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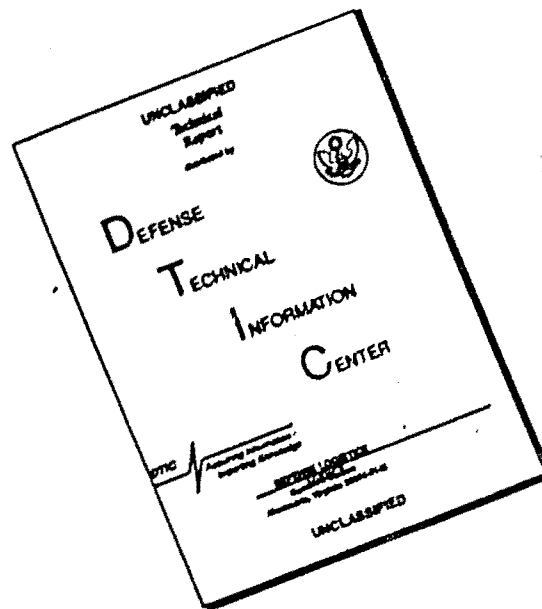
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July 1961



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YHU-1B CATEGORY I PERFORMANCE, STABILITY AND CONTROL TESTS

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Project Engineer

PAUL J. BALFE
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Project Pilot

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**AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE**

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ABSTRACT

The YHU-1B was tested by the Air Force Flight Test Center to gather limited performance and stability and control data to determine whether the helicopter will meet performance guarantees and to insure that no serious stability and control problems exist. The test program consisted of 18 flights totaling 24 hours and 40 minutes flight time between 5 October to 1 November 1960.

The YHU-1B is a single lifting rotor helicopter with a conventional tail rotor manufactured by the Bell Helicopter Company. It is powered by a Lycoming T53-L-5 gas turbine engine with a take-off rating of 960 shaft horsepower. For this program the fuel control was trimmed on the Lycoming test stand so the engine produced a maximum of 1100 shaft horsepower when corrected to standard sea level conditions. Test data from torquemeters showed that the installed engine was capable of producing 1085 shaft horsepower under the same conditions. Production aircraft will incorporate the T53-L-9 engine which is rated at 1100 shaft horsepower.

The design gross weight of this helicopter is 6600 pounds. Overload conditions allow 7660 pounds gross weight with all payload carried internally and 8500 pounds with an external load. During testing gross weight was varied from 5800 to 7660 pounds.

The test aircraft, S/N 5A-2078, is a modified HU-1 with HU-1B dynamic

components such as the rotor system, tail rotor, transmission, etc. No changes are programmed for the production HU-1B aircraft that will affect performance and stability.

The flying qualities of the YHU-1B are very good. In general, the flying qualities are improved over the earlier HU-1 series. This improvement stems primarily from the absence of the objectionable pitch and roll oscillations which were present in the HU-1. Control sensitivities are approximately equal, with a small decrease in pitch sensitivity being apparent in the YHU-1B. Control response of the two aircraft is approximately equal in pitch and roll, but the HU-1 develops a slightly greater yaw rate. Static and dynamic stability of the YHU-1B is generally good.

The helicopter meets all contractor guarantees for range, hovering, cruise speed, and service ceiling. However, it is felt that fuel capacity should be increased over the proposed 165 gallons to allow more adequate reserve for flight under instrument conditions. When compared to the HU-1 or HU-1A the YHU-1B has improved altitude performance, cruise speed, range and load carrying capabilities.

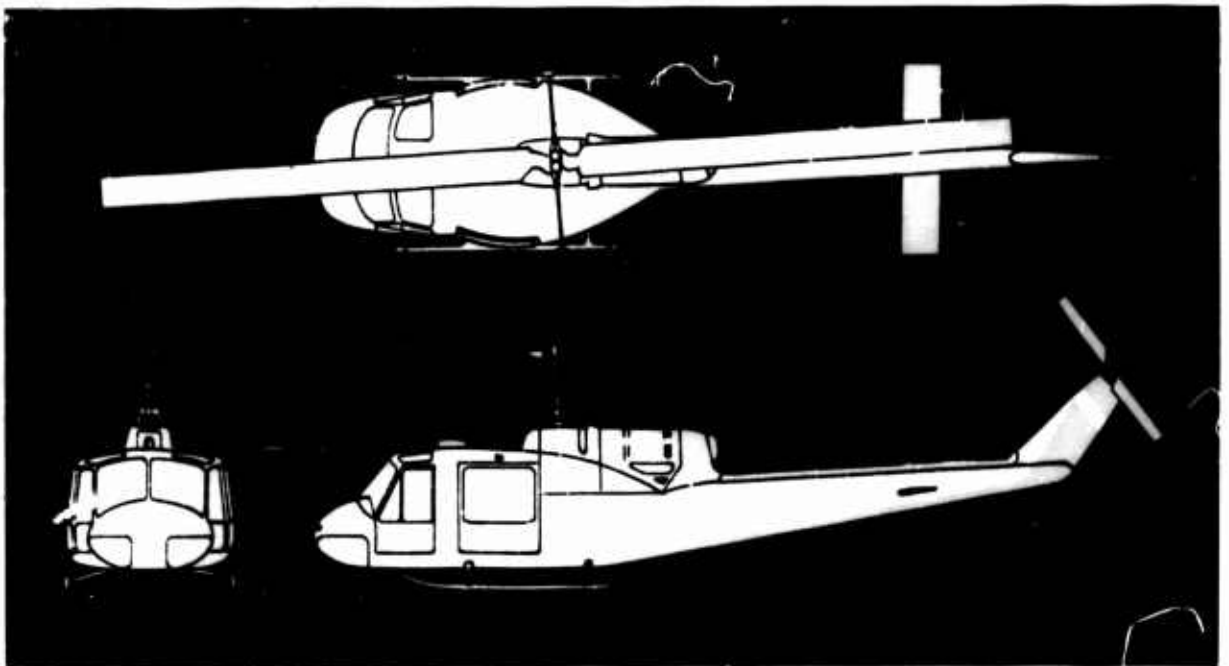
A general reduction in vibration is apparent with the YHU-1B. This is particularly significant at the higher airspeeds.

*This report
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approved*

Clayton L. Peterson
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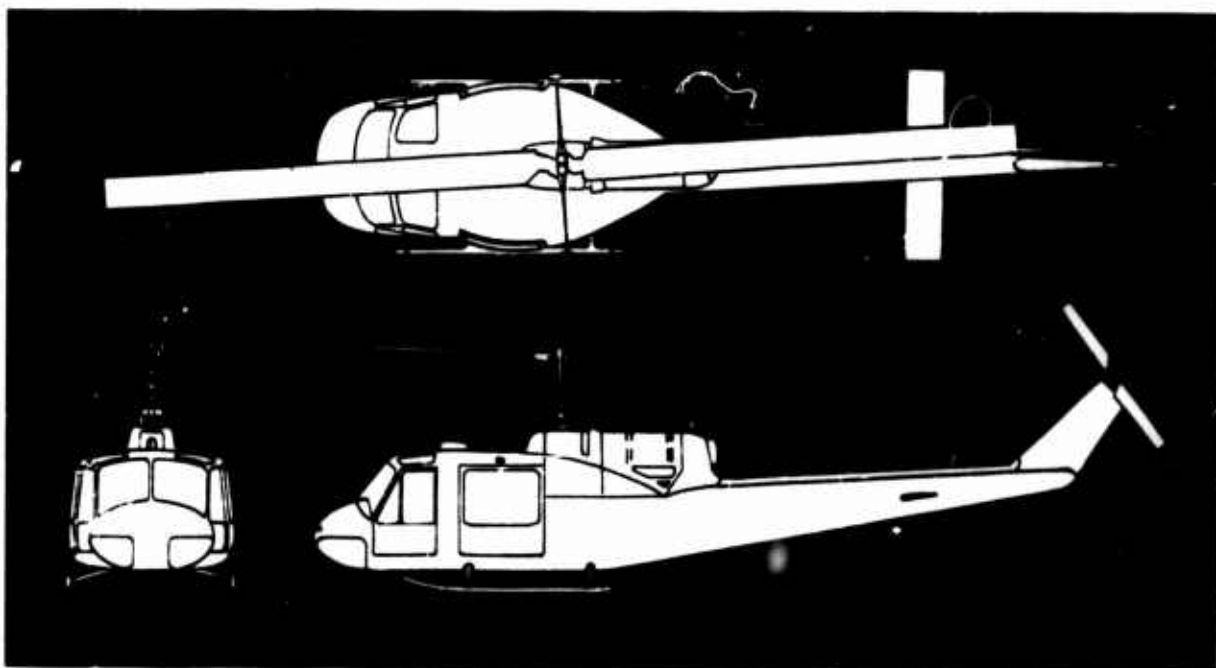


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Y H U - 1B



INTRODUCTION

This report presents the results of the Category I performance, stability and control tests performed on YHU-1B S/N 58-2078. These tests were conducted to determine if the helicopter could meet performance guarantees and determine if there were any stability and control deficiencies.

The test program was conducted at the Air Force Flight Test Center, Edwards Air Force Base, California from 5 October to 1 November 1960. Eighteen test flights were made for a total flight time of 24 hours and 40 minutes. The aircraft was maintained by the Bell Helicopter Corporation and the test instrumentation was installed and maintained by the AFFTC.

The YHU-1B is a single lifting rotor helicopter with a conventional tail rotor manufactured by the Bell Helicopter Company, Ft. Worth, Texas. It is powered by a Lycoming T53-L-5 gas turbine engine rated at 960 shaft horsepower at take-off. For the test program the engine fuel control was trimmed on the Lycoming test stand so the engine produced 1100 shaft horsepower. Installed in the helicopter the engine was capable of producing 1085 shaft horsepower at sea level on a standard day at maximum gas producer speed. Production HU-1B aircraft will be equipped with T53-L-9 engines rated at 1100 horsepower.

The design gross weight for this helicopter is 6600 pounds. The aircraft has an overload gross weight of 7660 pounds with only an internal load and a capability of 8500 pounds with an external load. The teetering or see-saw type two-bladed rotor is provided with a gyro stabilizing bar.

During testing the gross weight was varied from 5800 pounds to 7660 pounds and center of gravity position from 125.0 inches (forward) to 138 inches (aft). Flights will be conducted at 6500 pounds during Category II testing.

The control system is hydraulically boosted by a single hydraulic system incorporating irreversible valves. An artificial feel system for the cyclic controls and pedals is provided. A magnetic brake type stick centering system is provided to relieve control forces. The longitudinal cyclic control is connected to the horizontal stabilizer by a push-pull rod.

In this report, control positions are presented in the following manner:

1. Longitudinal and lateral cyclic displacements in inches from a neutral position where the stick is perpendicular to the floor. Full travel from neutral is ± 6.5 inches for both axes.

2. Pedal position in inches from a neutral position with pedals aligned fore and aft. Full travel is ± 3.5 inches from neutral.

The test aircraft, S/N 58-2078, is a modified HU-1 helicopter. It differs from a standard HU-1 in the following manner:

	<u>HU-1</u>	<u>YHU-1B</u>
Main rotor blade chord	15 in	21 in
Main rotor blade airfoil	NACA 0015	NACA 0012
Height (to top of rotor mast)	11.5 ft	12.5 ft
Transmission power limit	770 SHP at 6400 rpm	1100 SHP at 6600 rpm
Center of gravity range	Sta 128.0 to Sta 137.5	Sta 125.0 to Sta 138.0
Design gross weight	5725 lb	6500 lb

Contractor guarantees and specifications quoted in this report are to be found in Bell Helicopter Company report number 204-947-061A dated 20 January 1960, titled "Detail Specification for HU-1B Utility Helicopter".

Test data was released to the contractor as it became available. Final plots contained in Appendix I of this report were sent to Bell Helicopter Company on 1 June 1961.

PERFORMANCE TEST RESULTS



COCKPIT EVALUATION

The cockpit of the HU-1B is essentially the same as the HU-1A. The major difference is that the instrument panel in the HU-1B is extended 6 inches to the right. This places the instruments more nearly in front of the pilot which is considered desirable. The cockpit is easily entered from either side through wide hinged doors by using the step provided on the forward end of the skid. Access may also be gained to the cockpit from the cargo compartment by stepping over the center console. The collective pitch control is a slight hindrance to normal entry by the co-pilot and offers some obstruction to the pilot when taking his seat from the rear cabin area.

A wide console is provided between the pilot and co-pilot. The feature of functional grouping of switches and controls in individual removable panels is considered excellent. Generally, the instrument panel arrangement is satisfactory. The overhead panel contains the a.c. and d.c. power control switches, lighting controls, miscellaneous switches and the d.c. circuit breaker panel. Operation of these switches and breakers is satisfactory; however, it is necessary to lean over and to one side to read the identification markers.

The following deficiencies which were present in the HU-1A still exist:

1. The pilot and co-pilot's door handles are poorly located. The door handles are behind the pilot and are difficult to reach or operate due to the proximity of the seat to the door. The force required to operate the handle is

excessive and the sharp end of the handle digs into the pilot's hand when locking the door. (A 7)¹

2. The engine flight idle stop release system is unsatisfactory. The button electrically actuates a solenoid to retract a mechanical stop. It is possible to jam the stop by retarding the throttle prior to actuating the button. This requires adding throttle to release the pressure and allow the solenoid to retract the stop. Retarding the throttle while maintaining pressure on the button is difficult and inconvenient. In the event of complete electrical failure, it is not possible to shut down the engine since both the flight idle stop and shut off valves are electrically actuated. A positive mechanical flight idle stop should be provided that can be actuated by the pilot without removing his hand from the throttles. (A 1)

3. The collective pitch is too low when in the full "down" position for starting, run-up, or autorotation. During autorotation the pilot must bend forward and down, restricting his visibility. This condition is more serious in the HU-1B than the HU-1A because of the extended instrument panel. (A 3)

4. The throttle twist grip rotation is excessive. The pilot cannot rotate the throttle from full open to closed (full off) with one normal movement of his hand. Maximum allowable throttle travel is 150 degrees (HIAD J, 2-2.6.2.1). The HU-1B throttle travel exceeds this by approximately 90 degrees. (A 2)

¹ Numbers such as (A 7), etc., represent corresponding recommendation numbers tabulated in the Recommendations section of the report.

5. The a.c. circuit breaker panel located on the side of the console is hidden by the pilot's collective pitch stick and is not illuminated. The overhead circuit breaker panel is not illuminated. The a.c. circuit breaker panel should be relocated on the overhead panel and illumination should be provided for all circuit breaker panels. (A 6)

6. The cargo doors should be provided with jettison capability. These doors open aft by sliding on rollers and could be jammed in a crash landing. (B 3)

ENGINE START

Starting procedure for the gas turbine engine is relatively simple. During the start, the gas producer (consisting of the compressor, the combustors and the compressor turbine) rpm and exhaust gas temperature should be monitored closely along with the oil pressure indicators.

Following electrical power application to start the engine:

1. Place the fuel valve switch and oil valve switch in the "ON" position.
2. Check the fuel control switch in the automatic position.
3. Place the throttle twist grip in any position between ground idle and the flight idle detent.
4. Depress the starter switch and hold until 28 percent gas producer rpm or 490 degrees Centigrade exhaust gas temperature is reached. Operation of the starter switch is restricted to 40 seconds. A normal start will be accomplished in 20 to 25 seconds.

On two occasions a gas producer hang up was encountered. During these hang ups, the gas producer accelerated to ground idle rpm (42 percent) as in a normal start, but the EGT continued to rise to the red line at which time the throttle was placed in cut-off. An immediate successful second start was made following each hung start and no reason for the hung starts was determined. This is a potentially hazardous situation as the start appears normal to the pilot until the engine reaches ground idle speed (40 to 42 percent). It is imperative that the pilot monitor all engine instruments for several seconds after obtaining ground idle rpm to insure that the engine has stabilized at ground idle speed.

The aircraft is not equipped with a rotor brake; therefore, the rotor will begin to turn as the engine is started. After the engine reaches ground idle, the throttle is rotated to the full open position and maintained in that position during normal flight. As the throttle is opened, the rotor will accelerate up to the in-flight rpm range (5800 to 6600 power turbine rpm, 285 to 323 rotor rpm). The desired rpm within this range is selected by the pilot through the use of the power turbine (N₂) governor speed control switch (beep switch). No engine or transmission warm up is required.

LIFT-OFF

During lift-off an unsatisfactory loss of rotor speed was present. This loss occurs whenever collective pitch is applied. This condition is discussed more completely in the section on rpm droop.

The helicopter accelerates rapidly and smoothly from a hover to climb or cruise airspeed. As translational lift is reached, the amount of torque correction or left pedal must be reduced. This appears to the pilot as a large application of right pedal, but is not objectionable because the control forces are light.

HOVERING PERFORMANCE

Hovering performance was determined both in and out of ground effect utilizing tethered hovering and free flight techniques. Data was obtained tethered at a 1 foot and a 59 foot skid height. Rotor speeds were varied from 285 to 323 rpm. Hovering performance is summarized in Fig. 1, Appendix I. This summary plot is based on a free flight point at 13,200 feet and tethered flight points at 2300 feet. Free flight and tethered data are presented non-dimensionally in Fig. 2, Appendix I.

The helicopter exceeds the contractor guarantees for hovering ceiling out of ground effect.

The test aircraft exhibited a temperature rise at the engine bellmouth of 10 degrees Centigrade above ambient while hovering at the 1 foot skid height. This temperature rise is only 2 degrees Centigrade at 59 foot skid height. It is possible that hot exhaust gases are recirculated through the engine when the helicopter is hovered close to the ground. As a result, hovering ceiling in ground effect at the 1 foot skid height is decreased approximately 700 feet from what it would be if the temperature rise were only 2 degrees Centigrade. The contractor should conduct a study to determine whether this 10 degrees Centigrade temperature rise can be decreased. (B 4)

The YHU-1B utilizes the same power to hover out of ground effect as the YH-40. Since more power is available, hover ceilings are increased.

LEVEL FLIGHT PERFORMANCE

Level flight tests during this program were performed at density altitudes from 4,000 to 14,000 feet. Gross weight varied from 5600 to 7200 pounds and rotor speeds varied from 290 to 323 rpm. Individual test results are presented in Figs. 6 through 12 and summarized in non-dimensional form in Figs. 3, 4 and 5, Appendix I.

Two tests were performed at the same weight and thrust coefficient ($C_T = .00430$) with the center of gravity (cg) at the most forward limit (125 inch station) and the mid location (131 inch station). Center of gravity location apparently has a negligible effect on power required. Further tests will be conducted during the Category II to definitely establish the effect center of gravity position on power required.

Range:

Range performance was calculated from fuel flow data and power required curves obtained from level flight tests.

In making the calculations the following specifications for the guaranteed radius of action were used. This includes a full payload of 1230 pounds carried in both directions (crew included):

Engine start gross weight - 6600 pounds

Start engine (with full fuel load of 165 gallons = 1072 pounds)

Warm up and take-off (fuel usage equivalent to 2 minutes at normal rated power)

Cruise out at sea level

Land and shut down

Start engine

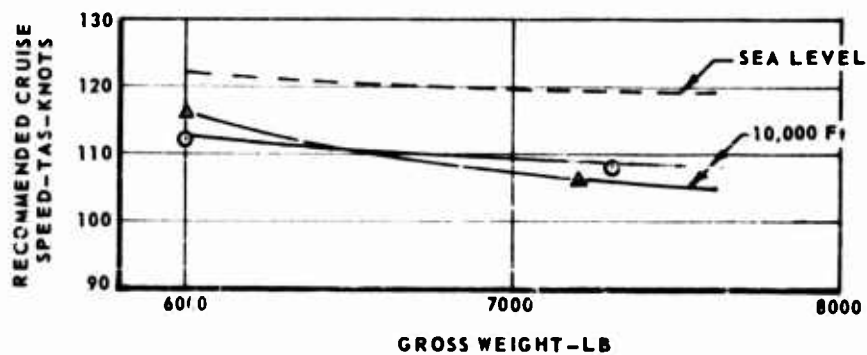
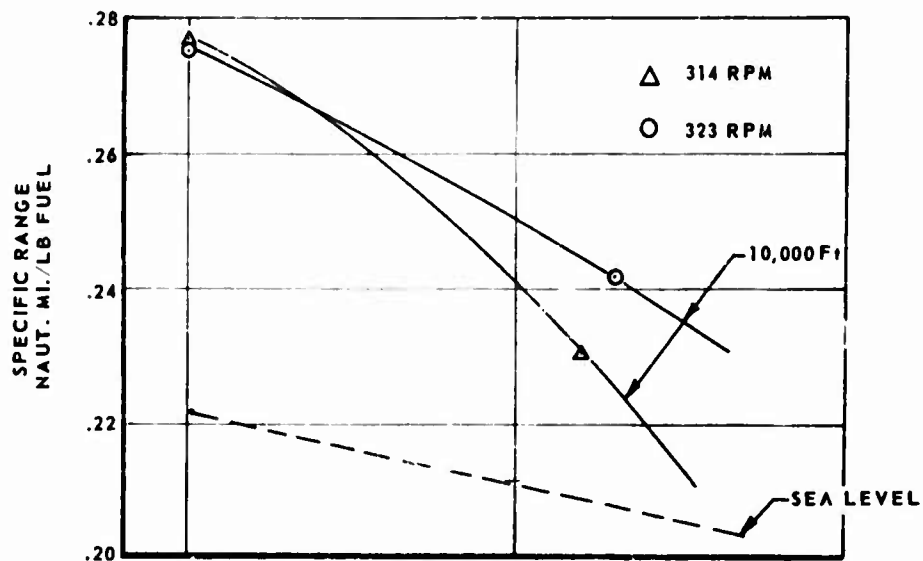
Warm up and take-off (fuel usage equivalent to 2 minutes at normal rated power)

Cruise back at sea level

Land and shut down with a reserve of 10 percent of initial fuel load

RANGE SUMMARY

YHU-1B S/N 58-2078
U.S. STANDARD DAY
CALCULATED FROM TEST DATA



The range for this mission is calculated to be 102 nautical miles at approximately 120 knots TAS and 314 rpm. This meets the contractor's guarantee of 100 nautical miles at a minimum cruise of 100 knots; however, these calculations do not include the 5 percent increase in fuel flow required by MIL-C-5011A. This range is for visual operation in smooth air using test instrumentation and carefully controlled pilot technique. The YHU-1B radius of action is increased approximately 22 nautical miles over that of the HU-1 at sea level. This gain in range is primarily due to the utilization of a higher and more efficient cruise speed made possible by lower vibration levels of the YHU-1B.

The mission requirements state that the aircraft be capable of operating in instrument conditions including light icing. The range of this helicopter will be seriously restricted in instrument flight as Army instrument rules require sufficient fuel to travel to the destination, to the alternate airport, plus 45 minutes of fuel reserve at cruising speed. An increased fuel supply would greatly expand the capabilities of the helicopter. (B 1)

The range summary figures shows a comparison of specific range (nautical air miles per pound of fuel) at sea level and 10,000 feet. The values shown are 70 percent of the maximum specific range calculated and the recommended true airspeed is the highest of the two airspeeds at which this specific range value occurs (Reference MIL-C-5011A). For the simulated mission, total range (radius of action) can be increased approximately 35 nautical miles at 10,000 feet; however, the average cruising speed will be 10 knots slower.

Rotor rpm has a negligible effect on range at sea level but a marked effect on cruise speed. For the described mission, range is decreased less than 1 nautical mile if 323 rpm or 307 rpm is used, but best cruising speed is reduced to approximately 110 knots TAS. The higher rpm becomes the more efficient rotor speed above 5000 feet at

heavier weights. The summary range figure shows a comparison of range at 10,000 feet for rotor speeds of 314 rpm and 323 rpm.

AUTOROTATION, APPROACH AND LANDING

Autorotational entries were performed with an aft center of gravity at speeds of 90, 100, and 110 knots CAS, and also at a forward center of gravity at 0, 10, 20 and 30 knots. For all conditions tested the entry is characterized by a mild pitch-up and a yaw to the left. Both the pitching and yawing moments are easily controlled. Quantitative rates of descents will be determined during the Category II performance test program.

Autorotational approaches and landings (touchdowns) were made at gross weights varying from 6000 to 7200 pounds. It was determined qualitatively that the minimum airspeed during approach to the flare and landing should be 60 knots IAS at gross weights greater than 6000 pounds. This recommended airspeed will provide sufficient rotor energy to slow the rate of descent and cushion the landing. A steep flare is required to stop the forward speed and break the rate of descent. This flare should be initiated 50 to 75 feet above the ground to avoid striking the tail boom on the ground. As the aircraft is leveled prior to ground contact, collective pitch should be slowly applied to control the rate of descent and cushion the touchdown. The ground sliding distance can be reduced if the collective pitch is lowered after the helicopter is firmly on the ground.

No control deterioration was noticed at any time during the tests; however, during the flare at gross weights above 6500 pounds, rotor speed exceeded 330 rpm. This was the power off limit during the Category I tests. The contractor has since raised this limit to 339 rpm. The feasibility of remaining within this limit will be determined quantitatively in the Category II test program.

Power approach characteristics are normal for a single rotor helicopter. The HU-1B is a relatively low drag helicopter and consequently is difficult to slow from cruise speed to final approach speed. Visibility on final approach is good. The pilot can see the intended landing area during all but the last 20 feet of the approach. There is no increase in the vibration level as the aircraft passes through translational lift. As a hover is reached a large amount of left pedal is suddenly required. Landing from hover is easy and the helicopter touches down nearly level on the skids.

ENGINE RPM DROOP

The engine rpm droop characteristics of the HU-1B are excessive. Whenever collective pitch is applied or a flight maneuver executed, this droop causes an excessive loss of rotor speed. The test aircraft was originally equipped with a standard HU-1A droop compensator cam that was not compatible with the test T53-L-5 engine. This cam was removed and a redesigned model was provided by Bell for the last two flights. This cam reduced the droop to the level of the HU-1A which amounts to a decrease of 10 rotor rpm at lift-off. The unsatisfactory droop characteristics of the HU-1A are discussed in AFFTC-TR-59-33. (A 4)

The power turbine (N₂) governor actuator rate is too slow. The unsatisfactory engine droop characteristics, aggravated by the slow N₂ governor or "beep" rate, makes precision rotor rpm control nearly impossible. The actuator cannot correct rapidly enough for large, rapid collective pitch movements. The collective control would have to move at an extremely slow rate to maintain a safe rotor speed during collective pitch changes. The present beep rate takes approximately 15 seconds from 285 rotor rpm to 323 rotor rpm. This time should be reduced to a maximum of 5 seconds. (A 5)

While this droop is evident during any power change or aircraft maneuver it is unsatisfactory during the lift-off, acceleration to forward flight, climb and landing phases.

During lift-off to a hover approximately 10 rotor rpm (200 power turbine or N₂ rpm) are lost and another 10 rotor rpm are lost as collective pitch is added to accelerate into forward flight and climb. In a maximum performance climb at 323 rotor rpm (6600 N₂ rpm), maximum available gas producer speed cannot be obtained. At 323 rotor rpm, N₁ rpm was 1 1/2 to 2 percent below the maximum. Maximum N₁ rpm can be obtained by bleeding rotor speed to 314 rpm (N₂ = 6400 rpm).

When the collective pitch is lowered and a slight flare initiated to slow the aircraft for landing, the rotor rpm will overspeed unless N₂ rpm is beeped down. The same condition is encountered when a final approach descent is initiated. As power is re-applied, the governor setting must be beeped up to provide the desired rotor rpm. If the helicopter is hovered using 323 rotor rpm, the governor setting must be reduced when the collective pitch is lowered to prevent overspeeding the rotor.

The contractor should initiate action to improve the engine droop characteristics and the governor actuator rate. (A 4)

VIBRATION

The vibration characteristics of the YHU-1B in level flight are a marked improvement over those of the HU-1A. At all weights and altitudes tested the helicopter was power limited rather than vibration limited. Vibration appears to change little with cg change. All vibration characteristics were evaluated qualitatively from pilot comments.

AIRSPPEED CALIBRATION

A boom airspeed system was installed for test purposes. This test system and the standard airspeed system were calibrated in level flight throughout the speed range to a maximum of 120 knots IAS. The calibration was accomplished by the ground speed course method. Results of this calibration are shown in Fig. 39, Appendix 1.

The calibration of the standard airspeed system shown in Fig. 39, Appendix I is not representative of the production HU-1B system. Prior to the last flight of the test program a second baffle was added at the standard system static source. The change is an attempt to produce a constant error. This new system will be evaluated in the Category II test program.

STABILITY AND CONTROL TEST RESULTS

HOVERING

The flying qualities while hovering are not entirely satisfactory; however, the YHU-1B is improved over the HU-1. The oscillations in pitch and roll, which were objectionable in the HU-1, are negligible, but random oscillation in yaw still exists which makes precision hovering in ground effect difficult. This yawing oscillation decreases at skid heights above 30 feet.

HOVERING DYNAMIC STABILITY

The dynamic stability characteristics were determined by 1 second pulse type control inputs about all axes. Prior to any control displacements, the aircraft was brought to a stabilized hover for several seconds.

Following a 1 inch forward longitudinal pulse the aircraft pitches down, moves forward, and then pitches up at translational lift. The resulting longitudinal oscillation is lightly damped with a period of approximately 5 seconds. The downward pitching is accompanied by left yawing and rolling which reverses when the aircraft pitches up. Following a 1 inch aft pulse, the aircraft pitches up, moves aft and pitches down. The aircraft rolls and yaws slightly right on the upward pitch and as the aircraft noses down, develops a right yaw of such magnitude that the maneuver must be discontinued.

Pedal pulses result in a nearly dead-beat oscillation in yaw. The accompanying roll oscillation is initially opposite

in direction to the pulse and heavily damped. Lateral pulses cause a divergent yawing in the direction of the pulse.

Time histories of these pulses are shown in Figs. 16 through 21, Appendix I.

All tests were performed at approximately 6600 pounds gross weight with a mid center of gravity at a rotor speed of 323 rpm.

HOVER CONTROLLABILITY

Control sensitivity during hover was determined by measuring the immediate maximum angular acceleration resulting from various step type control displacements from trim about all three axes. Control sensitivity about all axes is adequate. The following control sensitivities were obtained for a 1 inch control displacement from trim.

Axis	Sensitivity deg/sec ² /in	Time to Reach Maximum Acceleration - sec
Pitch	10	0.4
Roll	16	0.4
Yaw	40	0.4

No noticeable delay occurs between control movement and aircraft response. No "slop" or play is apparent in the control systems and control sensitivity is equal for control displacements in either direction. Results of tests at a mid center of gravity position and rotor speeds of 323 rpm are presented in Figs. 22 through 27, Appendix I.

Control response during hover was determined from various step type control inputs from trim about each axis. The resultant immediate maximum angular velocity was measured. Control response about all three axes is satisfactory. The following rates were measured:

Axis	Response deg/sec/in	Time to Reach Maximum Rate -sec
Pitch	9	2.0
Roll	8	0.9
Yaw	45	7 to 8

Longitudinal steps produce a pitch-up for aft and a pitch down for forward stick movements that continue until the maneuver is discontinued. An aft step is accompanied by a random yaw and roll that is initially to the right. A forward longitudinal step results in a yaw and roll that is also random but initially to the left.

Directional steps produce a yaw in the direction of the step that continues until recovery. The maximum rate is reached in 7 to 8 seconds. The directional control displacements also produce random oscillations in pitch and roll. With a lateral step the aircraft rolls in the direction of the displacement and a yaw slowly develops in the direction of the roll.

