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FINAL REPORT

USATECOM PROJECT NO. 4-4-0180-01

ENGINEERING FLIGHT TEST OF THE CH-37B

April 1964

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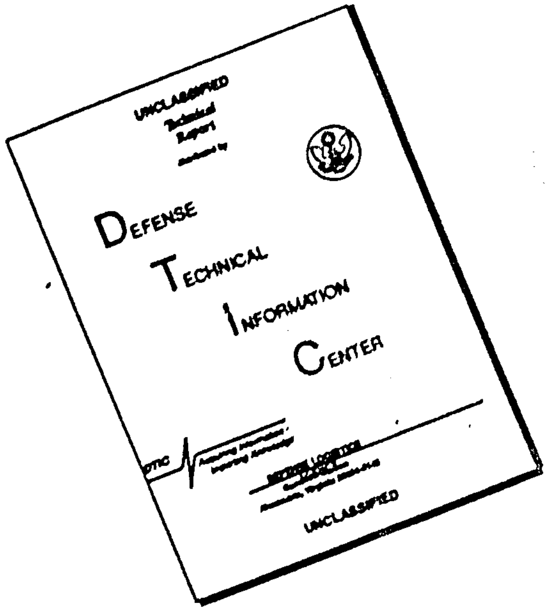

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FINAL REPORT

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ENGINEERING FLIGHT TEST

OF THE

CH-37B

April 1964

Robert N. Turk
Project Engineer

Joseph C. Watts
CWO, U. S. Army, TC
Project Pilot

ENGINEERING FLIGHT TEST
OF THE CH-37B

TABLE OF CONTENTS

SUMMARY

Engineering tests of the CH-37B were conducted to determine the effect of the changes from the CH-37A configuration on the performance and stability and control characteristics of the helicopter.

This report presents the results of a 29 hour flight test evaluation conducted during the period 5 September 1962 through 27 May 1963.

The test vehicle (U. S. Army S/N 54-0998) had been remanufactured from a CH-37A into a CH-37B by the incorporation of the following major changes:

(1) Installation of Automatic Stabilization Equipment (ASE).

(2) Relocation of the horizontal stabilizer to a position opposite the tail rotor.

(3) Installation of larger capacity oil tanks.

(4) Replacement of the split cargo door with a sliding cargo door.

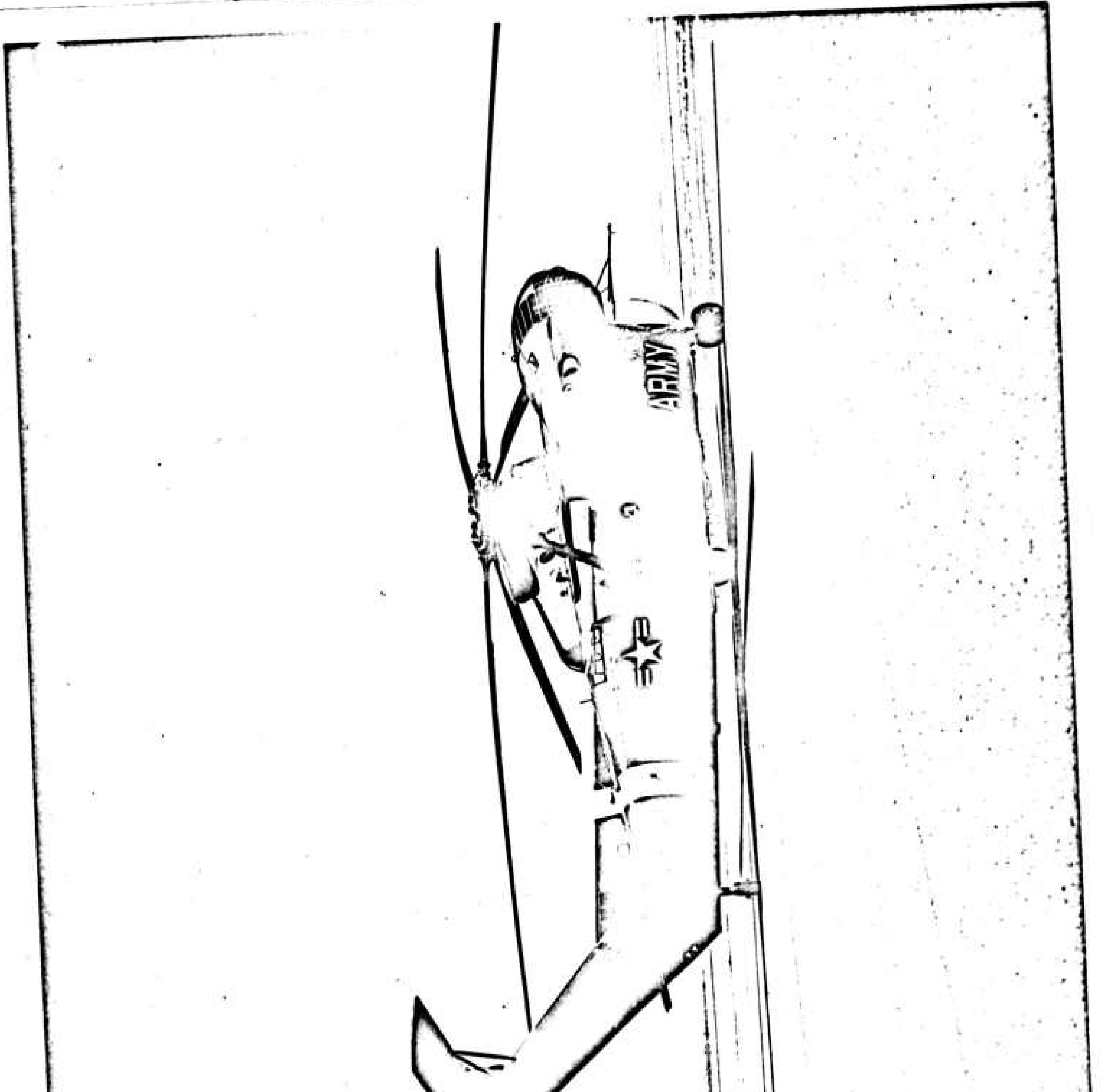
The performance data obtained during this test was compared with that presented in "Limited Evaluation of the H-37A Equipped With Wide Chord Blades" (AFFTC-TR-59-14) and the stability and control comparison is made with the "H-37A, Limited Stability and Control Evaluation" report (AFFTC-TR-60-15). Performance data comparison revealed that no significant differences exist which would make necessary revisions of the Operator's Handbook. This conclusion was based on level flight tests.

Not considering the ASE modification, the major stability and Control difference is experienced in the longitudinal axis. A larger degree of damping is present, and the short period oscillations of the CH-37B are approximately one-half those of the CH-37A. This change is attributed to the relocation of the horizontal stabilizer.

Laterally and directionally, there is no appreciable difference. Directional control response in a hover remains excessive.

The ASE improves the handling qualities of the helicopter to the extent that rate responses and attitude changes are subject to increased damping. This is accomplished without a reduction in control sensitivity.

The addition of the Automatic Stabilization Equipment is a definite improvement and improves the controllability of the aircraft while reducing pilot fatigue.


ARMY

PART I - GENERAL

ATA-TR-64-5
FINAL REPORT
USATECOM PROJECT NO. 4-4-0180-01
ENGINEERING FLIGHT TEST
OF THE
CH-37B
5 Sep 62 to 27 May 63

GENERAL

A. REFERENCES

A list of references will be found in Part III, Annex A.

B. AUTHORITY

This program was authorized on 11 May 1962 by the Commanding General, U. S. Army Aviation and Surface Materiel Command, St. Louis, Missouri, by means of an electrical message numbered TCMAC-EH-37-050-01249. The Directive, in part, stated, "By Reference 1", (Message TCMMD-AB-5-28-2 from DA, dated 8 May 62), "authority has been received to conduct a twenty (20) flying hour program on H-37B, S/N 54-998, to evaluate stability, control and performance changes resulting from changes incorporated during the H-37 remanufacturing program."

C. DESCRIPTION OF MATERIEL

1. The CH-37B is a twin-engine, single lifting rotor, all-metal transport helicopter manufactured by Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut.

2. This model aircraft results from a modification of the CH-37A. The following items outline the major changes incorporated in this program.

(a) Automatic Stabilization Equipment (ASE).

This equipment was designed and installed to improve the handling characteristics of the helicopter.

(b) Fixed Stabilizer.

The adjustable stabilizer located on the sides of the fuselage have been replaced with a fixed stabilizer located on the right side of the tail rotor pylon.

(c) Cargo Door.

A two section sliding door replaces the three section swing-out door of the CH-37A.

(d) Oil Tank.

A larger, rigid tank replaces the bladder-type tank of the CH-37A.

3. A detailed description of the CH-37B is presented in Annex D, Part III.

D. BACKGROUND

The CH-37A, S/N 54-998, arrived at Edwards Air Force Base, California, on 30 January 1957. It was returned to Sikorsky for modification and remanufacture on 17 October 1960. Major changes included the addition of Lear Automatic Stabilization Equipment and repositioning of the horizontal stabilizer.

On 14 July 1961, it was returned for testing to Edwards Air Force Base as a "B" Model of the CH-37. The U. S. Air Force Flight Test Center (AFFTC), Edwards AFB, California was requested to conduct an evaluation of changes due to modification in stability, control and performance characteristics of the "B" model. The aircraft, however, was reassigned to the U. S. Army Aviation Test Activity (formerly TMCATO) for testing after five hours of flight time conducted by the AFFTC.

Flight testing began on 5 Sep 1962 at Edwards AFB, California. Subsequent testing to 27 May 1963 was conducted at Meadows Field Airport, Bakersfield, California and Edwards AFB.

E. TEST OBJECTIVES

This test program was initiated for the purpose of evaluating the stability, control and performance differences resulting from changes incorporated during the H-37 remanufacturing program. This is as stated in the directive quoted in part in Section B.

F. FINDINGS

1. Performance

Level flight power required as a function of airspeed was determined at density altitudes from 5035 feet to 10,130 feet. Gross weights ranged to 24,290 pounds to 30,820 pounds and engine and rotor speeds were maintained at 2600 and 185.5 rpm respectively.

Individual test results are presented in Figures No. 3 through 6, Part II. Nondimensional summary plots are presented in Figures No. 1 and 2, Part II. The CH-37B fuel consumption versus shaft horsepower curves are presented in Figure No. 77, Part II. The manufacturer's engine performance curves are presented in Figures No. 78 and 79, Part III.

A comparison of the CH-37B test results with the findings in the report AFFTC-TR-59-14, "Limited Evaluation of the H-37A Equipped with Wide Chord Blades", indicates a small difference in level flight performance.[1] The shapes of the CH-37B speed-power curves are similar to those of the CH-37A. However, there is a displacement of the curve as illustrated in Figure A.

FIGURE A

(See next page)

The CH-37B apparently required greater power at high airspeeds and less power at low airspeeds than the CH-37A. The displacement of the curve may be attributed to any or all of the following:

(a) Airspeed Calibration. The airspeed calibrations used during the two test programs differed. The CH-37A airspeed data were obtained from the standard ship's system while the CH-37B was evaluated with airspeeds obtained from a test (boom) system. The different position errors from the calibrations resulted in CH-37A calibrated airspeeds which were generally 1 to 2 knots higher.

(b) Fuel Flow. The CH-37B evaluation was conducted with relatively high time engines (approximately 150 and 350 hours), and carburetors which had not been flow checked. An analysis of the fuel flow data shows that for the same horsepower output, these engines required a much greater fuel flow than the engines used in the CH-37A test aircraft.

(c) Method of Measuring Power. The engine power data for this report was obtained by using the manufacturers engine performance curves (power chart). The CH-37A power was determined from torquemeter data.

(d) Drag. Drag, while probably the least likely cause of the power-airspeed differential, should nevertheless be considered. The external configuration changes made to the CH-37A which could create a difference in drag are:

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[1] The method utilized in obtaining comparative values is outlined in "Data Analysis Methods and Test Techniques", Annex B, Part III.

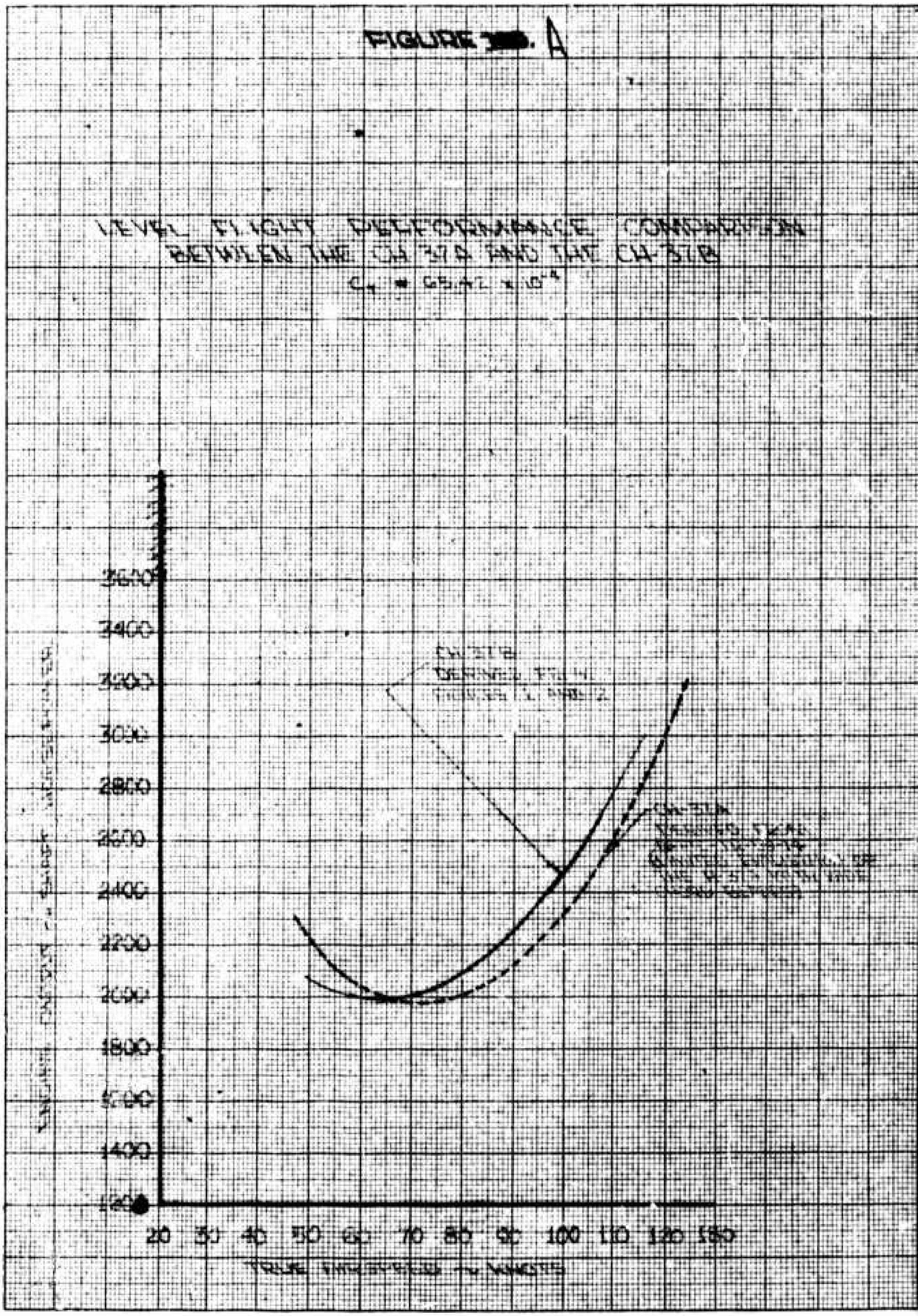


FIGURE A
LEVEL FLIGHT PERFORMANCE COMPARISON
BETWEEN THE CH-37A AND THE CH-37B
3600
3400
3200
3000
2800
2600
2400
2200
2000
1800
1600
1400
1200
CH-37B
CH-37A
20 30 40 50 60 70 80 90 100 110 120 130

(1) Relocation of the aft horizontal stabilizer

(2) Conversion of the side door

The frontal area of the H-37B horizontal stabilizer was reduced approximately one-half while the new sliding door protruded slightly into the airstream and may have added a small amount of drag. Thus, it appears that the drag difference between the two aircraft would be so small as to have little or no effect on the power required.

Consideration of the above indicates that the actual performance difference between the two configurations if less than 5 percent. Thus, no changes in the Operator's Manual are necessary, (Reference MIL-M-7700A, Paragraph 3.1.2.12.2; changes will be necessary if alternate configurations result in a performance variation of more than 5 percent).

2. Static Stability

a. General

The static longitudinal speed stability of the CH-37B was evaluated by recording the longitudinal, lateral and rudder pedal control positions required to vary airspeed about given trim conditions. These control positions were recorded during climb, level flight, and autorotation. The test conditions are presented in the following Table:

TABLE I

FLIGHT REGIME	DENSITY ALTITUDE -FT	GROSS WEIGHT -LB	ROTOR SPEED -RPM	C.G. LOCATION	TRIM AIRSPEED -KNOTS*
Climb	6290	30,710	192	242 (AFT)	65
Level Flt	4430	31,320	186	242 (AFT)	35,80,99
Autorotation	6290	30,710	200	242 (AFT)	65,90

* All airspeeds in the Stability and Control Sections of this report are calibrated airspeeds unless otherwise noted.

Test results are presented in Figures No. 7 and 8, Part II.

Static directional stability tests were conducted to determine the amount of pedal and lateral control required to maintain a stabilized angle of sideslip. Level flight and autorotation test results are presented and compared to CH-37A test results (Reference AFFTC-TR-60-15 "H-37A Limited Stability and Control Evaluation") in Figures No. 9 through 14, Part II. The test level flight trim airspeeds were 45 and 85 knots. The average density altitude was 5360 feet, the average gross weight was 30,770 pounds, the test rotor rpm was 186, and the center-of-gravity was located aft at Station 242 (C.G. limits of the CH-37B are from Station 228 forward to Station 245.1 aft). During autorotation the rotor speed was increased to 191 rpm and the test trim airspeeds were 42, 53 and 57 knots calibrated airspeed.

Low speed forward and rearward flight in-ground-effect was conducted to obtain data for analysis of the hovering characteristics during headwinds or tailwinds. Sideward flight tests were not conducted because there were no configuration differences which would indicate a change in characteristics for this flight regime. The tests were conducted at a wheel height of approximately 50 feet, a density altitude of 1900 feet, a rotor speed of 193 rpm, an engine speed of 2700 rpm, a c.g. at Station 236.5 (mid), and a gross weight of 30,020 pounds.

b. Static Longitudinal Speed Stability

Longitudinal speed stability was determined to be positive for all airspeeds above 45 knots at all flight conditions. There were no objectionable discontinuities and the speed stability became more positive as airspeed was increased. Qualitative pilot comments indicated that the static speed stability characteristics were the same with the Automatic Stabilization Equipment (ASE) both "on" and "off". Figure B compares the CH-37A and CH-37B longitudinal control positions as a function of calibrated airspeed.

<u>FIGURE B</u>

(see next page)

The CH-37B required an average of seven percent more forward longitudinal cyclic control and the speed stability gradient became more positive as the airspeed was increased. This forward control position and the more positive stability was attributed to the greater moment created by the new stabilizer location. Qualitatively, there was no detectable difference in static longitudinal stability between the CH-37A and the CH-37B.

c. Static Directional Stability

The CH-37B exhibited positive static directional stability in level flight. The tests results indicated increasing positive stability gradients as airspeed was increased. ASE operation did not

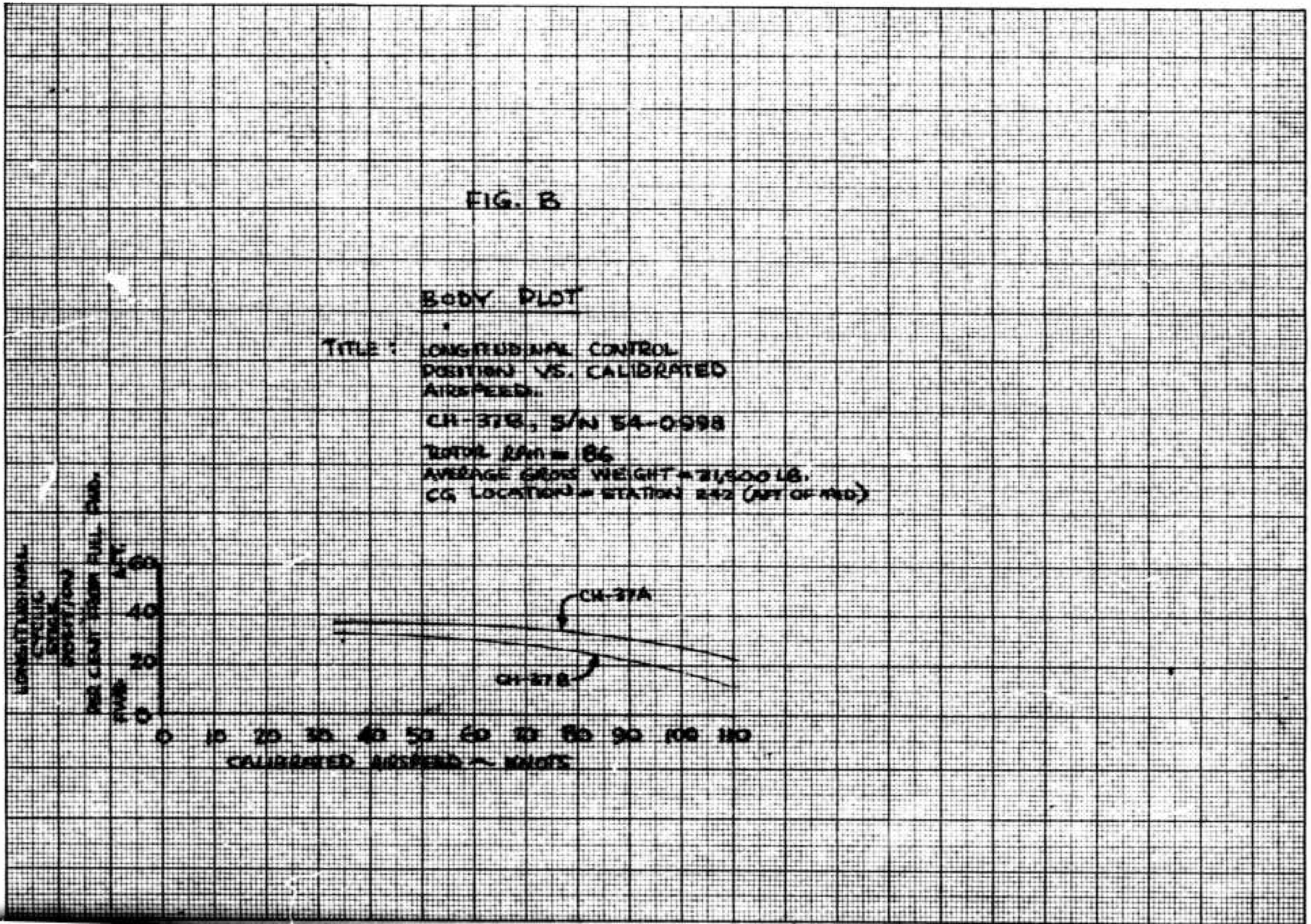


FIG. B
BODY PLOT
TITLE: LONGITUDINAL CONTROL POSITION VS. CALIBRATED AIRSPEED.
CH-37B, S/N 54-0998
ROTOR RPM = 86
AVERAGE GROSS WEIGHT = 21,500 LB.
CG LOCATION = STATION 242 (AFT OF RED)
CH-37A
CH-37B
LONGITUDINAL CYCLIC STICK POSITION PER CENT FROM FULL FWD.
AFT
FWD.
0
20
40
60
0 10 20 30 40 50 60 70 80 90 100 110
CALIBRATED AIRSPEED ~ KNOTS

have a significant influence on the static directional stability. Extrapolation of the test data indicated that at least a ten percent control position margin existed for stabilized sideslip angles of 45 degrees at low speeds and 15 degrees at high speeds.

The static directional stability in autorotation was determined to be weakly positive for airspeeds of 42 to 57 knots with only 0.16 inches of pedal input required to change the sideslip angle from 15 degrees left to 15 degrees right. Maintaining a stabilized sideslip angle in autorotation was extremely difficult. A "wallowing" motion was experienced and directional control inputs had little effect on yaw attitude. These tests results were similar to those reported during the CH-37A tests. As in the level flight case, no significant difference was noted between the ASE "on" and "off" data, however, qualitative pilot comments indicated that stability was improved with the ASE operative and the directional flying qualities were better. Sufficient directional control was available to produce sideslip angles similar to those obtained in level flight.

