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DEVELOPMENT OF A SERVO CONTROL SYSTEM FOR GUIDED MISSILES

OSRD Report No. 6418
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Auspices of Division 5, NDRC
Contract OEMsr—1013
Responsive to Navy Directive NO—115
Responsive to Army Directive AC—36

Servomechanisms Laboratory
Division of Industrial Cooperation
Massachusetts Institute of Technology
Project DIC 6168

October 31, 1945
Report Covers Period from March 15, 1945 to October 31, 1945

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October 31, 1945

Report Approved by Mr. H. H. Spencer, Division 5, NDRC, as of 4/15/46.

Distribution Authorized by Mr. Roger Cutting, Division 5, NDRC, as of 5/20/46.

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DEVELOPMENT OF A SERVO CONTROL SYSTEM FOR GUIDED MISSILES

Servomechanisms Laboratory
Massachusetts Institute of Technology
Cambridge, Massachusetts

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FOREWORD

This report, describing a control system for a guided missile of the glider type known as "Bat" is supplementary to a report covering the development of a similar control for a smaller missile known as "Pelican". The development methods are explained, the system devised is described, and the reasons for the choice of the system are given.

The development program culminated in a series of flight tests on a ship target, records of which are included in the report. These tests demonstrated that the servo system was capable of controlling the missile in a rapid and accurate manner.

A production design of the control system undertaken by Bell Telephone Laboratories, Inc., is also briefly described.

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1.0 INTRODUCTION

1.1 History and Objectives

In the spring of 1943 the Servomechanisms Laboratory, Massachusetts Institute of Technology, was requested to develop a control system for a guided missile of the homing type already being developed. The guided missile comprised a glider of heavy wing loading designed by the Bureau of Standards under Section 5.1 of NDRC, and an S-band radar receiver developed by the Radiation Laboratory, M.I.T. After a brief survey of the problem, development was begun at the Servomechanisms Laboratory in June of 1943. The general objective was to develop the simplest possible control system that would make the most extensive use of the capabilities of the glider and the homing intelligence.

Several sizes of gliders had been built or designed by the Bureau of Standards at the time the development of the control system was undertaken at M.I.T. The smallest of these gliders, known as Pelican, has an eight-foot wing span and is used both as a weapon and to provide field proofs of designs intended finally for larger size gliders. The initial program was to develop a control system for the eight-foot glider, with the possibility of adapting it to larger size units if flight tests were successful. The development of the servo for the eight-foot glider was completed in the latter part of 1944, and flight tests were made in February 1945. This work has been covered by "Development of the Servo Control System for Guided Missiles Preliminary Report", distributed by Division 5, NDRC.

The success of the work done on the control system for Pelican lead to the recommendation that the same type of control system be designed for a larger missile of the same general appearance known as Bat. Bat is a ten-foot glider-type missile carrying a 1000-pound bomb and with an S-band transmitter-receiver located in the nose of the missile to supply homing information. The development of the control system was carried through field tests and supplied assistance on the development of a production model. This report, which is the final report on Contract OEMsr-1013, covers the theory underlying the operation of the control system that was developed for Bat, detailed descriptions of the system and its components, a summary of the flight test results, and a brief description of the production unit that resulted from the development. This report is not intended to be complete in itself. The Preliminary Report preferably should be read first since many of the points covered in that report are not covered in the Final Report.

1.2 Summary of Results

Development of the control system for the ten-foot glider known as Bat was begun in March of 1945 and culminated in a series of flight tests in June, July, and August of 1945. These tests corroborated the tests made on the lighter and slower Pelican and proved that rapid and accurate control of this type of missile could be obtained with a relatively simple control system.

The control system designed for Bat was designed to be as close to the final production unit as possible. Since it was desirable to have field tests of these units at as early a date as possible, certain modifications in the design were employed to permit

the rapid manufacture of a small number of units. Ten units were prepared for field tests, three of which were dropped on a ground target and six of which were dropped on a ship (tanker) target. Results of these tests are covered fully in Chapter 6. Briefly, however, the results were as follows: Of the three units dropped on a ground target comprising a corner reflector approximately twenty feet on a side, one malfunctioned due to mechanical failure and flew only for a few seconds, one landed approximately sixty feet in front of the target and the other struck approximately seventy feet behind the target. Of the six units that were dropped on the ship target, one hit the ship approximately amidships and exactly at the water line, a second hit the target approximately amidships two feet above the water line, and a third flying in a thirty mile per hour cross wind hit the water only a few feet short of the bow. These three were classed as direct hits. The fourth struck the water approximately forty feet short of amidships and skipped into the ship. A fifth hit the water approximately fifty feet short of amidships and skipped into the ship. These two were classed as skip hits and would undoubtedly have caused damage although they might not have sunk the ship. The sixth passed over the ship approximately thirty feet high and was classed as a miss.

These tests were considered very successful.

1.3 General Description of Bat

1.3.1 The Air Stabilizer (Glider)

The glider (termed an air stabilizer by the Navy) was specifically designed to carry a heavy load with small wing area and to possess those aerodynamic properties necessary for its accurate and rapid control. Several sizes of air stabilizers have been developed. All are characterized by the fact that neither rudder nor elevator is used for control. Photographs of the ten-foot air stabilizer, known as Bat, are shown in Figures 1.1 and 1.2, and a pictorial of component location is given in Figure 1.3.

Instead of using conventional control surfaces, the glider is controlled entirely by what have been termed elevons. These are movable surfaces on the trailing edge of the wing, so designed that (1) they can be moved differentially as a conventional aileron is moved and (2) they can be moved up and down together in order to vary the lift characteristics of the wing. A differential displacement of the elevons rolls the glider just as ailerons roll a conventional airplane. A change in the average position of the elevons results in a change in the lift characteristics of the wing and changes the angle of the glide path of the glider. When the glider is rolled to an angle with the horizontal the resulting component of lift acting crosswise to the flight path results in a rate of yaw, and changes the directional heading of the glider. Thus the orientation of the glider in space is effected completely by the elevons alone.

The choice of elevons for controlling the air stabilizer was made for the following reasons:

1. Simplicity in manufacture of the air stabilizer is obtained.
2. Certain simplifications in the control mechanism are secured.
3. The glider can be so designed that the angle of attack remains relatively independent of the average position of the elevons.

The angle of attack is the angle between the direction in which the glider is pointing and its actual flight direction in space. In order to secure accurate results the homing intelligence must be boresighted in the direction in which the glider is traveling. The homing intelligence may be boresighted properly provided the angle of attack is known accurately. If the angle of attack should change appreciably during flight, the angle at which the homing intelligence is set with respect to the axis of the glider would have to be altered. When controlled by elevons the air stabilizer maintains a constant angle of attack during flight and therefore the homing intelligence can be boresighted at a fixed angle with respect to the glider before flight. If the homing intelligence is boresighted improperly, excessive acceleration of the glider is required toward the end of its flight, which considerably reduces the accuracy.

Unfortunately, the use of the elevons to control the glider possesses certain disadvantages:

1. The power required to control the vertical motion of the glider is relatively large. The power is required first by the high speed with which the elevons must be moved in order not to limit the maneuverability of the glider, and second, by the large hinge moments encountered in flight produced by the large angles through which the elevons must be displaced to secure the necessary changes in lift.

2. Certain adverse yawing moments are developed by moving the elevons differentially about an average elevon position different from the streamline position. This adverse yawing moment is caused by the fact that the drag on the air stabilizer becomes asymmetric as the elevons are displaced differentially about an average position which is different from the streamline position and a yawing moment is developed. Since no rudder is available on the glider it is not possible to supply a moment to counteract this adverse yawing moment.

3. Certain adverse pitching characteristics arise from the fact that when the elevons are suddenly displaced to a new average position the first effect on the glider is opposite to its final and desired effect. This adverse pitching effect arises from the fact that when the position of the elevons is changed rapidly the line of downwash on the tail stabilizer is altered and the elevons tend to act as elevators instead of elevons. For example, if the elevons are moved upward, a moment is produced tending to raise the nose of the glider while the elevons are in motion. This moment is soon counteracted by a more powerful moment tending to lower the nose of the glider in the desired direction. While the first effect lasts only a short time it appreciably reduces the maneuverability and controllability of the glider.

The effect of these adverse yawing and pitching moments is analyzed in Chapter 2 of this report covering the theory of operation.

1.3.2 Homing Intelligence

While several types of homing intelligence can be used in the air stabilizer, the particular one used in Bat is an S-band transmitter-receiver manufactured by the Western Electric Company and denoted as SWOD (Radar) Mk II Mod 0. The target is illuminated by the transmitter and the receiver obtains information as to which quadrant the receiver is in with respect to the axis of the receiving antenna, by offsetting the antenna and revolving it about the axis from which it was offset. The signal intensity from the

dish when pointed to the right is compared with the signal intensity when the dish is pointed to the left. The up and down signals are similarly compared. If the missile is traveling in the correct direction, the target will be located on the axis about which the antenna is revolving and no difference in intensities will exist between the right and left antenna positions nor between the up and down positions. If the missile is not aimed correctly a signal will be obtained indicating the direction and magnitude of the error in heading. This signal can be used to correct the flight of the missile.

Should the path of the missile suddenly change from its correct path, the signal indicating the error in heading will not appear instantly but will lag the motion of the glider. This delay is partly due to inherent lags in the receiver but mostly due to the fact that a certain amount of averaging of the signal must be employed because of its high noise content. This averaging, or filtering, is necessary to prevent the servo from reacting to false signals. If the filtering is excessive, the lag becomes too large and loss in servo control is encountered. The servo system must be designed to be as insensitive to lag as possible and in general must be designed to match the characteristics of the radar receiver. A photograph of the radar receiver and antenna attached to the front of a ten-foot air stabilizer is shown in Figure 1.2.

The particular design of the radar set made it very difficult to obtain reliable directional information closer than about twelve hundred feet from the target depending upon the type and size of the target. A revision of the set by the M.I.T. Field Experiment Station at the Bureau of Standards, reduced this minimum range to about four hundred feet. This figure appeared to be satisfactory for the type of operations for which Bat was intended. All field tests were made with the revised set.

1.3.3 Servo System

The servo control system for the missile must be designed to make full use of the aerodynamic properties of the missile and its controlling elevons. At the same time the control system must minimize certain of the adverse effects that accompany elevon control. In addition, the control system must make the most of the characteristics of the homing intelligence. Finally, the control system must be as simple as is commensurate with the above requirements.

The servo system was designed to obtain an inherently stable, accurate, and fast system. It was felt that such a system would require a minimum of compensating devices and would, therefore, prove most satisfactory. The principle of the system operation has been described in detail in the Preliminary Report. However, a brief description of of the principles of operation of the servo system is given in this report.

The control system was standardized as the SWOD (Control System) Mk 18 Mod 0. A schematic of this system illustrating the principles of operation is shown in Figure 1.4. The system employs a Sperry Mk 4 Bank and Climb Control Unit (vertical gyro) as a stabilizing instrument, a pair of specially developed clutch units as servomotors, electromechanical trim units for transforming electrical directional information into mechanical motion, and associated gear trains. A pictorial view of the unit is shown in Figure 1.5. The Sperry Mk 4 Bank and Climb Control Unit, as modified for the SWOD (Control System) Mk 18 Mod 0, includes an air driven gyro, rotary air valves coupled to the gimbals to measure roll and pitch motion, and a motor unit for remote caging and

uncaging the gyro. The air valve consists of a pilot (see Figure 1.4) which is coupled to the appropriate gyro gimbal and a sleeve surrounding the pilot which is driven by external means. The pilot of the valve measuring the roll position is geared directly to the roll gimbal and the pilot of the valve measuring the pitch position is connected to the pitch gimbal through a linkage system. If the pilot and sleeve of either valve are not aligned, a pressure differential is established across a corresponding air relay. The air relay is a pressure operated electrical switch comprising a metallic diaphragm with an electrical contact mounted at its center and enclosed in a sealed housing. The air relays are mounted in the housing located at the rear of the vertical gyro. (See Figure 1.5).

The sleeves in each air valve are driven by the spiders of differential gears located in the roll and pitch trim units. One side of each differential is geared to the elevon system and the second side is geared to the appropriate trim units. The gyro, air valves, and air relays are described in Chapter 4.

The clutch servomotor comprises a double-ended, multiple-disc, magnetically-actuated clutch. A cutaway of the clutch is shown in Figure 1.6 and an exploded view is shown in Figure 4.4.1. Detailed description of the clutch unit is given in Section 4.4.

The operation of the system can be described considering its operation under two conditions: (1) glider operation in the absence of homing signal but in the presence of disturbing air effects and (2) glider response to homing signals but with no interfering air disturbances. It is of help to refer to Figure 1.4 in following this description.

Should the glider receive a roll disturbance, the pilot of the roll valve is moved relative to the sleeve. The pressure differential is applied to the roll air relay, the clutch actuating magnet is energized, and the clutch is engaged in the appropriate direction. As the elevon differential position is changed by the clutch, its motion is fed back through the differential and the roll trim unit to the sleeve of the roll air valve in such direction as to cause the sleeve to follow the motion of the pilot. When the air valve is again in alignment, the roll air relay opens or reverses and the clutch of the elevon differential drive shaft disengages or reverses as the case may require. Thus the change in elevon differential position is made proportional to the change between the roll position of the glider as indicated by the pilot valve and a roll trim position as controlled by the position of the roll trim unit. The control in pitch is exactly similar to the control in roll.

When the glider is operating under the influence of homing signals, the roll and pitch references of trim angles are controlled by the output of the radar receiver. The manner in which this control is effected is described as follows (refer again to Figure 1.4):

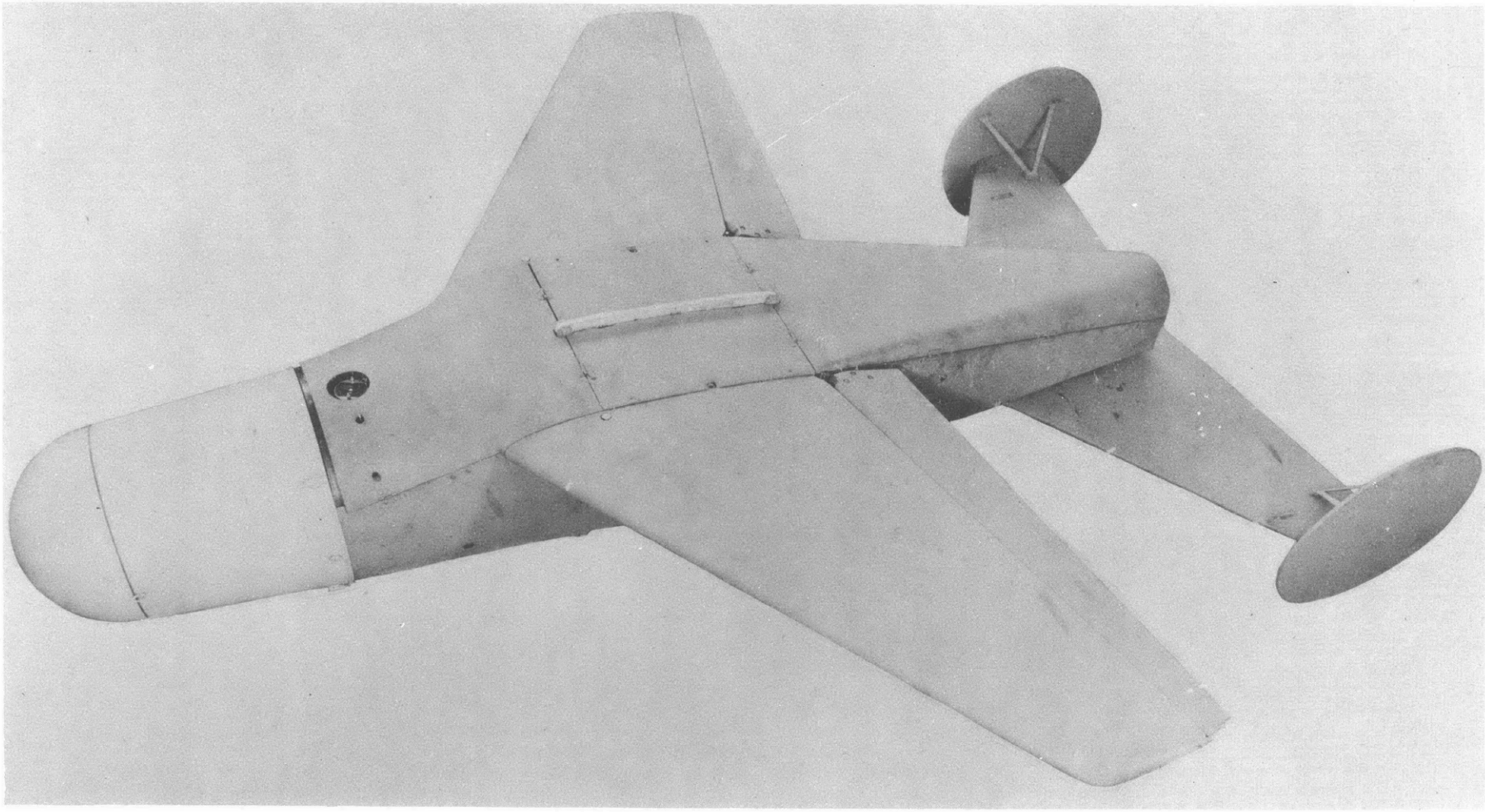
If a yaw heading error exists, a voltage from the receiver appears across terminals R and L of the roll trim amplifier. This signal is amplified by a two-stage, push-pull amplifier and operates one or the other of the roll trim relays. Depending upon which relay has been operated, the motor in the roll trim unit rotates in one direction or the other. As the motor revolves, it changes the position of one side of the differential located in the roll trim unit. The signal from the potentiometer, which is proportional to the angular displacement of the potentiometer from neutral, is fed into the cathode circuit in the first stage of the roll trim amplifier. When the roll trim

motor has revolved until the potentiometer signal counteracts the grid signal, the motor control relays will open and if necessary the opposite relays close and bring the motor to a stop. Thus the displacement of the sleeve of the roll air valve is made proportional to the yaw error as measured by the radar receiver.

As soon as the roll air valve sleeve is moved out of alignment with the roll air valve pilot, the air relay and the differential elevon clutch are operated and the glider is rolled to a new position at which the pilot and sleeve of the roll air valve are again aligned. Thus the angle through which the glider rolls is made proportional to the error in yaw and consequently the rate of yaw of the glider is made proportional to the error in yaw.

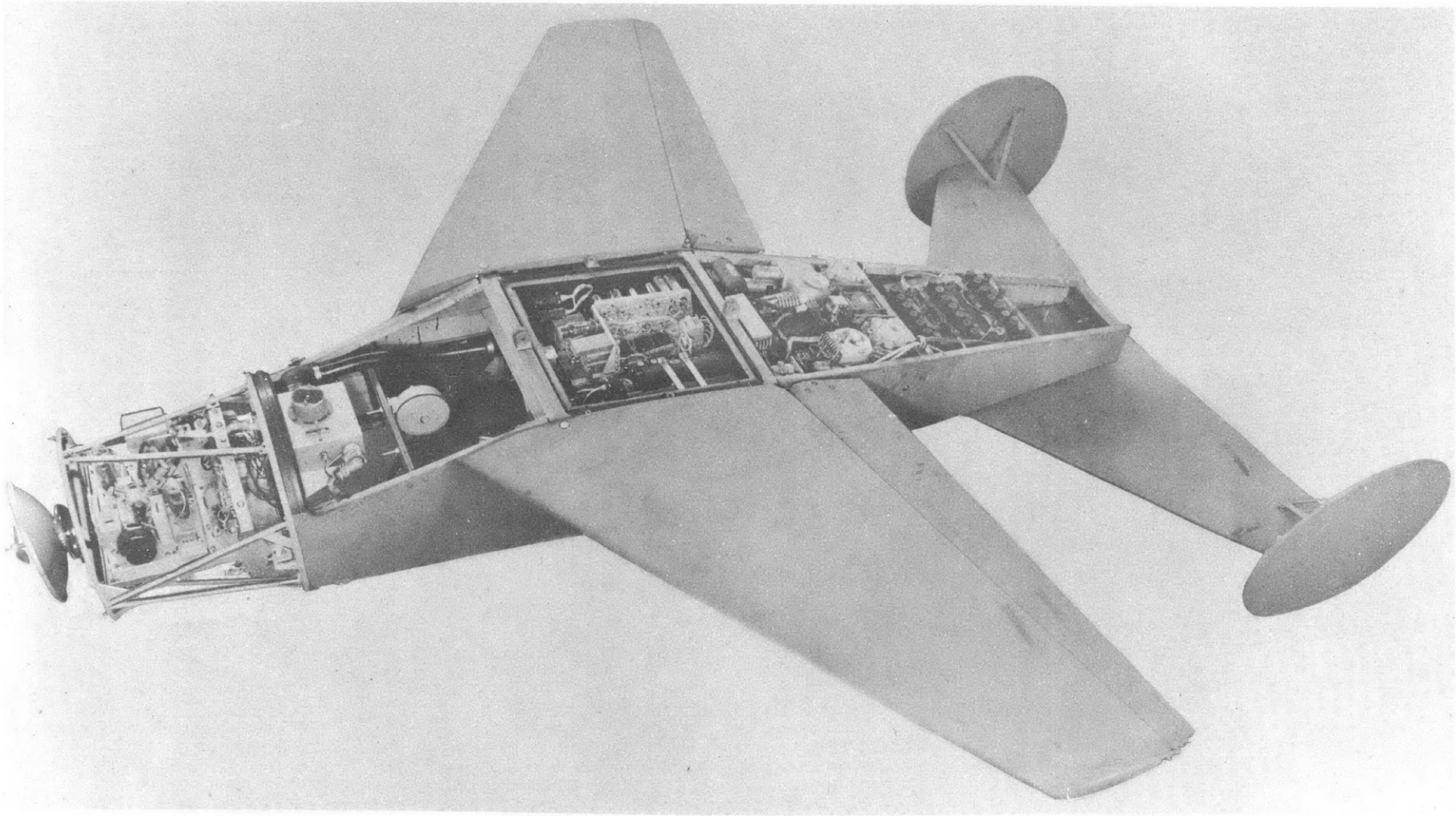
The pitch trim unit is operated in a similar fashion. If a pitch heading error exists, a voltage appears across terminals U and D of the pitch trim amplifier. One or the other of the pitch trim relays is operated, and the pitch trim motor rotates. As the motor turns, the sleeve of the pitch air valve is rotated relative to its pilot which is coupled to the pitch gimbal of the vertical gyro. In addition a small permanent magnet generator geared to the pitch trim motor is rotated. The voltage developed at the armature terminals of this generator is proportional to its rate of rotation and is fed back to the cathode circuit of the initial stage of the pitch trim amplifier. When the motor has reached such a rate that the generator voltage counterbalances the grid signal caused by the pitch heading signal, the closed relay opens and the motor decelerates. In actual operation the relays buzz at better than twenty cycles per second and close proportionality is obtained between the magnitude of the pitch heading error and the speed of rotation of the trim motor.

When the sleeve of the pitch air valve has been displaced from its position of alignment with its pilot, an air signal is supplied to the pitch air relay and the clutch which controls the average position of the elevons is engaged. The glider responds to this change in elevon position and as it changes its pitch orientation the pilot of the pitch air valve is realigned with its sleeve. Thus the rate of motion of the glider in pitch is proportional to the rate of rotation of the sleeve of the pitch air valve which in turn is proportional to the magnitude of the heading error in pitch.



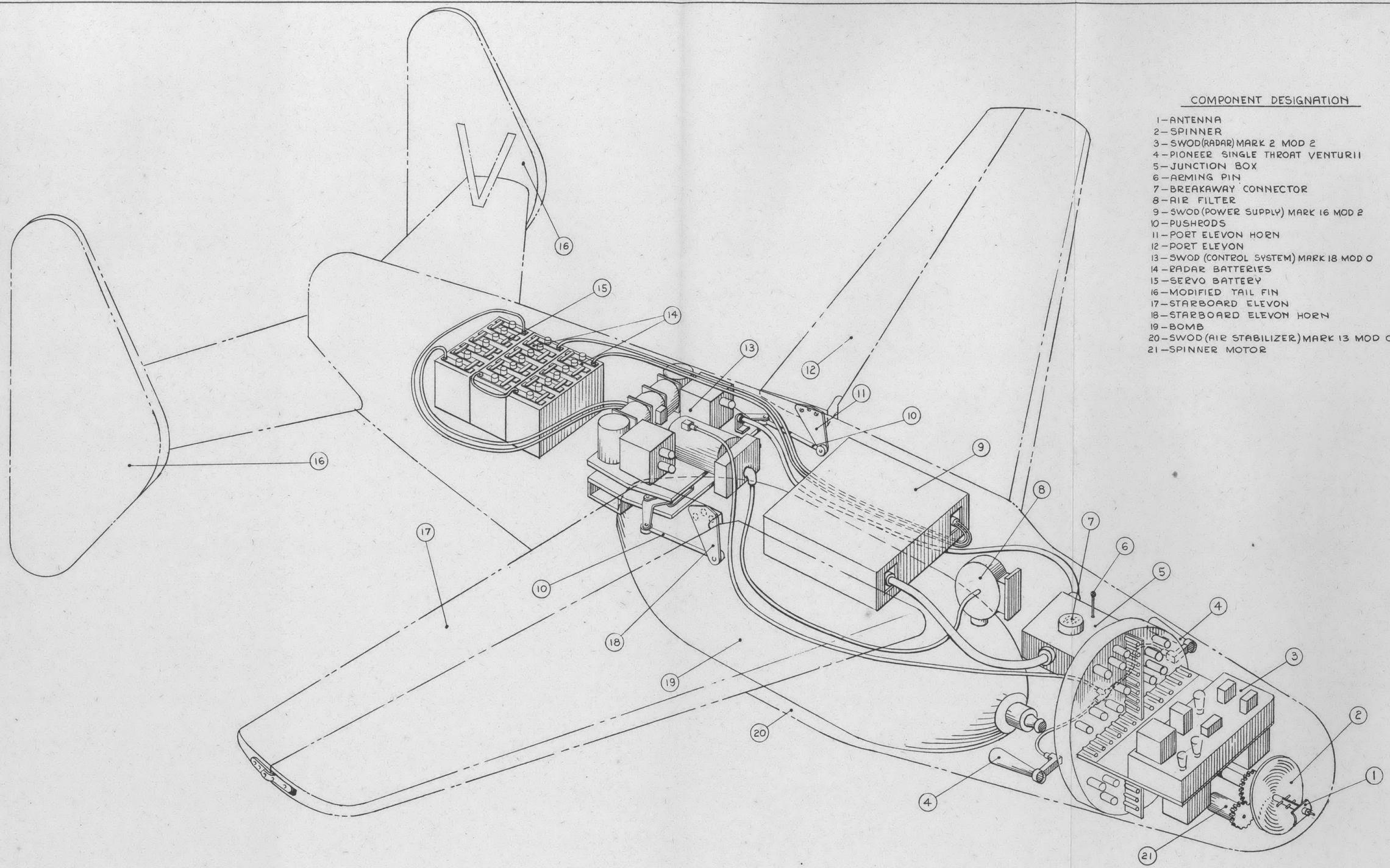
SWOD (Special Weapon) Mk 9 Mod 1

Figure 1.1



SWOD (Special Weapon) Mk 9 Mod 1 (Covers Removed)

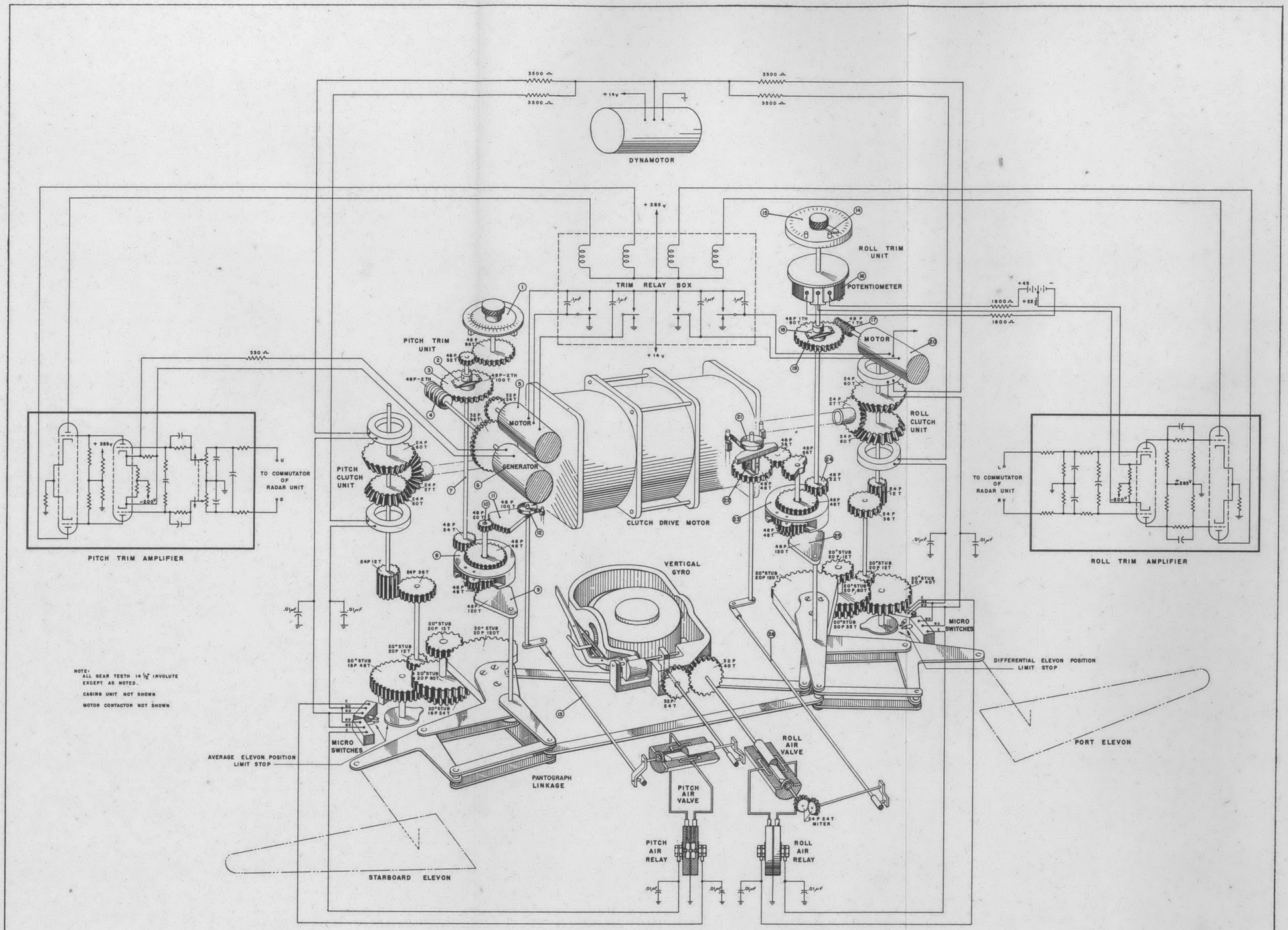
Figure 1.2



COMPONENT DESIGNATION

1—ANTENNA
2—SPINNER
3—SWOD(RADAR) MARK 2 MOD 2
4—PIONEER SINGLE THROAT VENTURII
5—JUNCTION BOX
6—ARMING PIN
7—BREAKAWAY CONNECTOR
8—AIR FILTER
9—SWOD(POWER SUPPLY) MARK 16 MOD 2
10—PUSHRODS
11—PORT ELEVON HORN
12—PORT ELEVON
13—SWOD (CONTROL SYSTEM) MARK 18 MOD 0
14—RADAR BATTERIES
15—SERVO BATTERY
16—MODIFIED TAIL FIN
17—STARBOARD ELEVON
18—STARBOARD ELEVON HORN
19—BOMB
20—SWOD (AIR STABILIZER) MARK 13 MOD 0
21—SPINNER MOTOR

SWOD (SPECIAL WEAPON) MARK 9 MOD 1

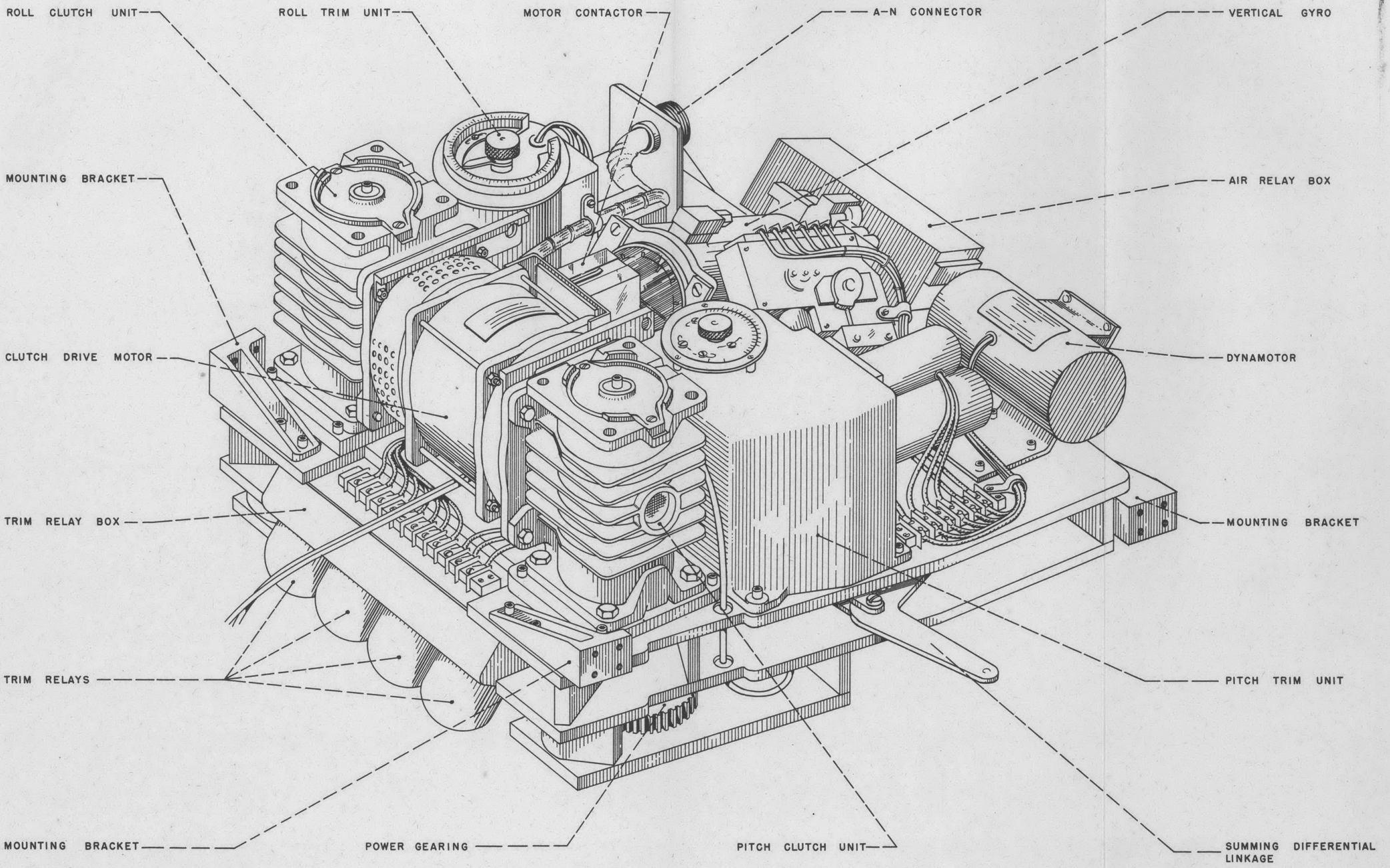


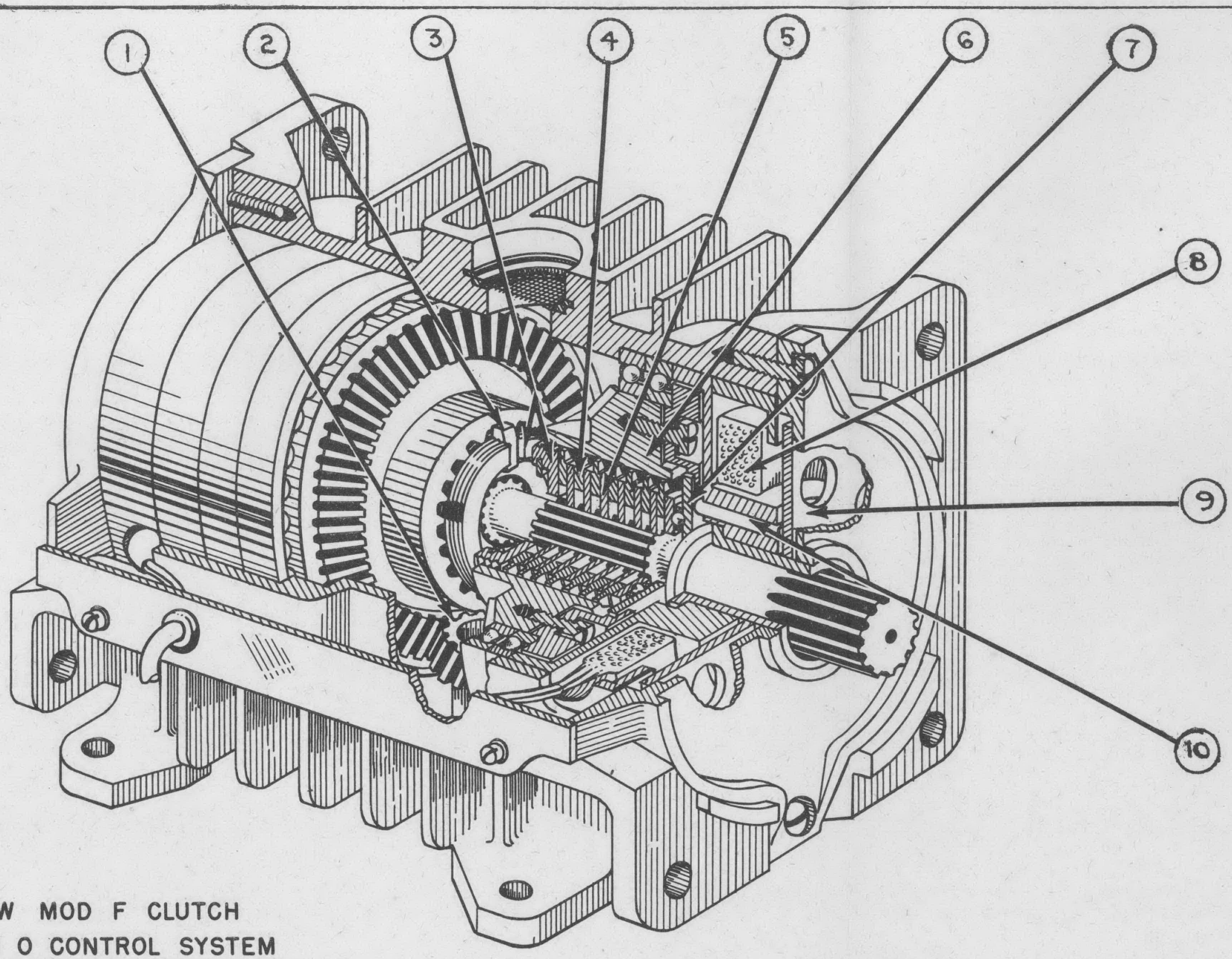
NOTE:
ALL GEAR TEETH 14 ½° INVOLUTE EXCEPT AS NOTED.
CAGING UNIT NOT SHOWN
MOTOR CONTACTOR NOT SHOWN

SCHEMATIC
MARK 18 MOD 0 CONTROL SYSTEM

PICTORIAL VIEW

MARK 18 MOD 0 CONTROL SYSTEM





CUTAWAY VIEW MOD F CLUTCH
MARK 18 MOD 0 CONTROL SYSTEM

2.0 DYNAMICAL ANALYSIS OF SYSTEM PERFORMANCE

2.1 Introduction

The difficulties of field testing guided missiles and the complexity of the control problem combine to make a thorough dynamical analysis of such systems almost imperative. The dynamical characteristics of the Bat system are determined by the three major system components, which, coupled together comprise the guided missile discussed in this report.

1. The radar intelligence unit. This unit has certain dynamical characteristics that can markedly influence the performance of the missile.

2. The airframe. The aerodynamical response of the missile to motions of its control surfaces and to wind disturbances is of prime importance. The response of the airframe is a complex problem since it is a body with six degrees of freedom.

3. The control system. In the ideal situation the dynamical characteristics of the control system are designed to "match" the characteristics of the radar and the airframe in order to secure optimum performance.

The control system should be so designed that it takes into account the dynamical characteristics of both the homing radar and the missile. In order to design properly the control system, therefore, a thorough dynamical analysis must be made of the complete system. This analysis not only determines the design of the control system itself, but assesses both the effects of lags in the radar homing unit and of certain aerodynamic peculiarities of the missile. Thus it permits specifications to be prescribed on both the radar characteristics and the airframe design.

Experience with a variety of control problems has proven that the system analysis must yield much more than an answer as to whether the system is stable or unstable. The analysis must lead to a design with a determinable and satisfactory margin of stability, must determine and control such factors as the natural frequencies of the complete system and its damping characteristics, and must establish the system sensitivity factors.

2.2 Method of Analysis

The method of analysis that was employed in the study of the system response had not previously been applied to the analysis of flight problems. It has been applied extensively to other types of automatic control problems, however, and is covered in a paper by Hall.[1] For the sake of completeness the salient points of the method are described in this section. A large portion of this description is to be published in an A.S.M.E. paper.[2]

[1] Hall, Albert C. The Analysis and Synthesis of Linear Servomechanisms. The Technology Press, Massachusetts Institute of Technology, 1943.

[2] Brown, Gordon S. and Hall, Albert C. Dynamic Behavior and Design of Servomechanisms. The American Society of Mechanical Engineers, Paper No. 45-A-20. Presented Nov. 26-29, 1945.

2.3 Frequency Response of an Automatic Control System

Most automatic control problems can be reduced to a simple block diagram as illustrated by Figure 2.1. The problem illustrated by this diagram is that of making an

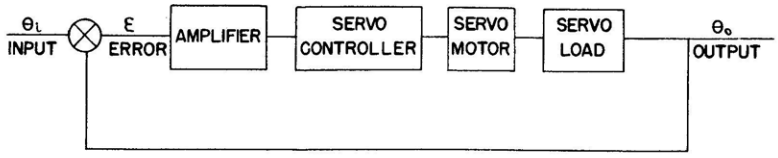


Figure 2.1

Block Diagram of an Automatic Control System

output shaft whose motion is designated by θ_o follow an input shaft whose motion is designated by θ_i. The servomechanism approach to the solution of this problem is to compare the output θ_o with the input θ_i and use the difference between these quantities to so operate the controller and servomotor that the difference between the output and input is made zero or minimized. The difference between the output and the input is termed the "error" and denoted by ϵ.

Two general methods of studying the response of such a system are in use. The first method studies the transient response of the system and the second method studies the sinusoidal response of the system. When studying the transient response, the procedure is to displace the input, θ_i, in accordance with some transient test function and measure or calculate the response of θ_o. A common test function is a step function as illustrated by Figure 2.2. The output can follow the input in a number of ways depending upon the characteristics of the amplifier-controller-servomotor combination. Certain types of output response to a step input are indicated in Figure 2.3. If the system parameters are mischosen, the system may be unstable and output will have a continuing oscillation illustrated by the unstable response. For another selection of parameters the output may approach its required position in a very slow fashion. This response is called overdamped and for many applications is equally unsatisfactory to the unstable response. For a variety of applications, the best system response is underdamped, but with a high damping factor as illustrated by the response curve so marked. The length of time that the output requires to reach a new position is a function of the damping and frequency characteristics of the oscillation. An estimate of the length of time required for a transient to disappear is given by a knowledge of these two factors.

The transient approach to the problem is satisfactory if the response characteristics as calculated or measured are satisfactory or will be satisfactory with only minor readjustment of system parameters. In general, however, it is very difficult to design or redesign a system from the transient standpoint.

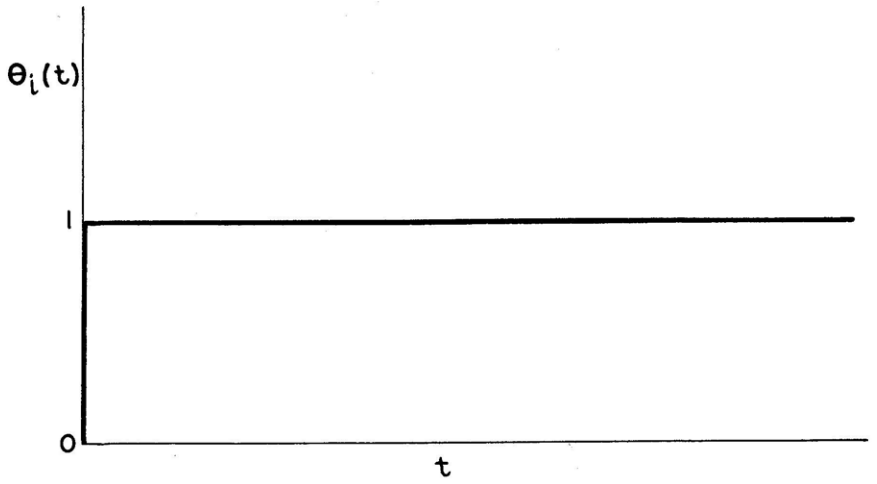


Figure 2.2

Input Step Displacement

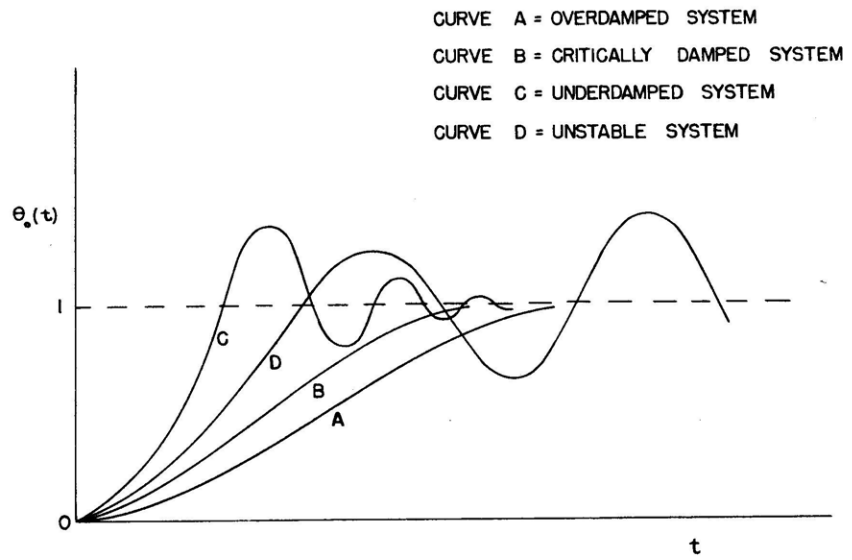


Figure 2.3

Responses of an Automatic Control System to a Step Displacement

The second method of approach to the design and analysis problem is to study the response of the system to a sinusoidal input. In this approach θ_i is made a sinusoidal function of constant magnitude and varying frequency and the steady-state performance of the output is calculated or measured.

The characteristics of the system to this type of input are prescribed by determining as a function of frequency, first, the ratio of the amplitudes of the output motion and, second, the relative phase between the output motion and the input motion. Mathematically speaking, if $\theta_i(t)$ is the input and $\theta_o(t)$ is the output of a servomechanism, then, if

$$\theta_i(t) = A \sin \omega t \tag{1}$$

and A is small, it will always be true that

$$\theta_o(t) = B \sin (\omega t + \phi). \tag{2}$$

In the above equations A is known as the amplitude of the input, B is known as the amplitude of the output, ω is the angular frequency (equal to $2\pi f$ where f is in cycles per second) of motion of $\theta_i(t)$ and $\theta_o(t)$, and ϕ is the relative phase angle between $\theta_i(t)$ and $\theta_o(t)$.

The amplitude ratio, B/A, and the phase angle, ϕ, when determined as functions of angular frequency, ω, comprise the <u>frequency response characteristic</u> of the system.

As shown above, the relationship between $\theta_o(t)$ and $\theta_i(t)$ when $\theta_i(t)$ is a sinusoidal function of time is specified by a magnitude, B/A, and an angle, ϕ, These two quantities can be considered as the defining properties of a vector whose amplitude is B/A and whose phase is ϕ. Thus it is frequently stated that a <u>vector relationship</u> exists between $\theta_i(t)$ and $\theta_o(t)$ when $\theta_i(t)$ varies sinusoidally with time. This vector relationship is represented symbolically by $\frac{\Theta_o}{\Theta_i}(j\omega)$ in which $j(=\sqrt{-1})$, itself, is generally thought of as a vector and emphasizes the vector properties of the ratio. The vector ratio $\frac{\Theta_o}{\Theta_i}(j\omega)$, is characterized by an amplitude, $\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right|$, and a phase, arc $\left[\frac{\Theta_o}{\Theta_i}(j\omega)\right]$.

The preceding development is briefly summarized as follows:

If in an automatic control system

$$\theta_i(t) = A \sin \omega t \tag{1}$$

then

$$\theta_o(t) = B \sin (\omega t + \phi). \tag{2}$$

By definition, the frequency response is denoted by $\frac{\Theta_o}{\Theta_i}(j\omega)$. The amplitude response is given by

$$\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right| = \frac{B}{A} \tag{3}$$

and the phase response is represented by

$$\text{arc}\left[\frac{\Theta_o}{\Theta_i}(j\omega)\right] = \phi. \tag{4}$$

The amplitude and phase response curves of a typical system are illustrated by Figure 2.4.

The curves of Figure 2.4 may be obtained (1) by calculation, if the constants of an actual or proposed design are available, or (2) by measurement, if the servomechanism itself is available. The frequency response characteristic is measured by moving the input sinusoidally at a fixed amplitude but at various frequencies. At each frequency

the amplitude of the output and its phase relation to the input motion is measured. The ratio of the amplitudes, plotted for each frequency, yields the first of the curves of Figure 2.4. The phase difference between the input and output, plotted for various frequencies, gives the second of the curves of Figure 2.4. The curves of Figure 2.4 are

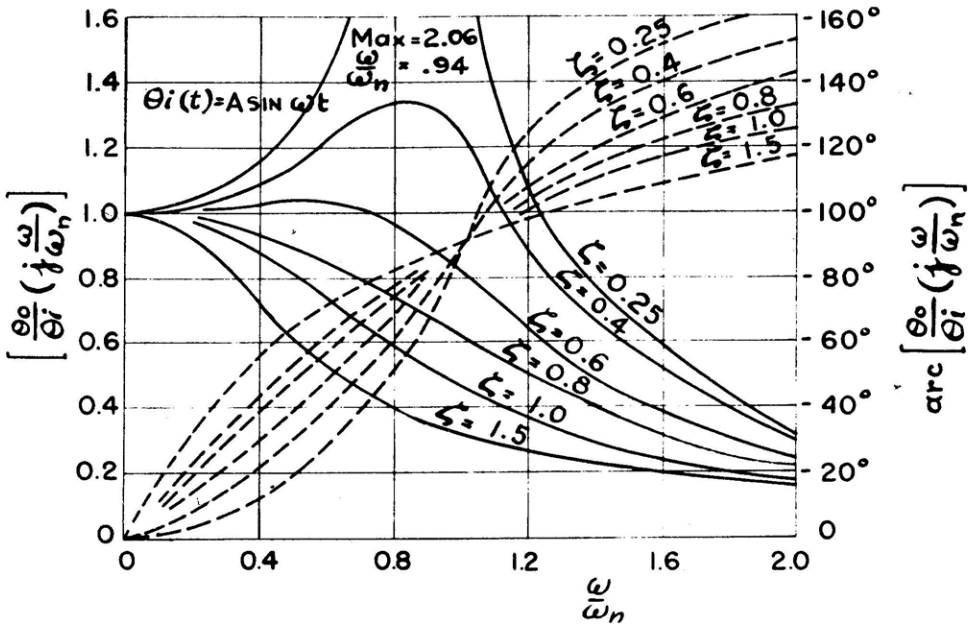


Figure 2.4

Frequency Response Characteristics

readily calculated if the constants of the system and the differential equation relating the output to the input is known. The calculation is effected by replacing the operator p by the frequency operator jω and applying conventional vector arithmetic.

The frequency response of a servomechanism may be closely correlated with its transient response. Important natural frequencies in the transient response are indicated by peaks in the amplitude response curve. The magnitudes of the peaks of the amplitude response are measures of the relative damping of the natural frequencies of the transient response. The frequency band over which the amplitude response has a substantially constant magnitude is a measure of the speed of response to transients, since a high natural frequency (and, therefore, a high speed of response) is linked with a high resonant frequency in the amplitude response. When, as indicated above, the frequency response characteristic is correlated with the transient response characteristic, the former becomes a powerful means of analysis.

The correlation between the sinusoidal and transient characteristics is illustrated by comparing the transient and frequency response of a simple system representable by a second-order differential equation.