LEVEL FLIGHT

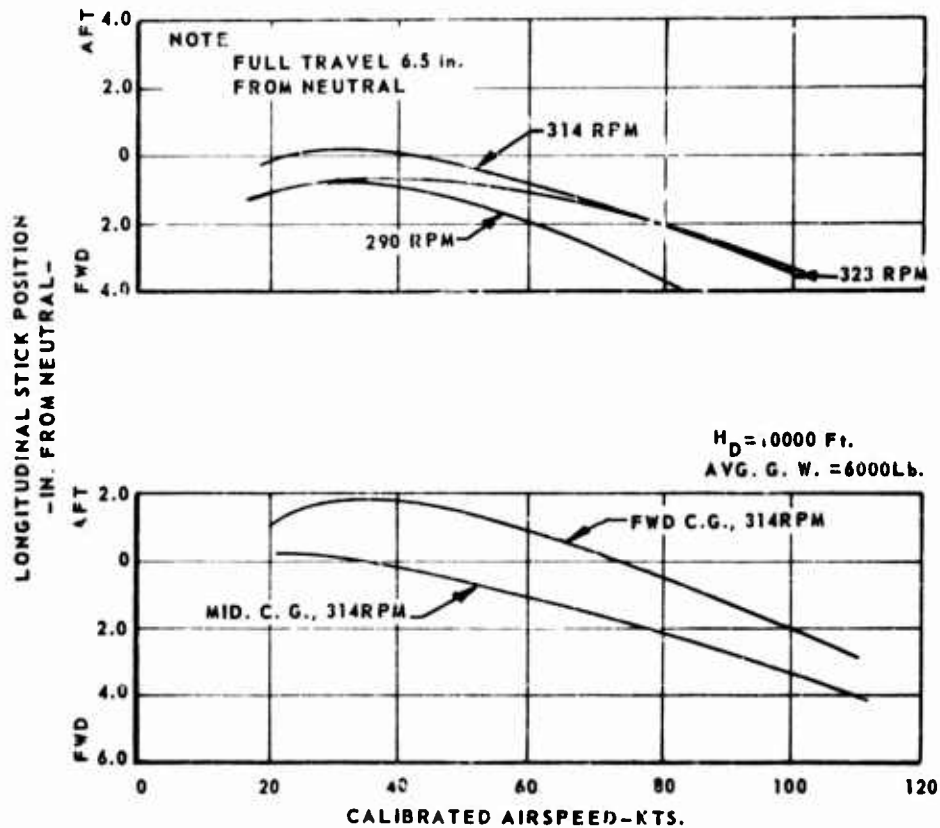
The level flight flying qualities are very good. Positive longitudinal stability, both static and dynamic, is demonstrated by the capability of the aircraft to fly hands off in light turbulence at speeds up to 100 knots CAS. The aircraft exhibits strong static and dynamic stability allowing pedal fixed turns to be easily accomplished. Roll rate is adequate even though the roll develops and then stops before continuing. This condition was also noted in the HU-1A.

Stability and control characteristics were determined for airspeeds from 34 to 104 knots CAS at average density altitudes of 5,000 and 10,000 feet. Gross weights for these flights were approximately 6000, 6600 and 7400 pounds at mid cg. Rotor speed was varied from 285 to 323 rpm. Longitudinal dynamic stability was tested qualitatively at forward and aft center of gravity locations. Prior to any control displacement the aircraft was stabilized in level flight at the trim airspeed with zero sideslip.

LONGITUDINAL CONTROL POSITION IN LEVEL FLIGHT

YHU-1B S/N 58-2078

$H_D = 9350$ Ft.
AVG. G. W. = 7400 Lb
AVG. C. G. = 130 in (MID)



STATIC STABILITY

Control positions during level flight were recorded as a function of airspeed at forward, mid and aft cg locations and at rotor speeds from 290 to 323 rpm. All cyclic control positions are measured from a position vertical to the floor. Full longitudinal cyclic travel is 6.5 inches forward and aft from vertical; full lateral travel is 6.5 inches left and right from vertical. Pedal travel is measured from neutral and full travel is 2.5 inches right and left from neutral. The apparent static longitudinal stability is positive above 30 knots CAS (Figs. 28 and 29, Appendix 1). At airspeeds below 30 knots

apparent static longitudinal appears to be slightly negative. A change of rotor speed from 290 to 314 rpm requires the cyclic to be moved aft 1 to 2 inches to maintain the same airspeed. The same is true of a change from 290 to 323 rpm for airspeeds from 50 to 90 knots. Lateral cyclic and pedal positions were normal at all speeds. An aft center of gravity location requires that the cyclic stick be moved approximately 2 inches farther right than for a forward cg at the same weight when the airspeed is less than 60 knots CAS. This lateral movement decreases to approximately

1 inch at 110 knots. Collective pitch control was found to be adequate at all rotor speeds.

The accompanying figure shows the effect of center of gravity location on longitudinal stick position at various airspeeds. The helicopter had adequate control margin during all Category I testing within the observed 120 knot IAS limit.

Static directional stability was investigated by obtaining the pedal position necessary to maintain various sideslip angles at several airspeeds. The same test was conducted using the contractor recommended climb speeds during both climb and descent. Results are presented in Figs. 30 through 34, Appendix I. The HU-1B has strong positive static directional stability. Good directional control effectiveness and light pedal forces assist the pilot in maneuvering the helicopter and in maintaining a heading.

LEVEL FLIGHT CONTROLLABILITY

Control sensitivity of the YHU-1B during level flight was determined by measuring the immediate angular acceleration resulting from various step type control displacements from trim about all three axes. These steps were performed at 34, 64 and 104 knots CAS at a rotor speed of 323 rpm. The directional steps for all speeds tested were repeated at rotor speeds of 285 rpm. Control sensitivity about all axes is satisfactory and approximately equal to those of the HU-1 except for slight decrease in pitch. The following sensitivities were obtained for a 1 inch displacement from trim.

Axis	Sensitivity deg/sec ² /in	Time to Reach Maximum Acceleration - sec
Pitch	10	0.4
Roll	28 right 26 left	0.4
Yaw	30	0.4

Results of these tests are presented in Figs. 22 through 27, Appendix I.

Control response of the YHU-1B was determined by measuring the immediate maximum angular velocity from the step type control inputs. Control response about all three axes is satisfactory. The HU-1 develops a slightly larger yaw rate, but the two aircraft have approximately the same response in pitch and roll. The following rates were measured:

Axis	Response deg/sec/in	Time to Reach Maximum Rate-sec
Pitch	9.0	1.9
Roll	13 left 16 right	1.2
Yaw, 323 rpm	11	1.1
285 rpm	11	1.1

Step inputs were made about all axes from stabilized level flight at airspeeds of 34, 64 and 104 knots CAS and rotor speeds of 323 rpm. Following a 1 inch forward longitudinal step, the aircraft pitches down with the pitch rate decreasing as airspeed increases. The pitching motion is accompanied by a gentle rolling and yawing to the left. An aft step results in opposite reactions about all axes. A 1 inch right lateral step produces a right roll followed by right yawing motion until recovery is executed. With a left lateral step a left roll develops, however, the yaw is initially slightly right and then becomes left until recovery from the maneuver.

Pedal steps were performed at airspeeds of 34, 64 and 104 knots CAS at rotor speeds of 285 and 323 rpm. At 34 knots a 1 inch directional step generates a hesitating turn in the direction of the step followed by an uneven rolling motion in the direction of the turn. This occurs at both 285 and 323 rpm.

A 1 inch left directional step at 64 knots CAS and a rotor speed of 285 rpm produces reactions similar to those that occur at 34 knots. A right directional step at 285 rpm produces a right yaw

that is stopped when a left roll develops. When the yaw is stopped the aircraft rolls right and the right yaw starts again.

A 1 inch left directional step at 104 knots CAS and a rotor speed of 323 rpm produces a small left turn and right roll. This right roll causes the left turn to stop and reverse to a right turn. A right step at 323 rotor rpm and steps in both directions at 285 rotor rpm produce a turn that becomes a steady state sideslip when sufficient roll has developed to stop the turn.

All directional steps cause a very slight nose down pitch.

LEVEL FLIGHT DYNAMIC STABILITY

Pulses to evaluate the dynamic stability were made from stabilized flight conditions at the same airspeeds and rotor speeds as the step inputs. Dynamic characteristics were determined to be good.

The short period oscillations excited by a longitudinal pulse are heavily damped. An aft pulse produces a heavily damped pitching motion initially up, accompanied by a well damped left rolling motion and a change in heading to the right until recovery is executed. A forward pulse produces the opposite reaction. During several of these maneuvers a long period oscillation, which was very lightly damped, was recorded. The period of this phugoid oscillation is approximately 27 seconds.

Directional pulses produce a heavily damped yawing motion initially in the direction of the pulse accompanied by a well damped rolling motion initially in the opposite direction of the yaw. The aircraft pitches up for a right yaw and down for a left yaw. These characteristics are summarized in Figs. 35 and 36, Appendix I.

A left lateral pulse produces a heavily damped initially left rolling motion and a turn to the left until recovery is effected. A right pulse produces a heavily damped

right rolling motion and a left turn that peaks in 5 seconds and then becomes a steady left sideslip.

SIDEWARD AND REARWARD FLIGHT

Tests were conducted to determine the hovering capabilities of the helicopter when operating close to the ground in crosswinds and tail winds. The sideward and rearward test flights were conducted at a gross weight of 7600 pounds, at the most forward cg (125 inches), and at a rotor speed of 323 rpm. These tests were flown in ground effect. Control positions obtained during these tests are presented in Figs. 37 and 38, Appendix I.

The YHU-1B moves easily into sideward flight with small pedal manipulations required below 10 knots TAS to maintain the desired heading. When translational lift occurs (10 to 20 knots), small, rapid pedal movements are required. Approaching translational lift oscillations in roll, yaw and pitch are encountered. When going to the left, above the speed for translation, there is a sudden requirement for a large amount of right rudder. For flight to the left 2.8 inches of right pedal are required at 30 knots, and 1.8 inches of left pedal are necessary for 30 knots to the right.

During sideward flight the cyclic stick moves aft and in the direction of flight laterally. Aft cyclic movements reach a maximum at approximately 27 knots TAS in either direction and then decrease slightly at 30 knots. Lateral cyclic stick position increases positively to approximately 27 knots at which point a slight reversal occurs. Collective pitch control is less than for a hover, with approximately the same amount necessary for similar speeds in either direction.

The HU-1B accelerates rearward nearly as easily as it does forward. There is a tendency to turn into the relative wind that requires rapid pedal movements to control. A nose down pitching moment at translational lift requires an increase of 1.4 inches of aft cyclic to control. As speed

increases longitudinal stick position moves from 2 inches aft of neutral at hover to a maximum of 1.3 inches or 10 percent from full aft at 30 knots TAS to the rear.

CONTROL FORCES AND TRIMMING

Control forces are satisfactory with the trim system on or off. Control forces are high with the control boost system off but no feedback from the rotor system is present. Boost-off control forces of the YHU-1B are slightly higher than those of the HU-1A but sufficient control is available to hover and land.

The trim system in the test aircraft is satisfactory at speeds from hover to 100 knots IAS. In this regime the centering device quickly removes all forces and the aircraft can be flown hands off for short periods. In light turbulence, however, at speeds greater than 100 knots the cyclic stick falls forward unless an excessive amount of friction is applied. When the needed friction is used to hold the stick, control forces become large. The trim system should be redesigned to give the same hands off capability at speeds of 120 knots or greater since this is the most efficient cruise speed for the helicopter at sea level. (B 2)

CONCLUSIONS

The YHU-1B helicopter is much improved over the earlier HU-1 and HU-1A series. The major improvements are:

1. Lower vibration levels.
2. Increase cruise speed and range.
3. Greater altitude performance.
4. Increased weight carrying capability.

Flying qualities of the YHU-1B are improved over the earlier models. This is primarily due to the absence of the objectionable pitch and roll oscillations which were present in the HU-1. Control sensitivities and response are approximately equal; however, in the HU-1 pitch sensitivity and yaw rate is slightly greater.

The HU-1B meets guarantees of range, cruise speed, and service ceiling. The hovering capabilities are good and meet guarantees; however, hovering ceiling in ground effect is reduced approximately 700 feet by a 10 degree decrease in bellmouth temperature over that occurring at the same conditions out of ground

effect. This temperature increase may be caused by hot exhaust gases recirculating through the engine.

The contractor has made no improvement in engine droop compensation. The excessive engine droop, aggravated by a slow power turbine (N₂) governor actuator rate, makes precision rotor rpm control nearly impossible. The N₂ actuator, which takes approximately 15 seconds to change rotor speed from 285 rpm to 323 rpm, cannot correct rapidly enough following large, rapid collective pitch movements.

The engine flight idle stop release system is unsatisfactory. This complaint was also made in YH-40 report AFFTC-TR-59-33 and HU-1 report AFFTC-TR-60-57. It is possible to jam the stop by retarding the throttle prior to actuating the electrical system which removes the stop. Throttle travel is excessive by HIAD standards, (HIAD J. 2. 2. 6. 2) and tends to make jamming the stop more probable. Furthermore a complete electrical failure would make engine shutdown impossible since no mechanical means of engine fuel shutoff is provided.

RECOMMENDATIONS

A. It is recommended that the following be action be taken as soon as possible:

1. Install a positive mechanical flight idle stop that can be actuated by the pilot without removing his hand from the throttle (page 3)
2. Limit the throttle twist grip rotation so that the pilot can rotate the throttle from full off to full open with one normal movement of his hand (page 3)
3. Raise the collective pitch stick (page 3)
4. Improve the droop characteristics to maintain ± 2 rotor rpm variation from the selected governed speed throughout the governing range under all flight conditions (page 8)
5. Reduce the time to change rotor speed from 285 rpm to 323 rpm to approximately 5 seconds (page 8)
6. Move the a.c. circuit breakers to the overhead panel and illuminate all circuit breakers (page 4)
7. Make the pilot's and co-pilot's door handles easier to operate (page 3)

B. It is recommended that studies be initiated as soon as possible to accomplish the following items:

1. Increase fuel capacity to make instrument flights more feasible (page 7)
2. Redesign the trim system so hands off flying capabilities are available at 120 knots IAS (page 4)
3. Make the cargo doors jettisonable (page 4)
4. Decrease the engine inlet temperature while hovering in ground effect (page 5)

APPENDIX I

data analysis methods

PERFORMANCE

General:

The equations and procedures used to correct the performance of this helicopter from test conditions to US standard atmosphere conditions are briefly described in this section.

Dimensional analysis of the major items affecting helicopter performance will yield two sets of dimensionless variables which may be used to present performance data in non-dimensional form. The C_p , C_T , method is used in this report. It should be noted that this non-dimensional method is useful only where compressibility effects are not significant. These variables are defined as follows:

$$C_p = \frac{SHP \times 550}{\rho A (\Omega R)^3}$$

$$C_T = \frac{W}{\rho A (\Omega R)^2}$$

$$\mu = \frac{V_T}{\Omega R}$$

where:

SHP = output shaft horsepower

ρ = air density - slugs/ft³

A = rotor disc area - ft²

Ω = rotor angular velocity - rad/sec

R = rotor radius - ft

W = gross weight - lb

V_T = true airspeed - ft/sec

Hovering:

The hovering data was obtained in tethered flight at two heights, one in and out of ground effect, at a pressure altitude of 2300 feet. Hovering was conducted in zero wind conditions. This data was reduced to C_p , C_T and is presented in Figure 1. The weight used to compute C_T consisted of the weight of the helicopter and tethering components plus the force (pounds) applied to the tie-down cable.

Level Flight:

The basis for correction of level flight speed power data lies in the C_p , C_T method. Each speed power was flown at an approximate C_T . This involves increasing altitude as fuel is burned. The data was corrected for C_p to an exact, constant C_T as follows:

$$C_{p_s} = C_{p_t} + \frac{\Delta C_p}{\Delta C_T} (C_{T_s} - C_{T_t})$$

where $\Delta C_p / \Delta C_T$ is the slope of the C_p versus C_T curve at constant μ and the subscripts s and t denote standard and test conditions, respectively. Shaft horsepower standard was then calculated using a standard rotor speed.

The non-dimensional parameters C_p , C_T and μ are used for correlation of the level flight data.

For each flight airspeed and power required are reduced to non-dimensional form and a plot is made of C_p versus C_T at the average C_T flown. A curve is faired through the points and faired line values are used to construct a carpet plot of C_p versus C_T . On this plot

lines of constant μ are then faired through the various test curves, thus defining power required for any altitude, gross weight, airspeed and rotor rpm. Non-dimensional summary plots are prepared from this carpet plot. These plots, Figs. 2, 3 and 4, and SHP versus V_T are presented in Appendix I.

Fuel flow data was reduced to fuel flow per SHP at various altitudes. These values were corrected to sea level conditions at the compressor inlet and are presented in the $SHP/\delta\sqrt{\theta}$ versus $W_f/\delta\sqrt{\theta}$ curve, Fig. 13.

Where:

SHP = output shaft horsepower

W_f = fuel flow - lb-fuel/hour

δ = ratio of test bellmouth inlet pressure to standard sea level pressure

θ = ratio of test bellmouth inlet temperature to standard ambient temperature at sea level

$SHP/\delta\sqrt{\theta}$, $W_f/\delta\sqrt{\theta}$, and $N_1/\sqrt{\theta}$ are designated as "Referred" shaft horsepower, "Referred" fuel rate, and "Referred" fuel rate, and "Referred" gas producer rpm in this report.

Power Determination:

The T53 gas turbine engine incorporates a hydro-mechanical torque meter as an integral part of the reduction gearing on the compressor end of the engine. This torque meter is essentially a piston which supplies pressure, in proportion to the output torque, on the contained hydraulic oil. This oil pressure is normally indicated on the pilot's panel and is used as an indication of engine torque. To obtain a more accurate indication of torque, the pressure of the oil vapor behind the piston is also measured and the difference between this pressure and the hydraulic oil pressure is found. The engine manu-

facturer calibrates the oil pressure and oil vapor pressure as a function of output shaft torque during the test cell qualification of each engine. Engine power output and fuel consumption characteristics are also determined during test cell operation. For the test engine these characteristics are presented in Figs. 1 through 3, Appendix II. The tests by the engine manufacturer are conducted using ideal intake and exhaust ducts. Consequently, compressor inlet conditions are considered equal to ambient conditions.

The equation from which output shaft horsepower was determined from in-flight torque meter readings was derived as follows:

$$SHP = \frac{2\pi}{12 \times 3300} \times N_e T$$

where:

SHP = output shaft horsepower

N_e = output shaft rotational speed - rpm

T = output shaft torque - in-lbs

The torque meter calibration as presented in Figure 1, Appendix II indicates that torque is the following function of torque pressure:

$$T = 236.5 \Delta P$$

where

ΔP = torque meter pressure minus inlet housing pressure - psi

Rotor speed is determined from engine output shaft speed as follows:

$$N_r = N_e/20.37$$

where

N_r = rotor rotational speed - rpm

Substituting the last two equations in the first, an equation for determining output shaft horsepower may be developed:

$$\text{SHP} = \frac{2\pi \times 236.5 \times 20.37}{12 \times 33000}$$

$$X N_T \Delta P = .0764 N_T \Delta P$$

During the test program, torque pressure from which SHP was calculated, was measured by taking the difference between the hydraulic oil pressure (high torque) and the oil vapor pressure (low torque). Engine characteristics were defined by curves of:

$$\frac{\text{SHP}}{\delta \sqrt{\theta}} \text{ vs } \frac{N_1}{\sqrt{\theta}}, \quad W_f / \delta \sqrt{\theta} \text{ vs } \frac{\text{SHP}}{\delta \sqrt{\theta}}$$

where

SHP = output shaft horsepower as calculated from torque pressure

N_1 = gas producer speed - percent

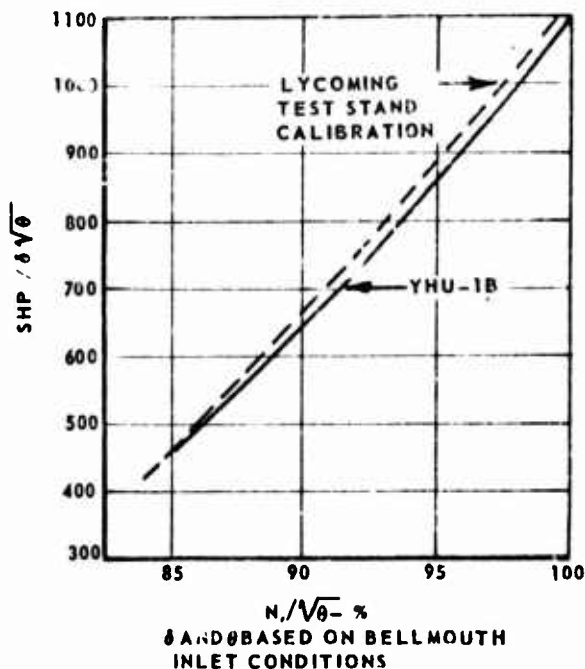
δ = ratio of test bellmouth inlet pressure to standard pressure at sea level

θ = ratio of test bellmouth inlet temperature to standard temperature at sea level

The accompanying figure shows a discrepancy between the characteristics of the test engine as determined by the manufacturer during the test stand calibration runs and those determined by flight tests.

ENGINE CHARACTERISTICS

T-53-L5 S/N LE03007



Test instrumentation was rechecked and found to be operating properly. Similar discrepancies have been found in previous tests (Reference AFFTC-TR-56-15 and AFFTC-TR-59-33).

Flight test data indicates that maximum power is available at 100 percent N_1 rather than 99.6 percent N_1 found by the engine manufacture at sea level standard day conditions. Limit N_1 for this engine is 100 percent. Maximum power available, Figure 15 of this Appendix, was calculated using flight test data.

STABILITY AND CONTROL

Definitions:

The stability and control characteristics of the YH-1B helicopter are discussed in terms of static stability, dynamic stability and controllability. These terms are defined as follows:

1. Longitudinal static stability is the apparent stability determined from an analysis of longitudinal control position with respect to airspeed. The collective position was treated as an independent variable. For each test point the collective stick position was determined by the position normally used in flight. A longitudinal control position-airspeed gradient obtained in this manner determines apparent static stability. The stability is called apparent because it is an indication of the longitudinal static stability from the pilot's viewpoint, but is not a direct measure of the speed stability or angle of attack stability of the aircraft. Static lateral directional stability was obtained by measuring control positions in steady-state side-slips.

2. The dynamic stability of the helicopter was determined by recording aircraft behavior, displacement, rate and angular acceleration following an artificial disturbance. This artificial disturbance was the result of a pulse type control input. The pulse input was made by rapidly displacing the control approximately 1 inch from trim position, holding for approximately 1 second, then rapidly returning to the trim position and holding the control fixed. A mechanical jig was used to guarantee precise input. The dynamic behavior of the aircraft in hover is presented by time histories (Figs. 15 through 20). The parameters presented are indicated values

traced from the oscillograph records. The longitudinal and directional dynamic stability data was reduced to damping ratios and period. The oscillations following a pulse input are heavily damped; therefore the method of counting cycles for the initial amplitude to damp to some fraction was not used to determine damping ratio. The time rise method was used to determine an approximate damping ratio. In this method the damping ratio (ξ) is found by the relationship of T_2/T_1 to ξ where T_1 is the time for the response to reach 20 percent of the steady state value and T_2 is the time to reach 74 percent of the steady state value. The accuracy of this method depends on how well the beginning of the response can be identified. The periods were determined from the following relationship:

$$T_n = T_d \sqrt{1 - \xi^2}$$

where

T_n is the undamped natural frequency

T_d is the damped natural frequency

3. Controllability is treated in two parts, namely sensitivity and response. Sensitivity is defined as the maximum angular acceleration (degrees per second²) of the aircraft per inch deflection of the cockpit control. Time to reach the maximum acceleration is included. Response is defined as the maximum angular velocity (degrees per second) of the aircraft per inch deflection of the cockpit control. Time to reach the maximum rate is included. The control deflections were stick fixed, sudden, step type inputs. The step input was made by rapidly displacing the control from trim and holding the control fixed until recovery was necessary. A mechanical jig was used to insure precise inputs.

GRAPHIC TEST RESULTS

Performance Plots

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Stability and Control Plots

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28	Control Positions	48
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36	Dynamic Directional Stability	52
37	Control Positions in Sideward Flight	53
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39	Airspeed Calibration	55

FIG. No 1
SUMMARY HOVERING PERFORMANCE
YHU-1B
323 ROTON RPM
STANDARD DAY
OGE - OUT-OF-GROUND EFFECT

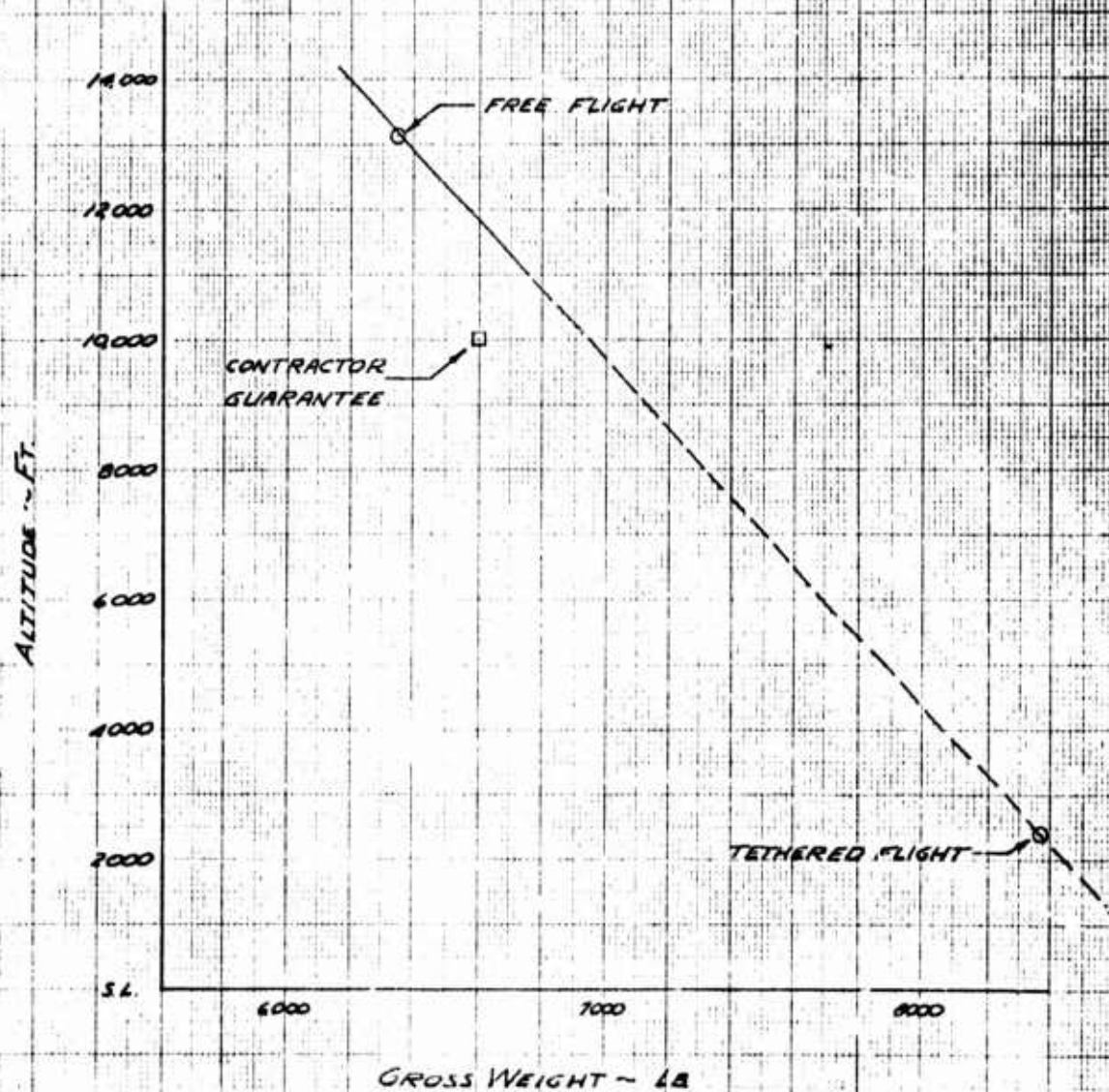


FIG No. 2
NON-DIMENSIONAL HOVERING PERFORMANCE
YHU-1B
TETHERED FLIGHT
S/N 58-2078

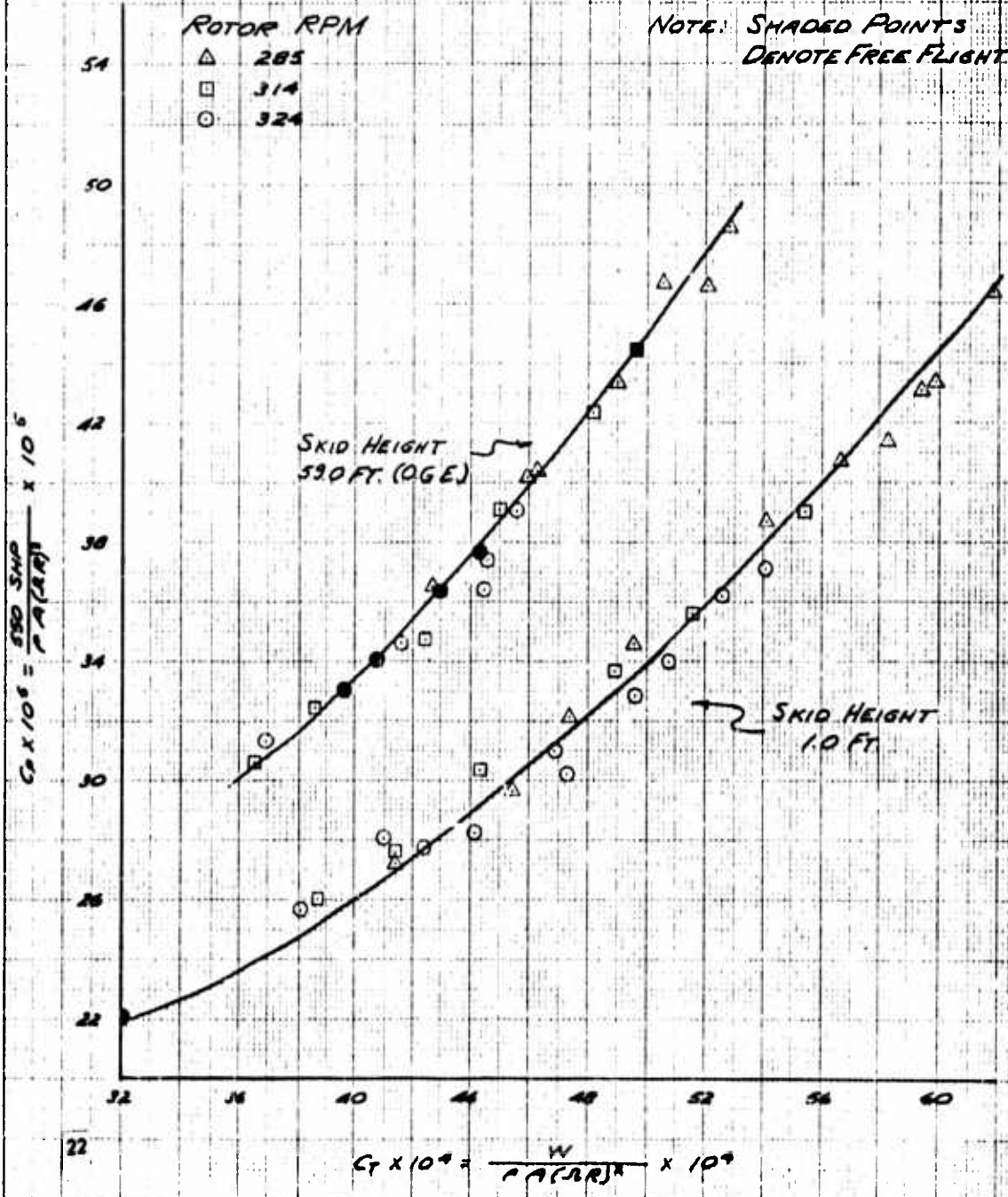


FIGURE No 3
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
YNU-1B

S/N 58-2078

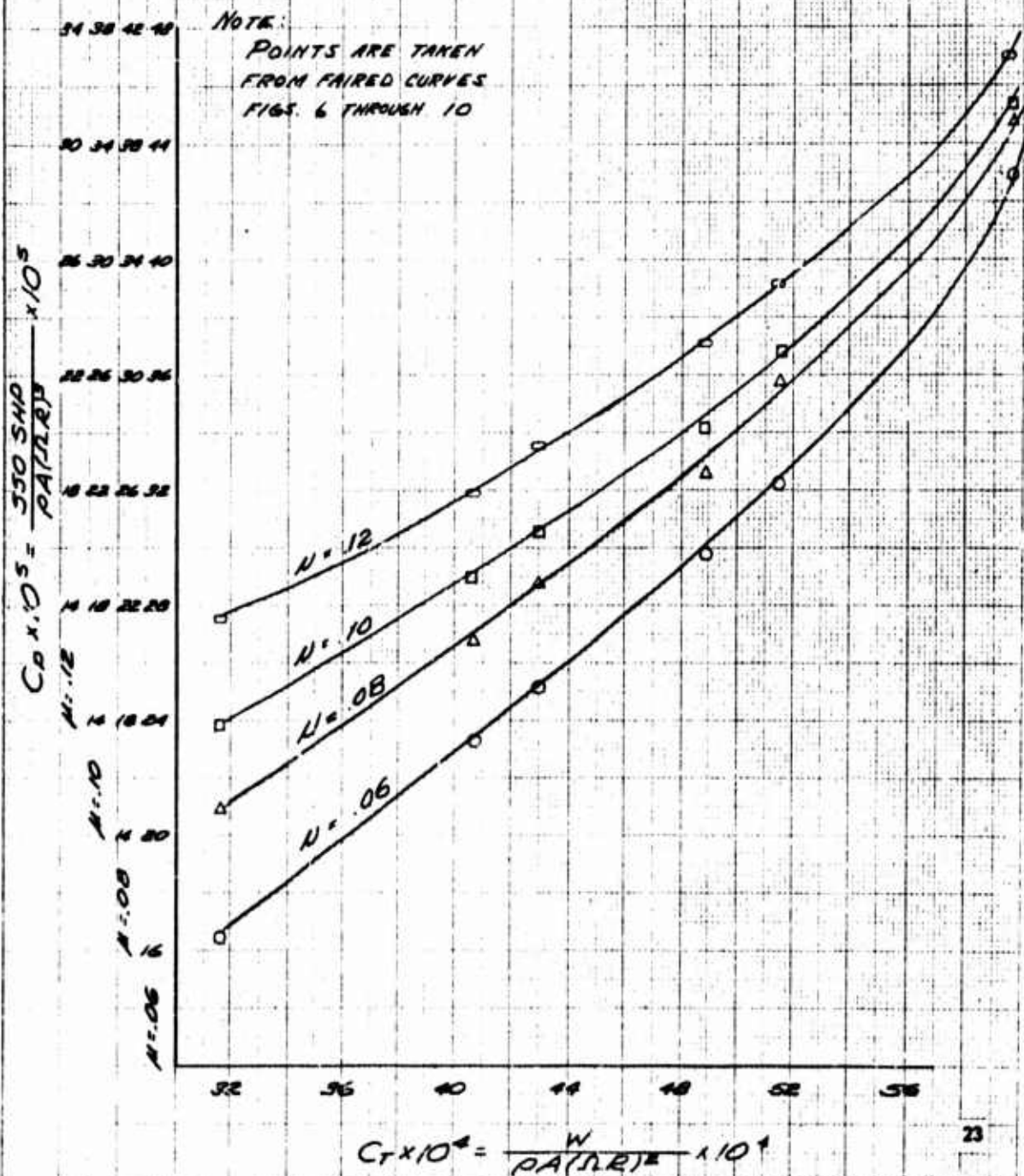


FIGURE No 4
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
YHU-1B
S/N 68-2078

