2-10 — 26TB3-6 AIL. POST TRANSD. PRI.

7V — (2)

2P1H — 2P1G — 26TB2-2 — Y.R. GYRO PICKOFF PRI.

(1)

2P1K

T12 ROLL AXIS BRI… TRANSFORMER

(6) — (12) 3V — (11) CT — (10) 3V — (9) 1.2V — TO ROLL TRIMPOT — (8) CT — (7) 1.2V — R-TB23-3

45V — (5)

R93 680 — R94 820 — 1P3F — E132

26V — (4)

10V — (3) — 1P3G — 26TB3-1

(1)

1P

9P1B
9P1A
2P3J
2P3N
ROLL RATE GYRO HTR.
2P4J
2P4N
PITCH RATE GYRO HTR.
2P2J
2P2N
EMERG. YAW RS. HTR.
8P1B
8P1A
8P1J
EMERG. LAT. ACCEL HTR
2P1J
2P1N
YAW RATE GYRO HTR.
26TB1-6
7P1B.
7P1A
7P1J
LAT'L. ACCEL. HTR.
13P1B
13P1A
13P1T
LAT'L. SCHED. ACCEL HTR.
10P1F
10P1E
10P1G
NORM ACCEL. HTR.
1P4 g̅
TEST
1P4 n̅
TEST
C11 .47
R164 7500
K14

T12 ROLL AXIS BRIDGE TRANSFORMER
6
12 3V
11 CT
10 3V
9 1.2V
8 CT
7 1.2V
TO ROLL TRIMPOT
R-TB23-9
4110P3m̅
4110P3N
27P1R
R-2TB23-10
CONNECT IF NO AFCS
4110P2e̅
4110P2b̅
27P1f̅
R-2TB23-8

T14 PITCH TRANSF
6
12 3V
11 CT
10 3V
9 1.2V
8 CT
7 1.2V
45V
E73
R55 680
R56 820
26V
E131
PIT
RE
1P7L
10V
1P7U
26TB3
7V
2

A36AA
A36
A36EE
NORM g.c CHAR VG PRI
C57 0.27
A36-cc
A36
A36-f̅
NORM g.c CHAR MOTOR
A27
18
19
NORM. YAW RATE HI-PASS V.G. PRI
C65 0.22
26
9
10
A27
NORM. YAW RATE HI-PASS MOTOR

1P7E
A23
18
19
ROLL TRIM COMM. V.G. PRI.
C42 0.22
E84
9
10
A23
ROLL TRIM COMM. INT. MOTOR
27P1S
45V
5
R93 680
R94 820
E132
1P3F
15P1H
15P1J
ROLL TRIM PISTON TRANSD PRI.
26TB2-7
26TB2-10
26V
4
21
A23
22
13
ROLL COMM. RESOLVER PRI.
10V
3
1P3G
26TB3-1
16P1A
16P1B
LEFT AIL. SERVO BAL. PRI
26TB2-5
17P1A
17P1B
RIGHT AIL. SERVO BAL. PRI.
15P1F
15P1E
PARALLEL SERVO BAL. PRI. AILERON
1
7V
2
2P3H
2P3G
26TB2-3
R.R. GYRO PICKOFF PRI.
2P3K

7P1H
26TB2-1
AL ACCEL. PRI
13
18P1A
26TB2-4
1. RUDDER SERVO BAL. PRI.
27P2G
26TB2-10
L. POST TRANSD. PRI.
2P1G
26TB2-2
Y.R. GYRO PICKOFF PRI.
2P1K
A35-1
A35-5
PG 29
A35

1P6m̄ (REF)

26TB2-1 1P6C (REF)

R-2TB21-1
4110 P1B TO AFCS

6TB1-1

6P2P 6P2G

26TB1-3

26TB1-4

26TB1-5

26TB1-6

2P3J 2P4J 2P2J 8P1B 2P1J 7P1B 13P1B 10P1F

ROLL RATE GYRO HTR. PITCH RATE GYRO HTR. EMERG. YAW RS. HTR. EMERG. LAT. ACCEL HTR YAW RATE GYRO HTR LAT'L. ACCEL. HTR. LAT'L. SCHED. ACCEL HTR NORM ACCEL. HTR.

2P3N 2P4N 2P2N 8P1A 2P1N 7P1A 13P1A 10P1E 10P1G

8P1I 7P1I 13P1I

J39

1P6D E111 C9 .4 R162 5100 K12

1P6R E112 C10 .47 R163 7500 K13

1P6f̄ E113 C11 .47 R164 7500 K14

1P4 q̄ TEST

1P4 m̄ TEST

T12 ROLL AXIS BRIDGE TRANSFORMER

T14 PITCH AX TRANSFORI

A36AA A36 A36EE
NORM g.c CHAR VG PRI

A36CC A36 A36-Y
NORM gc CHAR MOTOR

A27
NORM. YAW RATE HI-PASS V.G. PRI

A27
NORM. YAW RATE HI-PASS MOTOR

7P1H 26TB2-1
ACCEL. PRI

18P1A 26TB2-4
DDER SERVO BAL. PRI.

27P2G 26TB2-10
OST TRANSD. PRI.

2P1G 26TB2-2
GYRO PICKOFF PRI.

2 P1K

12 3V
11 CT
10 3V
9 1.2V
8 CT
7 1.2V
TO ROLL TRIMPOT

4110P3m̄ 4110P3N 27P1R R-2TB23-10
CONNECT IF NO AFCS
4110P2ē 4110P2b̄ 27P1f̄ R-2TB23-8
R-TB23-9

1P7J

A23
ROLL TRIM COMM. V.G. PRI.

C42 .22 E84 A23
ROLL TRIM COMM. INT. MOTOR

45V 5
R93 680 R94 820 E132

27P1S
26TB2-10

1P3F 15P1H 15P1J 26TB2-7
ROLL TRIM PISTON TRANSD PRI.

26V 4
A23
ROLL COMM. RESOLVER PRI.

10V 3 1P3G 26TB3-1
16P1A 16P1B 26TB2-5
LEFT AIL. SERVO BAL. PRI

17P1A 17P1B
RIGHT AIL. SERVO BAL. PRI.

15P1F 15P1E
PARALLEL SERVO BAL. PRI. AILERON

45V 5 E73
R55 680 R56 820
26V 4 E131
PITCH RES.
1P7L
10V 3 1P7U 26TB3-5

PITCH DISENGAGE LIMITER

Provision is made within the YG339B-1 Damper Systems for an acceleration limiting system. The acceleration or 'g' limiter consists of two force rebalance accelerometers, a pitch differential servo ram position pot, a pitch parallel servo ram position pot and an amplifier-calibrator.

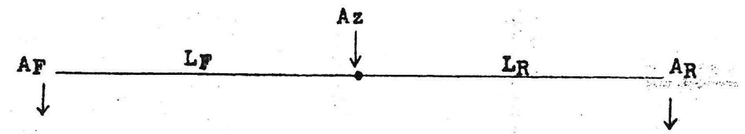
Pitch accelerations are sensed by the accelerometers, whose signals control 'g' disengage relays through a magnetic amplifier circuit. The disengage relay contacts are part of the damper pitch axis engage circuits and will disengage the pitch parallel servo or the differential and parallel servos when excessive 'g' forces are present.

The disengage function is represented by:

$$L = (Az + 162\dot{q})\frac{1 + .02S}{1 + .1S} + 12.5\delta D\,\frac{2S}{1 + 2S} + 15.75\delta p\,\frac{.5S}{1 + .5S}$$

where Az = normal acceleration at centre of gravity
$\dot{q}$ = pitch angular acceleration
δD = differential servo ram position
δp = parallel servo ram position

Pitch angular acceleration is not measured directly as angular acceleration but indirectly as linear acceleration at two separate points in the aircraft. One linear accelerometer is mounted forward of the centre of gravity and the other aft of the centre of gravity. The following sketch then applies:



The term L_F and L_R refer to the distances from the forward and aft accelerometers to the centre of gravity. A_F and A_R are linear accelerations as measured fore and aft respectively. A_z is normal acceleration at the centre of gravity. The following equations may now be written.

(1) $A_F = A_z + L_F\dot{q}$

(2) $A_R = A_z - L_R\dot{q}$

These equations are true because the fore and aft accelerometers will measure the algebraic sum of Az, the acceleration of the aircraft at its centre of gravity and of the angular acceleration of the aircraft about its centre of gravity, which at the fore and aft accelerometer, becomes a linear acceleration equal to angular acceleration multiplied by the distance of the centre of gravity to the fore and aft accelerometer.

Now the second equation is multiplied by a constant C giving:

(3) $C_{AR} = C_{Az} - C_{LR}\dot{q}$

Subtract equation (3) from equation (1):

(4) $A_F - C_{AR} = A_z - C_{Az} + L_F\dot{q} + C_{LR}\dot{q}$

or (5) $A_F - C_{AR} = A_z(1-C) + \dot{q}(L_F + C_{LR})$

Dividing through by 1-C:

(6) $$\frac{A_F - C_{AR}}{1 - C} = A_z + \left(\frac{L_F + C_{LR}}{1 - C}\right)\dot{q}$$

The right hand side of this equation can be made equal to the first term of the disengage function by selecting the proper value of C and adding a lag network whose transfer function is $\frac{1 + .02S}{1 + .1S}$.

Electrically, all of the above is accomplished by summing the two accelerometer signals using different size summing resistors. The difference in the summing resistors gives the constant C and the combination of the two signals at the summing point is equivalent to the subtraction of equation (3) from equation (1). The necessary lag networks are connected in series with the accelerometer signals and the result is an electrical circuit which exactly mechanizes the first term of the disengage function.

The second and third terms are high passed servo position signals which can be electrically supplied by servo position pots and high pass networks. These terms are needed because they give signals indicative of 'g' forces which the aircraft will feel a moment later due to damper action or commands from the stick. In other words, these two terms provide signals which anticipate 'g' force changes which will shortly follow movement of the elevator control surfaces. This permits the 'g' limiter system to cause disengagement before excessive 'g' forces are actually felt by the aircraft. If no such provisions were made it would be necessary to adjust the acceleration limits to cause disengagement at a 'g' level well below the aircraft 'g' limits to prevent 'g' forces from overshooting and possibly damaging the aircraft before disengagement can occur.

It should be remembered that the 'g' limiter will not disengage the pitch differential servos if they are driving in a direction to oppose the excessive 'g' forces. (Refer to section on DC engaging sequence) The pitch parallel servo will disengage regardless of its direction of travel.

The net electrical signal from the three terms of the limit function is passed through a pre-amplifier to the control windings of two magnetic amplifiers. One magnetic amplifier controls relay K2G and functions in the event of excess negative 'g' forces. The other magnetic amplifier controls relay K1G and functions in the event of excessive positive 'g' forces. These relays are normally energized and will de-energize when 'g' limits are exceeded. Contact of these

relays will open circuits in the engage paths to the differential and parallel servos when either relay de-energizes. Since the relays are normally energized, a power failure in the limiter will disengage the pitch axis as an additional safety factor.

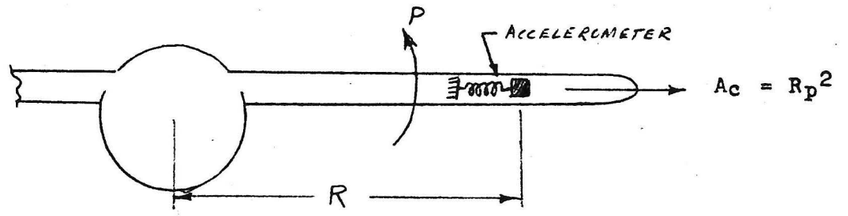
The acceleration limiting system contains a second identical but independent system to that described above. The second system controls relays K3G and K4G whose contacts are connected in series with those of K1G and K2G. In this way, failure of one system to operate will not jeopardize the safety of the aircraft, as the other system will then disengage the pitch axis.

ROLL RATE LIMITER

The YG339B-1 Damper System incorporates a roll rate limiter to protect the rudder from excessive structural loading due to high roll rates. This limiter will disengage the roll differential and parallel servos if the limiting roll rate is exceeded.

To obtain simplicity and ruggedness, rate of roll is not measured directly using a roll rate gyro. An accelerometer, properly mounted, is used to sense roll rate indirectly, and to open a sector switch when the limit has been exceeded.

The accelerometer is mounted well out in the right wing at a distance R from the longitudinal axis of the aircraft.



As shown above, the sensitive axis of the accelerometer is in the plane of the wing and perpendicular to the longitudinal axis of the aircraft.

When the aircraft is rolling, a centrifugal acceleration A_c will be sensed by the accelerometer due to the rolling rate p of the aircraft. Centrifugal acceleration and rolling rate are related as follows:

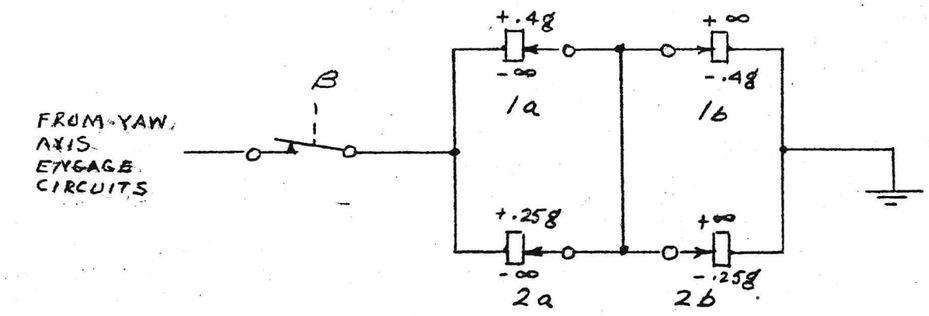
$$A_c = Rp^2$$

Since distance R is constant, the deflection of the accelerometer mass will be proportional to the square of roll rate. The accelerometer is manufactured such that it will open a sector switch at the desired rolling rate, and interrupt power to the roll axis engaging circuits (see section on DC engaging sequence)

RUDDER MONITOR

A rudder monitor system is included in the YG339B-1 damper to switch the system from normal operation to emergency operation if the aircraft exceeds lateral acceleration, rate of change of yaw rate, or crab angle limits. In this condition, both the pitch and roll axes of the damper will disengage, and the yaw axis will switch from the normal system to the duplicate emergency yaw axis. The system consists of two lateral switching accelerometers and a relative wind sensor switch.

The accelerometer switches and the relative wind sensor switch are connected as shown below:



It can be seen from the diagram that if the crab angle β (angle between the longitudinal axis of the aircraft and the direction of the relative wind) exceeds 10 degrees, the β switch will open and remove power from the yaw normal engaging circuits. Similarly, if switches 1a and 2a open or switches 1b and 2b open, power will be removed from the engage circuits.

The β switch is actuated by a relative wind sensing vane mounted on the nose boom of the aircraft. Lateral acceleration Ay and rate of change of yaw rate $\dot{r}$ are sensed by the two accelerometers. Accelerometer number one is mounted 40 feet ahead of the centre of gravity of the aircraft while accelerometer number two is mounted only 13 feet ahead of the centre of gravity. The accelerometers sector switches will open according to the following equations.

Accelerometer No. 1 $\quad .4g = Ay + 40 \frac{\dot{r}}{32.2}$

Accelerometer No. 2 $\quad .25g = Ay + 13 \frac{\dot{r}}{32.2}$

Lateral Acceleratio

LATERAL ACCELERATION SWITCHING

If a lateral acceleration Ay only exists on the aircraft (no $\dot{r}$), accelerometer switch 2a will open when the acceleration exceeds .25g. Since contacts of 1a are still closed, the circuit will not cause disengagement. However, when 1a senses .4g, both switches will be open and the damper system will switch to emergency.
Accelerometer No. 1 then controls switching due to excessive lateral acceleration.

RATE OF CHANGE OF YAW RATE SWITCHING

If a rate of change of yaw rate only (no Ay) is acting on the aircraft, sector switch 1a will open when:

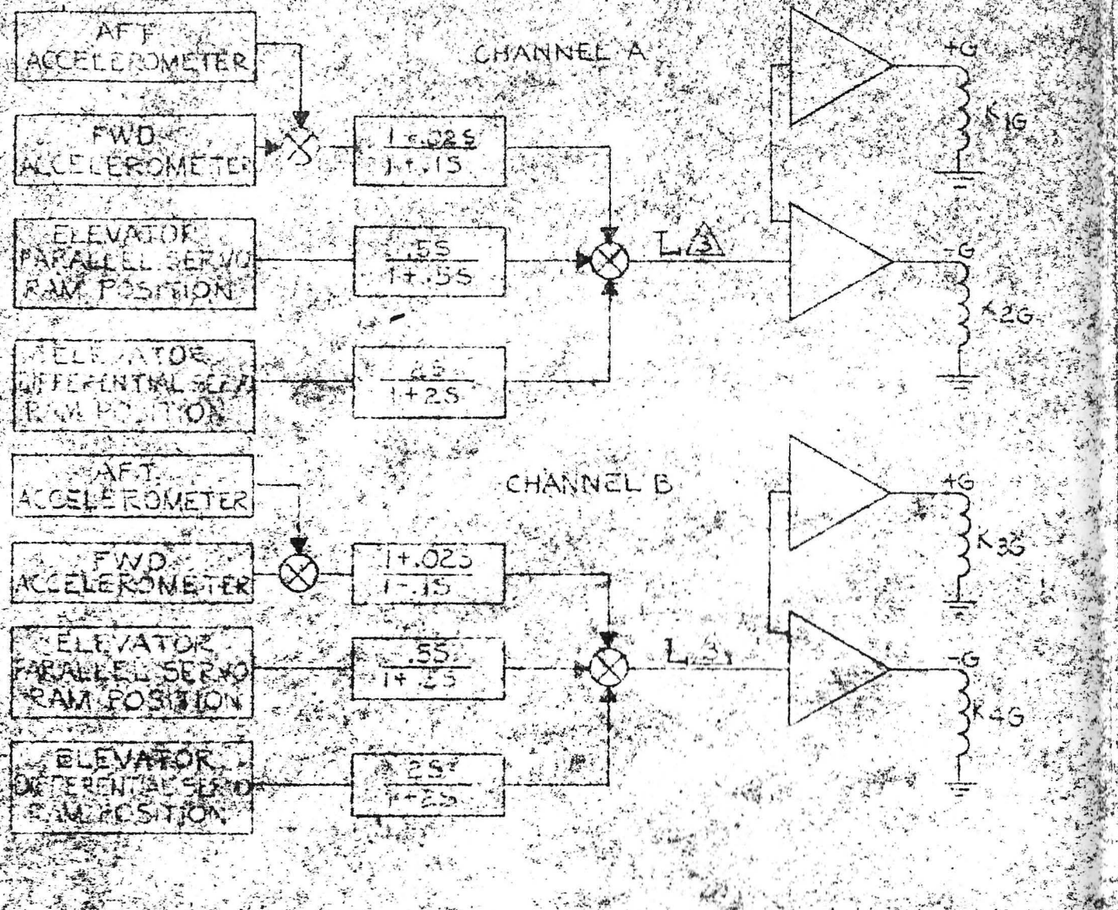
$$\dot{r} = \frac{.4 \times 32.2}{40} = .322 \text{ rad/sec}^2$$

Switch 2a will open when:

$$\dot{r} = \frac{.25 \times 32.2}{13} = .62 \text{ rad/sec}^2$$

Since both switches must be open to effect switching to emergency, accelerometer No. 2 controls switching due to excessive rate of change of yaw rate.

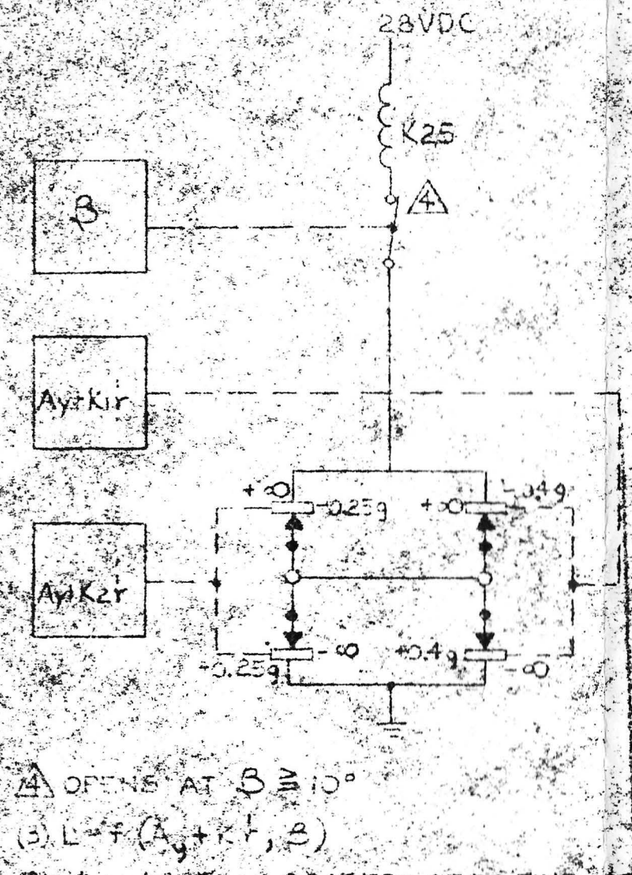
Accelerometer contacts 1b and 2b open when excessive accelerations or rates of change of yaw rate occur in the opposite sense to those described above.