If the input, $\theta_i(t)$, is a step displacement, and the output, $\theta_o(t)$ is determined for various values of the damping ratio, the set of transient responses illustrated by Figure 2.5 results. If $\theta_i(t)$ is made a sinusoidal function,

$$\theta_i(t) = A \sin \omega t \qquad (1)$$

and the amplitude response of the output, $\theta_o(t)$, is determined, the set of curves of Figure 2.4 are obtained. Comparison of the transient and frequency responses reveals a

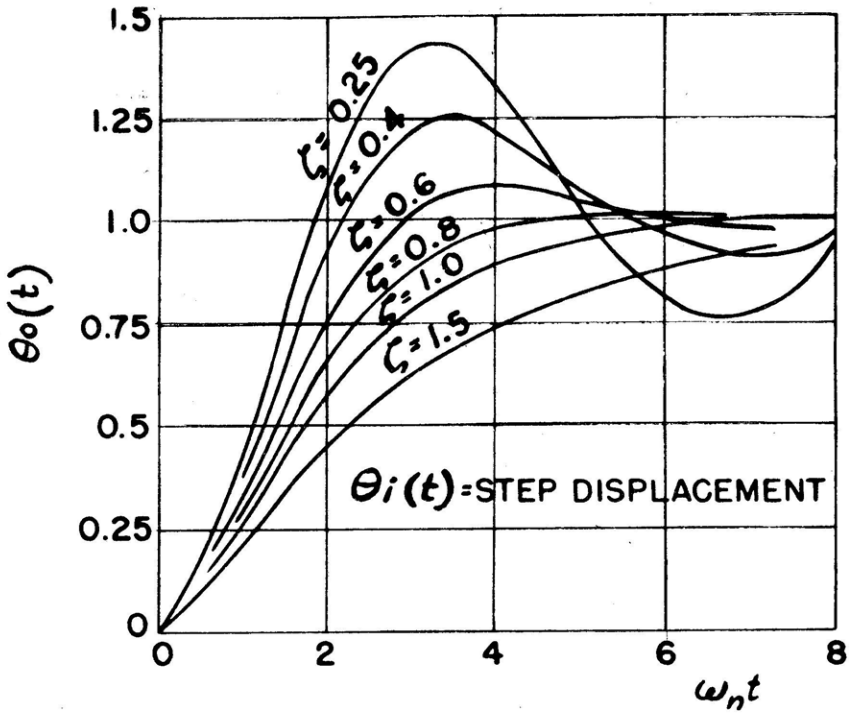


Figure 2.5

Responses of a Second-Order System to a Step Displacement

number of points of correspondence: (1) The frequency at which the transient response oscillates (the natural frequency) is approximately the same as the frequency at which the amplitude response has a peak (the resonant frequency). (2) As the damping ratio is reduced the transient response becomes more oscillatory, and the peak in the amplitude response is magnified. (3) When the damping ratio is made larger than unity, the transient response becomes sluggish and the amplitude response falls off rapidly without a peak.

It is frequently convenient to combine the information contained in the amplitude response curve and in the phase response curve of a system by a single graph. This can be accomplished by remembering that the amplitude ratio and the phase angle are quantities defining a vector which relates the output and input. As the frequency of the input is varied, this vector varies in phase and magnitude and the amplitude and phase response curves present the information on the manner in which these two properties of the vector change with frequency. The same information can be provided in an alternative way by plotting on polar coordinate paper the path followed by the tip of the vector as the frequency varies over the band of interest. If various points of the curve are labeled with the frequency to which they correspond one curve will supply the information contained in two curves when the amplitude and phase response are plotted separately. This graphical presentation is known as the locus of the frequency response characteristic. It is illustrated by Figure 2.6 which was drawn for the same second-order system to which the curves of Figures 2.4 and 2.5 apply. The principal advantage of presenting information in this fashion lies in the means it provides for visualizing the frequency response and in the fact that it emphasizes the very important relationship that always exists between the amplitude and phase responses of a system.

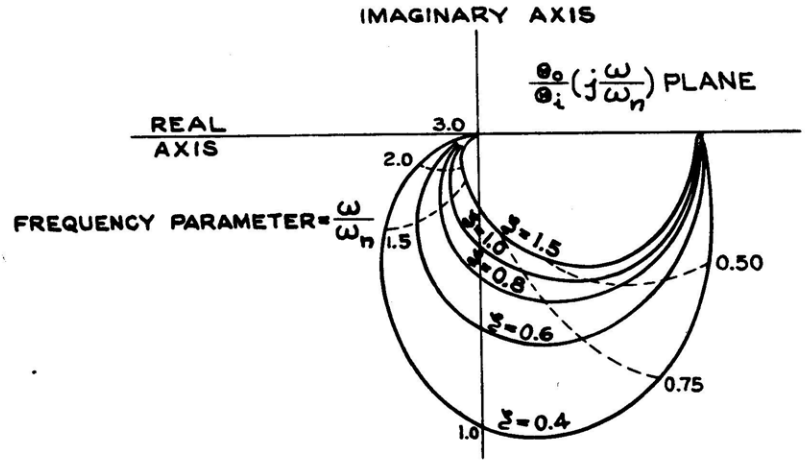


Figure 2.6

Complex Plane Plot of Sinusoidal Responses

2.4 Transfer Function of an Automatic Control System

It is clear from the block diagram of Figure 2.1 that a single function completely defines the performance of a servomechanism in which the feedback link contains no

frequency dependent elements. The defining function is the relationship between the servo output, $\theta_o(t)$, and the error, $\epsilon(t)$. If this relation is known in operational, sinusoidal or time response form, the performance of the system is completely defined for all conditions that may be imposed upon it. This function, relating the system output of the closed cycle system to its error, has been termed the Transfer Function of the system. When the sinusoidal or frequency response form of the transfer function is studied, it becomes a powerful analysis and synthesis tool. The transfer function may be derived from a known frequency response characteristic of the servo system, it may be measured directly, or, if system constants are known, it may be calculated. The transfer function of the system can be written in the form $\frac{\Theta_o}{E}(j\omega)$. This function is a vector quantity with an amplitude and a phase characteristic just as the ratio $\frac{\Theta_o}{\Theta_i}(j\omega)$ is a vector quantity.

The transfer function of a closed-cycle system is always the product of two parts, one that is invariant with frequency and a second that is frequency dependent. The fact that these two components exist is emphasized by writing the transfer function in the following form:

$$\frac{\Theta_o}{E}(j\omega) = KG(j\omega) \tag{5}$$

The term K represents that part of the transfer function which is invariant with frequency. This portion is known as the gain or the sensitivity factor and is a function of amplifier gain, etc. The second part of the transfer function is denoted by $G(j\omega)$ and represents the portion of the transfer function which changes with frequency.

The frequency response of the servomechanism is related to the transfer function by the following vector equation:

$$\frac{\Theta_o}{\Theta_i}(j\omega) = \frac{KG(j\omega)}{1 + KG(j\omega)} \tag{6}$$

The transfer function of a system is calculated by straightforward circuit analysis techniques, or by determining the differential equation relating the output to the error and replacing $\frac{d}{dt}$ by $j\omega$.

2.5 The Transfer Function Locus

The transfer function can be studied by means of its frequency characteristics just as other functions have been so studied. The amplitude and phase response curves of the transfer function can be drawn and these curves completely define the characteristics of the transfer function of the servomechanism and therefore completely define the system itself. Just as the phase and amplitude response curves were combined into a single polar plot with frequency as a parameter, so can the frequency and phase response curves of the transfer function be combined. This parametric polar plot of the transfer function has been called the transfer function locus, or simply the transfer locus of the servomechanism. The transfer locus completely defines the characteristics of the servo system. A study of its nature provides a very effective method for the synthesis of servomechanisms intended for particular applications and a useful general guide to the adjustment of servomechanism parameters in order to secure optimum performance.

If a parametric frequency plot of the transfer function, $KG(j\omega)$, is available, the magnitude and phase of the function $\frac{\Theta_o}{\Theta_i}(j\omega)$ may be found by graphical calculation.

The function $\frac{\Theta_o}{\Theta_i}(j\omega)$ has been related to the transfer function by Equation (6). A plot of a typical transfer function, $KG(j\omega)$, is illustrated by Figure 2.7. Suppose it is desired

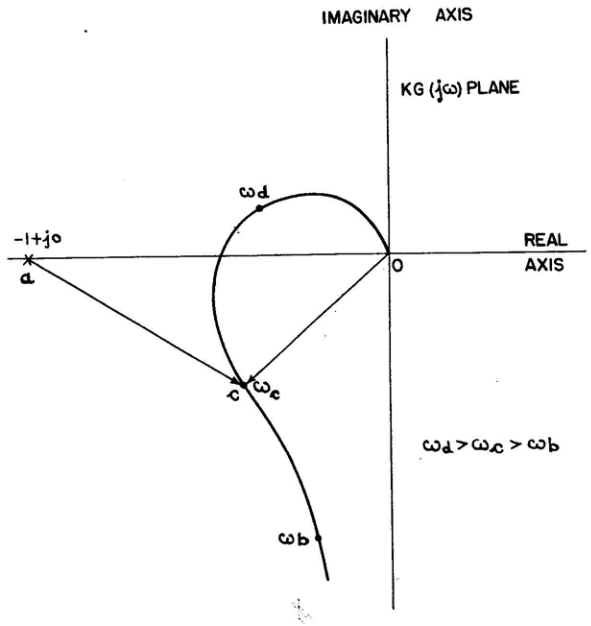


Figure 2.7

Transfer Function Locus

to calculate the magnitude and phase of $\frac{\Theta_o}{\Theta_i}(j\omega_c)$ for a particular frequency, ω_c, with only the transfer locus available. The transfer function at ω_c, $KG(j\omega_c)$, is represented by the vector $\overline{oc}$ while the vector $\overline{ac}$ represents the term $\left[1 + KG(j\omega_c)\right]$, since the point "a" is located at (-1 + j0). Therefore, the magnitude of $\frac{\Theta_o}{\Theta_i}(j\omega_c)$ is given by

$$\left|\frac{\Theta_o}{\Theta_i}(j\omega_c)\right| = \frac{\left|KG(j\omega_c)\right|}{\left|1 + KG(j\omega_c)\right|} = \frac{\left|\overline{oc}\right|}{\left|\overline{ac}\right|} \tag{7}$$

while the phase of $\frac{\Theta_o}{\Theta_i}(j\omega)$ is given by

$$\text{arc}\left[\frac{\Theta_o}{\Theta_i}(j\omega_c)\right] = \text{arc (oca)} \tag{8}$$

The angle, arc (oca), is negative. Thus the magnitude of $\frac{\Theta_o}{\Theta_i}(j\omega_c)$ is equal to the ratio of the magnitudes of the vectors $\overline{oc}$ and $\overline{ac}$ and the phase of $\frac{\Theta_o}{\Theta_i}(j\omega_c)$ is equal to the angle between these two vectors.

Equations (7) and (8) permit ready visualization or calculation of the magnitude and phase of $\frac{\Theta_o}{\Theta_i}(j\omega)$. At small frequencies, such as ω_b (see Figure 2.7), both vectors $\overline{oc}$ and $\overline{ac}$ are large and approximately equal, and their ratio is approximately unity. The angle between the two vectors, the phase of $\frac{\Theta_o}{\Theta_i}(j\omega)$, is small at this frequency. As the frequency increases the angle between the two vectors increases and their lengths become smaller so that differences in their lengths cause the ratio $\frac{|\overline{oc}|}{|\overline{ac}|}$ to depart from unity. Whether the ratio $\frac{|\overline{oc}|}{|\overline{ac}|}$ (the magnitude of $\frac{\Theta_o}{\Theta_i}(j\omega)$) increases or decreases as the frequency increases depends upon the shape of the curve relative to the origin and the point $(-1 + j0)$. A continuation of this reasoning for the remainder of the frequency range permits the general shape of the phase and magnitude of the servo output to be completely determined. If desirable, the phase and magnitude curves can be determined from the transfer locus with accuracy and ease by the use of a protractor and divider to measure the angles and lengths directly from the graph.

The general nature of the amplitude response may be obtained also by drawing in the complex plane curves of constant $\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right|$. These curves are circles and are shown in the upper part of Figure 2.8. If a servo could have a transfer function locus that lay

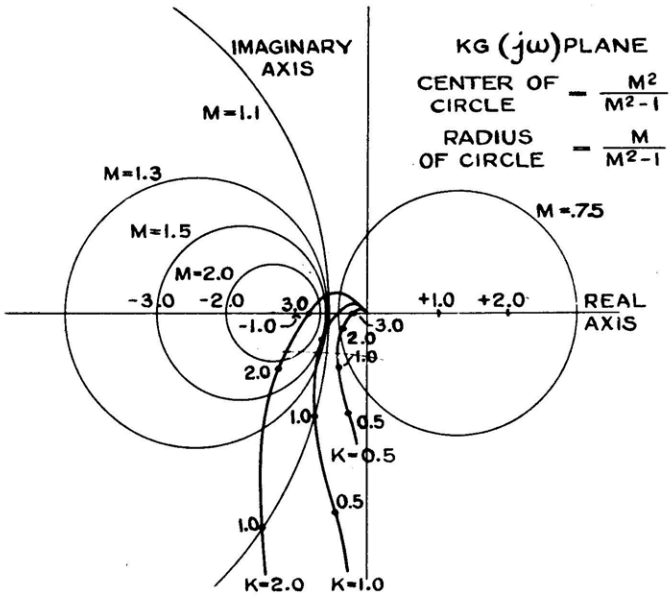


Figure 2.8

Determination of Servo Sensitivity

along one of these circles, the system amplitude response would be independent of frequency and equal to the value of M for which the circle was drawn. The center and radius of each circle is dependent upon M, and the relation is given in Figure 2.8. The

frequency at which the transfer function locus crosses one of these circles is the frequency at which the magnitude of $\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right|$ is equal to the value of M for which that circle is drawn. If the locus is tangent to a circle it indicates that a maximum or minimum in $\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right|$ occurs at that frequency. Thus the amplitude response of the system whose transfer function is plotted in Figure 2.8 has a peak of 1.5 at a frequency $\omega = 2$, if $k = 1.0$.

2.6 Absolute Stability Criterion

The primary requirement that almost every servomechanism must satisfy is that of stability. Although a servo system must be more than barely stable to be satisfactory, a stability criterion of one type or another is generally the first test applied to proposed servomechanism design. Several criteria exist; however, the one described here was developed primarily for application to feedback amplifiers.

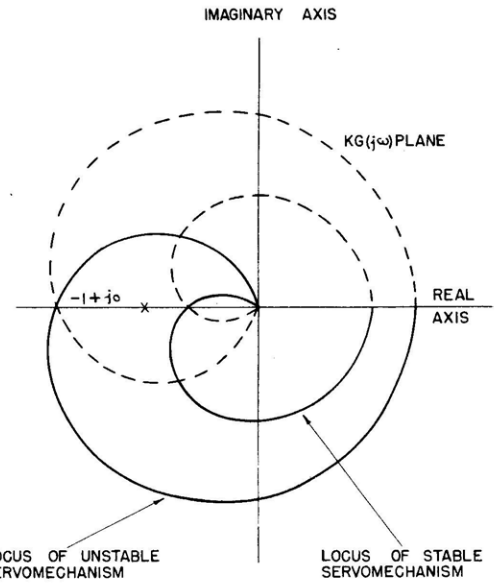


Figure 2.9

Transfer Loci of Closed Form

It is at once apparent to those familiar with feedback amplifier theory that the transfer locus of a servomechanism is analagous to the Nyquist diagram of a feedback amplifier. The term Nyquist diagram has been given to this type of plot for a feedback amplifier because of a very useful criterion developed by Nyquist [1] for determining the

[1] Nyquist, H. Regeneration Theory, Bell System Tech. Jour., January, 1932.

stability of a feedback amplifier. This criterion may be applied equally well to the transfer locus in order to determine from its shape and position whether or not the servomechanism for which it is drawn is stable. To apply the Nyquist stability criterion to servomechanisms the following procedure is employed: (1) The transfer locus ($KG(j\omega)$ plotted in polar form) is drawn for all frequencies from zero to infinity. (2) The conjugate of the transfer locus is drawn. The conjugate of a curve is the mirror image of the original curve about the real axis. (3) If the curves so formed enclose the point $(-1 + j0)$, the system is unstable. If the curves do not enclose this point, the system is stable. The application of this criterion is illustrated by Figure 2.9.

The above criterion applies to curves of closed form. That is, it applies to transfer loci of such character that the loci and their conjugates join at zero and at infinite frequency. Actually the transfer loci of most servomechanisms are of the open form and some extension is required in order to apply the stability criteria to these forms of transfer loci. The open form of the transfer locus can be changed into the closed form by connecting the curve and its conjugate at the zero frequency point by means of a circle of infinite radius. The connection should always be made in such a way that no phase discontinuity occurs along the path of the curve. This is illustrated by Figure 2.10 in which are plotted the transfer loci of two common types of servomechanisms, both of which are stable.

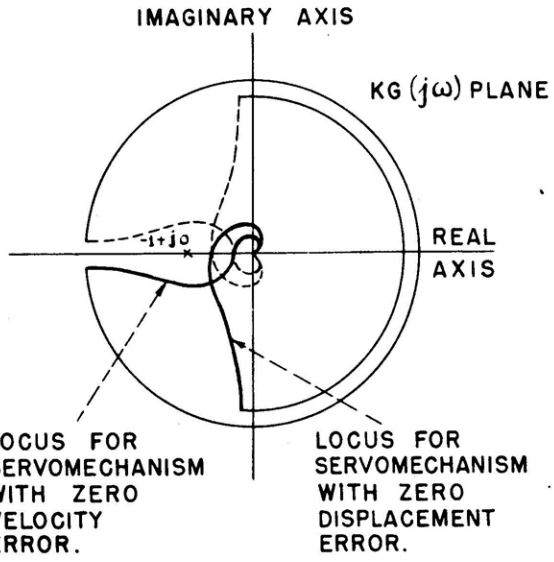


Figure 2.10

Transfer Loci of Open Form

2.7 Transfer Loci for Various Types of Steady-State Performance

The performance of a servomechanism under steady-state conditions is always of great importance. If the servomechanism is primarily a positional device, it is desirable that the servomechanism take up various positions without requiring an error to maintain it in that position. Similarly it is frequently necessary for the servomechanism to follow an input of constant velocity, that is, one in which $\theta_i(t) = kt$. In this case it is desirable for the servomechanism to follow various velocities as required without the necessity of a system error to maintain that velocity. Servomechanisms that satisfy the first condition frequently are termed zero-displacement-error servomechanisms and servos that meet the second condition similarly are called zero-velocity-error servomechanisms. It can be readily shown that if the transfer locus of a servomechanism approaches infinity along the negative imaginary axis, the servo will have zero-displacement-error. Similarly if the transfer locus of a servomechanism approaches infinity along the negative real axis that servo will have zero-velocity-error. The zero-velocity-error servo will, of course, have zero-displacement-error also. In Figure 2.10 is illustrated the transfer loci of a zero-velocity-error system and of a zero-displacement-error system. This concept can be extended to servo systems that will follow a constant input acceleration without steady-state error, and so forth.

2.8 Determination of the System Sensitivity

A very simple procedure exists for determining the system sensitivity, K, permitted by a prescribed maximum value of $\left|\frac{\theta_o}{\theta_i}(j\omega)\right|$. This procedure makes use of the fact that variations in the sensitivity, K, are equivalent to changes in scale of the plot of $KG(j\omega)$. Instead of plotting $KG(j\omega)$, the function $G(j\omega)$, only, is plotted. If the scale of this plot of $G(j\omega)$ is correctly altered, then this plot will represent $KG(j\omega)$ and the factor by which the scale must be altered is equal to K. The factor by which the scale of the plot of $G(j\omega)$ must be altered to transform it to a plot of $KG(j\omega)$ is determined by the maximum permissible value of $\left|\frac{\theta_o}{\theta_i}(j\omega)\right|$. The procedure is as follows:

It has been shown that if the plot $KG(j\omega)$ is tangent to a circle whose center is $\frac{M^2}{M^2-1}$ on the negative real axis, and whose radius is $\frac{M}{M^2-1}$, that the function $\left|\frac{\theta_o}{\theta_i}(j\omega)\right|$ corresponding to this $KG(j\omega)$ will have a maximum value of M. Now a circle whose center is at $\frac{M^2}{M^2-1}$ on the negative real axis and whose radius is $\frac{M}{M^2-1}$ will have an intercept on the real axis equal to $\frac{M}{M+1}$, and the following ratio will be maintained:

$$\frac{\text{Center of Circle}}{\text{Intercept on Real Axis}} = \frac{M}{M-1} \tag{9}$$

The scale factor, K, can be readily found if a circle can be located on the plot of $G(j\omega)$ that is tangent to the locus, $G(j\omega)$ and whose center on the negative real axis is $\frac{M}{M-1}$ times its real-axis intercept. The location of such a circle can be found by a cut-and-try process with a pair of dividers and is the work of but a few moments. Suppose such a circle has a center at A, on the $G(j\omega)$ plane. But it has been shown that if the scale were correctly chosen in order that the locus be a plot of $KG(j\omega)$, the center of the circle would be at $\frac{M^2}{M^2-1}$. Therefore the scale must be changed by the factor

$\frac{M^2}{A(M^2-1)}$ and the sensitivity K_M is equal to $\frac{M^2}{A(M^2-1)}$. The value of K_M is the system sensitivity that will provide a maximum value of $\left|\frac{\Theta_o}{\Theta_i}(j\omega)\right|$ equal to M.

The above procedure of plotting only $G(j\omega)$ and then determining the sensitivity, K, is easy to use and permits greater freedom in the study of transfer loci since it essentially non-dimensionalizes the plots as far as the sensitivity factor is concerned.

2.9 Application of Theory to Missile Control

Although the above design method has not been used previously in the design of control systems for airplanes, its adaptability to a variety of other control problems gave rise to the hope that it would prove equally effective in the solution of flight control problems. Accordingly it was employed in designing the control system for Bat. The success with which correct design parameters and flight results could be predicted justified the confidence placed in this method.

It is known that the performance of an airplane can be studied by considering its lateral and longitudinal motion separately and the analysis of the missile control is so handled here.

Figure 2.11 illustrates the lateral problem to be solved and Figure 2.12 presents the elements of the system in block diagram form.

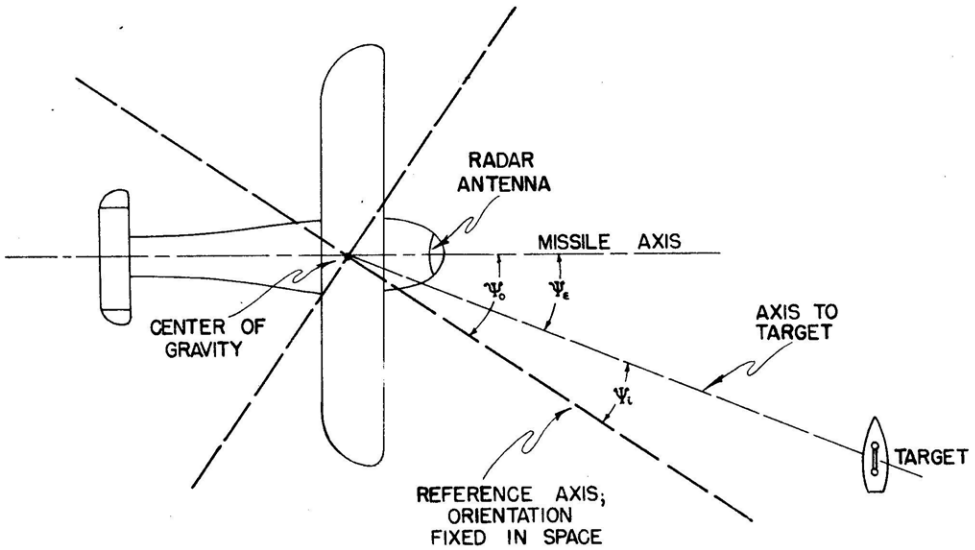


Figure 2.11

Axes used in Derivation of Lateral Motion

The relationship to the general block diagram of an automatic control system as shown in Figure 2.1 is obvious. Figure 2.13 presents the elements of both lateral and

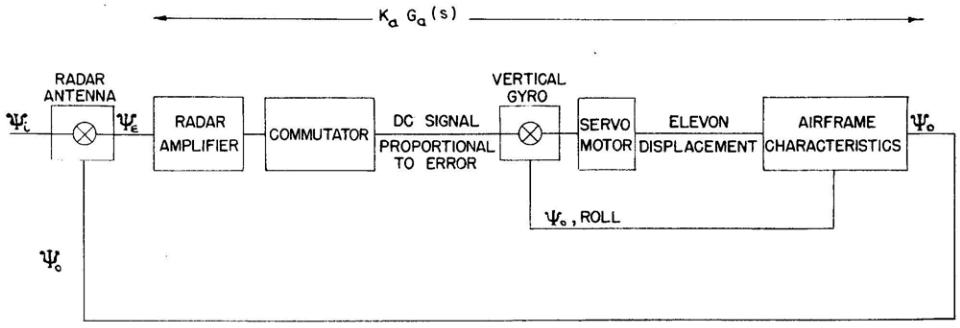


Figure 2.12

Block Diagram of Lateral System

longitudinal control in schematic form. In Figure 2.11 the input to the system is the target motion, the output is the orientation of the missile itself, and the error is the difference between the correct orientation of the missile and its actual orientation. The orientation of the missile and the position of the target are measured with respect to axes whose orientation is fixed in space and whose origin is located in the missile. The characteristics of the system are studied by determining the transfer function relating the error to the change in the missile orientation.

Because of the rush nature of the development of the SWOD (Control System) Mk 18 Mod 0, the choice of system constants depended greatly upon proper mathematical analysis of the complete dynamical system. While much experience had been gained in developing a similar system for Pelican, the differences between the two missiles were sufficient to require a complete analysis of the Bat system itself. In addition to a number of important aerodynamical differences that exist between the two airframes, other differences arose from the characteristics of the radar due to the fact that the radar for Bat is a transmitter as well as receiver, the fact that the weight of Bat is more than three times that of the test Pelican unit, and the fact that the speeds encountered with Bat were more than twice those encountered with Pelican.

It should be pointed out that the labor required to carry through the analysis of this problem with the sinusoidal approach is still large due to the highly complex nature of the system and due to the necessity of including many effects which could ordinarily be neglected in a problem in which the stability and response characteristics were less critical. The analysis given in this chapter, therefore, while simple in concept is lengthy algebraically. However, time spent on analysis paid for itself many fold in time gained in the development of the overall system due to the difficulty and expense of producing models for field tests.

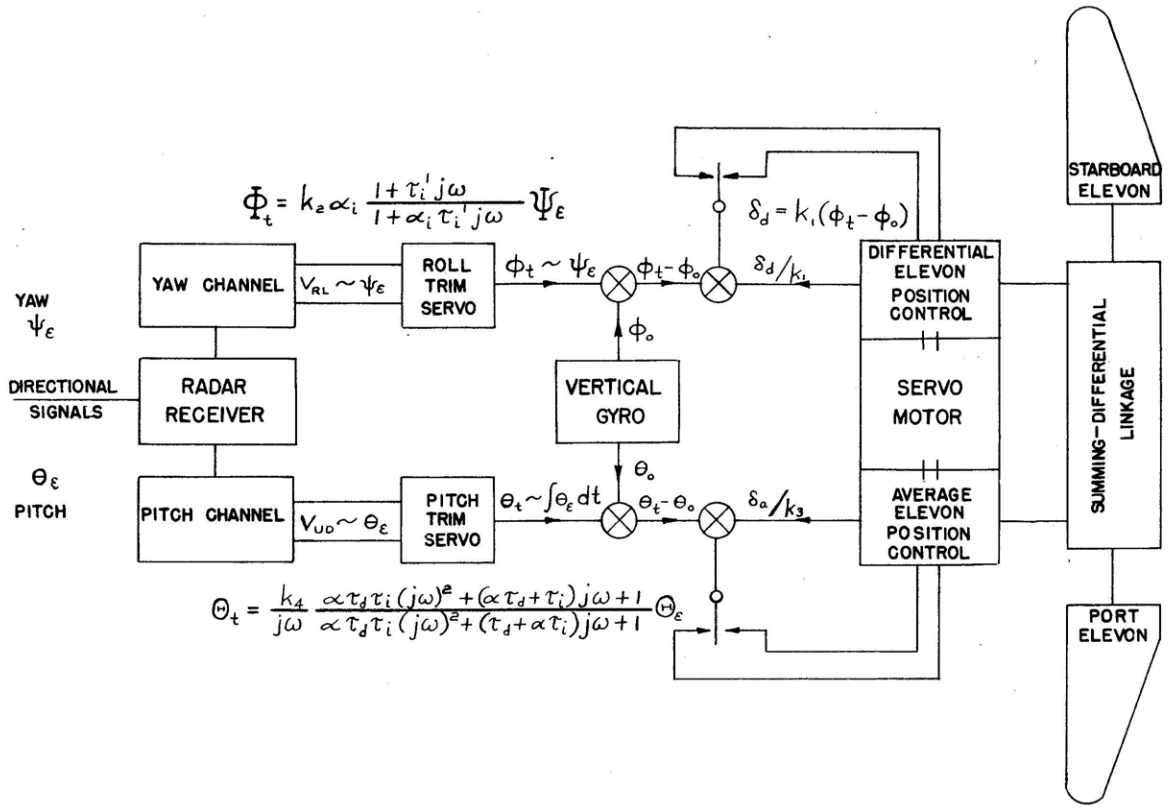


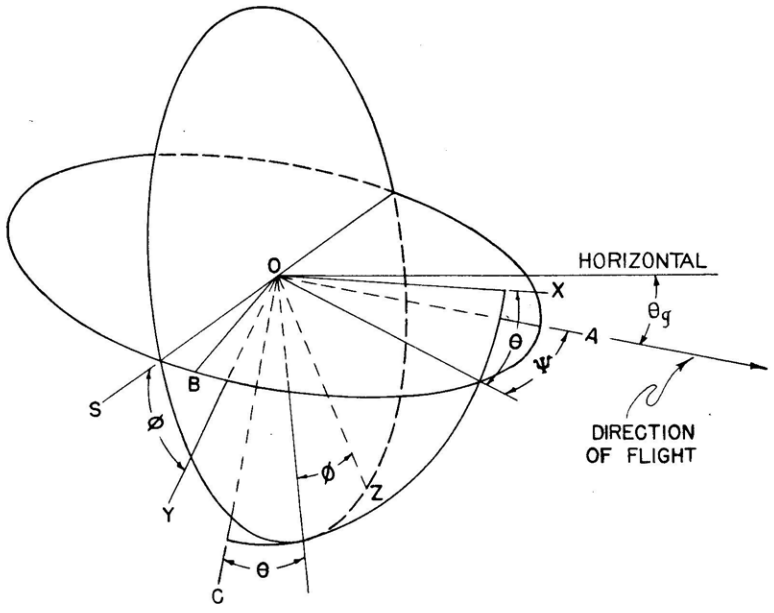
Figure 2.13

Functional Block Diagram of SWOD (Control System) Mk 18 Mod 0

2.10 Assumptions Made in the Analysis

Two principle assumptions were made in the system analysis: (1) The angular displacements of the missile from its steady-state position are small and (2) the lateral and vertical displacements made by the missile are small compared to its range from the target. The first assumption permits linear theory to be employed in the analysis of the performance of the missile. In determining the lateral characteristics, the assumption must be made that angular displacements are infinitesimal. This is a common assumption made in studying the lateral stability of airplanes and it has been found in practice that the results obtained for infinitesimal displacements apply well to moderately large angular displacements of the airplane. If this assumption is not made, the problem becomes too complex except when machine analysis is employed.

The significance of the second assumption is that the stability and dynamical performance of the glider is determined at long ranges from the target. The second assumption permits the analysis to be made on the basis that the signal received by the homing system is determined only by the angular rotation of the missile and not by its lateral or vertical displacements. Actually, of course, lateral and vertical displacements of the missile will affect the signal obtained by the homing system even though the angular attitude of the missile is unchanged. The effect of lateral and vertical displacements of the homing system is a function of the range from the missile to the target and is unimportant at long ranges. Such displacements become important at ranges of the order of 1000 feet. Since (1) the effect of signal changes due to displacements of the center of gravity of the missile occur much more slowly than the effect on the signal arising from changes in angular orientation of the missile and (2) the minimum range of the system is approximately 400 feet and (3) the size of the target is relatively large, it is felt that the effect of lateral and vertical displacements can be omitted from the dynamical analysis of the problem without appreciable loss in validity. Other less important assumptions in the analysis are indicated where they are used.



X, Y, Z are body axes, OX in flight direction.

A, B, C are space axes, OA in steady-state flight direction.

Point O is the center of gravity of the airframe.

Figure 2.14

System of Axes

2.11 General Equations of Motion

The equations of motion are written for a set of axes fixed in the missile. The angular orientation of the missile refers this set of axes to a set of axes of fixed orientation whose origin is at the center of gravity of the missile. The system of axes employed is illustrated by Figure 2.14. The symbols employed in the analysis are listed in Table 1.

The equations of motion of the airplane have been derived previously and are presented in references such as Aerodynamic Theory, Vol. 5, by Durand. The equations will not be derived here, but are presented for reference and development. It should be noted that the system of axes is tilted through an angle θ_g in order that the A axis will point along the steady-state flight direction.

The vector equations of motion follow:

$$M = I_y \frac{dq}{dt} - (I_z - I_x)rp - I_{xz}(r^2 - p^2) \tag{10}$$

$$N = I_z \frac{dr}{dt} - (I_x - I_y)qp - I_{xz}\left(\frac{dp}{dt} - rq\right) \tag{11}$$

$$L = I_x \frac{dp}{dt} - (I_y - I_z)qr - I_{xz}\left(\frac{dr}{dt} + pq\right) \tag{12}$$

$$\frac{X'}{m} + g(\cos \psi_o \cos \theta_o \sin \theta_g - \sin \theta_o \cos \theta_g) = \frac{du}{dt} - rv + qw \tag{13}$$

$$\frac{Y'}{m} + g(\cos \theta_g \cos \theta_o \sin \phi_o + \sin \theta_g \cos \psi_o \sin \theta_o \sin \phi_o - \sin \theta_g \sin \psi_o \cos \phi_o) = \frac{dv}{dt} - pw + ru \tag{14}$$

$$\frac{Z'}{m} + g(\cos \theta_g \cos \theta_o \cos \phi_o + \sin \theta_g \sin \quad \sin \phi_o + \sin \theta_g \cos \psi_o \sin \theta_o \cos \phi_o) = \frac{dw}{dt} - qu + pv \tag{15}$$

In the above six equations the moments M, N, and L and the forces X', Y', and Z' contain the control moments and forces applied to the missile to control its flight. In particular the moment L, and the force Z' contain terms which effect the control of Bat.

This is indicated mathematically by writing:

$$L_{\delta d}\delta_d = f_1(\phi_o, \psi_\epsilon) \tag{16}$$

$$Z_{\delta a}\delta_a = f_2(\theta_o, \theta_\epsilon) \tag{17}$$

This indicates (1) that the vertical force $Z_{\delta a}$ produced by changing the average position of the elevons is made a function of both the pitch orientation of the missile, θ_o, and the pitch error in heading θ_ϵ, and (2) that the rolling moment $L_{\delta d}$ produced by changing the differential position of the elevons is made a function of both the yaw error in heading, ψ_ϵ, and the roll orientation of the missile, ϕ_o.

After functions f_1 and f_2 are established, equations (10) to (17) completely determine the motion of the missile. The following two definition equations must be included, of course.

$$\theta_\epsilon = \theta_i - \theta_o \tag{18}$$

$$\psi_\epsilon = \psi_i - \psi_o \tag{19}$$

Equations (10) to (19) may be divided into two groups which are independent of each other as long as only small changes in angles are considered. The first group includes those equations which determine the lateral (or asymmetrical) motion of the missile, namely equations (11), (12), (14), (16), and (19). The second group includes those equations which determine the longitudinal (or symmetrical) motion of the missile, namely equations (10), (13), (15), (17), and (18). The two groups of equations will be considered independently.

2.12 Solution of the Lateral Equations of Motion

The lateral equations of motion are:

$$N = I_z \frac{dr}{dt} - (I_x - I_y)qp - I_{xz}\left(\frac{dp}{dt} - rq\right) \tag{11}$$

$$L = I_x \frac{dp}{dt} - (I_y - I_z)\,qr - I_{xz}\left(\frac{dr}{dt} + pq\right) \tag{12}$$

$$\frac{Y'}{m} + g\,(\cos\theta_g \cos\theta_o \sin\phi_o + \sin\theta_g \cos\psi_o \sin\theta_o \sin\phi_o - \sin\theta_g \sin\psi_o \cos\phi_o) = \frac{dv}{dt} - pw + ru \tag{14}$$

$$L_{\delta d}\delta_d = f_1(\phi_o, \psi_\epsilon) \tag{16}$$

$$\psi_\epsilon = \psi_i - \psi_o \tag{19}$$

Equations (11), (12), and (14) can be simplified as follows:

$$N = I_z \frac{dr}{dt} \tag{20}$$

$$L = I_x \frac{dp}{dt} \tag{21}$$

$$\frac{Y'}{m} + g \cos \theta_g \phi_o - g \sin \theta_g \psi_o = \frac{dv}{dt} + vr \tag{22}$$

The above simplification depends upon the following:

(1) The product of inertia is neglected.
(2) The variables ψ_o, ϕ_o, θ_o, r, p, q, v, and w are small.
(3) Products of small terms are neglected.
(4) The sine of a small angle can be replaced by the angle.
(5) The cosine of a small angle can be replaced by unity.
(6) The velocity, u, is essentially constant and equal to V, the steady-state forward velocity. Neglecting the products of inertia is considered permissible in studying the stability of an airplane and, in general, they are not known.

For this system of axes the following relations hold:

$$p = \frac{d\phi_o}{dt} \tag{23}$$

$$r = \frac{d\psi_o}{dt} \tag{24}$$

It is simpler to write the equations of motion using Laplacian transforms. In this case under consideration, the Laplacian form of the equations can be found by replacing " $\frac{d^n}{dt^n}$ " by "s^n". When the frequency response characteristic is calculated the operator "$j\omega$" replaces "s".

It is convenient at this point to expand the moments L and N and the side force, Y', in terms of their important components.

$$L = L_{\delta d}\delta_d + L_v v + L_p p + L_r r \tag{25}$$

$$N = N_{\delta d}\delta_d + N_v v + N_p p + N_r r \tag{26}$$

$$Y' = Y_v v \tag{27}$$

The equations determining the lateral motion of the missile can now be written as follows:

$$Y_v v + mg\cos\theta_g \phi_o - mg\sin\theta_g \psi_o = m(sv + vs\psi_o) \tag{28}$$

$$L_{\delta d}\delta_d + L_v v + L_p s\phi_o + L_r s\psi_o = I_x s^2 \phi_o \tag{29}$$

$$N_{\delta d}\delta_d + N_v v + N_p s\phi_o + N_r s\psi_o = I_z s^2\psi_o \tag{30}$$

$$\delta_d = f_3(\phi_o, \psi_\epsilon) \tag{16}$$

$$\psi_\epsilon = \psi_i - \psi_o \tag{19}$$

In order to solve the above equations, it is necessary to express the function, $f_3(\phi_o, \psi_\epsilon)$, in explicit form. As explained in the introduction, the control surfaces are moved by the servo system which interprets the signal supplied by the homing radar and the signal supplied by the vertical gyro. The vertical gyro measures the roll angle of the missile, ϕ_o, with respect to a reference which is called the roll trim angle, ϕ_t. The difference between ϕ_o and ϕ_t is used to actuate the elevons differentially. The relationship which is maintained is given by equation (31).

$$\delta_d = k_1 G_1 (\phi_t - \phi_o) \tag{31}$$

In equation (31), k_1 is the differential angle of elevons obtained by a unit difference between the roll trim angle ϕ_t and the roll orientation angle ϕ_o. The term G_o is ideally unity. Actually it will include a lag term which is principally a measure of the function of the finite velocity with which the elevons move. The speed at which the elevons move (90 degrees per second) has been made sufficiently high that the lag in reaching a new position does not affect the dynamical response of the system.

In steady flight ϕ_o equals ϕ_t, so that the missile flies at a roll angle equal to the roll trim angle. The roll trim angle ϕ_t is made a function of the error in missile heading as determined by the radar homing system. The relationship between the roll trim angle and the error in yaw heading, is expressed by equation (32).

$$\phi_t = k_2 G_2 \psi_\epsilon \tag{32}$$

The term G_2 in equation (32) may be unity in case lags in the radar and control system can be neglected, or can include a lag term which is a measure of the dynamic characteristics of the radar and control system, or may include certain compensating types of dynamical response functions in order to obtain the correct overall missile performance. Equations (31) and (32) imply that the vertical axis of the gyro is perpendicular to the direction of flight. This is approximately true.

In order to study the stability characteristics of the missile in accordance with the theory developed in Sections 2.2 through 2.8, it is necessary to determine the transfer function of the system. The transfer function for the asymmetric motion of the missile is denoted by $K_a G_a(s)$ and is defined as

$$K_a G_a(s) = \frac{\psi_o(s)}{\psi_\epsilon(s)} \tag{33}$$

It may be determined by solving equations (28), (29), (30), (31), and (32). When this solution is made, it is found that the transfer function is the following:

$$k_a G_a(s) = \frac{A_o}{B_1} k_2 G_2(s) \frac{A_3 s^3 + A_2 s^2 + A_1 s + 1}{B_5 s^5 + B_4 s^4 + B_3 s^3 + B_2 s^2 + s + B_o} \tag{34}$$

In equation (34)

$$K_a = \frac{A_o}{B_1} k_2 \tag{34a}$$

$$G_a(s) = \frac{A_3 s^3 + A_2 s^2 + A_1 s + 1}{B_5 s^5 + B_4 s^4 + B_3 s^3 + B_2 s^2 + s + B_o} \tag{34b}$$

The coefficients A_3 ... and B_5 ... involve complex expressions of the aerodynamic non-dimensional coefficients. For greater usefulness they have been determined in terms of the non-dimensional coefficients rather than in terms of the force and moment terms directly. The relations between the non-dimensional coefficients and the moment and force terms are given in Table (2). The expressions for A_3 ... and B_5 ... are given below:

$$A_o = \frac{2k_1 m^2 g \cos \theta_g}{\rho SV} (C_{n\delta d} C_{lv} - C_{l\delta d} C_{nv}) \tag{35}$$

$$A_1 = \frac{1}{A_o} \left[-\frac{1}{2} k_1 mb C_{yv} (C_{n\delta d} C'_{lp} - C_{l\delta d} C'_{np}) \right] \tag{36}$$

$$A_2 = \frac{1}{A_o} \left[\frac{2k_1 m I_x C_{n\delta d} C_{yv}}{\rho SVb} + \frac{k_1 m^2 b}{\rho SV^2} (C_{n\delta d} C'_{lp} - C_{l\delta d} C'_{np}) \right] \tag{37}$$

$$A_3 = \frac{1}{A_o} \left[\frac{-4k_1 m^2 I_x C_{n\delta d}}{(\rho S)^2 V^3 b} \right] \tag{38}$$

$$B_1 = \frac{1}{2} k_1 bm C_{yv} (C_{n\delta d} C'_{lr} - C_{l\delta d} C'_{nr}) + \frac{2k_1 m^2}{\rho S} (C_{n\delta d} C_{lv} - C_{l\delta d} C_{nv})$$
$$- \frac{bm^2 g \cos \theta_g}{\rho SV^2} (C'_{nr} C_{lv} - C'_{lr} C_{nv}) - \frac{bm^2 g \sin \theta_g}{\rho SV^2} (C'_{np} C_{lv} - C'_{lp} C_{nv}) \tag{39}$$

$$B_o = \frac{1}{B_1} \left[\frac{2k_1 m^2 g \sin \theta_g}{\rho SV} (C_{n\delta d} C_{lv} - C_{l\delta d} C_{nv}) \right] \tag{40}$$

$$B_2 = \frac{1}{B_1} \left[-\frac{mb^2 C_{yv}}{4V} (C'_{np} C'_{lr} - C'_{lp} C'_{nr}) \right.$$
$$+ \frac{4m^2 g I_z C_{lv} \cos \theta_g}{(\rho S)^2 bV^3} - \frac{4m^2 g I_x C_{nv} \sin \theta_g}{(\rho S)^2 bV^3} - \frac{m^2 b}{\rho SV} (C'_{np} C_{lv} - C'_{lp} C_{nv})$$
$$\left. - \frac{k_1 bm^2}{\rho SV^2} (C_{n\delta d} C'_{lr} - C_{l\delta d} C'_{nr}) + \frac{2k_1 m I_z C_{yv} C_{l\delta d}}{\rho SbV} \right] \tag{41}$$

$$B_3 = \frac{1}{B_1}\left[\frac{-mI_xC_{yv}C'_{nr}}{\rho SV^2} - \frac{mI_zC_{yv}C'_{lp}}{\rho SV^2} + \frac{(mb)^2}{2\rho SV^3}(C'_{np}C'_{lr} - C'_{lp}C'_{nr}) - \frac{4k_1m^2I_zC_{l\delta d}}{(\rho S)^2 bV^3} - \frac{4m^2I_xC_{nv}}{(\rho SV)^2 b}\right] \quad (42)$$

$$B_4 = \frac{1}{B_1}\left[\frac{4mI_xI_zC_{yv}}{(\rho Sb)^2V^3} + \frac{2m^2I_xC'_{nr}}{(\rho SV^2)^2} + \frac{2m^2I_zC'_{lp}}{(\rho SV^2)^2}\right] \quad (43)$$

$$B_5 = \frac{1}{B_1}\left[\frac{-8m^2I_xI_z}{(\rho S)^3b^2V^5}\right] \quad (44)$$

The numerical solution of the lateral equations is given in Section 2.14.

2.13 Solution of the Longitudinal Equations of Motion

The longitudinal equations of motion are:

$$M = I_y\frac{dq}{dt} - (I_z - I_x)rp - I_{xz}(r^2 - p^2) \quad (10)$$

$$\frac{X'}{m} + g(\cos\psi_o\cos\theta_o\sin\theta_g - \sin\theta_o\cos\theta_g) = \frac{du}{dt} - rv - qw \quad (13)$$

$$\frac{Z'}{m} + g(\cos\theta_g\cos\theta_o\cos\phi_o + \sin\theta_g\sin\psi_o\sin\phi_o + \sin\theta_g\cos\psi_o\sin\theta_o\cos\phi_o) = \frac{dw}{dt} - qu + pv \quad (15)$$

$$Z_{\delta a} = f_2(\theta_o, \theta_\epsilon) \quad (17)$$

$$\theta_\epsilon = \theta_i - \theta_o \quad (18)$$

Equations (10), (13), and (15) can be further simplified by making the following assumptions:

(1) The product of inertia is neglected.

(2) The angular displacements and rates are small and the products of small quantities can be neglected.

(3) The sine of a small angle can be replaced by the angle.

(4) The cosine of a small angle can be replaced by unity.

(5) The velocity of the missile in space is constant.

The analysis of the system being discussed is not particularly restricted by the above assumptions. The first three assumptions have been discussed previously. The assumption that the velocity of the missile remains constant removes the phugoid oscillation from the behavior of the missile. The phugoid oscillation is important in the

motion of an uncontrolled airplane but is relatively unimportant in the motion of a controlled airplane for the following reason:

The effect of the phugoid oscillation on an uncontrolled airplane is to change the speed and thus to change the angular orientation of the airplane in space. In a controlled airplane a properly designed control system can maintain a constant orientation of the airplane in spite of the phugoid because of the relative slowness with which the phugoid acts.

The change in speed of the airplane is important, however, in that it changes the stability characteristics of the airplane since they are in general functions of the velocity of the airplane. Since the velocity change is slow and, in the case of Bat, increases rather uniformly, the dynamical performance of the missile can be determined at various points in the speed range over which the missile operates. Thus, in the case of Bat, the velocity of the missile varies from approximately 300 feet per second at the instant that it is dropped from the carrying airplane to approximately 500 feet per second at the end of the trajectory. In general, the stability at the lower speeds is more critical and if the parameters are selected at this speed to obtain good dynamical characteristics, the dynamical response at higher speeds will be satisfactory. In the analysis presented in this report, the stability characteristics are determined for several velocities.

Under the above assumptions equation (13) is not needed to determine stability, and equations (10) and (15) can be simplified as follows:

$$M = I_y \frac{dq}{dt} \tag{45}$$

$$\frac{Z'}{m} + g \cos \theta_g + g \sin \theta_g \theta_o = \frac{dw}{dt} - Vq \tag{46}$$

Since the moment, M, and vertical force, Z', are commonly expressed in terms of angle of attack, α, it is convenient to introduce it at this point. The angle of attack is defined as

$$\alpha = \theta_o + \frac{w}{V} \tag{47}$$

The moment, M, is expressed as the following function of its important components:

$$M = M_\alpha \alpha + M_{\dot{\alpha}} \dot{\alpha} + M_{\delta a} \delta_a + M_{\dot{\delta a}} \dot{\delta}_a + M_q q \tag{48}$$

Correspondingly, the vertical force Z' may be written

$$Z' = -mg \cos \theta_g + Z_\alpha \alpha + Z_{\delta a} \delta_a \tag{49}$$

If Laplacian notation is employed, and the fact that in the coordinate system chosen

$$q = \frac{d\theta_o}{dt},$$

equations (45), (46), (48), (49), and (50) can be combined and the longitudinal equations of motion written:

$$M_\alpha \alpha + M_{\dot{\alpha}} s\alpha + M_{\delta a}\delta_a + M_{\dot{\delta a}} s\delta_a + M_q s\theta_o = I_y s^2\theta_o \tag{51}$$

$$Z_\alpha \alpha + Z_{\delta a}\delta_a + mg \sin \theta_g \theta_o = msw - Vs\theta_o \tag{52}$$

$$\alpha = \theta_o + \frac{w}{V} \tag{47}$$

$$\delta_a = f_4(\theta_o, \theta_\epsilon) \tag{53}$$

$$\theta_\epsilon = \theta_i - \theta_o \tag{18}$$

The above equations determine the longitudinal or symmetric performance of the missile. In order to solve the above equations, it is necessary to express the function, $f_4(\theta_o,\theta_\epsilon)$, in explicit form. As explained in the introduction, the control surfaces are moved by the servo system which interprets the signal supplied by the homing radar and the signal supplied by the vertical gyro. The vertical gyro measures the pitch angle of the missile, θ_o, with respect to a reference which is called the pitch trim angle, θ_t. The difference between θ_o and θ_t is used to change the average position of the elevons. The relationship which is maintained is given by equation (54).

$$\delta_a = k_3 G_3(\theta_t - \theta_o) \tag{54}$$

In equation (54), k_3 is the change in average position of the elevons obtained by a unit difference between the pitch trim angle θ_t and the pitch orientation angle θ_o. The term G_3 is ideally unity. Actually it will include a lag term which is principally a measure of the function of the finite velocity with which the elevons move. The speed at which the elevons move (90 degrees per second) has been made sufficiently high that the lag in reaching a new position does not affect the dynamical response of the system.

In steady flight θ_o is approximately equal to θ_t so that the missile flies at a pitch angle equal to the pitch trim angle. The pitch trim angle θ_t is made a function of the error in missile heading as determined by the radar homing system. The relationship between the pitch trim angle and the error in pitch heading is expressed by equation (55).

$$s\theta_t = k_4 G_4 \theta_\epsilon \tag{55}$$

The term G_4 in equation (55) may be unity in case lags in the radar and control system can be neglected, or can include a lag term which is a measure of the dynamic characteristics of the radar and control system, or may include certain compensating types of dynamical response functions in order to obtain the correct overall missile performance.

In order to study the stability characteristics of the missile in accordance with the theory developed in Sections 2.2 - 2.8, it is necessary to determine the transfer function of the system. The transfer function for the symmetric motion of the missile is denoted by $K_s G_s(s)$ and is defined as

$$K_s G_s(s) = \frac{\Theta_o(s)}{\Theta_\epsilon(s)} \tag{56}$$

It may be determined by solving equations (51), (52), (47), (54), and (55). When this solution is made, it is found that the transfer function is the following:

$$K_s G_s(s) = \frac{D_o}{E_o} k_4 G_4 \quad \frac{D_2 s^2 + D_1 s + 1}{s(E_3 s^3 + E_2 s^2 + E_1 s + 1)} \tag{57}$$

The coefficients D_2...and E_3 ...involve complex expressions of the aerodynamic non-dimensional coefficients. For greater usefulness they have been determined in terms of the non-dimensional coefficients rather than in terms of the force and moment terms directly. The relations between the non-dimensional coefficients and the moment and force terms are given in Table 2. The expressions for D_2 ... and E_3 ... are given below:

$$D_2 = \frac{mc}{\frac{1}{2}\rho S V^2} \quad \frac{C_{m\dot{\delta}a}}{C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}} \tag{58}$$

$$D_1 = \frac{c}{V} \frac{C_{l\delta a} C_{m\dot{\alpha}} - C_{m\dot{\delta}a} C_{l\alpha}}{C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}} \tag{59}$$

$$D_o = (\tfrac{1}{2}\rho S)^2 V^3 c \, (C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}) \, k_3 \tag{60}$$

$$E_3 = \frac{I_y m}{\frac{1}{2}\rho S V c} \quad \frac{1}{k_3(\frac{1}{2}\rho S V^2)(C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}) - mg \sin\theta_g C_{m\alpha}} \tag{61}$$

$$E_2 = \frac{cm}{\frac{1}{2}\rho S V^2 k_3 (C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}) - mg \sin\theta_g} \left[- 2C_{m\dot{\alpha}} - C_{mq} + k_3 C_{m\dot{\delta}a} - \frac{I_y C_{l\alpha}}{c^2 m} \right] \tag{62}$$

$$E_1 = \frac{1}{\frac{1}{2}\rho S V^2 k_3 (C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}) - mg \sin\theta_g} \left[\frac{1}{2}\rho S V c \left\{ C_{mq} C_{l\alpha} + k_3 (C_{l\delta a} C_{m\dot{\alpha}} - C_{m\dot{\delta}a} C_{l\alpha}) \right\} + mV(-2C_{m\alpha} + k_3 C_{m\delta a}) - \frac{cmg}{V} \sin\theta_g C_{m\dot{\alpha}} \right] \tag{63}$$

$$E_o = (\tfrac{1}{2}\rho S V c) \left[k_3 (\tfrac{1}{2}\rho S V^2) (C_{l\delta a} C_{m\alpha} - C_{m\delta a} C_{l\alpha}) - mg \sin\theta_g C_{m\alpha} \right] \tag{64}$$

The numerical solution of the symmetric equations is given in Section 2.14.