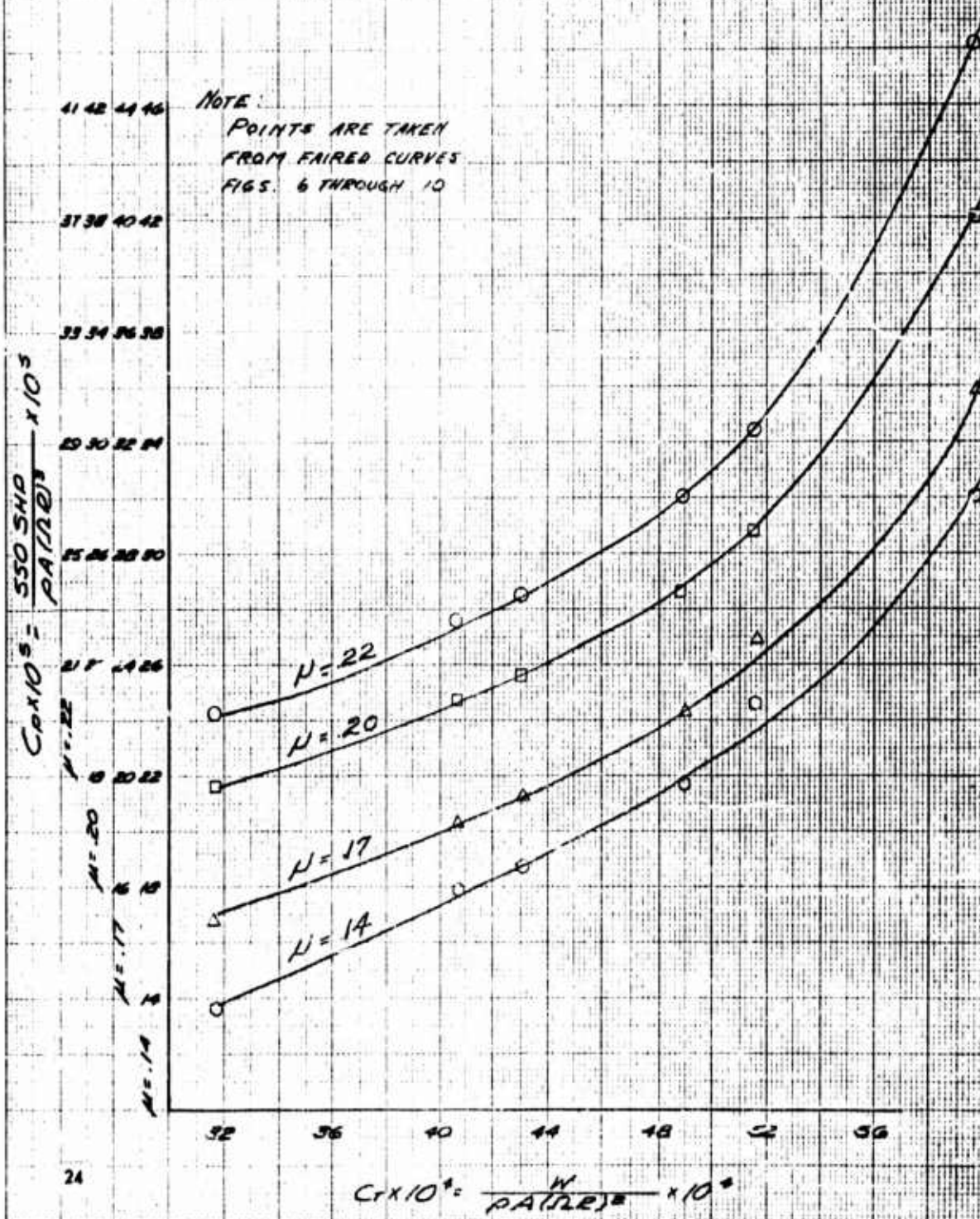


FIGURE NO. 5
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
YHU-1B
S/N 58-2078

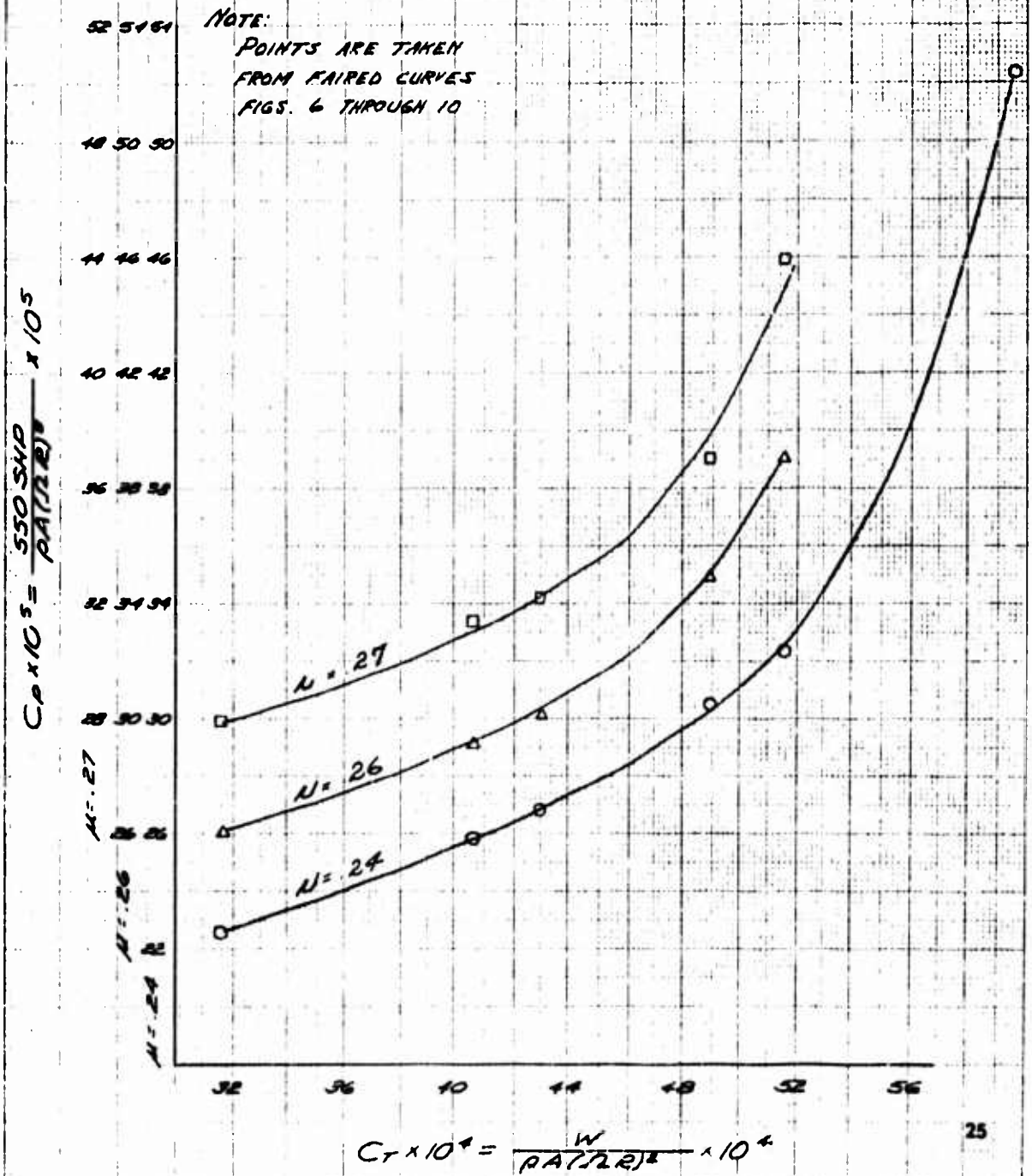


FIG. NO. 6
LEVEL FLIGHT PERFORMANCE
 YHU-1B
 GROSS WEIGHT 3600 LB.
 DENSITY ALT. 4000 FT.
 ROTOR RPM 323
 CT 0.00316

M.O. C.G.

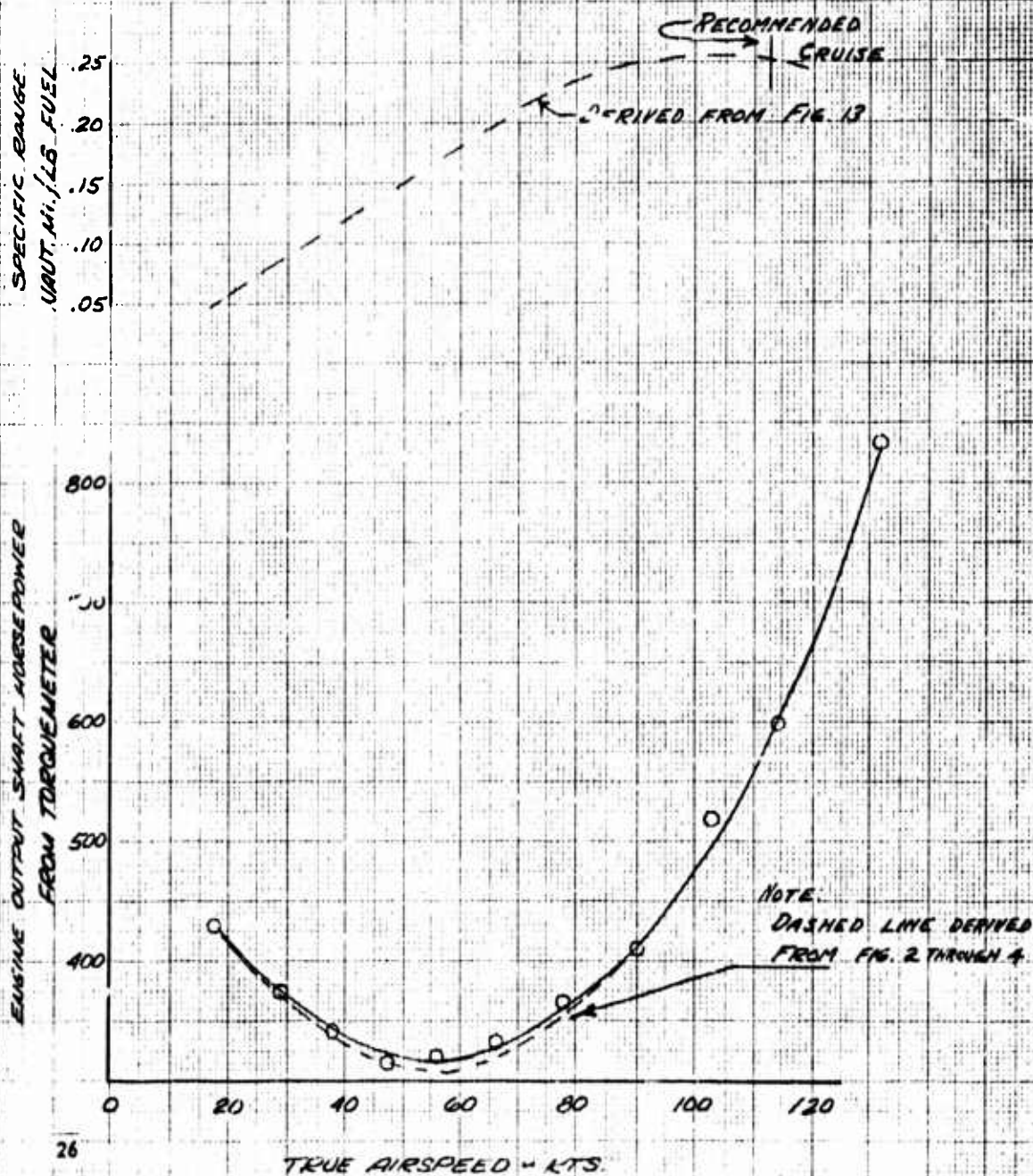


FIG. NO. 7
LEVEL FLIGHT PERFORMANCE

VHU-1B
GROSS WEIGHT
DENSITY ALT.
ROTOR RPM
CT

S/N 58-2078
6000 LB.
10000 FT.
323
0.00407

AFT C.G.

RECOMMENDED CRUISE

DERIVED FROM FIG. 13

SPECIFIC RANGE
NAUT. MI./LB FUEL

ENGINE OUTPUT SHAFT HORSEPOWER
FROM TORQUEMETER

.25
.20
.15
.10
.05

800
700
600
500
400

0 20 40 60 80 100 120

TRUE AIRSPEED "KTS.

NOTE:
DASHED LINE DERIVED
FROM FIGS 2. THROUGH 6.

FIG. NO. 8

LEVEL FLIGHT PERFORMANCE

VHU-1B

S/N 58-2078

GROSS WEIGHT

6000 LB

DENSITY ALT

10000 FT

ROTOR RPM

314

C_T

0.00430

○ FWD C.G.

△ AFT C.G.

RECOMMENDED
CRUISE

DERIVED FROM FIG. 13

SPECIFIC RANGE
NAUT. MI./LB. FUEL

.25
.20
.15
.10
.05

ENGINE OUTPUT SHAFT HORSEPOWER
FROM TORQUEMETER

800
700
600
500
400

0 20 40 60 80 100 120

28

TRUE AIRSPEED - KTS.

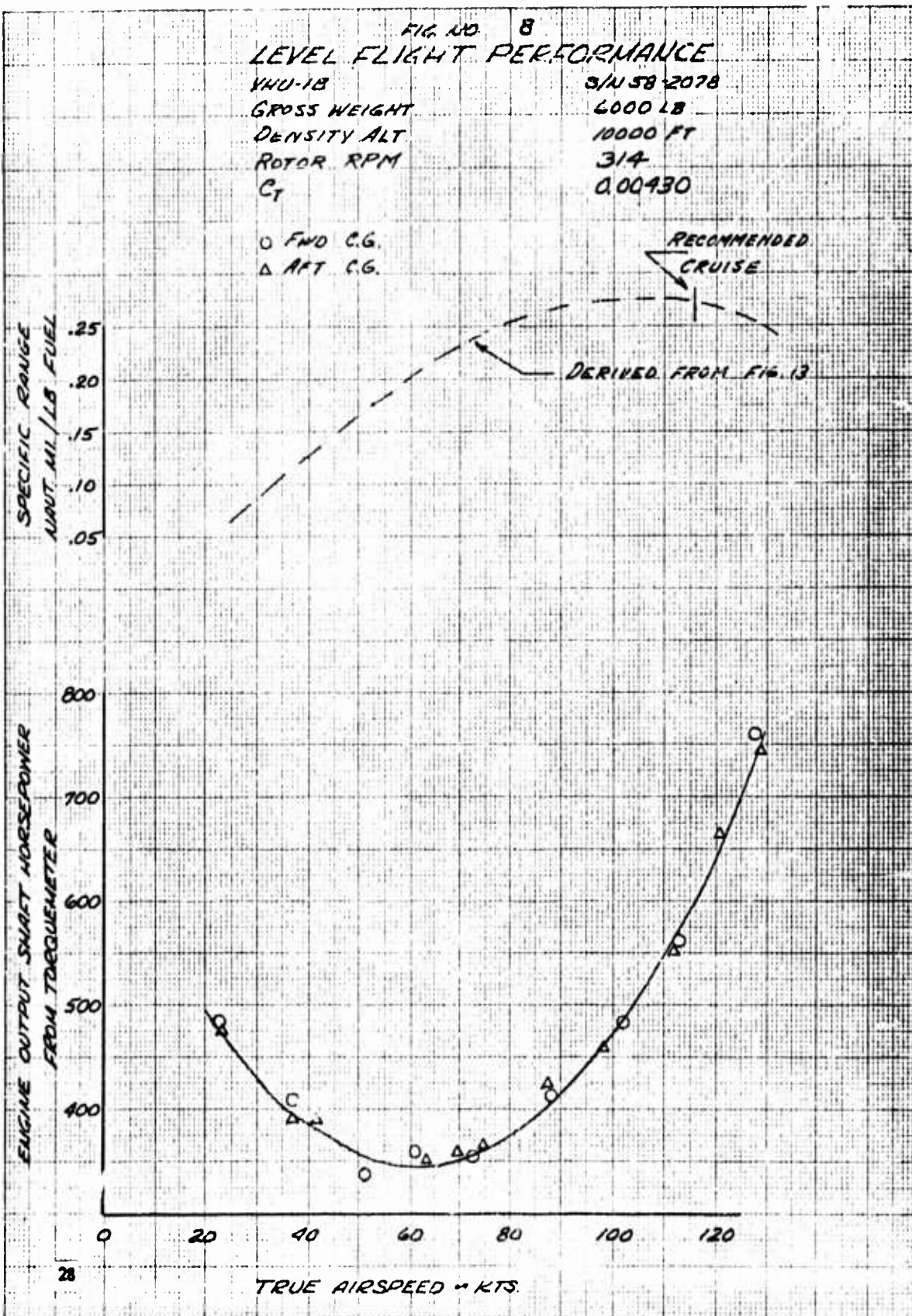


FIG. NO. 9
LEVEL FLIGHT PERFORMANCE

YHU-1B
GROSS WEIGHT
DENSITY ALT
ROTOR RPM
Ct

S/N 58-2078
7300 LB
10000 FT
323
0.00489

MID C.G.

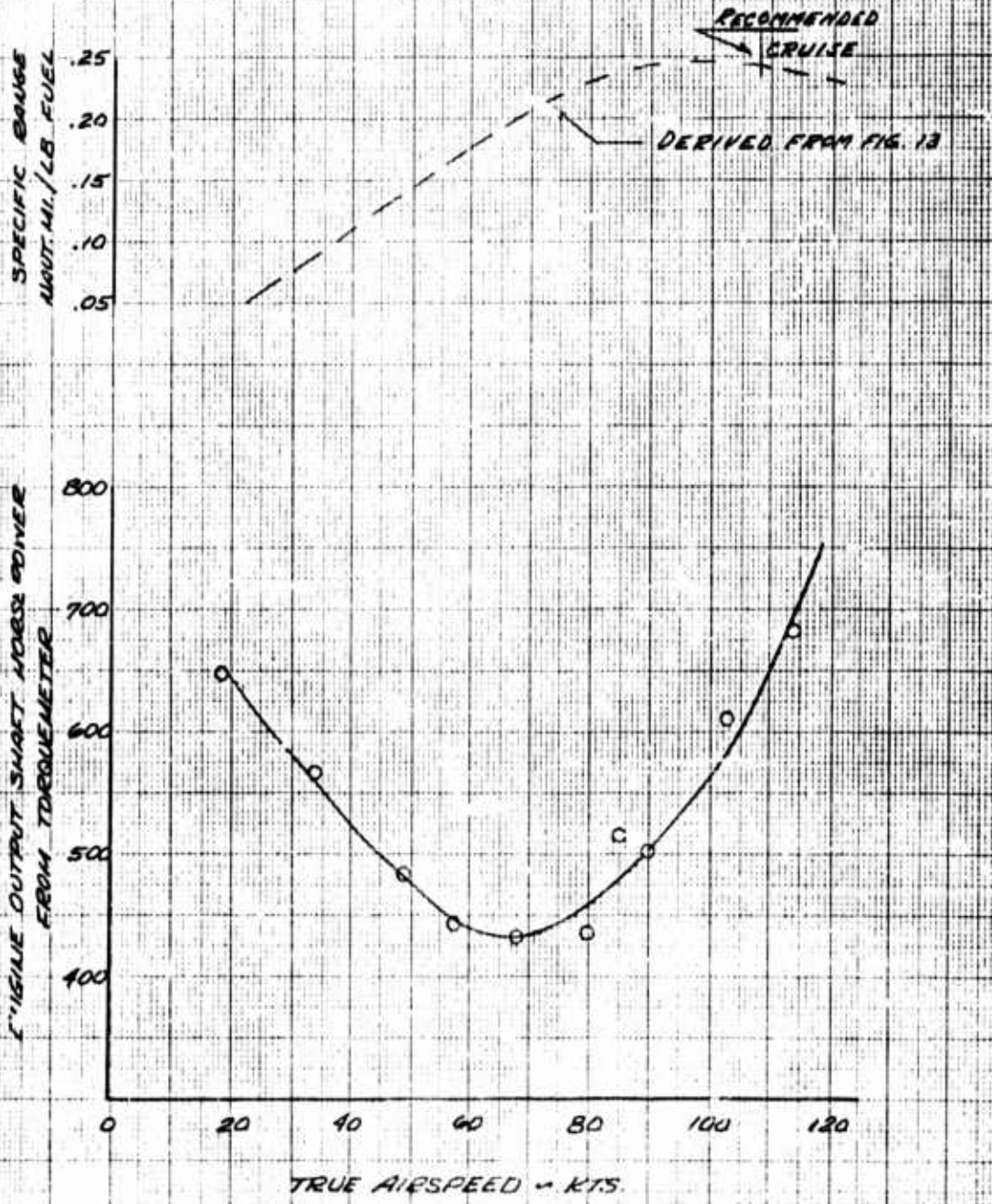


FIG. NO. 10
LEVEL FLIGHT PERFORMANCE
 YNU-1B
 GROSS WEIGHT 7200 LB
 DENSITY ALT 10000 FT.
 ROTOR RPM 314
 C_T 0.00515

MID C.G.

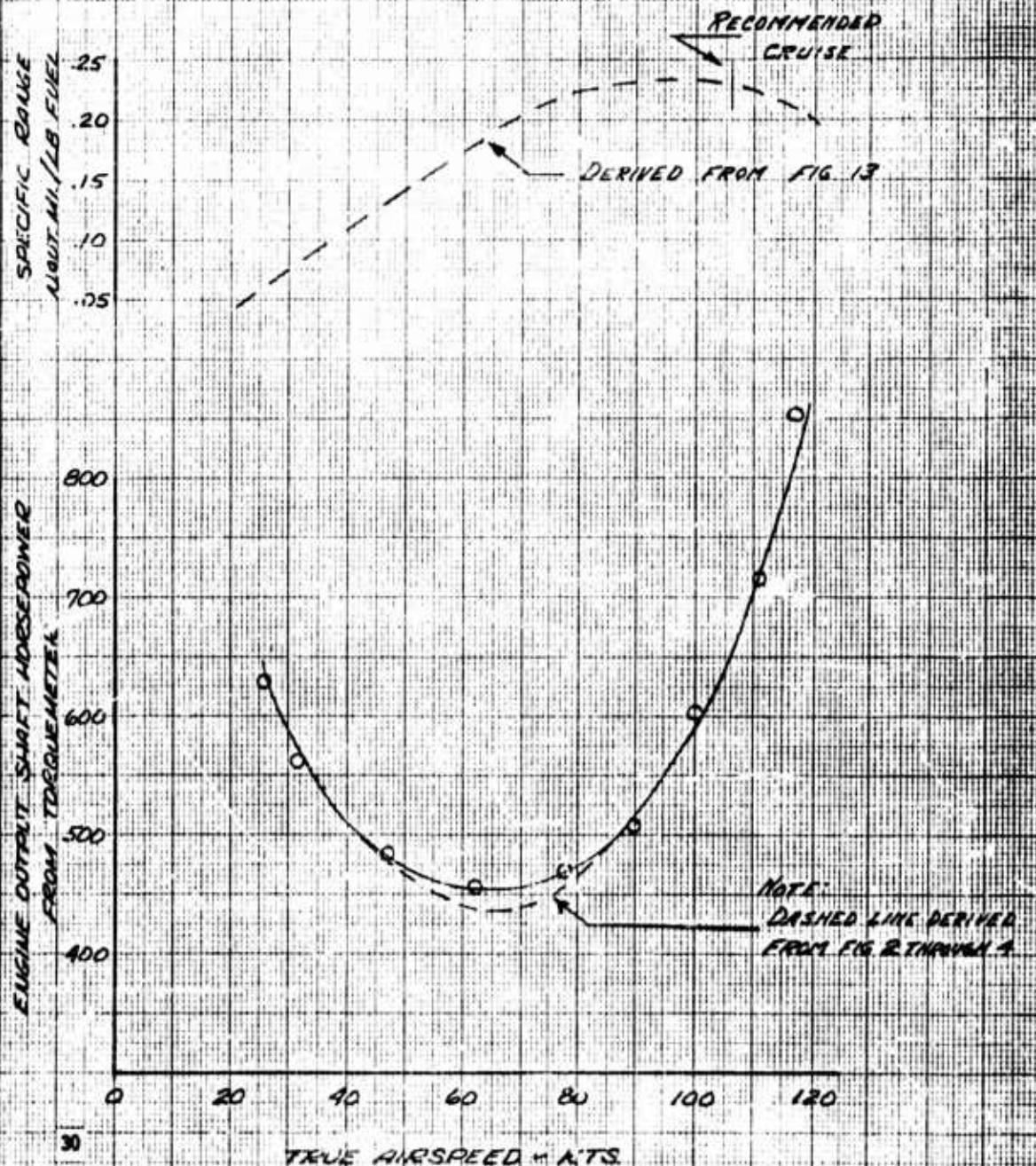


FIG. NO. 11
LEVEL FLIGHT PERFORMANCE
 YHU-1B
 GROSS WEIGHT 7200 LB
 DENSITY ALT 9500 FT.
 ROTOR RPM 290
 CT 0.00596

MID. C.G.

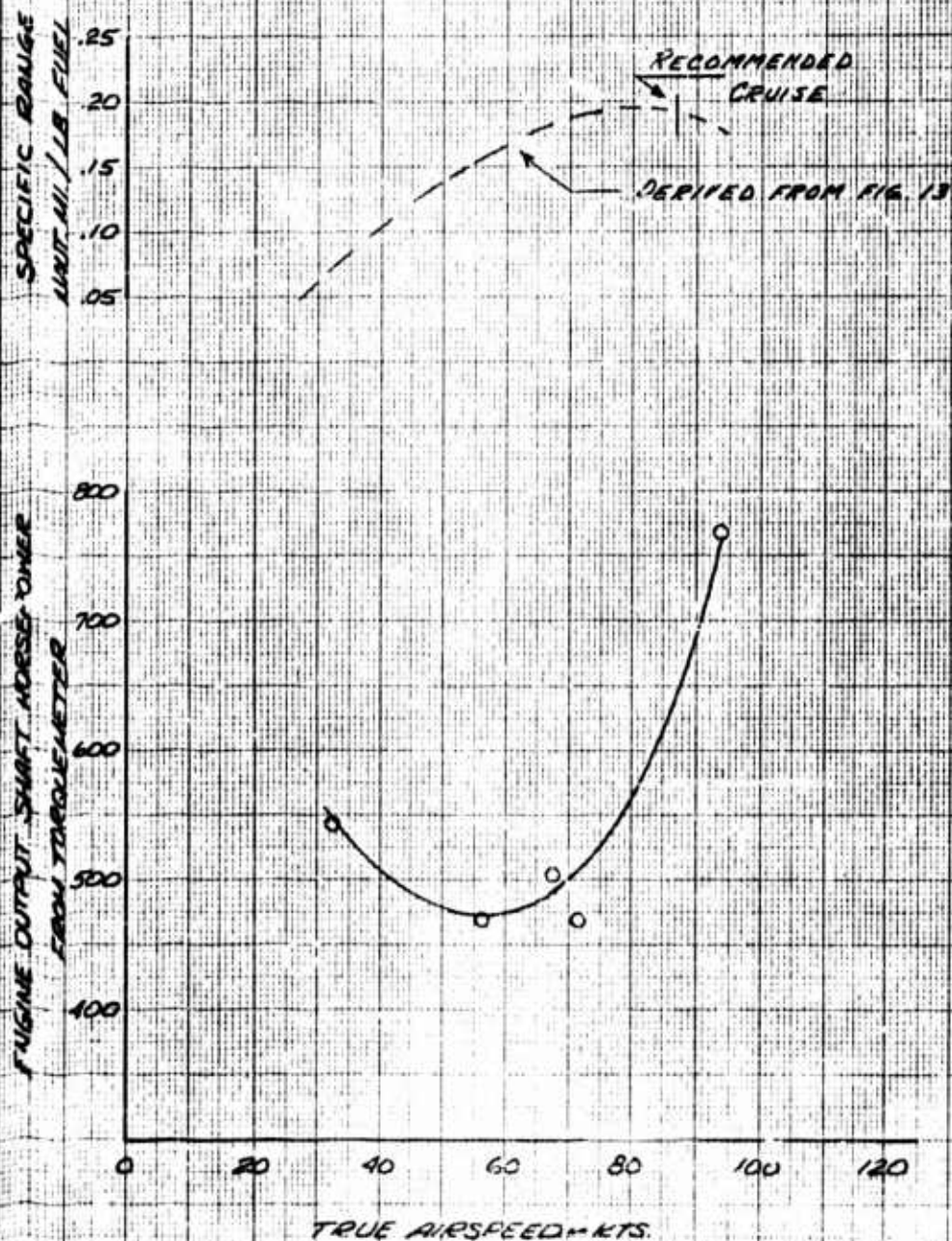


FIG. NO. 12 LEVEL FLIGHT PERFORMANCE

VHUB-3
GROSS WEIGHT
DENSITY ALT
ROTOR RPM
Ct

S/N 58-2078
6600 LB
14000 FT.
296
0.00604

MID C.G.