△1 $L=(A_z+16\ddot{\theta}_q)\frac{1+.025s}{1+.15}+12.5\theta_D\frac{2.5}{1+2.5}+15.75\theta_P\frac{.55}{11.55}$

FWD ACCELEROMETER LOCATED 22 FT. FWD OF C.G.
AFT ACCELEROMETER LOCATED 7 FT. AFT OF C.G.

PITCH DISENGAGE ACCELERATION LIMITER



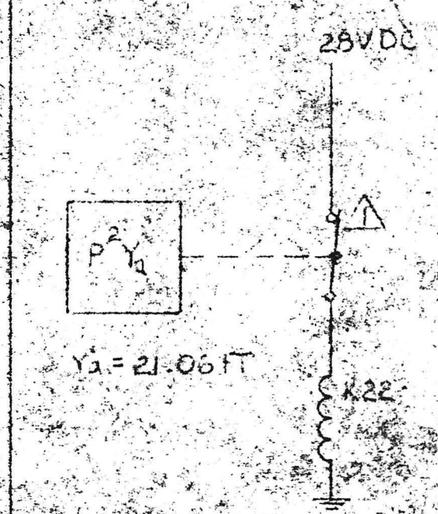
△4 OPENS AT $\beta \geq 10°$

(3) $L=f(A_y+K\dot{r},\beta)$

(2) .4g ACCEL. LOCATED 40 FT FWD OF C.G.

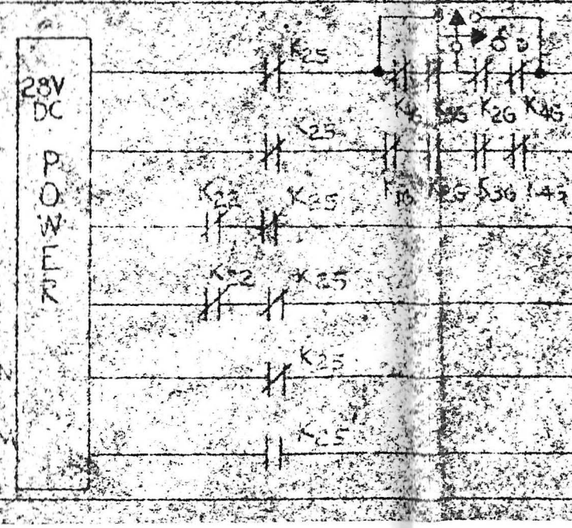
(1) .25g ACCEL. LOCATED 13 FT. FWD OF C.G.

RUDDER MONITOR



△1 OPENS AT p = 180°/SEC

ROLL RATE DISENGAGE LIMITER



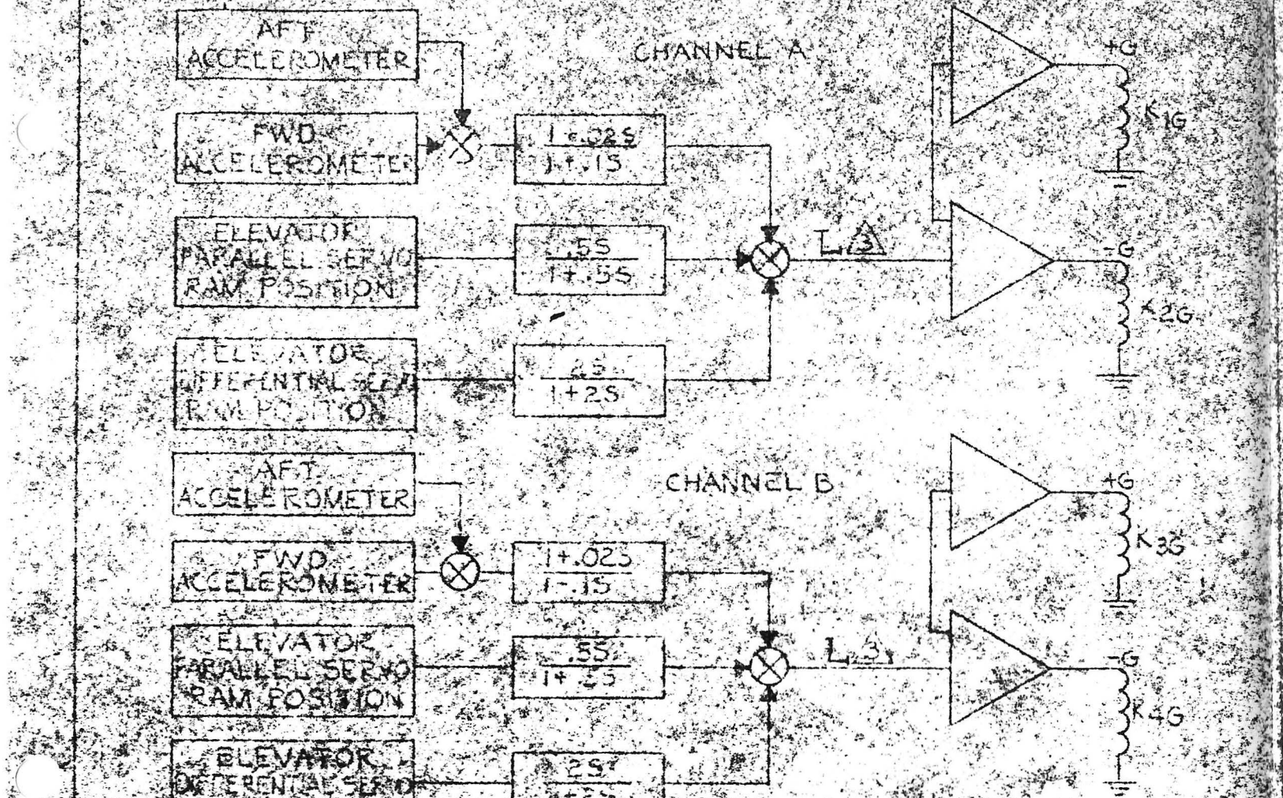
ALL RELAYS SHOWN IN THE ENERGIZED POSITION

ONLY THOSE RELAY CONTACTS REQUIRED TO EXPLAIN FUNCTIONS OF BLOCK DIAGRAM ARE SHOWN

SIMPLIFIED BLOCK DIAGRAM SAFETY MECHANIZATION YG339 DAMPER (NI-64)

5/7/58 ISSUE DATE

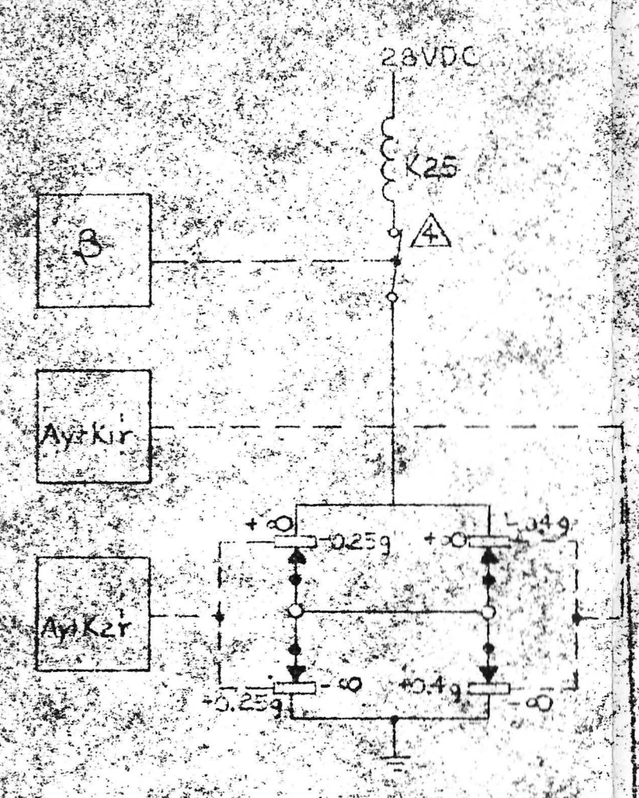
SK 66499



△1 $L = (A_z + 16\ddot{q}) \frac{1+.02S}{1+.1S} + 12.5\delta_D \frac{.25}{1+2S} + 15.7\delta_P \frac{.55}{1+1.5S}$

2 FWD ACCELEROMETER LOCATED 22 FT. FWD OF C.G.
AFT ACCELEROMETER LOCATED 7 FT. AFT OF C.G.

PITCH DISENGAGE ACCELERATION LIMITER



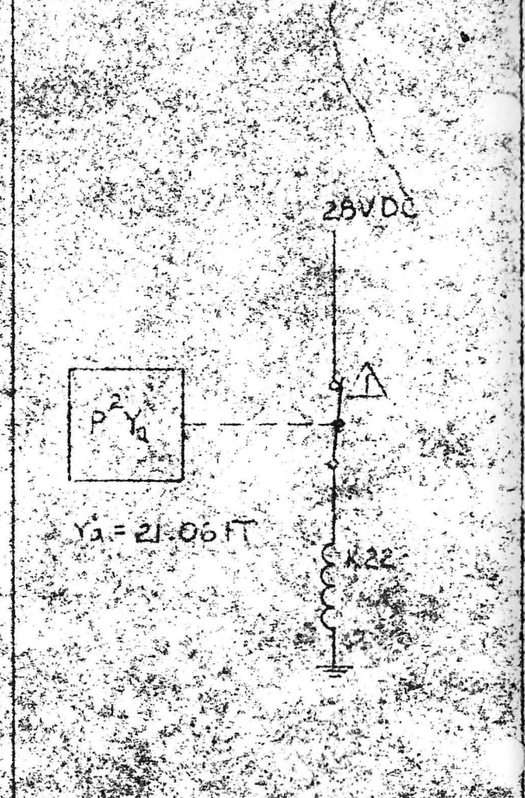
△4 OPENS AT $\beta \geq 10°$

(3) $L = f(A_y + K\dot{r}, \beta)$

(2) .4g ACCEL. LOCATED 40 FT. FWD OF C.G.

(1) .25g ACCEL. LOCATED 13 FT. FWD OF C.G.

RUDDER MONITOR

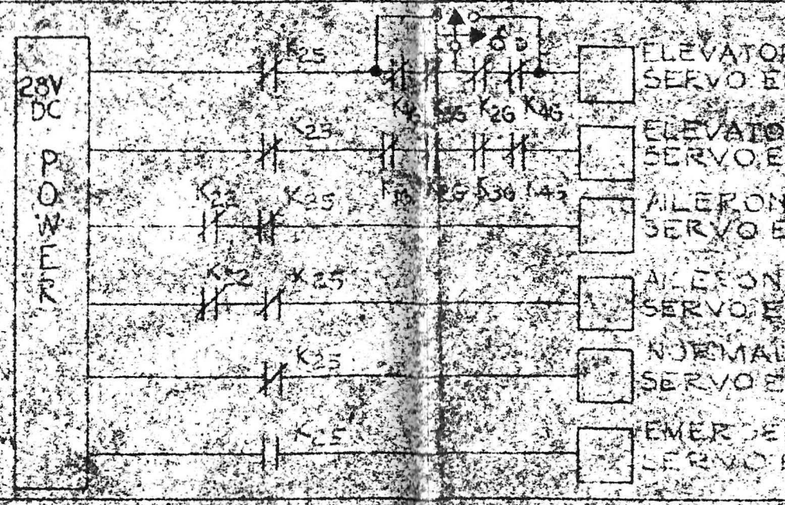


△1 OPENS AT $p = 150°/SEC$

ROLL RATE DISENGAGE LIMITER

2 ALL RELAYS SHOWN IN THE EMERGIZED POSITION

1 ONLY THOSE RELAY CONTACTS REQUIRED TO EXPLAIN FUNCTIONS OF BLOCK DIAGRAM ARE SHOWN



SIMPLIFIED BLOCK DIAGRAM SAFETY MECHANIZATION YG339 DAMPER UNIT

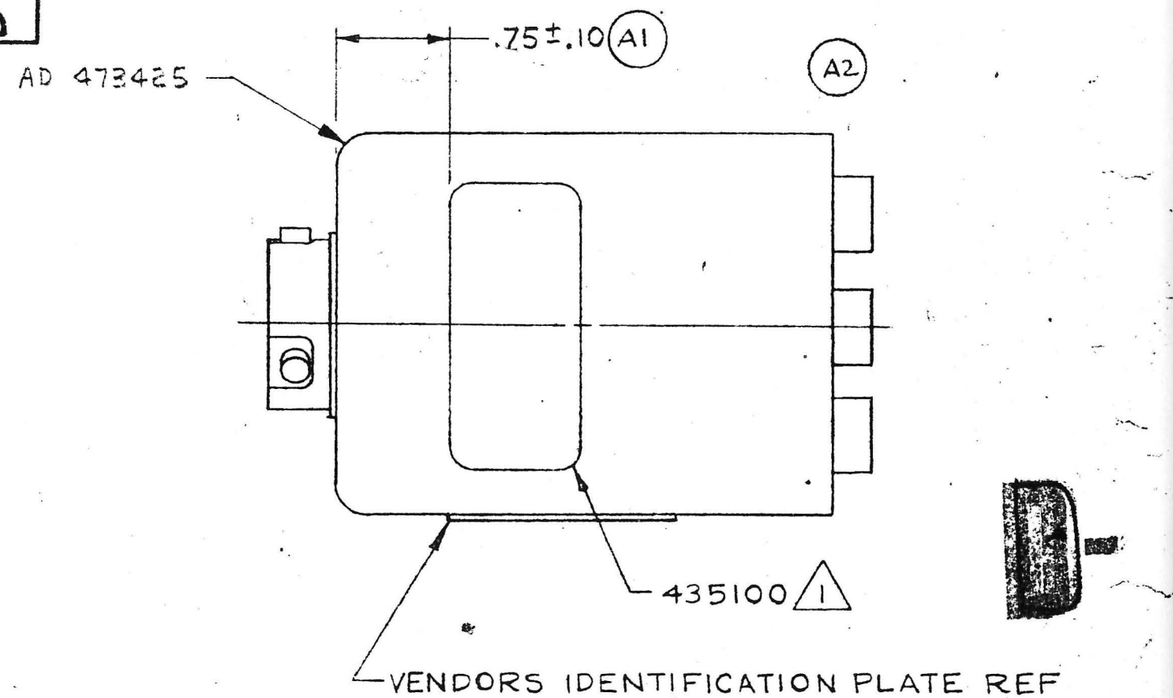
5/7/58 ISSUE DATE

SK 66499

D-GG 75A-1 SHEET 1 OF 2 A

SUPPLEMENTARY INFORMATION		
FIRST USED D-GG 75A-1		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	ADD (1) ITEM 2. 58 TITLE WAS "ACCELEROMETER GR ASSY ACFT" RELEASED VENDOR IDENT PL & 435100	7-29-58	9A-1617



1 APPLY METAL-CAL PER MH SPEC 4010

2	INSTALLATION	D
1	ASSEMBLY	A
SHEET	NAME	SIZE

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		+2° −1°

2021-22-01

SUPSD

GOVT MATL SPEC

DRAWN BY	R.R. Seda	4-14-58
DRAFTING √	[illegible]	4-18-58
MATL & PIN. √		
DESIGNER √	[illegible]	4/14/58
DESIGN √	[illegible]	4-18-58
APPROVED	[illegible]	4-22-58

REF MH MATL (SPEC)

FINISH (SEE NOTE)

ACCELEROMETER ASSEMBLY- G LIMITER

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE FULL SIZE

WT ACT. CALC LB

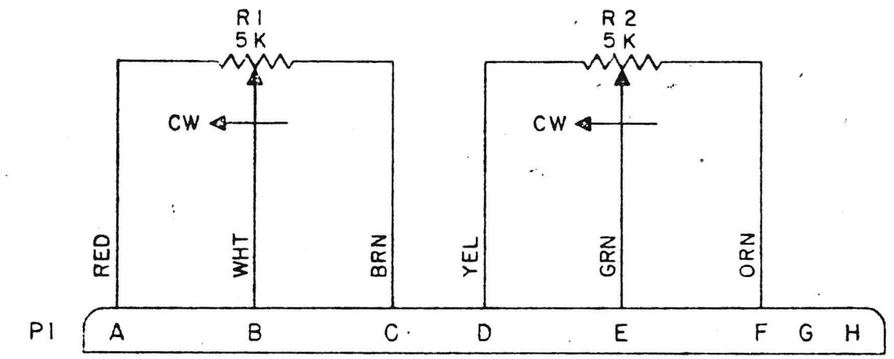
D-GG 75A-1
SHEET 1 OF 2

C 7688

SUPPLEMENTARY INFORMATION		
FIRST USED D-LG16K-1		
REFERENCE		
LG16K-1	S1 THRU	C7688

FIRST CLASS

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL



AN3106A-20-7P REF

1 - CW MOVEMENT OF INPUT SHAFT WHEN FACING INPUT END OF TRANSMITTER

D-

TOLERANCE UNLESS NOTED		
ONE PLACE	(.)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
FORMED ANGLES		±1°
OTHER ANGLES		±1°

01196-30 AD

GOVT MATL SPEC

DRAWN BY	RR Sharp	8-28-7
DRAFTING		
MATL & FIN		9-16-57
DESIGNER		
DESIGN		
APPROVED		

REF MH MATL (SPEC)

FINISH (SEE NOTE)

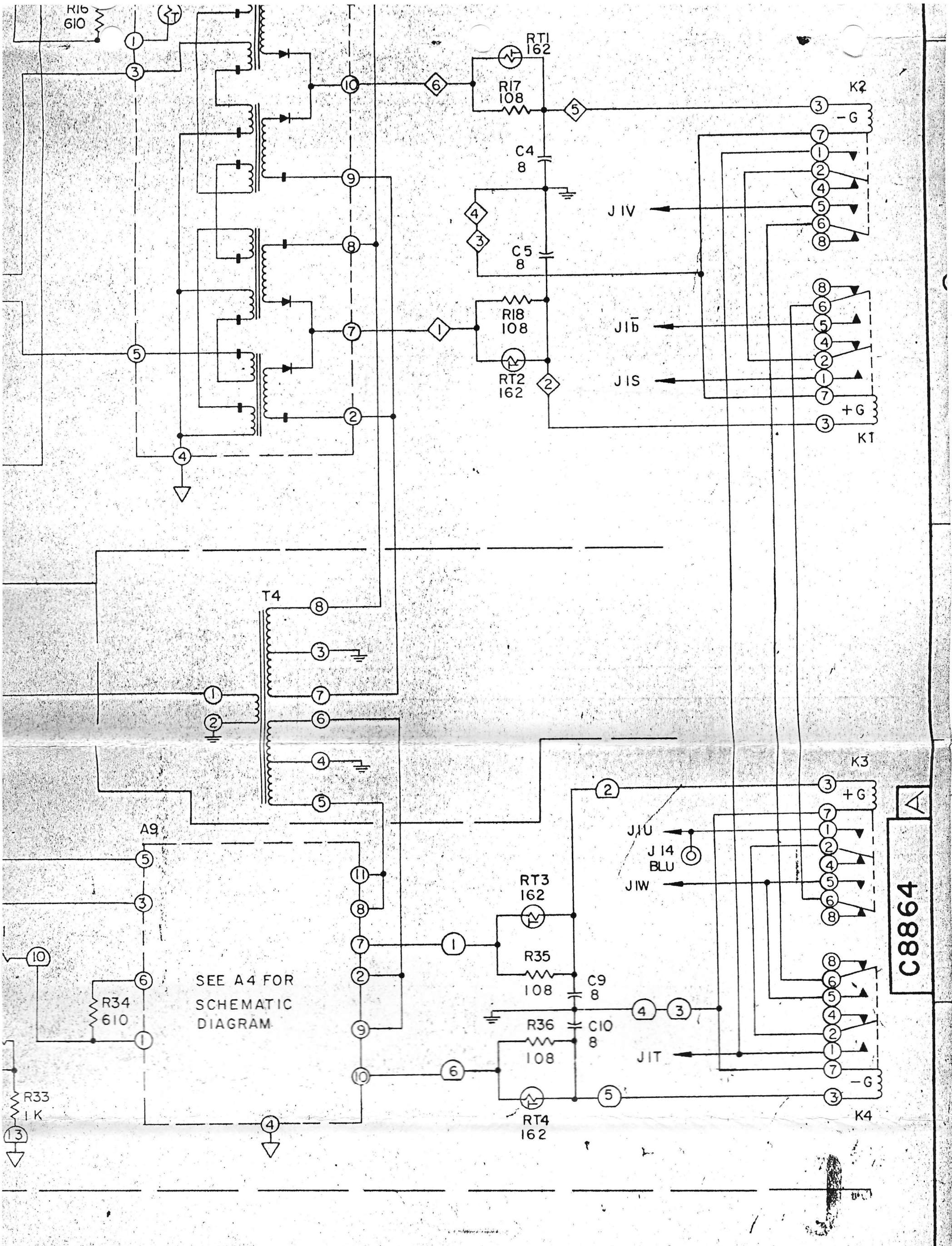
SCHEMATIC DIAGRAM— TRANSMITTER, POSITION

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT (LBS)

C 7688



TOLERANCE UNLESS NOTED ONE PLACE (.0) ±.020 TWO PLACE (.00) ±.010 THREE PLACE (.000) ±.005 90° FORMED ANGLES ±20'	DRAWN BY M. Lyssy 5-6-58 DRAFTING √ C Ryan 5-14-58 MATL & FIN. √ DESIGNER √ 7/29/58	SCHEMATIC DIAGRAM — CALIBRATOR — NORMAL ACCELERATION LIMITER	MINNEAPOLIS-HONEYWELL REGULATOR COMPANY AERONAUTICAL DIVISION MINNEAPOLIS, MINNESOTA
DEV 2021-11-01	DESIGN C Ryan 5-14-58 APPROVED 7-29-58		C8864
SUPSD	REF MH MATL (SPEC)		
GOVT MATL SPEC	FINISH (SEE NOTE)	SCALE NONE WT ACT. CALC LB	