Dihedral effect, as indicated by lateral control positions during steady sideslip, was positive at all speeds in level flight and in autorotation above 50 knots. During level flight, dihedral effect increased with airspeed and was the same with the ASE both "on" and "off". This positive dihedral effect, coupled with the strong static directional stability, gave the helicopter good pedal fixed maneuvering capability in level flight. During autorotation, the dihedral effect was weakly positive for airspeeds above 50 knots and was neutral or negative below 50 knots. The rotor rpm ranged from 190 to 192. This weak or neutral dihedral effect, in addition to the marginal static directional stability, made it extremely difficult to maneuver or maintain yaw attitudes during autorotational descents. Turbulence increases the pilot effort and might prevent adequate maneuvering required for a safe autorotational landing.

d. Control Positions in Forward and Rearward Flight

Forward and rearward flight test results are illustrated in Figure No. 15, Part II. Notable differences were found between the CH-37A and CH-37B collective and pedal positions. This was mainly attributed to the CH-37A tests being conducted at a rotor rpm of 186 as compared to 193 for the CH-37B. Some variation was also attributed to the change in the location of the horizontal stabilizer. More than 30 percent of the aft longitudinal stick travel remained at 30 knots in rearward flight. There were no unusual lateral or directional control requirements as rearward flight speed increased up to 30 knots, TAS.

e. Miscellaneous

It should be noted that the data for the report AFFTC-TR-60-15, H-37A Limited Stability and Control Evaluation, were obtained from CH-37A, S/N 54-0998, while that helicopter was fitted with wide chord plates, but still rigged for narrow chord blades. In essence this means the comparative CH-37A control position curves presented on the static stability plots in this report may vary somewhat from curves that would have resulted had the blades been rigged in the wide chord configuration as they were in the CH-37B test aircraft. Static longitudinal speed stability, static directional stability, and control positions in forward and rearward flight are the plots affected. Performance data and dynamic stability plots are not influenced by this rigging difference. The blade pitch variations are as follows:

TABLE II

BLADE READINGS

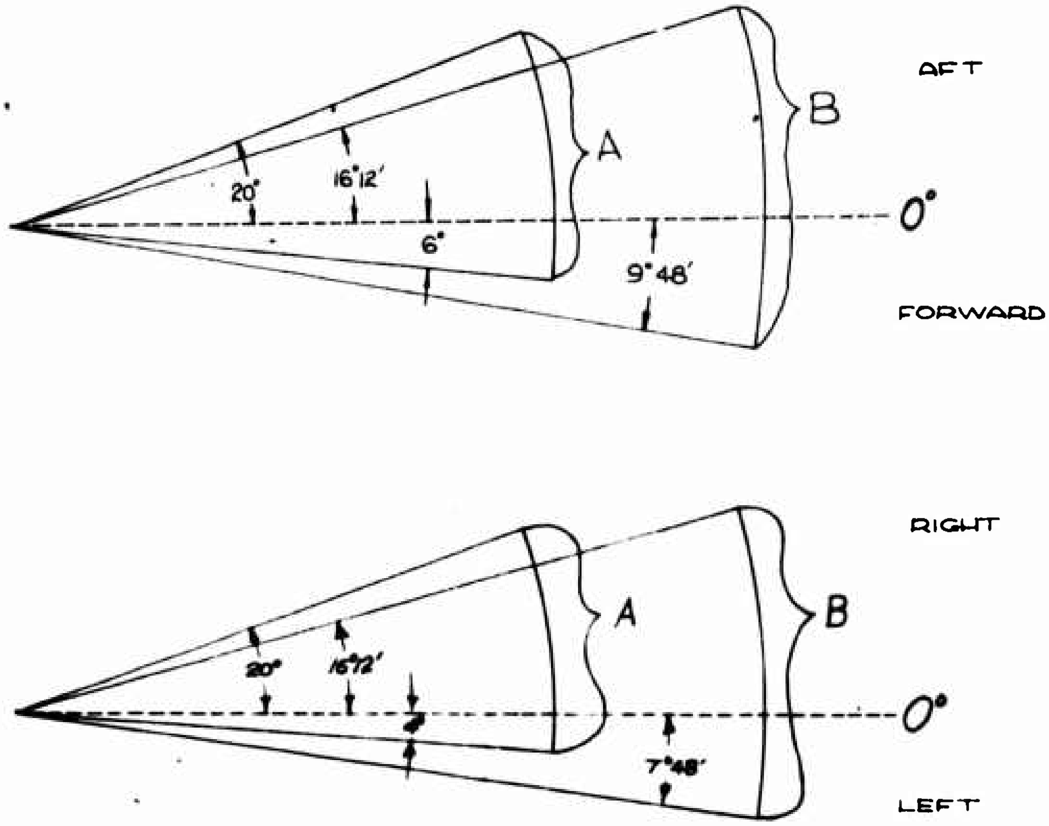
	A - Narrow	B - Wide
Left	-4° ± 1°	-7° 48' ± 1°
Right	20° ± 1°	16° 12' ± 1°
Fwd.	-6° ± 1°	-9° 48' ± 30'
Aft.	20° ± 1°	16° 12' ± 30'

The different rigging configurations are presented in the accompanying sketch, Figure C, for comparative purposes.

FIGURE C

(See next page)

FIGURE C



A—Blade pitch variation of the CH-37A
B—Blade pitch variation of the CH-37B

3. Dynamic Stability

a. General

The CH-37B dynamic stability characteristics were determined by analyzing the time histories of the helicopter motions resulting from pulse type control inputs. This analysis considered damping, control lag, curve shape, and qualitative pilot comments. Typical time histories are presented in Figures No. 16 through 33, Part II.

Tests were conducted in hover and level flight about the longitudinal, lateral and directional axes with the ASE both "on" and "off". Hovering dynamic stability tests were conducted in calm air at an average density altitude of 1500 feet, a mid center-of-gravity location (Station 236.5), a rotor rpm of 194 and an average gross weight of 30,000 pounds.

Level flight tests were conducted at 6000 feet average density altitude, aft center-of-gravity location (Station 242), average rotor speed of 186 rpm, and an average gross weight of 30,000 pounds. Tests were conducted at both low and cruise airspeeds.[1]

b. Longitudinal Dynamic Stability

The initial aircraft motion following a longitudinal pulse control input was in the proper direction. In all cases with the ASE "off", damping decreased as the airspeed increased. At cruise airspeeds the short period pitching motion was divergent. Pulse inputs at low airspeeds and in hovering flight resulted in an oscillatory divergent motion that required pilot recovery during the second half of the cycle.

The longitudinal motions with the ASE "off" in both low and cruise speed flight had oscillation periods which were approximately one-half those recorded for the CH-37A in the report, AFFTC-TR-60-15. The relocation of the horizontal stabilizer apparently provided greater damping in pitch during forward flight.

Longitudinal dynamic stability was improved considerably by the addition of the ASE. All pitching oscillations were damped in less than one cycle. Damping was higher at low airspeeds than in hovering flight and some reactions exhibited dead-beat characteristics.

c. Lateral Dynamic Stability

In all cases initial helicopter motion following a lateral pulse input was in the direction of the control input. Damping was greatest in hovering flight and at low airspeeds. With the ASE inoperative pilot recovery was necessary because of divergent lateral oscillations. Pitching motions and lateral-directional coupling were present in all flight regimes and in a hover the yaw motion was divergent.

---

[1] Henceforth, in this report low airspeed denotes approximately 45 knots CAS and cruise airspeed denotes approximately 85 knots CAS.

d. Directional Dynamic Stability

The helicopter yawed in the same direction as the pulse control input, in all cases.

Little difference in dynamic stability was noted between the CH-37A and the CH-37B with the ASE "off". In all flight regimes a right pedal input resulted in a turn to the right with an oscillation about the yaw axis while left pedal pulses created an oscillation about the trim axis with no heading change. Yaw rates from the oscillations were high but the relatively long time required to reach maximum allowed the pilot to recover without difficulty. The high tail rotor location resulted in an initial small adverse lateral-directional coupling caused by changes in the tail rotor thrust. This coupling was prevalent at both low and high airspeeds. An initial nose-up pitching motion followed a left pedal pulse during hovering flight. In all other conditions, the helicopter pitched nose-down after the directional input. Ground effect and translation influenced motion about the pitch axis when conducting hovering stability tests.

With the ASE "on" all oscillations damped to zero within one cycle. Attitude returned to trim and there was a small opposite roll contributed by the tail rotor moment about the roll axis. Characteristics are similar for all airspeeds tested, including hover.

4. <u>Controllability</u>

Controllability tests were conducted to determine the helicopter response characteristics to control inputs. Step control inputs were used to evaluate these characteristics. The analysis of the data included control lag, maximum values, time to reach maximum values, discontinuities in the curves, and the resulting helicopter attitudes. The time histories are presented in Figures No. 40 through 59.

Hovering flight tests were conducted in-ground-effect at 500 to 1,000 foot density altitudes with average gross weights varying from 30,150 to 31,025 pounds. The center-of-gravity was at Station 236.5 (mid) and rotor speeds utilized were from 192 to 195 rpm. All tests were conducted in a stabilized hover.

The level flight tests were conducted within a density altitude range of 4000 to 6000 feet and with average gross weights between 30,000 and 31,000 pounds. The center-of-gravity was maintained at Station 242 (aft of mid), and the rotor speed was 186 rpm. Characteristics in forward flight were evaluated at speeds of 45 and 85 knots CAS.

a. Control Sensitivity

The control sensitivity was determined by analyzing the angular accelerations resulting from step-type control inputs. No significant differences were found when comparing ASE "on" and ASE "off" data. The maximum values obtained and characteristic shapes of the acceleration curves are approximately the same.

Maximum angular acceleration versus control displacement are presented in Figures No. 34, 35 and 36, Part II. These plots indicate that sensitivity is non-linear at higher airspeeds in the lateral and directional axes. This non-linearity is not objectionable since the large control inputs required to reach the non-linearity condition are seldom required during normal operations.

Table III provides a control sensitivity comparison between the CH-37A and the CH-37B.

TABLE III

CONTROL SENSITIVITY COMPARISON BETWEEN CH-37B & CH-37A
(Sensitivity is measured in degrees per second^2)

Flight Conditions:

CH-37A; refer to AFFTC-TR-60-15
CH-37B; refer to Figures No. 34 through 36 .

	Longitudinal		Lateral		Directional	
	Fwd.	Aft.	Left	Right	Left	Right
Hovering (OGE)						
CH-37B	8.0	8.0	16.3	16.3	25.0	21.5
CH-37A	5.5	6.8	29.5	29.5	22.5	19.2
Low Airspeed 45 Knots CAS						
CH-37B	8.0	8.0	14.3	14.3	25.0	21.5
CH-37A	5.5	6.8	22.0	22.0	22.5	19.2
Cruise Airspeed 85 Knots CAS						
CH-37B	9.8	9.8	13.0	19.0	32.5	32.5
CH-37A	8.0	9.0	28.0	28.0	27.0	22.5

b. Control Response

(1) Longitudinal response (Reference Figure No. 37, Part II).

The maximum longitudinal angular velocities recorded in hover and at low airspeed without ASE were 8.5 degrees per second per inch of longitudinal cyclic input in either direction. The time required to achieve the maximum values of angular velocity averaged 2.8 seconds in a hover and 2.9 seconds at low airspeeds. Time histories of attitude indicate a pure divergence in both directions. However, the requirements of MIL-H-8501A, paragraph 3.2.11.1 are met. The long periods of the rate oscillations and the tendency for the angular acceleration to remain above zero also bear this out. During low speed flight the trim airspeed had changed only one to three knots at the time of recovery. At cruise airspeeds the pitch angular velocity increased to a value of 11.5 degrees per second for a one inch control input in either direction. This maximum was achieved approximately 2.5 seconds after the initial control deflection. The c.g. normal acceleration changed approximately 0.3g's in just over two seconds. No significant differences in control response existed between the CH-37A and the CH-37B at cruise airspeed.

With the ASE operative during hover and low airspeed flight, the control response was 4.5 degrees per second per inch of forward or aft cyclic movement. Maximum rates were reached in approximately 1.3 seconds in a hover and 1.2 seconds at low airspeeds. The helicopter pitched according to the direction of the control input and returned approximately to the initial trim attitude within 8 seconds. When trimmed at 45 knots an aft step caused the calibrated airspeed to steadily decrease to approximately 25 knots. The calibrated airspeed increased from 45 knots to approximately 70 knots in 7 seconds after a forward step, and c.g. normal acceleration slowly decreased to 0.8g's. The very slight adverse lateral-directional coupling which resulted from an aft step was not objectionable. At cruise airspeeds the control response was 5.5 degrees per second per inch of control input and time required to reach this value averaged 1.1 second. These values were the same for both forward and aft inputs. The helicopter restabilized in an attitude several degrees from trim and the airspeed change was 15 or 20 knots within 10 seconds. The c.g. normal acceleration varied approximately 0.3g's depending on the direction of input.

(2) Lateral Response (Reference Figure No. 38, Part II).

During hover and low airspeed flight with the ASE "off", a lateral cyclic step input produced a rolling velocity of 12.5 degrees per second per inch of control displacement in approximately 2 seconds.

Aircraft motion following a right step input was right roll, right yaw and pitch nose-down. A left cyclic stick input resulted in a left roll, an initial right yaw, and a pitch nose-up. The right yaw changed to left yaw after several seconds, and developed into a coordinated left turn. A right lateral step at cruise airspeed resulted in a roll angular velocity of 20 degrees per second while the rate response to the left was only 11.5 degrees per second. In general, the helicopter rolled and turned in the direction of lateral cyclic control input. The CH-37A exhibited similar characteristics (Reference AFFTC-TR-60-15).

Lateral cyclic movements in hovering or during low airspeed flight with the ASE "on" resulted in an angular rolling velocity of 6.6 degrees per second to the left and 7.3 degrees per second to the right. Maximum values occurred 1.1 seconds after control input. In hovering flight, the maneuvers caused loss of altitude and early recoveries were necessary because of the close proximity to the ground. At low airspeeds some bank attitudes reached 30 degrees before recovery was initiated, but the roll rate damped to approximately 5 degrees per second. Response to left and right lateral cyclic inputs at cruise airspeeds with the ASE operative resulted in average maximum roll rates of 7.5 and 11.5 degrees per second respectively. These maximum values were attained in approximately 1 second. The pitch and yaw attitude changes encountered previously were not present during operation of the ASE.

(3) Directional Response (Reference Figure No. 39, Part II).

With the ASE "off" directional step inputs provided by abrupt pedal movements in a hover resulted in maximum control responses of 41.0 and 62.0 degrees per second per inch of control input for left and right inputs respectively. These maximum values were reached in 3.7 seconds. At low airspeed the maximum rates dropped to 18 degrees per second for inputs in either direction and peaked in 1.8 seconds. These yaw rates are excessive, especially in a hover, however, pilot recovery is not difficult because of the time required to reach the maximum value. Although rates differed considerably between the two flight conditions, the helicopter attitude changes were similar. A right pedal input made the helicopter yaw right, roll right, and pitch down. Inputs to the left initiated a left yaw, a slight right roll followed by a left roll, and a slight pitch up. The CH-37A had essentially the same response characteristics (Reference AFFTC-TR-60-15). At cruise airspeed the maximum control response increased to 17.8 degrees per second which was achieved in 1.4 seconds after the initial input. Pedal inputs in either direction resulted in coordinated oscillating turns. The helicopter pitched nose down slightly after a right pedal step and slightly nose up after a left input. The CH-37A exhibited similar responses.

Maximum yaw rates attained during hover with the ASE "on", were 15.0 degrees per second in either direction and reached maximum in approximately 0.9 seconds. At low airspeeds the helicopter entered a turn with a small oscillation about the yaw axis. Control responses were 9.0 degrees per second attained in 1.3 seconds after control input. At cruise airspeed the maximum angular velocity resulting from one inch pedal inputs with the ASE "on" was 9 degrees per second attained in 1.1 seconds. Motions with ASE "on" were similar to those with the stability system inoperative. The high damping provided by the ASE does not prevent adequate maneuvering capability.

5. Automatic Stabilization Equipment Malfunctions

Tests were conducted to determine the aircraft's reaction to an ASE failure and the pilot effort required to control this reaction. The two types of failures analyzed were hard-overs and feedback circuit failures (oscillating hard-overs). A remote panel was provided by the aircraft manufacturer for the purpose of simulating the desired type of failure. This simulator panel is illustrated in Figure D.(Page 1.18)

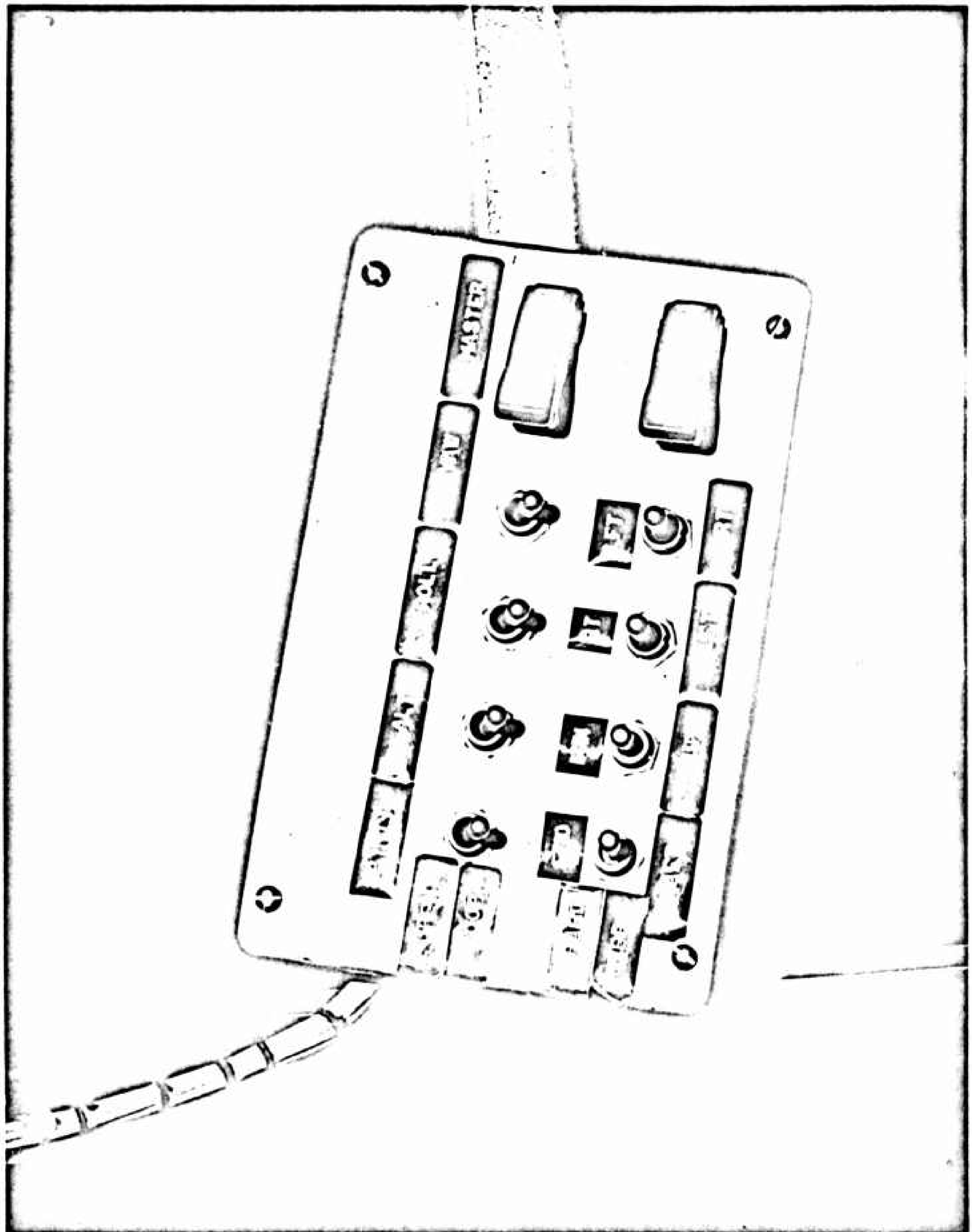
Two types of pilot response were utilized for this test:

1. The controls were held fixed until recovery was necessary

2. An immediate recovery attempt upon sensing an ASE failure.

a. Hardovers

ASE servo hard-overs may result from any of the following equipment malfunctions:


TASTER

(1) A broken feedback link or pilot valve.

(2) Loss of adjustment of a connecting linkage or pilot valve.

(3) Improper action of the feedback linkage preventing proper servo follow-up.

In the first two types of malfunctions, the power piston of the affected channel is driven to its extreme position. In the third case a constant force drives the cyclic control stick to its extreme position.

An actual system hard-over is generated by a step-type ASE control input which has the magnitude of the maximum authority (20 percent) of the ASE system. The aircraft motion should be similar to that resulting from a pilot-induced, step-type control input of the same magnitude with the ASE inoperative. Analysis of the data indicates that the remote simulator panel did not provide true hard-overs during this test. The inputs generated by the panel were initially of the proper magnitude, but slowly returned to the trim position after approximately 1 second. The helicopter reaction was similar to that resulting from a 10 percent cyclic or a 20 percent pedal pulse input, instead of the step-type control movement. Therefore, in order to properly evaluate the resulting motions of the CH-37B to ASE failures which result in hard-overs, it was necessary to analyze both the simulator panel results and the results from the pilot-induced step-type control inputs.

From the simulator panel results it was found that the initial motion was in the same direction as the control movement which resulted from the failure. With the controls fixed the reaction was usually a long period, lightly damped oscillation. The only time it was necessary for the pilot to initiate immediate recovery was following a forward cyclic failure at high speed. In this case the motion was a divergent pitch down and recovery was initiated two and one-half seconds after the failure. Recovery was accomplished without excessive control inputs and before extreme attitudes resulted. When an immediate recovery was initiated the trim attitude could be maintained with only small corrections. Time history illustrations of these results are presented in Figures No. 60 through 68, Part II.

For a pilot induced hard-over, which was a one-inch step input with the ASE inoperative, immediate corrective action was required to prevent excessive rates and extreme attitudes from developing. A complete discussion of this aircraft motion can be found in Section 4.b, Part I, and the time histories are presented in Figures No. 40 through 59, Part II.

b. ASE Feedback Circuit Failures (Reference Figures No. 68 through 73, Part II).

The helicopter responded to a feedback circuit failure by oscillating about the failure axis. Rates and angular accelerations were large in all cases, however, the directions reversed too quickly to allow large attitude variations. A feedback circuit failure in the directional channel resulted in rates and accelerations that were of sufficient magnitude to create personal discomfort and concern about the structural integrity of the aircraft.

Pilot attempts to override the control inputs and maintain attitude results in amplification of the helicopter motions. This was apparently caused by closed loop control response due to pilot reaction time. The best pilot reaction to a feedback circuit failure is to hold the control fixed and immediately turn off the ASE.

c. Three-axis Hard-overs.

The helicopter reaction to a three-axis hard-over was investigated by actuating the two-position "Override Check" switch during flight. This resulted in a combined nose-down, left roll, and left yaw maneuver (presented as a time history in Figure No. 74, Part II). Actuation of the switch in the opposite direction resulted in a combined nose-up, right roll, and right yaw maneuver.

d. Actual Malfunctions

Several actual vibrational disturbances, one of which is presented as a time history in Figure No. 75, Part II, were encountered during testing. The malfunction that caused these disturbances was never determined except that it probably was in the ASE. It was not possible to determine whether the malfunction was peculiar to this installation or is inherent in all CH-37B helicopters.

6. <u>Control Forces</u>

Tests were conducted to determine the force gradient and friction forces present in the control system. These tests were performed with the helicopter on the ground with the rotor stationary. During the tests, the utility hydraulic system, the main servos, and the ASE servos were operated in a manner which simulates various normal flight emergency conditions. The results of these tests are presented in the following Table:

TABLE IV

(See next page)

TABLE IV

Utility Hydraulic System	Main Servos	ASE Servos	Longitudinal Stick Force -pounds	Lateral Stick Force - pounds	Directional Stick Force -pounds
Off	On	Off	5.5	6.5	-
On	On	Off	3.5	4.5	35.0
On	On	On	1.5	2.5	20.0

Longitudinal and lateral breakout forces are satisfactory and there is no apparent dead-band region. Incorporation of the ASE servo in the CH-37B reduced the forces and is a decided improvement over the CH-37A with all systems operating. With the stick trim turned "off" and all servos operating, less than one pound of force was required to move the cyclic control stick through full travel.

The high directional control friction forces are unsatisfactory. The values recorded for the CH-37B are greater than those reported for the CH-37A in the report AFFTC-TR-60-15. These high pedal forces in addition to the high sensitivity increase the tendency for the pilot to overcontrol and make precision flying difficult especially while hovering.

