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CONTINUED ENGINEERING FLIGHT TEST OF THE YOH-6A HELICOPTER

FINAL REPORT
BY

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Project Pilot

MAY 1967

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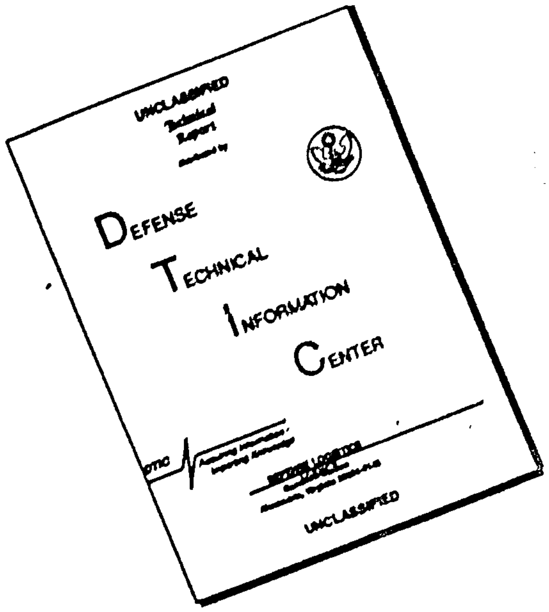

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U.S. Army Aviation Test Activity
Edwards Air Force Base, California

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ABSTRACT

As a result of an Army study requirement, a light observation helicopter design competition was initiated, prototypes were built and tested, a selection was made, and a production contract was awarded. Due to changes to the helicopter configuration required for type certification by the Federal Aviation Agency, additional testing was accomplished by the Army to provide data for the initial operator's manual.

Tests were conducted near Bishop, Bakersfield, and Edwards Air Force Base, California. Test sites used varied in field elevation from sea level to 11,500 feet. A total of 161 flights, accounting for 172.5 flight hours, were made from 27 January 1965 to 30 March 1966.

At design weight (2085 pounds), the YOH-6A had a significant increase in cruising airspeed and maximum airspeed compared with previous observation helicopters. Its climb performance was also good. Although the YOH-6A exhibited some handling quality shortcomings, it was described by the pilots as being very "agile." Takeoff and hovering performance were only fair and decreased rapidly with increasing weight or ambient temperature.

No deficiencies were found which would prohibit further testing or release for service use. Correction of the following shortcomings, however, would increase the flight safety and improve mission effectiveness of the OH-6A.

The height presented in the YOH-6A and OH-6A operator's manuals for safe autorotational landings after engine failure should be increased substantially until further height-velocity testing can be accomplished. Minimum touchdown speed practice autorotational landings should not be attempted. Collective pitch should not be lowered rapidly and aft cyclic control should not be applied after touchdown during autorotational landings to help prevent the main rotor blades from contacting the tail boom.

Airspeeds should be limited to those recommended in this report in cruising flight because of a pitchup tendency. Maneuvering stability was unacceptable near pitchup airspeed and airspeed should be reduced at least 10 percent below V_{ne} during maneuvering flight or flight in turbulent air. A description of the pitchup tendency and method of recovery should be included in the operator's manual.

The present airspeed limit of 10 knots for sideward and rearward flight at overload weights (above 2085 pounds) should be retained because of the small aft control margin remaining with a forward longitudinal center of gravity(C.G.). Optimum rotor speed and power setting should be determined near service ceiling in climbing flight in future tests since increasing power from maximum continuous to takeoff power decreased service ceiling at overload weight (2700 pounds). Flight in moderate to severe turbulence should be avoided. Sliding takeoffs can be made from suitable surfaces and will significantly increase the weight or altitude for takeoff. Sliding takeoffs should be investigated quantitatively in future OH-6A testing.

Starting capability (either air or ground) was marginal using internal battery power under all conditions above 2600 feet. The power turbine speed select switch was unsatisfactory and should be replaced with a more reliable thumb operated switch. Sufficient forward trim authority should be provided for all flight conditions. The rapid rate of build up of dirt in the compressor requiring frequent cleaning should be reduced or eliminated.

The change in the horizontal stabilizer (E.O. 6D-369-2504) had no significant effect in relieving instability in lightweight climbs. The "loss of power phenomenon" experienced previously in the YOH-6A was determined to be a significant decrease in tail rotor power requirements with forward flight.

FOREWORD

Authority for this test was provided by reference a and b. The U.S. Army Aviation Test Activity (USAAVNTA) was responsible for preparing the test plan, executing the tests, and preparing the test report.

In addition to the responsibilities of the authors in this test program, Mr. William A. Anderson and Captain Donald P. Wray assisted as pilots and Mr. Connie Statum assisted as engineer.

Aircraft maintenance was provided by the Maintenance Branch of the Logistics Division, USAAVNTA. Mr. Paul Meyers was the assigned crew chief. He was assisted at various times by Mr. Charles F. Blum, Jr., Mr. Thomas D. Dye, or Mr. Charles E. Benner.

Instrumentation was installed and maintained by the Instrumentation Branch of the Logistics Division, USAAVNTA. Mr. Fred J. Menick was the assigned instrumentation analyst. He was assisted at various times by Mr. Adam R. Stickles, Mr. James K. Slack, and Mr. Billy G. Roberts.

Data reduction and analysis, and report preparation were accomplished by the Engineering Division, USAAVNTA. Mrs. Mildred L. Christopherson was assigned as the engineering technician. She was assisted by Mr. Larry O. Deeds.

SECTION 1 INTRODUCTION

1.1 BACKGROUND

As a result of Army Study Requirement 1-60 on Army light observation aircraft, a decision was made to use light observation helicopters and to phase light observation airplanes out of the Army inventory. The Light Observation Helicopter (LOH) Design Competition was initiated on 14 October 1960. Three prototype designs were tested by the U.S. Army during the last half of FY 1964 for the purpose of selecting the most suitable design for Army use. The OH-6A was selected and a production contract was awarded in May 1966. It was decided to obtain data on the prototype OH-6A so that a more complete operator's manual for the production OH-6A could be released for service use as soon as aircraft production began.

The U.S. Army Test and Evaluation Command (USATECOM) issued a test directive to USAAVNTA, 14 January 1965, to develop the additional data required to complete the operator's manual for the OH-6A helicopter. Plan of test for the completion of engineering flight test of the OH-6A helicopter was submitted by USAAVNTA in January 1965, and as modified 8 March 1965, and 6 April 1965, was approved by the LOH Project Manager, U.S. Army Materiel Command (USAMC). The Deputy Chief, LOH Field Office, 3 August 1965, requested that the OH-6A (prototype) be designated the YOH-6A.

Testing was conducted at Edwards Air Force Base, Bakersfield, and Bishop, California. A total of 161 flights, including ferry flights, accounting for 172.5 flight hours were made from 27 January 1965 to 30 March 1966.

1.2 DESCRIPTION OF MATERIEL

The YOH-6A is a single-main-rotor, four-place, light observation helicopter. The four-bladed main rotor is fully articulated. The control system is conventional and completely unboosted. No automatic stabilization equipment is employed. The YOH-6A is powered by a T63-A-5 free turbine engine derated to 250 shaft horsepower (SHP) at 6000 rpm (limited by main transmission torque) for takeoff power. Provisions are made for installation of various armament kits.

Photo 1 - YOH-6A prior to modification order
E.O. 369-2264 and E.O. 6D-369-2504

The YOH-6A was modified subsequently to previous YOH-6A tests and prior to this test by engineering order (E.O.) 369-2264, "Installation of Cooling Air Scoops," Photo 2 and Photo 3, (reference h) and E.O. 6D-369-2504, "Modified Horizontal Stabilizer Assembly," Photo 4, (reference i).

A more complete description of the aircraft can be found in appendix III.

Photo 2 - Cooling air scoops

Photo 3 - Cooling air scoops

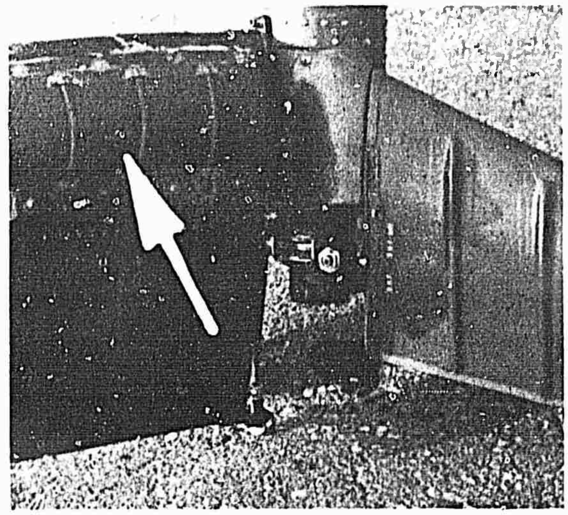


Photo 4 - Modified horizontal stabilizer assembly

1.3 TEST OBJECTIVES

The objectives of this test program were to: (a) develop the additional data required to complete the operator's manual for the OH-6A helicopter; (b) verify test data obtained during the original YOH-6A program (references j and k); and (c) determine the effects of the external modifications to the YOH-6A (E.O. 369-2264 and E.O. 6D-369-2504).

1.4 SUMMARY OF RESULTS

The significant results of the YOH-6A operator's manual tests are summarized in the following paragraphs.

1.4.1 Takeoff

At the design gross weight (2100 pounds) on a 35-degree-centigrade (C) day, the YOH-6A could take off vertically (hover out of ground effect (OGE) at pressure altitudes up to 3280 feet. It could make a running takeoff without skid-ground contact from a level surface (hover at 2 feet) at 6000 feet pressure altitude and clear a 50-foot obstacle in approximately 600 feet. At maximum overload gross weight (2700 pounds), vertical takeoffs could not be made at any conditions of altitude and temperature. Running takeoffs without skid-ground contact could be made up to 3390-foot pressure altitude on a standard day. On a 35-degree-C day, takeoffs without skid-ground contact were not possible.

Of the two takeoff techniques tested, the level acceleration technique was considered better than the climb and acceleration technique for four reasons: (a) it required less takeoff distance to clear a 50-foot obstacle at conditions under which the helicopter could not hover OGE; (b) it required less time in the "avoid" area of the height-velocity diagram (height and airspeed from which a safe landing could not be made if engine failure occurred); (c) it decreased the possibility of "fall through" due to the climbout speeds being too low: and (d) it was easier and required less experience to fly correctly.

Optimum climbout (minimum distance) airspeed for running takeoffs varied from approximately 15 knots true airspeed (TAS) at conditions under which the YOH-6A had just sufficient power to hover OGE to approximately 30 knots TAS at conditions under which the helicopter could only hover at 2 feet. When takeoff area was not restricted, however, a climbout airspeed of approximately 30 knots indicated airspeed (IAS) is recommended for three reasons: (a) this was the minimum airspeed for which the airspeed system was reliable. Below this speed airspeed had to be estimated from wind-corrected ground speed. (b) there was no possibility of "fall through." "Fall through" occured at speeds below optimum climbout speed. As the 2-foot hovering ceiling was approached, optimum climbout airspeed approached this recommended climbout airspeed. (c) with a 30-knot IAS climbout, no time was spent in the "avoid" area of the height-velocity diagram when the level acceleration technique was used. With the climb and acceleration technique, minimum time was required in the "avoid" area.

Sliding takeoffs could be made at weights, altitudes, and ambient temperatures at which the YOH-6A could only hover light on the skids at takeoff power. This required, however, that more than 1000 feet of smooth level surface be available. This ability significantly increased the weight and/or altitude at which takeoffs could be made.

The apparent loss-of-power phenomenon experienced during takeoffs in earlier programs with the YOH-6A was found to be caused by a significant decrease in tail-rotor power required with forward flight when transitioning from a hover to forward flight.

1.4.2 Hover

Hovering performance obtained during this program agreed with data obtained during the original YOH-6A performance tests (reference k). While the YOH-6A was hovering in winds greater than

5 knots from varying directions, hovering height could not be maintained. At conditions under which the helicopter had just sufficient power to hover OGE, a rate of climb or a rate of descent sufficient to cause ground contact would develop as wind velocity or direction changed.

1.4.3 Climb

At 2085-pound gross weight and takeoff power on a standard day, the sea-level rate of climb was 1950 feet per minute (fpm) and the service ceiling was 19,700 feet. At maximum overload gross weight (2700 pounds), the sea-level rate of climb was 1170 fpm; however, the service ceiling of 10,500 feet was 1100 feet lower than that obtainable with maximum continuous power. This decrease was apparently due to increased blade stall with the increased collective needed to obtain takeoff power.

The change to the horizontal stabilizer (E.O. 369-2264) did not significantly reduce the dynamic longitudinal instability experienced during high rates of climb at optimum climb airspeed at weights of 2085 pounds and less. This instability resulted in airspeed excursions up to 10 knots calibrated airspeed (CAS). Climb performance was probably not affected significantly because of the small effect airspeed had on rate of climb at the conditions at which the instability occurred. Increasing airspeed above 60 knots CAS or reducing rate of climb eliminated the instability.

At 2700 pounds and 2085 pounds gross weight, a lateral cyclic stick vibration which increased in amplitude as service ceiling was approached was encountered. It occurred at a frequency of approximately 32 cycles per second (cps) (4 per rotor revolution), and at service ceiling the stick displacement due to vibration was greater than 1/2-inch double amplitude at the grip.

During the climbs a lack of adequate forward stick force trim authority was noted. Full trim authority was reached at approximately 60 percent of service ceiling and a maximum forward stick force of 8 to 10 pounds was required at service ceiling.

In climbing flight, pitchup airspeed appeared to be lower than for level flight (see Level Flight, paragraph 1.4.4).

1.4.4 Level Flight

The modifications (external oil cooler airscoops and modified horizontal stabilizer) to the YOH-6A resulted in approximately 3-knots TAS decrease for a constant power at recommended cruise airspeed. The range summary plot, figure 25, reference k, is incorrect and the method used to derive it gives erroneous results. (The range summary presented in the present report should be used).

The XM-7 armament kit resulted in an increased drag of .7 square feet equivalent flat plate area. This agreed with the findings in reference k. This increased drag resulted in less than a 5-knot TAS reduction at maximum continuous power.

Sideslip was found to have a significant effect on power required. In left sideslip, a slight decrease in power required was apparent. In right sideslip a fairly large increase in power required was evident. This condition should be investigated during future tests. A method of determining sideslip to insure best cruise performance was not provided. The pilot should be given some method of determining sideslip to insure best cruise performance. A yaw string would probably suffice.

Pitchup was present in unaccelerated level flight in smooth air at airspeeds near the contractor-published never-exceed airspeed (V_{ne}). A lower recommended V_{ne} based on 90 percent of the speed for pitchup is presented in figure 50. This recommended V_{ne} should be used in the operator's manual. This V_{ne} limited best cruise airspeed at the heavier weights and higher altitudes. At 2100 pounds gross weight and less, recovery from pitchup had to be initiated immediately to "avoid" extreme nose-high attitudes and excessive load factor (g). At heavy gross weight (2600 pounds), ample vibration warning occurred prior to pitchup. At 2100 pounds gross weight and less, little or no vibration warning was given. Pitchup appeared to be a deterioration of already poor angle-of-attack stability which was aggravated by blade stall. Forward center of gravity (C.G.) and low rotor speed decreased the airspeed at which pitchup occurred. In turbulent air or maneuvering flight, pitchup airspeed was lower than for level flight in smooth air. For flight in turbulent air or maneuvering flight, airspeed should be reduced at least 10 percent below that shown in figure 50. In climbing flight, pitchup airspeed appeared to be lower than for level flight. The pitchup tendency and method of recovery should be included in the operator's manual.

Very low damping and poor dynamic stability, particularly in the yaw axis, necessitated continuous pilot attention while flying in light turbulence. In moderate to severe turbulence, 100-percent pilot effort was required to maintain airspeed, altitude, and heading during cruising flight. Flight in moderate to severe turbulence should be avoided.

1.4.5 Autorotational Descents

Extrapolation of the data obtained indicated that at gross weights less than approximately 1400 pounds at sea level rotor speed could not be maintained above minimum power-off rotor speed (400 rpm). At gross weights heavier than approximately 2450 pounds at sea level, full-down collective could not be used during stabilized autorotational descent since the maximum limit power-off rotor speed (514 rpm) would be exceeded.

1.4.6 Power-on Landings

Although limited data were obtained, it was found that power-on landings could be made in shorter distances than required for takeoffs at any given conditions. Out-of-ground-effect hover capability was necessary, however, to prevent hard landings at approach airspeeds less than 20 knots CAS.

1.4.7 Autorotational Landings

Only a limited amount of autorotational landing (height-velocity) data were obtained because testing was terminated after an accident which occurred during an autorotational landing. Major damage to the aircraft resulted when a rotor blade struck the tail boom after touchdown.

It appeared that the heights presented in the YOH-6A and OH-6A operator's manuals (references l and m) for safe autorotational landings could not be met using the technique required by MIL-H-8501A (reference n). These height-velocity diagrams were developed using the Federal Aviation Agency (FAA) technique which requires only a 1-second delay between throttle chop and lowering the collective and does not require minimum touchdown airspeed. Autorotational landings from minimum heights require extremely good judgment by pilots with substantial experience in the aircraft. Aft cyclic stick application and rapid lowering of collective at touchdown can cause the main rotor blades to strike the tail boom. Touchdown airspeeds of 15 knots or less may not be practical in the OH-6A.

With a 2-second delay (required by MIL-H-8501A) between simulated power failure and lowering collective, rotor speed will fall below the minimum power-off limit (400 rpm). The 2-second delay is realistic since the only indication of power failure is decreasing rotor speed and a very slight yaw to the left. The engine-out warning system is essentially useless except for practice autorotations.

1.4.8 Sideward and Rearward Flight

At approximately 2500 pounds at forward longitudinal and left lateral C.G. above 17 knots to the left and 25 knots to the rear, an average of less than 10-percent aft longitudinal control remained. Above 9 knots to the left or to the rear, intermittent contact was made with the aft control stop. Continual cyclic and pedal inputs of up to ± 2 inches were required to maintain speed and heading in sideward and rearward flight.

1.4.9 Maneuvering Stability

Maneuvering flight characteristics were satisfactory at an airspeed of 55 knots CAS at the 2100-pound gross weight tested. Under these conditions, positive stick position and stick force gradients were exhibited. These gradients became less with increasing airspeed. They became unsatisfactory above 95 knots CAS, where the stick position gradient became zero and a negative stick force gradient (push force with increasing g) was encountered.

This deterioration in maneuvering stability was caused by the pitchup exhibited by this aircraft. This condition detracts from the capability of the aircraft since the pilot must devote his attention to maneuvering the aircraft rather than to performing the mission.

1.4.10 Engine Acceleration Characteristics

Insufficient data were obtained to make summaries or draw conclusions. The engine response of the YOH-6A, however, was generally very poor, particularly when compared with the aircraft response to other controls.

1.4.11 Engine Starts

Air starts were marginal under all conditions and required a good battery and a recently cleaned compressor. Ground starts were marginal at 4200 feet and above using external power. When internal battery power was used, ground starts were marginal at 2600 feet, marginal to impossible at 4200 feet, and impossible at 11,500 feet. The standard turbine outlet temperature (TOT) indicator had a greater lag than the test indicator; this resulted in lower apparent TOT's and less time at these temperatures. A more responsive TOT indicator should be installed to give the pilot a more accurate indication of TOT during transient conditions (starting and accelerations).

1.4.12 Airspeed Calibration

The standard airspeed system was stable and reliable above approximately 25 knots CAS. The position error remained the same in climbs, level flight, and descents. It varied from 4 knots at 30 knots IAS to zero at 90 knots IAS and remained zero up to the maximum airspeed tested. The airspeed calibration made during the reference k tests was in error because the test airspeed boom affected the position error and was not removed. The test boom was removed for this calibration.

1.4.13 Maintenance Problems

In general, little maintenance was required on the YOH-6A. The engine required the most maintenance since the compressor had to be cleaned approximately every 30 hours. In addition, the compressor discharge pressure (CDP) filter had to be replaced every 5 to 10 hours. Two engines failed during the test. One engine was removed after cracks were found in the plastic compressor liner. A second engine was removed after large metallic chips were found on the engine chip detector plug. It was suspected that the main power turbine bearing had failed. All of the fuel control-engine combinations used had different starting characteristics.

Continual problems were experienced with the power turbine speed selector switch.

1.5 CONCLUSIONS

At design gross weight (2085 pounds), the YOH-6A had a significant increase in cruising airspeed and maximum airspeed compared with previous observation helicopters. Its climb performance was also good, with a service ceiling slightly less than 20,000 feet and a sea-level rate of climb slightly less than 2000 fpm. (paragraphs 2.2.4 and 2.2.3)

Although the YOH-6A exhibited some handling quality shortcomings, it was described by the pilots as being very "agile". This is because control response was rapid and positive and control forces remained light throughout the majority of the flight envelope.

The YOH-6A includes other good features. Its weight was decreased approximately 600 pounds while the same useful load as previous observation helicopters was retained. It has an internal cargo area or a four-place seating capability. Its length and rotor diameter are less than other observation helicopters. Relatively little maintenance was required by the airframe because of the lack of hydraulic and stability augmentation systems and the exclusive use of bearings which did not require periodic lubrication. Apparent vibration at design gross weight was very low because of the relatively high predominant frequency (32 cps). Except for the tail rotor noise, the YOH-6A would be relatively quiet.

The takeoff and hovering performance deteriorated very rapidly with increasing temperature, primarily due to the decrease in power with increasing temperature. Performance also decreased rapidly with increasing weight. At 2500 pounds gross weight, the YOH-6A could not hover OGE at any altitude or temperature and could not be operated safely near the ground in crosswinds or tailwinds greater than 10 knots. At 2700 pounds (maximum allowable gross weight), takeoff distance to clear a 50-foot obstacle would be greater than 600 feet on a 35-degree centigrade (C) day at sea level. Service ceiling was 10,500 feet. During climbs at 2700 pounds gross weight, a large-amplitude lateral cyclic stick vibration would occur at a frequency of 32 cps above 6000 feet.

Stability and controllability deteriorated with increasing weight. Very low damping and poor dynamic stability, particularly in the yaw axis, necessitated continuous pilot attention while flying in light turbulence. In moderate to severe turbulence, 100-percent pilot effort was required to maintain airspeed, altitude, and heading during cruising flight.

At conditions under which the helicopter had just sufficient power to hover OGE, hovering height could not be maintained while hovering in winds greater than 5 knots from varying directions. A rate of climb or a rate of descent sufficient to cause ground contact would develop as wind velocity or direction changed.

At airspeeds near the contractor-published V_{ne}, pitchup occurred in level unaccelerated flight in smooth air. During pitchup, extreme pitch attitudes may develop and limit load factors may be exceeded. This pitchup tendency at 2100 pounds and 5000 feet density altitude causes the maneuvering stability to be unacceptable above 95 knots CAS.

Safe autorotational landings would be difficult or impossible for the average pilot from the entry heights and airspeeds presented in the YOH-6A or OH-6A operator's manual. These height-velocity diagrams were developed using the FAA technique which requires only a 1-second delay and does not require minimum touchdown airspeed. Using the 2-second delay between throttle chop and lowering collective (required by MIL-H-8501A) the minimum rotor speed limitations were exceeded. Autorotational touchdown speeds of less than 15 knots (required by MIL-H-8501A) may not be practical in the OH-6A. Rapid lowering of collective and aft cyclic stick application during touchdown may cause the main rotor blades to strike the tail boom. The engine-out warning system is essentially useless except for practice autorotations.

Airstart capability was marginal at all conditions. Ground starts using the aircraft's battery were marginal above 2600 feet and impossible above 4200 feet.

The hovering performance determined during this program agrees with that of the original YOH-6A program (reference k). The climb instability experienced in that program was not alleviated by the changed horizontal stabilizer. The range performance presented in reference k is not correct. The longitudinal C.G's presented in both references j and k are incorrect. An error in the basic weight and balance determination of both aircraft was found during this program. The problem is explained in detail in appendix III.

Additional shortcomings and undesirable characteristics are included in section 1.6 "Recommendations."

1.6 RECOMMENDATIONS

1.6.1 The following deficiencies in the operator's manual should be corrected to reduce possible safety of flight conditions:

a. The minimum height for safe autorotational landing (height-velocity curve) presented in the OH-6A operator's manual should be doubled until further height-velocity testing using the standard "2-second delay-minimum touchdown airspeed" technique required by MIL-H-8501A can be accomplished. (paragraph 2.2.7)

b. A complete description of the technique used to establish each area of the FAA height-velocity curve in the operator's flight manual should be included in the operator's manual. (paragraph 2.2.7)

1.6.2 The following warning notes should be included in the operator's manual to preclude possible damage to the aircraft:

a. Minimum touchdown speed practice autorotational landings should not be attempted. (paragraph 2.2.7)

b. After touchdown, during autorotational landings, collective should not be lowered rapidly and aft cyclic stick should not be applied except to prevent impending nose-over. These procedures will help prevent the main rotor blades from striking the tail boom. (paragraph 2.2.7)

c. At speeds near V_{ne}, the YOH-6A exhibits a pitchup tendency. The speed at which pitchup occurs will be lower in turbulence or when maneuvering. If pitchup occurs, recovery should be made immediately by lowering collective and applying sufficient forward stick force to maintain a level attitude. (paragraph 2.3.3)

1.6.3 The following caution notes should be included in the operator's manual to provide safer aircraft operation:

a. Flight in moderate to severe turbulence should be avoided. (paragraph 2.3.3)

b. The airspeed limit should be reduced at least 10-percent below that shown in figure 50 for flight in turbulent air or maneuvering flight. (paragraph 2.3.3)

c. Vertical oscillations during OGE hovering flight at maximum power of sufficient magnitude to cause ground contact occur when hovering in gusty wind conditions. (paragraph 2.2.2)

d. At conditions under which the helicopter cannot hover OGE, approaches for landing should be made at 20 knots CAS or above. (paragraph 2.2.6)

e. Hovering in crosswinds and downwinds (sideward and rearward flight) should not be accomplished in winds in excess of 10 knots at gross weights above 2085 pounds. (paragraph 2.3.1)

1.6.4 The following changes should be incorporated immediately to correct shortcomings in the aircraft:

a. Increase forward cyclic stick trim authority. (paragraph 2.2.3)

b. Install a low (minimum-power-on) and high (maximum-power-on) rotor speed warning system in the OH-6A to provide a more rapid and adequate engine-failure warning system than the present low gas producer (N_1) speed indication. (paragraph 2.2.7)

c. Incorporate a twist grip override to prevent operation during engine starts and practice autorotations. (paragraph 2.2.7)

d. Retain the present low N_1 speed warning system to provide engine-out warning during practice autorotations. (paragraph 2.2.7)

1.6.5 The following information should be included in the operator's manual:

a. The level acceleration takeoff technique should be used at conditions under which the helicopter cannot hover OGE since it results in the shortest distance to clear 50 feet and requires the least time in the "avoid" area of the height-velocity curve. (paragraph 2.2.1.2)

b. A minimum climbout airspeed of 30 knots IAS should be used when maximum performance (shortest distance to clear a 50-foot obstacle) is not required. (paragraph 2.2.1)

c. The hovering performance presented in figures 9 through 15, reference k, and reconfirmed in figures 23 through 27, appendix I, should be included. (paragraph 2.2.2)

d. During autorotation, rotor speeds below the minimum power-off limit (400 rpm) will result at gross weights less than 1400 pounds at sea-level density altitude. At higher altitudes, the gross weight at which the minimum power-off rotor speed cannot be maintained is less. The change is approximately 50-pound weight decrease per 1000-foot density-altitude increase above sea level. (paragraph 2.2.5)

e. The YOH-6A can land in a shorter distance than required for takeoff at any condition of weight, altitude, and ambient temperature. (paragraph 2.2.6)

f. The V_{ne} presented in figure 50 should be used. (paragraph 2.3.3)

g. Range performance presented in this report should be used until the results of the production OH-6A tests become available. (paragraph 2.2.4)

1.6.6 Studies or flight tests should be conducted to accomplish the following:

a. Investigate running takeoffs (sliding) during production OH-6A testing since the takeoff "ceiling" (altitude or weight) can be substantially increased using this technique. (paragraph 2.2.1.4)

b. Reduce or eliminate the high lateral cyclic stick vibration and fuselage vibration experienced near service ceiling. (paragraph 2.2.3)

c. Determine the effect of power setting (collective setting and rotor speed on service ceiling at gross weights heavier than 2085 pounds. (paragraph 2.2.3)

d. Determine climb performance of the production aircraft at airspeeds above the optimum climb speed so that the climb instability can be avoided. (paragraph 2.3.2)

e. Determine the effect of sideslip on power required and range. (paragraph 2.2.4)

f. Provide a means for the pilot to determine sideslip angle. This can be accomplished with a yaw string. (paragraph 2.2.4)

g. Investigate pitchup airspeed in high-speed climbing flight when testing the production OH-6A. (paragraph 2.3.3)

h. Investigate maneuvering stability at heavier weights (above 2100 pounds) and airspeeds (above 95 knots CAS) during production OH-6A testing. (paragraph 2.3.4)

i. Improve the starting capability of the OH-6A so that repeated battery starts (both air and ground) can be made throughout the altitude envelope specified by the engine manufacturer. This should be possible with the battery in a normal service condition and the engine in the poorest reasonable condition that might be expected between major engine overhauls. The airstart capability should be demonstrated on several production OH-6A's by the airframe manufacturer. (paragraph 2.4.2)

j. Determine if the compressor cleaning requirements of the T-63-A-5A engine are reduced so that the engine will deliver specification performance (power and fuel flow) without cleaning for at least the amount of time specified for periodic inspections when operating in reasonably dust-free conditions. (appendix IV)

k. Provide a responsive TOT indicator to give the pilot an accurate indication of TOT during transient conditions. (paragraph 2.4.2.2)

SECTION 2 DETAILS of TEST

2.1 INTRODUCTION

Performance, stability and control, and other tests were conducted on the YOH-6A, S/N 62-4212 to obtain additional data necessary for completion of the operator's manual. The tests were conducted by the U.S. Army Aviation Test Activity at Edwards Air Force Base, Bakersfield, and Bishop, California. Test sites varied in field elevation from sea level to 11,500 feet. A total of 161 flights were made from 27 January 1965 through 30 March 1966, accounting for 172.5 flight hours including ferry flights.

Test methods specified in the plan of test (reference c) were generally used for all tests conducted. Some of the tests were expanded to obtain sufficient data for presentation. A brief description of test methods is included in each subtest paragraph when considered necessary for understanding the test and the results obtained. A complete description of test methods, data acquisition, and data reduction methods is included as Appendix II.

Test results are compared with previous results or military specification requirements where applicable.

2.2 PERFORMANCE

2.2.1 Takeoff

Fourteen takeoff curves were obtained at field elevations of 4200 feet and 11,500 feet. All takeoffs were made in winds less than 2 knots. Density altitudes ranged from 3400 feet to 12,800 feet. Ambient temperatures varied from -3 degrees C to 28.5 degrees C. Aircraft gross weight was varied between 1900 pounds and 2700 pounds with a mid longitudinal C.G. location. Takeoff power (250 horsepower or 738 degrees C (TOT) whichever occurred first) and a standard rotor speed of 469 rpm were used throughout.

Two takeoff techniques were used: level acceleration from a 2-foot hover and simultaneous climb and acceleration from light on the skids.

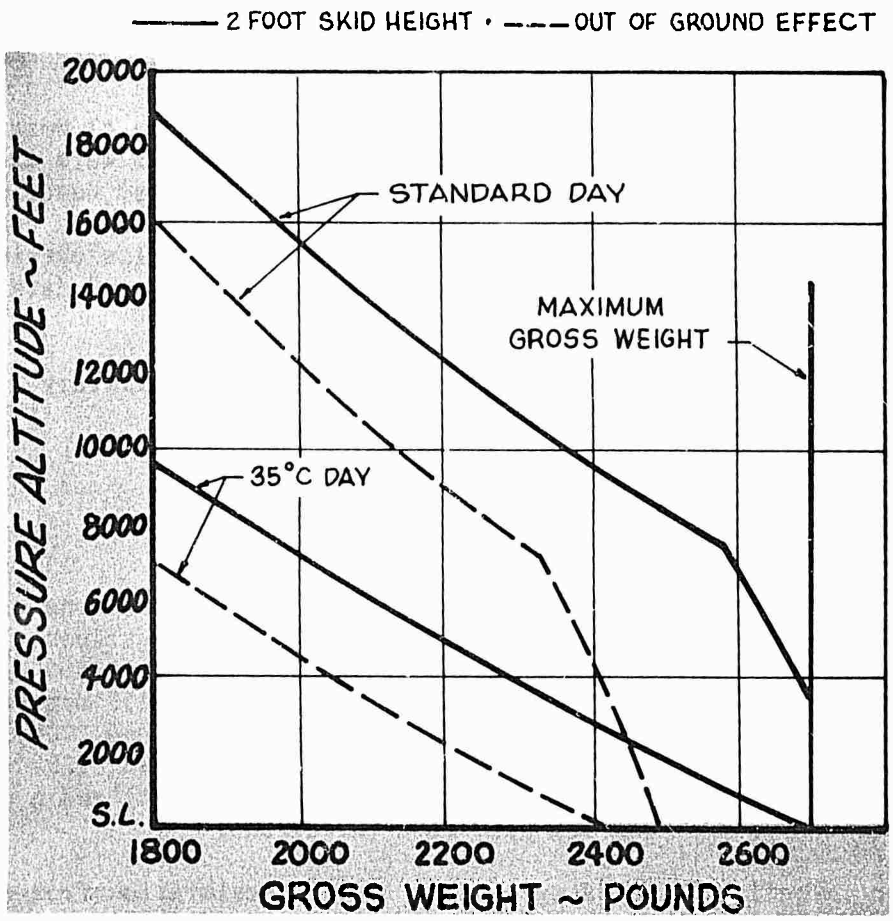
Test results are presented in figures 7 through 10 and figures 12 through 22, appendix I. Non-dimensional summaries (distance versus differential power coefficient (ΔCp) and airspeed) are presented in figures 6 and 11. Dimensional summaries are presented in figures 1 through 5 for standard-day and 35-degree-C hot-day conditions.

Hovering performance gives a good indication of takeoff performance. OGE and a 2-foot-skid-height hovering ceiling are presented for standard-day and 35-degree-C day conditions in figure A.

FIGURE A

YOH-6A HOVER CEILING

TAKEOFF POWER



At conditions under which the helicopter can hover OGE, a vertical takeoff can be made and takeoff distance is essentially zero. If hovering capability is less than a 2-foot skid height, the skids contact the ground during takeoff from a level surface.

Extrapolation of the data (figures 1 through 5, appendix II) indicates that when the helicopter has just sufficient power to hover at 2 feet, the takeoff distance required to clear a 50-foot obstacle will be greater than 600 feet. When the hovering capability is less that 2 feet, sliding takeoffs can be made if a smooth level surface is available (see paragraph 2.2.1.4). Distance required to clear a 50-foot obstacle, however, is substantially increased.

At 2700 pounds, 35-degree-C day and at sea level or above, the YOH-6A would require a sliding takeoff. Above 3400 feet on a standard day, a sliding takeoff would be required at 2700 pounds.

The takeoff performance decrease with altitude and temperature is due primarily to a decrease in power available. Below engine critical altitude, where the power available is constant, takeoff ceiling (hovering ceilings) increase approximately 8000 feet per 100-pound decrease in weight. Above engine critical altitude, where power available decreases with increasing altitude or temperature, takeoff ceiling increases approximately 2000 feet per 100-pound decrease in weight.

Ambient temperature has the greatest effect on takeoff performance. At 6000 feet on a standard day (ambient temperature = +3 degrees C), the YOH-6A can take off without ground contact (hover at 2 feet) at a weight of 2620 pounds and take off vertically (hover OGE) at 2360 pounds. On a 35-degree-C day, at 6000 feet, the YOH-6A can take off without ground contact at 2100 pounds and can take off vertically at 1870 pounds. Each 6-degree-C rise in ambient temperature, therefore, will require approximately 100 pounds to be off-loaded to maintain the same takeoff performance.

For any given excess power, there is some optimum climbout airspeed which will minimize the distance required to clear a 50-foot obstacle. For the YOH-6A, this airspeed varies from approximately 15 knots TAS at conditions under which sufficient power is available to hover OGE to slightly more than 30 knots TAS at conditions under which the helicopter can just hover at 2 feet. Since the airspeed system is unreliable below about 30 knots IAS (34 knots CAS), however, the only method the pilot has to determine airspeed is by judging apparent ground speed and correcting for wind. The pilot, therefore, cannot determine airspeeds accurately below 30 knots IAS. At conditions under which the helicopter cannot hover OGE, attempting to climb out at airspeeds approximately 5 knots slower than optimum airspeed will result in insufficient power to climb (fall through). If this occurs, the helicopter must be accelerated to a higher airspeed and the distance to clear an obstacle will be greatly increased. If "fall through" occurs when the takeoff distance available is marginal, it is recommended that the takeoff be aborted and begun again with climbout delayed until a higher airspeed is reached. To avoid the possibility of "fall through", it is recommended that 30 knots IAS be used as the minimum climbout airspeed when maximum performance (shortest distance to clear a 50-foot obstacle) is not required. This airspeed will also allow takeoffs to be made outside the "avoid" area of the height-velocity diagram (i.e., "safe" autorotational landings will be possible if engine failure occurs).

2.2.1.1 Takeoff Techniques

The level acceleration from a 2-foot hover technique consisted of the following: The helicopter was stabilized at a 2-foot skid height hover. A rotor speed 2-3 rpm higher than the desired 469 rpm (100 percent power turbine speed (N_2)) was selected while hovering to compensate for steady-state droop which occurred when takeoff power was applied. Takeoff power (250 SHP or 738-degree-C TOT, whichever limit occurred first) was applied by increasing collective pitch, the helicopter was moved into forward flight and a constant skid height acceleration was made. Two to 5 knots before the desired climbout airspeed was reached, rotation to climbout attitude was initiated. The climbout was then accomplished at constant airspeed.

The climb and acceleration from light on the skids technique consisted of the following: Power was increased until the helicopter was light on the skids. As in the level acceleration from a 2-foot hover technique, a rotor speed 2-3 rpm higher than the desired 469 rpm was selected. Takeoff power was applied, liftoff was accomplished, and a helicopter pitch attitude that would give the desired airspeed at 50 feet was established. This attitude was held throughout the takeoff. Some experience with this technique was necessary before the correct pitch attitude for the excess power available could be chosen to achieve the desired airspeed.

While using both techniques, as forward flight was initiated, it was necessary to continue to increase collective pitch to maintain takeoff power. As the helicopter gained translation lift, forward stick application was necessary to maintain attitude. Throughout the takeoff, the rotor speed was held approximately constant and the power was maintained at the maximum takeoff power attainable.

2.2.1.2 Comparison of Takeoff Techniques and Data

At the same conditions of weight, altitude, temperature, and airspeed, the maximum difference in distance to clear a 50-foot obstacle between the 2 techniques used was less than 50 feet within the range of conditions tested. At conditions where the helicopter could hover OGE, the climb and acceleration from light on the skids technique required slightly shorter distances to clear a 50-foot obstacle. As excess power decreased (altitude, weight, or temperature increase), the level acceleration from a 2-foot hover technique yielded a shorter takeoff distance. The level acceleration technique also gave increasingly better performance than the climb and acceleration technique as optimum climbout airspeed (approximately 15 to 30 knots TAS) was approached. As excess power decreased and takeoff distance available decreased, the level acceleration technique became more advantageous when compared with the climb and acceleration technique. The level acceleration technique also had the advantage of requiring less time in the "avoid" area of the height-velocity diagram.

Obstacle height has an effect on determining the takeoff technique that gives the shortest takeoff distance. At conditions under which both techniques give the same distance over a 50-foot obstacle, the climb and acceleration technique gives the shortest distance over an obstacle lower than 50 feet and the level acceleration technique yields the shortest distance over an obstacle higher than 50 feet. The level acceleration technique gives a longer period of level flight near the ground with a steeper climbout angle and higher rate of climb. The climb and acceleration technique gives a short-or no-ground run with a shallower climbout angle and lower rate of climb at any given condition and climbout airspeed. Takeoff distance as a function of obstacle height was not investigated quantitatively.

Takeoff distances for 2100 pounds on a 35-degree-C day using the level acceleration technique for data from the original YOH-6A Performance Program (reference k) and this program are presented in figure B, page 20.

A difference of up to 50 feet exists between takeoff distances indicated by the original YOH-6A data and data obtained during this program. This difference is probably caused by the differences in pilot technique or in interpretation of the data.

2.2.1.3 Loss-of-Power Phenomenon During Takeoff

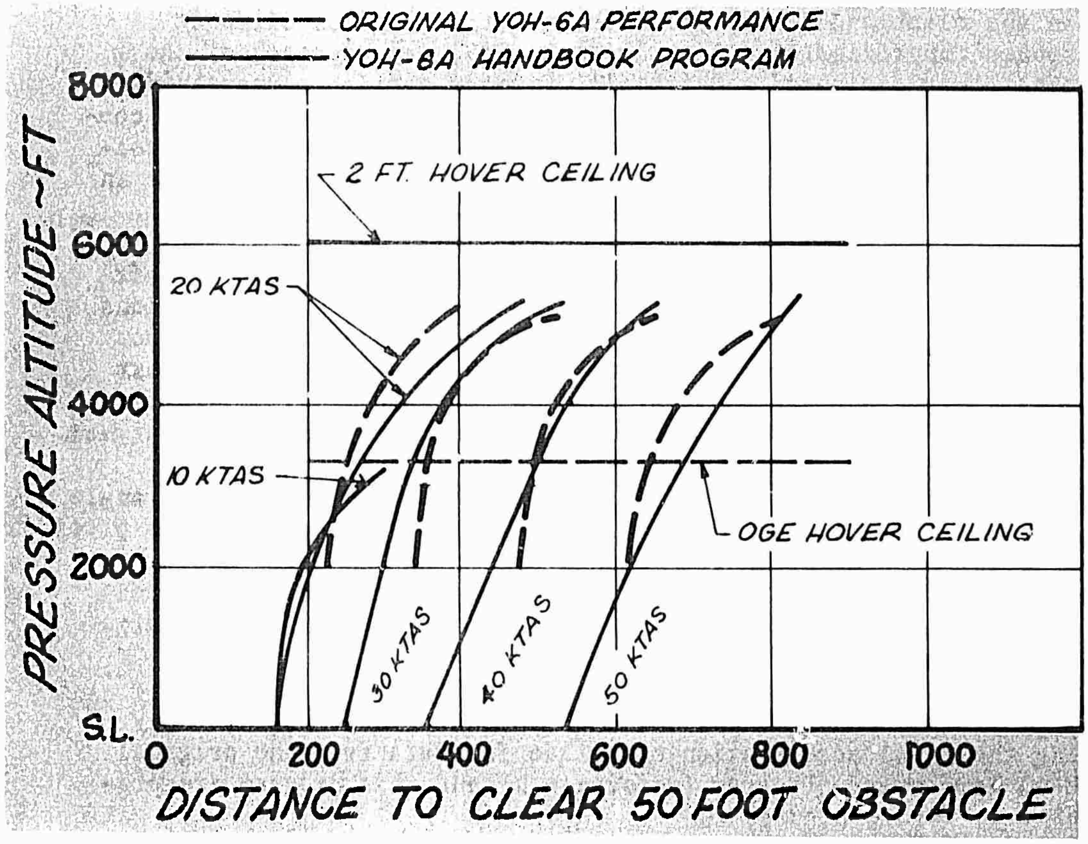
During the previous YOH-6A takeoff tests (reference k), a "fairly large" decrease in TOT occurred as forward flight was initiated. In an effort to determine the cause of this decrease in TOT, level accelerations were made with pedals fixed and also with zero yaw (constant-heading). During these level accelerations, collective was increased to takeoff power during the first second and held fixed for the remainder of the acceleration. Data com-

[redacted]

FIGURE B

YOH-6A LEVEL ACCELERATION TAKEOFF PERFORMANCE

GROSS WEIGHT = 2100 LBS. • 35°C DAY



paring these level accelerations are shown in figure 22. The data show a relatively constant TOT of 738 degrees C (takeoff power TOT limit) for the fixed-pedal accelerations and a decreasing TOT from 738 degrees C to approximately 720 degrees C after 15 seconds for the constant-heading accelerations.

Power could not be determined accurately during the early portion of the accelerations because of a lag in the torque sensing system. After torque pressure had stabilized, however, approximately 11 inches of mercury torque pressure (18 horsepower) difference existed between the 2 types of accelerations. Less power was required for the constant-heading takeoffs. It was concluded that the decrease in TOT (loss of power) experienced was actually a substan-

[redacted]

tial decrease in tail-rotor power required as the helicopter moved into forward flight. This effect is present in all helicopters with tail rotors. It was more apparent in the YOH-6A, however, because of the relatively high tail-rotor power required in a hover.

2.2.1.4 Minimum-Skid-Height-Takeoff

Several takeoffs were made when the helicopter was only able to become light on the skids. Quantitative data, however, were not obtained. One takeoff was made from a level-sod field at an elevation of 9500 feet. The ambient temperature was approximately 18 degrees C and the gross weight was approximately 2200 pounds. Under these conditions the helicopter was just able to become light on the skids at takeoff power. During the takeoff, continual ground contact was made up to a ground speed of approximately 15 knots. Intermittent ground contact occurred between 15 knots ground speed and 25 to 30 knots IAS. Climbout was initiated at approximately 35 knots IAS. Distance was not recorded; however, approximately 1000 to 1500 feet were required before climbout was started and approximately 2000 feet were required to gain a 50-foot altitude. Takeoffs were also made at a field elevation of 4200 feet on a paved runway under conditions at which the helicopter could not hover at 2 feet. Characteristics were similar to those encountered at 9500 feet.

Successful takeoffs can be made at conditions where sufficient power is available to allow the helicopter to begin forward movement. Suitable terrain, however, either a prepared surface or smooth level grass, must be available. Being able to make sliding takeoffs would substantially increase the weight, altitude, and temperature at which takeoffs could be made. Takeoffs under conditions under which the helicopter cannot hover at 2 feet should be investigated during future testing.

2.2.2 Hover

Hovering performance data were obtained at skid heights of 2, 5, 10, and 50 feet in winds of less than 2 knots. The data were obtained at field elevations of sea level, 4200, 9500, and 11,500 feet. Gross weight was varied from approximately 1600 pounds to 2700 pounds. Density altitude varied from -500 feet to 13,040 feet. A mid longitudinal C.G. was maintained. Rotor speed was varied from 464 to 482 rpm. Non-dimensional data are presented in figures 23 through 27, appendix I. Fairings on these figures are taken from original YOH-6A performance data (figures 16 through 19, and a cross plot of figure 13, reference k). The free-flight method of determining hovering performance was used. A complete description of this method is included in appendix II.

The data obtained during this program agree with the data obtained during the original YOH-6A program. The same helicopter, YOH-6A, S/N 62-4212, and the same main rotor blades were used to obtain the hovering performance data during both programs.

These data show that the maximum altitude the YOH-6A could hover OGE on a 35-degree-C day is approximately 3400 feet at 2085 pounds gross weight (design weight). On a sea-level standard day the maximum OGE hovering weight is 2490 pounds. This hovering performance is based on takeoff power, as defined by the T-63-A-5 engine model specification 580E (reference o).

The hovering performance summary plots presented in reference k, figures 9 through 15, are valid and correct. No hovering performance summaries are presented in this report.

It was found while hovering in light winds greater than 5 knots from varying directions, that it was not possible to maintain a constant skid height. At conditions (weight, altitude and temperature) under which the helicopter had sufficient power just to maintain OGE hover with zero wind, changes in wind velocity or direction would cause the helicopter to attain a rate of descent sufficient to cause ground contact or a rate of climb. Also, relatively large and continual pedal input was required to maintain heading. This contributed to the inability to maintain skid height.

Hovering was also conducted at skid heights of .5, 1, 1.5, and 3 feet in an effort to determine the cause of the loss-of-power phenomenon described in paragraph 2.1.4, reference k. No unusual or discontinuous power effects that were encountered might explain this loss-of-power phenomenon. The cause was determined to be a large decrease in tail-rotor power required with forward speed, reference paragraph 2.2.1.3. Hovering data at these skid heights are not presented.

2.2.3 Climb

Continuous climbs to service ceiling were conducted from sea level with climb start gross weights of 1600, 2085, and 2700 pounds at a mid longitudinal C.G. Takeoff power (250 SHP or 738 degrees C TOT, whichever occurred first) was used. The climbs were flown at 469 rpm, zero sideslip, and an airspeed schedule determined from level flight data (speed for minimum power at each weight and altitude condition). Level flight data did not extend to service ceiling so the airspeed schedule was extrapolated to the unique airspeed at which absolute ceiling could be maintained.

This airspeed was determined during preliminary climbs. Power and weight correction factors determined during the original YOH-6A performance program (reference k) were used to correct test-day climbs to standard-day conditions. No power management problems were encountered during the climb tests. During the portion of the climb below critical altitude for the engine, collective control was gradually increased to maintain the limit torque value. Small adjustments in power turbine speed (N_2) were required during this portion of the climb since some droop in rotor rpm occurred. After the critical altitude for the engine was reached and the TOT limit was obtained, collective control remained essentially fixed for the remainder of the climb.