C8864 A

SUPPLEMENTARY INFORMATION		ZONE	REVISIONS			
FIRST USED D-BG128A-1			SYM	DESCRIPTION	DATE	APPROVAL
NEXT ASSEMBLY	REFERENCE		A	ROUTING FOR J1S WAS FROM K2-1; ADD NOTE 11	8-18-58	9A-1660

A4 AD(473230)

R16 610

RT1

R13 1K

1K

1K

RT1 162

R17 108

C4 8

C5 8

R18 108

RT2 162

K2

-G

+G

K1

J1V

J1b

J1S

J1U

J1W

J 14 BLU

K3

R31 4K

R34 610

SEE A4 FOR SCHEMATIC DIAGRAM

RT3 162

R35 108

C9 8

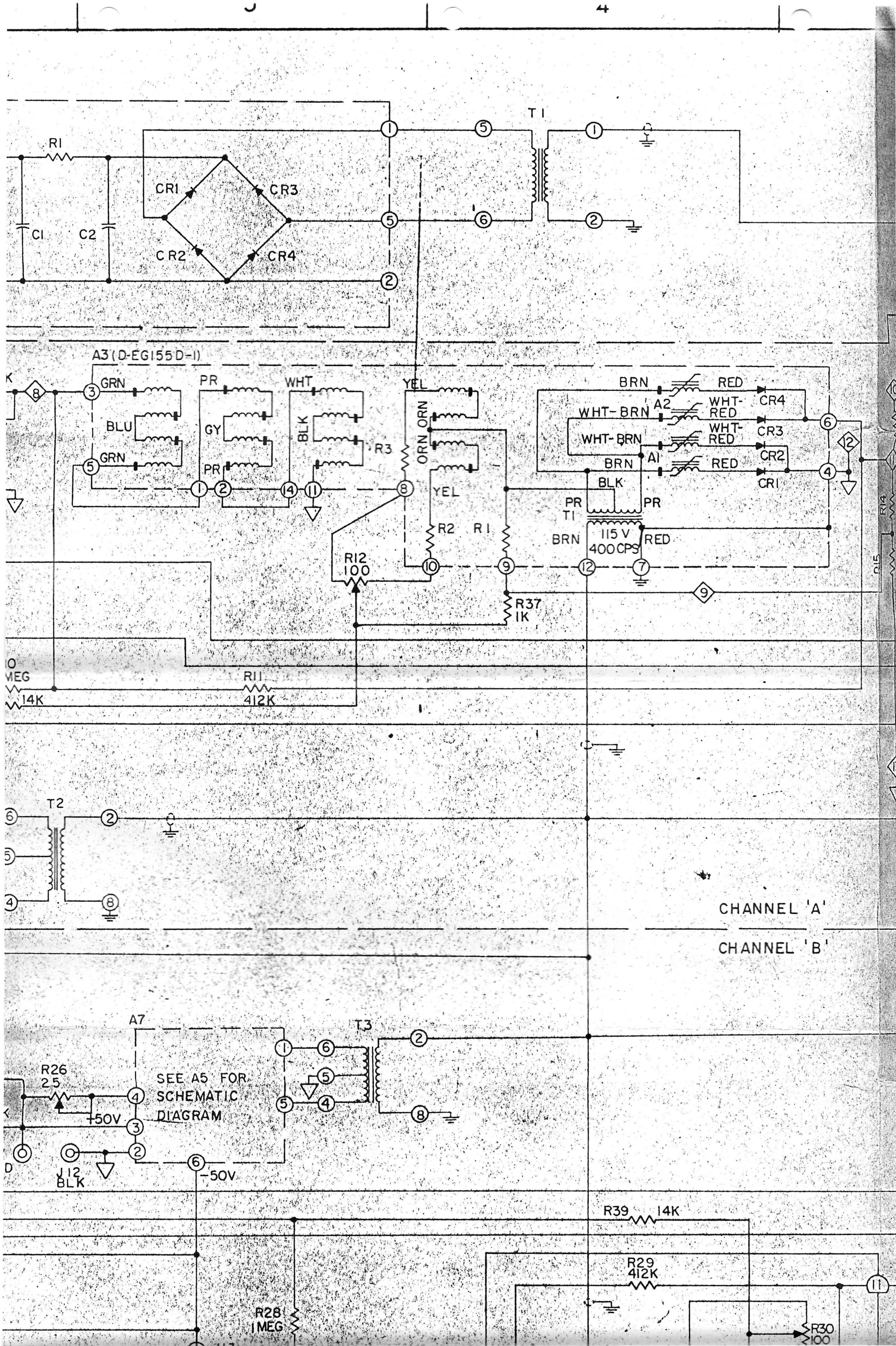
R36

C10

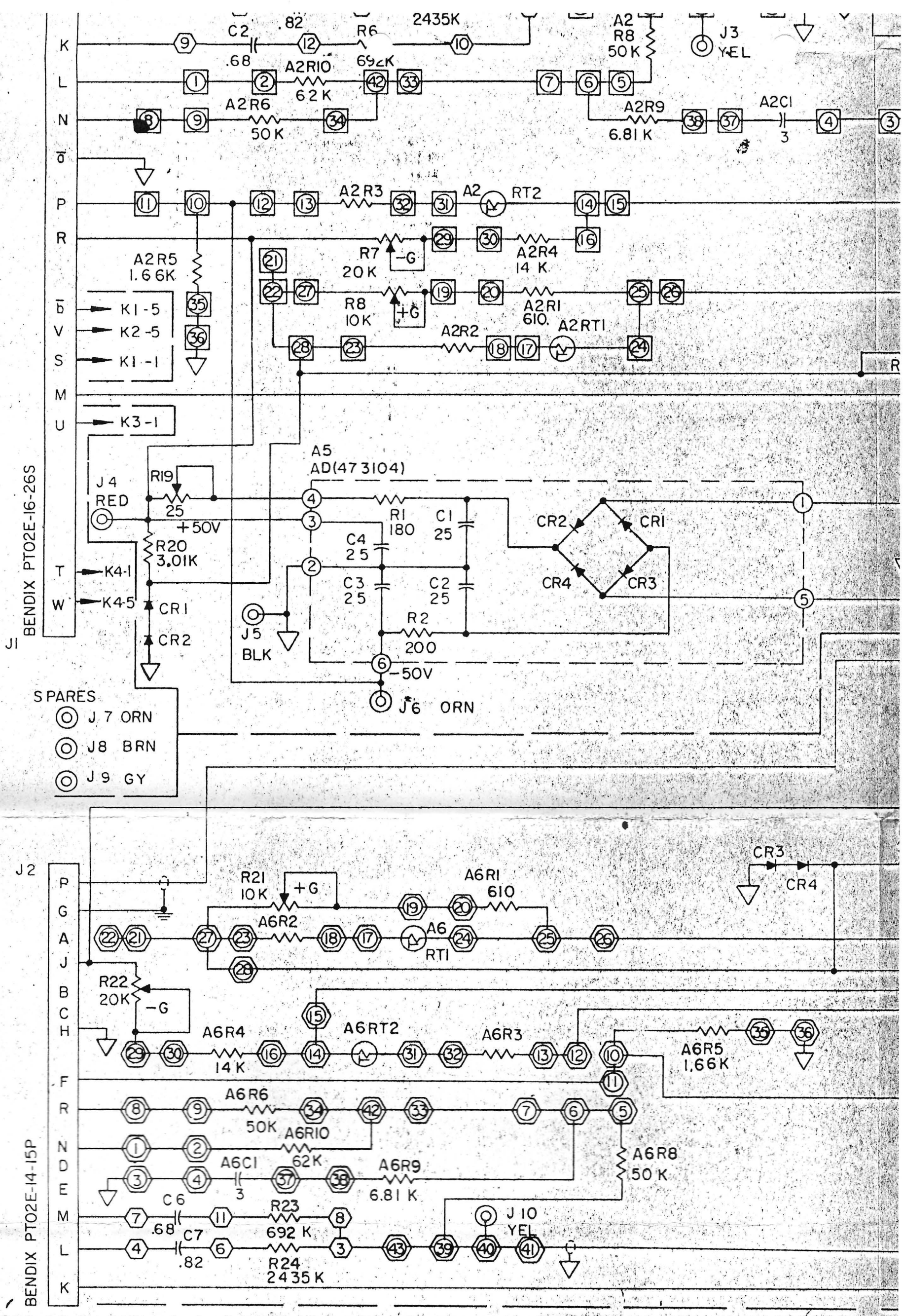
T4

A9

C8864 A


T1
R1
CR1
CR3
C1
C2
CR2
CR4
A3 (D-EG155D-1)
GRN
BLU
GRN
PR
GY
PR
WHT
BLK
R3
ORN
ORN
YEL
YEL
R2
R1
R12
100
R37
1K
BRN
RED
A2
WHT-BRN
WHT-
RED
CR4
WHT-BRN
WHT-
RED
CR3
BRN
A1
RED
CR2
CR1
BLK
PR
T1
PR
BRN
115 V
400 CPS
RED
R14
R15
MEG
14K
R11
412K
T2
CHANNEL 'A'
CHANNEL 'B'
A7
T3
R26
25
SEE A5 FOR
SCHEMATIC
DIAGRAM
+50V
J12
BLK
-50V
R39
14K
R29
412K
R28
1 MEG
R30
100

BLU
GRN
GY
PR
BLK
R3
ORN ORN
YEL
R2
R1
R12
100
R37
1K
WHT-BRN
A2
WHT-
RED
CR4
WHT-BRN
WHT-
RED
CR3
BRN
A1
RED
CR2
CR1
BLK
PR
PR
T1
BRN
115 V
400 CPS
RED
R15
R10
1 MEG
14K
R11
412K
T2
CHANNEL 'A'
CHANNEL 'B'
A7
T3
R26
25
+50V
SEE A5 FOR
SCHEMATIC
DIAGRAM
J12
BLK
-50V
R39
14K
R29
412K
R28
1 MEG
J13
GRN
R30
100
R40
1K
C8
.47
R27
47.5 K
A8
SEE A3 FOR
SCHEMATIC DIAGRAM



11 - SIGNAL GRD OF CHANNEL 1 & 2 & POWER GRD ARE TIED TOGETHER NEAR J1

10 - T2-3, T2-7, T3-3, T3-7, A5-7, A7-7 ARE SPARE PINS USED AS TIE POINTS ONLY

9 - R37 - R40 ARE NOMINAL VALUES, RESISTORS ARE SELECTED FOR VOLTAGE COMPENSATION DURING MFG

8 - ▽ INDICATES SIGNAL GROUND

7 - ⊙ INDICATES TEST JACK

6 - ⬡ INDICATES TERMINALS ON A6

5 - ▣ INDICATES TERMINALS ON A2

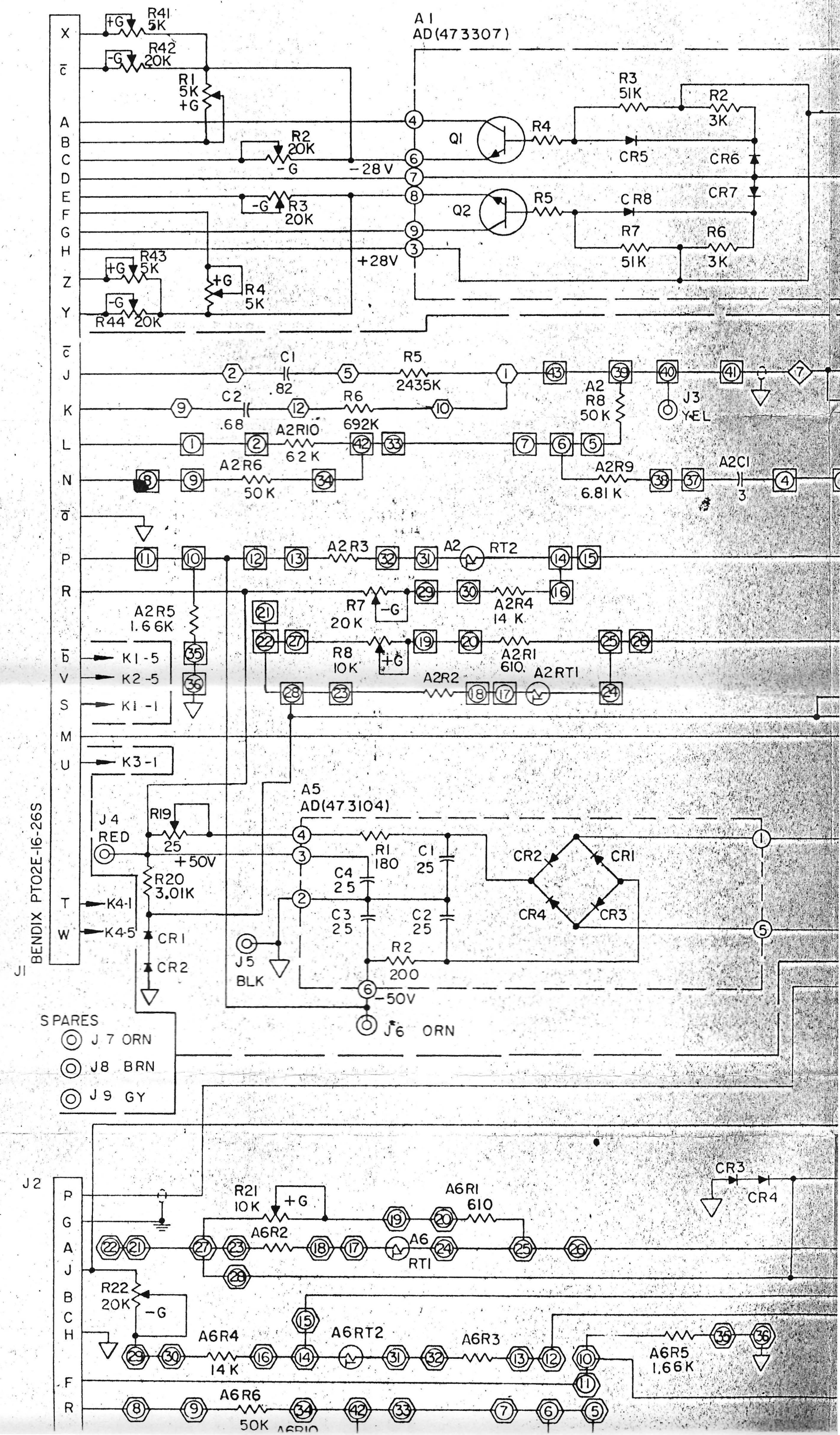
4 - ◇ INDICATES TERMINALS ON TB3

3 - ⌓ INDICATES TERMINALS ON TB2

2 - ⬡ INDICATES TERMINALS ON TB1

1 - ALL RESISTANCES ARE IN OHMS AND ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

8 | 7 | 6


A1
AD(473307)
+G R41 5K
-G R42 20K
R1 5K +G
R2 20K -G
-G R3 20K
+G R4 5K
-G R44 20K
+G R43 5K
-28 V
+28V
Q1
Q2
R4
R5
R3 51K
R2 3K
R7 51K
R6 3K
CR5
CR6
CR7
CR8
C1 .82
C2 .68
R5 2435K
R6 692K
A2R10 62K
A2R6 50K
A2R9 6.81K
A2C1 3
A2R8 50K
J3 YEL
A2R3
A2
RT2
A2R5 1.66K
R7 20K -G
A2R4 14K
R8 10K +G
A2R1 610.
A2R2
A2RT1
K1-5
K2-5
K1-1
K3-1
K4-1
K4-5
J1 BENDIX PT02E-16-26S
J4 RED
R19 25
+50V
R20 3.01K
CR1
CR2
J5 BLK
-50V
J6 ORN
A5
AD(473104)
R1 180
C1 25
C4 25
C3 25
C2 25
R2 200
CR1
CR2
CR3
CR4
SPARES
J7 ORN
J8 BRN
J9 GY
J2
R21 10K +G
A6R1 610
A6R2
A6
RT1
R22 20K -G
A6R4 14K
A6RT2
A6R3
A6R5 1.66K
A6R6 50K

SUPPLEMENTARY INFORMATION	
FIRST USED D-BG128A-1	
NEXT ASSEMBLY	REFERENCE

T1

A4 AD(473230)

RT1

R16 610

R13 4K

R14 1K

R15 1K

R37 1K

R2

R1

YEL

ORN

ORN

BRN

RED

A2

WHT-BRN

WHT-RED

CR4

CR3

CR2

CR1

A1

BLK

PR

T1

115 V 400 CPS

CHANNEL 'A'

CHANNEL 'B'

T4

A9

R39 14K

R29 412K

R31 4K

R30 100

R32

R34 610

SEE A4 FOR SCHEMATIC DIAGRAM

WHT-BRN RED CR3
WHT-BRN WHT-RED CR2
BRN A1 RED CR1
BLK
PR T1 PR
BRN 115 V 400 CPS RED
ORN ORN
YEL
R2 R1
R37 1K
R13 4K
R14 1K
R15 1K
610
CHANNEL 'A'
CHANNEL 'B'
T4
A9
R39 14K
R29 412K
R31 4K
R30 100
R40 1K
R32 1K
R33 1K
R34 610
A8
SEE A4 FOR SCHEMATIC DIAGRAM
SEE A3 FOR SCHEMATIC DIAGRAM

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±1°

DEV 2021-11-01
SUPSD
GOVT MATL SPEC

DRAWN BY	M. Lyzyk	5-6-58
DRAFTING √	C Ryan	5-14-58
MATL & FIN. √		
DESIGNER √		7/29/58
DESIGN	C Ryan	5-14-58
APPROVED		7-29-58

REF MH MATL (SPEC)
FINISH (SEE NOTE)

A1
AD(473307)

R3
51K
R2
3K
R4
Q1
CR5
CR6
CR7
-28V
Q2
R5
CR8
R7
R6
51K
3K
+28V
R1
C1
C2
CR1
CR3
CR2
CR4

A3(D-EG155D-1)

R9
47.5K
C3
.47
GRN
BLU
GRN
PR
GY
PR
WHT
BLK
R3

R5
2435K
R6
692K
A2
R8
50K
J3
YEL
A2R9
6.81K
A2C1
3
A2R3
A2
RT2
R7
-G
A2R4
14K
+G
A2R1
610.
A2R2
A2RT1
R10
1MEG
R38
14K
R11
412K
R12
100

3104)

R1
180
C1
25
C2
25
R2
200
CR2
CR1
CR4
CR3
T2
-50V
J6 ORN

A7
T3
R26
2.5
+50V
R25
3.01K
CR3
CR4
J11
RED
J12
BLK
SEE A5 FOR
SCHEMATIC
DIAGRAM
-50V

A6R1
610
A6
RT1
6RT2
A6R3
A6R5
1.66K
R28
1MEG
J13

R6
692K
2435
A2
R8
50 K
J3
YEL
C3
.47
BLU
GRN
GY
PR
BLK
A2R9
6.81 K
A2C1
3
2R3
A2
RT2
R7
-G
A2R4
14 K
+G
A2R1
610.
A2R2
A2RT1
R10
1MEG
R38
14K
R11
412K
R12
100

7 3104)

R1
180
C1
25
C2
25
R2
200
CR2
CR1
CR4
CR3
T2
-50V
J6 ORN

A7
T3
R26
25
R25
3.01K
+50V
SEE A5 FOR
SCHEMATIC
DIAGRAM
CR3
CR4
J11
RED
J12
BLK
-50V
A6R1
610
A6
RT1

6RT2
A6R3
A6R5
1.66K
R28
1MEG
J13
GRN
A6R8
50 K
A6R9
6.81 K
J10
YEL
C8
.47
R27
47.5 K

RE TIED TOGETHER NEAR J1
IS USED AS TIE POINTS ONLY
RE SELECTED FOR VOLTAGE COMPENSATION DURING MFG

ANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

D-GG 75A-1 SHEET 2 B

D-GG 75A-1 | C8960

THIS DWG APPLIES TO THE ABOVE UNITS | SCHEMATIC DIAG REF

TOLERANCE UNLESS NOTED

PLACE	(0)	±.020
PLACE	(.00)	±.010
EE PLACE	(.000)	±.005
FORMED ANGLES		±1°

2021-22-01

SD

T MATL SPEC

DRAWN BY	R. Ryndy	4-15-58
DRAFTING	Ryndy	4-17-58
MATL & FIN.		
DESIGNER	GDO	4/15/58
DESIGN	Ryndy	4-18-58
APPROVED	RHH	4-22-58

REF MH MATL (SPEC)

FINISH (SEE NOTE)