7. Airspeed Calibration (Reference Figure No. 75, Part II).

The sensitive airspeed system fitted to the test aircraft was calibrated by the ground speed course method. The helicopter was flown in OGE level flight at each airspeed on reciprocal headings to nullify wind effects. The engines were operated at normal mixture and 2600 rpm. The average gross weight was 26,000 pounds and the center-of-gravity was located at Station 236.5 (mid). There were no external stores and the landing gear was down.

G. CONCLUSIONS

The limited performance tests conducted during this program did not determine that there were any required changes in the Operator's Manual. However, significant variations in level flight maximum airspeed, fuel flow characteristics, power required, and power available were found during the evaluation. These variations were not considered to be a result of the change in the horizontal stabilizer location or the side door conversion. The variation is attributed to differences in the engine characteristics and power measuring techniques employed in the different tests. Analysis of the test results do not indicate sufficient justification for changing the performance data presented in Operator's Manual.

Static longitudinal speed stability is satisfactory and was found to be slightly more positive than that reported for the CH-37A. This increase in positive stability is attributed to greater pitching moments from the relocated horizontal stabilizer. Static longitudinal stability was not affected by the ASE.

For all flight conditions other than low airspeed autorotation, the static directional stability is satisfactory and dihedral effect is positive. The helicopter has good pedal fixed maneuvering capability in level flight. During low speed autorotation a "wallowing" motion is prevalent and directional control inputs have little effect on yaw attitude.

Forward and rearward flight tests indicate that more than 30 percent of the aft longitudinal stick travel remains at airspeeds up to 30 knots TAS in rearward flight.

With the exception of longitudinal motion, no significant changes in the dynamic stability are apparent between the CH-37A and CH-37B with the ASE "off". With the ASE "on" the dynamic stability is improved considerably in all cases. Relocation of the horizontal stabilizer apparently provides greater longitudinal damping in forward flight.

Control sensitivity is essentially the same for the CH-37B as for the CH-37A, and is not significantly affected by operation of the ASE.

Control response about the longitudinal and lateral axes is satisfactory with the ASE "off" and is comparable to the CH-37A test results. The directional control response is excessive, and, combined with the high pedal forces, causes frequent overcontrolling in hovering flight. For all axes, operation with the ASE "on" lowers the control response and provides better flying qualities, particularly during precision hovering.

ASE failures which result in maximum authority control inputs are controllable if recoveries are initiated within a reasonable time period. Excessive delay results in extreme aircraft attitudes. The helicopter responds to an ASE feedback circuit failure (oscillating full authority control inputs) by oscillating about the failure axis. The best pilot technique for recovery is to fix the control and immediately turn off the ASE.

H. RECOMMENDATIONS

It is recommended that accomplishment of the following items be given consideration:

1. No changes should be made to the Operator's Manual based on the performance data in this report.

2. Automatic Stabilization Equipment should be installed in all CH-37 aircraft to improve the handling qualities.

3. A study should be conducted to verify the structural integrity of the tail rotor pylon. Rapid yaw movements resulting from large pedal inputs or directional hard-overs may be of sufficient magnitude to allow the forces to exceed the structural limits of the aircraft.

4. A note should be added to the Operatot's Manual (TM55-1520-203-10) briefly describing the helicopter's response to ASE failures in the various modes and the best method of recovering from each.

RICHARD J. KENNEDY, JR.
Lieutenant Colonel, TC
Commanding

PART II- TEST DATA

FIGURE NO. 1

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

CH-37B, U.S.A., S/N 54-0998

C_T	AVERAGE GROSS WEIGHT ~ POUNDS	AVERAGE DENSITY ALTITUDE ~ FEET	RPM
61.63×10^{-4}	24,920	5085	2500/185.5
68.72×10^{-4}	27,800	5250	2500/185.5
76.45×10^{-4}	30,880	5370	2500/185.5
79.76×10^{-4}	27,780	10,130	2500/185.5

C.G. LOCATION = STATION 286.5 (MID)

POINTS REPRESENT FAIRED LINE VALUES FROM FIGURES TO

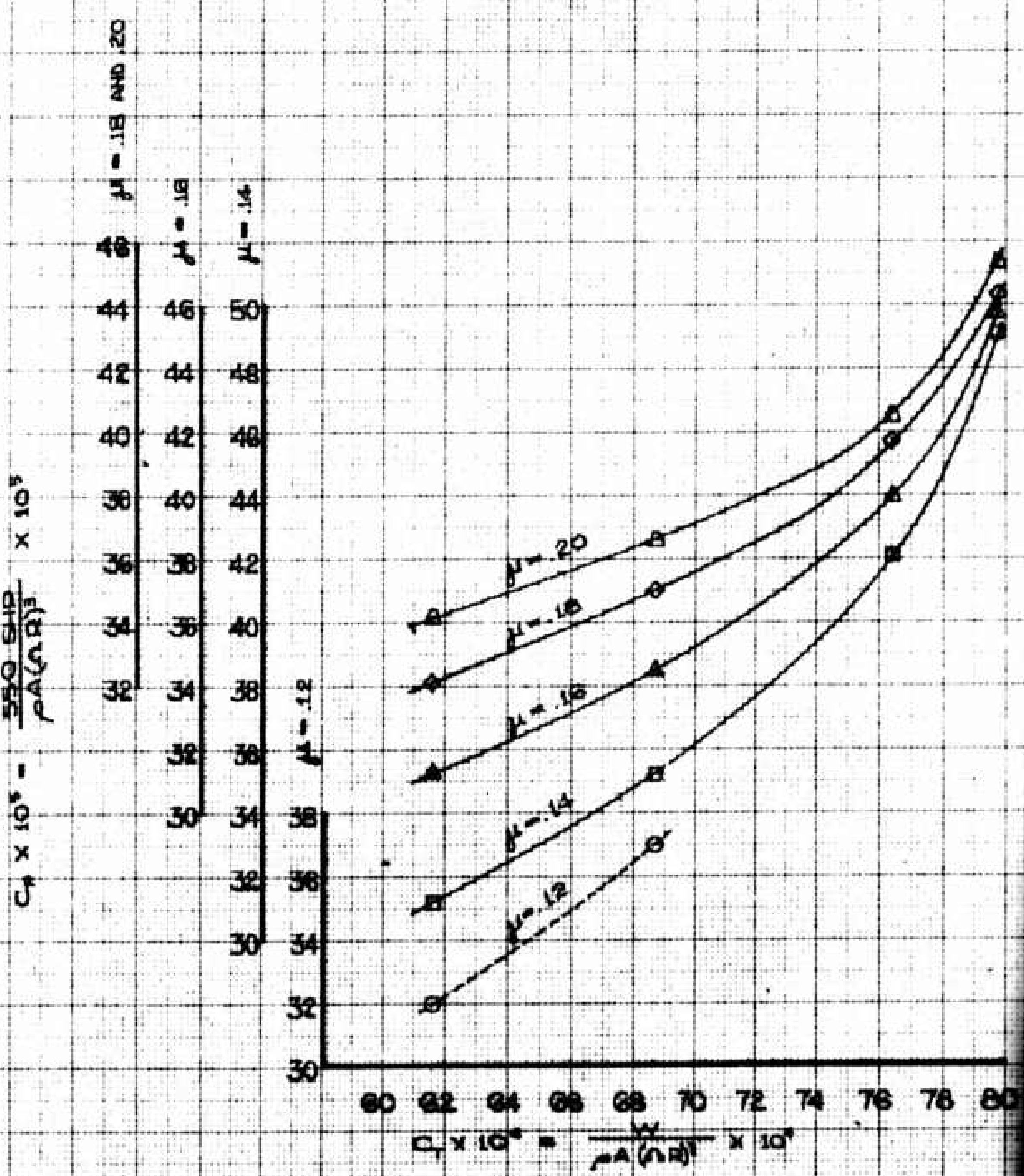


FIGURE NO. 2

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

CH-37B, U.S.A., S/N 54-0998

C_T	AVERAGE GROSS WEIGHT ~ POUNDS	AVERAGE DENSITY ALTITUDE ~ FEET	RPM
61.63×10^{-4}	24,920	5035	2600/185.5
68.72×10^{-4}	27,800	5250	2600/185.5
76.45×10^{-4}	30,820	5370	2600/185.5
79.76×10^{-4}	27,780	10,130	2600/185.5

C.G. LOCATION = STATION 286.5 (MID)

POINTS REPRESENT FAIRED LINE VALUES FROM FIGURES TO

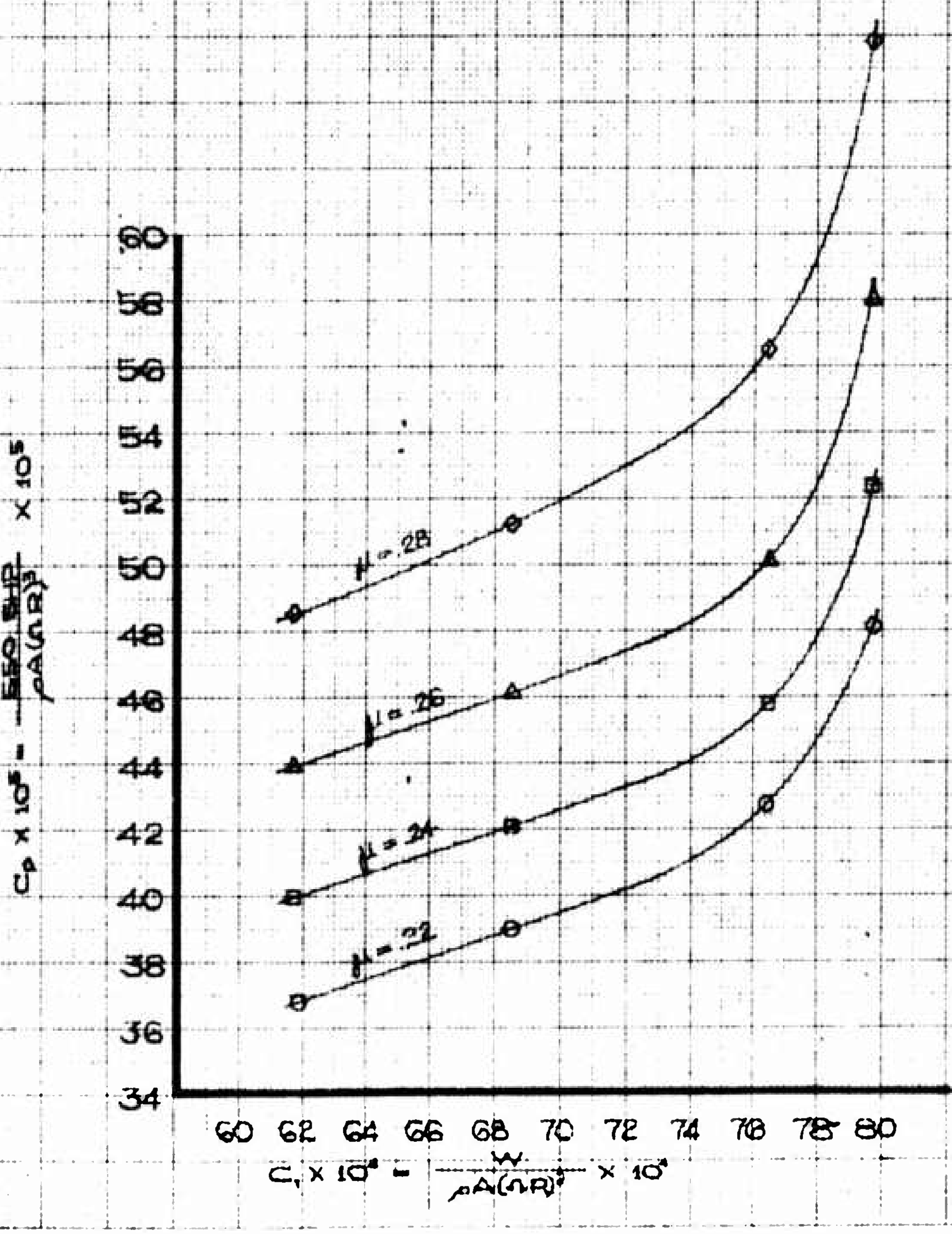


FIGURE NO. 3
LEVEL FLIGHT PERFORMANCE
CH-37B, U.S.A., S/N 54-0998

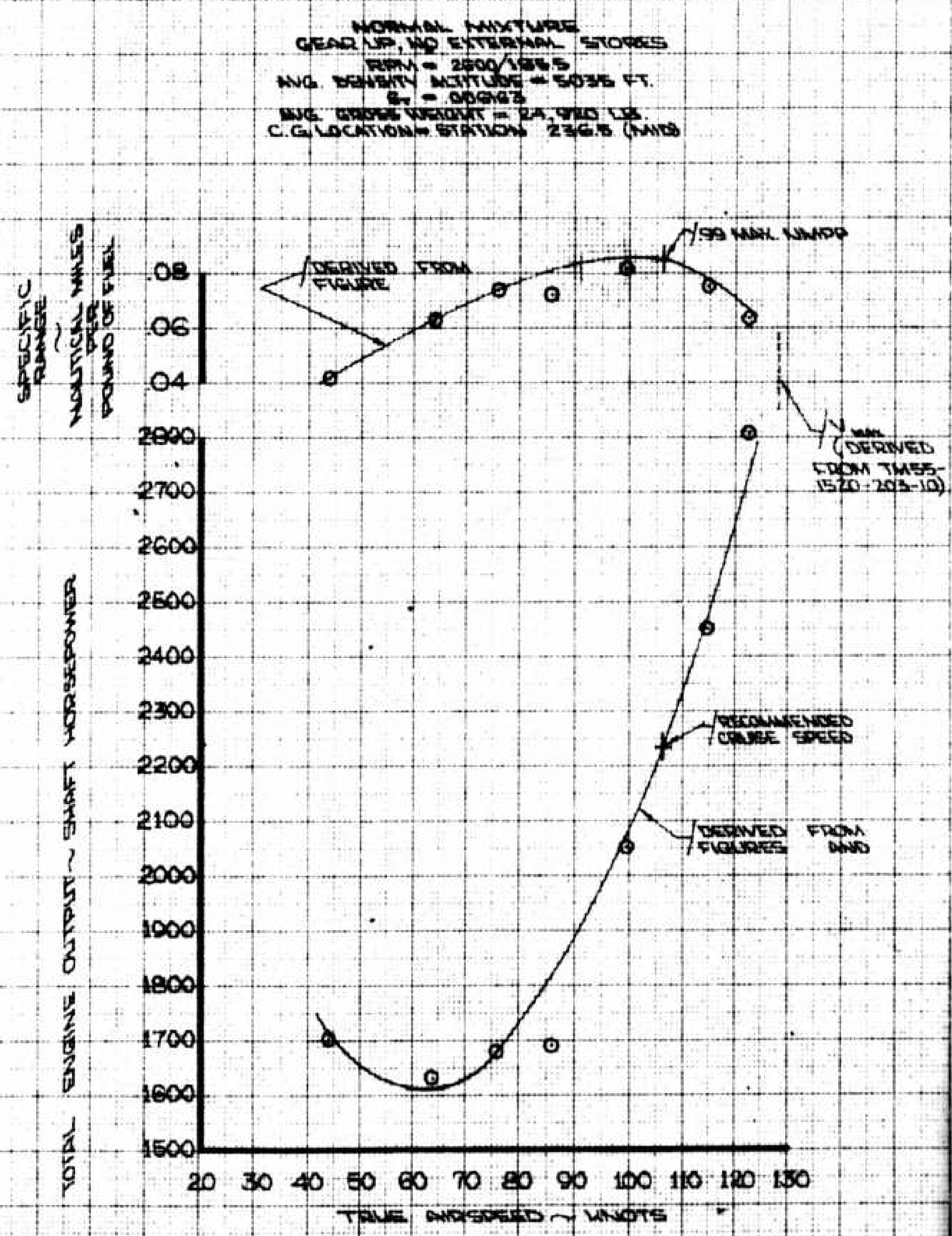


FIGURE NO. 4

LEVEL FLIGHT PERFORMANCE

CH-37B, U.S.A., S/N 54-0998

NORMAL MIXTURE
GEAR UP, NO EXTERNAL STORES
RPM = 2800/185.5
AVG. DENSITY ALTITUDE = 5250 FT.
C_T = .006872
AVG. GROSS WEIGHT = 27,500 LB.
C.G. LOCATION - STATION 236.5 (MID)

NOTE: SPECIFIC RANGE POINTS ARE OMITTED WHERE FUEL FLOW DATA WERE NOT AVAILABLE.

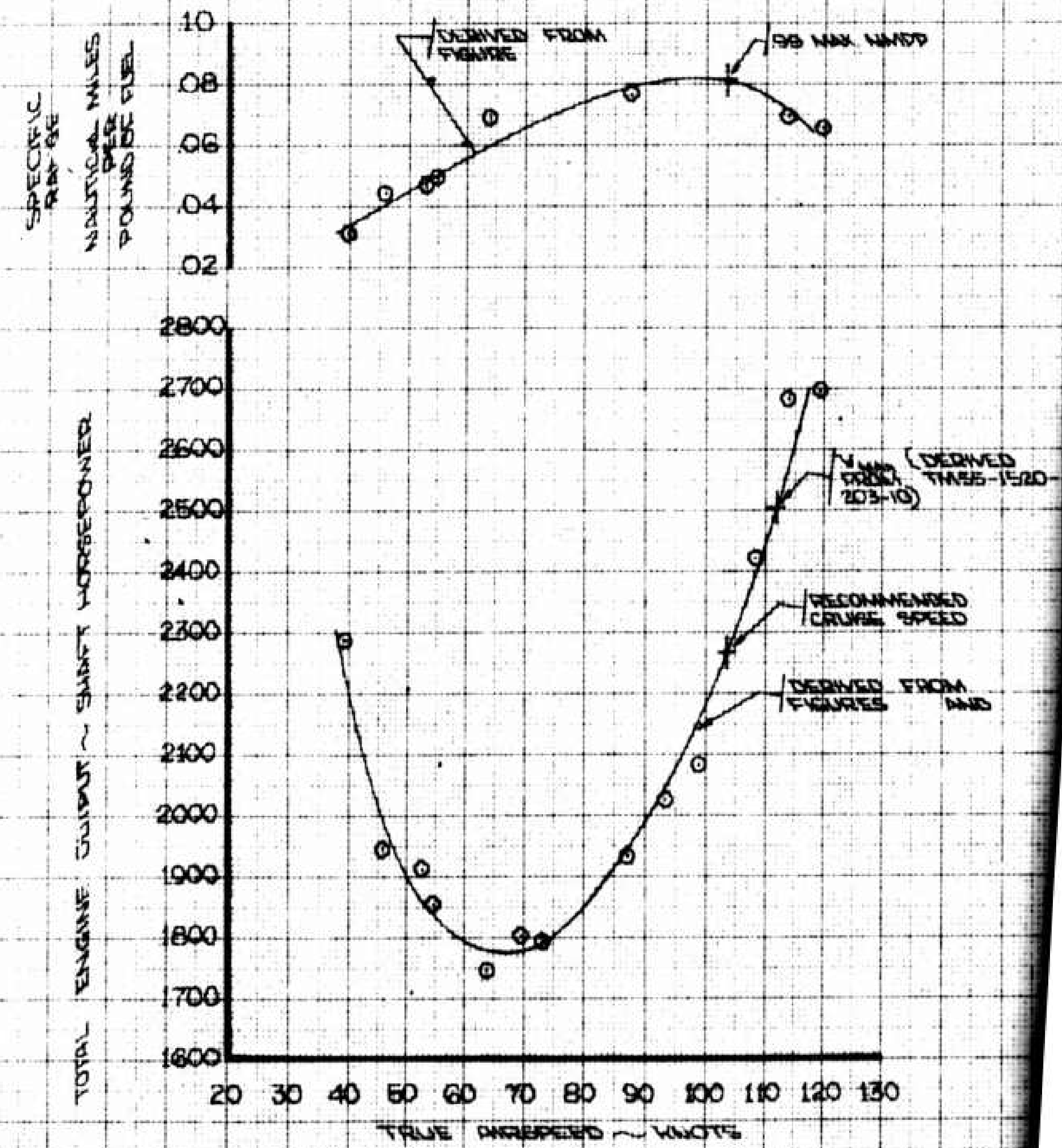


FIGURE NO. 5

LEVEL FLIGHT PERFORMANCE

CH-37B, U.S.A., S/N 54-0998

NORMAL MIXTURE
GEAR UP, NO EXTERNAL STORES
RPM = 2800/185.5
AVG. DENSITY ALTITUDE = 5370 FT.
C_T = .007645
AVG. GROSS WEIGHT = 30,820 LB.
C.G. LOCATION = STATION 236.5 (AVG)

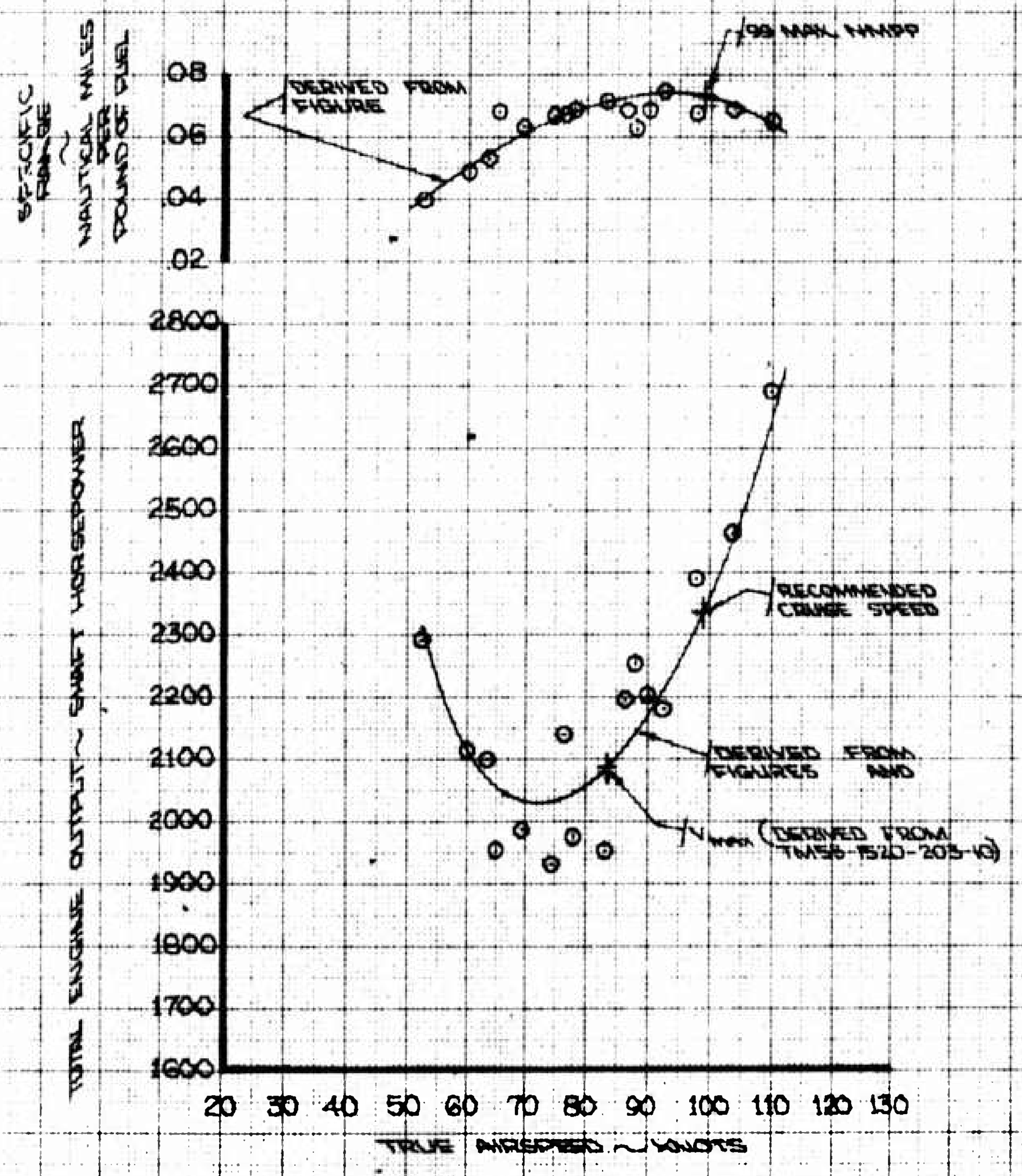


FIGURE NO. 6

LEVEL FLIGHT PERFORMANCE

CH-37B, U.S.A., S/N 54-0998

NORMAL MIXTURE
GEAR UP, NO EXTERNAL STORES
RPM = 2800/1885
AVG. DENSITY ALTITUDE = 10,130 FT.
C_T = .007978
AVG. GROSS WEIGHT = 27,780 LB.
C.G. LOCATION = STATION 232.5 (MID)