SPECIFIC RANGE
NAUT. MI. / LB. FUEL

.25
.20
.15
.10
.05

RECOMMENDED
CRUISE

DERIVED FROM FIG. 13

ENGINE OUTPUT SHORT-NOISE POWER
FROM TORQUEMETER

800
700
600
500
400

0 20 40 60 80 100 120

TRUE AIRSPEED - KTS.

FIG. NO. 13
ENGINE CHARACTERISTICS
 YHU-1B S/N 58-2078
 ENGINE T53-L5 S/N LEO3027

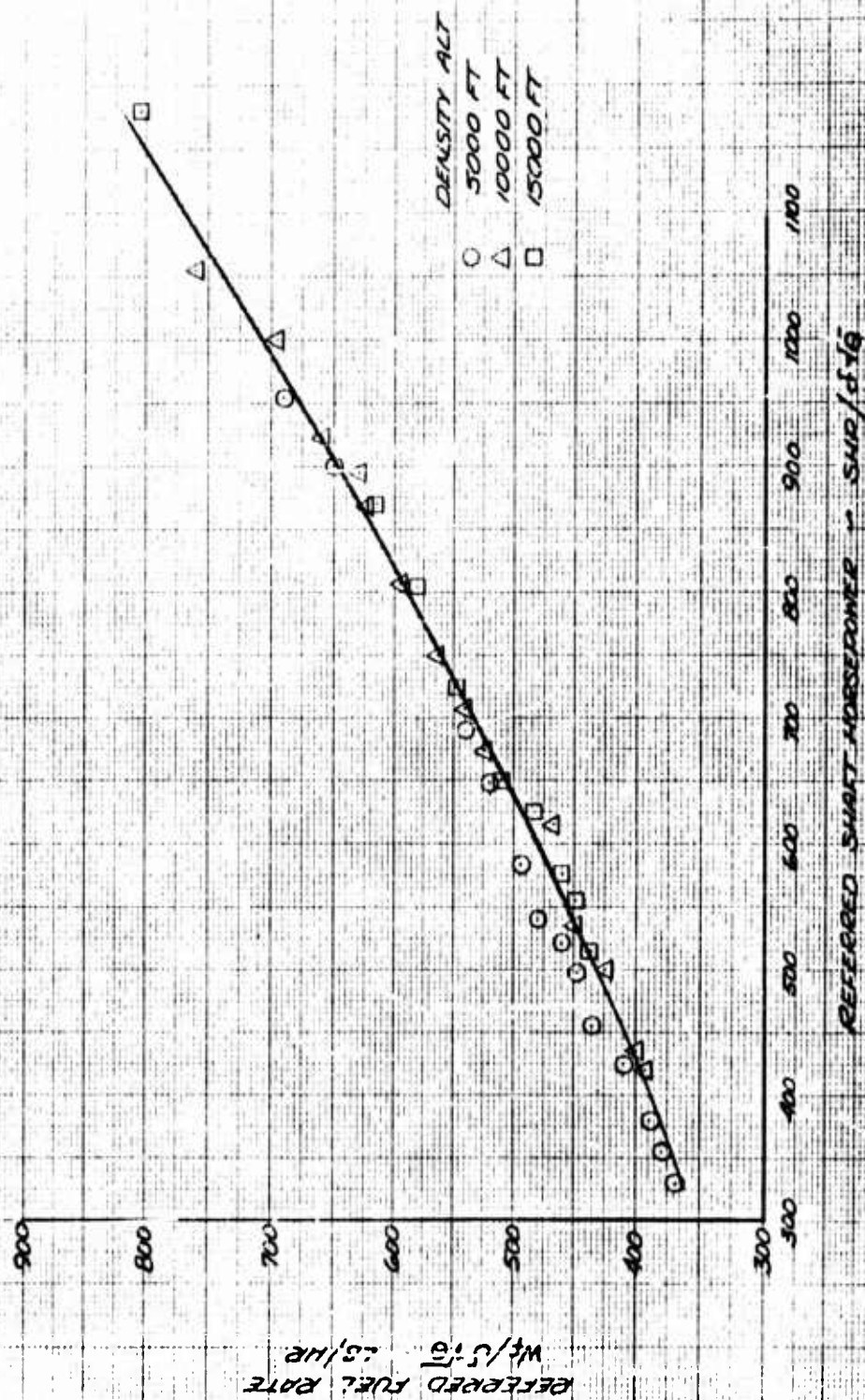


FIGURE NO. 14
ENGINE CHARACTERISTICS
YHU-1B S/N 58-207A
T-53-L5 S/N 1603007

NOTE:

1. $\sqrt{S}$ BASED ON BELLMOUTH INLET AIR TEMPERATURE
2. S BASED ON STATIC BELLMOUTH INLET AIR PRESSURE
3. SHP BASED ON LYCOMING DIFFERENTIAL TORQUE PRESSURE CALIBRATION.
TORQUE = 236.5 ΔP
100% = 25150 RPM

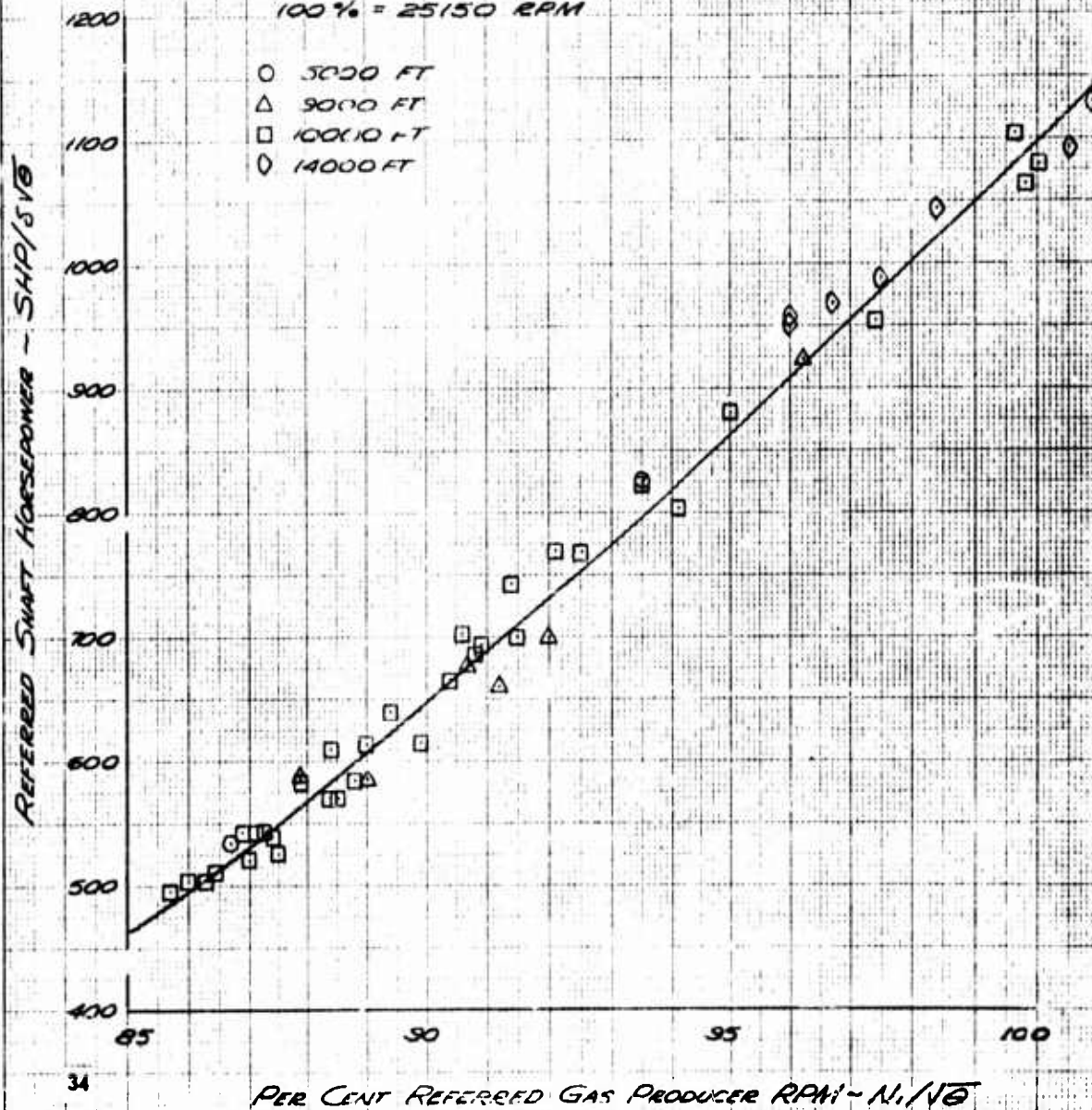


FIG. NO. 15
SHAFT HORSEPOWER AVAILABLE
 YHU-1B S/N 58-2078
 ENGINE T53-L5 S/N LE03007

$N_1 = 25150 \text{ RPM (100\%)}$
 AT 15°C COMPRESSOR INLET
 TEMPERATURE

NOTE.

- 1) COMPRESSOR INLET TEMPERATURE ASSUMED = AMBIENT $+2^\circ\text{C}$ O.S.B.
- 2) LIMIT N_1 VARIES WITH TEMP PER LYCOMING BIAS CURVE
- 3) SHP DETERMINED FROM CURVE OF $\text{SHP}/\sqrt{N_1}$ VS $N_1/\sqrt{N_1}$ AS ESTABLISHED FROM TORQUEMETER READINGS TAKEN IN FLIGHT.

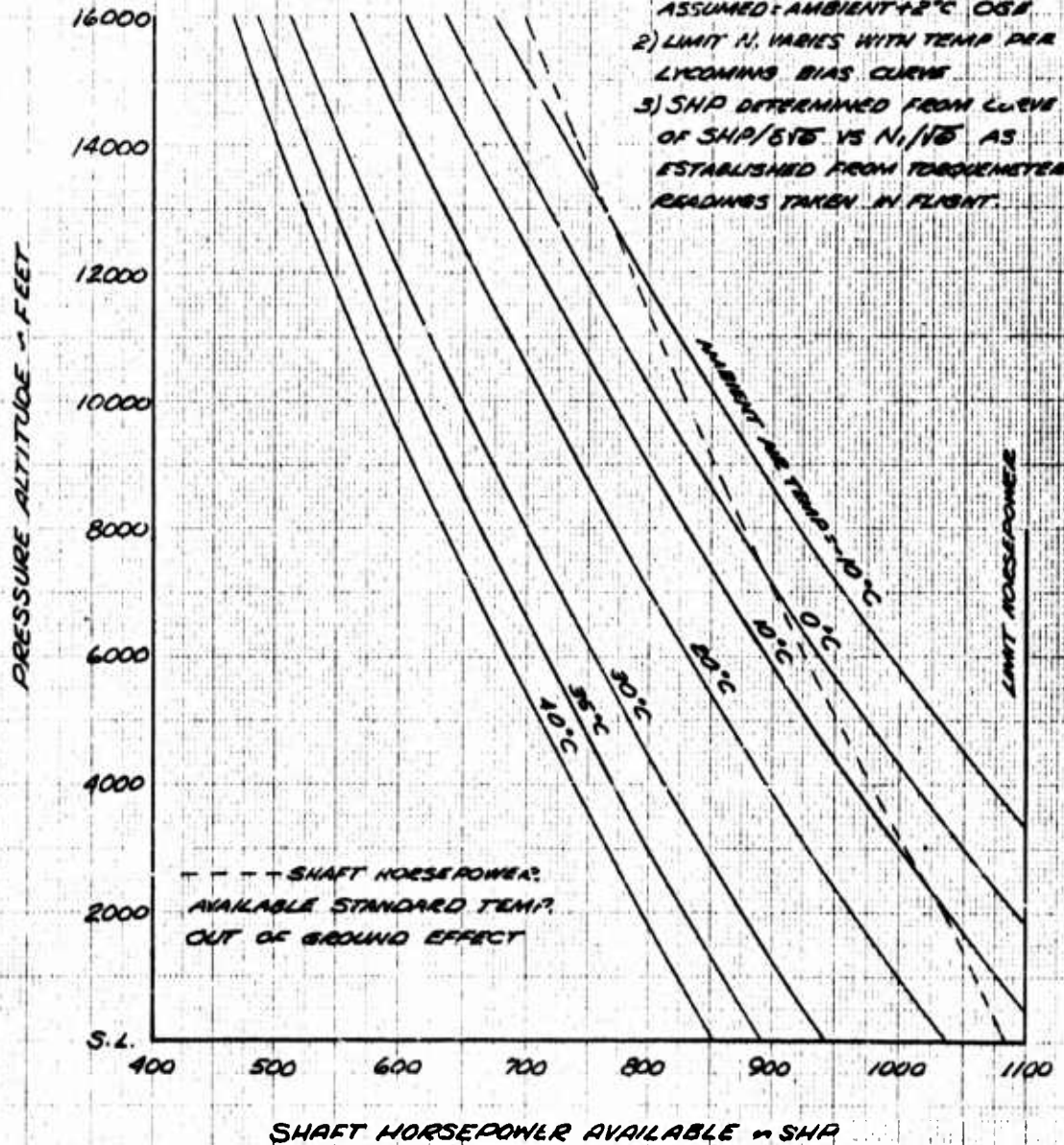
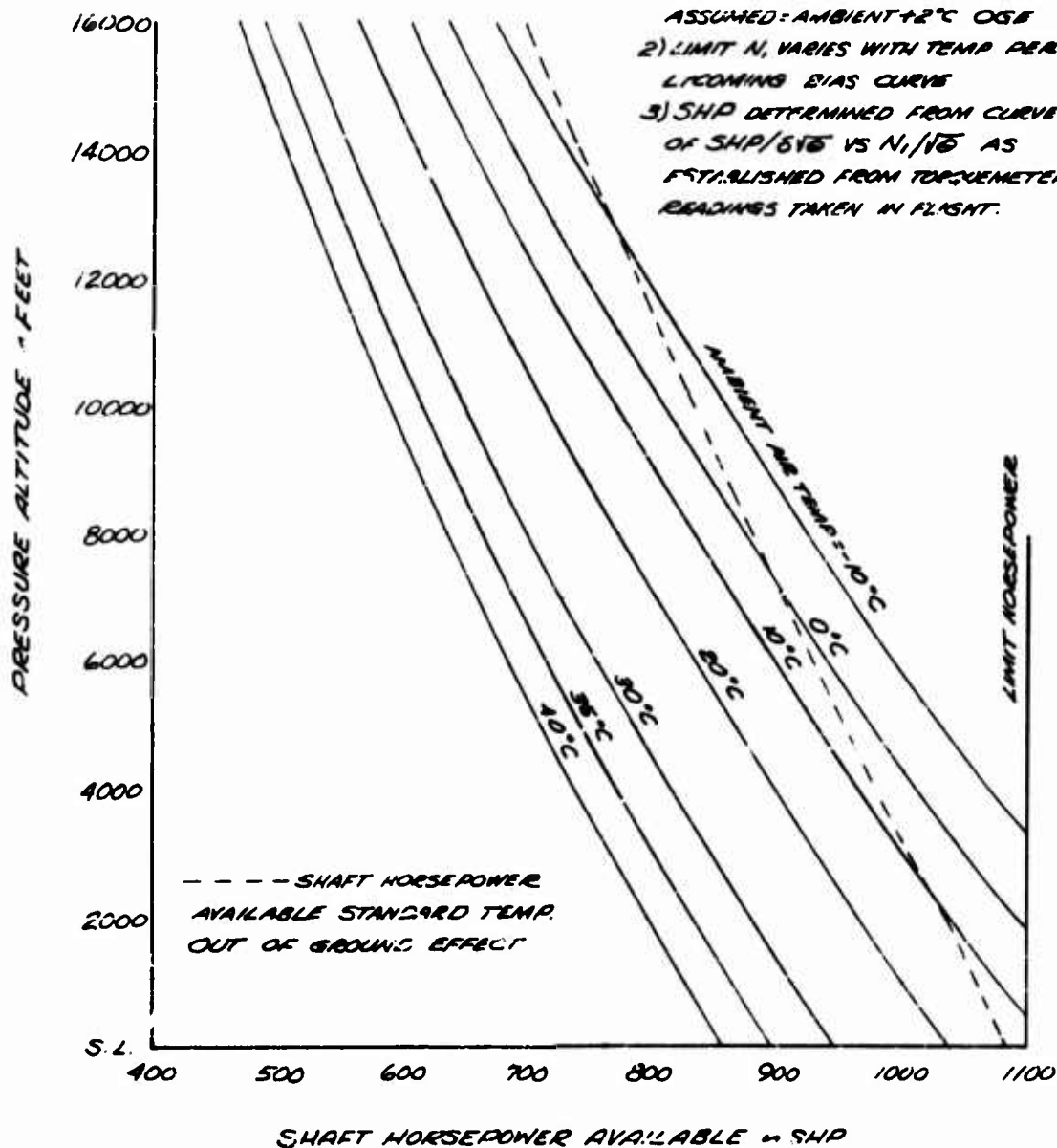


FIG. NO. 15
SHAFT HORSEPOWER AVAILABLE
 YHU-1B S/N 58-2078
 ENGINE T53-L5 S/N LE03007

$N_1 = 25150 \text{ RPM (100\%)}$
 AT 15°C COMPRESSOR INLET
 TEMPERATURE

NOTE:

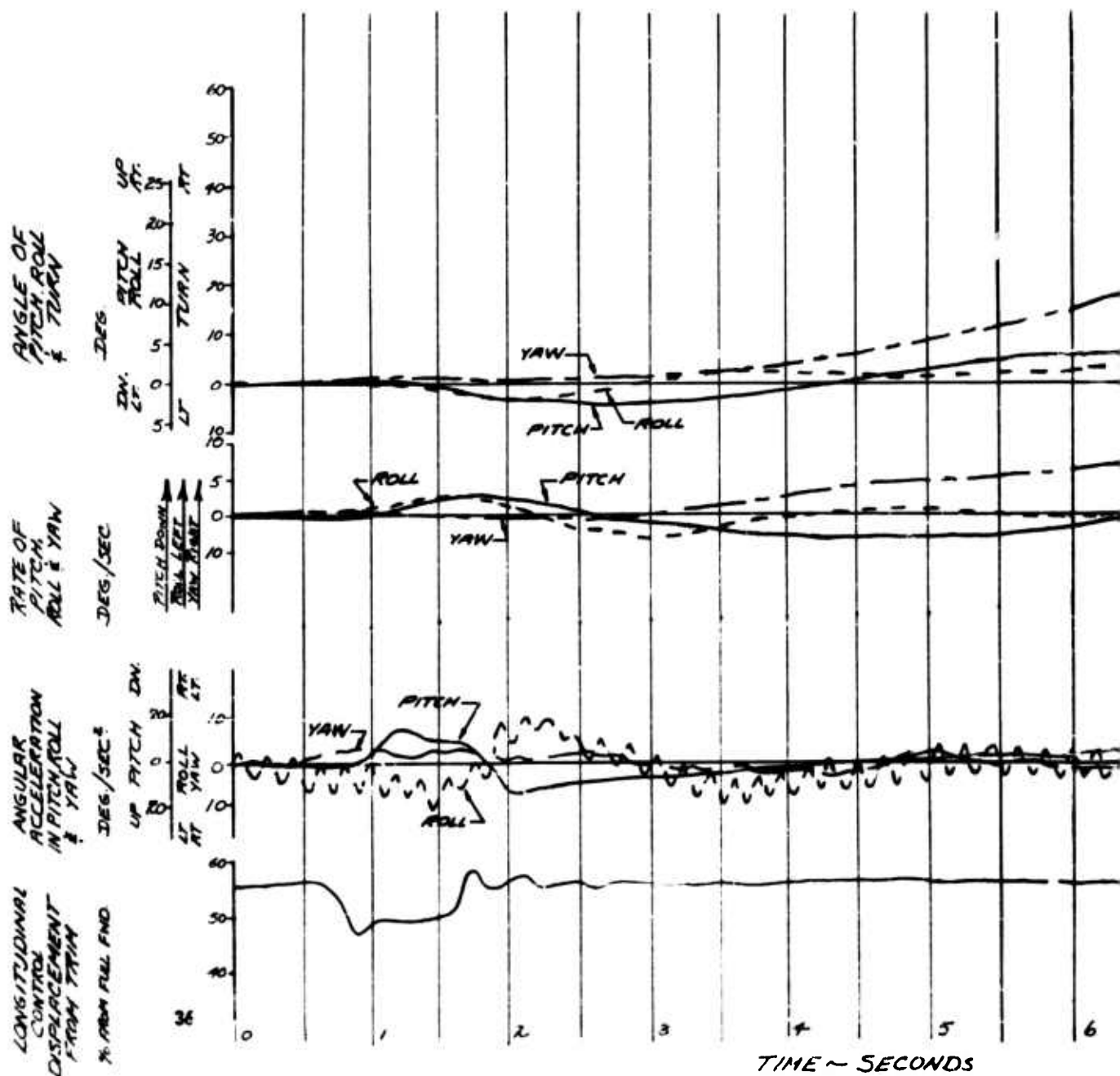
- 1) COMPRESSOR INLET TEMPERATURE
 ASSUMED = AMBIENT $+2^\circ\text{C}$ OGE
- 2) LIMIT N_1 VARIES WITH TEMP PER
 LIFETIME BIAS CURVE
- 3) SHP DETERMINED FROM CURVE
 OF $\text{SHP}/\sqrt{N_1}$ VS $N_1/\sqrt{N_1}$ AS
 ESTABLISHED FROM TORQUEMETER
 READINGS TAKEN IN FLIGHT.



1

FIGURE NO. 16

RESPONSE TO A FORWARD LONGITUDINAL PULSE
YHU-18
CG LOCATION ~ 131.0 (MID) DENSITY ALTITUDE ~ 3000 FT.
ROTOR SPEED ~ 323 GROSS WEIGHT ~ 6600 LB.
HOVER



SE
178
3000 FT.
2 LB.

2

PITCH

ROLL & BANK

YAW & TURN

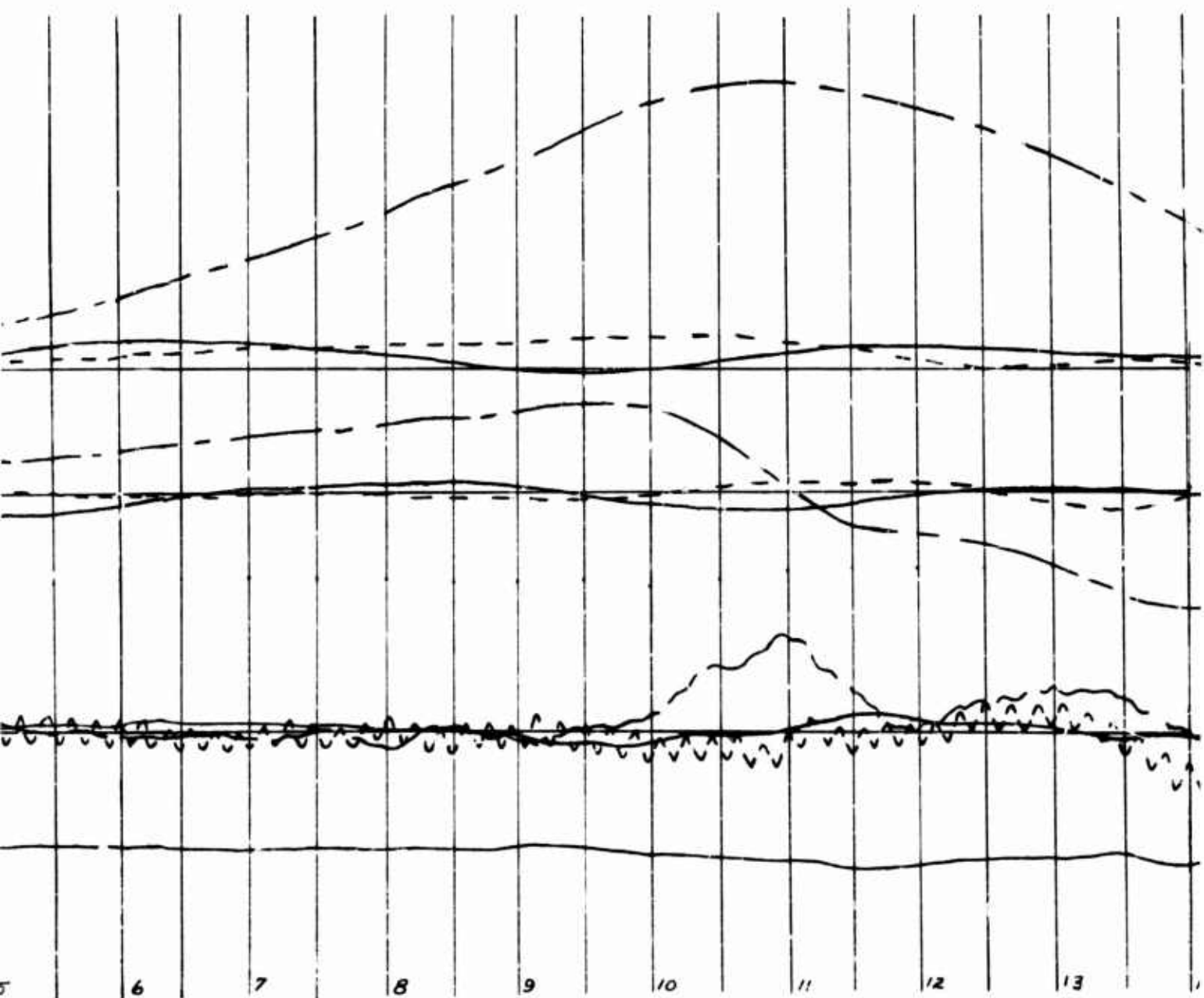
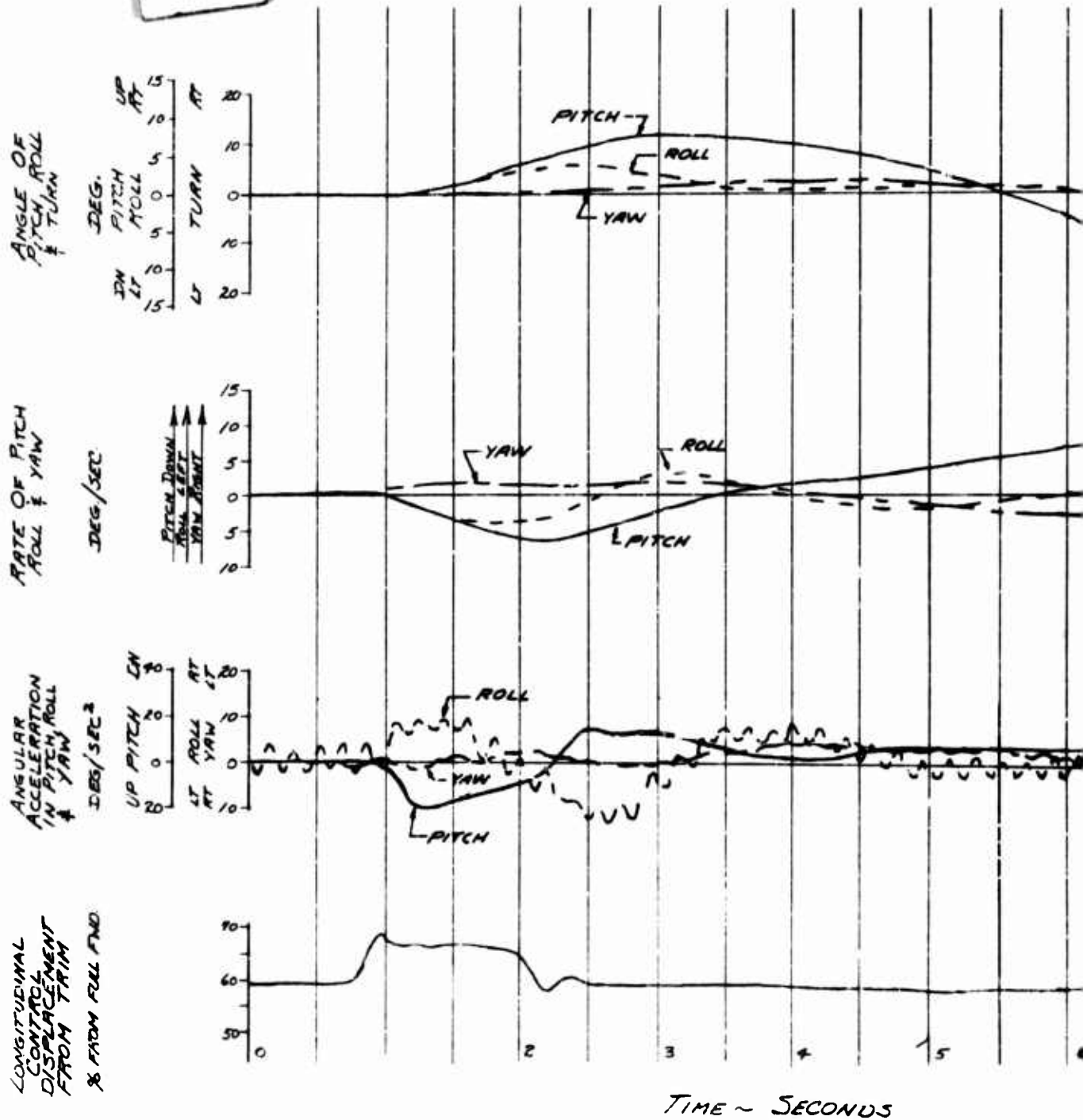


FIGURE No. 17

RESPONSE TO AN AFT LONGITUDINAL PULSE
 YHU-1B
 CG. LOCATION ~ 131.0 (MID) DENSITY ALTITUDE ~ 3000 FT.
 ROTOR SPEED ~ 323 GROSS WEIGHT ~ 6600 LB.
 HOVER



E
78
1000 ft
500 LB.