ACCELEROMETER
INSTALLATON - G LIMITER

SCALE FULL SIZE | WT ACT 1.5 MAX LB

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY

AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

D-GG 75A-1

SHEET 2

A

3 2 1

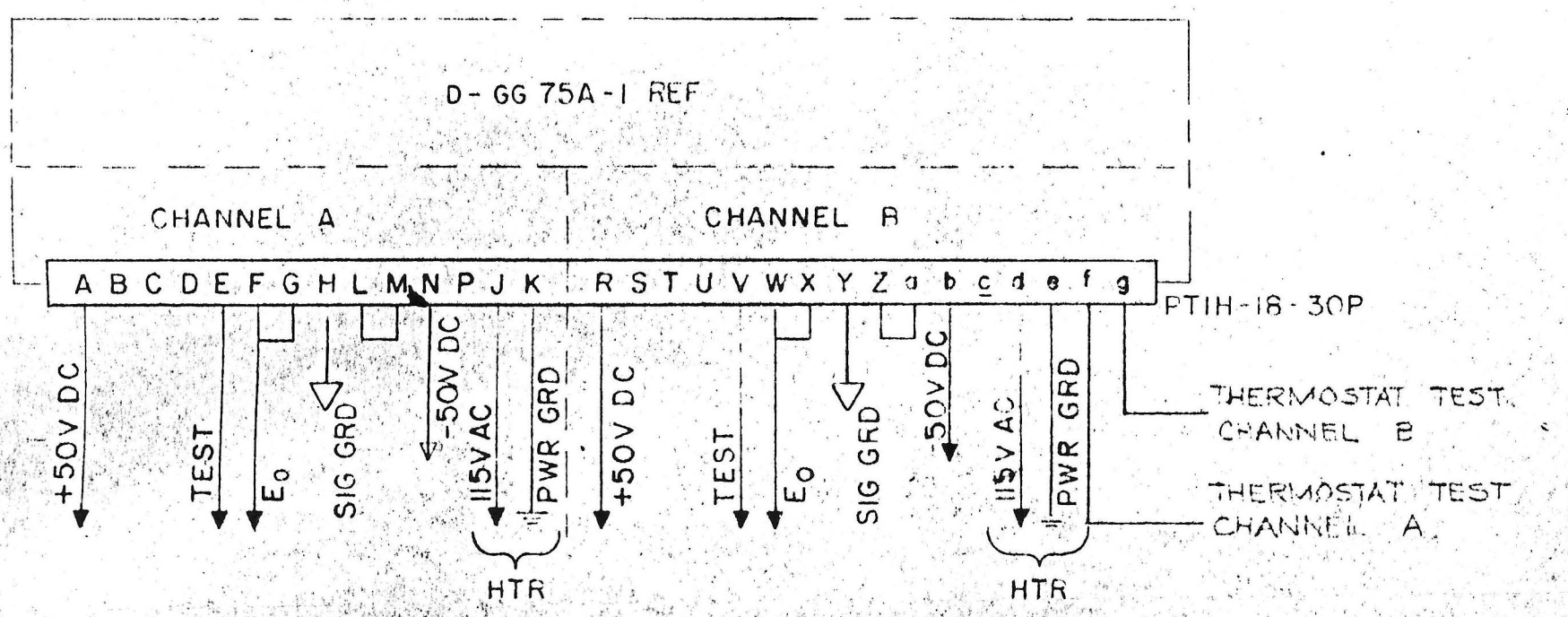


FIG. 1

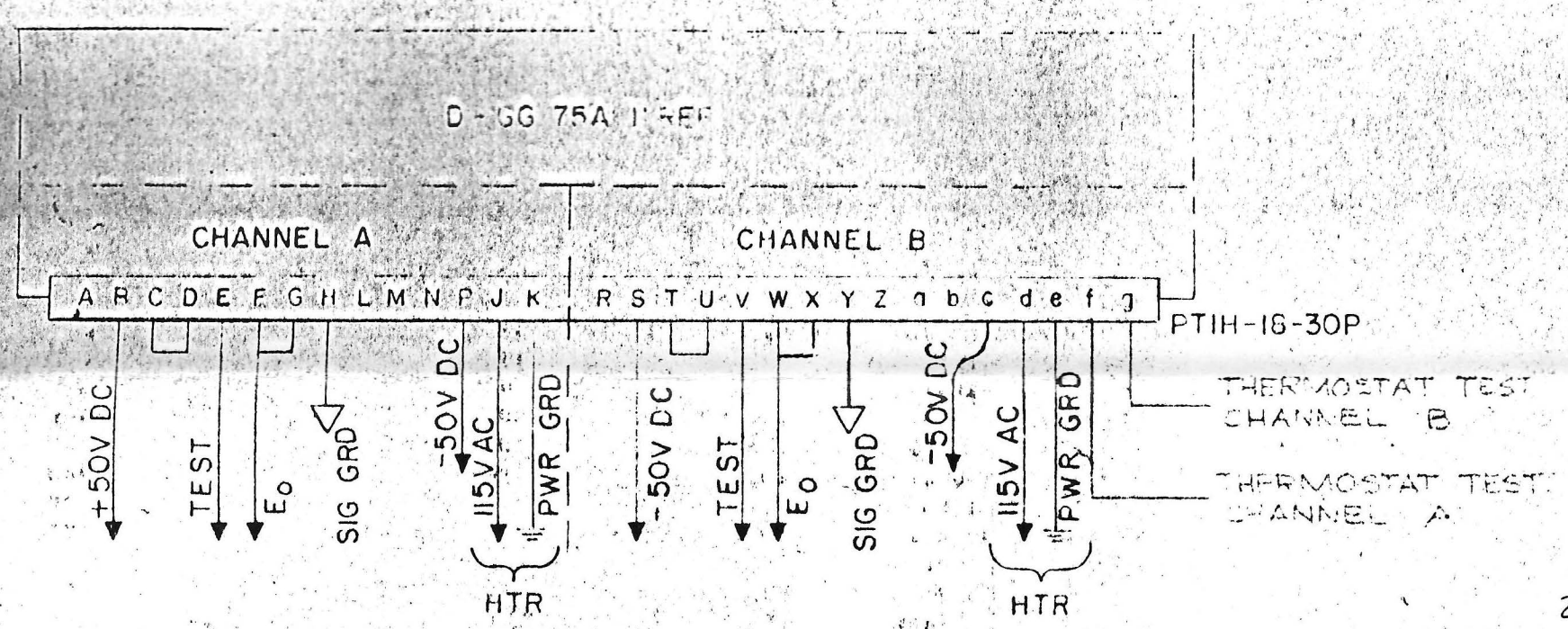


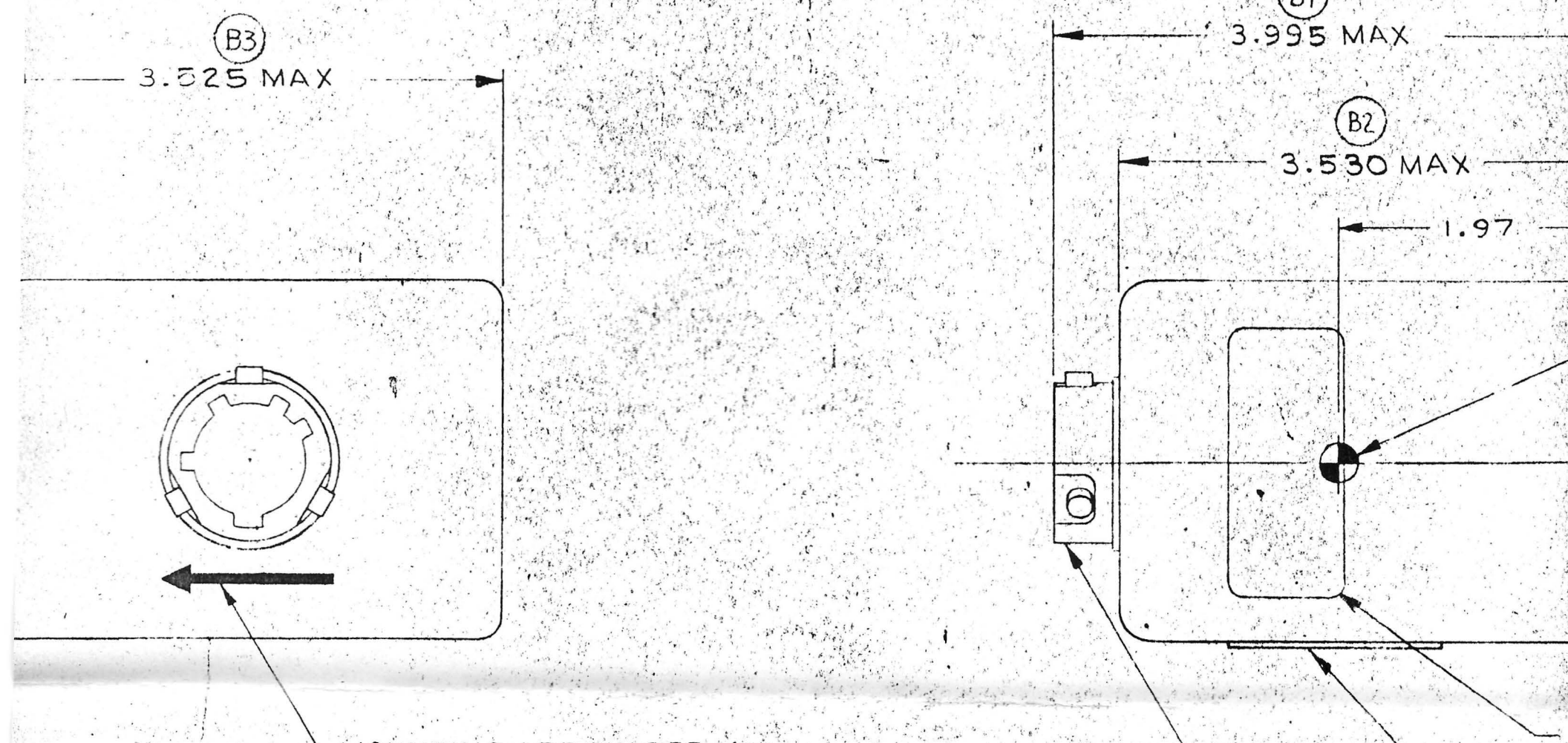
FIG. 2

1 - THE ACCELEROMETER RANGE IS +7.5g ABSOLUTE AND -4.0g ABSOLUTE. SIGNAL OUTPUT POLARITY DEPENDS UPON MOUNTING AND ELECTRICAL CONNECTIONS

2 - WHEN THE OUTPUT CHARACTERISTICS OF FIG.3 ARE DESIRED THE ACCELEROMETER IS TO BE MOUNTED WITH THE MOUNTING ARROW DIRECTED AWAY FROM THE CENTER OF THE EARTH. WIRE PER FIG. 1

3 - WHEN THE OUTPUT CHARACTERISTICS OF FIG4 ARE DESIRED THE ACCELEROMETER IS TO BE MOUNTED WITH THE MOUNTING ARROW DIRECTED TOWARDS THE CENTER OF THE EARTH. WIRE PER FIG. 2

4 - SENSITVE AXIS IS ON ℄ SHOWN REF


B3
3.525 MAX
B1
3.995 MAX
B2
3.530 MAX
1.97
MOUNTING ARROW REF
PTIH-18-30P
BENDIX

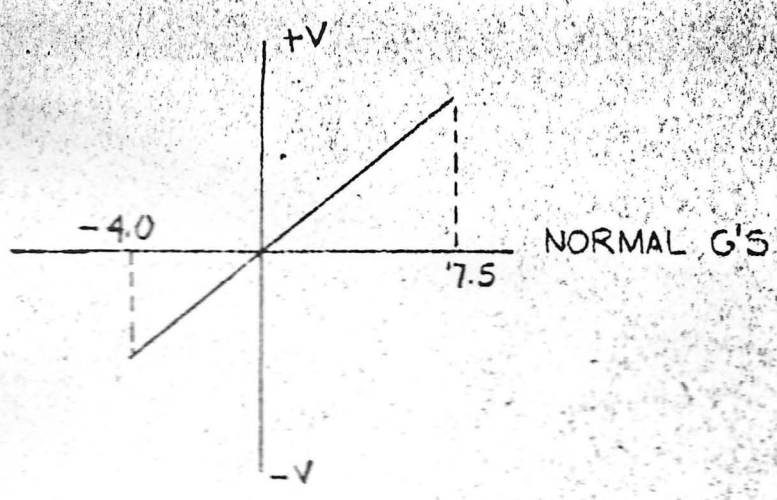

+V
-4.0
7.5
NORMAL G'S
-V

FIG. 3

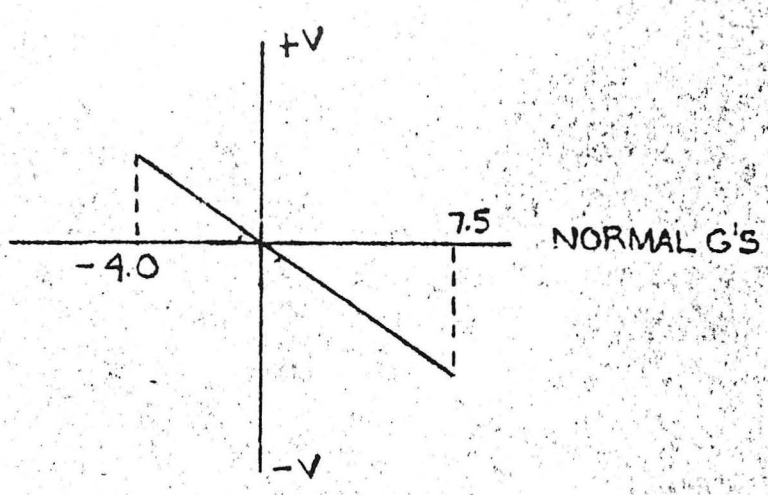
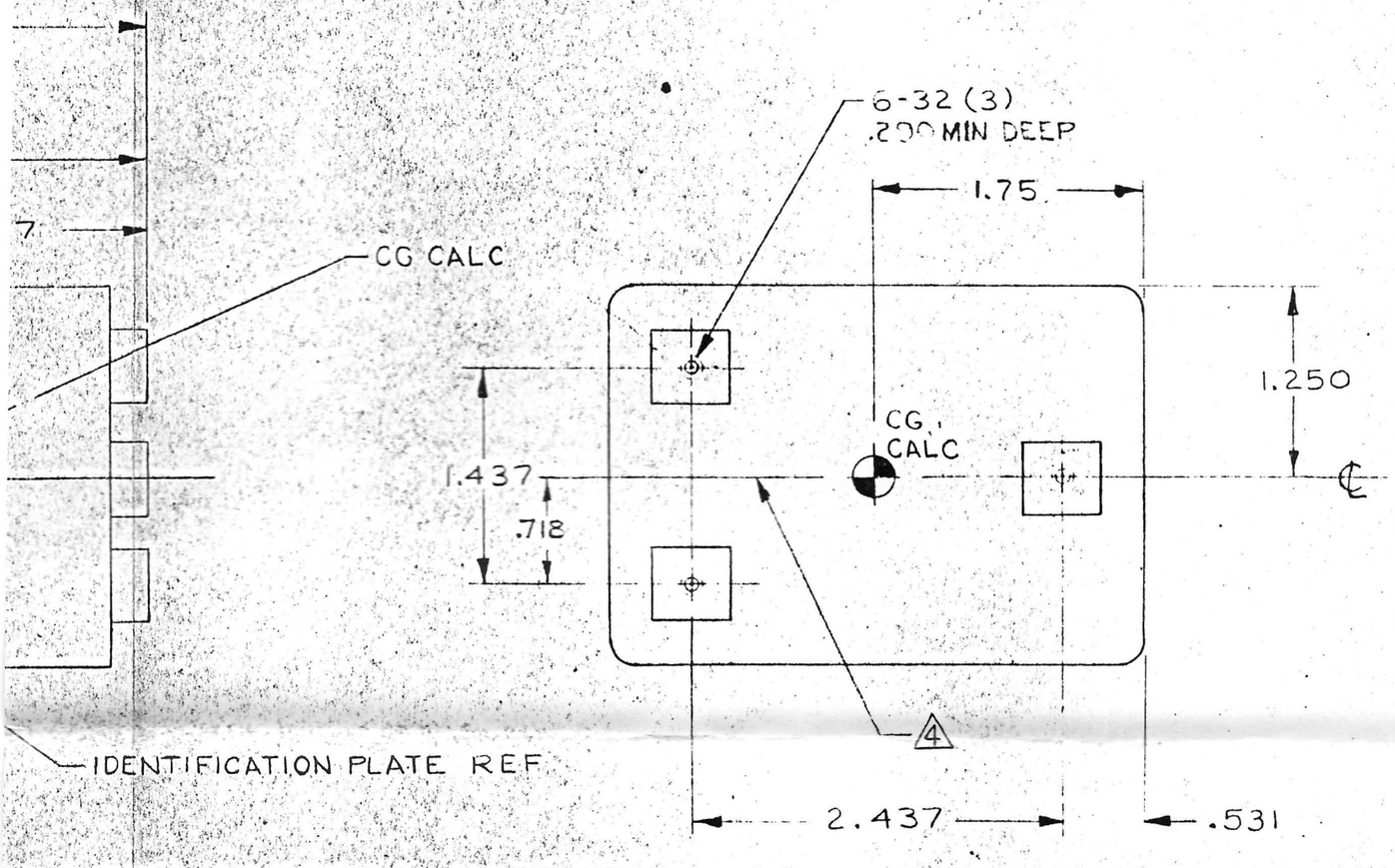

+V
7.5
-4.0
NORMAL G'S
-V

FIG. 4



D

D-GG 75A-1	C8
THIS DWG APPLIES TO THE ABOVE UNITS	SCH DIAG

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
30° FORMED ANGLES		±1°

DRAWN BY	R. P. Grady	4-15-58
DRAFTING		5-19-58
MATL & FIN.		
DESIGNER		4/16/58

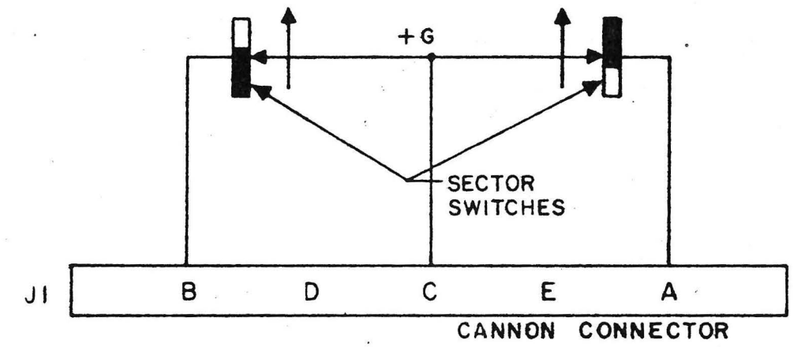
MINNEAPOLIS-HONE
REGULATOR COM

C7608 A

SUPPLEMENTARY INFORMATION		
FIRST USED D-GG62C		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	TITLE WAS "SCHEMATIC DIAGRAM—INERTIA "G" SWITCH	11-8-57	9A-1329

DEVICE NO.	CONNECTOR
D-GG62C-1	GS02-16S-8P-111
D-GG62C-2	GS02-14S-5P-111



D-

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	± .010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±2°

DEV 01196-39

GOVT MATL SPEC

DRAWN BY	L. Johnson	6-13-57
DRAFTING	Gulachek	6-19-57
MATL & FIN.		
DESIGNER		7-19-57
DESIGN	Gulachek	6-19-57
APPROVED		7-19-57

REF MH MATL (SPEC)

FINISH (SEE NOTE)

SCHEMATIC DIAGRAM—
ACCELEROMETER
LATERAL SWITCHING

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT ACT. CALC LB

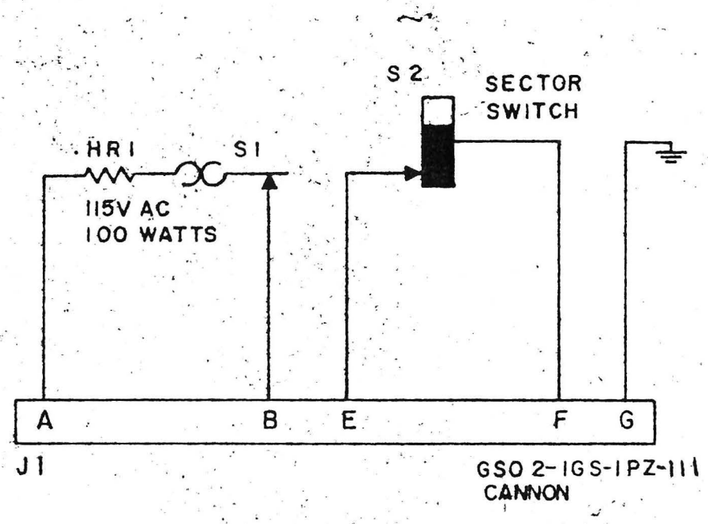
C 7608

8-2799

C6996

SUPPLEMENTARY INFORMATION		
FIRST USED GG59A-1		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL



1- CASE CONNECTED TO PIN G FOR GROUNDING OF CASE

D-

TOLERANCE UNLESS NOTED		
ONE PLACE	(0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±2° −1°

01196-15
SD

GOVT MATL SPEC

DRAWN BY	3-20-57
DRAFTING	√
MATL & FIN.	√
DESIGNER	√ 6-6-57
DESIGN	√
APPROVED	6-7-57
REF MH MATL (SPEC	)
FINISH (SEE NOTE	)