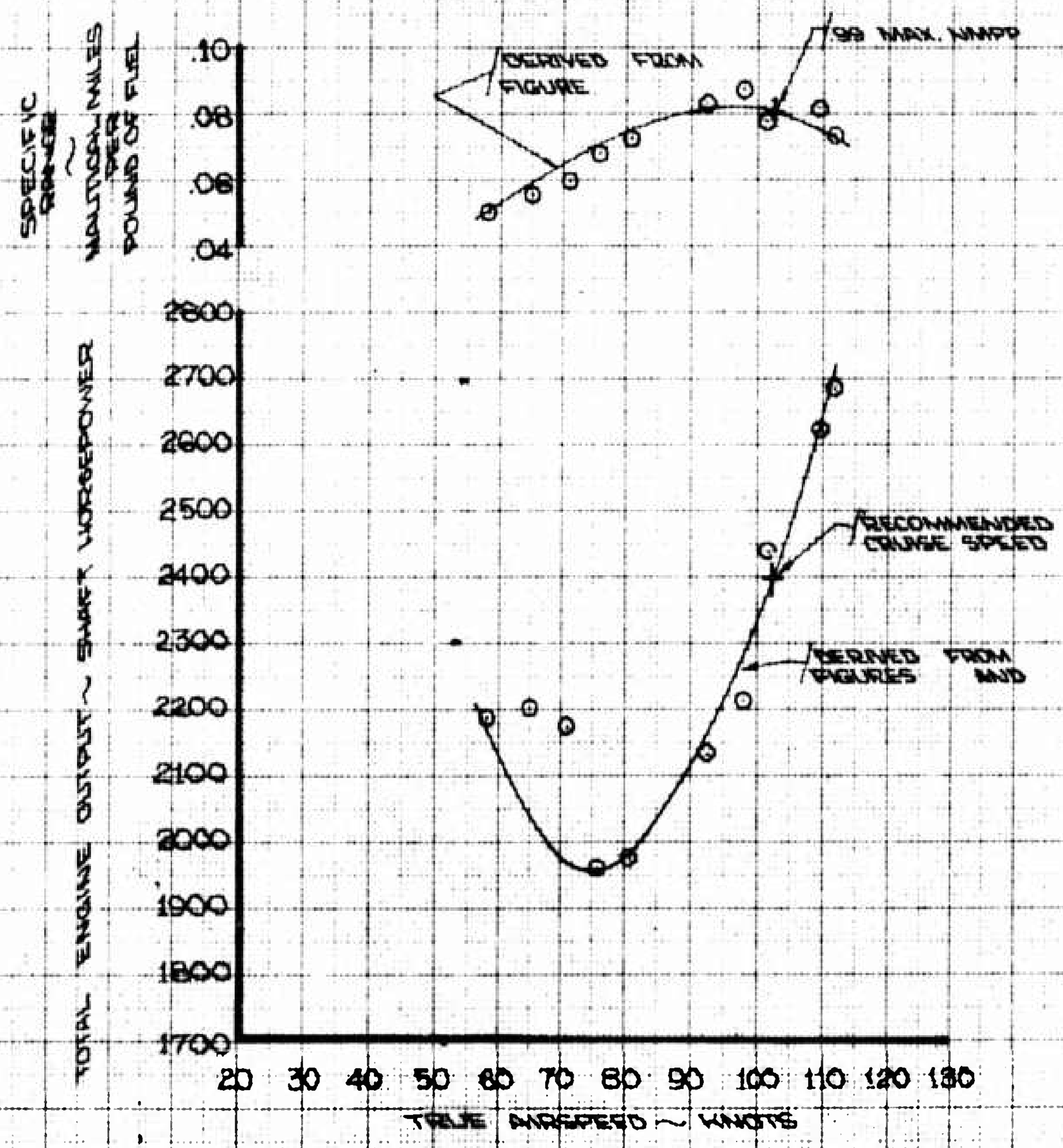


FIGURE NO. 7

STATIC LONGITUDINAL SPEED STABILITY

CH-37B, U.S.A. S/N 54-0998

LEVEL FLIGHT ~ ASE OFF

CALIBRATED TRIM AIRSPEED	SYMBOL
[illegible] KNOTS	○
[illegible] KNOTS	□
[illegible] KNOTS	△

CONTROL	FULL TRAVEL ~INCHES	TRAVEL FOR ONE INCH~PER CENT
LONGITUDINAL	15.9	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3
COLLECTIVE	10.0	10.0

[illegible]

AVERAGE GROSS WEIGHT = 31,320 LB.

AVERAGE DENSITY ALTITUDE = 4430 FT.

GEAR DOWN

NO EXTERNAL STORES

C.G. LOCATION ~ STATION 242 (AFT OF [illegible])

NOTE: SHADED POINTS DENOTE TRIM CONDITIONS

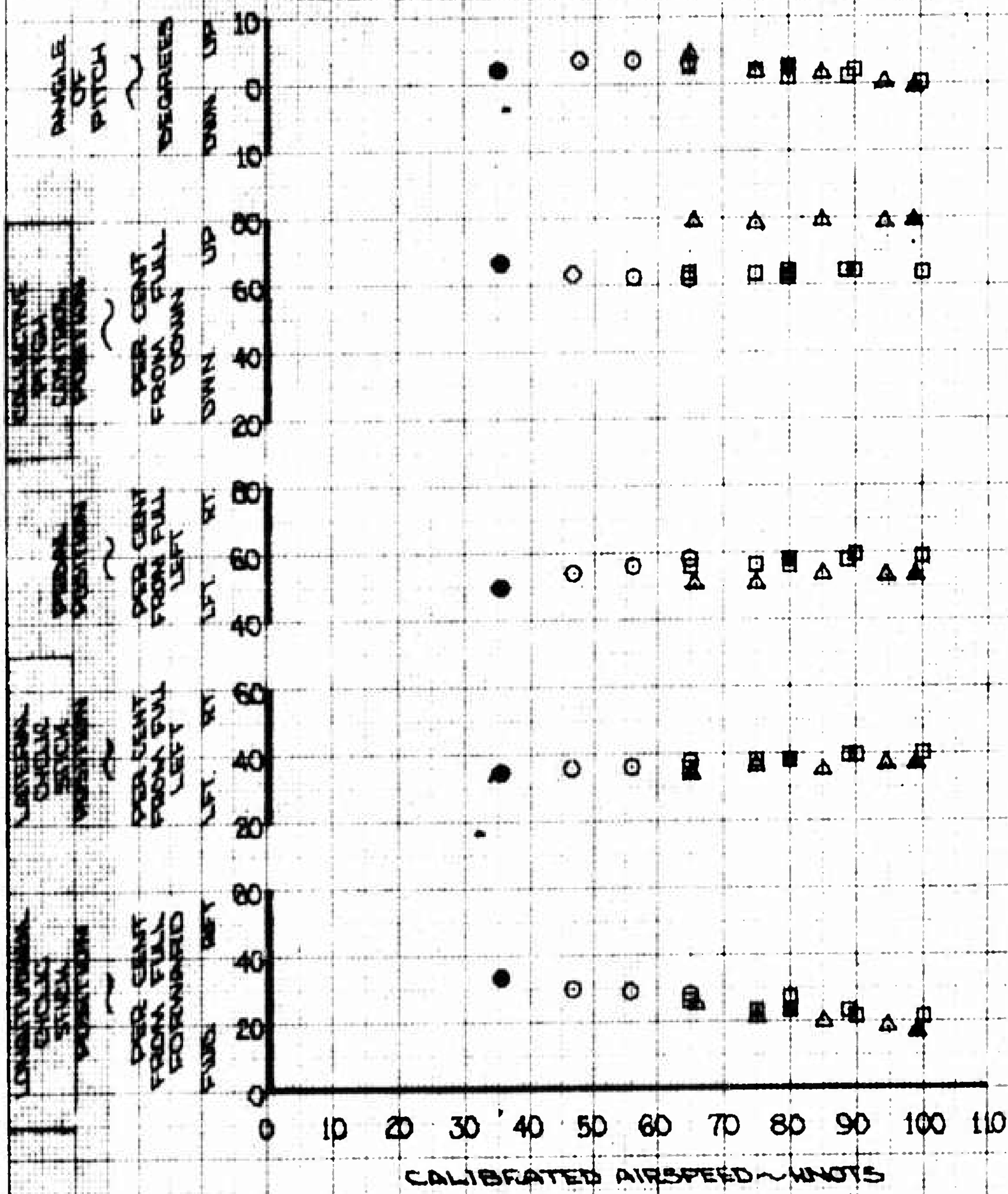


FIGURE NO. 8

STATIC LONGITUDINAL SPEED STABILITY

CH-37B, U.S.A., S/N 54-0998

CLIMB AND AUTOROTATION ~ ASE OFF

FLIGHT CONDITION	CALIBRATED TRIM AIRSPEED	SYMBOL
CLIMB	65 KNOTS	○
AUTOROTATION	65 KNOTS	□
AUTOROTATION	90 KNOTS	△

CONTROL	FULL TRAVEL ~ INCHES	TRAVEL FOR ONE INCH ~ PER CENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3
COLLECTIVE	10.0	10.0

ROTOR RPM = 192 @ 65 KTS., 200 @ 90 KTS.
AVERAGE GROSS WEIGHT = 30,710 LB.
AVERAGE DENSITY ALTITUDE = 6280 FT

GEAR DOWN
NO EXTERNAL STORES
C.G. LOCATION = STATION 242 (AFT OF MID)

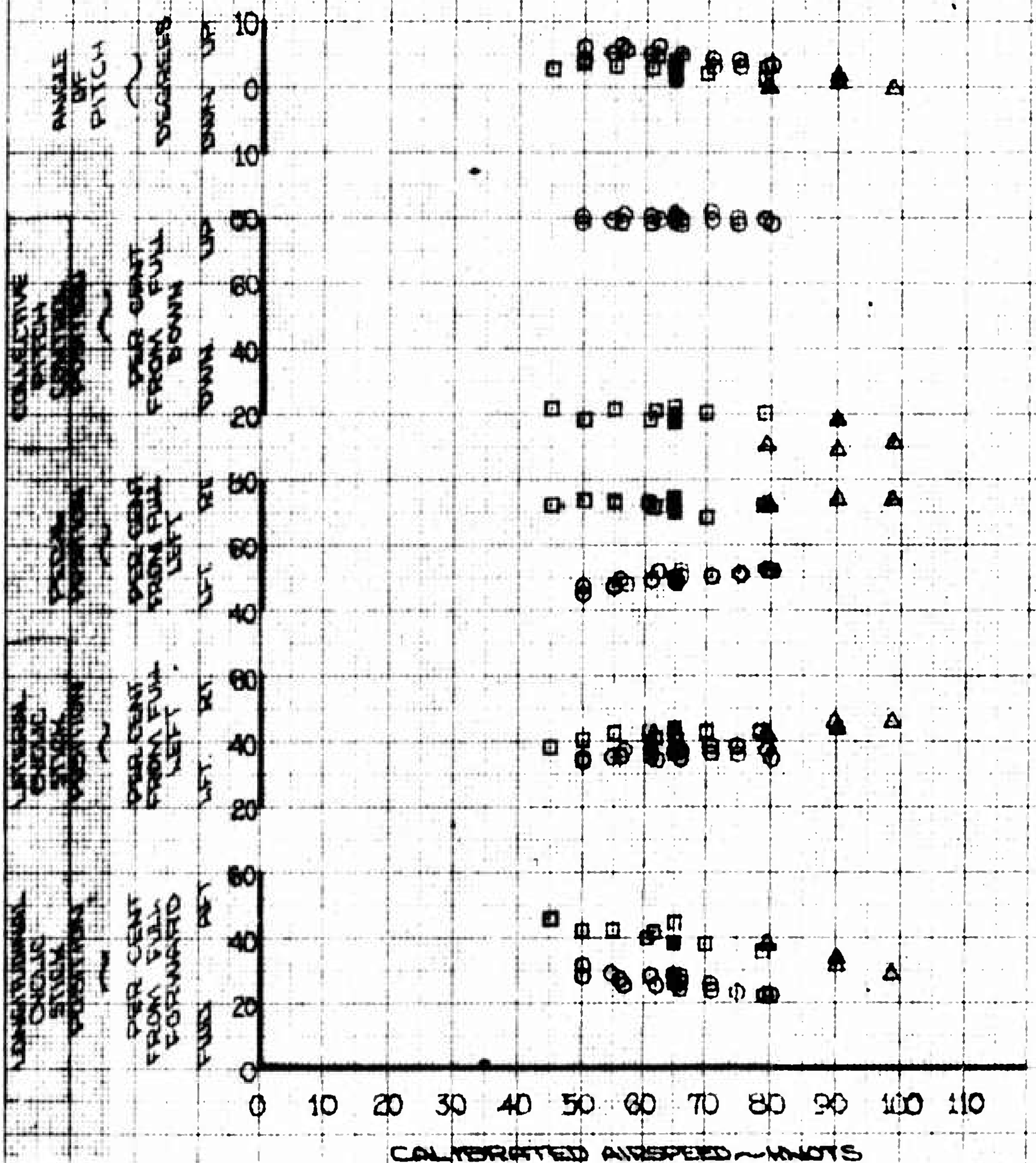


FIGURE NO. 9

STATIC DIRECTIONAL STABILITY

CH-37B, U.S.A., S/N 54-0998

LEVEL FLIGHT ~ ASE OFF

GEAR DOWN, NO EXTERNAL STORES
C. G. LOCATION = STATION 242 (AFT OF MID)
AVERAGE DENSITY ALTITUDE = 5860 FT
AVERAGE GROSS WEIGHT = 30,810 LB
RPM = 26[illegible]/187

45 KNOTS CALIBRATED TRIM AIRSPEED

CONTROL	FULL TRAVEL ~ INCHES	TRAVEL FOR ONE INCH ~ PER CENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3

NOTE: SHADED POINTS DENOTE TRIM CONDITIONS

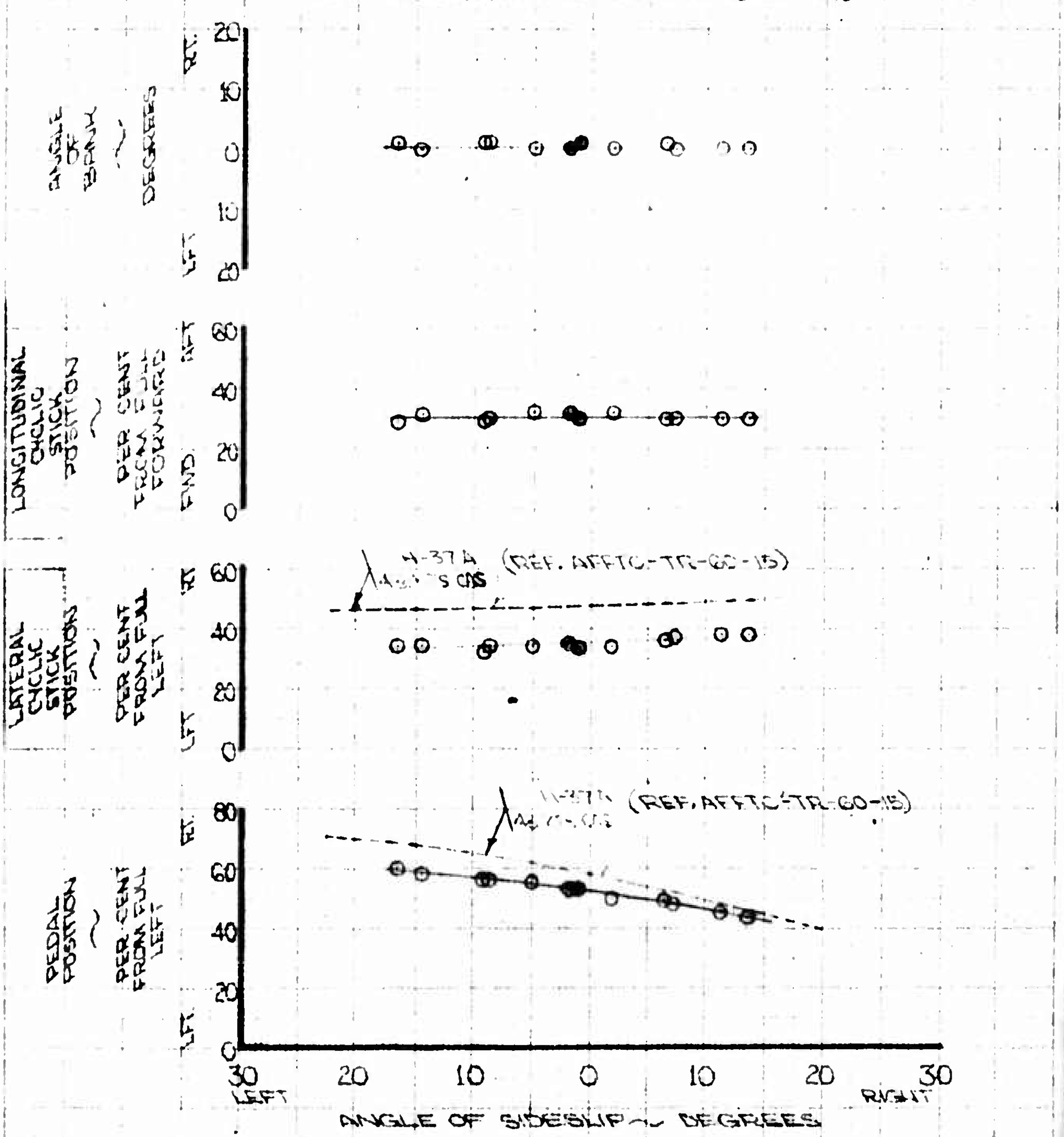


FIGURE NO. 10

STATIC DIRECTIONAL STABILITY

CH-37B, U.S.A., S/N 54-0998

LEVEL FLIGHT - ASE OFF

GEAR DOWN, NO EXTERNAL STORES
[illegible]
[illegible] DENSITY ALTITUDE = 5950 FT
AVERAGE GROSS WEIGHT = 30,470 LB
RPM = 2205, 182

85 KNOTS CALIBRATED TRIM AIRSPEED

CONTROL	TOTAL TRAVEL INCHES	TRAVEL FOR ONE INCH PER CENT
LONGITUDINAL	[illegible]	6.25
LATERAL	15.0	6.67
PEDAL	[illegible]	[illegible]

NOTE: SHADED POINTS DENOTE TRIM CONDITIONS.

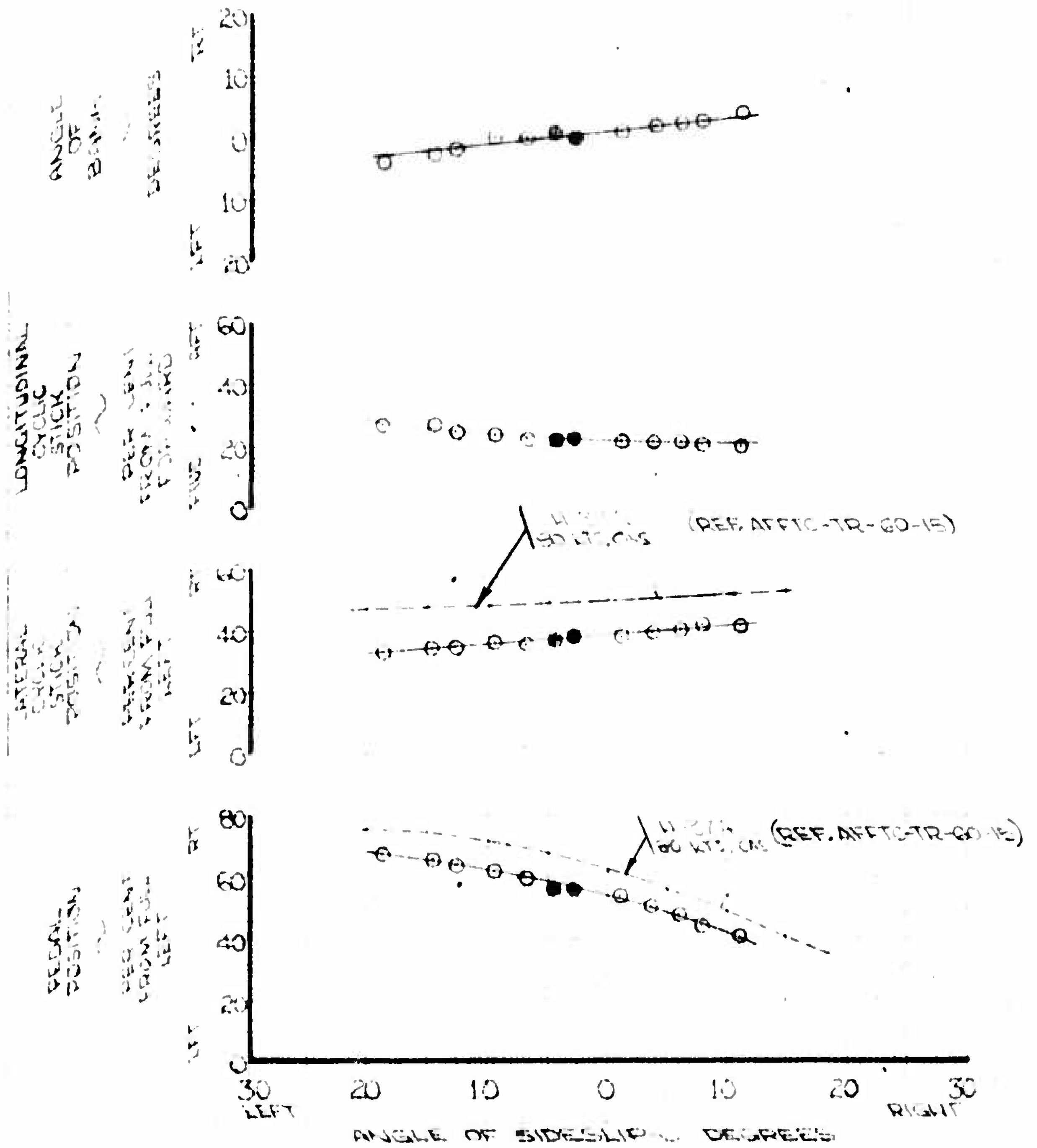


FIGURE NO. 11

STATIC DIRECTIONAL STABILITY

CH-37B, U.S.A., S/N 54-0998

LEVEL FLIGHT ~ ASE ON

GEAR DOWN, NO EXTERNAL STORES
C.G. LOCATION = STATION 242 (AFT OF MID)
AVERAGE DENSITY ALTITUDE = 4850 FT.
AVERAGE GROSS WEIGHT = 31,100 LB.
RPM = 2605/186

45 KNOTS CALIBRATED TRIM AIRSPEED

CONTROL	FULL TRAVEL ~INCHES	TRAVEL FOR ONE INCH~PER CENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3

NOTE: 1. TOTAL PEDAL INPUT IS PEDAL POSITION PLUS PEDAL ASE POSITION.
2. SHADED POINTS DENOTE TRIM CONDITIONS.

