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ARMY PRELIMINARY EVALUATION I
YO-3A AIRPLANE

FINAL REPORT

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AUGUST 1970

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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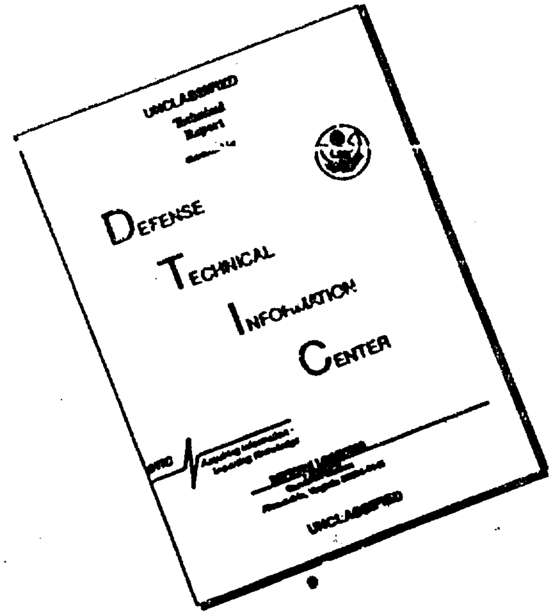

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ABSTRACT

The Army Preliminary Evaluation I (APE I) of the YO-3A airplane, manufactured by the Lockheed Missiles and Space Company of Sunnyvale, California, was conducted from 18 June through 9 July 1969. Twenty-seven test flights encompassing 36.0 flight hours were flown from Moffet Field Naval Air Station and Crows Landing Naval Auxiliary Landing Field, California. Performance and stability and control tests were conducted to determine the suitability of the airplane to perform a classified mission. The test airplane exhibited an excellent mission concept with potential for the satisfactory accomplishment of the intended mission. However, the test airplane as evaluated was not sufficiently developed for operational deployment. The test program revealed 8 safety-of-flight deficiencies, 28 deficiencies for which correction is mandatory, and 23 shortcomings for which correction is desirable. The engine and fixed-pitch propeller combination results in reduced performance which compromises the takeoff, climb and maximum speed capabilities of the YO-3A airplane. With the exception of the stall speed and maximum level-flight airspeed guarantees, the test airplane did not meet the performance guarantees listed in the Detail Specification. The performance capability of the test airplane was marginal; however, an acceptable airplane capable of performing the intended mission would result with the correction of the following deficiencies: engine overheating, excessive directional control forces, excessive control system breakout plus friction force, unintentional braking during the takeoff ground roll, lack of an acceptable mixture leaning indicator and lack of static power rpm limits. Stall characteristics were acceptable with all controls effective throughout the stall sequence. An adverse sideslip characteristic combined with high rudder force requirements compromises the satisfactory accomplishment of the intended mission.

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INTRODUCTION

BACKGROUND

1. The development of the YO-3A airplane was predicated on the results of a classified evaluation of the QT-2PC airplane. The QT-2PC airplane was designed and developed by the Lockheed Missiles and Space Company (LMSC) of Sunnyvale, California. The YO-3A airplane concept, also designed and developed by LMSC as a follow-on to the QT-2PC, was presented as an unsolicited proposal to the US Army Aviation Systems Command (USAAVSCOM). A cost-plus incentive fee contract was let by USAAVSCOM for the manufacture of the YO-3A airplane. The Army Preliminary Evaluation I (APE I) as directed by the USAAVSCOM test directive 68-47 (ref 1, app I) was conducted by personnel from the US Army Aviation Systems Test Activity (USAASTA). The APE I commenced at the completion of the contractor's Phase I flight test program. The security classification guide for this APE is presented in reference 15, appendix I.

TEST OBJECTIVES

2. The test objectives were:

a. To provide quantitative and qualitative engineering flight test data to serve as a basis for an estimate of airplane suitability for its intended mission.

b. To assist in determining the flight envelope to be used by Army pilots for future service tests.

c. To allow early correction of deficiencies, as well as provide a basis for evaluation of changes incorporated to correct deficiencies.

d. To provide preliminary airplane performance data for operational use.

DESCRIPTION

3. The YO-3A is a light-weight, fixed-wing, single-engine, two-place, observation airplane manufactured for the US Army by Lockheed Missiles and Space Company, Sunnyvale, California. The low-wing airframe is of all metal, semimonocoque construction (except for fabric ailerons and rudder, fiberglass engine cowl, aft canopy fairing, exhaust shroud, wing root fairings and wheel-well fairings). The electrically actuated, retractable main landing gear consists of two air and hydraulic main struts and wheels. The tail wheel mounted on a torsion tube is steerable and releases for full swivel after 30 degrees of tail travel. The cockpit is tandem configured with the observer's station forward of the pilot's station. Mechanically interconnected, reversible, conventional flight controls are provided in each cockpit. A detailed description of the flight control system is presented in appendix IV. The airplane is powered by an air-cooled fuel-injected Continental IO-360-D reciprocating engine rated at 210 shaft horsepower (shp) at standard day, sea level (SL), static conditions. Power is transmitted through a fixed-pitch six-bladed wooden propeller which is driven through a 3.33:1 pulley and belt reduction system.

SCOPE OF TESTS

4. The test airplane was evaluated to determine its suitability to perform a classified mission.

5. Flight restrictions and operating limitations issued by USAAVSCOM are presented in reference 2, appendix I. Test conditions are presented in appendix V.

6. Flight tests were conducted at Moffett Field Naval Air Station and Crows Landing Naval Auxiliary Landing Field, California, during June and July 1969. A total of 27 test flights were performed encompassing 36.0 hours of productive flight testing.

7. The Handling Qualities Rating Scale (HQRS) presented as appendix IX was used to augment the pilot's qualitative comments recorded during stability and control testing. The HQRS table was extracted from the National Aeronautics and Space Administration (NASA) report number TN-D-5153 (ref 3, app I).

METHOD OF TESTS

8. Performance and stability and control test techniques, as outlined in references 4 through 9, appendix I, and also in appendix VIII, were adhered to in obtaining, reducing and analyzing test data. Qualitative comments in conjunction with an analysis of quantitative data were used in formulating the conclusions and recommendations. Flights were conducted in nonturbulent and lightly turbulent atmospheric conditions to ascertain the flying qualities of the airplane under conditions representative of intended operations.

9. An airplane equipped with sensitive calibrated instrumentation was used to obtain the data presented in this report. A detailed list of cockpit, photopanel and oscillograph test instrumentation is presented as appendix III.

CHRONOLOGY

10. The chronology of the APE I is as follows:

Test directive received	29 January 1969
Test plan submitted	14 February 1969
Test plan approved	3 June 1969
Ground school and flight training at contractor's facility	10 through 18 June 1969
Test airplane accepted	19 June 1969
Tests initiated	19 June 1969
Tests completed	9 July 1969
Contractor debriefed	25 July 1969
Preliminary report submitted	22 September 1969

RESULTS AND DISCUSSION

GENERAL

11. The performance and stability and control tests were conducted under the test conditions and configurations listed in tables a and b, appendix V, respectively. The test airplane exhibited an excellent mission concept with potential for the satisfactory accomplishment of the intended mission; however, the airplane as evaluated was not sufficiently developed for operational deployment. With the exception of the stall speed and maximum level-flight airspeed guarantees, the test airplane did not meet the performance guarantees (table 1) listed in the detail specification (ref 11, app I). The performance characteristics of the test airplane were marginal but would be acceptable for the accomplishment of the intended mission after correction of the following deficiencies: engine overheating, excessive directional control forces, excessive control system breakout plus friction force, unintentional braking during the takeoff ground roll, lack of an acceptable mixture leaning indicator and lack of static power rpm limits. Test results revealed 8 safety-of-flight deficiencies, 28 additional deficiencies and 23 shortcomings. The engine and fixed-pitch propeller combination results in reduced performance which compromises the takeoff, climb and maximum speed capabilities of the YO-3A airplane. Stall characteristics were acceptable with all controls effective throughout the stall sequence. The adverse sideslip characteristic combined with high rudder-force requirements derogates the satisfactory accomplishment of the intended mission. An airplane-propeller gyroscopic coupling characteristic was exhibited which should be avoided in future designs. The spoiler control mechanism enhances mission effectiveness and should be included in future designs.

WEIGHT AND BALANCE DETERMINATION

12. Prior to commencing flight testing, a weight and balance determination was conducted on the instrumented YO-3A test airplane, S/N 69-18000. Weighing was accomplished in a hangar using the Cox and Stevens electronic scales located under the aircraft jack points. The results of the weight and balance are shown in table 2.

Table 1. Performance Guarantee Compliance Summary.

Condition[1]	Guarantee	Gross Weight 3500 Pounds With Mission Equipment Installed		Gross Weight 3500 Pounds Without Mission Equipment Installed		Gross Weight 3381 Pounds Without Mission Equipment Installed	
		Result	Percent Non-compliance (%)	Result	Percent Non-compliance (%)	Result	Percent Non-compliance (%)
Maximum speed, level flight, sea level	Not less than 120 knots	113.7 KTAS	5.3	118.5 KTAS	1.3	120.0 KTAS	Note[2]
Stall speed, idle power, sea level	Not greater than 66 knots	61.1 KTAS	Note[2]	61.1 KTAS	Note[2]	60.0 KTAS	Note[2]
Endurance at loiter speed of not less than 1.15 stall speed, idle power, 1000-ft altitude	Not less than 5 hours	3.31 hours	33.8	3.63 hours	27.4	3.96 hours	20.8
Range at cruise speed of not less than 80 KTAS, 1000-foot altitude	Not less than 390 nautical miles (NM)	231 NM	40.8	252 NM	35.4	261 NM	33.1
Rate of climb at maximum continuous power, sea level, landing gear retracted	Not less than 850 feet per minute (fpm)	595 fpm	30.0	655 fpm	23.1	664 fpm	21.9
Takeoff over 50-foot obstacle at maximum continuous power, sea level	Not more than 2300 feet	2588 feet	12.5	2588 feet	12.5	2430 feet	5.7

[1]Standard day.
[2]Guarantee satisfied.

Table 2. Weight and Balance.

Crew[1]	Fuel Tanks	Gross Weight (lb)	CG Location - FS (in.)
With	Empty	3364	85.62
	Full	3520	85.34
Without	Empty	3136	83.27
	Full	3292	83.08

[1]Crew consists of a 200-pound pilot and a 28-pound parachute located at fuselage station (FS) 117.9.

13. For the "with crew" conditions, a 0.28-inch forward shift in center of gravity (cg) location existed between fuel-empty and fuel-full conditions. With the takeoff restrictions specified in reference 2, appendix I, the cg takeoff limits as shown in table 3 were used. This allowed a maximum change of 2.18 inches in cg location for this evaluation. The cg takeoff limits actually experienced during testing were 84.66 to 86.08 in. FS.

Table 3. Takeoff Center of Gravity Limits.

CG Location Takeoff Limit	Gross Weight (lb)	CG Location - FS (in.)
Aft	3537	86.08
Forward	3580	83.90

PITOT-STATIC SYSTEM CALIBRATION

14. The boom and standard pitot-static systems of the test airplane were calibrated during stabilized flight utilizing a calibrated trailing bomb towed by a pace UH-1C helicopter. The method of calibration is discussed in appendix VIII, and the test conditions are presented in appendix V. For the calculation of all test airspeeds, the boom system calibration shown in figure 2, appendix II, was used. The standard system was calibrated in order to evaluate its serviceability. Within the scope of the tests, the ship's standard

pitot-static system is satisfactory for the intended mission. The test results are presented in figure 1, appendix II. The airspeed position error (ΔV_{pc}) is linear and has a value of +6 knots at mission cruise airspeeds from 75 to 80 KIAS.

PERFORMANCE

Takeoff and Landing Performance

15. Takeoff performance tests were conducted to determine total horizontal distance required to clear a 50-foot obstacle as a function of the true airspeed at lift-off. Ground roll distance was also determined as a function of the true airspeed at lift-off. Testing was conducted from a dry concrete runway with a downhill slope of 0.425 percent and under the conditions listed in appendix V. With the mixture control setting at full RICH, full throttle was applied and brakes released after the propeller rpm had stabilized. The contractor-recommended rotation and lift-off airspeeds were utilized with climbout at the lift-off airspeed. Lift-off airspeeds were decreased to just above stall speed where optimum results were achieved. The aircraft was rotated at 38 KIAS to a thrust-line-level attitude, and takeoff rotation was initiated at 57 KIAS. Climbout at 60 KIAS was accomplished at the maximum coefficient of lift as determined by the angle of attack indicator.

16. A Fairchild flight analyzer camera was utilized to record time histories of the takeoffs. Takeoff distance data have been corrected to SL, standard day, 3500-pound grwt and level runway as discussed in appendix VIII. Winds were calm during these tests. The test results are presented in figure 3, appendix II, as a function of ground roll distance and total distance required to clear a 50-foot obstacle versus true airspeed at lift-off.

17. The guarantee for the horizontal takeoff distance required to clear a 50-foot obstacle was 2300 feet. Testing was conducted without mission equipment installed. Corrections for gross weight were made as discussed in appendix VIII. The results of these tests are shown in table 4.

Table 4. Takeoff Distance Guarantee Compliance.

Gross Weight (lb)	Takeoff Distance to Clear 50 Feet (ft)	Percent Noncompliance (%)
3500	2588	12.5
3381	2430	5.7

18. The optimum technique for a maximum performance takeoff over a 50-foot obstacle was accomplished by holding the brakes locked while advancing the throttle to full power. Immediately upon achieving a stabilized rpm, the brakes were released, and a slight right directional yaw was experienced. Prolonged brake application while at the stabilized rpm only aggravated the engine overheating characteristic. At 30 KIAS, full forward stick was applied, and a thrust-line-level attitude was achieved at approximately 38 KIAS. The ground roll acceleration was continued until 57 KIAS was attained, and the airplane was then rotated to a slightly nose-high attitude. The airplane was accelerated to 60 KIAS, and this climbout airspeed was maintained until the obstacle had been cleared. During these tests, the gear remained in the down position.

19. The takeoff distances experienced during sea level tests will increase with density altitude (H_d). The high density altitudes experienced in Southeast Asia and the scarcity of sufficiently long runways dictate a requirement to define the YO-3A takeoff performance at density altitudes up to 5000 feet.

20. Qualitative landing performance tests were conducted in conjunction with the takeoff tests, and the landing roll distances achieved were representative of the airplane's landing performance. Maximum performance landings, as typified by the O-1 and U-6 airplanes, were not practical in this airplane because of the basic wing planform which does not include flaps and does not permit high sink rate landings.

21. The following technique was used to obtain minimum landing roll: incremental spoiler and power were used to maintain an airspeed of 75 KIAS and 100 feet of altitude on short final approach until the intended landing point was approximately 45 degrees below the horizontal. Full spoilers were then applied and throttle reduced to approximately 1800 rpm. The nose was lowered slightly and airspeed decreased until 3 to 5 feet above the intended touchdown point. Airspeed was then approximately 65 KIAS, and the

airplane was placed in a three point attitude for landing. Immediately upon touchdown, maximum braking was applied while maintaining directional control. Landing roll distances averaged 1400 feet.

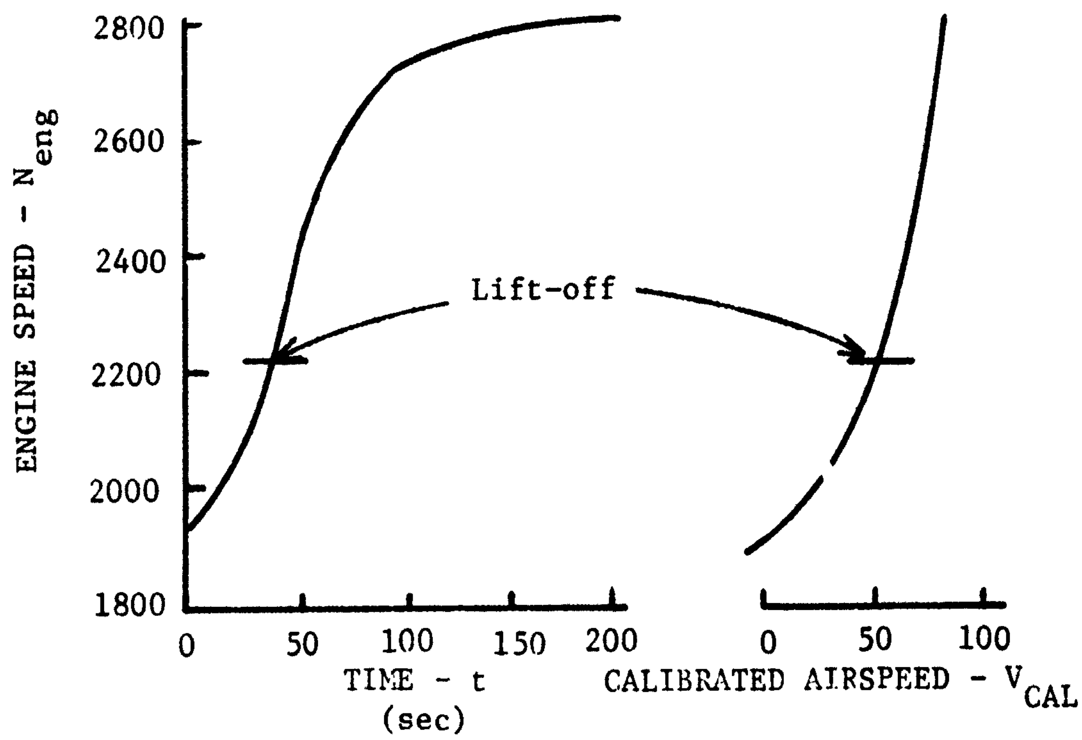
22. The recommended landing technique required that the pilot fly a 75 KIAS base leg and long final using spoilers as necessary to reach the desired touchdown point. Throttle was reduced to approximately 1600 rpm on final, and a gradual 70 KIAS descent was maintained until the landing flare was executed. Landing distances averaged 1800 feet. The spoilers permitted excellent glide slope control and consistent pinpoint landings. This feature enhances the mission effectiveness of the test airplane and should be included in future designs.