The dynamic longitudinal instability discussed in paragraph 2.5.5.1, reference j, was experienced during the 1600-and 2085-pound climbs. This instability is discussed further in reference d, paragraph 2.3.1.2.

A lateral cyclic-stick vibration was noted during the 2085- and 2700-pound climbs. This vibration was very disconcerting to the pilot. No vibration instrumentation was installed but the following characteristics were noted. The vibration occurred at a frequency of 4 per rotor revolution (approximately 32 cps). It became perceptible approximately at the altitude that full forward trim authority was reached (reference paragraph 2.3.2) and increased in amplitude as service ceiling was approached. At 2700 pounds near service ceiling the lateral cyclic stick displacement, due to the vibration, was in excess of 1/2-inch double amplitude at the grip. Fuselage vibration although of much less amplitude, was also perceptible. The amplitudes and frequency were such that they tended to numb the pilot's hand and the pilot's and observer's feet. It was necessary for the pilot to lock the collective friction and change hands on the cyclic stick to retain feeling in his hand. As the climb was terminated and descent begun, a large decrease in vibration amplitude was noted. Perceptible vibration, however, was still present. This condition exceeds vibration limits specified in MIL-H-8501A, paragraph 3.7.1. It is recommended that vibration during climbs be investigated quantitatively during future testing.

Climb performance data are presented in figures 28 through 30, appendix I. Data reduction methods are described in appendix II. A summary of these climbs and climbs conducted at maximum continuous power (212 SHP or 693 degrees C TOT, whichever occurred first) during the original YOH-6A performance tests are presented in table 1.

[redacted]

TABLE 1

YOH-6A CLIMB PERFORMANCE SUMMARY

STANDARD 469 ROTOR RPM MID CG

Power Setting	Sea Level Weight lb	Sea Level Rate of Climb ft/min	Service Ceiling ft	Service Ceiling Weight lb	Time To Service Ceiling min
Takeoff	1600	2440	26,000	1561	18.5
Takeoff	2085	1950	19,700	2046	12.0
Takeoff	2700	1170	10,500	2662	13.6
* Maximum Continuous	2085	1450	19,000	2045	22.0
* Maximum Continuous	2700	910	11,600	2653	19.0

* Obtained from reference k, figures 20 and 21.

An increase of 700 feet in service ceiling at a climb start weight of 2085 pounds and a decrease of 1100 feet in service ceiling at a climb start weight of 2700 pounds were obtained by conducting the climbs at takeoff power instead of maximum continuous power. The decrease in service ceiling at 2700 pounds is partially due to the increased drag caused by modifications to the aircraft (reference paragraph 1.2) and the fact that service ceiling was reached at a slightly higher weight. The decrease in service ceiling, however, was probably caused primarily by an increase in blade stall as evidenced by the high lateral cyclic stick vibrations (described earlier). This was caused by the fact that increasing collective pitch to maintain takeoff power rather than maximum continuous power aggravated the blade stall. Increasing blade stall could reduce the rotor efficiency enough so that the net gain in climb performance would be negative. That is, the effect of increasing power from maximum continuous power to takeoff power would decrease service ceiling and rate of climb near service ceiling. The data indicates that this is what has happened. Increasing rotor speed from 469 rpm to 482 rpm should decrease blade stall and, therefore, improve climb performance. Also, there will be some optimum power schedule (collective pitch schedule) less than takeoff power to obtain maximum service ceiling and rate of climb. This phenomenon was not investigated during this program but should be investigated when the production OH-6A is tested.

[redacted]

2.2.4 Level Flight

Speed-power tests were conducted to determine the following:

a. The change in drag caused by the external modifications to the aircraft (reference paragraph 1.2).

b. The change in drag caused by the XM-7 armament kit installation over an increased range of conditions.

c. The drag caused by the test boom installation.

Tests were conducted over the following range of conditions:

a. Clean configuration: gross weights from 1745 pounds to 2475 pounds; longitudinal C.G. mid; density altitudes from 1960 feet to 10,310 feet; rotor speeds from 468.5 rpm to 482 rpm; and sideslip of zero.

b. With the XM-7 armament kit installed: gross weights from 1835 pounds to 2085 pounds; longitudinal C.G., mid; density altitudes from 7150 feet to 10,370 feet; rotor speeds from 469 rpm to 482 rpm; and sideslip of zero.

c. With the test boom off: gross weights of 2080 pounds and 2660 pounds; longitudinal C.G., mid; density altitudes of 5010 feet and 5450 feet; rotor speed of 469 rpm; and zero bank angle.

Results of the speed power tests in the clean configuration are presented in figures 32 through 41 appendix I. A range summary is presented in figure 31.

Power required for both the original configuration (reference k)and modified configuration are presented on each speed-power plot. The modifications resulted in an airspeed decrease of approximately 3 knots TAS for a constant shaft horsepower at recommended cruise speed.

The range summary presented in reference k, figure 25, is not valid. It implies that range varies linearly with weight. The range summary presented in figure 31, shows this is not true at all altitudes.

The recommended cruise speed is limited by never-exceed airspeed (V_{ne}) at 2700 pounds at all altitudes. As weight decreases, the altitude at which recommended cruise speed is limited by V_{ne} increases. For a standard day the recommended cruise speed does not exceed the speed for max:.mum continuous power for any weight and altitude condition in the range tested. For a hotter than standard day, however, recommended cruise speed may be limited by maximum continuous power (693 degrees C TOT). This condition was not investigated.

Data from the 3 speed-power tests with the XM-7 armament kit installed are presented in figures 42 through 44, appendix I. The XM-7 armament kit was found to contribute an increase in drag of .7 square feet of equivalent flat plate area at recommended cruise speed throughout the range of altitudes and weights tested. This agrees with the findings in reference k.

Two speed-powers were flown with the test airspeed boom removed to determine its effect on drag. Boom "off" data are compared with basic data with the boom "on" and presented in figures 45 and 46, appendix I. The equivalent flat plate area of the boom could not be determined because sideslip angle has a significant effect on power required and no sideslip indication was available with the boom removed.

The power requirements as a function of sideslip were investigated at 1 condition and are shown in figure C.

FIGURE C EFFECT OF SIDESLIP ON POWER REQUIRED

YOH-6A USA S/N 62-4212

AVE. DENSITY ALTITUDE = 5000 FT. · ROTOR SPEED = 469 RPM

CLEAN CONFIGURATION (BOOM ON)

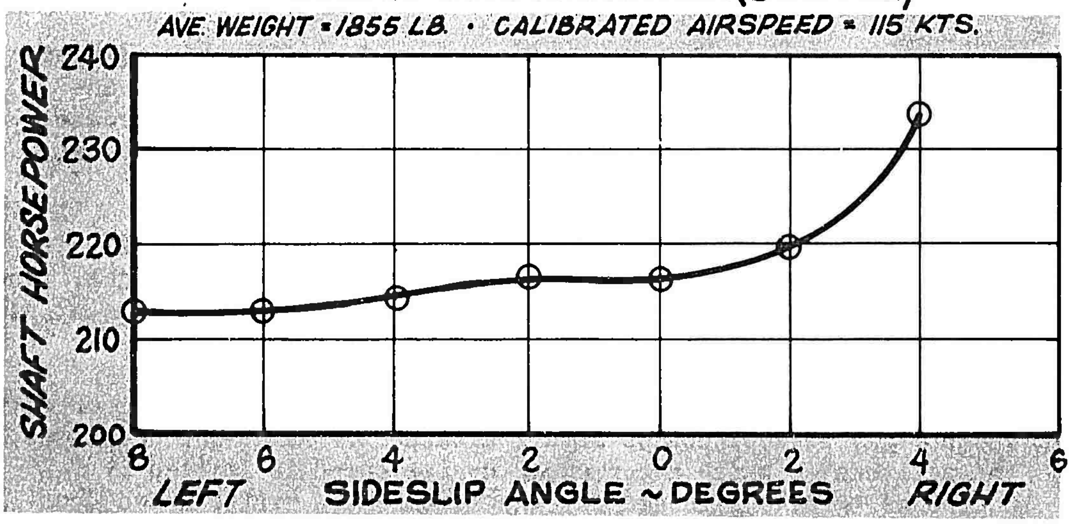


Figure C shows that slightly less power was required in left sideslip than in zero sideslip and that power increased significantly in right sideslip at this condition.

The following conclusions can be made. Power required at a given condition (airspeed, weight, and altitude) varies significantly with sideslip. Some optimum (minimum power required) sideslip condition other than zero sideslip may exist. Range can be significantly decreased or possibly increased by flying at other than zero sideslip. The pilot, therefore, should be given some means of determining sideslip to insure that the best range performance is obtained. (The use of a yaw string should suffice). The effect of sideslip angle on level flight performance should be investigated further.

2.2.5 Autorotational Descent

Autorotational descents were conducted to determine the altitude for full-down collective and to determine if any gross weight-altitude conditions existed where the minimum rotor speed limit would be exceeded. Results are presented and summarized in figure 47, appendix I.

As altitude is decreased during an autorotational descent, the collective is lowered to maintain a constant rotor speed. At a certain altitude, depending on gross weight, rpm and airspeed, full-down collective is reached. As the aircraft descends below this altitude, the rotor speed decreases and may exceed the minimum power-off limit.

Extrapolation of the data obtained indicates that at gross weights less than approximately 1400 pounds at sea level, rotor speed cannot be maintained above the minimum power-off rotor speed (400 rpm). At gross weights more than approximately 2450 pounds at sea level during stabilized autorotational descent, full-down collective cannot be used since the maximum limit power-off rotor speed (514 rpm) would be exceeded. At higher altitudes, the gross weights at which the rotor speed limits are reached with full-down collective are less. The change is approximately 50-pound decrease per 1000-foot increase. The rotor speed change with altitude for full-down collective is 6-rpm decrease for each 1000-foot density altitude decrease. This information should be included in the operator's manual.

2.2.6 Power-on-Landing

Power-on landings were conducted to determine the minimum distance required to land over a 50-foot obstacle at various weights and altitudes. YOH-6A operator's manual testing was terminated before adequate data could be obtained. From the testing accomplished, however, it was determined that a landing could be made in a shorter distance than required for takeoff at any given weight, altitude, and temperature condition. It was also determined that OGE hover capability was necessary to make an approach and touchdown at speeds less than 20 knots IAS since flare effectiveness did not exist below this speed and ground effect was not sufficient to cushion touchdowns.

2.2.7 Autorotational Landing (Height-Velocity)

It was intended to conduct autorotational entries and landings to determine the minimum height from which safe autorotational landings could be made following an engine failure. An accident occurred on 30 March 1966, during the autorotational landing buildup work, however, and testing was terminated. Prior to the accident, 4 flights including 14 autorotational landings had been accomplished to develop the technique and buildup to minimum height points.

The buildup including autorotational entries was at a density altitude of approximately 5000 feet, an entry rotor speed of 469 rpm, and airspeeds from zero to 85 knots CAS. Gross weights were from 1900 to 2100 pounds with C.G.'s at station 98.8 to station 100.5 (C.G. limits from station 97.0 to station 104.0). Autorotational landings were made at density altitudes of approximately 3000 feet and 6000 feet, a gross weight of approximately 1900 pounds, C.G. at station 99.4, and entry speeds of zero, 20, and 35 knots CAS. All landings were made on level paved surfaces. During this buildup work, 45 knots CAS was determined to be the minimum effective flare speed.

The helicopter flying and ground handling qualities military specification MIL-H-8501A (reference n) requires that a minimum of 2-second time delay from loss of power to collective change shall be used to simulate pilot reaction time. At no time during this maneuver shall the rotor speed fall below a safe minimum. The helicopter shall also be capable of making safe autorotational landings at touchdown speeds of 15 knots or less. Zero touchdown speed is highly desirable.

The following technique was used at entry airspeeds below minimum effective flare speed (45 knots CAS). After the aircraft was stabilized at the desired conditions, power failure was simulated by rapidly rotating the throttle to the ground-idle detent (throttle chop). Approximately 2 seconds later collective was rapidly lowered to full down. The aircraft was pitched nosedown and accelerated to the minimum effective flare speed. The nosedown pitch attitude was approximately 35 degrees when initiated from a hover and approximately 20 degrees when initiated with a forward speed of 35 knots CAS. After the minimum effective flare speed (45 knots CAS) had been achieved, the aircraft attitude was adjusted to maintain that speed. At approximately 50 feet above the ground a cyclic flare was then initiated to a noseup attitude of approximately 20 degrees which was sufficient to arrest the rate of descent. During the cyclic flare, collective was applied to slow the forward speed. The flare was held until the aircraft started to settle. The aircraft was then leveled and collective applied to cushion the touchdown which was made in a level attitude. After touchdown, the collective was lowered rapidly to full down to maintain as much rotor rpm as possible for directional control. This technique should be modified slightly in that collective should be lowered at a moderate rate following touchdown to help avoid contacting the tail boom with the rotor blades, even though it compromises directional control.

Following the throttle chop, a mild left yaw was the only apparent attitude change. This was easily corrected with a small pedal input. In turbulent air, this attitude change could be easily masked. During the 2-second delay after the throttle chop, airspeed or altitude did not change significantly.

Rotor speed decayed rapidly until collective was lowered. The decay rate varied with collective position (power setting prior to simulated failure), the maximum being approximately 65 rpm per second in a hover and at 85 knots CAS in level flight and the minimum being approximately 40 rpm per second at the speed for minimum power required. At heavier weights (above 2100 pounds), higher airspeeds, or in climbing flight, the decay rate would be higher. These rotor speed decay rates were attained approximately .5 seconds after the throttle chop and continued until after collective was lowered. With the engine operating at ground-idle, some power (less than 35 horsepower) was delivered to the rotor below 45 rotor rpm. Following an actual engine failure, therefore, slightly higher decay rates would occur.

The decay rates encountered using a 2-second delay between throttle chop and lowering of collective resulted in minimum rotor speeds from 370 rpm to 410 rpm, depending on entry airspeed. Minimum rotor speed limit is 400 rpm. This does not comply with military specification MIL-H-8501A. Although control effectiveness was reduced at these lower rotor speeds, sufficient control effectiveness remained to control the aircraft. Rotor speeds down to 350 rpm had been demonstrated previously (reference k); therefore, the required 2-second delay was used throughout these tests. The 400-rpm minimum rotor speed limit, however, may be a structural limit rather than a control limit.

Lowering collective resulted in a nosedown pitch attitude. The amount of nosedown pitching increased with airspeed and the rate at which collective was lowered. Above 45 knots CAS, the rate at which collective was lowered was decreased so this nosedown pitching tendency could be more easily controlled. Below 35 knots CAS, some additional forward cyclic input was used to attain the desired pitch attitude.

Rotor speed rate of increase after collective was lowered varied from a minimum of approximately 40 rpm per second from entry at a hover to 70 rpm per second at 85 knots CAS. At the higher airspeeds, it was necessary to reapply collective to prevent rotor overspeed.

The minimum flare airspeed was attained approximately 3 seconds after lowering collective when the aircraft was entering autorotation from a hover. In forward flight less time was required. To avoid overshooting the desired airspeed (45 knots CAS) and to maintain this airspeed until flare, it was necessary to decrease the nosedown pitch attitude almost immediately after attaining it. This required a continuous cyclic input from trim to forward to aft to a new position required to maintain 45 knots CAS.

During the maneuver, high rates of descent were observed. The rate of descent varied from 3750 feet per minute (fpm) with entry from a hover to 2500 fpm with entry at 35 knots CAS. The rates of descent obtained were not stabilized and were a result of diving the helicopter to gain airspeed and/or rotor speed. These rates of descent were attained approximately 2 seconds after collective was lowered and continued until the cyclic flare became effective. It was necessary to arrest the rate of descent at a height sufficient to prevent the tail skid from contacting the ground, but not so high that a high sink rate would develop after the aircraft was leveled ("fall through") and before touchdown was made. This required a very high degree of pilot judgment in selecting the height at which to start the flare and the flare rate to use. The aft cyclic stop was contacted during several of the landing flares but sufficient control was available to achieve the desired flare attitude. With a more forward C.G. or at heavier weight, aft control available could become a problem.

During the landing, full-up collective was sometimes needed to cushion touchdown. The normal acceleration load factor did not exceed 2 g during any landing. Initial landing speeds were approximately 20 to 25 knots. Several landings were made at touchdown speeds of less than 5 knots. During landing slides, a mild pitching oscillation occurred; the magnitude was a function of initial touchdown attitude, speed, and load factor. The maximum oscillation of 6 degrees nosedown to 2 degrees noseup occurred with the highest landing speed and maximum load factor. No tendency to nose-over was noted during any autorotational landings or subsequent slides.

Engine-Failure Warning System

The engine-failure warning system in this aircraft is essentially useless except for practice autorotations. It is activated when the gas producer speed (N_1) falls below 55 percent. Gas producer speed (N_1) decay rates observed indicate this would not occur in less than 2 seconds. Also, an engine failure in the power train or output shaft would cause no engine-out warning. A low rotor speed warning set just below minimum power on rotor speed (465 rpm) would give a warning in approximately .2 seconds. A twist-grip override could be incorporated so that the warning system would be active only when the twist-grip is in the flight-idle position (full open). This would permit engine starts and practice autorotations without the warning system operating. The present engine-failure warning system should be retained to provide a warning system during practice autorotations.

Accident

An accident occurred during an autorotational landing on 30 March 1966. Entry was made at 19 knots IAS at 213 feet above the ground. The normal procedure described previously was used. An autorotational landing at these entry conditions had been accomplished successfully before. After touchdown had occurred, a main rotor blade struck the tail boom. The tail boom was completely severed and extensive damage to the main rotor blades was incurred. The primary cause was determined by the Accident Investigation Board to be rapid lowering of collective pitch (approximately 10 inches/second) at the time of highest g load (1.91 g). Contributing factors were:

a. Aft longitudinal cyclic stick application

b. Low rotor speed (263 rpm)

c. Nosedown pitch rate

d. Slight quartering tailwind (approximately 4 knots)

e. Low touchdown speed (1-2 knots)

f. Forward C.G. (station 99.4)

No damage or out-of-tolerance condition that could not be attributed to the blade strike was found. All of the factors considered to have contributed to the accident had been equaled or exceeded individually during previous autorotational landings.

In view of the autorotational entry and landing characteristics observed and the accident encountered, it appears that:

a. The heights presented in the YOH-6A or OH-6A operator's manual (reference m) for safe autorotational landings cannot be met using the technique required by MIL-H-8501A (2-second delay and minimum touchdown speed).

b. Autorotational landings from minimum heights require extremely good judgment by pilots with substantial experience in the aircraft. It would be difficult or impossible, however, for the average pilot to make safe autorotational landings from the minimum heights presented in the YOH-6A manual.

c. Aft cyclic application and rapid collective lowering after touchdown during autorotational landings may cause the main rotor blades to strike the tail boom.

d. Touchdown speed of 15 knots or less during autorotational landings may not be practical in the OH-6A.

2.3 STABILITY AND CONTROL

2.3.1 Sideward and Rearward Flight

Sideward and rearward flights were conducted in calm air to evaluate the hovering capability of the YOH-6A in crosswind or tailwind conditions. The first sideward and rearward tests were conducted at the following conditions: gross weight of 2585 pounds, longitudinal C.G. of 95.8, lateral C.G. of 3.4 inches left, and density altitude of -100 feet. It was intended to conduct the tests at the forward C.G. limit (station 97.0); however, after the flight it was found that the forward C.G. limit had been exceeded. This weight and balance problem is discussed in appendix III. The tests were then reflown at the following conditions: gross weight of 2495 pounds, longitudinal C.G. of 97.0, lateral C.G. of 3.9 inches left, and density altitude of 3560 feet. Airspeeds were from 35 knots to the left to 35 knots to the right and from 35 knots to the rear to 35 knots forward. The flights were conducted in ground effect (IGE) at a height sufficient to permit recovery if unusual flight attitudes or control problems were encountered (10-15 feet). The weight was limited by power available to obtain the desired height. Full lateral C.G. limit (5 inches) could not be reached with the instrumentation installed using internal loading. A calibrated ground pace vehicle was used to determine the airspeed.

Average control positions in sideward and rearward flight are presented in figures 48 and 49, appendix I. In both sideward and rearward flights, continual pedal and cyclic control inputs up to ± 1 inch were required to maintain heading and speed. While the aircraft was gaining translational lift (10 to 20 knots), larger inputs of ± 1 to 2 inches were required. This increased the already large amount of pilot effort required to maintain speed and heading. No objectionable control reversals or discontinuities were encountered in either sideward or rearward flight.

In sideward flight to the left an average of less than 10-percent aft control remained above 17 knots. Above 9 knots, intermittent contact was made with the aft control stop. This did not comply with MIL-H-8501A.

In rearward flight an average of less than 10-percent aft longitudinal control remained above 25 knots. Above 9 knots, intermittent contact was made with the aft control stop. This does not comply with MIL-H-8501A.

It is recommended that the present 10-knot crosswind and tailwind limitation at overload weights (above 2085 pounds) be retained because of the small amount of longitudinal control margin when compared with the control required under crosswind or tailwind conditions.

2.3.2 Climb Instability

The dynamic longitudinal instability experienced during climbs flown during the original YOH-6A testing (references j and k) was investigated throughout the range of conditions specified in the Test plan, reference c, paragraph 2.2.1(25-55 knots CAS; zero and 5- to 7-degrees right sideslip) to investigate the effect of the horizontal stabilizer modification on climb stability. The instability was still present up to 55 knots CAS in either zero sideslip or 5- to 7-degrees right sideslip in climbing flight at takeoff power. The change to the horizontal stabilizer (E.O. 369-2264) had no significant effect on the dynamic longitudinal stability during high-powered climb flight.

At 1600 pounds during climbs made to service ceiling, the instability resulted in airspeed excursions up to 10 knots CAS from the desired climb speed. This condition continued to a density altitude of approximately 14,000 feet. During 2085-pound climb, the condition was less severe and resulted in airspeed excursions of approximately 5 knots. At 2700 pounds, the instability was not present.

During routine climbs to altitudes required for other tests, airspeed was increased to 60 to 70 knots CAS to avoid this instability and the slightly lower rate of climb was accepted. The decrease in rate of climb at the higher speeds was not determined. The instability could also be eliminated at optimum climb speed by reducing rate of climb below 1500 fpm. Climbs at higher than optimum airspeed should be investigated quantitatively in future testing.

The lack of adequate forward stick-force trim authority that occurred during the original YOH-6A performance test climbs was also noted during the continuous climbs flown during this program. At 2085 pounds, full trim authority was reached at 12,000 feet density altitude. At 2700 pounds full trim authority was reached at approximately 6000 feet density altitude. As the climbs were continued above these altitudes, gradually increasing forward stick force was required to maintain the desired airspeed. This force reached a maximum of 8 to 10 pounds near service ceiling. This condition did not comply with MIL-H-8501A.

2.3.3 <u>Level Flight Pitchup</u>

The pitchup experienced at speeds near never-exceed airspeed (V_{ne}) during the original YOH-6A program (references j and k) was investigated at gross weights of 1800 to 2600 pounds, longitudinal C.G.'s of 97 (forward limit) to 102.6 (aft), density altitudes of 3000 to 12,000 feet, and rotor speeds of 463 and 469 rpm. Results are presented in figures 51 and 52, appendix I. A recommended V_{ne} is presented in figure 50 based on 90 percent of the airspeed for self-induced pitchup. All references to V_{ne} in this report pertain to figure 50.

Data were obtained in smooth air in level flight with no control inputs. Airspeed was gradually increased until pitchup occurred. Flights were conducted in hotter-than-standard atmosphere. This caused speed to be power-limited in some cases where it would not normally be power-limited with a specification engine in standard or colder conditions. When this occurred, a small aft cyclic stick pulse was used to induce pitchup. The size of the pulse required and the severity of the resulting maneuver gave some indication of the unaccelerated pitchup speed.

Pitchup airspeed was primarily a function of gross weight and density altitude; however, rotor speed and longitudinal C.G. were also found to affect pitchup airspeed.

At heavy weight (approximately 2600 pounds), the pitchup is relatively mild. Approximately 10 to 15 pounds of forward stick force are needed to maintain attitude when pitchup starts to occur. Prior to pitchup, a large increase in vibration level occurs. Near design weight (2100 pounds), the pitchup is more severe and little vibration warning is given. To recover from pitchup, collective must be lowered and a forward force of 20 to 30 pounds on the cyclic stick must be applied. At weights of 2100 pounds or less, recovery from pitchup must be initiated immediately to avoid extreme nose-high attitudes. When this occurs, limit load factors may be reached or exceeded if recovery is not initiated immediately. At lighter weights (1800 pounds), self-induced pitchup airspeed may be beyond takeoff power-limit airspeed for altitudes near sea level.

Decreasing rotor speed from 469 rpm to 463 rpm caused airspeed for self-induced pitchup to decrease by approximately 2 knots. Increasing rotor speed to maximum power on rotor speed (484 rpm) may delay pitchup to a slightly higher airspeed. Changing the longitudinal C.G. from aft to forward caused the airspeed for self-induced pitchup to decrease approximately 1 knot.

Some pitchup tendency is present at all airspeeds above minimum power speed. This tendency to pitchup becomes greater as airspeed increases. Turbulent air or maneuvering flight reduces the speed at which pitchup occurs. A moderate rate of descent (500 fpm or more) delays pitchup airspeed beyond the maximum airspeed (V_D) demonstrated by the contractor. Climbing at high power may reduce the speed for self-induced pitchup. This tendency was noted during climbs to the high-altitude test sites at airspeeds above best rate-of-climb airspeed.

Pitchup appears to be a deterioration of the already poor angle-of-attack stability which is aggravated and goes divergent because of blade stall. This is evidenced by the fact that reduced rotor speed causes pitchup to occur at lower airspeed.

In addition to pitchup, low yaw damping contributed to high pilot workload during cruising flight. During flights in light turbulence, continual small cyclic and pedal inputs were required to maintain direction, airspeed, and altitude. This increased the pilot effort required to fly the helicopter and added to pilot fatigue. In moderate turbulence, the control inputs required to maintain straight and level flight were increased. In severe turbulence, 100-percent pilot effort was required to maintain straight and level flight. The most noticeable effect of turbulence was yaw and oscillations. Sideslip limitations may be exceeded during these oscillations.

It is recommended that airspeed in cruising flight in smooth air be limited to that shown in figure 50. Flights in moderate to severe turbulence should be avoided. For flight in turbulent air or maneuvering flight, airspeed should be reduced at least 10 percent below that shown in figure 50. These limitations should be included in the operator's manual. An explanation of the pitchup tendency and method of recovery should also be included. Pitchup during climbing flight at high airspeeds should be investigated in future tests.

2.3.4 Maneuvering Stability

Maneuvering stability tests were conducted to determine the longitudinal cyclic stick position and force gradients with increasing load factor (g). Tests were conducted at approximately 2085 pounds gross weight, forward and aft longitudinal C.G., approximately 5000 feet density altitude, 469 rotor rpm, 55 and 95 knots CAS. Data are presented in figures 53 through 56, appendix I. A summary of results is presented in table 2.

At approximately 2100 pounds, the maneuvering stability was acceptable below 95 knots CAS. Above 95 knots CAS, the pitchup tendency limited maneuvering.

Stick position gradients decreased as forward speed increased (table 2). With collective fixed, stick position gradients were positive except at forward C.G. at 95 knots CAS where they were zero. Of the conditions tested, this corresponded to the greatest pitchup tendency (see paragraph 2.3.3). Stick position gradients decreased when collective was applied (power increased).

Longitudinal stick forces at all conditions tested were 3 pounds or less; therefore, stick force gradients were difficult to determine. At 55 knots CAS the stick force gradient was approximately 20 to 30 pounds per g from 1.0 to 1.1 g and appeared to be neutral (no change in force with change in load factor) from 1.10 to 1.47 g with approximately 2 to 3 pounds pull required. At 95 knots CAS, the stick force gradient appeared to be negative with approximately a 1-pound push force required at the highest load factor obtained (1.35 g).

It was intended to obtain maneuvering stability data at best cruise speed (101 knots CAS); however, above 95 knots CAS the pitch- up tendency was too large to obtain stabilized conditions. Load factor with collective fixed was limited by the ability to maintain stabilized condition. At some conditions collective was applied to obtain higher load factors. Load factors achieved by increasing collective pitch were limited by apparent blade stall (large increase in 4-per-rev vibration). Higher load factors could be achieved during transient maneuvers but not under stabilized conditions. Limit load factors were not obtained at any condition during these tests.

[redacted]

TABLE 2

YOH-6A MANEUVERING STABILITY SUMMARY

Calibrated Airspeed kt	Average Density Altitude ft	Average Weight lb	Longitudinal Center of Gravity in	Longitudinal Stick Position Gradient in/g	Maximum Load Factor Achieved g	Collective Pitch Control
55	5000	2090	98.7	+ 3.2	1.47	Fixed
55	5000	2080	102.6	+ 4.0	1.36	Fixed
55	6500	2120	102.4	0	1.76	As Required
95	5000	2090	98.7	0	1.31	Fixed
95	5000	2080	102.6	+ 1.2	1.35	Fixed
95	5000	2100	102.4	- .6	1.6	As Required

[redacted]

It is desirable to have both positive force and position gradients (increasing pull force and aft stick displacement with increasing load factor). Even though these characteristics were not always met, maneuvering stability in the YOH-6A at 2100 pounds was considered acceptable below 95 knots CAS. This was because limit load factors could not be reached, control response was rapid and positive, and little pilot effort was required to maneuver. Pilot effort was small because of the light forces required even though the force gradients were sometimes large, negative, or nonlinear. Above 95 knots CAS, maneuvering stability became unacceptable because of an increasingly large negative force gradient which was due to the pitchup tendency. Maneuvering characteristics at higher weights should be investigated in future testing.

2.4 MISCELLANEOUS

2.4.1 Engine Acceleration

Engine acceleration characteristics were investigated using the three techniques specified in the Test Plan (reference c). Power recoveries from autorotation were made IGE and OGE. Power demands were made from a flare IGE. Power recoveries OGE yielded the most consistent data. At each condition specified, collective was applied at three different rates. This did not give sufficient data to make summaries or draw complete conclusions. At each condition, at least six collective application rates would be necessary to obtain sufficient data. Data are not presented in this report but will be retained for comparison with data from the production OH-6A with the T-63-A5-A engine installed.

The engine response of the YOH-6A was generally considered very poor, particularly when compared with the aircraft response to other controls. Engine response was considered objectionable for three reasons:

a. It was slow. Engine acceleration from a low power to a high power normally required 3 to 5 seconds.

b. It was unpredictable. Acceleration time varied with engine condition. When the compressor or compressor discharge pressure filter was dirty, acceleration times increased above 5 seconds and sometimes the engine would "hang up" (would not accelerate) if N_1 speed fellbelow approximately 62 percent.

c. Acceleration was nonlinear. Very little power would be delivered during the early portion of the acceleration and the majority of the power change would come during the last second of the acceleration. Some "overshoot" which would occur compromised directional control. Moderate yaw oscillations occurred when making a large power demand even after the pilot had become familiar with the engine response characteristics and could anticipate the power delivery rate.

2.4.2 Engine Start

Engine air starts were attempted and ground starts monitored to evaluate the starting performance of the engine. Ground starts were made at altitudes of sea level, 2600, 4200 and 11,500 feet and airstarts were made at altitudes from 12,500 to 4000 feet. The starting procedure specified in the YOH-6A operator's manual was used for all starts. Allflying was done in apparent dust-free conditions. The compressor, however, generally needed to be cleaned at flying time intervals of approximately 30 hours. It was sometimes necessary to clean the compressor more frequently when a large decrease in power (approximately 10 percent) was apparent. Three different engines and several fuel controls were used during this program. Each engine-fuel control combination exhibited different starting characteristics. Engine model specification 580-E requires that starting times not exceed 1 minute.

2.4.2.1 Ground Start

An auxiliary power unit (APU) was normally used for ground starts because of the increased possibility of a hot start's occurring when the aircraft's battery was used. Also, starting time when the aircraft's battery was used was much longer than when an APU was used under the same conditions. APU's with capacities of 400 amperes and 1000 amperes were used. The engine-starter motor was rated at 100 amperes; therefore, any starting deficiencies when an APU was used were due to the engine-starter motor combination. An apparently inadequate aircraft battery contributed to starting problems when battery starts were attempted.

A summary of approximate starting times experienced is shown in table 3.

TABLE 3

APPROXIMATE YOH-6A GROUND STARTING TIMES

	FLIGHT TIME SINCE COMPRESSOR CLEANING			
	0-10 hr	10-30 hr	0-10 hr	10-30 hr
	APU Starts		Battery Starts	
Field Elevation	Starting Time	Starting Time	Starting Time	Starting Time
Sea Level	Less than 1 min	1 to 5 min	Less than 1 min	1 to 5 min
2600 ft	Less than 1 min	1 to 5 min	Less than 1 min	1 to 5 min
4200 ft	Less than 1 min	1 to 5 min	1 to 5 min	Not possible
11,500 ft	1 to 5 min	Marginal	Not possible	Not possible

At a field elevation of 4200 feet, over-temperatures were experienced randomly during starts when either the APU or the helicopter's battery was used. Hot-starts occurred more frequently when the battery was used. As the compressor became dirty or when ambient temperatures were above standard, hot starts occurred even more frequently. With a clean compressor, in temperatures colder than standard (-10 to +6 degrees C) during the first start of the day, peak starting TOT's would be from 350 degrees C to 450 degrees C (low) and starting time would be in excess of 1 minute. During subsequent starts (warm engine) peak TOT temperatures would be 50 to 100 degrees C hotter and starting time would decrease. At times, the initial start was aborted even though a hot-start did not occur and a second start was initiated so starting time would not be excessive.

At a field elevation of 11,500 feet when an APU was used and after approximately 10 flying hours had been accumulated since compressor cleaning, it was necessary to control manually the fuel during start to avoid an over-temperature condition. This was accomplished by rotating the twist-grip from start position to cut-off position and back intermittently until the engine had accelerated sufficiently to leave the twist-grip in the start position without causing an over-temperature condition. Battery starts were not possible at this field elevation.

2.4.2.2 Air Start

Thirteen airstart attempts were made at airspeeds from 50 to 90 knots CAS. Gas producer speed (N_1) and power turbine speed (N_2) were allowed to go to zero before the starts were initiated. Five of these attempts resulted in successful starts at altitudes up to 12,500 feet. The rest were aborted because of impending over-temperatures or actual over-temperatures. The rapid rise of TOT (200 to 300 degrees C per second) after ignition and the poor twist-grip design (which nearly required a two-handed starting operation) required very good judgment to determine if a start should or should not be aborted. All successful starts were made with a recently cleaned compressor.

The condition of the aircraft battery was the major factor influencing airstart capability. No successful airstarts were made with the installed battery in normal service condition. One successful start in 6 attempts was made after servicing the battery (discharging and recharging according to the prescribed schedule). A new battery produced 4 successful starts in 4 attempts. There was no method of determining battery condition.

The standard TOT indicator was used for the last 4 successful starts. This contributed to starting capability since the standard indicator had a greater lag than the test indicator. This resulted in lower apparent TOT's and less time at these temperatures. A more responsive TOT indicator should be installed to give the pilot a more accurate indication of TOT during transient conditions (starting and accelerations).

The successful airstarts required approximately 2 minutes. Approximately 3000 feet of altitude was lost during successful airstarts at the airspeed for minimum rate of descent (55 knots CAS).

2.4.3 Airspeed Calibration

The standard YOH-6A airspeed system was calibrated to determine the position error. The calibration was made in level flight and checked in climbs at takeoff power and descents up to 1000 fpm at 469 rotor rpm and 2100 pounds gross weight. Results are presented in figure 57, appendix I.

The standard airspeed system was found to have a position error of 4 knots at 34 knots CAS. The position error decreased to zero at approximately 90 knots CAS and remained zero up to the maximum airspeed tested (123.5 knots CAS). The position error in climbs and powered descents was the same as in level flight. The airspeed system was stable above approximately 25 knots CAS. Below this speed it became unstable and reliable. Position error during full autorotational descents could not be determined because a trailing bomb was used during the calibration and the danger of fouling the cable in the tail rotor existed.

These results do not agree with those obtained during the original tests (reference k). The airspeed calibration made during the original tests was made with the test boom installed. This was found to influence the position error because of the close proximity of the test boom to the standard pitot source. The test boom was removed for this calibration.

SECTION 3 APPENDICES

Appendix I TEST DATA

FIGURE NO. 1

SUMMARY TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM

WIND VELOCITY ≤ 2 KNOTS — TAKE-OFF POWER

GROSS WEIGHT = 1900 LB

TECHNIQUE:

——— LEVEL ACCELERATION FROM A 2 FOOT HOVER

- - - - - CLIMB AND ACCELERATION FROM LIGHT ON SKIDS

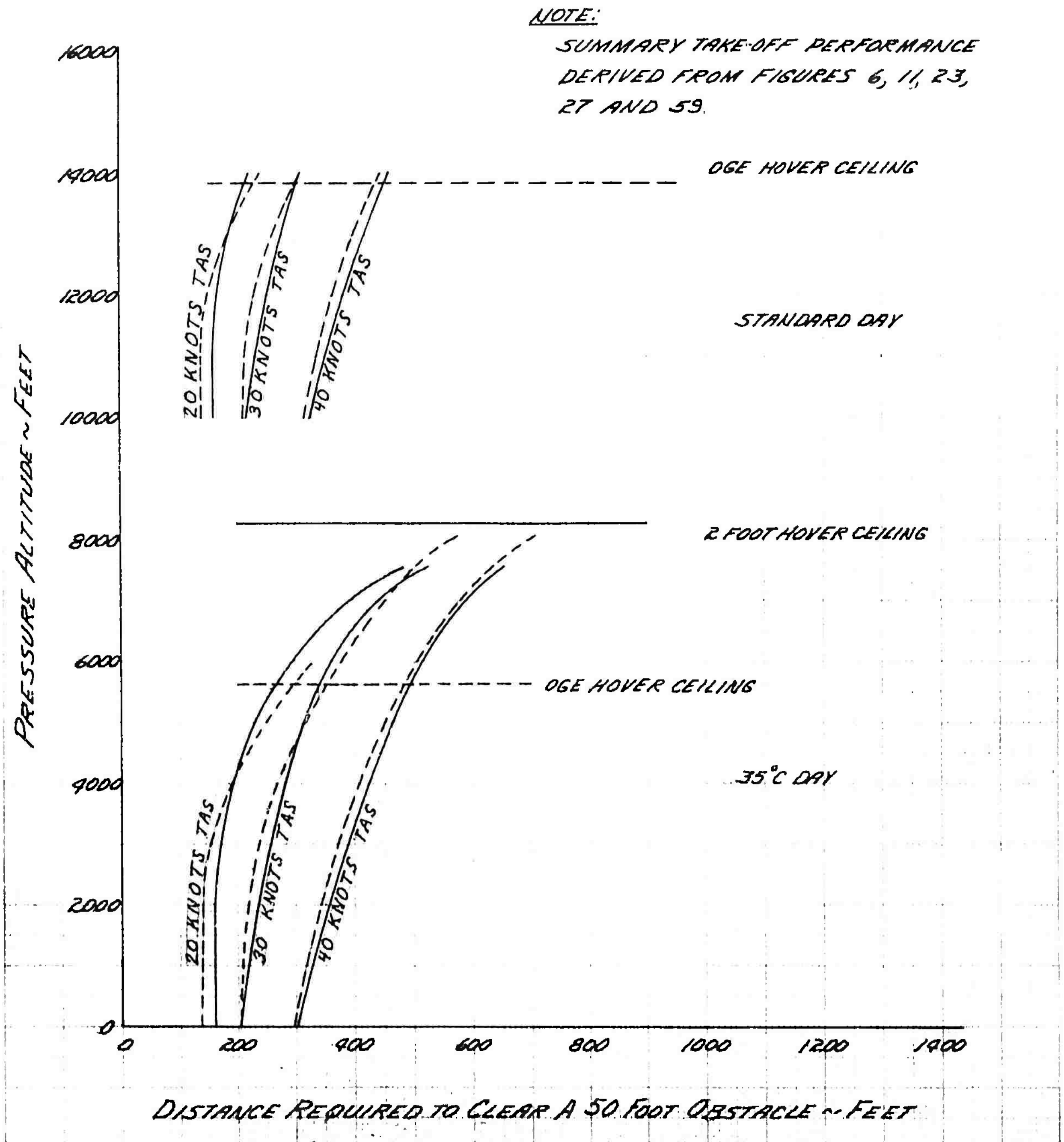


FIGURE NO. 2
SUMMARY TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM

WIND VELOCITY ≤ 2 KNOTS — TAKE-OFF POWER

GROSS WEIGHT = 2100 LB.

TECHNIQUE:

———— LEVEL ACCELERATION FROM A 2 FOOT HOVER

- - - - - CLIMB AND ACCELERATION FROM LIGHT ON SKIDS

NOTE:
SUMMARY TAKE-OFF PERFORMANCE DERIVED FROM FIGURES 6, 11, 23, 27 AND 59.

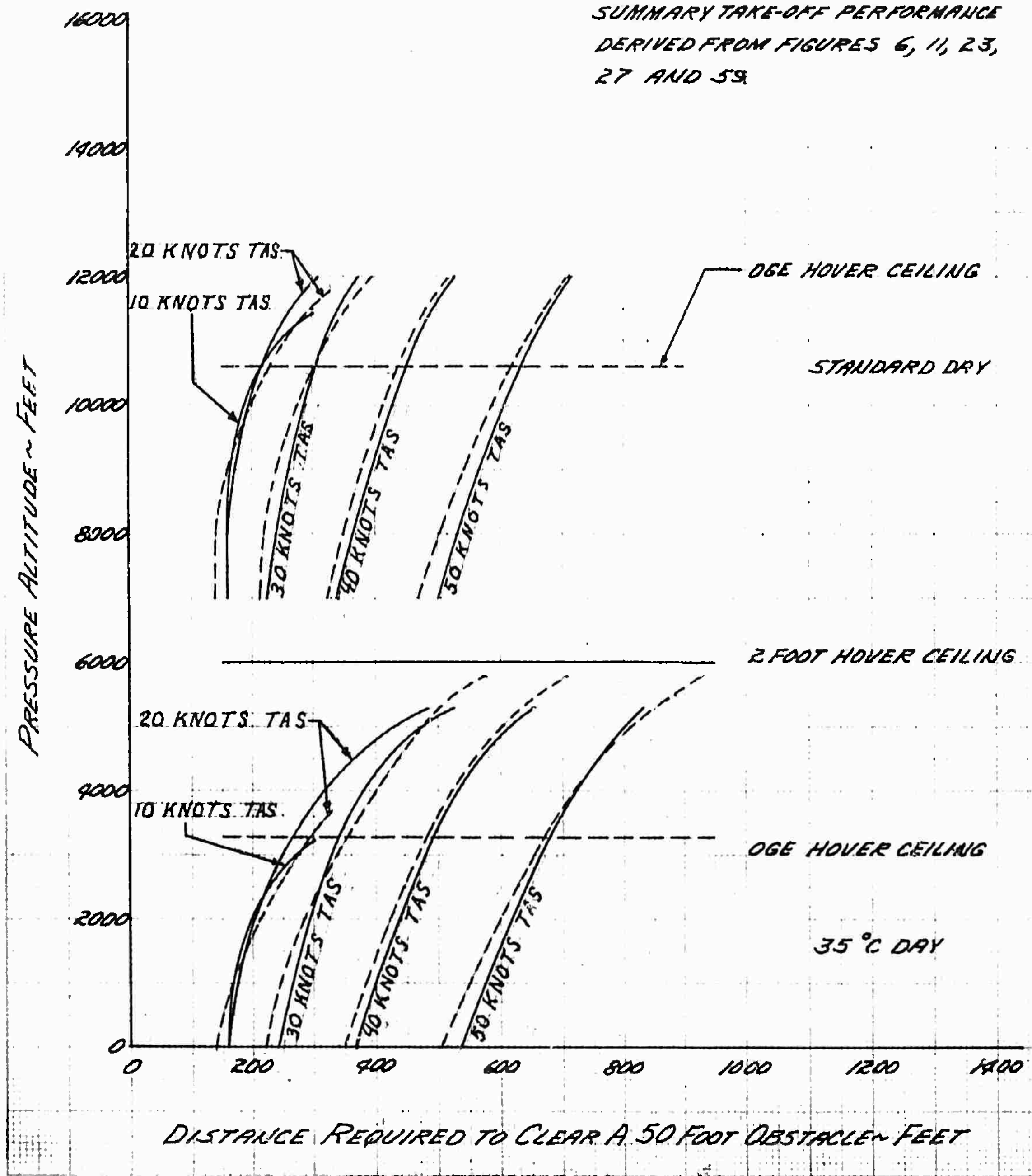


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FIGURE NO. 3
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM
WIND VELOCITY ≤ 2 KNOTS — TAKE-OFF POWER
GROSS WEIGHT = 2300 LB

TECHNIQUE:
——— LEVEL ACCELERATION FROM A 2 FOOT HOVER
- - - - CLIMB AND ACCELERATION FROM LIGHT ON SKIDS

NOTE:
SUMMARY TAKE-OFF PERFORMANCE DERIVED FROM FIGURES 6, 11, 23, 27 AND 59.

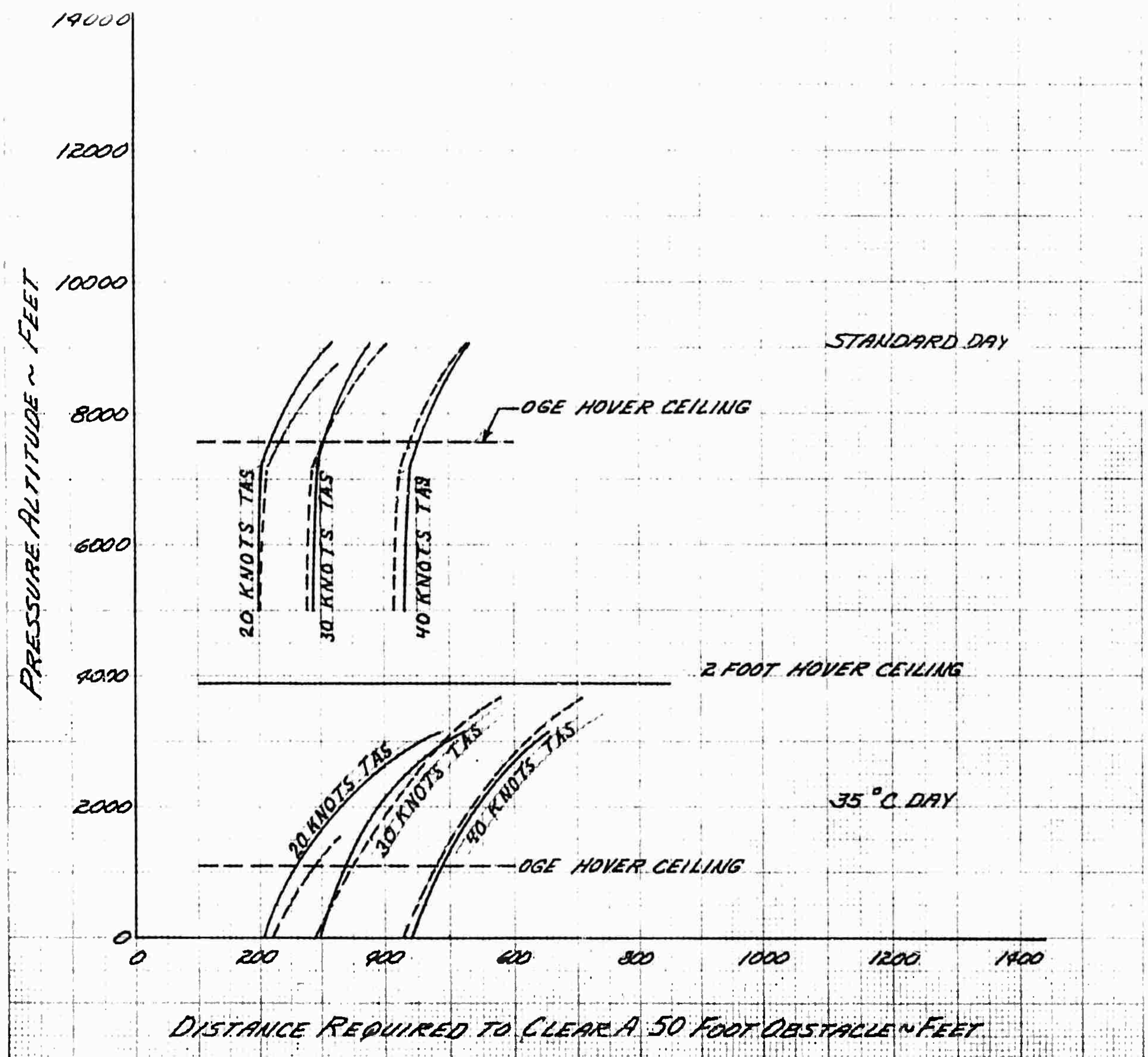

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FIGURE NO. 4
SUMMARY TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM

WIND VELOCITY ≤ 2 KNOTS — TAKE-OFF POWER

GROSS WEIGHT = 2500 LB

TECHNIQUE:

——— LEVEL ACCELERATION FROM A 2 FOOT HOVER

- - - - - - CLIMB AND ACCELERATION FROM LIGHT ON SKIDS

NOTE:

SUMMARY TAKE-OFF PERFORMANCE DERIVED FROM FIGURES 6, 11, 23 AND 59.