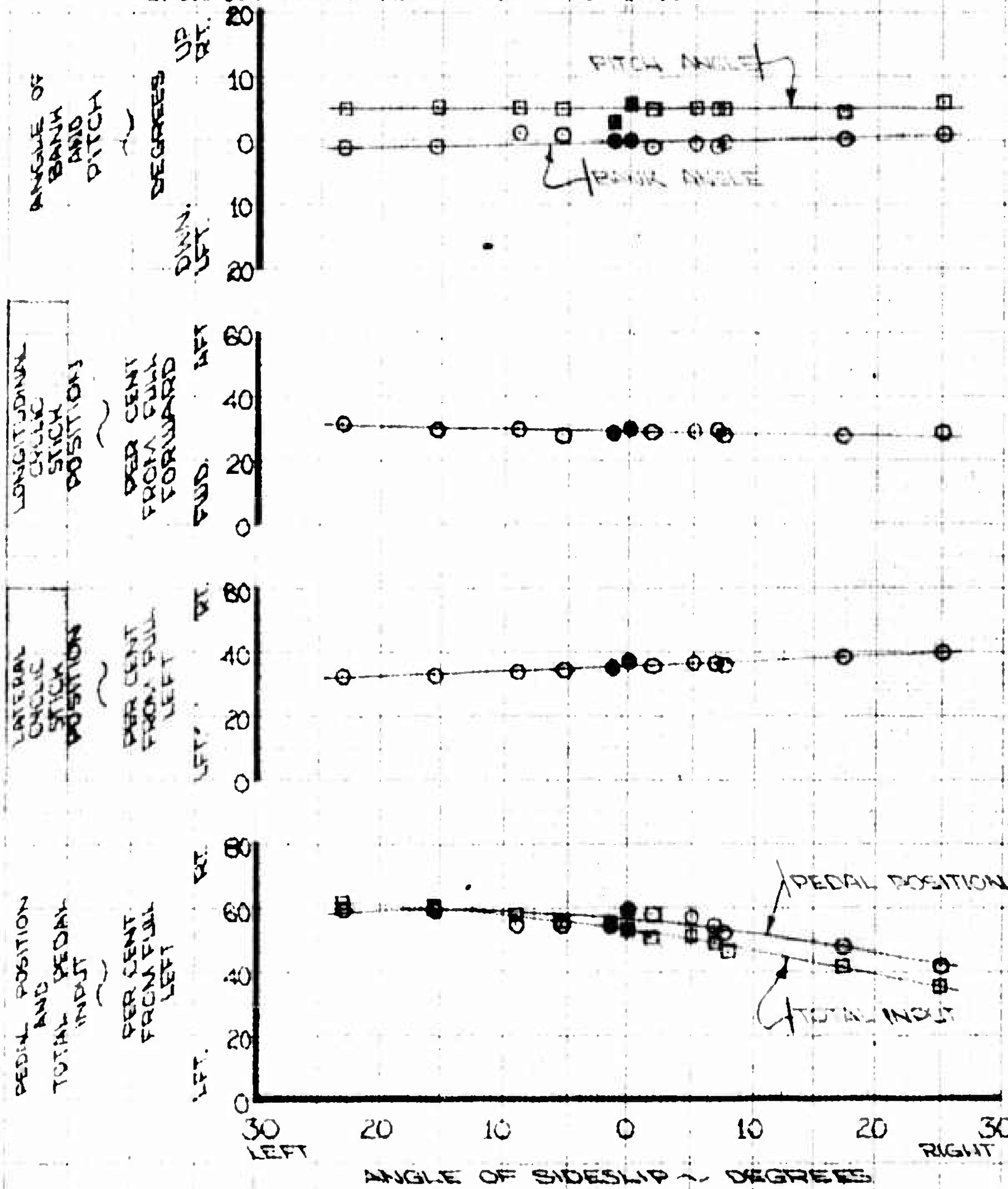


FIGURE NO. 12

STATIC DIRECTIONAL STABILITY

CH-37B, U.S.A. S/N 54-0998

LEVEL FLIGHT - ASE ON

GEAR DOWN, NO EXTERNAL STORES
C.G. LOCATION = STATION 242 (AFT OF MID)
AVERAGE DENSITY ALTITUDE = 4900 FT.
AVERAGE GROSS WEIGHT = 30,720 LB.
R.P.M. = 2305/136

85 KNOTS CALIBRATED TRIM AIRSPEED

CONTROL	FULL TRAVEL ~ INCHES	TRAVEL FOR ONE INCH ~ PER CENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3

NOTE: 1. TOTAL PEDAL INPUT IS PEDAL POSITION PLUS PEDAL ASE POSITION.
2. SHADED POINTS DENOTE TRIM CONDITIONS

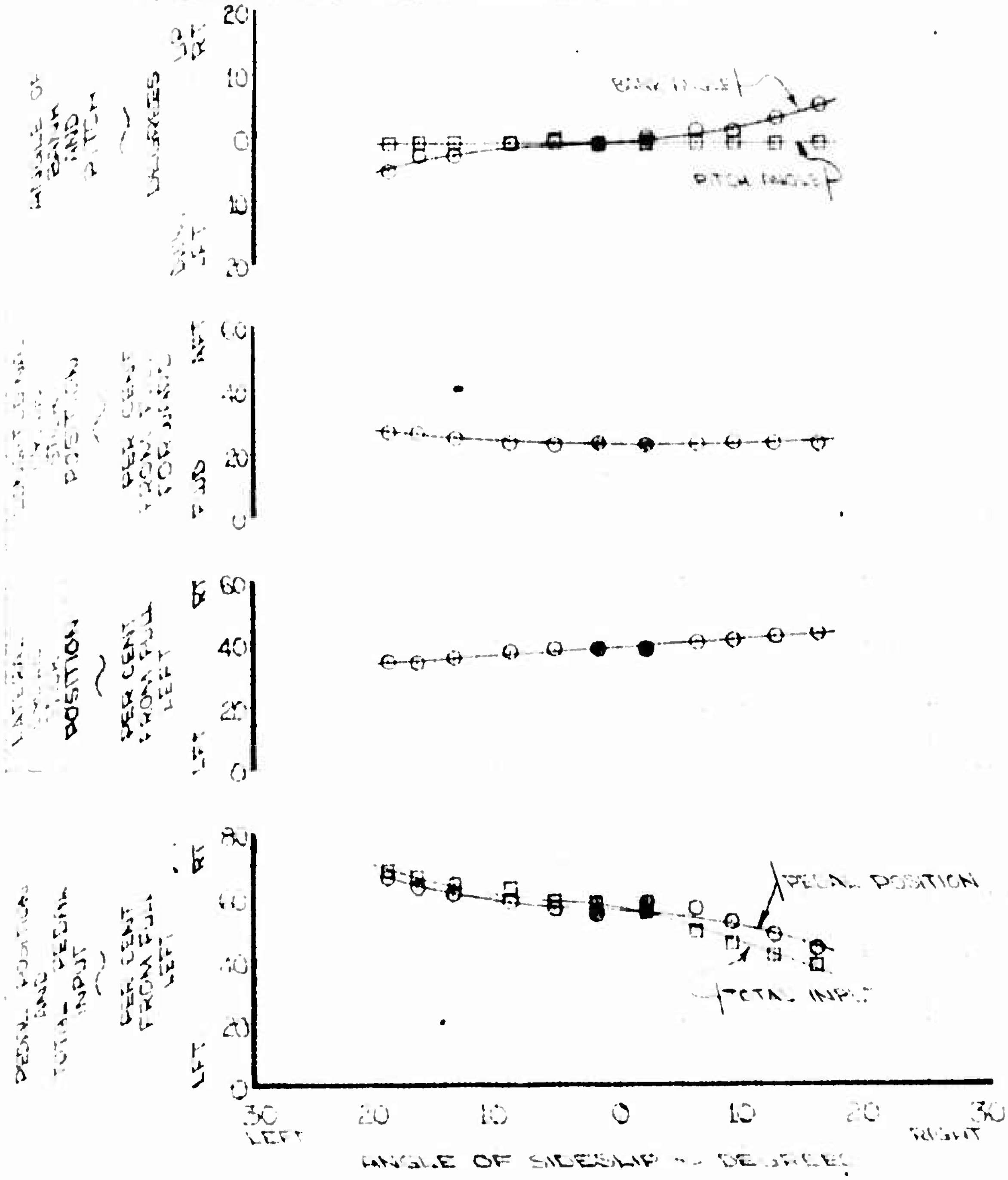


FIGURE NO. 13

STATIC DIRECTIONAL STABILITY

CH-37B, U.S.A., S/N 54-0998

AUTOROTATION ~ ASE OFF

CALIBRATED TRIM AIRSPEED	SYMBOL
42 KNOTS	△
53 KNOTS	○
57 KNOTS	□

CONTROL	FULL TRAVEL ~ INCHES	TRAVEL FOR ONE INCH ~ PER CENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	3.8	26.3

RPM = 192
AVERAGE GROSS WEIGHT = 30,500 LB.
AVERAGE DENSITY ALTITUDE = 5200 FT

GEAR DOWN
NO EXTERNAL STORES
C.G. LOCATION = STATION 242 (AFT OF MID)

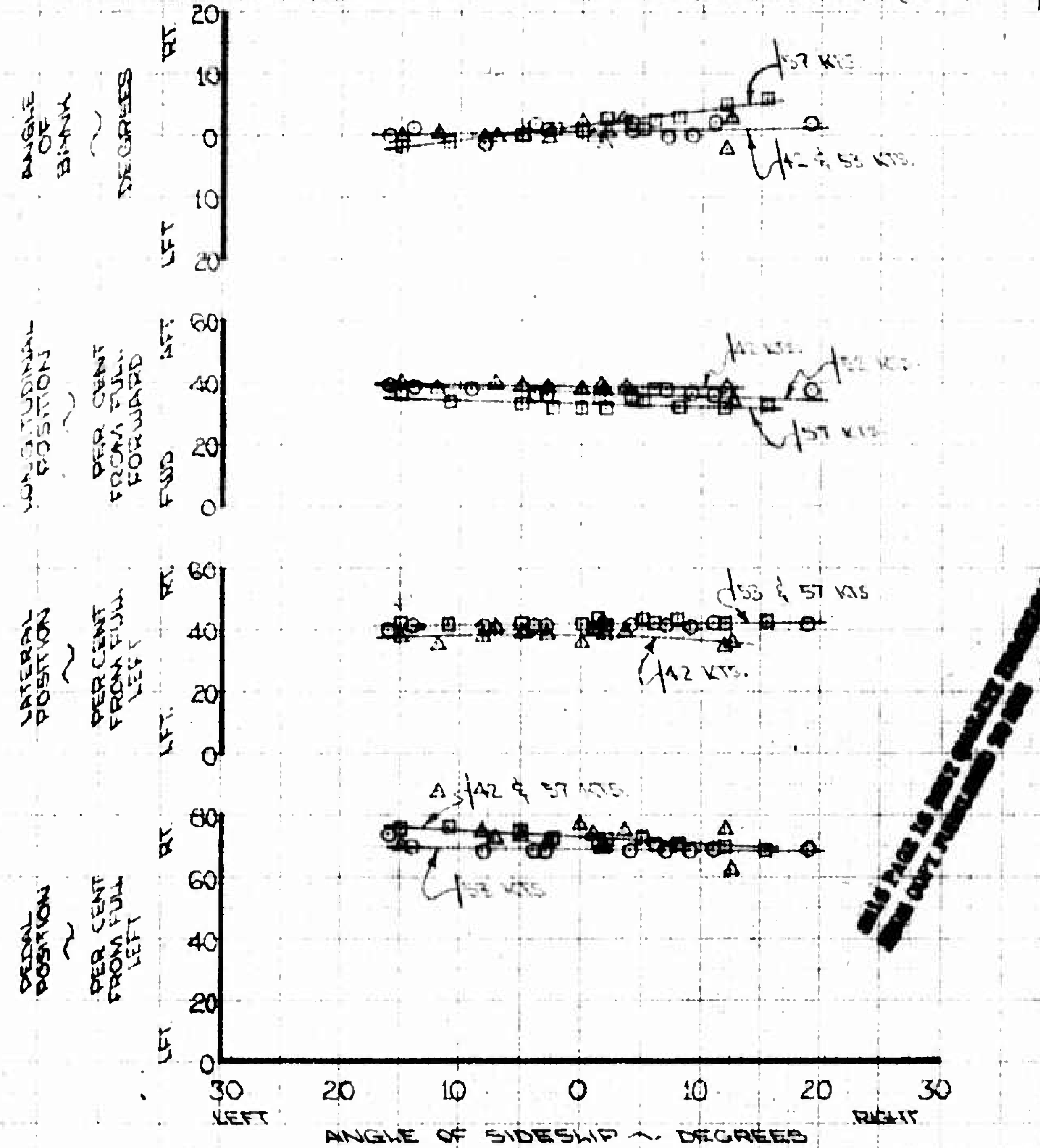


FIGURE NO 14

STATIC DIRECTIONAL STABILITY

CH-47B U.S.A. S/N [illegible]

AUTOROTATION ~ ASE ON

CALIBRATED TRIM AIRSPEED	SYMBOL
42 KNOTS	○
[illegible] KNOTS	□

CONTROL	[illegible] TRAVEL INCHES	TRAVEL FOR ONE INCH PERCENT
LONGITUDINAL	16.0	6.25
LATERAL	15.0	6.67
PEDAL	5.8	[illegible]

RPM = 1[illegible]
AVERAGE GROSS WEIGHT = [illegible] LB
AVERAGE DENSITY ALTITUDE = 4200 FT

GEAR DOWN
NO EXTERNAL STORES
C.G. LOCATION = STATION 242 (AFT OF MID)

NOTE: FLAGGED SYMBOLS DENOTE TOTAL PEDAL INPUT WHICH IS PEDAL POSITION PLUS PEDAL ASE INPUT

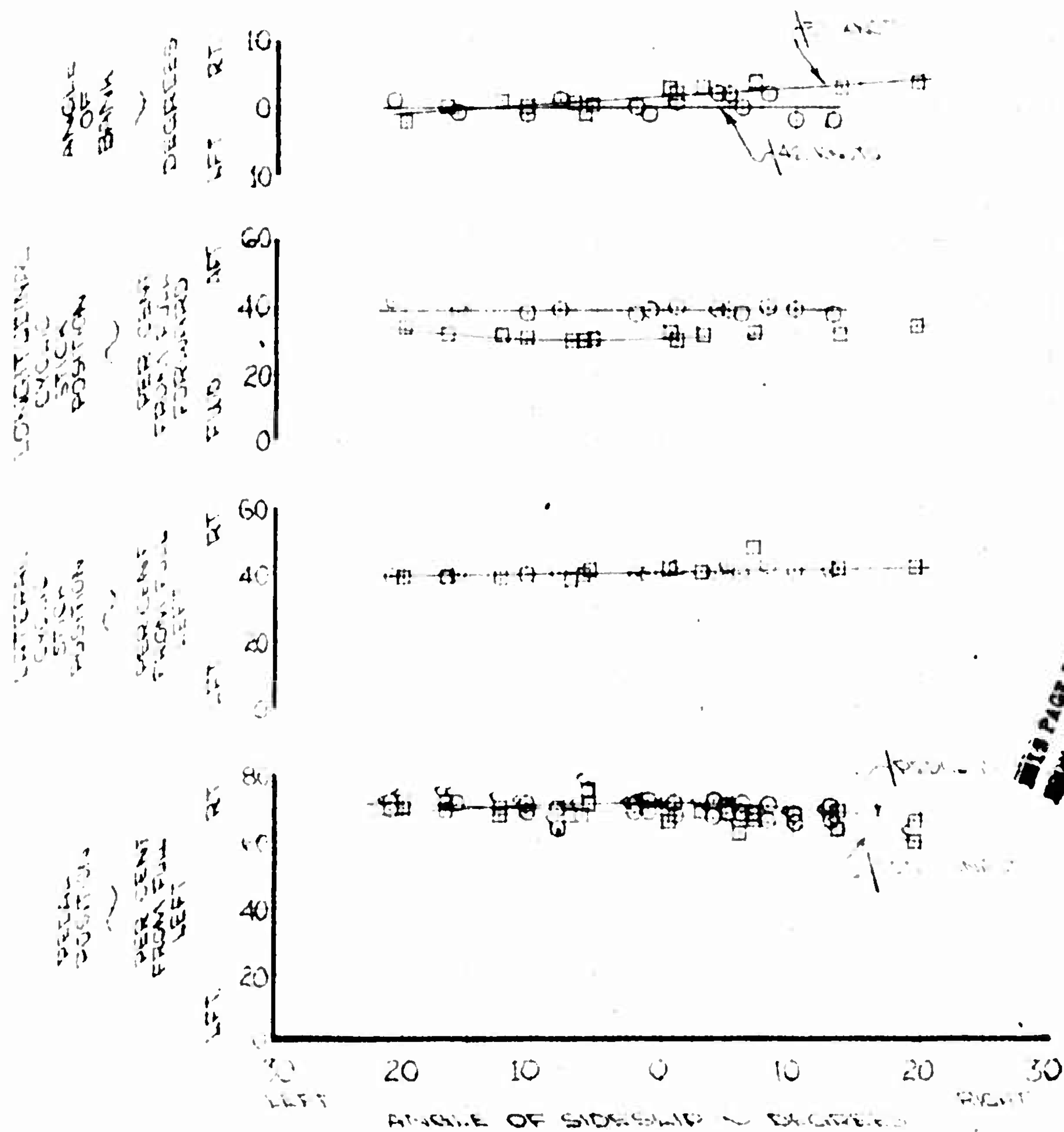


FIGURE NO. 15

CONTROL POSITIONS IN FORWARD AND REARWARD FLIGHT

CH-37B, U.S.A., S/N 54-0998

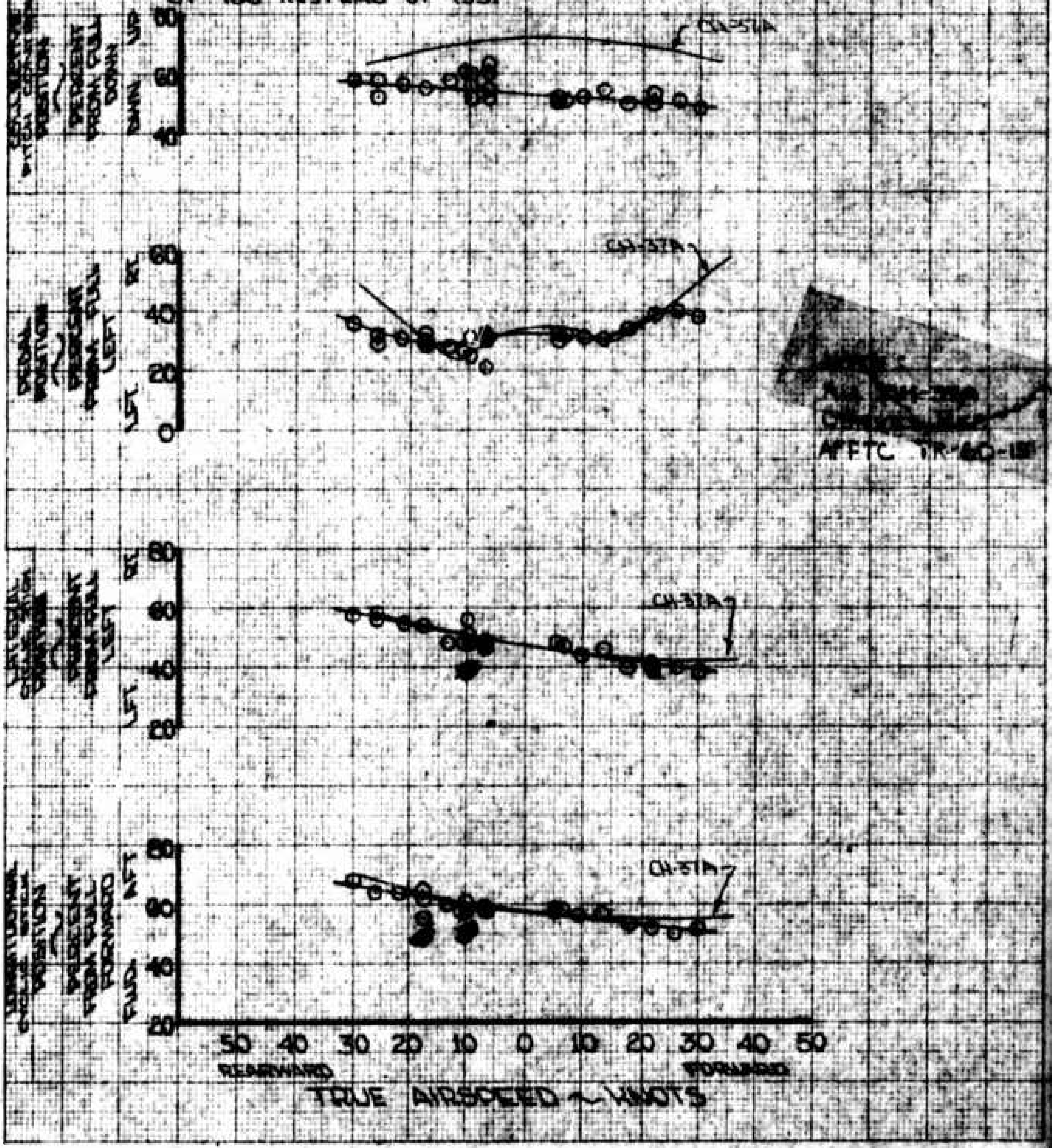
LEVEL FLIGHT ~ ASE OFF

GEAR DOWN, NO EXTERNAL STORES
AVERAGE DENSITY ALTITUDE = 1500 FT.
AVERAGE GROSS WEIGHT = 30,000 LB.
C.G. LOCATION = STATION 256.8 (MID)
RPM = 2700/193

IN GROUND EFFECT

CONTROL	FULL TRAVEL ~ INCHES	TRAVEL FOR ONE INCH ~ PERCENT
LONGITUDINAL	18.0	[illegible]
LATERAL	15.0	6.67
PEDAL	3.8	26.3
COLLECTIVE	10.0	10.0

NOTE: ALL CH-37A FLIGHT CONDITIONS COMPARE EXCEPT FOR A ROTOR RPM OF 189 INSTEAD OF 193.



DYNAMIC STABILITY

(See "Pulse" Time Histories)

Figures No. 16 - 33 Inclusive

2.17 through 2.34

FIGURE NO. 34

LONGITUDINAL CONTROL SENSITIVITY

CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED ~KNOTS	AVERAGE DENSITY ALTITUDE ~FEET	AVERAGE GROSS WEIGHT ~POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	1020	31,025	236.5 (MID)	2405/183
□	45	4200	30,850	242 (AFT)	2605/186
●	85	5810	30,925	242 (AFT)	2605/186

NOTE: 1. FULL LONGITUDINAL CONTROL TRAVEL = 16.0 INCHES.
2. ONE INCH LONGITUDINAL TRAVEL = 6.25 PER CENT
3. APPROXIMATE TIME REQUIRED TO REACH MAXIMUM ACCELERATION = 0.67 SECONDS @ HOVER AND 45 KNOTS, AND 0.74 SECONDS @ 85 KNOTS
4 ALL CH-37A CURVES REF. AFFTC TR.-60-15

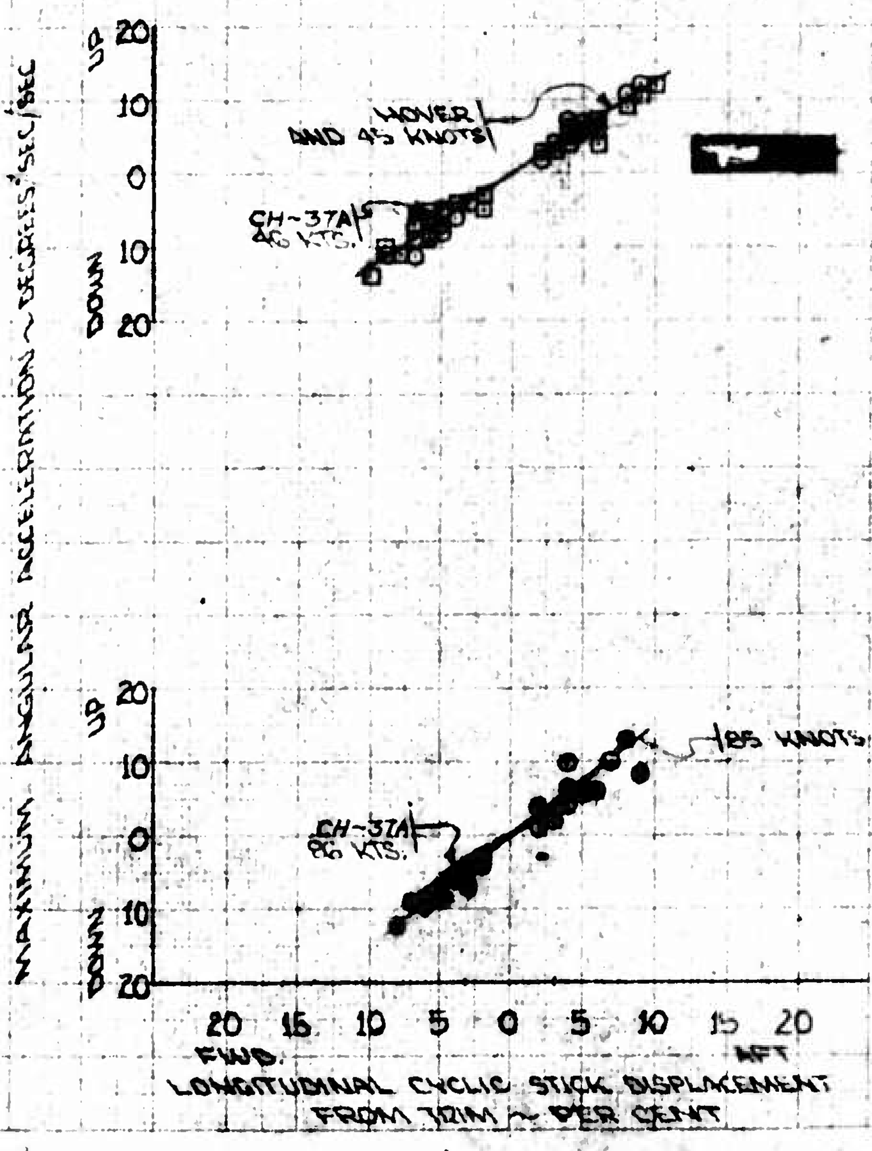


FIGURE NO. 35

LATERAL CONTROL SENSITIVITY

CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED ~KNOTS	AVERAGE DENSITY ALTITUDE ~FEET	AVERAGE GROSS WEIGHT ~POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	525	30,650	235.6 (MID)	2680/192
□	45	5870	30,675	242 (AFT)	2605/186
●	86	5235	30,575	242 (AFT)	2605/186