23. Crosswind landings were difficult to perform. A 6-knot, 90-degree crosswind component to the left of the landing runway resulted in the pilot having to hold left aileron and excessive right rudder force to maintain adequate runway alignment. The sideslip technique with spoilers extended, resulted in a large and sudden increase in rate of descent, and to counter this effect, spoilers were retracted. The large rate of descent was halted just prior to touchdown; however, without spoilers activated, the airplane had a tendency to balloon into the air after touchdown. At this point, lateral control was ineffective, and the crosswind effect caused the airplane to drift across the runway. Activation of full spoilers was then necessary to rapidly bring the airplane back to the runway surface so that the brakes and steerable tail wheel could be utilized to counter the crosswind effect. Crosswind takeoffs were equally disconcerting; in that, once a thrust-line-level attitude was achieved, the airplane had not yet developed sufficient lateral control effectiveness. The airplane then started to drift or skip across the runway. Due to the insufficient lateral control effectiveness, takeoffs and landings should not be conducted with a 90-degree crosswind component in excess of 10 knots. Correction of the poor crosswind landing and takeoff characteristics is mandatory for the satisfactory accomplishment of the intended mission (HQRS 7).

24. A time history of engine speed and calibrated airspeed during takeoff is presented as figure A. This figure reveals the unusually long acceleration period necessary before obtaining takeoff power. In order to achieve takeoff power at 2800 rpm, the airplane had to be accelerated to approximately 95 knots calibrated airspeed (KCAS). This condition occurred at 200 feet above ground level and almost 2 minutes after full throttle application and brake release. The normal procedure for airplanes with fixed-pitch propellers is to lock the brakes, apply full throttle and observe the indicated power (rpm). If takeoff power (rpm) is achieved,

Figure A. Takeoff Characteristics.
YO-3A, S/N 69-18000
Mission Equipment Installed

Config	Density Altitude (ft)	Gross Weight (lb)	Center of Gravity Location (FS) (in.)
TO	165	3530	86.08



the brakes are released and takeoff started. The inability of the YO-3A airplane to achieve takeoff power prior to airplane rotation is unsatisfactory. The feasibility of a constant-speed or two-speed propeller on the YO-3A airplane should be investigated. The takeoff and landing performance of the YO-3A airplane is marginally acceptable for accomplishment of the intended mission.

Level Flight Performance

25. Level flight performance tests were conducted to determine level-flight power required, range, endurance and maximum level-flight airspeed. The optimum values of range and endurance and their corresponding airspeeds also were determined. Test conditions are listed in appendix V. Level flight performance testing was conducted while recording data at stabilized airspeeds. The stabilized airspeeds were varied in approximate 10-knot increments from 60 KIAS to maximum level-flight airspeed as determined by maximum power available. During this test sequence, a

noticeable and objectionable engine/propeller vibration was experienced during flight at 70 to 80 KIAS. Correction of this deficiency is mandatory for the satisfactory accomplishment of the intended mission.

26. Test results are presented in appendix II for power and fuel flow requirements. For power requirements, the most basic results presented are generalized power-velocity (P_{iw}-V_{iw}) plots for mission equipment installed, mission equipment not installed and cowl-open/cowl-closed comparison in figures 4, 5 and 6, appendix II, respectively. Also presented are engine brake horsepower required versus true airspeed plots for these same configurations (figures 7, 8 and 9, respectively) at a 3500-pound grwt. Coefficient of lift versus coefficient of drag curves (C_L-C_D) are presented in figures 10 and 11.

27. Fuel flow requirements are presented, as specific range and specific endurance versus true airspeed with and without mission equipment installed, in figures 12 and 13, appendix II, respectively.

28. There were three contractor guarantees on the YO-3A level flight performance: maximum speed, level flight, SL; range at cruise speed at not less than 80 KTAS, 1000-foot altitude; endurance at loiter speed of not less than 1.15 stall speed, idle power, 1000-foot altitude, all at standard day conditions.

29. The guarantee for maximum level-flight airspeed at SL was 120 KTAS. Testing was accomplished without mission equipment installed. Corrections for gross weight and mission equipment were made according to the methods discussed in appendix VIII. The results are shown in table 5.

Table 5. Level-Flight Guarantee Compliance.

Mission Equipment	Gross Weight (lb)	Maximum Airspeed (KTAS)	Percent Noncompliance (%)
Installed	3500	113.7	5.3
Not installed	3500	118.5	1.3
Not installed	3381	120.0	Satisfactory

30. The guarantee for range at the 1000-foot altitude was 390 NM at the airspeed for 0.99 maximum specific range but not less than 80 KTAS. Testing was done with the fuel mixture setting at full RICH, with and without mission equipment installed. An easily operated and accurate fuel/cruise guide indicator is essential for optimum range and endurance performance. Correction of this deficiency is mandatory for satisfactory mission accomplishment. Corrections for gross weight were made according to the methods discussed in appendix VIII. The resultant curve of specific range is shown in figure 14, appendix II. The results are shown in table 6. The airspeeds shown are recommended for maximum range.

Table 6. Range Guarantee Compliance.

Mission Equipment	Gross Weight (lb)	Range (NM)	Percent Noncompliance (%)	Airspeed (KTAS)
Installed	3500	231	40.5	85.3
Not installed	3500	252	35.4	80.0
Not installed	3381	261	33.1	80.0

31. During the test program it became evident through an investigation of power available and power required that the maximum specification engine shaft horsepower was not being obtained. In numerous tests the propeller rpm redline restriction was the limiting parameter. Rarely, if ever, were both the manifold pressure and propeller rpm limits achieved simultaneously. The lack of a calibrated engine and engine power-loss test data, to substantiate the curves provided by LMSC, prevented the precise determination of engine power characteristics which are so vital for evaluating performance guarantees.

32. The guarantee for endurance at the 1000-foot altitude was 5.0 hours at an airspeed not less than 1.15 times stall airspeed. Testing was done both with and without mission equipment installed. Corrections for gross weight were made according to the methods discussed in appendix VIII. The resultant curve of specific endurance is shown in figure 14, appendix II. The results are shown in table 7. The airspeeds shown are recommended for maximum endurance. The level flight mission performance of the YO-3A airplane is mediocre without an operational fuel/cruise guide indicator. This results in a marginally acceptable airplane for accomplishment of the intended mission.

Table 7. Endurance Guarantee Compliance.

Mission Equipment	Gross Weight (lb)	Endurance (hr)	Percent Noncompliance (%)	Airspeed (KTAS)
Installed	3500	3.31	33.8	[1]70.3
Not installed	3500	3.63	27.4	[1]70.3
Not installed	3381	3.96	20.8	[2]69.0

[1]1.15 stall airspeed at a 3500-pound grwt.
[2]1.15 stall airspeed at a 3381-pound grwt.

Climb Performance

33. Sawtooth climb tests were conducted to determine rates of climb and the airspeeds for maximum rates of climb as a function of pressure altitude. The test conditions are presented in appendix V. Using full throttle, sawtooth climbs were performed with airspeed varied in approximate 10-knot increments from 70 KIAS to the maximum airspeed at which a positive rate of climb was achieved.

34. The rate of climb has been corrected for temperature, power and weight variations as discussed in appendix VIII. The test results are presented in figure 15, appendix II, as rate of climb versus calibrated airspeed for the different pressure altitudes (Hp) at which tests were performed. This figure shows the maximum rate of climb airspeed to be 85 KCAS based on data up to a 5000-foot H_P.

35. To check the maximum rate of climb guarantee at SL, the test maximum rate of climb was plotted versus pressure altitude and the curve extrapolated to SL (fig 16, app II). The SL rate of climb guarantee was 850 fpm. Testing was done with mission equipment installed. Corrections for gross weight and mission equipment installed were made according to the methods discussed in appendix VIII. The results are shown in table 8. The climb performance of the YO-3A airplane is marginally acceptable for accomplishment of the intended mission.

Table 8. Sea Level Rate of Climb Guarantee Compliance.

Mission Equipment	Gross Weight (lb)	Rate of Climb (fpm)	Percent Noncompliance (%)
Installed	3500	595	30.0
Not installed	3500	655	23.1
Not installed	3381	665	21.9

Descent Performance

36. Sawtooth descent tests were conducted to determine rates of descent and airspeeds for minimum rate of descent as a function of pressure altitude. The test conditions are presented in appendix V. Using idle power, airspeed was varied in 10-knot increments from 70 KIAS to 100 KIAS. The results were corrected for temperature and weight variations in the same manner as the sawtooth climb data. The resulting rates of descent versus calibrated airspeed for the different pressure altitudes are presented in figure 15, appendix II. This plot shows that the minimum rates of descent were not achieved during testing and that the rate of descent will decrease as the stall speed is approached. The descent performance of the YO-3A airplane is satisfactory for accomplishment of the intended mission.

Stall Performance

37. Stall performance tests were conducted to determine calibrated stall airspeeds ($V_{cal_{stall}}$) for the configurations and conditions presented in appendix V. Stall speed was determined by trimming the airplane at 1.2 times the stall airspeed and then decreasing airspeed at slightly less than 1 knot per second until the stall occurred.

38. Stall speeds have been corrected for weight variations as discussed in appendix VIII. The results are presented in table 9.

39. The guaranteed stall speed was 66 knots at SL in the glide configuration. Testing was conducted with mission equipment installed. Corrections for gross weight were made according to appendix VIII. The installation of mission equipment does not affect stall speed. The true airspeed results for guarantee comparison are shown in table 9. Stall performance results

for all configurations are shown in table 10. The stall performance of the YO-3A airplane is satisfactory for accomplishment of the intended mission.

Table 9. Stall Performance Guarantee Compliance.

Gross Weight (lb)	Stall Speed (KTAS)	Percent Compliance (%)
3500	61.1	7
3381	60.0	9

Table 10. YO-3A Stall Performance.[1]

Configuration	Calibrated Airspeed at Stall Gross Weight 3381 Pounds (knots)	Calibrated Airspeed at Stall Gross Weight 3500 Pounds (knots)
Cruise	56.3	57.3
Power approach	57.0	58.0
Land	57.4	58.4
Glide	60.0	61.1
Takeoff	57.2	58.2
Dive	61.4	62.4

[1]Mid cg location (FS 85.37).

STABILITY AND CONTROL

Static Longitudinal Stability

40. Static longitudinal stability tests were conducted at constant throttle settings under the conditions and configurations specfied in appendix V. Test results are presented in figure 17, appendix II. When trimmed at, or below, the minimum power required airspeed, the stick force gradient decreased gradually and exhibited a neutral to slightly positive gradient as airspeed was decreased from trim (HQRS 4). The breakout plus friction force characteristic masked the actual stick force gradient primarily at airspeeds below

trim and below the minimum power required airspeed. At trim airspeeds greater than the minimum power required airspeed, the variation of stick force with velocity exhibited a positive gradient throughout the range of airspeeds tested. At these trim airspeeds, the positive stick force gradient provided the pilot with satisfactory airspeed cues (HQRS 3). Correction of the static longitudinal instability tendency at airspeeds below trim and below the minimum power required airspeed is desirable for improved mission accomplishment.

41. The gradient of stick position versus airspeed was only slightly positive and provides the pilot with minimal airspeed cues (fig. 17, app II). The longitudinal control stick position variation over a 20-knot airspeed change was approximately 0.5 inch. Therefore, the pilot uses stick force as the primary airspeed cue due to the shallow gradient of stick position versus airspeed (HQRS 5).

Neutral Point Determination

42. An analysis of the static longitudinal stability test results was conducted to determine the stick position and stick force neutral points. The results achieved were inconclusive due to the small changes in cg location.

Dynamic Longitudinal Stability

43. Dynamic longitudinal stability tests were conducted under the conditions and configurations specified in appendix V. Test results are presented in figures 18 through 21, appendix II. During short period qualitative dynamic testing, the airplane exhibited an aperiodic subsidence after a "doublet" input. The heavily damped short period motion provides the pilot with satisfactory angle of attack or attitude response characteristics during maneuvering tasks (HQRS 3).

44. Dynamic longitudinal long-period characteristics were excited by releasing the stick at an airspeed approximately 10 knots below the trim airspeed. The long-period or phugoid motion of the test airplane, during all configurations tested, with the exception of the power (climb) configuration, exhibited a negatively damped oscillation as shown in table 11 and figure B. The long-period oscillation, by itself, was not bothersome to the pilot but it did contribute to a considerable increase in pilot workload while attempting to stabilize at a specific trim airspeed (HQRS 5). The 62-pound downspring, while alleviating a static longitudinal instability characteristic, could contribute to the long-period divergence and should be avoided in future designs. Correction of the slightly divergent long-period oscillation is desirable for improved service use.

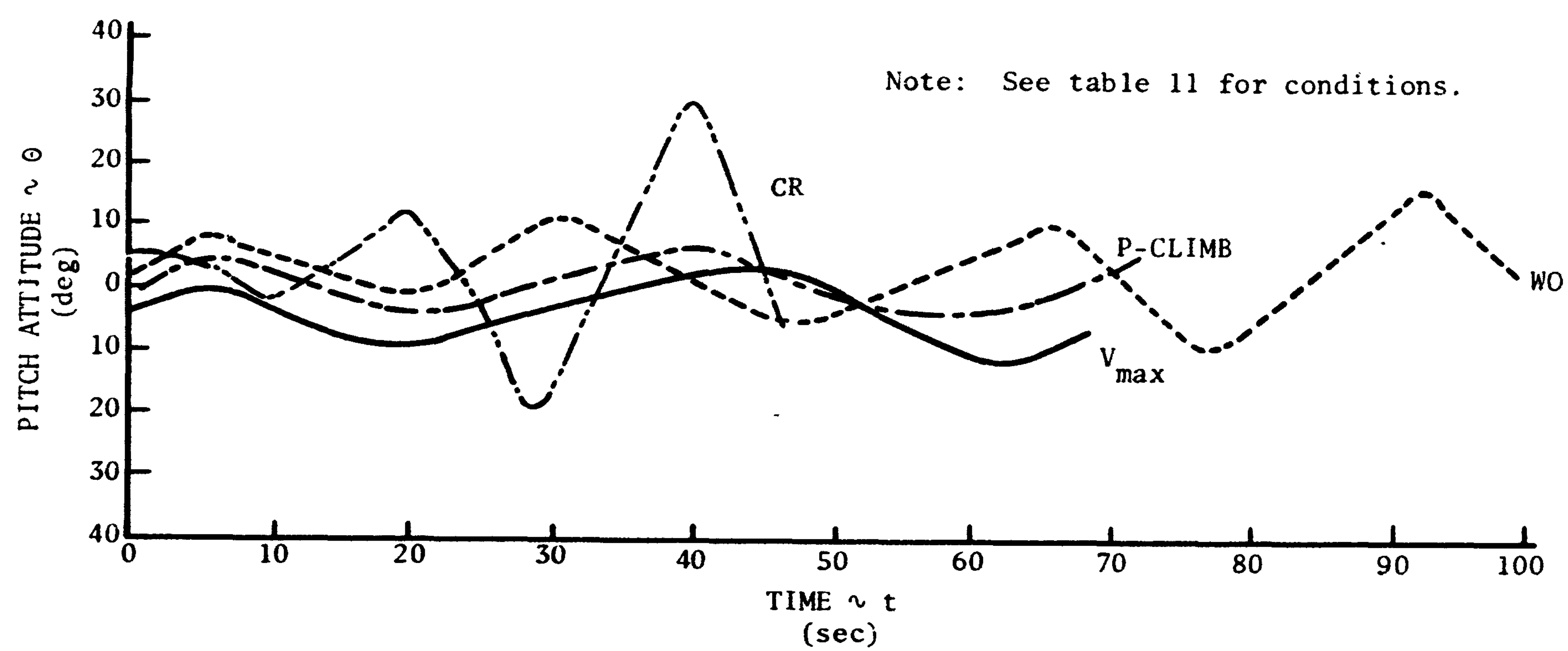
Table 11. Longitudinal Long-Period Characteristics - Stick Free.

Configuration	Trim Airspeed (KTAS)	Density Altitude (ft)	Gross Weight (lb)	CG Location (FS) (in.)	Damping Ratio ζ	Period T (sec)	Undamped Natural Frequency ω_n (rad/sec)	Cycles to Double Amplitude C_2 (cycles)
Cruise	63.0	5000	3500	85.37	-0.221	18	0.358	0.486
Waveoff	68.0	6460	3500	86.08	-0.083	30	0.210	1.324
Power climb	82.0	6460	3500	86.08	+0.010	35	0.178	[1]20.0
Cruise	92.5	6460	3500	86.08	-0.070	38	0.165	1.57

[1]Cycles to one-half amplitude ($C_{1/2}$).

Figure B. Summary of Long Period Characteristics.

YO-3A S/N 69-18000



Static Lateral-Directional Stability

45. Steady-heading sideslip tests were conducted under the conditions and configurations specified in appendix V. Test results are presented in figures 22 through 26, appendix II. The variation in rudder force with sideslip revealed a positive directional stability characteristic. The variation in rudder force with sideslip angle exhibits a gradient which is linear about a trim airspeed. As the sideslip limits are approached, the rudder force gradient tends to decrease, and a definite trend towards reversal is exhibited (figs. 22 and 24). Although rudder force lightening at the sideslip limits was evident in the test data, the effect was not noticeable to the pilot. The gradient of rudder position versus sideslip angle is essentially linear throughout the sideslip envelope.

46. The variation in lateral stick force with sideslip angle reveals an essentially neutral gradient. This characteristic is indicative of neutral effective dihedral. A similar and approximately identical trait was exhibited by the variation in lateral stick position with sideslip angle. Evaluating effective dihedral by attempting to raise a lowered wing with opposite rudder was inconclusive since propeller gyroscopic effects masked any dihedral analysis (HQRS 4). The neutral gradients of lateral stick force and lateral stick position with sideslip angle do not meet the requirements of reference 13, appendix I. This characteristic is not objectionable but should be avoided in future designs.

47. Bank angle variation with sideslip, a side force characteristic, was essentially linear and positive throughout the range of sideslip angles tested. For identical ranges of sideslip angles tested, the bank angle required in right sideslip was twice as large as the bank angle required in left sideslip. Since the variation in lateral stick position and stick force versus sideslip angle was essentially neutral, the side force characteristic was primarily due to the influence of the propeller, wings and fuselage. The positive sideforce characteristic provides the pilot with acceptable cues (the balance ball) during coordinated turning maneuvers (HQRS 3).