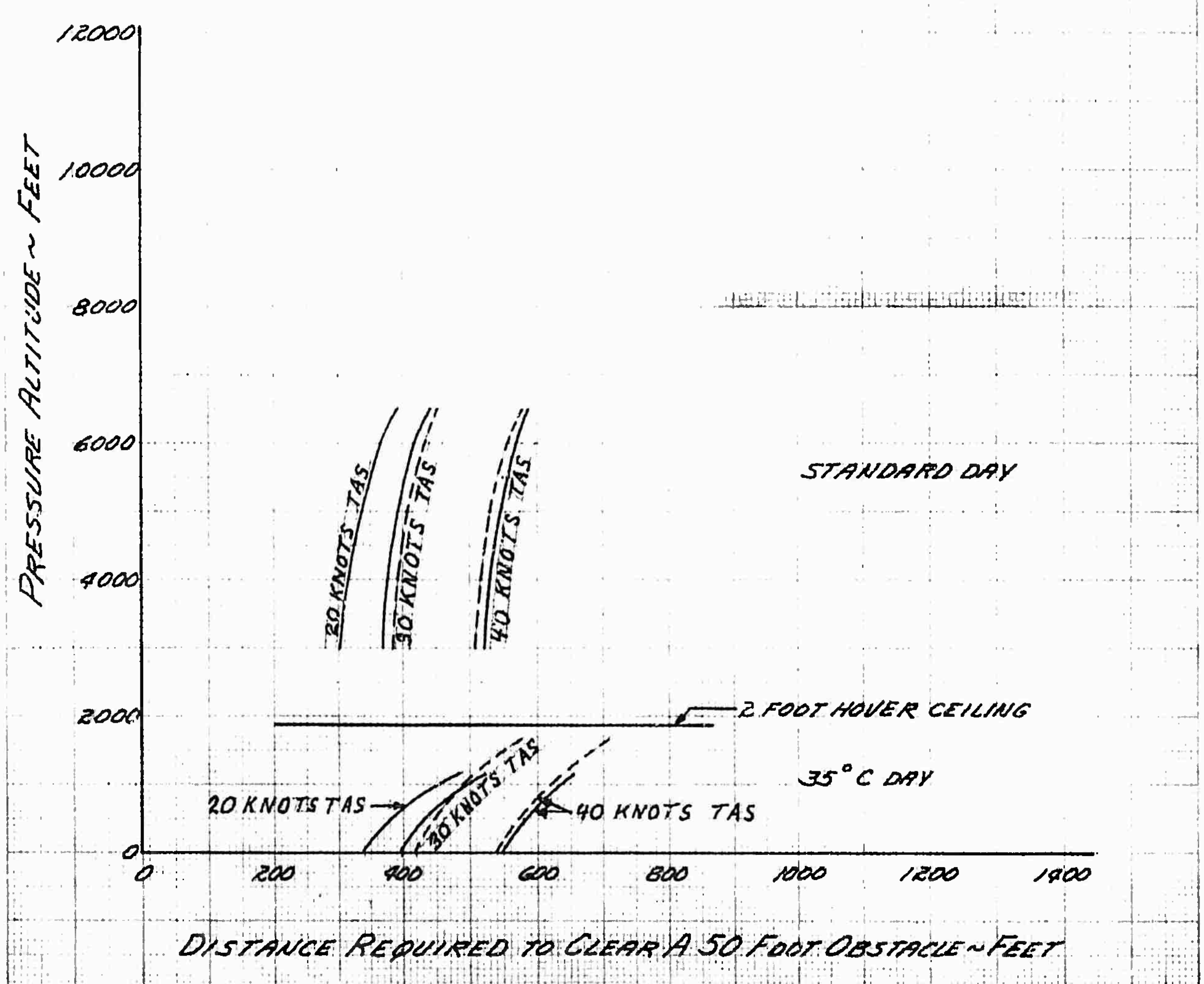

FIGURE NO 5

SUMMARY TAKE OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM

WIND VELOCITY ≤ 2 KNOTS — TAKE-OFF POWER

GROSS WEIGHT = 2700 LB

TECHNIQUE:

----- CLIMB AND ACCELERATION FROM LIGHT ON SKIDS

NOTE:

SUMMARY TAKE-OFF PERFORMANCE DERIVED FROM FIGURES 11, 23 AND 59.

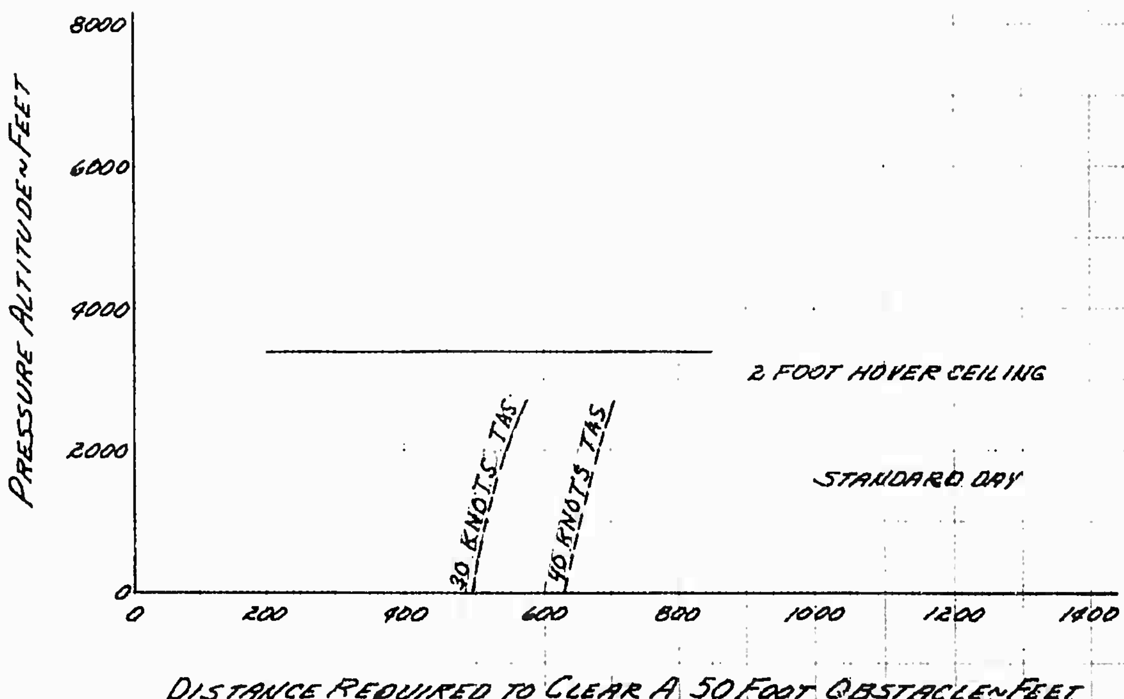

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FIGURE No. 6
VARIATION OF TAKE-OFF DISTANCE WITH
CLIMB-OUT AIRSPEED AND ΔC_P
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION   ROTOR SPEED = 469 RPM
WIND VELOCITY ≤ 2 KNOTS   TAKE-OFF POWER
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

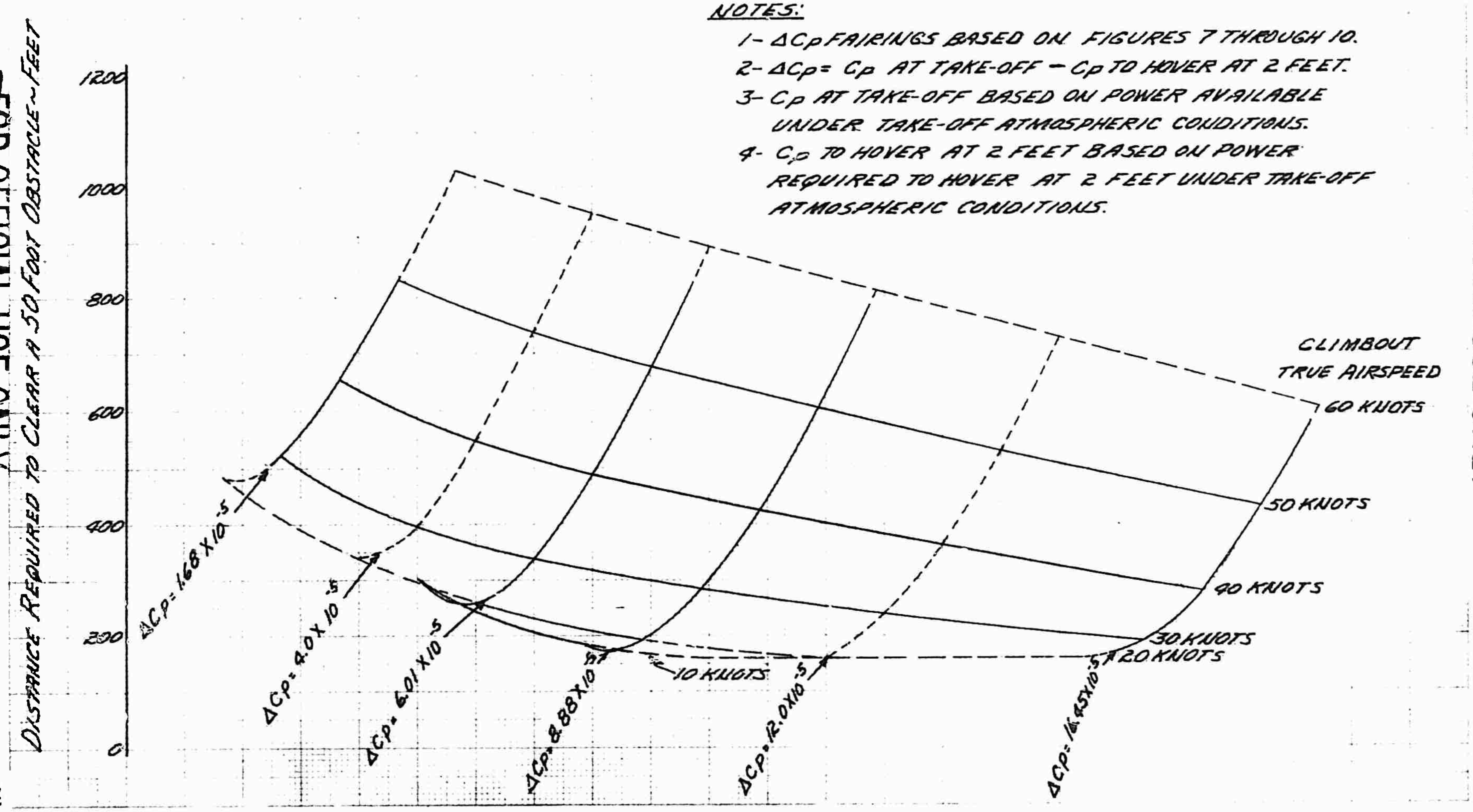

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FIGURE NO. 7

TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2690 LB.
FREE AIR TEMP. = 2. °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_p = 1.68 \times 10^{-5}$
PRESSURE ALTITUDE = 4030 FT.
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

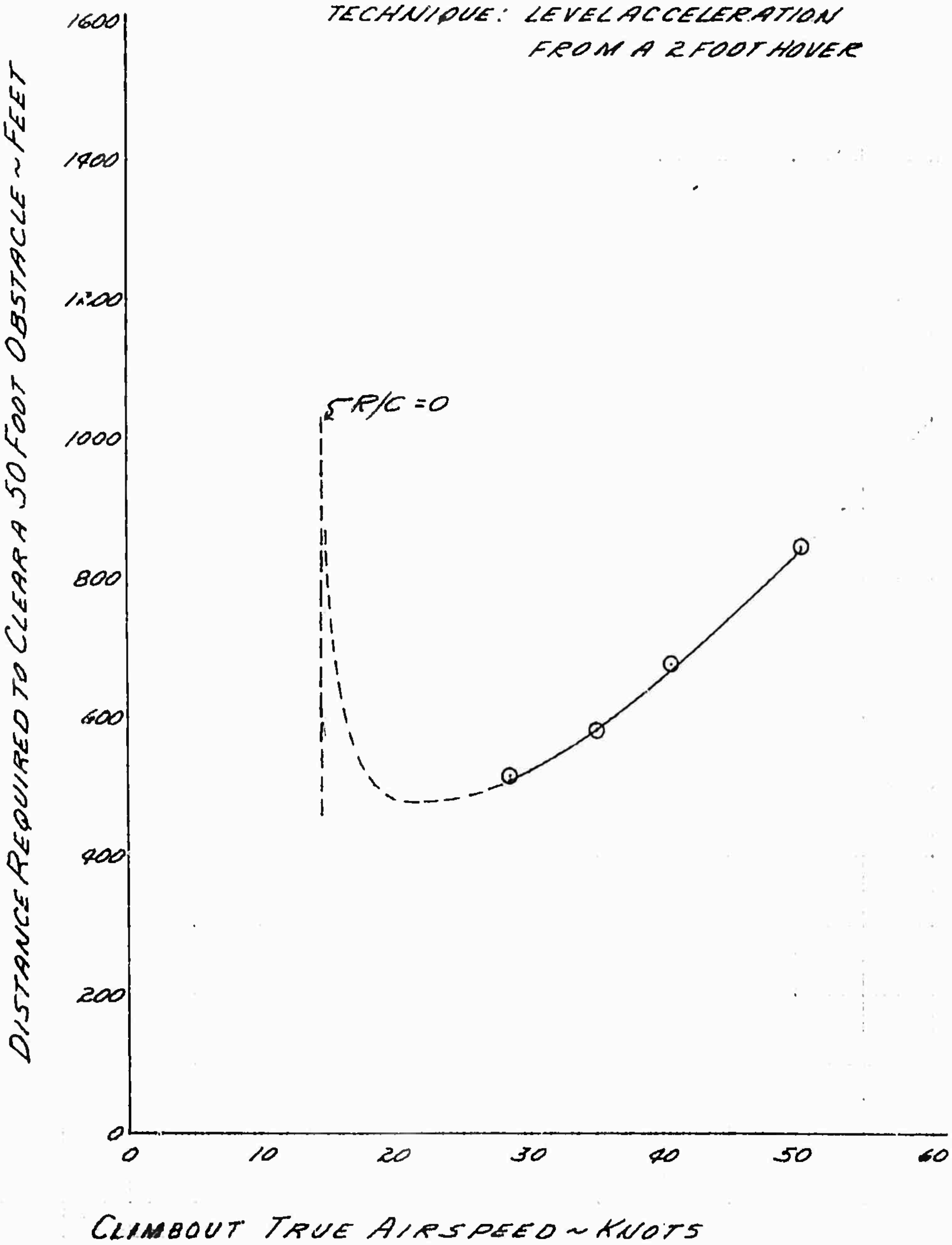

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FIGURE No. 8

TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2490 LB.
FREE AIR TEMP. = 7 °C
WIND VELOCITY ≤ 2 KTS
$\Delta C_p = 6.01 \times 10^{-5}$
PRESSURE ALTITUDE = 4160 FT.
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER

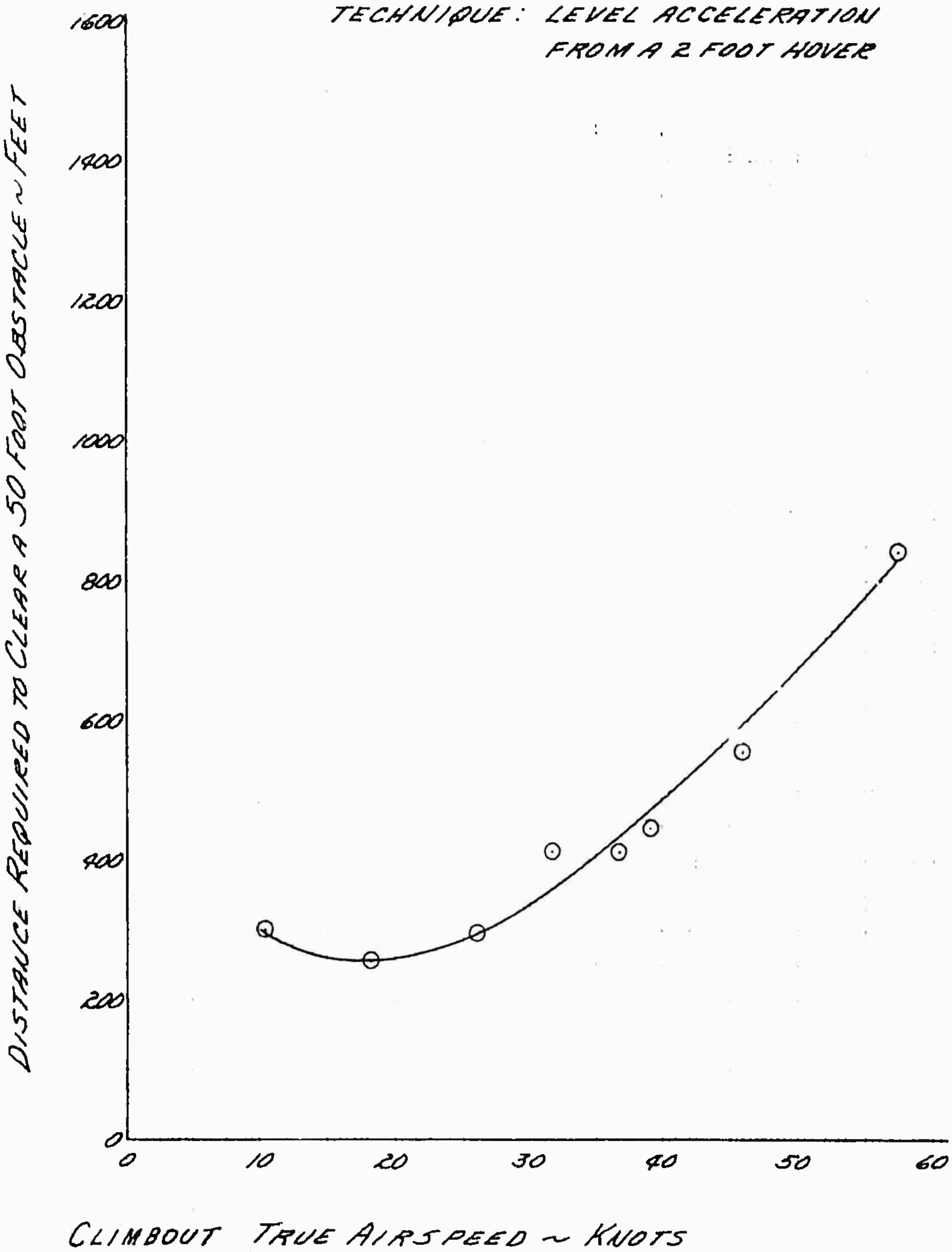

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FIGURE NO. 9

TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2300 LB.
FREE AIR TEMP. = 19 °C
WIND VELOCITY ≤ 2 KTS.
ΔC_p = 8.88×10^{-5}
PRESSURE ALTITUDE = 3770 FT.
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER

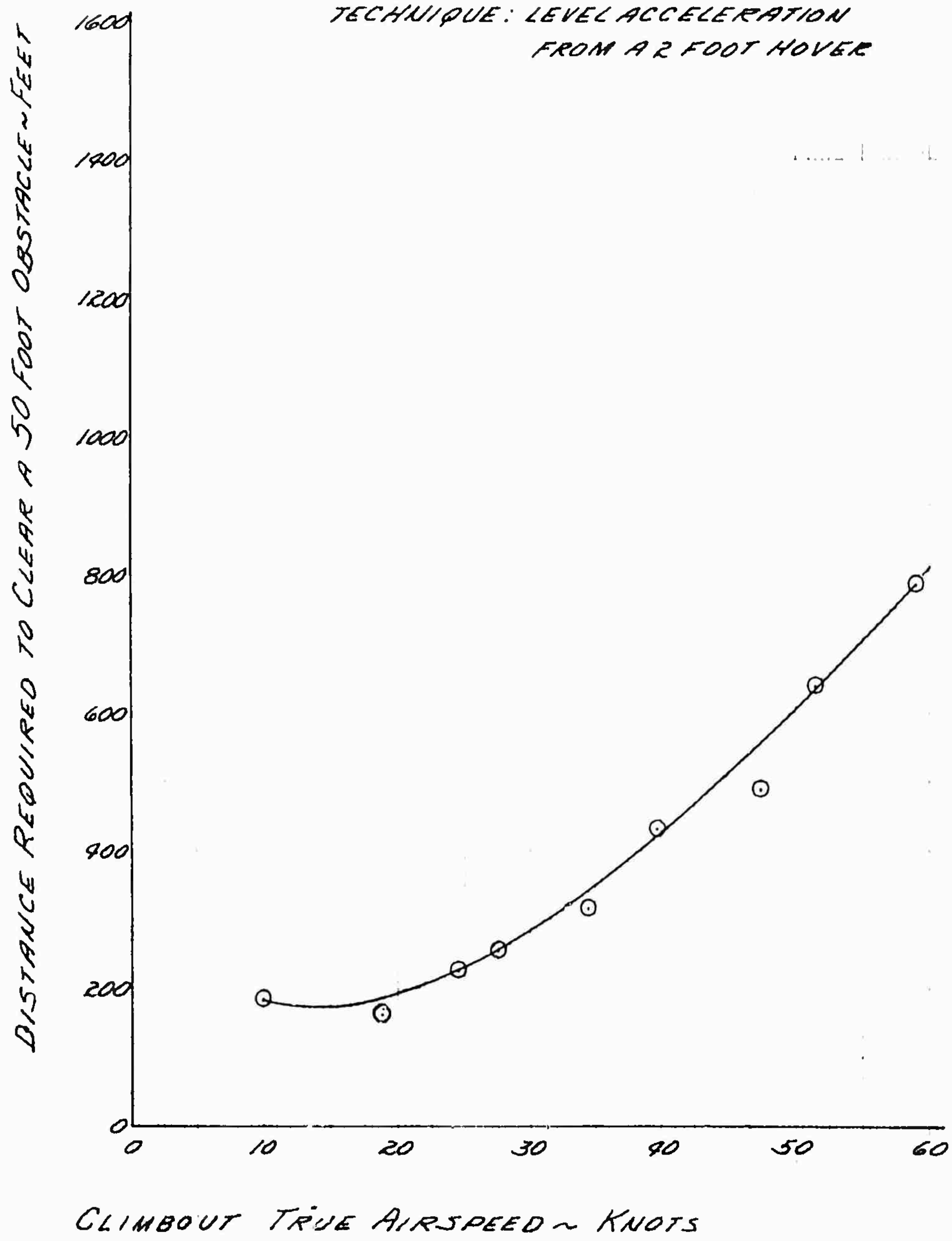

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FIGURE No. 10

TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION

ROTOR SPEED = 469 RPM

GROSS WEIGHT = 1900 LB.

FREE AIR TEMP. = 17 °C

WIND VELOCITY ≤ 2 KTS.

$\Delta C_p = 16.45 \times 10^{-5}$

PRESSURE ALTITUDE = 4070 FT.

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

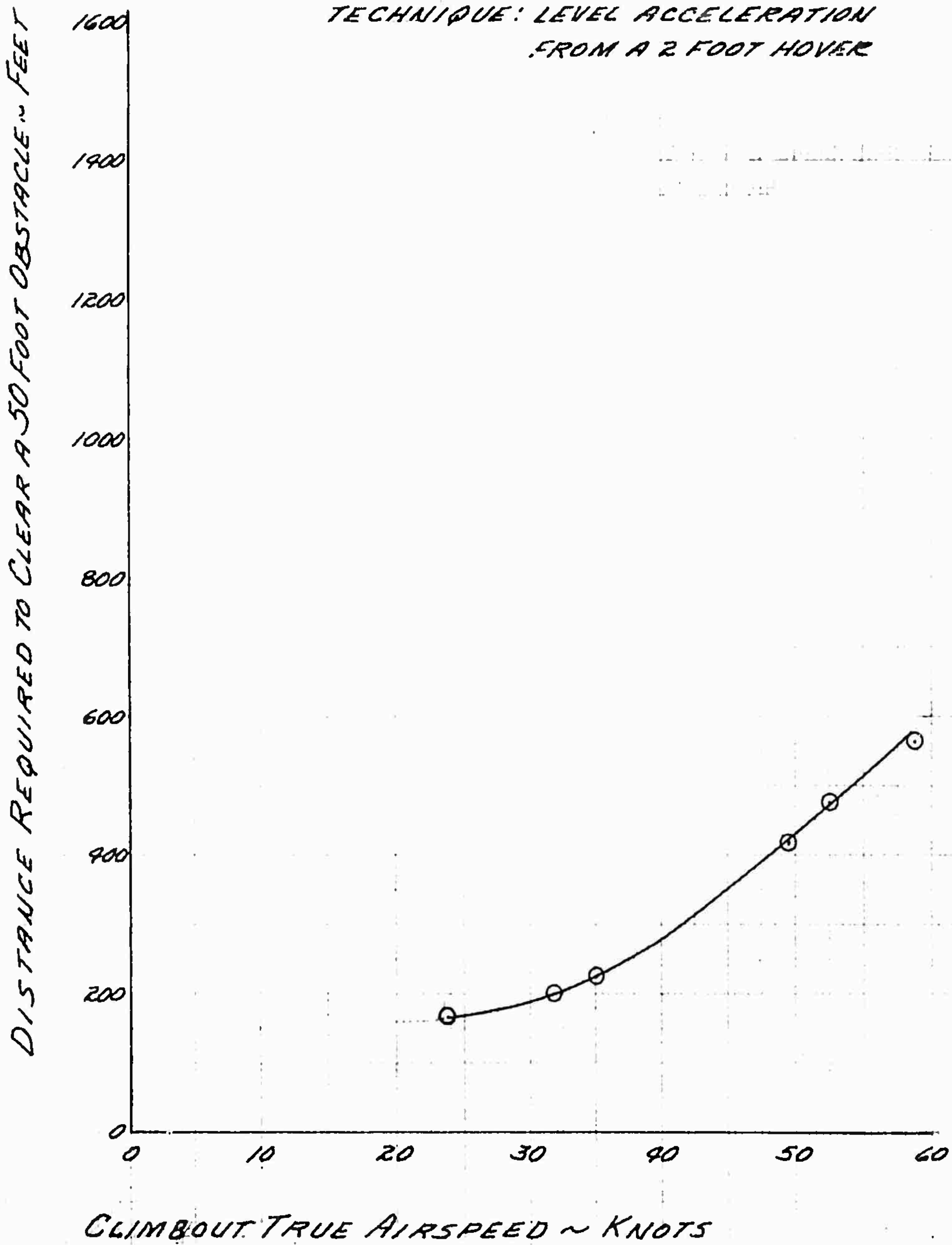

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FIGURE NO. 11

VARIATION OF TAKE-OFF DISTANCE WITH CLIMB-OUT AIRSPEED AND ΔC_p

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION ROTOR SPEED: 469 RPM

WIND VELOCITY ≤ 2 KNOTS TAKE-OFF POWER

TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

NOTES:

1- ΔC_p FAIRINGS BASED ON FIGURES 12 THROUGH 21.
2- ΔC_p = C_p AT TAKE-OFF − C_p TO HOVER AT 2 FEET.
3- C_p AT TAKE-OFF BASED ON POWER AVAILABLE UNDER TAKE-OFF ATMOSPHERIC CONDITIONS.
4- C_p TO HOVER AT 2 FEET BASED ON POWER REQUIRED TO HOVER AT 2 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS.

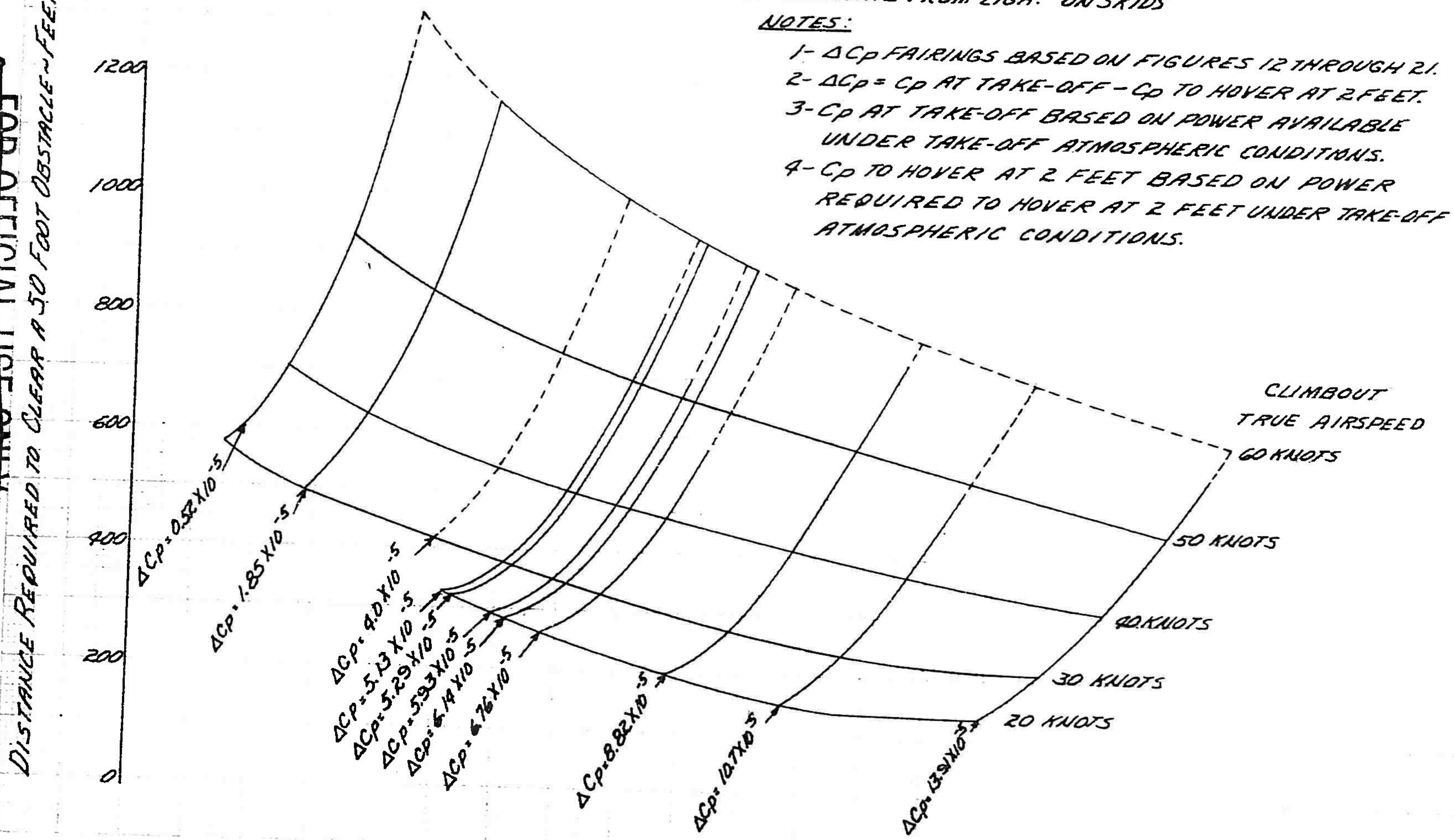

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FIGURE NO. 12
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2550 LB.
FREE AIR TEMP. = 22 °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_P = 0.52 \times 10^{-5}$
PRESSURE ALTITUDE = 4030 FT.
TECHNIQUE: CLIMB AND ACCELERATE
FROM LIGHT ON SKIDS

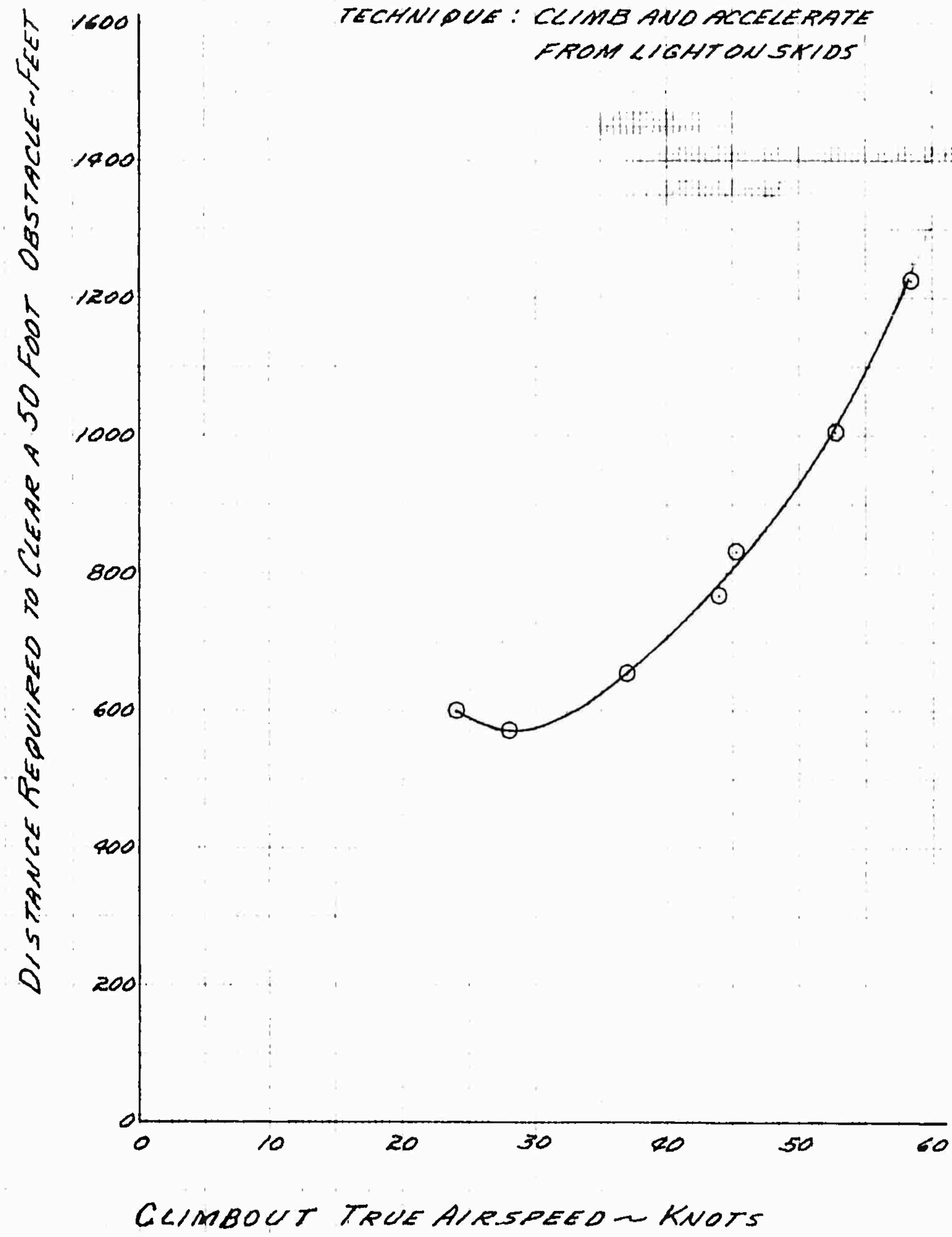

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FIGURE NO. 13
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION
ROTOR SPEED = 471 RPM
GROSS WEIGHT = 2080 LB.
FREE AIR TEMP. = -4. °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_p = 1.85 \times 10^{-5}$
PRESSURE ALTITUDE = 11710 FT.
TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

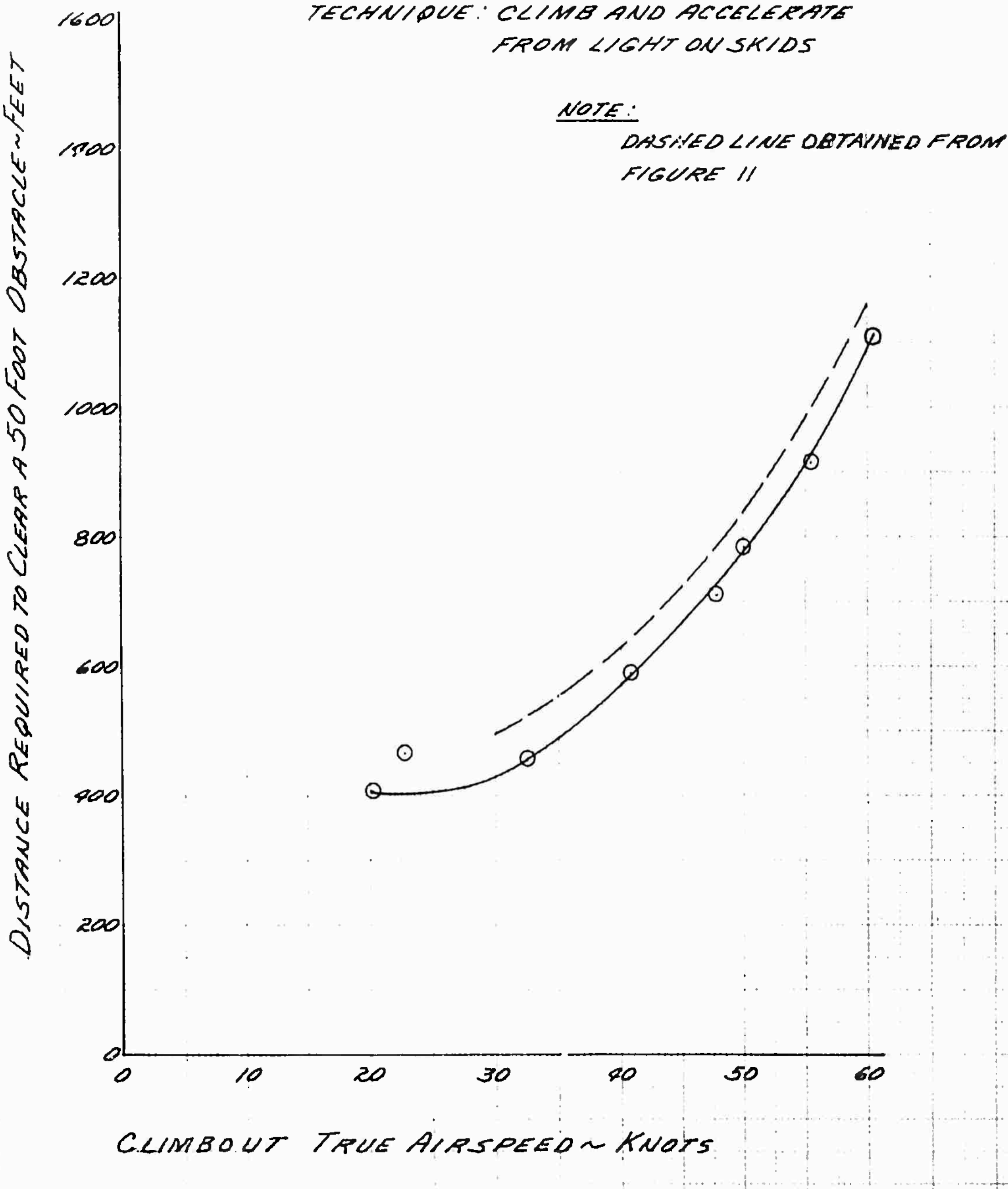

FIGURE NO. 14
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION
ROTOR SPEED: 469 RPM
GROSS WEIGHT: 2470 LB.
FREE AIR TEMP.: 16.5 °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_p = 5.13 \times 10^{-5}$
PRESSURE ALTITUDE: 4100 FT.
TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

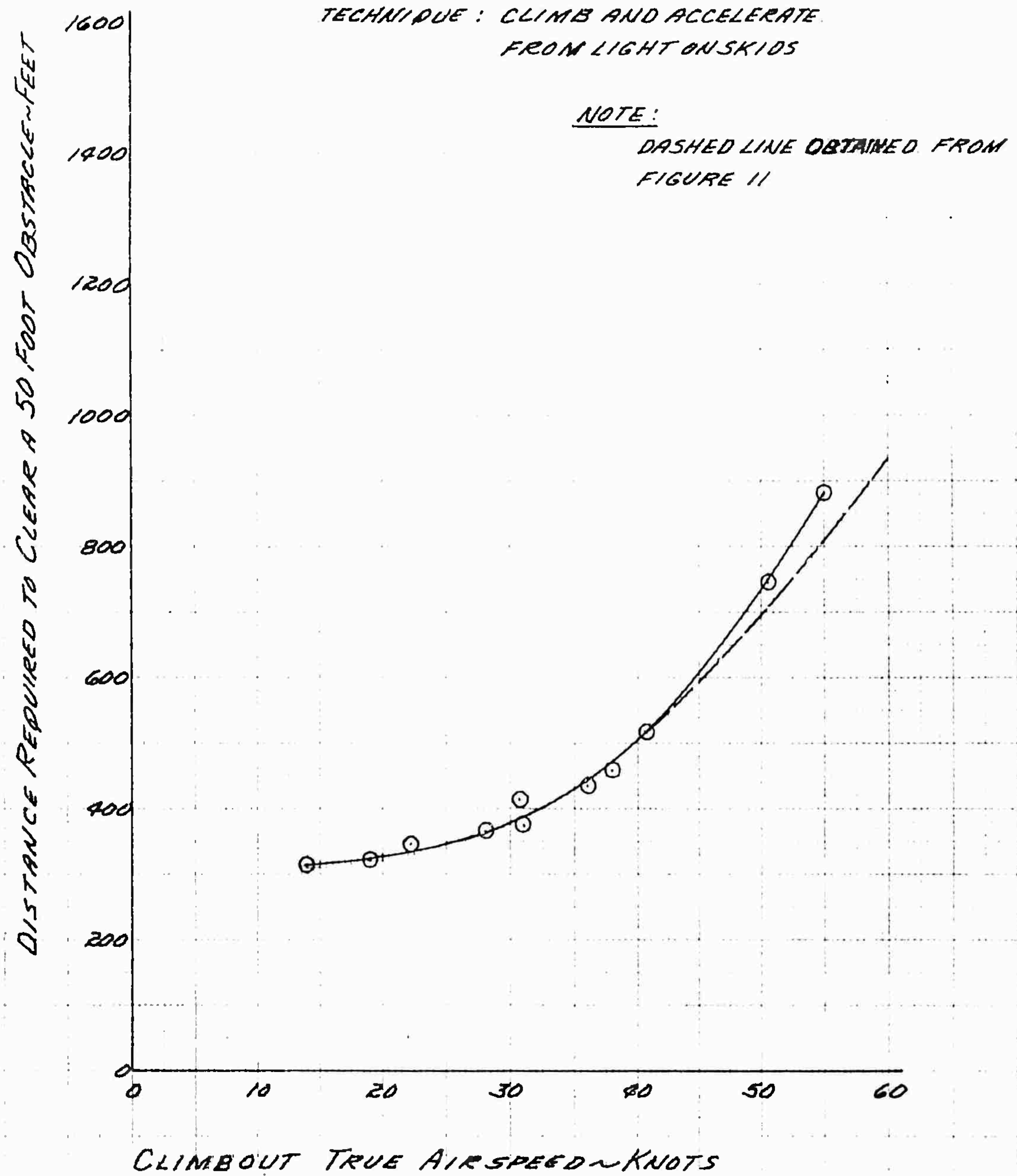

FIGURE NO. 15
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM.
GROSS WEIGHT = 2220 LB.
FREE AIR TEMP. = 26.5 °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_p = 5.29 \times 10^{-5}$
PRESSURE ALTITUDE = 3990 FT.
TECHNIQUE: CLIMB AND ACCELERATE
FROM LIGHT ON SKIDS

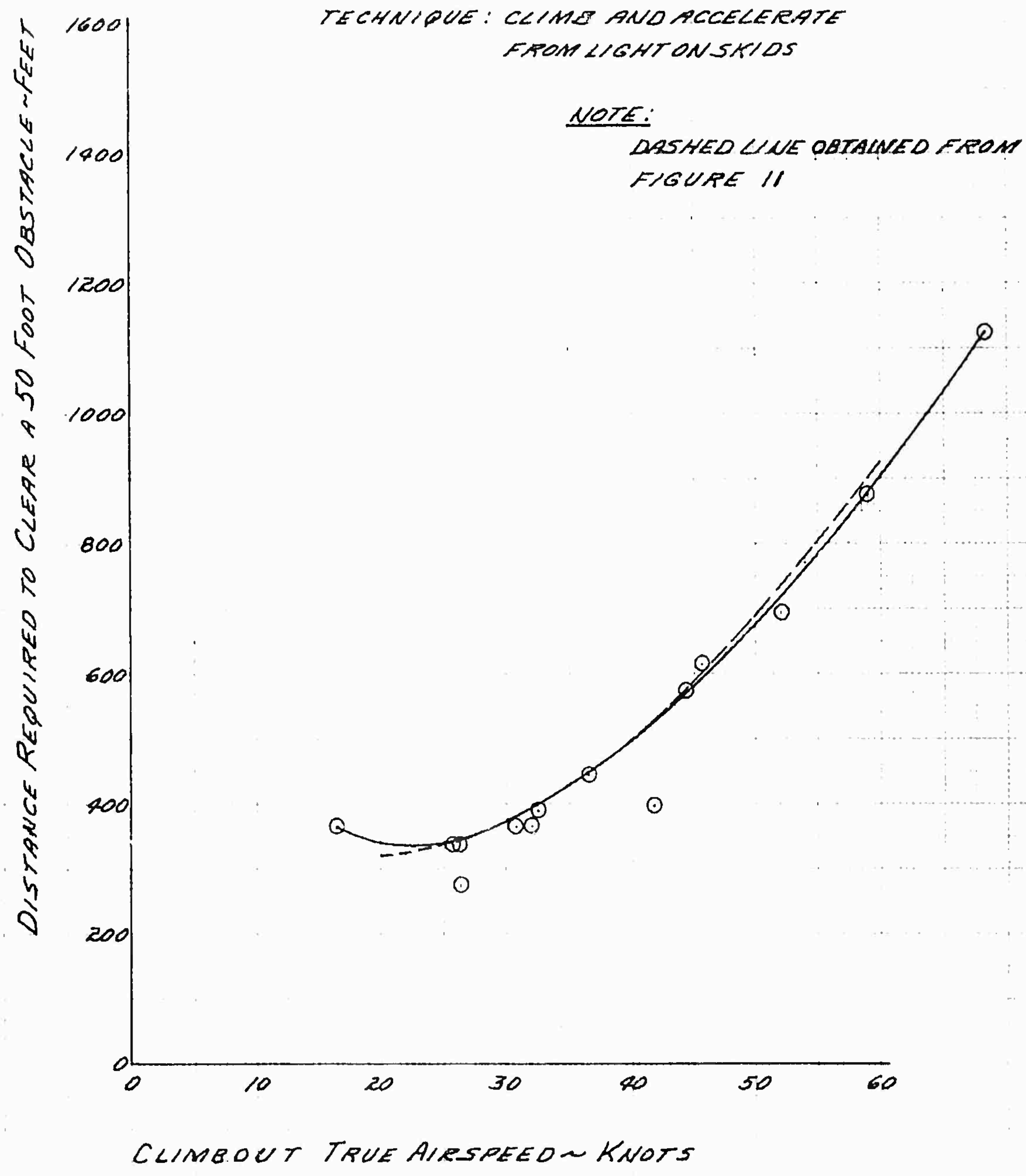

FIGURE NO. 16

TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION

ROTOR SPEED: 469 RPM

GROSS WEIGHT = 2120 LB.

FREE AIR TEMP. = 28.5 °C

WIND VELOCITY ≤ 2 KTS.

$\Delta C_P = 5.93 \times 10^{-5}$

PRESSURE ALTITUDE = 3990 FT.

TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

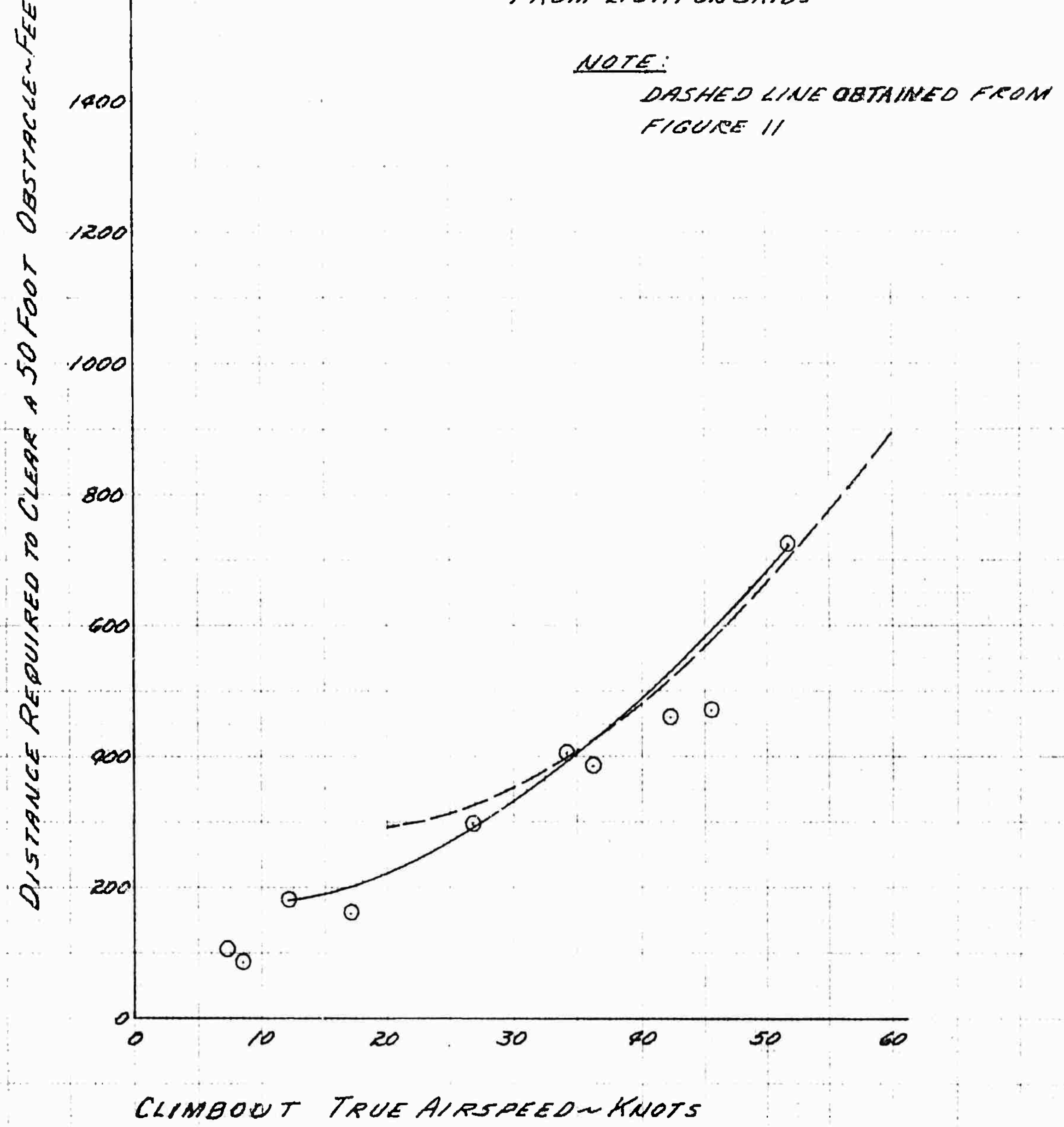

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FIGURE NO. 17

TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FOOT OBSTACLE

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION

ROTOR SPEED: 469 RPM

GROSS WEIGHT: 1970 LB.

FREE AIR TEMP.: 3 °C

WIND VELOCITY ≤ 2 KTS.

$\Delta C_p = 6.14 \times 10^{-5}$

PRESSURE ALTITUDE = 11560 FT.

TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

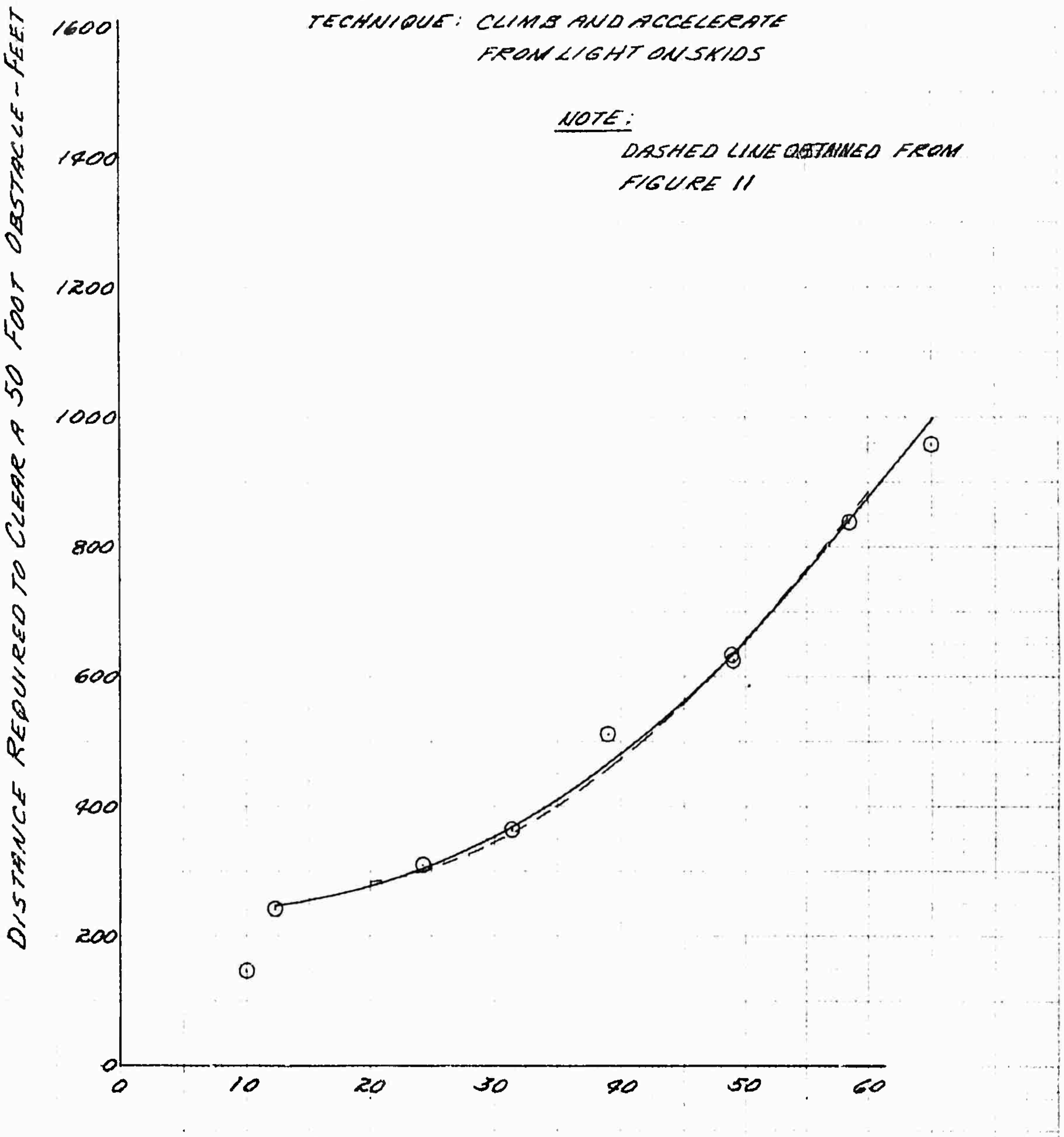

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FIGURE NO. 18
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

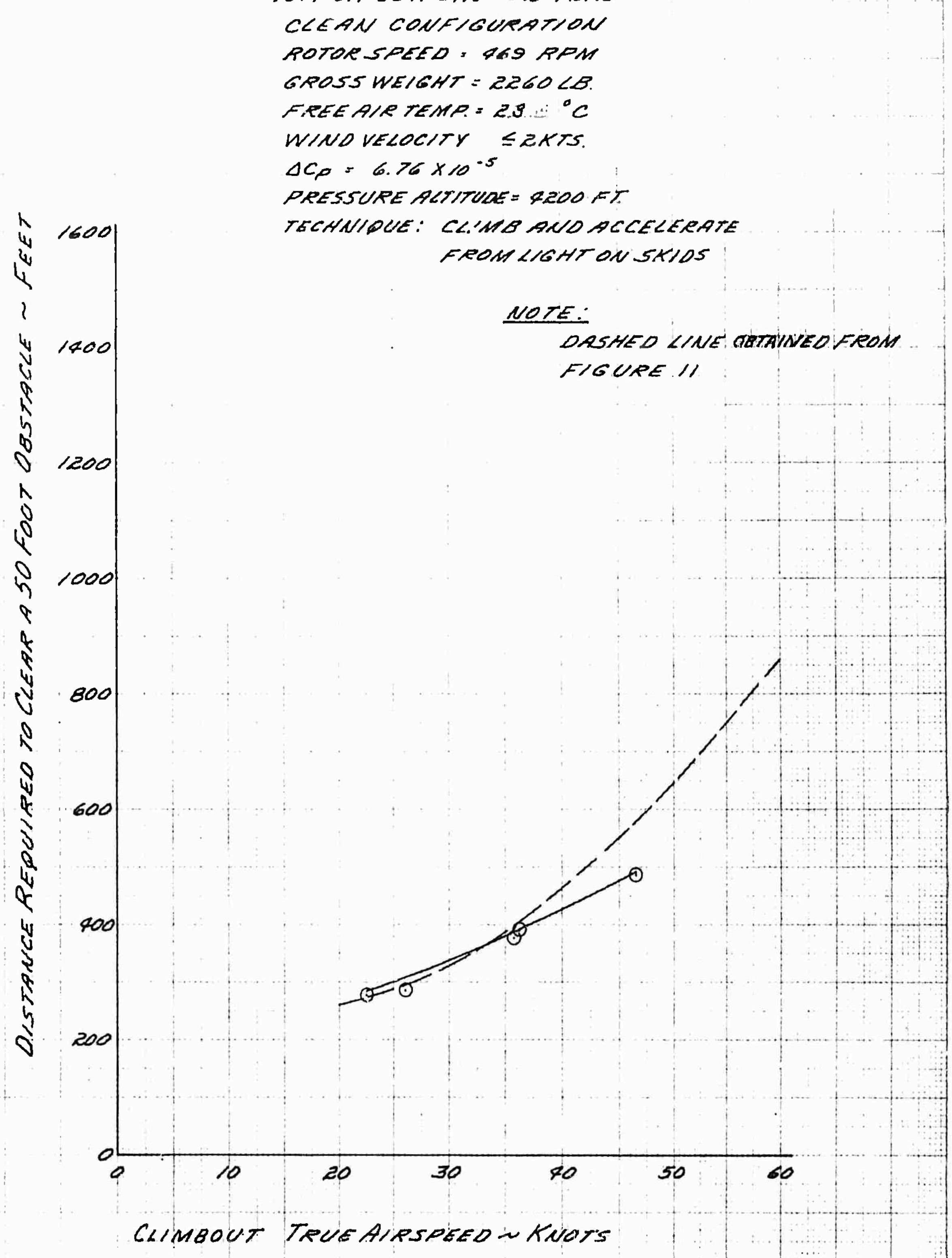

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FIGURE NO. 19
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2070 LB.
FREE AIR TEMP. = 27 °C
WIND VELOCITY ≤ 3 KTS.
ΔC_p = 8.82×10^{-5}
PRESSURE ALTITUDE = 3980 FT.
TECHNIQUE : CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

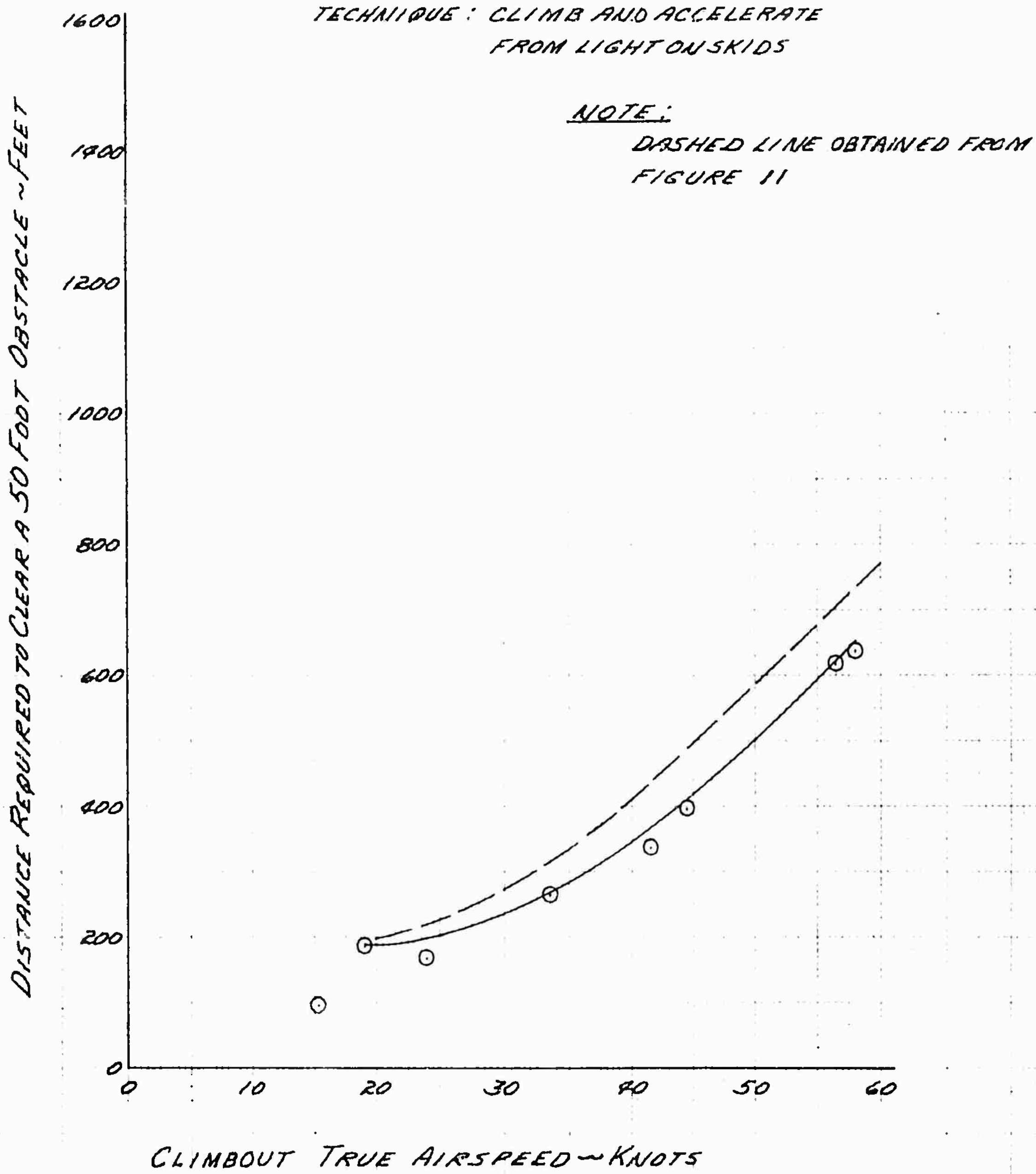

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FIGURE NO. 20
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2100 LB.
FREE AIR TEMP. = 25 °C
WIND VELOCITY ≤ 2 KTS
ΔC_P = 10.7×10^{-5}
PRESSURE ALTITUDE = 4100 FT.
TECHNIQUE: CLIMB AND ACCELERATE FROM LIGHT ON SKIDS

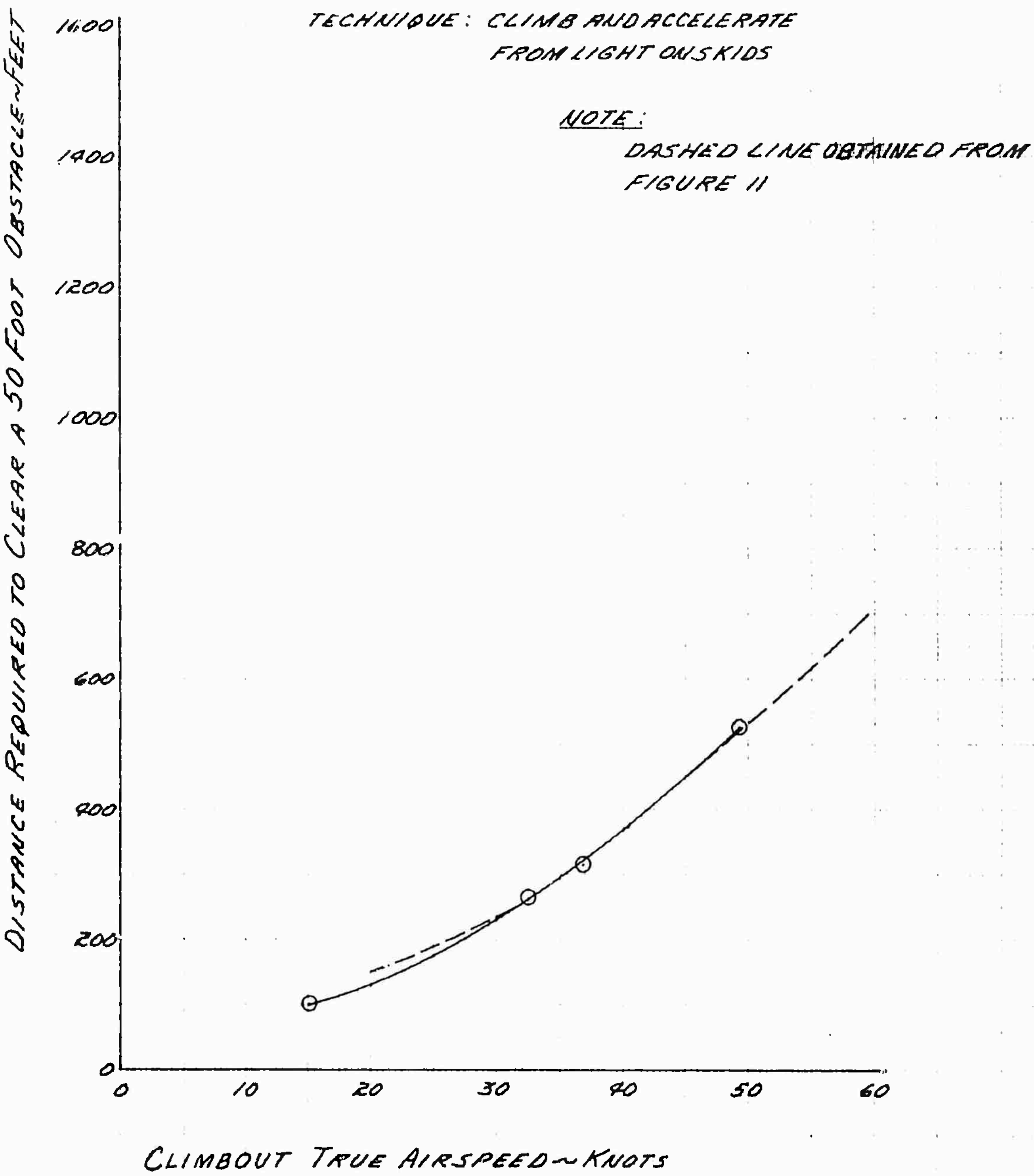

FIGURE NO. 21
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION
ROTOR SPEED = 469 RPM
GROSS WEIGHT = 2050 LB.
FREE AIR TEMP. = 1.9 °C
WIND VELOCITY ≤ 2 KTS.
$\Delta C_p = 13.91 \times 10^{-5}$
PRESSURE ALTITUDE = 4300 FT.
TECHNIQUE: CLIMB AND ACCELERATE
FROM LIGHT ON SKIDS

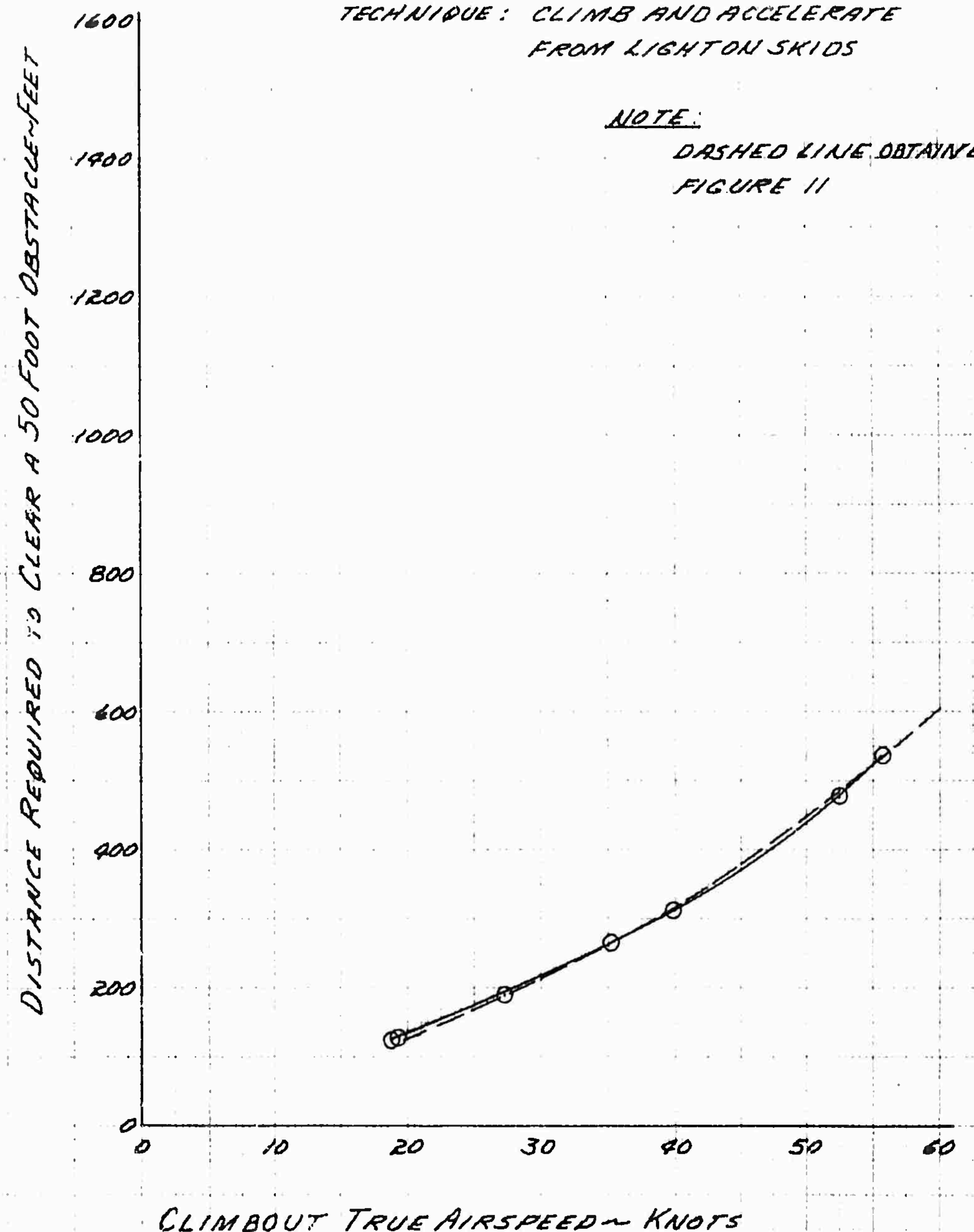

FIGURE NO. 22
EFFECT OF PEDAL POSITION ON POWER REQUIRED DURING LEVEL TAKE-OFF ACCELERATIONS WITH FIXED COLLECTIVE
YOH-6A USA S/N 62-4212
CLEAN CONFIGURATION

ROTOR SPEED = 469 RPM
FREE AIR TEMP. = 26 °C
WIND VELOCITY ≤ 2 KTS
GROSS WEIGHT = 2100 LB. (APPROX)
PRESSURE ALTITUDE = 4100 FT.
SKID HEIGHT = 2 FT.

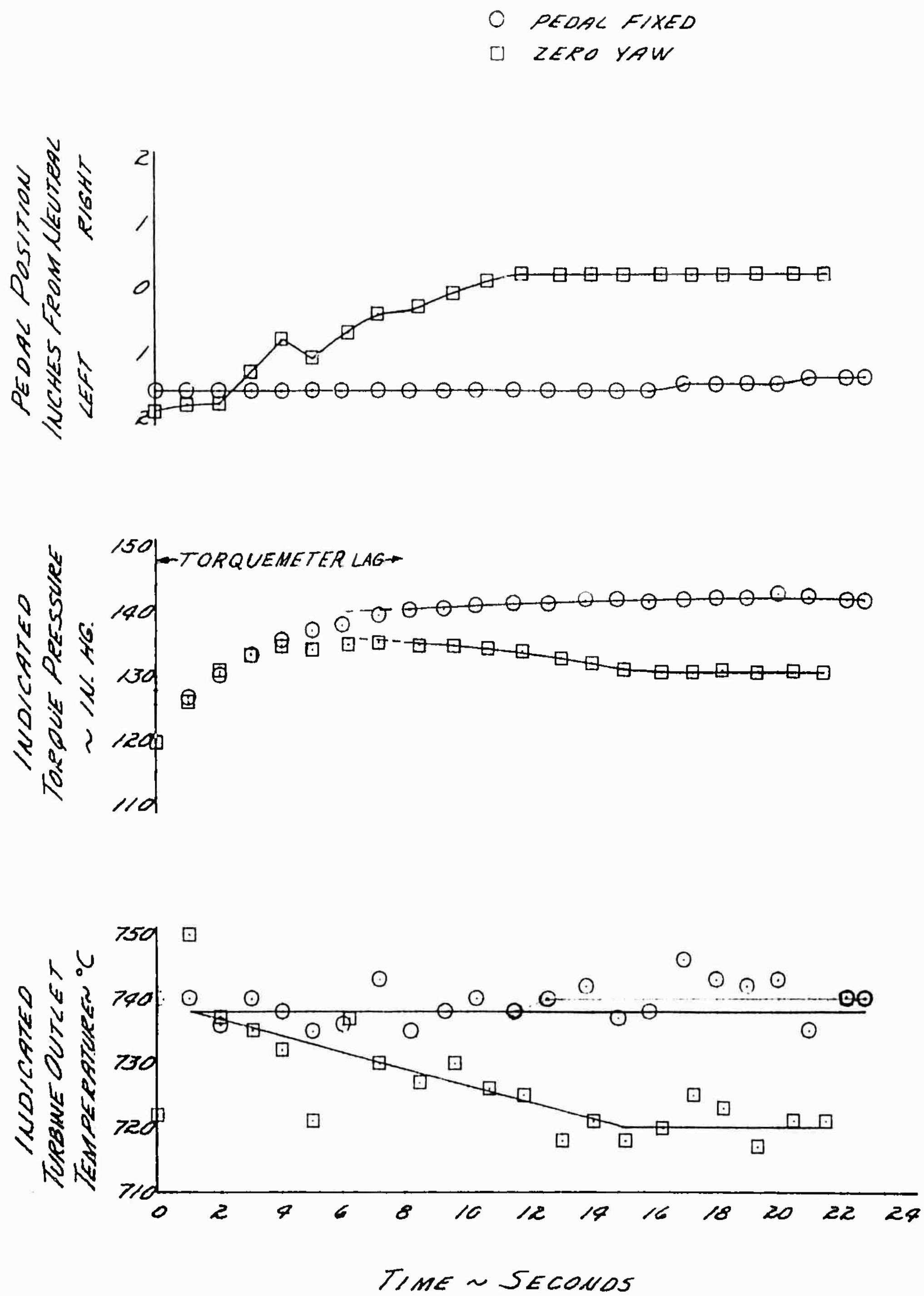

FIGURE No. 23
HOVERING PERFORMANCE
YOH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 2 FT.

CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS

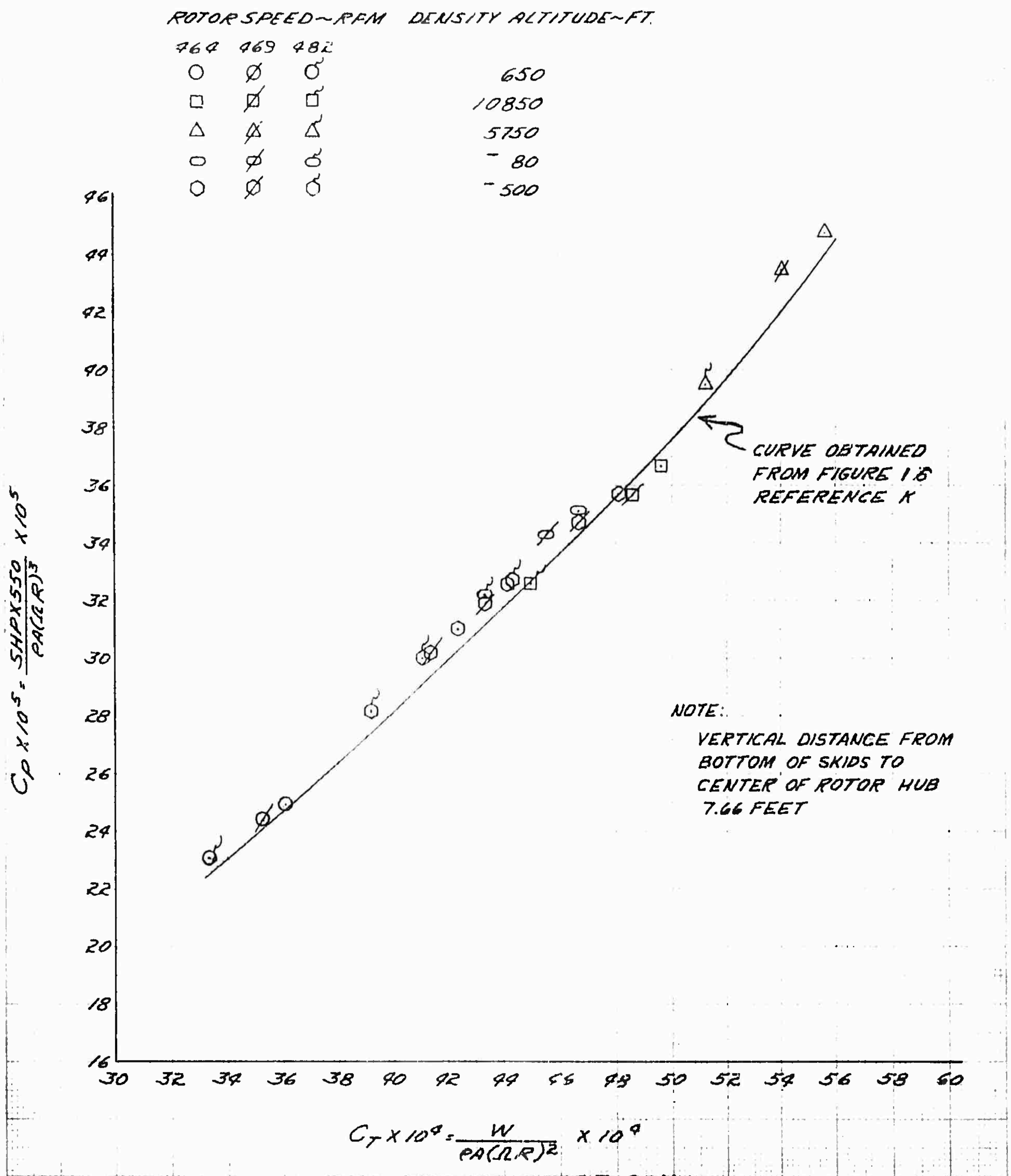

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FIGURE No. 24

HOVERING PERFORMANCE

YOH-6A USA S/N 62-4212

FREE FLIGHT METHOD

SKID HEIGHT = 5 FT

CLEAN CONFIGURATION

WIND LESS THAN 2 KNOTS

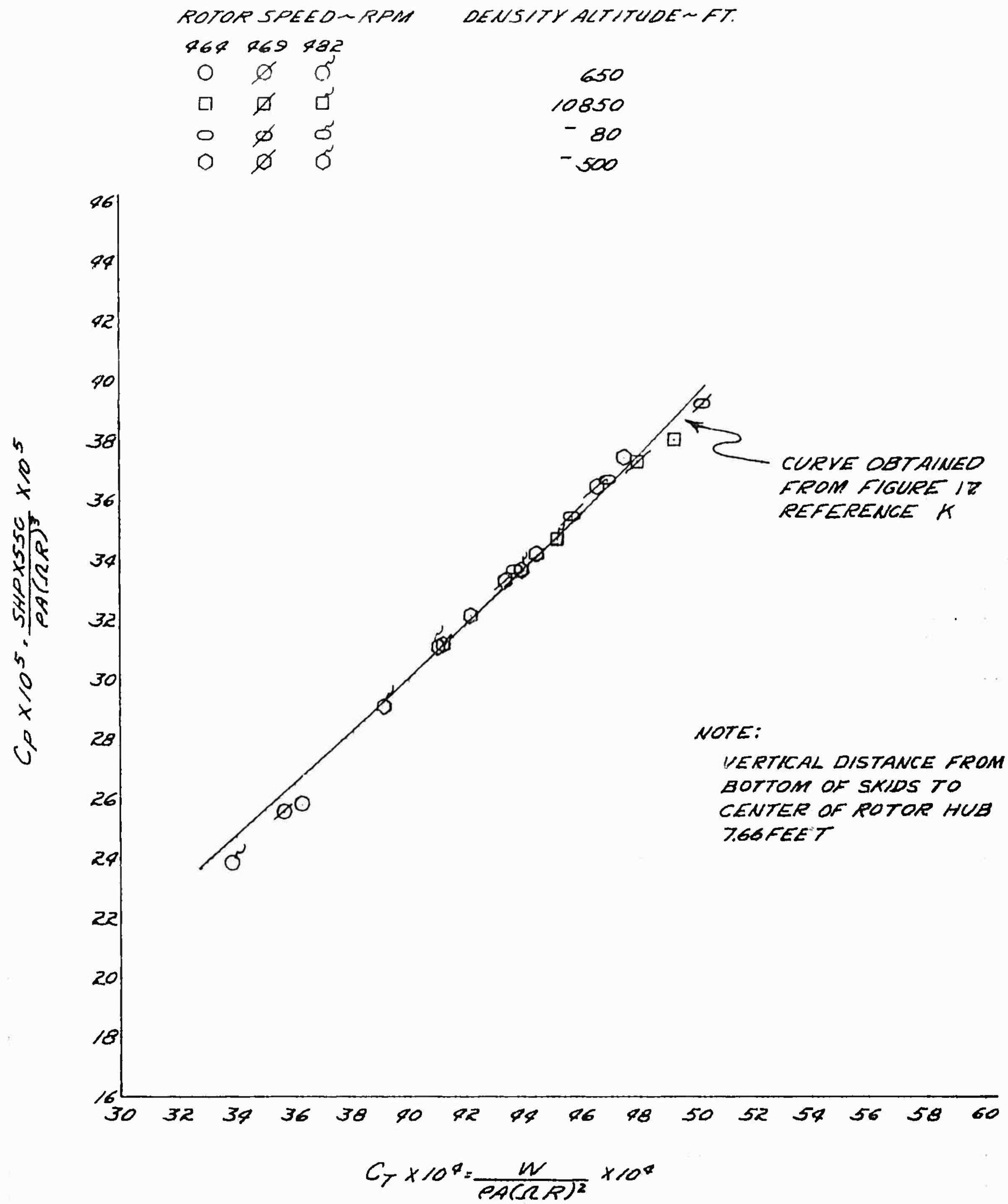

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FIGURE NO. 25
HOVERING PERFORMANCE
YOH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 10 FT.

CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS

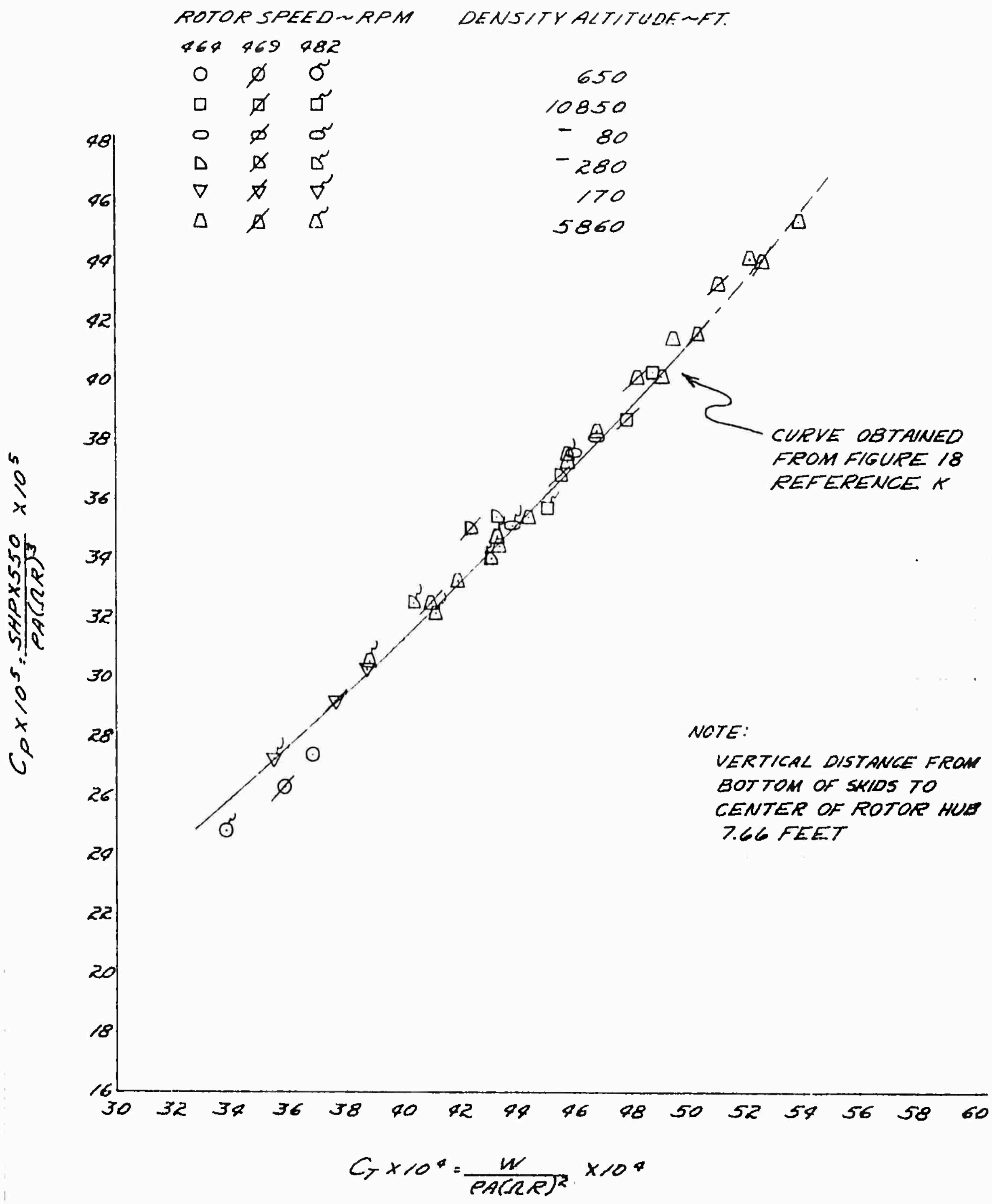

FIGURE No. 26
<u>HOVERING PERFORMANCE</u>
YOH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 15 FT.

CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS

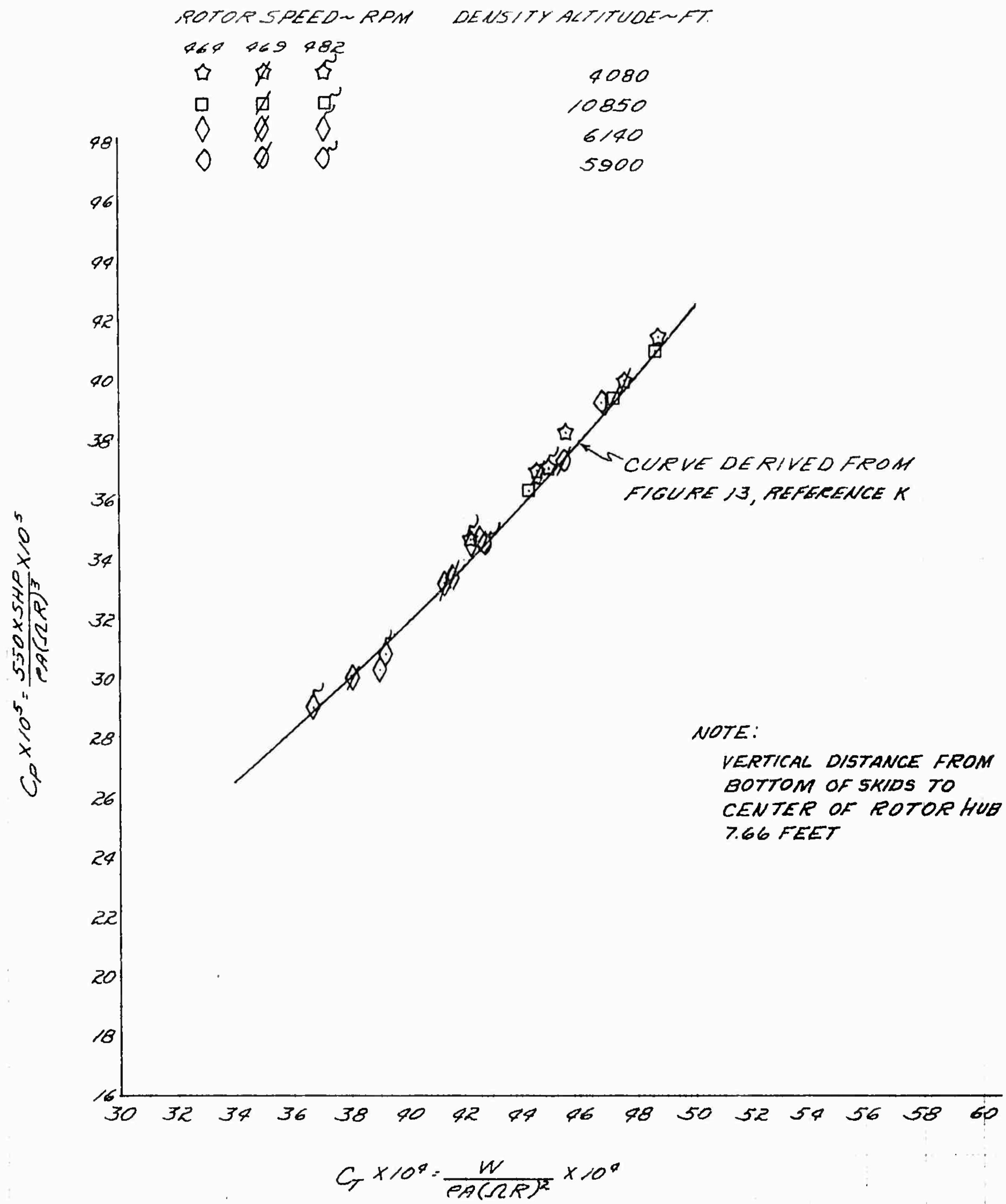

FIGURE NO. 27
HOVERING PERFORMANCE
YOH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 50 FT.

CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS

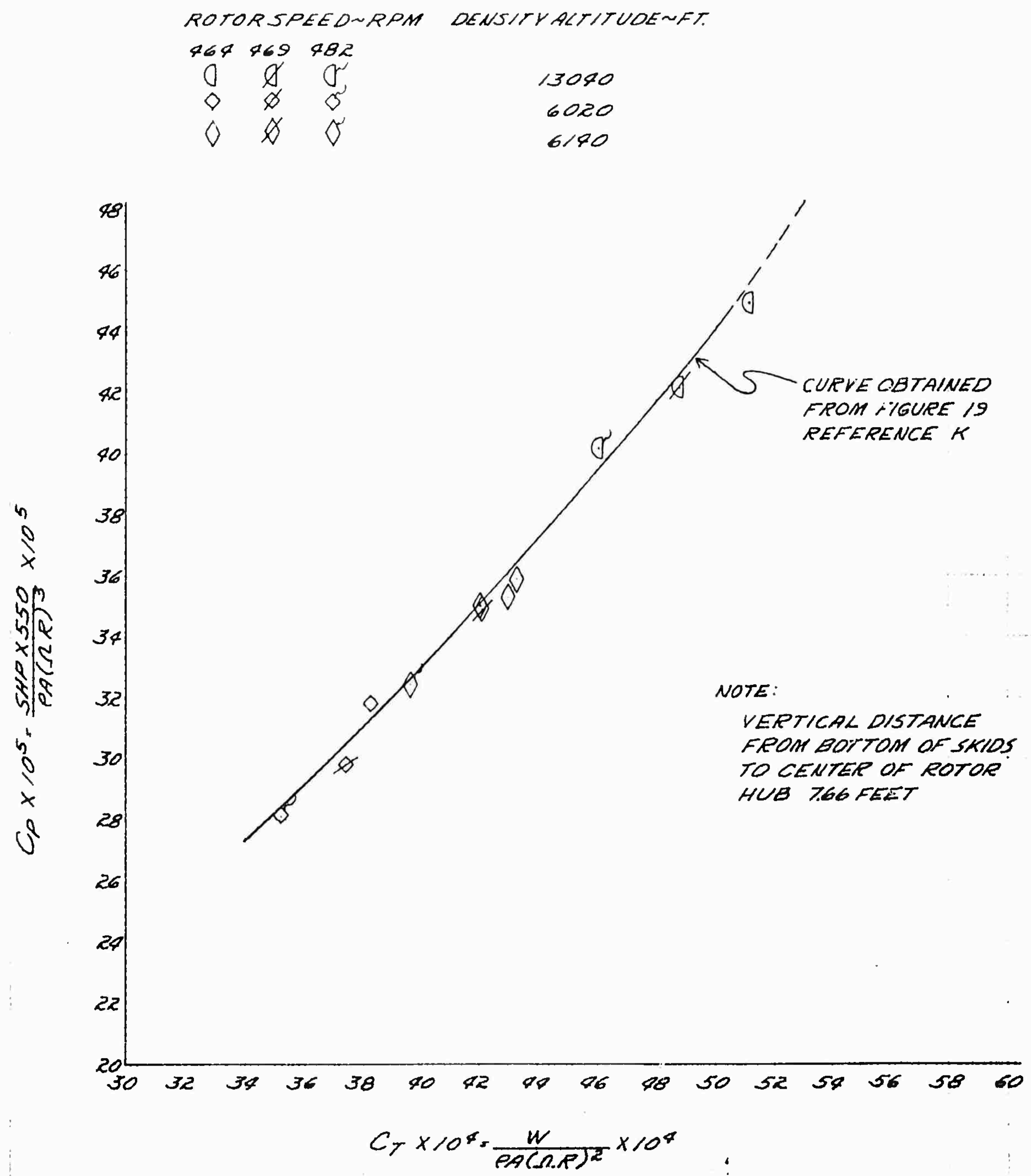

FIGURE No. 28
CLIMB PERFORMANCE
YOH-6A USA S/N 62-4212

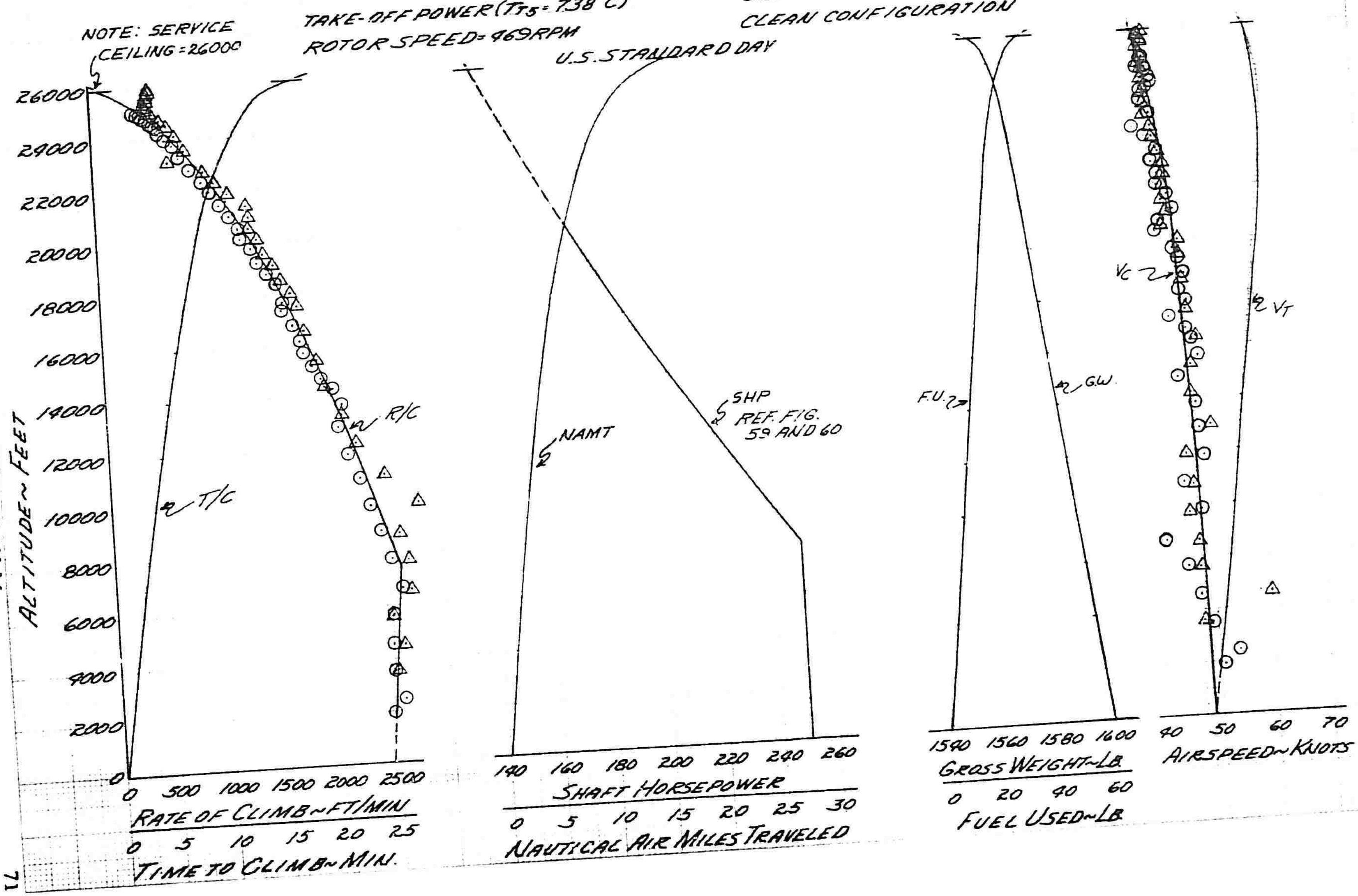

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FIGURE NO. 29
CLIMB PERFORMANCE
YOH-6A USA S/N 62-4212

TAKE-OFF POWER (T_{t5} = 738°C)
ROTOR SPEED = 469 RPM
C.G. = STATION 100.5 (MID)
CLEAN CONFIGURATION
U.S. STANDARD DAY

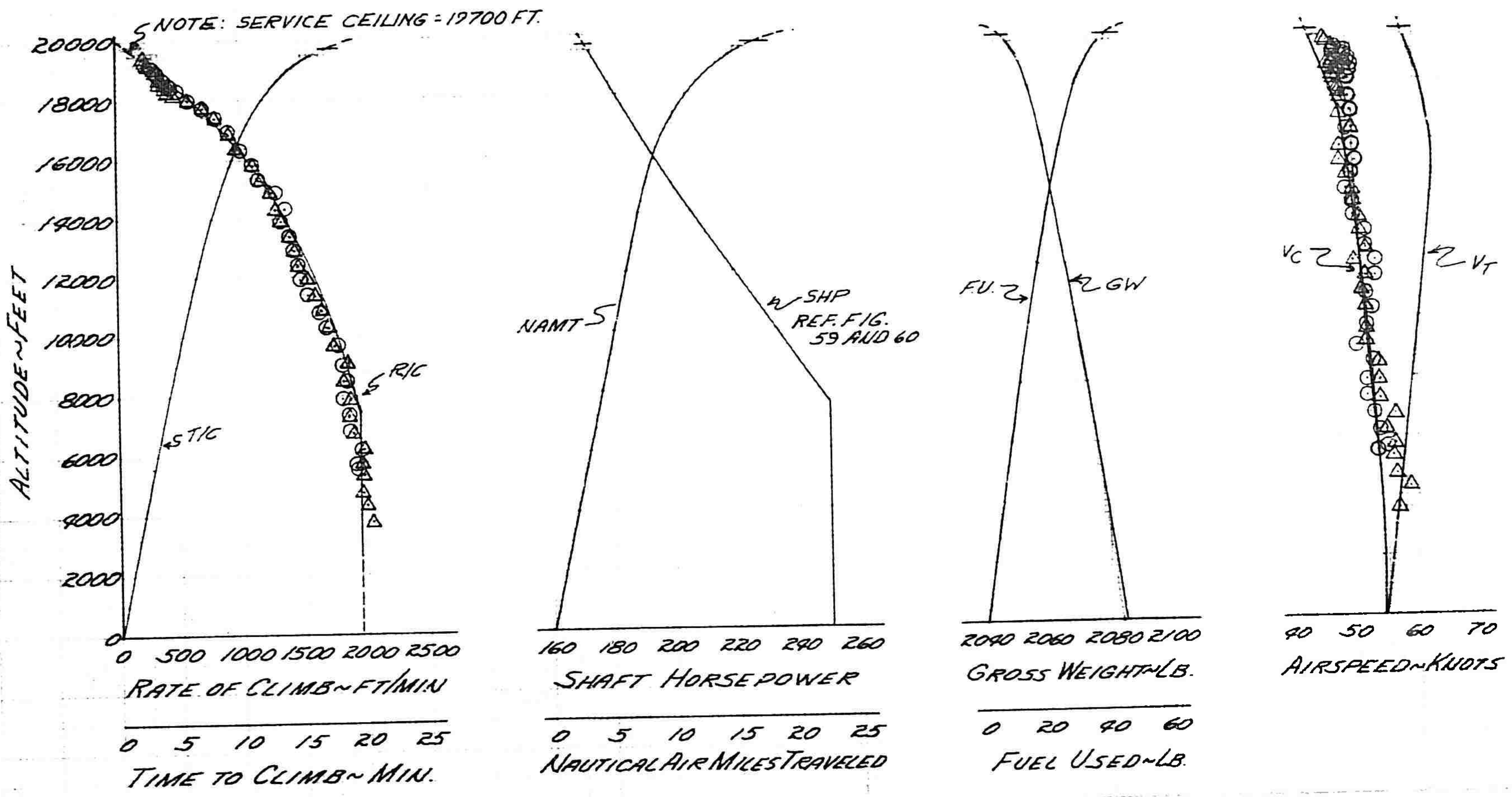

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FIGURE No. 30
CLIMB PERFORMANCE
YOH-6A USA S/N 62-4212

TAKE-OFF POWER (T_{T5} = 738°C) C.G. = STATION 100.5 (MID)
ROTOR SPEED = 469 RPM CLEAN CONFIGURATION
U.S. STANDARD DAY

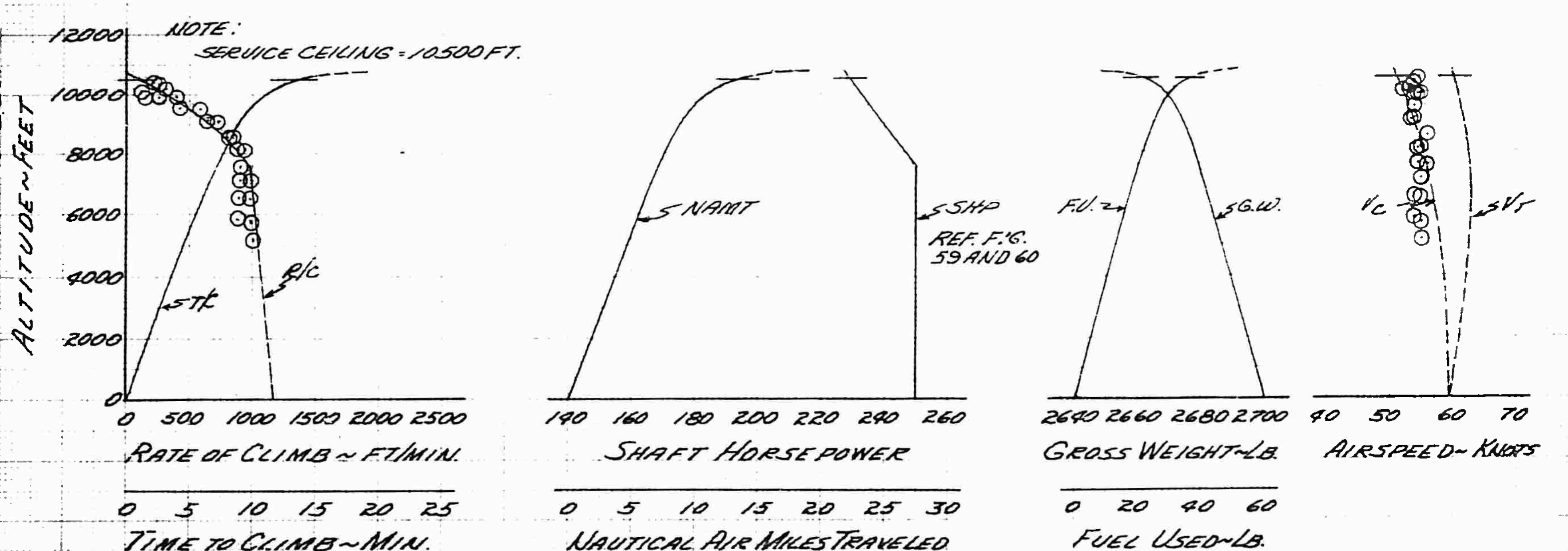

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FIGURE NO. 31
RANGE SUMMARY

YOH-6A USA S/N 62-4212
ROTOR SPEED = 469 RPM
CLEAN CONFIGURATION
U.S. STANDARD DAY

——— MAX TAS FOR .99 MAX SPECIFIC RANGE
— — — V_{NE} (REFERENCE FIGURE 48)
—·—·— EXTRAPOLATED ABOVE ALTITUDE FOR WHICH LEVEL FLIGHT DATA WAS OBTAINED

NOTE:
RANGE PERFORMANCE CALCULATED FROM FIGURES 32 THROUGH 34 AND 67.

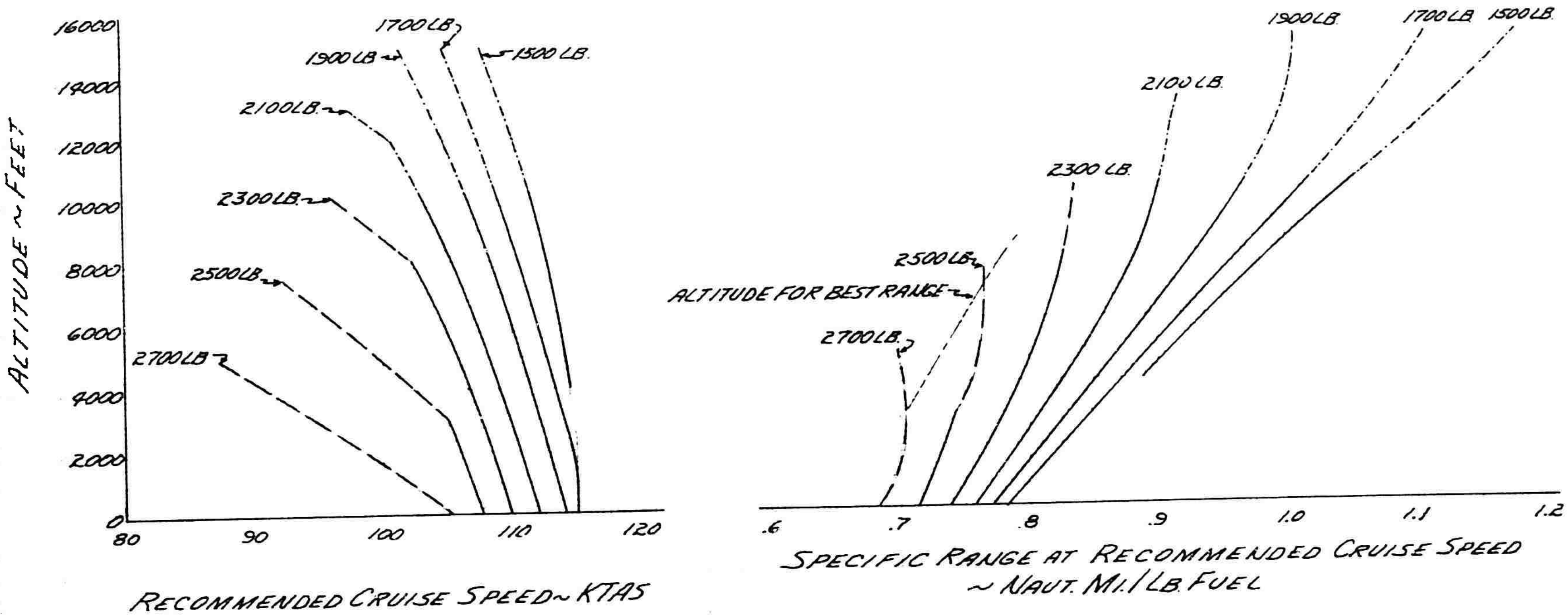

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FIGURE No. 32
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
C. G. STATION = 100.5 (MID)
CLEAN CONFIGURATION

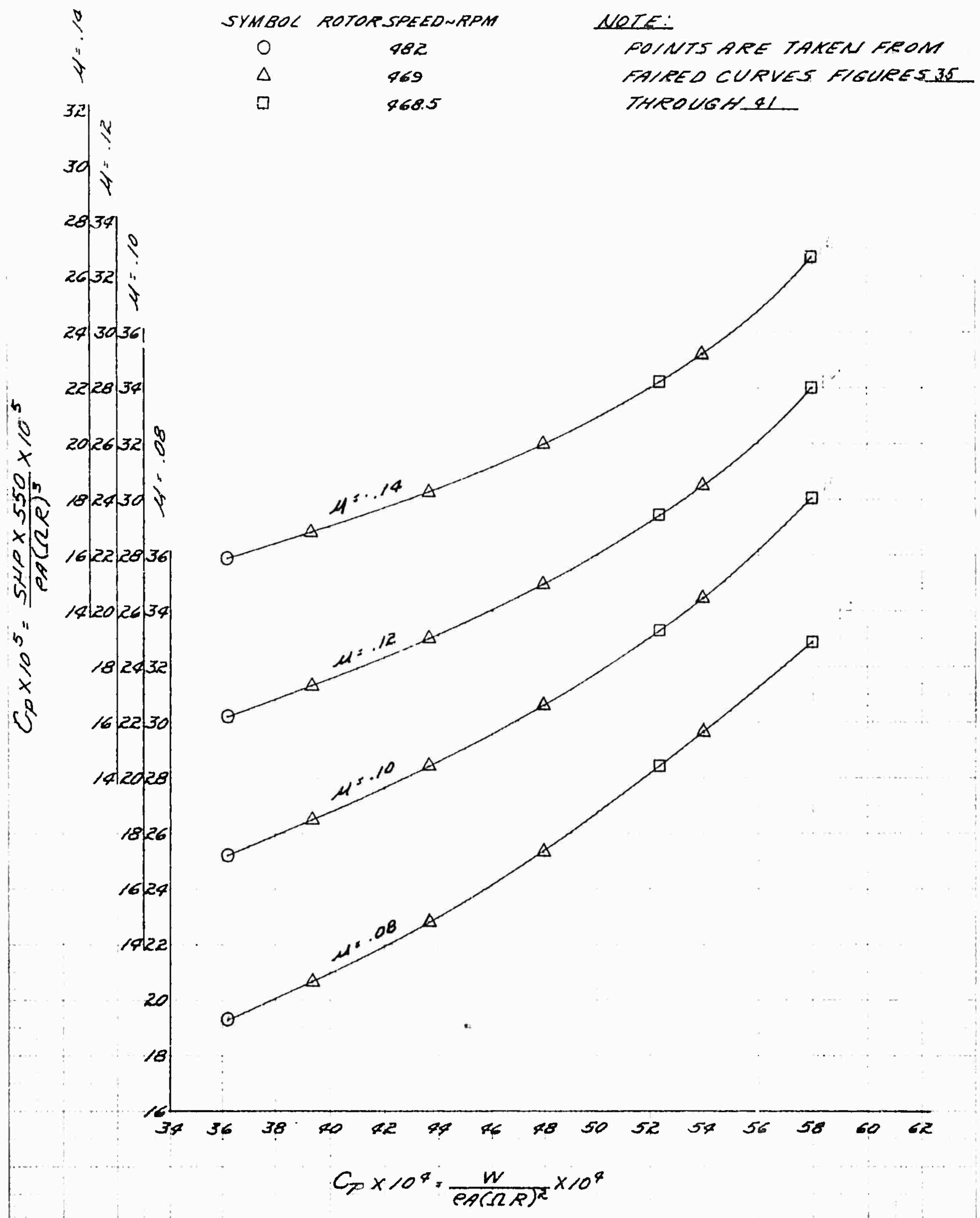

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FIGURE NO. 33

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

C.G. STATION :100.5 (MID)

CLEAN CONFIGURATION

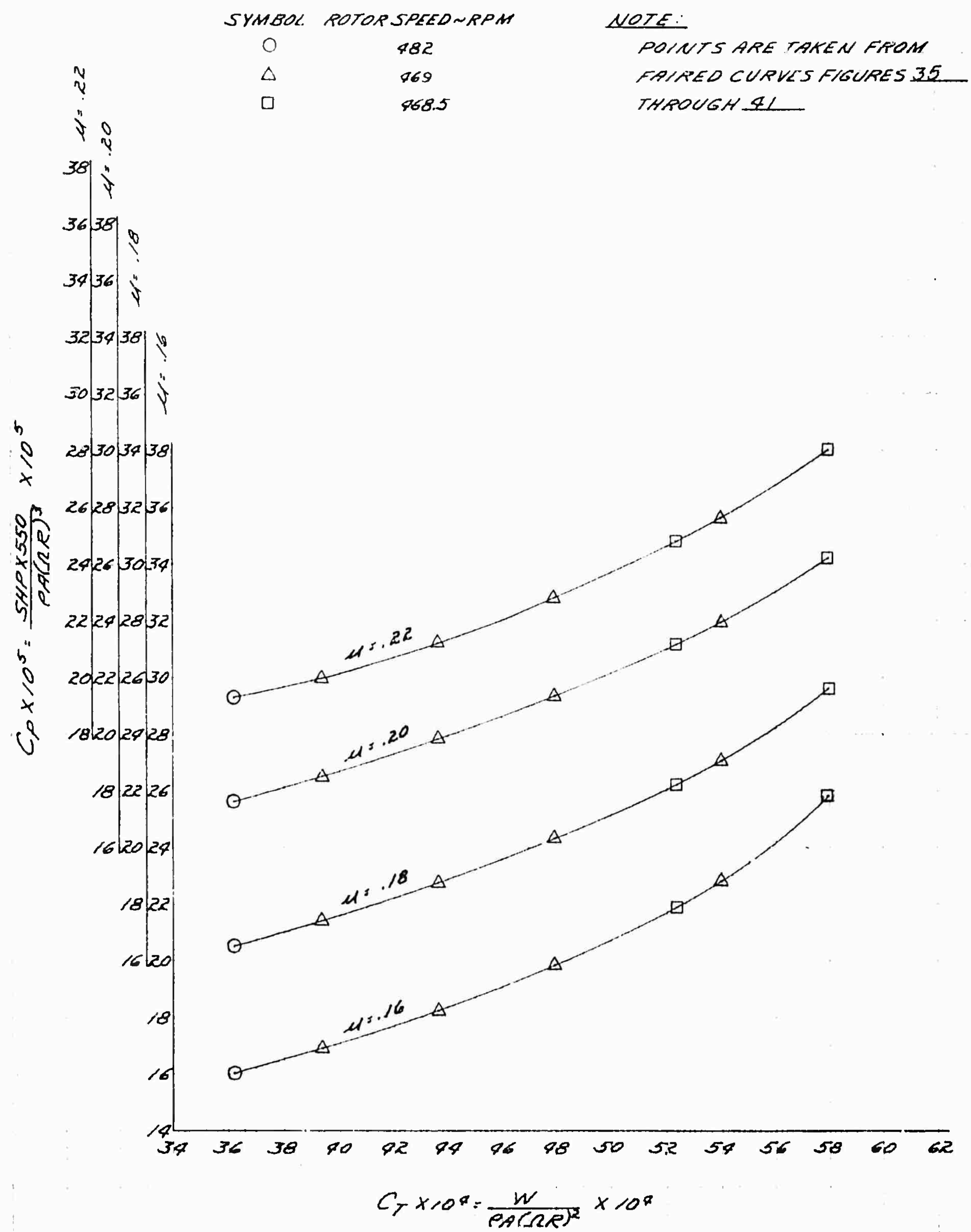

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FIGURE No. 34
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
C.G. STATION = 100.5 (MID)
CLEAN CONFIGURATION

SYMBOL	ROTOR SPEED~RPM
○	482
△	469
□	468.5

NOTE:
POINTS ARE TAKEN FROM FAIRED CURVES FIGURES 35 THROUGH 41

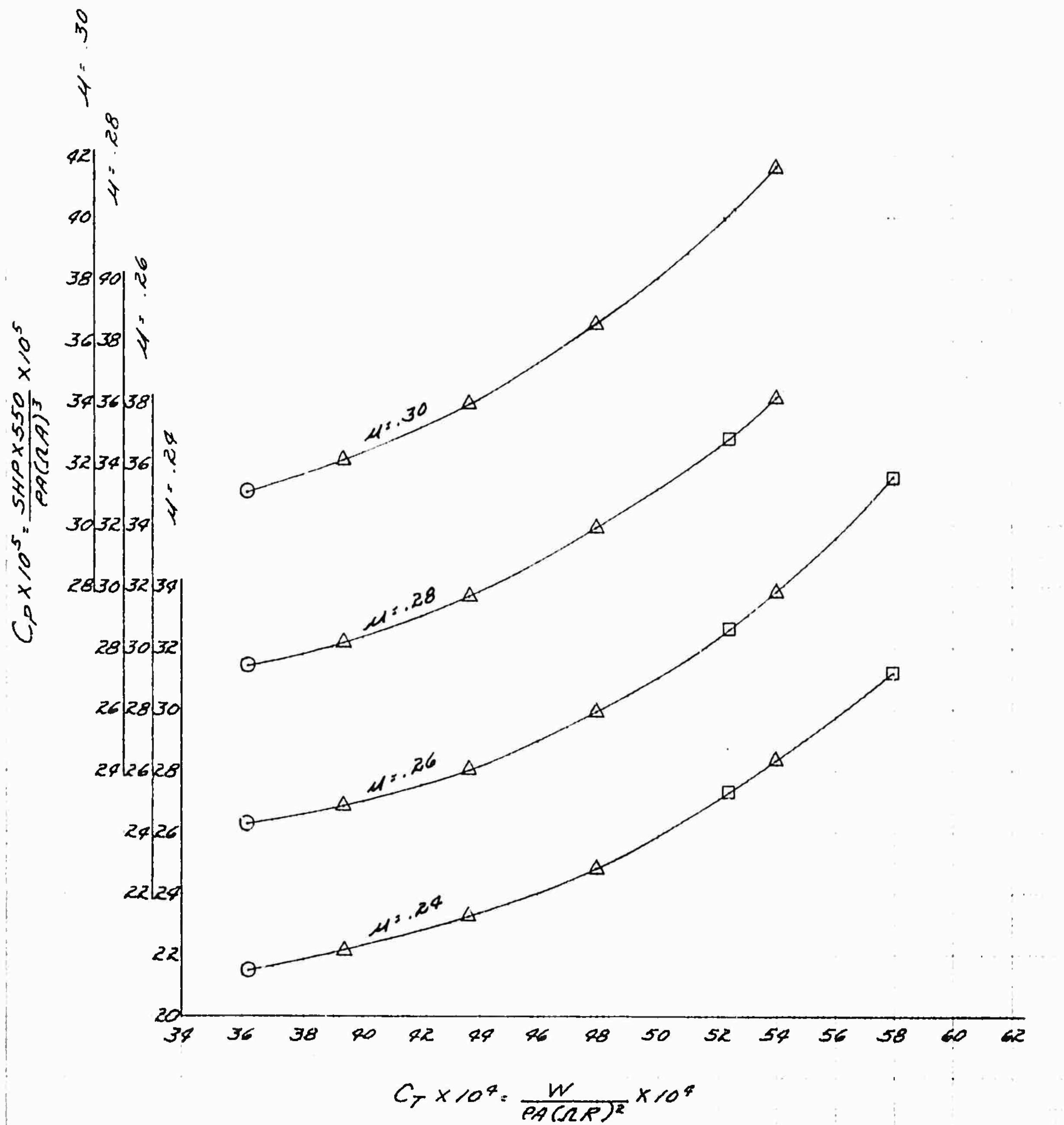

FIGURE NO. 35
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 1745 LB.
DENSITY ALTITUDE = 5630 FT.
ROTOR SPEED = 482 RPM
C.G. STATION = 100.5 (MID)
C_T = .00362
CLEAN CONFIGURATION
ZERO SIDESLIP

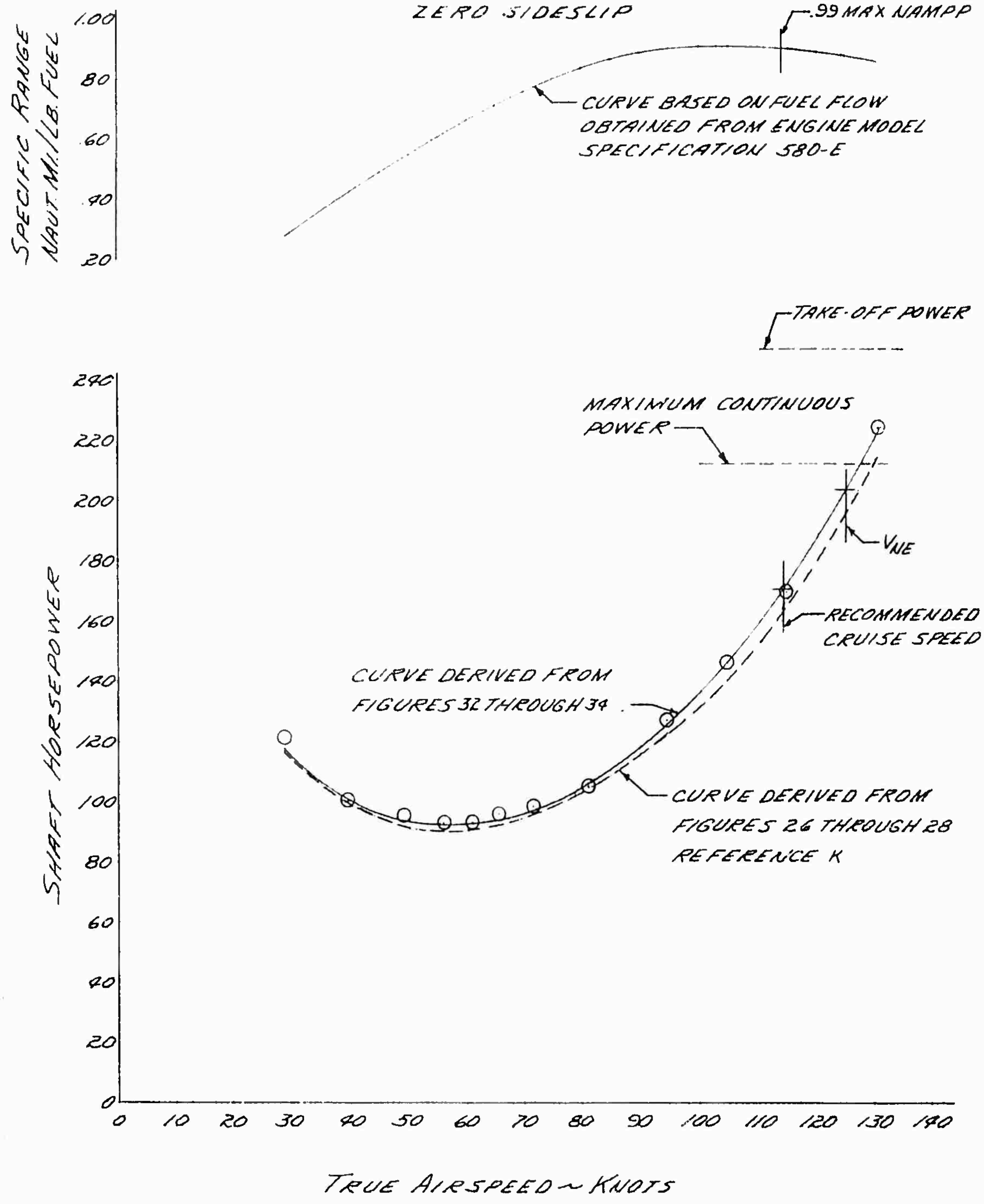

FIGURE NO. 36

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

GROSS WEIGHT = 1975 LB.

DENSITY ALTITUDE = 2530 FT.

ROTOR SPEED = 469 RPM

C. G. STATION = 100.5 (MID)

C_T = .003935

CLEAN CONFIGURATION

ZERO SIDESLIP

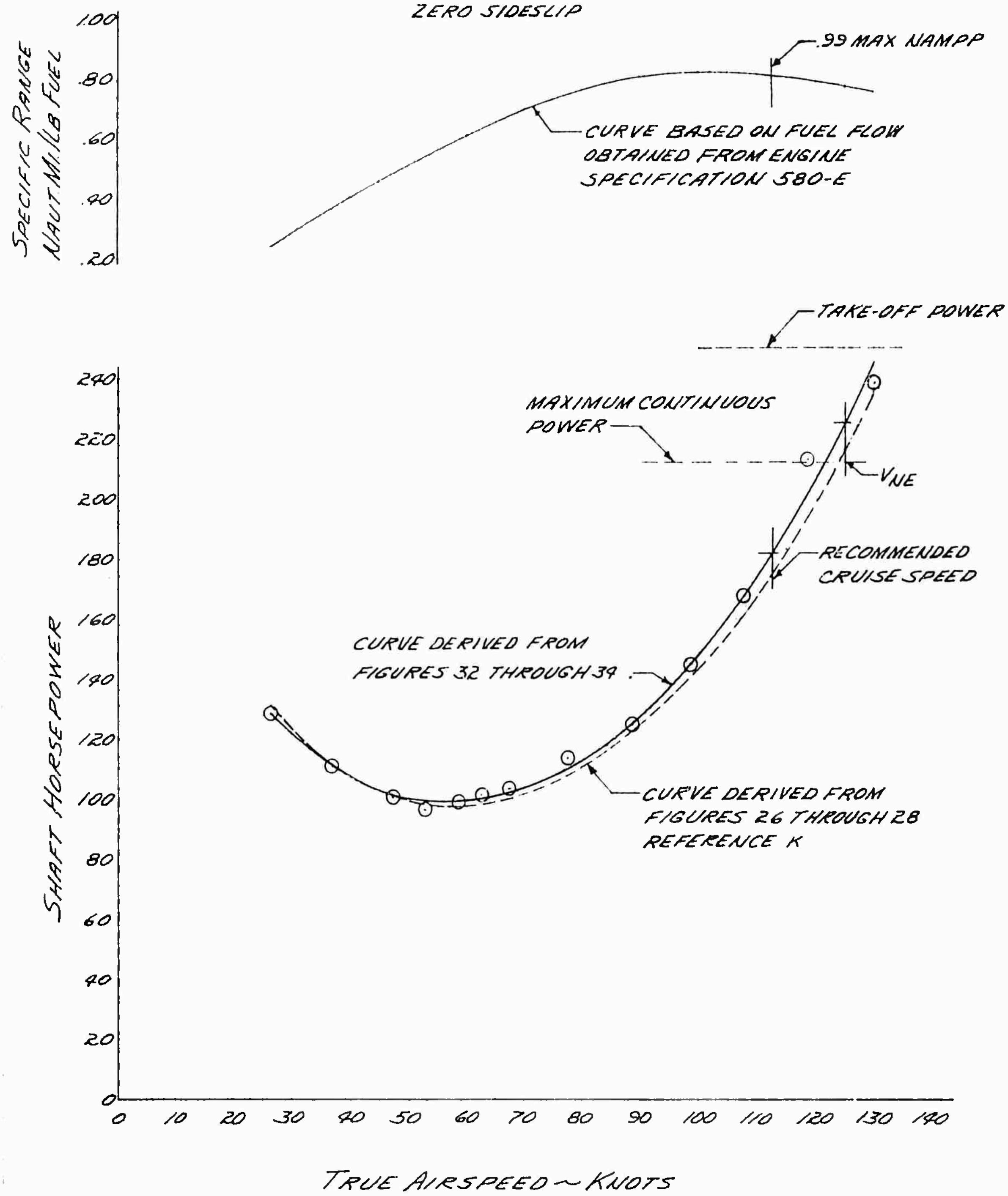

FIGURE No. 37

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

GROSS WEIGHT = 2225 LB.

DENSITY ALTITUDE = 1960 FT.

ROTOR SPEED = 469 RPM

C.G. STATION = 100.5 (MID)

C_T = .004365

CLEAN CONFIGURATION

ZERO SIDESLIP

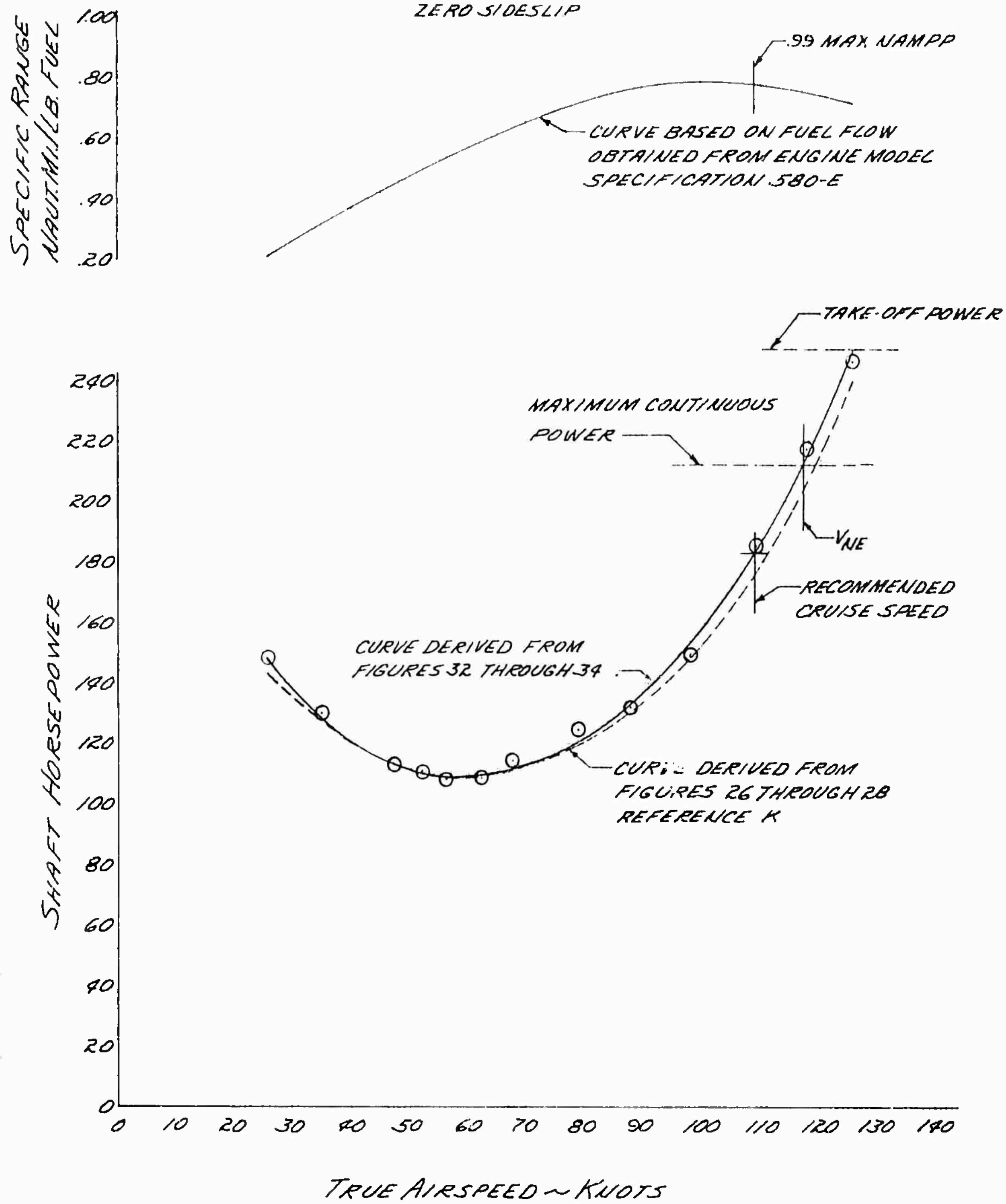

FIGURE No. 38
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 1895 LB.
DENSITY ALTITUDE = 10290 FT.
ROTOR SPEED = 469 RPM
C.G. STATION = 100.5 (MID)
C_T = .004794
CLEAN CONFIGURATION
ZERO SIDESLIP

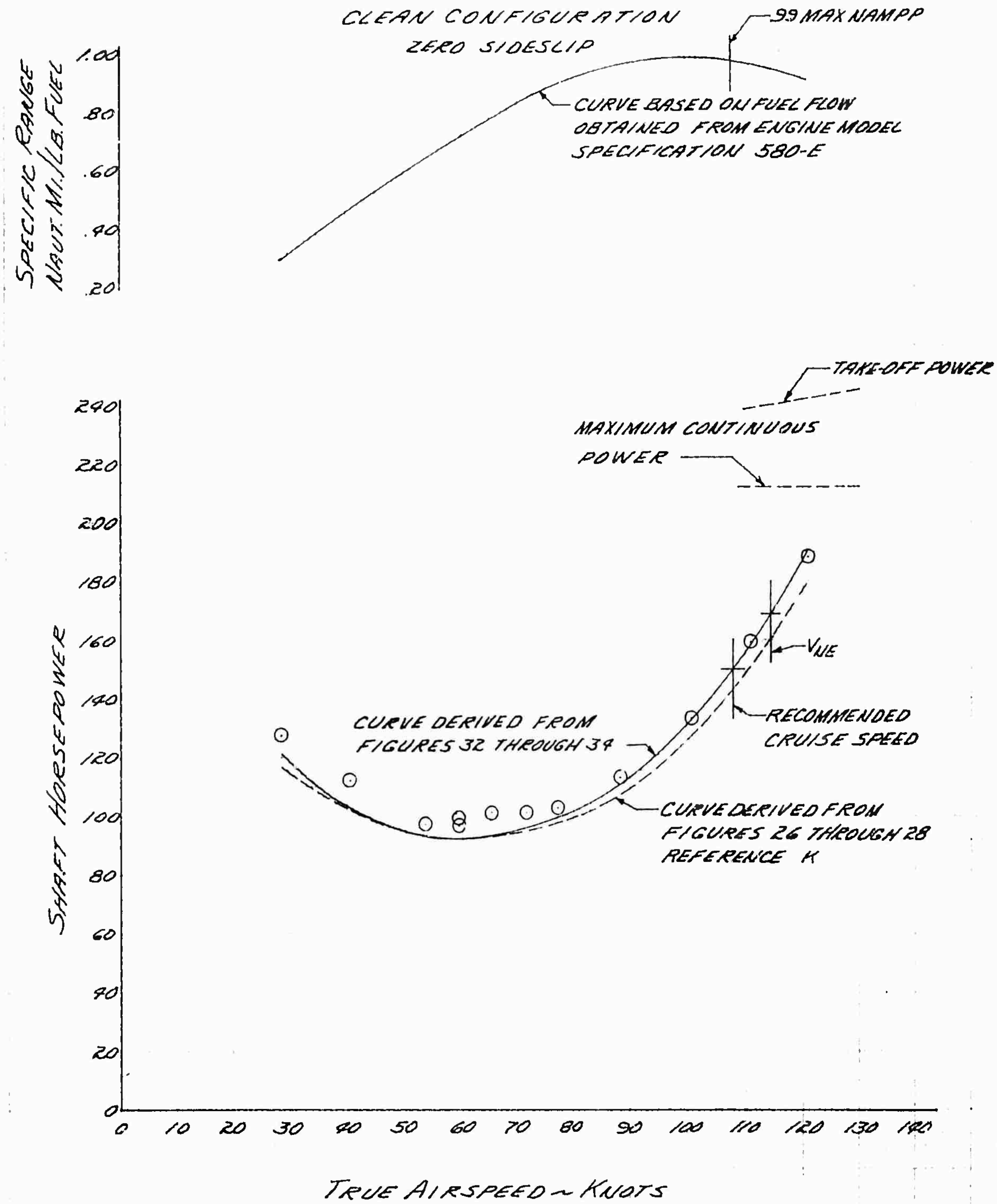

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FIGURE No 39
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 2040 LB.
DENSITY ALTITUDE = 10310 FT.
ROTOR SPEED: 468.5 RPM
C.G. STATION = 100.5 (MID)
C_T = .005235
CLEAN CONFIGURATION
ZERO SIDESLIP

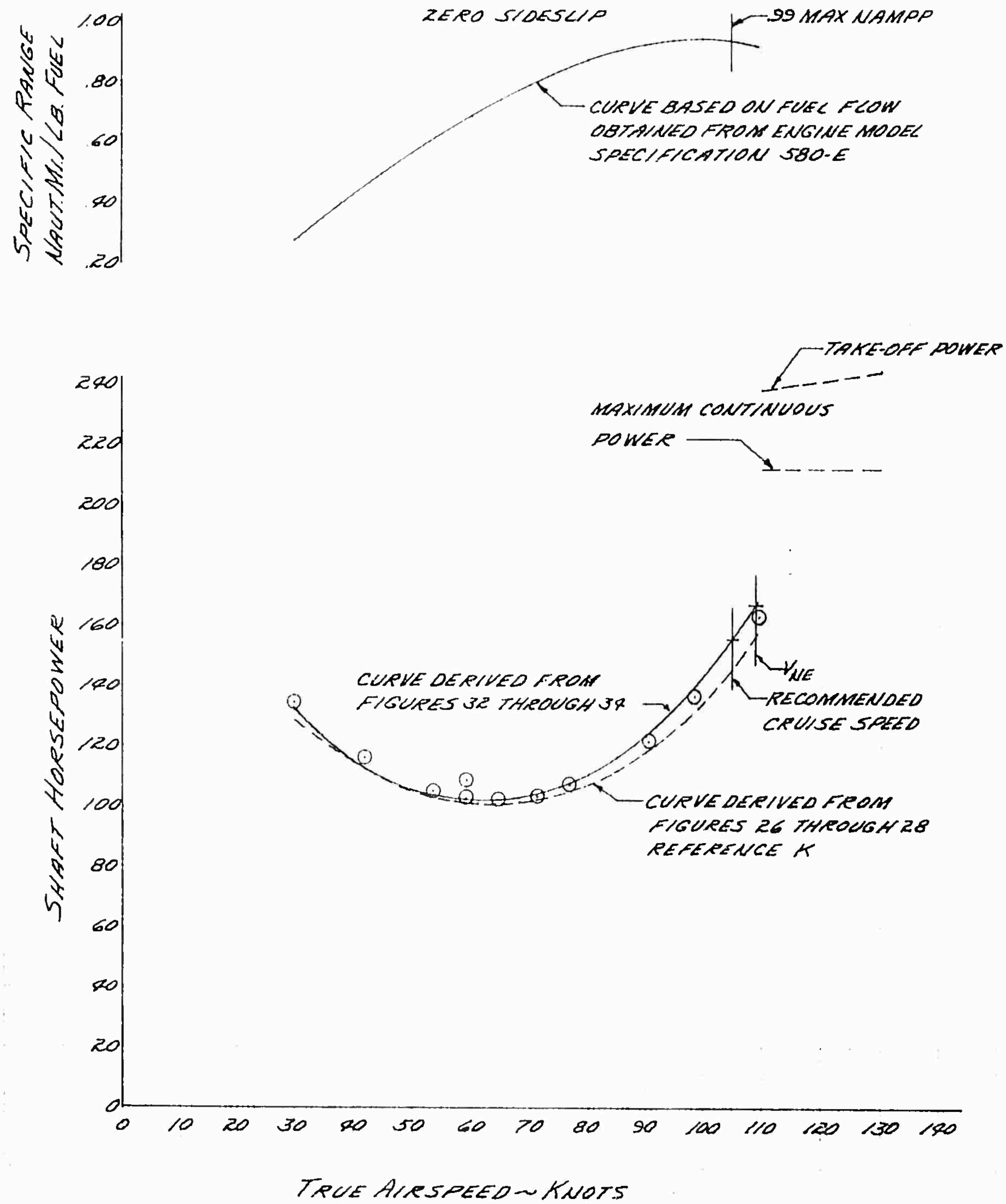

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FIGURE No. 40

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

GROSS WEIGHT = 2475 LB.

DENSITY ALTITUDE: 5450 FT.