2.14 Numerical Solution and Study of the Lateral Equations of Motion

When the study of the lateral equations of motion was undertaken, the initial purpose was to determine the correct values of the roll sensitivity factor, k_1, and the yaw sensitivity factor, k_2. These factors have been defined in Section 2.12. Other results that were sought were (1) to determine the minimum speed with which the elevons could be

moved without reducing the inherent response possible in the missile and (2) to determine the maximum lag allowable in the radar circuits without affecting adversely the response of the missile. As the study progressed, however, two points became apparent; first, that the missile had insufficient lateral weathercock stability and second, that an integral circuit (to be explained later) was necessary to obtain the desired missile accuracy. The study lead to recommendations that the effective vertical fin area be increased. This was done and flights were made with a lateral weathercock stability approximately twice that of the original missile. For the sake of clarity, therefore, the study of the lateral response of the missile will be undertaken in the following order:

1. Study of the effect of vertical fin area.
2. Study of the effect of yawing moment produced by differential displacement of the elevons.
3. Determination of the roll sensitivity factor.
4. Study of the effect of missile velocity.
5. Study of the effect of integral control.
6. Determination of allowable lags in the radar circuit and of a finite speed of moving the elevons.

The values of the missile constants, aerodynamic constants, mass, etc., and the calculated values of the parameters for equation (34) are shown in Tables (3), (4), and (5) for the cases with which the study is concerned.

1. Study of the Effect of Vertical Fin Area

In order to obtain high accuracy, it is desirable to have as high a yaw sensitivity factor, k_2, as possible. It was found that in Bat this sensitivity factor was primarily limited by the yawing moment produced by a differential motion of the elevons. The sign of this yawing moment may be either positive or negative depending upon whether the average position of the elevons is above or below the streamline position. In aircraft terminology, this yawing moment is called either favorable or adverse depending upon whether the yawing moment is in a direction to help the airplane turn in the desired direction or to oppose its turning in the desired direction. Actually a study of the dynamics of the situation revealed that the presence of the yawing moment unfavorably effected the response of this moment whether it was "adverse" or "favorable". This yawing moment can be almost cancelled by attaching vertical flippers to the missile and making their displacement the correct function of both the average and differential displacement of the elevons. This would complicate both the control system and the manufacture of the airframe.

A second method of reducing the effect of this yawing moment is to increase the lateral weathercock stability of the missile. This was finally decided to be the best solution. Figure 2.18 (see end of chapter) shows the loci plots of the lateral transfer function ($G_a(j\omega)$ for 3 values of fin area. These are plots of equation (34b) for $G_2(s)$ = unity and "s" replaced by "$j\omega$". The curve corresponding to unity fin area is for the original value of fin area of the Bat airframe. The curves marked fin area equals 2 or 4 are plotted for fin areas of twice and four times the original area. The coefficient of the yawing moment produced by differential elevon displacement, $C_{n\delta d}$, is taken as

- .013. This was the most unfavorable value of this coefficient that was expected to be encountered. Corresponding curves for other values were plotted but are not included in this report. The lateral sensitivity possible for each value of fin area was calculated as explained in Section 2.7, that is, by locating a circle corresponding to a peak value of the sinusoidal response of the missile of 1.33. The sensitivities obtained are listed in the summary table on Figure 2.18. It is seen that if the original value of fin area is doubled, the possible sensitivity is almost doubled and if the fin area is quadrupled, the possible sensitivity is almost doubled again. It proved to be impractical to use a fin area of four times the original area but it was possible to use fins whose effective area was twice the original area of the fins. Actually the fins were not much larger in physical size but by making them of triangular shape rather than circular and moving them a few inches to the rear, the effective area was doubled.

2. Study of the Effect of Yawing Moment Produced by Differential Displacement of The Elevons

A study of the wind tunnel tests on Bat and the similar airframe, Pelican, indicated that the yawing moment produced by differential elevon displacement would be zero for a value of δ_a approximately 5 degrees down and would range from approximately -.013 to + .013 for the range of average elevon position expected to be encountered in flight. The transfer loci corresponding to these three cases ($C_{n\delta d}$ = - .013, 0, and +.013) for an effective vertical fin area twice the original value is shown in Figure 2.19. The summary of the sensitivities possible for the three cases is given in the summary table on the curve sheet It is clear that a small value of $C_{n\delta d}$ is very desirable. When $C_{n\delta d}$ = + .013 a peculiar form of locus results and the possible sensitivity is reduced to about eleven degrees of roll per degree of yaw error. If $C_{n\delta d}$ = - .013 the locus is not quite as peculiar in shape but the sensitivity must be reduced to less than eight degrees of roll per degree of yaw error.

The curves of Figures 2.18 and 2.19 are drawn for a flight speed of 400 feet per second. At a flight velocity of 300 feet per second, which is the speed at the point at which the Bat is dropped from the carrying airplane, the permissible sensitivity is even less. (See Figure 2.21). The small permissible sensitivity is one of the reasons why it was necessary to incorporate an integral control circuit in the system. It was decided that the system with this integral control network was operable and should give good results. However, it is important to notice that if the missile had no yawing moment produced by differential elevons, the sensitivities in the system might have been more than doubled.

3. Determination of the Roll Sensitivity Factor

The roll sensitivity factor is a factor that controls the amount of differential elevon displacement that is obtained from the control system for an error in the roll orientation of the missile. For a high degree of stabilization this factor should be as large as possible since it permits the system to strongly resist rolling moments produced by air disturbances. It should also be high to minimize heading errors produced by asymmetries in the design of the missile. Thus if an asymmetry in the airframe produces an effective roll torque on the missile which must be overcome by a differential elevon displacement, such a displacement may be obtained only by changing the roll

orientation of the missile. If the factor k_1 is large, the effective change in roll orientation that the missile must take up in order to correct for such asymmetries is small. If the sensitivity factor, k_1, is too large, however, oscillations result primarily in roll but occurring also in yaw, whenever the missile receives a roll air disturbance. Figure 2.20 presents plots of the transfer loci of the system $G_a(j\omega)$, (equation (34b) when $G_2(j\omega) = 1$) for sensitivity factors of 0.25, 0.50, and 1. A factor of 1 signifies that 1° of differential elevon is obtained for a degree difference between the roll trim angle and the roll orientation angle, in accordance with equation (31). One degree of differential elevon is defined as 1/2 degree displacement on each elevon, in opposite directions. A roll sensitivity factor of 1/2 was chosen as the optimum. This value of sensitivity factor permits twice the yaw sensitivity factor that is obtainable for a roll factor of $k_1 = 1$. If k_1 were reduced to .25, the yaw sensitivity factor could be increased by another 50 per cent, but only at the expense of sacrificing response to roll disturbances. Also, the yaw error produced by asymmetries in the airframe is inversely proportional to the product of the roll and yaw sensitivity factors and this product is a maximum for a roll sensitivity factor, k_1, of approximately 1/2. If $C_{n\delta d}$ were zero, k_1 could have been considerably increased. The roll sensitivity factor could also have been increased if the missile possessed larger inherent damping in roll (corresponding to a larger C_{lp}). Larger damping, however, can only be obtained by a larger wing spread or by including rate terms in the servo control unit itself.

4. Study of the Effect of Missile Velocity

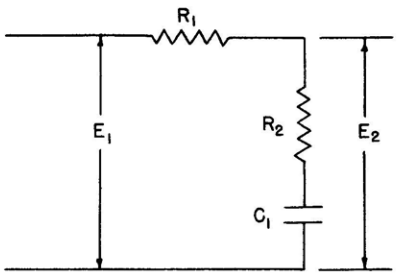
It was found that the missile velocity varies from approximately 300 feet per second when the missile is released from the carrying airplane to as much as 500 feet per second when it reaches the target. Figure 2.21 is a plot of the lateral transfer function $G_a(j\omega)$ for speeds of 300, 400, and 500 feet per second. The system is more stable and permits a higher sensitivity at higher speeds which is fortunate since it is important to obtain the greatest stability at the end of the trajectory. However, it is important that the system be stable for all parts of the course, and, therefore, the stability constants were selected to obtain good stability at speeds of 300 feet per second. The summary on Figure 2.21 indicates that a sensitivity of 4.6 degrees of roll per degree of yaw error is permissible and this was the factor which was selected for the design. It would be preferable if this factor were at least three times this size. However, the integral control circuit to be described in the next section compensated to a large extent for this low factor.

One reason for making sure that the missile is stable throughout its flight is the following: If the missile were unstable at the early part of the flight the larger drag produced by the oscillation of the missile might prevent the speed of the missile from increasing sufficiently to make it stable. In such cases instability might exist throughout the flight and the miss might be excessive.

5. Study of the Effect of Integral Control

It early appeared that the integral control circuit should be incorporated in the yaw channel for two reasons: (1) to increase the effective yaw sensitivity and (2) to provide the missile with a certain amount of "memory" to enable it to handle intermittent radar signals and to continue on a more correct course for the last part of the trajectory

during which the radar signal fails. The type of circuit employed is shown in Figure 2.15. This circuit permitted the effective sensitivity to be increased by a factor of



$$\frac{E_2}{E_1}(j\omega) = \frac{1 + \tau_1 j\omega}{1 + \alpha_1 \tau_1 j\omega}$$

$$\tau_1 = R_2 C_1$$

$$\alpha_1 = \frac{R_1 + R_2}{R_2}$$

Figure 2.15

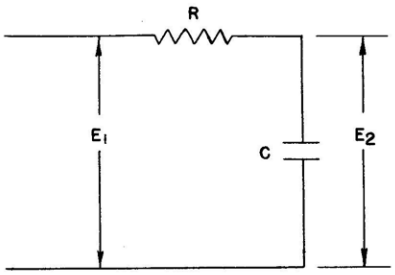
Integral Control Circuit

about 10. Thus instead of obtaining 4.7 degrees of roll per degree of yaw error, the missile was capable of flying with a sensitivity of about 45 degrees of roll per degree of yaw error. This high sensitivity was obtained with a time constant of about 10 seconds which means that the sensitivity of 45 degrees of roll per degree of yaw error does not apply to rapidly changing transients. This high sensitivity factor enables the missile to follow a true pursuit course much more accurately and thus to reduce errors arising from limited acceleration at the last part of the flight. The second reason for employing this type of circuit is that it enables the missile to remember its correct course if the homing signal fails. Without such a circuit the missile always attempts to fly a straight line course in the absence of homing signals. If the radar signals fail with the integral circuit, the missile attempts to fly a curved course whose curvature is equal to the curvature of the path it was following when operating under homing signals.

The transfer locus of the system with the integral control circuit incorporated is shown in Figure 2.22. This curve is a plot of $G_a(j\omega)$ of equation (34b) in which the function $G_2(j\omega)$ is given on the curve. If the time constant is too long, the system requires an excessive time to reach its ultimate sensitivity. If the time constant is too short the maximum value of the output to the input is excessive. The time constant in the integral control circuit is made as small as possible without increasing excessively the maximum value of the output to input ratio. A time constant of τ_1 equal to one second was chosen and the allowable value of M was increased to 1.7.

6. Determination of Allowable Lags in the Radar Circuits

The time constants of the radar circuit can be approximated by the circuit shown in Figure 2.16. The transfer function of this circuit is given by $\frac{1}{1 + j\omega\tau_d}$ in which τ_d is equal to the product RC. The effect of the circuit on the transfer locus of the system is to shift the locus toward the unity point, thus destabilizing the system unless



$$\frac{E_2}{E_1}(j\omega) = \frac{1}{1 + j\omega\tau_d}$$

$$\tau_d = RC$$

Figure 2.16

Time Lag Circuit

the sensitivity is reduced in which case a loss of sensitivity is encountered. The maximum time constant that this circuit may have without affecting the dynamical response of the missile is easily determined by requiring the shift that the circuit produces in the system locus to be not more than approximately five degrees at the resonant frequency. Thus in Figure 2.21 the locus drawn for a speed of 300 feet per second has a resonant frequency of $\omega = 6.3$. If the preceding criterion is applied, and the phase shift is specified as five degrees at this frequency, the time constant of the radar circuit should be less than about .02.

A similar type of calculation can be employed to determine the minimum speed with which the elevons can move without affecting the dynamical response of the system. This speed was calculated to be 150 degrees per second. Tests on the flight table indicated that this speed could be reduced to ninety degrees per second. Accordingly, this figure was decided upon, and the test and production models of the SWOD (Control System) Mk 18 Mod 0 move the elevons at ninety degrees per second.

2.15 Numerical Solution and Study of the Longitudinal Equations of Motion

When the study of the longitudinal equations of motion of Bat was undertaken, the initial purpose was to determine the optimum values of the average elevon displacement factor, k_3, and the pitch sensitivity factor, k_4. These factors have been defined in Section 2.13. Other results that were sought were (1) to determine the minimum speed with which the elevons could be moved without diminishing the dynamic characteristics of the the missile and (2) to determine the maximum lag allowable in the radar circuits that would not adversely effect the response of the missile. As the study progressed, it became evident that a compensating circuit was required in order to obtain the necessary missile characteristics in pitch. A compensating circuit was chosen and the parameters of this circuit were determined from a study of the longitudinal transfer function of the system. The study of the system also revealed that the pitching moment produced by a

change in the elevon average position was very important in determining stability in performance of the system.

The missile airframe is designed so that changes in average elevon position will produce no pitching moment over a small range of elevon positions. On either side of this range, however, a pitching moment is produced by the elevons and it was found that this factor was important in determining overall response in pitch. Accordingly the study of the symmetric response of the missile is discussed in the following order:

1. Study of the effect of pitching moment produced by a change in the elevon average position.
2. Determination of the average elevon displacement sensitivity factor, k_3.
3. Determination of the constants of the compensating network.
4. Study of the effect of the pitching moment produced by a change in the elevon average position with a compensating network.
5. Determination of the allowable lags in the radar signal and the effect of finite speed of moving the elevons.

The values of the missile constants, aerodynamic constants, mass, etc., and the parameters for equation (57) are given in Tables (6), (7), and (8).

1. Study of the effect of pitching moment produced by a change in the elevon average position.

As explained in the preceding paragraph the constant $C_{m\delta a}$ may be zero, negative, or positive depending upon the location of the mean position of the elevons. Figure 2.23 is a plot of the transfer function $G_s(j\omega)$ for 3 values of $C_{m\delta a}$ covering the range that is expected in practice. The function $G_4(j\omega)$ is set equal to 1 for the plots on Figure 2.23. A table summarizing possible sensitivities is presented on the same curve sheet. It is seen that $C_{m\delta a}$ positive, corresponding to upward elevon position, permits a much higher sensitivity than if $C_{m\delta a}$ is zero or negative. When $C_{m\delta a}$ is positive, the sign of the coefficient is such that the pitching moment produced by a motion of the elevons is in the same direction as the pitching moment that is produced by the change in angle of attack caused by the change in life characteristics. When $C_{m\delta a}$ is negative, the locus is swung toward the negative unity point and the sensitivity must be reduced. The shape of the locus makes it desirable to sacrifice somewhat in M value (the maximum value of the ratio of the output to the input) in order to increase permissible sensitivities. It also appears necessary because of the low sensitivity factors that are required by the negative coefficient if the M value is kept reasonable to design the system primarily for a $C_{m\delta a}$ of zero, and take advantage of the fact that it is generally unnecessary to work in a region in which $C_{m\delta a}$ is negative. When $C_{m\delta a}$ is negative, the damping characteristics will be much poorer but as long as the system is not actually unstable it seems permissible.

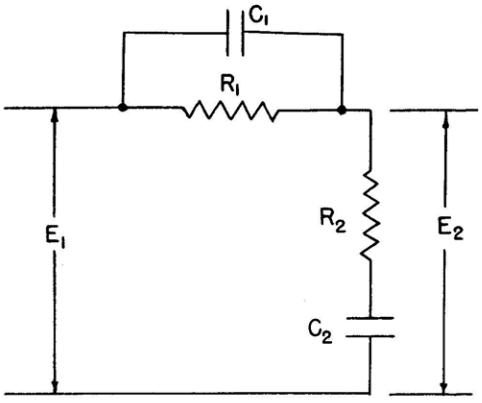
2. Determination of the average elevon displacement sensitivity factor, k_3.

The factor k_3 relates the change in elevon average position produced by the unit change and difference between the pitch trim angle and the pitch orientation angle. The factor is defined by equation (54). For a high degree of stabilization, this factor

should be as large as possible since it permits the missile to resist strongly pitching moments produced by air disturbances. If the factor is too large transients resulting from air disturbances in pitch are excessively underdamped. Figure 2.24 is a plot of the longitudinal transfer function for 3 values of k_3. It is seen that larger values of k_3 permit a higher value of pitch sensitivity factor k_4. However, it is also necessary that the pitch response characteristics of the missile be satisfactory in the absence of radar information and studies made of this case which are not included in this report indicated that the factor k_3 should probably be kept under 6. The system mechanism by which k_3 is controlled also operates better if k_3 is of the order of 4 rather than of the order of 6. For these reasons the value of k_3 equal to 4 was chosen for the missile.

3. Determination of the constants of the compensating network

The preceding two studies indicated that a sensitivity factor of $k_4 = 0.9$ could be employed if the factor k_3 was set equal to 4 and the missile flew so that the factor $C_{m\delta a}$ was not negative. A factor of k_4 equal to 0.9 signifies that the missile will turn at a rate of 0.9 degree per second if a pitch error of one degree exists in the system in pitch heading. A factor of this value was considered too small to permit the necessary accuracy in pitch. It is important that as much accuracy as possible be obtained in pitch because of the fact that the missile approaches the target on a relatively flat glide path and a small error in heading produces a large miss. Therefore, it was thought necessary to attain a sensitivity almost twice as high as 0.9 degree per second per degree. The sensitivity can be increased by employing a compensation lead integral network. The type of network employed is illustrated by Figure 2.17. The transfer function of this network



$$\frac{E_2}{E_1}(j\omega) = \frac{(\alpha\tau_d\tau_i)\ (j\omega)^2 + (\alpha\tau_d + \tau_i)j\omega + 1}{(\alpha\tau_d\tau_i)\ (j\omega)^2 + (\tau_d + \alpha\tau_i)j\omega + 1}$$

$$\alpha\tau_d\tau_i = C_1C_2R_1R_2$$

$$\alpha\tau_d + \tau_i = C_1R_1 + C_2R_2$$

$$\tau_d + \alpha\tau_i = C_1R_1 + C_2R_1 + C_2R_2$$

Figure 2.17

Combination Lead and Integral Network

is given in Figure 2.17. The effect of this network on the longitudinal transfer function of the system is indicated by Figure 2.25. It is seen that values of $\tau_d = .26$ and $\tau_i = 13.1$ gave considerably better results than $\tau_d = .13$ and $\tau_i = 6.56$. However, the larger time constants require capacitors of twice the size and space was sufficiently crowded that it was felt that the smaller time constants should be employed. Smaller time constants permitted a "proportional sensitivity" of 1.6 degree per second per degree and an ultimate sensitivity of about 4 degrees per second per degree. The sensitivity in response to sudden transients is of the order of 4.6 but the sensitivity in response to a constant error is of the order of 4.8. The compensating network thus permits a much higher pitch sensitivity to be employed with a consequent reduction in miss.

4. Study of the effect of the pitching moment produced by a change in the elevon average position with a compensating network

Figure 2.26 is a plot of the effect of varying $C_{m\delta a}$ on the transfer locus of the system including the compensating network. It is seen that the performance of the system is satisfactory except for $C_{m\delta a} = -0.4$, in which case only marginal stability is possible. Since it was not expected that this value of the capacitance $C_{m\delta a}$ would be encountered for more than a very short time in flight, this adjustment was considered to be satisfactory.

5. Determination of the allowable lags in the radar signal and the effect of finite speed of moving the elevons

The effect of lags in the radar circuit and of finite speed of moving elevons was determined by the same methods indicated under the discussion relating to the lateral equations of motion. It was found that essentially the same specifications existed for both the pitch and yaw channels.

Table I

Definition of Symbols

Symbol	Definition	
L	Moment about OX (rolling moment)	Positive: Starboard wing down.
M	Moment about OY (pitching moment)	Positive: Nose up.
N	Moment about OZ (yawing moment) (See Figure 2.14)	Positive: Nose to starboard.
X'	Force along OX.	Positive: Forward direction.
Y'	Force along OY.	Positive: Starboard direction.
Z'	Force along OZ.	Positive: Downward direction.
I_x	Moment of inertia about OX	
I_y	Moment of inertia about OY	
I_z	Moment of inertia about OZ	
I_{xz}	Product of inertia about OY	
p	Rate of angular turning about OX	Positive direction: Same as L, M, N.
q	Rate of angular turning about OY	
r	Rate of angular turning about OZ	
u	Linear velocity along OX	Positive direction: Same as X', Y', Z'.
v	Linear velocity along OY	
w	Linear velocity along OZ	
δ_a	Change in average position of elevons	
δ_d	Differential position of elevons	
α	Angle of attack	
τ_i	Time constant in supplementary circuit	
τ_i	Time constant in supplementary circuit	

Table I

Definition of Symbols

(Cont.)

Symbol	Definition
θ_i	Pitch angle from reference axis to target axis.
θ_o	Pitch angle from reference axis to body axis.
θ_ϵ	Pitch error = $\theta_i - \theta_o$.
θ_t	Pitch trim angle.
θ_g	Glide angle measured from horizontal.
ψ_i	Yaw angle from reference axis to target. (See Figure 2.11)
ψ_o	Yaw angle from reference axis to body axis.
ψ_ϵ	Yaw error = $\psi_i - \psi_o$.
ϕ_o	Roll orientation of missile (zero if wing are level).
ϕ_t	Roll trim angle.
m	Mass of missile.
ρ	Air density.
S	Wing area.
b	Span
c	Mean aerodynamic chord = S/b.
g	Acceleration of gravity.
V	Total forward velocity.
$K_aG_a(j\omega)$	Transfer function for the asymmetric motion of the missile.
$K_sG_s(j\omega)$	Transfer function for the symmetric motion of the missile.
$G_1(j\omega)$	See equation (31) and text following.
$G_2(j\omega)$	See equation (32) and text following.
$G_3(j\omega)$	See equation (54) and text following.
$G_4(j\omega)$	See equation (55) and text following.

Table I

Definition of Symbols

(Cont.)

Symbol	Definition
k_1	Roll sensitivity factor.
k_2	Yaw sensitivity factor.
k_3	Average elevon displacement sensitivity factor.
k_4	Pitch sensitivity factor.
A_3---A_o	Parameters of transfer functions.
B_5---B_o	Parameters of transfer functions.
D_2---D_o	Parameters of transfer functions.
E_3---E_o	Parameters of transfer functions.

Symbol of Derivative And Its Non-Dimensional Coefficient	Definition
$L_{\delta d}$, $C_{l\delta d}$	Rolling moment produced by a unit differential deflection of the elevons.
L_p, C'_{lp}	Rolling moment produced by a unit rate of turning about OX.
L_r, C'_{lr}	Rolling moment produced by a unit rate of turning about OZ.
L_v, C_{lv}	Rolling moment produced by a unit rate of displacement along OY.
$M_{\delta a}$, $C_{m\delta a}$	Pitching moment produced by a unit change in average elevon position.
M_α, $C_{m\alpha}$	Pitching moment produced by a unit change in angle of attack.
$M_{\dot{\delta}e}$, $C_{m\dot{\delta}e}$	Pitching moment produced by a unit rate of change of δ_a.
$M_{\dot{\alpha}}$, $C_{m\dot{\alpha}}$	Pitching moment produced by a unit rate of change of angle of attack.
M_q, C_{mq}	Pitching moment produced by a unit rate of turning about OY.
$N_{\delta d}$, $C_{n\delta d}$	Yawing moment produced by a unit differential elevon angle.
N_p, C'_{np}	Yawing moment produced by a unit rate of turning about OX.
N_r, C'_{nr}	Yawing moment produced by a unit rate of turning about OZ.
N_v, C_{nv}	Yawing moment produced by a unit rate of displacement along OY.

Symbol of Derivative And Its Non-Dimensional Coefficient	Definition
$Z_{\delta a}$, $C_{l\delta a}$	Vertical force produced by a unit change in δ_a.
Z_{α}, $C_{l\alpha}$	Vertical force produced by a unit change in angle of attack.
Y_v, C_{yv}	Side force produced by a unit rate of displacement along OY.

Table 2

Relations Between Non-Dimensional Coefficients and the Moment and Force Terms

$L_{\delta d} = \frac{1}{2}\rho SV^2 b \; C_{l\delta d}$	$N_{\delta d} = \frac{1}{2}\rho SV^2 b \; C_{n\delta d}$
$L_p = \frac{1}{4}\rho SVb^2 \; C'_{lp}$	$N_p = \frac{1}{4}\rho SVb^2 \; C'_{np}$
$L_r = \frac{1}{4}\rho SVb^2 \; C'_{lr}$	$N_r = \frac{1}{4}\rho SVb^2 \; C'_{nr}$
$L_v = \frac{1}{2}\rho SV^2 b \; C_{lv}$	$N_v = \frac{1}{2}\rho SV^2 b \; C_{nv}$
$M_{\delta a} = \frac{1}{2}\rho SV^2 c \; C_{m\delta a}$	$Z_{\delta a} = \frac{1}{2}\rho SV^2 \; C_{l\delta a}$
$M_\alpha = \frac{1}{2}\rho SV^2 c \; C_{m\alpha}$	$Z_\alpha = \frac{1}{2}\rho SV^2 \; C_{l\alpha}$
$M_{\dot{\delta a}} = \frac{1}{2}\rho SVc^2 \; C_{m\dot{\delta a}}$	
$M_{\dot{\alpha}} = \frac{1}{2}\rho SVc^2 \; C_{m\dot{\alpha}}$	$Y_v = \frac{1}{2}\rho SV^2 \; C_{yv}$
$M_q = \frac{1}{2}\rho SVc^2 \; C_{mq}$	

Table 3

Values Invariant for All Lateral Studies

$I_x = 45$ slug ft^2	$C'_{np} = -.01$
$I_z = 200$ slug ft^2	$C_{l\delta d} = +.036$
$m = 50$ slugs	$C'_{lp} = -0.3$
$\rho = .00237$ slug/ft^3	$C'_{lr} = +.066$
$s = 24.4$ ft^2	$C'_{lv} = -.000425$
$b = 10.0$ ft	$C_{yv} = -0.003$
$\theta_g = 12.5$ degrees	
$g = 32.2$ ft/sec^2	

Table 4

Values Varying for Different Studies (Lateral)

Study No. 1			Study No. 2			Study No. 3			Study No. 4			Study No. 5
Fin Area			$C_{n\delta d}$			k_1			V			$G_2(j\omega) \neq 1$
1	2	4	0	-.013	+.013	0.25	0.50	0.10	300	400	500	$\tau_1 = 1.0$ $\tau_1 = 9.4$
C'_{nr}			$C'_{nr} = -0.4$									
-0.2	-0.4	-0.8										
C'_{nv}			$C'_{nv} = \frac{+.34}{400}$									
$\frac{+.17}{400}$	$\frac{+.34}{400}$	$\frac{+.68}{400}$										

The following values of fin area, $C_{n\delta d}$, k_1, V, and $G_2(j\omega)$ are used in all studies except those in which they are specifically indicated as varying in the above table:

Fin Area	= 2
$C_{n\delta d}$	= -.013
k_1	= 0.50
V	= 400
$G_2(j\omega)$	= 1

Table 5 Calculated Values of the Parameters (Lateral)

	Study No. 1			Study No. 2		
	Fin Area			$C_{n_{\delta d}}$		
	1	2	3	0	-.013	+.013
A_o	-3.32×10^{-2}	-8.54×10^{-2}	-18.9×10^{-2}	-1.04×10^{-1}	-8.54×10^{-2}	-1.23×10^{-1}
A_1	-4.79×10^{-2}	-1.88×10^{-2}	-0.84×10^{-2}	-1.30×10^{-3}	-1.88×10^{-2}	1.08×10^{-2}
A_2	-1.84×10^{-1}	-7.16×10^{-2}	-3.24×10^{-2}	-4.67×10^{-3}	-7.16×10^{-2}	3.58×10^{-2}
A_3	-4.10×10^{-2}	-1.60×10^{-2}	-0.72×10^{-2}	0	-1.60×10^{-2}	1.11×10^{-2}
B_1	-4.34×10^{-1}	-1.11	-2.46	-1.35	-1.11	-1.59
B_o	1.70×10^{-2}	1.71×10^{-2}	1.71×10^{-2}	1.71×10^{-2}	1.71×10^{-2}	1.71×10^{-2}
B_2	3.92×10^{-1}	2.86×10^{-1}	2.50×10^{-1}	2.36×10^{-1}	2.86×10^{-1}	2.01×10^{-1}
B_3	1.28×10^{-1}	8.45×10^{-2}	6.89×10^{-2}	6.94×10^{-2}	8.45×10^{-2}	5.90×10^{-2}
B_4	9.89×10^{-3}	4.33×10^{-3}	2.38×10^{-3}	3.57×10^{-3}	4.33×10^{-3}	3.03×10^{-3}
B_5	2.1×10^{-3}	8.21×10^{-4}	3.70×10^{-4}	6.75×10^{-4}	8.21×10^{-4}	5.74×10^{-4}

Table 5 Calculated Values of the Parameters (Lateral)
(Cont.)

	Study No. 3			Study No. 4			Study No. 5
	k_1			V			$G_2(j\omega) \neq 1$ $\tau_1 = 1.0$ $\alpha_1 = 9.4$
	0.25	0.50	1.0	300	400	500	
A_o	-4.26×10^{-2}	-8.54×10^{-2}	-1.71×10^{-1}	-1.52×10^{-1}	-8.54×10^{-2}	-5.46×10^{-2}	-8.54×10^{-2}
A_1	-1.88×10^{-2}	-1.88×10^{-2}	-1.88×10^{-2}	-1.41×10^{-2}	-1.88×10^{-2}	-2.35×10^{-2}	-1.88×10^{-2}
A_2	-7.16×10^{-2}	-7.16×10^{-2}	-7.16×10^{-2}	-7.16×10^{-2}	-7.16×10^{-2}	-7.16×10^{-2}	-7.16×10^{-2}
A_3	-1.60×10^{-2}	-1.60×10^{-2}	-1.60×10^{-2}	-2.13×10^{-2}	-1.60×10^{-2}	-1.28×10^{-2}	-1.60×10^{-2}
B_1	-5.61×10^{-1}	-1.11	-2.20	-1.98	-1.11	-7.09×10^{-1}	-1.11
B_o	1.69×10^{-2}	1.71×10^{-2}	1.72×10^{-2}	1.71×10^{-2}	1.71×10^{-2}	1.72×10^{-2}	1.71×10^{-2}
B_2	5.46×10^{-1}	2.86×10^{-1}	1.55×10^{-1}	2.94×10^{-1}	2.86×10^{-1}	2.83×10^{-1}	2.86×10^{-1}
B_3	1.52×10^{-1}	8.45×10^{-2}	5.04×10^{-2}	1.12×10^{-1}	8.45×10^{-2}	6.79×10^{-2}	8.45×10^{-2}
B_4	8.57×10^{-3}	4.33×10^{-3}	2.19×10^{-3}	7.67×10^{-3}	4.33×10^{-3}	2.78×10^{-3}	4.33×10^{-3}
B_5	1.62×10^{-3}	8.21×10^{-4}	4.15×10^{-4}	1.94×10^{-3}	8.21×10^{-4}	4.21×10^{-4}	8.21×10^{-4}

Table 6

Values Invariant for all Studies (Longitudinal)

$I_y = 180$ slug ft^2	$C_{m\alpha} = -0.4$
$m = 50$ slugs	$C_{mq} = -2.8$
$\rho = .00237$ slug/ft^3	$C_{m\dot{\alpha}} = -1.4$
$s = 24.4$ ft^2	$C_{m\dot{\delta a}} = -0.3$
$c = 2.75$ ft	$C_{l\alpha} = -4.0$
$\theta_g = 12.5$ degrees	$C_{l\delta a} = -0.8$
$g = 32.2$ ft/sec^2	

Table 7

Values Varying for Different Studies (Longitudinal)

Study No. 1			Study No. 2			Study No. 3			Study No. 4		
$C_{m\delta a}$			k_3			$G_4(j\omega)$			$C_{m\delta a}$		
+.028	0	-.04	3.0	4.0	6.0	1	τ_d=0.13 τ_i=6.56	τ_d= 0.26 τ_i=13.10	+.028	0	-.04
$G_4(j\omega) = 1$ $k_3 = 4.0$			$G_4(j\omega) = 1$ $C_{m\delta a} = 0$			$k_3 = 4.0$ $C_{m\delta a} = 0$ $\alpha = 2.87$			$k_3 = 4.0$ $G_4(j\omega) = \begin{cases} \tau_d = 0.13 \\ \tau_i = 5.56 \end{cases}$ $\alpha = 2.87$		

Table 8

Calculated Values of the Parameters (Longitudinal)

	Study No. 1			Study No. 2		
	$C_{m\delta a}$			k_3		
	+.028	0	- .04	3.0	4.0	6.0
D_2	-2.07×10^{-2}	-2.80×10^{-2}	-5.59×10^{-2}	-2.80×10^{-2}	-2.80×10^{-2}	-2.80×10^{-2}
D_1	2.80×10^{-1}	1.72×10^{-3}	1.08	1.72×10^{-3}	1.72×10^{-3}	1.72×10^{-3}
E_3	3.55×10^{-2}	4.80×10^{-2}	9.58×10^{-2}	6.40×10^{-2}	4.80×10^{-2}	3.20×10^{-2}
E_2	1.09×10^{-1}	1.47×10^{-1}	2.94×10^{-1}	2.05×10^{-1}	1.47×10^{-1}	8.81×10^{-2}
E_1	2.30	2.76	4.44	3.69	2.76	1.84

	Study No. 3	Study No. 4		
	$G_4(j\omega) \neq 1$	$C_{m\delta a}$ $\left[G_4(j\omega) \neq 1\right]$		
		+.028	0	-.04
D_2	-2.80×10^{-2}	-2.07×10^{-2}	-2.80×10^{-2}	-5.59×10^{-2}
D_1	1.72×10^{-3}	2.80×10^{-1}	1.72×10^{-3}	1.08
E_3	4.80×10^{-2}	3.55×10^{-2}	4.80×10^{-2}	9.58×10^{-2}
E_2	1.47×10^{-1}	1.09×10^{-1}	1.47×10^{-1}	2.94×10^{-1}
E_1	2.76	2.30	2.76	4.44

Note: D_o and E_o have been omitted, since it was found that they effectively canceled each other.

Figure 2.18

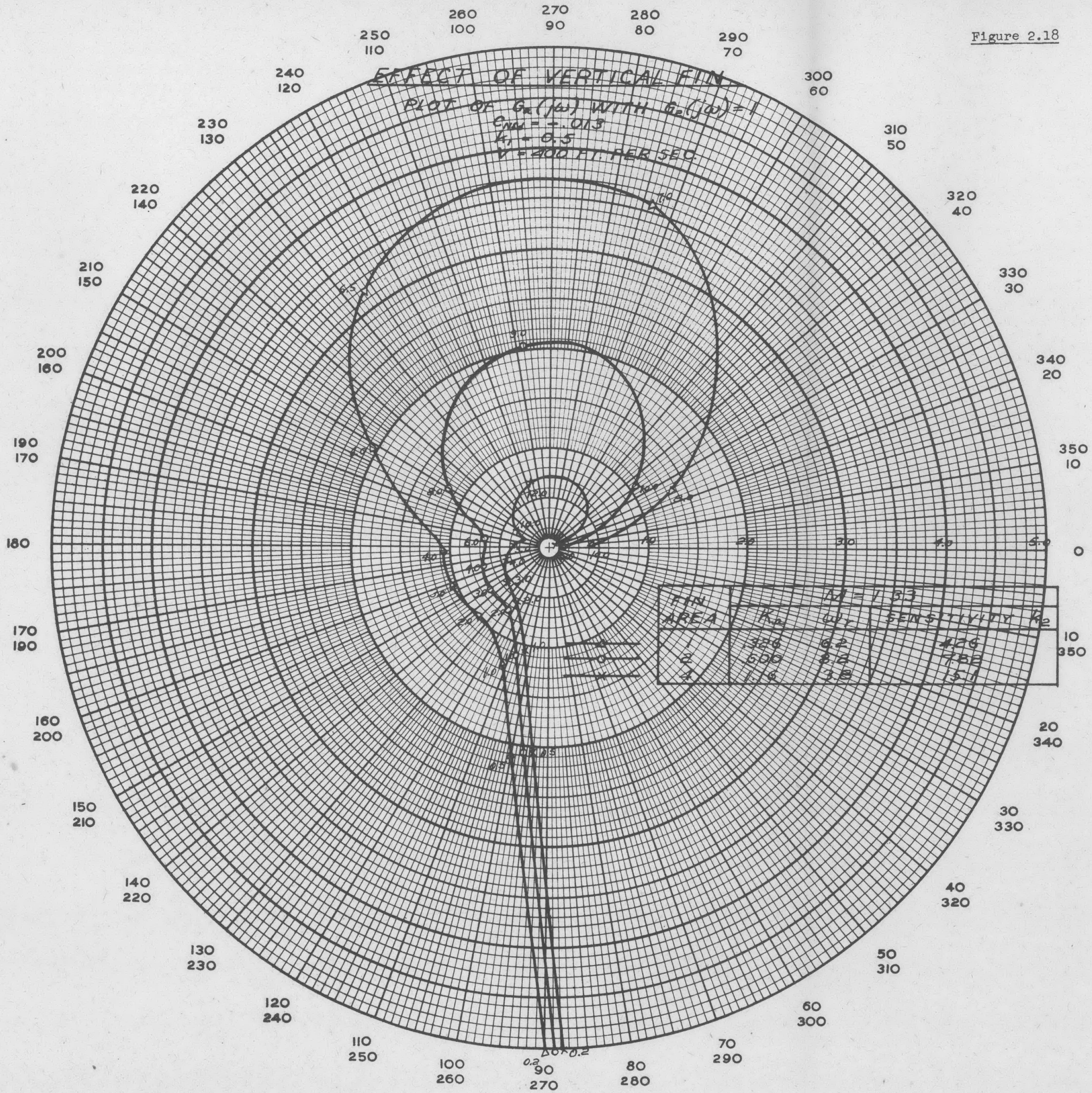

EFFECT OF VERTICAL FIN
PLOT OF $G_R(j\omega)$ WITH $G_2(j\omega) = 1$
$C_{NA} = -.013$
$M = 0.5$
$V = 400$ FT. PER SEC.
FIN AREA
K_D
ω_T
$M = 1.53$
SENSITIVITY K_2
2
4
326
600
1.6
6.2
8.2
3.8
4.26
7.52
5.1
260 100
270 90
280 80
250 110
290 70
240 120
300 60
230 130
310 50
220 140
320 40
210 150
330 30
200 160
340 20
190 170
350 10
180
0
170 190
10 350
160 200
20 340
150 210
30 330
140 220
40 320
130 230
50 310
120 240
60 300
110 250
70 290
100 260
80 280
90 270
0.2
ΔO×0.2

Figure 2.19

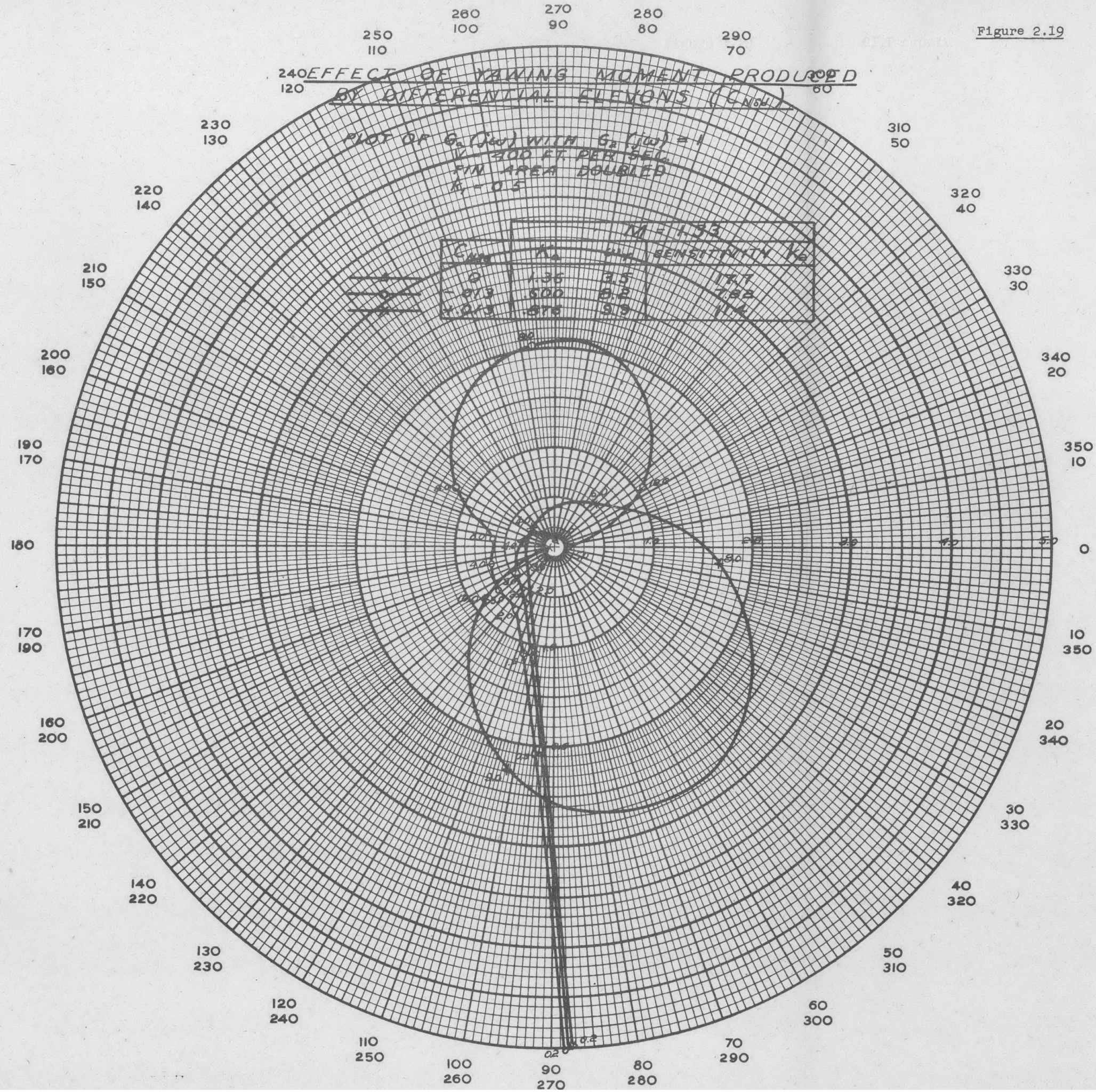

EFFECT OF YAWING MOMENT PRODUCED BY DIFFERENTIAL ELEVONS ($C_{n_{\delta_E}}$)
PLOT OF $\sigma_a(j\omega)$ WITH $\sigma_2(j\omega) = 1$
300 FT. PER SEC.
FIN AREA DOUBLED
$k_1 = 0.5$
M = 1.33
$C_{n_{\delta_E}}$
K_a
ω_n
SENSITIVITY k_a
0
1.36
3.5
17.1
-0.13
6.00
6.2
7.82
+0.13
8.75
9.9
1.74
90
80
70
60
50
40
30
20
10
0
350
340
330
320
310
300
290
280
270
260
250
240
230
220
210
200
190
180
170
160
150
140
130
120
110
100
90

Figure 2.20

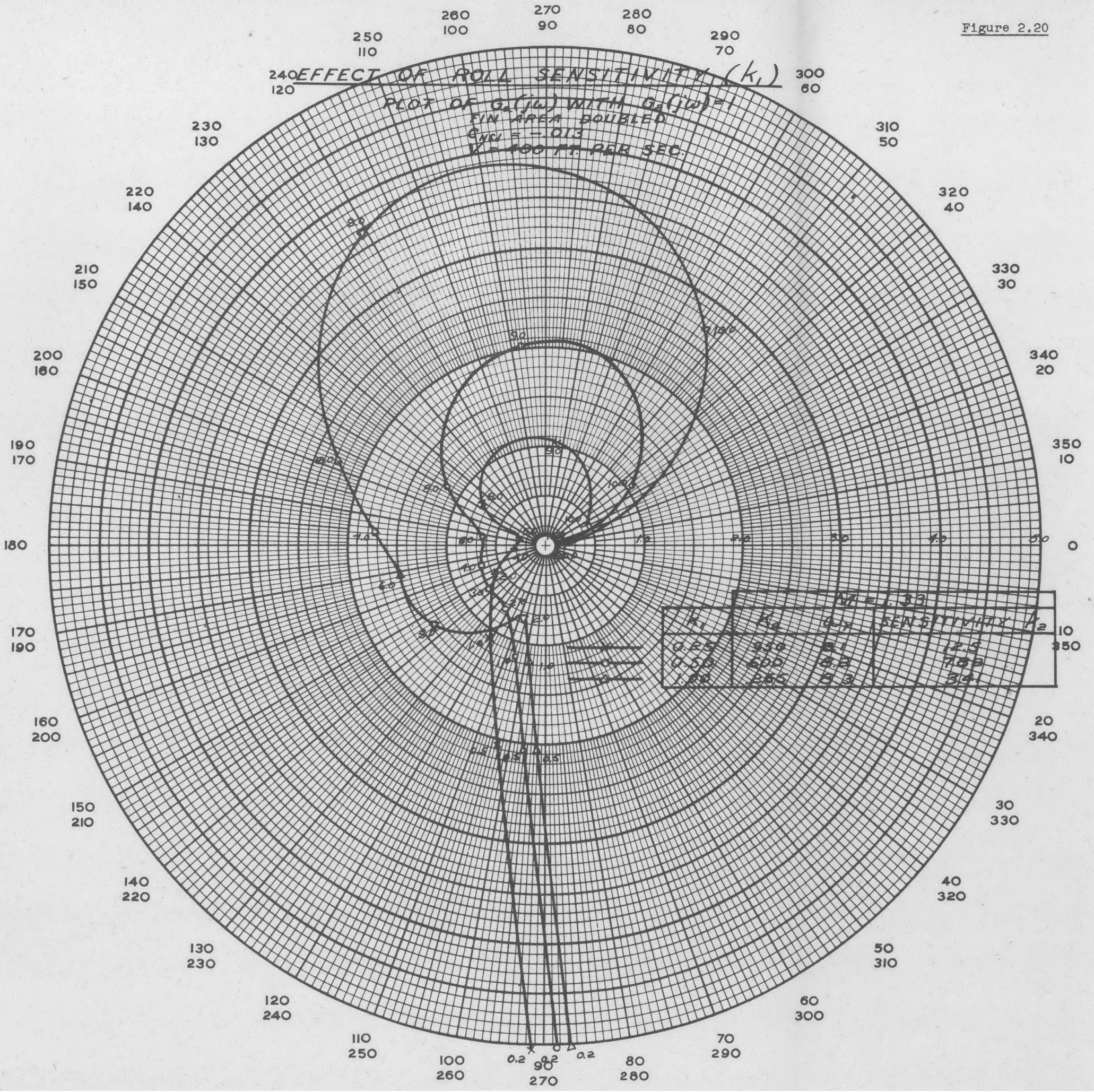

EFFECT OF ROLL SENSITIVITY (K_1)
PLOT OF $G_a(j\omega)$ WITH $G_2(j\omega)=1$
FIN AREA DOUBLED
$C_{N\alpha}$ = -.013
V = 100 FT. PER SEC.
M_α = 1.33
K_1
K_2
SENSITIVITY K_2
0.25
0.50
1.00
2.5
10.0
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160
170
180
190
200
210
220
230
240
250
260
270
280
290
300
310
320
330
340
350

Figure 2.21

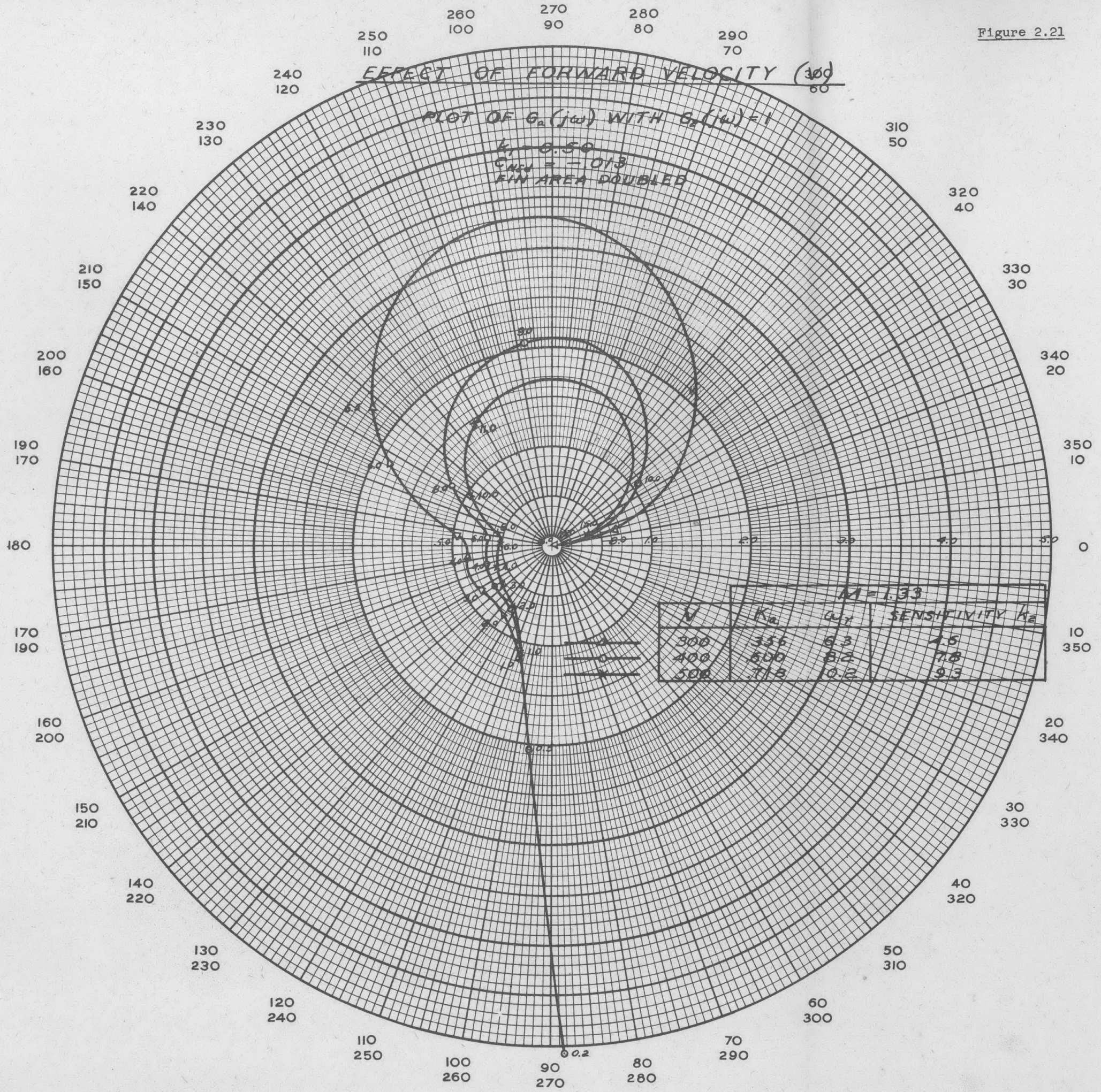

EFFECT OF FORWARD VELOCITY (ω0)
PLOT OF $G_a(j\omega)$ WITH $G_a(j\omega)=1$
$K_1 = 0.50$
$C_{NA} = -.013$
FIN AREA DOUBLED
M = 1.33
V | K_a | ω_r | SENSITIVITY K_a
300 | 356 | 6.3 | 1.5
400 | 500 | 8.2 | 7.8
500 | 718 | 10.2 | 9.3
90 270
80 280
70 290
60 300
50 310
40 320
30 330
20 340
10 350
0 360
100 260
110 250
120 240
130 230
140 220
150 210
160 200
170 190
180