48. The variation in longitudinal control force versus sideslip angle was linear through the trim point in left sideslips; however, the gradient reversed at a 5-degree right sideslip. The longitudinal stick pull-force required in left sideslips (right rudder) is apparently a manifestation of the propeller aerodynamic moments. During right sideslips, a reversal from a longitudinal push-force at a 5-degree sideslip to a longitudinal pull-force at a 15-degree sideslip causes the pilot to reorient his reactions to stick force cues (HQRS 5).

Dynamic Lateral-Directional Stability

49. Dynamic lateral-directional stability tests were conducted under the conditions and configurations specified in appendix V. Test results are presented in figures 27 through 29, appendix II. Quantitative and qualitative results of the dynamic lateral-directional stability characteristics were obtained by releases from steady heading sideslips, by directional control doublets and in flights during gusty atmospheric conditions. The Dutch roll response was characterized by a "snaking" motion which was moderately damped (table 12). The moderate damping of the Dutch roll oscillation is advantageous and would normally tend to reduce its derogating or nuisance influence. However, moderate rudder or lateral inputs produced large sideslip excursions which are disconcerting to the pilot and result in considerable pilot compensation required during precise and rapid heading corrections (HQRS 5). The test airplane met the lateral-directional damping requirements specified in reference 13, appendix I (fig. C).

Figure C. Lateral-Directional Damping Requirements.

YO-3A, S/N 69-18000

Mission Equipment Installed

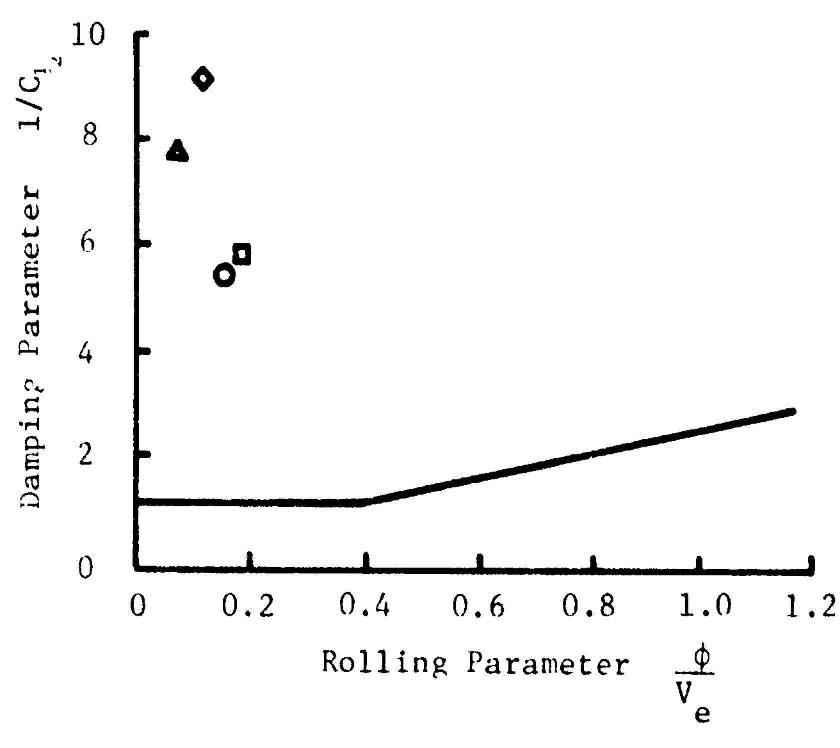


Table 12. Summary of Dutch Roll Characteristics.

Configuration	Density Altitude (ft)	Trim Airspeed (KCAS)	CG Location (FS) (in.)	Gross Weight (lb)	Damping Ratio ζ	Undamped Natural Frequency $\omega_{n}DR$ (rad/sec)	Roll-to-Sideslip Ratio ϕ/β	Cycles-to-½ Amplitude $C_{1/2}$
Cruise	6200	66.6	86.08	3500	0.53	1.60	1:3.5	0.176
Cruise	5000	73.0	85.37	3500	0.55	1.63	1:3.0	0.167
Cruise	8300	73.0	86.08	3500	0.71	1.64	1:3.8	0.109
Power climb	5550	81.5	86.08	3500	0.77	1.75	1:3.0	0.0914
Cruise	7350	[1]104.0	86.08	3500	0.65	2.17	1:5.3	0.128

[1]Maximum airspeed.

50. The low values of the roll-to-sideslip ratio (ϕ/β) are characteristic of the "snaking-type" Dutch roll oscillation and confirm the low effective dihedral characteristic brought out in the static lateral-directional data.

51. Lateral stick pulses or stick raps resulted in the control stick returning to an approximate center position; however, the aileron surface remained displaced and caused the airplane to enter a rapidly descending spiral. This effect was identical for both left and right stick raps. Correction of the lateral control oscillation deficiency is mandatory for the satisfactory accomplishment of the intended mission (HQRS 5).

52. The undesirable characteristics of the Dutch roll oscillation can be minimized by increasing the mission cruise airspeed from 68 to approximately 75 KCAS. The dynamic lateral-directional handling qualities characteristics are acceptable for the intended mission.

Spiral Stability

53. Spiral stability tests were conducted under the conditions and configurations specified in appendix V. Spiral stability tests were conducted in accordance with reference 9, appendix I. From a stabilized bank attitude with the flight controls returned to the trimmed, hands-off condition, a time history of the resulting motion was obtained. Both left and right banked turns were performed. The tests revealed a divergent spiral characteristic with both left and right turns diverging (figs. D and E). At the airspeed for minimum drag (approximately 73 KCAS) with mission equipment installed, the time to double amplitude (14 seconds) does not meet the requirements of reference 13 by 6 seconds (30%). Although the spiral divergence at the airspeed for minimum drag does not meet the referenced requirement, pilot effort to achieve satisfactory performance is not objectionable (HQRS 4). The spiral stability characteristics of the test airplane, at a cruise airspeed of 73 KCAS, are satisfactory for the intended mission.

Figure D. Spiral Stability.
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Mission Equipment Installed

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	6200	3500	66.08	66.6

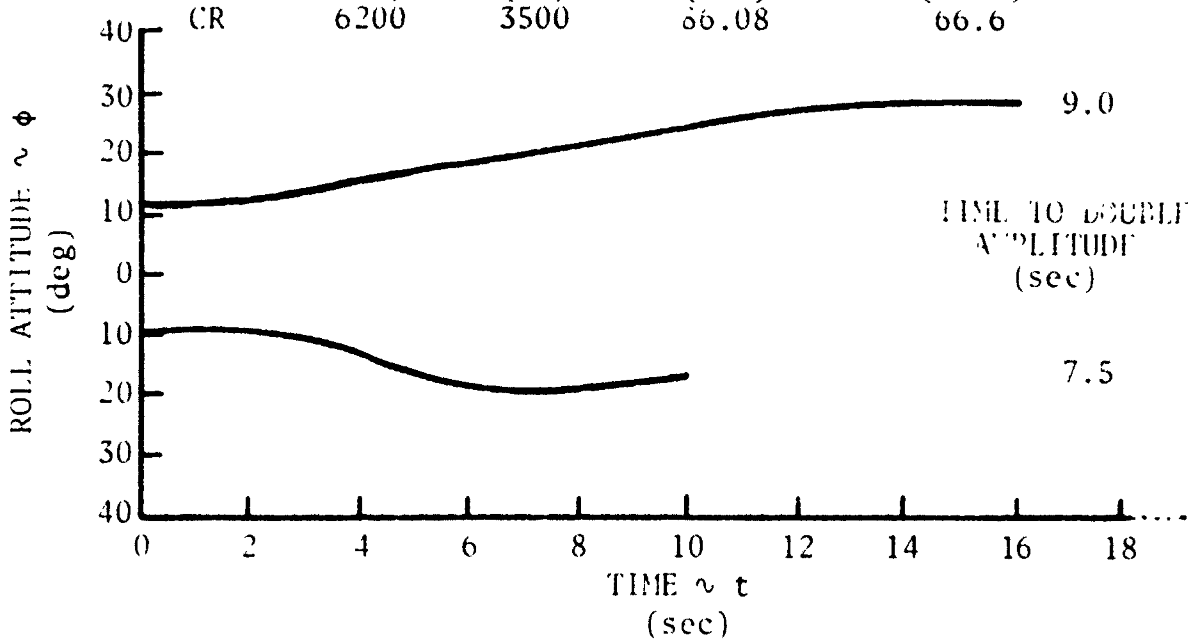
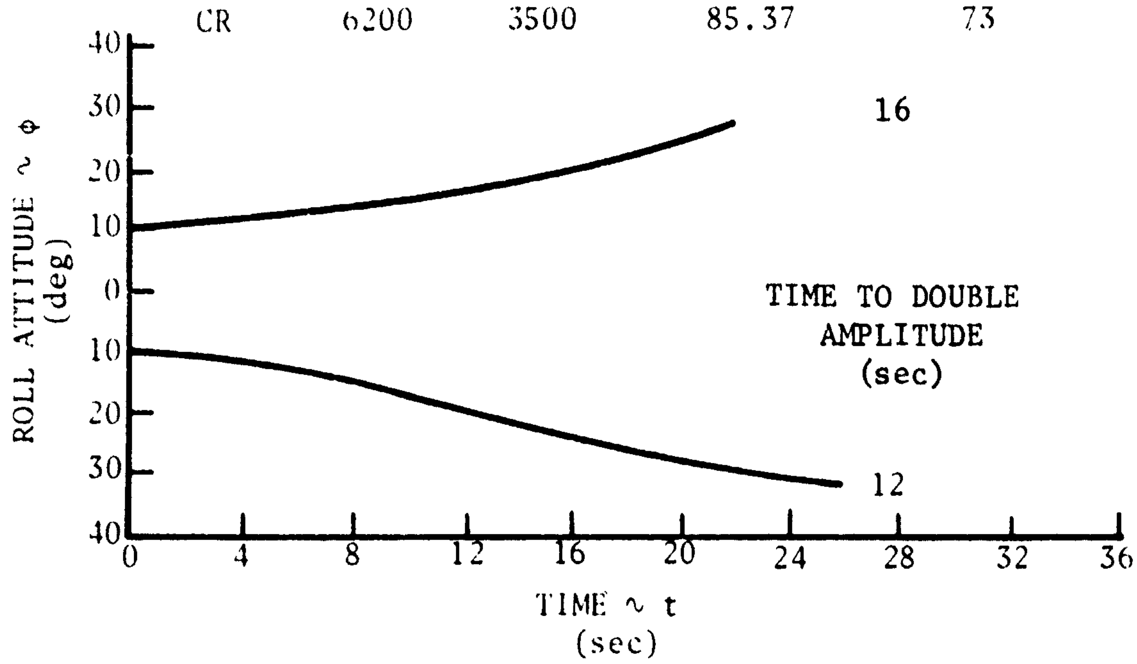


Figure E. Spiral Stability.
YO-3A S/N 69-18000
Mission Equipment Installed

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	6200	3500	85.37	73



Adverse Sideslip

54. Adverse sideslip characteristics were evaluated under the conditions and configurations specified in appendix V and were conducted using the methods described in reference 6, appendix I. Test results are presented in figures 30 through 32, appendix II. In the power approach and cruise configurations, the maximum sideslip change was 17 degrees, which exceeds the maximum specification limit by 2 degrees (7.5 percent) (ref 10, app I). The large sideslip excursion is undesirable and results in a reduction of roll control effectiveness. Correction of this shortcoming is desirable for improved mission accomplishment.

55. Full lateral control stick movement is restricted during rolls to the left or right, because of contact with the pilot's leg (photo 1, app VI). Left lateral control stick movement was more restrictive than right lateral stick movement, and was attributed to the pilot's left leg being forced against the throttle control, protruding from the left bulkhead (photo 2, app VI). The inability to achieve full left lateral control displacement is a deficiency, the correction of which is mandatory.

56. Although the adverse sideslip characteristic is inherent in the basic airframe design, the combination of large rudder pedal forces and restricted lateral control results in a derogation of roll control effectiveness. Additionally, during power approach configured flight and during abrupt turning entries; for example, a rapid runway realignment maneuver on final approach, the adverse sideslip characteristic resulted in poor roll response and extensive pilot compensation was required (HQRS 6).

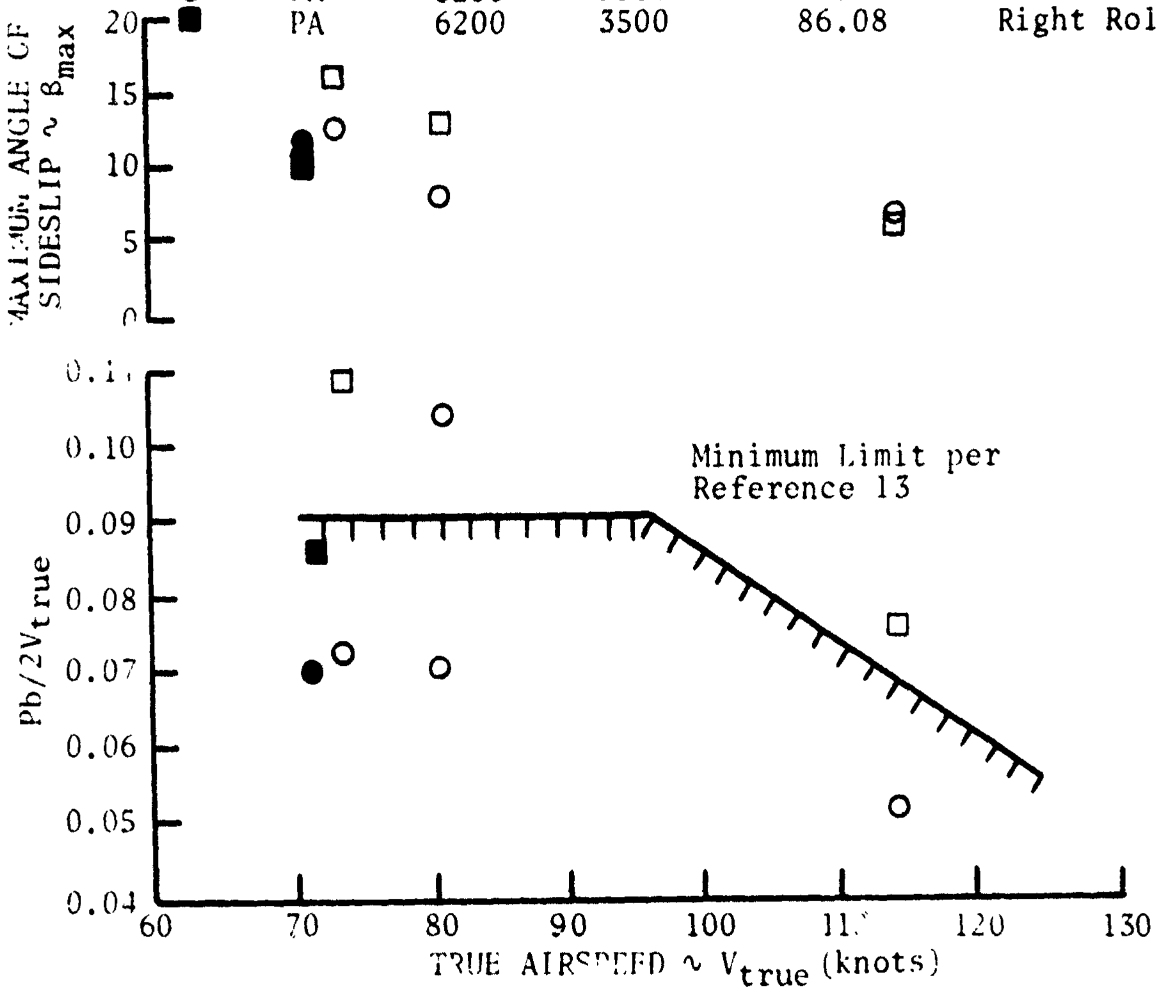
Roll Control Effectiveness

57. Roll control effectiveness tests were conducted under the conditions and configurations specified in appendix V. Test results are presented in figure F and in figure 33, appendix II. All roll control effectiveness data presented herein are derived from full (or maximum available) lateral control deflections unless otherwise specified. In the power approach configuration, the time required to achieve a 30-degree angle-of-bank change (ϕ_t = 30 degrees) was 1.5 second for right rolls and 2.0 seconds for left rolls. The corresponding roll performance requirement, in terms of the wing tip helix angle parameter (pb/2V) was 0.087 radians to the right and 0.07 radians to the left. The large amount of pedal force required to negate the adverse sideslip generated during rolling maneuvers from lateral stick-only inputs resulted in an initial roll

response in the proper direction. As the adverse sideslip began to increase the yaw rate decreased toward zero. Then, as the adverse sideslip effect decreased, the yaw rate began to increase. The combination of low effective dihedral (C_{ℓ_β}) and a large, adverse sideslip characteristic is disconcerting to the pilot. This combination is especially noticeable during night (instrument) operations where the turn entry is characterized by a decrease in yaw rate immediately following stick input. Rolls to the right exhibited better performance than rolls to the left. This characteristic was attributed to the left lateral stick contacting the pilot's leg. The time required to achieve a 60-degree bank angle change (ϕ_t = 60 degrees) during flight at 73 KTAS was 3.0 seconds. This characteristic is indicative of mediocre roll performance and results in considerable pilot compensation to achieve adequate performance (HQRS 5).