SCHEMATIC DIAGRAM —
INERTIA G SWITCH

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT ACT. CALC LB

C6996

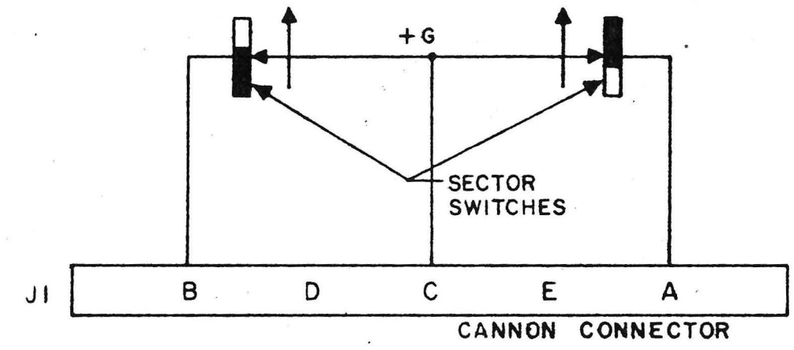
A

C 7608

SUPPLEMENTARY INFORMATION		
FIRST USED D-GG62C		
NEXT ASSEMBLY		REF

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	TITLE WAS "SCHEMATIC DIAGRAM—INERTIA "G" SWITCH	11-8-57	9A-1329

DEVICE NO.	CONNECTOR
D-GG62C-1	GS02-16S-8P-111
D-GG62C-2	GS02-14S-5P-111



D-

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	± .010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±2°

DEV 01196-39

GOVT MATL SPEC

DRAWN BY	L. Johnson	6-1357
DRAFTING	Gulachek	6-19-57
MATL & FIN.		
DESIGNER		7-19-57
DESIGN	Gulachek	6-19-57
APPROVED		7-19-57

REF MH MATL (SPEC)

FINISH (SEE NOTE)

SCHEMATIC DIAGRAM—
ACCELEROMETER
LATERAL SWITCHING

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

SCALE NONE

WT ACT. CALC LB

C 7608

8-2799

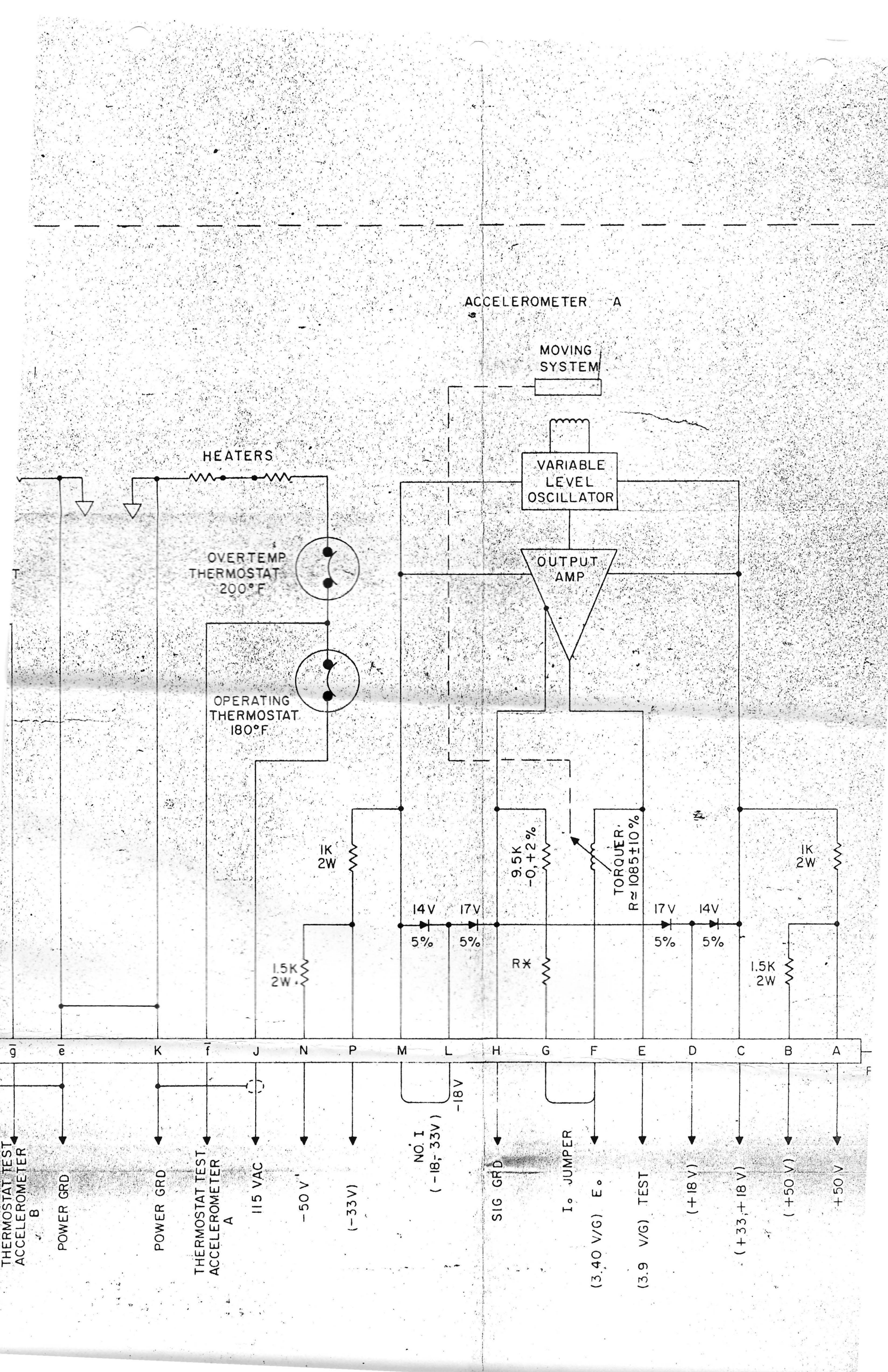
3 – ▽ DENOTES PWR GROUND (CASE)
2 – TOLERANCE ON VOLTAGES IN PARENTHESES ARE ±10%
1 – UNITS ARE SHOWN CONNECTED FOR OPERATING RANGES OF +7.59 TO −4.09
FOR OPERATING RANGES OF +4.09 TO −7.59 CHANGE CONNECTIONS AS FOLLOWS
MOVE JUMPER NO.1 TO PINS C AND D , JUMPER NO.2 TO PINS T AND U
REMOVE WIRE FROM PIN A AND ATTACH TO PIN B
REMOVE WIRE FROM PIN N AND ATTACH TO PIN P
REMOVE WIRE FROM PIN R AND ATTACH TO PIN S
REMOVE WIRE FROM PIN b̄ AND ATTACH TO PIN c̄

A-0692

SIG

I₀
(3.40 V/G)
(3.9 V/G) T
(+
(+33,+
(+
+

TOLERANCE UNLESS NOTED ONE PLACE TWO PLACE THREE PLACE 90° FORMED ANGLES	DRAWN BY R. Olson 7/28/58 MAT'L SPEC Ryan 8-6-58 MAT'L APPD DESIGNED DES APPD Ryan 8-6-58 APPD ENG R.O.H 8-6-58	SCHEMATIC DIAGRAM — ACCELEROMETER GROUP , AIRCRAFT	MINNEAPOLIS-HONEYWELL REGULATOR COMPANY AERONAUTICAL DIVISION MINNEAPOLIS MINNESOTA
DEV 2021-22-01 SUPSD			C8960
GOVT MAT SPEC	MFR MH MAT SPEC FINISH MFR NOTE	SIZE NONE WT ACT CAL LB	


ACCELEROMETER A
MOVING SYSTEM
HEATERS
VARIABLE LEVEL OSCILLATOR
OVERTEMP THERMOSTAT 200°F
OUTPUT AMP
OPERATING THERMOSTAT 180°F
1K 2W
9.5K -0, +2%
TORQUER R≈1085±10%
1K 2W
14V 5%
17V 5%
17V 5%
14V 5%
1.5K 2W
R*
1.5K 2W
g e K f J N P M L H G F E D C B A
THERMOSTAT TEST ACCELEROMETER B
POWER GRD
POWER GRD
THERMOSTAT TEST ACCELEROMETER A
115 VAC
-50 V
(-33V)
NO. 1 (-18,-33V)
-18V
SIG GRD
I₀ JUMPER
(3.40 V/G) E₀
(3.9 V/G) TEST
(+18 V)
(+33,+18 V)
(+50 V)
+50 V

ACCELEROMETER B

MOVING SYSTEM

VARIABLE LEVEL OSCILLATOR

OUTPUT AMP

HEATER

OVERTEM THERMOS 200°F

OPERATING THERMOSTA 180°F

1K 2W

TORQUER R=1085±10%

9.5K -0, +2%

1K 2W

14V 5%

17V 5%

17V 5%

14V 5%

1.5K 2W

R*

1.5K 2W

R S T U V W X Y Z ā c̄ b̄ d̄

+50 V

(+50 V)

(+33, +18 V)

(+18 V)

(3.9 V/G) TEST

(3.4 V/G) E_o

I_o JUMPER

SIG GRD

-18 V

NO.2 (-18, -33V)

(-33 V)

-50 V

115 VAC

ELEROMETER B

MOVING SYSTEM

VARIABLE LEVEL OSCILLATOR

OUTPUT AMP

HEATERS

OVERTEMP THERMOSTAT 200°F

OPERATING THERMOSTAT 180°F

HEATERS

OVERTEMP THERMOSTAT 200°F

OPERATING THERMOSTAT 180°F

TORQUER R=1085 ± 10%

9.5K -0, +2%

17V 5%

14V 5%

R*

1K 2W

1.5K 2W

1K 2W

1.5K 2W

14V 5%

W X Y Z ā c̄ b̄ d̄ ḡ ē K f̄ J N P M L

(3.4 V/G) E_o

I_o JUMPER

SIG GRD

-18 V

NO. 2 (-18, -33V)

(-33 V)

-50 V

115 VAC

THERMOSTAT TEST ACCELEROMETER B

POWER GRD

POWER GRD

THERMOSTAT TEST ACCELEROMETER A

115 VAC

-50 V

(-33V)

NO. I (-18, -33V)

-18V

ACCELEROMETER B

MOVING SYSTEM

VARIABLE LEVEL OSCILLATOR

OUTPUT AMP

HEATERS

OVERTEMP THERMOSTAT 200°F

OPERATING THERMOSTAT 180°F

1K 2W

TORQUER R≈1085 ± 10%

9.5K -0, +2%

1K 2W

14V 5% 17V 5%

17V 5% 14V 5%

1.5K 2W

R*

1.5K 2W

R S T U V W X Y Z a c b d g e

+50 V

(+50 V)

(+33, +18 V)

(+18 V)

(3.9 V/G) TEST

(3.4 V/G) E_o

I_o JUMPER

SIG GRD

-18 V

NO. 2 (-18, -33 V)

(-33 V)

-50 V

115 VAC

THERMOSTAT TEST ACCELEROMETER B

POWER GRD

09683

PTIH-18-30P BENDIX

ACCELEROMETER A

MOVING SYSTEM

VARIABLE LEVEL OSCILLATOR

OUTPUT AMP

1K 2W

1.5K 2W

14V 5%

17V 5%

TORQUER R≈ 1085±10 %

9.5K -0, +2 %

R*

A +50 V

B (+50 V)

C (+33, +18 V)

D (+18 V)

E (3.9 V/G) TEST

F (3.40 V/G) E.

I. JUMPER

G

H SIG GRD

TRANSISTORS

Definition

As a working definition it might be said that a transistor is a valve for controlling electron flow in a solid (as contrasted to a vacuum tube, which controls electron flow in a vacuum). The transistor has the added advantage, however, of being able to control the flow of positive charges, or "holes" as well as electrons.

There are basically two types of transistors, point-contact and junction. The junction type may be further classified as either PNP or NPN. When employed in circuits, they are connected in one of three basic configurations: common base, common emitter or common collector. Since M-H uses PNP junction type transistors in a common emitter circuit, the discussion here will be more or less limited to that particular application.

The junction type transistor is more effective as a low and medium frequency amplifier than PC types. It is more stable and provides more gain in electrical power. PNP consturction is used simply because it is more prevalent among transistor suppliers and not necessarily because it has any advantages over the NPN type. The common emitter circuit is used because it has the highest gain of the three circuit configurations, it requires only one bias source and it features a moderate input impedance and a fairly high output impedance.

Manufacture

Germanium is used in transistors because it is an element that is neither a good conductor nor a good insulator. It is a semi-conductor. (Silicon is also a semi-conductor used in transistors, but its use is not as widespread as germanium).

Germanium is found in many minerals, but two chief sources have been the dust in the stacks of zinc smelters and also the ashes from coal. (Enough ashes and soot are shoveled out in England alone every year to provide enough germanium for 15 billion transistors.) Germanium dioxide is extracted from coal ash at a cost of about $140. per pound and delivered to transistor manufacturers as a white powder, which is heated to 650°C in a hydrogen atmosphere to separate the oxygen from the germanium.

The temperature is then raised to 950°C to melt the powder; upon cooling it becomes a silver-gray metallic ingot with impurities of about one part in 10,000,000 - still too impure for use in transistors, so further purification is necessary. One way of doing this is by a process called "zone melting". The ingot is melted in sections slowly from one end to the other, and the impurities, which prefer molten germanium to frozen germanium, are swept along in the molten mass to the end of the ingot, which is cut off. The process is repeated until there is only one foreign atom to every one billion germanium atoms.

Purity is measured by the amount of resistance the material offers to current flow. Resistance increases as the material becomes more pure and "pure" germanium after zone melting has a resistance of about 60 ohms per cc. By introducing controlled amounts of impurities at this stage the resulting germanium will have a given amount of free electrons available for current conduction - or a given amount of "holes" (absence of electrons) available for conduction, depending on the type of impurity added.

Although the germanium is pure enough after zone melting, it is in polycrystalline form; that is, it is composed of hundreds of individual germanium crystals whose boundaries are at random angles to each other presenting serious obstacles to smooth electron flow.

The next step in building a transistor simultaneously introduces impurities in controlled amounts and forms a single crystal. A small single crystal, a "seed", is dipped into molten germanium and slowly raised. The germanium freezes to the seed in the same crystal orientation as the seed, and after several hours of slow withdrawal a long single crystal is formed (possibly 10 - 12" long) with enough germanium for thousands of transistors. This process is called "pulling" Before this crystal growth is started, impurity is added to the molten germanium in the ratio of one millionth of a gram of impurity to 100 grams of crystal.

To produce N type germanium (which will have an excess of electrons) arsenic is the most commonly used impurity, although antimony and boron may be used also. To produce P type (a deficiency of electrons or holes) indium or gallium may be added.

A PNP transistor is made by fusing a portion of indium on each side of a wafer of N type germanium about .001 in. thick. Only a portion of the indium on each side acts as the P-type layer, just the portion that consists of a mixture of germanium and indium in the correct ratio. The remainder of the indium serves as a handy point for soldering two leads, and a third lead is attached to the thin layer of N-type germanium. The finished transistor is coated with plastic to give it mechanical strength and to protect it from moisture, dust, light or anything else which might alter its electrical characteristics.

Transistor Action:

Before considering the action of a PNP transistor (a triode), let us look first at the transistor diode rectifier, which is formed by joining a section of N type with a section of P type germanium as shown in Fig. 1.

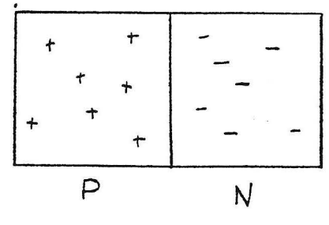


Fig. 1 A Diode Rectifier

The + signs in the P type indicate the presence of holes, and the - signs in the N type indicate the presence of electrons. (It is well to keep in mind that electron current flow, in transistors, is like that in any electrical circuit. The movement of electrons is from the point of highest negative potential through the circuit to those points of less negative potential.)

The PN junction (point at which the two types are joined) may act either as a conductor or as an insulator, depending on the polarity of its external electrical connections. If the positive terminal of a battery is connected to the P section, as in Figure 2, the holes are repelled away from the connection toward the junction.

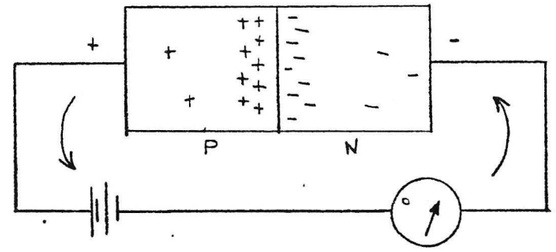


Fig. 2 Diode Rectifier Biased in the Forward Direction
Arrow Indicate Electron Flow

If the negative terminal of the battery is connected to the N section, the electrons are repelled away from the connection toward the junction. There are now many current carriers at the junction or, in other words, the resistance at the junction is low, and a large current flows. Actually, there is a fairly free movement of carriers across the junction, holes from the P material moving into the N material and vice versa. This type of battery connection is called forward bias.

If the battery terminals are reversed, as in Figure 3, the holes in the P section are pulled away from the junction, and the electrons in the N section are likewise pulled away, leaving no carriers at the junction.

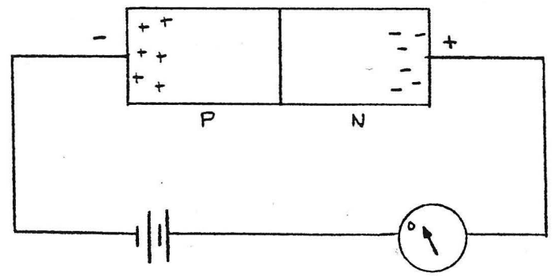


Fig. 3 Diode Rectifier Biased in Reverse Direction
Practically no current flows

The resistance is high, and only a very low leakage current will flow. This is called reverse bias.

A PNP triode transistor is simply the combination of two diodes with N types joined (PNNP). The N type center material is about 0.001 in. thick.

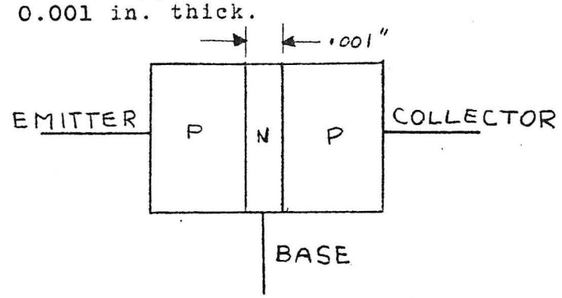


Fig. 4

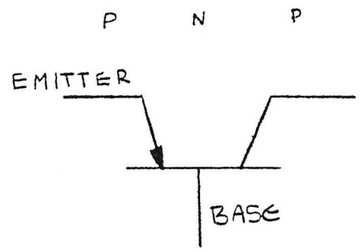


Fig. 5

The terminals are labeled as shown in Fig. 4. The schematic symbol is shown in Fig. 5, with the arrow representing the direction of hole flow. (If the term "hole flow" is confusing, imagine a line of cars waiting for a light to change.)

When the light does change, the first car - an electron - will move forward, leaving the space, or hole, where it used to be. The second car will move forward, filling up this hole, and the hole will move back to the space where the second car used to be. Thus, the cars - electrons - move forward, and the spaces - holes - move in the opposite direction.)

The emitter-base junction is biased in the forward direction (See Fig. 2) and the collector-base junction is biased in the reverse direction. (See Fig. 3) This means that a good current will flow between emitter and base, since that junction has a low resistance (around 500 ohms). Without this current, there would be practically no current flowing between collector and base, since that junction has a high resistance (around 500K ohms).

However, with the E-B junction forward biased, it might be said that holes are "squirted" into the N material from the emitter, thus providing a quantity of carriers available to the collector. Since the majority of the holes originally present in the collector were pulled to the terminal connection, the junction side of the collector was left relatively free of holes, and thus the collector will "collect" more holes from the base, or N material.

This means, then, that holes (current carriers) are moving across the C-B junction. The number of holes that move across depends directly on the number made available in the N material which, in turn, depends on the strength of the E-B current. If that current is varied, the number of holes in the center (N) material will change, and similar variations will occur in the C-B current flow. In practice, change in E-B current is varied by adding a small AC signal in series with the bias.

Currents in the two circuits may have about the same value, but an appreciable power gain may be realized between the two circuits because of the difference in impedances.

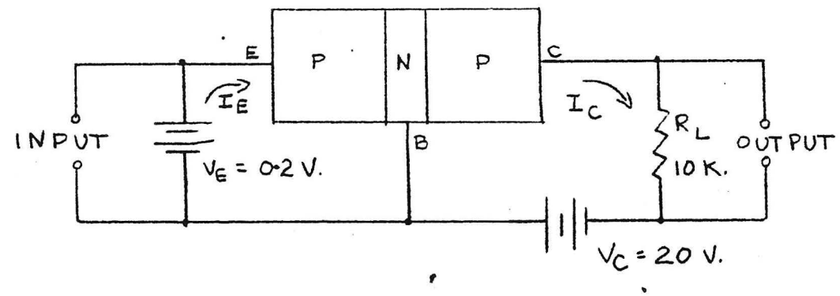


Fig. 6 Bias on a PNP Triode. Arrows indicated hole flow.

MAGNETIC AMPLIFIERS

1. General

The requirements of modern electric systems demand increased reliability and ruggedness from the components used in the systems. Designers have endeavoured to find substitutes for the vacuum tube which has been responsible for much of the unreliability of electronic systems in the past. In many applications, the transistor may be used as a reliable and rugged replacement for the vacuum tube. However, when it is desired to control relatively high power circuits, use of the transistor, while more reliable than a vacuum tube, may increase the cost of the system beyond reasonable limits. The magnetic amplifier, which can control high power circuits is reliable and rugged, as well as being less costly than an equivalent transistorized system.

2. Principles of Operation

Diagram 1(a) shows an idealized magnetization curve for the magnetic core shown in 1(b). On the core is wound a coil with N turns with a direct current I flowing through it. As shown in 1(a) an increase of ampere turns (N x I) causes an increase in flux (ϕ) which is rapid and nearly linear up to the point A. At this point, the rate of increase of flux becomes much lower and the core is said to be saturated.