NOTE:
1. FULL LATERAL CONTROL TRAVEL = 15.0 INCHES
2. ONE INCH LATERAL TRAVEL = 6.67 PER CENT
3. APPROXIMATE TIME REQUIRED TO REACH MAXIMUM ACCELERATION = 0.68 SECONDS
4. ALL CH-37A CURVES REF AFFTC TR-60-15

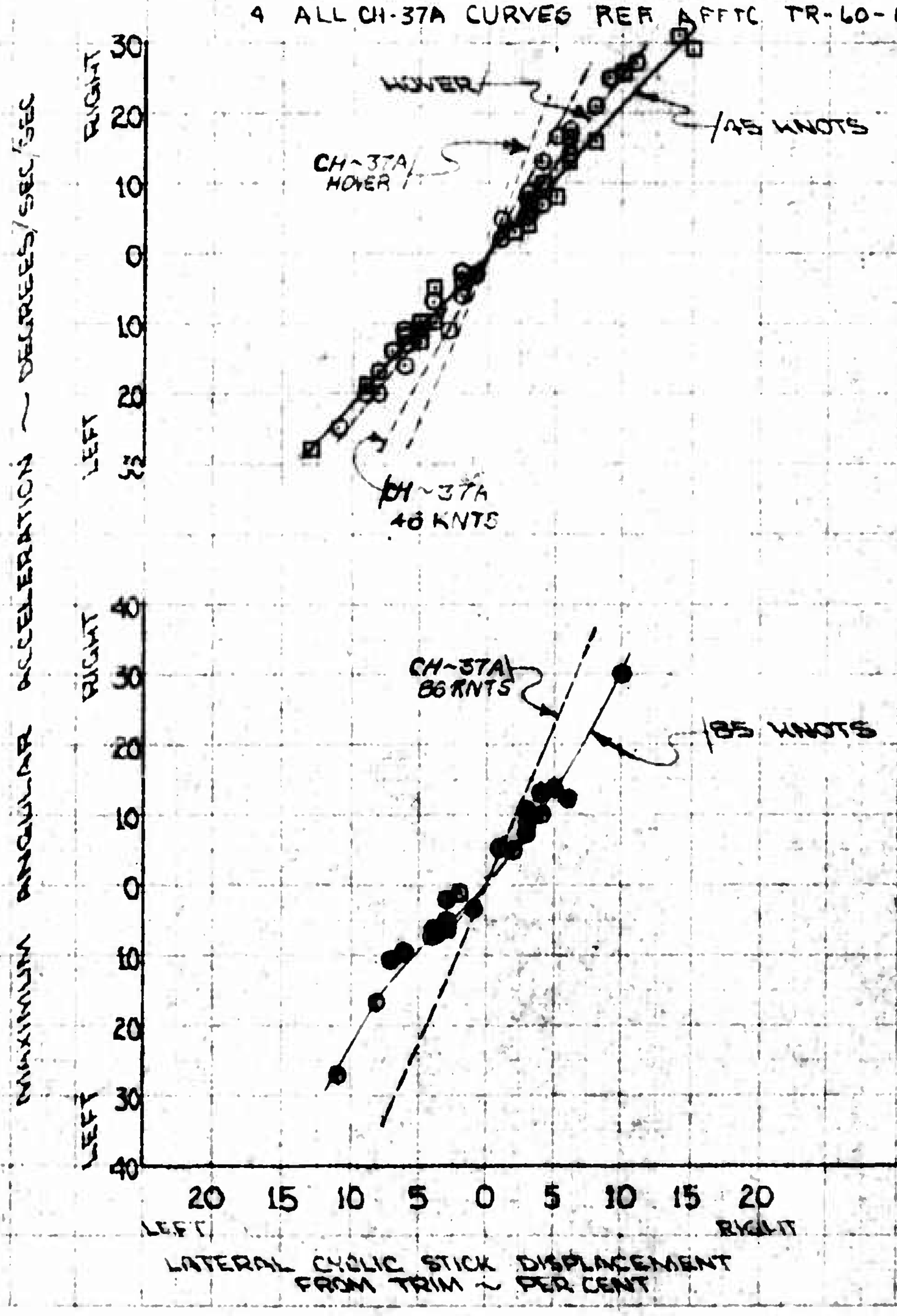


FIGURE NO. 36

DIRECTIONAL CONTROL SENSITIVITY
CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED ~ KNOTS	AVERAGE DENSITY ALTITUDE ~ FEET	AVERAGE GROSS WEIGHT ~ POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	1520	32,150	[illegible] (MID)	[illegible]
□	45	3960	30,650	242 (AFT)	[illegible]
●	86	3400	30,700	242 (AFT)	[illegible]

NOTE: 1. FULL DIRECTIONAL CONTROL TRAVEL = 3.8 INCHES
2. ONE INCH DIRECTIONAL TRAVEL = 26.3 PER CENT
3. APPROXIMATE TIME REQUIRED TO REACH MAXIMUM ACCELERATION = 0.80 SECONDS
4. ALL CH-37A CURVES REF. AFFTC TR-60-15

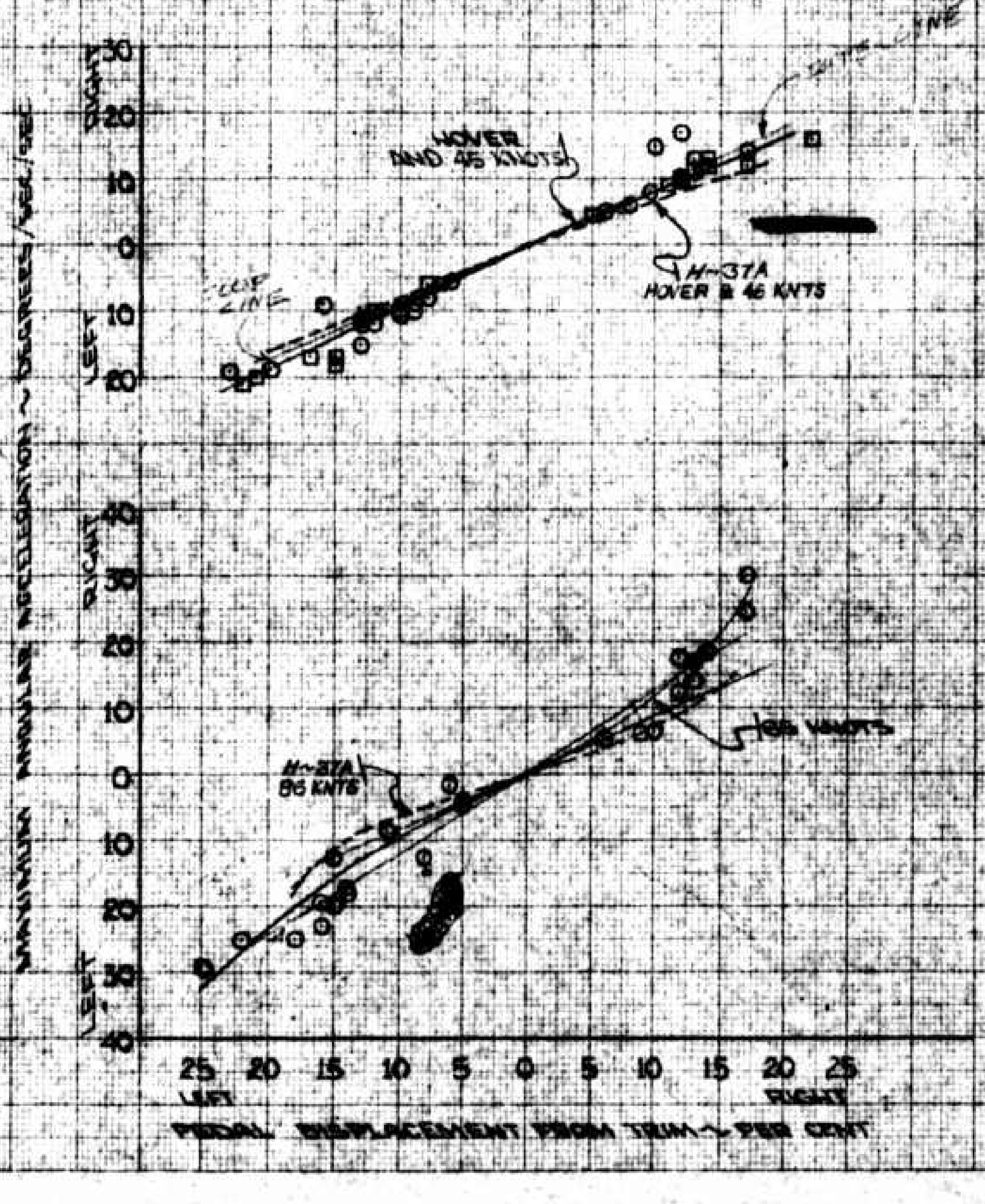


FIGURE NO. 37

LONGITUDINAL CONTROL RESPONSE

CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED ~ KNOTS	AVERAGE DENSITY ALTITUDE ~ FEET	AVERAGE GROSS WEIGHT ~ POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	1020	31,025	286.5 (MID)	2705/193
□	45	4200	30,850	242 (AFT)	2605/186
●	85	5810	30,925	242 (AFT)	2605/186

NOTE:
1. FULL LONGITUDINAL CONTROL TRAVEL = 16.0 INCHES
2. ONE INCH LONGITUDINAL TRAVEL = 6.25 PER CENT
3. APPROXIMATE TIME IN SECONDS REQUIRED TO REACH MAXIMUM RESPONSE

	HOVER	45 KNOTS	85 KNOTS
ASE ON	1.3	1.2	1.1
ASE OFF	2.8	2.5	2.5

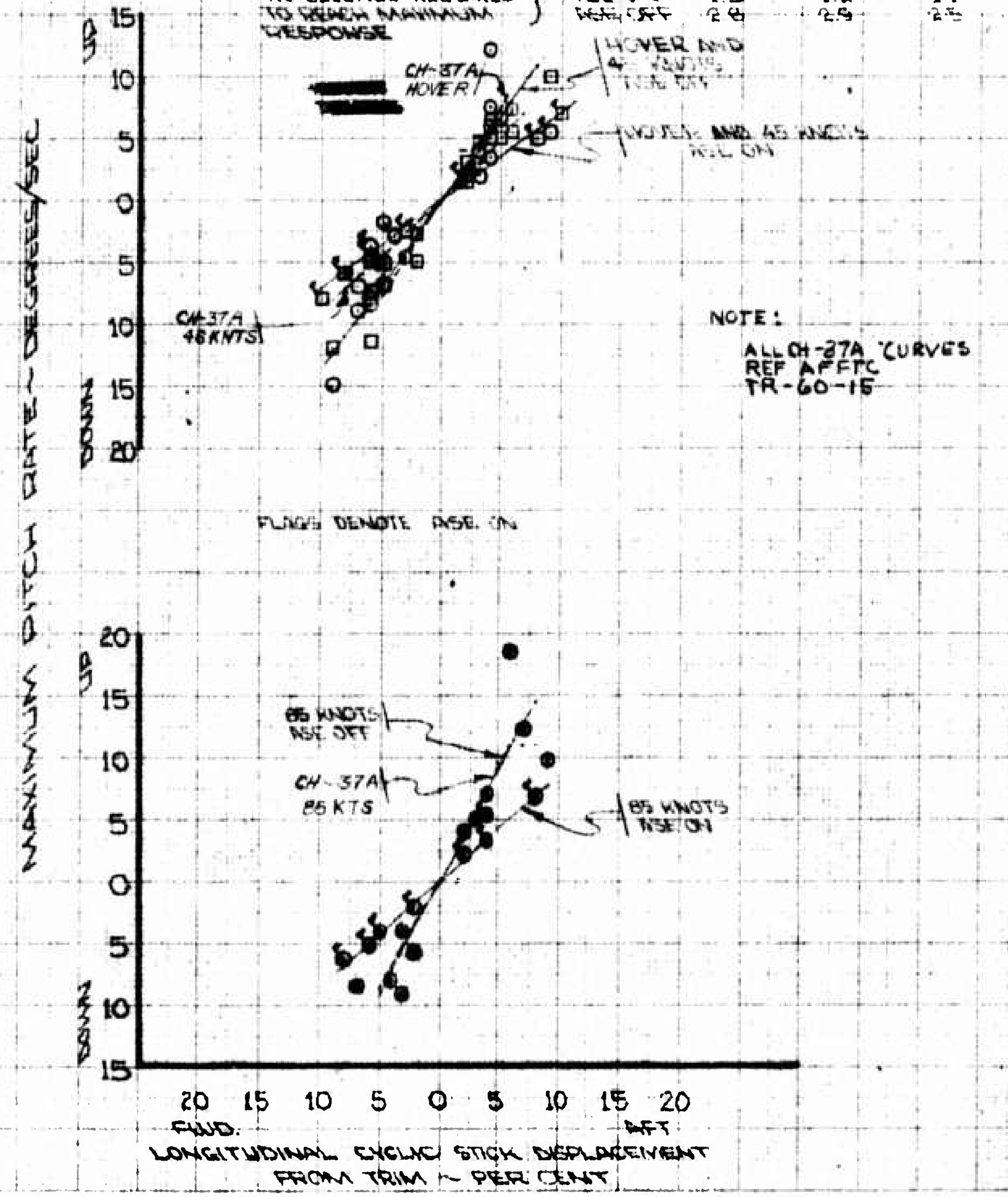


FIGURE NO. 38

LATERAL CONTROL RESPONSE

CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED ~ KNOTS	AVERAGE DENSITY ALTITUDE ~ FEET	AVERAGE GROSS WEIGHT ~ POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	525	30,650	235.6 (MID)	2680/192
□	45	5870	30,875	242 (AFT)	2805/186
●	85	2,375	30,975	242 (AFT)	2805/186

NOTE:
1. FULL LATERAL CONTROL TRAVEL = 15.0 INCHES
2. ONE INCH LATERAL TRAVEL = 6.67 PER CENT
3. APPROXIMATE TIME IN SECONDS REQUIRED TO REACH MAXIMUM RESPONSE.

	HOVER	45 KNOTS	85 KNOTS
ASE ON	1.1	1.1	1.0
ASE OFF	2.0	2.0	[illegible]

4. ALL CH-37A CURVES REF AFFTC TR-60-15

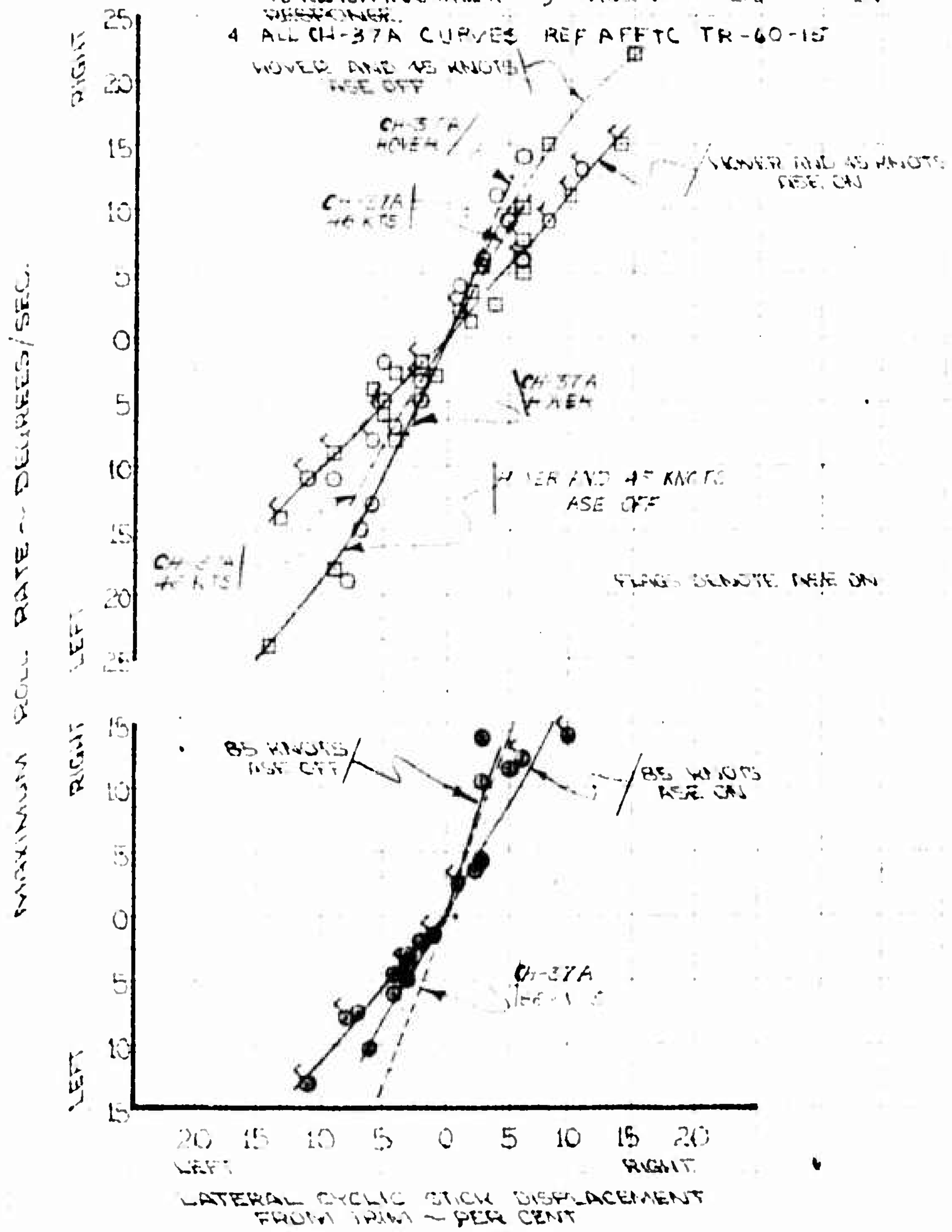


FIGURE NO. 39

DIRECTIONAL CONTROL RESPONSE

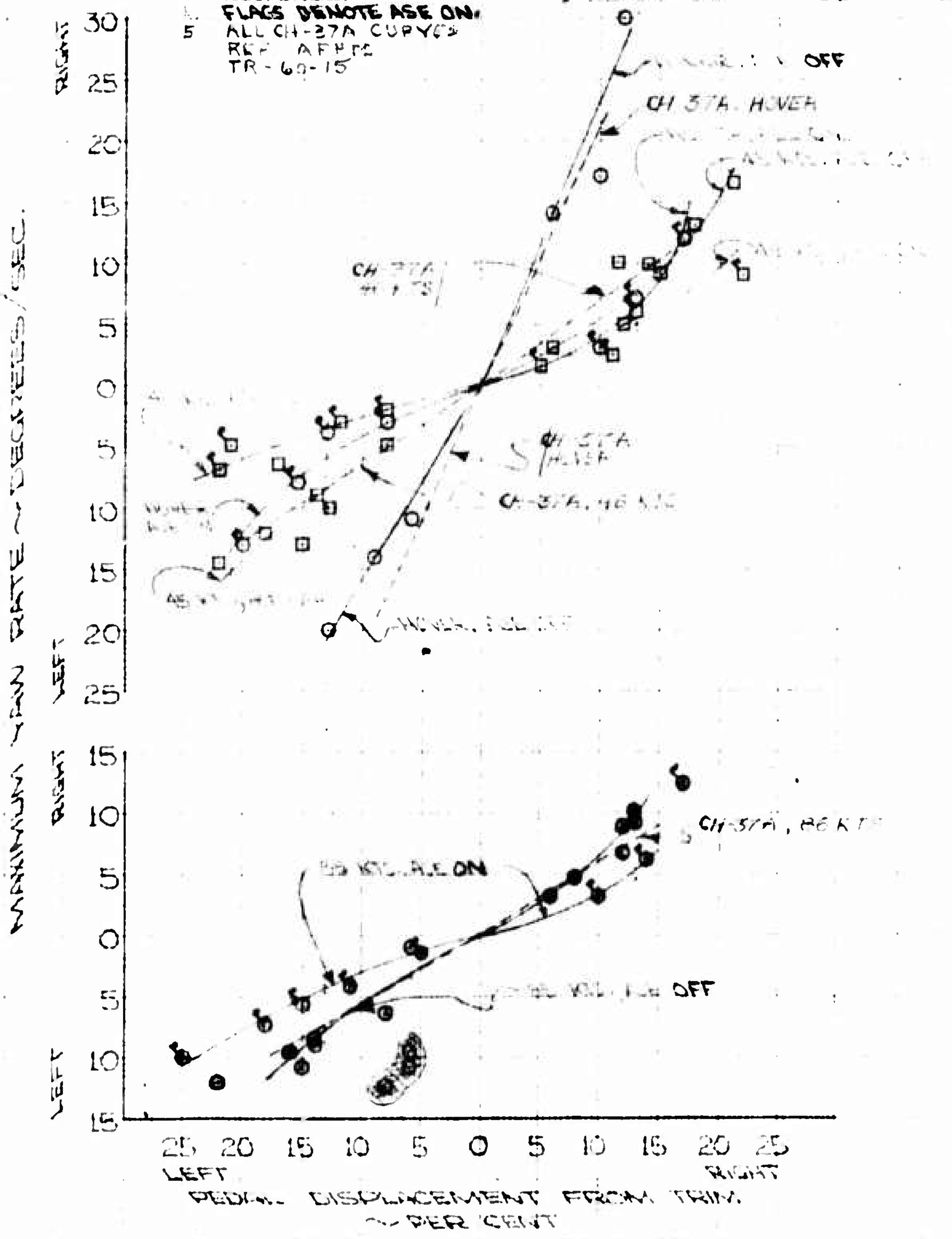
CH-37B, U.S.A., S/N 54-0998

SYMBOL	TRIM CALIBRATED AIRSPEED - KNOTS	AVERAGE DENSITY ALTITUDE - FEET	AVERAGE GROSS WEIGHT - POUNDS	C.G. LOCATION (STATION)	RPM
○	HOVER (IGE)	520	30,150	236.5 (MID)	2730/195
□	45	5960	30,650	242 (AFT)	2600/185.5
●	85	5400	30,700	242 (AFT)	2600/185.5

NOTE:
1. FULL DIRECTIONAL CONTROL TRAVEL = 3.8 INCHES
2. ONE INCH DIRECTIONAL TRAVEL = 26.3 PER CENT
3. APPROXIMATE TIME IN SECONDS REQUIRED TO REACH MAXIMUM RESPONSE.