Figure F. Roll Control Effectiveness-Pedal Fixed.
YO-3A S/N 69-18000
Mission Equipment Installed

SYM	CONFIG	DENSITY ALT (ft)	GROSS WT (lb)	CG (FS)	FLIGHT COND
○	CR	6200	3500	86.08	Left Roll
□	CR	6200	3500	86.08	Right Roll
●	PA	6200	3500	86.08	Left Roll
■	PA	6200	3500	86.08	Right Roll

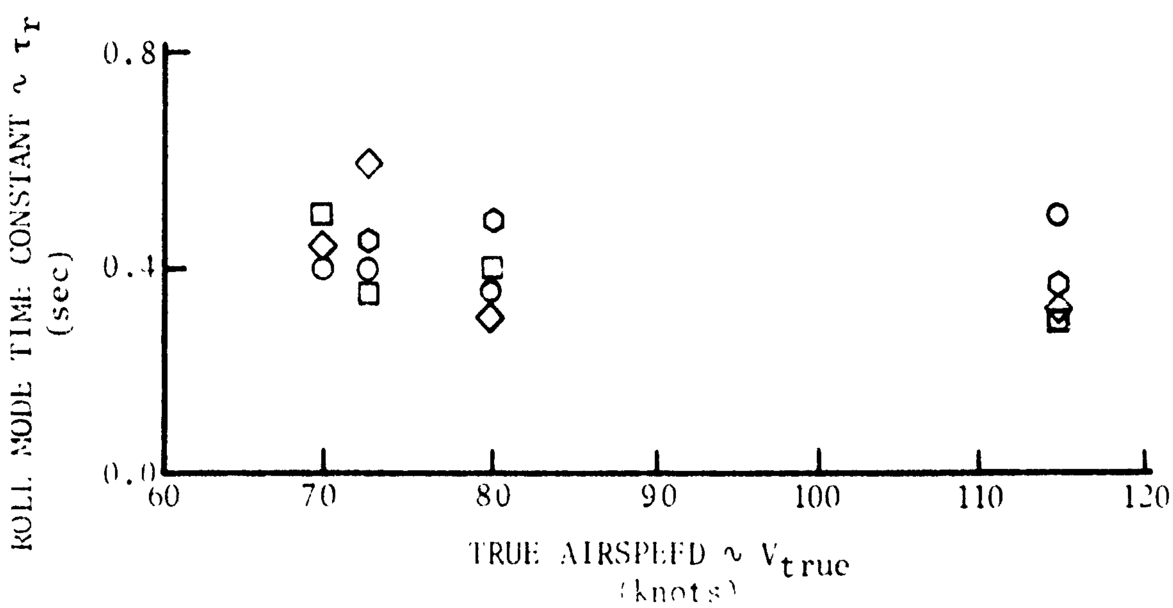


58. Roll control effectiveness with respect to the wing tip helix angle parameter (pb/2V) is presented in figure F. Rolls to the right in the cruise configuration met the requirements of reference 13, appendix I. As stated previously, the restriction of left-lateral stick displacement is the primary cause of the derogation of left rolling performance. The roll control effectiveness characteristics of the test airplane result in considerable pilot compensation to obtain adequate performance for the intended mission (HQRS 5).

59. The roll mode time constant (τ_r) which describes the manner in which the initial roll rate builds and decays after the application of lateral control inputs was also analyzed, and the results are presented in figure G. The average value of τ_r achieved during the roll control effectiveness testing was 0.4 second which is indicative of satisfactory initial roll rate entry characteristics (HQRS 2).

Figure G. Roll Mode Time Constants.
YO-3A S/N 69-18000

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	FLIGHT CONDITION
○	CR	5200	3500	86.08	Left Roll - Rudder Fixed
□	CR	5200	3500	86.08	Right Roll - Rudder Fixed
○	CR	5200	3500	86.08	Left Roll - Rudder Free
◇	CR	5200	3500	86.08	Right Roll - Rudder Free



Maneuvering Stability

60. Maneuvering stability tests were conducted from 1.0 to 2.5 g's normal acceleration under the conditions and configurations specified in appendix V. Test results are presented in figure H. These results were derived from stabilized turns, steady pull-ups and sudden pull-ups, with and without mission equipment installed. The test airplane's stick force per normal acceleration (F_s/g) gradient met the requirements of reference 13, appendix I. Right turns revealed a steeper gradient of F_s/g than did left turns, and steady pull-ups exhibited the smallest gradient (fig H). The gradient for sudden pull-ups was steeper than the gradient for steady pull-ups. Maneuvering stability tests with and without mission equipment were essentially identical. When the operational airplane

Figure H. Maneuvering Stability Summary.

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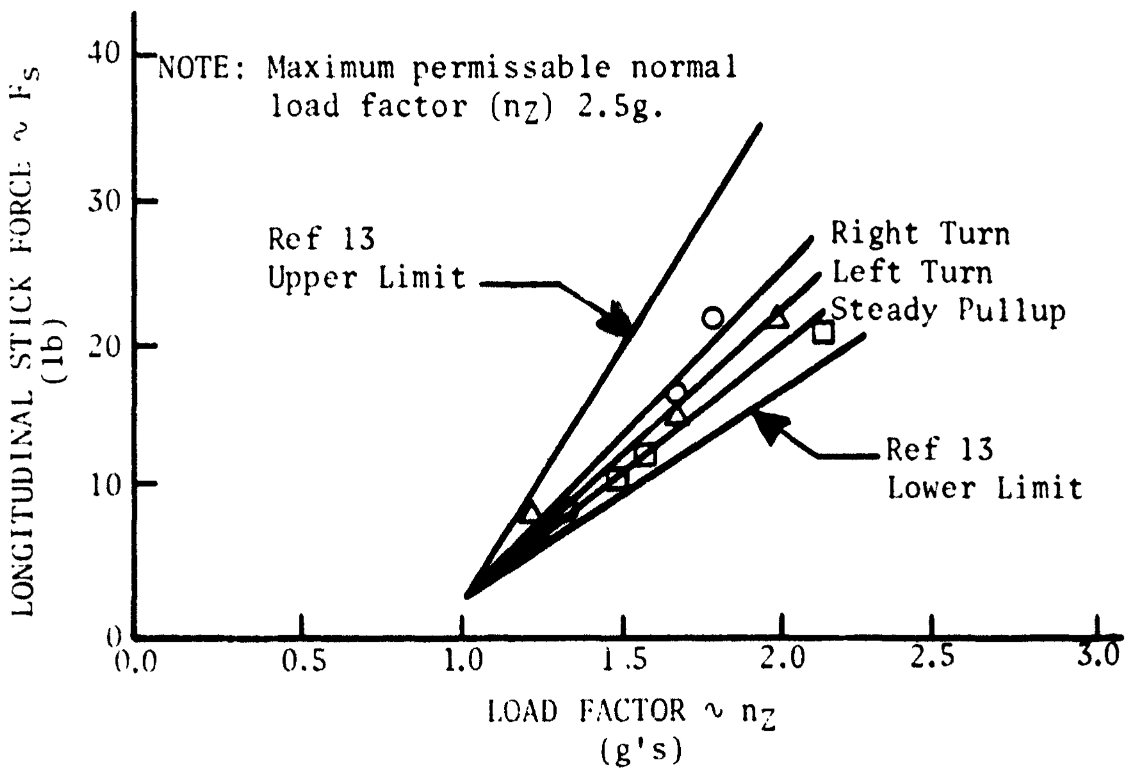
Mission Equipment Installed

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)	FLIGHT CONDITION
○	CR	5000	3500	86.08	75	Right Turn
△	CR	5000	3500	86.08	75	Left Turn
□	CR	5000	3500	86.08	75	Steady Pullup



is cleared to +3.8g normal acceleration, the gradient (F_S/g) would not, qualitatively, based on present test data, meet the requirements of reference 13. Additional maneuvering stability tests should be conducted at higher values of normal acceleration. Within the scope of this test, the gradient of F_S/g is satisfactory for the accomplishment of the intended mission (HQRS 3).

Control Effectiveness During Takeoff

61. Control effectiveness during takeoff testing was conducted throughout the test program. Configurations and conditions applicable to this test are specified in appendix V. The longitudinal stick push-forces (F_S) required to achieve a thrust-line-level attitude are presented in table 13.

Table 13. Control Forces Required to Achieve a Thrust-Line-Level Attitude.[1]

Trim Setting (units)	Center of Gravity Location (FS) (in.)	Airspeed Stabilator Effective (KIAS)	Push-Force[2] (lb)	Specification Noncompliance (%)
2 nose down	85.37	39	29	45
0	86.08	39	32	60
0	84.66	38	33	65

[1]Gross weight, 3500 pounds.
[2]Maximum push-force allowed per reference 13, appendix I, is 20 pounds.

62. The operation of the longitudinal trim wheel permits settings which reduce the push-forces required to achieve a thrust-line-level attitude. With an increase in the airplane nose down trim setting, the reduction in push-force is slight; however, there results a subsequent increase in the pull-force required for rotation. At an aft cg condition, a 3 or 4 unit airplane nose down trim setting should provide acceptable push-forces and pull-forces during tail wheel and main gear lift-off. At a forward cg configuration (the most normal operational condition), push-forces required to achieve a thrust-line-level attitude during the takeoff

roll are excessive at any trim setting. The pilot workload during this phase of the takeoff roll is extensive due to the high control forces involved (HQRS 6). Correction of this deficiency is mandatory for satisfactory mission accomplishment.

63. The longitudinal stick pull-forces required to achieve rotation are presented in table 14. At an aft cg condition, the pull-force required to achieve rotation for takeoff was 2 pounds. The pull-force required in the aft cg condition is satisfactory and results in reduced pilot workload during the takeoff phase. At mid and forward cg conditions, the excessive pull-forces required to effect rotation at takeoff result in extensive pilot compensation in order to achieve satisfactory takeoff performance (HQRS 6). The correction of this deficiency is mandatory.

Table 14. Control Forces Required at Rotation.[1]

Trim Setting (units)	Center of Gravity Location (FS) (in.)	Airspeed at Rotation (KIAS)	Pull-Force[2] (lb)	Specification Noncompliance (%)
2 nose down	85.37	60	28	180
0	86.08	58	2	Satisfactory
0	84.66	60	22	120

[1]Gross weight, 2500 pounds.
[2]Maximum pull-force allowed per reference 13, appendix I, is 10 pounds.

64. The stabilator control does not become effective until 38 KIAS and results in increased ground roll distance prior to achieving a thrust-line-level attitude. Achieving a thrust-line-level attitude as rapidly as possible results in a quicker reduction of drag, thereby providing increased acceleration, which enhances takeoff performance. The inability of the stabilator control to effect a more rapid thrust-line-level attitude is a shortcoming, the correction of which is desirable for improved mission accomplishment.

Control Effectiveness During Landing

65. Control effectiveness during landing testing was conducted under the conditions and configurations specified in appendix V. The test airplane was trimmed at 66 KIAS in the PA configuration

during the downwind leg of the landing approach. A long and relatively flat approach was made at an airspeed of 75 KIAS. Power was reduced to idle (approximately 1600 rpm) when the airplane was directly over the runway. At this time a very noticeable and objectionable engine/ propeller vibration was experienced throughout the airframe. Correction of this engine/propeller vibration during glides at idle power is mandatory for the satisfactory accomplishment of the intended mission. An inherent 5-degree left sideslip condition resulted in the pilot having to maintain an excessive left rudder force in order to achieve proper runway alignment. The stall occurred at 57 KIAS, but was preceded by an obvious airframe buffet at 58 KIAS with a simultaneous activation of the audible stall warning indicator. The maximum longitudinal pull force required to achieve the stall was 18 pounds. The longitudinal control pull force required and the stabilator control effectiveness met the requirements of reference 13, appendix I, and are satisfactory for the intended mission.

Trimmability

66. Longitudinal trimmability tests were conducted under the conditions and configurations specified in appendix V. Qualitative results were also obtained during trimming tasks during other programmed test flights. Longitudinal trimmability can be defined as the facility with which a pilot can effectively reduce the longitudinal control forces to zero at a precise airspeed and have the airplane maintain that trimmed condition without pilot attention. During climbing, cruising and descending flight, the longitudinal trim control exhibited satisfactory rates of operation; however, the sensitivity or ability to achieve a precise airspeed was extremely poor and resulted in extensive pilot compensation to maintain a specified airspeed (HQRS 6). This characteristic of the trim control is attributed to the large friction band and slightly divergent phugoid. Correction of the longitudinal trimmability shortcoming is desirable for improved mission use.

67. In order for the pilot to use the longitudinal trimming device he must remove his right hand from the control stick and adjust the trim wheel located on the right bulkhead (photo 3, app VI). This technique is unsatisfactory in that numerous trim changes are required after takeoffs, during landings and in maneuvering flight. For every power or airspeed change, and spoiler activation, a trim change is required. The pilot workload under the present trim adjustment procedure (trimming with the right hand) is extensive and correction of this deficiency is mandatory for satisfactory mission accomplishment (HQRS 6).

68. The longitudinal trim tab control permitted the pilot to trim into the stall. The combination of a large breakout plus friction force and a neutral to slightly positive stick force gradient at airspeeds below the trimmed mission cruise airspeed requires a restriction on the nose up trim. Correction of this deficiency is mandatory for the satisfactory accomplishment of the intended mission.

69. There was no method of trimming the test airplane in the lateral or directional axes. The lateral control system of the test airplane exhibited a free-play characteristic (slop) of three-eighths of an inch or approximately 4 percent of the available lateral control. The lateral free-play did not meet the requirements of reference 13, appendix I, and resulted in a considerable increase in pilot effort while maintaining a lateral trim (HQRS 6). Correction of the excessive lateral free-play is mandatory for satisfactory mission accomplishment. The amount of rudder force required to return the airplane to a wings level, zero sideslip condition, within a range of sideslip angles that will be encountered operationally, was excessive and required maximum tolerable pilot compensation (HQRS 7). These operations may include but are not limited to: cruise at high airspeeds, coordinated turns, tracking ground targets, crosswind takeoffs and landings, wingovers or other evasive tactical maneuvers and descents at idle power. See figure J for a time history of the rudder forces developed during a typical takeoff. The lack of an inflight, cockpit-adjustable, rudder trim tab requires that the pilot provide maximum tolerable compensation and yet not achieve satisfactory handling qualities at airspeeds other than the contractor's recommended mission cruise speed (HQRS 7). The lack of an inflight, cockpit-adjustable, rudder trim tab is a deficiency, the correction of which is mandatory for satisfactory mission accomplishment.

Longitudinal Pitch Trim Changes

70. Longitudinal pitch trim change tests were conducted under the conditions and configurations specified in table 15. Table 15 also summarizes the test results. Longitudinal pitch trim changes, as a result of gear activation to the up position while in the takeoff configuration, were practically nonexistant. This highly desirable characteristic results in low pilot effort required for this configuration change (HQRS 1). Although the specifications (refs 6 and 13, app I) state that the control forces shall not exceed ±10 pounds following a longitudinal trim change, the location of the pitch trim wheel justifies a more stringent requirement. Longitudinal pitch trim changes resulting in control forces in excess of 5 pounds push or pull (conditions 2, 4, and 5, table 15) are

excessive and result in extensive pilot compensation for adequate performance (HQRS 6). The extensive pilot compensation required is due to the location of the longitudinal pitch trim wheel on the right bulkhead. Relocating the longitudinal pitch trim wheel, which will permit the pilot to trim the airplane without removing his right hand from the control stick, is mandatory for the satisfactory accomplishment of the intended mission.

Figure J. Rudder Forces During Takeoff.
YO-3A S/N 69-18000

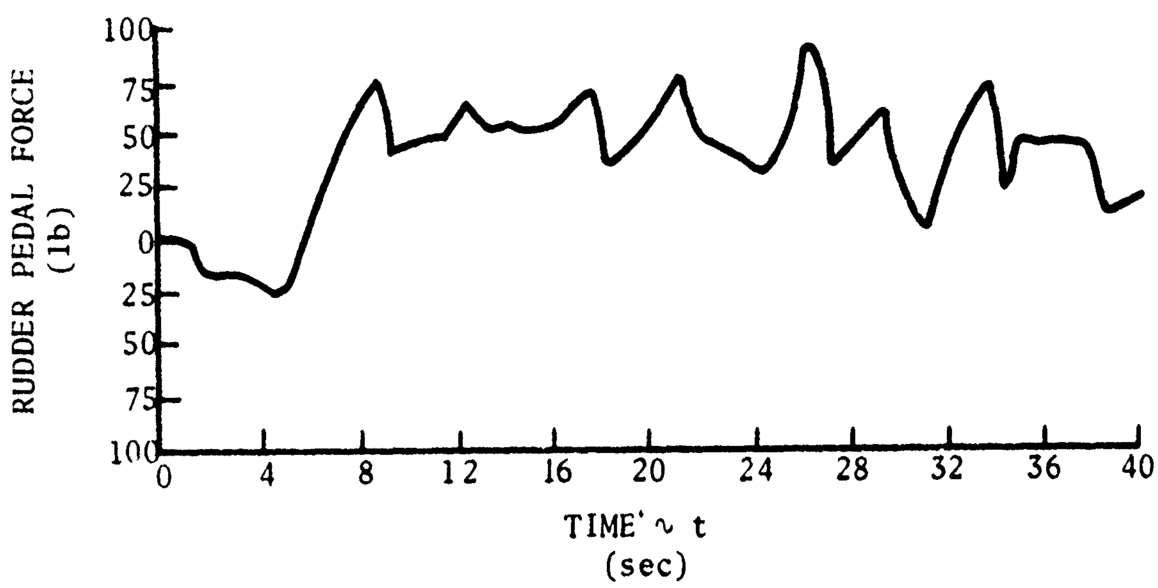


Table 15. YO-3A, S/N 69-18000, Longitudinal Trim Changes.

Condition	Flight Phase	Initial Trim Condition[1]				Configuration Change	Parameter Held Constant	Maximum Longitudinal Force (lb)
		Airspeed KCAS	Landing Gear	Spoilers	Power			
1	Takeoff	74.8	Down	Closed	MRP	Gear up	Rate of climb	3 Push
2	Cruise	112.0	Up	Closed	NRP[2]	Idle power	Altitude	10 Pull
3	Cruise	77.0	Up	Closed	PFLF[3]	Gear down	Altitude	3 Pull
4	Cruise	68.5	Down	Closed	PFLF	Power to MRP	Altitude	20 Push
5	Waveoff	69.0	Down	Closed	500-fpm R/D	Gear up, power to MRP	Airspeed	8 Push
6	Cruise	68.5	Up	Closed	PFLF	Open spoilers	Airspeed	5 Push

[1]Center of gravity location is 85.37 in. (FS).
Gross weight is 3500 lb.
Density altitude is approximately 3500 ft.
[2]Normal rated power.
[3]Power for level flight.

Stall Characteristics

71. Stall characteristics were evaluated under the conditions and configurations specified in appendix V. The flight testing sequence involved an investigation of: the approach to the stall, the fully developed stall, and the stall recovery during normal 1 g stalls and accelerated stalls. A summary of the stall characteristics is presented as table A, appendix II. For purposes of this report the stall speed is defined as the minimum attainable steady airspeed at 1 g, normal to the flight path, wherein abrupt or significant changes occur with pitching or rolling moments which result in temporary or partial loss of control.