Diagram 2(a) show the core of 1(b) with an additional winding and an alternating current i flowing through it. Diagram 2(b) shows the combined action of the direct current I and the alternating current i. If the direct current is zero, the alternating current will operate about the origin and will change the flux in the same manner as would direct current variations. The resultant flux change is shown. If the direct current is increased so that the operating point is at X, an alternating current of equal amplitude to the current acting at the origin will produce the same flux change as the current acting at the origin. But if the direct current is increased until the operating point is at point y, the core is saturated. The flux change caused by the same alternating current i will now be very small.

Since inductance of any magnetic coil is determined by the ratio of flux change to current change, the inductance of the core will be much greater from 0 to A than to the right of A. We can see that by varying the direct current in a control winding we may vary the inductance and therefore the impedance of a main or gate winding.

Diagram 3 shows the current output of the gate winding against control winding current for a system connected as in 2(a). It should be noted that increasing the control current in either direction from zero will increase the alternating current output. The increase of current is limited of course, by the internal resistance of the core and the resistance of the load, R_L. The

diagram indicates as well that output current does not go to zero when control winding current goes to zero.

If a large number of turns is used on the control winding, a small direct current may be used to control relatively large amounts of A.C. power.

3. Biasing

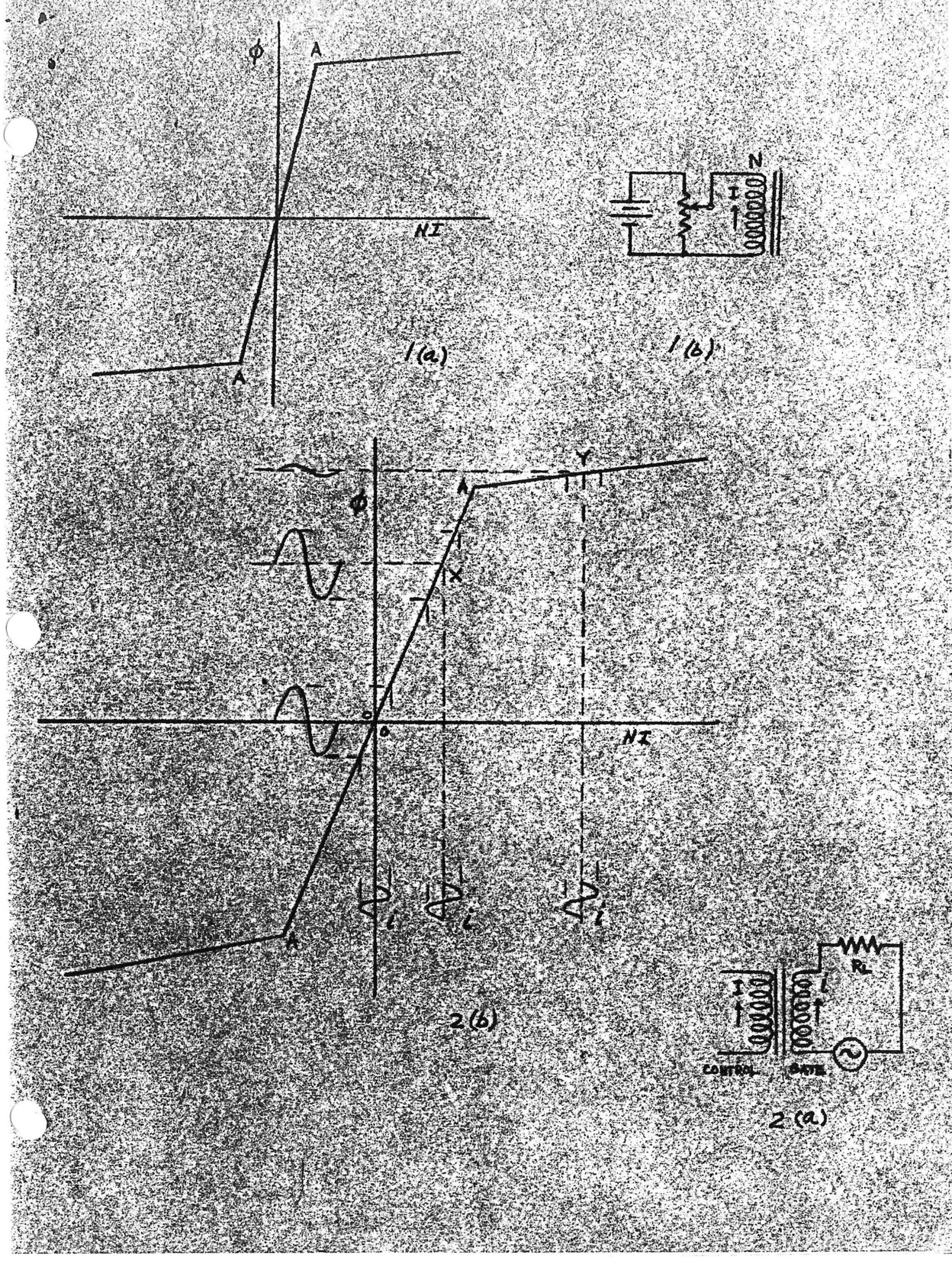
If the magnetic amplifier is to provide increasing and decreasing power as the direct current control is increased and decreased, operation must be confined to the linear portion of the curve in Diagram 3 between A and B. By setting up a fixed bias current in the control winding or by using a separate bias winding, control current changes about this bias point can cause both increased and decreased power. It should be remembered that the output current will never be zero using this method. For adequate control action, the inductive reactance of the unsaturated gate winding must be high compared to the resistance of the load.

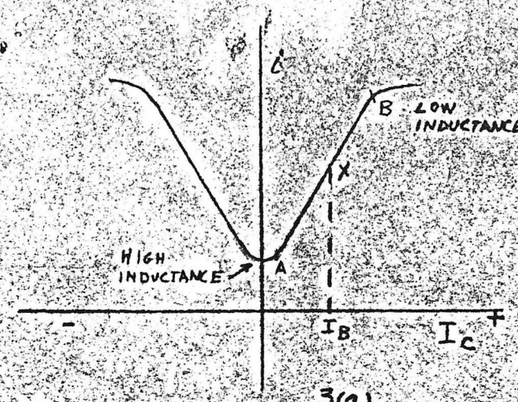
4. Transformer Action

Alternating currents flowing through the gate winding of a magnetic amplifier will induce currents in the control winding. These currents, flowing back through the control current source will produce an undesirable power loss. By connecting two magnetic amplifiers as shown in Diagram 4, the voltages induced in the control windings will cancel and no power loss will result.

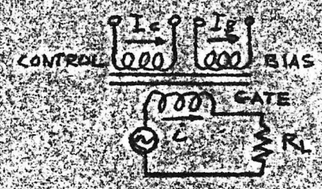
5. Self Bias

By connecting a rectifier in series with the gate winding of a magnetic amplifier, a pulsed output will be applied to the load, but in addition, an average direct current will flow through the gate winding. This direct current will be proportional to the alternating current flowing in the winding. The curve of 5(b) shows the effect of this current with no current acting in the control winding. With control current zero, an alternating current A will flow in the gate winding. This will produce an average current B which in turn will increase the alternating current to C by acting in the same manner as a control current. This process will continue until the operating point becomes D, near the maximum output of the amplifier. Introduction of very small control currents can cause this action to reverse. The net affect of the self bias action is to increase the gain of the magnetic amplifier by a very large factor. The characteristic curve of the self biased magnetic amplifier can be redrawn as shown in 5(c). The slope of the linear portion of the curve is much higher, indicating that the gain of the amplifier has been increased. It can be seen from the curve, that the minimum output current point has shifted to some negative value of control current rather than being at zero control current as in an unbiased amplifier.

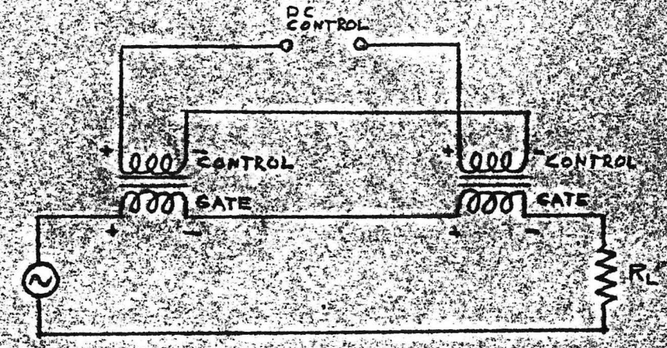

φ
A
NI
A
N
I
1(a)
1(b)
φ
A
Y
X
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i
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2(b)
RL
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CONTROL
GATE
2(a)



3(a)



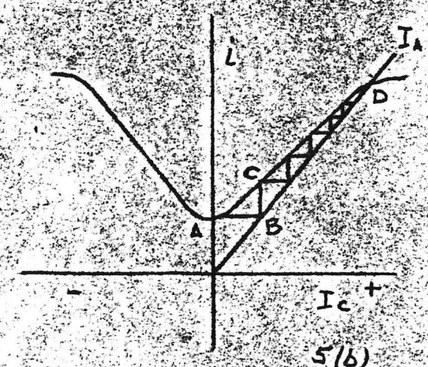
3(b)



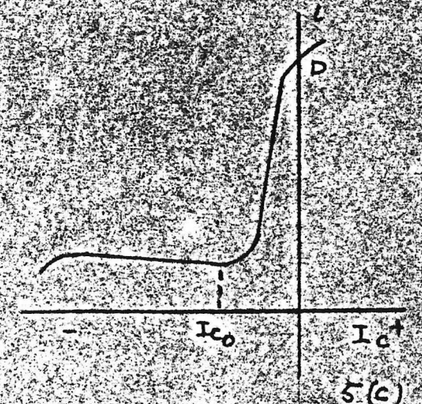
4



5(a)



5(b)



5(c)

AMPLIFIERS

A. PITCH RATE GYRO AMPLIFIER EG153A-1

1. The EG153A-1 Amplifier is a completely transistorized two stage amplifier with a self contained power supply. It consists of one preamplifier stage, a push-pull Class B output stage, and a full wave diode rectifier.

2. The circuit of C7554 shows the EG153A-1 Amplifier. The input is applied between terminals one and three and is applied in series with a negative feedback voltage from the output transformer T3, to the input transformer T1. The amplifier circuits are conventional; however several diodes have been added to improve performance. CR5 and CR1 are used to improve the linearity of the amplifier stages without the large gain loss resulting if negative feedback alone were used. CR4 is used as a voltage regulator. If the rectifier output voltage exceeds the reverse voltage breakdown point of CR4, a large current will flow through R6 and CR4 to reduce the excessive voltage at CR4.

3. Three output windings are provided on T3, the output transformer. Winding 2 is the feedback winding, 3 is the output to the pitch axis of the damper and 4 is the output to the normal yaw axis of the damper.

B. SERVO AMPLIFIERS AND TRIM AMPLIFIERS EG113B-1 AND D-1(C6626 and C6949)

1. The schematic diagrams of the amplifiers are identical. Differences will be found in the values of R6 and R1 only. The amplifier consists of a two stage direct coupled pre-amplifier, a full wave demodulator, and a full wave diode recitifer power supply.

2. Pre-Amplifier

The bias voltage of the first stage transistor Q1 is produced by the voltage drop across R4 in the second stage circuits. Feedback of signals between the second and first stage is prevented by the inductive choke L1. Direct coupling is used between the first and second stages to remove the requirement for a bulky coupling capacitor.

The second stage is conventional except for the inclusion of CR9 in the emitter circuits. To explain the action of CR9, we must first assume that CR9 is replaced by a fixed resistor, R. Since the base of Q2 is at the relatively high collector potential of Q1, this resistor R must be relatively high in value to limit the emitter bias current. To prevent feedback, and the attending gain loss, this resistor, R must be by-passed by a large capacitance. The combination of R and its by-pass will hold a fixed DC voltage across R.

The capacitor, however, will be bulky. By using a diode, CR9 connected in reverse polarity, we may replace the resistor and capacitor by a very small component. If the diode is selected so that its reverse breakdown voltage is the voltage we would have obtained across R, then its action will be the same as the resistor and capacitor combination. When a diode is operated at its reverse voltage breakdown point, its impedance is very low. This means that very little feedback will be developed across it. The output of the second stage is produced in transformer T2. Winding number three of this transformer makes a feedback voltage available if desired.

3. Demodulator

To describe the action of the demodulator, we will examine the circuits of Q4 only. The collector of Q4 is supplied by two power transformer windings through two diodes CR4 and CR5. It should be noted that CR4 is fed from the end E of winding two, while CR5 is fed from the beginning B of winding number three. This means that B3 and E2 are out of phase. Let us assume that B3 is in phase and E2 is out of phase. If an in-phase signal is present at the base of Q4, conduction can occur through Q4 and CR5 during the first half cycle. During the second half cycle, E2 will be positive, but Q4 base will be negative going, and no conduction will occur through CR4. This means that a half wave output current can be produced in circuits connected to terminal 5.

If an out of phase signal is applied to the base of Q4, no current will flow through CR5 during the first half cycle, since the transistor will be cut off. On the second half cycle, however, E2 will be positive at the same time as the transistor base goes positive, and conduction will occur through CR4. This will produce a half wave output current in circuits connected to terminal 8. In summary, an in phase signal produces a half wave output at terminal 5, and an out of phase signal produces a half wave output at terminal 8.
Q3 acts in the same manner as Q4, and its action, combined with the action of Q4 produces full wave demodulation.

4. As in the EG153 amplifier, CR2 is used to improve linearity, and CR1 is used as a voltage regulator.

C. INTEGRATOR AMPLIFIER EG129A-1

1. The EG129A-1 is a hybrid amplifier consisting of a transistorized amplifier demodulator and a magnetic amplifier. The transistorized portion of this amplifier is similar in circuitry and operation, to the EG113 series of amplifiers. By referring to schematic C6462, we can see that the demodulator will control the currents flowing in the control windings of the magnetic amplifier.

2. The Magnetic Amplifier

The purpose of the magnetic amplifier is to apply 115 volt, 400 cycle power to either terminal 1 or terminal 4 by using control currents from the demodulator. In the schematic, the magnetic amplifier looks to be quite complex. We may simplify it by neglecting some of the windings for the present. The mag-amp consists of four cores A, B, C, and D. Each core has five windings. Winding 8-10 is the gate winding, and windings 2-16 and 1-15 are control windings. Windings 7-9, 11-3 and 12-5 can be neglected for the present. This leaves us with four gate windings connected in series with four diodes, and four sets of two control windings.

If we assume first that no control current is flowing, we can see that each of the gate windings will produce self bias and will remain in the fully "on" position. This means that gate C and CR11 will apply positive half cycles to terminal 1, and gate D and CR10 will apply negative half cycles to terminal 1, producing a complete sine wave at this terminal. Similarly gate A and B will apply a full sine wave to terminal 4. With no control current, both terminal 1 and terminal 4 have power applied to them. To produce control action, power must be removed from either terminal 1 or 4.

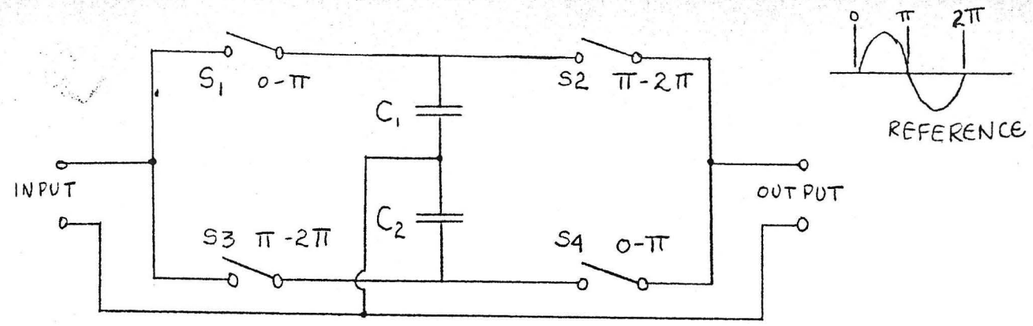
For simplicity, we will designate the four control windings connected in series with R9 as the R9 control and the four control windings connected in series with R10 as the R10 control. If current increases in the R9 control, current flow through A and B control windings will be in a direction which will oppose the self bias action of the A and B gates and will tend to turn off these gates and remove power from terminal 4. Current flow through R9 controls C and D will be in a direction to aid the self biasing action of gates C and D, so that these gates will continue to supply terminal 1.

Similarly, if R10 control current increases, (R9 control zero) then gates A and B will remain on, and C and D will remove power from terminal 1.

Windings 11-3 and 12-5 of each core are connected such that currents will flow in them only during the half cycle in which the gate 8-10 of the core is not conducting. These windings have a reset action during the off half cycle to return the magnetic amplifier operating point to a consistent starting point on the hysteresis loop of the core. Winding 7-9 is used to reduce harmonic distortion of the output waveform.

D. QUADRATURE FILTER AG26A-1

1. The action of the quadrature filter can be shown by the simplified diagram below.



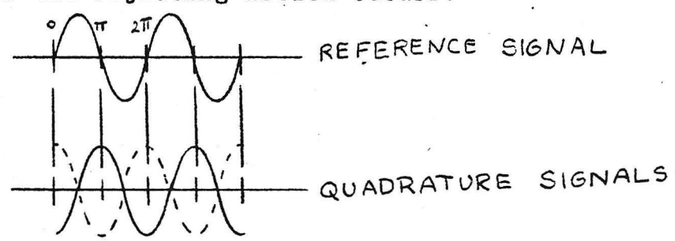
1. Quadrature Filter

Switches S1, S2, S3 and S4 close during the portion of the reference wave indicated. If an in phase or out of phase signal is applied at the input, the following action takes place. S1 will close during the first half cycle of the reference (0-π), and the first half cycle of the signal will be applied to capacitor C1, which will charge to the average value of the first half cycle. During the second half cycle of the reference (π - 2π) S2 closes and S1 opens. C1 will discharge to the output circuits.

S3 is also closed during this period (second half cycle) so that capacitor C2 will charge to the average value of the second half cycle of the input signal. During the third half cycle, S3 will open and S4 will close, discharging C2 to the output circuits. The output voltage will be alternating in nature but will be of opposite phase to the input signal. It should be noted that during the third half cycle, S1 is again closed and the whole cycle repeats.

2. Quadrature Rejection

If we assume that a quadrature signal is applied to the filter input we can see how the rejecting action occurs.



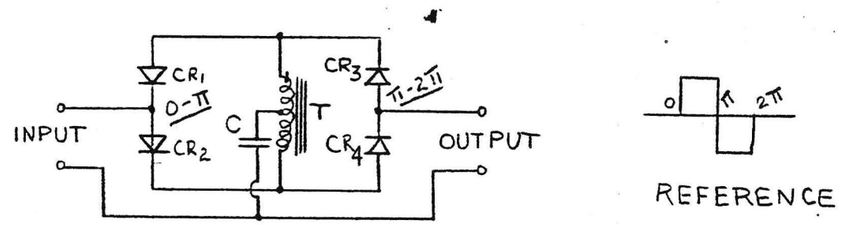
2. Quadrature Signal

Diagram 2 above shows the relation of a quadrature signal to the reference signal. By definition, it is displaced from the reference phase by 90 degrees. As shown by the solid and dotted lines, a quadrature signal can either lead or lag the reference by 90 degrees.

We have mentioned above that switch S1 closes during the first half cycle (0-π) of the reference, and the input signal is applied to the capacitor C1. The capacitor will charge to the average value of the input signal during that period (0-π). As shown in the diagram above, the average value of a quadrature signal during the period 0-π of the reference, must be zero, since an equal area of the quadrature signal lies below the zero line as lies above the zero line. Therefore, when S2 closes during the period π-2π of the reference, no output will be produced, since the average charge on the capacitor will be zero. When S3 closes, also during the period π-2π the charge applied to C2 will average to zero as well, and no output will be produced during the time that S4 closes. By this action, quadrature signals are prevented from passing through the filter, while signals which are in phase or out of phase with the reference will pass through, undergoing a phase reversal.

3. Diode Switch

The simplified diagram below shows how the switches S1 and S2 are mechanized. A duplicate system to that shown below is used to mechanize S3 and S4.



3. Diode Switch

"T" is a secondary winding of a transformer, the primary of which is supplied with a square switching wave at the reference frequency and phase. During the period 0-π, CR1 and CR2 will conduct, since the reference wave is positive. CR1 and CR2 will provide a low impedance path between point A and point B. Any signal present at A will charge C1. During this same period (0-π) CR3 and CR4 are biased off, and no low impedance path exists to the output.

During the period π-2π of the reference square wave, CR1 and CR2 are biased off and CR3 and CR4 conduct. A low impedance path is now present between point B and point C, so that any charge on the capacitor can discharge to the output circuits. The action described above is the same as the action of switches S1 and S2 in Diagram 1. By using the duplicate system, but

using C as the input and A as the output, the action of switches S3 and S4 can be mechanized.

Schematic C7632 shows the AG26 in detail. Two complete filters are contained in the same unit. Diode CR1 clips an input reference sine wave to produce the square switching wave desired on the primary of transformer T2. Secondaries 1 and 1', 2 and 2' operate the two diode switch sections. The two tapped windings 4 and 5, are used to balance the currents flowing in the diode switches so that no output is produced when no input is applied.