ROTOR SPEED: 469 RPM

C. G. STATION = 100.5 (MID)

C_T = .005394

CLEAN CONFIGURATION

ZERO SIDESLIP

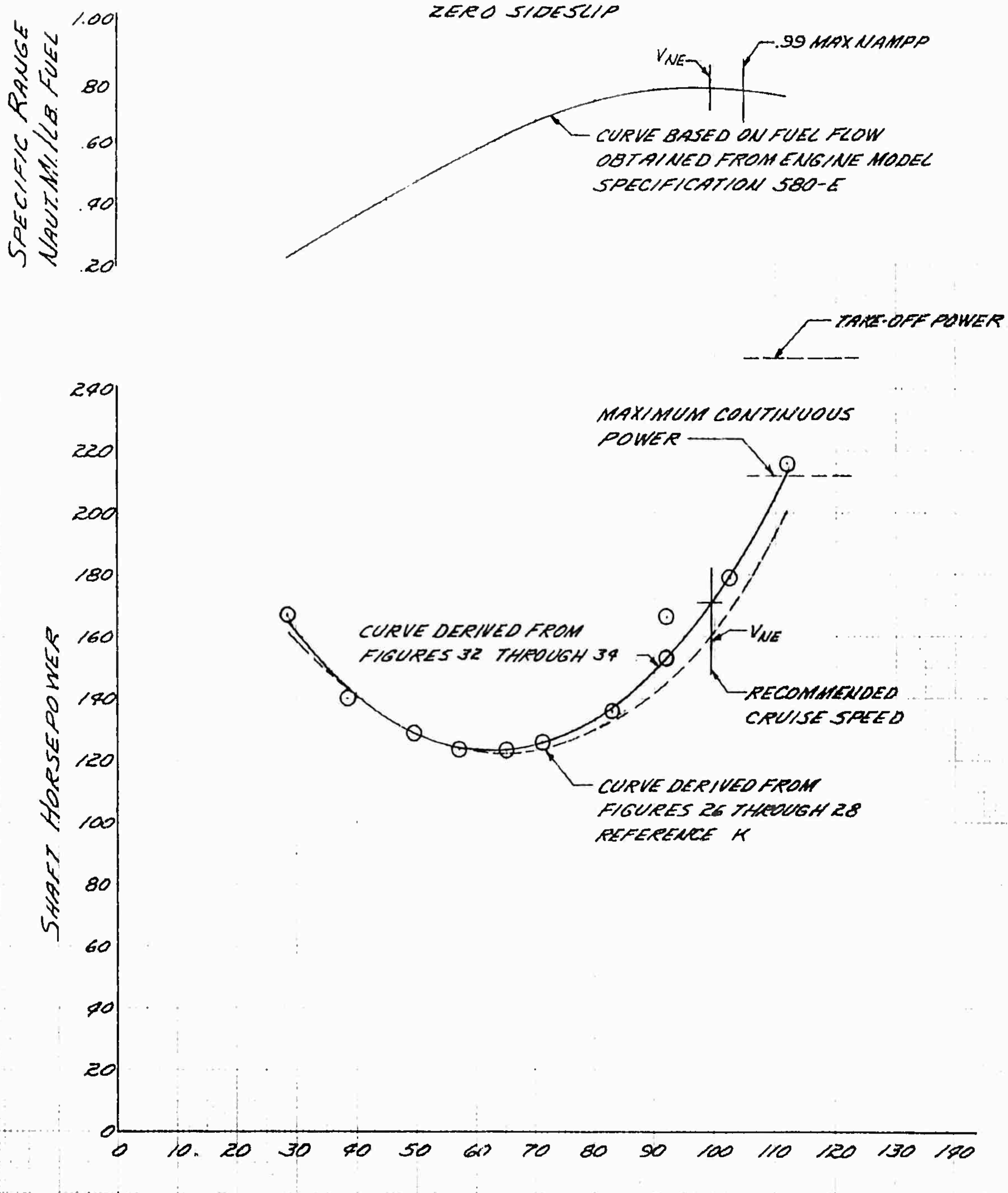

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FIGURE NO. 41
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 2285 LB.
DENSITY ALTITUDE = 10270 FT.
ROTOR SPEED = 468.5 RPM
C.G. STATION = 100.5 (MID)
C_T = .005794
CLEAN CONFIGURATION
ZERO SIDESLIP

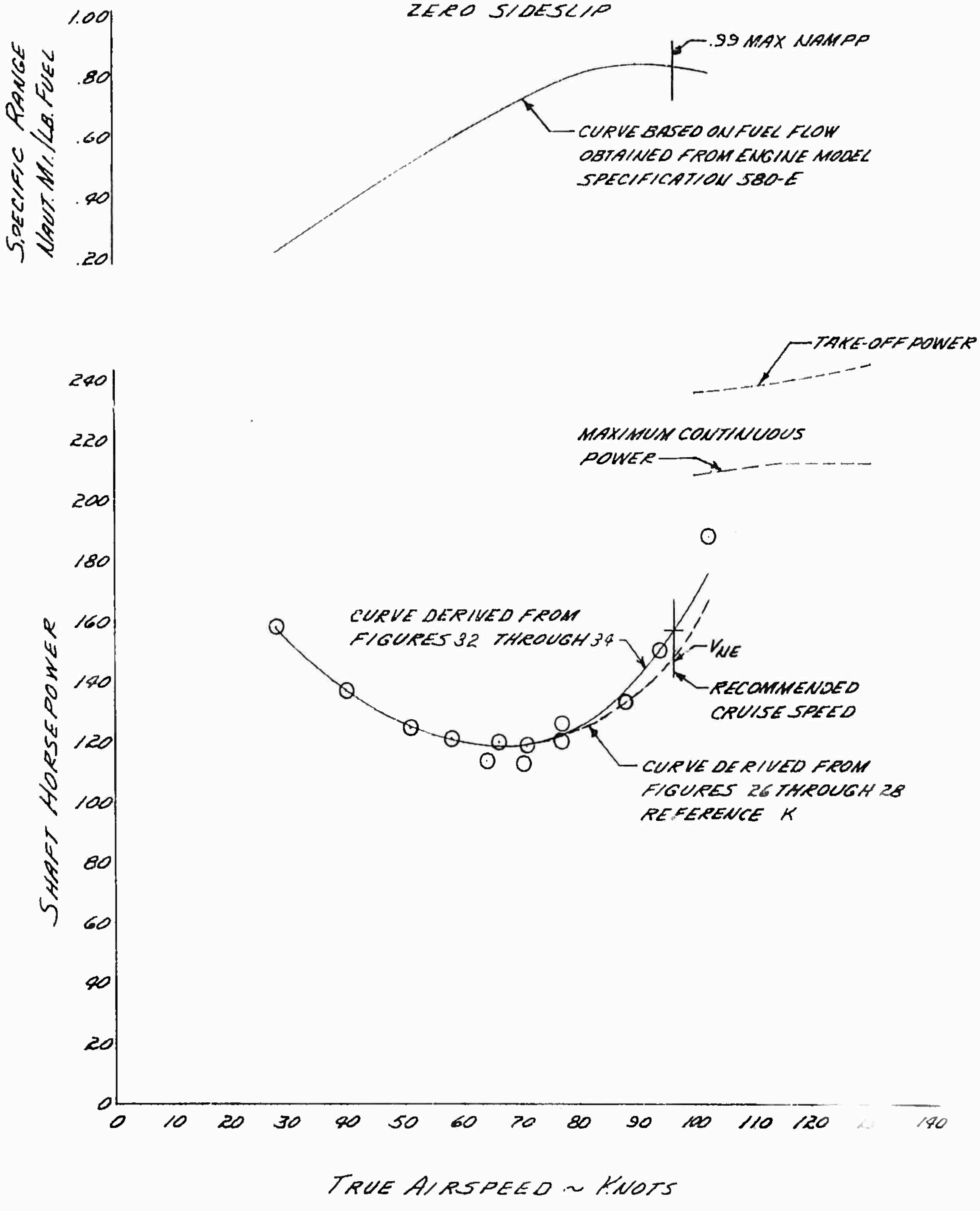

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FIGURE NO. 42
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 1835 LB.
DENSITY ALTITUDE = 7150 FT
ROTOR SPEED = 482 RPM
C.G. STATION = 100.7 (MID)
C_T = .003986
XM-7 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION

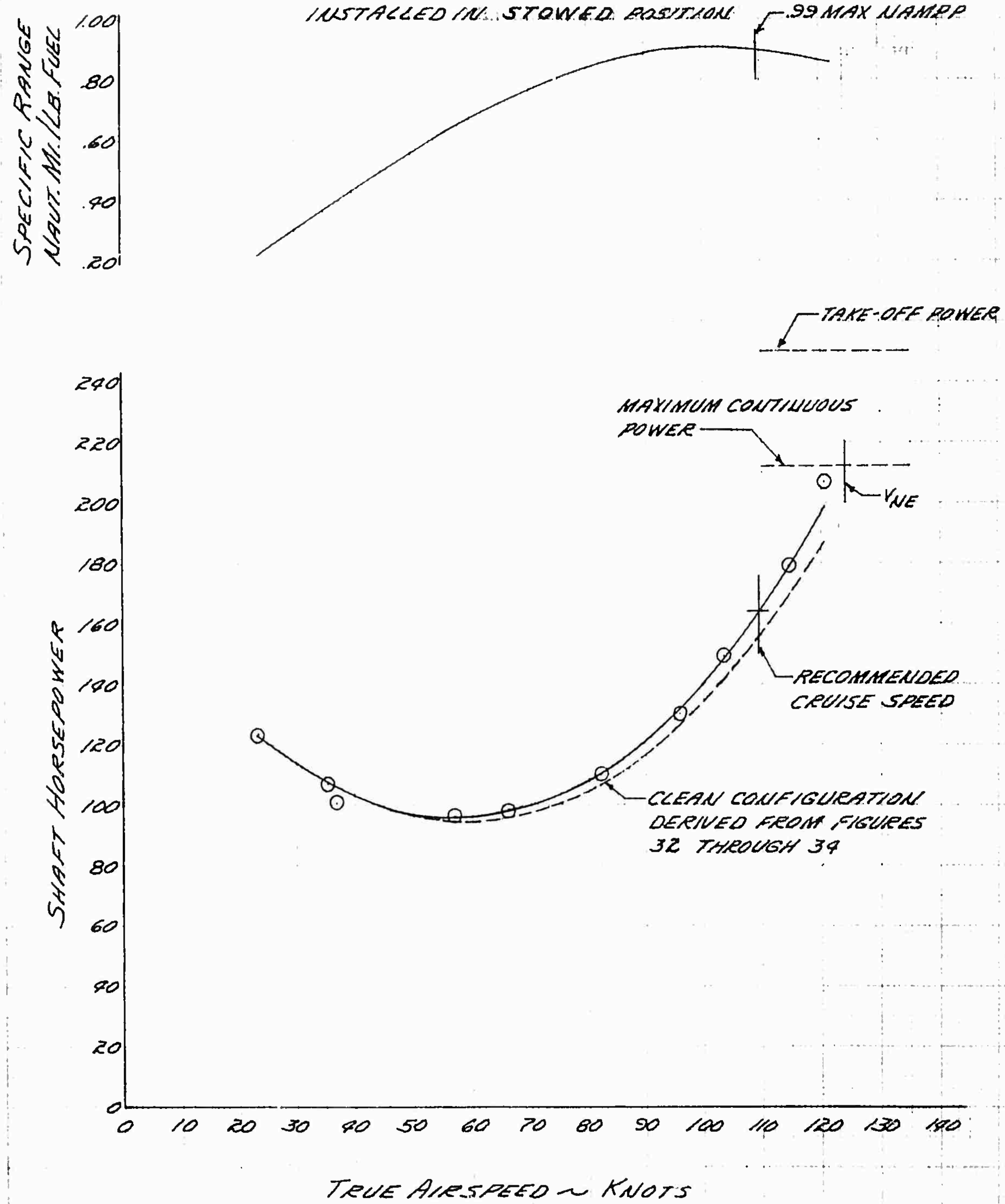

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FIGURE NO. 43

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

GROSS WEIGHT = 2020 LB.

DENSITY ALTITUDE = 8900 FT.

ROTOR SPEED = 469 RPM

C. G. STATION = 100.6 (MID)

C_T = .004886

XM-7 ARMAMENT SYSTEM

INSTALLED IN STOWED POSITION

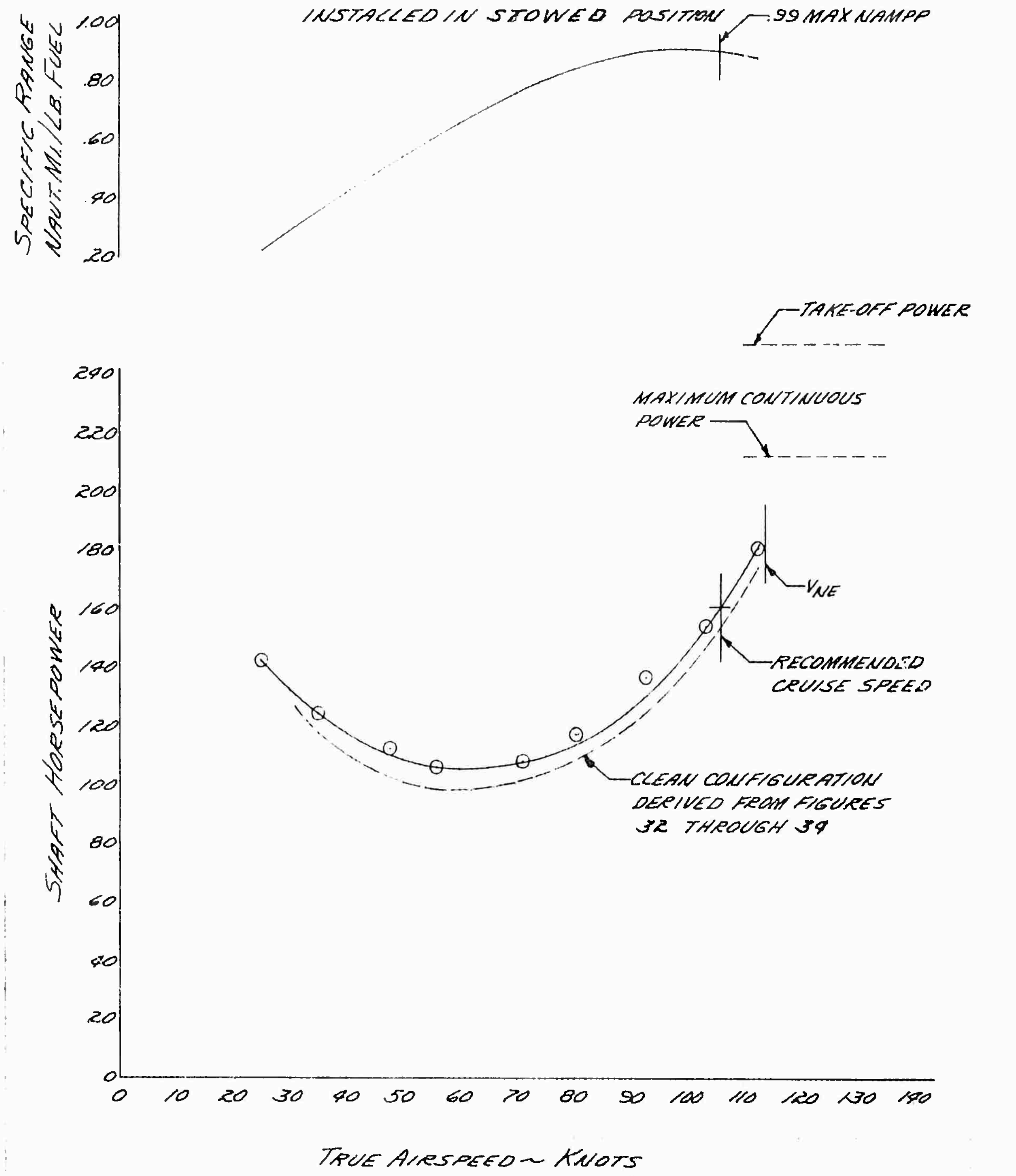

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FIGURE NO. 44
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT = 2085 LB.
DENSITY ALTITUDE = 10370 FT.
ROTOR SPEED = 469 RPM
C.G. STATION = 100.3 (MID)
C_T = .005285
XM-7 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION

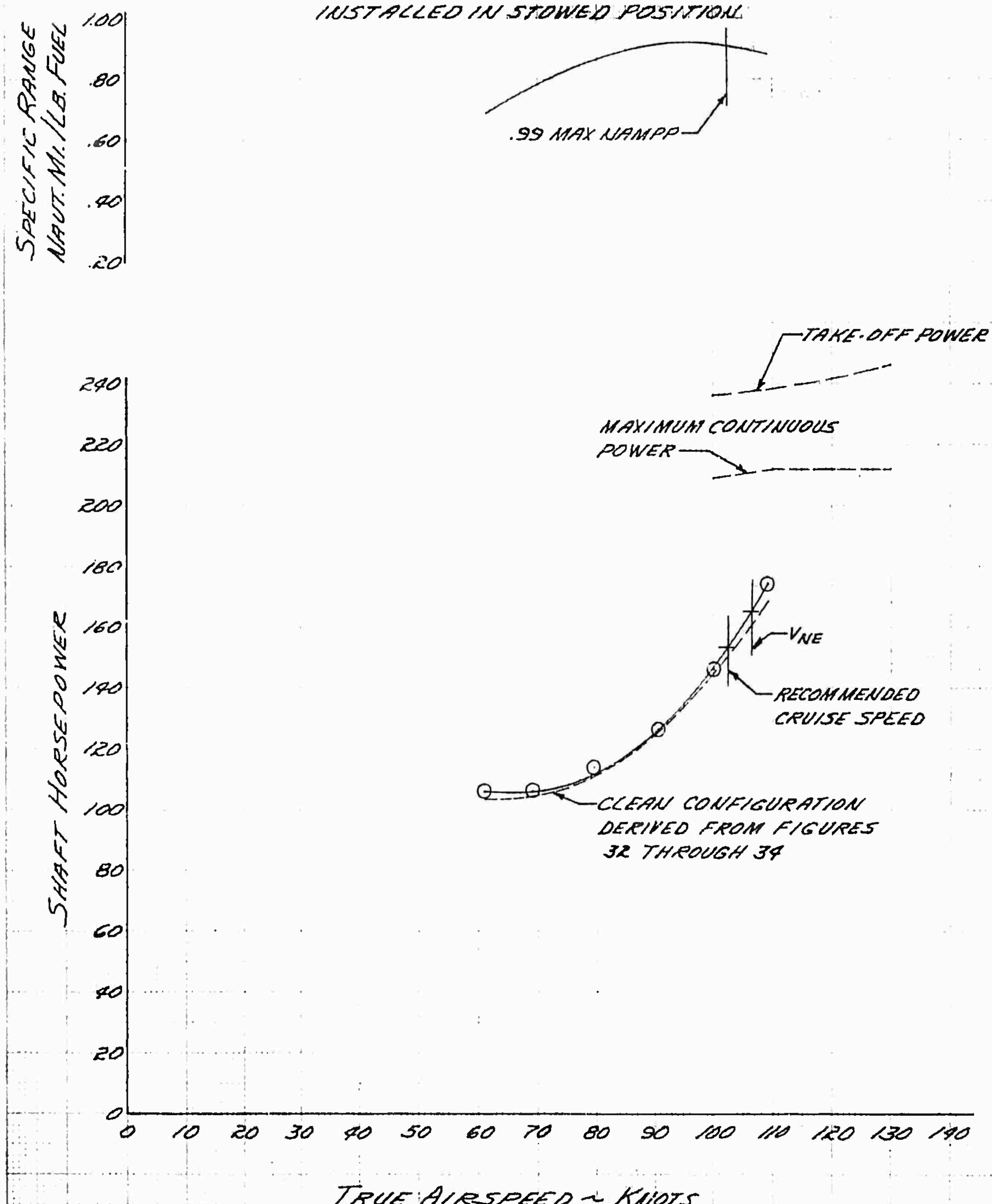

FIGURE No. 45

LEVEL FLIGHT PERFORMANCE

YOH-6A USA S/N 62-4212

GROSS WEIGHT = 2660 LB.

DENSITY ALTITUDE = 5450 FT.

ROTOR SPEED = 469 RPM

C.G. STATION = 100.5 (MID)

C_T = .005795

BOOM OFF CONFIGURATION

NEEDLE AND BALL CENTERED

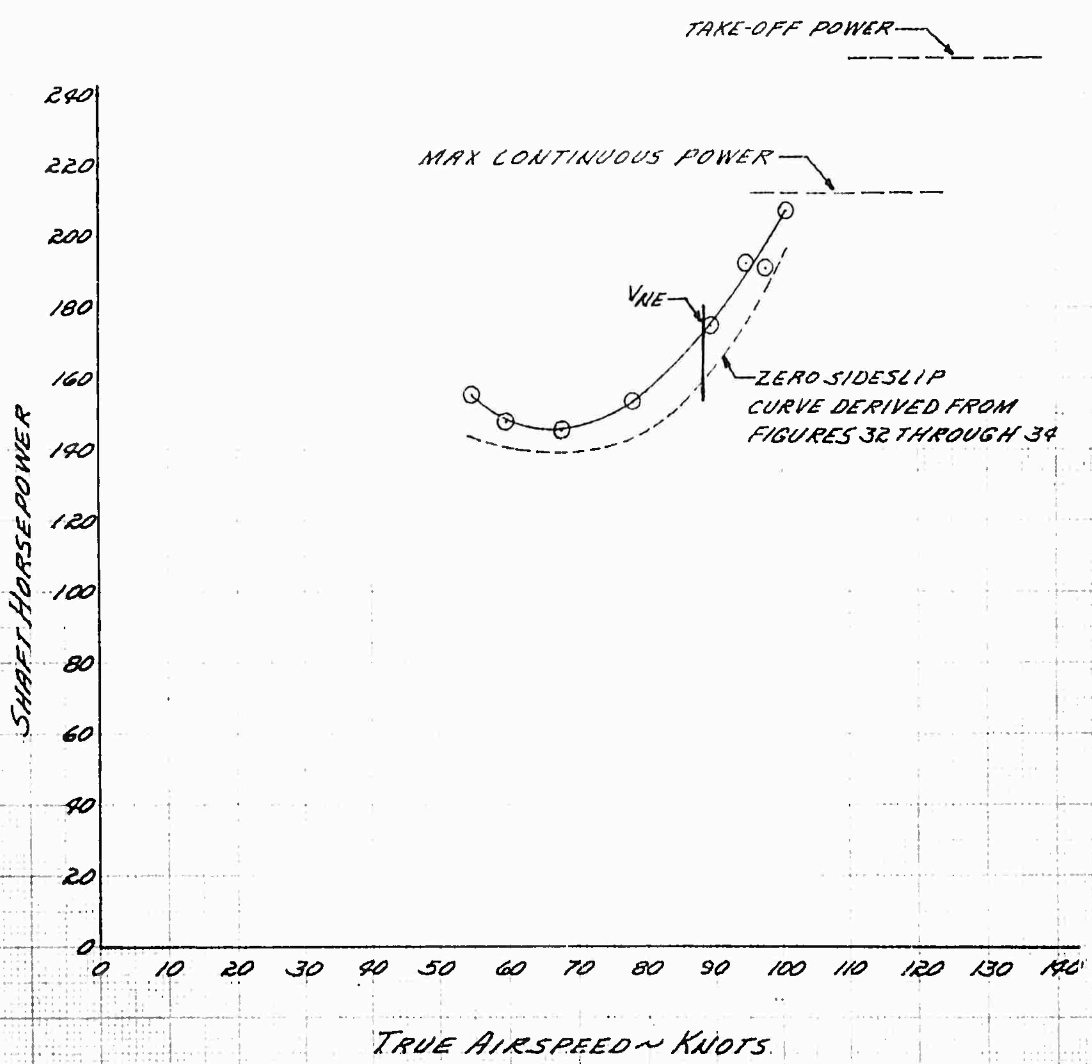

FIGURE NO. 46
LEVEL FLIGHT PERFORMANCE
YOH-6A USA S/N 62-4212
GROSS WEIGHT: 2080 LB.
DENSITY ALTITUDE = 5010 FT.
ROTOR SPEED = 469 RPM
C.G. STATION = 100.5 (MID)
C_T = .00496
BOOM OFF CONFIGURATION
NEEDLE AND BALL CENTERED

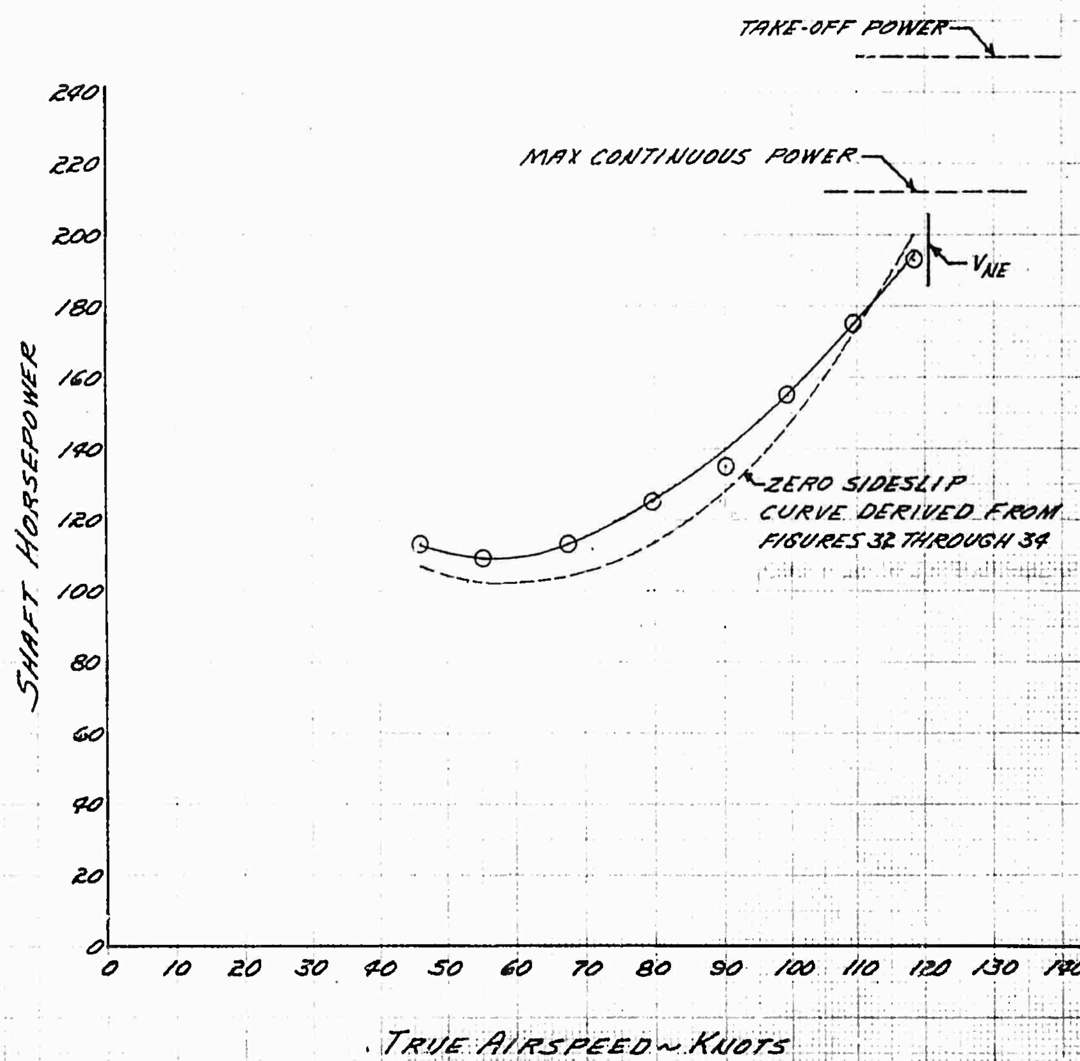

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FIGURE NO. 47
AUTOROTATIONAL CHARACTERISTICS
YOH-6A USA S/N 62-4212

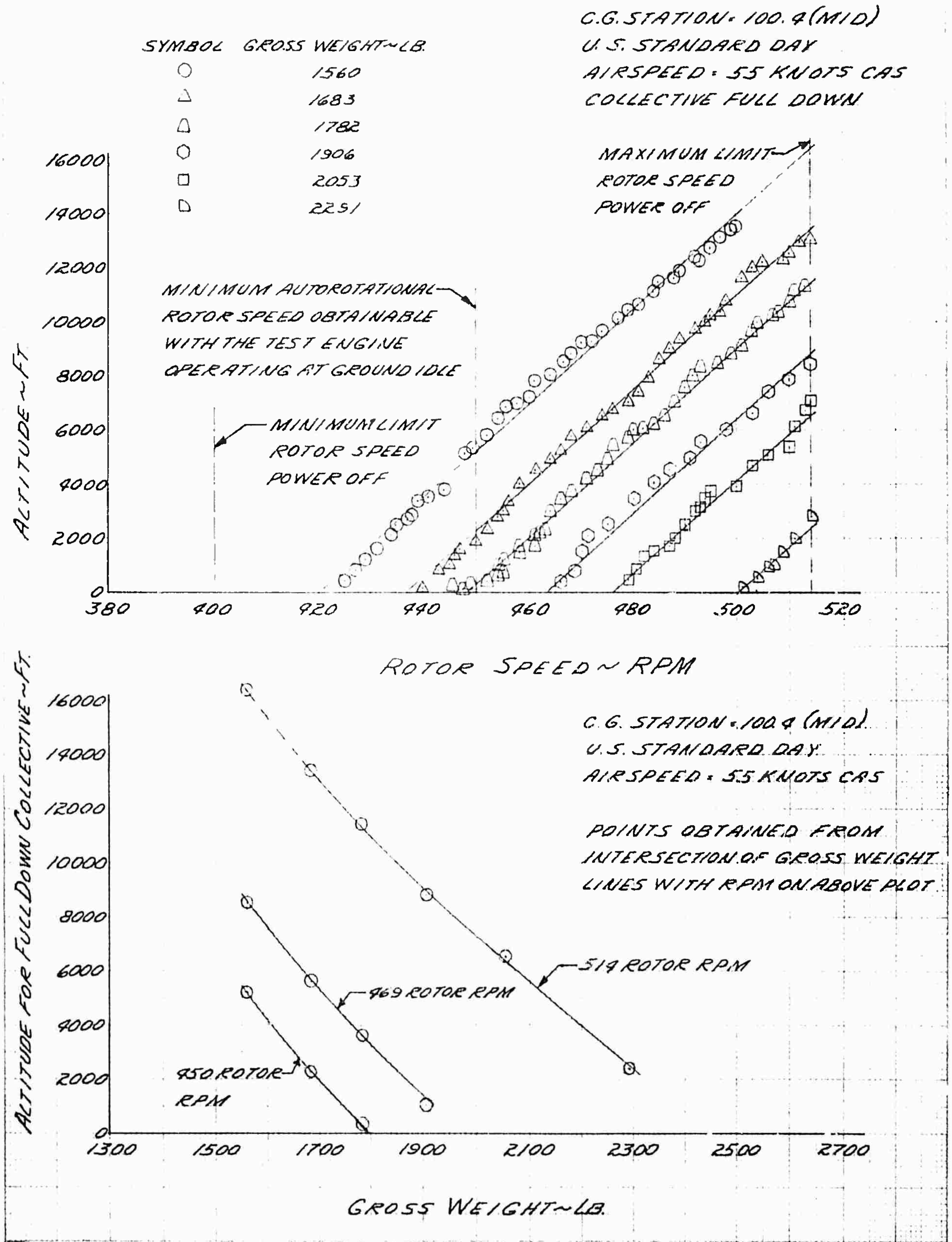

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FIGURE No. 48

CONTROL POSITIONS IN SIDEWARD FLIGHT

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION — ROTOR SPEED = 469 RPM

WIND = CALM

SYMBOL	AVERAGE DENSITY ALTITUDE~FT.	AVERAGE GROSS WEIGHT~LB.	LONGITUDINAL C.G.~IN.	LATERAL C.G.~IN.	FLIGHT CONDITION
○	-100	2585	95.8 (FWD)	3.4 LEFT	IGE (10-15FT)
□	3560	2495	97.0 (FWD LIMIT)	3.9 LEFT	IGE (10-15FT)

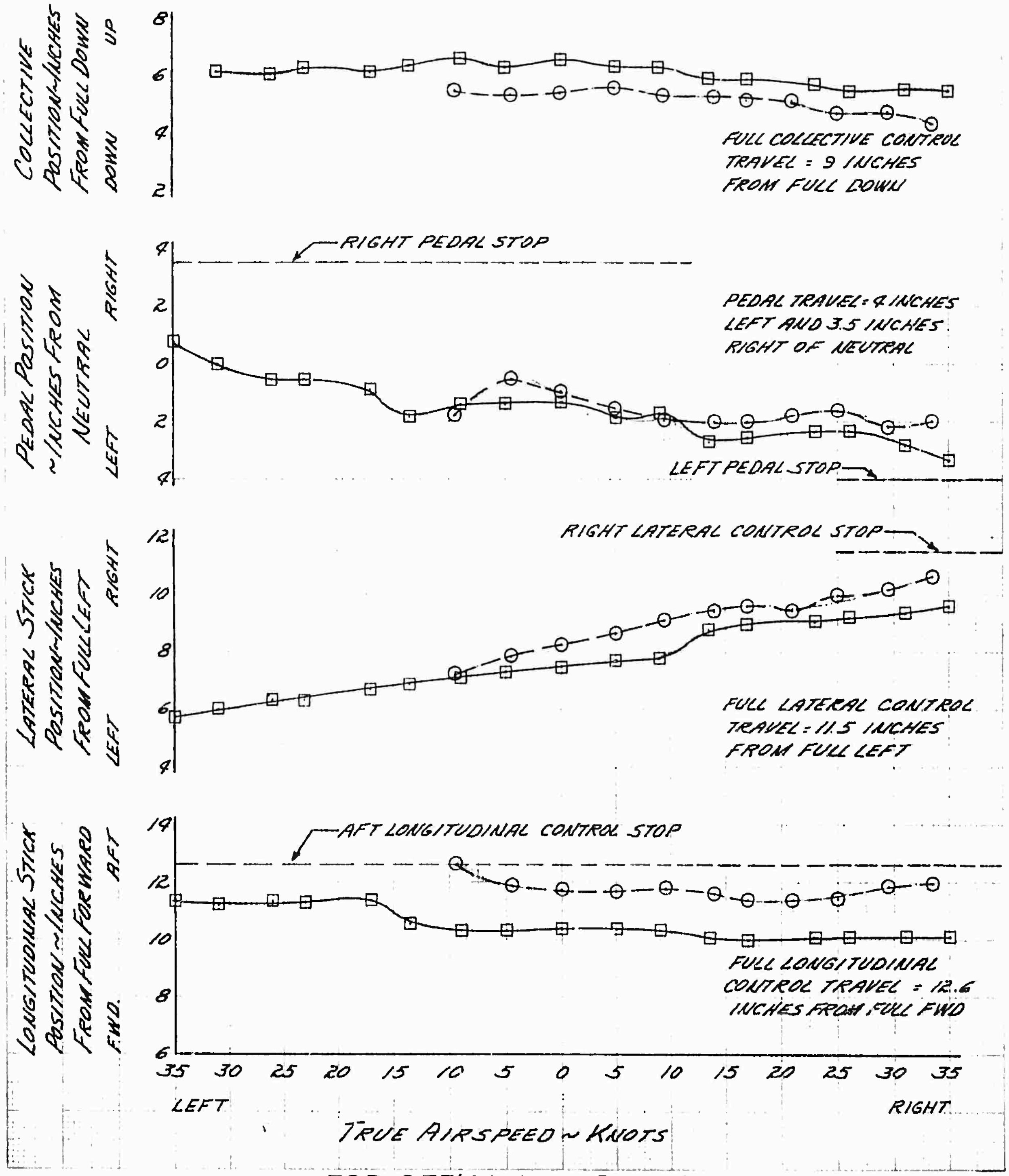

FIGURE NO. 49

CONTROL POSITIONS IN REARWARD FLIGHT

YOH-6A USA S/N 62-4212

CLEAN CONFIGURATION ROTOR SPEED = 469 RPM

WIND: CALM

SYMBOL	AVERAGE DENSITY ALTITUDE ~ FT.	AVERAGE GROSS WEIGHT ~ LB	LONGITUDINAL C.G. ~ IN.	LATERAL C.G. ~ IN.	FLIGHT CONDITION
○	− 100	2585	95.8 (FWD)	3.4 LEFT	IGE (10-15 FT.)
□	3560	2495	97.0 (FWD LIMIT)	3.9 LEFT	IGE (10-15 FT.)

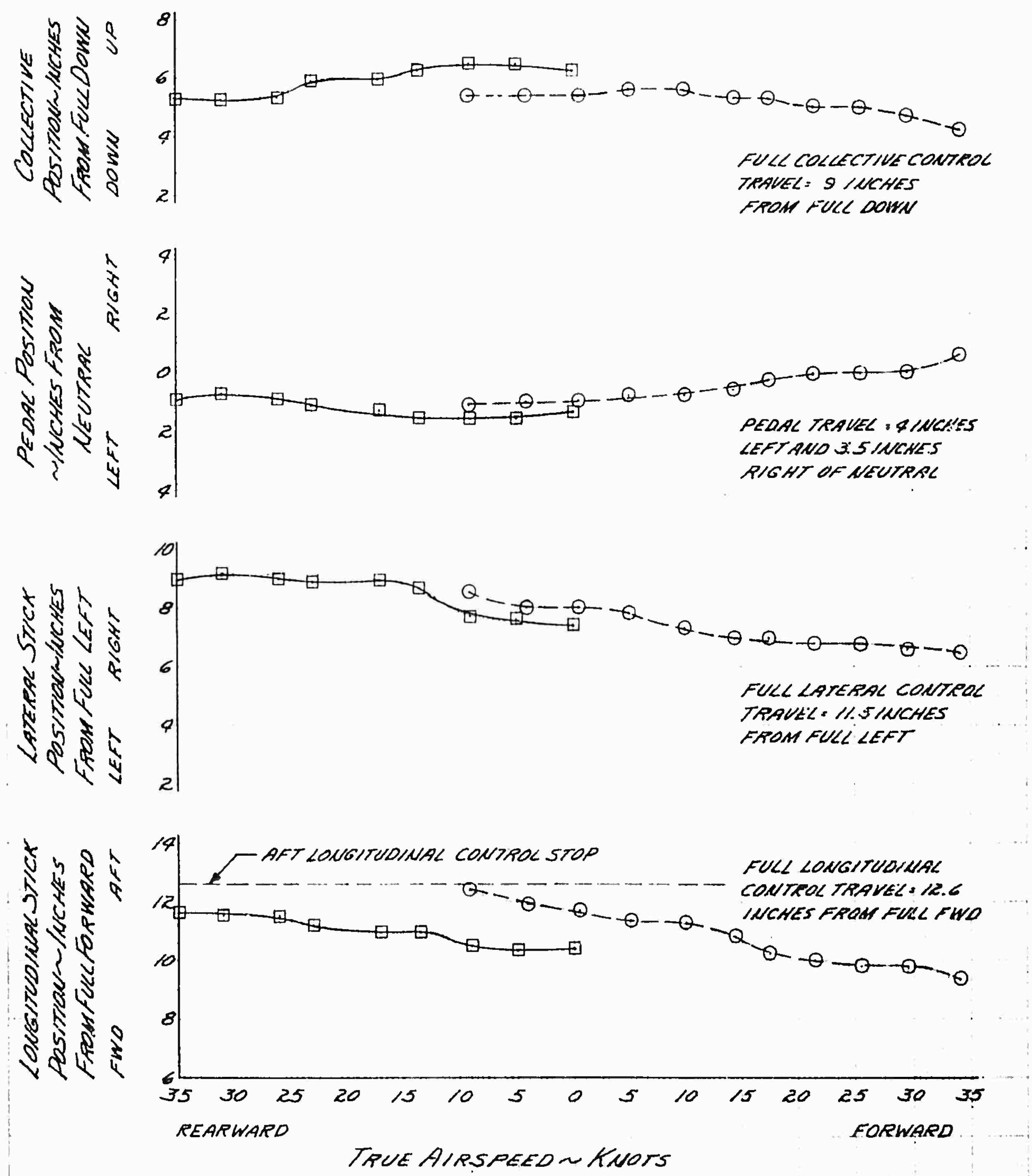

FIGURE No. 50

RECOMMENDED AIRSPEED LIMIT ENVELOPE

YOH-6A USA S/N 62-4212

ROTOR SPEED = 469 RPM (100%)

CLEAN CONFIGURATION

C.G. STATION = 102.6 (AFT)

BASED ON .9 AIRSPEED FOR UNACCELERATED PITCH-UP

REFERENCE FIGURE 49

NOTES:

1- REDUCE LIMIT AIRSPEED BY ONE (1) KNOT FOR FWD C.G.
2- REDUCE LIMIT AIRSPEED BY TWO (2) KNOTS FOR EACH PER CENT ROTOR SPEED BELOW 100%.
3- DASHED LINES ARE EXTRAPOLATIONS BASED ON PROTOTYPE FLIGHT MANUAL V_{NE}.

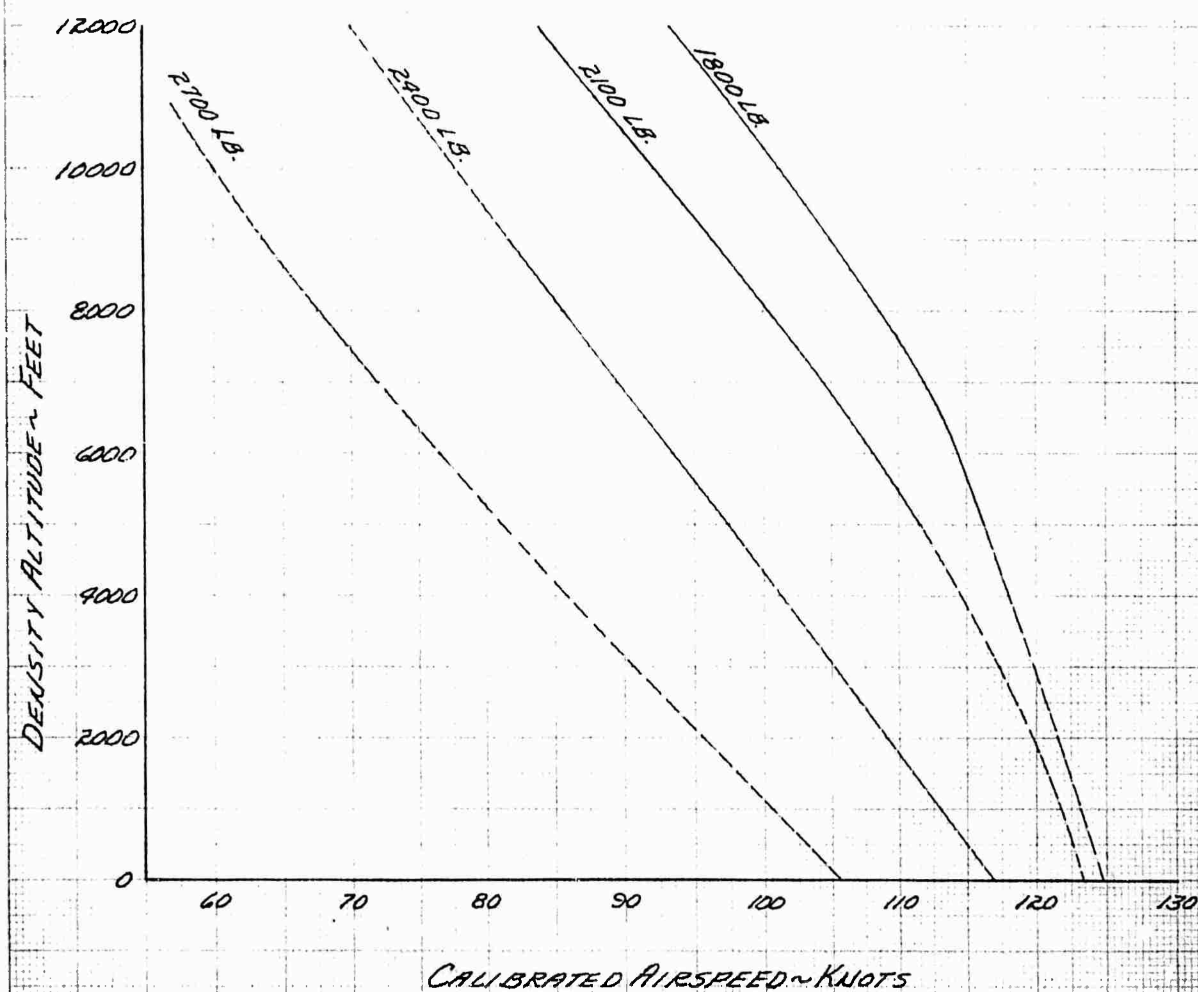

FIGURE NO. 51

UNACCELERATED PITCH-UP AIRSPEED

YOH-6A USA S/N 62-4212

ROTOR SPEED = 469 RPM (100%)

CLEAN CONFIGURATION

C.G. STATION = 102.6 (AFT)

SYMBOL	DENSITY ALTITUDE ~ FEET
○	3000
□	5000
◇	7000
◁	9000
▽	10900
△	12000

NOTES:

1- SYMBOLS WITHOUT TAILS INDICATE C.G. = STATION 102.6 (AFT).
2- SYMBOLS WITH TAILS INDICATE C.G. = STATION 97.0 (FWD).
3- SYMBOLS WITH FLAGS INDICATE TEST DAY POWER LIMITED AIRSPEED.

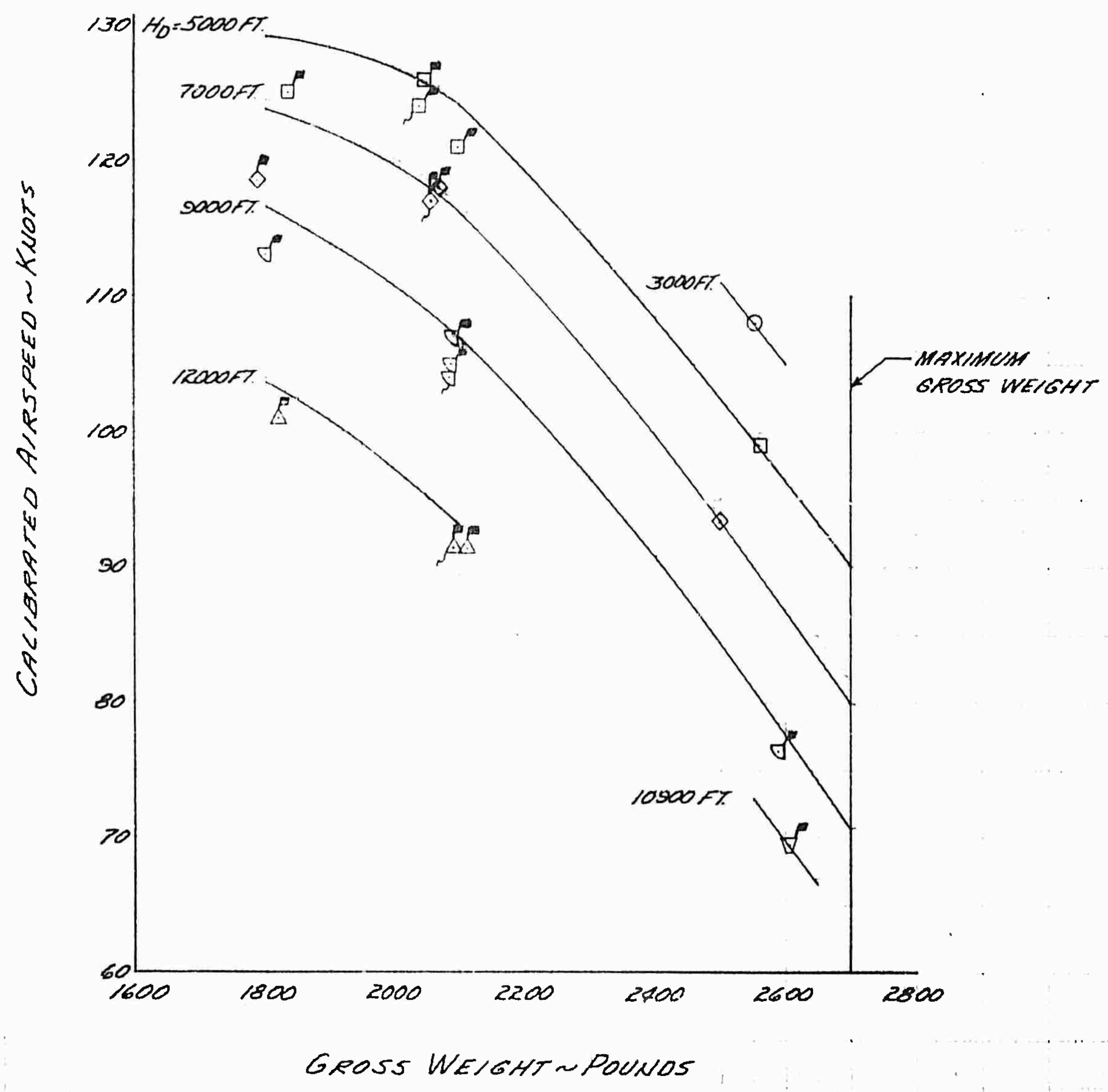

FIGURE NO. 52

UNACCELERATED PITCH-UP AIRSPEED

YOH-6A USA S/N 62-4212

ROTOR SPEED: 463 RPM

CLEAN CONFIGURATION

C.G. STATION = 102.6 (AFT)

SYMBOL	DENSITY ALTITUDE~FEET
○	3000
□	5000
◇	7000
◁	9000
▽	10900
△	12000

NOTES:

1- SYMBOLS WITHOUT TAILS INDICATE C.G. STATION 102.6 (AFT).

2- SYMBOLS WITH TAILS INDICATE C.G. STATION 97.0 (FWD).

3- SYMBOLS WITH FLAGS INDICATE TEST DAY POWER LIMITED AIRSPEED.