Approach to the Normal Stall:

72. The approach to a normal stall in the cruise configuration, when trimmed for level flight, was evaluated at the airspeed for 0.99 maximum range and at the contractor's recommended mission cruise airspeed. The initial stall warning occurred at 56 KIAS and was characterized by a slight airframe buffet and a simultaneous activation of the aural stall warning device. As airspeed was decreased below 60 KIAS, a slight but not objectionable, airplane nose-up pitch rate was experienced. A neutral to slightly negative stick force versus velocity gradient was noticeable between 60 and 57 KIAS. Adequate control about all axes was available during the approach to the stall. In all configurations in the approaches to the stall, the lateral axis consistently exhibited the initial derogation from control effectiveness. In the glide configuration, the approach-to-stall characteristics were essentially identical to the cruise configuration characteristics except for the lack of the slight nose-up pitching tendency. The airframe buffet was also obvious, as in the cruise configuration, and provided the pilot with earlier stall warning cues than did the aural stall warning device which activated simultaneously with the stall. The airframe buffet stall warning, which preceded the stall, was undistinguishable from the engine/propeller vibrations which was present at idle power in the glide configuration. Correction of this deficiency is mandatory for satisfactory operational use. As in the cruise configuration, the effectiveness of the flight controls during glide configuration stalls was adequate about all axes. In the dive configuration, the approach to the stall was characterized by a definite airframe buffeting which occurred 5 knots prior to and continued into the stall. The aural stall warning indicator activated 1 to 2 knots prior to the stall. Normal stalls in the power approach and landing configurations exhibited characteristics during the approach to the stall which were essentially identical to those described above. In all of the configurations described above,

the aural stall warning device did not provide sufficient warning in advance of the stall to allow pilot prevention of the stall by normal control manipulation or within normal pilot reaction time (HQRS 6). Under the present stall warning arrangement, once the aural stall warning device had activated, the pilot had no other choice but to proceed through the stall and execute stall recovery. Application of full power and reduction in angle of attack was insufficient to negate the onset of 1 g stalls once the aural stall warning device had activated, since engine acceleration was inordinately slow in the region of stall airspeed. Ideally, the artificial stall warning indicator should be activated by an angle of attack sensor instead of an airspeed sensor. The aural stall warning indicator should be armed to activate at least 5 knots prior to the stall. The inability of the aural stall warning indicator to provide sufficient warning of an impending stall is a deficiency and correction is mandatory prior to the airplane's acceptance for service use.

Fully Developed Normal Stalls:

73. Fully developed normal stalls in the cruise configuration were characterized by an initial loss of lift with a subsequent nose-down pitching moment with a slight lateral divergence, usually to the right. The directional control was effective in maintaining a wings-level attitude into and throughout the stall. The directional control was more effective in maintaining a wings-level attitude than was the lateral control. The lateral control was only slightly effective in inducing a roll to the right or left during the stall. In the glide and dive configurations, the fully developed stalls exhibited characteristics similar to the cruise configuration stall. When trying to maintain a wings-level attitude in the stall, excessive rudder application should be discouraged since an inadvertent spin entry could easily be achieved. The stall in the takeoff configuration exhibited a mild left wing drop-off (approximately 15 degrees) with a slight nose-down pitching moment.

Normal Stall Recovery:

74. Stall recovery in all configurations was rapid following angle-of-attack reduction and simultaneous application of full throttle. Adequate control about all axes was exhibited and control forces were not excessive. Altitude loss to regain level flight from a cruise configured stall averaged 150 feet. In the glide and power approach configurations, altitude loss averaged 300 feet. The optimum stall recovery technique consisted of the immediate reduction in angle of attack, leveling of the wings and simultaneous application of full power. As airspeed increased to 65 KIAS

a gradual increase in angle of attack was accomplished until the level flight was achieved. The unusually slow engine acceleration characteristic made it incumbent upon the pilot to immediately reduce angle of attack and simultaneously apply full throttle at the first indication of impending stall in order to preclude excessive altitude loss. This is particularly noteworthy during low-level, night operations.

75. With the exception of the late activation of the aural stall warning indicator the normal stall characteristics of the test airplane are satisfactory. Minimal pilot compensation is required for desired performance (HQRS 3).

Accelerated Stalls

Approach to Accelerated Stalls:

76. In the cruise configuration, while trimmed in level flight at the contractor's recommended mission cruise speed, accelerated stalls were evaluated from level turns at bank angles up to 45 degrees. Airframe buffet and simultaneous activation of the aural stall warning device occurred at 1 knot prior to the stall. In the dive configuration, the approach to the stall was characterized by an obvious airframe buffet just prior to the stall. Activation of the aural stall warning device occurred at stall. During this test sequence, the reduction of throttle to idle resulted in activating the landing gear aural warning horn. The landing gear aural warning horn was identical in tone and frequency to the aural stall warning device. Identical artificial warning devices which alert the pilot to two dissimilar occurrences are unsatisfactory and correction of this deficiency is mandatory. The approach to a power approach configured accelerated stall, in a wings-level attitude, was characterized by immediate airframe buffet and activation of the aural stall warning device with aft stick input. The stick force versus airspeed gradient exhibited a smooth, slight reversal with a subsequent nose-up pitch rate just prior to the stall. The stick force reversal and pitch up were not objectionable to the pilot. Accelerated stalls were also evaluated in the power approach configuration during descending left and right turns with similar approach characteristics evident prior to stall.

Fully Developed Accelerated Stall:

77. In the cruise configuration, during a coordinated left turn, the stalled condition was characterized by a significant rate of descent, approximately 600 fpm. When excessive left rudder was used in the left turn (ball to the topside) the left wing dropped slightly in the stall. With excessive right rudder during the left

turn entry to a stall, the bank angle began to decrease as the stall continues and, eventually, the bank angle decreased toward zero and the airplane flew out of the stall. In the cruise configuration, accelerated stalls during right turns exhibited similar characteristics as described for left turns but in the opposite direction, with the exception of a slight nose-down (10 degrees) pitching moment at the stall. In the dive configuration, the stall was characterized by a rapid 45-degree, nose-down pitching moment. The pitch attitude change occurred rapidly, but could not be classified as a "snap" type maneuver. The accelerated stall in the power approach configuration from a wings-level pullup was characterized by a mild 30-degree, nose-down pitch change. During descending left turns in the power approach configuration, the accelerated stall exhibited a left wing dropoff and a nose-down pitch change of 15 degrees. During descending right turns, the stall was characterized by a large increase in rate of descent (from 500 fpm prior to the stall to 1500 fpm in the stall). In this case, the large increase in rate of descent defined the stall condition.

Accelerated Stall Recovery:

78. In all accelerated stall tests the stall condition could be terminated by the immediate reduction in angle of attack and normal acceleration (n_z). Control effectiveness about all axes was adequate with no tendency toward secondary, or post stalls being exhibited. As altitude was lost and airspeed increased during the stall recovery, the pilot should be cognizant of a propeller overspeed possibility of airspeed increases beyond 90 KIAS. The accelerated stall characteristics of the test airplane exhibited satisfactory characteristics for the intended mission with the exception of the aural stall warning device (HQRS 3).

Flight Control System Characteristics

79. A detailed description of the YO-3A test airplane's flight control system is presented in appendix IV. The characteristics of the flight control system, discussed below, were derived from quantitative and qualitative analysis of engineering data accumulated during actual flight tests. A plot of the available control stick motion of the test airplane is presented as figures a and d, appendix IV. From the forward lateral limit, and proceeding aft, the control stick made continuous contact with the pilot's leg thereby preventing maximum stick travel. At the full aft position, the control stick also contacted the pilot's parachute seat pack. The restriction of full control stick displacement is a deficiency, correction of which is mandatory.

80. The test airplane's control system exhibited three-quarters of a pound breakout force with a total breakout plus friction force of 2½ pounds (fig 34, app II). Although the breakout plus friction force characteristic is within the limits of reference 13, appendix I, it is, nevertheless, excessive for the satisfactory accomplishment of the intended mission. The large breakout plus friction band complicates and masks the stick force versus airspeed gradient and results in minimal pilot cues. The combined effect of a large breakout plus friction band and divergent phugoid results in an extensive pilot workload in order to achieve adequate performance (HQRS 6). The longitudinal breakout plus friction force is a deficiency and correction is mandatory for the satisfactory accomplishment of the intended mission.

81. Control stick centering from a trimmed condition, after control stick displacement and release, exhibited positive, although not absolute, centering. The control stick centering characteristic is probably complicated by the large friction band.

82. Lateral breakout force was measured at three-quarters of a pound, and the rudder pedal breakout force approximately 5 pounds. The lateral and directional control system breakout force characteristics are satisfactory for the accomplishment of the intended mission.

83. The control harmony characteristics of the test airplane are unsatisfactory because of the excessive rudder pedal forces and the large control system breakout plus friction force. Rapid movement of the lateral and directional controls during maneuvering flight requires extensive pilot compensation to achieve adequate performance (HQRS 6). Correction of the unsatisfactory control harmony characteristics is mandatory for the satisfactory accomplishment of the intended mission.

MISCELLANEOUS TESTS

Cockpit Evaluation

84. A day and night cockpit evaluation of the production airplane, S/N 69-18002, and test airplane was performed during the APE with the following safety-of-flight deficiencies noted:

a. Throughout the entire APE the contractor's recommended maximum power static rpm check (2000 rpm at 9.0 psi metered fuel pressure at sea level) was rarely achieved. The failure of the propeller/engine combination to consistently produce this recommended static power rpm limit is unsatisfactory. It is imperative that a list of "go" and "no-go" static power rpm limits as a function of density altitude be developed by the contractor and placed within the cockpit for the operational pilot's use.

b. On several test flights during climbing flight at 80 to 85 KIAS, the odor of raw gasoline fumes permeated the cockpit area. The climbing flight was performed with wings level and no unusual attitudes were involved. The occurrence of raw gasoline fumes during night missions with mission equipment in the operating mode is particularly hazardous.

c. Emergency control devices, e.g. emergency canopy release handle, emergency gear extension handle, emergency spoiler control handle, were not identified.

d. The cockpit floor partition in the pilot's compartment was inadequate in that spaces permitted objects to fall into the fuselage among the control cables and control rods.

e. Only one emergency gear extension handle was installed for use by the pilot and the observer.

f. The fuel tank selector rod failed during contractor testing and the replacement was inadequate.

85. The following cockpit deficiencies were noted:

a. Unintentional braking occurred during the takeoff groundroll.

b. Detent positioning of the fuel tank selector handle was unsatisfactory.

c. There was no spoiler position indicator for use during night and ground control approach operations.

d. The cockpit canopy could not consistently be closed and locked without outside ground crew assistance.

e. The compass heading card on the ID 1351/A did not have sufficient lighting for night operations.

f. An emergency canopy shattering tool was not supplied for emergency ground egress.

g. The observer's mission equipment had no integral night lighting.

h. On several test flights inconsistent fuel quantity readings were noted.

86. The following cockpit shortcomings were noted:

a. The emergency canopy-release air bottle could not be serviced or preflighted without removing the observer's parachute.

b. Excessive cockpit temperatures were experienced during moderate ambient air temperature conditions.

c. The fuel quantity indicators were difficult to read due to poor location.

d. The pilot's shoulder harness lock mechanism was difficult to quickly locate due to cables, rods and wires in the immediate vicinity.

e. The engine rpm was difficult to read precisely due to the small size and coarse scale of the indicator.

f. There was no integral lighting for the circuit breaker panel.

g. The lighting for the fuel tank selector, auxiliary fuel pump switch, altenator switch, bus trim switch and master switch was inadequate.

h. The instrument/map light was ineffective due to poor location (on the lower left bulkhead where the pilot's leg interfered) and insufficient cord length.

i. The pilot's and observer's seats and rudder pedals were not adjustable when the seats were occupied.

j. The landing gear toggle switch was directly adjacent the cowl flap toggle switch.

k. The cowl flap indicator flashed intermittently during high-speed flight with the cowl flap closed.

87. The canopy defogging system was not installed in the test airplane. An evaluation of this system should be accomplished on a production airplane prior to operational deployment.

Airplane Evaluation

88. During the APE there were numerous occasions, while taxiing out for takeoff, when the engine cylinder head temperature gage and engine oil temperature gage were indicating over red line. The outside air temperature varied from +12 to +20 degrees centigrade on these occasions. The total ground run time, from engine start to takeoff, ranged from 5 to 10 minutes. The cylinder head overtemperatures varied from the red-line limit to approximately 20 to 30 degrees above this limit. This condition constitutes a safety-of-flight hazard.

89. While flying at maximum level-flight airspeed, 1¼ inches of aft throttle movement from the maximum power setting was required prior to any indication of power reduction. This characteristic is disconcerting to the pilot, and correction of this shortcoming is desirable for improved mission use.

90. Due to a continually popping circuit breaker the rotating beacon (Grimes Light) was not available for use. This condition should be corrected.

Ground Handling Characteristics

91. The APE test airplane exhibited satisfactory ground handling characteristics. RPM was set at 1300 and a fast-walk taxi speed was achieved. The tail wheel configuration results in a nose-high airplane attitude during taxiing operations and "S" turns are consequently required for straight ahead field of view. There was no unintentional braking due to rudder-only application in the instrumented test airplane. This was attributable to the temporary installation of wooden toe blocks on the rudder pedals for use of the contractor's pilot and left there for the Army pilots. During crosswind taxi operations, the lack of sufficient lateral control was compensated for by the spoiler control.

92. The taxi light was satisfactory for night taxi operations; however, it was unsatisfactory for use as a landing light.

Maintenance Characteristics

93. The APE test airplane and production airplane evaluated during the test program exhibited certain maintainability characteristics as follows:

a. There was no provision for towing the test airplane with a towbar. Correction of this deficiency is mandatory for the satisfactory accomplishment of the towing requirement.

b. The engine cowlings had no "hold-open" support device for use while performing engine maintenance. The addition of engine cowling support devices is desirable to reduce the airplane mechanic's workload.

c. An unguarded, stainless steel exhaust tube along the lower right exterior cockpit bulkhead was capable of causing burns if contacted shortly after engine shutdown. A caution or warning label should be placed in an obvious location to preclude unnecessary burns. Correction of this shortcoming is desirable.

d. Reed and Prince, or Phillips head machine screws were used to secure the engine cowling to the airframe. Many of these screws exhibited slots that were becoming rounded from excessive use. During a preflight inspection, it was observed that several (approximately six) machine screws in the engine cowling could not be secured. Common hardware screws were used for attaching the false leading edge and upper cap to the vertical fin of the production airplane, while the test airplane used Phillips head machine screws. This is indicative of poor quality control during the production of the airplane. More rigid quality control procedures must be utilized and Dzus fastners, or another type quick disconnect fastner, be used in place of machine screws in the engine cowling.

e. Access to the tire air valve was blocked by the landing gear fairing. Correction of this shortcoming is desirable for improved service use.

f. The fuselage and control surfaces did not have drain holes. Correction of this deficiency is mandatory for satisfactory mission accomplishment.

g. The non-skid walkway on the test airplane had lost its effectiveness through wear, and the large wing root camber made cockpit ingress and egress rather precarious. Correction of this shortcoming is desirable.

h. The clearance existing between the landing gear fairing door and the ground surface was approximately 1½ inches. This characteristic is unsatisfactory and correction of this deficiency is mandatory for satisfactory mission utilization.

CONCLUSIONS

GENERAL

94. The following conclusions were reached upon completion of the YO-3A Army Preliminary Evaluation:

a. The YO-3A airplane concept exhibits excellent potential for the satisfactory accomplishment of the intended mission. However, the test airplane as evaluated was not sufficiently developed for operational deployment.

b. The YO-3A airplane did not meet any of the contractor guarantees as specified in the detail specification with the exception of the maximum level-flight airspeed and stall speed guarantees.

c. The performance and handling qualities characteristics of the YO-3A airplane were marginal for the accomplishment of the intended mission. However, the satisfactory performance of the intended mission would result with the correction of the deficiencies listed below.

SAFETY OF FLIGHT DEFICIENCIES

95. Correction of the following safety-of-flight deficiencies is mandatory for acceptance of the YO-3A airplane:

a. Excessive engine cylinder head temperatures during normal taxi, run-up and takeoff operations (para 88).

b. Excessive engine oil temperatures during normal taxi, run-up and takeoff operations (para 88).

c. Raw gasoline fumes in the cockpit during climbing flight at 80 KIAS (para 84).

d. Absence of placarded limits for the maximum static power rpm check (para 84).

e. Structural failure of the fuel tank selector rod (para 84).

f. Lack of emergency control device identification (para 84).

g. Only one emergency gear handle for the pilot's and observer's positions (para 84).

h. An imcomplete cockpit floor partition in the pilot's compartment (para 84).

DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSICN ACCOMPLISHMENT

96. Correction of the following deficiencies is mandatory for the satisfactory accomplishment of the intended mission:

a. Failure of the engine and propeller combination to consistently produce the contractor's recommended static power rpm at full throttle (para 84).

b. Excessive directional control forces (para 69).

c. Lack of an accurate and simplified fuel/cruise guide indicator (para 30).

d. An excessive longitudinal control system breakout plus friction band (para 80).

e. Pilot required to remove right hand from stick control to trim out longitudinal control forces (paras 67 and 70).

f. Excessive longitudinal stick push-force required to achieve a thrust-line-level attitude during the takeoff roll (para 62).

g. Excessive longitudinal stick pull-force required to rotate the airplane at takeoff (para 63).

h. Lack of a spoiler position indicator (para 85).

i. Poor rudder pedal design, in that, rudder application results in unintentional braking (para 85).

j. Outside ground crew assistance required to close and lock the canopy (para 85).

k. The lateral control system exhibits excessive free play (three-eighths of an inch or 4 percent) (para 69).

l. Unsatisfactory simulated lateral gust response characteristics (para 51).

m. Poor crosswind landing and takeoff characteristics (para 23).

n. Unsatisfactory control harmony characteristics (para 83).

o. Excessive engine vibration during cruising flight at 70 to 80 KIAS and during glides at idle power (paras 25 and 65).

p. Insufficient airspeed margin (2 knots) between aural stall warning actuation and stall (para 72).

q. The aural stall warning and aural landing gear warning horns are identical in tone and frequency (para 76).

r. Longitudinal trim tab control permitted trimming into the stall (para 68).

s. Full left-lateral control displacement restricted due to the control stick contacting the pilot's leg (para 79).

t. Insufficient clearance (approximately 1½ inches) between landing gear fairings and ground surface (para 93).

u. Unsatisfactory detent positioning of the fuel tank selector (para 85).

v. Full aft unrestricted motion of the control stick prevented by the parachute seat pack (para 79).

w. Lack of an emergency canopy shattering tool for emergency ground egress (para 85).

x. Insufficient night lighting for the compass heading card on the ID 1351/A (para 85).

y. Lack of integral night lighting capability on the observer's mission equipment panel (para 85).

z. Lack of a provision for towing the airplane with a towbar (para 93).

aa. Inconsistent readings of fuel quantity indicators (para 85).

ab. Lack of drain holes for fuselage and control surfaces (para 93).