Figure 2.22

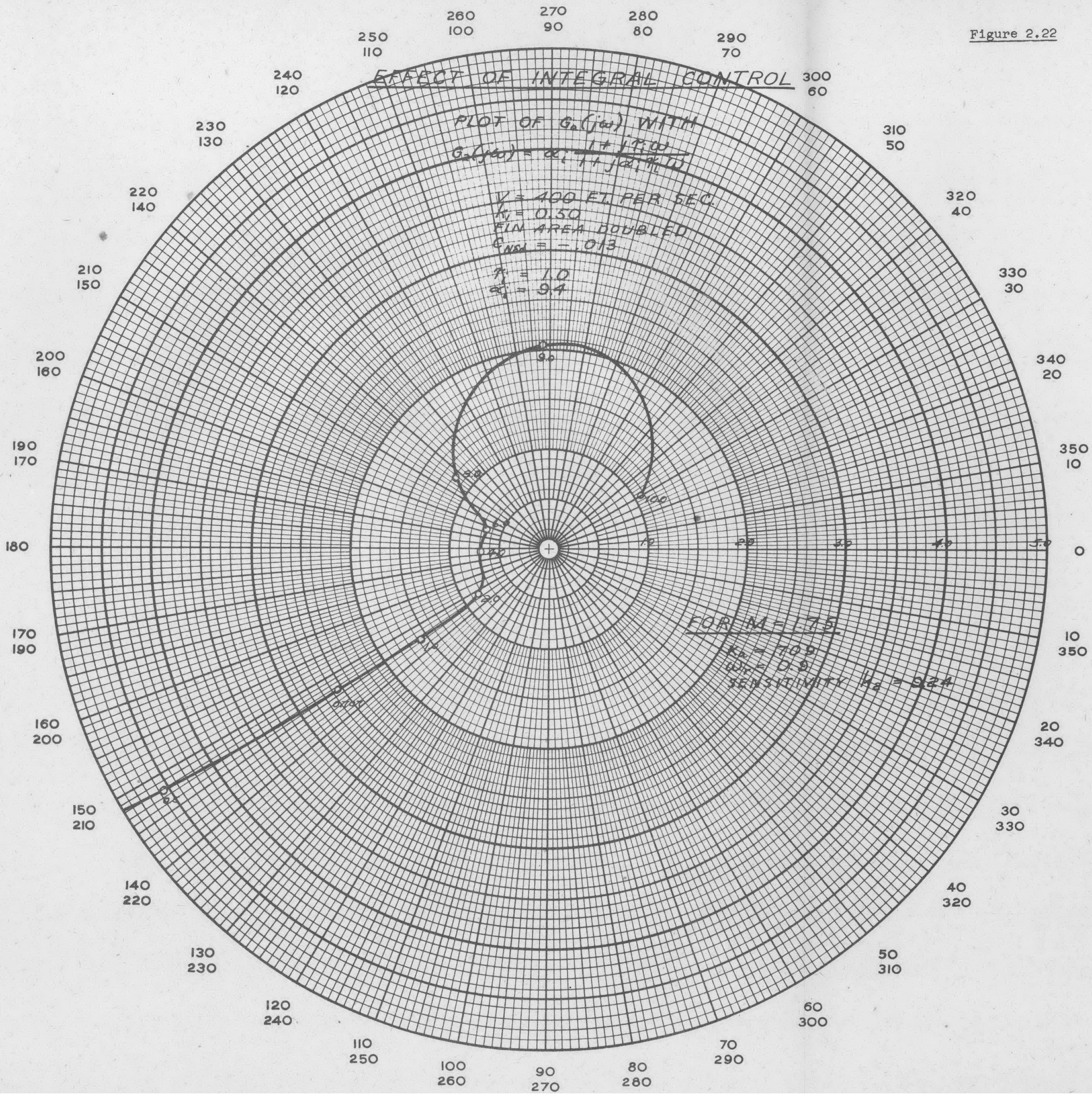

EFFECT OF INTEGRAL CONTROL
PLOT OF $G_a(j\omega)$ WITH
$G_a G(j\omega) = \alpha_c \frac{1 + j T_1 \omega}{1 + j \alpha_c T_1 \omega}$
V_t = 400 FT. PER SEC.
T_1 = 0.50
FIN AREA DOUBLED
G_{NSL} = −.013
T_1 = 1.0
α_c = .94
FOR M = 1.75
K_a = 1.09
ω_n = 0.9
SENSITIVITY K_a = 0.24
90
270
80
280
100
260
110
250
120
240
130
230
140
220
150
210
160
200
170
190
180
0
10
350
20
340
30
330
40
320
50
310
60
300
70
290

Figure 2.23

EFFECT OF PITCHING MOMENT PRODUCED BY CHANGE IN ELEVON AVERAGE POSITION ($C_{m\delta a}$)

PLOT OF $G_s(j\omega)$ WITH $G_a(j\omega)=1$

$k_3 = 4.0$

$C_{m\delta a}$	$K_a = K_{a1}$ M=1.33	1.50	1.75	2.0	ω_1 M=1.33	1.50	1.75	2.0
+.020	.790	1.50	6.37	11.1	.55	.70	.67	.70
0	.543	.697	.910	1.21	.35	.45	.50	.60
-.04	.191	.225	.282	.333	.16	.20	.22	.25

Figure 2.24

EFFECT OF AVERAGE ELEVON DISPLACEMENT SENSITIVITY FACTOR (K_3)

PLOT OF $G_s(j\omega)$ WITH $G_A(j\omega) = 1$

$C_{MS_2} = 0$

$K_3 = K_4$

K_3	M=1.33	1.50	1.75	2.0	ω_n M=1.33	1.50	1.75	2.0
3.0	.380	.522	.709	.889	.26	.33	.41	.48
4.0	.545	.667	.910	1.21	.35	.45	.50	.60
6.0	.760	1.00	1.33	1.67	.55	.64	.80	.90

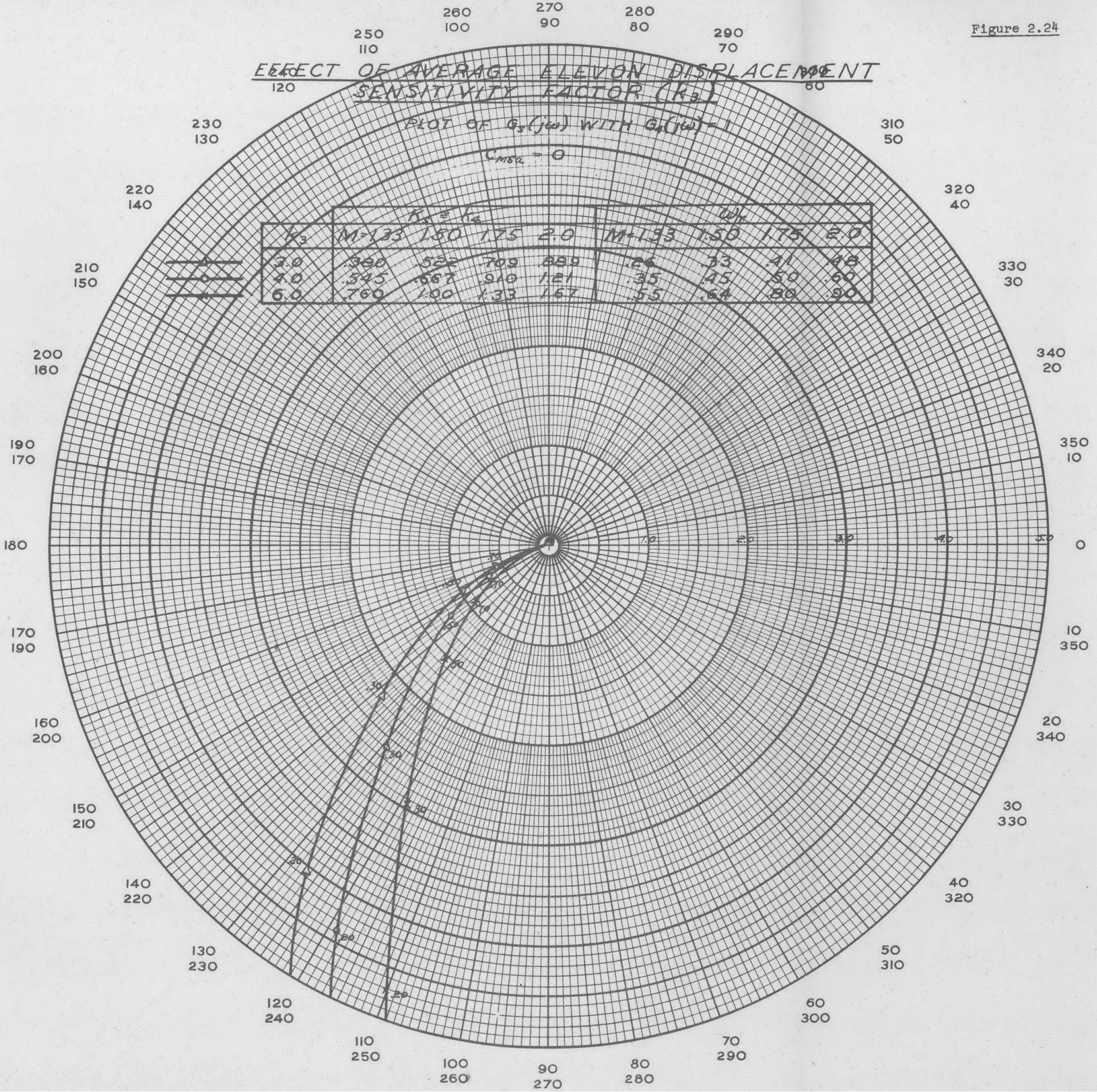


Figure 2.25

EFFECT OF COMPENSATING NETWORK

PLOT OF $G_s(j\omega)$ WITH

$$G_s(j\omega) = \frac{\alpha T_1 T_c (j\omega)^2 + (\alpha T_1 + T_c) j\omega + 1}{\alpha T_2 T_1 (j\omega)^2 + (T_2 + \alpha T_1) j\omega + 1}$$

$\alpha = 2.87$

$f_{13} = 4.0$

$C_{MSE} = 0$

$G_s(j\omega)$	$K_s = f_{12}$ M=1.35	1.50	1.75	2.0	M=1.33	1.50	1.75	2.0
$T_2 = 3, T_1 = .656$	.445	.667	.910	1.21	.35	.45	.60	.80
$T_2 = .86, T_1 = 3.1$	.200	.300	.150	3.33	.09	.10	.30	.30
	.43	.252	13.2	19.0	.35	.10	1.5	3.0

Figure 2.26

EFFECT OF C_{min} WITH COMPENSATING NETWORK

PLOT OF $G_s(j\omega)$ WITH

$$G_s(j\omega) = \frac{\alpha\tau_0\tau_1(j\omega)^2 + (\alpha\tau_0 + \tau_1)j\omega + 1}{\alpha\tau_0\tau_2(j\omega)^2 + (\tau_2 + \alpha\tau_0)j\omega + 1}$$

$\alpha = 2.87$
$\tau_0 = .13$
$\tau_1 = 0.56$
$K_s = 40$

FOR $K_s = 1.60$

C_{min}	PM	ω_x
.028	1.40	25
0	1.75	40
.04	BARELY STABLE	35

260 100 270 90 280 80
250 110 290 70
240 120 300 60
230 130 310 50
220 140 320 40
210 150 330 30
200 160 340 20
190 170 350 10
180 0

3.0 SYSTEM DESCRIPTION

3.1 Specifications

In a rush program there is generally insufficient time to develop exact specifications for the equipment under development. Instead, estimates of the important specifications are made and the system designed to meet these specifications. The estimates are generally made with an estimated factor of safety to protect the designer from his lack of knowledge of the exact specifications. It is particularly necessary to adopt such a procedure in designing as complex a system as a guided missile system. The general considerations that were set up as a guide to the design of the control system are presented below. In the subsequent sections of this chapter the way in which these specifications were met is presented. The final specifications on the control system are divided into two parts: (1) those specifications primarily controlling missile performance and (2) those specifications primarily concerned with matters of design. Those characteristics that determine the missile performance are stability, torque loads, accuracy of control, and speed of response. Those matters that are controlled by design specifications are space requirements, weight requirements, power supply limitations, and mechanical and production requirements. The above points are briefly discussed in the following:

3.1.1 Performance Specifications

a. Stability

While it is unnecessary under certain conditions to require the missile to have a stable flight, it is generally accepted as necessary in missiles which lose target information at some minimum range for if such a missile were unstable at that point, it might be headed wrong due to unstability and the miss increased. Therefore, a requirement on this system is that the missile have a stable flight throughout its trajectory.

b. Torque Loads

The control system must be so designed that it can provide the torque loads imposed on the control surfaces throughout its flight with a satisfactory margin of safety. This was a difficult requirement to meet at the speeds that it was believed necessary to move the elevons. Hinge moments on the control surfaces were not accurately known; however, they were estimated to be about twelve hundred (1200) inch pounds per elevon at a deflection of thirty degrees.

c. Accuracy of Control

The accuracy requirements are set by two general factors. First, that the missile be pointed in the correct direction within an allowable error factor at the time that the homing information fails and second that the missile follow a pursuit course sufficiently accurately that accelerations at the end of the trajectory be minimized. It was felt that if the missile were to have an error of no more than plus or minus one half (1/2) degree in both yaw and pitch, the operation should be satisfactory.

d. Speed of Response

The speed of response of the missile should be high to permit it to correct rapidly for disturbances during the latter part of the trajectory and to enable it to

follow as accurately as possible a true pursuit course. Speed of response is obtained by correct dynamical design of the system and by designing the servomotor to move the control surfaces with the proper speed. The speed with which the control surfaces were moved was set at ninety degrees per second and the natural frequencies in yaw and pitch were set at one cycle per second.

3.1.2 Design Specifications

a. Space Requirements

The space which the SWOD (Control System) Mk 18 Mod 0 could occupy was definitely limited in Bat since the complete unit was designed and laid out before the SWOD (Control System) Mk 18 Mod 0 had been designed. It was found necessary to design the SWOD (Control System) Mk 18 Mod 0 to fit in a space at the rear of the power supply and over the tail of the bomb (see Figure 1.2). It was possible to so design the system that it fitted into the space although relatively small clearances were used.

b. Weight Requirements

The weight requirements of the system were not particularly rigid. They were primarily set by balancing requirements on the missile. It was expected that this requirement would be met if the unit weighed less than one hundred pounds.

c. Electrical Considerations

In addition to good design practice and design of the system in order that it could be manufactured easily, the principal electrical requirement was to make use of the available power supplies without exceeding their capacity. The battery already provided in the missile is a fourteen-volt Exide storage battery type 9 TAS-9E. This battery when properly prepared for use will deliver one hundred amperes for periods several times as long as a normal flight. The discharge curve of this battery is shown in Figure 3.1. Under low temperature conditions a battery heater must be used. The main electrical motor used as a prime mover for the clutch servomotors has a current drain within the battery requirements.

d. Mechanical Considerations

Mechanical specifications are those encountered in any application of this sort. They include adequate strength, protection against vibration, protection against climatic conditions including high humidity, low temperature, etc. In addition, since the equipment is expendable, emphasis must be placed upon ease of manufacture. This requires the avoidance of too close tolerances, complicated machinery, and special assembly techniques.

3.2 General Description

The SWOD (Control System) Mk 18 Mod 0 is considered to include the following two parts:

A. The electronic portion referred to as the control amplifier.

B. The mechanical portion referred to as the gyropilot.

The electronic portion comprises the pitch trim and roll trim amplifiers which together form the data link between the output signal at the commutator of the SWOD (Radar) Mk 2 Mod 2 and the gyropilot.

The control amplifier, Figure 4.3.3, consists of two channels, one for conveying right-left homing information from the radar unit to the gyropilot, and one for conveying up-down homing information from the radar unit to the gyropilot. The former is called the roll trim amplifier, the latter the pitch trim amplifier. These will be discussed in detail in the section under trim unit.

The stabilizing element of the gyropilot is the vertical gyroscope. This is a Sperry Mk 4 Bank and Climb Gyroscope, modified for the present application. The rotor element is air-driven and is suspended in gimbals. It has an erecting mechanism and a caging mechanism.

Each gyro also has a pair of air valve pick-offs so that information of the attitude of the gyro case, and thus the missile, with respect to the vertical (as established by the gyro rotor) can be conveyed to other elements in the servo system. This information takes the form of pressure signals to a pressure-operated switch, or air relay. There are two air relays, one connected to the roll pick-off valve, and one connected to the pitch pick-off valve.

The air relays actuate magnets in the clutch units. The roll and pitch clutch units are identical. Each consists of two magnets, two sets of driven clutch plates, and two sets of driving clutch plates. The two clutch units are attached to either end of the clutch drive motor, which has a shaft extension at both ends. The double-ended drive motor is geared to the gear drives. The driven clutch plates are forced against the driving clutch plates by either magnet in each clutch unit and then drive the elevons through power gearing, a pantograph linkage, and connecting rods. The sense of the elevon motion is determined by which way the air pick-off valve in the gyro is operated, which in turn determines which way the air relay will close, which determines which clutch magnet will be actuated.

The pressure signal supplied by an air valve to an air relay depends upon the relative position of the pilot and follow-up sleeve that make up the air valves. The valve pilot is connected to the gyro gimbals, and therefore acts as a vertical reference in both the pitch and roll axes. The valve sleeve is connected to the output of a trim unit, one for roll and one for pitch. The elevon pitch and roll motions are fed back to the respective valve sleeves through mechanical differentials.

It can, therefore, be seen that relative motion between the valve pilot and valve sleeve can occur in any or all of the following ways:

1. Maneuver of the glider in pitch or roll
2. Trim unit operation
3. Elevon operation

In flight, all three actions occur simultaneously. Let us assume that a right signal (target to the right) is fed to the roll trim amplifier. This will cause the following train of events (see Figure 1.3):

1. The roll trim unit motor will turn until feedback voltage from potentiometer equalizes the input signal to amplifier.
2. The roll valve sleeve will rotate.
3. The roll air relay will be actuated.
4. The roll clutch will operate to move elevons differentially.
5. This differential elevon motion is fed back to the roll valve sleeve, reducing the original excursion caused by the trim unit.
6. The elevons will continue to move differentially until the valve sleeve is brought back to the neutral position with respect to the valve pilot.
7. This elevon differential will cause the glider to roll to the right.
8. This roll is measured by the gyro and causes relative motion between the valve sleeve and the valve pilot.
9. This actuates the air relay in such a direction as to reduce the differential elevon angle.
10. The glider continues to roll until the elevons are returned to neutral.

Note: Since the roll axis is underdamped the glider will overshoot this position (corresponding to neutral elevon) and the elevons will move opposite to their original displacement to reduce the overshoot.

11. At this point, if the signal to the roll amplifier is maintained at constant value, the glider will remain in a bank, the elevons will be at neutral, and the glider will yaw to the right (towards the target).
12. This yaw motion reduces the right-left signal, which means that the roll trim unit will move to a new position nearer that corresponding to zero right-left signal.
13. The change in position of the roll trim unit shaft results in motion of the valve sleeve. The resulting elevon motion reduces the angle of roll.
14. Therefore, for the new reduced value of right-left signal the elevons will be at neutral and the roll angle will be reduced, giving a reduced rate of yaw.
15. This action is continuous and progressive, finally resulting in minimum right-left signal, with minimum angle of roll and both the trim unit and the elevons at neutral.

Similarly, if the glider is on course, so that there is no right-left signal, an air disturbance causing roll of the glider will result in the following set of circumstances (assume roll to the right):

1. The roll is measured by the gyro and produces a relative motion between the valve sleeve and valve pilot.
2. The elevons move to correct the roll.
3. The elevons stop when the feedback from the elevons has zeroed the valve sleeve.
4. The glider rolls toward neutral.
5. As the glider roll angle decreases, the elevon angle decreases.

In pitch the action is similar (assume the target moves up, causing an up pitch error):

1. The pitch trim unit motor will run and reach a speed such that the generator voltage fed back equalizes the input signal. Thus the speed of the trim unit is proportional to the pitch error.

2. The pitch valve sleeve will rotate.

3. The pitch air relay will be actuated.

4. The pitch clutch will operate to move the elevons down.

5. This elevon motion is fed back through the differential and moves the sleeve back to neutral.

6. This will result in an elevon deflection proportional to the output angle of the trim unit.

7. The elevon motion will result in a rate of pitch of the missile.

8. As the missile turns, the gyro will measure this rotation and change the relative position of the pilot and sleeve in the air valve, thereby reducing the deflection of the elevons.

9. The net result is that the missile pitches with a rate proportional to the pitch error.

3.3 Reasons for Particular System

The SWOD (Control System) Mk 18 Mod 0 for Bat evolved from earlier systems designed for Pelican. (See Preliminary Report - Development of a Servo Control System for Guided Missiles.) The M.I.T. Pilot Mk II Control System was the final design for a guided missile of the Pelican class and it was decided to adapt it for use in Bat. This system had never been flight-tested but had the same dynamic characteristics as the earlier M.I.T. Pilot Mk IIA Control System which was used in several successful Pelican flights. The dynamics of this system are discussed in the preliminary report. The differences between the M.I.T. Pilot Mk II and the SWOD (Control System) Mk 18 Mod 0 (The Servo Control System for Bat) are largely in mechanical design. The evolution of this system is contained in the Pelican report, and will be reviewed here. The reasons for this particular dynamic system are made clear in the chapter on Theory.

The Sperry Mk IV Bank and Climb gyro was chosen as the stabilizing element because of its availability, general adaptability, and relative small size and weight. This gyro possesses a caging mechanism, enabling the gyro to be caged and uncaged at will. The air valve pick-offs of the Sperry gyro were readily adaptable, through air relays, to the control of clutches.

The most important reason for using a clutch servomotor is its increased efficiency over the reversing-motor type servomotor. With a relay-operated reversing motor, the losses in the motor and the power used in accelerating and decelerating are high. In a clutch servomotor, the drive motor is unidirectional at a speed giving high efficiency. For a more detailed description of a clutch servomotor see Section 4.4.1. Although the clutch unit was originally developed for Pelican, the possible application to larger missiles was anticipated. With larger missiles the disadvantages of the relay-operated reversing motor are greater, because larger currents cause difficult relay and commutation problems.

The reason for using a pantograph linkage is the limit stop problem as outlined in the Pelican report. If each elevon is driven by its own servomotor, the limit stops must

be on each elevon if great complexity is to be avoided. This means that the glider loses roll stability if pitch motion causes one elevon to reach its limit. Since it was found desirable to have a high pitch gain, such a condition is very likely to occur. The pantograph linkage has one servomotor driving the elevons differentially (to produce roll motion) and the other servomotor driving average elevon (to produce pitch motion). With this arrangement the limit stops are on each servomotor, and limit differential elevon motion and average elevon motion. In this way roll stability is retained even when the pitch servomotor is in the limit.

One of the important characteristics of the SWOD (Control System) Mk 18 Mod 0 is the feedback of elevon motion. This results in elevon motion proportional to the change in the glider attitude in pitch and roll. This proportional arrangement was necessary to obtain the optimum dynamical characteristics from the missile. These considerations are more fully discussed in the chapter on Theory.

3.4 Manner in Which the System Meets Requirements

Since the fundamental requirements of the servo control system is that the missile score hits, the first criterion of performance is the results indicated by the actual flight tests. The results of such tests are described in detail in Chapter 4. A more specific measure of merit of the servo control system is obtained by considering the specifications of Section 3.1, and how they have been fulfilled. Here, too, flight record test data can be used as a reference for the performance.

3.4.1 Flight Stability

Referring again to the flight records, it is seen that the glider flies in a stable fashion when responding to radar information. With no homing information the flight of the glider is also stable. Therefore, the requirement is met that, in the absence of radar information, the glider fly level and at constant glide angle, with the following exception:

If the glider is in a bank caused by a right-left radar signal, and the signal is suddenly removed, the glider will return to level flight gradually, with a time constant of several seconds. Similarly, if there is an up-down radar signal, causing a change in pitch attitude, the pitch motion of the glider will not cease immediately, if the radar signal is suddenly removed. Instead, the pitch motion will slow down gradually, and stop after several seconds. This "memory" effect is more fully described below in the section on "Trim Units and Amplifiers." Its action is to give more accurate course following in the presence of jittery radar signals.

The control system is designed to allow a maximum bank of 60°, thus providing a maximum rate of turn of 4° per second at a forward speed of 400 feet per second.

3.4.2 Output Torque

The peak torque required of the servomotor is 2400 in-lbs at the elevon output shaft. Since the clutch unit will deliver 18 in-lbs minimum, and the gear ratio is 150:1, the minimum torque available is 2700 in-lbs. The production clutch unit torque is 22 in-lbs.

3.4.3 Accuracy of Control and Speed of Response

The accuracy which the servo control system maintained in flight can be obtained from the records made by the glider cameras. By means of instrumentation in the glider, photographs of errors, as indicated by meters in appropriate circuits, show that for a representative flight the right-left error and the up-down error were never more than one degree, except for the initial and final transients. This accuracy is well within the requirements, when considered with the minimum range and the high speed of response of the system. The speed of response of the missile is approximately one cycle per second, and the time constant is approximately nine seconds.

3.4.4 Space and Weight

Although the SWOD (Control System) Mk 18 Mod 0 very nearly fills the space over the bomb at the rear, there is adequate clearance between the control system and the airframe.

The weight of the servo unit is approximately 100 pounds. This is not excessive, compared to a total weight for the missile of over 1500 pounds. The balance of the glider is affected by the addition of the SWOD (Control System) Mk 18 Mod 0, but rebalancing presents no problem.

3.4.5 Electrical Considerations

Tests were run on the clutch drive motor to determine the current drain under various load conditions. These tests show that the load current will normally not exceed 80 amperes. This is well within the capacity of the storage battery for the time of flights expected plus a suitable safety factor.

3.4.6 Mechanical Considerations

The mechanical design is considered in some detail in Section 4.1. The reasons for various elements in the design are discussed. Therefore, it is only necessary to state that the design does in general meet the requirements met in actual use. There was only one case of any trouble in the field. (See Flight Tests, Section 6.2.) This was easily corrected.

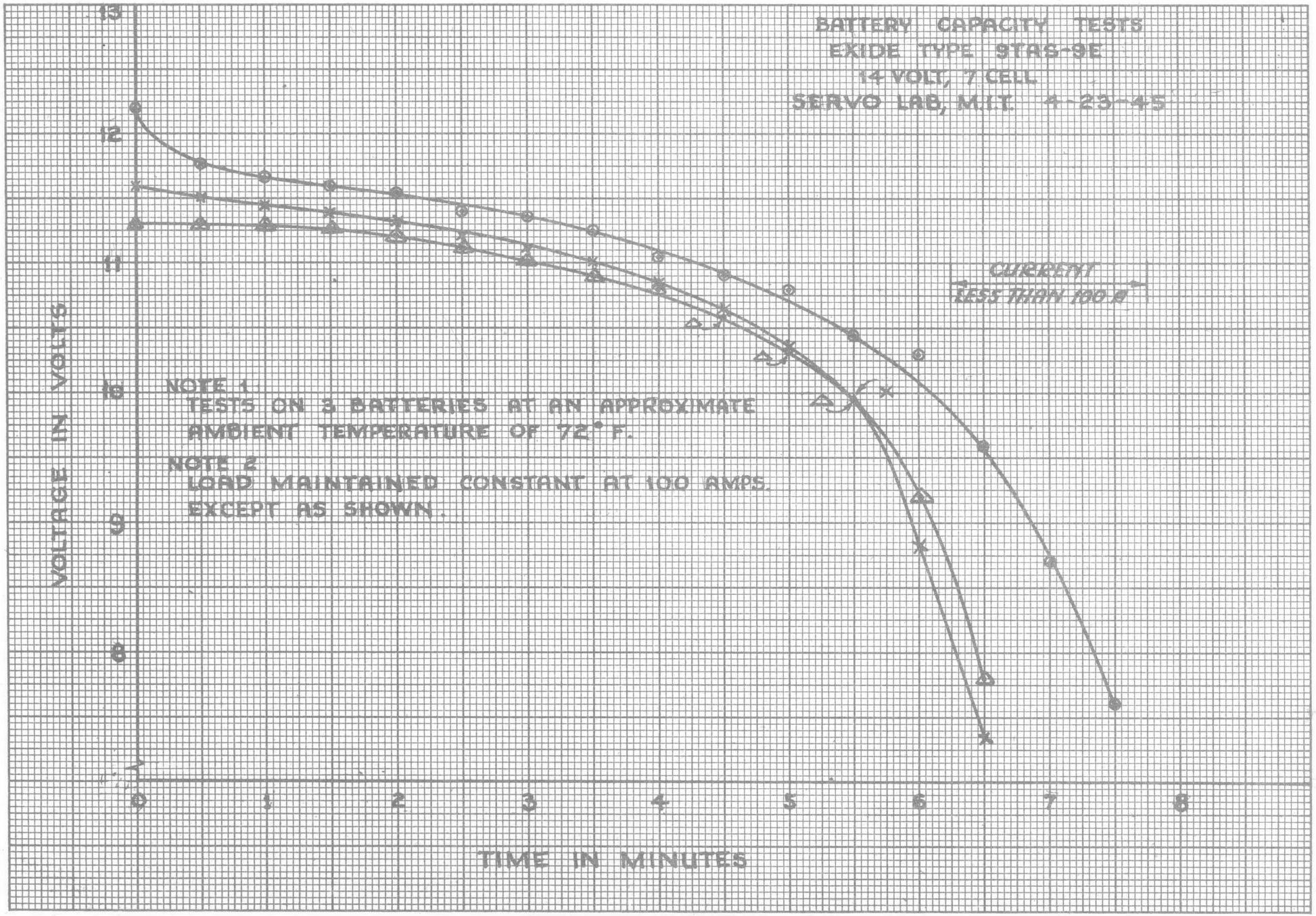

BATTERY CAPACITY TESTS
EXIDE TYPE 9TAS-9E
14 VOLT, 7 CELL
SERVO LAB, M.I.T. 4-23-45
CURRENT LESS THAN 100 A
NOTE 1.
TESTS ON 3 BATTERIES AT AN APPROXIMATE AMBIENT TEMPERATURE OF 72° F.
NOTE 2
LOAD MAINTAINED CONSTANT AT 100 AMPS. EXCEPT AS SHOWN.
VOLTAGE IN VOLTS
13
12
11
10
9
8
0
1
2
3
4
5
6
7
8
TIME IN MINUTES

Figure 3.1

4.0 COMPONENT DESCRIPTION

4.1 General Mechanical Design

4.1.1 Function

The purpose of the mechanical section of the servo control unit as is evident from reference to the Schematic, Figure 1.3, is to position the elevons according to signals generated by the homing radar and the stable vertical.

4.1.2 Design Considerations

The mechanical design of the control system was seriously influenced by a number of factors. The specifications rather rigidly prescribed the dynamic characteristics such as power output, speed of response of the elevons, etc., while the aerodynamic design of the airframe imposed certain space and weight limitations. A third factor controlling the overall design was the desirability of using some components already proven in previous experiments. A fourth factor, namely, the extremely short space of time that could be allotted to the design of a mechanical system, required that choice and arrangement of components, as well as materials used in construction, be chosen on the basis of sound engineering judgment rather than as a result of exhaustive tests. Difficulty in rapidly procuring manufactured items and construction materials was a further serious consideration.

4.1.3 Available Space

In the airframe, SWOD (Air Stabilizer) Mk 13 Mod 0, the same compartment was utilized as for the Bureau of Standards Servo, SWOD (Servo) Mk 4 Mod 3, except that the increased volume of equipment made it necessary to use as much as possible of the entire space available in this compartment. For instance, all of the limiting surfaces (front and rear bulkheads, airframe sides, bomb, and cover) were approached with minimum practicable clearance in order to obtain sufficient room for the servo control unit.

4.1.4 Choice of Components

The following components, most of which had proven satisfactory in previous applications, were chosen to form the nucleus of an integrated design for the control servo.

1. For the stable element, a Sperry Mk 4 Bank and Climb Gyro was chosen because of its availability, compact size, moderately light weight, and general adaptability.
2. The power drive section was based upon the previously developed Model F clutch, used in conjunction with a 1/2 hp Emerson d-c motor.
3. A satisfactory pantograph linkage for operating the elevons was already in production and proven in use; hence, it was desirable to utilize this component with a minimum of modifications.

4. The basic design for the roll and pitch trim units had already been evolved, although the design details were not particularly convenient for the new design. The basic design, however, served as a valuable guide for function and space requirements.

5. John Oster Type A-14B-3A motors were selected for operating the trim units and gyro caging mechanism on the basis of previous experience. More detailed information concerning these motors will be found in Section 4.3.4.2. The potentiometer used in the roll trim unit is also described in the above section. The Elinco generator used in the pitch trim unit is described in Section 4.3.7.2.

4.1.5 Arrangement of Components

A preliminary design study was begun on March 15, 1945, given the information regarding function, chosen components, and available space. Two-dimensional model cut-outs were made and arranged on a plan view of the compartment. The primary determinants were the plane of action of the pantograph, the size and shape of the power drive assembly (clutches and drive motor), and the size and shape of the gyro. Rather quickly it became evident that considerable advantage was to be gained by keeping the pantograph in the same plane of action as used in the standard airframe. This left the possibilities of moving the pantograph fore or aft and turning it to face forward or astern. The power drive was then studied for position. It was found that the only way in which it would fit was to place it above and astern of the pantograph. The clutch output shafts had to be on the bottom in order to design a gear train which would go as directly as possible from the power drive to pantograph input shafts.

Next in line was the gyro, which was tried in several different coordinate positions, the best apparent location being on the longitudinal center line, as far forward and as high as space permitted, with the face of the gyro to the rear. Another requirement was that the gyro longitudinal axis be parallel within a degree or two with the airframe center line. Since the pantograph plane of action is at an angle of 17° from the airframe center line, it was necessary to rotate the gyro case longitudinally by the same amount.

The relationship of the power drive to the pantograph indicated that most of the equipment would integrate most easily on planes parallel to the pantograph plane of action. Since the gyro was required to be at a different angle, the problem arose of transferring signal motions from one plane to another. Two possibilities were available: flexible cable with pulleys, or a combination of cranks, ball and socket joints, and connecting rods. Previous experience with the cable and pulleys suggested that another method be tried, and the ball and socket joints appeared promising. These were accordingly chosen. (See Figure 1.3.) It was also desired to shock mount the gyro to prevent its being damaged in shipment. The use of ball and socket connections thus satisfied all the conditions of motion transfer between different planes, rigid signal transmission, and adaptability to shock mounting. Following these decisions, the gyro was required to be positioned with the face aft in order to get the cranks for the signal transmission sufficiently separated that the transmission be reasonably linear. This then determined that the roll trim equipment would be on the port side, and the pitch trim on starboard.

Sufficient definition of arrangement had been accomplished at this stage to determine that the pantograph linkage would lie mostly forward of the pantograph input shafts.

An approximate layout of the necessary shafts and gears into this equipment pointed the way to the design and placement of all the details, which work was begun about March 25, 1945.

The limited time available, together with the necessity for a few pilot models, precluded the use of special protective finishes for parts used in the mechanical section.

Lubrication was selected on the basis that one application at assembly would provide sufficient protection to moving parts to serve throughout testing and field use. Delicate parts and bearings were given a light coating of Esso Handy Oil. Colonial Beacon OS-1350 grease was used on the medium duty parts such as worm gears and pantograph bearings. Lubriplate, a grease lubricant, manufactured by Fiske Brothers Refining Co. was selected for the power drive gearing. Oilite bearings for the clutches were fabricated from Oilite bar stock and were given a thorough soaking in SAE 10 machine oil before assembly. This was considered suitable for the pilot models although it is not to be recommended where low temperature conditions may be encountered. The Oilite bearings used for power drive shafts were used as obtained from the manufacturer.

To expedite procurement and assembly, stock gears were specified for the design wherever possible. Because of the extremely high loading conditions, the power drive gearing was specially cut and treated.

4.1.6 Deck Assembly

This assembly forms the principal structure upon which the other assemblies are mounted, and through which the pantograph reactions are carried into the airframe. It consists of decks at three levels (Figure 1.4) with spacers introduced between, and held together with bolts and dowels.

Castings as compared with plates and spacers for the deck construction were duly considered. Foreseeing considerable time required to obtain complex patterns and castings, and increased assembly complexity, it was decided that machined plate stock separated by spacers would be used. Aluminum was used for all parts, where practicable, to reduce weight.

On the top deck are mounted the drive unit, the relay assembly (to the rear and below the drive unit), the pitch trim unit, the roll trim unit, the gyro unit, and a number of auxiliary items such as dynamotor, electrical connector, terminal strips, cable clamps, etc. The adjustable mounting lugs which attach the pilot to the sides of the airframe are secured at approximately the four corners of the top deck.

Between the top and center decks are located some of the gears and shafting through which power is transmitted from the clutches to the input shafts of the pantograph.

The power gearing presented a difficult problem in compromise. Due to the high gear ratio from the clutches to the pantograph, the gear tooth loads become extremely high toward the pantograph end of the train of gears. The small amount of space available seriously limited the permissible gear tooth face widths. Also, the best gear materials were not immediately available for use. A further consideration was that of reducing to the minimum gearing inertia reflected to the clutches. These various conditions were met by using stub tooth gears, the best material (NE Steel No. 9450 hardened to 300

Brinnell) before precision machining. To reduce face widths, the material was stressed to the maximum permissible values.

The shaft centers for the power gearing were obtained on the deck plates by first manufacturing a boring plate which was made on a jig borer to considerable precision. This plate was then clamped on the deck plates individually and machining done.

The entire pantograph linkage is also within this space between the top and center decks, as are the linkages from the trim units to the gyro crank arms.

The lower decks, of which there is one on each side, are smaller than the upper two, and serve as the lower bearing plate for the pantograph limit stops and the remainder of the pantograph drive gearing. Associated with the mechanical limit stops, which physically prevent the power gearing from rotating beyond specified limits, are the electrical limit stops whose microswitches are actuated by small cams fixed to the rotating members.

The electrical limits function by switching opposite current into the clutch coils and forcing the unit to rotate in the direction to recede from the limiting position.

The pantograph linkage is essentially the same as that used previously. Complete linkages were obtained from Harris-Seybold-Potter and altered slightly to suit. It was deemed advisable to transmit the torque for elevon operation directly to the linkage arms rather than via taper pins into and out of the pantograph input shafts. To accomplish this improvement, the driving gear sector was sandwiched between the linkage arms which therefore had to be of special shape and were accordingly redesigned. This constituted the only modification performed on the linkage as presently produced.

Because of procurement problems on ball bearings, and because plain bearings seemed sufficiently satisfactory and practically free of contamination troubles, ball bearings were not used in the power drive. To seat the plain bearings (Oilites were used) in some cases, where rivets or pins could not be used, a force fit sufficed with the added security of a slightly splined hole into which each bearing was seated.

4.2 General Electrical Design

4.2.1 Function

The purpose of this section is to describe the function of the circuits and electrical components used in the control system. Some of this material is more specifically covered elsewhere in this report. However, most of it will be repeated in this section for the sake of completeness.

The discussion will be divided into sections roughly corresponding to the various circuits contained in the control system. Because there is considerable overlapping in the functions of the circuits in some cases it is not possible to completely separate the description of the various sections.

4.2.2 Main Power Circuit

The function of this circuit is to provide a connection from the 14 volt storage battery to the clutch drive motors. It includes a single-pole, single-throw contactor for switching purposes.

The drive motor is designed to operate from a 14 volt supply and to drive the clutches at an input speed of approximately 5,000 rpm. The normal rating of the motor is 1/2 hp. A more complete summary of the drive motor specifications is given in Section 4.4.4.4.

A General Electric contactor, catalogue No. CR2792B-116A2, was used for motor switching. The nominal contact rating was 50 amperes, and the nominal rating of the coil was 12 volts. The use of the coil on a 14 volt circuit provided a safety factor in the operation of the contactor.

The motor leads were of wire size AWG No. 8. One motor lead is grounded permanently to the main base plate. The leads from the control system to the battery were of wire size AWG No. 8, and were terminated in closed lugs, Aircraft and Marine Products Co. solderless type.

The battery leads were connected permanently into the control system thereby eliminating the need for a connector or terminal block.

The battery, Exide Type 9TAS-9E, was a seven cell lead plate type with a capacity of approximately 20 ampere hours. The charge on the battery was such that it was capable of high peak currents at some sacrifice of long life.

4.2.3 Clutch Coil Circuit

The purpose of this circuit is to provide excitation for the clutch coils. This excitation is provided by a small dynamotor with full load terminal rating of 250 volts at 50 milliamperes. The current in the clutch coils is switched on and off by means of the air relays. The circuit also contains current limiting resistors, limit switches and condensers for the protection of the air relay contact points.

The dynamotor was a standard Signal Corps type DMA 25. Performance curves are given elsewhere in this report. (See Figure 4.4.9.) It suffices here to state that the nominal input voltage is 12 volts, and the nominal output rating is as given above. The dynamotor was used on the 14 volt battery so that the output was slightly higher than its normal rating.

In the production unit the clutch coil excitation is obtained from a rectified supply in the main power supply. Each of the two clutch coil circuits is provided with a 3500 ohms, 10-watt resistor to limit the current in the clutch coil. This resistance in series with the coil also serves to reduce the effective time constant of current build-up in the coil.

The current is switched to the proper coil by means of air operated relays. Each relay is essentially a single-pole, double-throw switch which provides a means of exciting one coil or the other in the clutch unit. Because the clutch coils are highly inductive it is necessary to provide protection for high voltage at the relay contacts. This was done by connecting a 0.01 mfd condenser across the contact points.

Also in series with the air relay are a pair of microswitches in each clutch coil circuit. These microswitches serve as limit stops for the output linkage of the control unit. When the linkage moves into the limit position the proper microswitch is operated so that the clutch unit reverses the direction of motion of the output linkage. Contact protection for the microswitches is provided by the same condenser used to protect the air relay contact.

4.2.4 Arming Circuits

In the operation of the control system it is necessary to keep the gyro element caged prior to the actual switching on of the unit. Therefore, when the unit is switched on, provision must be made for uncaging the gyro at the same time. This is accomplished by providing the caging unit on the gyro with switches which have a two-fold function:

1. They act as caging limits.
2. They act as switches for starting other elements in the control system.

The caging mechanism from electrical considerations is a small motor and two microswitches. The motor, John Oster Co. Type A-14B-3A, has a shunt field which is excited directly from the 26 volt radar battery. The armature voltage is provided by a reversing switch which provides 13 volts of either polarity depending upon the direction of rotation desired. The microswitches serve as limit stops to cut the motor off when the desired number of revolutions are obtained. These switches also provide voltage for the clutch motor contactor coil and the armature circuits of the pitch and roll trim motors. Also the circuit contains two pilot lamps which operate from the switches for the purpose of indicating positively the caged and uncaged positions. The voltage provided by the caging circuit for the pitch trim motor is not applied directly to the trim motor circuit. It is first applied to the three-second delay switch located in the Bat junction box. This switch causes the pitch trim unit to remain inoperative until three seconds after the missile drops away from the airplane. The roll trim motor operates as soon as the gyro is uncaged, but right-left radar signals are withheld from the roll trim unit until three seconds after the missile is dropped from the airplane. These functions are described in more detail in the section of this report dealing with the trim servos. (See Section 4.3).

4.2.5 Pitch Trim Servo

The function of the pitch trim servo is to transmit the up-down radar signals into mechanical motion suitable for trimming the pitch reference of the control system. The electrical circuit consists of a trim motor, a pair of reversing relays, and a small d-c generator. The radar signals in the form of d-c currents operate one or the other of the reversing relays providing the proper direction of rotation for the trim motor. The generator provides a d-c voltage proportional to the trim rate which is fed back into the radar unit to stabilize the rate at which trim action occurs.

The pitch trim motor is identical to the caging motor. The reversing relays and the motor armature must have contact arc suppression both to protect the contact points and eliminate the possibility of generating radio interference. This arc suppression is accomplished by means of 0.25 mfd condensers placed across the motor armature. In this manner both the relay contacts and the motor armature are protected.

4.2.6 Roll Trim Servo

The roll trim servo performs a function similar to the pitch trim servo in that it transmits the right-left radar information into mechanical trimming of the control system. The electrical components consist of a trim motor, a pair of reversing relays, a potentiometer, and a small dry battery. The function of the motor and relays are identical

to those in the pitch trim unit. The potentiometer and battery combination performs a function similar to the generator in the pitch trim unit, except that the voltage fed back to the radar unit is proportional to the position of the trim unit rather than its rate. The potentiometer, General Radio Type 214-A, has a resistance of 2000 ohms. The potentiometer is redesigned so that it is an integral part of the trim unit. The battery, Burgess Type W30BP, is a 45-volt, center-tapped dry battery. The percentage of the battery voltage which is applied to the potentiometer terminals is determined by dropping resistors. Associated with the battery is a small relay which operates when the gyro is uncaged. The purpose of the relay is to keep the battery open-circuited until the trim unit is actually operated. In the production unit the battery is replaced by power from the main power supply.

4.2.7 Summary of Requirements

The general requirements for the electrical design of the control system can be enumerated as follows:

1. The design must make for ease of assembly and disassembly so that modification and changes can be made on the control system with a minimum of difficulty. This means that terminal blocks with screw type terminals were used wherever a unit subassembly was connected electrically into the overall system.

2. Because of the space requirements it was necessary to make a compromise between flexibility and a minimum number of components. This is especially true of electrical connections to the external unit. One AN type connector was used for connections to the bomb junction box. The connections to the storage battery were made with leads that were permanently attached to the control system.

3. The main wiring assembly was made up into a harness and all terminations were in closed type insulated lugs. This type of construction enabled considerable handling without causing deterioration of the wiring.

4. High voltage is present on some of the terminals and therefore care had to be taken to avoid accidental contact. The danger was minimized principally by locating exposed high voltage points so that accidental contact would be at a minimum. In the production model an insulating barrier was placed over the high voltage contacts with the words "Danger High Voltage."

The AN connector used in the control system was Type AN3102-22-14P. This connector has 19 contacts which made soldering connections somewhat difficult and the spacing of the exposed points small. This condition actually caused no trouble but was accepted only because of difficult spacing requirements.

All terminal blocks were of Howard B. Jones Co. Type 140 and were mounted permanently on the control unit. On each terminal block was placed a Bakelite pad 1/32" thick. This pad served a dual purpose in that it provided extra voltage protection and also provided a means of numbering each terminal of the terminal block.

The wire used for most of the wiring was a type manufactured according to JAN specifications, and was of conductor size AWG No. 20 or No. 18. Some parts of the circuit required shielded wire. The wire used was Cornish Wire Co. No. 957. The shielded wire was connected into the unit by means of terminal lugs on both the conductors and

the shield on each end. The shield is connected to the base plate of the control unit at both ends.

The electrical design was carried out in a manner which permitted most of the electrical work to be completed independently of the mechanical assembly. In this way the final assembly of the unit was not hampered by the necessity of doing the electrical assembly work on the final mechanical assembly. In actual operation practically no trouble was encountered with the electrical system.

4.3 Trim Units and Amplifiers

4.3.1 General Description

The method of elevon control employed in the SWOD (Control System) Mk 18 Mod 0 requires a mechanical feedback link to the attitude-detection apparatus (air valves) on the vertical gyro. (See Figure 1.3.) If it is desired to change the glider attitude in response to signals from a homing device, the reference position of the feedback links must be mechanically displaced by the amount necessary to secure the desired change.This is accomplished by inserting a differential gear in the feedback link between the clutch output and the sleeve of the air valve. The differential is fed by an electromechanical amplifier, designated as a trim servo, which converts the electrical signal from the radar homing device into the required mechanical displacement of the feedback reference. (See Section 3.2 for fuller description.)

The trim servo comprises the following elements:

1. A balanced electronic amplifier which amplifies the difference between the radar signal and a feedback signal which is a function of the output.

2. A pair of sensitive relays connected as a reversing switch, with coils energized by the amplifier.

3. A reversible d-c motor, the armature supply of which is controlled by the relays.

4. A potentiometer or electrical tachometer, employed to produce a feedback signal proportional to output position or speed, respectively.

5. Limit stops which prevent an excessive change in reference position.

6. A slip clutch to prevent stalling of the motor when the output is against a limit stop.

7. A differential gear which connects the clutch output position to the trim output position and applies the resultant position to the air valve sleeve.

8. Gearing between the motor and feedback device and between the feedback device and the differential so designed that these devices are operated at the proper speeds for stable glider flight and for stable and accurate servo performance.

4.3.2 Input Signal Characteristics

The signal supplied by the homing device, a SWOD (Radar) Mk 2 Mod 0, is, to a first approximation, a d-c voltage whose magnitude is proportional to the error in

heading and whose polarity is determined by the direction of the error. A detailed description of the operation of the radar will be found in the Western Electric Service Handbook for the set.

The signal is obtained from quadrant commutator segments spaced around the axis of the radar antenna and separated by short insulating segments. The reflecting part of the antenna is set at an angle to the axis of the radiator, and is rotated about it by a small motor, so that it looks successively down, right, up, and left. The commutator segments are simultaneously contacted by a brush which is mechanically connected to the rotating part of the antenna. The segments are thus designated down, right, up, and left. The output of the radar receiver is fed to the rotating brush by means of a second brush and slip ring.

If an echo is received from a target which is displaced from the axis of rotation, the received signal will vary from a maximum when the antenna is pointed toward the target to a minimum when it is pointed away. Since the antenna rotates at 1800 to 2000 rpm, the received signal will be modulated at 30 to 33 cps. The signal impressed on the commutator is the a-c component of the 30 to 33 cycle modulation, plus a pulse ripple having a frequency of 2000 pps. The modulation amplitude is a function of the displacement of the target from the axis of rotation, and the phase is a function of the direction of the target.

The commutator separates the signal into up-down and right-left components. If the target is directly above the axis of rotation, a positive voltage will appear at the up segment during the time the brush is in contact. A negative voltage will appear at the down segment, while the right and left segments will have zero average voltage. The 30 and 2000 cycle ripple is attenuated by filter networks. The sum of the up and down voltages is then supplied to the pitch trim servo amplifier. The sum of the right and left voltages is similarly supplied to the roll trim servo amplifier.

The signal supplied to the trim amplifiers departs greatly from the ideal. The faults encountered are listed below:

1. The signal is a d-c voltage proportional to the angular displacement of the target from the axis of rotation of the antenna, for small displacement angles. At about 10° (two-way transmission), the slope of the signal vs. displacement curve begins to drop off and at very large angles the signal magnitude decreases for increasing angular displacement.

2. Time lags in the commutator filter and preceding circuits cause the signal to lag behind rapid motions of the target.

3. Since the antenna is not stabilized, rolling of the glider introduces false signals.

4. The nature of the signal supplied to the commutator is such that an up-down signal produces a small false right-left signal. This effect, known as rotation of the axes, can be minimized by proper design of the commutator input and output networks.

5. The signal is subject to fading, interference due to noise and other radars, and to occasional random changes.

6. At very close ranges the strength of the signal becomes so great as to cause saturation in the r-f and i-f sections of the radar. This saturation causes a considerable loss of sensitivity. For large targets, reversal of information may result.

7. A small proportion of the 30-cycle ripple remains after filtering.

The control system, and especially the trim servos, must be designed to minimize any harmful effects on glider performance caused by the defects in the signal.

4.3.3 Output Characteristics

As previously described, Section 3.2.1, the control of the motion of the glider in yaw is accomplished by producing a roll angle proportional to the homing error in yaw. This requires that the roll trim servo displace the gyro reference through an angle proportional to the yaw error. Ideally, this angle should be attained instantaneously, but in practice it is sufficient that it be reached at a rate large in comparison with the rate of turn of the elevons and the maximum rate of roll.

The requirements for instantaneous and average accuracy of the trim servo output are determined by entirely different considerations. Since the roll axis inertia of the glider is high, rapid oscillations of the elevons, if of moderate amplitude, will not cause appreciable rolling. It is, therefore, permissible to allow the roll trim servo, as well as the elevon drive servo, to oscillate at a high frequency and small amplitude. In order to keep the speed of response of the trim servo high and thus pervent the introduction of additional time lags in the system, the amplitude of oscillation was made small and the frequency high in comparison with the amplitude and frequency of the elevon servos. The power required to drive the elevon servos is decreased at the same time. The actual values obtained were about $\pm 5/8^{\circ}$ amplitude at each elevon and about 7 cps frequency.

The average accuracy is determined by the permissible homing error in yaw. In order to keep the yaw error small, it is necessary to secure a large ratio of roll angle to yaw angle. If this is secured, the roll axis control can be designed at a comparatively low accuracy level without impairing the ultimate accuracy in heading. Actual values of the roll-yaw ratio and accuracy will be discussed in a later section.

In order to obtain the correct dynamic performance the up-down signal is integrated and applied to the gyro pitch reference. A simple and accurate means of performing this integration was secured in the trim servo by employing rate feedback. The output rate of this servo is proportional to the input signal; thus, the output angle is proportional to the time integral of the signal.

The accuracy requirement is higher in pitch than in roll, since inaccuracies in the trim and gyro feedback directly affect the pitch heading. However, an error which persists long enough for the integrating process to take place (for example, a slow drift of the gyro vertical) will be corrected by the trim servo. Backlash errors are more difficult to remove; consequently, the feedback links are designed to operate with the least friction and backlash that could be obtained with parts that were sufficiently rugged and easy to manufacture. Any remaining backlash was taken up by a spring provided for the purpose.