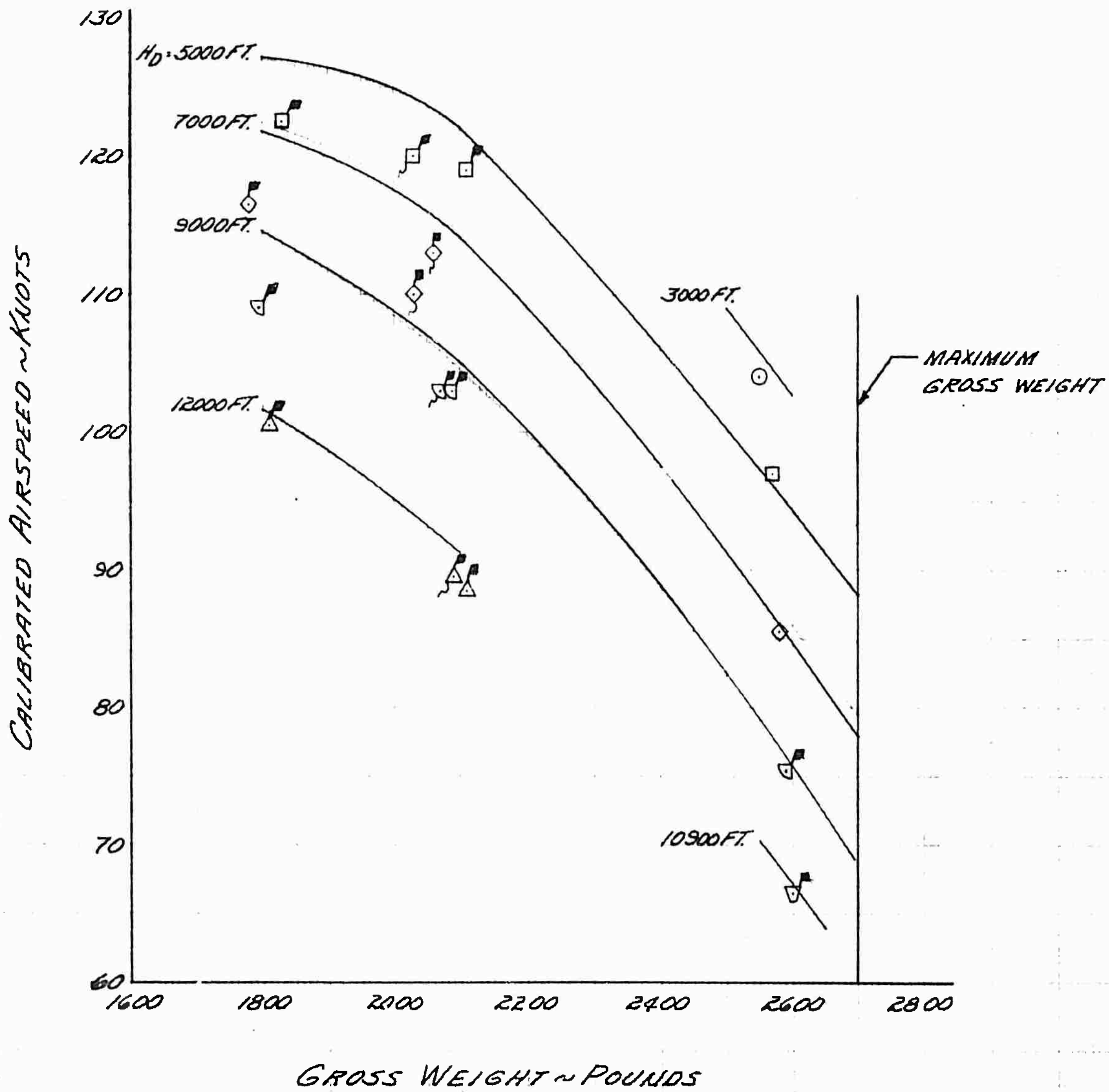

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FIGURE NO. 53
CONTROL POSITIONS IN ACCELERATED FLIGHT
YOH-6A USA S/N 62-4212
METHOD: STEADY TURNS
CLEAN CONFIGURATION
ZERO SIDESLIP

SYMBOL TRIM	SYMBOL RIGHT	SYMBOL LEFT	AVERAGE DENSITY ALTITUDE~FT	CALIBRATED AIRSPEED~KTS	AVERAGE GROSS WEIGHT~LB	LONGITUDINAL C.G.~IN	ROTOR SPEED ~RPM	FLIGHT CONDITION
●	○	○	6500	55	2120	102.4	470	TRIMMED FOR
■	□	□	5000	55	2080	102.6	469	LEVEL FLIGHT

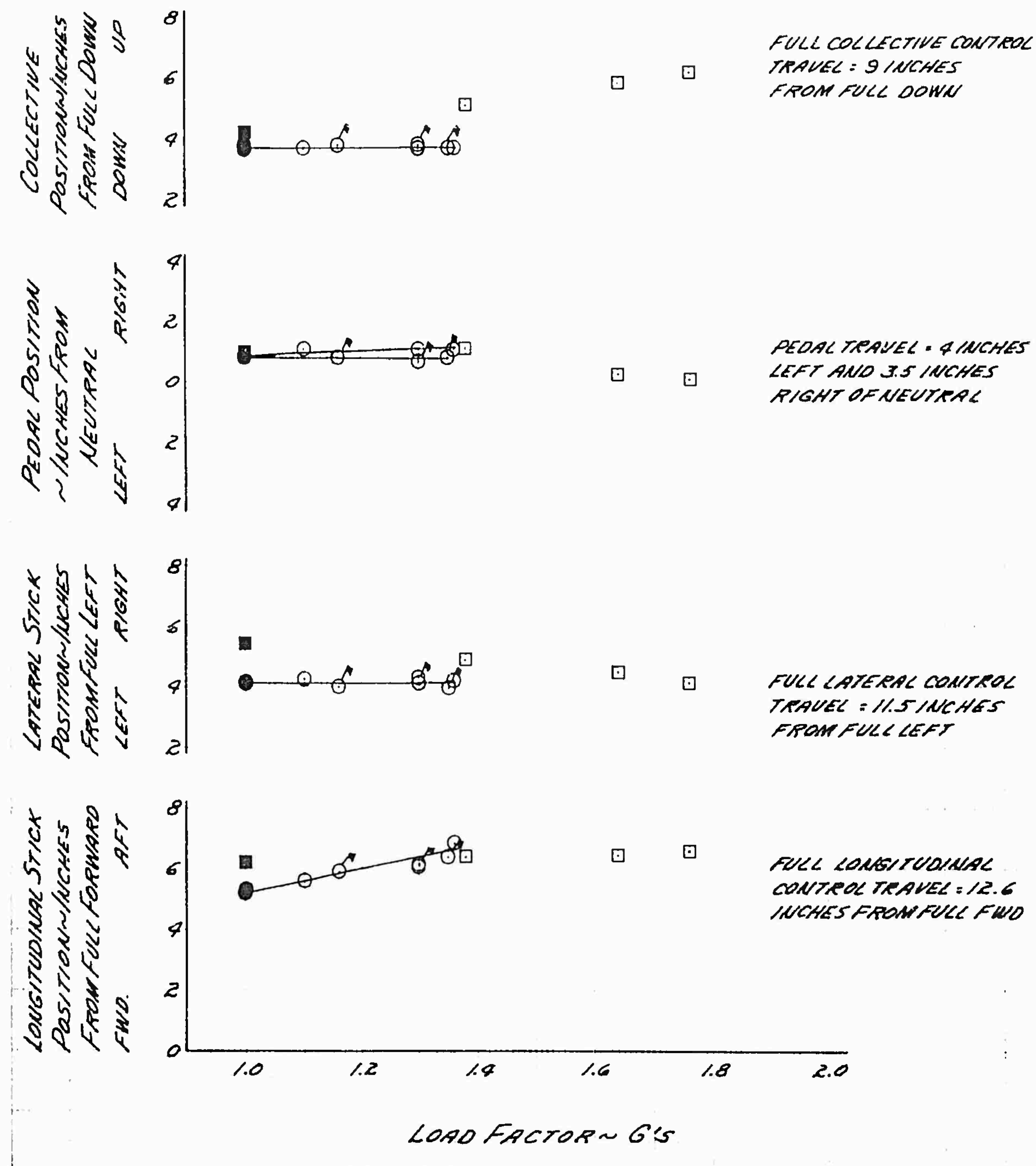

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FIGURE No. 54

CONTROL POSITIONS IN ACCELERATED FLIGHT

YOH-6A USA S/N 62-4212

METHOD: STEADY TURNS

CLEAN CONFIGURATION

ZERO SIDESLIP

SYMBOL TRIM	SYMBOL RIGHT	SYMBOL LEFT	AVERAGE DENSITY ALTITUDE ~ FT	CALIBRATED AIRSPEED ~ KTS	AVERAGE GROSS WEIGHT ~ LB	LONGITUDINAL C.G. ~ IN	ROTOR SPEED ~ RPM	FLIGHT CONDITION
●	○	○ (flagged)	5000	55	2090	98.7	469	TRIMMED FOR LEVEL FLIGHT

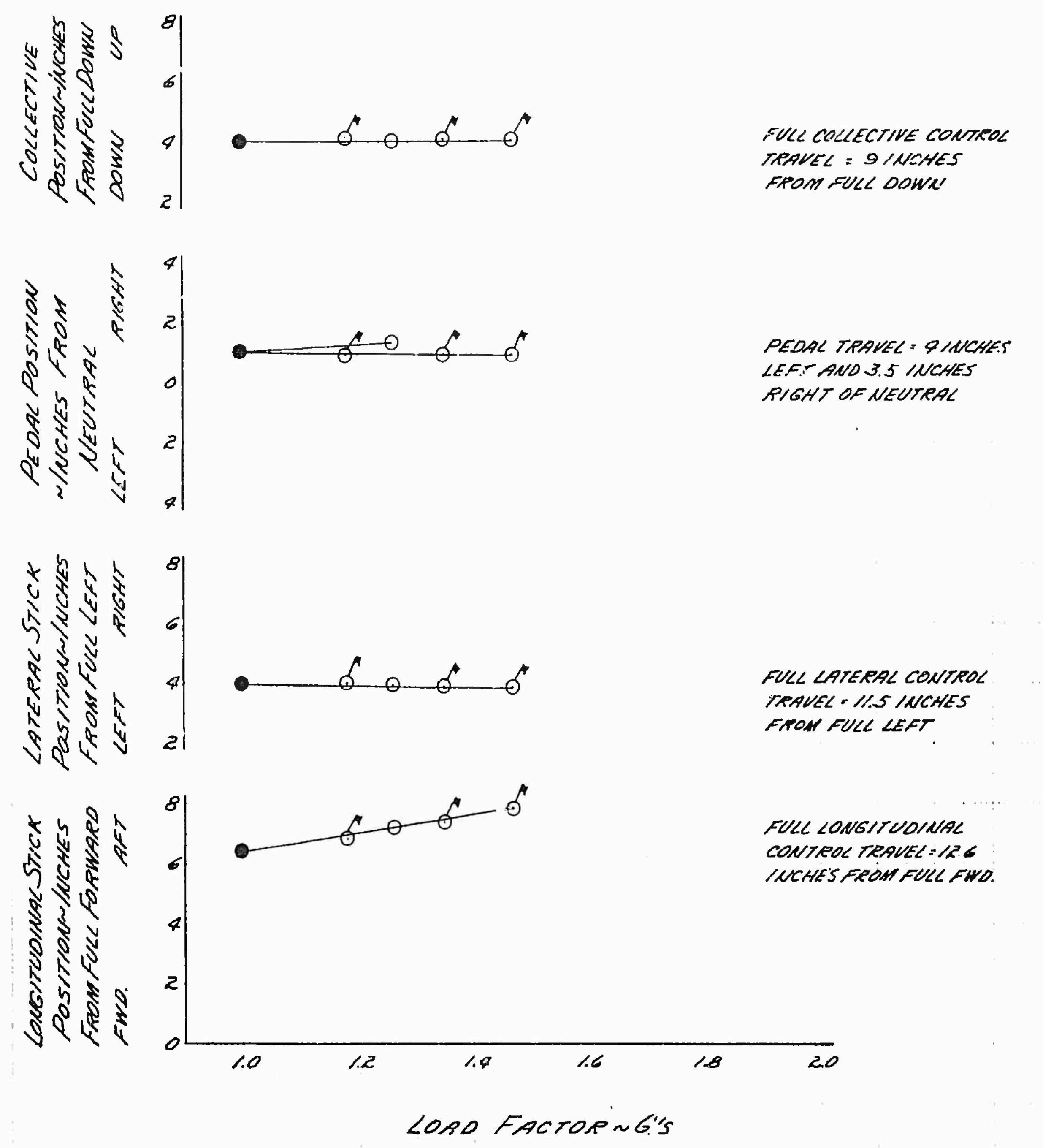

FIGURE NO. 55
CONTROL POSITIONS IN ACCELERATED FLIGHT
YOH-6A USA S/N 62-4212
METHOD: STEADY TURNS
CLEAN CONFIGURATION
ZERO SIDESLIP

SYMBOL TRIM	SYMBOL RIGHT	SYMBOL LEFT	AVERAGE DENSITY ALTITUDE ~ FT	CALIBRATED AIRSPEED ~ KTS	AVERAGE GROSS WEIGHT ~ LB	LONGITUDINAL C.G. ~ IN	ROTOR SPEED ~ RPM	FLIGHT CONDITION
●	○	○ (flagged)	5000	95	2100	102.4	470	TRIMMED FOR
■	□	□ (flagged)	5000	95	2080	102.6	469	LEVEL FLIGHT

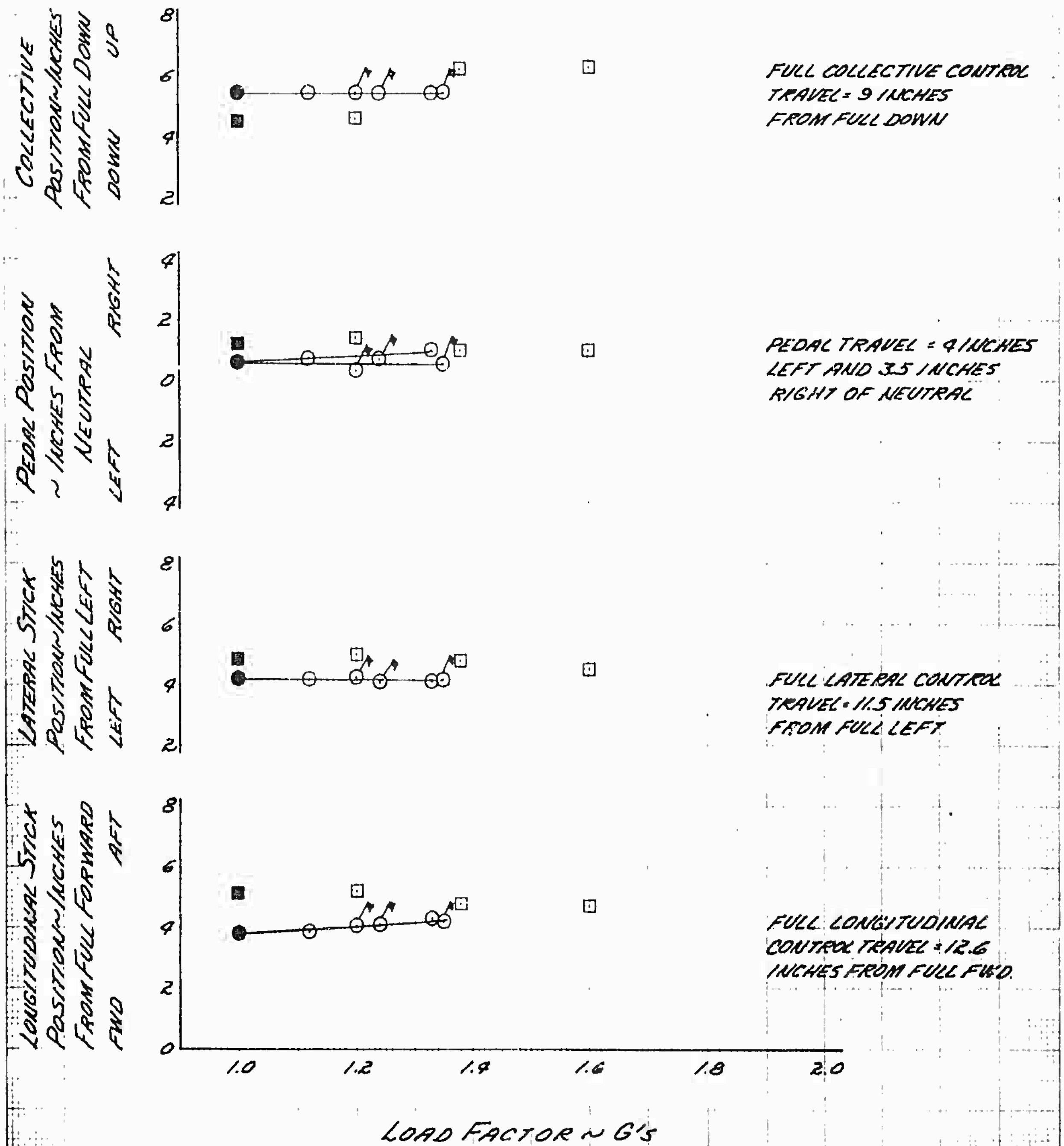

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FIGURE NO. 56

CONTROL POSITIONS IN ACCELERATED FLIGHT

YOH-6A USA S/N 62-4212

METHOD: STEADY TURNS

CLEAN CONFIGURATION

ZERO SIDESLIP

SYMBOL TRIM	RIGHT	LEFT	AVERAGE DENSITY ALTITUDE ~ FT.	CALIBRATED AIRSPEED ~ KTS.	AVERAGE GROSS WEIGHT ~ LB.	LONGITUDINAL C.G. ~ IN.	ROTOR SPEED ~ RPM	FLIGHT CONDITION
●	○	♂	5000	95	2090	98.7	469	TRIMMED FOR LEVEL FLIGHT

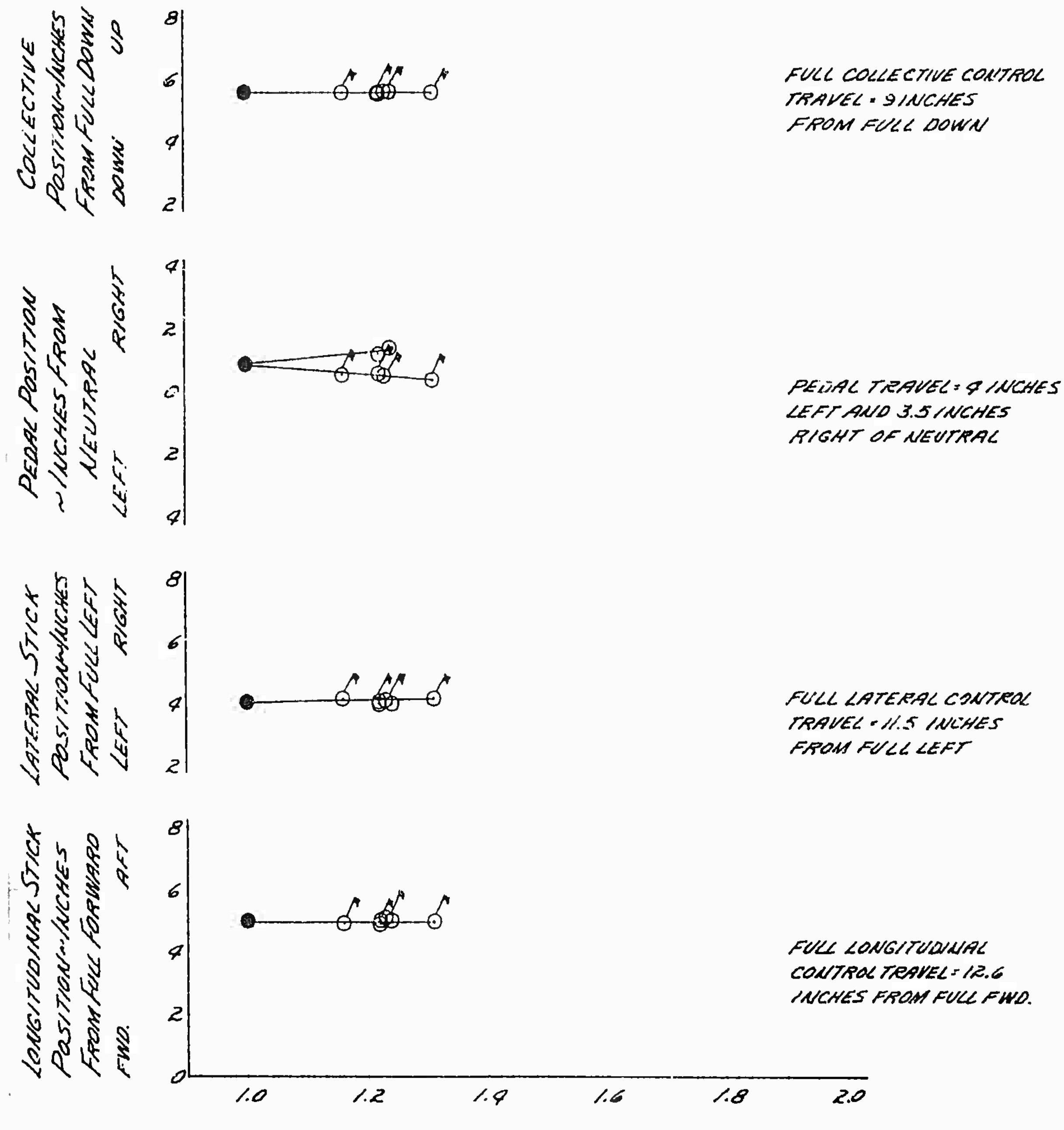

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FIGURE NO. 57
AIRSPEED CALIBRATION
YOH-6A USA S/N 62-4212
STANDARD SYSTEM
BOOM OFF

AVERAGE C.G. = STATION 100.2 ROTOR SPEED = 469 RPM

SYMBOL	AVERAGE DENSITY ALTITUDE ~ FT	AVERAGE GROSS WEIGHT ~ LB.	FREE AIR TEMP. ~ °C	CONDITION	METHOD
○	3200	2120	23.5	LEVEL FLIGHT	TRAILING BOMB
△	4960	2100	22.0	CLIMB ~ 2000 FT/MIN.	TRAILING BOMB
◺	4500	2090	20.5	DESCENT ~ 1000 FT/MIN.	TRAILING BOMB
□	2890	2160	22.0	LEVEL FLIGHT	T-28 PACER

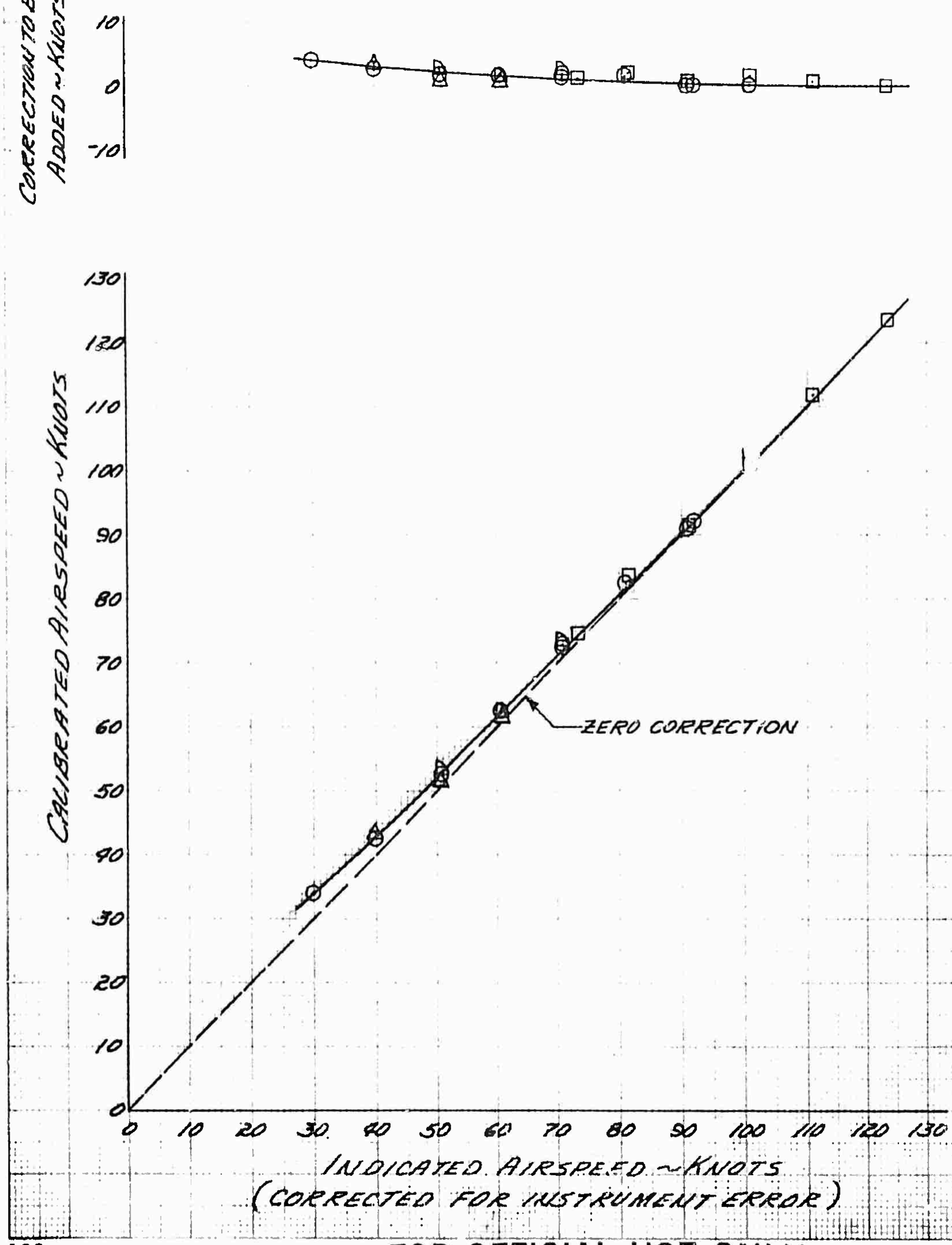

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FIGURE NO. 58
AIRSPEED CALIBRATION
YOH-6A USASN 62-4212
BOOM SYSTEM
GROUND SPEED COURSE
CLEAN CONFIGURATION

AVERAGE DENSITY ALTITUDE~FT.	AVERAGE GROSS WEIGHT~LB.	AVERAGE C.G.~IN.	FREE AIR TEMP.~°C	ROTOR SPEED ~RPM
2360	2100	100.1	12	469

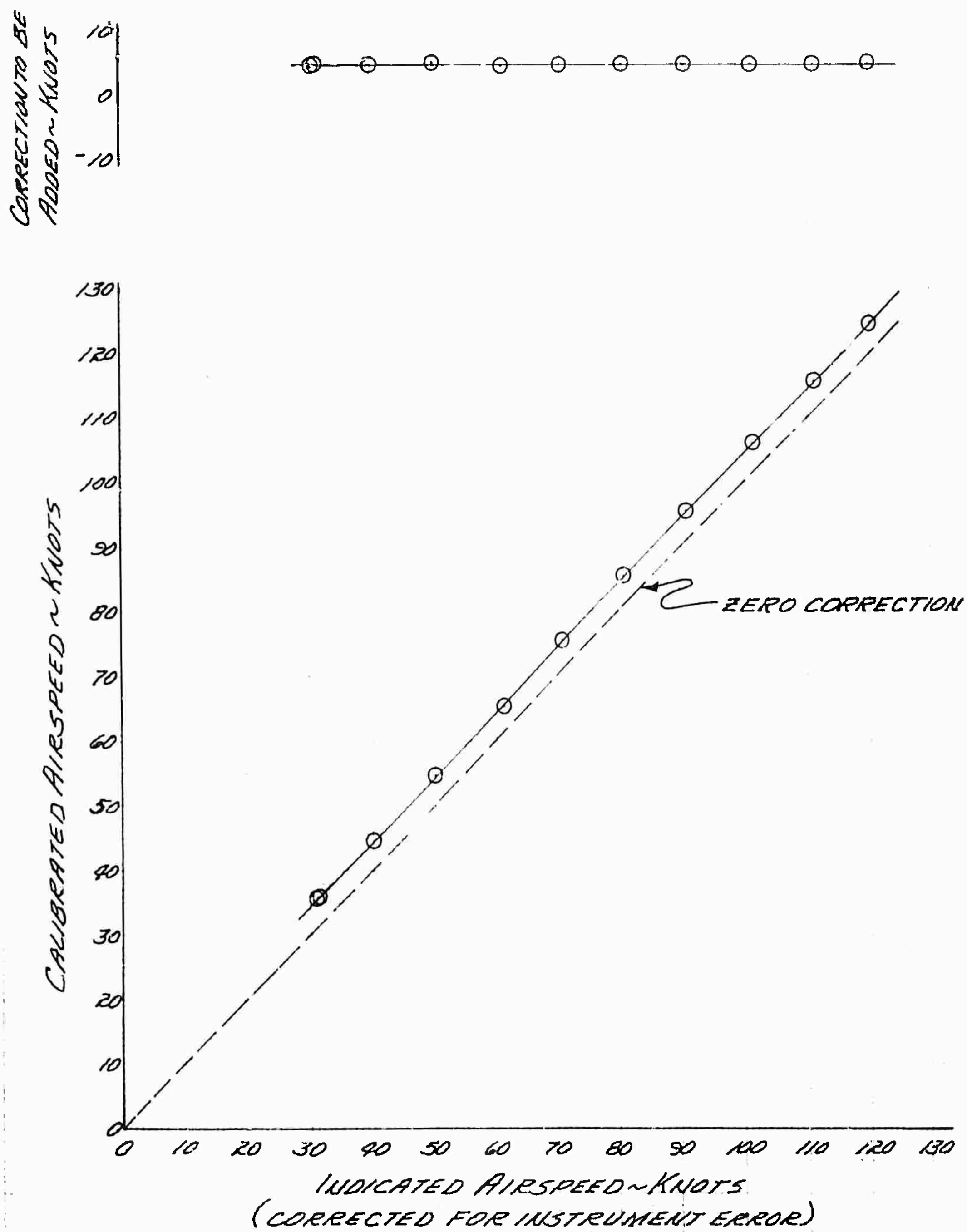

FIGURE No 59

SHAFT HORSEPOWER AVAILABLE

YOH-6A USA S/N 62-4212

T-63-A-5

TAKE-OFF POWER

$T_{T_5} = 738°C$

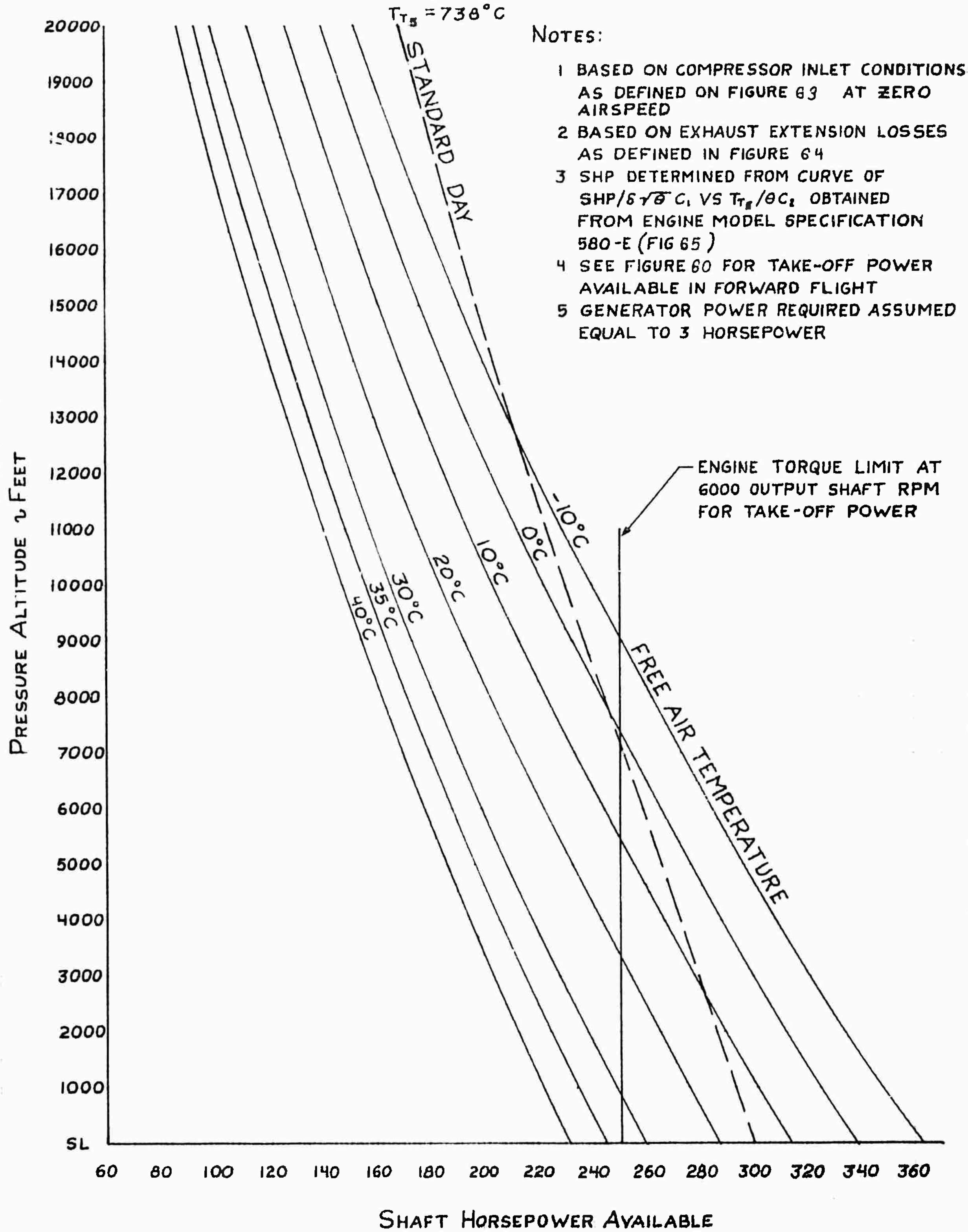

FIGURE No 60
SHAFT HORSEPOWER AVAILABLE
YOH-6A USA S/N 62-4212
T63-A-5
TAKE-OFF POWER T_{T_5} = 738 °C
US STANDARD DAY

NOTES:
1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 53 REFERENCE K
2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 64
3. SHP DETERMINED FROM CURVE OF $SHP/\delta\sqrt{\theta}\,C_1$ VS $T_{T_5}/\theta C_2$ AS OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIGURE 65)
4. SEE FIGURE 59 FOR TAKE-OFF POWER AVAILABLE IN A HOVER
5. GENERATOR POWER REQUIRED ASSUMED EQUAL TO 3 HORSEPOWER

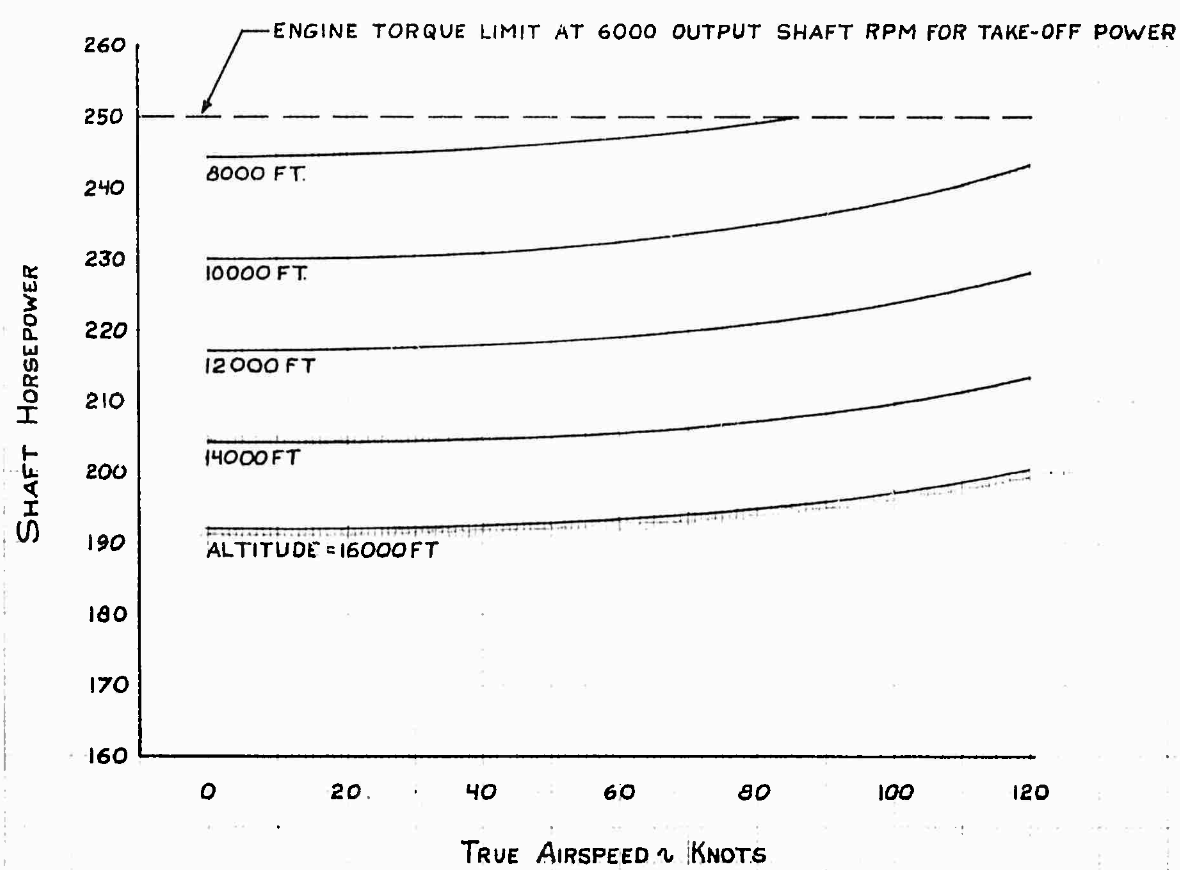

FIGURE NO 61

SHAFT HORSEPOWER AVAILABLE

YOH-6A USA S/N 62-4212

T-63-A-5

MAXIMUM CONTINUOUS POWER

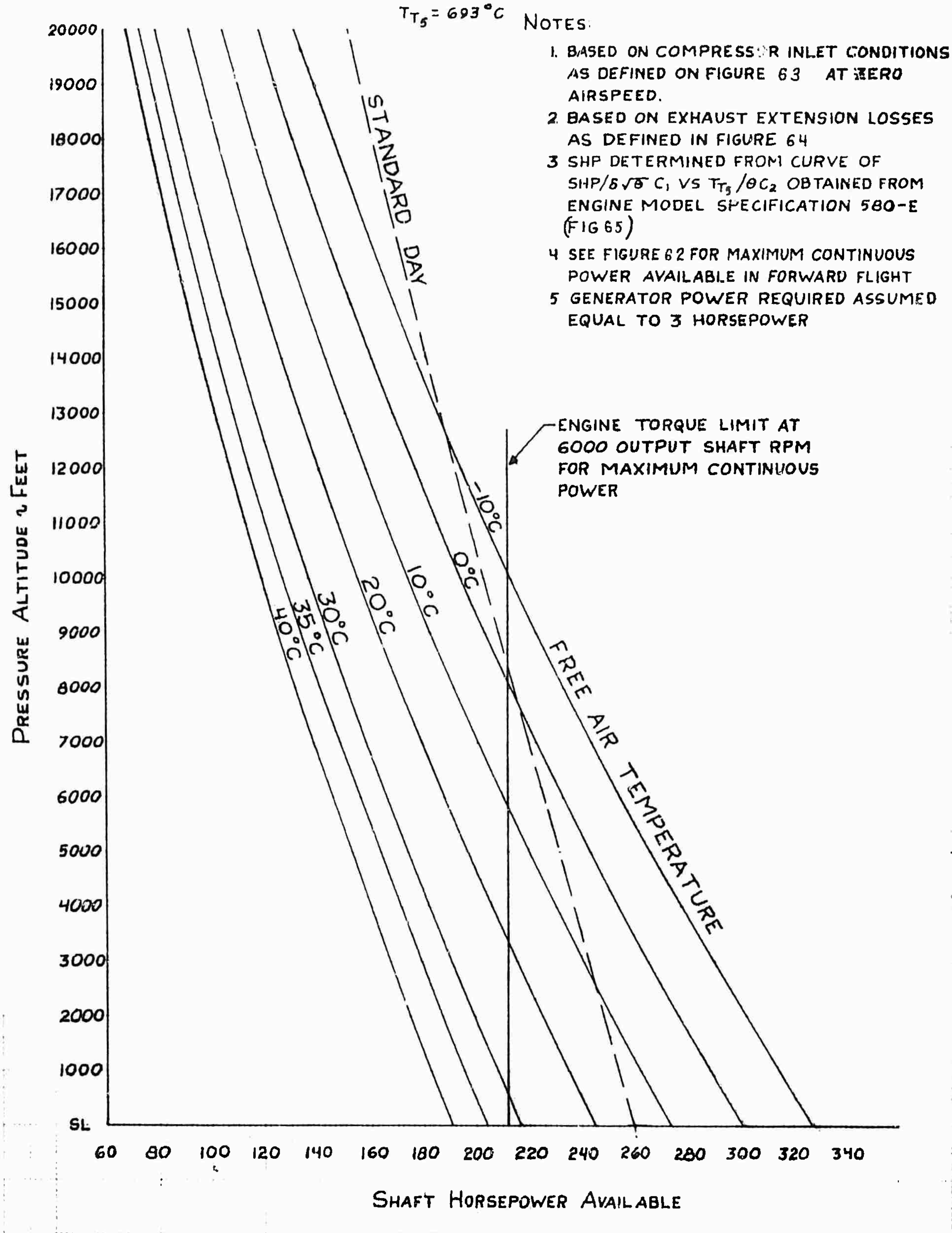

FIGURE No 62

SHAFT HORSEPOWER AVAILABLE

YOH-6A USA S/N 62-4212

T63-A-5

MAXIMUM CONTINUOUS POWER T_{T_5} = 693°C

US STANDARD DAY

NOTES:

1 BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 53 REFERENCE K

2 BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 64

3 SHP DETERMINED FROM CURVE OF $SHP/\delta\sqrt{\theta}\,C_1$ VS $T_{T_5}/\theta C_2$ AS OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIGURE 65)

4 SEE FIGURE 61 FOR MAXIMUM CONTINUOUS POWER AVAILABLE IN A HOVER

5 GENERATOR POWER REQUIRED ASSUMED EQUAL TO 3 HORSEPOWER

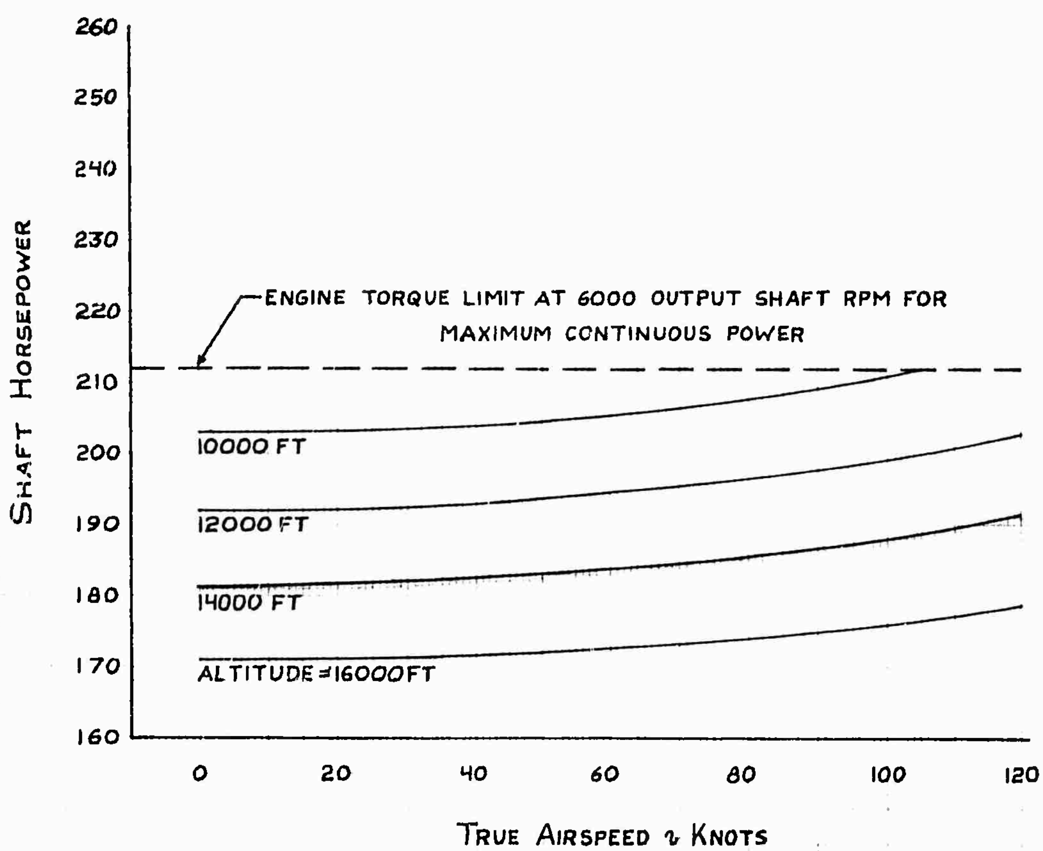

FIGURE NO. 63
ENGINE INLET CHARACTERISTICS
YOH-6A USA S/N 62-4212

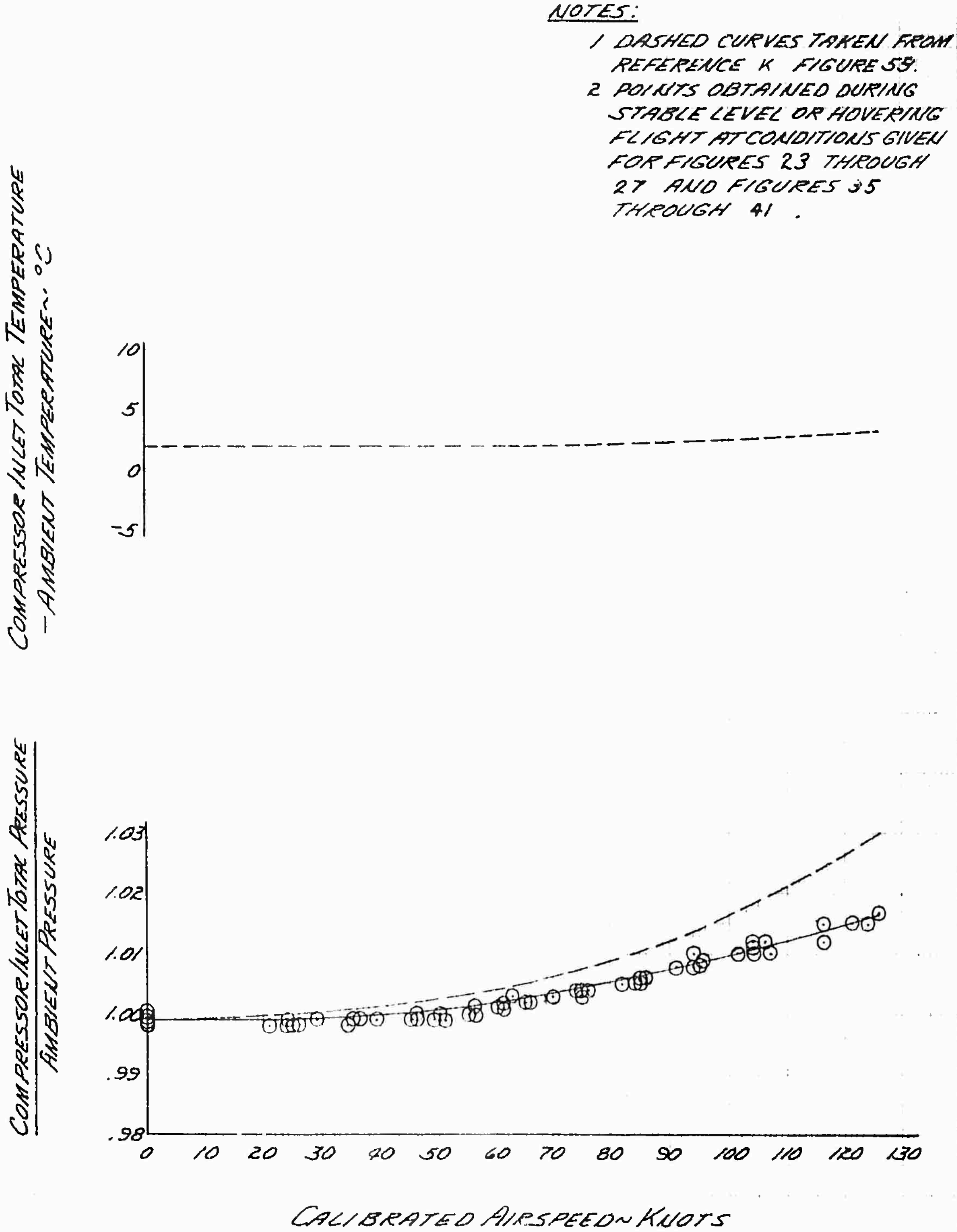

FIGURE NO. 64

EFFECT OF YOH-6A EXHAUST EXTENSION ON T63-A-5 EXHAUST STATIC PRESSURE

NOTES:

1 BASED ON ENGINE MANUFACTURER'S TEST STAND CALIBRATION.
2 P_{S7} = STATIC PRESSURE AT EXIT OF ENGINE EXHAUST COLLECTOR.
3 $P_{AMBIENT}$ = STATIC PRESSURE AT EXIT OF YOH-6A EXHAUST EXTENSION.