2

PITCH
ROLL & BANK
YAW & TURN

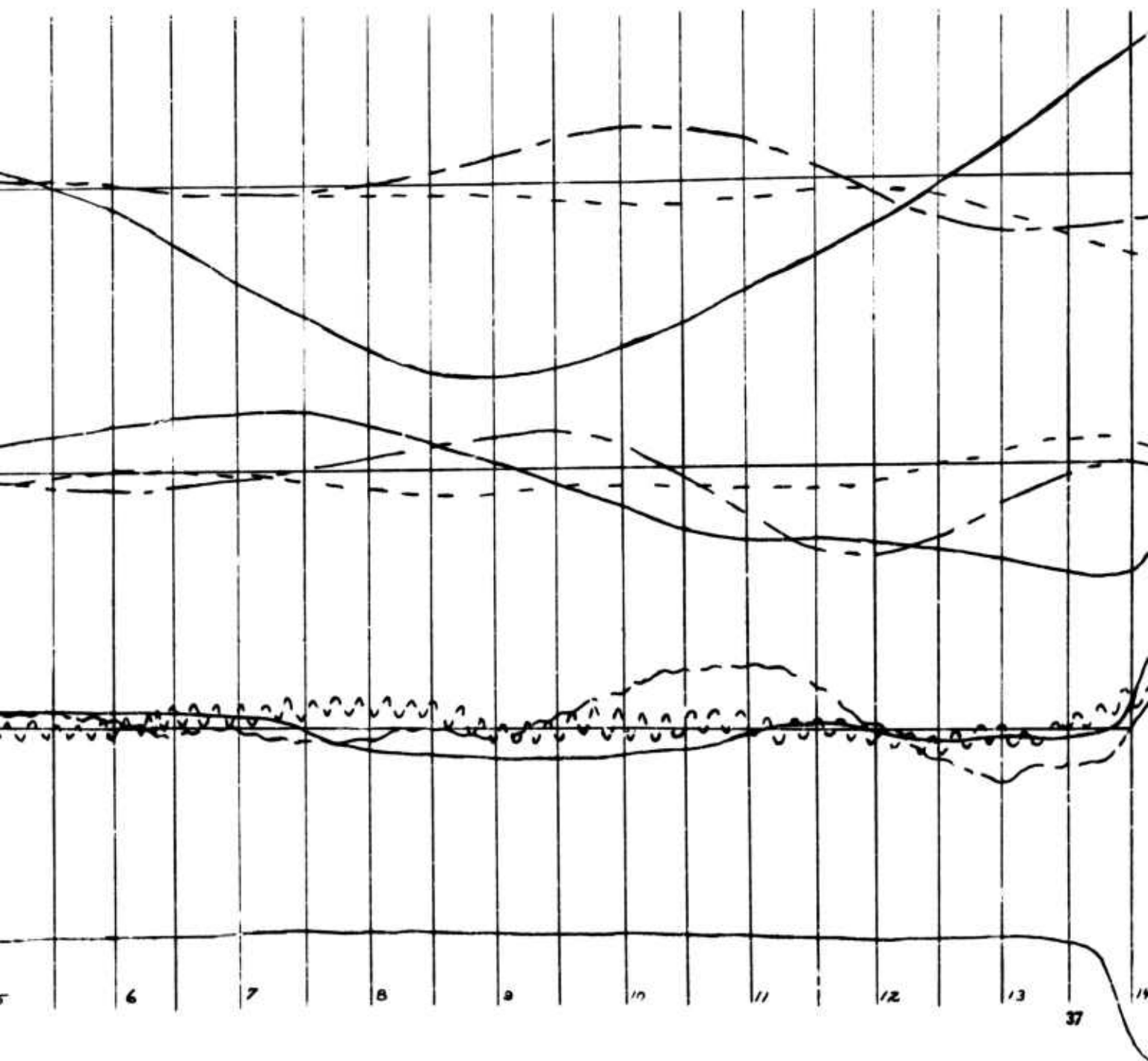


FIGURE NO. 16

RESPONSE TO A RIGHT LATERAL PULSE

YHU-1B

SN 58-2078

CG. LOCATION ~ 131.0 (MID)

DENSITY ALTITUDE ~ 3000 FT.

ROTOR SPEED ~ 323

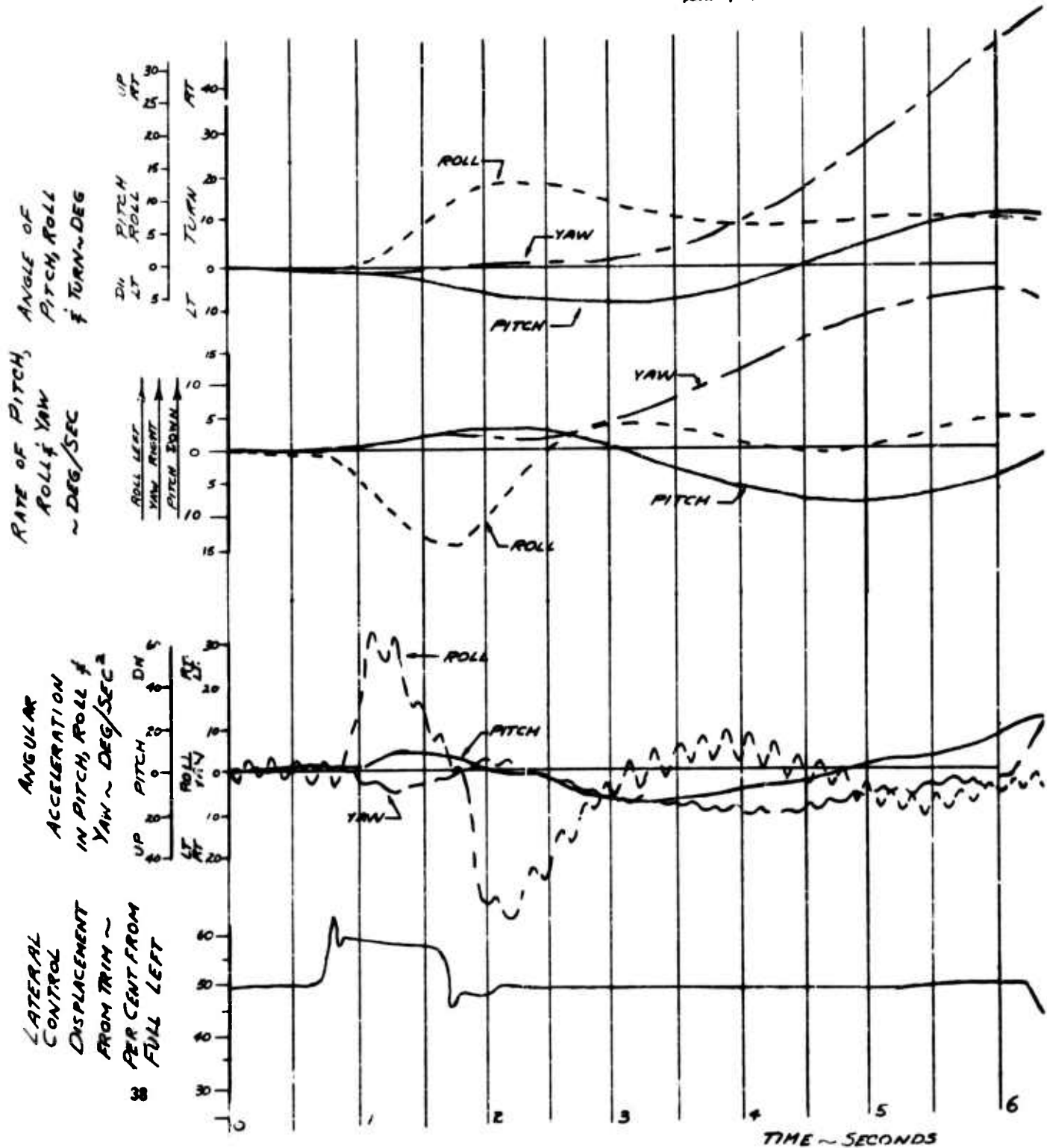
GROSS WEIGHT ~ 6600 LB.

HOVER

PITCH

ROLL & BANK

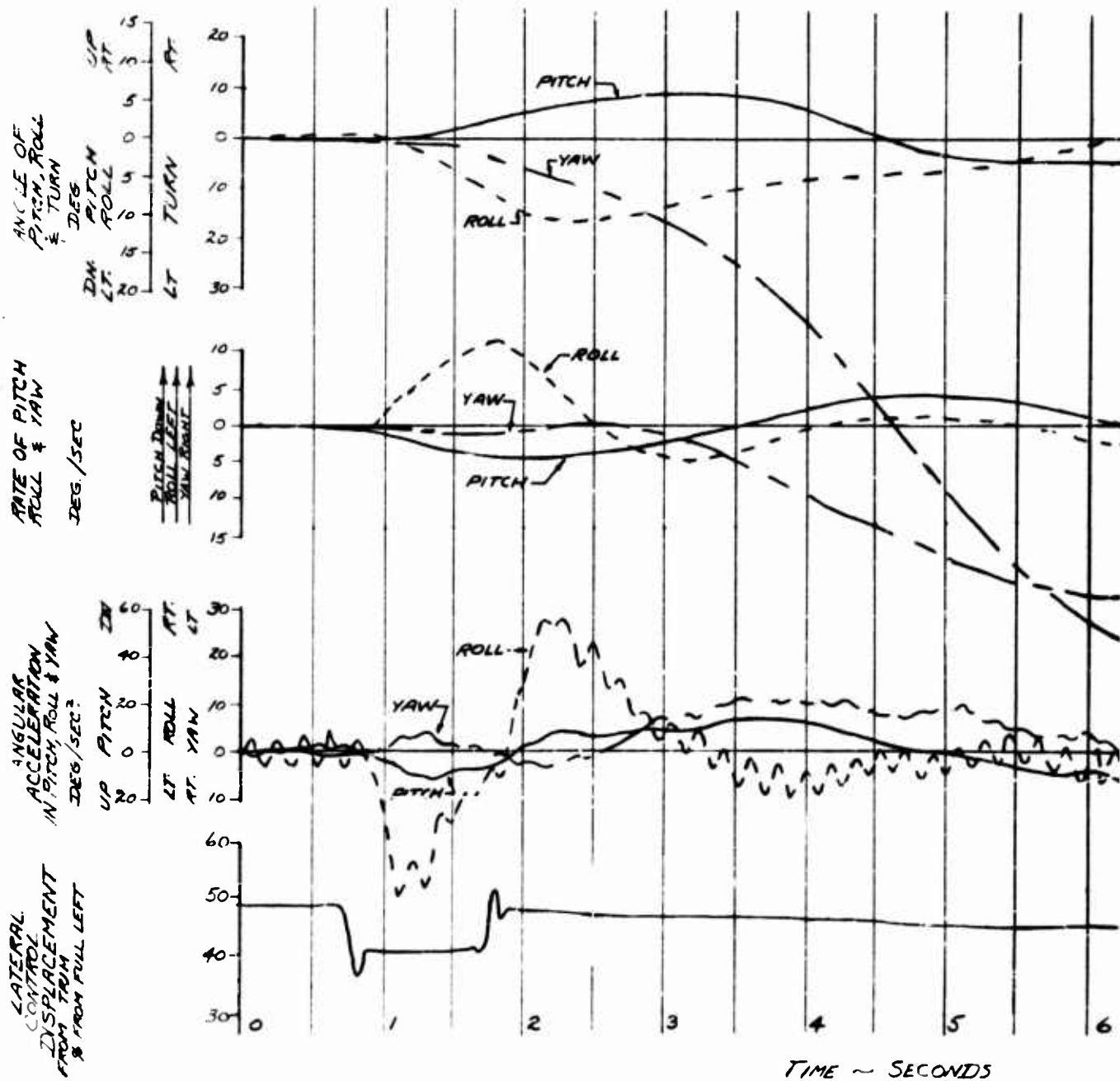
YAW & TURN



1

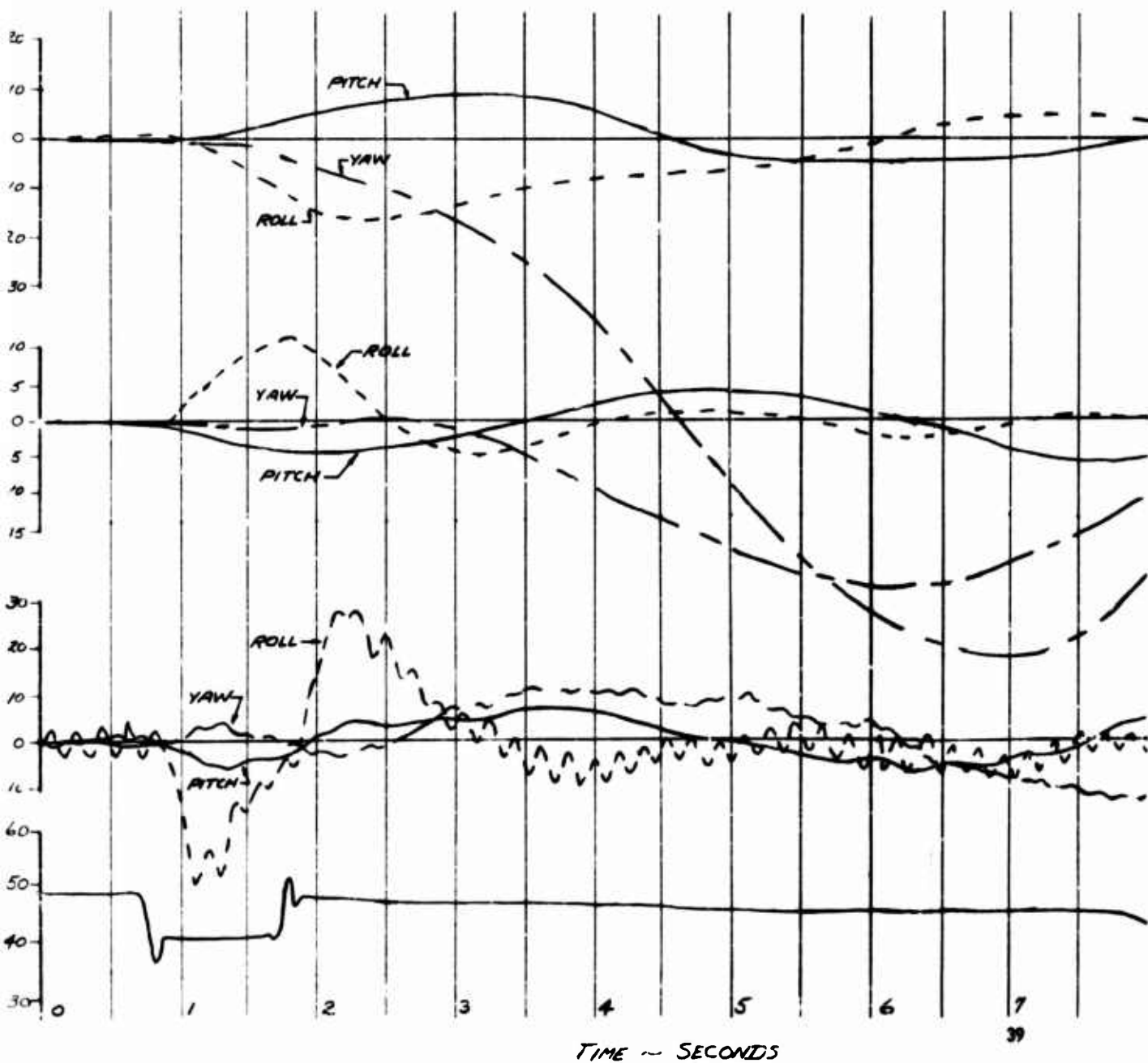
RESPONSE TO A LEFT LATERAL PULSE
YHU-1B 3458-2072
C.G. LOCATION - 131.0 (MID) DENSITY ALT. ~ 3000 FT.
ROTOR SPEED ~ 323 GROSS WEIGHT ~ 6600 LB.
HOVER

PITCH —
ROLL & BANK —
YAW & TURN —



RESPONSE TO A LEFT LATERAL PULSE
 YHU-1B 3/4 58-207A
 C.G. LOCATION ~ 131.0 (MID) DENSITY ALT. ~ 3000 FT.
 ROTOR SPEED ~ 323 GROSS WEIGHT ~ 6600 LB.
 HOVER

PITCH _____
 ROLL & BANK -----
 YAW & TURN - - - - -



1

FIGURE NO. 20

RESPONSE TO A RIGHT DIRECTIONAL PULSE

YHU-1B

S/N 58-2078

CG LOCATION ~ 131.0 (MID) DENSITY ALTITUDE ~ 3000 FT.

ROTOR SPEED ~ 323

GROSS WEIGHT ~ 6600 LB.

HOVER

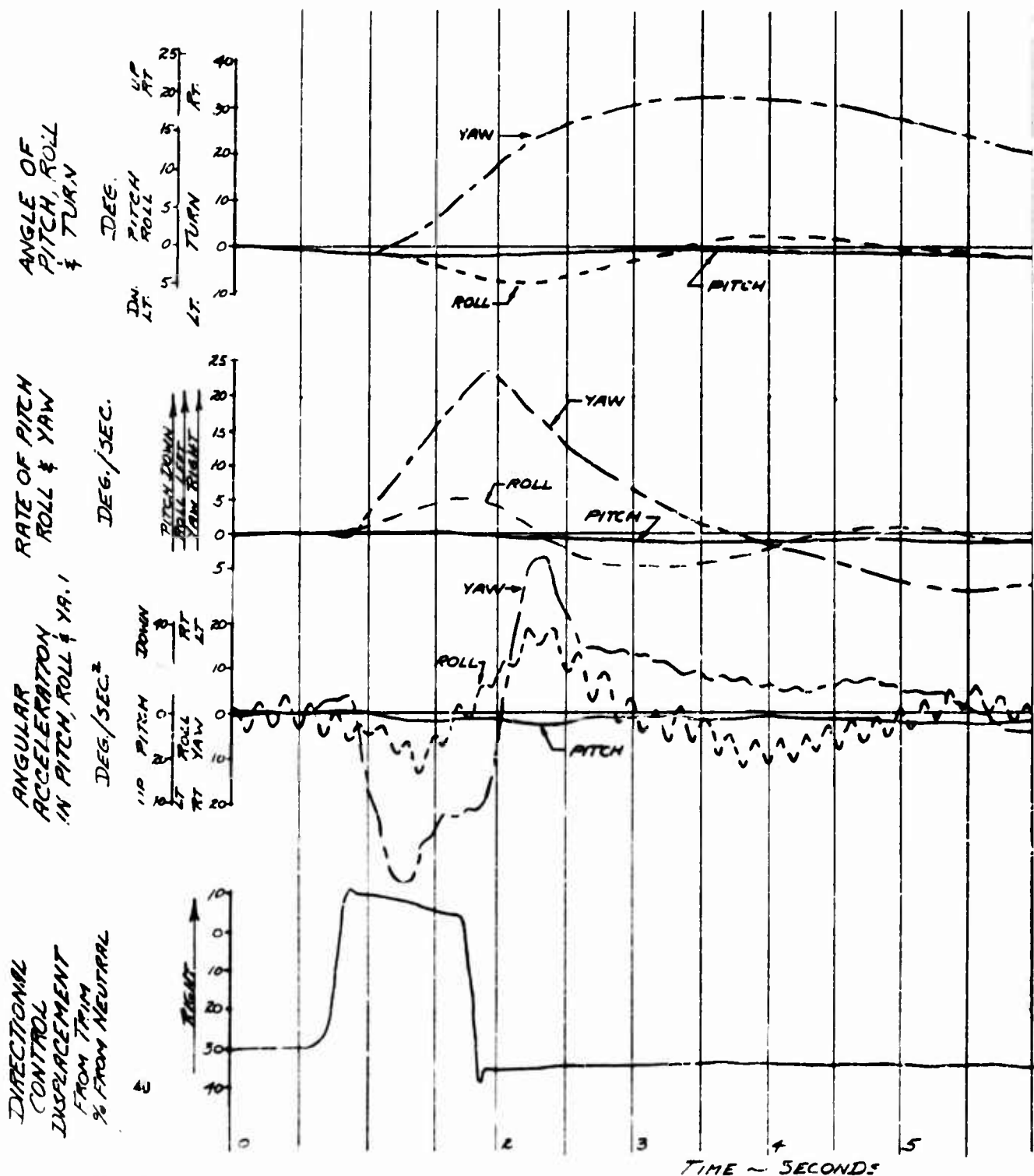


FIGURE NO. 20

2

RESPONSE TO A RIGHT DIRECTIONAL PULSE

YHU-1B

5/11 58-2078

CG LOCATION ~ 131.0 (MID) DENSITY ALTITUDE ~ 3000 FT.

ROTOR SPEED ~ 323

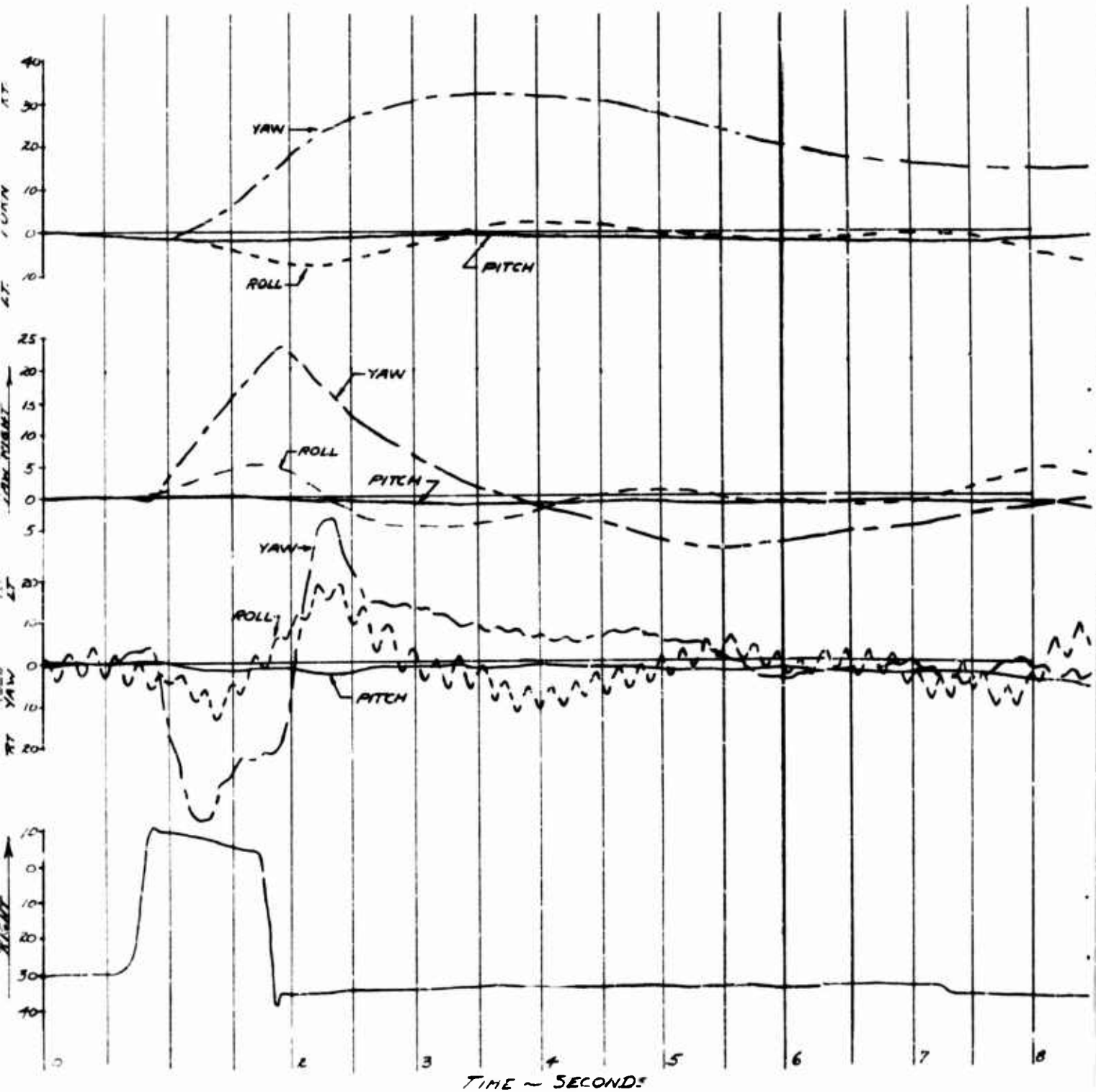
GROSS WEIGHT ~ 6600 LB.

HOVER

PITCH ———

ROLL & BANK ———

YAW & TURN - - - -



1

FIGURE NO 21

RESPONSE TO A LEFT DIRECTIONAL PULSE

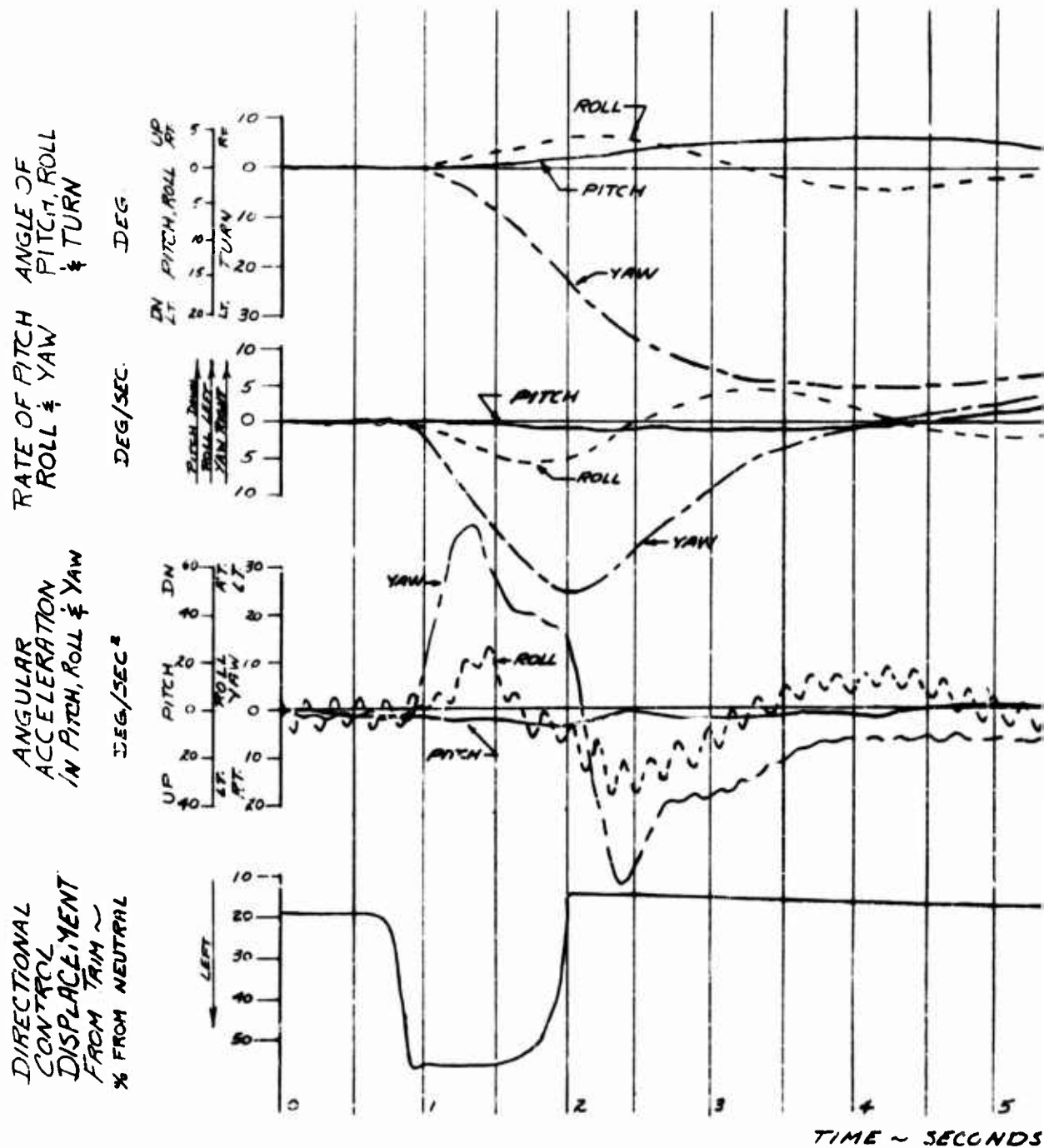
HU-1B

S/N 58-2078

CG. LOCATION ~ 1310 (MID) DENSITY ALTITUDE ~

ROTOR SPEED ~ 323 GROSS WEIGHT ~ 6600

HOVER



TIME ~ SECONDS

2

FIGURE NO 21

RESPONSE TO A LEFT DIRECTIONAL PULSE

YHU-18

SN 58-2078

CG. LOCATION ~ 1310 (MID) DENSITY ALTITUDE ~ 3000 FT.

ROTOR SPEED ~ 32.7 GROSS WEIGHT ~ 6600 LB.