The rate accuracy of the output motion does not affect the accuracy in heading. Too high a ratio of output rate to input signal will cause instability, while too low a ratio will reduce the maneuverability of the glider. An adjustment is provided in the electronic amplifier to set this ratio.

It is essential that the rate servo have a clearly defined zero or dead zone. Unless a dead zone is present, the trim servo will tend to drift away from its last setting if the signal is temporarily lost. (See Section 4.3.8.)

In general, the output of the pitch trim servo is characterized by very slow rates of change and very small angular changes. The design problem is much simpler than that for the roll trim servo.

4.3.4 Roll Trim Servo

4.3.4.1 Description

The roll trim servo is designed to position the sleeve of the roll air valve in the gyro in response to an electrical signal from the R-L output of the radar and a mechanical signal from the roll elevon drive. The servo must have a high speed of response, moderate accuracy, and an output torque sufficiently large to overcome bearing friction, vibrational forces, and take-up spring torque. The servo is physically divided into two units, a mechanical drive which is enclosed in the casting shown in Figure 1.4, and at the right of Figure 5.3.7, and an electronic amplifier which is incorporated in the radar nose assembly. In addition, the motor control relays, the battery supply for the potentiometer, and the battery-switching relay are mounted on the deck plates where space is available. (See Figure 1.4).

The essential parts of the mechanical drive are shown schematically in Figure 1.3. The roll trim servomotor is a low-voltage d-c shunt motor. It carries on its shaft a 48-pitch, single-thread worm (17) which is meshed with an 80-tooth, 48-pitch, single-thread worm gear (19). The worm gear rotates freely on its shaft, but is coupled to it by means of a friction clutch (18). The central shaft, which serves as a bearing for the worm wheel, carries the driven side of the friction clutch, the wiper arm of a modified General Radio Type 214-A potentiometer (16), the external dial pointer (14), and a small gear (24), which drives the upper sun gear of the mechanical differential (23). The travel of the potentiometer is limited by fixed stops on the pointer. These stops are set at 136° each side of the center of the potentiometer winding in order to prevent the wiper arm from leaving the end of the winding.

The lower sun gear of the differential is coupled to a sector gear (25) on the roll input shaft of the pantograph linkage. The output angle of rotation of the differential, obtained from the spider and shaft, equals 1/2 the sum of the angle fed back from the roll elevon drive and the angle set in by the trim servo.

The differential output is geared to the override clutch (22). This clutch is employed to limit the travel of the air valve sleeve to $\pm 36^{\circ}$ (the maximum allowable travel -- Section 4.5). The driven side of the clutch carries fixed stops at -25.8°. Any greater rotation is taken up by a coil spring. Since there is no permanent slippage through the clutch, the trim servo and air valve will return to their proper alignment when the limit stop is disengaged. The coil spring (21) at the driving side of the override clutch is employed to take up any backlash in the mechanical data-transmission system. The driven side of the clutch is coupled to the air-valve sleeve by means of an adjustable linkage (26).

4.3.4.2 Components and Design

The major requirements for the roll trim servomotor are as follows:

1. High ratio of torque to inertia
2. Equal torque in both directions of rotation
3. Intermittent operation
4. Operation from 14-volt, d-c supply
5. Low bearing and brush friction
6. Size as small as is consistent with effective heat dissipation
7. Minimum stalled torque = 3 in.oz.
8. Speed at 1 in.oz. load = 5000 rpm

The motor chosen for this application was a d-c shunt motor designed for operation on 28 volts at approximately 1/150 hp. The motor was manufactured by the John Oster Co. to Bell Laboratories specification KS-5968 List 1 (manufacturer's Type A-14). The A-14 motor, when operated with 28-volts field supply and 14-volts armature supply, meets the above requirements. The torque-speed curve and other pertinent data will be found in Figure 4.3.1.

The potentiometer was a modification of the General Radio Co. Type 214-A. Improved shaft bearings were employed, a pigtail added to give a positive connection to the wiper arm, and the wiper arm modified to give a sharper point of contact.

The differential is a standard Sperry part, No. 642487-H. This differential has considerable backlash, but it is taken up by the anti-backlash spring.

The entire roll trim servo assembly is housed in an aluminum casting which is secured to the upper deck plate, while the motor, potentiometer, and dial are supported by the casting. All shafts run in ball bearings with the exception of one idler shaft and the clutches. All gears were lubricated with a light, low-temperature grease.

4.3.4.3 Measurements and Adjustments

A dial, (15) in Figure 1.3, is placed above the potentiometer to indicate the trim reference angle. The pointer of this dial is aligned with the potentiometer arm. The dial plate can be adjusted with respect to the potentiometer so that the reading will be zero when the arm is at the midpoint of the winding. The dial is calibrated in equivalent degrees of differential elevon angle, the angle between the limit stops being divided 50°–0–50°. The equivalent roll angle is twice the dial reading. For example, a dial reading of 10° counterclockwise will cause the port elevon to move 5° down and the starboard 5° up (10° differential). The glider will eventually reach a roll angle of 20° to starboard.

An alternative method of measuring trim angle is illustrated in Figure 4.3.2. The voltmeter can be calibrated to read the same as the dial.

In addition to the dial adjustment mentioned above, two adjustments are provided for setting up the initial alignment of the roll data system. The entire potentiometer assembly, including the dial plate, can be rotated with respect to the housing, thus providing an adjustment of the zero position of the trim angle. The length of the push rod in the linkage which connects the trim servo to the air valve can be changed to adjust the zero of the air valve.

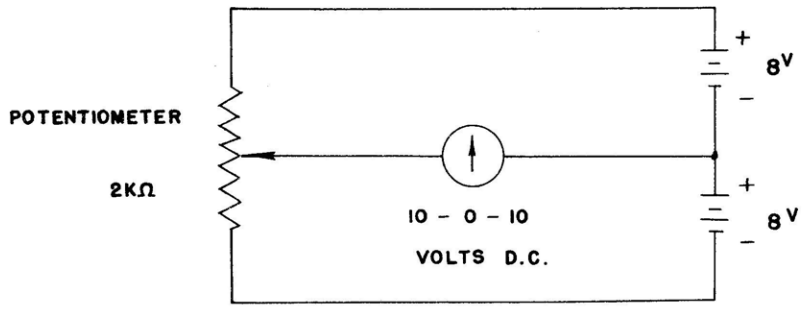


Figure 4.3.2

Alternative Method of Measuring Trim Angle

The alignment procedure is as follows:

1. Before assembly, set the dial and pointer approximately at zero, and set the pantograph linkage at its neutral position.

2. During assembly of roll trim casting to deck plate, adjust output sector gear so that it is positioned at the center of its travel.

3. With the air valve at zero, set the gyro linkage lever at the specified angle, and pin to the shaft.

4. Zero the roll dial and pointer accurately, using voltmeter method to determine potentiometer center.

5. Connect the push rod and make rough alignment by changing its length.

6. Make final adjustment either by changing length of the push rod or position of potentiometer. It was found by experience that this adjustment could be made most easily by adjusting the push rod length.

4.3.5 Roll Trim Amplifier

The electronic amplifier for the roll trim servo is located in the nose assembly. The amplifier comprises two tubes, a 6SL7-GT voltage amplifier and adder and a 7F8 power amplifier, together with the associated input and interstage coupling networks. D-c input signals are obtained from the R-L commutator segments and from the potentiometer in the trim servo. The output is delivered to two motor-control relays located in the control system. Filament and plate power is obtained from the radar power supply.

The roll trim amplifier is shown at the lower right of Figure 4.3.3. The signal from the commutator appearing at points L and R is first filtered and then passed through an integrating network to the grids of the adder-amplifier tube, V_7. The filter circuit consists of R_{37}, R_{39}, C_{15} in one channel, and R_{38}, R_{40}, and C_{16} in the other. R_{39} and R_{40} are required to provide a d-c ground return for the grids of V_7. The filter time constant is .02 seconds, while the d-c attenuation through the circuit is 1.7.

The circuit comprising R_{41}, R_{42}, R_{43}, R_{44}, and C_{14} is commonly designated an under-compensating integral network. The response of this circuit to signals which are changing rapidly with respect to the time constant of the network is approximately that which would be obtained from the circuit with C_{14} short-circuited. The attenuation with the condenser short-circuited is 9.4, and will be called the transient attenuation. If a constant signal is applied for a time considerably longer than the circuit time constant, C_{14} will be charged to the signal voltage. The attenuation of the network will be unity, and is designated the static attenuation. The time constant of the voltage applied to the grids is 9.4 seconds. The resistance and capacitance is proportioned so as to use the highest resistance possible without causing grid current instability.

In operation, the integrating network has an attenuation of 9.4 to a rapidly changing signal. The gain of the following circuits is adjusted to obtain the desired degree of missile stability. In order to secure greater precision when a constant error is present, as in following a target moving with constant velocity, the network is designed to produce a steady decrease in attenuation. The attenuation will fall to approximately 5.5 at a time 9 seconds after the application of the error. Eventually it will fall to 1. The stability of the system is unimpaired, since the attenuation to transient signals is still 9.4.

A set of normally closed contacts on relay R_{y1} is employed to short-circuit the integrating condenser. The relay is actuated by an arming switch located in the junction box. The arming pin (6 in Figure 1.2) is pulled out when the glider is dropped. Removal of the pin actuates a group of switches and also initiates a timing mechanism. After 3 seconds have elapsed, a second group of switches is actuated. One set of contacts in this group energizes relay R_{y1} and removes the short-circuit from C_{14}. Thus no extraneous voltage can build up across C_{14} during the bombing run and initial dropaway. If the condenser were permitted to charge up, an extremely violent roll transient might result when the servo is first turned on. After 3 seconds the glider is essentially on course and corrections in heading can be introduced with safety.

The feedback signal from potentiometer R_{54} is introduced across the cathode resistors, R_{45} and R_{46}. The grid-cathode voltage impressed on either section of V_7 will then be the difference between the grid-ground signal voltage and one-half the feedback voltage. If a difference exists between these two voltages, an amplified differential voltage will appear between the plates of V_7. This voltage will be further amplified by V_8, will actuate relays R_{y3} and R_{y4}, and will thereby cause motor M_1 to drive the feedback potentiometer. The resultant change in feedback voltage will be in such a direction as to reduce the differential voltage. The system will thus tend to equalize the grid-cathode voltages of V_{7A} and V_{7B}.

The grid bias for V_7 is obtained principally from cathode resistor R_{36}. By employing a very high resistance at this point the circuit is made highly degenerative to a simultaneous increase or decrease in the plate currents of both tube sections. The circuit is thus stabilized for changes in plate supply voltage, cathode temperature, grid current, and the like. It will also discriminate against non-differential signals appearing at the commutator. Such signals are always present, the magnitude varying with the magnitude of the differential signal to such an extent that, if fixed bias were employed, the servo would become completely inoperative.

The difference between the signal and feedback voltages is amplified by V_7 and applied to a pair of lead networks, R_{49}-R_{51}-C_{17} and R_{50}-R_{52}-C_{18}. These networks improve the high-frequency response of the amplifier, and thereby reduce the amplitude and increase the frequency of the servo oscillation. The lead network attenuation is approximately 3, and the time constant is .03 seconds.

The output of the lead networks is amplified by V_8, a Type 7F8 dual triode. The 7F8 has about the same power output as a 6SN7, but the transconductance is about twice as great. However, because of close spacing between the elements, the tube tends to develop excessive grid current.

The plate current of each triode of V_8 flows through a relay R_{y3} or R_{y4}. These are sensitive, high-speed, hermetically-sealed relays, Allied Control Type BGHx-1.

The cathode resistor of V_8 is so adjusted that, with signal equal to feedback voltage, equal currents of 4.0 milliamperes flow in each relay coil. The relays may be either picked up or dropped out, depending on the previous history of operation. If sufficient differential current is applied so that R_{y3} is picked up and R_{y4} dropped out, current will flow downward through the armature of motor M_1. If the direction of the differential current is reversed, R_{y4} will pick up, R_{y3} will drop out, and the armature current will flow upward. Since the field is separately excited, the motor will be reversed.

A typical sequence of operations in the servo loop will be described. Consider the circuit to be at rest with no signal input, no feedback voltage (wiper at center of potentiometer winding), balanced relay currents, and both relays dropped out. Assume a displacement of the potentiometer arm in the direction of R_{55}. Then this arm and the cathode of V_{7A} will become more positive, the plate current of V_{7A} will decrease, the grid of V_{8A} will become more positive, and the plate current of V_{8A} will increase. The plate current of V_{8B} will be correspondingly decreased. R_{y4} will pick up, R_{y3} will drop out, and the motor will rotate in the direction to move the potentiometer arm toward R_{56}. This motion will continue until the arm has produced a reverse feedback voltage of sufficient magnitude to reverse the relays. The potentiometer arm will thus be driven back and forth about the position of zero voltage, the frequency and amplitude of the oscillation depending on the gain of the amplifier, torque-to-inertia ratio of the motor, the magnitude of the time lags in the system, and the constants of the lead network. If a signal voltage is now instantaneously applied, the arm of the potentiometer will rotate at constant speed until a new equilibrium position is reached, when the normal oscillation will be resumed.

Certain auxiliary equipment is shown in Figure 4.3.3. Condensers C_{19} and C_{20} are employed for contact protection and interference elimination. Battery B_1 is a small dry battery which supplies the feedback potentiometer. Relay R_{y2} connects the battery to its load when power is applied to motor M_1, thus preventing drain on the battery when the servo is not operating. Switch S_1 is the 3-second delay arming switch previously mentioned in this section. It supplies power to the integral condenser relay, the motor, and the battery relay.

4.3.6 Electrical Component Characteristics

The majority of the electrical components are standard items covered by various JAN specifications. The sensitive relays, motor, potentiometer, and battery are

exceptions. The motor and potentiometer are described in a previous section (4.3.4.2). The important specifications for the sensitive relay are given below:

Manufacturer	Allied Control Company
Type No.	BGHX-1
Contact Arrangement	SPDT
Contact Material	Tungsten
Coil Resistance	12,000 ohms
Pickup Current	5.0 ± 0.2 milliamperes
Dropout Current	3.5 ± 0.2 milliamperes
Pickup Time	3.0 milliseconds (maximum)
Construction	Hermetically sealed in metal can; plug-in base

Battery B_1 is a small 45-volt dry battery of the type designed for portable radios. It is manufactured by the Burgess Battery Company as their Type W30BP. The battery has two disadvantages in this application. In the first place, the voltage drops off appreciably at low temperatures. Secondly, the voltage drops off during use to such an extent that replacement would be necessary if testing were continued for more than an hour or so. (See Figure 4.3.4.) Since the roll servo gain is directly proportional to the potentiometer supply voltage, constancy of voltage is the most important characteristic of the battery. Because of its small size, convenience, and availability, the battery was employed in the flight test model, but was replaced by a rectifier-filter supply in later models.

4.3.7 Pitch Trim Servo

4.3.7.1 Description

The pitch trim servo is designed to position the sleeve of the pitch air valve in the gyro in response to an electrical signal from the U-D output of the radar and a mechanical signal from the pitch elevon drive. The output requirements are similar to those for the roll trim servo, but the speed of response need not be so high. The mechanical drive is contained in the casting on the pitch side of the control system (Figure 1.4.) (See also the left-hand unit of Figure 5.3.7.) The electronic amplifier is incorporated in the radar nose assembly, while the motor control relays are mounted, together with the roll trim servo relays, in the relay box on the under side of the deck plate (Figure 1.4.)

The essential parts of the mechanical drive are shown schematically in Figure 1.3. The servomotor (5) is the same as the one used in the roll trim servo. It is geared through a 4:1 reduction to a d-c voltage generator having a permanent-magnet field (6). The generator shaft is coupled to an extension which carries a 48-pitch, double-thread worm (4). The worm is meshed with a 100-tooth, 48-pitch, double-thread worm gear (3), which is coupled to the central shaft (7) by means of a slip clutch (2), identical to the roll trim slip clutch. The central shaft drives the adjusting dial (1) through a 3:1 reduction. The dial carries limit stops which are normally set for $+33\text{-}1/3^\circ$ and -300° from the neutral position of the output.

The central shaft is geared to the upper sun gear of a differential (8) identical to that in the roll trim servo. The lower sun gear is meshed with a sector gear (9) on

the pitch input shaft of the pantograph linkage. The output of the differential is the sum of the average elevon angle and the pitch reference angle.

The differential output shaft is geared (10 and 11) to an adjustable linkage (13) which drives the sleeve of the pitch air valve. A backlash take-up spring (12) is provided as in the roll trim servo, but an override clutch is not necessary, since the sum of the elevon angle and reference angle is never greater than 25° at the valve.

4.3.7.2 Components

The requirements for the servomotor and differential in the pitch trim servo are much the same as those in roll. Identical units were accordingly employed.

The generator is required to produce a voltage proportional to speed over the range from zero to about 2000 rpm. The calibration should be linear, permanent, and identical for either direction of rotation. The generator chosen for this application was the Electric Indicator Company's Type C-47. Permanence of calibration and reversibility is achieved by employing a permanent-magnet field and metallic brushes set accurately on the electrical neutral. A typical curve of generated voltage vs. rpm is shown in Figure 4.3.5. The volts per rpm may vary by ±10 per cent between units. For this and other reasons, a gain control was included in the pitch trim servo amplifier.

4.3.7.3 Measurements and Adjustments

The dial provided on the pitch trim servo is calibrated directly in glide angle from 0 to 20 degrees dive, with additional reference marks at the limit stop settings of 5° climb and 45° dive. In making the initial glide path setting previous to a flight, it is necessary to compensate for the attitude of the glider when the gyro is uncaged. This attitude is dependent on the angle at which the glider is supported in the bomb rack and on the angle of attack of the dropping plane. A fixed scale marked "Launch Angle" serves as the index for the dial. When the desired glide path is set opposite to the estimated launch angle, the correction is made automatically.

The initial alignment of the pitch data system is made in a manner similar to that employed for the roll axis, with the exception that only one adjustment, the length of the push rod to the gyro, is provided. The procedure is as follows:

1. Before assembly, set the pitch dial so that +5 of the dial is opposite the 0 index mark. The dial adjustment screws should be at the center of their slots.
2. Before assembly, set the pantograph linkage at its neutral position.
3. During assembly of pitch trim casting to deck plate, adjust output sector gear so that it is positioned at the center of its travel.
4. With the air valve at zero, set the gyro linkage lever at the specified angle and pin to the shaft.
5. Connect push rod and adjust its length to bring the air valve back to zero.

As a result of setting the dial plate at +5° (the angle of attack of the glider) when the control system is at its neutral position, the dial is made direct-reading in glide angle. This may be visualized as follows:

If the glider is dropped with the dial set at +5° and no radar information, it will attempt to fly with its longitudinal axis level. The glide path will then be 5 degrees below horizontal.

If a glide path of, for example, 13 degrees is desired, it will be necessary to turn the dial 8 degrees from its initial position. This will bring the +13° point opposite the index.

In adjusting the trim servo gain it is necessary to know the speed of rotation of the output. This is most easily accomplished by measuring the time required to turn from one limit stop to the other. The speed can also be determined by measuring the voltage output of the generator. This is not an accurate method unless the generator has been calibrated.

4.3.8 Pitch Trim Amplifier

The pitch trim amplifier (Figure 4.3.3) is basically similar to the roll trim amplifier, but differs in some details of circuit design. The following discussion is principally concerned with the differences between these units.

The amplifier comprises an adder-amplifier tube, a power tube, associated input and interstage coupling networks, and two motor-control relays. The input is obtained from the U-D commutator segments through a filter network having a time constant of .026 seconds. The d-c attenuation of the filter can be varied from 1.2 to infinity by means of the dual potentiometer R_{21}-R_{22}. This potentiometer is the pitch sensitivity control, and is employed to adjust the ratio between trim reference speed and U-D error signal.

The integral circuit (R_{23}-R_{24}-R_{25}-R_{26}-C_{14}) has the same function as the roll integral circuit, but has a time constant of 19 seconds and an attenuation of 2.9.

Condensers C_{11} and C_{12} were added after results of the first two flight tests indicated that additional maneuverability in pitch was desirable, particularly in the last few seconds of flight. These condensers, together with resistors R_{23}, R_{24}, R_{25}, and R_{26} form a lead network. The function of this lead network is not, as in the roll trim amplifier, to improve the servo performance. Its only effect is to increase the response of the glider to a rapidly-varying pitch signal.

The plates of V_5, the 6SL7 adder-amplifier tube, are directly connected to the grids of V_6, a type 6SN7, with no lead network. The lead network can be dispensed with as the servo is not required to oscillate in position at a high frequency. Resistors R_{31} and R_{32} form a bleeder to set the proper grid bias on V_6. With the lead network eliminated, less gain is required in the output stage. A 6SN7 supplies enough gain without the disadvantages of the 7F8 (Section 4.3.5.)

The description of the action of the adder tube and the cathode bias resistor in the roll trim amplifier applies equally well to the corresponding elements in the pitch trim amplifier. Relays R_{y5} and R_{y6} operate in the same fashion as R_{y3} and R_{y4}.

The manner in which the pitch servo produces a rate of turn proportional to signal input voltage is analogous to that in which the roll servo produces a position proportional to signal. The action is most easily visualized with the aid of a diagram such as Figure 4.3.6 which presents an ideal representation of output speed vs. time for a succession of input signsls, ϵ_p.

If we assume a pitch error angle ϵ_{p1} to be instantaneously applied to the radar system, a differential voltage will be produced at the trim amplifier input. Neglecting time lags in the trim servo, the output shaft of the servo will start to move at time = 0. The speed will increase at a constant acceleration determined by the motor torque and

load until a speed V_{1a} (Figure 4.3.6) is reached. At this speed the feedback voltage from the generator is large enough to reverse the relays. The motor then decelerates to a speed V_{1b}, when the relays again reverse and the cycle is repeated. An average speed V_1 is determined which is half-way between V_{1a} and V_{1b} and has the value $V_1 = k\epsilon_{p1}$, where k is a constant determined by the gain of the feedback generator, the radar receiver, and the resistance networks in the signal and feedback loops. Since the amplifier has a high gain and the torque-inertia ratio of the servo is high, the rate oscillation has a frequency of at least 20 cps and an amplitude which is so small as to be barely detectable.

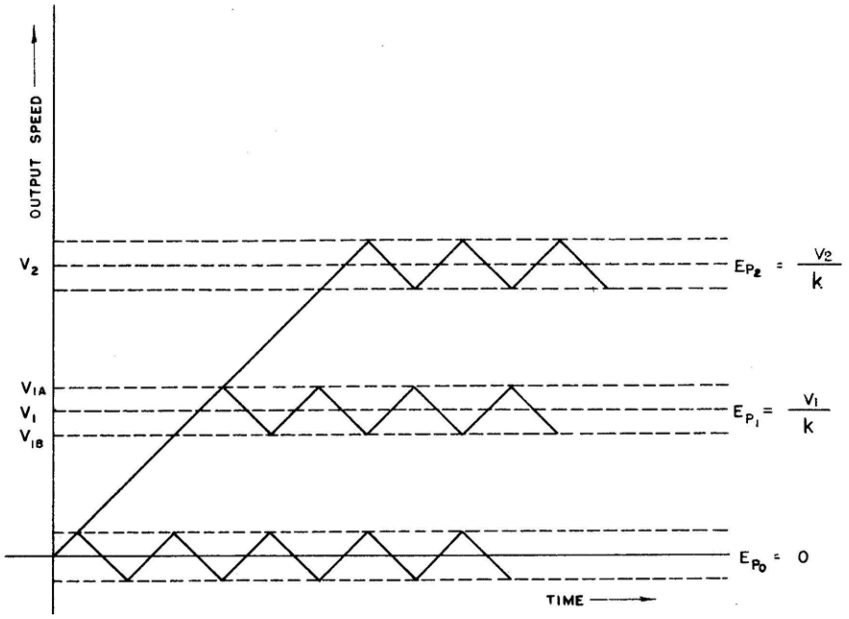


Figure 4.3.6

Simplified Representation of Pitch Trim Servo Operation

If the signal is increased to a value ϵ_{p2}, the servo will oscillate about a new average rate, $V_2 = k_{\epsilon p2}$, Figure 4.3.6.

If the radar signal is lost ($\epsilon_p = 0$), the output will remain stationary, since the signal will not be great enough to trip the relays. A small amount of unbalance in the amplifier can be tolerated, provided it is not so great as to exceed the dead space indicated by the dashed lines.

4.3.9 Electrical Component Characteristics

The discussion in previous sections covers all components of the pitch trim amplifier.

4.4 Clutch Units

4.4.1 Choice of Servomotor

The first problem involved in the design of a servomechanism to operate the elevons of the glider was the choice of a servomotor. As has been previously pointed out in this report the type of control proposed by M.I.T. required that the position of the elevons be set rather than merely controling the direction of their rotation. There are two fundamental types of servomechanisms which can be considered for such an application. The so-called proportional control type of system permits smooth control and high accuracy. This is the type of servo commonly used in such applications as antiaircraft fire control. A second type, the relay servo, or on-off control, merely permits full torque to be developed in the appropriate direction by the servomotor. In general the relay servo comprises smaller and simpler components than the proportional type, since it is not necessary for the servo controller to handle a large amount of the power involved. Furthermore, this type of servo does not require measuring devices whose response is accurately linear. For these reasons it was decided to concentrate on the development of a relay servo for this application.

Since on-off servos are inherently unstable in the presence of load torques it is particularly important that the amplitude of oscillations be held to a relatively small value. It can be shown that the requirements of speed of response for these systems are much more stringent than for the proportional type servo. A very rapid speed of response means that the servomotor must be capable of very high accelerations. On this basis alone it was felt that present-day types of electric motors would be excluded from consideration, since the maximum acceleration of such motors are limited to about 20,000 radians per second squared. There are only two basic types of servomotors commonly available which can substantially exceed this value of maximum acceleration: hydraulic motors and devices which clutch into a constant speed source.

In choosing between these two types of motors the problem of power supply is of first importance. In earlier models of the Bat control system provision had been made for a 14-volt storage battery in the design of the airframe. Because of the urgency of the program it was felt highly desirable to make use of this battery instead of alternative arrangements, since time could be saved by complete elimination of the power supply problem. It was felt that such schemes as hydraulic accumulators to supply a source of pressure for hydraulic controls might possibly offer long term advantages, but such schemes were laid aside because it was felt that more time would be required in their development.

Having determined on the 14-volt storage battery as a primary source of power it was evident that an electric motor and hydraulic pump would be required in any sort of hydraulic control. In such a system a piston type servomotor would permit the elimination of a gear train which would be an advantage from the standpoint of size and simplicity.

Like the hydraulic control a clutch type system would require an electric drive motor but in place of the pump, valve, and piston would require only a relatively small clutch. Perhaps the determining factor in the final choice was the desire to eliminate hydraulic oil from the system. With hydraulic controls there is always the problem of preventing oil leakage and difficulties with increased viscosity at extremely low temperatures.

The servo developed by the Bureau of Standards had used a clash gear arrangement to reverse the direction of rotation of the elevons. The servomotor in that system was required only to rotate the elevons at constant velocity in the appropriate direction to correct the glider's course, while in the M.I.T. system it was felt desirable to control the position rather than merely the direction of rotation of the control surfaces. This scheme requires a much higher frequency of reversal, and for this reason it was felt that clash gearing would be unsatisfactory. Accordingly, it was decided to concentrate on the development of a friction clutch capable of relatively high frequency reversals.

4.4.2 Analysis of Clutch Design Problem

4.4.2.1 General Configuration

The space available for the power controls for both axes was very restricted, being of the order of 600 cubic inches, including gear reductions. A preliminary estimate of the power of drive motor had been made, being about 3/8 hp for most of the flight with transients to 3/4 or 1 hp. Since the axis of rotation of elements of the pantograph was vertical, and it was desired that the power gear train utilize spur gears only, for simplicity and efficiency, it appeared that the axis of the motor would have to be vertical. This was found to be unfeasible for ordinary proportions of motors of the required rating, there not being enough height available to accommodate a reasonable motor length. Accordingly, it was decided to incorporate bevel gears in the clutches, thus permitting the motor axis to be horizontal, and gaining other advantages in the construction of the clutch itself. A double-ended motor of compact design was selected, provision being made for mounting one clutch on either end. See Figure 1.5. In order to obtain the smallest frame size the speed was made reasonably high, namely 5000 rpm.

4.4.2.2 Speed of Response

There was not much concern about the time required to change the position of the elevons with this system. The chief concern was to make the engagement of the clutch sufficiently rapid to limit the amplitude of oscillation to a reasonable value, say $\pm 2^{\circ}$ or 3°. This amplitude is determined primarily by two time lags, that necessary to develop force at the electro-magnet, and that necessary to decelerate the clutch rotor from full speed to a stop. It is during this deceleration time that power is wasted as heat, so it is desirable that this be short for a second reason. To accomplish high acceleration with a given torque the effective inertia of the gear train and clutch rotor must be a minimum. The effective inertia of the clutch can be minimized in two ways (a) by making the rotor have the least possible rotational inertia and (b) by interposing the clutch at as low a speed in the gear train as practicable. Method (a) is limited primarily by permissible bearing pressure on the clutch facings while method (b) is limited primarily by the size of the clutch at low speeds and high torques, and also by the excessive forces required of the electro-magnet. Since the output or elevon speed was specified by the requirements of the control surfaces as about 90 degrees per second and the drive motor speed had been tentatively established at 5000 rpm, the overall gear reduction was fixed at about 330:1. A few exploratory designs indicated that

introducing the clutches at a reduction of about 2.2:1 from the drive motor would result in approximately optimum performance as a result of compromise among the relevant factors. Accordingly, this established the torque and speed at which the clutch was to operate, thus satisfying the demands of method (b) above.

In order to minimize both the inertia and size of the clutch unit a multi-disc type seemed desirable, since it permits a greater torque to be developed at a smaller radius than an equivalent single disc unit, thus obtaining a higher maximum acceleration or torque-to-inertia ratio. As finally worked out, the clutch had six driven discs for each direction of rotation. There is an inherent problem with such a multi-disc clutch, however, in providing for coincidence in the position of the armature required to compress the discs with that necessary to establish the desired air gap. This difficulty was effectively overcome by using an adjusting nut as shown in the cross-section view (Figure 1.6) to secure this coincidence regardless of the variation in thickness of the pile of clutch discs. Since this scheme is not self-compensating and is based on the assumption that the original thickness of clutch discs is to be maintained accurately, metallic rather than cork or asbestos facings were indicated. The coefficient of friction for most such facings is relatively constant around 0.2.

The remaining design problem, that of a high speed electromagnetic actuator, was analyzed as follows: The force required to develop the necessary torque was known. By utlizing as high a current as possible the most rapid action would be obtained. Tests on the dynamotor, which was to supply this power, showed that by overloading the dynamotor, to what was felt to be a safe amount for this short life application, about 50 milliamperes at 285 volts could be used for each clutch. Experimental air relays had been operated for some time at about one-half this rating and it was felt that no great trouble would be encountered in doubling their dissipation. Therefore, 50 milliamperes was chosen as a basis for the magnet design.

In order to avoid unusually delicate construction it was thought that 38 AWG was about as fine as would be desirable for a winding. The magnet proportions were designed on the basis of maximum force-to-power ratio and the coil size checked for heat dissipation. After two or three models were made and tested the proportions were varied slightly and the final design obtained. The air gap was made about .012 average with tolerances varying as much as ±.002. By removing metal in the armature a saturation effect was introduced to make the slope of this force-distance curve fairly flat in the operating range.

Having determined the magnet coil resistance, additional series resistance was added to limit the steady-state current to 50 milliamperes, plus reducing the time-constant of the circuit.

The speed of response of the complete clutch was found to be determined primarily by the time constant of the magnetic circuit and the time required for the clutch rotor to accelerate to full speed. In this particular application the time to build up maximum flux in the magnet is of the order of six milliseconds, while the time required for the rotor to accelerate to full speed is about one millisecond.

4.4.3 Construction and Operating Details

The clutch unit finally developed is shown in the cross-section drawing, Figure 1.6, and in the exploded view, Figure 4.4.1. The electrical circuit used to actuate the clutch is shown in Figure 4.4.2.

The size of each clutch is approximately 6" x 4" x 4" and the weight is about 7.4 pounds. One clutch is capable of transmitting about 1 hp as used in the Bat application where the output speed is about 2200 rpm and the available torque about 25 in-lb. The life of the unit is at least 1/2 hour under the most abusive operation that can be encountered in this application, i.e., controlling the maximum torque the motor can supply without stalling. The housing is designed to mount on a standard aircraft motor flange. The most significant characteristic of the clutch is its extremely high maximum oscillation which is of the order of 250,000 radians per second squared.

Basically the function of this clutch is to reverse the direction of rotation of a gear train whenever its position differs from that called for by the automatic pilot. Referring to Figure 1.6 the driving power from the electric motor rotates the bevel pinion (1) at a speed of approximately 5000 rpm. Two identical bevel gears (6) mesh with this pinion on opposite sides, so that they rotate in opposite directions. A gear reduction of about 2.2 to 1 is obtained at this mesh. The function of the clutch is to engage the clutch shaft with whichever bevel gear is turning in the required direction. Seven driving discs (4) are splined to each bevel gear and six driven discs (5) are splined to each end of the clutch shaft. These discs are arranged alternately so that pressure on the end of the stack produces 12 driving surfaces. A thrust bearing is mounted at one end of the stack of discs and rotates with them. Engagement is accomplished by the three thrust pins (10) which are magnetically actuated whenever the air relay closes a circuit through the coil (8). When the control circuit is closed the armature (9) is drawn toward the pole faces of the magnet, but prevented from touching them by the pins (10) which transmit the force to the thrust bearing (7). The adjusting nut (3) which backs up the stack of clutch plates on the opposite end can be turned to adjust the magnitude of the magnet air gap. The unit is so designed that adjustment can be made before final assembly. A small retainer (2) can lock the adjusting nut in any of 18 positions per revolution, permitting positive locking of this nut within .001" of any desired position.

The photograph, Figure 4.4.1, shows the disassembled clutch. Here it can be seen how all of the rotating components are carried in the bevel gear as a subassembly, thus simplifying the problem of interchangeable manufacture. At the left of the photograph parts from the left end of the clutch are spread out to show the details of their design, while on the right these identical parts are shown assembled.

The more important technical data describing the design and performance of the clutch are tabulated in the following pages. Curves and data plots referred to in the technical data section appear under Section 4.4.4.5.

4.4.4 Technical Data

4.4.4.1 Clutch Physical Characteristics

A. Overall weight of unit 7.4 lbs.

B. Rotor
(Consists of 6 driven discs for each direction of rotation splined to output shaft but axially free.)

1. Driven Disc Specifications, M.I.T. Dwg. A-17212:

OD	1-1/8 inch
Thickness	.047
Material	Steel
	SAE 1095, Rockwell C45-50 connected to shaft by spline with .004-.008 chordal backlash.

Note: Production unit uses stainless steel, Type 420 at Rockwell C45-50.

2. Driving Disc Specification, M.I.T. Dwg. A-17210-2:

OD (Effective)	1-1/8 inch
ID (Effective)	13/16 inch
Thickness	.078 inch
Material	Raybestos powdered metal facing No. 1908-B.

3. Contact Area Per Face: 0.47 square inch

4. Effective Friction Radius for Perfect Contact: 0.50 inch

5. Effective Coefficient of Friction:

a. Dry	.16 to .17
b. Well oiled	.042

Note: See Curve 4.4.2.

6. Moment of Inertia:

a. Shaft	3.22×10^{-5} in.lb.sec.^{2}
12 Discs	5.95×10^{-5} in.lb.sec.^{2}
Total	9.17×10^{-5} in.lb.sec.^{2}
or	.035 in.^{2}lb.

7. Output Pinion Specifications: 14-1/2° involute, enlarged profile, cut by 1-7/8D hob, 12 teeth, 24 E.P., 3/8 active face.

Material	Nitralloy 135 modified
Fatigue strength	80,000-90,000 psi
Surface hardness	Above Rockwell C-60

C Magnet:

1. Material	Cold Rolled Steel SAE 1020 or 1112, unannealed

Note: Production units use annealed Armco magnet iron

2. Coil Specifications:	5600 turns, No. 38 wire 2100 ohms, resistance Class A insulation Tested at 2000 volts rms to ground

Note: Production test, 1000 volts rms

3. Air Gaps:

Minimum	.008 inch
Maximum	.015 inch

4. Pole Face Areas:

a. Inner	1.02 square inch
b. Outer	1.05 aquare inch

5. Force distance curve can be derived from Torque Air Gap Curve 4.4.4 noting that 1 in. lb. torque = 1.02 lb. force. Torque-Current Curves are shown in 4.4.5.

6. Coil inductance measured by impedance bridge at 1000 cycles:

a. In magnet	approximately 3.5 henries at .012 air gap
b. Coil alone	approximately 1.7 henries

D Clutch Gear Ratio:

Input to output	2222 to 1

4.4.4.2 Clutch Operating Characteristics:

A Pitch

1. Input Speed	5000 rpm
Output Speed	2250 rpm, maximum

2. Effective Inertia (at clutch) Including Gear Train:

1.18×10^{-4} in.lb.sec.^{2}

or

.045 in.^{2}lb.

3. Actuating Circuit (See Figure 4.4.2):

a. Supply	280 volts
b. Limiter	3500 ohms
c. D-c steady-state current	50 milliamperes

d. Peak voltage at break with .02 MF across contacts: 1000 volts approx.
e. Effective time lag due to magnet delay: .005-.010 seconds approx.

4. Output Torque:

Minimum at .015 gap	.8 in.lb.
Maximum at .008 gap	34 in.lb.

Note: See Curve 4.4.3

This yields the following range of torque-to-inertia ratios (maximum acceleration)

Clutch alone	163,000-370,000 rad/sec^2
With gear train	128,000-287,000 rad/sec^2
Time to bring gear train to full speed	1.85 milliseconds. to 0.83 milliseconds.

5. Dead Zone at Clutch: ±150° due to +1/8° at air valve, which is amplified 1300 times

6. Time Lag in Air Relay: Definite figures were not independently determined for this quantity. An idea of the magnitude is given by Curve 4.4.6.

7. Backlash (approximate) at Clutch from Air Valve:

40°-60° total on Model F Test Stand where data in this report was taken. This should be reduced to about 1 degree due to spring loading in final design.

8. Clutch Facing Data:

a. Maximum relative contact speed: 670 fpm
b. Maximum contact pressure: 65 psi for 30 lbs. magnet force

9. Clutch Pinion Data: See Section 4.4.4.1 - B7

Maximum torque	35 in.lb.
Maximum speed	3000 rpm, .002 error in action
a. Maximum static load	140 lb.
b. Maximum dynamic load	5.2 lb. x 1.35 (for pulsating load)
c. Wear load (Indefinite Operation)	220 lb.
d. Beam stress at maximum dynamic load	87,000 psi approx.

B Roll

All operating constants are identical except the following: static dead zone at clutch (estimated) ±8° due to ±1/8° at air valve, which is amplified 62 times for air valve = 2.4 x elevon speed.

4.4.4.3 Clutch Performance Characteristics

A Pitch

1. Under conditions enumerated in 4.4.4.2-A the no load amplitude and frequency are roughly:

   $\pm 300^{\circ}$-375° at clutch or

   $\pm 2.0^{\circ}$-2.5° at elevons at 8-10 cps

2. Performance in limit:

   Oscillates with considerably smaller amplitude and higher frequency than above.

3. Power consumption at 5000 rpm input speed with clutches vertical (approx.): for one clutch:

   a. Motor driving, but magnets unexcited: .05 hp
   b. Clutch oscillating at no load (12 cps): .15 hp

      (Driving test gear train with approx. 7.0×10^{-5} in.lb.sec.^{2} additional inertia at clutch)

   c. Clutch oscillating at 8 in.lb. torque: .28 hp
   d. Clutch torque = 16 in.lb.: .73 hp

4. Drag torques at room temperature: (Defined as torque necessary to turn shaft due to internal friction)

   a. With clutch vertical:

Input	approx. 2-3 in.oz.
Output	approx. 8 in.oz.

   b. With clutch horizontal:

Input	approx. 1-2 in.oz.
Output	approx. 1-2 in.oz.

5. Heat rise information is included in Curve 4.4.6.

B Roll

(All characteristics are identical with pitch except the following:)

Under conditions enumerated in 4.4.4.2-B the no load amplitude and frequency are roughly:

$\pm 210^{\circ}$-240° at clutch or

$\pm 1.4^{\circ}$-1.6° at elevons at 12-14 cps

C The following conditions appear to be best for running-in the units and serve as a check on operation:

Input shaft speed	1800 rpm
Dead zone	$\pm 6^{\circ}$
Series resistance	3500 ohms

D-c steady state current	50 milliamperes
Additional contactor inertia	7.0×10^{-5} in.lb.sec.^{2}

Under these conditions clutch should oscillate at $\pm 30^\circ$ maximum at 30 cps, minimum, and run-in time should be at least 1 hour.

4.4.4.4 Drive Motor Characteristics

A Name plate data:

Emerson Electric	
Frame - D34FJ183	Volts 14
Style - 445 - 0417	Amps. 55
HP - 1/2	RPM 5000
Winding -	Compound

For characteristics see Curve 4.4.8.

B Temperature Rise (Emerson Data):

1/2 hp for 2 hours	Frame 38° C Field 48° C
1/2 hp for 2 hours, followed by 3/4 hp for 10 min.	Frame 43° C Field 56° C

C Approximate current requirements supplying both axes:

a. Standby	32 amperes
b. Roll in electrical limit, pitch neutral	38 amperes
c. Pitch in electrical limit, roll neutral	70 amperes
d. Both axes in electrical limit	90 amperes
e. Clutch output stalled	Considerably greater than 150 amperes

4.5 Gyroscope

A number of alterations are required on the Sperry Mk 4 Bank and Climb Gyro shown in Figures 4.5.1, 4.5.2, 4.5.3, and 4.5.4. The parts removed and not used are the caging knob, shaft, and parts up to and not including the gear into the case; the rear end cover, the bellows assemblies; the front mounting plate, trimming knobs, shafts, and parts; the air manifold. In the air valves, the sleeves are replaced with a somewhat different design and the loading springs are removed.

External changes include the removal of the caging parts outside the case, the rear cover, and the front mounting plate because they served no use and occupied space required for other apparatus. The trim and signal adjustment parts are removed and discarded. A redesigned rear cover is used, and new parts are added so that rubber shock mounts may be used to support the gyro. This cover also supports and locates bearings for the output shafts of the air valves and provides an air inlet through which the driving air is supplied. The cover must be located accurately and properly sealed to prevent air leakage. In order to clear the top of the bomb, it is necessary to cut away about 1/4 inch of the bottom flange at the rear of the gyro case.

An entirely different means of caging is used in order to cage and uncage remotely (see Figures 4.5.2 and 4.5.4) with a motor doing the work, and an override loading spring that maintains an appreciable caging load upon the gyro as long as it is in the caged position. This eliminates stalling of the caging motor at the caged position and yet allows the gyro to be positively caged over a long period of time without becoming loose as a result of vibration or small amounts of motor rotation. Cams are mounted on the caging assembly and properly set to disconnect the motor in the two extremes of motion and at the same time to generate a signal which indicates to the dropping operator when either the caged or uncaged position is actually obtained.

Internal changes to the gyro include alterations to the air valves, alteration of the bulkhead, and changing of the gear ratio from roll air valve pilot to gyro gimbal. The air valves have the loading springs removed and the roll side uses a new loading spring and the pitch side uses the original spring except that it is moved to the opposite end of the valve. Both valves have the outboard shaft end of the sleeve replaced with a new and specially adapted design. The bulkhead has two special dowel holes added for positioning the end cover. The original gear ratio from air valve pilot to gimbal was unity. This is changed to 0.6 (32 pitch gears, 24T to 40T on Figure 1.3) to enable the gyro to tilt farther without moving the pilot valve to the reverse signal position. With the air valve limited to $\pm 36^{\circ}$, this then allows the gyro to tilt 60° before the signal limits are reached. This allows more maneuverability.

Due to rough handling before arrival in the laboratory all the gyros used had to be completely overhauled. This included disassembly, pivot repair, cleaning, re-assembly, balancing, adjustment, and testing. A certain amount of such work is mandatory even with the gyros in good condition since the extent of the alterations exposes critical parts to contamination and loss of adjustment. For repair information refer to Sperry Gyro-Pilot Repair Manual in which it will be noticed that the overhaul procedure is rather complex and involved. These particular gyros are specified for a shelf life of six months after which they should be disassembled and again overhauled. This arises from small amounts of corrosion and evaporation of lubricant and very often from damage as a result of shipment and mishandling.

4.6 Air Relays and Air Supply Components

4.6.1 Air Relays

Two relays are used to translate the roll and pitch directional information from the gyro into correcting action at the clutch unit. Each relay is equivalent to a

single-pole, double-throw switch mechanically operated by differential pressure from the air valve pickoffs on the gyro. The switching action serves to energize either of the two coil magnets controlling the output direction of a roll or pitch clutch drive unit.

Figure 1.3 shows small views of the air relays in relation to the entire assembly. The relays comprise two discs fastened together with a thin metal diaphragm between them. Each disc is provided with an air inlet connection leading to a recessed chamber in the disc and has an insulated contact screw which matches corresponding tungsten contact points of the diaphragm. The diaphragm material is flat, half-hard brass sheet .001" thick and approximately one inch in diameter.

An alternative diaphragm may be made of thin paper impregnated to resist moisture and give dimensional stability. The diaphragm of about one inch working diameter is provided with several concentric corrugations to increase the linearity of deflection with pressure. A small metal tongue having attached tungsten points is cemented to the diaphragm for electrical contact.

The relay sensitivity, as measured by the speed of response of the servo unit, is a function of the thinness, and kind of diaphragm material, the working diameter, and the grip setting. These factors are interdependent on each other for a given servo response so that the final choice of diaphragm specifications represents a compromise for optimum performance.

Because the switching takes place in an inductive circuit, it is necessary to protect the contact points against pitting and burning. Contact protection is provided through the use of small resistors and condensers which limit the arcing voltage and peak currents to safe values. The choice of these components must be such that the circuit operation is not affected appreciably while maintaining adequate contact protection.

4.6.2 Air Supply Components

The vertical gyro and its pickoffs are operated by a suction of four inches of mercury. It is the purpose of this section to describe components necessary to provide this suction at the gyro. First, the air intake must be fitted with a filter. At the recommendation of the Sperry Gyroscope Company, manufacturers of the gyro, a Purolator Type PA-12 was used. Air passages connecting the air filter to the gyro and the gyro to the source of suction had to be made large enough to prevent excessive pressure drops caused by the flow of air through them. At the same time the physical size of the piping and tubing could not exceed dimensions allowed by space considerations. It was felt that if sufficient suction could be provided to overcome a moderate amount of line pressure drop the lines could serve a two-fold function. The first function is, of course, providing a passage for the air, and the second that of pressure regulation. Excessive suction on the gyro element itself is minimized by permitting considerable pressure drop in the air lines. In this case it was possible to eliminate a pressure regulator.

The means of providing a suction was through the use of two single-throat venturis projecting into the air stream on each side of the airframe.

Performance data on the single-throat venturi and on a double-throat venturi was available for limited conditions. Exact information as to which venturi would be necessary to create the required amount of suction under the flying conditions to be encountered was not available. Therefore, it was deemed necessary to make a series of

flight tests. These tests were carried on by the Experimental Test Group at Johnsville, Pennsylvania, in April 1945. The bomb airframe was equipped with the proposed air supply system including air filter, gyro, venturi, and connecting tubing. In addition, two pressure taps, one on each side of the gyro, were brought out through the airframe. The airframe was mounted under a Navy SB2C and the external pressure taps brought up into the rear cockpit. The airplane was then flown under a series of flight conditions while the suction created by the venturi tubes was recorded. In this way sufficient information was obtained to determine a suitable air supply system. The system decided upon consisted of two single-throat venturi tubes, the above described filter, and interconnecting tubing. This tubing consisted of two parts; 3/8 inch ID copper tubing wherever possible and 1/4 inch ID flexible tubing for connection to the gyro itself. The flexible tubing was required because of the necessity of shock mounting the gyro. This air supply system proved satisfactory for the test drops of the first ten M.I.T. pilot models. Pressure data was obtained throughout the flight and indicated that the suction increased to at least ten inches of mercury. This indicates that the gyro and the air valves were receiving at least six inches of mercury suction toward the end of the homing run. At the point of dropping the gyro receives at least four inches of mercury suction. For test purposes it has been found that four inches of mercury suction on the gyro, including the flexible connections, is sufficient suction for satisfactory operation. The air supply system as described above has been found satisfactory in all tests to date and has been included with no major changes in the production model of the control system.

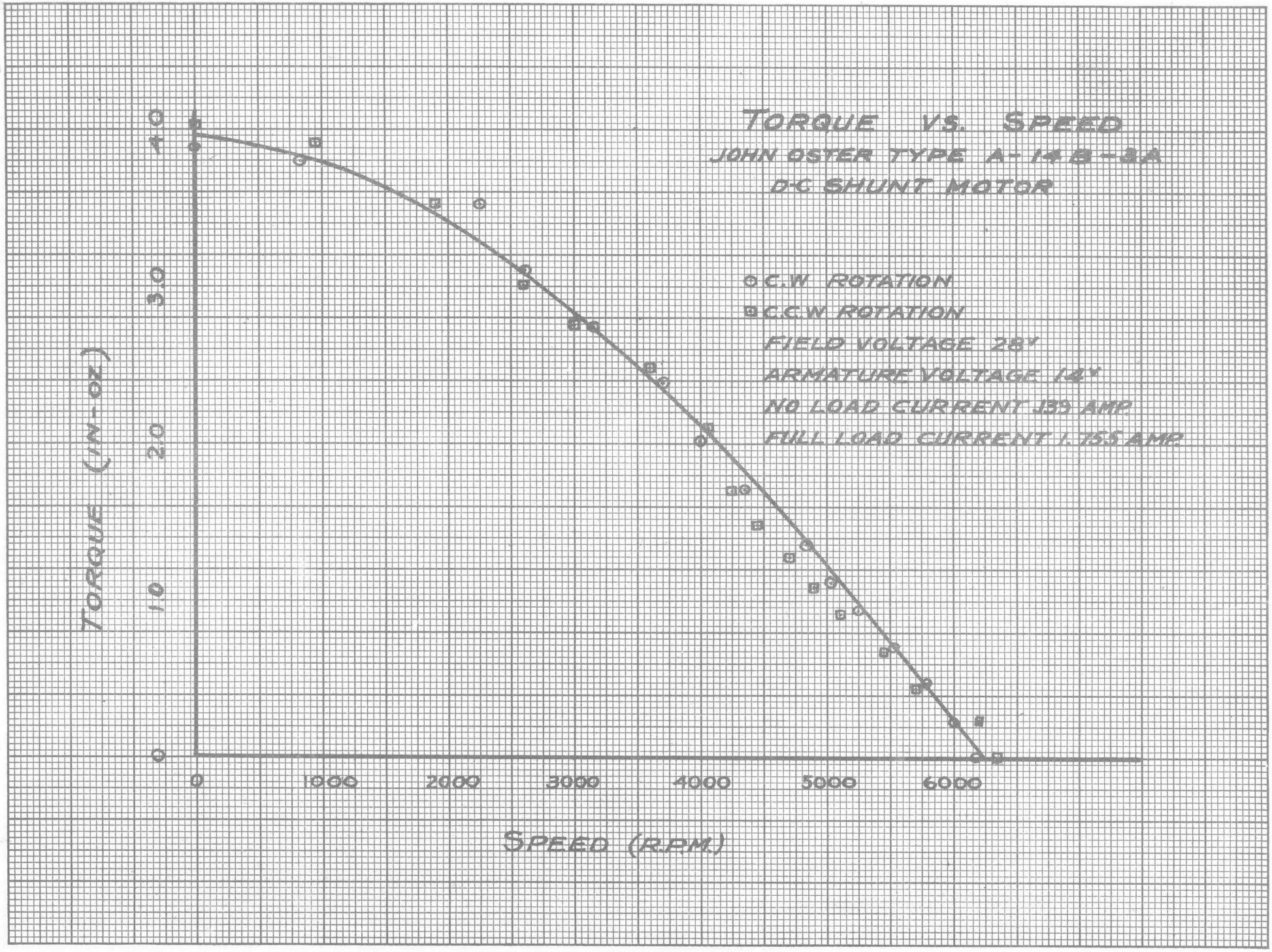

TORQUE VS. SPEED
JOHN OSTER TYPE A-14 B-3A
D-C SHUNT MOTOR
C.W ROTATION
C.C.W ROTATION
FIELD VOLTAGE 28V
ARMATURE VOLTAGE 14V
NO LOAD CURRENT 139 AMP
FULL LOAD CURRENT 1.75 AMP
TORQUE (IN-OZ)
0
10
20
30
40
0
1000
2000
3000
4000
5000
6000
SPEED (R.P.M.)