Aeronautical Division

C.E.L. NO. C-105-293
DATE February 24, 1958

CUSTOMER ENGINEERING LETTER

Mr C V Lindow
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AVRO Aircraft Limited
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cc C N Lucas
AVRO Purchasing

MH-64 DAMPER; ARROW
QUALIFICATION TESTING
REVISED INFORMATION

REF: 1. CEL No C-105-109, dated April 8, 1957
2. CEL No C-105-216, dated October 7, 1957
3. CEL No C-105-227, dated October 18, 1957
4. CEL No C-105-246, dated November 27, 1957

Gentlemen

MH, in a meeting with AVRO personnel at Minneapolis on February 3 and 4, presented a revised estimate of costs for the production design and qualification program.

The accompanying tables show the tests MH will conduct on the YG339 (MH-64) Damper System components on the revised qualification on which the discussion in the above meeting was based.

The tests which will be conducted are indicated by an X in the table. The tests to be conducted is referenced to an applicable paragraph of MH Document R-ED-889, including Amendment C. R-ED-889 will be rewritten to reflect the contents of this letter.

The following tests have been eliminated from the qualification program.

Paragraph 4.3.2.1.6	Salt Spray
Paragraph 4.3.2.1.7	Sand and Dust
Paragraph 4.3.2.1.8	Fungus Resistance
Paragraph 4.3.2.1.11	Explosion Proof

MH has removed these tests from experience with similar units because failures on previous damper and autopilot components due to the above four tests have been no existant.

The LG16K-1 Parallel Servo Position Potentiometer is similar to production devices which have been previously qualified and therefore no qualification type tests are required on this device.

CONTROL VALVE

TABLE I

		Elevator Valve	Aileron Valve
Individual Tests	(4.4.5.1.1) (4.4.6.1.1)	X	X
Humidity	(4.3.2.1.5)		X
Shock	(4.3.2.1.9)		X
Vibration	(4.3.2.1.10)		X
Flow Displacement	(4.3.2.1.17.1)	X	X
Operational	(4.3.2.1.17.2)	X	X
Flow Pressure Force	(4.3.2.1.17.3)	X	X
Low Temperature	(4.3.2.1.17.4)	X	
Rapid Warm-Up	(4.3.2.1.17.5)	X	
High Temperature	(4.3.2.1.17.6)	X	
Endurance	(4.3.2.1.17.7)	X	
Burst	(4.3.2.1.17.8)	X	X
Ultimate Load	(4.3.2.1.17.9)	X	

SERVOS

TABLE II

	Minimum Clearnace MG55N	Minimum Clearance MG51D	Maximum Clearance MG51D	Minimum Clearance MG62B	Maximum Clearance MG62B
Individual Tests (4.4.1.1.1) (4.4.3.1.1) (4.4.4.1.1)	X	X	X	X	X
Humidity (4.3.2.1.5)	X	X			
Shock (4.3.2.1.9)	X		X		X
\ ration (4.3.2.1.10)	X	X		X	
Radio Interference (4.3.2.1.12)	X		X	X	

TABLE II - continued

	Minimum Clearance MG55N	Minimum Clearance MG51D	Maximum Clearance MG51D	Minimum Clearance MG62B	Maximum Clearance MG62B
Voltage Stability (4.3.2.1.13)	X		X	X	
High Potential (4.3.2.1.14)		X			X
Extreme Temperature (4.3.2.1.16.1) (Minimum Clearance)	X	X		X	
High Temperature Life (4.3.2.1.16.2) (Maximum Clearance)			X		X

ELECTRONICS

TABLE III

Accelerometers shall include: GG47D, GG58A, GG58B, GG59A, GG62B, GG62C, GG75A.
(tests to be conducted on each type listed)

	Accelerometers	GG77A	GG72A	QG100A	BG92B	BG128A
Individual Tests (per applicable paragraph)	X	X	X	X	X	X
Low Temperature (4.3.2.1.3)	X	X	X		X	X
Altitude (4.3.2.1.4)					X	X
Humidity (4.3.2.1.5)					X	X
Shock (4.3.2.1.9)	X	X	X	X	X	X
Vibration (4.3.2.1.10)	X (see note 1)	X	X	X	X	X
Radio Interference (4.3.2.1.12)	X	X	X		X	X
Voltage Stability (4.3.2.1.13)	X	X	X		X	X
High Potential (4.3.2.1.14)	X	X	X		X	X
High Temperature (4.3.2.1.15)	X		X		X	X

Note 1: The GG75A Accelerometers will be vibrated at 5g operating and 10g non-operative.

Very truly yours

C S Herrala
Project Coordinator

R G Patterson
Autopilot Design JRM

M D Peters
Component Design

AVRO AIRCRAFT LIMITED

BOX 4004
TERMINAL 'A'
TORONTO. ONT.

July 9th, 1957.
Reference No. 9301/02E/J

Minneapolis-Honeywell Regulator Co.,
Aeronautical Division,
MINNEAPOLIS, Minnesota.

Attention: Mr. R. G. Patterson,
Project Engineer,
Autopilot Design.

Copy: Mr. C. S. Herrala,
Project Coordinator,
Service Engineering Dept.

Via: Mr. P. J. Nuss,
Minneapolis-Honeywell Regulator Co.,
Vanderhoof Avenue,
LEASIDE, Ontario.

Subject: PROPOSED QUALIFICATION SPECIFICATION M-H 64 DAMPER

Gentlemen:

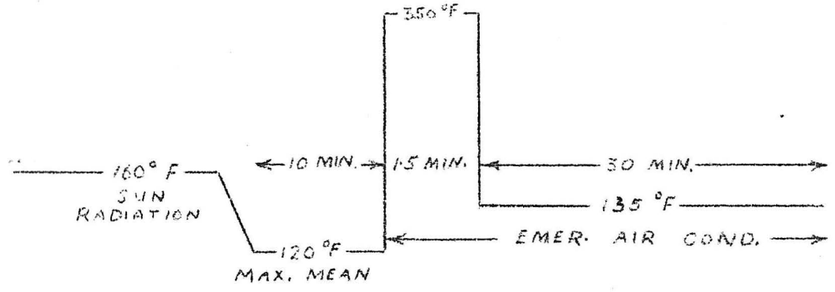
Your preliminary test specification for the MH-64 Damper (CEL-105-109) has been considered; our comments below are numbered as per your specification.

Sand, dust and fungus tests will be required, as per MIL 5272 procedure 1 for sand and dust and amendment 1 for the fungus tests.

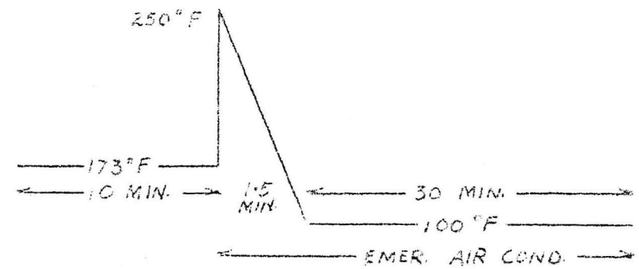
4.1 High Temperature

(a) Procedure I MIL-E-5272 for the Function Selector.
Comment: 160°F is high for cockpit operations. The unit should be soaked at 160°F and then operated at a lower temperature (90 to 100°F).

(b) Procedure I MIL-E-5272 at 160°F for 20 hours is satisfactory but additional cycles should be done as outlined below:



(c) Procedure I MIL-E-5272 at 220°F for 10 hours and additional cycles to the following schedule.



4.2 Low Temperature-
Procedure II Mil-E-5272
Comment: Procedure I is more applicable to our requirement ie. soak at -65°F not -80°F. Operation at room temperature is satisfactory.

4.3 Humidity
Procedure I MIL-E-5272 for 10 cycles
Comment: Satisfactory

4.4 Altitude
Procedure III MIL-E-5272 at 60,000 feet and + 70°F for ten hours
Comment: Satisfactory

4.5 Salt Spray
MIL-E-5272
Comment: Satisfactory

4.6 Vibration
Procedure I MIL-E-5272 for non-shock mounted units and for shock mounted units on shock mounts.
Comment: Satisfactory except that vibration should be conducted in accordance with the temperature schedule given in the MIL-spec.

4.7 Rain
Procedure I MIL-E-5272
Comment: Unnecessary

4.8 Explosion Proof
Procedure I MIL-E-5272
Comment: Satisfactory

4.9 Shock-
Procedure II MIL-E-5272
Comment: Satisfactory

4.10 Radio Interference
MIL-I-6181 dated 14 June 1950
Comment: Satisfactory.

4.11 Voltage Stability
MIL-E-7894 for a type I power source. The system shall not be required to engage below 21 volts D.C. and shall not be required to remain engaged for direct current transients below 10 volts.
Comment: Satisfactory

4.12 High Potential
Comment: Satisfactory

Yours very truly,

AVRO AIRCRAFT LIMITED,

C.S.R. Marshall
Technical Coordinator.

F. H. Brame
Asst. Chief of Technical Design.

R.L.P./np

Comments on the Environmental Tests proposed in MH letter 5th April 1957. C-E-L N. 6-105-109

4.1. (a) Procedure I of MIL-E-5272 calls for soaking at +160°F.

Comment. This is high for cockpit operation. The unit should be soaked at +160°F and then operated at a lower temperature (say 90 to 100°F)

(b) Soak at 160°f and then cycle.

Comment 160°f is OK but I question the 220°f in view of the flight envelope.

(c) Likewise as in (b) above. The 250°f soak for 15 minutes is excessive; could be 10 minutes.

4.2. Procedure II of MIL-E-5272.

Comment Procedure I is more applicable to our requirement is, soak at -65°f, not -80°f

Operation at room temp is satisfactory.

4.3. O.K.

4.4. Altitude. 60,000 feet and +70°F

Comment. The normal test requirements are 60000 ft and -65°f, but if it is considered that air conditioning will provide the necessary differential then +70°f would suffice.

4.5. OK

4.6 OK. except that vibration should be conducted in accordance with the temperature schedule given in the MIL-Spec.

4.7. Unnecessary

4.8 OK

4.9. OK.

4.10. O.K.

Aeronautical Division

C.E.L. NO. C-105-267
DATE January 14, 1958

CUSTOMER ENGINEERING LETTER

Mr Guest Hake
Project Designer
AVRO Aircraft Limited
Box 4004, Terminal A
Toronto, Ontario, Canada

MH-64 DAMPER; ARROW
LIFE TESTS OF MH-64 DAMPER COMPONENTS

Gentlemen

During theforthcoming Coordination meeting at AVRO on January 21, 22, MH plans to discuss the subject matter as one of the agenda items.

We feel that the subject matter is of sufficiently significant importance that we attach a summary of our conclusions and recommendations for your consideration in advance of the meeting.

Very truly yours

C S Herrala
Project Coordinator

R G Patterson
Autopilot Design

cc C N Lucas
AVRO Purchasing

Attachment to CEL C-105-267
dated January 14, 1958

LIFE TESTS OF MH-64 DAMPER

BACKGROUND

At the Specification Meeting, December 9 - 10, AVRO expressed a desire to include life tests as a part of the qualification program on the MH-64 Damper. This qualification program is contained in R-ED-889 and was originally proposed in CEL C-105-105 and later modified per AVRO request in CEL C-105-231. MH appreciates the need for such tests. It is the purpose of this paper to present recommendations for such a program.

In order to plan for these tests, let us examine the objectives to be achieved.

1. Uncover deficiencies in design.
2. Assure that the life of the system is adequate.
3. Assure that the mean time to failure is adequate for the mission of the aircraft.
4. Determine which components if any have a "wear out" time shorter than the design objective life of the system.
5. Assure maintenance of quality of manufacturing on a life test sampling basis.

Since most of these objectives require redesign if the equipment does not meet the requirements it is desirable to complete the initial tests as soon as possible in order that any changes can be made prior to production. It appears likely that delivery requirements on the ARROW MH-64 Damper program will dictate production build-up immediately following, if not concurrent with, the production qualification tests. Therefore if life tests are included as part of the qualification program, it will be impossible to make the design changes required effective in the first production units. This would result in:

1. Lowered reliability of the first block of systems.
2. Complex retrofit programs in the factory and field.
3. Late deliveries
4. An end product which would be more difficult to maintain and service due to the changes made after the device was released to production.

RECOMMENDATIONS

In light of these considerations, MH recommends the following three fold program.

1. Concurrently with the development program begin life tests on typical components which are essentially in their final form and on sub-components which have not previously been life tested on other programs.

2. Life test all components as soon as they are ready for production release. These tests to be included as part of the qualification program.

3. Conduct life tests on a sampling basis during production.

All of these life tests would be performed on the components individually, at room conditions. The duration of the tests would be specified to be equivalent of 1000 hours of operation of each unit. It would be desirable to conduct some tests on an accelerated basis with either more rapid cycling or more sever temperature conditions than would be experienced in actual flight. Mutually acceptable tests of this kind could be agreed upon.

In the first phase of this program it is suggested that the following components and subcomponents be tested:

Quantity	Device No.	Description
1	DVG17C-1	Valve, Control, Elevator
1	DMG55N-3	Servo, Parallel, Elevator
1	DMG62B-2	Servo, Differential, Rudder
1	DGG72A-2	Rate Gyro, Roll
1	AD463550	Fader
1	AD463549	Signal Control Trim
1	DEG129A-1	Amplifier, Magnetic
1	DPG29A-1	Differential Pressure Transmitter
1	DPG32A-1	Compensator
1	DGG58B-1	Accelerometer, Lateral
1	DGG62B-1	Accelerometer, Lateral Switching

It is suggested that AVRO request a proposal to cover the first phase of this program at once and that they request a requote on the qualification program including life tests. The third phase should be included in the production program.

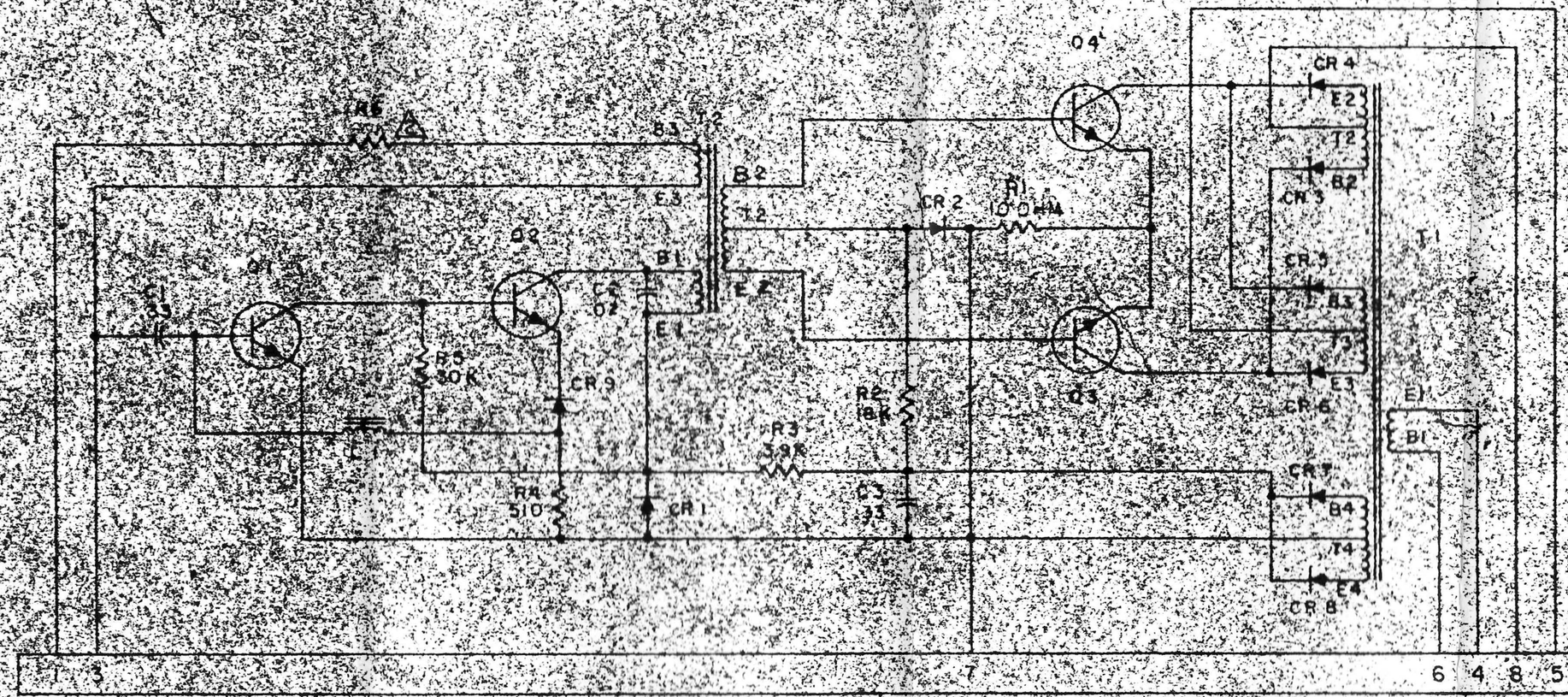
8 67693

SUPPLEMENTARY INFORMATION

PURCHASE DE 6113D-1

REFERENCE DE 6113B-

SYM	DESCRIPTION	DATE	APPROVAL
A	R6 WAS 90K, R1 WAS R1A PER ECS	7-22-57	9A-1172
B	REM AD FROM BW& NO.	11-14-57	9A-1327



D

TOLERANCE UNLESS NOTED		DRAWN BY Diamant 1-7-57
ONE PLACE (.X) ±2%		DRAFTING Sanford 1-15-57
TWO PLACE (.XX) ±1%		MATL SPEC
THREE PLACE (.XXX) ±.5%		DESIGNER BCH 5-20-57
30' FORMED ANGLES ±1°		DESIGN APPROVED [illegible] 5-22-57
DEV 01196-22		MFR MH MATL SPEC
SUPSO		
GOVT MATL SPEC		
FINISH SEE NOTE	SCALE NONE	WT ACT CALC LB

SCHEMATIC DIAGRAM — TRANSISTORIZED PREAMPLIFIER, FULL WAVE DEMODULATOR

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

AC6949

2 CALIBRATION RESISTOR, SELECTED VALUE

1- ALL RESISTANCES ARE IN OHMS AND ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

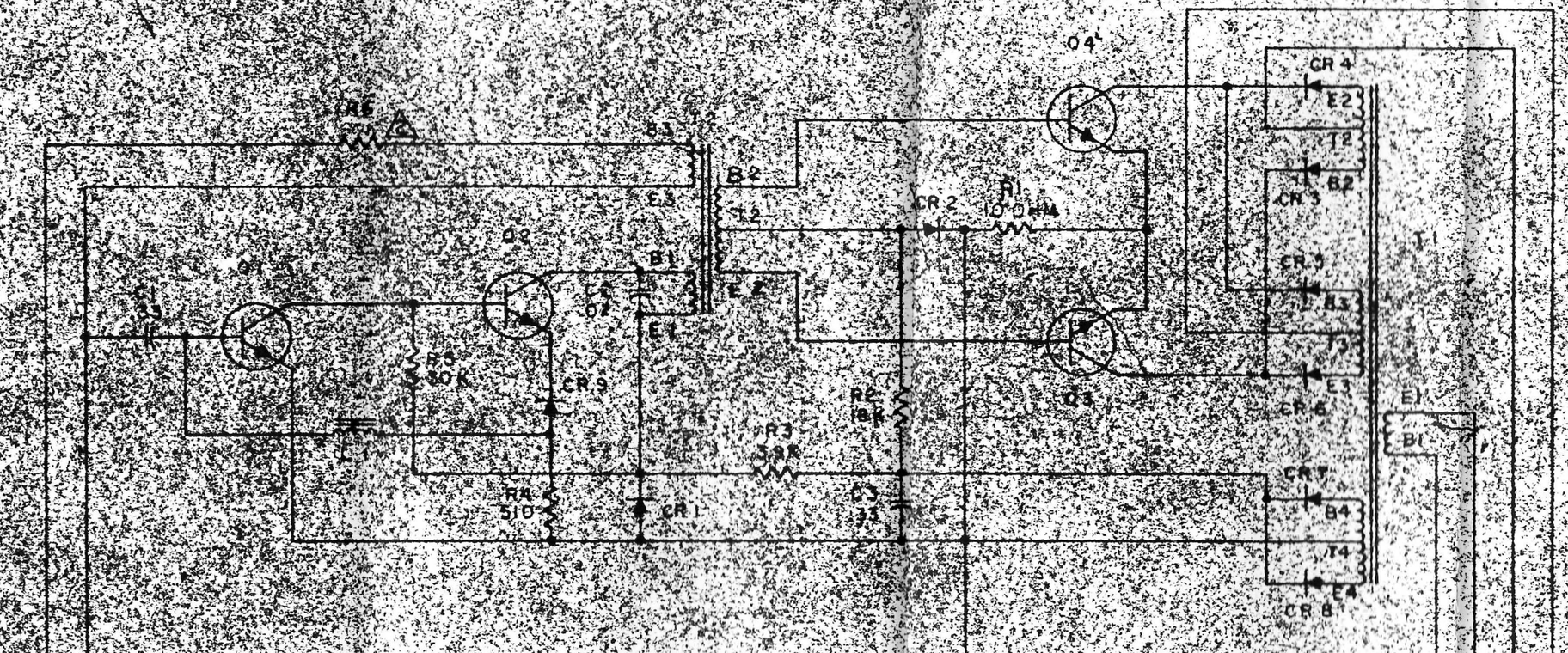
8 67693

SUPPLEMENTARY INFORMATION

PURCHASE ORDER 61130-1

REFERENCE DRG 61138

SYM	DESCRIPTION	DATE	APPROVAL
A	R6 WAS 90K, R1 WAS R1045	7-22-57	9A-1172
B	REM AD FROM BWG NO.	11-14-57	9A-1327



ELECTRONIC INDUSTRIES
60GS/90W-HS-BA

D

2 CALIBRATION RESISTOR, SELECTED VALUE

1 ALL RESISTANCES ARE IN OHMS AND ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

TOLERANCE UNLESS NOTED	
ONE PLACE	.0 ± .1
TWO PLACE	.00 ± .01
THREE PLACE	.000 ± .005
IN FORMED ANGLES	± 1°
DEV 01196-22	
SUPSD	
GOVT MATL SPEC	