97. Correction of the following shortcomings is desirable for improved operation and mission capabilities:

a. Inadequate cockpit cooling system (para 86).

b. Neutral to slightly positive stick force versus airspeed gradient at airspeeds below trim during flight in the region from stall to the minimum power required (para 40).

c. Poor longitudinal trimmability characteristics (para 66).

d. The slightly divergent long-period oscillation (para 44).

e. Stabilator will not effect a rapid thrust-line-level attitude during takeoff (para 64).

f. Engine rpm indicator difficult to read precisely (para 86).

g. Poorly located fuel quantity indicators (para 86).

h. Continuous popping of the rotating beacon (Grimes Light) circuit breaker (para 90).

i. Power lever required 1¼ inches of aft movement from the full-throttle position prior to any indication of power reduction (para 89).

j. Unguarded exhaust tube along right bulkhead (para 93).

k. Adjacent landing gear toggle switch and cowl flap toggle switch (para 86).

l. Pilot's shoulder-harness lock difficult to locate due to cables and rods in the immediate area (para 86).

m. Slippery nonskid-walkway on the inboard wing section (para 93).

n. Unadjustable pilot's and observer's seats and rudder pedals when occupied (para 86).

o. Inadequate instrument lighting for the fuel selector, auxiliary fuel pump switch, alternator switch, bus trim switch and master switch (para 86).

p. Lack of an integral circuit-breaker-panel lighting capability (para 86).

q. Poorly-located instrument map light (para 86).

r. Insufficient length of instrument map light cord (para 86).

s. Access to tire air valve blocked by landing gear fairing (para 93).

t. Cowl flap indicator-light operates intermittently (para 86).

u. Large variation of sideslip (15 degrees) caused by the adverse sideslip characteristic (para 54).

v. Removal of the observer's parachute required for servicing or preflighting the emergency canopy release bottle (para 86).

w. Lack of a "hold-open" support device on the engine cowling for engine maintenance (para 93).

RECOMMENDATIONS

98. The safety-of-flight deficiencies, for which correction is mandatory for a satisfactory degree of flight safety, should be corrected prior to acceptance of the YO-3A airplane.

99. The deficiencies, for which correction is mandatory, should be corrected prior to the Army pilot's training program.

100. The shortcomings, for which correction is desirable, should be corrected prior to operational deployment.

101. The US Army should evaluate the feasibility of installing a constant-speed, or two-speed, propeller on the YO-3A airplane (para 24).

102. Further takeoff performance testing should be conducted at density altitudes up to 5000 feet prior to operational deployment (para 19).

103. The takeoff and landing limitation for operational use should be a 90-degree crosswind component at 10 knots (para 23).

104. The canopy defogging system should be evaluated prior to airplane deployment (para 87).

105. More rigid quality control procedures should be utilized during the production line inspection sequence (para 93).

APPENDIX I. REFERENCES

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11. Specification Report, Lockheed Missiles and Space Company, LMSC No. 5580127, *Detail Specification, YO-3A Aircraft*, 15 November 1968, (CONFIDENTIAL).

12. Operator's Handbook, Lockheed Missile and Space Company, LMSC No. D148159, *Preliminary Operator's Handbook, YO-3A Aircraft*, 1 January 1969.

13. Design Report, Lockheed Missiles and Space Company, LMSC No. 681689, *Design Requirements for YO-3A Observation Aircraft*, with revision, 9 September 1968.

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APPENDIX II. TEST DATA

FIGURE 1

PITOT STATIC SYSTEM CALIBRATION
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED
SHIP STANDARD SYSTEM

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	FLIGHT CONDITION
△	CR	1000	3500	85.37	Level Flight
◇	PA	1000	3500	85.37	Level Flight

Note: Calibration: Pace aircraft with trailing bomb.

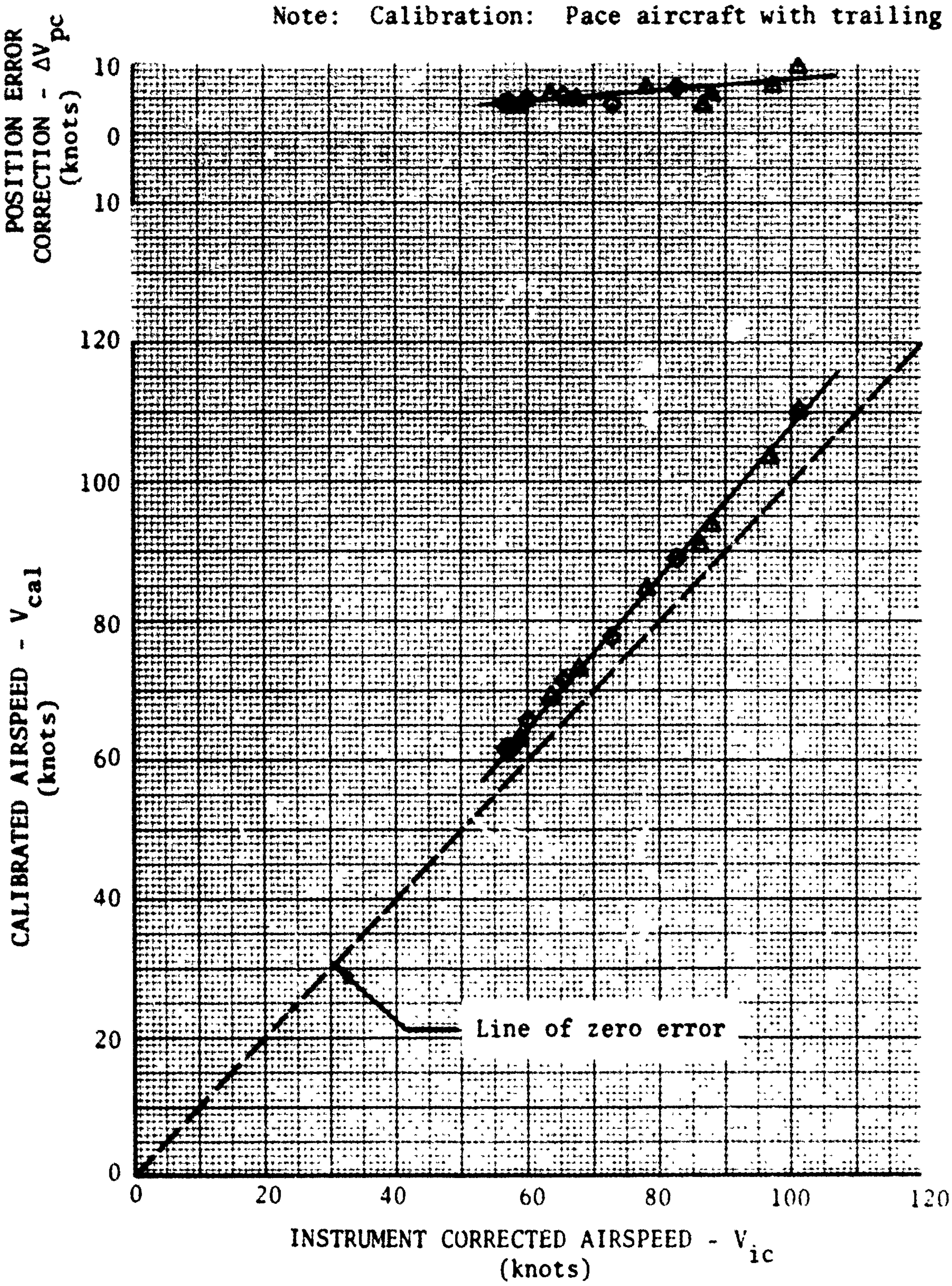


FIGURE 2

PITOT STATIC SYSTEM CALIBRATION
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED
BOOM SYSTEM

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	FLIGHT CONDITION
△	CR	1000	3500	85.37	Level Flight
◇	PA	1000	3500	85.37	Level Flight
○	CR	6500	3500	85.37	Level Flight
□	PA	6500	3500	85.37	Level Flight

Note: Calibration: Pace aircraft with trailing bomb.

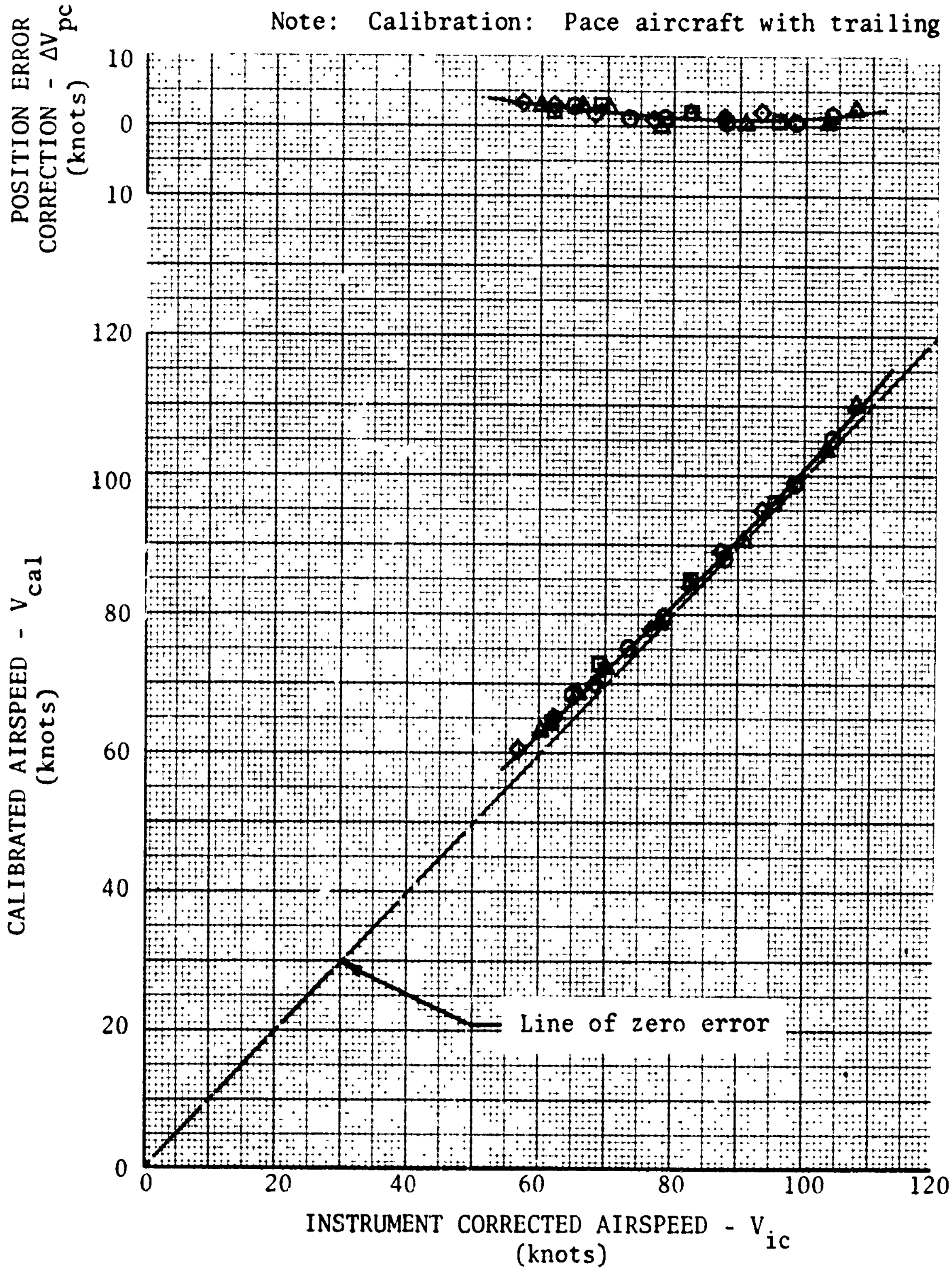
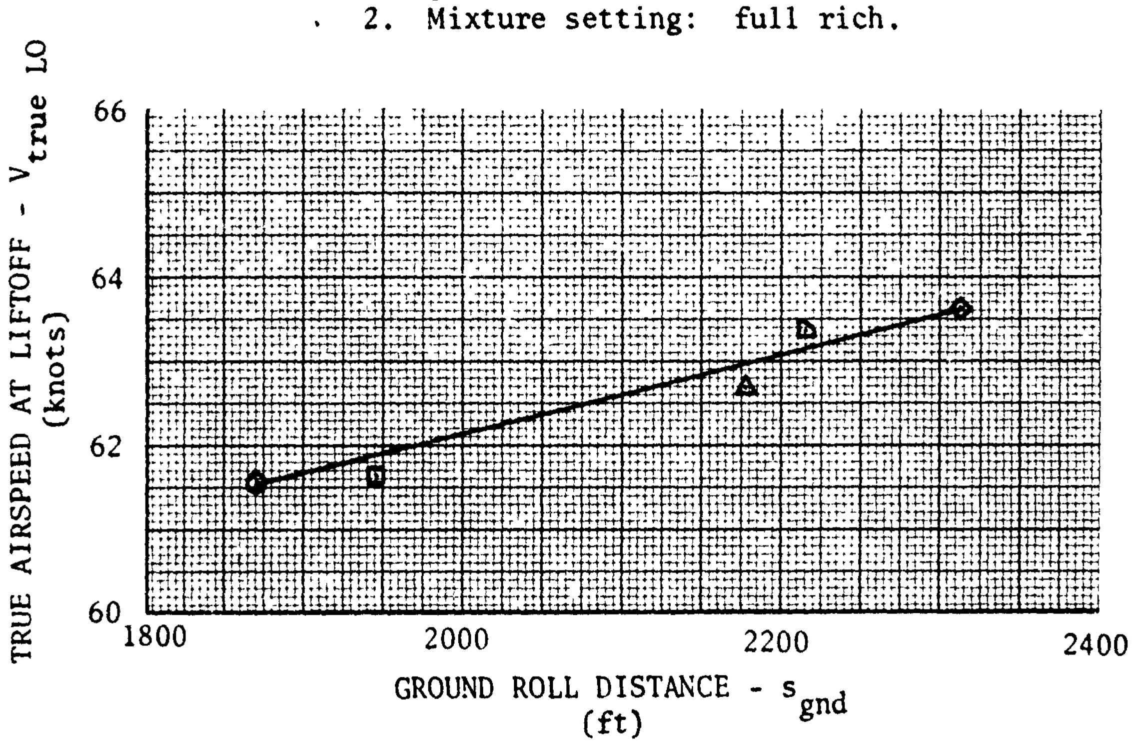


FIGURE 3

TAKEOFF PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
TO	150	3500	85.35

Notes: 1. Results corrected to sea level standard day, 3500 pounds grwt, and level runway. Zero wind existed.
2. Mixture setting: full rich.



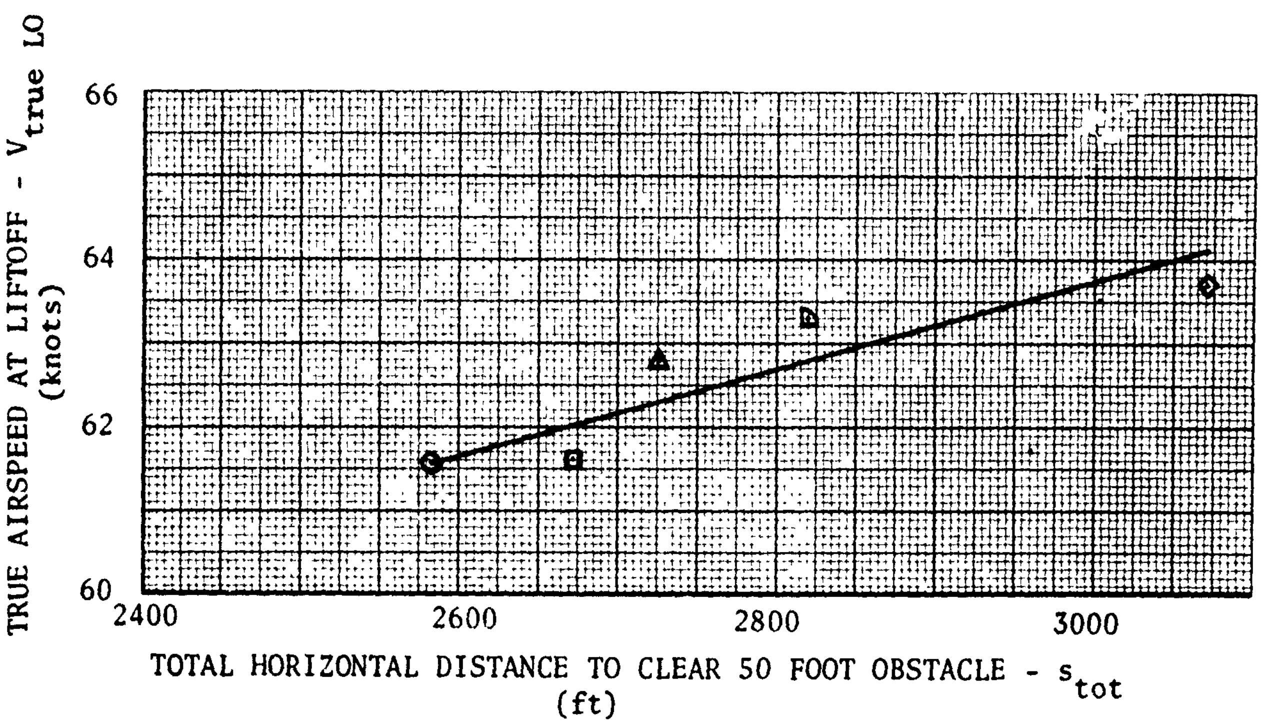


FIGURE 4

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
○	CR	1000	3500	85.37
□	CR	5000	3500	85.37
◇	CR	10000	3500	85.37
Q	PA	1000	3500	85.37

Notes: 1. Fuel mixture: full rich.
2. Cowl flaps closed.