	HOVER	45 KNOTS	85 KNOTS
ASE ON	0.9	1.3	1.1
ASE OFF	3.7	1.8	1.4

4. FLAGS DENOTE ASE ON.
5. ALL CH-37A CURVES REF AFFTC TR-60-15



CONTROLLABILITY

(See "Step" Time Histroies)

Figures No. 40 - 59 Inclusive

ASE FAILURES

(See Time Histories)

Figures No. 60 - 75 Inclusive

FIGURE NO. 76

BOOM AIRSPEED CALIBRATION

CH-37B, U.S.A., S/N 54-0998

NORMAL MIXTURE
GEAR DOWN, NO EXTERNAL STORES
RPM = 2600/185.5
AVG. GROSS WEIGHT = 29,000 LB.
C.G. LOCATION = STATION 236.5 (MID)
GROUND SPEED COURSE, LEVEL FLIGHT (OGE)

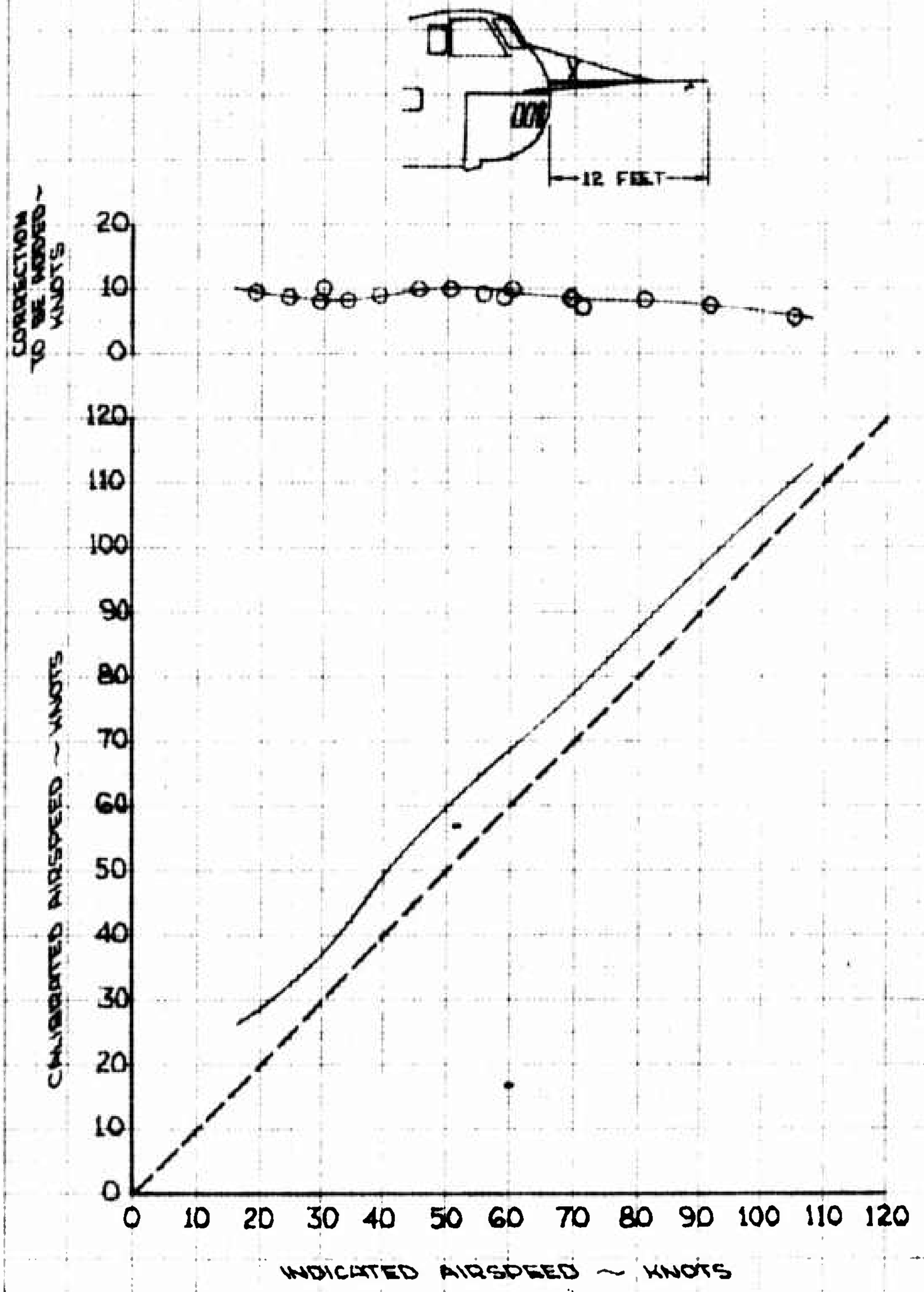


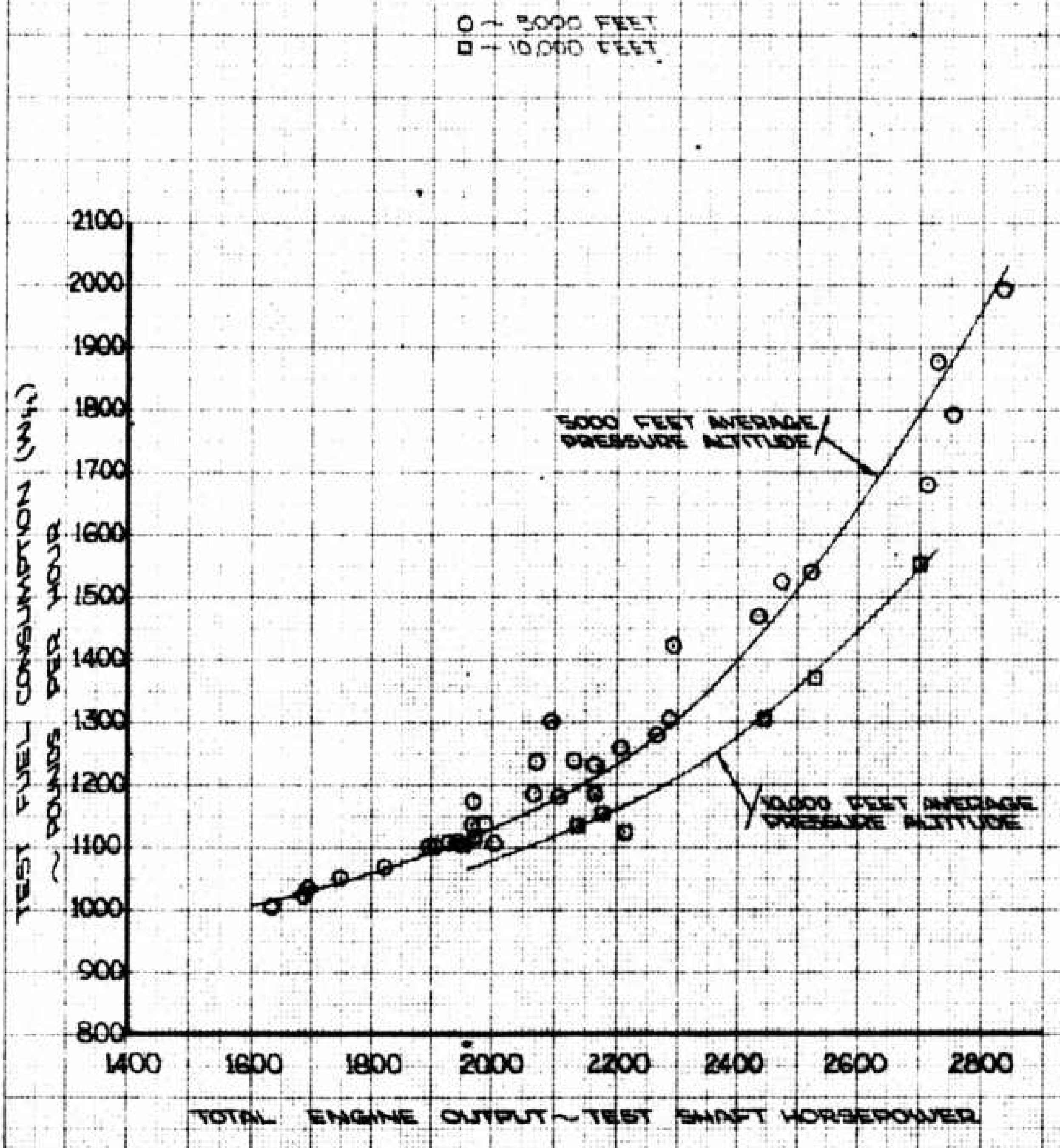
FIGURE NO. 77

FUEL CONSUMPTION

CH-37B, U.S.A., S/N 54-0998

NORMAL MIXTURE
COLD CARBURETOR AIR
RPM = 2600/1853

OBTAINED IN STABILIZED LEVEL FLIGHT
(REFER TO FIGURES 3 THROUGH 6)



PART III - ANNEXES

ANNEX A

REFERENCES

Information pertinent to this report was taken from the following list of references:

1. Flight Test Engineering Manual. AF Technical Report No. 6273 Revised Jan, 1953.

2. Helicopter Flying and Ground Handling Qualities, General Spec. for MIL-H-8501 and MIL-H-8501A.

3. Limited Stability and Control Evaluation, H-37A, AFFTC-TR-60-15, June 1960.

4. Limited Evaluation of the H-37A Equipped With Wide Chord Blades, AFFTC-TR-59-14, May 1959, ASTIA Document No. AD-214005.

5. The Effect of Gear Extension on Level Flight Performance of the H-37A, AFFTC-TN-58-27, October 1958, ASTIA Document No. AD-203728.

6. Operator's and Crew Member's Instructions; Army Model H-37A and H-37B Helicopters (Sikorsky), TM 55-1520-203-10.

7. Flying Qualities of the H-37B Helicopter With a High Pylon Stabilizer, (S1509-2000). Sikorsky Aircraft Report No. SER-56161.

8. Performance of H-37B with Lear ASE Installation; Sikorsky Aircraft Serial No. SER-56194, June 27, 1961.

ANNEX B

I. TEST TECHNIQUES AND DATA ANALYSIS METHODS

A. General

A brief description of testing techniques, methods of analysis, and the equations used in the correction of performance and stability and control data to Standard-Day conditions are outlined in this section.

B. Performance

The non-dimensional parameters used in the data analysis of the major items affecting helicopter performance are defined as follows:

$$C_P = \frac{BHP_t \times 550}{\rho_t A (\Omega R)^3}$$

$$C_T = \frac{W_t}{\rho_t A (\Omega R)^2}$$

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

Where:

C_P = Power Coefficient

C_T = Thrust Coefficient

μ = Rotor Tip Speed Ratio

BHP = Brake Horsepower - 33,000 ft.-lb/min

ρ = Air Density - slugs/ft.^{3}

A = Rotor Disc Area - sq. ft.

Ω = Angular Velocity - radians/sec.

R = Rotor Radius - ft.

W = Aircraft Gross Weight - lb.

V_T = True Airspeed - Knots

Subscripts used in this report are as follows:

s = standard

t = test

This nondimensional method assumes that there are no significant compressibility or blade stall effects on the rotor.

Constant W/ρ ratios corresponding to approximate density altitudes of 5000 feet and 10,000 feet were maintained during the level flight tests.

Torque meters, driven by the two main drive shafts, were found to produce data with excessive scatter, therefore brake horsepower was derived from the engine manufacturer's power chart illustrated in Figures No. 78 and 79, Part III. The manufacturer's ratings of power required to drive the engine cooling fan and generator (95 and 0.8 percent engine BHP respectively at 2600 engine rpm) were then subtracted to find shaft horsepower. The equation:

$$SHP_s = SHP_t \left(\frac{\rho_s}{\rho_t}\right)$$

was then used to correct test power to Standard-Day conditions.

True Speed was calculated using the following equation:

$$V_T = \frac{V_{calibrated}}{\sqrt{\sigma}}$$

In order to compare CH-37A and CH-37B performance data, the following was accomplished:

(1) Values of C_P, C_T, and μ were found for the CH-37A (from AFFTC-TR-59-14) at the same test conditions as the CH-37B.

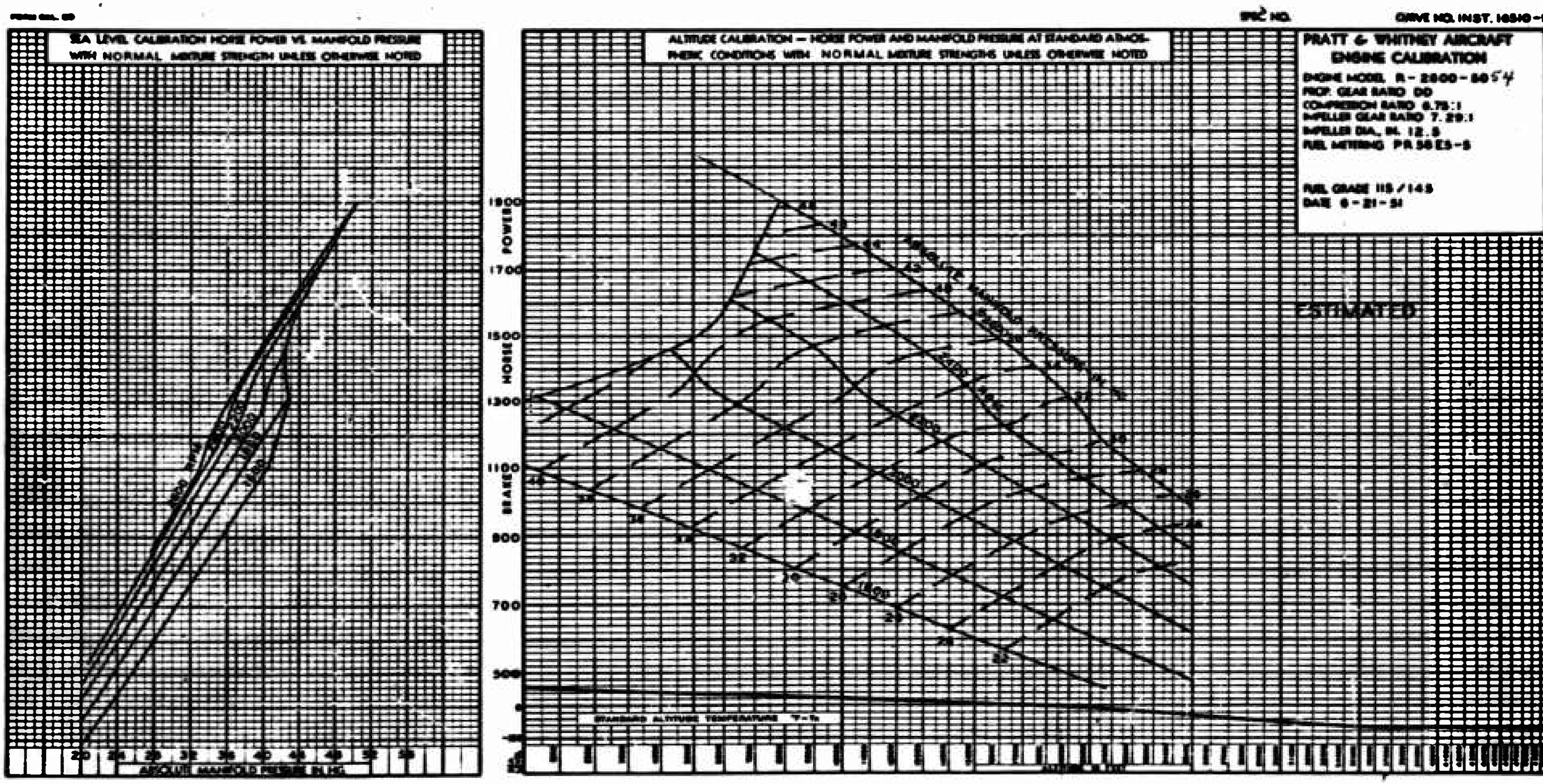
(2) CH-37B C_P and μ values were then obtained from the summary plots (Figures No. 1 and 2, Part II) at the CH-37A C_T values.

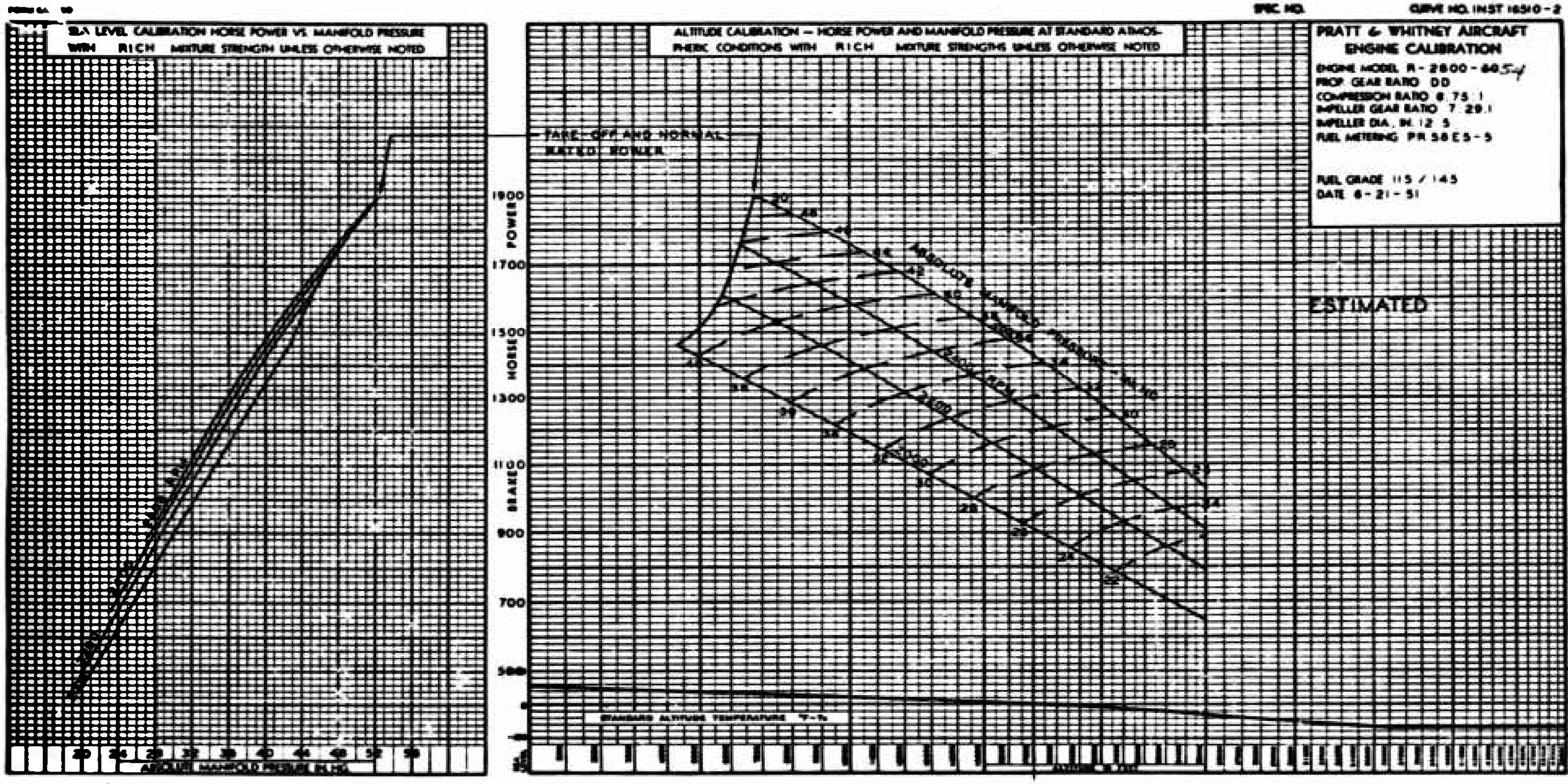
(3) The C_P and μ values for both aircraft were converted to SHP and V_T terms and then plotted at the common C_T values.

The resulting plots provided a comparison between the two helicopters as illustrated in Figure A, Part I.

Fuel flow data was reduced to specific range (nautical miles per pound of fuel) by the following method:

Specific range = V_T / W_{f_s}


SEA LEVEL CALIBRATION HORSE POWER VS. MANIFOLD PRESSURE
WITH NORMAL MIXTURE STRENGTH UNLESS OTHERWISE NOTED
ABSOLUTE MANIFOLD PRESSURE IN HG.
ALTITUDE CALIBRATION — HORSE POWER AND MANIFOLD PRESSURE AT STANDARD ATMOSPHERIC CONDITIONS WITH NORMAL MIXTURE STRENGTHS UNLESS OTHERWISE NOTED
BRAKE HORSE POWER
ABSOLUTE MANIFOLD PRESSURE IN. HG.
STANDARD ALTITUDE TEMPERATURE
ESTIMATED
NORMAL
CURVE NO. INST. 16510-1
PRATT & WHITNEY AIRCRAFT
ENGINE CALIBRATION
ENGINE MODEL R-2800-50 54
PROP. GEAR RATIO DD
COMPRESSION RATIO 6.75:1
IMPELLER GEAR RATIO 7.29:1
IMPELLER DIA., IN. 12.5
FUEL METERING PR 58 E5-5
FUEL GRADE 115/145
DATE 6-21-51


SPEC. NO.
CURVE NO. INST 16510-2
SEA LEVEL CALIBRATION HORSE POWER VS. MANIFOLD PRESSURE
WITH RICH MIXTURE STRENGTH UNLESS OTHERWISE NOTED
ALTITUDE CALIBRATION — HORSE POWER AND MANIFOLD PRESSURE AT STANDARD ATMOSPHERIC CONDITIONS WITH RICH MIXTURE STRENGTHS UNLESS OTHERWISE NOTED
PRATT & WHITNEY AIRCRAFT
ENGINE CALIBRATION
ENGINE MODEL R-2800-54
PROP GEAR RATIO DD
COMPRESSION RATIO 6.75:1
IMPELLER GEAR RATIO 7.29:1
IMPELLER DIA., IN. 12.5
FUEL METERING PR 58 E5-5
FUEL GRADE 115/145
DATE 6-21-51
ESTIMATED
RICH
TAKE-OFF AND NORMAL RATED POWER
ABSOLUTE MANIFOLD PRESSURE — IN. HG.
BRAKE HORSE POWER
1900
1700
1500
1300
1100
900
700
500
20
24
28
32
36
40
44
48
52
56
ABSOLUTE MANIFOLD PRESSURE IN. HG.
STANDARD ALTITUDE TEMPERATURE °F

3.6

Where:

w_f = fuel flow

Fuel flow values found under test conditions were corrected to standard conditions by use of the SHP_t versus W_{f_t} at each altitude. The plot was entered at test conditions, and, by moving parallel to the curve, a standard fuel flow corresponding to a BHPs was determined.

C. Stability and Control

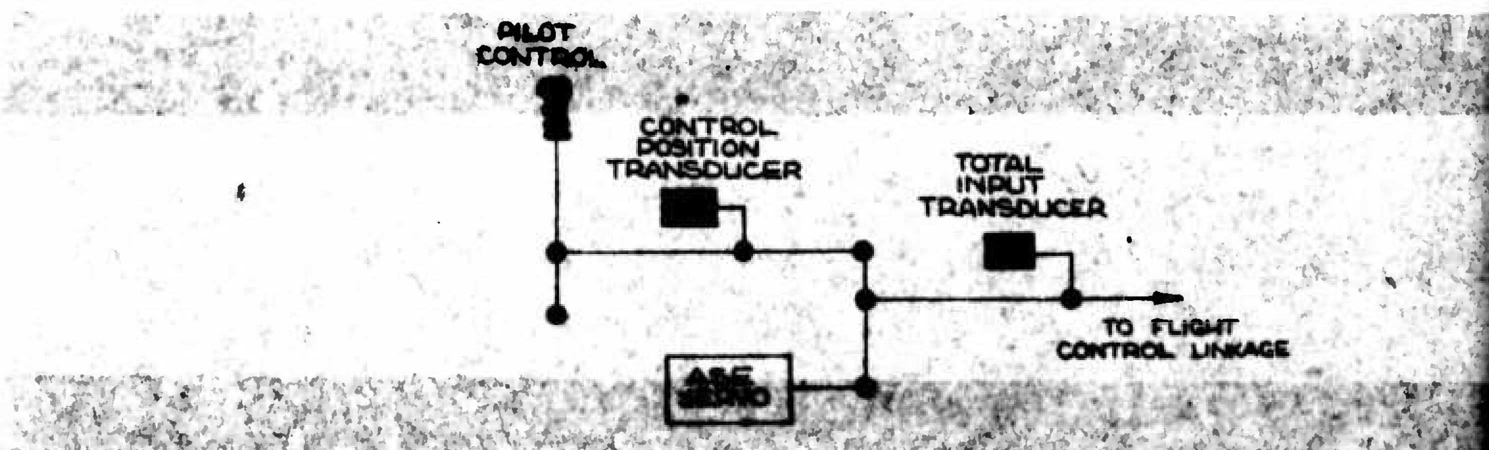
Static and dynamic stability and controllability of the aircraft was determined from an analysis of the helicopter motion resulting from pulse and step type control inputs. Tests were conducted for both ASE "off" and ASE "on" conditions.

Pulse inputs were obtained by rapidly displacing the control from trim position, holding the new position for a period of approximately 1 second, then returning the control to trim and maintaining this position until recovery was necessary. The linearity of the stability and controllability characteristics were determined by various size control inputs up to a maximum of more than one inch.

Precise inputs were insured by use of a control jig which was operated by the copilot. Each axis was investigated separately and all other controls were held fixed during the maneuver.

The effect of the Automatic Stabilization Equipment (ASE) on the total input was recorded through use of a pickup located directly behind the ASE system. This instrumentation feature made it possible to isolate the total input (which includes the ASE input) from the pilot input. A simplified illustration of this is presented in Figure E.

FIGURE E



(Minor movements in control position that are not evident from the total input position pickup can be attributed to "slop" in the mechanical and electrical linkages located between the control position pickup and the total input pickup.)

Simulated hard-over and oscillating hard-over type failures were evaluated by inducing the proper signals into the ASE system.

ANNEX C

DESCRIPTION OF MATERIEL

1. General

The CH-37B is a twin-engine, all-metal cargo/transport manufactured by Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut.

a. Power - Power is provided by two Pratt and Whitney R2800-54, 18 cylinder, twin-row, radial engines, each equipped with a single stage, single speed supercharger. Each engine is rated at 2100 BHP maximum power (5 minute limit) and 1900 BHP normal rated power (maximum continuous). The engines are mounted in nacelles at the ends of short wings which slant downward from the fuselage at an approximate 12.5 degree angle.

(1) Carburetion. Each engine has a rectangular barrel, pressure type, down draft carburetor equipped with automatic mixture control. Two carburetor air levers, mounted on the engine control quadrant, mechanically actuate doors in the carburetor air intake duct by means of control cables and linkages. Mixture control is available in three stages: rich, normal, and idle cut-off. The fuel priming system is an integral part of the carburetor. Fuel is directed to discharge nozzles up stream of the impeller section of the engines for starting the engine.

(2) Ignition. A low-tension type ignition system is provided for each engines. Direct current flows from the circuit breaker to the starter relay, to the induction vibrators, and then to the ignition switch. After the engine is started, the magneto supplies the power for firing the plugs.