Figure 4.3.1

B1	BURGESS TYPE W30BP DRY BATTERY
RY1, RY2	STERLING TYPE 2LL44P RELAY
RY3, RY4 RY5, RY6	ALLIED CONTROL TYPE BGHX-1 RELAY
M1, M2	JOHN OSTER TYPE A-14 MOTOR
G1	ELINCO TYPE C-47 PERMANENT-MAGNET GENERATOR
S1	3-SECOND ARMING SWITCH NO.
R54	GENERAL RADIO TYPE 214-A POTENTIOMETER

INTEGRATOR

INDICATOR

PITCH TRIM

ROLL TRIM

ROLL TRIM

PITCH TRIM

JONES P-304-AB

NOTES:
1. 1.05 AMP. ADDITIONAL LOAD ON T301.
2. ALL OTHER RESISTOR TOLERANCES ± 10%.
3. ALL OTHER RESISTOR RATINGS 1 WATT FOR VALUES BELOW 1 MEG.; 1 MEG. AND ABOVE 1/2 WATT.
4. ALL POTENTIOMETERS 1 WATT, COMPOSITION TYPE, SCREWDRIVER ADJUSTMENT, WITH LOCK.
5. (J) PINS ON CONNECTOR AN-3102-22-14P (SEE C-17030).
6. * COMPONENTS RETAINED FROM SWOD (RADAR) MK2 MOD 0 OR MOD 1.
7. (S) SHIELDED WIRE IN CABLE.

T 301
KS 9090

Trim Servo Wiring

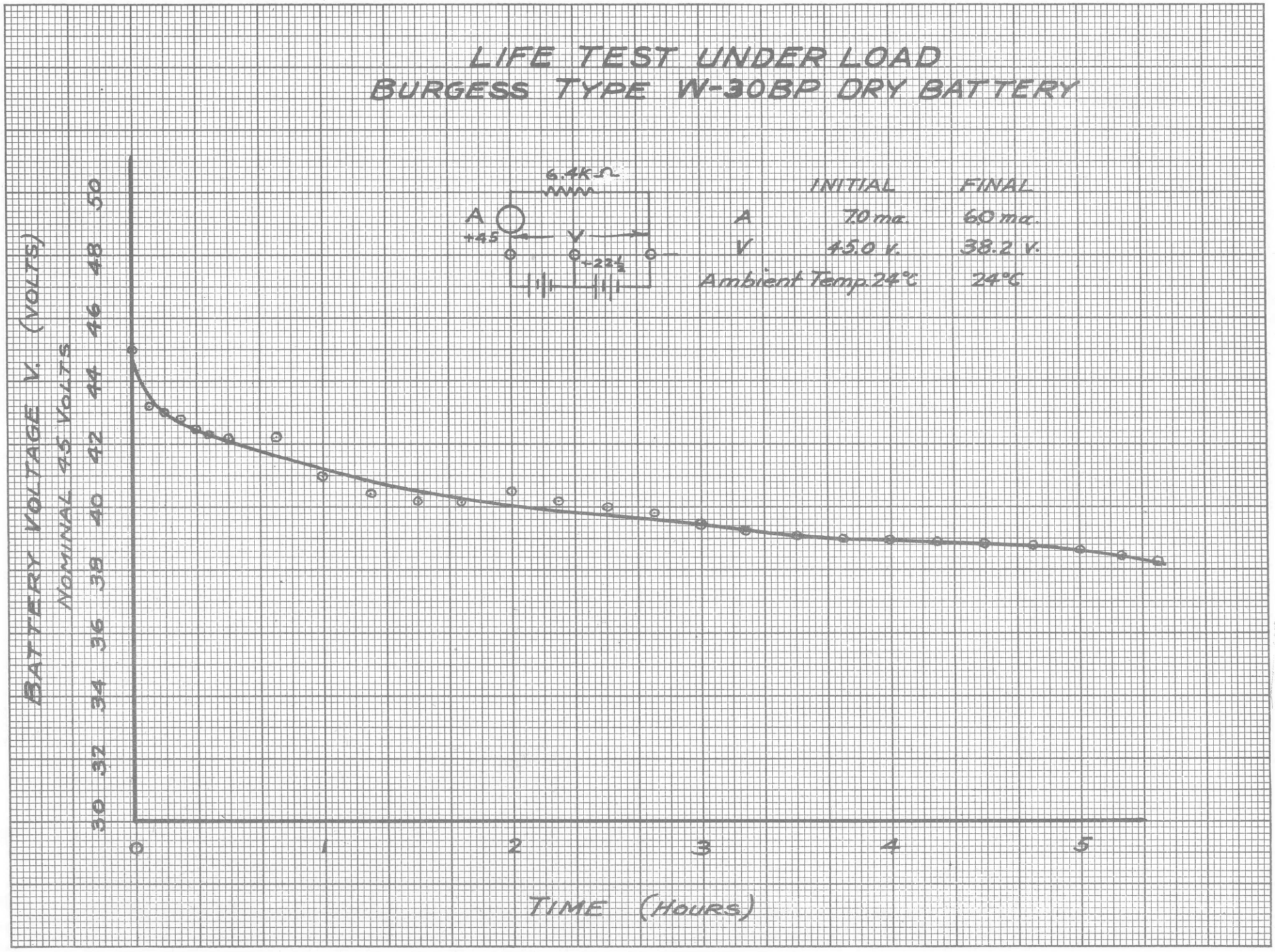

LIFE TEST UNDER LOAD
BURGESS TYPE W-30BP DRY BATTERY
6.4K Ω
A
+45
V
-22½
INITIAL FINAL
A 70 ma. 60 ma.
V 45.0 v. 38.2 v.
Ambient Temp. 24°C 24°C
BATTERY VOLTAGE V. (VOLTS)
NOMINAL 45 VOLTS
30 32 34 36 38 40 42 44 46 48 50
0 1 2 3 4 5
TIME (HOURS)

Figure 4.3.4

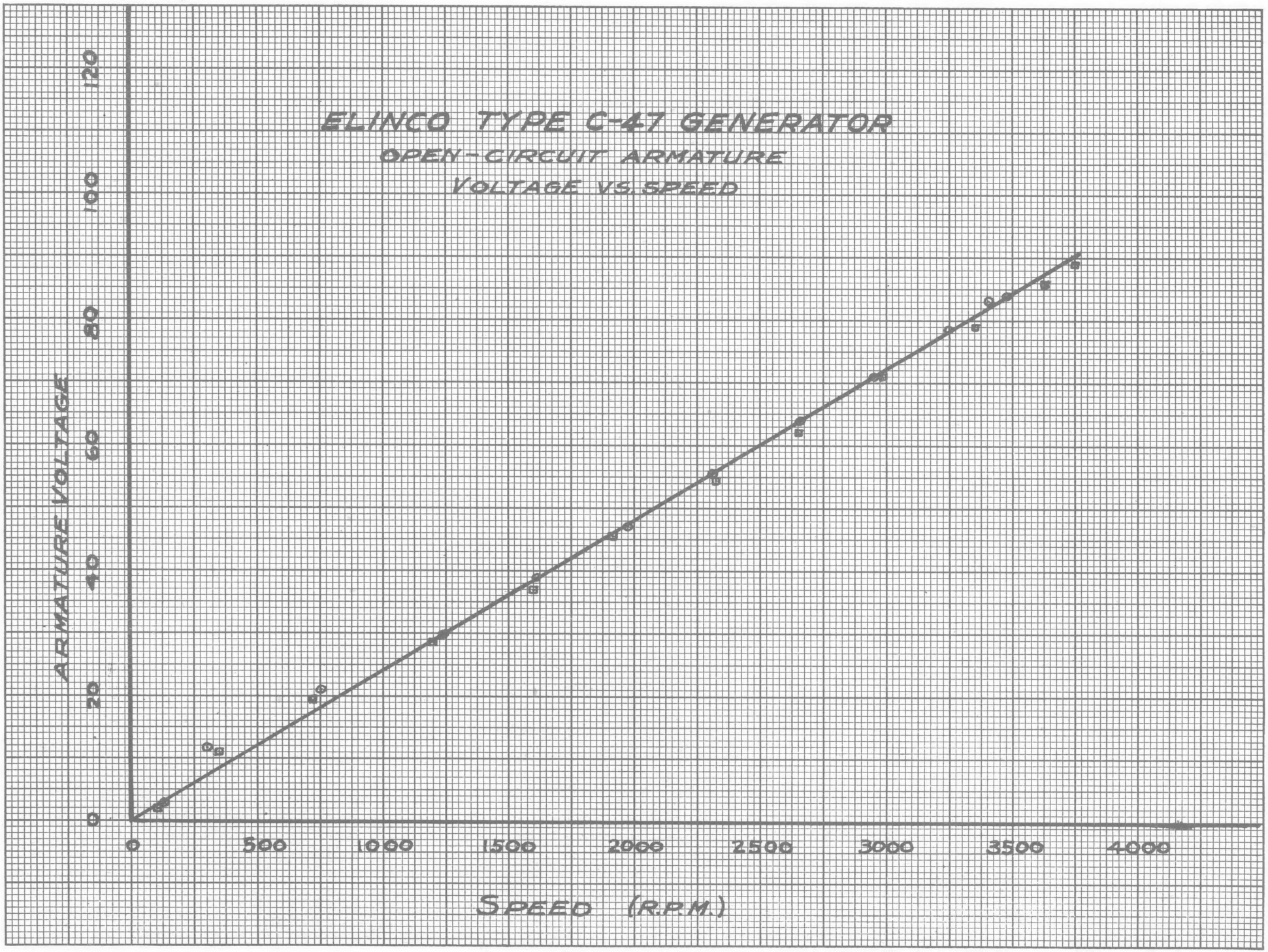
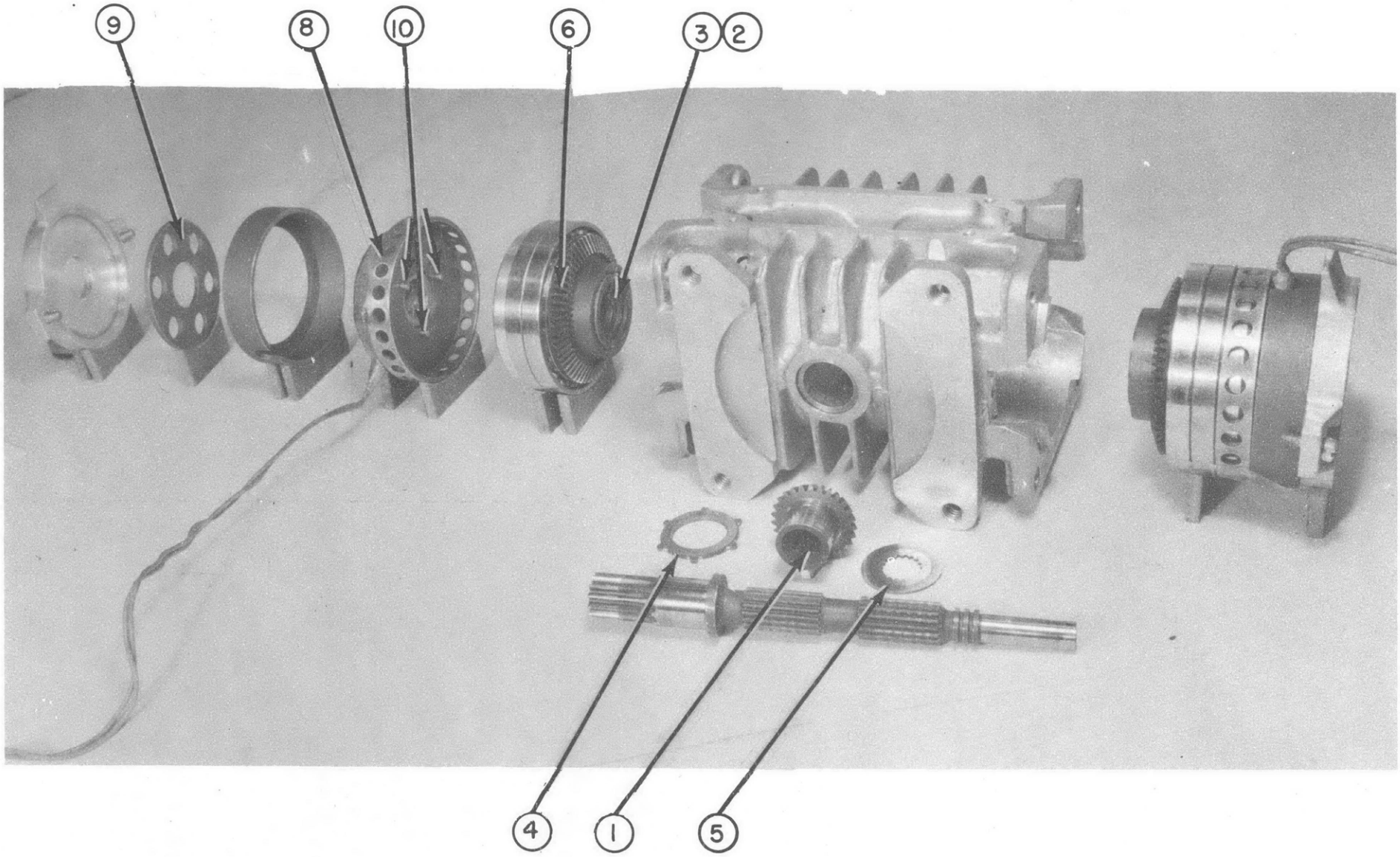
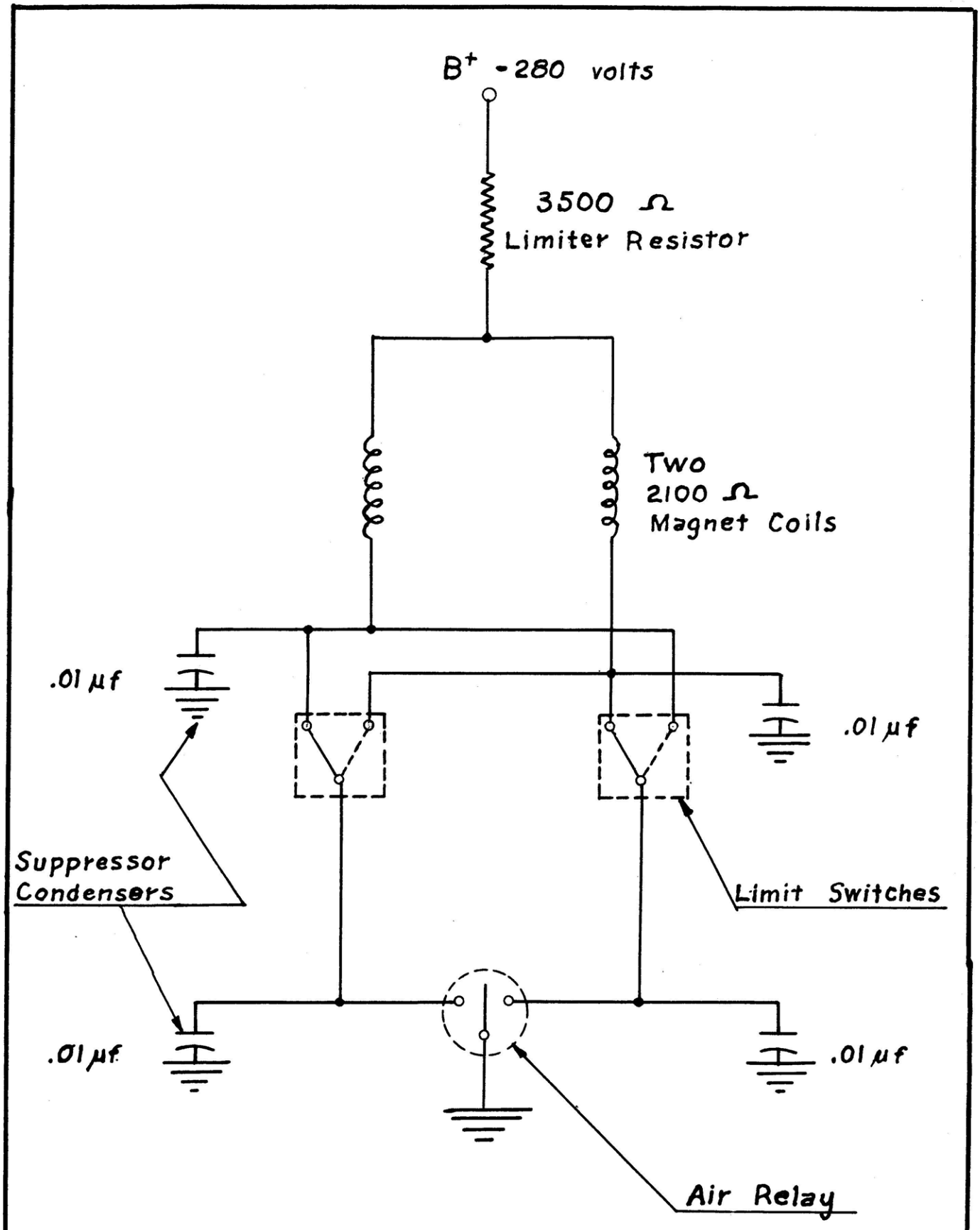

ELINCO TYPE C-47 GENERATOR
OPEN-CIRCUIT ARMATURE
VOLTAGE VS. SPEED
ARMATURE VOLTAGE
0
20
40
60
80
100
120
0
500
1000
1500
2000
2500
3000
3500
4000
SPEED (R.P.M.)

Figure 4.3.5



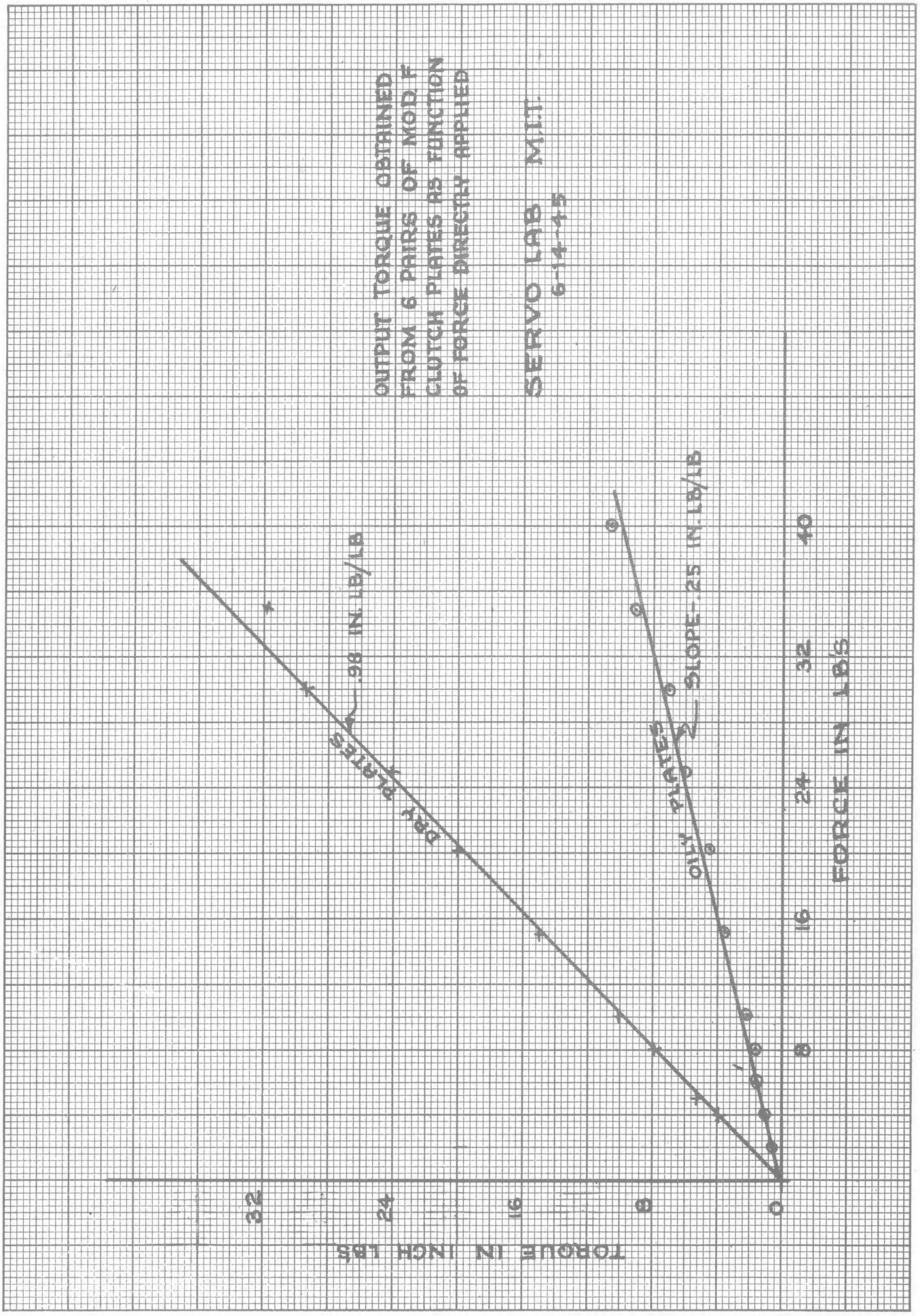
Exploded View, Mod F Clutch

Figure 4.4.1



ACTUATING CIRCUIT
ELEVON SERVO

Figure 4.4.3


OUTPUT TORQUE OBTAINED
FROM 6 PAIRS OF MOD F
CLUTCH PLATES AS FUNCTION
OF FORCE DIRECTLY APPLIED
SERVO LAB M.I.T.
6-14-45
TORQUE IN INCH LBS
FORCE IN LBS
0
8
16
24
32
40
DRY PLATES
SLOPE-.98 IN. LB/LB
OIL'D PLATES
SLOPE-.25 IN. LB/LB

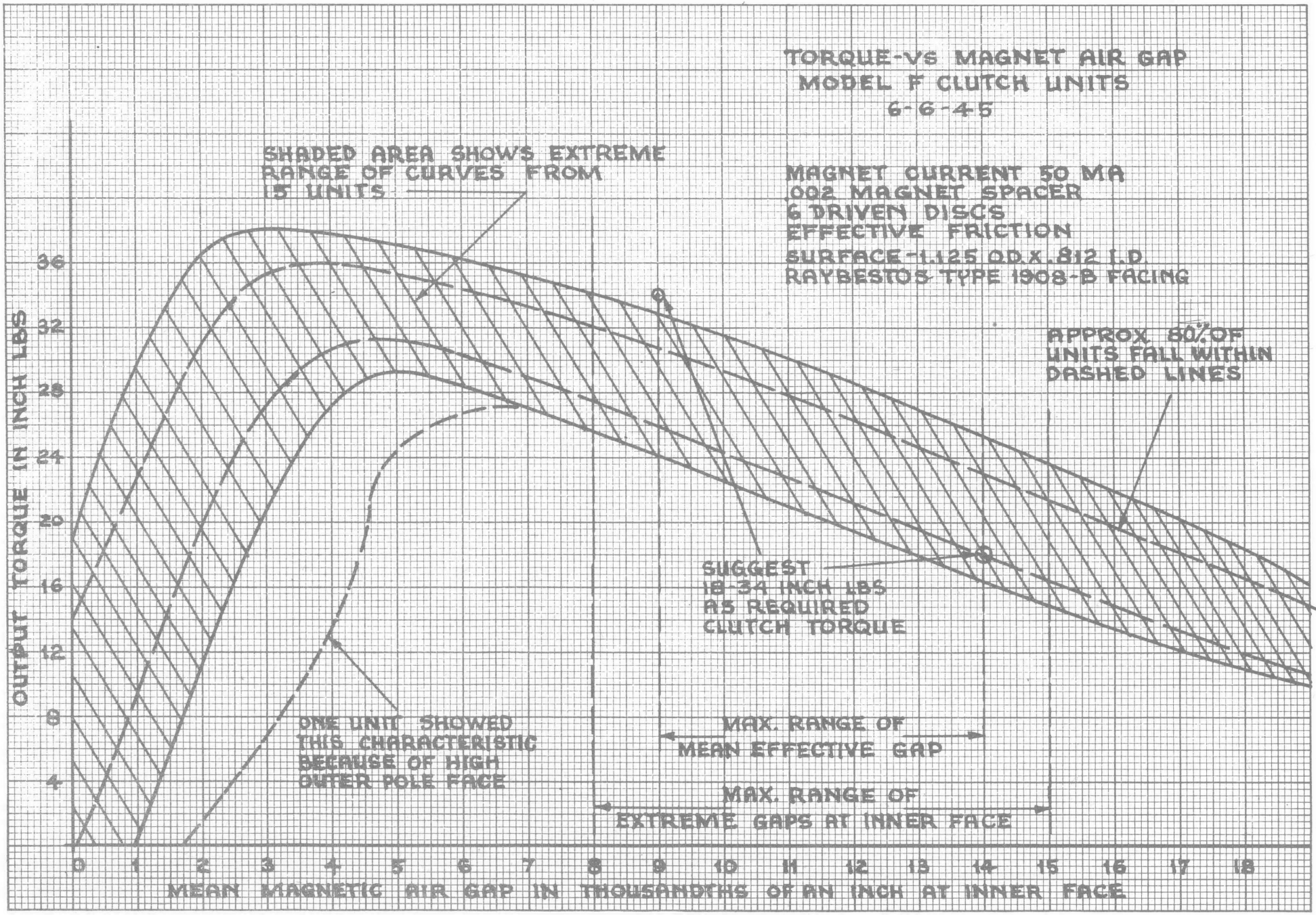

TORQUE-VS MAGNET AIR GAP
MODEL F CLUTCH UNITS
6-6-45
SHADED AREA SHOWS EXTREME RANGE OF CURVES FROM 15 UNITS
MAGNET CURRENT 50 MA
.002 MAGNET SPACER
6 DRIVEN DISCS
EFFECTIVE FRICTION SURFACE-1.125 O.D. X .812 I.D.
RAYBESTOS TYPE 1908-B FACING
APPROX 80% OF UNITS FALL WITHIN DASHED LINES
SUGGEST 18-34 INCH LBS AS REQUIRED CLUTCH TORQUE
ONE UNIT SHOWED THIS CHARACTERISTIC BECAUSE OF HIGH OUTER POLE FACE
MAX. RANGE OF MEAN EFFECTIVE GAP
MAX. RANGE OF EXTREME GAPS AT INNER FACE
OUTPUT TORQUE IN INCH LBS
4
8
12
16
20
24
28
32
36
0 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18
MEAN MAGNETIC AIR GAP IN THOUSANDTHS OF AN INCH AT INNER FACE

Figure 4.4.4

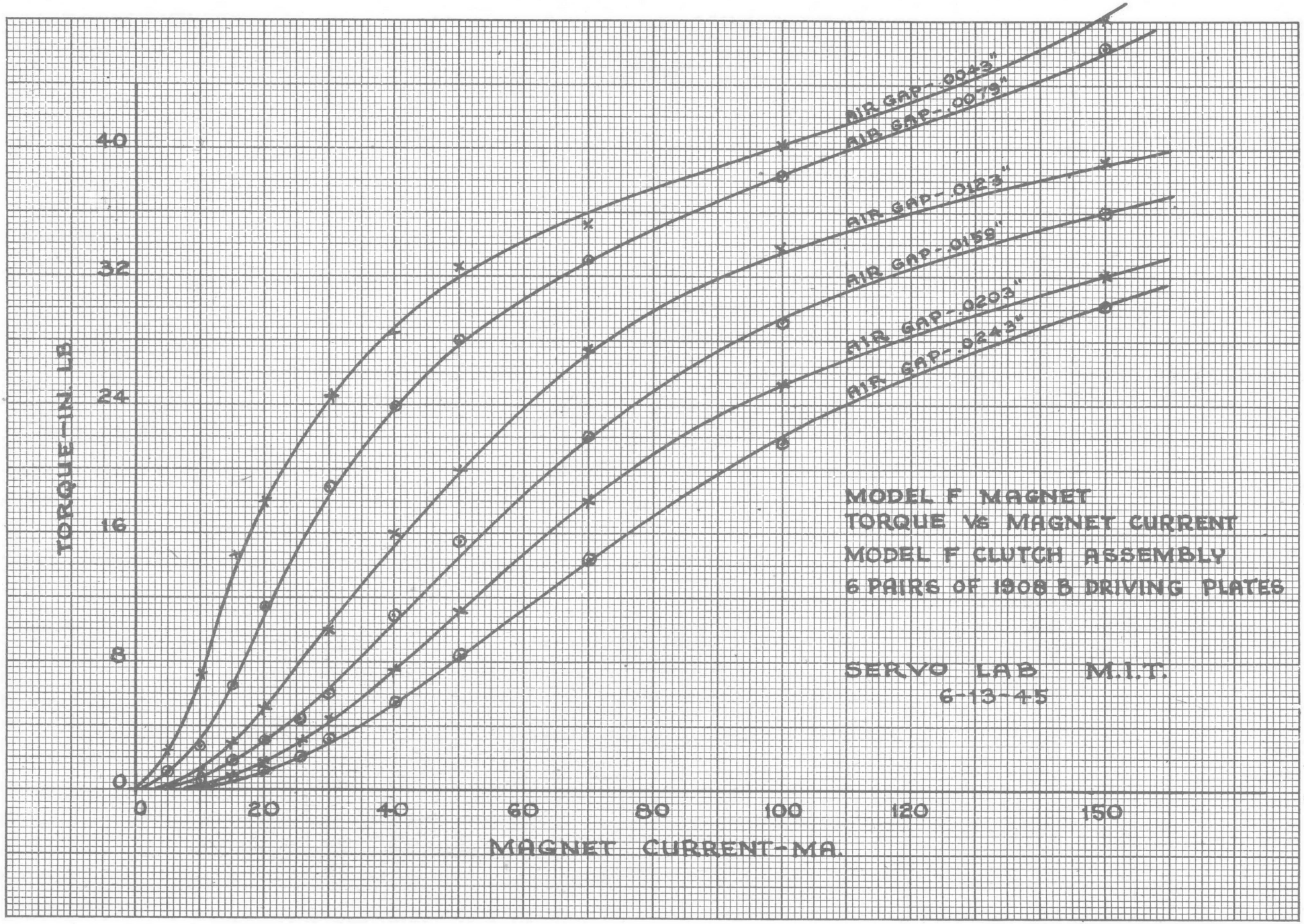

AIR GAP-.0043"
AIR GAP-.0079"
AIR GAP-.0123"
AIR GAP-.0159"
AIR GAP-.0203"
AIR GAP-.0243"
MODEL F MAGNET
TORQUE VS MAGNET CURRENT
MODEL F CLUTCH ASSEMBLY
6 PAIRS OF 1908 B DRIVING PLATES
SERVO LAB M.I.T.
6-13-45
TORQUE-IN. LB.
MAGNET CURRENT-MA.
0
8
16
24
32
40
0
20
40
60
80
100
120
150

Figure 4.4.5

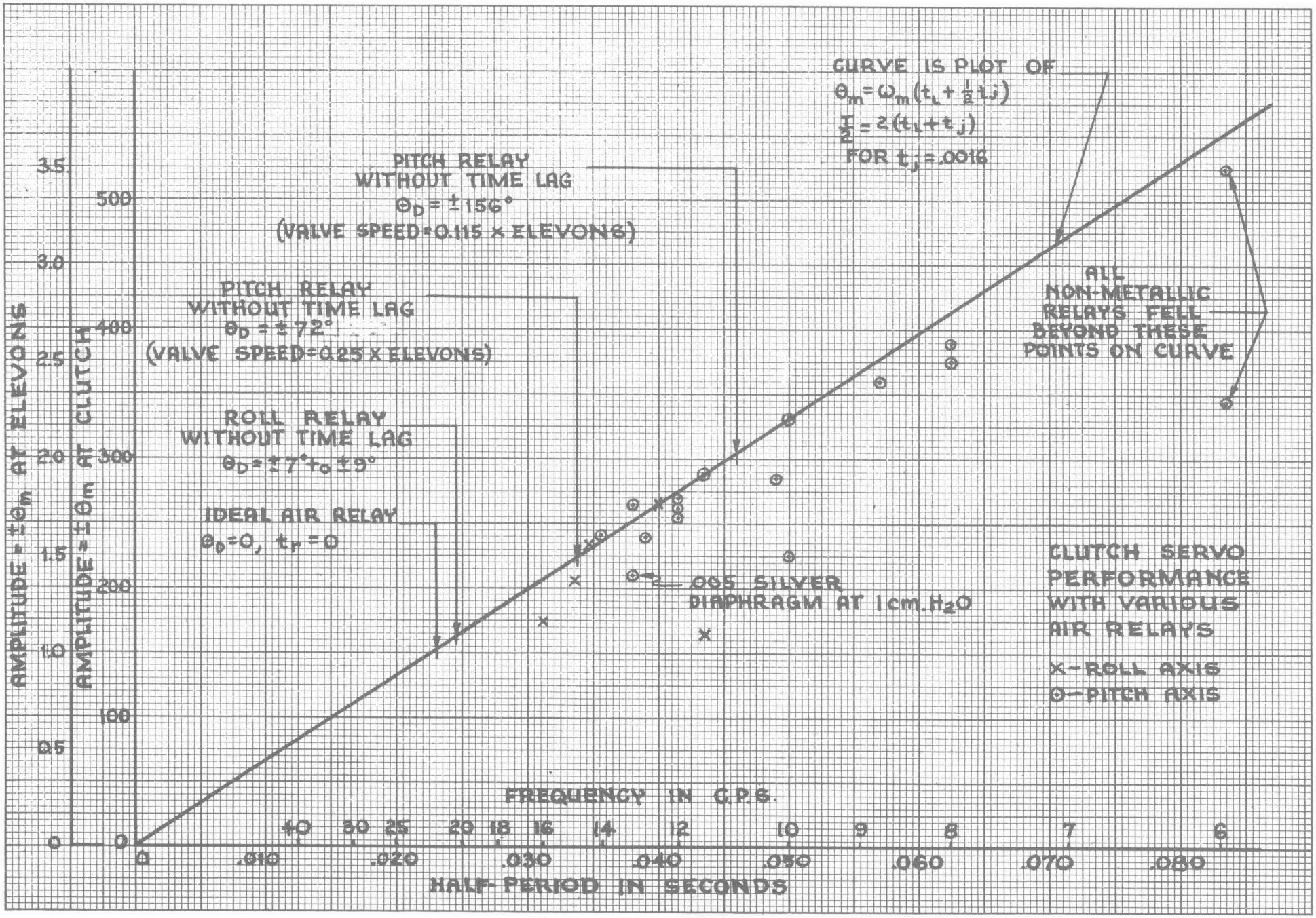

CURVE IS PLOT OF
$\theta_m = \omega_m (t_L + \frac{1}{2} t_j)$
$\frac{T}{2} = 2(t_L + t_j)$
FOR $t_j = .0016$
PITCH RELAY
WITHOUT TIME LAG
$\theta_D = \pm 15°$
(VALVE SPEED = 0.115 × ELEVONS)
PITCH RELAY
WITHOUT TIME LAG
$\theta_D = \pm 7.2°$
(VALVE SPEED = 0.25 × ELEVONS)
ROLL RELAY
WITHOUT TIME LAG
$\theta_D = \pm 7° + \theta_0 \pm 9°$
IDEAL AIR RELAY
$\theta_D = \theta_j$, $t_r = 0$
ALL
NON-METALLIC
RELAYS FELL
BEYOND THESE
POINTS ON CURVE
.005 SILVER
DIAPHRAGM AT 1cm. H_2O
CLUTCH SERVO
PERFORMANCE
WITH VARIOUS
AIR RELAYS
×—ROLL AXIS
⊙—PITCH AXIS
AMPLITUDE = $\pm\theta_m$ AT ELEVONS
AMPLITUDE = $\pm\theta_E$ AT CLUTCH
FREQUENCY IN C.P.S.
HALF-PERIOD IN SECONDS
3.5
3.0
2.5
2.0
1.5
1.0
.5
0
500
400
300
200
100
0
40
30
25
20
18
16
14
12
10
9
8
7
6
0
.010
.020
.030
.040
.050
.060
.070
.080

Figure 4.4.6

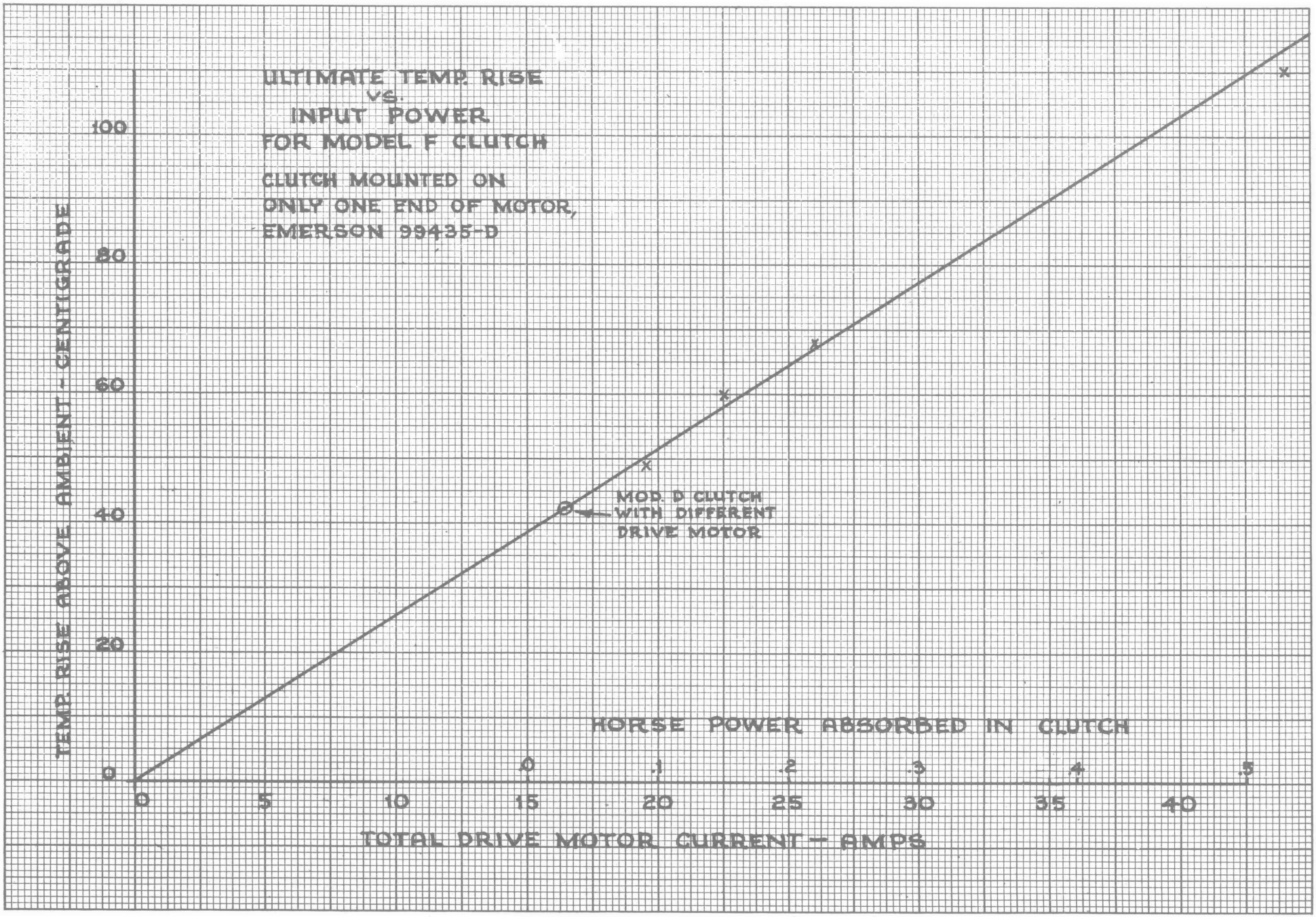

ULTIMATE TEMP. RISE
VS.
INPUT POWER
FOR MODEL F CLUTCH
CLUTCH MOUNTED ON
ONLY ONE END OF MOTOR,
EMERSON 99435-D
MOD. D CLUTCH
WITH DIFFERENT
DRIVE MOTOR
TEMP. RISE ABOVE AMBIENT - CENTIGRADE
0
20
40
60
80
100
HORSE POWER ABSORBED IN CLUTCH
0
.1
.2
.3
.4
.5
0
5
10
15
20
25
30
35
40
TOTAL DRIVE MOTOR CURRENT — AMPS

Figure 4.4.7

Figure 4.4.8

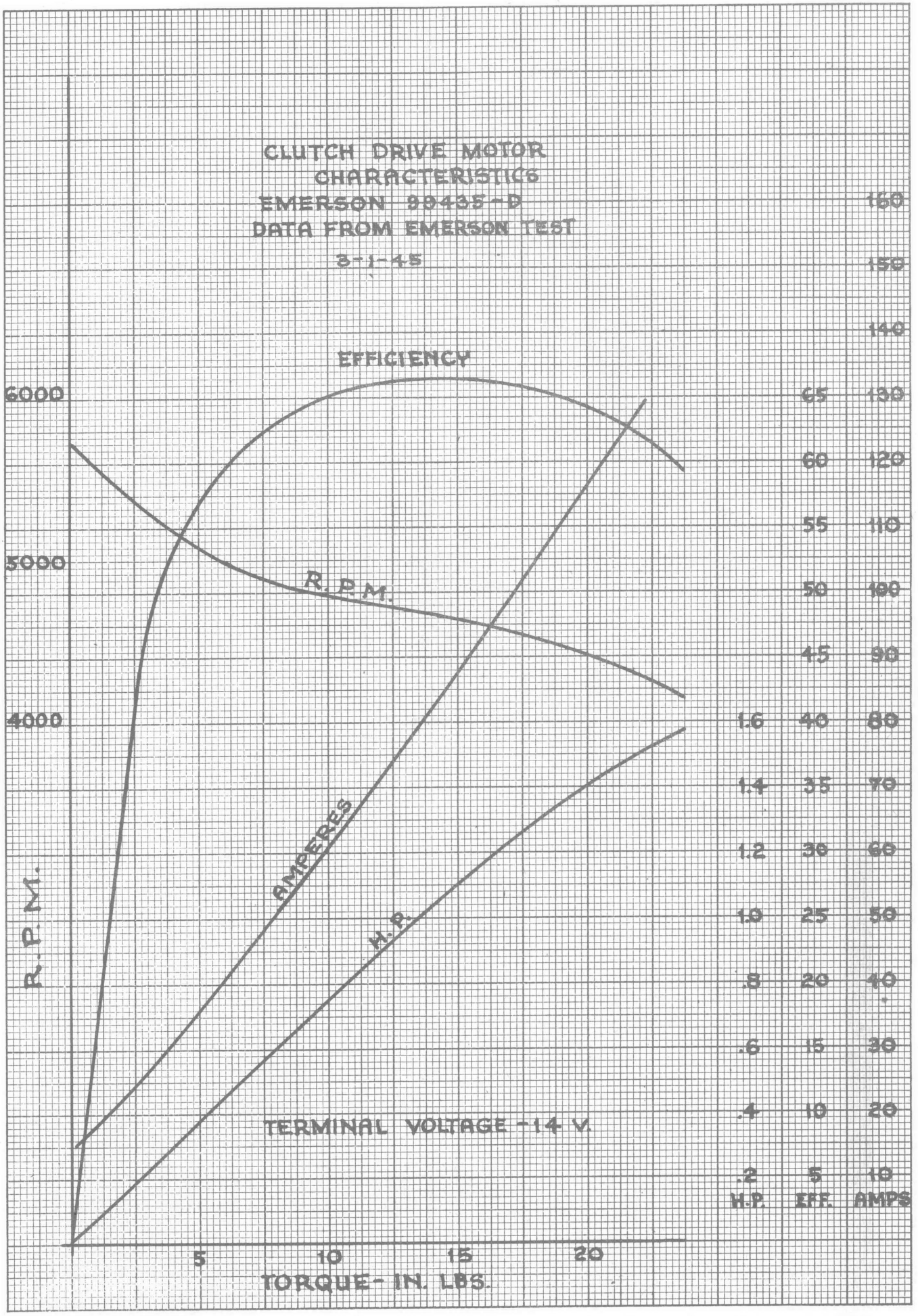
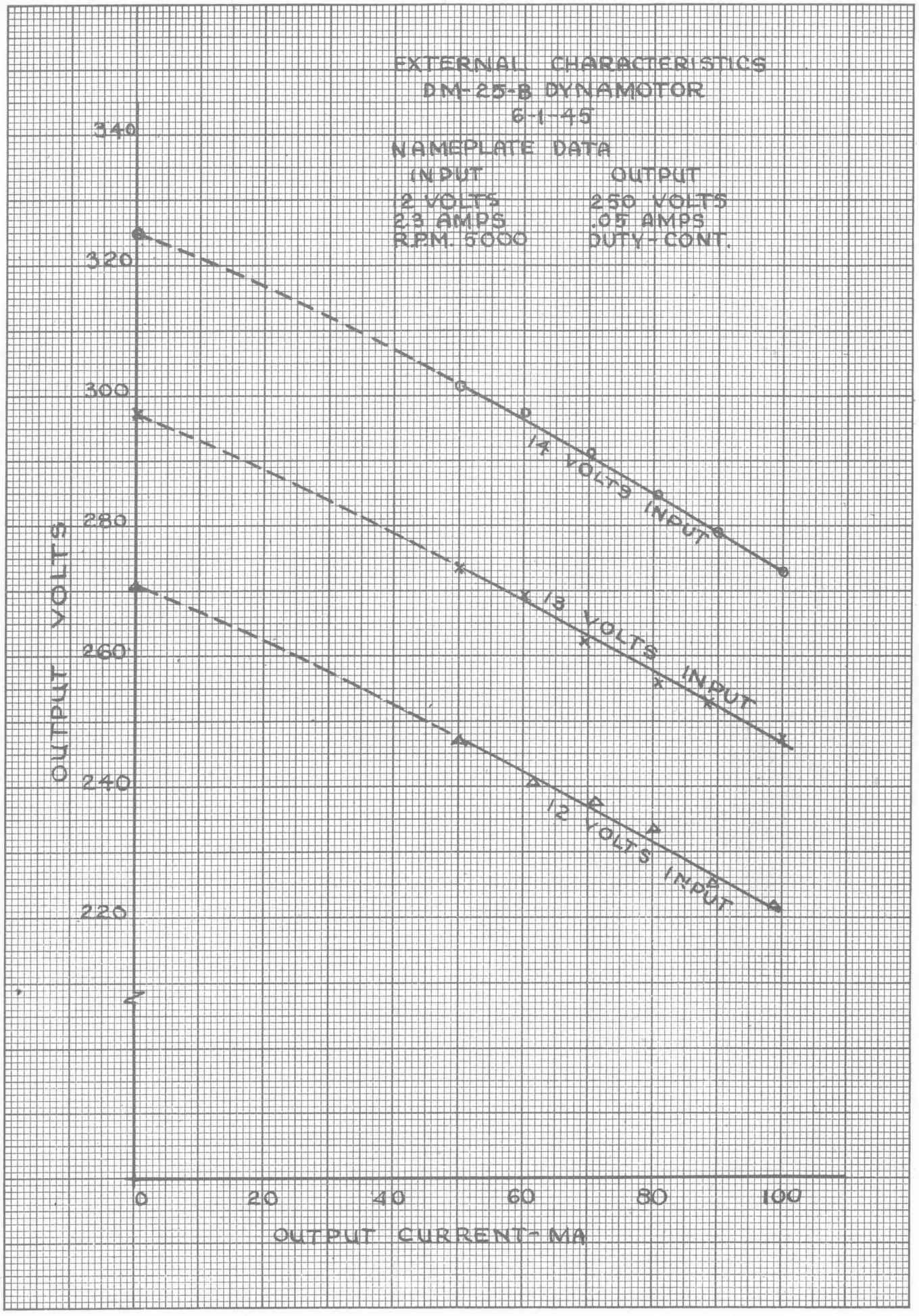
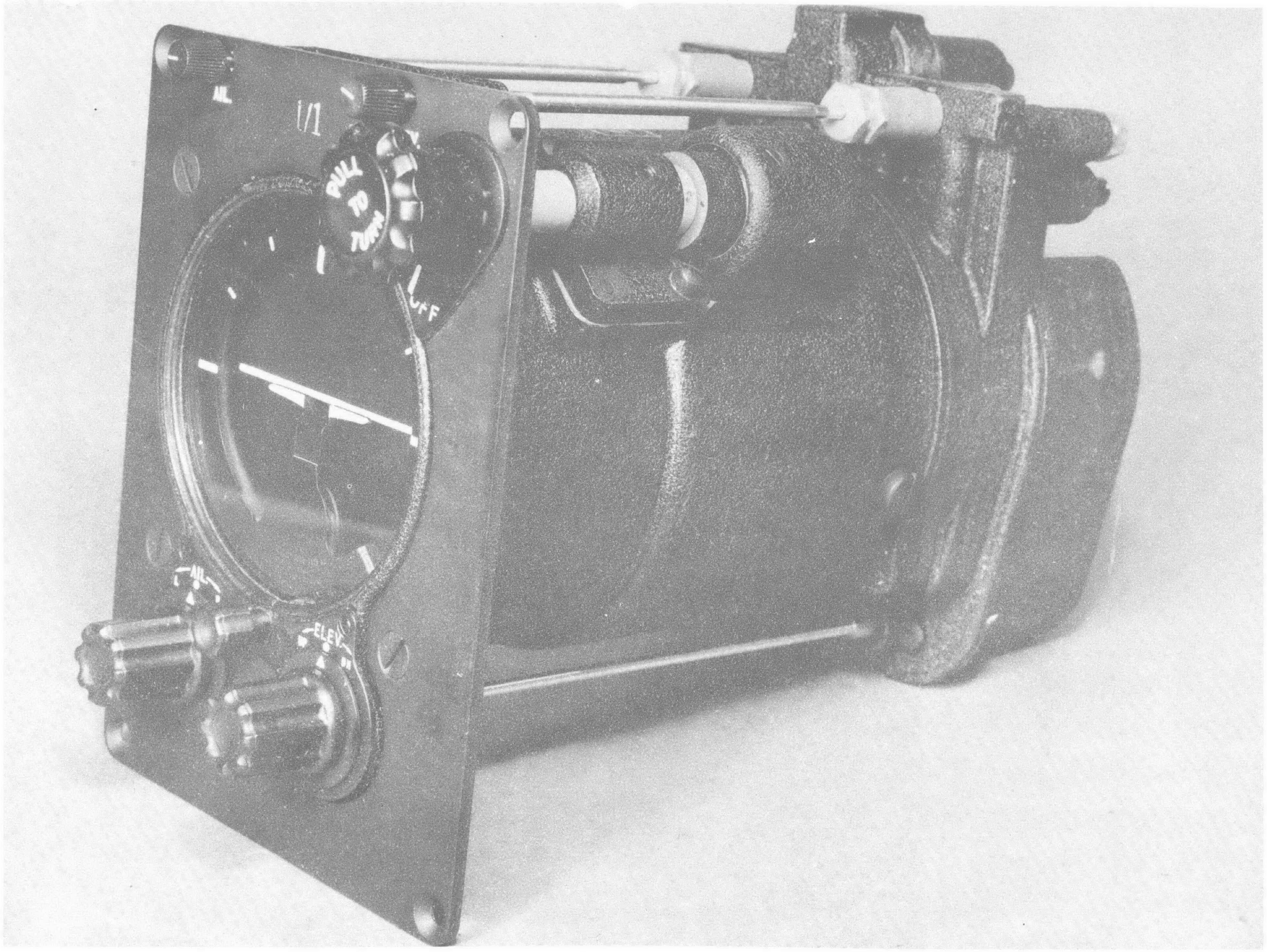

CLUTCH DRIVE MOTOR
CHARACTERISTICS
EMERSON 99435-D
DATA FROM EMERSON TEST
8-1-45
EFFICIENCY
R.P.M.
AMPERES
H.P.
TERMINAL VOLTAGE - 14 V.
R.P.M.
6000
5000
4000
5
10
15
20
TORQUE - IN. LBS.
65
60
55
50
45
1.6
1.4
1.2
1.0
.8
.6
.4
.2
H.P.
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
EFF.
AMPS