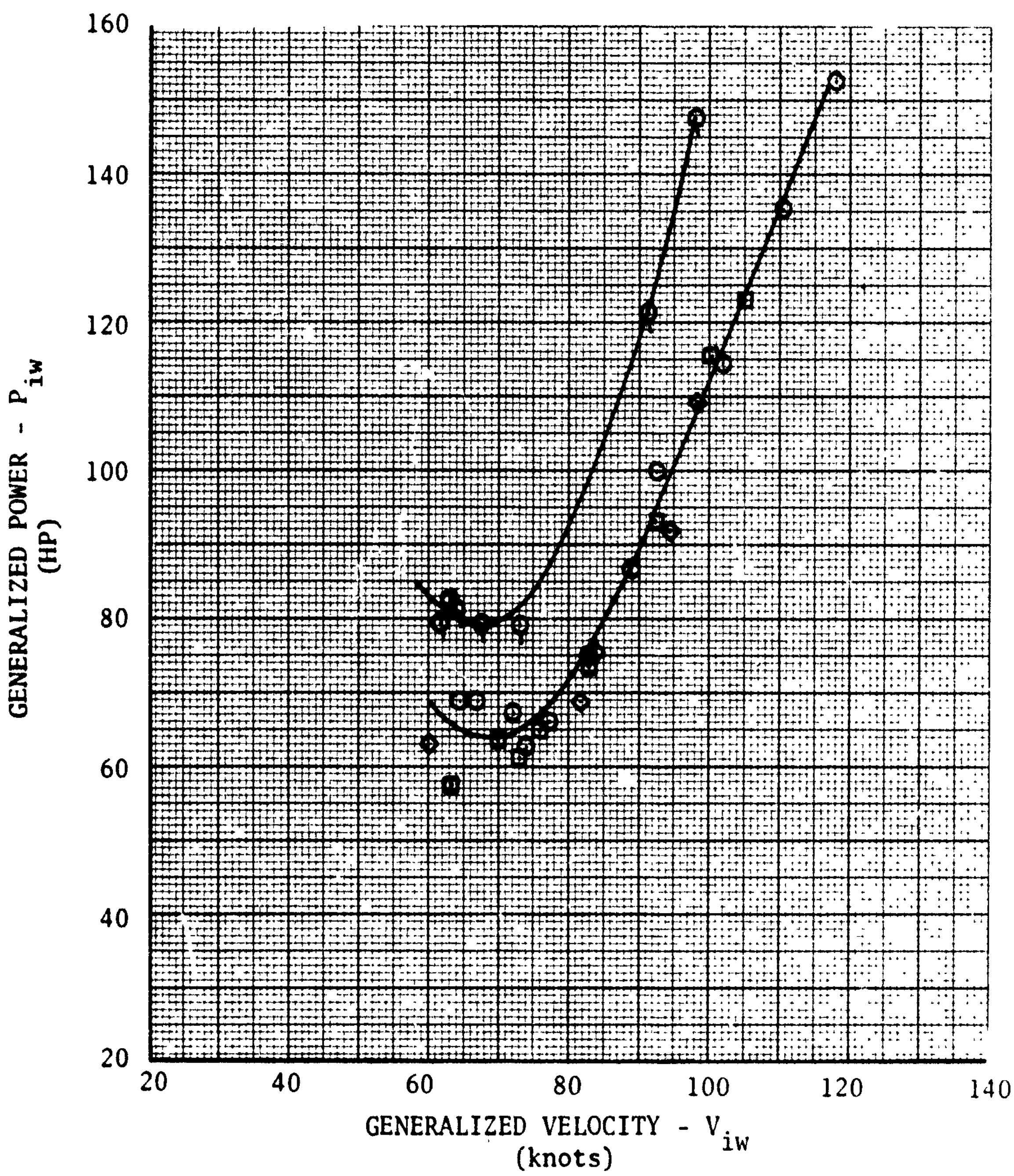


FIGURE 5

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
○	CR	1000	3500	85.34
□	CR	5000	3500	85.34
◇	CR	10000	3500	85.34

Notes: 1. Fuel mixture: full rich.
2. Cowl flaps closed.

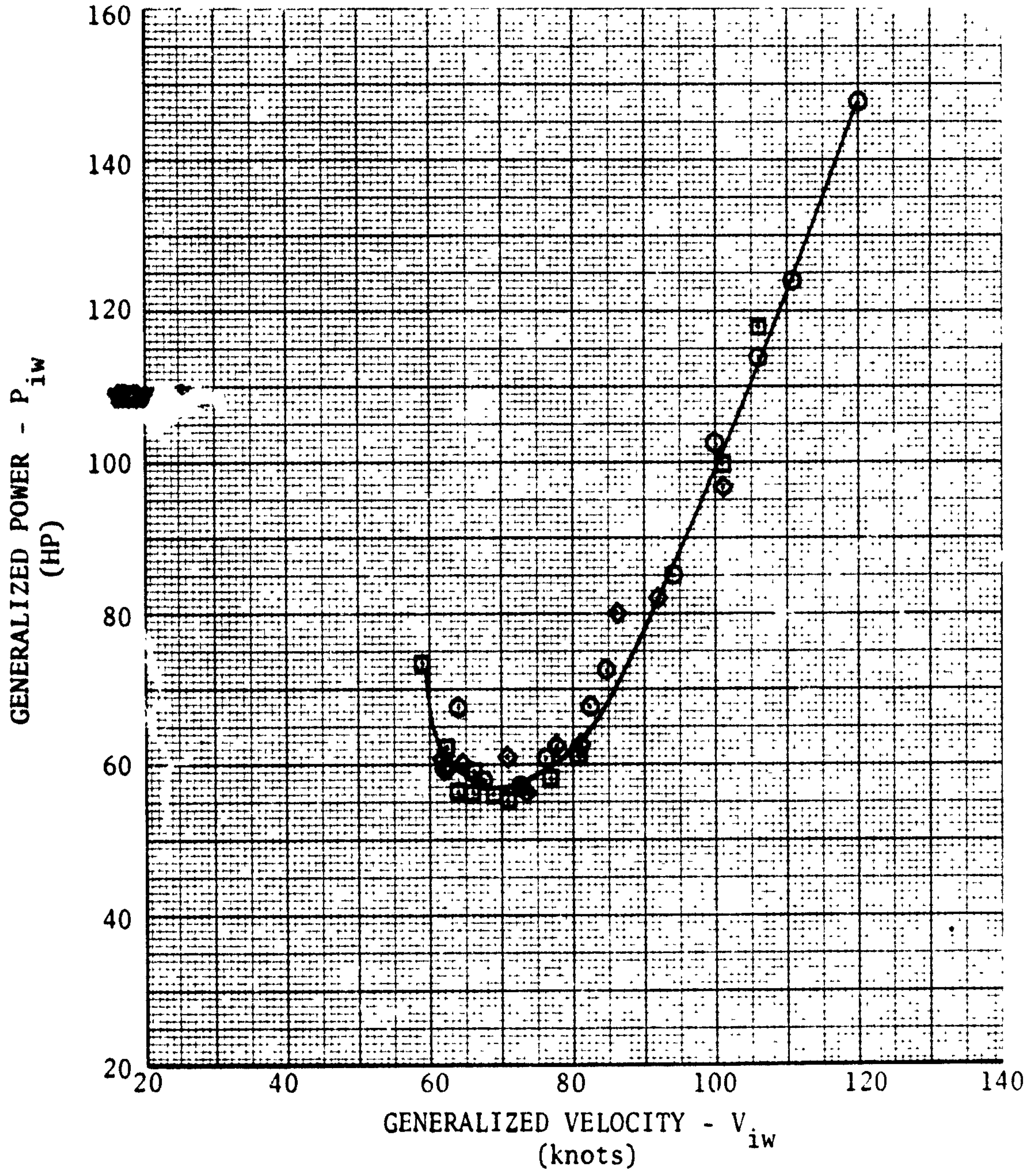


FIGURE 6

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-180[illegible]
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	POSITION OF COWL
○	CR	1000	3500	85.37	Closed
△	CR	1000	3500	85.37	Open

Note: Fuel mixture: full rich.

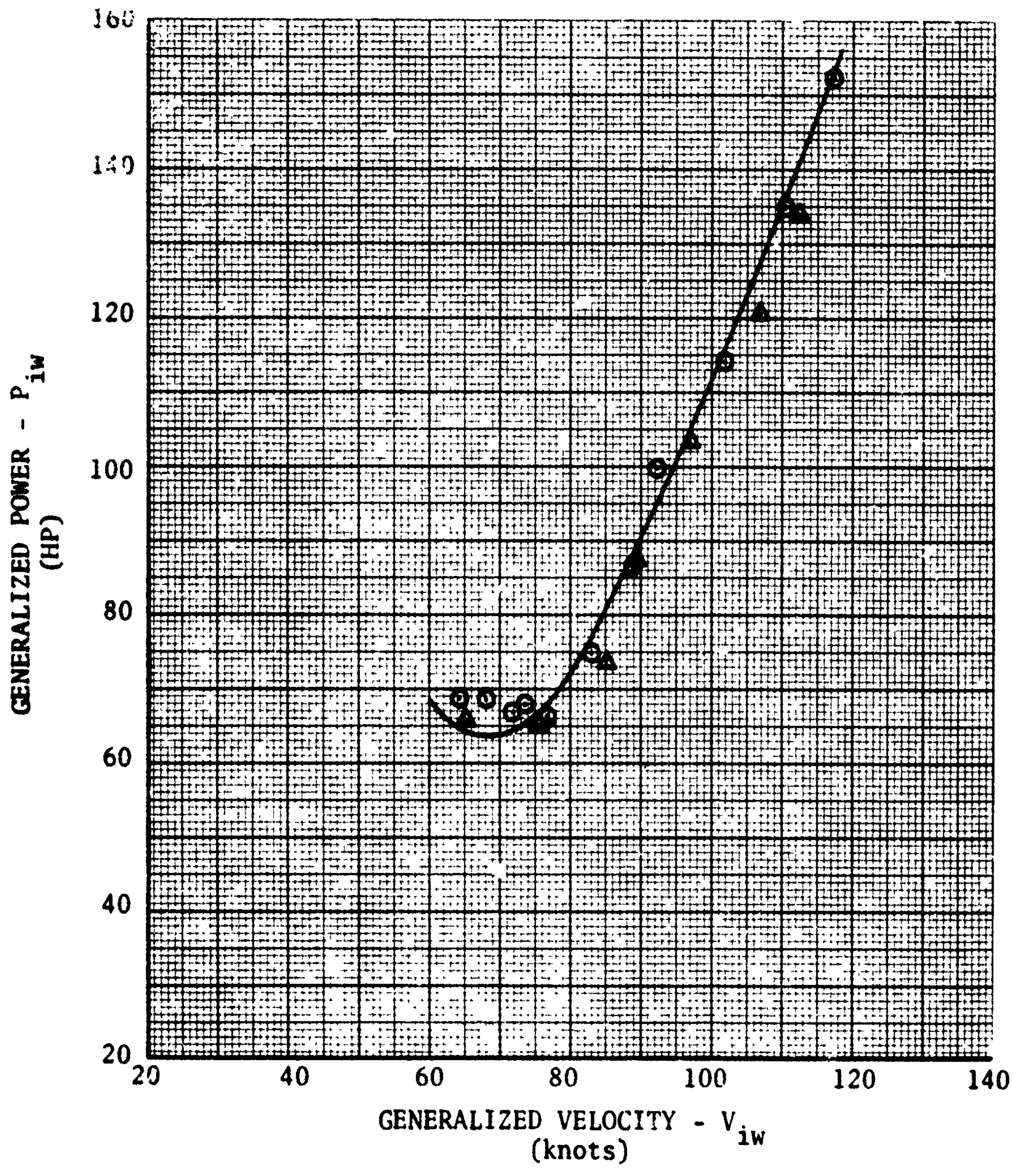


FIGURE 7

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
○	CR	1000	3500	85.37
□	CR	5000	3500	85.37
△	CR	10000	3500	85.37
○ (flagged)	PA	1000	3500	85.37

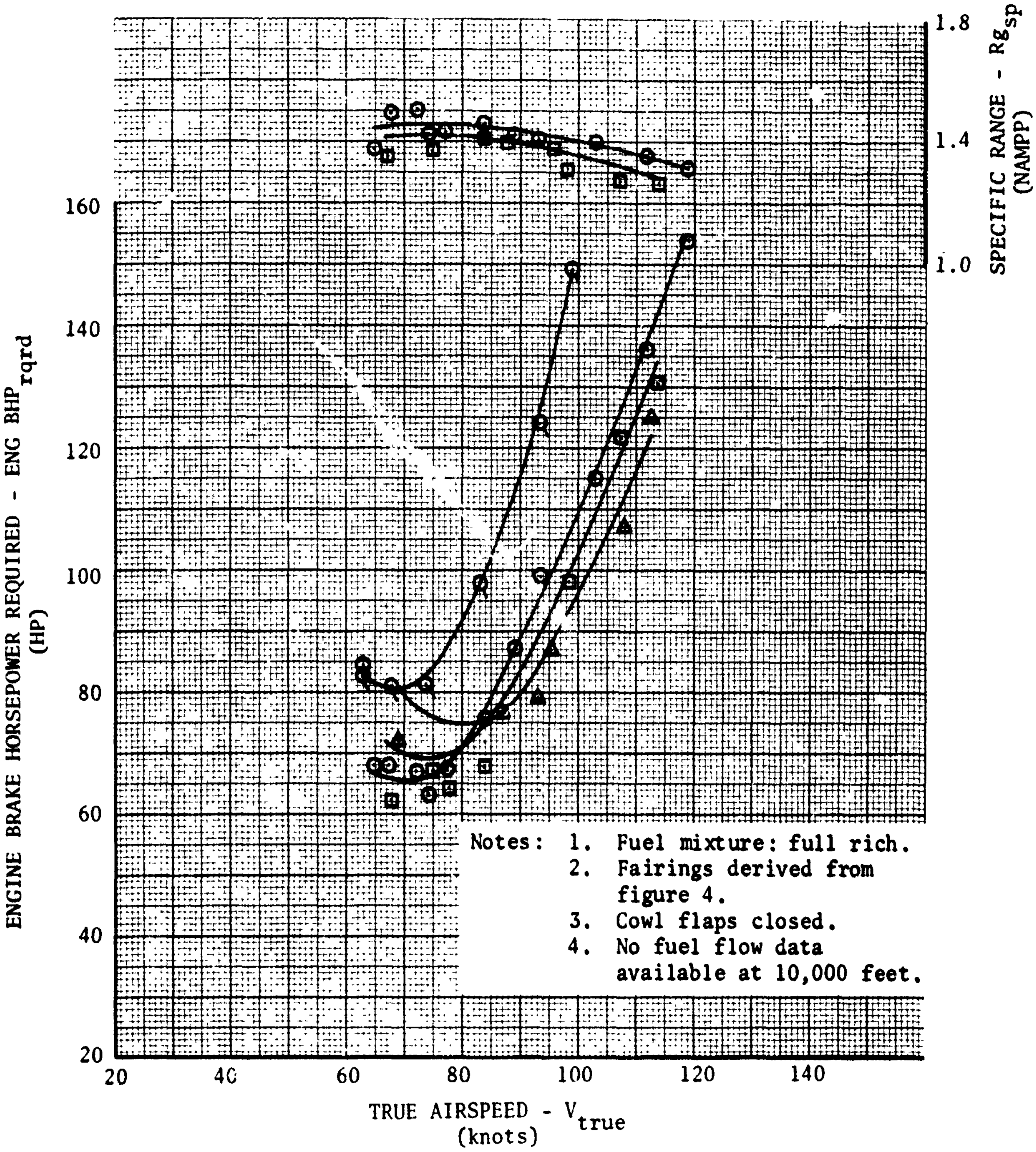


FIGURE 8

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
○	CR	1000	3500	85.34
□	CR	5000	3500	85.34
△	CR	10000	3500	85.34