HOVER

PITCH

ROLL & BANK

YAW & TURN

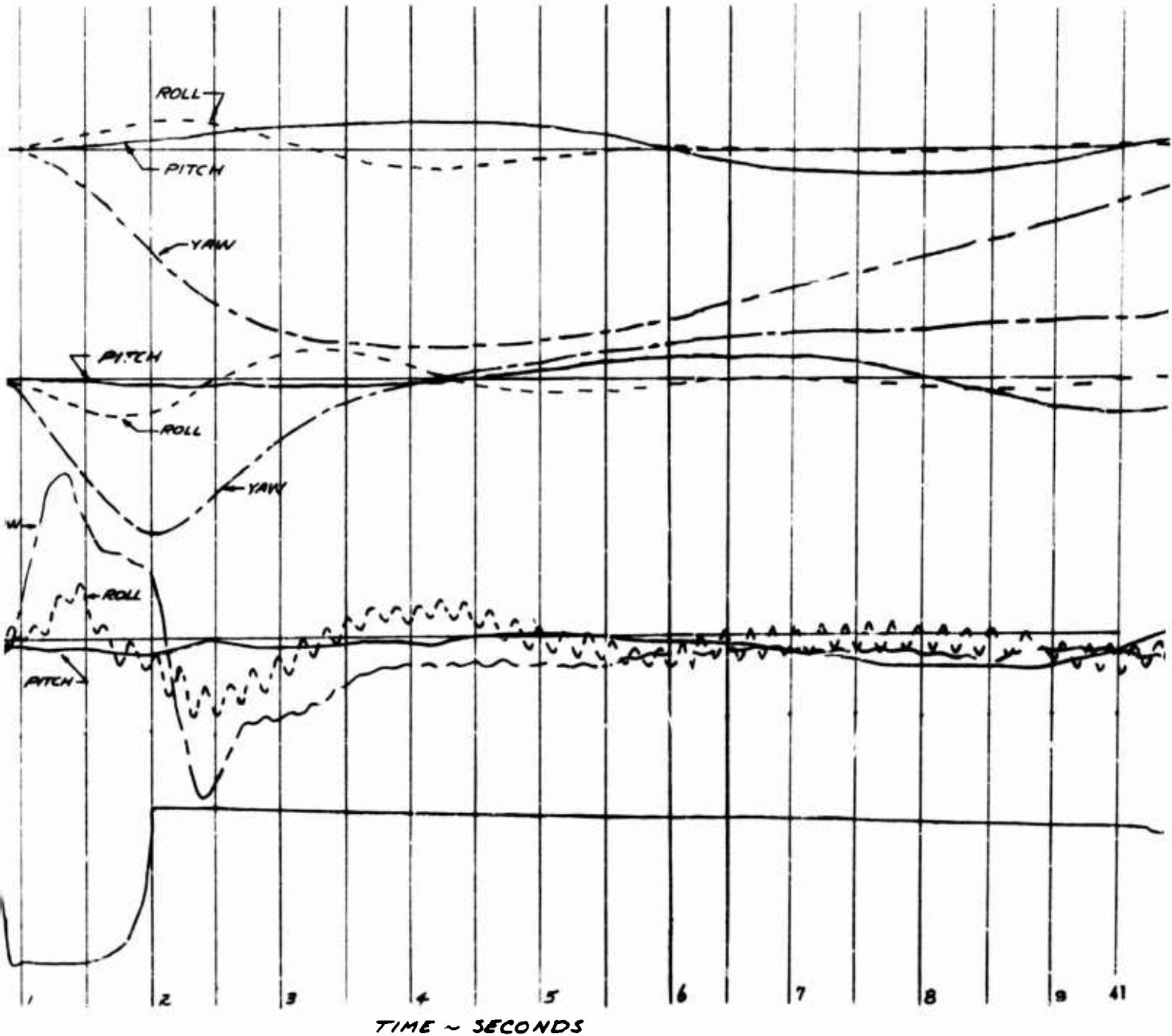
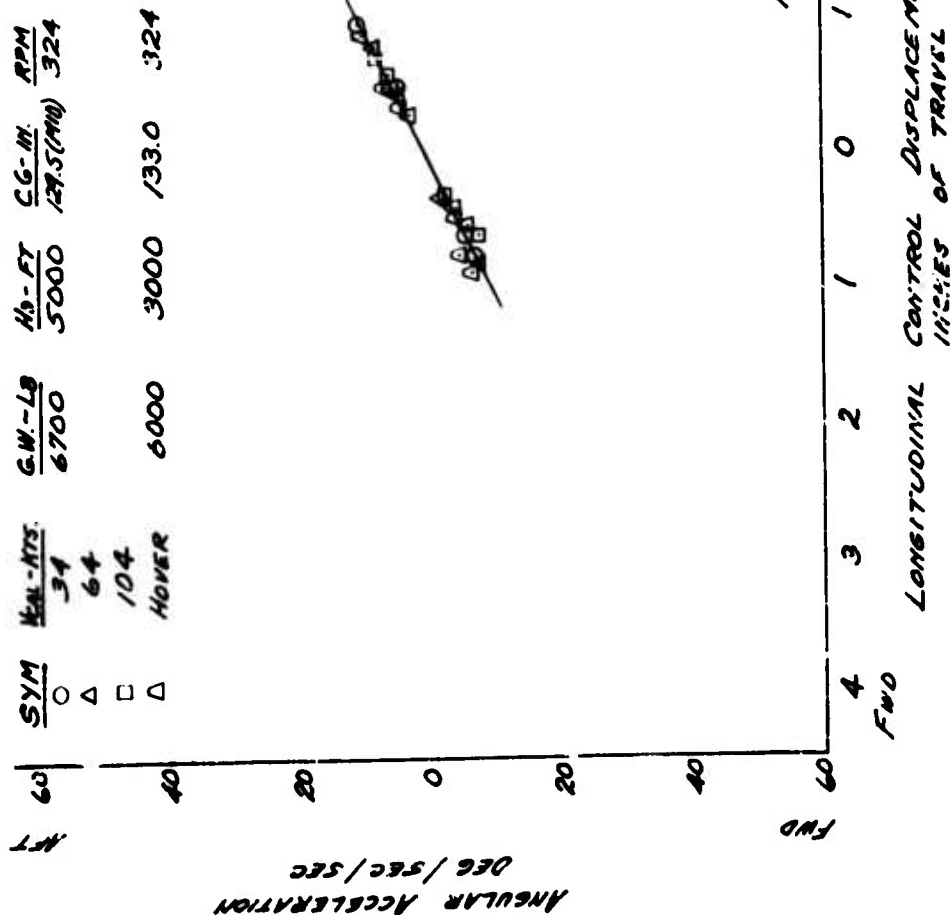


FIG NO 22
 LONGITUDINAL CONTROL SENSITIVITY
 YHU-1B
 SYN 58-2078



NOTE: MINIMUM ANGULAR ACCELERATION
 REACHED 0.4 SECONDS AFTER
 CONTROL DISPLACEMENT

FIG NO 22
LONGITUDINAL CONTROL SENSITIVITY
YHU-18
SN 58-2078

SYM	WAL-NTS	GM-LB	NO-FT	CG-IN	RPM
○	34	6700	5000	128.5 (110)	324
△	64				
□	104				
△	HOVER	6000	3000	133.0	324

ANGULAR ACCELERATION
DEG/SEC/SEC

NOTE: MAXIMUM ANGULAR ACCELERATION
REACHED 0.4 SECONDS AFTER
CONTROL DISPLACEMENT

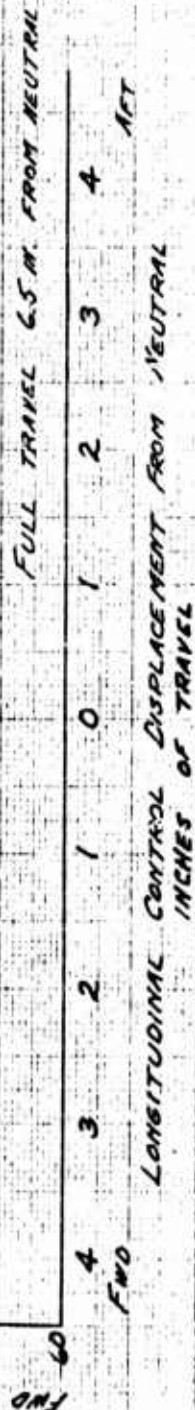
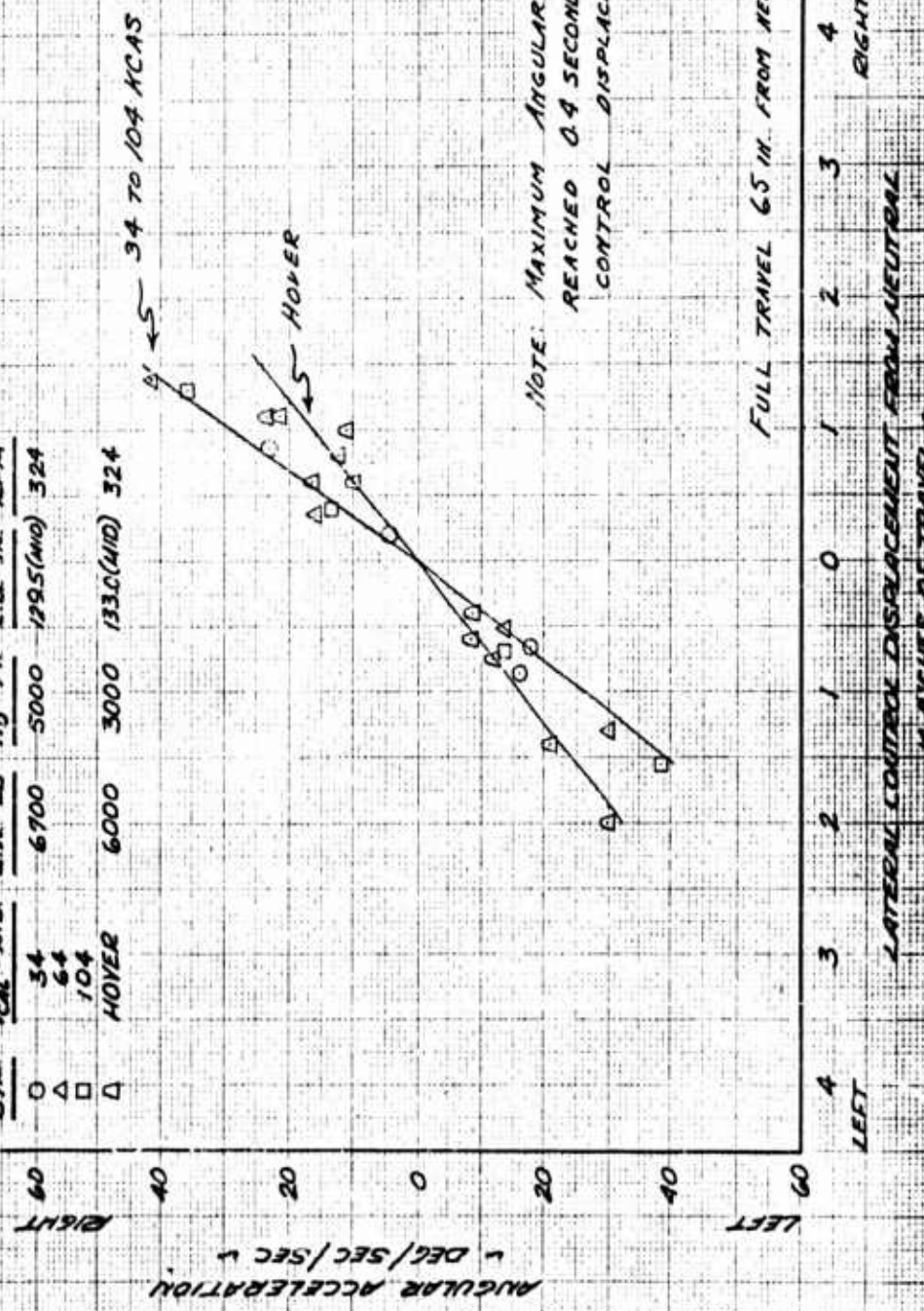


FIG. NO. 23
LATERAL CONTROL SENSITIVITY
YHU-1B
SN 58-2078

SYM	V _{CL} - KTS	G.W. - LB	H ₀ - FT	C.G. - IN	RPM
○	34	6700	5000	1295(MID)	324
△	64				
□	104				
◇	HOVER	6000	3000	1330(MID)	324



FILE NO. 24
DIRECTIONAL CONTROL SENSITIVITY
 YHU-1A S/N 58-2078

SWAY	$Y_{100} = \text{WTS}$	GM = LB	$H_0 = \text{FT}$	CG = IN	RPM
0	34	6700	5000	1295(MID)	324
0	34				285
Δ	64				324
Δ	64				285
□	104				324
□	104				285
□	HOVER	6000	3000	1330(MID)	324

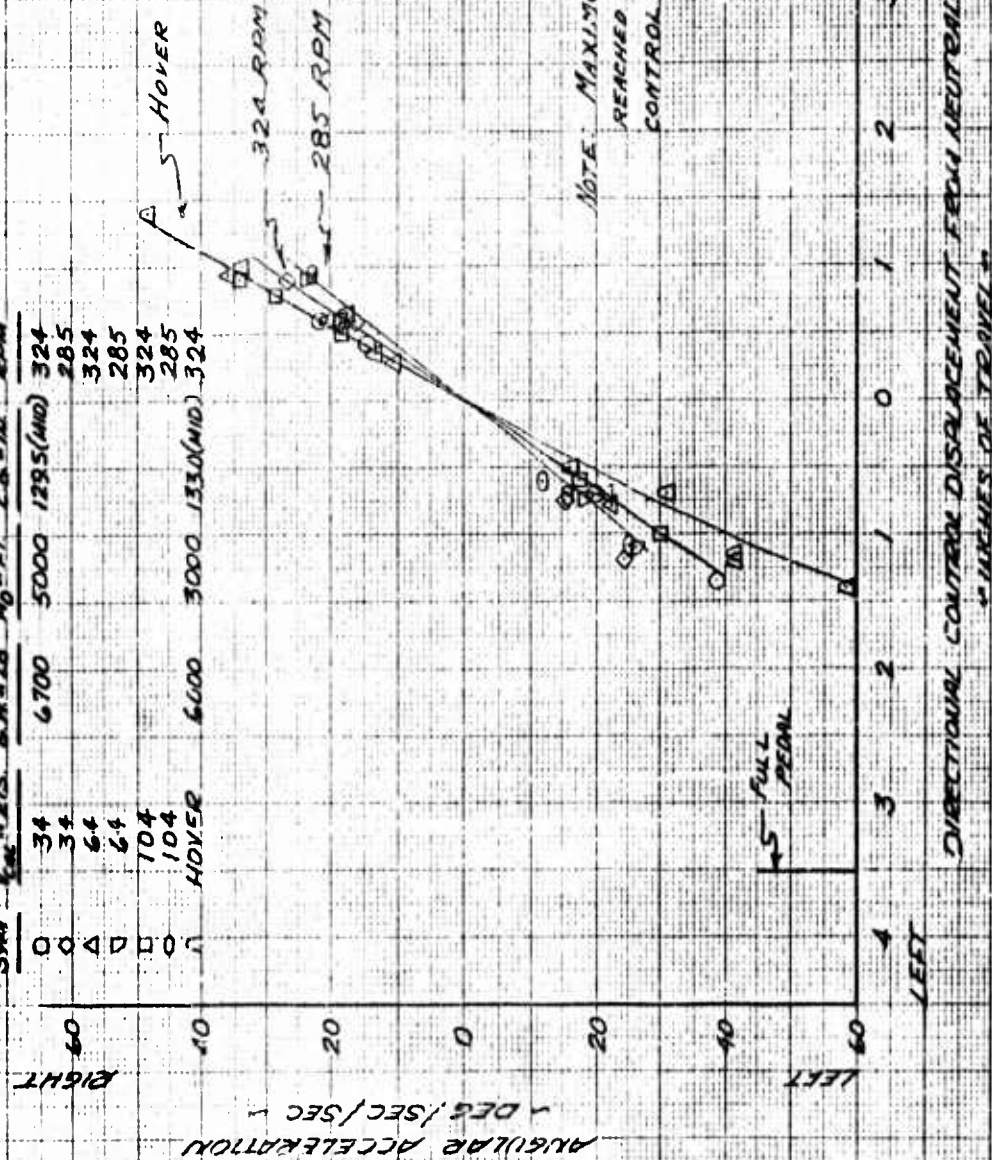


FIG NO 25
 LONGITUDINAL CONTROL RESPONSE
 S/W 58-2078

SWAY	V_{SWAY}	SWAY	M_0	SWAY	$C.G.$	SWAY
0	34	6700	5000	1295(MID)	324	
Δ	64	"	"	"	"	"
□	104	"	"	"	"	"
▽	HOVER	6000	3000	1330(MID)	324	

60
 AFT

40

20

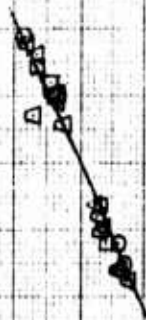
0

20

40

60
 FWD

DITCH RATE
 DEG/SEC



NOTE: MAXIMUM ANGULAR VELOCITY
 REACHED 19-20 SECONDS AFTER
 CONTROL DISPLACEMENT

FULL TRAVEL 6.5 IN. FROM NEUTRAL

4
 FWD

3

2

1

0

1

2

3

4

45

LONGITUDINAL CONTROL DISPLACEMENT FROM NEUTRAL
 IN INCHES OF TRAVEL

FIG. NO. 24
LATERAL CONTROL RESPONSE
S/N 58-2078
VA-11B

SYM.	V_{CL} - KTS.	G.W. - LB	H_0 - FT	C.G. - IN	RPM
○	34	6700	5000	129.5 (MID)	324
△	64				
□	104				
◇	HOVER	6000	3000	133.0 (MID)	324

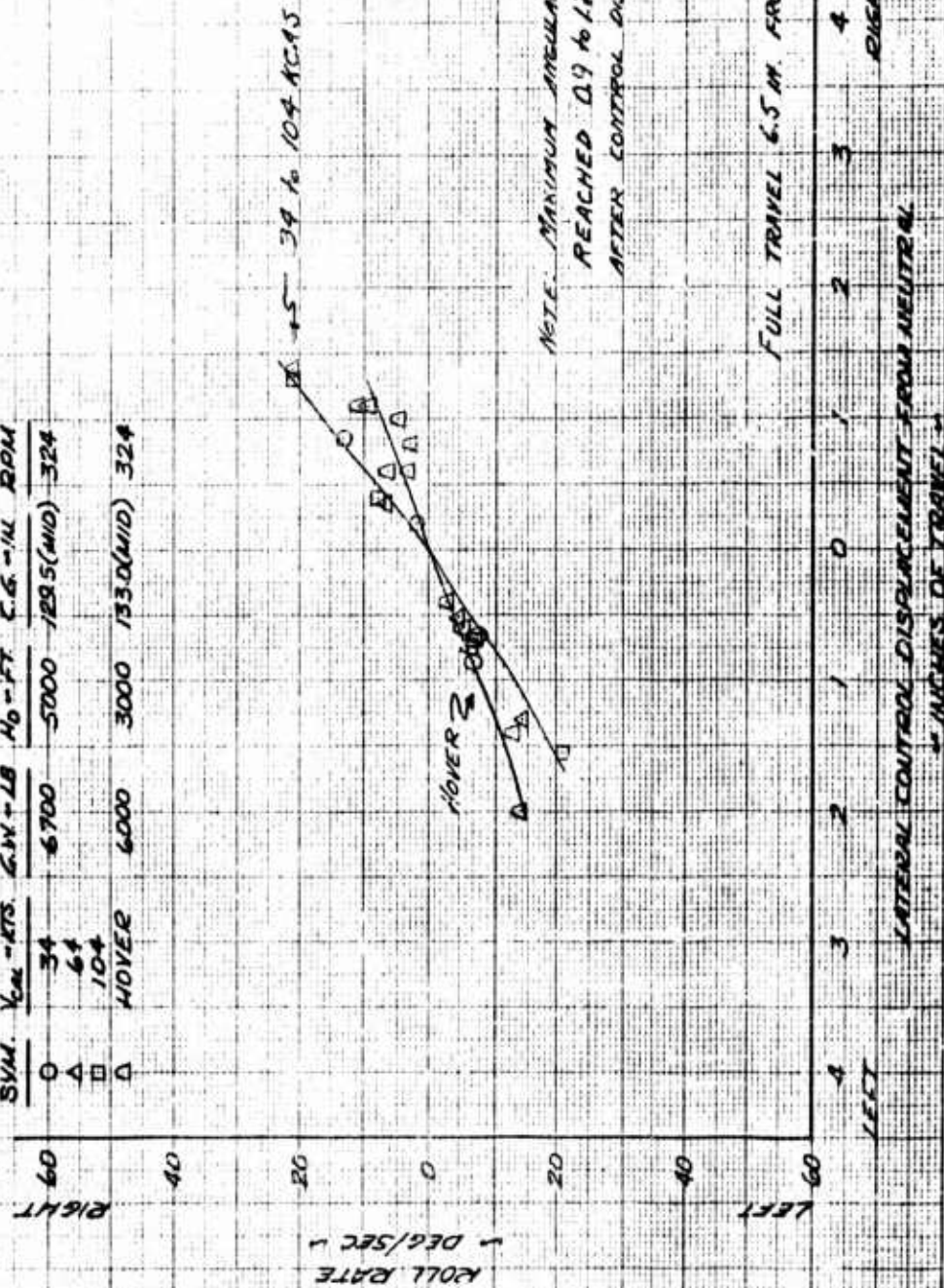


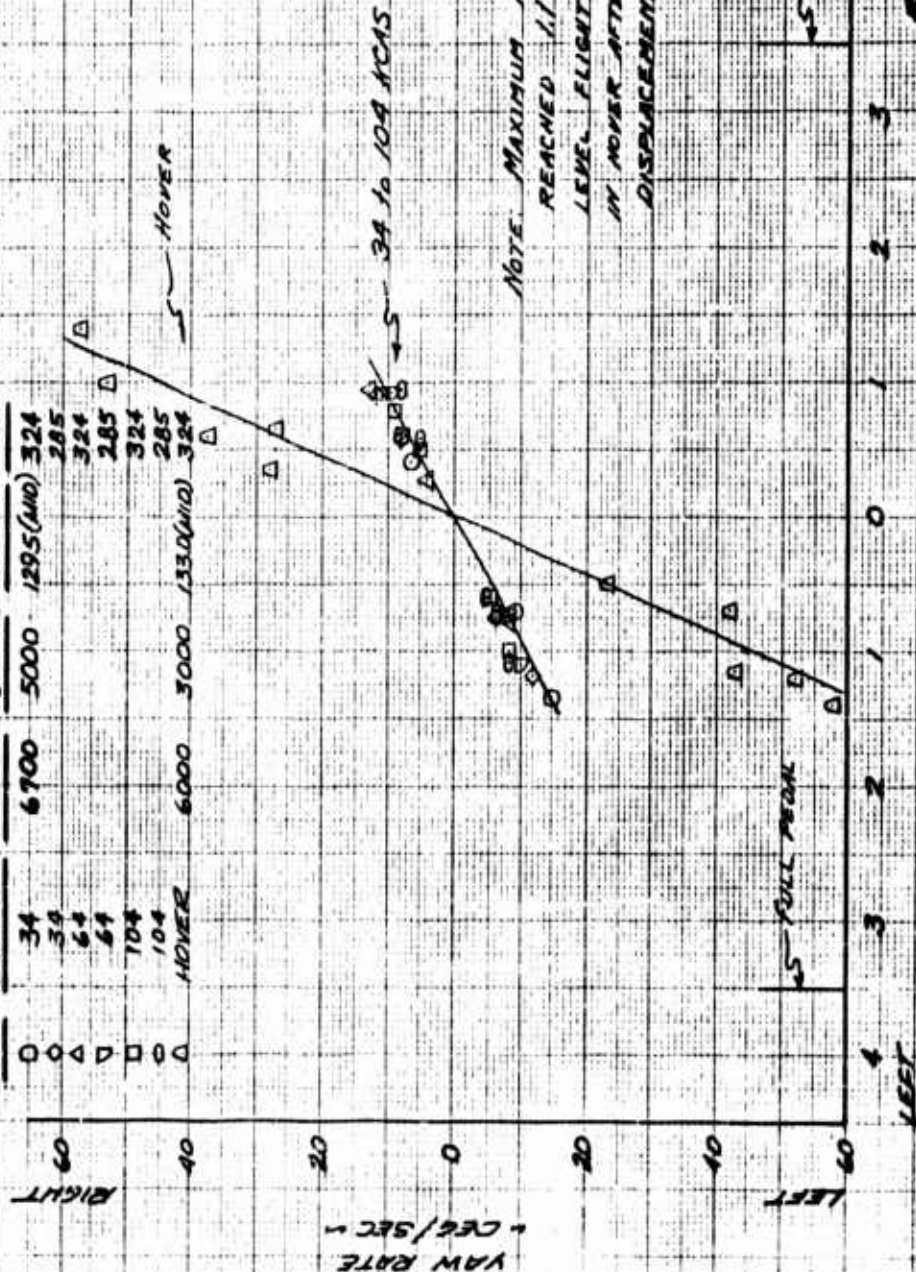
FIG. NO. 27

DIRECTIONAL CONTROL RESPONSE

YMU-1B

SNW 58-2078

SYM	V _{CR} = KTS	C _N = 1.8	H ₀ = FT.	C _G - IN.	RPM
○	34	6700	5000	1295 (MID)	324
○	34				285
△	64				324
△	64				285
□	104				324
□	104				285
△	HOVER	6000	3000	1330 (MID)	324



DIRECTIONAL CONTROL DISPLACEMENT FROM NEUTRAL

INCHES OF TRAVEL

FIG. NO. 28
CONTROL POSITIONS IN LEVEL FLIGHT
YHU-12 S/N 58-2078

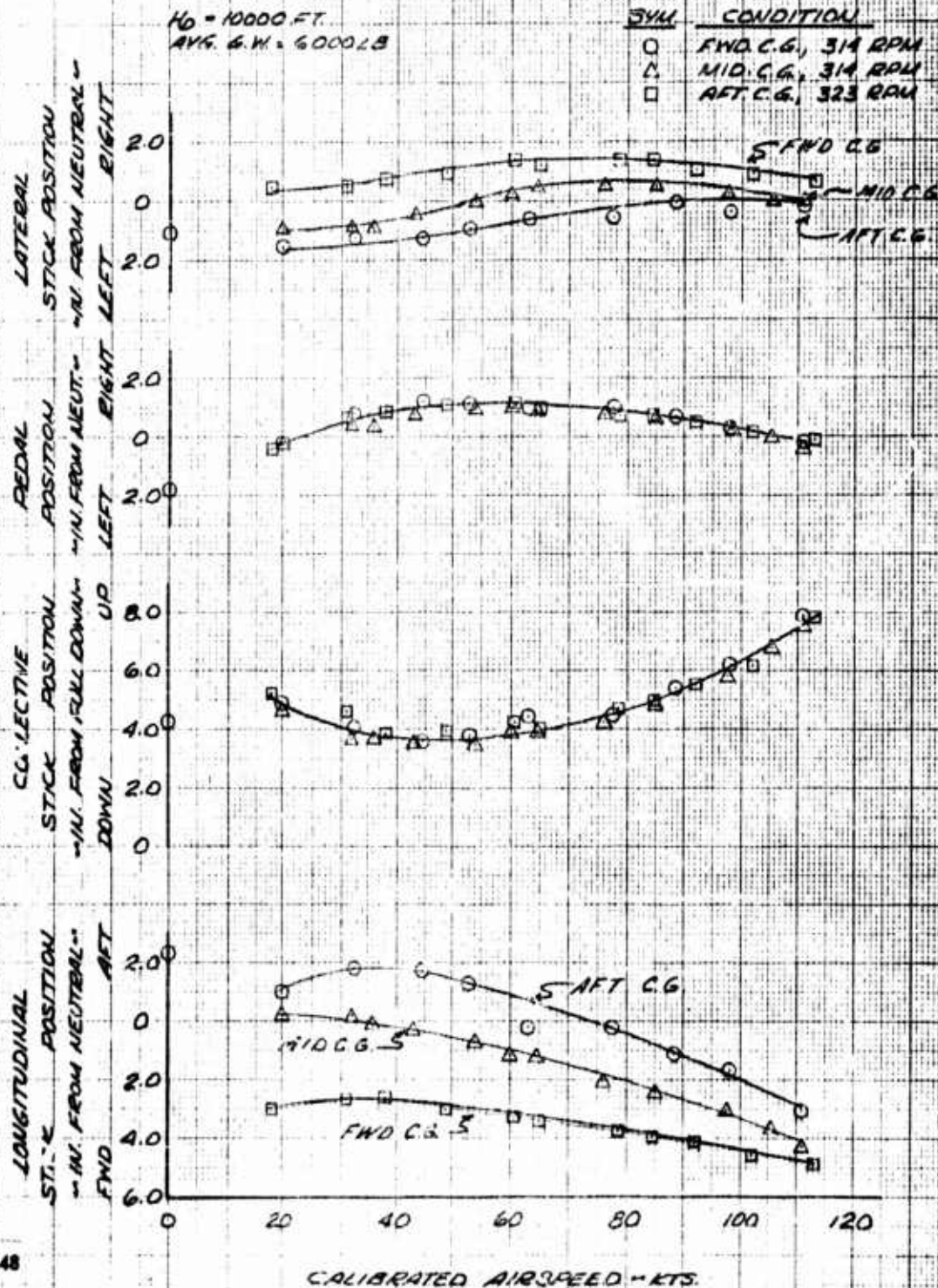
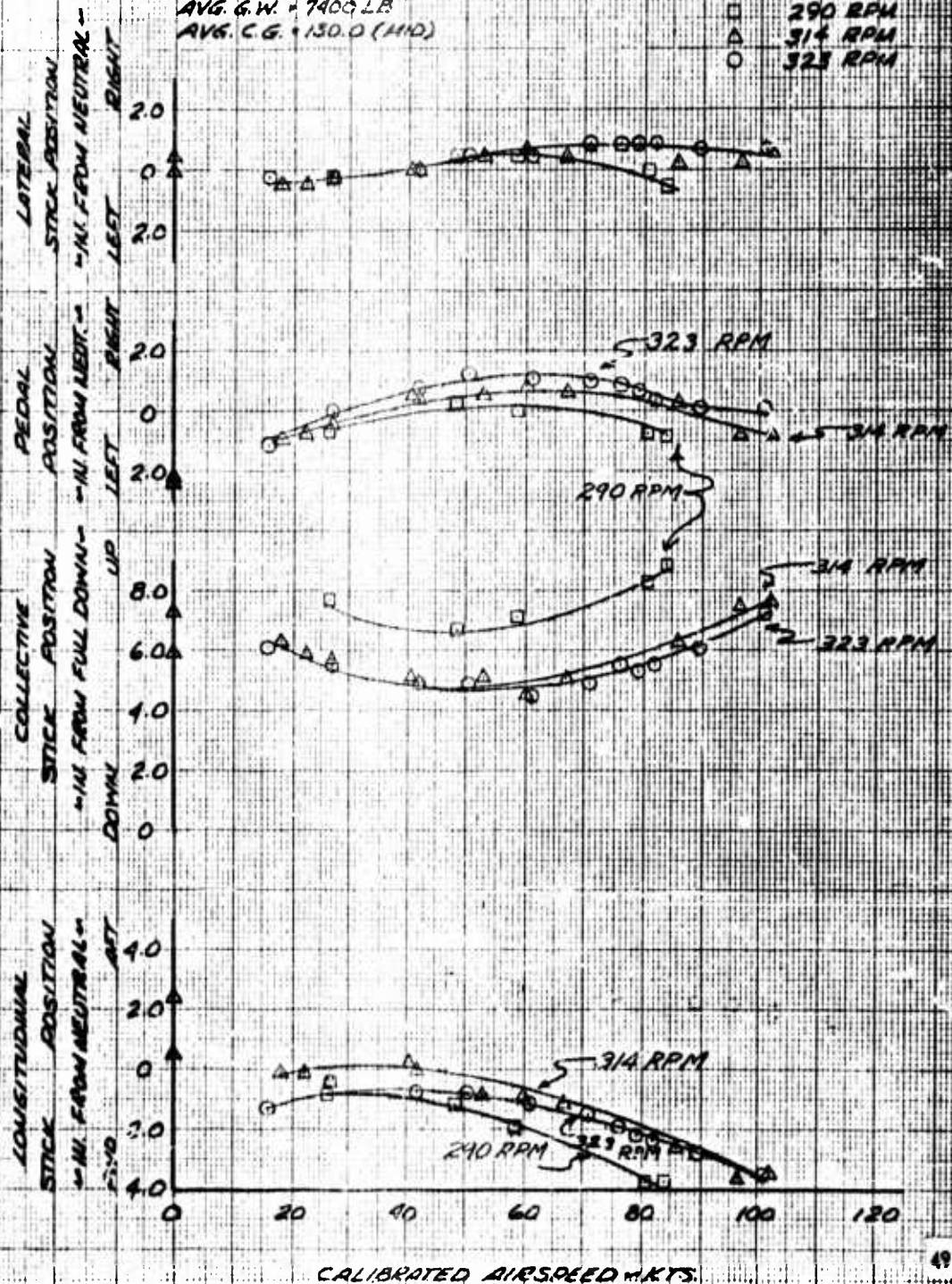


FIG. NO. 29
CONTROL POSITIONS IN LEVEL FLIGHT
 YHU-1B
 S/N 58-2078

HD = 9350 FT.
 AVG. G.W. = 7400 LB.
 AVG. C.G. = 130.0 (HND)

SYM. CONDITION
 □ 290 RPM
 △ 314 RPM
 ○ 323 RPM



CALIBRATED AIRSPEED - KTS.

FIG. NO. 30
 STATIC DIRECTIONAL STABILITY
 YHU-1B
 LEVEL FLIGHT

$V_T = 37.0$ KTS
 $H_D = 5000$ FT.
 AVG. G.W. = 7660 LB
 AVG. C.G. = 129.8 (MID)
 RPM = 323

SYM.
 Δ ROLL, ϕ
 $\square$ ANGLE OF ATTACK, α
 $\circ$ LONG., PEDAL, LAT. STICK POSITION.