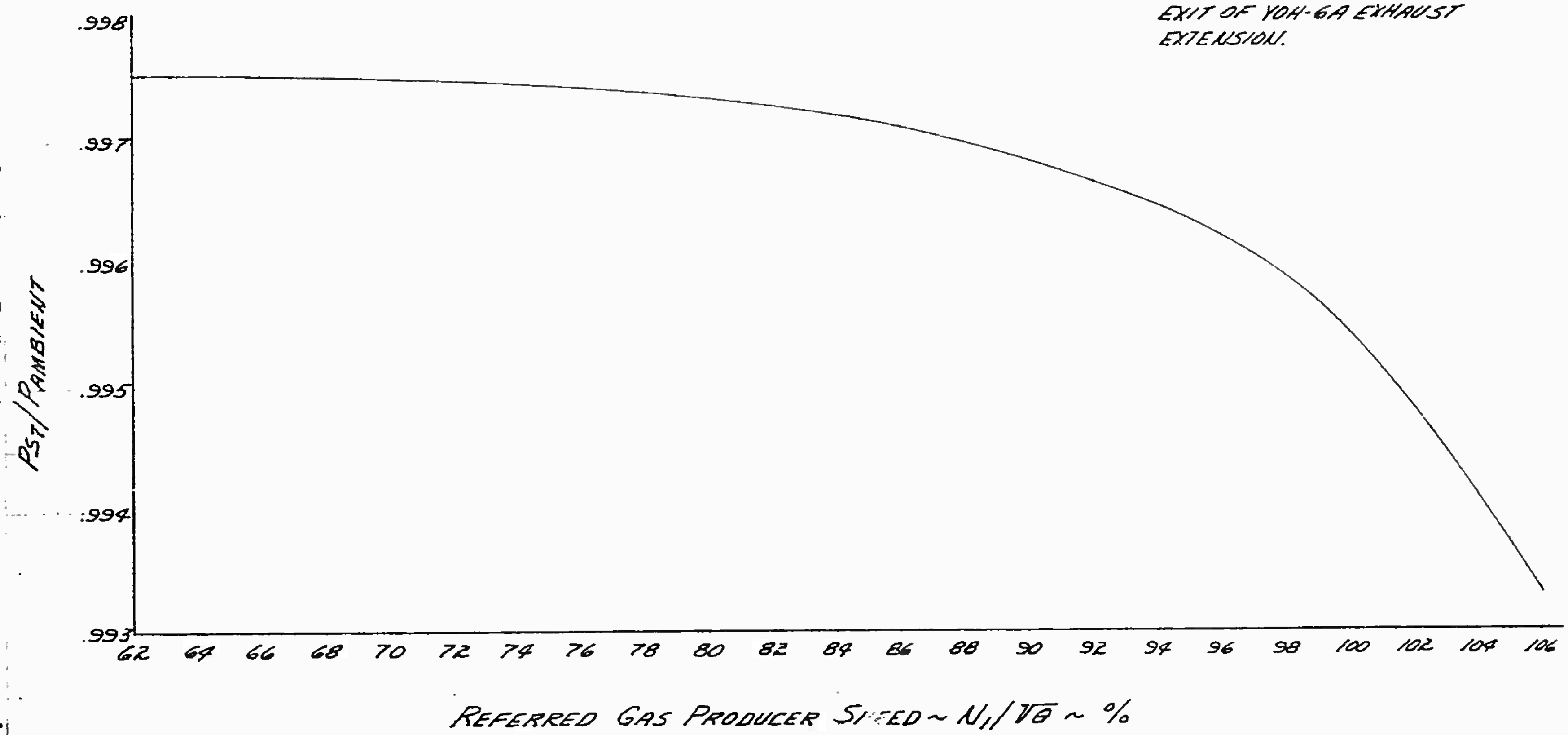

FIGURE No. 65
ENGINE CHARACTERISTICS
T63-A-5
ENGINE MODEL SPECIFICATION
580-E FOR SEA LEVEL STANDARD DAY

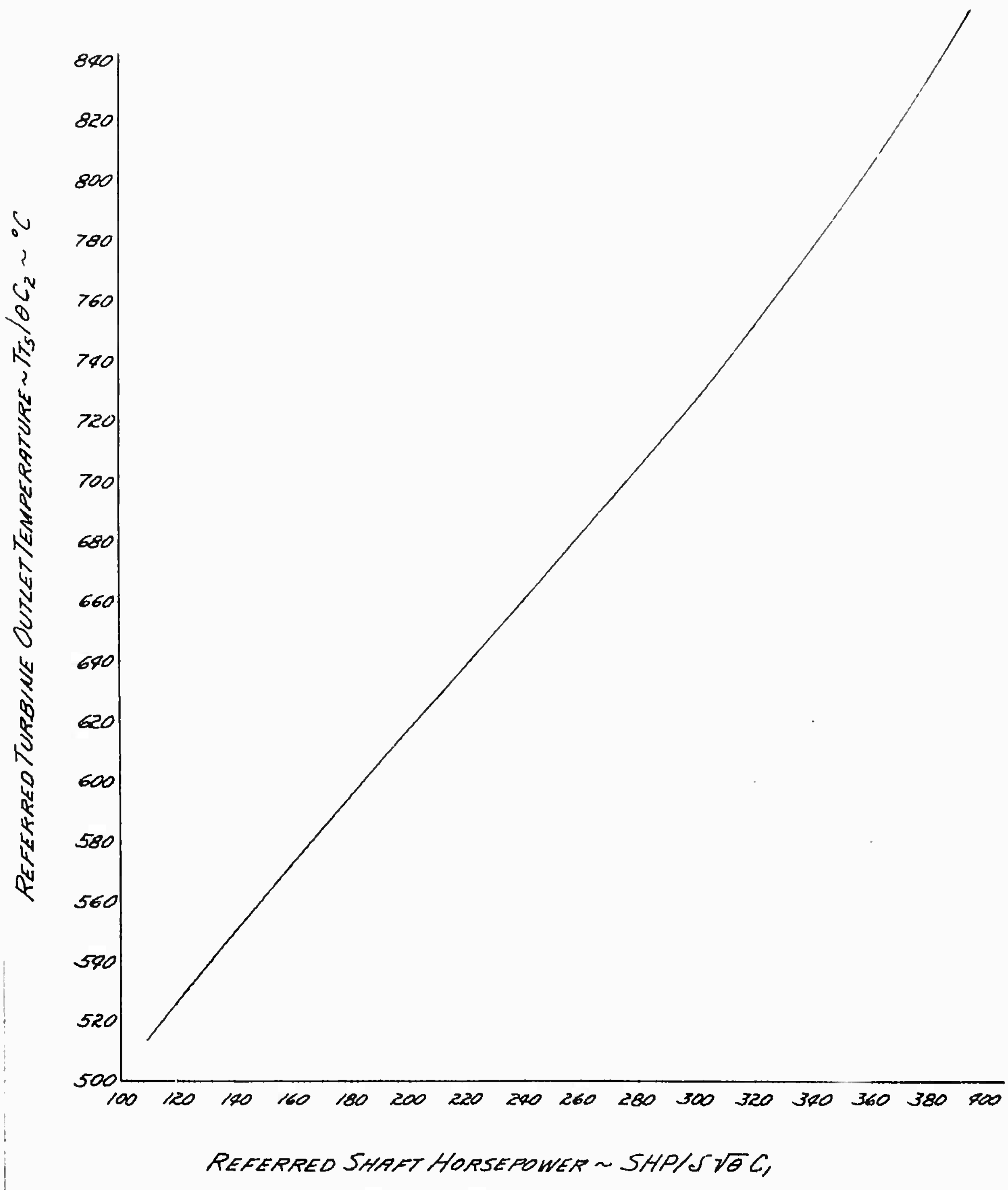

FIGURE NO. 66
ENGINE CHARACTERISTICS
T63-A-5
ENGINE MODEL SPECIFICATION
580-E FOR SEA LEVEL STANDARD DAY

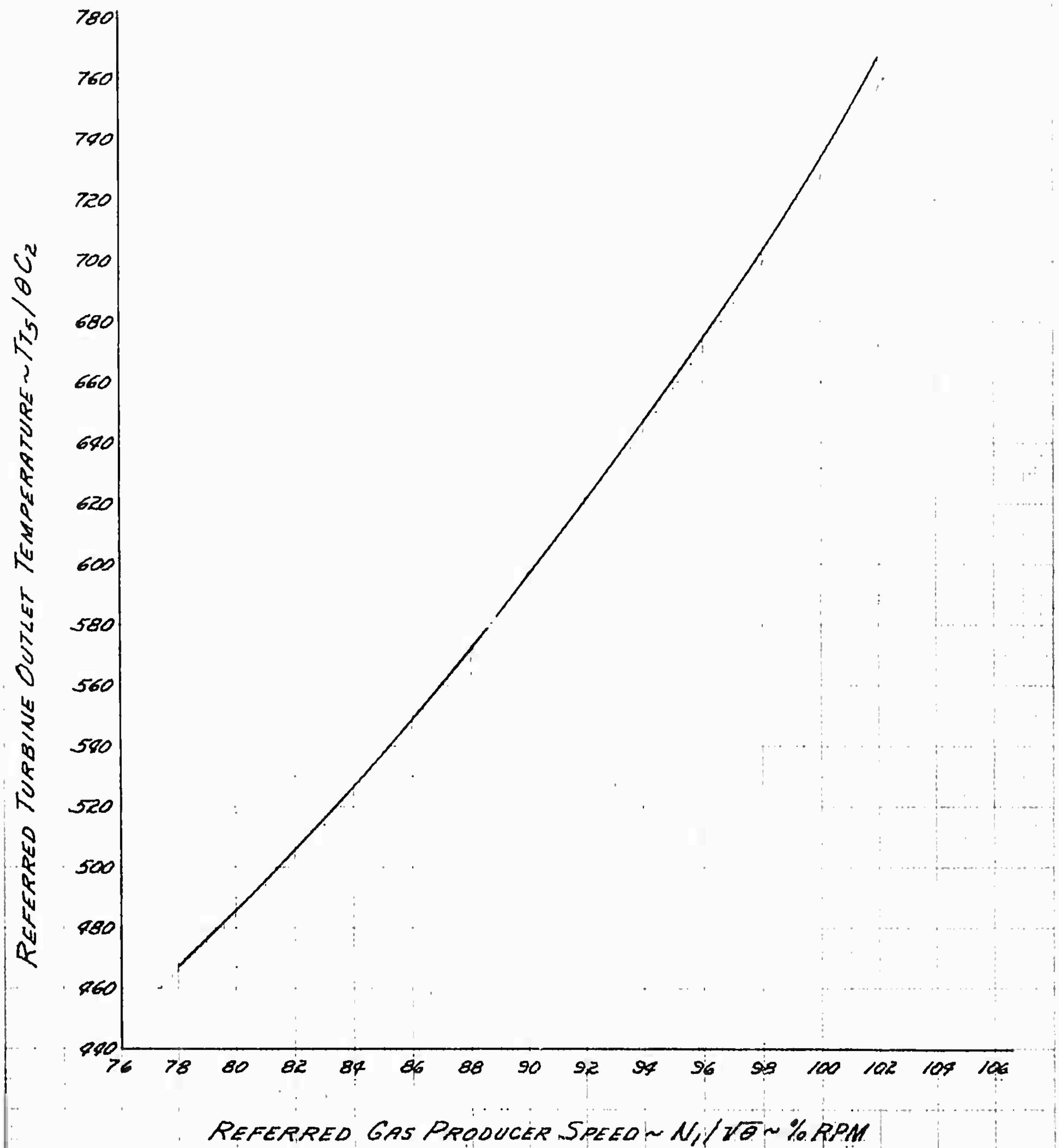

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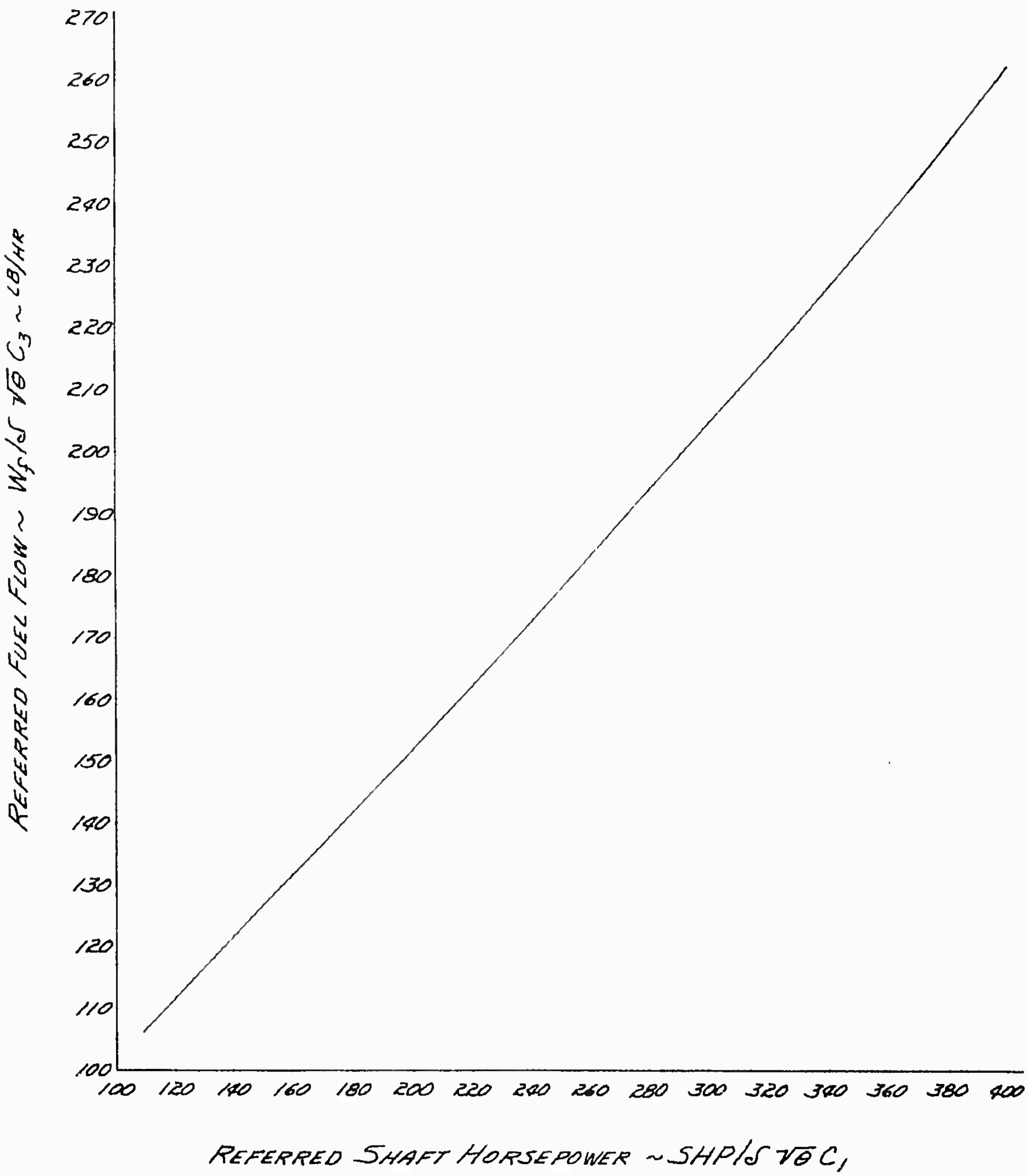
FIGURE No. 67

ENGINE CHARACTERISTICS

T63-A-5

ENGINE MODEL SPECIFICATION

580-E FOR SEA LEVEL STANDARD DAY



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Appendix II

TEST METHODS and DATA REDUCTION PROCEDURES

1.0 GENERAL

The general method used to reduce and analyze the test data was based on non-dimensional analysis of the terms effecting helicopter performance. This method allowed more concise and complete summaries to be made and made it easier to derive particular conditions. It should be noted that the non-dimensional method is useful only where compressibility and blade stall effects are not significant. The following non-dimensional terms were used:

Thrust Coefficient

$$C_T = \frac{W}{\rho A(\omega r)2}$$

Power Coefficient

$$C_P = \frac{SHP\ (550)}{\rho A\ (\omega r)^3}$$

Advance Ratio

$$\mu = \frac{V_T \times 1.6889}{r}$$

Where:

W = Gross weight, lb

ρ = Ambient Air Density, slugs/ft^3

A = Rotor Disk Area, ft^2

ω = Rotor Angular Velocity, radians/sec

r = Rotor Radius, ft

SHP= Engine Output Shaft Horsepower

V_T = True Airspeed, kt

In most cases data were reduced to standard-day conditions so that comparisons could be made. The U.S. Standard Atmosphere described in NACA Report 1235 was used. This is an approximate average of conditions experienced at various latitudes, time of day, and season of the year presented as a function of height above sea level.

2.0 POWER DETERMINATION

Engine output power was determined from readings obtained from sensitive torque pressure gages and rotor speed tachometers. Power was computed using the following equation:

$$SHP = \frac{2\pi \times T \times N_E}{33000}$$

Where:

N_E = output shaft speed, rpm

T = output shaft torque, ft-lb

Calibration of the engine torque system indicated that torque was the following function of torque pressure:

$$T = C \times P$$

Where:

C = Constant (slightly different for each engine)

P = Torque Meter Pressure, in Hg

Two corrections in addition to the instrument correction were made to the torque pressure. The weight of the piston in the engine torquemeter had to be balanced when the engine was not in a horizontal position (the engine is canted 43 degrees in the YOH-6A). This was corrected by adding 1.55 inches of mercury torque pressure to the indicated reading. The second correction, head effect of the torque pressure oil, resulted from the engine torque sending unit being at a different level than the gage. This was corrected by subtracting the static pressure reading from the indicated readings obtained in flight. Considering these 2 corrections the torquemeter pressure was obtained from torquemeter readings as follows:

$$P = P_{IND} + 1.55 - P_s$$

Where:

P_{IND} = Indicated torquemeter reading corrected for instrument error

P_s = Static torquemeter reading corrected for instrument error

Rotor speed may be determined from engine output shaft speed as follows:

$$N_R = N_E \times .078125$$

By substituting the last 3 equations in the first, a convenient equation for determining output shaft horsepower may be developed.

$$SHP = \frac{2\pi}{33000 \times .078125} C\, N_R\, (P_{IND} + 1.55 - P_s)$$

$$= .00243588\, C\, N_R\, (P_{IND} + 1.55 - P_s)$$

3.0 TAKEOFF

Takeoff tests were conducted to obtain curves of climbout airspeed versus distance required to clear 50 feet. Each curve was obtained by conducting a series of takeoffs using various climbout airspeeds. During each series ballast was added or removed as necessary so as to maintain the desired excess power available conditions as fuel was consumed and ambient temperature varied. A ground operated Fairchild Flight Analyzer was used to produce a photographic record of time, horizontal distance, and vertical distance for each takeoff. The climbout airspeed range used for each series of takeoffs varied from the minimum achievable to the maximum practical airspeed (approximately 50 knots IAS). All takeoff tests were performed in winds of 2 knots or less.

The excess power method of takeoff analysis was used. Power coefficient (C_P) available was computed using the power available at the atmospheric conditions and rotor speed for the test. Cp required to hover at a skid height of 2 feet was determined by calculating thrust coefficient (C_T) at takeoff conditions (weight, density, altitude, and rotor speed) and entering the 2-foot hover curve at this C_T to obtain the corresponding Cp. Then excess power was determined as follows:

$$\Delta C_P = C_P \text{ Available} - C_P \text{ Required at 2 ft}$$

Distance to clear 50 feet was obtained by plotting a time history of height and distance from the Fairchild plate and reading the horizontal distance at the 50-foot skid height point. A wind correction was made to the distance by multiplying the time required from takeoff initiation to 50 feet by the headwind component velocity and adding this resulting distance to the distance obtained from the Fairchild plate. The climbout airspeed was determined from the height-distance time history by calculating the horizontal and vertical velocities and the determining of the resultant velocity along the flight path. The corrected distance was plotted versus climbout true airspeed for each takeoff.

Takeoffs were conducted at 2 field elevations to show that the ΔC_P method of takeoff analysis was valid for more than 1 altitude for this helicopter.

4.0 HOVER

The free-flight method of determining hovering performance was used. Using this method the helicopter was ballasted to obtain the desired thrust (weight). Skid height was determined by using a weighted cord attached to a skid. The cord length was changed to the correct length for each desired height. An observer on the ground continuously monitored the skid height and relayed the information so that the pilot could maintain the desired height. All hovering data were obtained in winds of 2 knots or less. Wind velocity was determined by reference to the smoke from a tire fire. In the YOH-6A winds of 2 knots or less were necessary to completely stabilize power and skid height. Rotor speeds throughout the allowable power on speed range were used. All data were reduced to non-dimensional terms (C_P and C_T) using ambient atmospheric conditions, rotor speed, weight, and test power.

5.0 CLIMBS

All climbs were flown at constant rotor speed, zero sideslip, and constant power setting. When significant winds were present, climbs were flown on a crosswind heading to minimize wind effect. An airspeed schedule determined primarily from level flight data at the speed for minimum power required was used. Since level flight data did not extend to service ceilings, it was necessary to extrapolate the airspeed schedule. During preliminary climbs airspeed was varied slightly ($\pm$ 5 knots) at absolute ceiling to determine what airspeed was optimum (highest ceiling) for a given weight. It was necessary to maintain each airspeed for approximately 1 minute so that zoom effects would be eliminated. It was also necessary to consider the effect of continuous decreasing weight due to fuel consumption. The airspeed schedule determined from level flight data was extrapolated to this optimum airspeed at absolute ceiling.

Test rate of climb was determined from pressure altitude variation with time and corrected for altimeter error caused by non-standard temperature using the following equation:

$$R/C_{test} = \frac{d\,H_p}{d\,t} \times \frac{T_{A\ test}}{T_{A\ STD}}$$

Where:

$T_{A\ test}$ = Test-day absolute temperature at pressure altitude

$T_{A\ STD}$ = Standard-day absolute temperature at pressure altitude

Test-day rate of climb was corrected for variations in power from test-day power available to standard-day power available at each density altitude using the following equation:

$$R/C_{power} = R/C_{test} + K_p \times 33000 \times \frac{SHP_{STD} - SHP_{test}}{G\,W_{test}}$$

Where:

SHP_{STD} = Standard-day shaft horsepower available determined from engine model specification 580-E corrected for installation losses.

SHP_{test} = Shaft horsepower available during the climb.

$G\,W_{test}$ = Test-day gross weight

K_p = Power correction factor determined from flight test (reference k).

Standard-day weight was determined by the following equation:

$$G\,W_{STD} - G\,W_{S.L.\ STD} - W_{f\ STD}\ \frac{\Delta H_D}{60 \times Avg\ R/C_{power}}$$

Where:

$G\,W_{S.L.\ STD}$ = Climb start gross weight at sea level on a standard day.

$W_{f\ STD}$ = Standard-day fuel flow determined from the engine model specification 580-E corrected for installation losses.

ΔH_D = Altitude increment

Avg R/C_{power} = Average power corrected rate of climb for altitude increment.

The power-corrected rate of climb was corrected for test-day deviations in weight to obtain standard-day rate of climb using the following equation:

$$R/C_{STD} = R/C_{power} + K_w \times 33000 \times \frac{SHP_{STD}\ (G\ W_{test} - G\ W_{STD})}{G\ W_{test} \times G\ W_{STD}}$$

Where:

K_w = weight correction factor determined from flight test (reference k).

After rate-of-climb standard had been obtained, time to climb, distance traveled, and fuel used were obtained using the following equations:

$$\text{Time to Climb} = \frac{\Delta H_D}{\text{Avg } R/C_{STD}} \quad \text{Minutes}$$

Where:

Avg R/C_{STD} = Average standard rate of climb for altitude increment

$$\text{Distance traveled} = \frac{\Delta H_D}{60 \times \text{Avg } R/C_{STD}} \quad V_T2 - (.9875 \times R/C_{STD})^2$$

Nautical Miles

$$\text{Fuel Used} = \frac{\Delta H_D}{60 \times \text{Avg } R/C_{STD}} \quad W_{f\ STD} \quad \text{Pounds}$$

Rate-of-climb standard, time to climb, shaft horsepower standard, distance traveled, weight, fuel used, calibrated airspeed, and true airspeed were plotted as a function of standard altitude and are presented in appendix I.

6.0 LEVEL FLIGHT

Speed power tests were conducted primarily to determine power requirements with speed variation at various weights and altitudes.

In addition to power required, fuel flow, N_1 speed, turbine outlet temperature, and engine inlet conditions were recorded so that engine performance could be compared to the engine model specification. From this information range, endurance, and power limit airspeeds could be determined. The tests were conducted in the clean configuration with the test boom on, with the test boom removed, and with the XM-7 armament kit installed so that the incremental drag of the test boom and the XM-7 armament kit could be determined. Each flight was conducted in non-turbulent air at constant rotor speed. The clean configuration and XM-7 test were flown at zero sideslip. The boom off tests were flown at zero bank since no sideslip reference was available.

Airspeed was varied in approximately 10 knot increments from the maximum attainable to the minimum for which airspeed could be determined accurately. Data was taken at each stabilized airspeed. As the speed power test was being flown, altitude was increased slightly for each point (50-200 feet) so a constant value of weight divided by ambient air density could be maintained as fuel was used. This was necessary to keep the non-dimensional variable C_T constant for each flight.

Data was reduced to the non-dimensional terms C_P, C_T, and μ and summarized. From this summary and the engine model specification a range summary was derived and power limit airspeeds were determined.

7.0 AUTOROTATION

Autorotational descents were conducted to determine the altitude for full down collective and if any gross weight-altitude conditions existed where the minimum rotor speed limit would be exceeded.

Stabilized autorotational descents were begun above the altitude at which full down collective could be reached without rotor overspeed. The descents were conducted at 55 knots CAS (airspeed for minimum rate of descent) with the engine operating at ground idle. Rotor speed was maintained at 514 rpm (maximum allowable rotor speed) until full down collective was reached. The descent was continued to minimum practical altitude. With the engine at ground idle, 450 rotor rpm was the minimum autorotational rotor speed. Below 450 rpm, the engine would deliver some power to the rotor and it was necessary to extrapolate rotor speed below 450 rpm. The tests were repeated at several different weights over the practical weight range of the YOH-6A. Results were reduced and summarized as rotor speed for full down collective versus density altitude for constant weights.

From this summary the variation of rotor speed at full down collective with weight and altitude could be determined. And, therefore, the weight and altitude combination could be determined where the minimum and maximum rotor speed limitations would be exceeded with full down collective.

8.0 POWER ON LANDINGS

An adequate method of determining power on landing performance which is both practical and lends itself to data acquisition, reduction, and presentation has not yet been determined. Landing performance determination would be similar to takeoff performance determination except the additional independent variables of descent power, rate of descent, and touchdown criteria are involved.

9.0 AUTOROTATIONAL LANDINGS (HEIGHT VELOCITY)

The test technique and analysis methods are presented completely in section 2.2.7.

10.0 SIDEWARD AND REARWARD FLIGHT

Sideward and rearward flight tests were conducted by stabilizing the helicopter in sideward or rearward flight and recording the required control positions. A ground vehicle with a calibrated speedometer was used as an aid in stabilizing the helicopter and as an airspeed reference. Tests were done in winds of less than 3 knots.

11.0 CLIMB INSTABILITY

Climb instability tests were made by conducting full power climbs at selected values of weight, airspeed, and sideslip angle within the conditions at which the instability occurred during the reference j tests. Time histories of aircraft attitude, airspeed and control positions were recorded.

12.0 LEVEL FLIGHT PITCHUP

The test technique and analysis methods are presented completely in section 2.3.3.

13.0 MANEUVERING STABILITY

The helicopter was trimmed at the desired airspeed in level flight. Zero sideslip descending turns were made at increasing bank angle and load factors (g's) at the trim airspeed. Turns were made both to the right and to the left. At each stabilized load factor, control positions and longitudinal control force were recorded. Load factors were obtained by holding collective fixed at the trim position and also by increasing collective pitch.

14.0 ENGINE ACCELERATION CHARACTERISTICS

The following three methods were used to determine engine acceleration characteristics:

a. Autorotational Method

The collective pitch setting required to obtain 110 percent of the power required to hover IGE was established for each weight and altitude condition. A descent was initiated 500 feet above the test altitude at a calibrated airspeed of 50 knots. Instrumentation was started 5 seconds prior to recovery. Collective pitch was then increased steadily to the position previously established to give 110 percent of the power required to hover IGE. Collective application rates were varied to give times of 3, 2, and 1 second.

b. Flare Method

This method was used to determine engine response characteristics in ground effect at speeds of less than 20 knots and with zero sink rate. The aircraft was rapidly decelerated from 70 knots near the ground with rotor speed being maintained at 470 and 480 rpm. When the aircraft started to sink, the attitude was leveled and collective pitch was applied in 3, 2, and 1 second. Data was taken for each flare.

c. Power Recovery

Power recoveries were made to determine the combined effects of the foregoing 2 methods. An autorotational approach was made with the throttle closed (ground-idle) and a flare made as the ground was approached. During the late stages of the approach, the throttle was opened and collective was applied as required to establish a hover. Data was taken for each recovery.

For each of these 3 methods minimum rpm droop was determined and plotted versus rate of collective application. Three rates of collective application did not yield sufficient data to completely define the acceleration characteristics. At least 6 collective application rates should be used during future tests.

15.0 ENGINE START

The test technique and analysis methods are presented completely in section 2.4.2.

16.0 AIRSPEED CALIBRATION

The standard airspeed system was calibrated by comparing the readings to a true source. A trailing bomb, calibrated in a wind tunnel, was suspended from the helicopter with an 80-foot cable to avoid proximity effects. The helicopter was then stabilized throughout its airspeed range in level flight, climb, and powered descents. By comparing the airspeed corrected for instrument errors of the standard system to the referenced bomb the systems position error was defined.

Appendix III GENERAL AIRCRAFT INFORMATION

1.0 SOURCES OF INFORMATION

The information contained in this appendix was obtained from the FAA Approved Flight Manual, the FAA Type Inspection Authorization, the preliminary Type Certificate, and directly from the engine and airframe manufacturers.

2.0 DESIGN DATA

2.1 Overall Dimensions

Length (rotors turning)	30 ft 3-3/4	in
Length (blades removed)	22 ft 9-1/2	in
Height (overall)	8 ft 1-1/2	in
Width (fuselage)	4 ft 6-1/4	in
Width (tread)	6 ft 9-1/4	in
Rotor diameter	26 ft 4	in

2.2 Weights

Empty weight	1070	lb
Design gross weight	2085	lb
Maximum overload gross weight	2700	lb

2.3 Main Rotor Design Data

Number of blades	4	
Diameter	26 ft 4	in
Blade chord (root to tip)	6.75	in
Blade airfoil	NACA 0015	
Blade twist	8-1/2	deg
Blade movement relative to centerline of mast		
(1) Collective pitch travel at .75 R	6-7	deg up
(2) Cyclic pitch	6-7	deg down
Longitudinal forward	16	deg
Longitudinal aft	8	deg
Lateral left	7.25	deg
Lateral right	6.25	deg

2.4 Tail Rotor Design Data

Number of blades	2	
Diameter	4 ft 3	in
Blade chord	4.81	in
Blade airfoil	NACA 0015 modified	
Control travel blade movements		
Left pedal	+27 (thrust to right)	deg
Right pedal	-12 (thrust to left)	deg
Swashplate	$\pm$0.63	in

2.5 Derived Data

(1) Main Rotor

Disk area (total swept area)	544.6	ft^2
Blade area (including hub)	29.6	ft^2
Solidity	.0544	
Disc loading at 2085 lb	3.83	lb/ft^2
at 2700 lb	4.96	lb/ft^2
Blade loading at 2085 lb	70.4	lb/ft^2
at 2700 lb	91.2	lb/ft^2
Power loading at takeoff power	(250 SHP)	
at 2085 lb	8.34	lb/HP
at 2700 lb	10.80	lb/HP
Rotor RPM at 6000 engine output RPM (100% N_2)	468.75	rpm
Average rotor tip speed at 6000 engine RPM	646.1	fps

(2) Tail Rotor

Disk area (total swept area)	14.2	ft^2
Blade area (excluding hub)	1.17	ft^2
Solidity	.0824	
Rotor RPM at 6000 engine RPM	3019.5	rpm
Average rotor tip speed at 6000 engine RPM	671.8	fps

3.0 FLIGHT LIMITATIONS

3.1 Engine and Transmission

(1) Power rating (sea-level standard day)

Takeoff (5-min limit)	250	SHP
Maximum continuous	212	SHP

(2) Output shaft torque (airframe transmission limited)

Takeoff (5-min limit at 6000 RPM)	219	ft-lb(91%)
Maximum continuous (at 6000 RPM)	180	ft-lb(77%)

(3) Turbine outlet temperature

Takeoff (30-min limit)	738	deg C (1360 deg F)
Maximum continuous	693	deg C (1280 deg F)
Maximum transient (0-10 sec)	738 to 927	deg C (1360 to 1700 deg F)

(4) Engine speed

Gas producer (N_1)

Maximum	102	%
Minimum	55	%

Power turbine (N_2)

Power-on maximum	103	%
minimum	99	%
Transient (15 sec)		
Takeoff power	105	%
Flight idle	110	%
Power-off normal	97	%

3.2 Rotor

(1) Design maximum

Power-on	484	rpm
Power-off	514	rpm

(2) Design Minimum

Power-on	465	rpm
Power-off	400	rpm

3.3 Airframe

(1) Loading

Design weight	2085	lb
Overload weight	2700	lb
Maximum forward C.G.	Station 97	
Maximum aft C.G.	Station 104	
Maximum lateral C.G. (2085 lb)	± 3	in
Maximum lateral C.G. (2700 lb)	± 5	in
Maximum cargo loading (Station 78.5 to 125)	130	lb/ft^2

(2) Maximum load factor (2085 lb)

Power-on	+ 2.58	g
Power-off	+ 2.91	g

3.4 Airspeed Limitations

(1) Forward flight

Density Altitude	Sea Level	2000	4000	6000	8000	10,000	12,000
2100 lb V_{ne} KIAS	128	125	117	109	99	89	79
V_{dive} KIAS	142	138	130	121	110	99	88
2700 lb V_{ne} KIAS	111	100	90	82	74	67	60
V_{dive} KIAS	123	111	100	91	82	74	56

(2) Sideward and rearward flight

Weight	Sideward	Rearward
2085 lb	35 KIAS	35 KIAS
2700 lb	10 KIAS	10 KIAS

3.5 Sideslip Limitations

Airspeed, KIAS	30	40	60	80	100	110	120
Sideslip angle, deg	90	73	46	38	16	12	9

4.0 AIRCRAFT SYSTEMS

4.1 Power Plant

The YOH-6A is powered by an Allison T-63-A-5 free turbine engine rated at 275 SHP which incorporates a pneumatic fuel control system which provides automatic speed governing, acceleration control, altitude compensation, and temperature compensation. The drive to the rotor is through a single gearbox which is rated at 250 SHP at 100-percent N_2 (6000 rpm). The tail rotor drive is taken off the same gearbox.

The engine is a free turbine type. The compressor consists of 6 axial stages and 1 centrifugal stage. Compressor speed at 100 percent is 51,120 rpm. The combuster section consists of a single chamber into which a regulated flow of fuel is injected to support continuous combustion. The power turbine has 2 axial stages. Power turbine speed at 100 percent is 35,000 rpm. The high speed of the power turbine is reduced in the accessory gearbox to 6000 rpm for the engine output speed. Engine operated accessories are also driven from the accessory gearbox.

The DP-D3 gas turbine fuel control is pneumatically operated by compressor discharge air. The fuel control senses input from 3 sources. These sources are the pilot's twist grip, the flyball governor connected to the gas producer, and the power turbine governor. In addition, both altitude and temperature compensation are provided. The function of the fuel control is to integrate the inputs so that the power turbine speed selected by the pilot is maintained under varying load demands.

4.2 Control System

The flight control system used on the YOH-6A helicopter is the conventional stick and pedal type. The flight control system consists of the collective stick, the cyclic stick and the pedals. Movement of the collective pitch control changes the angle of attack on all 4 blades by means of the collective control push rod. Forward and aft movement of the cyclic stick provides longitudinal control by tilting the swashplate forward or backward, which in turn, causes a complete cyclic pitch change per rotor revolution. Control of the tail rotor thrust is accomplished by means of control rods and bellcranks that are connected to the pedals.

4.3 Fuel System

Two fabric-reinforced rubber fuel cells are located under the flooring in the passenger-cargo compartment and have a useable capacity of 382 pounds. The fuel system consists of a boost pump, electrically actuated low pressure out-off valve, fuel filter and engine-driven fuel pump. Range extension torso tanks can be fitted.

4.4 Electrical Systems

A 28-volt, 65-ampere-hour nickel-cadmium battery provides D.C. power for all electrical services, including engine starting, which are protected by circuit breakers on the center console. Generator power is controlled by a master electrical selector switch and provision is made, under the left seat and inside the aircraft, for an external power receptacle. In-flight electrical power is provided by a 28-volt, 100-ampere generator driven by the engine and controlled by a voltage regulator. A load meter is provided in the cockpit and the battery can be isolated by moving the master control switch from "NORMAL" to "GENERATOR."

4.5 Landing Gear

A skid landing gear is fitted with 4 air/oil damped shock struts. Ground handling wheels are also provided.

5.0 WEIGHT AND BALANCE

The test aircraft was weighed with all test instrumentation installed. The weighing was done in a closed hangar using an electronic weighing kit. As weighed, the aircraft basic weight (full oil and trapped fuel) was 1360 pounds with the longitudinal C.G. located at station 105.9 inches and the lateral C.G. .4 inches left of the aircraft centerline. Ballast was used to obtain the desired gross weight and C.G. location for the flight tests.

In September 1965, an error in the method of determining longitudinal C.G. was discovered. The aircraft had a decal on the side with a longitudinal station number and a hash mark on it. It was interpreted that the station number referred to the hash mark on the decal. This information was used to determine longitudinal C.G. during weighings. It was found that the station number on the decal referred to the leveling target on the cargo compartment floor. This discrepancy resulted in a difference of 3.5 inches in longitudinal C.G. at a basic weight of 1360 pounds. All longitudinal C.G. locations presented in this report have been corrected.

The erroneous method of determining longitudinal C.G. was used during the original YOH-6A Program (references j and K). Therefore, the longitudinal C.G.'s presented in references j and k are incorrect. An approximate correction can be made by using the following method:

$$\text{Approximate True C.G.} = \text{Presented C.G.} - \left(\frac{\text{Basic wt}}{\text{Presented wt}}\right) \times (\text{erroneous basic C.G.} - \text{true basic C.G.})$$

Therefore for YOH-6A S/N 62-4212 (reference k tests)

$$\text{Approximate True C.G.} = \text{Presented C.G.} - \left(\frac{1360}{\text{Presented wt}} \times 3.5\right)$$

And for YOH-6A S/N 62-4214 (reference j tests)

$$\text{Approximate True C.G.} = \text{Presented C.G.} - \left(\frac{1171}{\text{Presented wt}} \times (111.72 - 107.92)\right)$$

or

$$\text{Approximate True C.G.} = \text{Presented C.G.} - \left(\frac{1171}{\text{Presented wt}} \times 3.8\right)$$

The decal mentioned could cause a hazardous situation to exist (flight beyond the forward C.G. limit). It should not be put on the production OH-6A.

6.0 TEST INSTRUMENTATION

The test instrumentation used during this evaluation was supplied, installed, and maintained by the Logisitics Division of the U.S. Army Aviation Test Activity. A swivel-mounted pitot-static airspeed head was installed on a nose boom mounted approximately 5 feet forward of the nose of the helicopter. The static pressure ports of this pitot-static head were the pressure source for the sensitive altimeter as well as the sensitive boom airspeed indicator.

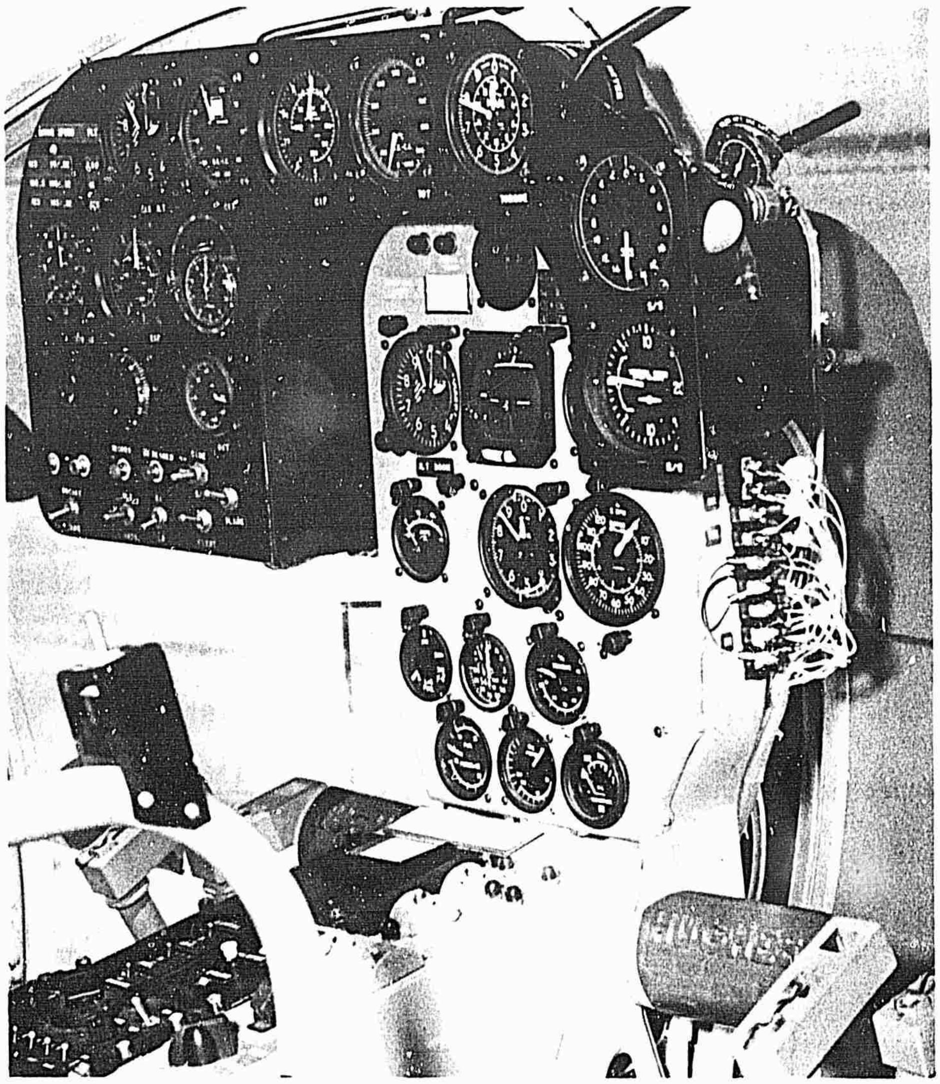

Photo-5 YOH-6A INSTRUMENTATION PANEL

The airspeed position error for this installation is shown in figure 58, appendix I. Sensitive instrumentation was installed prior to initiation of the test flights to measure the following parameters:

a. <u>Pilot-Engineer Panel</u>

Boom System Airspeed
Standard System Airspeed
Boom System Altitude
Angle of Sideslip
Free Air Temperature

Rotor Speed
Gas Producer Speed (N_1)
Torquemeter Oil Pressure
Turbine Outlet Temperature
Compressor Inlet Total Pressure
Compressor Inlet Total Temperature
Cockpit Absolute Pressure (reference)
Total Fuel Used
Fuel Flow
Collective Stick Position
Cockpit Normal Acceleration
Photo Panel Frame Counter
Oscillograph Record Counter

b. Photo Panel

Boom System Airspeed
Boom System Altitude
Free Air Temperature
Rotor Speed
Gas Producer Speed (N_1)
Power Turbine Speed (N_2)
Torque Oil Pressure
Compressor Inlet Pressure
Total Fuel Used
Time
Collective Stick Position
Throttle Position
Fuel Flow Rate Indicator
Photo Panel Frame Counter
Oscillograph Record Counter

c. Recording Oscillograph

Longitudinal Stick Position
Lateral Stick Position
Pedal Position
Collective Stick Position
Throttle Position
Pitch Attitude
Roll Attitude
Yaw Attitude
Pitch Rate
Roll Rate
Yaw Rate
Angle of Attack
Angle of Sideslip

Power Turbine Speed
Rotor Speed
Torque Pressure
Voltage Monitor
Engineer Event Marker
Pilot Event Marker

[redacted]

Appendix IV MAINTENANCE REQUIREMENTS

1.0 GENERAL

In general, relatively little maintenance was required by the YOH-6A. No unscheduled maintenance was required during the first 6 months of this program. The majority of the maintenance was required by the engine. Airframe maintenance was small because of its lack of hydraulic or stability augmentation systems, the use of bearings not requiring periodic lubrications, and its general simplicity.

2.0 ENGINE CLEANING

It was necessary to clean the engine compressor approximately every 30 hours of flying time to maintain specification engine power. Flying was done in apparent dust-free conditions. On the original engines with the "wide" centrifugal compressor shroud, the engine had to be removed, cleaned, and reinstalled. This took 2 men approximately 8 hours under field conditions. On later engines with the narrower shroud, the technique using brouline (high grade detergent) and water was successful in cleaning the compressor. Two or 3 cycles of pouring approximately 1 pint of brouline solution into the engine inlet, motoring the engine for 30 to 60 seconds, and then rinsing the brouline out with water were necessary to clean the compressor. The solution was allowed to drain out the combuster drain hole at all times. This procedure required approximately 30 minutes.

The engine manufacturer reports that the T-63-A-5A engine, to be used on the production OH-6A, will have greatly reduced or no compressor cleaning requirements.

3.0 COMPRESSOR DISCHARGE PRESSURE FILTERS

It was necessary to change compressor discharge pressure filters (CDP filters) every 5 to 10 hours flying time. This took approximately 10 to 15 minutes to accomplish. CDP filters were changed whenever acceleration time from ground idle to 100 percent N_2 exceeded the 7 seconds specified in Engine Model Specification 580-E.

The T-63-A-5A engine does not have a compressor discharge pressure filter.

[redacted]

4.0 POWER TURBINE SPEED SELECTOR SWITCH

Continual problems were experienced with the power turbine speed selector switch (N_2 Beep Switch). A combination of complicated circuitry, excessive switch clearances, and weak or broken switch springs resulted in the switch's actuating itself at times or when operated by the pilot moving in the opposite direction to the desired direction. This made it difficult to set or maintain power turbine speed accurately. This problem was lessened slightly by replacing the switch springs when they became broken or weak. One of two relays used in the N_2 speed select circuit failed during this program. Because of its inaccessability, it took approximately 2 days to replace.

The power turbine speed selector switch in the production OH-6A has been changed to the conventional, thumb-operated, direct-power type similar to those found on other turbine engine powered helicopters.

5.0 ENGINE CHANGES

Three T-63-A5 engines were used during this program. Engine serial number 400037 was replaced after cracks in the plastic compressor liner were found, during one teardown for cleaning. Engine serial number 400067 was installed. During an engine start, a chip detector warning light was noted. Large metallic chips were found on one of the engine chip detector plugs. A teardown was not accomplished but it was suspected that the main power turbine bearing had failed. Engine serial number 400061 was installed and no problems were encountered with this engine.

6.0 FUEL CONTROLS

Several different fuel controls were used during this program in an attempt to improve starting characteristics. Each fuel control has been bench-flowed at the manufacturer's facility prior to installation. Each fuel control exhibited slightly different starting characteristics although generally no improvement in starting was noted.

Appendix V REFERENCES

a. Letter, AMCPM-LH-T, U.S. Army Materiel Command (USAMC), 30 November 1964, subject: "Continuation of the OH-5A and OH-6A Testing," with Inclosure: "Performance Test Directive of the OH-5A and OH-6A."

b. Letter, AMSTE-BG, Hq, U.S. Army Test and Evaluation Command (USATECOM), 14 January 1965, subject: "Test Directive, USATECOM Project No. 4-3-0250-78, Continued Military Potential Testing (Handbook Data), Light Observation Helicopters (LOH)."

c. Coordinated Plan of Test, USATECOM Project No. 4-3-0250-78: "Plan of Test for Completion of Engineering Flight Tests of the OH-6A Helicopter," USAAVNTA, January 1965.

d. Letter, AMCPM-LHT, Hq, USAMC, 8 March 1965, subject: "Proposed Engineering Plan of Test of Completion of Engineering Flight Tests of the OH-5A and OH-6A."

e. Letter, AMCPM-LH, Hq, USAMC, 6 April 1965, subject: "Proposed Engineering Plan of Test of Completion of Engineering Flight Tests of the OH-6A."

f. Type Inspection Authorization No. CH-1205-4DM, Federal Aviation Agency, 30 January 1964.

g. Letter, AMCPM-LHFO-M, Hq, USAMC, 3 August 1965, subject: "Designation of OH-6A Observation Helicopter to YOH-6A."

h. Engineering Order (E.O.) 369-2264, "Installation of Cooling Air Scoops," Hughes Tool Company, Aircraft Division.

i. E.O. 6D-369-2504, "Modified Horizontal Stabilizer Assembly," Hughes Tool Company, Aircraft Division.

j. Report, U.S. Army Test and Evaluation Command (USATECOM) Project No. 4-3-0250-51/52/53, Part One of Two Parts of the "Engineering Flight Test, Stability and Control Phase, of the OH-6A Helicopter, Unarmed (Clean) and Armed with the XM-7 or XM-8 Weapon Subsystem," U.S. Army Aviation Test Activity, August 1964.

[redacted]

k. Report, U.S. Army Test and Evaluation Command (USATECOM) Project No. 4-3-0250-51/52/53, Part Two of Two Parts of the "Engineering Flight Test, Performance Phase, of the OH-6A Helicopter, Unarmed (Clean) and Armed with the XM-7 or XM-8 Weapon Subsystem," U.S. Army Aviation Test Activity, August 1964.

l. Operator's Flight Manual, Hughes 369 (Army Model OH-6A) Helicopter FAA Approved Flight Manual, approved June 30, 1964.

m. Operator's Manual, POMM 1520-214-10, "Operator's Manual, Helicopter Observation OH-6A (Hughes)," March 1966.

n. Military Specification MIL-H-8501A, "General Requirements for Helicopter Flying and Ground Handling Qualities," 7 September 1961.

c. Model Specification No. 580-E, Engine Model T63-A-5, Allison Division of General Motors, 24 June 1963.

p. Letter Report, U.S. Army Aviation Test Activity, subject: "Confirmatory Hovering Tests of the OH-6A," USATECOM Project No. 4-3-0250-51/52/53, December 1964.

q. Letter, USAAVNTA, subject: "Review of Proposed OH-6A Handbook Performance Data," 26 May 1966.

[redacted]

Appendix VI SYMBOLS and ABBREVIATIONS

SYMBOL	DEFINITIONS	UNIT
TAS (Vt)	True Airspeed	knots
CAS (Vc) (Vcal)	Calibrated Airspeed	knots
KCAS	Knots Calibrated Airspeed	knots
KIAS	Knots Indicated Airspeed	knots
V_{ne}	Never Exceed Airspeed	knots
V_{max}	Maximum Airspeed Attainable	knots
V_D	Maximum Permissible Dive Speed	knots
OGE	Out of Ground Effect	
IGE	In Ground Effect	
C.G.	Center of Gravity	inches
GW	Gross Weight	pounds
RPM	Revolutions per Minute	rpm
°C	Degrees Centigrade	degrees
°F	Degrees Fahrenheit	degrees
SL	Sea Level	
SHP	Shaft Horsepower	
R/D	Rate of Descent	feet per minute
R/C	Rate of Climb	feet per minute
T/C	Time to Climb	minutes

SYMBOL	DEFINITION	UNIT
C_p	Power Coefficient	
C_T	Thrust Coefficient	
NAMT	Nautical Air Miles Traveled	
NAMPP	Nautical Air Miles Per Pound of Fuel	
N_1	Gas Producer Speed	percent rpm
N_2	Power Turbine Speed	percent rpm
Hd	Density Altitude	feet
H_p	Pressure Altitude	feet
Tt_5 (TOT)	Turbine Outlet Temperature	degrees
ρ (rho)	Air Mass Density	$\frac{lb\text{-}sec^2}{ft^4}$

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13 ABSTRACT

As a result of an Army Study Requirement, a light observation helicopter design competition was initiated, prototypes were built and tested, a selection was made, and a production contract was awarded. Due to changes to the helicopter configuration required for type certification by the Federal Aviation Agency, additional testing was accomplished by the Army to provide data for the initial operator's manual. Tests were conducted near Bishop, Bakersfield, and Edwards Air Force Base, California. Test sites used varied in field elevation from sea level to 11,500 feet. A total of 161 flights, accounting for 172.5 flight hours, were made from 27 January to 30 March 1966. At design weight (2085 pounds), the YOH-6A had a significant increase in cruising airspeed and maximum airspeed compared with previous observation helicopters. Its climb performance was also good. Although the YOH-6A exhibited some handling quality shortcomings, it was described by the pilots as being very "agile." Takeoff and hovering performance were only fair and decreased rapidly with increasing weight or ambient temperature. No deficiencies were found which would prohibit further testing or release for service use. Correction of the shortcomings, however, would increase the flight safety and improve mission effectiveness of the YOH-6A.

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