(3) Cooling. Air is forced over the engines by engine-driven fans. Cooling air is necessary as ram air is not available during hovering and ground operations.

(4) Fuel System. A main fuel system and an auxiliary fuel system are provided for each engine. These systems are interconnected to permit use of all systems with one engine in case of an emergency. This also permits compensation for different fuel consumption rates between engines.

(a) Main Tanks. The two main tanks consist of two interconnected bladder-type fuel cells each, one located in the wing section and the other in the nacelle. Electrically operated sump-mounted booster pumps in each wing cell supply fuel under pressure to the system. Fuel flows from the tank through strainers and valves to the engine-driven fuel pumps and then to the carburetors.

(b) The Auxiliary Fuel System. This system consists of either two 150 gallon or two 300 gallon capacity drop tanks positioned approximately at cabin floor level and extending slightly forward of the wings. Auxiliary tanks were not used during this evaluation.

b. Transmission.

(1) Engine Drive Shafts. These shafts slant inboard and forward at approximately 10 degrees and are each splined to a hydro-mechanical rotor clutch.

(2) Clutches. The clutches are connected to the main gear box.

(3) Main Gear Box. This unit, containing a two-stage planetary gear system, reduces engine rpm at a ratio of 14.01:1.

(4) Tail Rotor Drive Shaft. Extending aft from the main gear box, this shaft drives the intermediate gear box.

(5) Rotor Brake. The rotor brake is located on the tail rotor drive shaft just aft of the main gear box.

(6) Intermediate Gear Box. Located at the base of the tail rotor pylon, this gear box changes the direction of the torque transmission to the tail rotor, and provides a disconnect point for folding the tail rotor pylon.

(7) Tail Gear Box. The tail gear box located at the top of the pylon contains a right-angle bevel gear reduction drive system to transmit engine torque to the tail rotor.

c. Rotor System

This sytem consists of the main rotor system and the tail rotor system, which are driven through the transmission system and tcontrolled by the flight control system.

(1) Main Rotor System. The main rotor head and the five main rotor blades make up the main rotor system.

(a) Main Rotor Head. The main rotor head supports the five main rotor blades and provides means for transmitting the movements of the flight controls to the blades. The following items comprise the main rotor head: the main rotor hub, which consists of an upper and lower plate, hinge assemblies, sleeve-spindle assemblies, and five dampers; the star assembly, which consists of a rotating star and a stationary star; restrainers; rods and assemblies; scissors; and locks.

(b) Main Rotor Blades. The five all-metal main rotor blades are constructed of aluminum alloy except for steel cuffs at the root ends. The chord length is 23.65 inches. The blade hinging is fully articulated. Restrainers and stops limit the motions. The leading edge of each blade is a hollow extruded spar; the trailing edge consists of individual pockets of honey-comb ribbed core construction bonded to the leading edge spar.

(2) Tail Rotor System. Four all-metal blades, a rotor assembly, and a pitch change mechanism make up the tail rotor system. The blades are fully articulated. The tail rotor drive shaft is hollow to facilitate the blade pitch changing mechanism.

d. Flight Control System. This system consists of a main control system, the cyclic stick trim system, the tail rotor flight control system, the flight control servo hydraulic system, and the automatic stabilization equipment (ASE).

(1) Main Rotor Flight Control System. This system provides longitudinal, lateral and vertical control by mechanical and hydraulic means. The cyclic stick changes the pitch of the main rotor blades to create lift as they rotate, thus effectively tilting the tip path plane and providing horizontal as well as vertical thrust. Hydraulically operated flight control servos assist the mechanical linkage.

(2) Cyclic Trim System. The cyclic trim system permits trimming of the cyclic stick by means of the two spring loaded struts connected to magnetic brakes.

(3) Tail Rotor Flight Control System. This control system compensates for main rotor torque and permits directional control. Control action is assisted by hydraulically operated flight control servos. Dampers prevent abrupt movements.

(4) Flight Control Servo Hydraulic System. The servo hydraulic system eliminates high stick forces and because of nonreverseability, reduces the main rotor vibratory loads.

(5) Automatic Stabilization Equipment (ASE). See Section C.2.a. for description.

2. Major Differences Between the CH-37B and the CH-37A.

Listed below are the items which might have had an effect on the performance, stability, or control of the CH-37B. These items are either changes or additions, as noted, to the CH-37A.

a. Automatic Stabilization Equipment (ASE) (Additional).

The purpose of the ASE is to improve the handling characteristics of the helicopter to permit automatic cruising flight and hands-off hovering. The ASE provides improved dynamic stability within the center-of-gravity limitations of the helicopter. The ASE incorporates four control channels: pitch, roll, yaw, and altitude. In each channel an appropriate electrical displacement signal is initiated, modified, and amplified to provide a control voltage for the servo motor assembly. The servo motor assembly actuates the helicopter's flight control system in such a manner as to dampen the helicopter's motion. The control action of the ASE is limited to approximately 25 percent of the range of the helicopter flight control system authority. An ASE block diagram is presented in Figure F, Part III.

b. Fixed Stabilizer. (Change)

The adjustable stabilizer located on each side of the aft fuselage section of the CH-37A was removed and a fixed stabilizer was attached on the CH-37B to the top right-hand side of the pylon opposite the tail rotor. The fixed stabilizer is installed with a 10 degree dihedral angle and a zero degree incidence setting.

FIGURE F.

PART III

c. Cargo Door (Change). The CH-37B has a sliding cargo door located in the aft section of the cabin on the right hand side of the fuselage. The door consists of forward and aft sections that ride on tracks above and below the door. This improved door replaces the 3-section CH-37A door.

d. Oil Tank (Change). A rigid fiberglass oil tank with a normal capacity of 30 gallons replaces the bladder-type oil tank of 13.3 gallons normal capacity which was incorporated in the CH-37A.

3. Dimensions and Design Data.

a. Main Rotor Disc Diameter 72 ft.

b. Tail Rotor Disc Diameter 15 ft.

c. Width (overall)

(1) Maximum (with rotors stationary)

- - - (approx) 68 ft. 5.75 in.

(2) Minimum (with rotors stationary)

- - - (approx) 65 ft. 1.5 in.

(3) Minimum (with main rotor blades folded or removed)

27 ft. 4.0 in.

(4) Width (at tail cone)

17 ft. 10 in.

d. Length (overall)

(1) Maximum (both rotors at extreme position)

88 ft.

(2) Minimum

(a) (Both rotors at minimum positions)

79 ft. 6.64 in.

(b) (Main rotor at minimum; tail rotor at extreme)

- - -(approx) 81 ft. 4.35 in.

(c) (Main rotor at extreme; tail rotor at minimum)

- - -(approx) 85 ft. 11.61 in.

(3) Minimum (blades and pylon folded)

55 ft. 8.0 in.

e. Height (overall)

(1) Maximum (tail rotor at high position)

22 ft.

(2) Minimum (tail rotor at low position)

20 ft. 0.23 in.

(3) Height (pylon folded at tail cone)

(rotor at 35.75 degrees)

15 ft. 1 in.

2. Main Rotor Blades

a. Number of Blades 5

b. Weight (approx) 350 lb.

c. Airfoil Section (curve identification) - NACA 0010.9

d. Total Blade Area (five blades)

- - - - - - - - - 287.5 sq. ft.

e. Area Per Blade - - - - - - - - - 57.5 sq. ft.

f. Area of Rotation (rotor Disc area)

- - - - - - - - - 4071.5 sq. ft.

g. Blade Radius - - - - - - - - - 36 ft.

h. Chord at Root - - - - - - - - - 23.65 in.

i. Chord at Tip - - - - - - - - - 23.65 in.

j. Disc Loading (at normal gross weight)

- - - - - -- -(Approx) 7.228 lb/sq ft.

k. Rotary Solidarity Ratio (effective)

- - - - - - - - 0.0865

l. Angle of Incidence in Neutral (all blades)

- - - - - - - - 10 deg. 33 min. at root

- - - - - - - - 6 deg. 48 min. at 75% chord

m. Ground Clearance (rotating)

Minimum ---------- (approx) 14 ft. 4 in.

n. Ground Clearance (static)

- - - - - (approx) 12 ft.

3. Tail Rotor Blades

a. Number of Blades - - - - - - - - 4

b. Airfoil Section - - - - - - - - NACA 0012 (modified)

c. Tail Blade Area (4 blades) - - - - - - - - 31.35 sq. ft.

d. Area Per Blade - - - - - - - - 7.8375 sq. ft.

e. Area of Rotation (rotor disc area) - - - 176.71 sq. ft.

f. Blade radius - - - - - - - - 7 ft. 6 in.

g. Chord at Root - - - - - - - - 13.5 in.

h. Chord at Tip - - - - - - - - 13.5 in.

i. Rotor Solidarity Ratio (total blade area divided by disc area)

- - - - - - - - 0.1774

j. Ground Clearance (rotating) - - - - - - 7 ft. 0.23 in.

k. Ground Clearance (static) - - - - - - 7 ft. 0.23 in.

4. Wing

a. Total Area, Including: - - - - - - 250.6 sq. ft.

(1) Fuselage - - - - - - 51.2 sq. ft.

(2) Nacelles (both) - - - - - - 187.2 sq. ft.

b. Chord at Root (aircraft centerline) - 5 ft. 8 in.

c. Chord at Tip (theoretical extended section) 5 ft. 8 in.

5. Horizontal Stabilizer (Fixed)

a. Area - - - - - - 24.5 sq. ft.

b. Span - - - - - - 6 ft. 5 in.

c. Chord at Root - - - - - - 46.44 in.

d. Chord at Tip - - - - - - 46.44 in.

e. Dihedral - - - - - - 10 deg.

f. Airfoil at Root - - - - - - NACA 14 0.0015 (MOD.)

g. Airfoil at Tip - - - - - - NACA 16 0.0009 (MOD.)

h. Angle of Incidence - - - - - - 0 deg.

6. Fuselage (without Main and Tail Rotor Blades)

a. Maximum Width (to outside of nacelles)

- - - - - - 27 ft. 4 in.

b. Maximum Length - - - - - - 64 ft. 10.69 in.

c. Height - Maximum (without landing gear)

- - - - - - 15 ft. 2.78 in.

Maximum (with landing gear)

- - - - - - 17 ft. 2 in.

d. Height of Door Level Above Ground (Static)

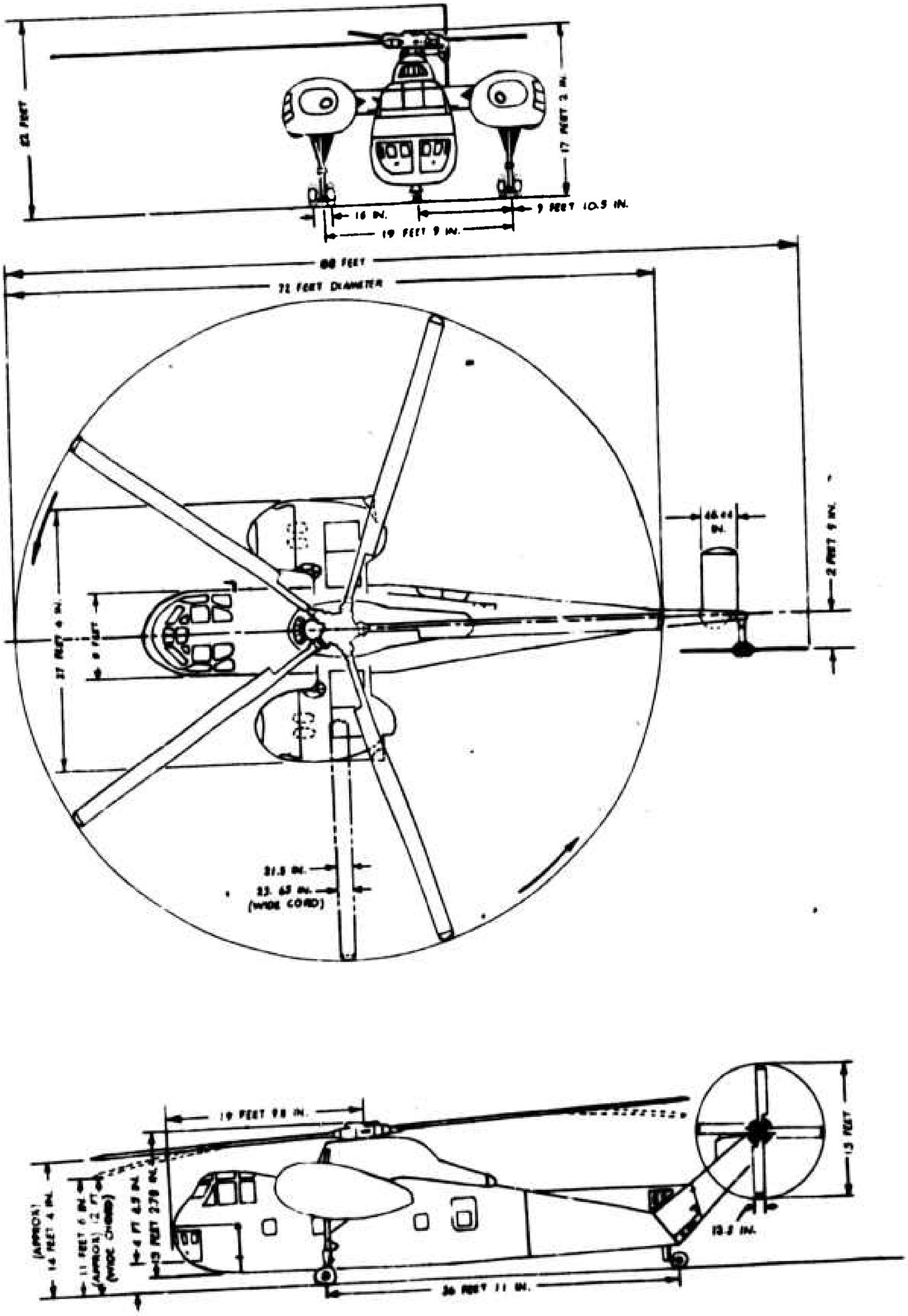
- - - - - - 2 ft. 11.4 in.

e. Door Dimensions (cargo)

(1) Width - - - - - - 5 ft. 9.8 in.

(2) Height - - - - - - 6 ft.

f. Total Cubic Feet of Cargo Space - - 1252.7 cu. ft.


17 FEET 2 IN.
16 IN.
9 FEET 10.5 IN.
19 FEET 9 IN.
88 FEET
72 FEET DIAMETER
27 FEET 4 IN.
2 FEET 9 IN.
21.5 IN.
(WIDE CORD)
13 FEET
13.5 IN.
36 FEET 11 IN.

ANNEX C

II. TEST INSTRUMENTATION - CH-37B

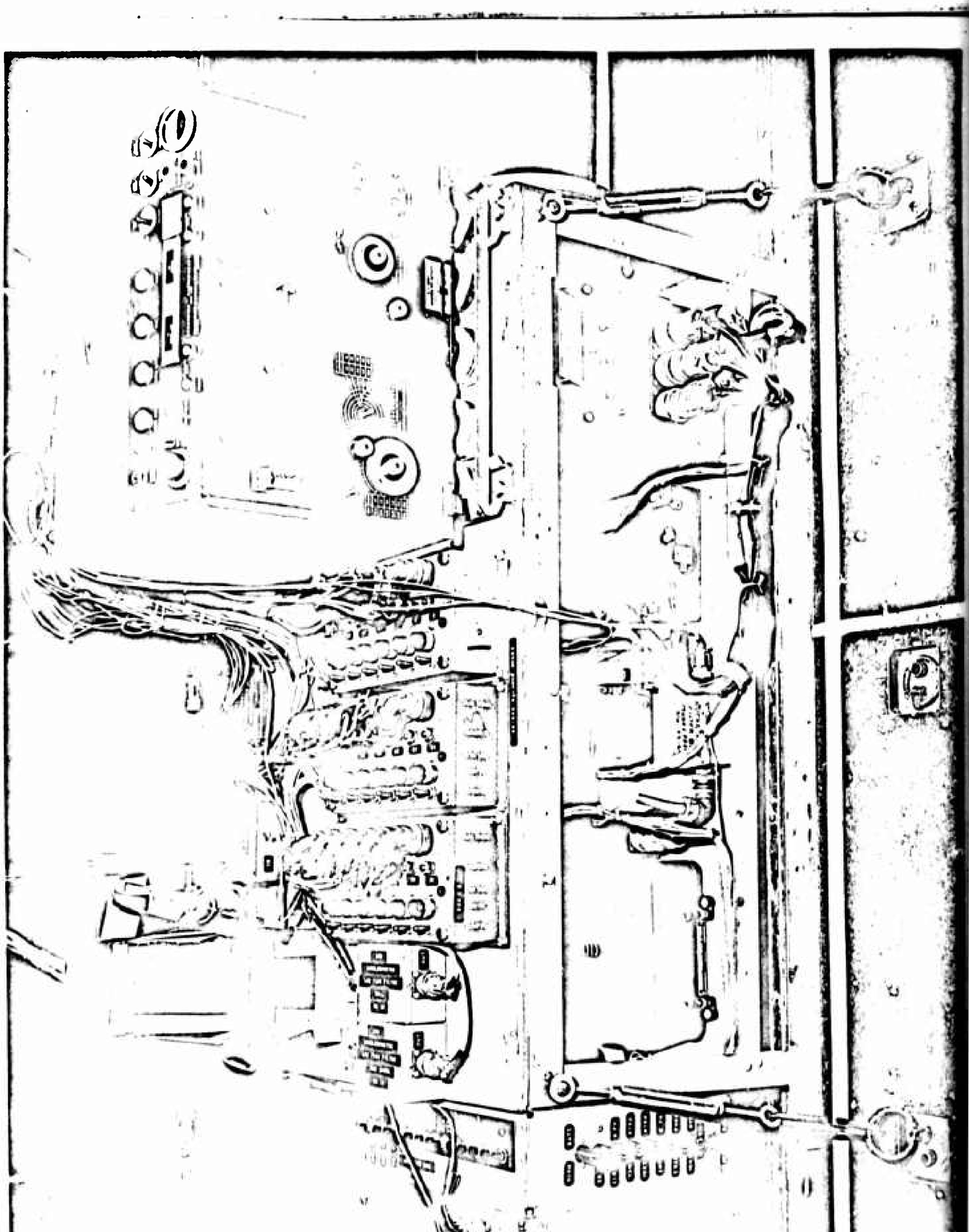
Sensitive instrumentation to measure the following parameters was supplied, installed, and maintained by the Instrumentation Branch of the Logistics Division, U. S. Army Aviation Test Activity. All of the instrumentation listed below was calibrated by the Air Force Flight Test Center with the exception of the items marked with an asterisk (*), which were calibrated by the USAATA.

A. Pilot's Panel

1. Airspeed (Boom)
2. Altitude (Boom)
3. Airspeed (STD)
*4. Free Air Temperature
5. Rate of Climb (Boom)
*6. Angle of Sideslip
7. Engine Speed (left and right engine)
8. Manifold Pressure (left and right engine)
*9. Carburetor Air Temperature (left and right engine)
10. Main Rotor Speed
11. Torque (left and right engine)
12. Total fuel Used (left and right engine)
13. Fuel Flow (left and right engine)

B. Oscillograph (illustrated in Figure H, Part III)

*1. Longitudinal Control Position
*2. Lateral Control Position
*3. Pedal Position
*4. Collective Pitch Position
*5. Angle of Attack
*6. Angle of Sideslip



7. Angle of Pitch
8. Angle of Roll
9. Angle of Yaw
10. Rate of Pitch
11. Rate of Roll
12. Rate of Yaw
13. Angular Acceleration in Pitch
14. Angular Acceleration in Roll
15. Angular Acceleration in Yaw
16. Normal Acceleration at the C.G.

*17. Boom Airspeed

18. Boom Altitude
19. Rotor RPM (linear)

*20. Total Longitudinal Control Input [1]

*21. Total Lateral Control Input [1]

*22. Total Directional Control Input [1]

[1]Total input is control position plus the respective ASE position.

ANNEX C

III. WEIGHT AND BALANCE

The following loadings were used during the performance, stability and control evaluation of the CH-37B:

A. 26,500 lb Mid C.G.

Basic Weight (Full Oil, 50 Gal)	23,259 lb
Crew (4)	800
Fuel	2,100
Ballast	541
	26,700 lb

B. 28,500 lb Mid C.G.

Basic Weight (Full Oil, 50 Gal)	23,259 lb
Crew (4)	800
Fuel	2,351
Ballast	2,090
	28,500 lb

C. 30,500 lb Mid C.G.

Basic Weight (Full Oil, 50 Gal)	23,259 lb
Crew (4)	800
Fuel	2,351
Ballast	4,090
	30,500 lb

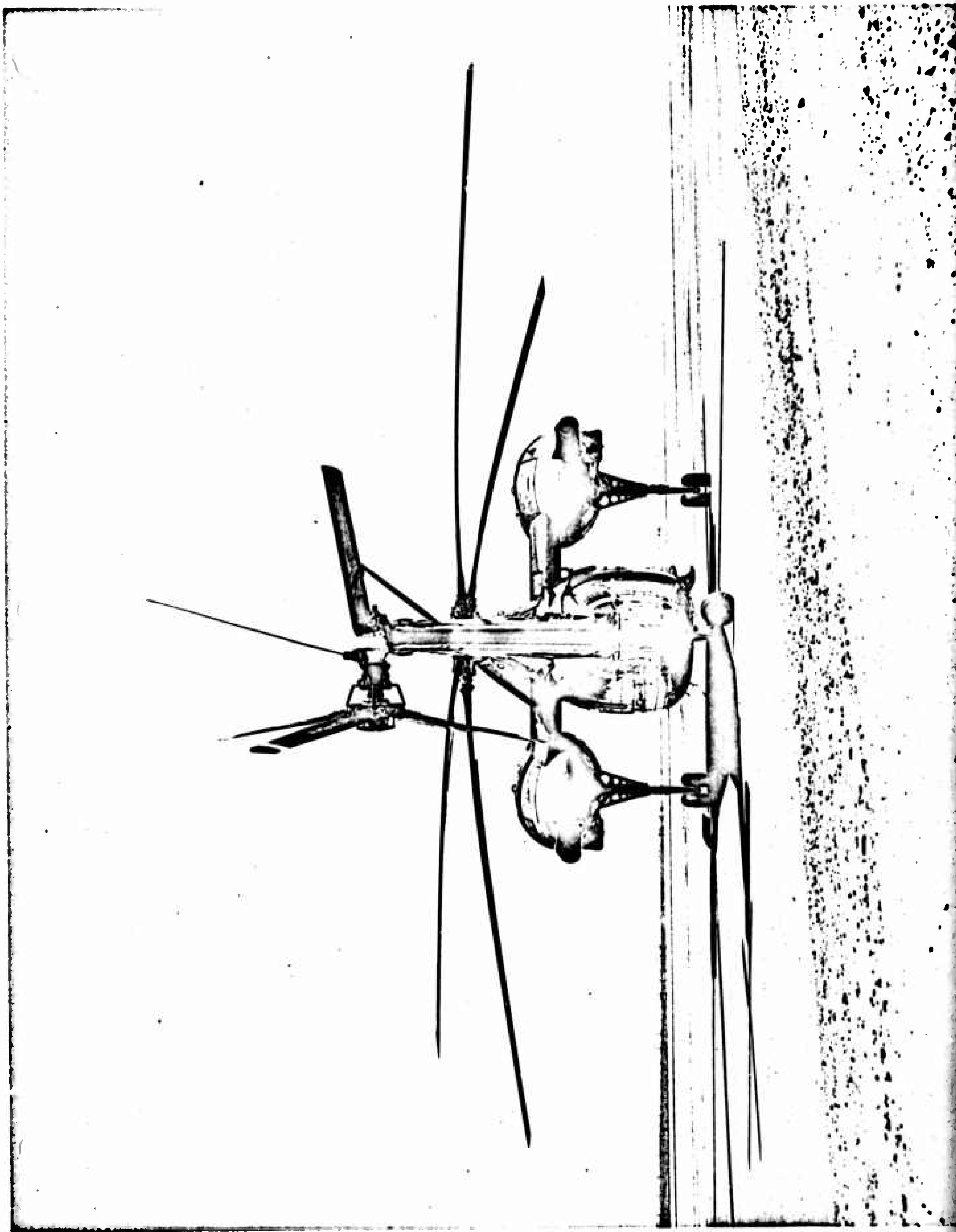
D. 31,500 lb Mid C.G. and Aft C.G.

Basic Weight (Full Oil, 50 Gal)	23,259 lb
Crew (4)	800
Fuel	2,351
Ballast	5,090
	31,500 lb

In all cases, the ballast was distributed in the cargo area to provide the desired C.G. location.

PART III

ANNEX D - PHOTOGRAPH



PART IV

DISTRIBUTION LIST

(Will be added at a later date)