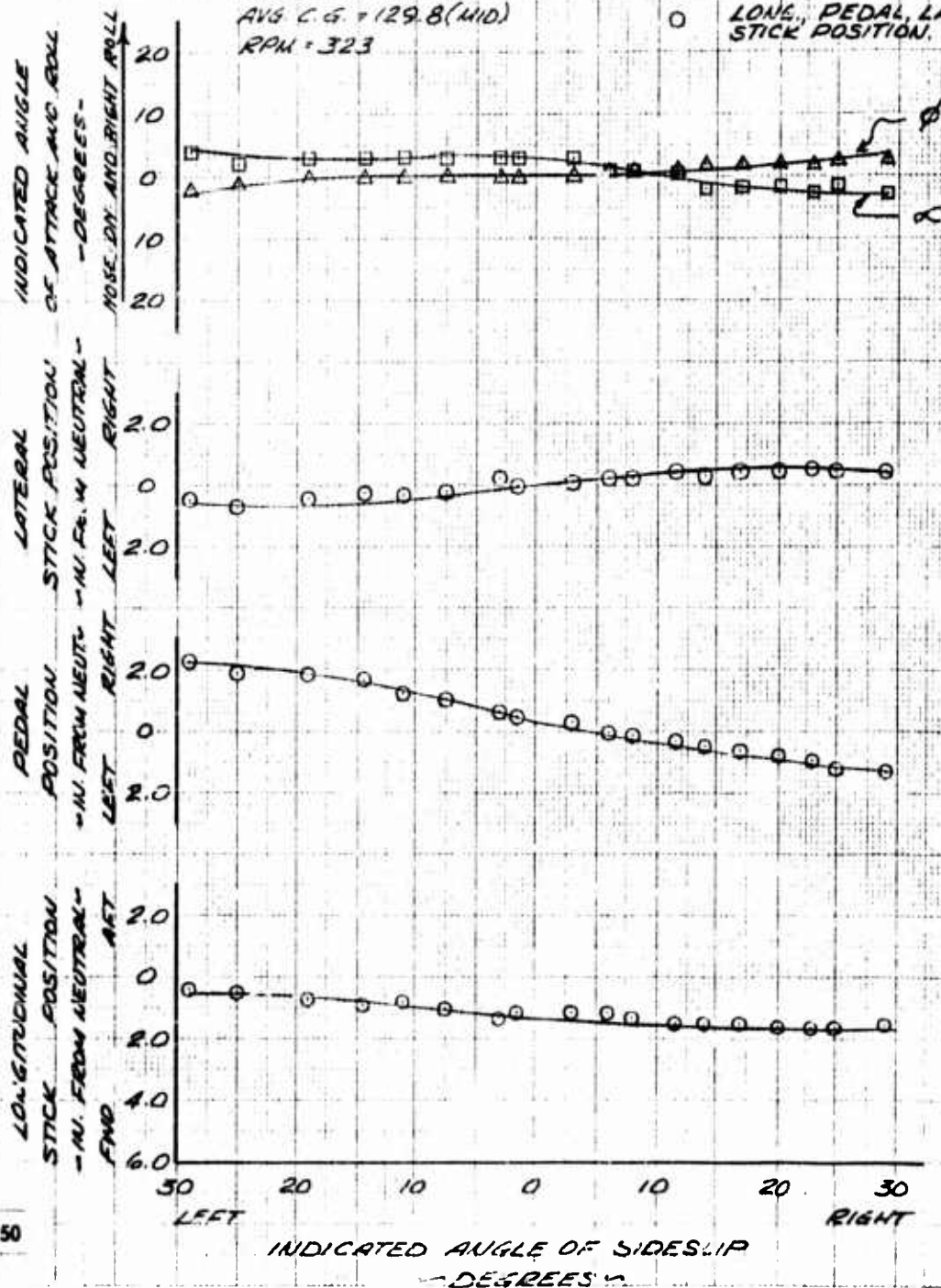


FIG. NO. 31
 STATIC DIRECTIONAL STABILITY
 VNU-1B
 3/11/58 2078
 LEVEL FLIGHT

$V_T = 68.5 \text{ KTS.}$
 $H_0 = 5000 \text{ FT.}$
 $\text{AVG. G.W.} = 7660 \text{ LB}$
 $\text{AVG. C.G.} = 129.8 \text{ (IN.)}$
 $\text{RPM} = 325$

SYM.
 Δ ROLL, ϕ
 $\square$ ANGLE OF ATTACK, α
 $\circ$ LONG. PEDAL, LOT STICK POSITION

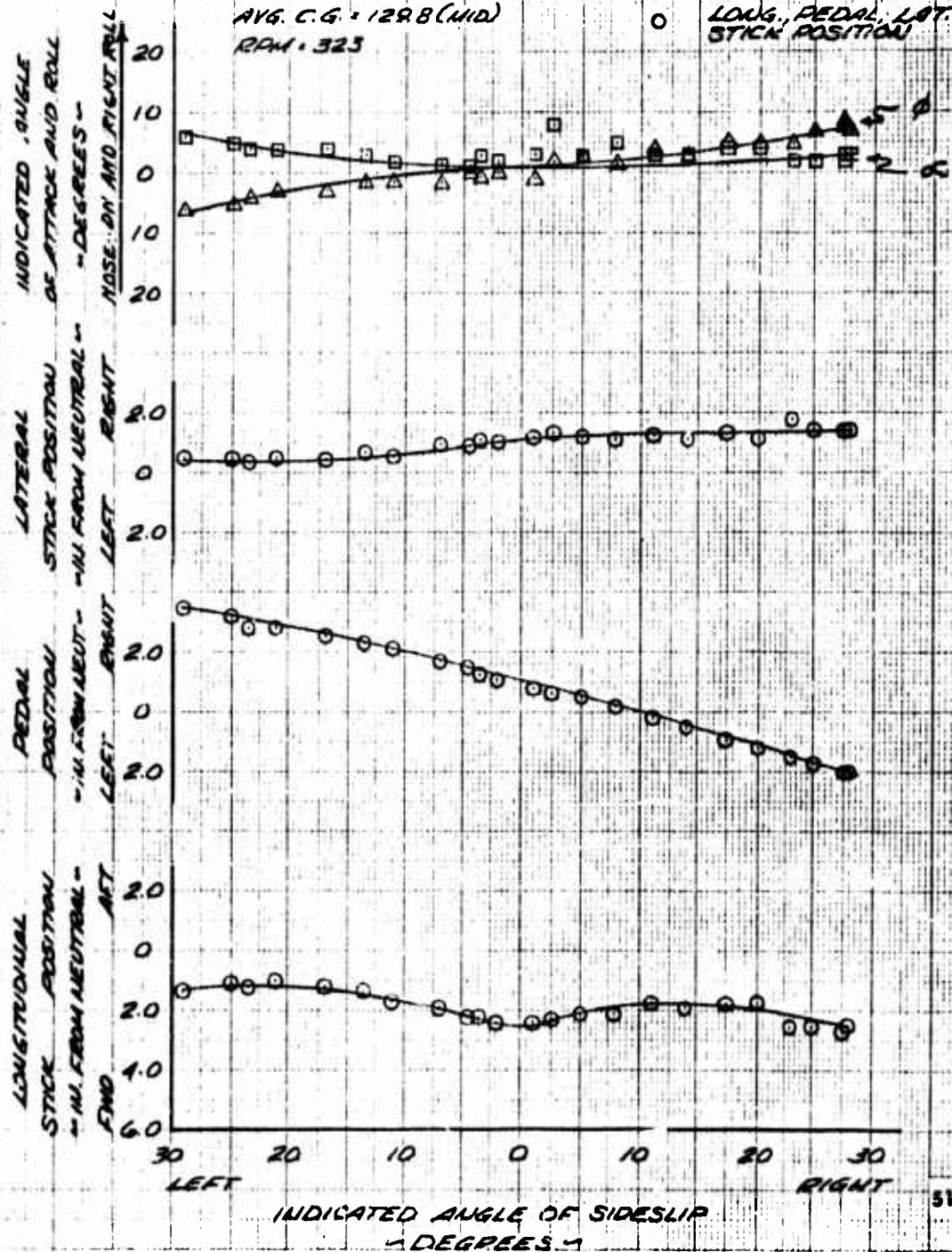


FIG. NO. 32
 STATIC DIRECTIONAL STABILITY
 YHU-1B
 LEVEL FLIGHT

$V_T = 112.5$ KTS.
 $H_D = 5000$ FT.
 AVG. G.W. = 7660 LB
 CVG. C.G. = 129.8 (MID)
 RPM = 323

SYM
 Δ ROLL, ϕ
 $\square$ ANGLE OF ATTACK, α
 $\circ$ LONG. PEDAL, LAT. STICK POSITION

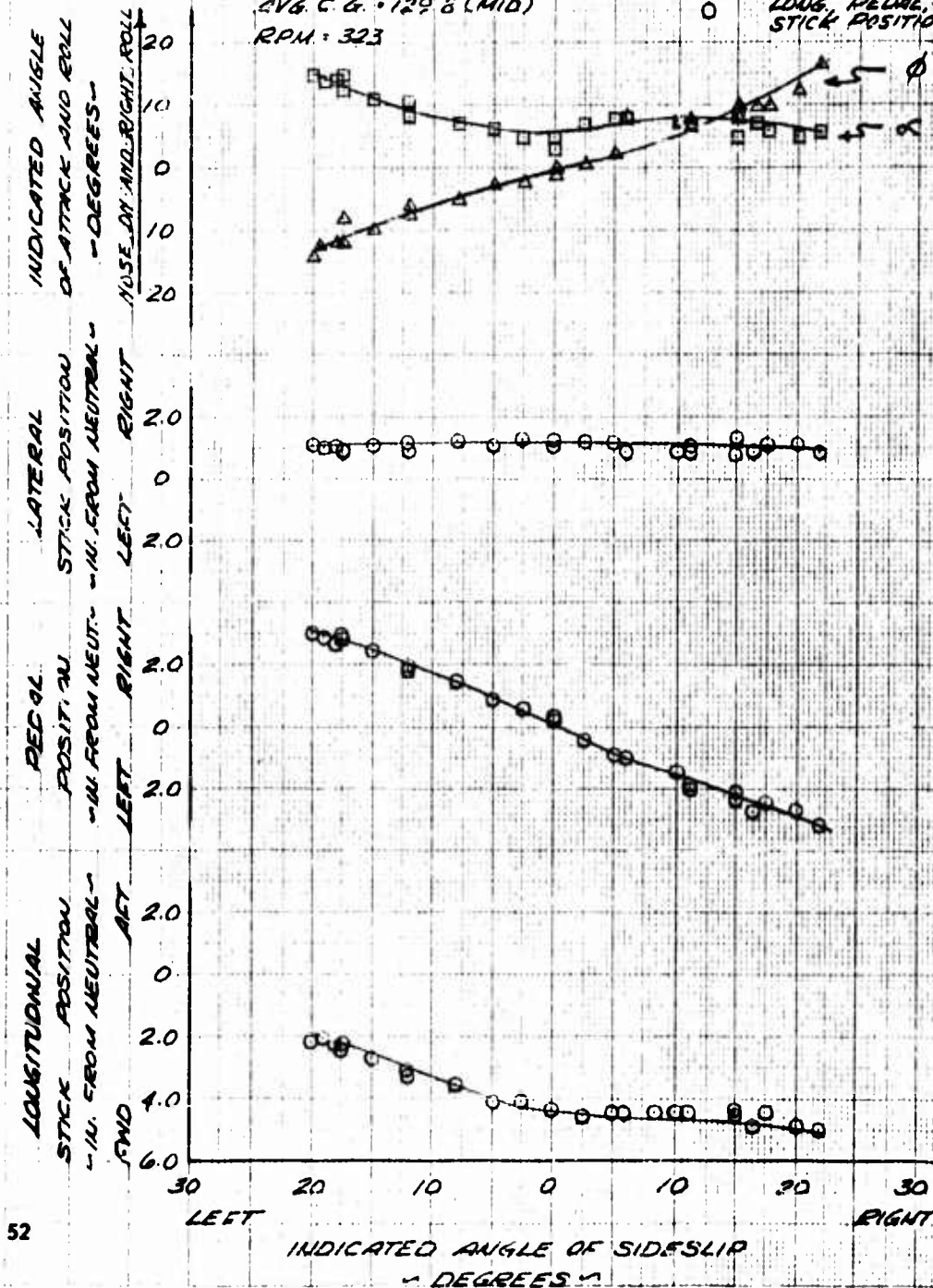


FIG. NO. 99
STATIC DIRECTIONAL STABILITY
 YHU-18
 S/N 58-2078
CLIMB

ROM, 323
 $V_T = 52.0$ KTS.
 $H_0 = 10000$ FT.
 AVG. G.W. = 6360 LB
 AVG. C.G. = 130.6 (INCH)

SYM.
 Δ ROLL, ϕ
 $\square$ ANGLE OF ATTACK, α
 $\circ$ LONG, PEDAL, LAT.
 STICK POSITION