DRAWN BY	Diehart 1-7-57
DRAFTING	Sanford 1-15-57
MATL SPEC	
CHECKED	6CH 5-20-57
DESIGN APPROVED	5-22-57
REF MN MATL SPEC	
FINISH SEE NOTE	

SCHEMATIC DIAGRAM — TRANSISTORIZED PREAMPLIFIER, FULL WAVE DEMODULATOR

SCALE NONE | WT ACT CALC LB

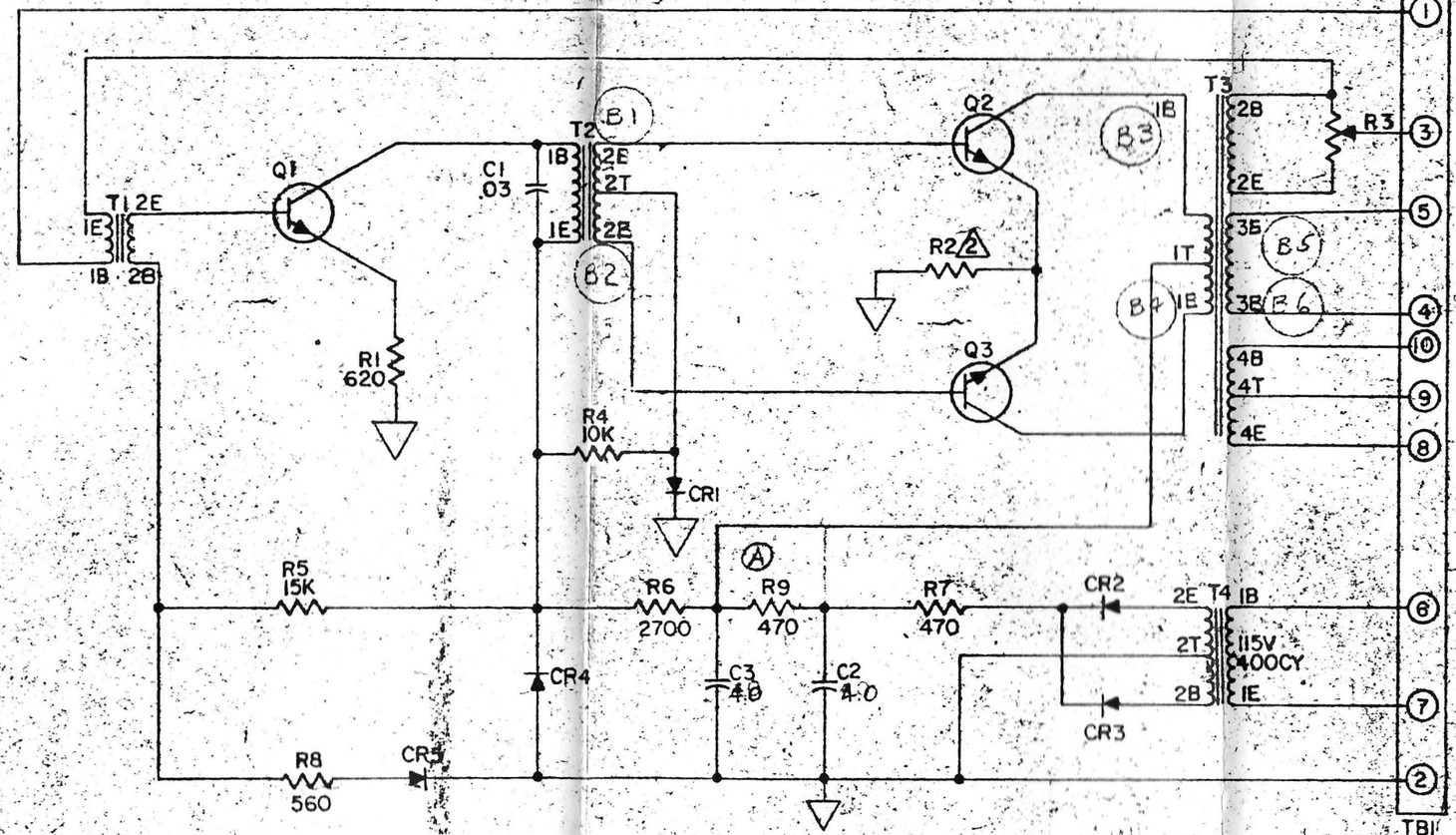
MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

AD C6949

C7554

SUPPLEMENTARY INFORMATION	
FIRST USED EG153A-1	
NEXT ASSEMBLY	REFERENCE
EG 153A	

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL
A	REM A.D FROM DWG NO.	11-11-57	9A-1327



D-

3-▽ DENOTES SIGNAL GRD

2-△ VALUES TO BE SELECTED

1-ALL RESISTANCES ARE IN OHMS AND ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
30° FORMED ANGLES		±1°

DEV 01196-25

SUPSD

GOVT MATL SPEC

DRAWN BY	Seven 5-2-57
DRAFTING	J Dorsey 5-21-57
MATL & FIN	
DESIGNER	
DESIGN	J Dorsey 5-21-57
APPROVED	W.K 5-6-57
REF MH MATL (SPEC	
FINISH (SEE NOTE	

PREAMPLIFIER-
ELECTRONIC CONTROL

SCALE NONE | WT ACT CALC LB

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

C7554

C6462 A

SYM	DESCRIPTION	APPROVED
A	ADD C5, C6, R9, & R10	12-6-57 9A-1366

SUPPLEMENTARY INFORMATION

EQ129A

R6
90.9K

T2

B3

E3

B2

T2

E2

B1

SELECTED R8

C2 .1

C6 .005

E1

Q1

Q2

Q4

Q3

CR2

R1
SELECTED

C1
.33

R5
30K

CR9

L1

R4
510

R2
24K

R3
3.9K

C3
.33

CR1

CR4

CR3

CR5

CR6

CR7

CR8

T1

B1

E1

B4

T4

E4

C4 .05

R9
1.5K

R7
5.1K

C5
.15

R10
1.5K

CR10

CR11

CR12

CR13

J1
HEADER
424075-34

8 6 9 2 5 3 4 1

SIGNAL

GAIN ADJ

VEL. GEN.

SIG. GR.

PWR. GR.

115V, 400~

OUTPUT

OUTPUT

D

1-ALL RESISTANCES ARE IN OHMS & ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

TOLERANCE UNLESS NOTED

01196-19

Rasmussen 12-22-56

2448 4-12-57

4-15-57

SCHEMATIC DIAGRAM — TRANSISTORIZED, MAGNETIC AMPLIFIER

NONE

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

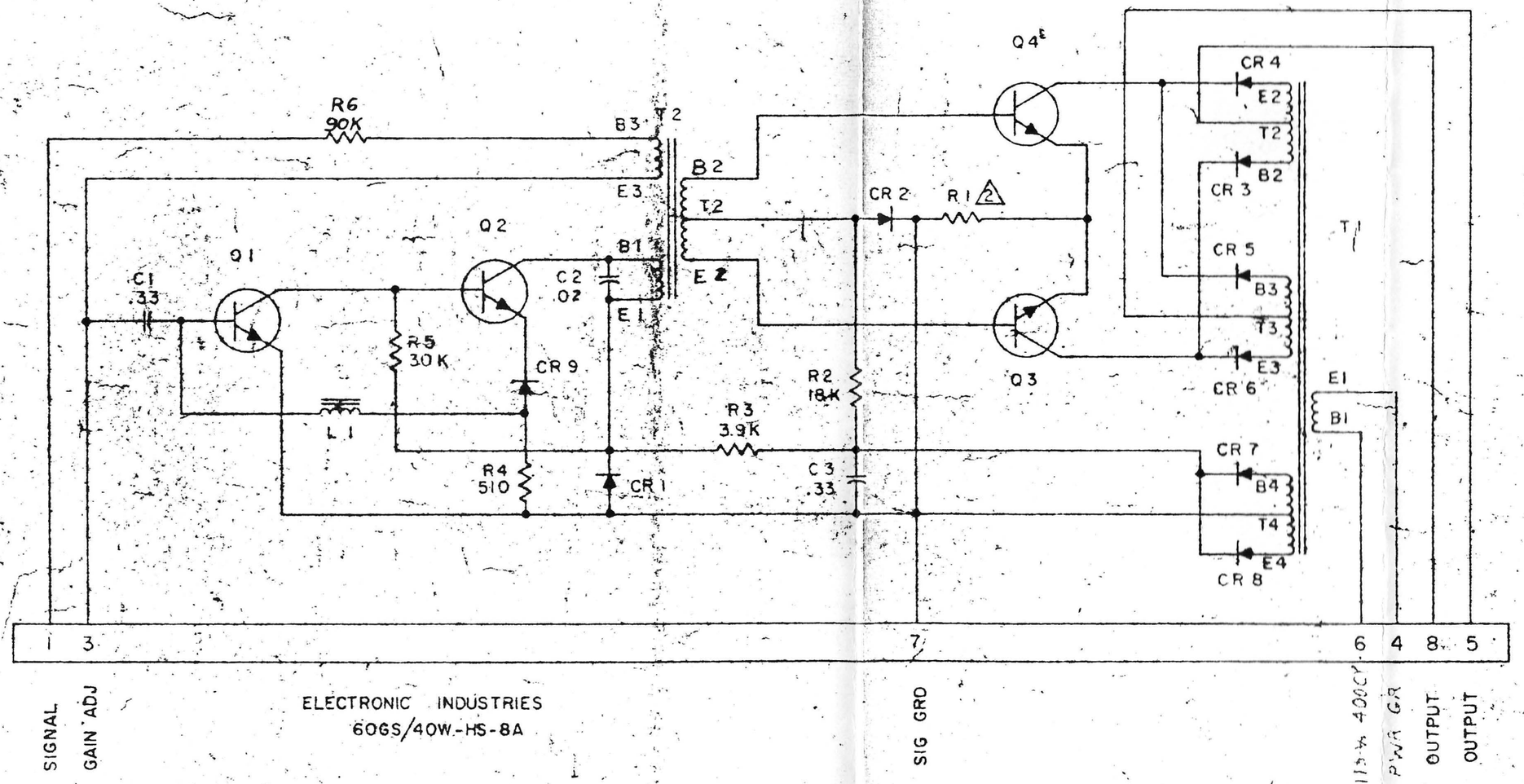
C6462

AD C6626 B

SUPPLEMENTARY INFORMATION	
FIRST USE	DEG1138-1
REFERENCE	
DEG1138-1	

REVISIONS

SYM	DESCRIPTION	DATE	APPROVAL
A	IN J1, 6 WAS 4, 4 WAS 6; 8 WAS 5, & 5 WAS 8	4-13-57	9A-1086
B	REM AD FROM DWG NO.,	11-11-57	9A-1327



ELECTRONIC INDUSTRIES
6065/40W-HS-8A

2 CALIBRATION RESISTOR; SELECTED VALUE

1- ALL RESISTANCES ARE IN OHMS AND ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	.020
TWO PLACE	(.00)	.010
THREE PLACE	(.000)	.005
90° FORMED ANGLES		1°

DEV 01026-06

SUPSD

GOVT MATL SPEC

DRAWN BY	Diethert 1-7-57
DRAFTING	Sanford 1-23-57
MATL & FIN.	
DESIGNED	[illegible] 1-21-57
DESIGN APPROVED	Storm 1-24-57
REF MH MATL ISPEC	
FINISH (SEE NOTE)	
SCALE	NONE
WT	ACT / CALC LB

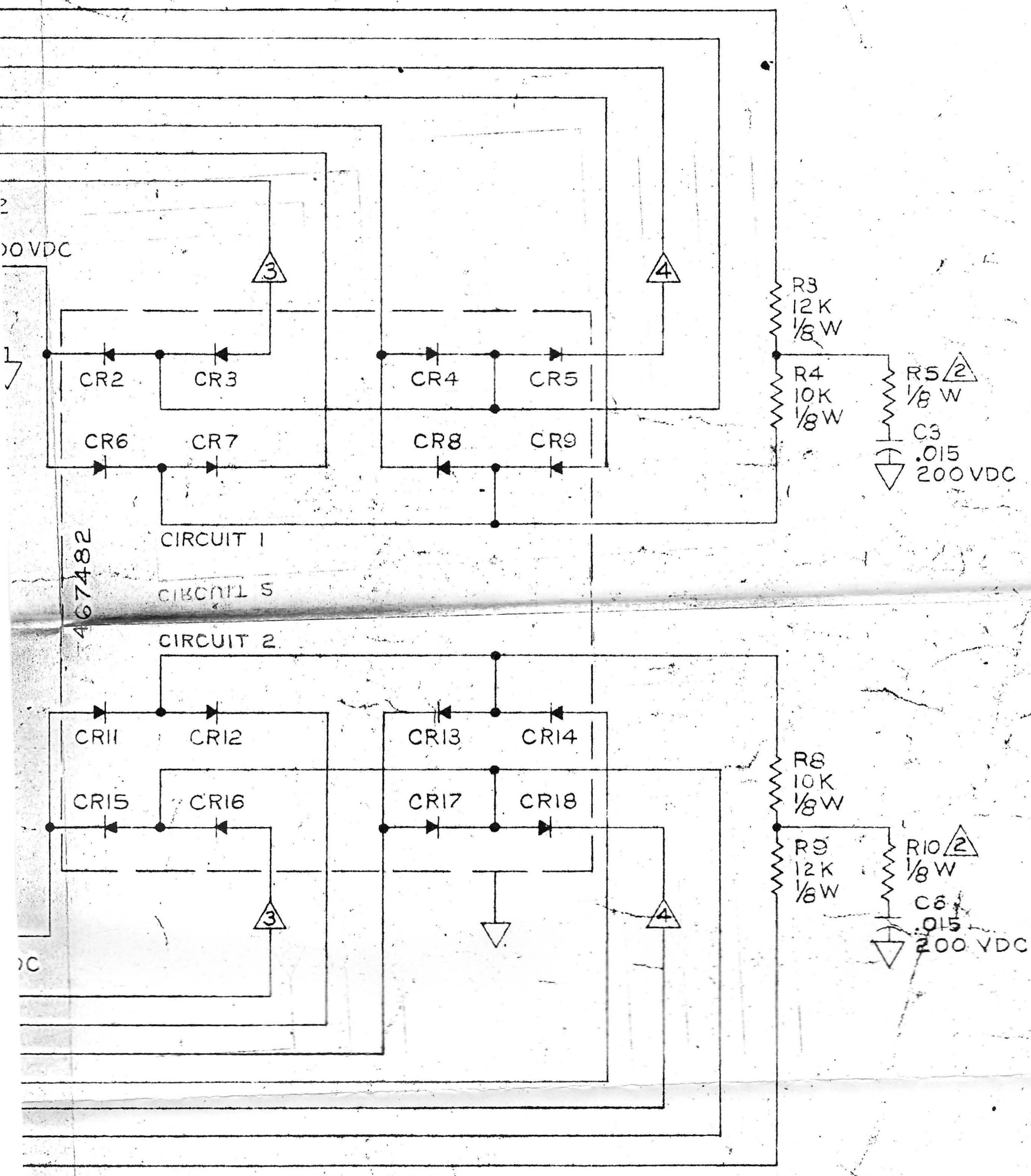
SCHEMATIC DIAGRAM — TRANSISTORIZED PREAMPLIFIER, FULL WAVE DEMODULATOR

MINNEAPOLIS-HONEYWELL REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS MINNESOTA

C6626

SUPPLEMENTARY INFORMATION		
FIRST USED D-AG26A-1		
REFERENCE		
AG 26A-1	S1 THRU	C7632

REVISIONS			
SYM	DESCRIPTION	DATE	APPROV



467482

TOLERANCE UNLESS NOTED		
ONE PLACE	(.0)	±.020
TWO PLACE	(.00)	±.010
THREE PLACE	(.000)	±.005
90° FORMED ANGLES		±1°

DEV 01196-31	
SUPSD	
GOVT MATL SPEC	

DRAWN BY	D.S. 6-27-57
DRAFTING	7-22-57
MATL & FIN.	
DESIGNER	
DESIGN	7-22-57
APPROVED	9-19-57
REF MH MATL (SPEC)	
FINISH (SEE NOTE)	

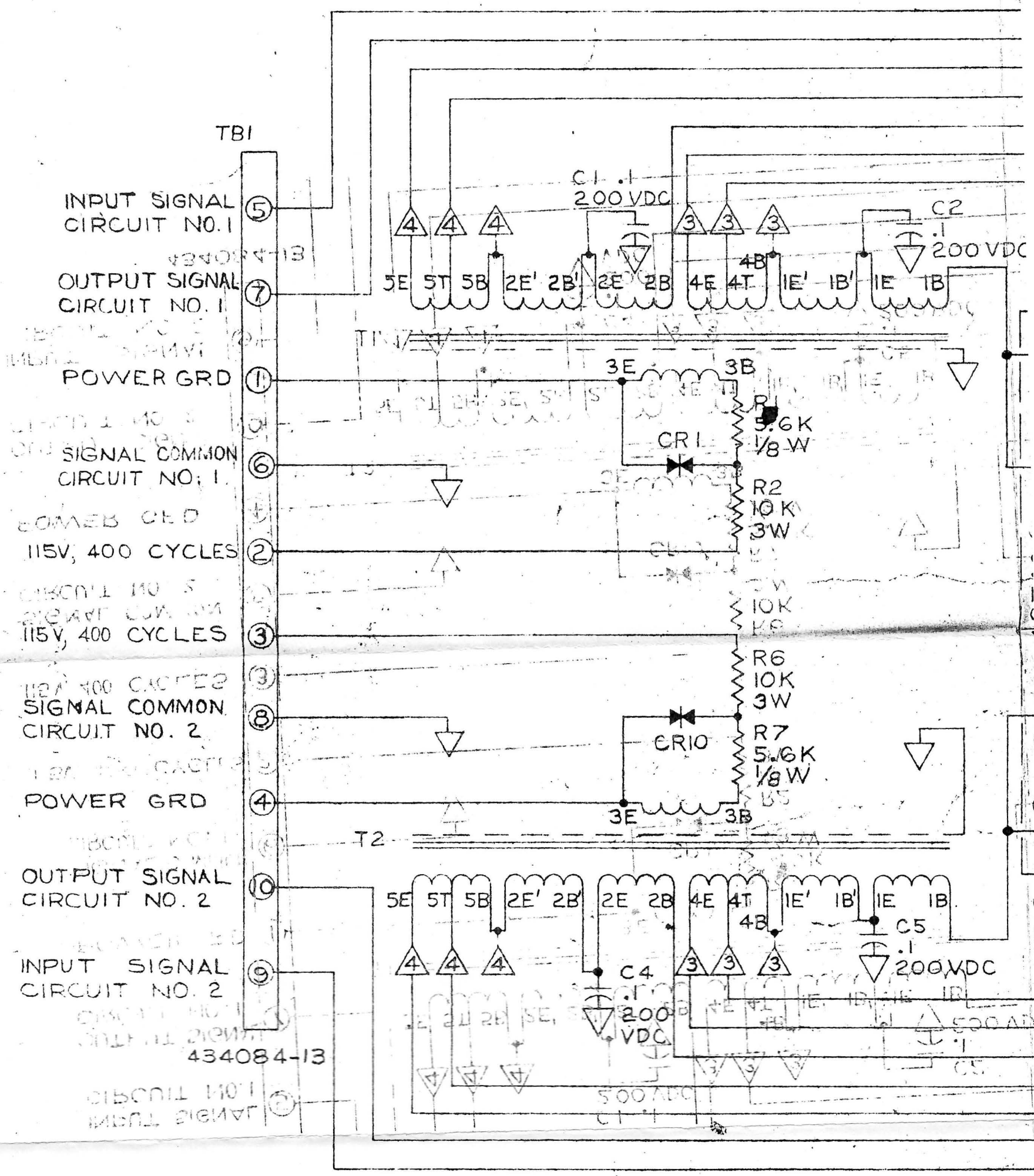
SCHEMATIC DIAGRAM—
FILTER, QUADRATURE
SIGNAL

SCALE NON WT ACT. CALC LB

MINNEAPOLIS-HONEYWELL
REGULATOR COMPANY
AERONAUTICAL DIVISION
MINNEAPOLIS, MINNESOTA

C7632

D-



5- ▽ DENOTES SIGNAL GRD

4- ANODE OF CR9, CR14 AND CATHODE OF CR5 & CR18 CONNECTED TO SELECTED TAP OF 5B2E', 5T OR 5E IN ACCORDANCE WITH MH SPEC

3- ANODE OF CR3, CR16 AND CATHODE OF CR7 & CR12 CONNECTED TO SELECTED TAP OF 4B1E', 4T OR 4E IN ACCORDANCE WITH MH SPEC

2- CALIBRATION RESISTOR; SELECTED VALUE

1- ALL RESISTANCES ARE IN OHMS & ALL CAPACITANCES ARE IN UF UNLESS OTHERWISE SPECIFIED