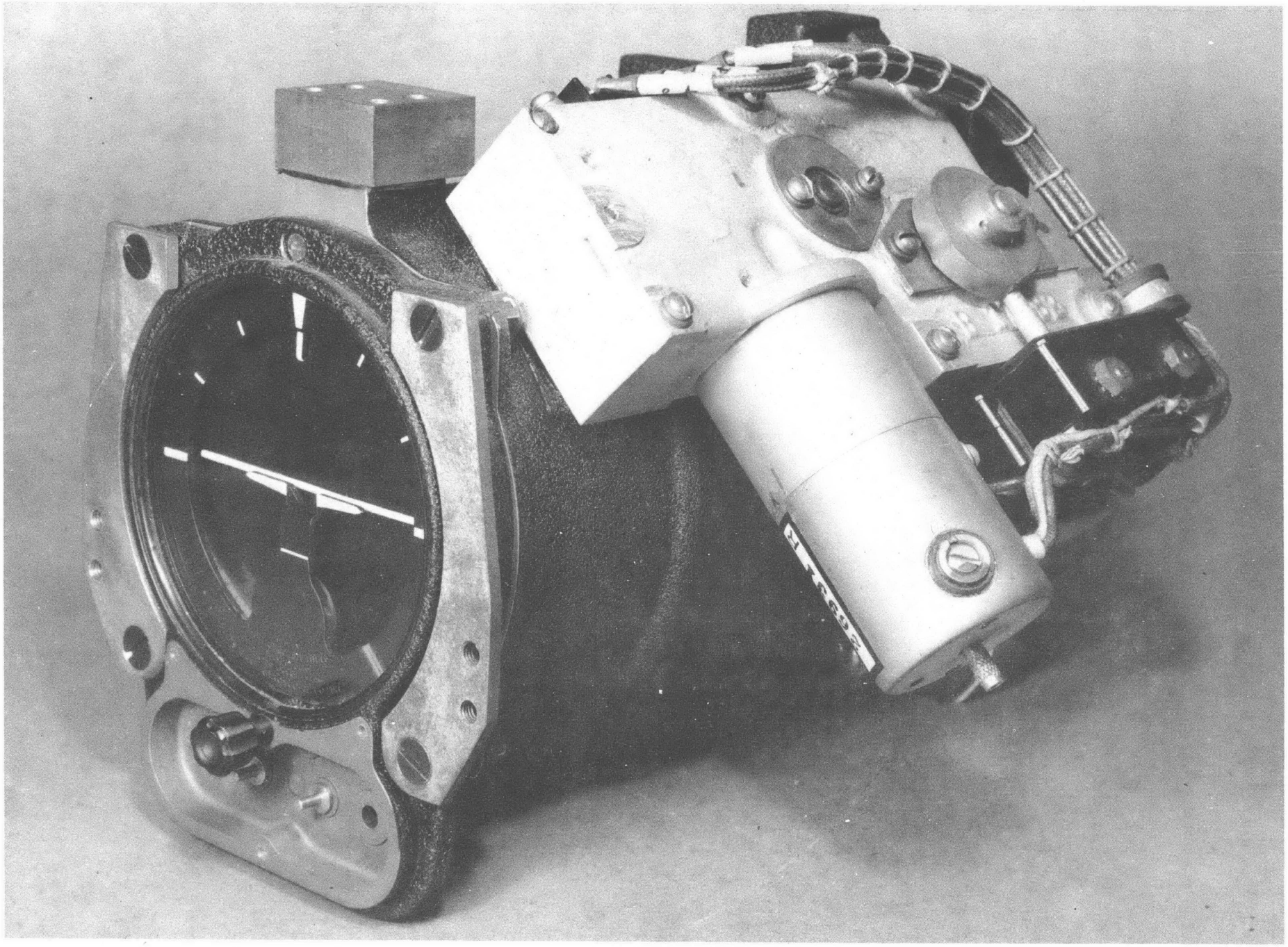
Figure 4.4.9


EXTERNAL CHARACTERISTICS
DM-25-B DYNAMOTOR
6-1-45
NAMEPLATE DATA
INPUT
12 VOLTS
23 AMPS
R.P.M. 6000
OUTPUT
250 VOLTS
.05 AMPS
DUTY-CONT.
340
320
300
280
260
240
220
OUTPUT VOLTS
14 VOLTS INPUT
13 VOLTS INPUT
12 VOLTS INPUT
0
20
40
60
80
100
OUTPUT CURRENT-MA



Sperry Mk IV Bank and Climb Gyro Front View, Before Modification

Figure 4.5.1



Sperry Mk IV Bank and Climb Gyro Front View, After Modification

Figure 4.5.2

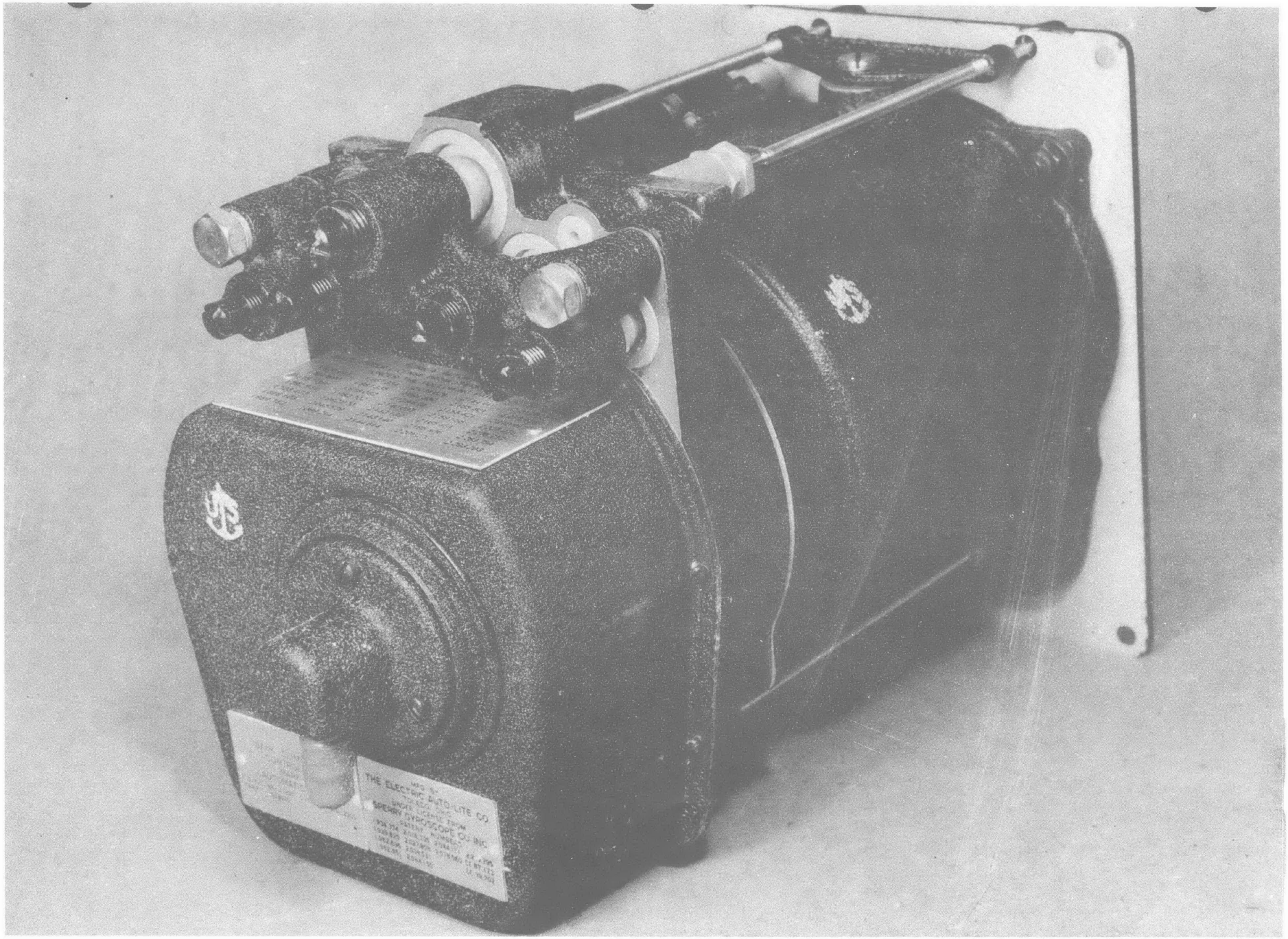
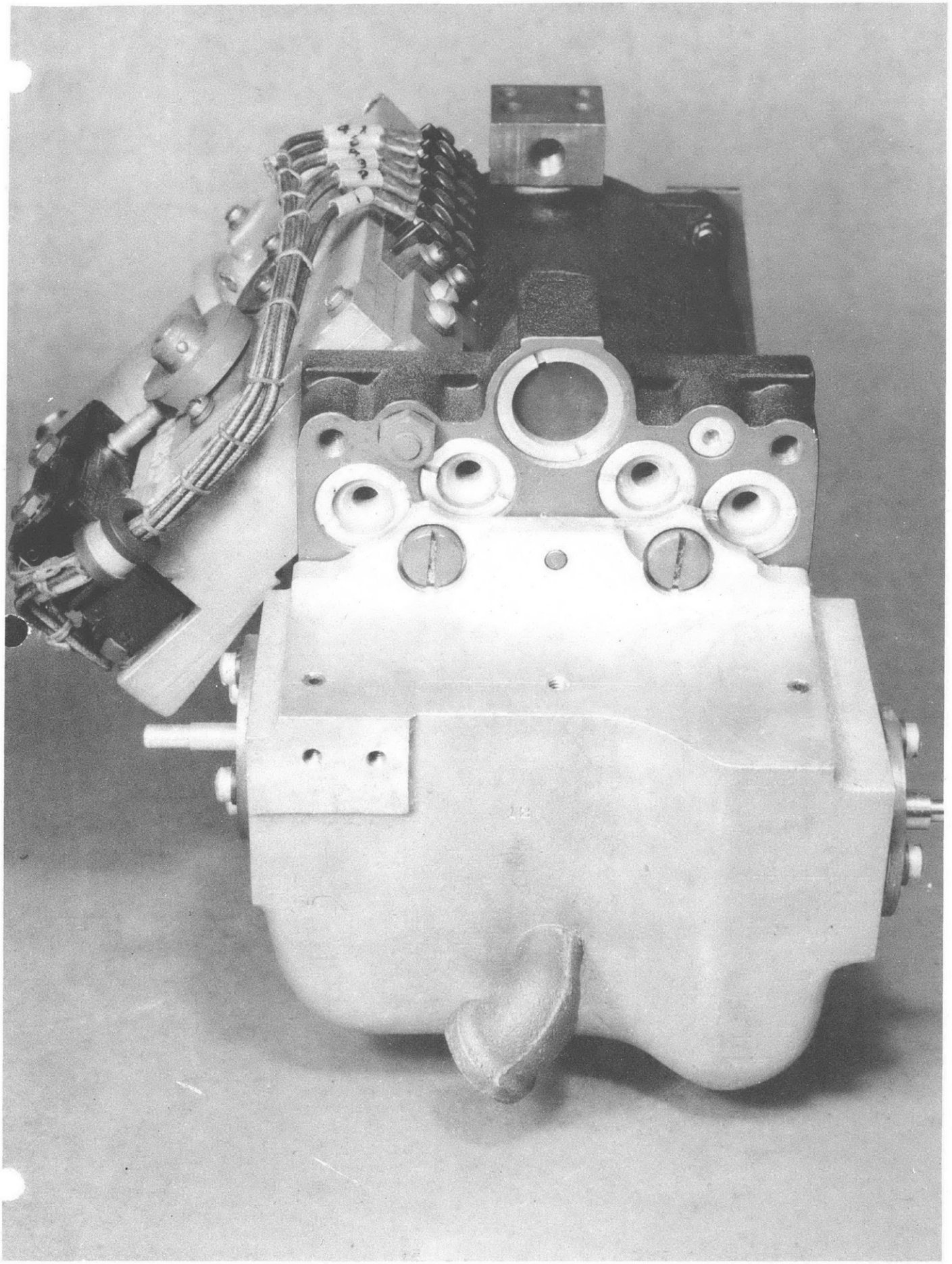


Figure 4.5.3

Sperry Mk IV Bank and Climb Gyro Rear View, Before Modification

Figure 4.5.4



Sperry Mk IV Bank and Climb Gyro
Front View, After Modification

5.0 TEST EQUIPMENT

Since the control equipment in a glider bomb has only one opportunity to perform properly, it is of the utmost necessity that the system and its components be thoroughly ground and air tested before the actual drop is made to reduce the possibility of failure. For this reason the testing procedure carried on in connection with the SWOD (Control System) Mk 18 Mod 0 is far more thorough than is customary with similar equipment intended for operation where a failure can be corrected.

5.1 Testing Procedure

The testing was carried out in the following stages:

1. All major manufactured components such as motors, dynamotors, tubes, air relays, etc. were subjected to an acceptance test. All mechanical parts were inspected for conformance to the drawings.
2. Gear trains, limit stops, etc. were adjusted during the assembly operation.
3. The major subassemblies, i.e., the pitch and roll trim servos, the gyro, and the clutches, were given rigid performance tests.
4. The complete control system was given a final laboratory test which simulated as closely as possible the actual operation.
5. At the flight testing base the control system was tested with the accompanying radar unit after installation in the glider. The radar unit had been previously tested by the manufacturer and by the M.I.T. Field Experiment Station at the Bureau of Standards.
6. The glider was given a final test to make certain that it would respond to radar signals and would operate from its own power supply after installation on the airplane.
7. Before finally dropping the glider, dummy runs were made on the target. Operation of the control system was ascertained by visual observation of the elevons.The radar was checked by observation of the indicator oscilloscope.

5.2 Types of Test Equipment

The test equipment required for the test program may be divided into six classes, as listed below:

1. Equipment used in testing the components.

   Air relay tester
   Sensitive relay tester

2. Equipment used in testing the subassemblies of the control system.

   Clutch performance test stand
   Trim servo test amplifier

3. Equipment used in testing the complete control system in the laboratory.

   Volter-indicator
   Control system test box
   Loading springs and angle indicators

4. Equipment built to simulate the control system for tests on the radar unit.

   Dummy load unit
   Trim amplifier test unit
   Volter-indicator (previously listed)

5. Equipment used in the ground field testing of the complete glider.

   Control system test box (previously listed)
   Volter-indicator (previously listed)

6. Control and observation equipment installed in the dropping airplane.

   Bomb monitor
   Indicator
   Pilot's indicator

5.3 Description and Function of Test Equipment

5.3.1 Air Relay Tester

In order to make certain that the air relay diaphragms had the proper stiffness, curves of displacement vs. differential pressure were made on all diaphragms to be used in the control system. A special fixture was made which supported the diaphragm as it would be supported in the air relay housing. The fixture carried two micrometer heads in place of the contact points. Air pressure was measured with a water manometer, and contact between the micrometer head and the diaphragm was determined with a buzzer.

5.3.2 Sensitive Relay Tester

This unit was built up to allow rapid measurement of the pick-up and drop-out characteristics of the Allied Control type BGHX-1 sensitive relays. It combines in one unit a variable-current supply, a meter to indicate coil current, a means of indicating continuity through the relay contacts, and provision for making high-potential tests.

5.3.3 Clutch Performance Test Stand

Since the design and construction of the clutch units was completed well in advance of the complete control system, an elaborate test stand was constructed to simulate as nearly as possible the servo loop in which the clutch acts as a driving member. Figures 5.3.1 and 5.3.2 are photographs of this unit. The roll side of this system is shown in Figure 5.3.1. It consists of a gear train driving a torsion rod which supplies the load, and a pointer which indicates the output angle. A wire and pulley feedback is provided to the roll air valve of a modified Sperry Mk 4 Bank and Climb Control Unit which can be seen above and behind the drive motor. The air relays are connected to the output of the air valves.

Figure 5.3.2 shows the pitch side of the system. This is identical to the roll side except that the gear ratios are different. A sine-function director (partially visible in back of the clutch) is provided in order to stroke the servo over the entire output range.

The test stand was used primarily to determine the effect of the air data system and gear train backlash on the servo performance. No attempt was made to test all clutches on this system. Instead, the performance to be expected was determined by testing a few clutch units on the stand. This performance was then duplicated with a simpler unit which consisted merely of a constant speed a-c driving motor coupled to the clutch unit under test and an adjustable contactor coupled to the output side of the clutch, and controlling the clutch current.

One clutch unit was tested to failure by operating it under load with the sine-wave director supplying the input. Failure eventually occurred from erosion of the friction surface of the clutch plates. Before this happened, however, the clutches were hot enough to smoke the insulation of the coils.

5.3.4 Trim Servo Test Amplifier

In order to test roll and pitch trim servo subassemblies, a test amplifier was built which includes the trim amplifier circuits of Figure 4.3.3, a filament and plate supply, a battery to supply the potentiometer, the sensitive relays, and switches to select the operation desired. In order to avoid duplication of controls, the unit was designed to be used with the control system test box (see below), AN connectors being provided for connection with this unit. A plug is provided for connection to the volter-indicator (see below) which provides signals similar to those supplied from the radar. Figure 5.3.3 is a photograph of the test amplifier.

5.3.5 Volter-Indicator

The volter-indicator provides dummy radar signals for tests on the trim servos and amplifiers. The unit is shown in the photograph of Figure 5.3.4. Referring to the photograph, the left-hand side of the panel contains the controls for the volter. The selector switch (upper left) selects the up-down or right-left channel and disconnects the batteries in the "off" position. The zero-center voltmeter measures the differential voltage across the output terminals. A scale switch provides ranges of 20 or 40 volts. The knob below the meter varies the output voltage over the range of ±40 volts. Tip jacks are provided at the bottom of the panel in order to connect a more accurate voltmeter for calibration purposes. The switch between the jacks reverses the polarity of the voltage at these terminals.

The indicator unit at the right of the panel connects to the indicator circuit in Figure 4.3.3 in place of the cross-pointer meter. When the sensitivity switch is in the "high" position the calibration may be adjusted by means of the potentiometers located below each meter. In the "low" position the sensitivity is fixed. In use, the indicators, together with the cathode followers which feed them, are calibrated by means of the volter. The indicators may then be used to measure the commutator of the radar set in determinations of radar width, etc.

5.3.6 Control System Test Box

The test box is used to provide individual control over the various parts of the control system during laboratory and ground field tests. The unit is shown Figure 5.3.5.

The switches marked "roll", "pitch", and "clutch" control the power to the motors in the corresponding sections of the control system. The switch marked "int.—in—out" controls the relay that short-circuits the integral condensers. Terminals are provided at the upper left-hand corner of the panel for connecting an external 28-volt supply for the caging circuit. The switch marked "ext.—int." switches the caging circuit from external to internal supply.

At the upper right-hand side of the panel are lights which indicate whether the gyro is caged or uncaged. The switch between the lights controls the caging motor.

The zero-center voltmeter is provided to measure the feedback voltages from the roll and pitch trim servos. The switch marked "gen.—pot." provides a change-over from the pitch feedback (generator) to the roll feedback (potentiometer). Separate calibration controls are provided for the two circuits.

The switch marked "run—test" when in the "test" position connects the controls and indicators previously described. In the "run" position the wires from the control system circuits are connected through the test box to the points at which they are normally connected. Thus, the bomb monitor and arming circuits can be tested without the necessity of removing the test box from the circuit.

In order to measure the sensitive relay coil currents, closed-circuit phone jacks are mounted on the side of the test box. Milliammeters are connected as in Figure 5.3.6.

5.3.7 Loading Springs and Angle Indicators

In order to simulate as closely as possible the actual loading on the control system a set of straight leaf loading springs was designed which would supply a load equal to the expected wind load on the elevons. Push rods were connected between the ends of the pantograph and the free ends of the springs. The spring constant was about 140 pounds per inch.

It was necessary to measure the elevon angles in order to determine the amplitude of oscillation, the gain factors of various parts of the system, and the positions at which the limit stops tripped. Angle indicators were designed for this purpose, each consisting of a scale calibrated in degrees of elevon deflection and placed under the end of the pantograph.

5.3.8 Dummy Load Unit

The dummy load unit, shown in the photograph, Figure 5.3.6, was designed to simulate the relay load on the trim amplifiers, to provide a means for measuring relay currents, and to provide a source of feedback voltage. A switch is provided to transfer the source of variable feedback voltage from pitch to roll. In the "normal" position of the switch no feedback voltage is introduced and the feedback circuits are loaded with their normal impedance. Jacks are provided for the milliammeters and for a voltmeter to read feedback voltage.

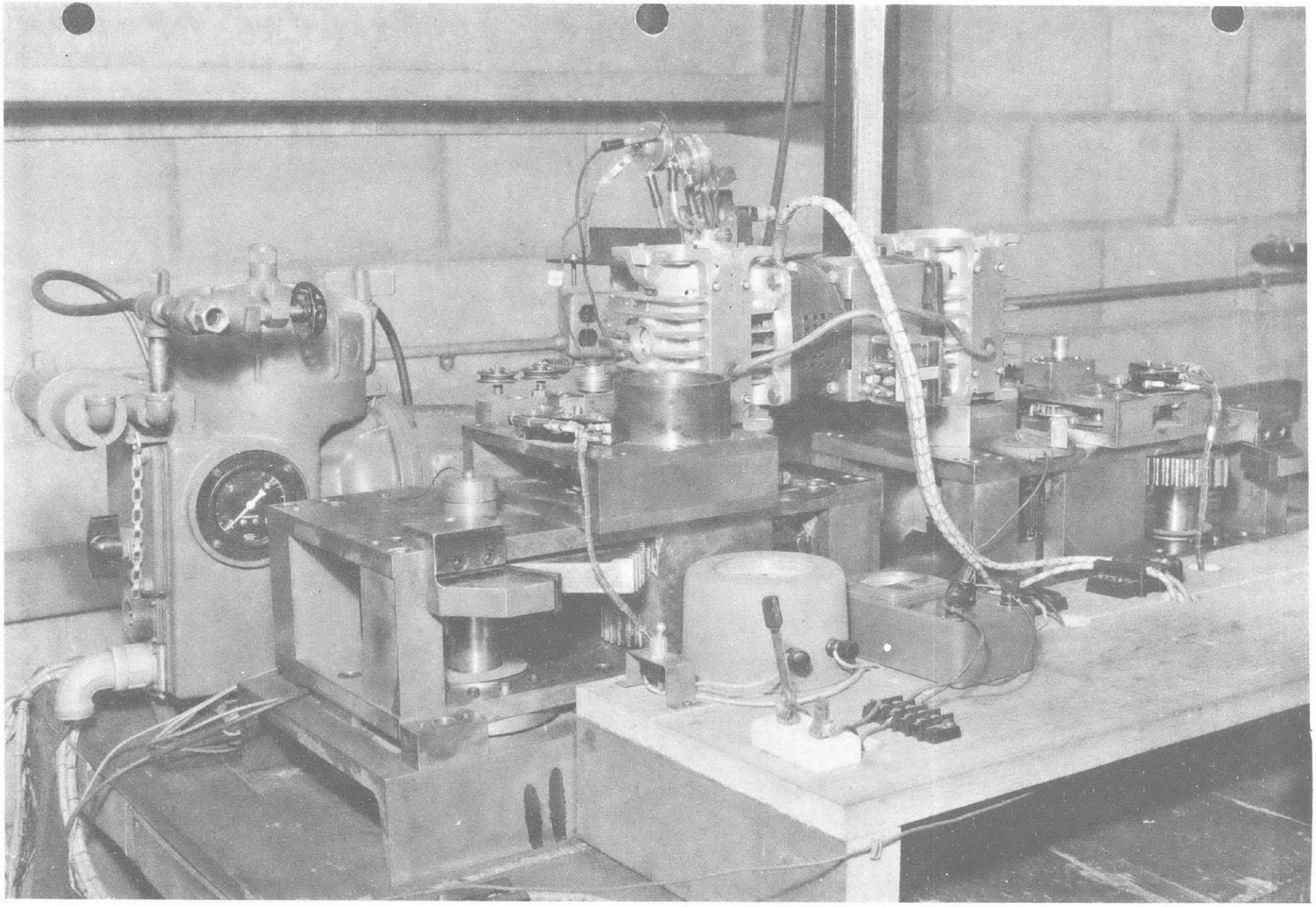
5.3.9 Trim Amplifier Test Unit

The test unit shown in Figure 5.3.7 provides a means of testing the trim amplifier with a standard set of trim servos. The test unit contains all the components shown in

the section at the lower right-hand side of Figure 4.3.3, plus switches for control of the various functions. The use of this unit and the dummy load unit for testing trim amplifiers is covered in a report by F. J. Norvik entitled "Modification of the SWOD (Radar) Mk 2 Mod 0 and Mod 1 for operation of the SWOD (Control System) Mk 18 Mod 0."

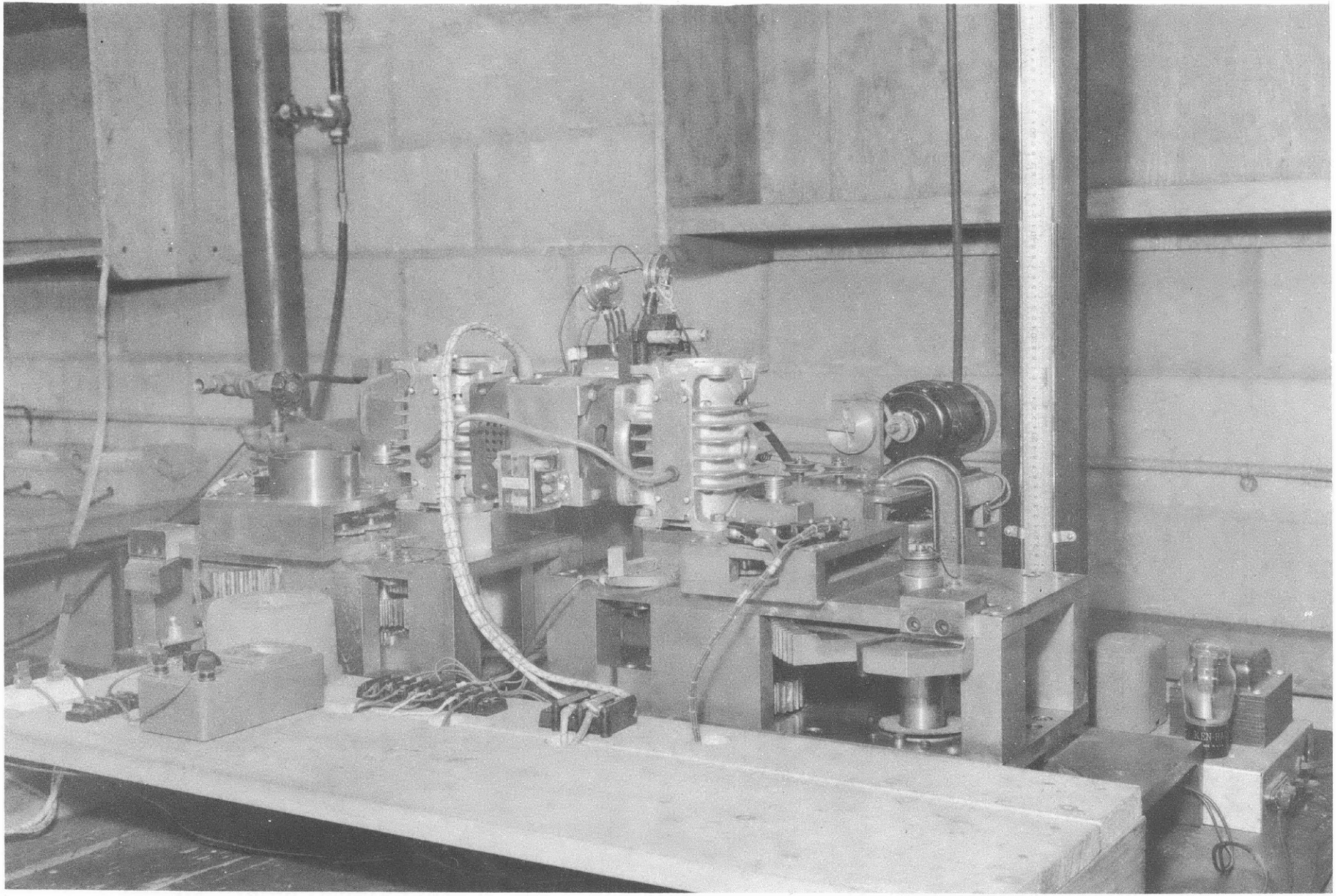
5.3.10 Bomb Monitor, Indicator, and Pilot's Indicator

These units are fully described in the Western Electric Service Handbook, Bulletin No. 21. Minor modifications were made in the bomb monitor to obtain a switch position for uncaging the gyro. The other units were unmodified.



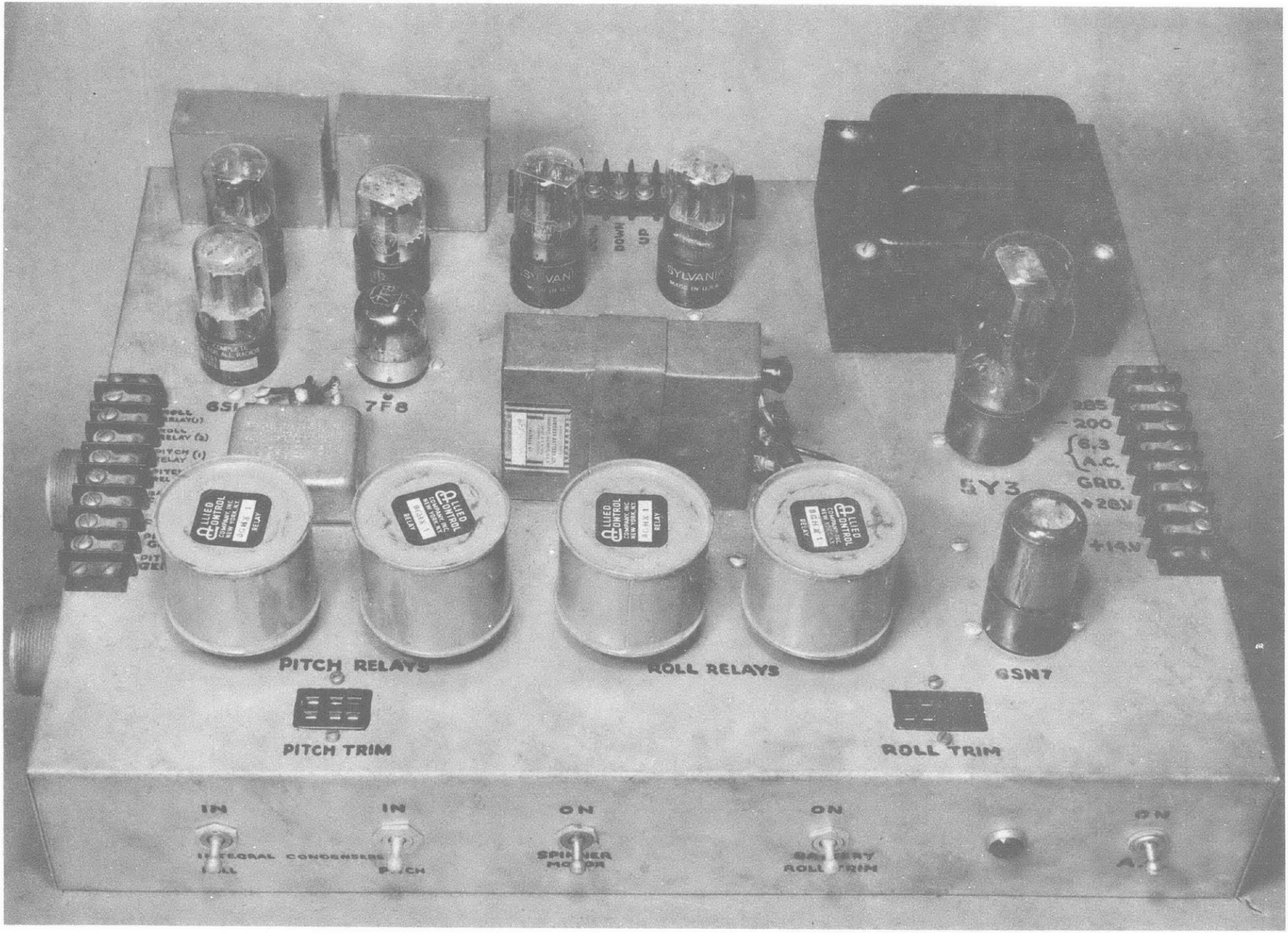
Clutch Performance Test Stand (Roll Side)

Figure 5.3.1



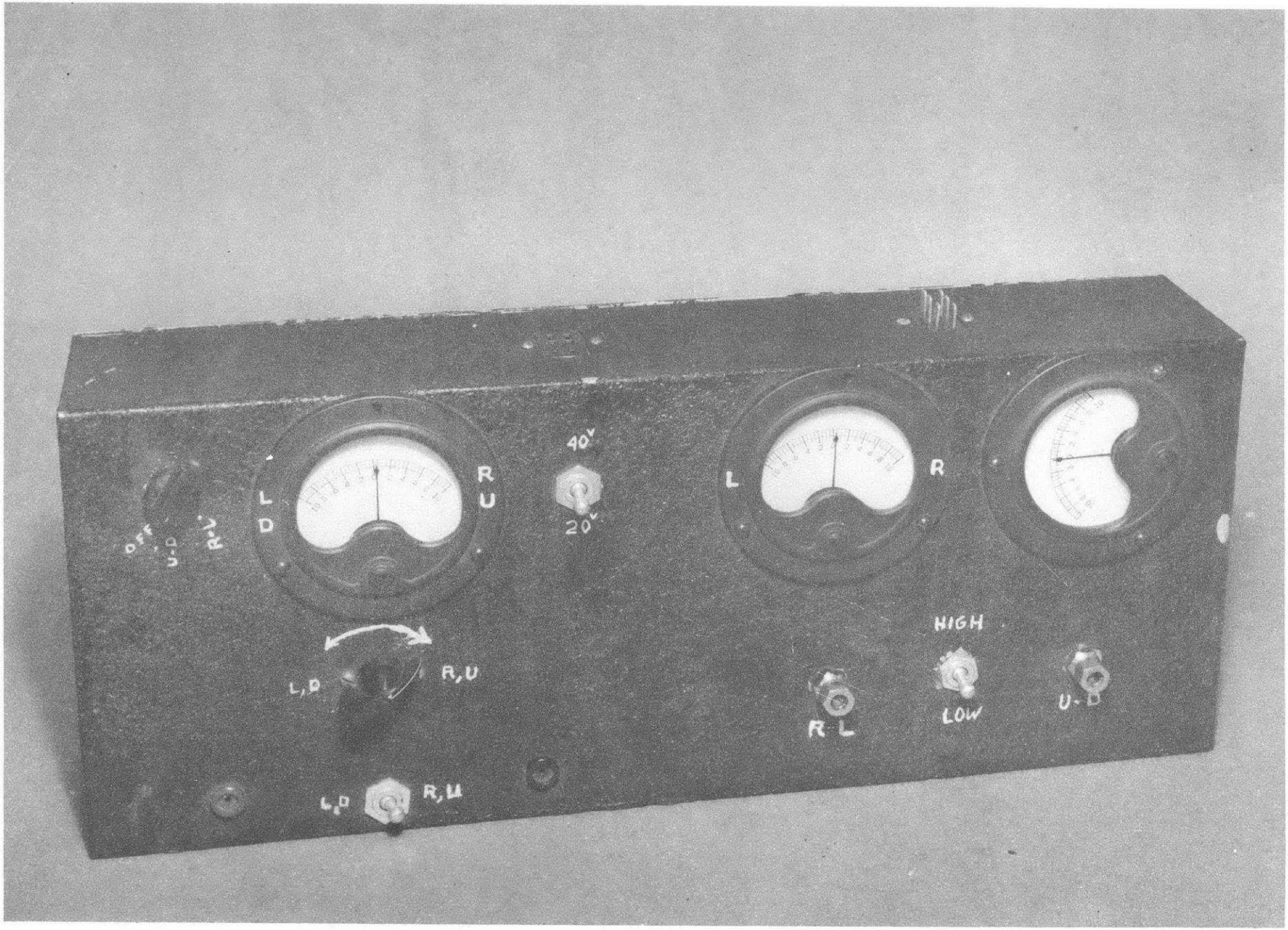
Clutch Performance Test Stand (Pitch Side)

Figure 5.3.2



Trim Servo Test Amplifier

Figure 5.3.3

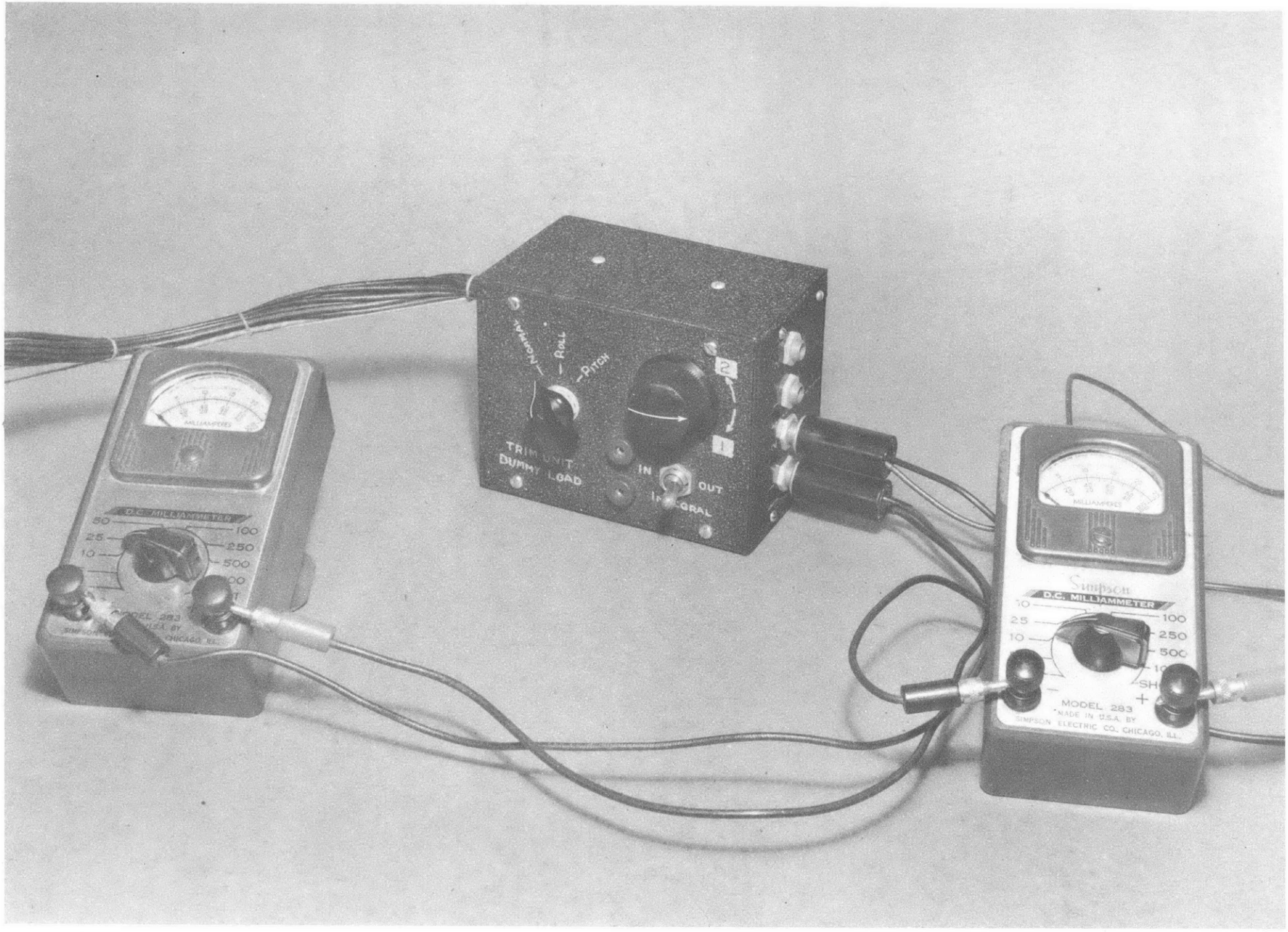


Volter-Indicator

Figure 5.3.4

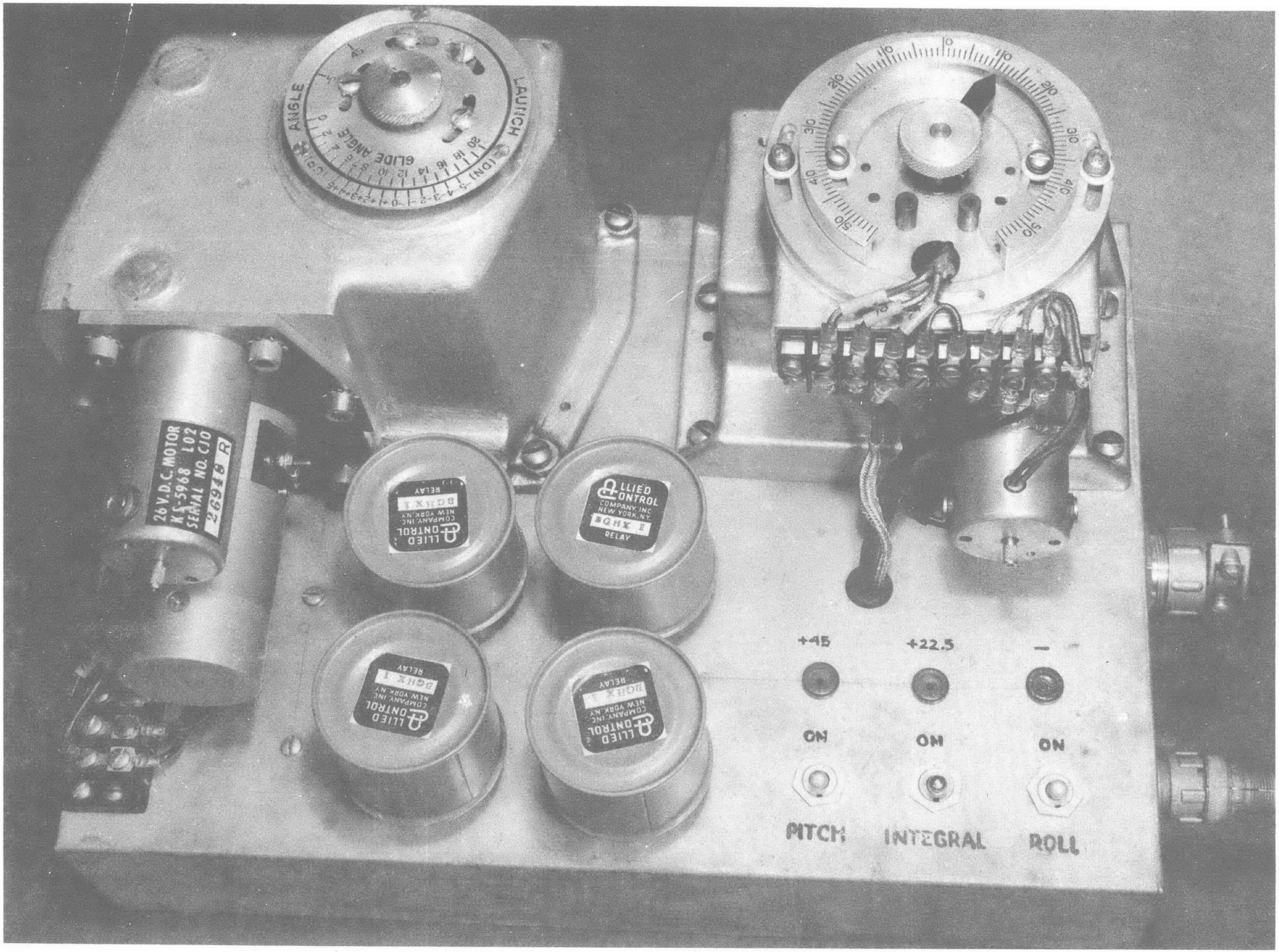


Control System Test Box



Dummy Load Unit

Figure 5.3.6



Trim Amplifier Test Unit

Figure 5.3.7

6.0 FLIGHT TESTS

6.1 Description

In March 1945 a program was started in the Servomechanisms Laboratory to prepare ten SWOD (Control Systems) Mk 18 Mod 0 for flight test. These tests were to be performed as soon after the first of June as possible. The construction of the units for test is described elsewhere in this report.

In April 1945 some preliminary flight tests were made to determine the suitability of the venturi tubes which are used to supply the gyro element in the control system. These tests covered a period of one week. The description of the procedures and results are given in the section of this report dealing with the design of the air supply for the SWOD (Control System) Mk 18 Mod 0.

In June 1945 personnel and equipment were sent to the Bureau of Standards, Washington, for the purpose of modifying the SWOD (Radar) Mk 31 Mod 0 and preparing the SWOD (Airframe) Mk 13 Mod 0 for the required tests. The description of the modification and test of the radar units is covered in a separate report issued by the M.I.T. Field Experiment Group at the Bureau of Standards, Washington, D. C. This report is designated as "Modification of the SWOD (Radar) Mk 2 Mod 0 and Mod 1 for operation of the SWOD (Control System) Mk 18 Mod 0." The preparation of the airframes for the installation of the radar unit and the control unit was done at the Bureau of Standards.

It was decided to ship these airframes to the test group at Johnsville, Pennsylvania, where the SWOD (Control Systems) Mk 18 Mod 0 were being installed. In June 1945 three control systems were shipped to Johnsville. The next three were shipped early in July 1945, followed closely by the remaining four.

The proposed test plan was divided into two parts:

1. A series of three test drops were to be made at Warren Grove, New Jersey, using a corner reflector as a target. All units in this group were to contain instruments and cameras for the purpose of obtaining data during flight.

2. A group of 14 tests were to be made at Pamlico Sound, off North Carolina, using a ship target. Several of these units were to be instrumented to obtain data during flight. Seven units were to contain SWOD (Control System) Mk 18 Mod 0. Seven were to contain the SWOD (Control System) Mk 4 Mod 3.

All units were to have a complete checkout at Johnsville before flight. The units which were to be tested in North Carolina were to receive additional checkout upon arrival at Manteo, North Carolina.

The first three units were to receive a special radar test at Johnsville to determine if any "squint" was present in the radar units. (Squint may be defined as the deviation between the true position of the target and the apparent position indicated by the radar.) This test was performed by mounting the radar unit on a tracking stand and tracking an airplane target through a boresighted telescope while checking for "squint" on the directional meters in the radar unit. A rough check of the directional sensitivity of the radar unit also was made at this time. A complete description of these tests is included in the report designated as Final Report on Developments in Radar Homing Missiles OSRD Report No. 6413.

The servo unit received a checkout preliminary to installation in the airframe and a second checkout after they were installed. Directional sensitivity tests were made to determine the proper settings for each control unit. A complete radar checkout was also to be performed by personnel from the M.I.T. Field Experiment Group, assisted by Navy personnel at Johnsville. After hanging the completed weapon on the dropping airplane a final checkout was to be made to make sure that the controls were operating satisfactorily.

6.2 Results of Tests

The first three flight tests were made at Warren Grove, New Jersey, between July 5, 1945, and July 26, 1945. All three units were dropped from an altitude of 6000 feet on a glide ratio of 4.5:1. The dropping speed was 200 knots. The dropping signal was given by the ground crew. The target for this series of tests was a corner reflector comprising three 20-foot squares. The radar signal back from this target was equivalent to that from a medium size merchant ship.

The first unit dropped hit the ground approximately 60 feet in front of the corner of the reflector, and scattered debris through the right wall of the reflector. The second unit dropped flew just over the reflector and landed approximately 70 feet beyond the corner of the reflector. The third unit dropped malfunctioned and dove into the ground immediately after dropping. The principal difference between the first and second units was the setting of the antenna looking angle. The looking angle of the first unit was set to 5 degrees below the axis of the airframe. The looking angle of the second unit was set to 6-1/2 degrees. The adjustments of the third unit are not significant in the light of the malfunctioning of the unit. The first two flights were considered very satisfactory in that both hits were within 15 feet of the target measured in a vertical plane. Investigation of the wreckage of the third unit showed that the elevon driving links had been rendered inoperative because of a pivot pin which became loose. Precautions were taken on all succeeding units to prevent a reoccurrence.

The seven remaining units incorporating the SWOD (Control System) Mk 18 Mod 0 and seven Bureau of Standards units incorporating the SWOD (Servo) Mk 4 Mod 3 were tested simultaneously at Manteo, North Carolina. The results of these tests were even more encouraging than the results of the tests at Warren Grove. Of the 14 units which were taken to North Carolina, 12 were actually dropped. The tests were run in pairs, using one of each type of unit for each test. All radar units were given equal attention so that as nearly as possible a direct comparison of the merits of the two types of control systems could be obtained. The airframes containing the SWOD (Control System) Mk 18 Mod 0 were equipped with special triangular tail fins whereas the Bureau of Standards airframes were equipped with the standard circular tail fins. In addition to the change in tail fins, the horizontal stabilizer was moved back four inches from the position used on the Bureau of Standards units.

Given below is a brief summary of the results based on the Field Test Activity Report (Navy) covering these tests. The target ship referred to in the summary was 260 feet long, 45 feet wide, and had a wooden lattice built up on the deck to a height of 30 feet above the water line.

Units containing SWOD (Control System) Mk 18 Mod 0

No. 98, Skip Hit	Hit water approximately 50 feet short of the target and 34 feet to the left of midships with bomb skipping through wooden framework and scattering debris on both sides of ship.
No. 96, Skip Hit	Hit water approximately 40 feet short of target, bomb skipping and passing through wooden framework at about three to five feet above deck and 30 feet forward of midships.
No. 480, Direct Hit	Hit target 20 feet aft of midships, two feet above water line.
No. 94, Direct Hit	Hit target 45 feet forward of midships exactly at water line.
No. 258, Miss	Passed over target at bow. Impact about 103 feet beyond target (cross wind).
No. 102, Skip Hit	Hit water 15 feet short of bow and skipped into bow and hit keel in 25 knot cross down wind.
No. 99, No Test	Unit damaged when PB4Y-2 went over end of runway. Unit returned to Washington.

Units containing SWOD (Servo) Mk 4 Mod 3

No. 261, Near Miss	Touched uppermost structure of forward superstructure above bridge. Impact 65 feet beyond target and 103 feet forward of midships.
No. 463, Near Miss	Hit top three slats on superstructure at midships. Impact 70 feet beyond target.
No. 38, Hit	Hit pilot house on bow of target, 112 feet forward of midships.
No. 257, Miss	Passed near pilot house at bow, hitting water about 130 feet beyond target.
No. 45, Miss (wild)	Unit started to home, but lost intelligence and dove in approximately 3/4 mile short of target.
No. 464, Miss	Passed 140 feet off bow of target and hit water approximately 300 feet from stern in 25 knot cross down wind condition.
No. 259, No Test	Unit developed low receiver sensitivity in flight and was not dropped. Unable to clear trouble at Manteo and unit was returned to Johnsville.

The above summary indicates briefly the results of the flight tests. The following discussion will deal with the results in greater detail giving in addition the operating constants and flight conditions.

Unit	54V
Flight No.	B54
Date of Flight	July 5, 1945
Decalage Angle	2.0 degrees
Spinner Angle	5.0 degrees
Right-Left Control Constants	
Sensitivity	6.1 degrees roll/degree off course
Integral time constant	9.4 seconds
Attenuation factor	9.4 (See Section 4.3.5, paragraphs 3 and 4)
Up-Down Control Constants	
Sensitivity	1.6 degrees/second/degree off course
Integral time constant	19 seconds
Attenuation factor	2.9 (See Section 4.3.8, paragraph 3)
Dropping Altitude	6000 feet
Dropping Speed	200 knots
Glide Ratio	4.5:1

The radar unit for this test was a SWOD (Radar) Mk 2 Mod 0 modified for the SWOD (Control System) Mk 18 Mod 0 and for special instruments contained within the bomb.These instruments were a cathode ray oscilloscope with "A" type presentation, an AVC meter, up-down and right-left directional meters, and a suction gauge indicating the operating pressure of the gyro. These instruments were photographed during flight. A second camera was boresighted with the axis of the airframe so that it would provide actual directional information by photographing the terrain during the flight of the missile. The test was made at Warren Grove using a corner reflector as a target.

The flight of the missile was on the whole very satisfactory. The middle part of the flight, however, contained a noticeable rolling oscillation which cannot be readily explained. This oscillation ceased, however, after approximately two-thirds of the flight and from there on in the flight was very smooth. The missile struck the ground approximately 60 feet in front of the corner reflector and scattered debris through the right half of the reflector.

Unit	91V
Flight No.	B55
Date of Flight	July 7, 1945
Decalage Angle	2.0 degrees
Spinner Angle	6 degrees, 30 minutes
Right-Left Control Constants	
Sensitivity	5.2 degrees roll/degree off course
Integral time constant	Same as unit 54V
Attenuation factor	Same as unit 54V
Up-Down Control Constants	
Sensitivity	1.4 degrees/second/degree off course
Integral time constant	Same as unit 54V
Attenuation factor	Same as unit 54V
Dropping Altitude	6000 feet
Dropping Speed	200 knots
Glide Ratio	4.5:1

This unit was identical to unit 54V and was to be tested under identical conditions. One change was made, however. Because unit 54V hit short of the target it was felt that if the spinner angle was increased there would be a better chance of a direct hit. The spinner angle was increased from $5^{\circ}\ 00'$ to $6^{\circ}\ 30'$. The flight of this unit was satisfactory throughout. The roll oscillation which was apparent in the previous unit was completely absent. The unit passed within ten feet of the top of the corner reflector and struck the ground approximately 70 feet beyond. The camera records obtained from this flight were very satisfactory.

Unit	58V
Flight No.	B50
Date of Flight	July 26, 1945
Decalage Angle	1 degree, 54 minutes
Spinner Angle	5 degrees, 30 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	9.4 seconds.
Attenuation factor	9.4
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	19 seconds
Lead time constant	0.37 seconds
Attenuation factor	2.9
Dropping Altitude	6000 feet
Dropping Speed	180 knots
Glide Ratio	4.5:1

This unit was similar to the previous two units except that it contained extra modification in the up-down channel of the directional amplifier in the radar unit. A network was added to improve the pitch stability thus permitting a larger pitch sensitivity to be used. The pitch sensitivity was increased above that used in previous flights.

This test was unsuccessful because the unit malfunctioned during flight and dove into the ground immediately after dropping. Examination of the wreckage showed that a pivot pin in the elevon driving linkage had worked loose. Therefore no useful information was obtained from this flight.

Unit	98V
Flight No.	B64
Date of Flight	August 22, 1945
Decalage Angle	1 degree, 52 minutes
Spinner Angle	5 degrees, 22 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5200 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

This was the first unit to be dropped on a ship target (Fleetwood.) The preparation of the bomb was similar to that used in the previous three units and also contained the extra modification in the up-down channel in the directional amplifier. This modification was made on all succeeding units and has been incorporated into the circuit design. The bomb contained instruments as in the previous three flights.