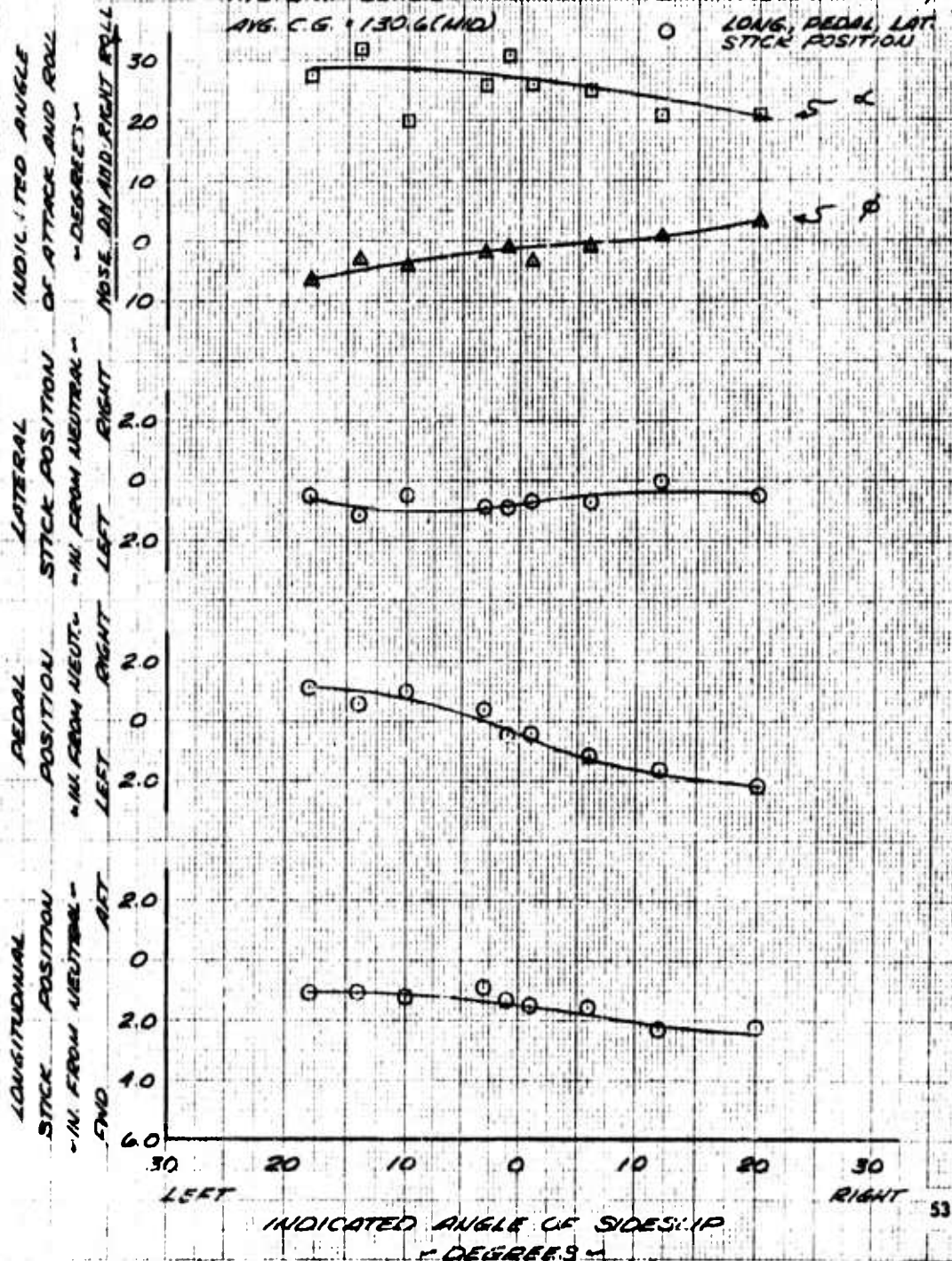


FIG NO. 3*

STATIC DIRECTIONAL STABILITY

YHU-1B SIN 58-2078

AUTOROTATION

$V_T = 52.0$ KTS.
 $H_D = 10000$ FT.
 AVG. G.W. = 6360 LB
 AVG. C.G. = 130.6 (INCH)
 RPM = 323

SYM

Δ ROLL, ϕ
 $\square$ ANGLE OF ATTACK, α
 $\circ$ LONG. PEDAL LAT. STICK POSITION

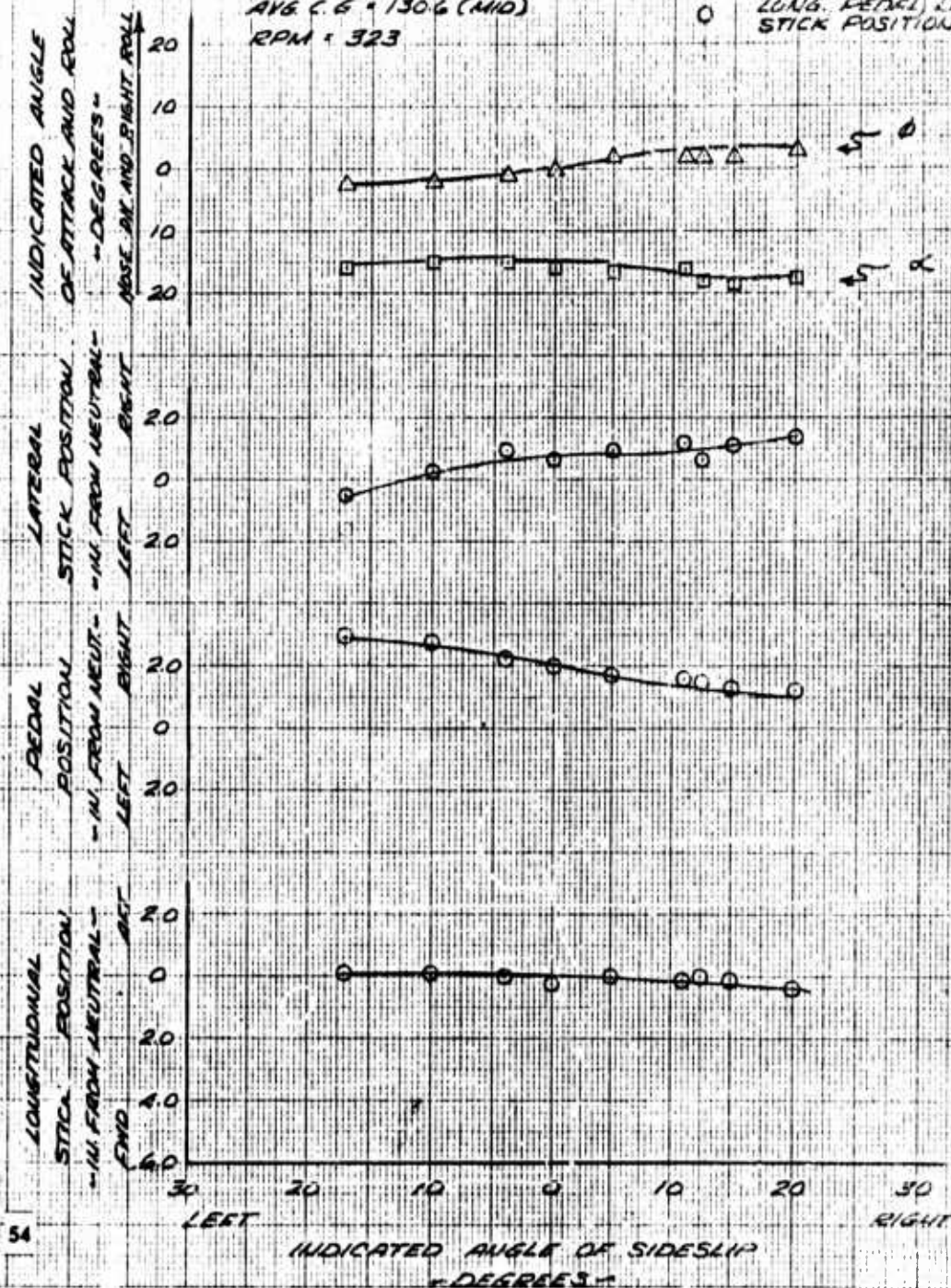


FIG. NO. 35
 DYNAMIC LONGITUDINAL STABILITY
 YH-13

S/N 58-2078

H₀ = 5000 FT.
 AVG. G.W. = 6700 LB
 AVG. C.G. = 130.0 (MID)
 RPM = 324

SYM	CONDITION
□	FWD CYCLIC
■	AFT CYCLIC

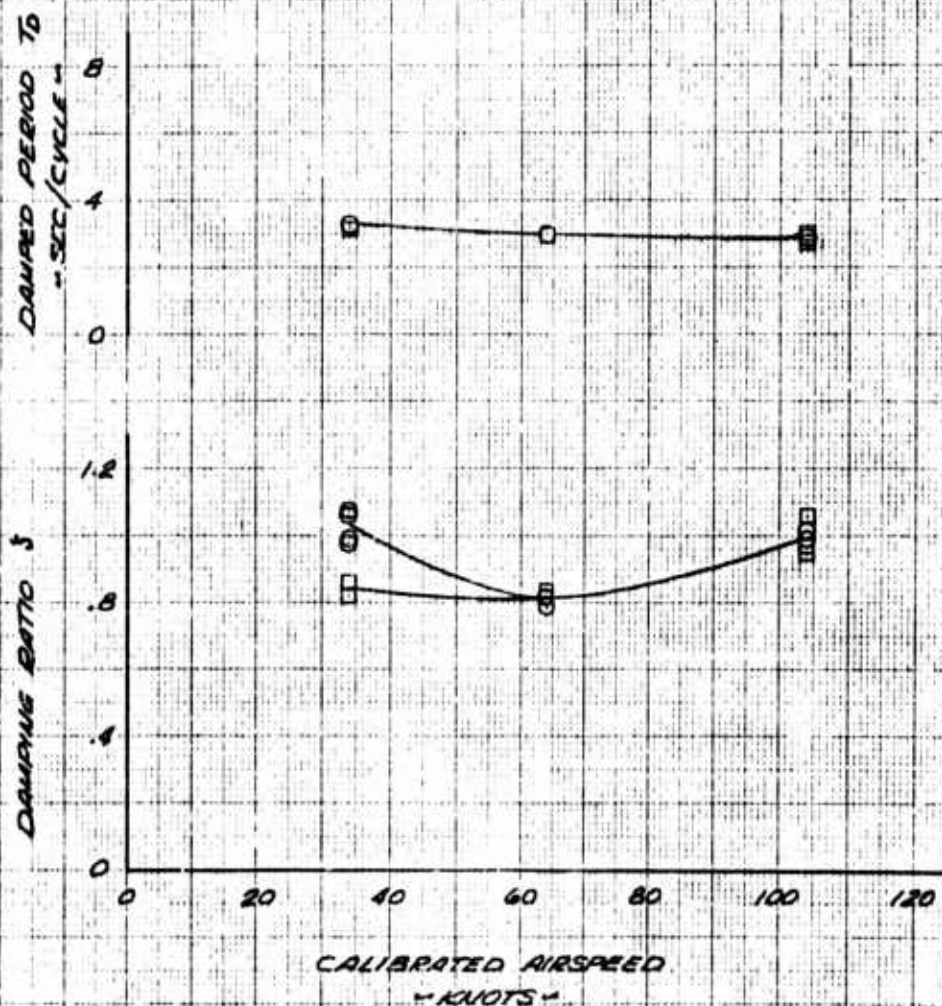


FIG. NO. 36
 DYNAMIC DIRECTIONAL STABILITY
 YHU-1B

S/N 58-2078

$H_0 = 5000$ FT.
 AVG. G.W. = 6700 LB
 AVG. C.G. = 130.0 (MID)
 RPM = 285 & 324

SYM. CONDITION
 ○ LEFT PEDAL
 □ RIGHT PEDAL

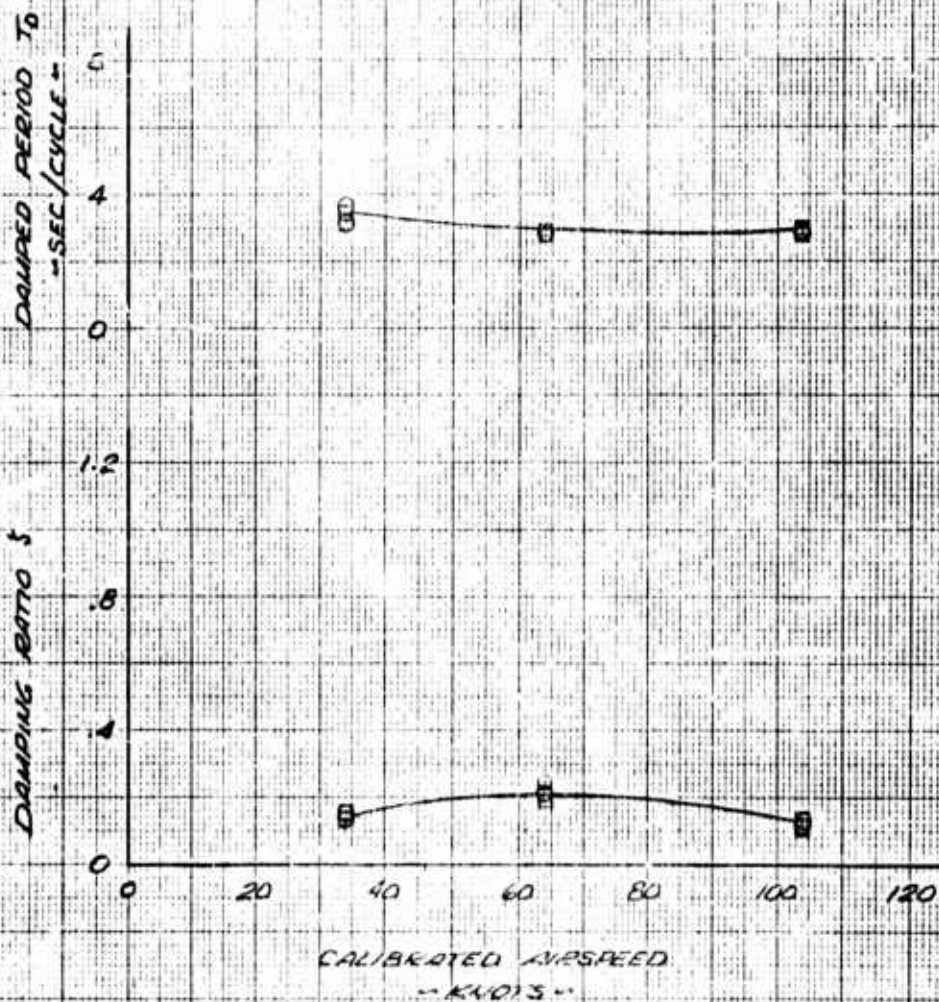


FIG. NO. 37
CONTROL POSITIONS IN SIDWARD FLIGHT
VH-1B

SN 58-2078

H₀ = 3000 FT.
AVE. G.W. = 7600 LB
RPM = 324
AVE. C.G. = 125.8 (FWD)
IN GROUND EFFECT

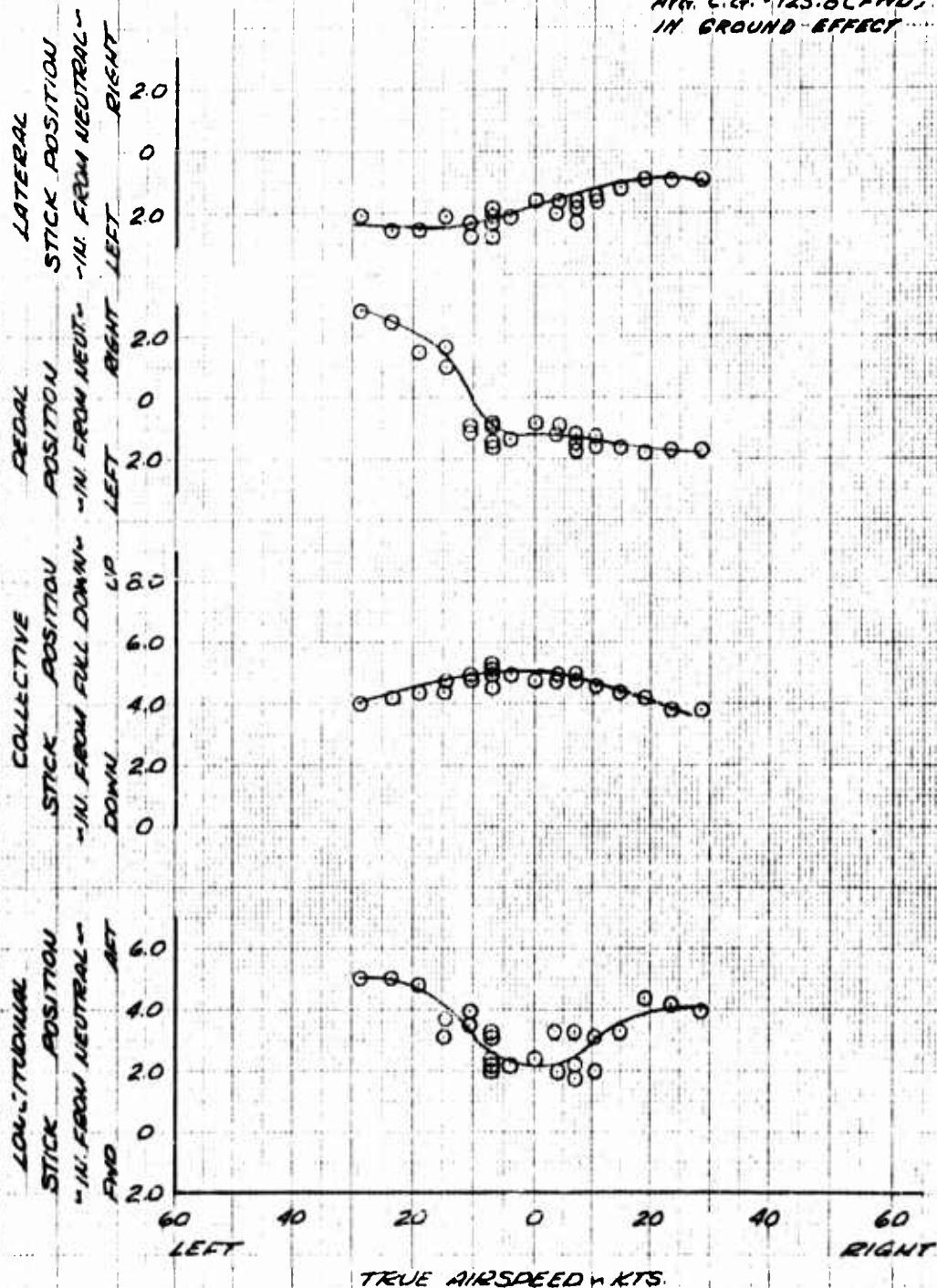


FIG. NO. 38 CONTROL POSITIONS IN FORWARD AND REARWARD FLIGHT

YHU-1B

S/N 58-2078

H_D = 3000 FT.
AVG. G. W. = 7600 LB
RPM = 324
AVG. C. G. = 125.0 (FWD)
IN GROUND EFFECT

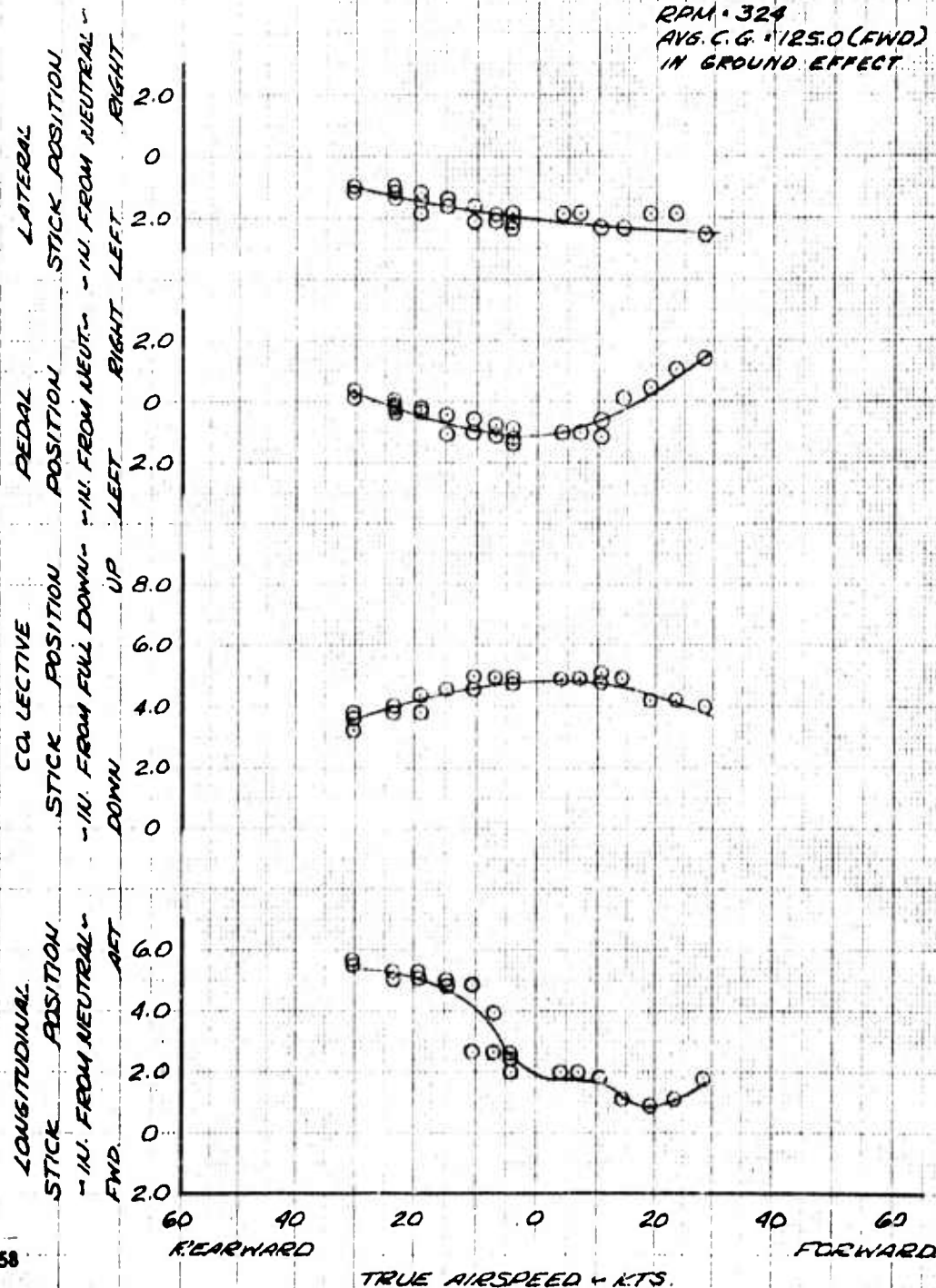


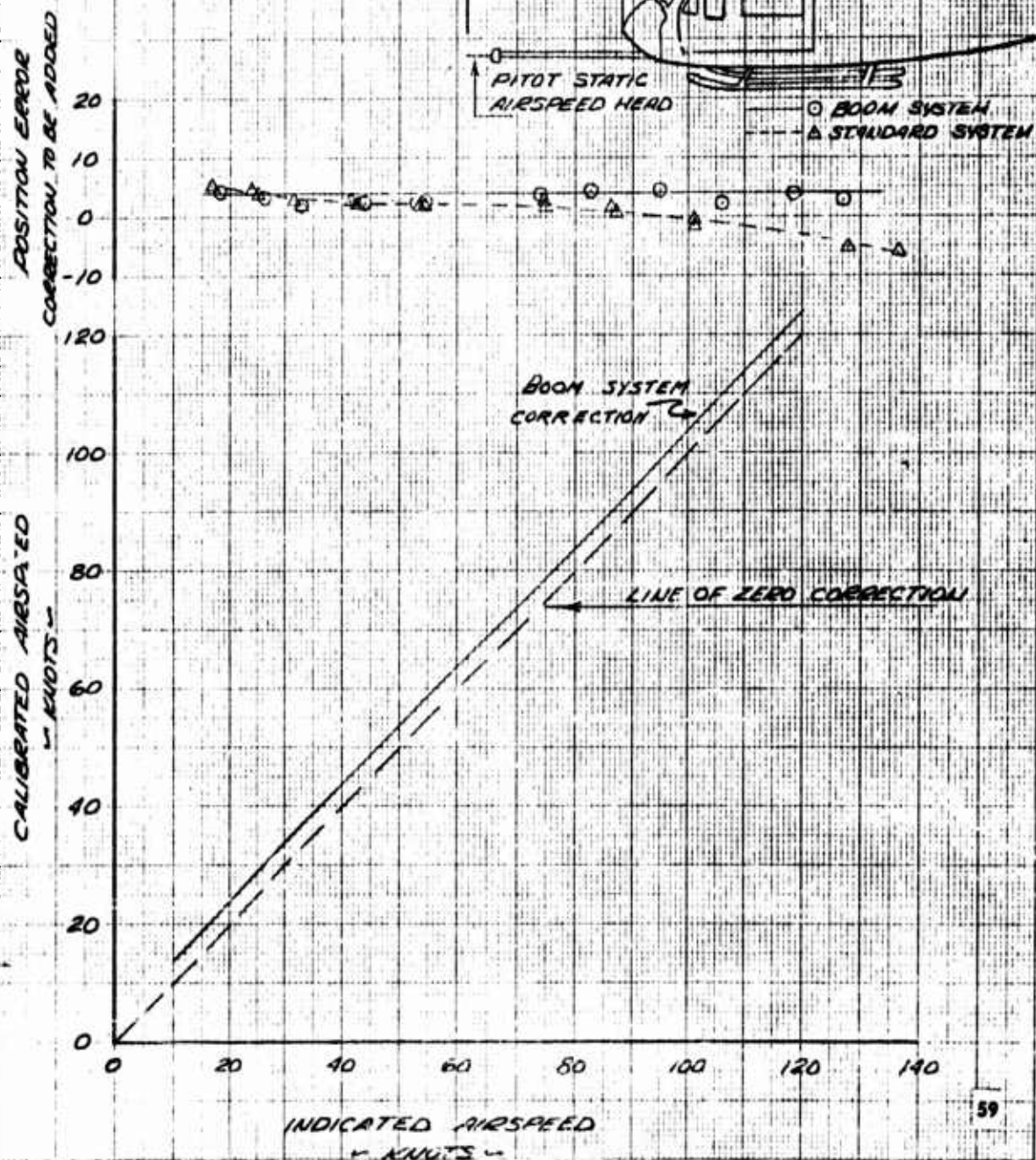
FIG. NO. 39
AIRSPED CALIBRATION
 YHU-1B S/N 58-2078

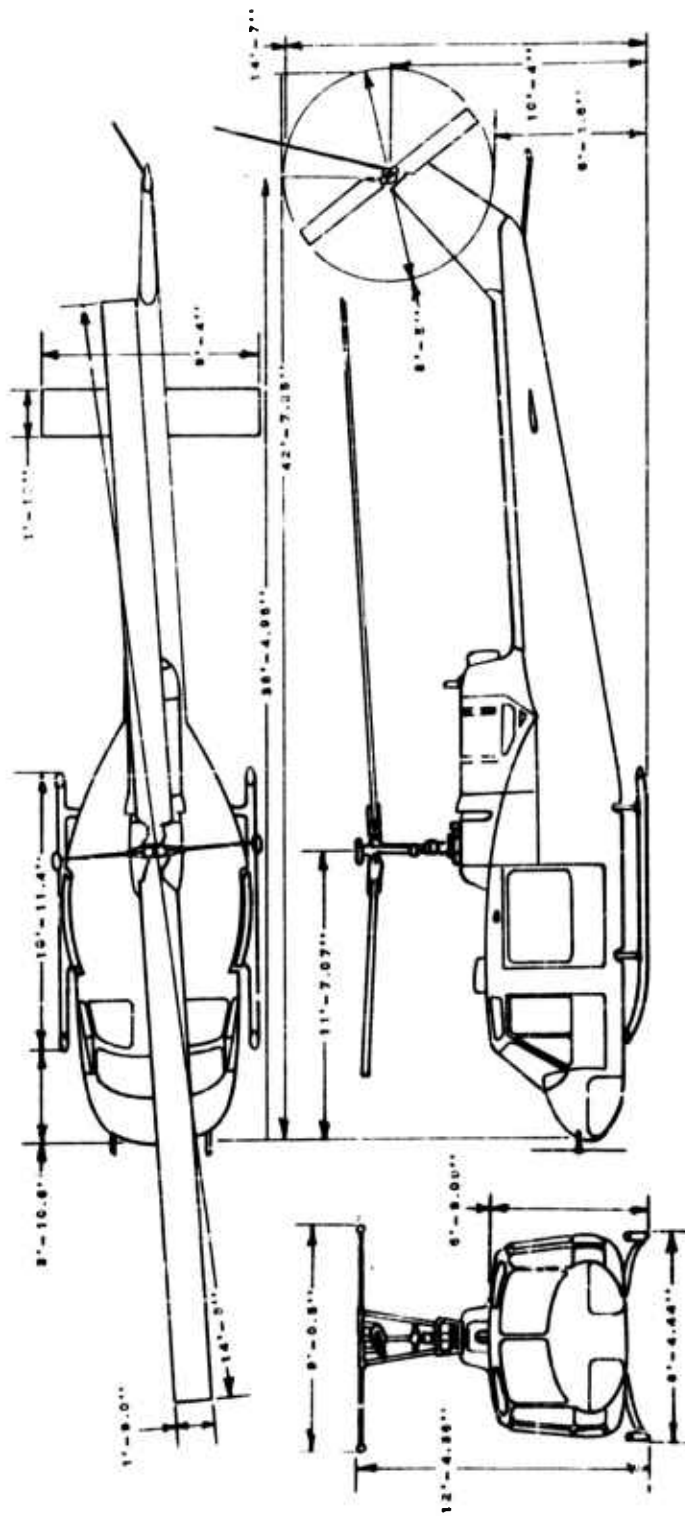
GROUND SPEED COURSE METHOD

AVG. G.W.: 6700 LB.

H₀ = 3000 FT.

ROTOR RPM = 314





APPENDIX II

general aircraft Information

DIMENSIONS AND DESIGN DATA

Overall Dimensions:

Aircraft length (nose to tail skid)	39.5 ft
Aircraft length (rotors turning)	54 ft
Width of skids	8.4 ft
Height (to top of turning tail rotor)	14.7 ft
Height (to top of rotor mast)	12.5 ft

Main Rotor:

Number of blades	2
Rotor diameter	44 ft
Rotor solidity	0.0509
Swept area	1520.5 ft ²
Blade area (each)	38.3 ft ²
Blade chord (root to tip)	21 in
Blade airfoil (root to tip)	NACA 0012
Flapping angle	±12 deg
Collective pitch angle limits (at 75 percent radius)	0 to +12 deg
Preconing angle	4 deg

Tail Rotor:

Number of blades	2
Rotor diameter	8.5 ft
Rotor solidity	0.105
Swept area	56.8 ft ²
Blade area (each)	2.98 ft ²
Blade chord (root to tip)	0.7 ft
Blade airfoil (root to tip)	NACA 0015
Blade twist	0 deg
Flapping angle	±8 deg

Gear Ratios:

Power turbine to engine output shaft	3.22 to 1
Engine output shaft to rotor	20.37 to 1
Engine output shaft to tail rotor	3.97 to 1

Flight Limits:

Forward Ait	Sta. 125.0
Rotor hub centerline	Sta. 138.0
Design minimum rotor speed (power on and power off)	Sta. 131.2'
Design maximum rotor speed for continuous operation (power on)	285 rpm
Maximum governed rotor speed for test aircraft	323 rpm
Maximum rotor speed for autorotation	323 rpm
Structural limit rotor speed (power on and power off)	330 rpm*
Limit dive speed	356 rpm
Design maximum side-ward speed	168 KTAS
Design maximum rear-ward speed	30 KTAS
	30 KTAS

*Changed to 339 rpm, Revision A to Design Specification.

Control Travel:

Cyclic, full forward to full aft	13 in
full left to full right	13 in
Pedal, full left to full right	7 in
Collective, full down to up	12.2 in

POWER PLANT

The test aircraft was equipped with a Lycoming T53-L-5 gas turbine engine S/N LE 03007. This engine is designed to produce 960 shaft horsepower for take-off at 6600 rpm (engine output shaft speed) with sea level standard day conditions.

For this test the fuel control was trimmed so the engine could produce 1100 shaft horsepower.

A torquemeter is installed integral with the reduction gearing of this engine. Torque is found by measuring the difference between the torquemeter hydraulic pressure and the inlet housing pressure. Lycoming calibrated the engine-torquemeter combination prior to delivery of the engine. The results of this calibration are presented as Figs. 1, 2 and 3 of this Appendix. The torquemeter calibration (Fig. 1) was used to determine power during the test program.

SYSTEMS

General:

The rotor and control systems and the engine fuel control are essentially the same as those of the earlier HU-1 series which is adequately described in AFFTC-TR-59-33.

Transmission:

The transmission consists of a single stage bevel gear and a two stage planetary gear train. This unit is connected to the engine output shaft, through a free wheeling unit, by a short drive shaft. Engine output shaft rpm is reduced to main rotor speed at a ratio of 20.37 to 1.

This transmission is designed to transmit 1100 shaft horsepower at 6600 rpm power turbine speed. Near the end of the program the transmission was limited because of a failure during the transmission qualification runs. At the time of this writing the following restrictions are in force:

990 SHP at 6600 rpm

120 knots maximum indicated
airspeed

WEIGHT AND BALANCE

The test aircraft was delivered partially instrumented. Therefore the aircraft was weighed fully instrumented. In this condition the aircraft was found to have a basic weight of 4870 pounds. The center of gravity was at station 138.7.

Tests were flown at weights ranging from 5800 pounds to the maximum internal load of 7660 pounds. Most tests were flown at a station 131.5 (mid) center of gravity, however two tests were flown at station 125.0 (forward) and one flight was made with the center of gravity located at station 138 (aft).

TEST INSTRUMENTATION

The instrumentation used during the tests was supplied, calibrated and maintained by the Instrumentation Branch of the Air Force Flight Test Center. The following sensitive instrumentation was installed:

Cockpit Instrument Panel:

Rotor rpm
Gas producer rpm
Exhaust gas temperature
High torque pressure
Low torque pressure
Outside air temperature
Total fuel used
Airspeed (ship's system)
Airspeed (boom)
Altitude (boom)
Angle of sideslip (boom)
Rate of climb
Stepper Motor

Photo-Recorder:

Outside air temperature
Compressor inlet temperature
Total fuel used
Airspeed (boom)
Altitude (boom)
High torque pressure
Low torque pressure
Combustor static pressure
Compressor discharge total pressure
Tailpipe static pressure
Rotor rpm

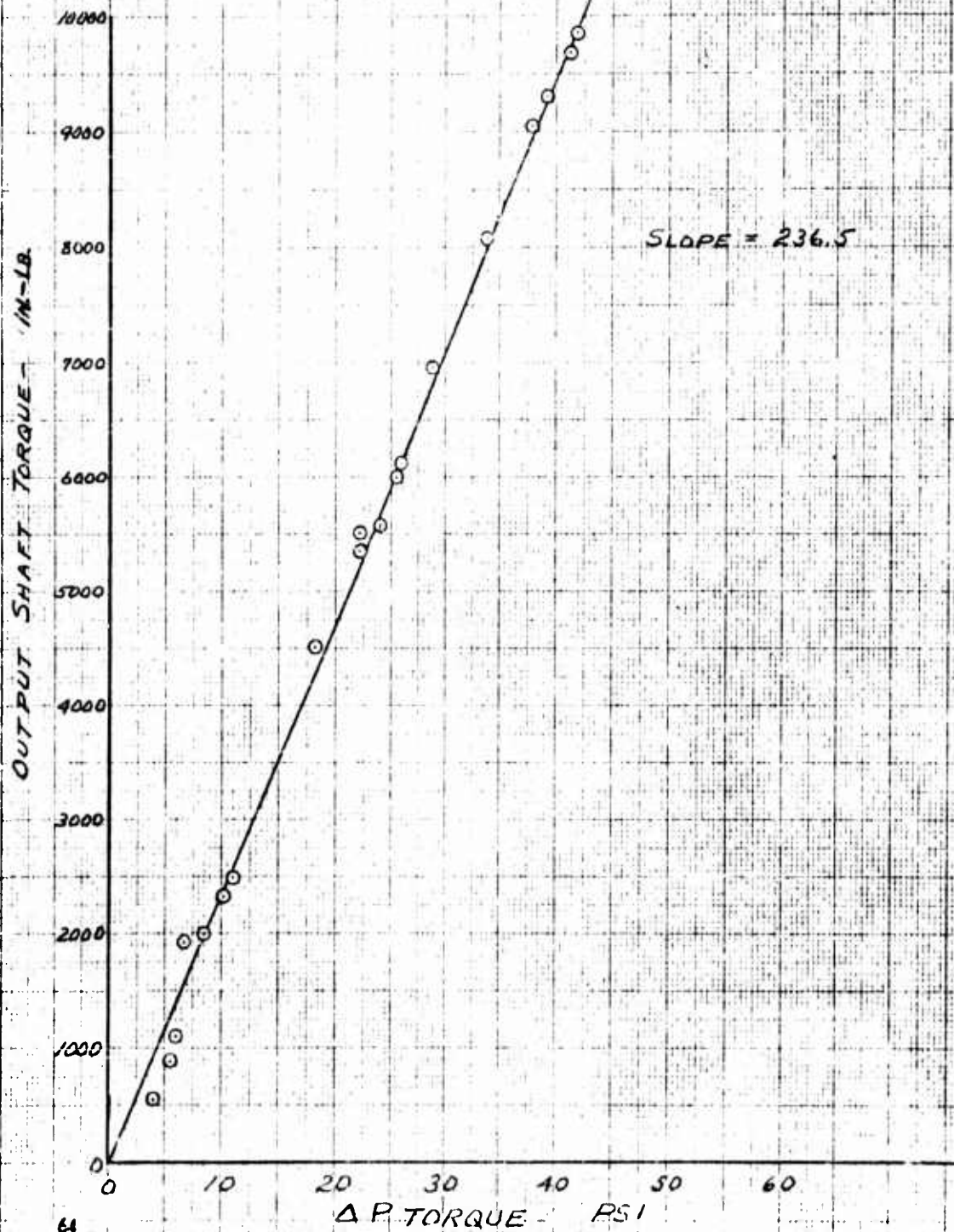
Gas producer rpm
Compressor inlet pressure static
Bellmouth inlet total pressure
Compressor discharge total
temperature
Exhaust gas temperature
Clock
Event marker

The following parameters were recorded
on the oscillograph:

Airspeed (boom)
Altitude (boom)
High torque pressure
Pedal position
Longitudinal stick position
Lateral stick position
Collective stick position
Angle of bank
Angle of pitch
Angle of turn
Rate of pitch
Rate of roll
Rate of yaw
Angular acceleration in pitch
Angular acceleration in roll
Angular acceleration in yaw
Total fuel used
Rotor rpm

ENGINE CHARACTERISTICS
T53-L5 S/N LE 63007
BASED ON TEST STAND
CALIBRATION
(LYCOMING DATA)

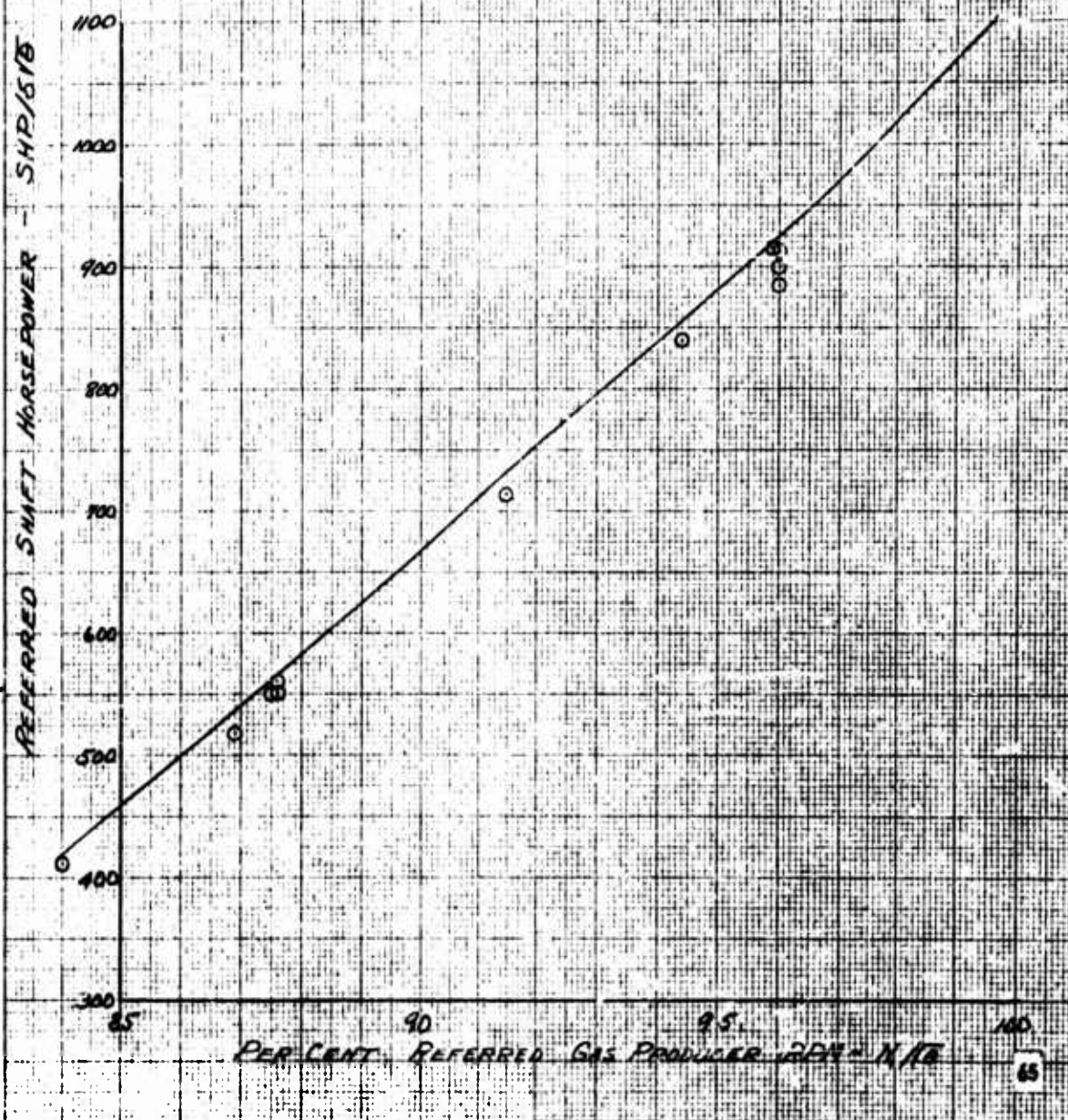
FIGURE 1



ENGINE CHARACTERISTICS
 T53-L5 S/N LE 03007
 BASED ON TEST STAND
 CALIBRATION
 (LYCOMING DATA)

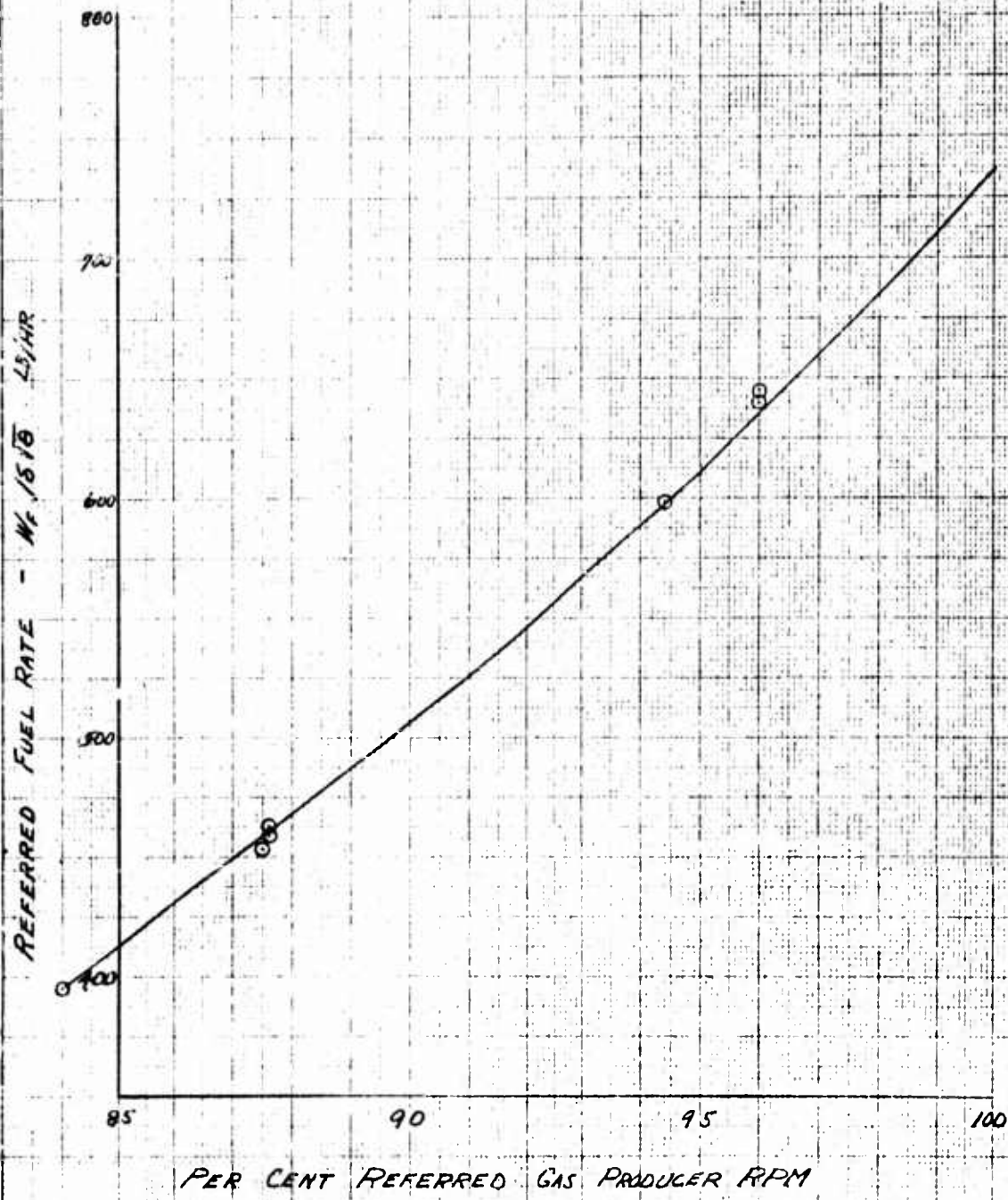
FIGURE 2

100% = 25150 RPM



ENGINE CHARACTERISTICS
T53-L5 S/N LE03007
BASED ON TEST STAND
CALIBRATION
(LYCOMING DATA)

FIGURE 3



APPENDIX III

test
data
corrected
for
instrument
error

Pilot CAPT. BALFE

Date 5 Oct 60

FSW 6.34 lb/gal.

Remarks

Pilot CAPT BALFE

Date 5 Oct 60

FSW 6.34 lb/gal.

[illegible]

Remarks

Pilot CAPT BALFE

Date 5 Oct 60

FSW 6.34 lb/gal.

[illegible]

Remarks

Pilot CAPT BALFE

Flight No. 4

Date 5 Oct 60

ESGW 6700 1b.

FSW 6.37 lb/gal.

[illegible]

Remarks

Pilot CAPT BALFE

Date 6 Oct 60

FSW 6.23 lb/gal.

Remarks

TEST DATA CORRECTED FOR INSTRUMENT ERROR
YHU-1B USA S/P 51-2076

Pilot *Chipi Balfi*

Test SPEED POWER

Flight No. 6

Date 6 Oct 60

ESGW 7625 1b.

FSW 623

lb/gal.

[illegible]

Репарка

Pilot CAPT BILFE

ESGW 76.74 lb. FSW 6.34 lb/gal.

[illegible]

Remarks

TEST DATA CORRECTED FOR INSTRUMENT ERROR
YHU-1B USA S/N 50-2076

Pilot CAPT BALLEE

Test	SPEED	POWER
------	-------	-------

Flight No. 8

Date 10 Oct 60

ESGW 7674 1b.

FSW 6.34 lb/gal.

[illegible]

Remarks

TEST DATA CORRECTED FOR INSTRUMENT ERROR
YHU-1B USA S/N 5P-20-6

Pilot Capt Baker

Test SPED Power Flight No. 10 Date 12 Oct 60

ESGW 6374 lb. FSW 629 lb/gal.

[illegible]

Remarks

TEST DATA COLLECTED FOR INSTRUMENT ERROR
YHU-1B USA S/N 58-2076

Pilot

Copy Right

Test SPEED Power

Flight No. 10

Date 13 Oct 60

ESGW 6374 1b.

FSW 6.29 lb/gal.

[illegible]

Remarks

Pilot, CAPT BALFE

ESGW 6360 lb. FSW 6.30 lb/gal.

Remarks

Pilot Capt BALFE

Date 14 Oct 60

FSW 6.30 lb/gal.

[illegible]

Remarks

Pilot CAPT BALFE

MSGW 6360 lb. PSW 6.30 lb/gal.

Remarks

Pilot CAPT BALLE

ESGW 6740 lb. FSW 6.28 lb/gal.

[illegible]

Remarks

Pilot CAPT BULFE

ESGW 6740 lb. FSW 6.28 lb/gal.

Remarks

Pilot CAPT BALFE

NSGW 5800 lb. FOW 6.28 lb/gal.

Remarks

Pilot CAPT BALFE

Date 25 Oct 69

1b/gal.

Remarks

Pilot CAPT BALFE

Date 1 Nov 60

10/21.

[illegible]

Remarks

TEST DATA CORRECTED FOR INSTRUMENT ERROR
YHC-1B USA S/N 5P-2076

Pilot CAPT BALFE

Test ENGINE CALIBRATION

Flight No. 21

Date 1 Nov 60

ESGW

16.

PSW

16/gal.

[illegible]

Remarks

Pilot Capt. BALLE

Date 1 Nov 60

1b/cel.

Remarks

Pilot CAPT BALFE

Flight No. 21

Date / Nov 60

FSW _____ lb/gal.

Remarks

TEST DATA CORRECTED FOR INSTRUMENT ERROR
YHQ-10 USA S/N 50-2074

Pilot Capt. RALPH

Test ENGINE CALIBRATION

Flight No. 21

Date 1 Nov 60

PLCW

1b.

PSM

lb/gal.

[illegible]

Remarks

Pilot Lt. Colvin

Date 19 APR 61

1b/gal.

Remarks

Pilot LT COLVIN

Date 14 APR 61

lb/gal.

Remarks

Pilot *Dr. Couvins*

Date 14 Apr 61

16/gal.

Remarks

Pilot Lt COLVIN

Date 19 APR 61

!h/{a}.

Remarks

Pilot CAPT BALFE

Date 25 April 61

1b/gal.

Remarks

Pilot CAPT BALFE

Date 25 APR 61

lb/gal.

Remarks

Pilot LT COLVIN

Date 17 May 1961

1b/gal.

Remarks

ASTIA DOCUMENT NO. AD-

Air Force Flight Test Center
Flight Test Engineering Division
Edwards AFB, California

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The test aircraft was a modified HU-1 with HU-1B dynamic components such as the rotor system, tail rotor, transmission, etc. No changes are programmed for the production HU-1B aircraft that will affect performance and stability.

The flying qualities of the YHU-1B are very good. In general, the flying qualities are improved over the earlier HU-1 series. This improvement stems primarily from the absence of the objectionable pitch and roll

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The helicopter meets all contractor guarantees for range, hovering, cruise speed, and service ceiling. However, it is felt that fuel capacity should be increased over the proposed 165 gallons to allow more adequate reserve for flight under instrument conditions. When compared to the HU-1A the YHU-1B has improved altitude performance, cruise speed, range and load carrying capabilities.

A general reduction in vibration is apparent with the YHU-1B. This is particularly significant at the higher airspeeds.

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