The unit was dropped broadside to the ship and had a good signal at the time of dropping. The unit flew steadily and struck the water 68 feet short of midships. (See Figure 6.2.1.) Debris was scattered through the wooden framework and the dye marker contained in the bomb colored the water on both sides of the ship. It should be noted at this point that the deck of the ship at the point where the bomb struck is only four feet above the water line.

Efforts made to recover the cameras were unsuccessful. Therefore, no film records were obtained of this flight.

Unit	96V
Flight No.	B65
Date of Flight	August 23, 1945
Decalage Angle	1 degree, 53 minutes
Spinner Angle	5 degrees, 27 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5300 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

This unit differed from the preceding unit only in that it contained a concrete bomb instead of the special instrumented bomb. The conditions of the drop were the same as for unit 98V. The wind was light and the course of the drop was broadside. This unit showed more activity during its flight, rolling at times as much as 20 degrees right and left. The impact occurred 77 feet short of midships, the bomb hitting the ship's railing at the base of the wooden framework. (See Figure 6.2.2.) The bomb itself passed through and struck the water on the other side of the ship. Most of the debris from this unit remained on deck. In azimuth the impact was 20 feet forward of midships.

Unit	480V
Flight No.	B68
Date of Flight	August 24
Decalage Angle	1 degree, 52 minutes
Spinner Angle	5 degrees, 30 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5200 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

This unit flew in a manner similar to previous units except for an apparent change in homing direction during the middle of the flight. The new direction of homing did not last long and the unit returned to the original direction and continued into the target. The impact occurred approximately two feet above the water line just aft of midships. This can be considered a perfect hit. (See Figure 6.2.3.)

Unit	94V
Flight No.	B69
Date of Flight	August 25, 1945
Decalage Angle	1 degree, 54 minutes
Spinner Angle	5 degrees, 45 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5150 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

This unit was dropped in a light cross wind and corrections for drift were obvious to the flight observers. The impact occurred exactly at the water line. This can be considered a perfect hit. (See Figure 6.2.4.)

Unit	102V
Flight No.	B73
Date of Flight	August 26, 1945
Decalage Angle	1 degree, 59 minutes
Spinner Angle	5 degrees, 50 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5200 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

The conditions for this drop were exactly the same as for the preceding drop. There was a 24 knot wind from the right and the unit showed constant corrections for the drift resulting from the wind. The flight appeared to be more steady than the preceding flight. The impact occurred 15 feet short of the target directly in line with the bow of the ship. (See Figure 6.2.5.)

Unit	258V
Flight No.	B72
Date of Flight	August 26, 1945
Decalage Angle	1 degree, 50 minutes
Spinner Angle	5 degrees, 26 minutes
Right-Left Control Constants	
Sensitivity	4.5 degrees roll/degree off course
Integral time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Up-Down Control Constants	
Sensitivity	1.7 degrees/second/degree off course
Integral time constant	Same as unit 58V
Lead time constant	Same as unit 58V
Attenuation factor	Same as unit 58V
Dropping Altitude	5300 feet
Dropping Speed	180 knots
Dropping Range	3.8 nautical miles

The ratio between the decalage angle and the spinner angle was reconsidered in the light of results of the previous flights. These results indicated that a ratio of approximately 3:1 would be better than the ratio of 2.5:1 used on the previous units. This unit was dropped with the intention of determining the effect of the increased ratio. However, another factor entered into the test. This was a cross wind of approximately 24 knots. The unit was dropped approximately broadside and was required to make large corrections to the right to maintain its course. At the end of the flight the unit banked steeply to the right and passed over the target at the bow. The impact was approximately 200 feet beyond the ship. (See Figure 6.2.6.)

The above described flights show that a reasonable percentage of hits can be obtained under a variety of conditions. Although the number of tests made was small, a fairly reliable estimate can be made of the possibility of the weapon. At the same time it is obvious that a more complete series of tests will have to be made to obtain a complete evaluation. At this time there is not sufficient information on which to base an estimate of the accuracy of the weapon. However, it is encouraging to note that the center of the scatter of the hits seems to be midships and at the water line of a ship target. This fact, plus the number of hits made during these tests, seems to form sufficient evidence of the practicability of this bomb.

6.3 Records of Flights

The following records are graphical plots of data obtained by means of cameras placed at strategic points to observe the flight path of the missile. The glider number appearing in the upper right-hand corner of the graphical plot corresponds to the unit number given in Section 6.2 under Results of Tests. Records are also included showing flight results of missiles controlled by SWOD (Servo) Mk 4 Mod 3 (Bureau of Standards), but these records are not discussed under Section 6.2.

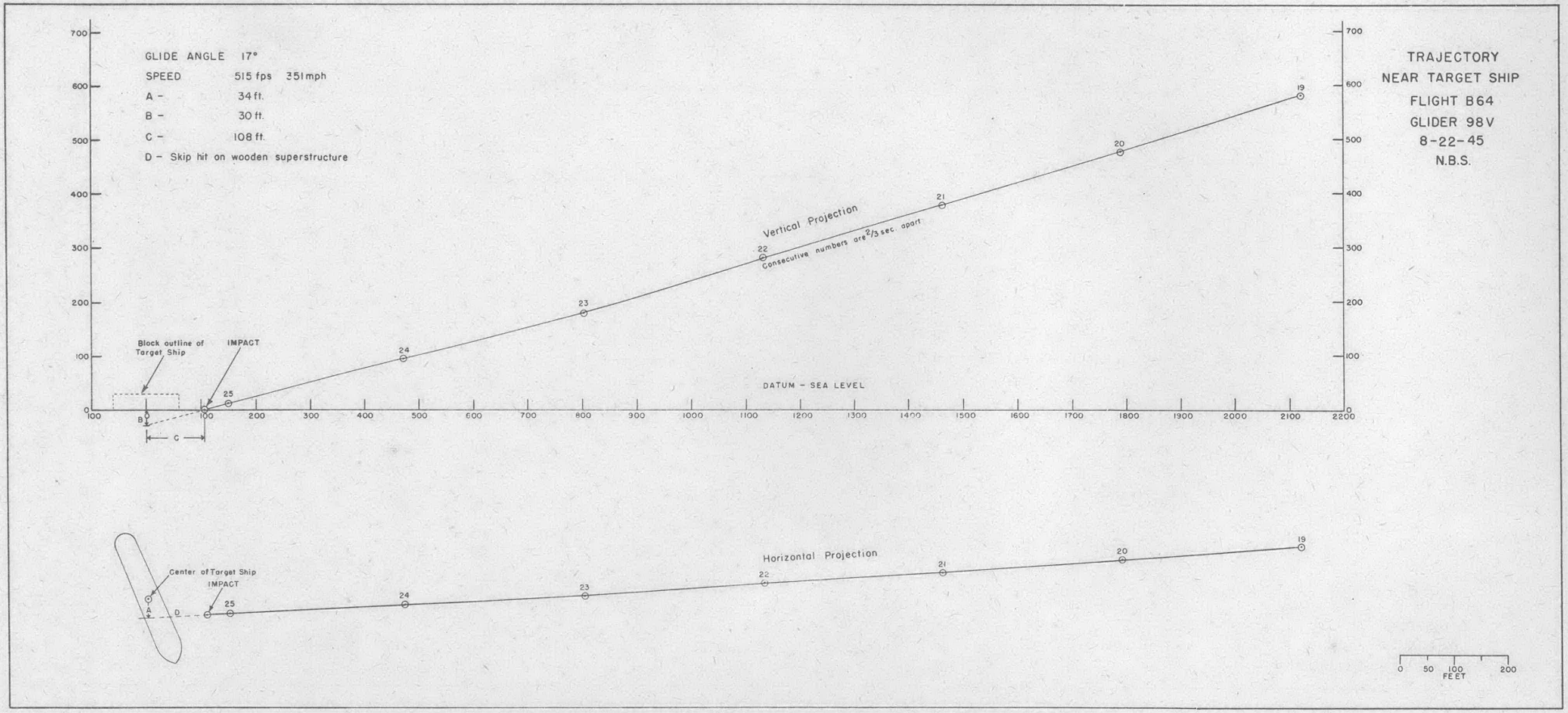
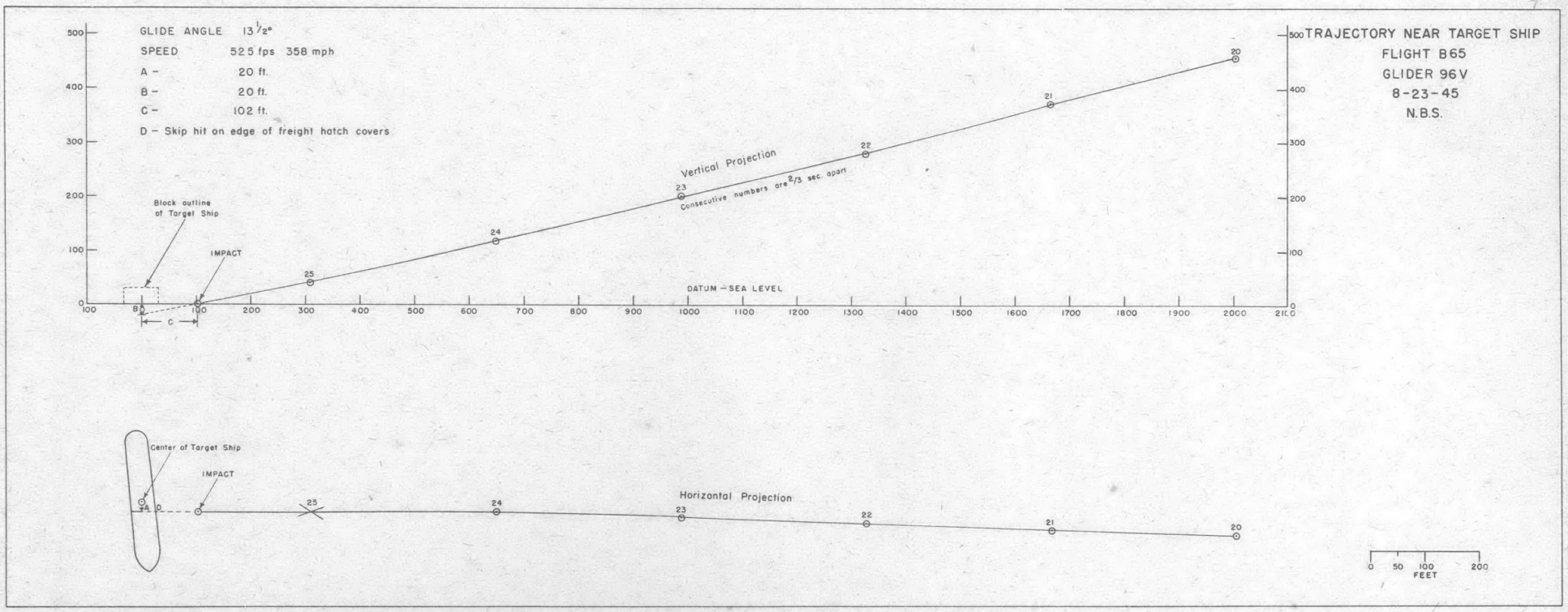

GLIDE ANGLE 17°
SPEED 515 fps 351 mph
A – 34 ft.
B – 30 ft.
C – 108 ft.
D – Skip hit on wooden superstructure
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B64
GLIDER 98V
8-22-45
N.B.S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
DATUM – SEA LEVEL
Block outline of Target Ship
IMPACT
Horizontal Projection
Center of Target Ship
IMPACT
0 50 100 200
FEET

Figure 6.2.1

Figure 6.2.2


GLIDE ANGLE 13 1/2°
SPEED 525 fps 358 mph
A – 20 ft.
B – 20 ft.
C – 102 ft.
D – Skip hit on edge of freight hatch covers
TRAJECTORY NEAR TARGET SHIP
FLIGHT B65
GLIDER 96V
8-23-45
N.B.S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
Block outline of Target Ship
IMPACT
DATUM – SEA LEVEL
Center of Target Ship
IMPACT
Horizontal Projection
0 50 100 200
FEET

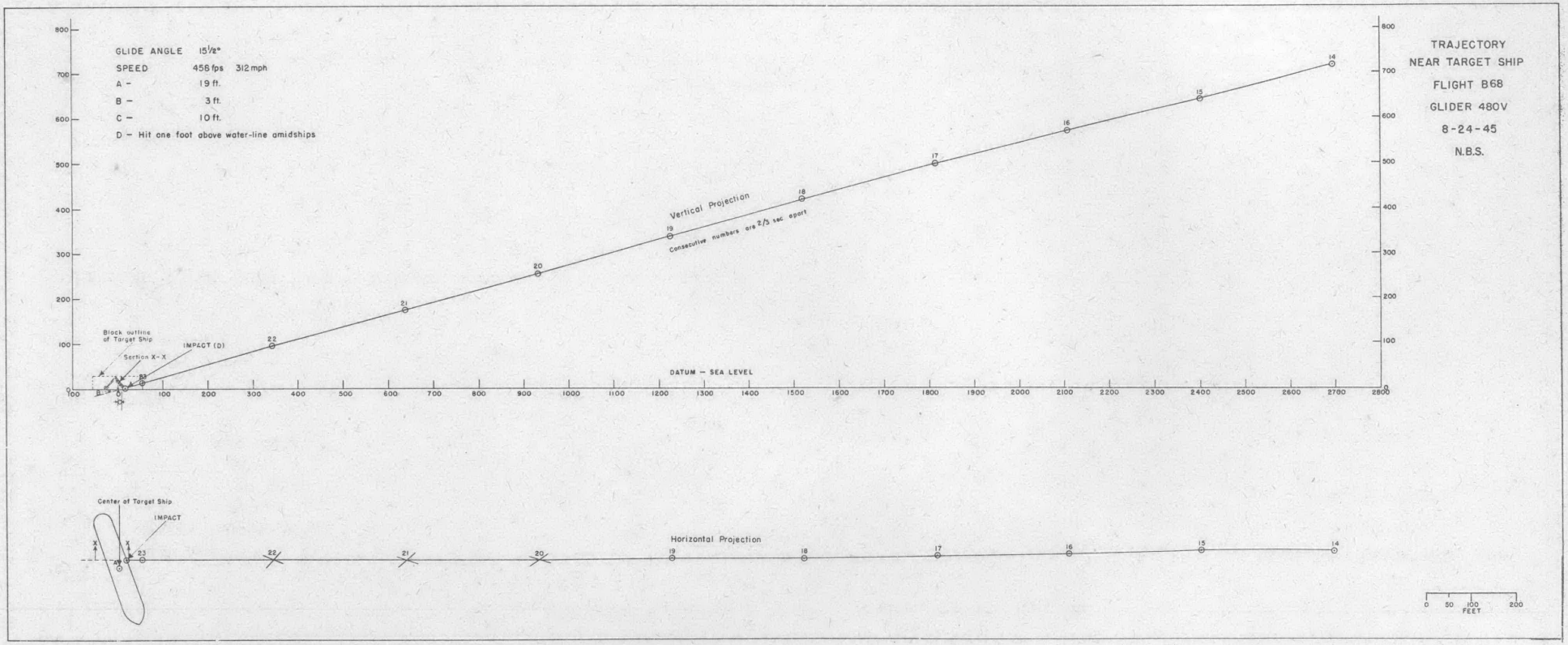
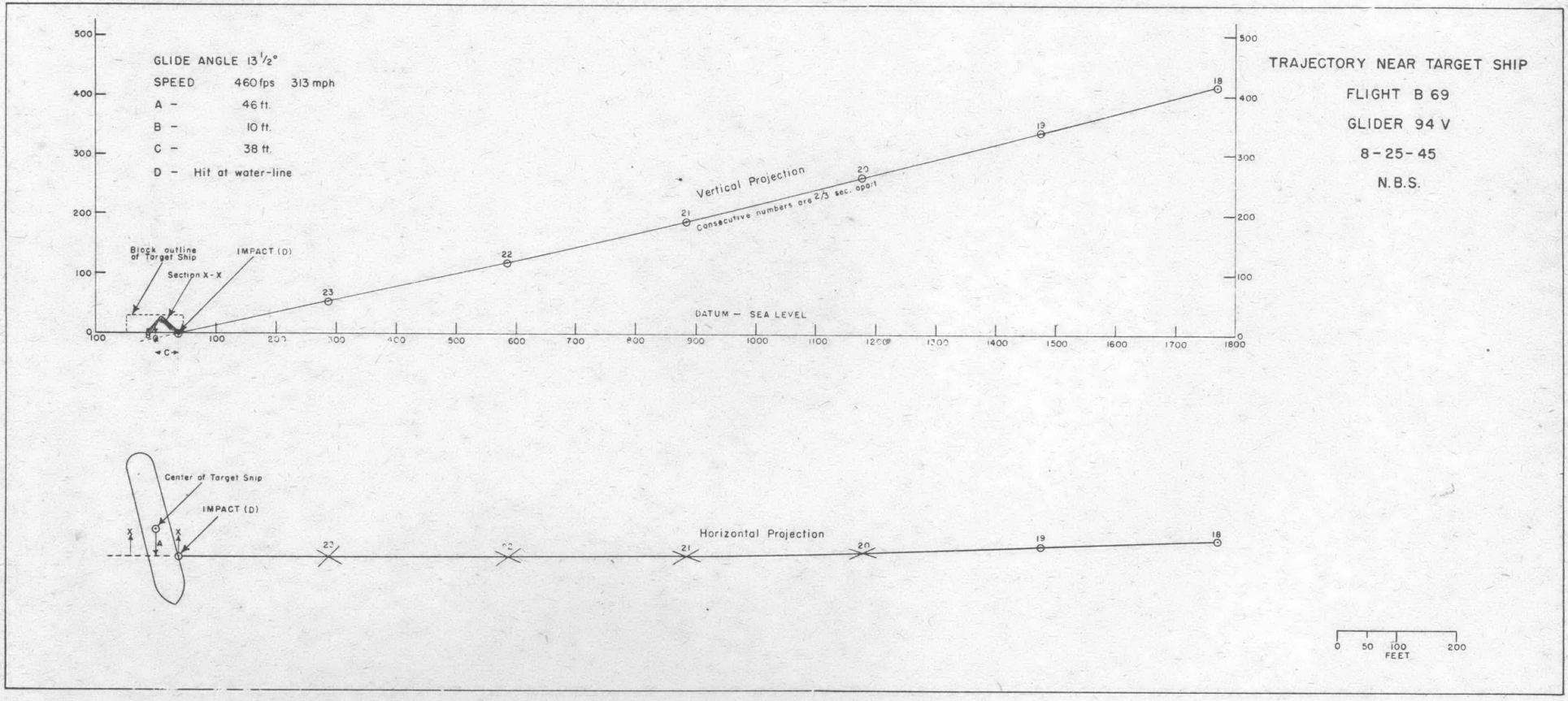

GLIDE ANGLE 15½°
SPEED 458 fps 312 mph
A – 19 ft.
B – 3 ft.
C – 10 ft.
D – Hit one foot above water-line amidships
Vertical Projection
Consecutive numbers are 2/3 sec apart
DATUM – SEA LEVEL
Block outline of Target Ship
Section X-X
IMPACT (D)
Center of Target Ship
IMPACT
Horizontal Projection
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B68
GLIDER 480V
8-24-45
N.B.S.
0 50 100 FEET 200

Figure 6.2.3

Figure 6.2.4


GLIDE ANGLE 13 1/2°
SPEED 460 fps 313 mph
A – 46 ft.
B – 10 ft.
C – 38 ft.
D – Hit at water-line
Block outline of Target Ship
Section X-X
IMPACT (D)
Vertical Projection
Consecutive numbers are 2/3 sec. apart
DATUM – SEA LEVEL
TRAJECTORY NEAR TARGET SHIP
FLIGHT B 69
GLIDER 94 V
8-25-45
N.B.S.
Center of Target Ship
IMPACT (D)
Horizontal Projection
0 50 100 200
FEET

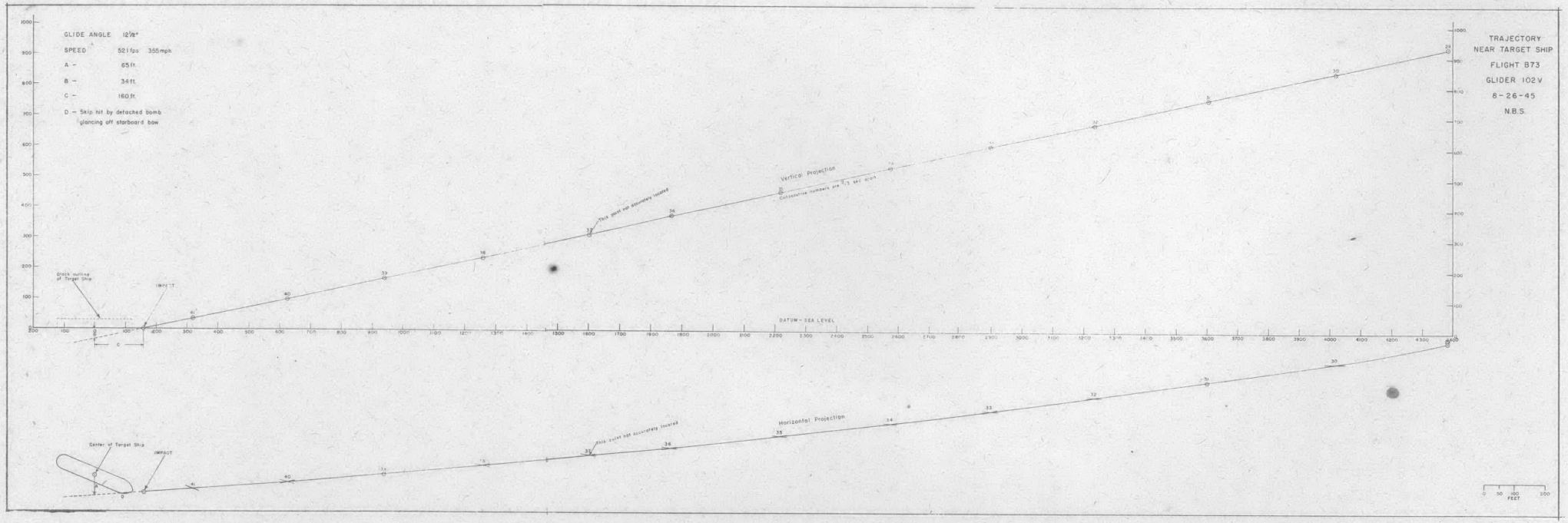
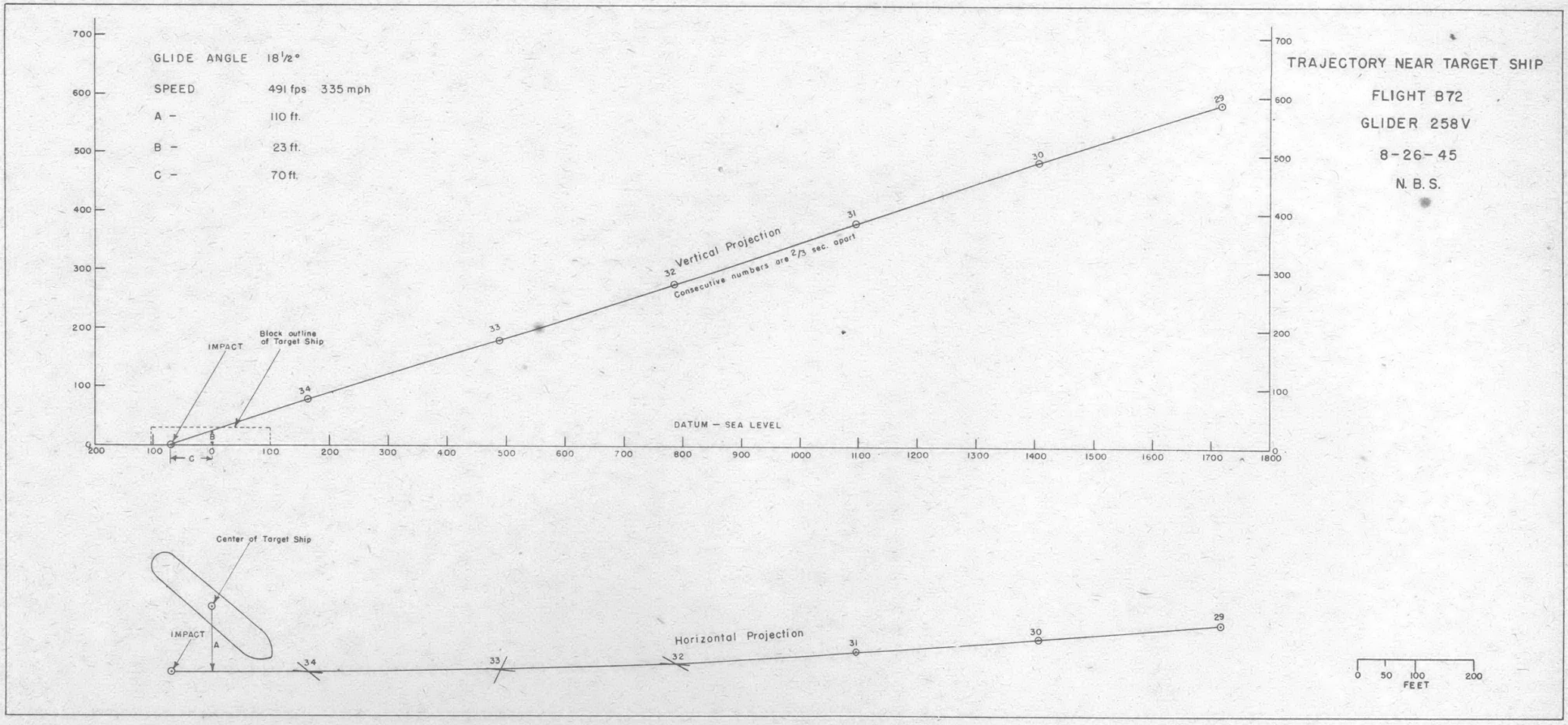

GLIDE ANGLE 12½°
SPEED 521 fps 355 mph
A – 65 ft.
B – 34 ft.
C – 160 ft.
D – Skip hit by detached bomb glancing off starboard bow
TRAJECTORY
NEAR TARGET SHIP
FLIGHT 873
GLIDER 102 V
8-26-45
N.B.S.
Vertical Projection
Consecutive numbers are 0.1 sec apart
This point not accurately located
Glass outline of Target Ship
IMPACT
DATUM – SEA LEVEL
Horizontal Projection
Center of Target Ship
IMPACT
This point not accurately located
0 50 100 200
FEET

Figure 6.2.5

Figure 6.2.6


GLIDE ANGLE 18 1/2°
SPEED 491 fps 335 mph
A – 110 ft.
B – 23 ft.
C – 70 ft.
TRAJECTORY NEAR TARGET SHIP
FLIGHT B72
GLIDER 258V
8-26-45
N. B. S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
IMPACT
Block outline of Target Ship
DATUM — SEA LEVEL
Center of Target Ship
Horizontal Projection
0 50 100 200
FEET

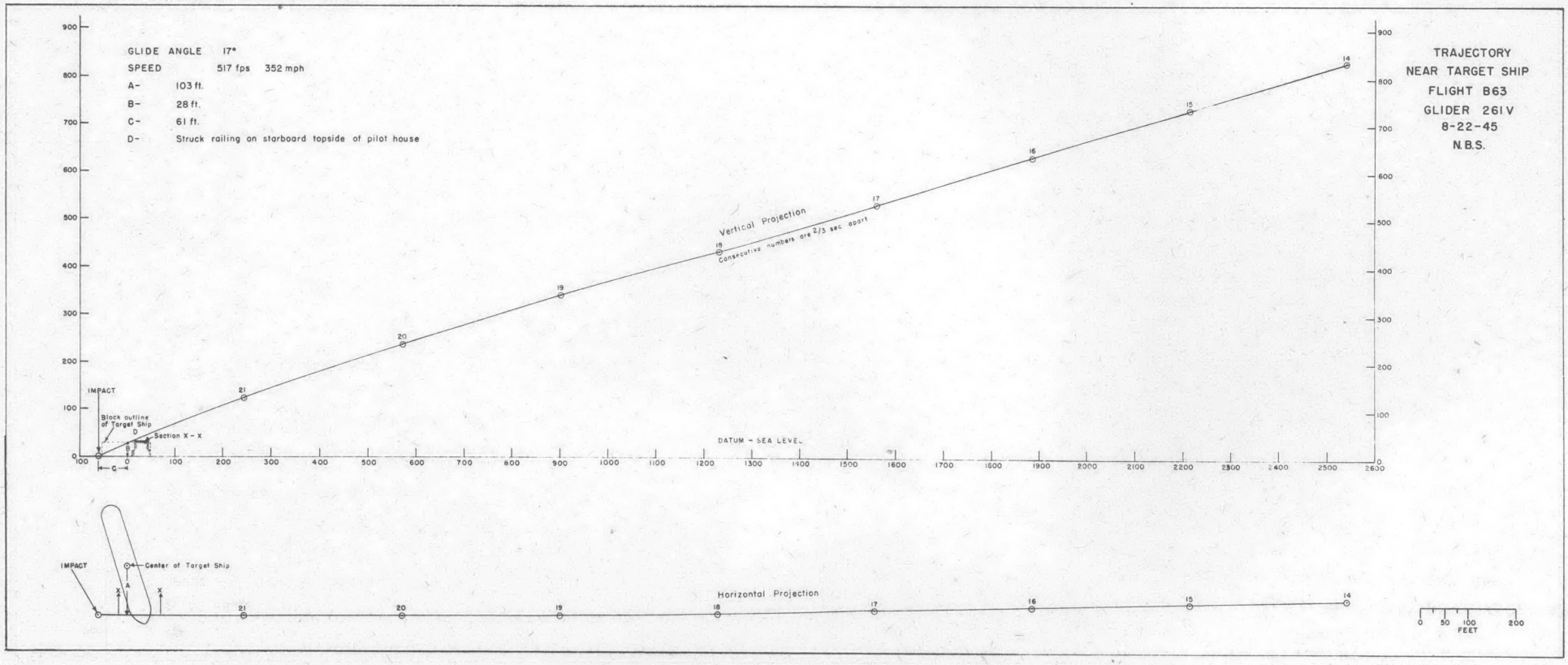
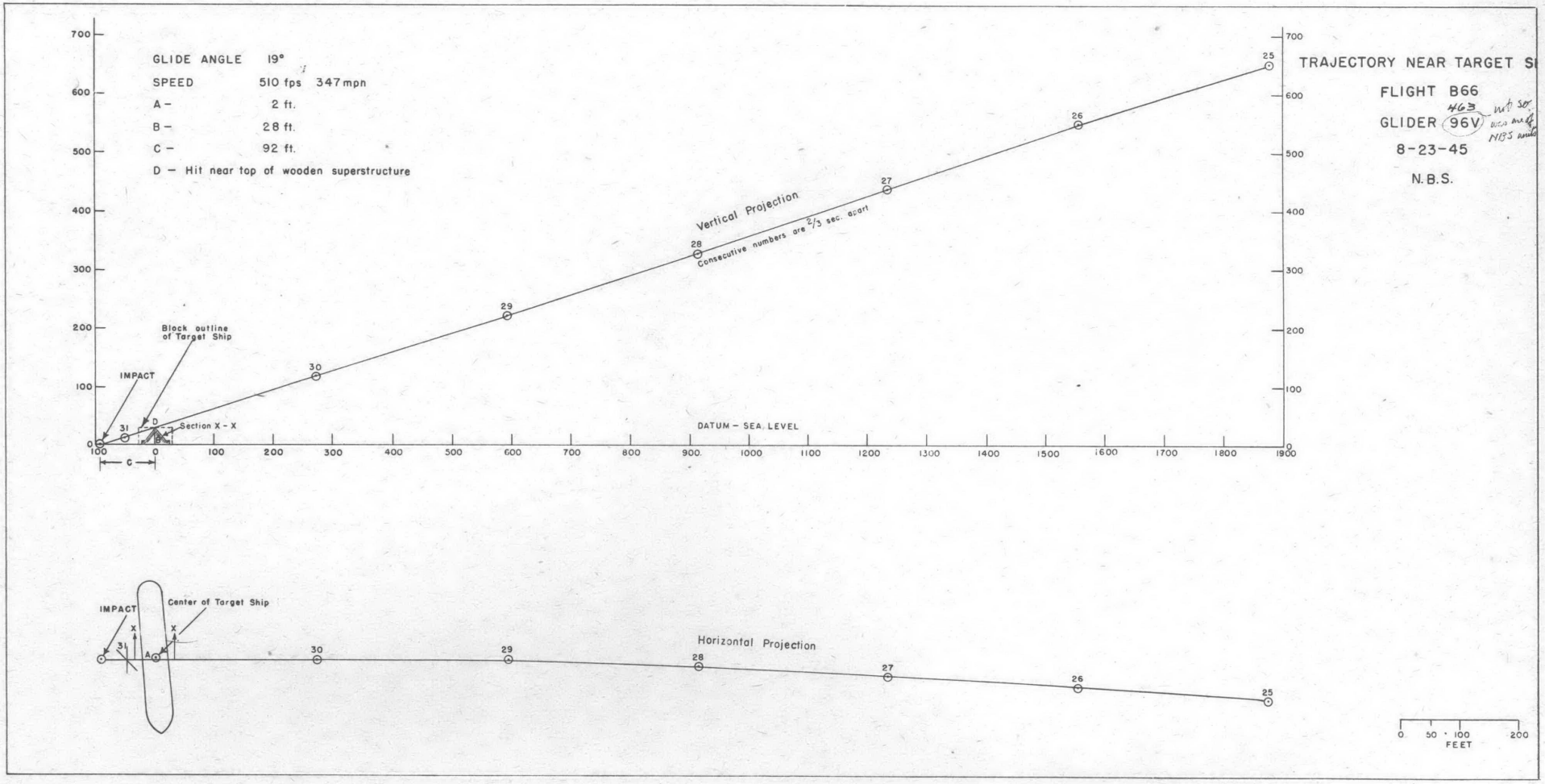

GLIDE ANGLE 17°
SPEED 517 fps 352 mph
A- 103 ft.
B- 28 ft.
C- 61 ft.
D- Struck railing on starboard topside of pilot house
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B63
GLIDER 261V
8-22-45
N.B.S.
Vertical Projection
Consecutive numbers are 2/3 sec apart
DATUM - SEA LEVEL
IMPACT
Block outline of Target Ship
Section X - X
Horizontal Projection
Center of Target Ship
FEET

Figure 6.2.7

Figure 6.2.8


GLIDE ANGLE 19°
SPEED 510 fps 347 mph
A – 2 ft.
B – 28 ft.
C – 92 ft.
D – Hit near top of wooden superstructure
TRAJECTORY NEAR TARGET SI
FLIGHT B66
GLIDER 96V
8-23-45
N.B.S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
Block outline of Target Ship
IMPACT
Section X - X
DATUM – SEA LEVEL
Horizontal Projection
Center of Target Ship
0 50 100 200
FEET

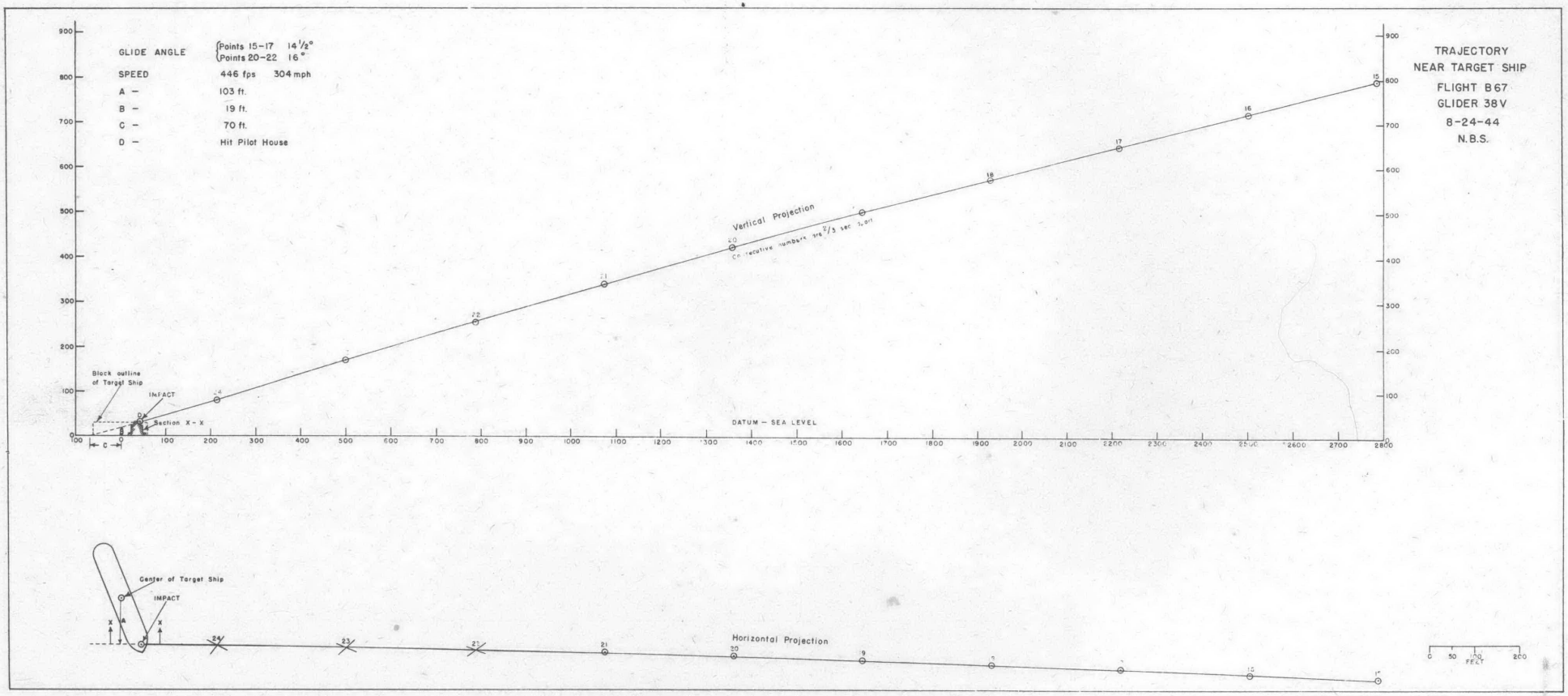
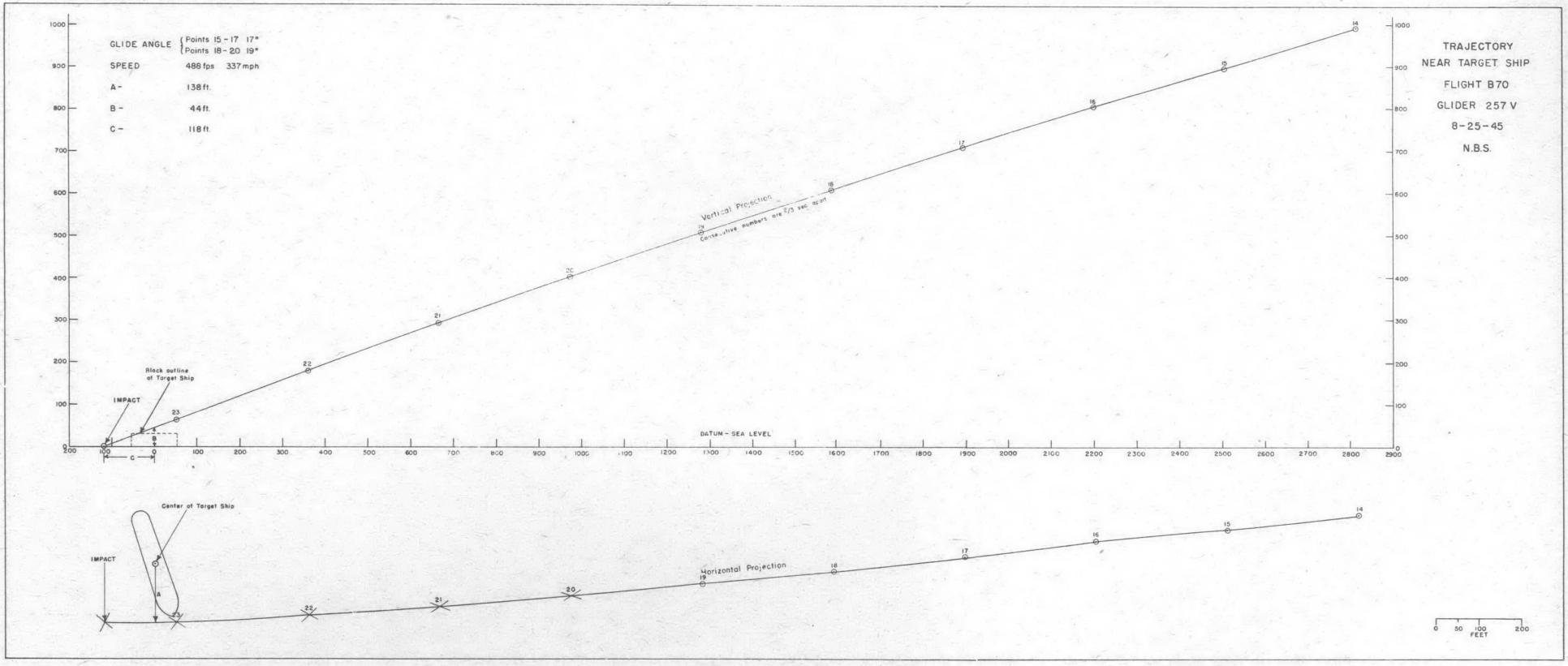

GLIDE ANGLE
Points 15–17 14 1/2°
Points 20–22 16°
SPEED 446 fps 304 mph
A — 103 ft.
B — 19 ft.
C — 70 ft.
D — Hit Pilot House
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B67
GLIDER 38V
8-24-44
N.B.S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
DATUM — SEA LEVEL
Block outline of Target Ship
IMPACT
Section X – X
Center of Target Ship
IMPACT
Horizontal Projection
0 50 100 200
FEET

Figure 6.2.9

Figure 6.2.10


GLIDE ANGLE Points 15-17 17°
Points 18-20 19°
SPEED 488 fps 337 mph
A- 138 ft.
B- 44 ft.
C- 118 ft.
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B70
GLIDER 257 V
8-25-45
N.B.S.
Vertical Projection
Consecutive numbers are 1/5 sec. apart
DATUM - SEA LEVEL
Black outline of Target Ship
IMPACT
Center of Target Ship
Horizontal Projection
0 50 100 200
FEET

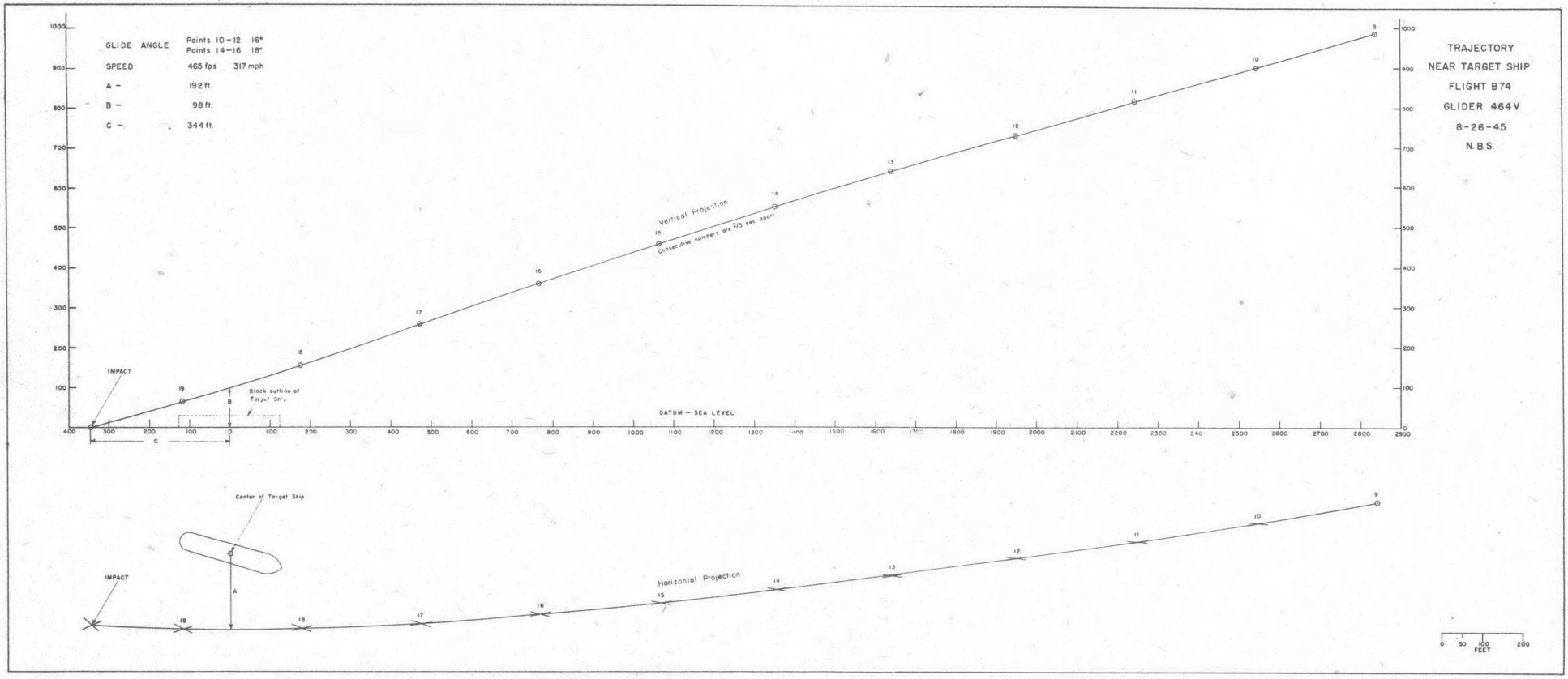

GLIDE ANGLE
Points 10 – 12 16°
Points 14 – 16 18°
SPEED 465 fps 317 mph
A – 192 ft.
B – 98 ft.
C – 344 ft.
TRAJECTORY
NEAR TARGET SHIP
FLIGHT B74
GLIDER 464V
8-26-45
N. B. S.
Vertical Projection
Consecutive numbers are 2/3 sec. apart
IMPACT
Block outline of Target Ship
DATUM – SEA LEVEL
Center of Target Ship
Horizontal Projection
0 50 100 200
FEET

Figure 6.2.11

7.0 PRODUCTION MODEL

On March 10, 1945, the first work was started on the experimental design of the mechanical section of the SWOD (Control System) Mk 18 Mod 0. Within a few days representatives of Bell Telephone Laboratories were at M.I.T. to follow the design as it was developed. It was the general aim to design the experimental model as well as possible with consideration of production needs in order that a minimum of time and effort would be required of Bell Telephone Laboratories in producing a production design. Due to the urgency of the program, however, it was necessary to make use of standard components, such as gears, shafts, etc., and a different type of construction than would be advisable for production. Engineers from Bell Telephone Laboratories, working in close proximity, were kept in touch with such expedients. Later, the BTL design work was moved to a Boston office but close liaison was continued.

Before describing the detailed differences between the experimental and production designs, it seems appropriate to point out the general improvement in the production design. Although the components and general arrangement of the experimental design was retained, numerous improvements throughout resulted in a mechanism which was smaller, lighter, had fewer parts, and was relatively easy to produce in quantity.

7.1 The Complete Assembly

One improvement is the addition of a metal cover over the entire unit. This cover should do much toward preventing the infiltration of foreign matter into the power gearing in particular, and into the unit in general; and also should prolong the life of electrical insulation.

A more fundamental difference, one frequently encountered in stepping from an experimental to a production design, is the greater use of castings in the structural members. In the experimental design for the Bat, it was expeditious to use plates and spacers for forming the deck structure because of time saved in not requiring fabrication of complicated patterns for castings. For production, however, it was far better to use castings, and in so doing, many fewer parts were needed.

Throughout the production unit more engineering attention was given to materials, anti-corrosive treatment, and general finished appearance; whereas its predecessor was made up of available material, since the corrosion problem was unimportant for the test equipment.

One section of the unit changed very little, principally because it had enjoyed a reasonable development program in the Servomechanisms Laboratory prior to use in either experimental or production design. The clutch unit was quite complete and ready for use in the experimental design and was very satisfactory in the production design.

As mentioned previously, the production unit is lighter, weighing some 15 pounds less than the experimental unit whose weight was about 100 pounds, and thus is easier to handle, ship, and install. Due to more compact and efficient design of the components, more working space is available for the assembly of major components.

7.2 Bearing Plate Assembly

Three aluminum castings form the decks, onto which all the major components are attached as well as numerous auxiliary parts. Oilite bearings are pressed into bored

holes in the deckplates. These press fits are relied upon to prevent the Oilites from spinning or losing their seating. This was brought up because the Servomechanisms Laboratory previously had experienced difficulties due to this point. However, the decision was made by BTL to simply press the bearings into place. The three-piece construction saves a large number of parts and allows much better assembly procedures.

An improved electrical harness was developed which passes below the bearing plate and thus provides more assembly working space. Better brackets and more suitable AN connectors are used.

The mechanical limit stops were designed without shock pads so that metal to metal contact is obtained when the stops act. Since these stops act only when the electrical limits fail, it is only an emergency mode of operation, and on this basis may prove satisfactory, although actual tests should be carried out to check this point. Furthermore, the mechanical limits were designed as part of the pantograph segment gears, which is considered good practice, and thereby eliminates several parts from a design such as the experimental one.

The electrical limit stops were designed with roller type microswitches operating on cams, which is better than the sliding pins riding on cams as in the experimental design.

Concerning the power gearing, more conservative design for strength increases the reliability, although the experimental design suffered no failures even though subjected to severe testing conditions, unless it can be considered a failure when the tooth faces exhibited slight surface deterioration after an hour of severe testing.

7.3 Pitch Trim Unit Assembly

The new unit is slightly smaller than the M.I.T. design and the casting walls are thinner. Most important is the inclusion of the differential assembly as an integral part of this assembly. Doing this increases the complexity of the housing, but seems worthwhile in the more positive location of the differential without extra parts and assembly work.

Of relatively secondary importance is the use of Oilite bearings rather than ball bearings, and the addition of motor and generator replacement features. The latter are attained through the use of special couplings and mountings whereby the motor or generator may be replaced without disassembly of any of the rest of the subassembly from the unit. This feature depends upon the vulnerability of the motor and generator for its value.

Assembly is facilitated by the use of a zero lead worm drive from the generator to the differential. The adjustment of the slip clutch is made easier by the use of two jam nuts.

7.4 Roll Trim Unit Assembly

As in the Pitch Trim Unit, this unit includes the differential assembly as an integral part, making the assembly slightly more complicated but very worthwhile in the saving of assembly time in quantity production.

Oilite bearings were used in place of ball bearings.

The unit is smaller than the M.I.T. design and affords more room for working around the outside.

In the potentiometer, a better connection from slide to terminal was effected by using flexible braid and keeping the braid enclosed within the plastic shell of the potentiometer.

7.5 Gyro Assembly

The major difference is the design of the roll and pitch input shafts to the air valves so that the shaft supports are integral with the air valve body castings. In the M.I.T. design the shafts were either wholly or partly supported on the end casting. The B.T.L. design is considered an improvement.

On the roll input to the air valve sleeve is a pair of small but accurate miter gears taken from the bleeder adjustment mechanism which is otherwise not used. This is considered to be a useful expedient, although the miter gears seem rather small for their application.

As elsewhere in the unit, Oilite bearings have replaced ball bearings. In the gyro unit, this does not seem the best place for plain bearings; however, if practice proves them satisfactory, no comment is in order. The objection felt is that the plain bearings offer more friction than ball bearings and may be more difficult in operation at low temperatures. Another slight objection from the standpoint of friction is the use of Garlock seals on the input shafts for holding the vacuum in the case against air leakage. The use of such seals is certainly advantageous in sealing, but excessive friction is to be avoided, otherwise the loading springs in the Roll Trim Unit Assembly may prove too weak to positively actuate the gyro air valve sleeve.

The redesign of shaft supports already mentioned, along with redesign of the end casting eliminated the accurate positioning required of the end casting on the M.I.T. design. The new design makes use of holes already in the gyro bulkhead for end cover mounting, thus improving over the M.I.T. design which used special holes. Assembly procedure was simplified thereby. The air connection has been improved on the end casting.

In the caging unit several improvements were made. First, since the whole pilot is covered, the caging casting was left open on the sides, making a simpler casting and an easier assembly. Through the use of special gears and better spacing, the gear centers were designed in better relation to each other from the standpoint of casting strains affecting center distances. In the M.I.T. design binding of the caging gears resulted from casting strains when improperly mounted.

The caging motor is mounted on a plate which allows the motor to be assembled or replaced from the outside, without disassembly of the caging castings, which is an improvement over the M.I.T. design.

A gear ratio from motor to caging input was increased from 60:1 to 100:1 making for a larger factor of safety for full caging against the loading springs. As a result of binding trouble in the M.I.T. loading springs small washers were added to eliminate binding. The use of Oilite bearings in this mechanism rather than ball bearings is considered an improvement. A larger cam radius for the caging limit switches was used to prevent binding of the microswitch actuating pins.

The shock mounting brackets have been refined in design and improved in appearance.

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