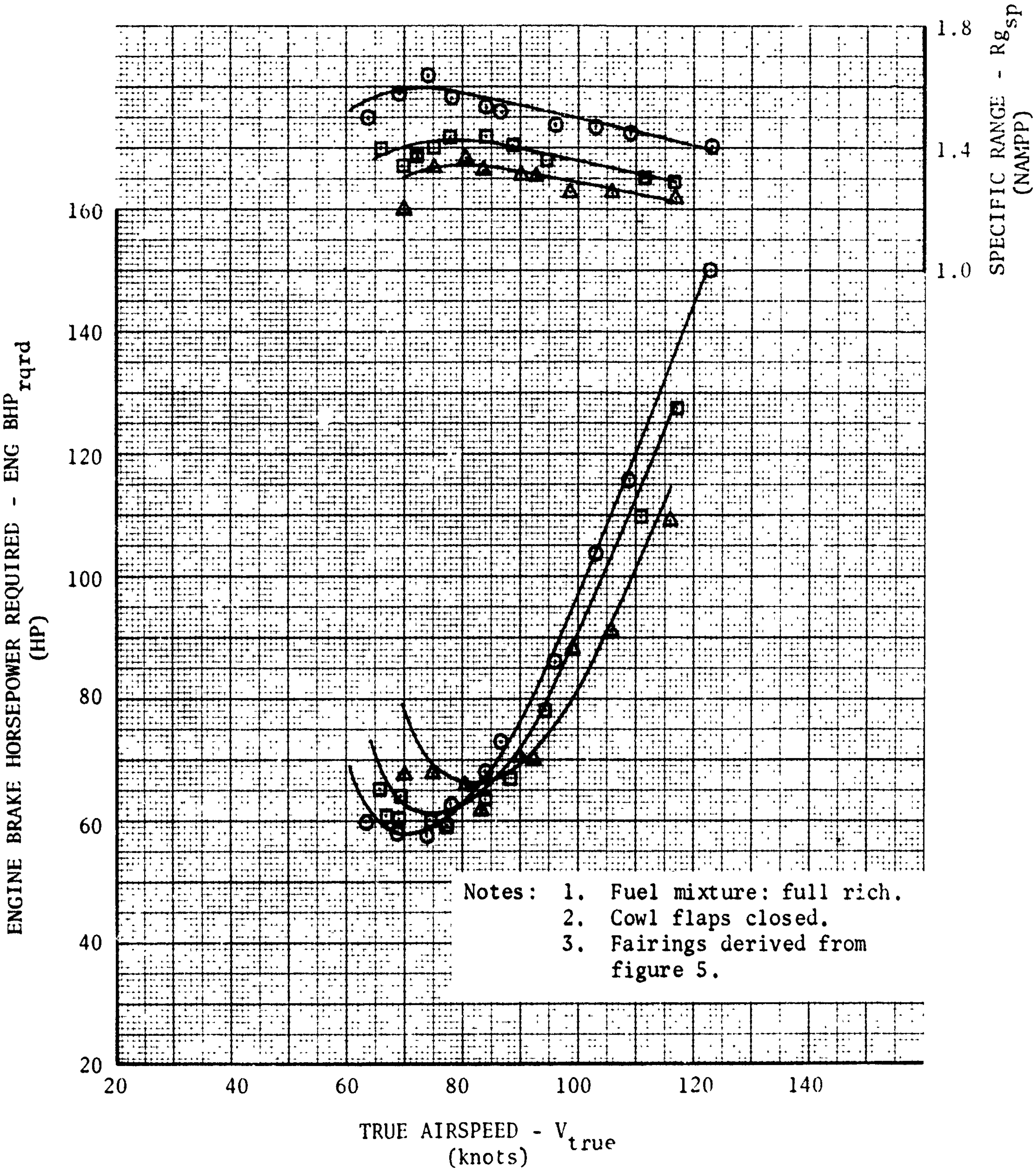


FIGURE 9

LEVEL FLIGHT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	POSITION OF COWL
○	CR	1000	3500	85.37	Closed
△	CR	1000	3500	85.37	Open

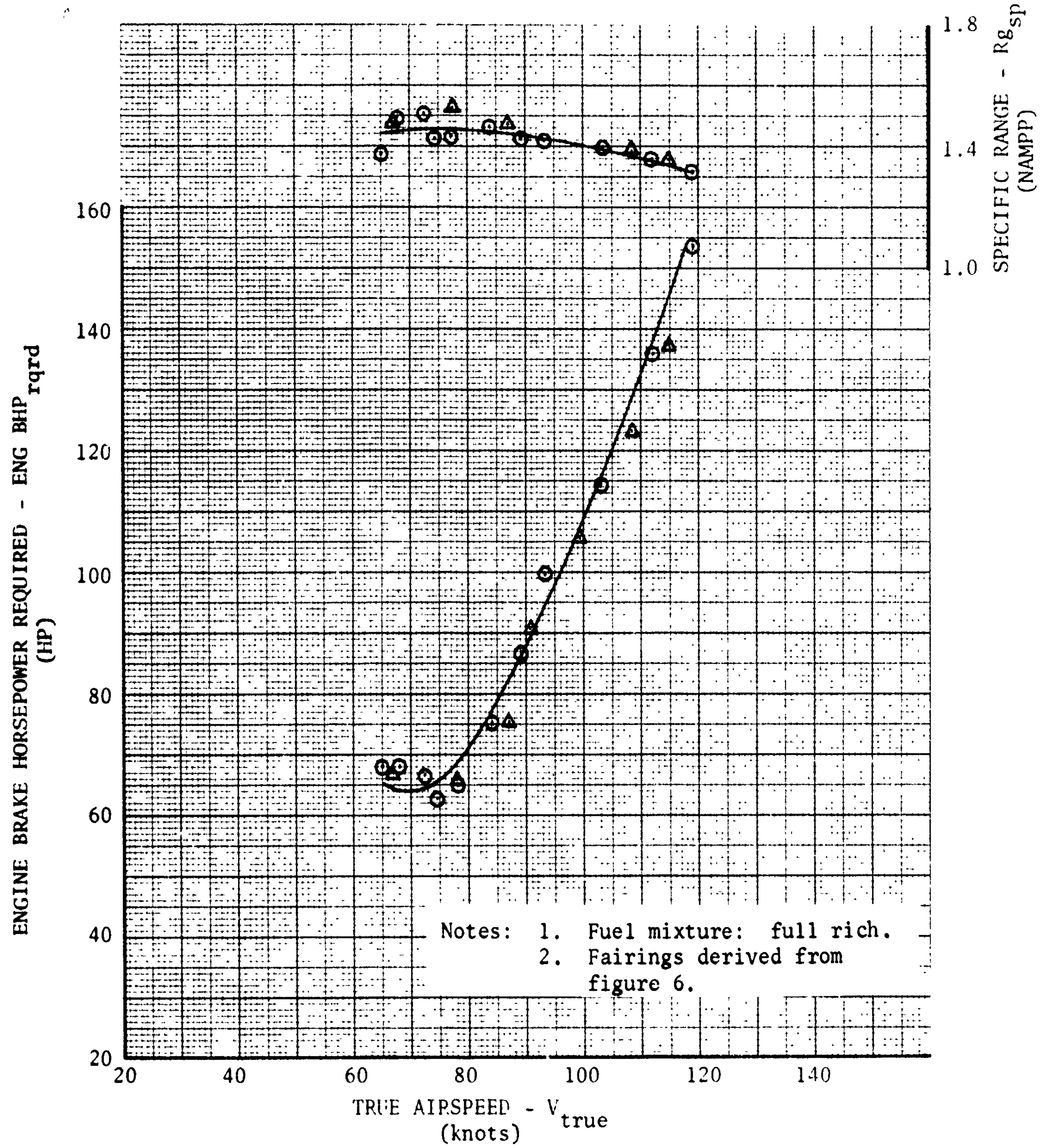


FIGURE 10

DRAG POLAR
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	CG LOCATION (FS) (in.)
○	CR	1000	85.37
□	CR	5000	85.37
◇	CR	10000	85.37
Q	PA	1000	85.37

Note: Wing area = 213.1 ft^2.

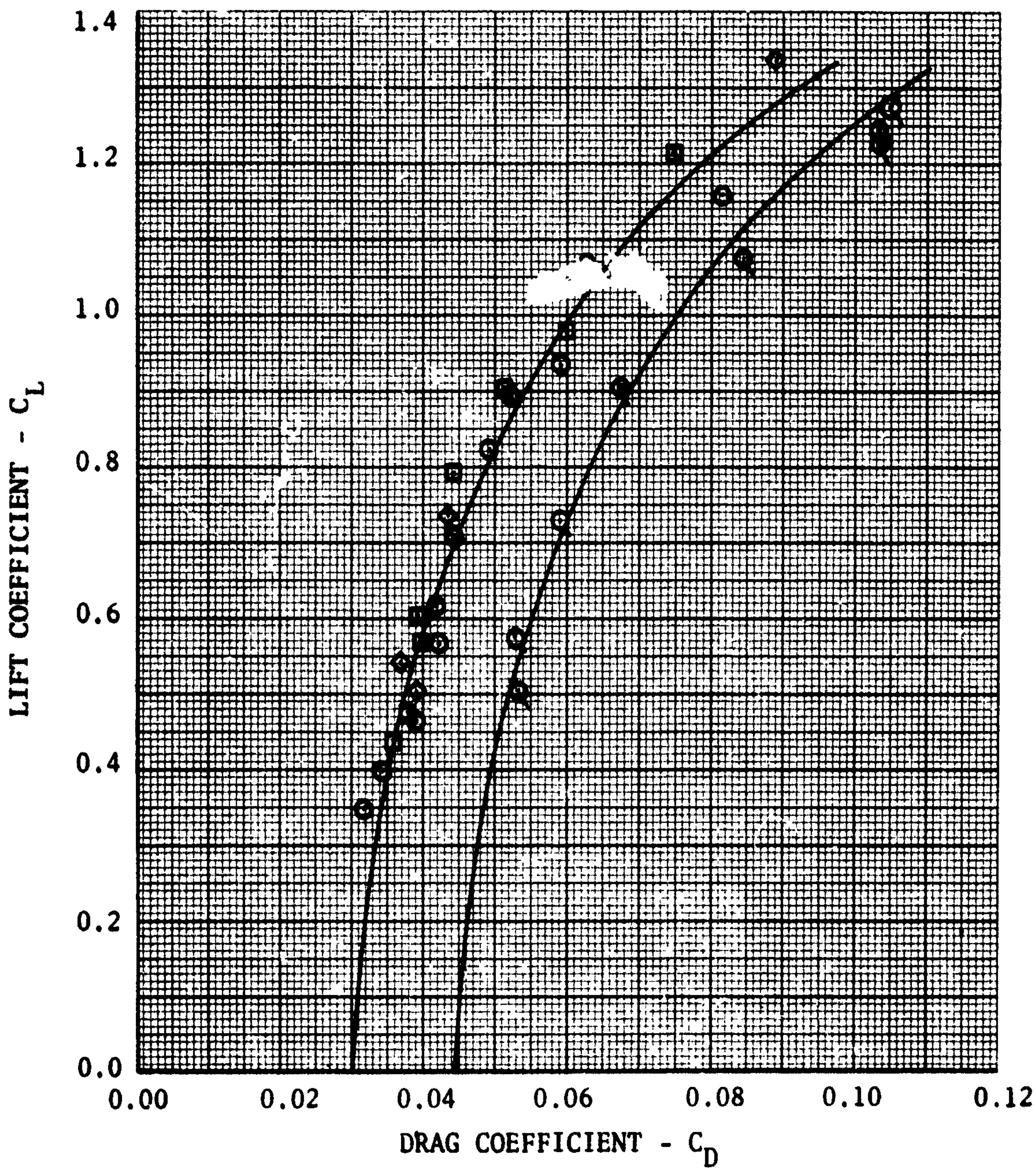


FIGURE 11

DRAG POLAR
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	CG LOCATION (FS) (in.)
○	CR	1000	85.37
□	CR	5000	85.37
◇	CR	10000	85.37

Note: Wing area = 213.1 ft^2.

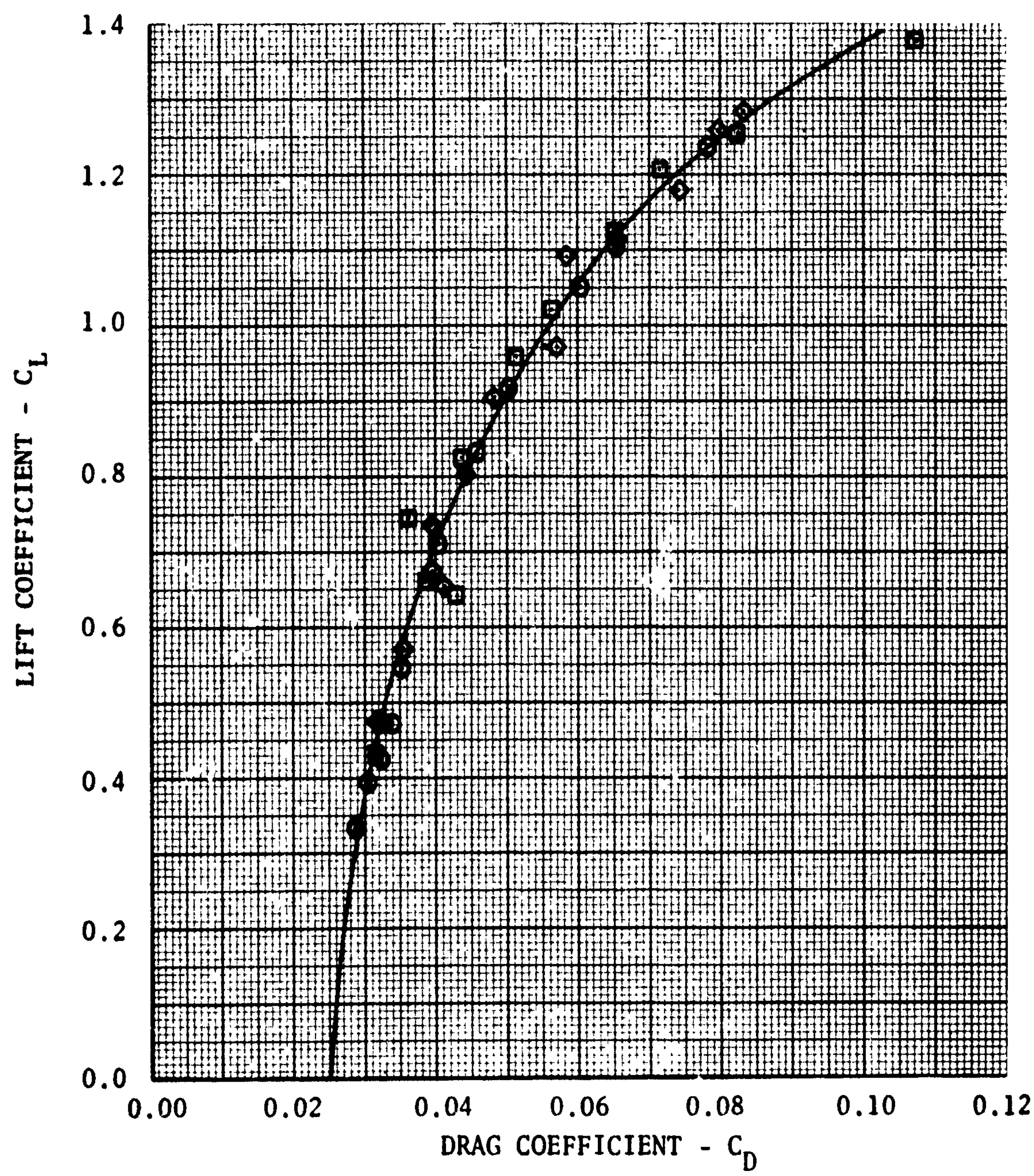


FIGURE 12

SPECIFIC RANGE AND ENDURANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
CR	1000	3500	85.37
CR	5000	3500	85.37

Notes: 1. Fuel mixture: full rich.
2. Curves derived from figure 7.

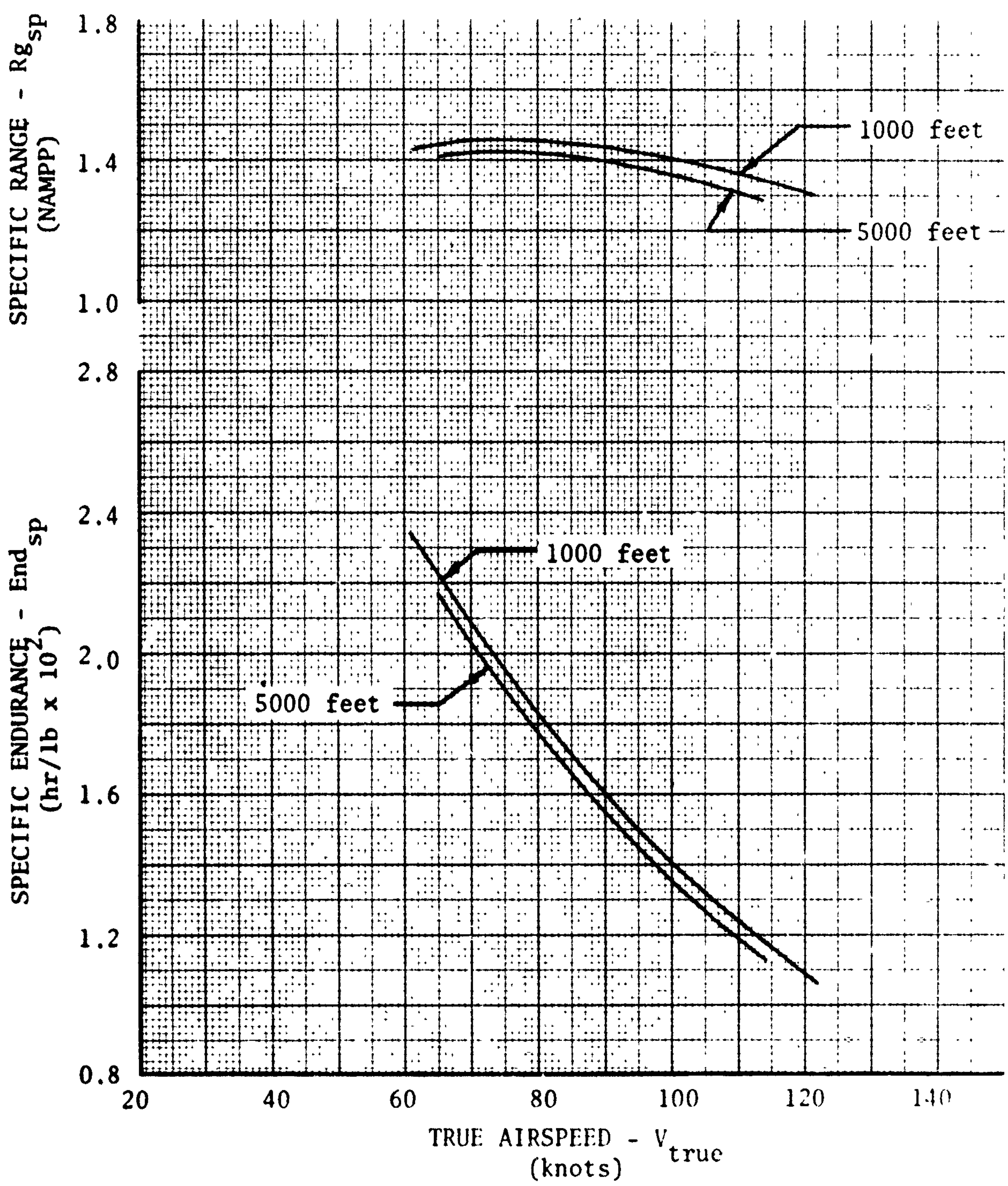


FIGURE 13

SPECIFIC RANGE AND ENDURANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
CR	1000	3500	85.34
CR	5000	3500	85.34
CR	10000	3500	85.34

Notes: 1. Fuel mixture: full rich.
2. Curves derived from figure 8.

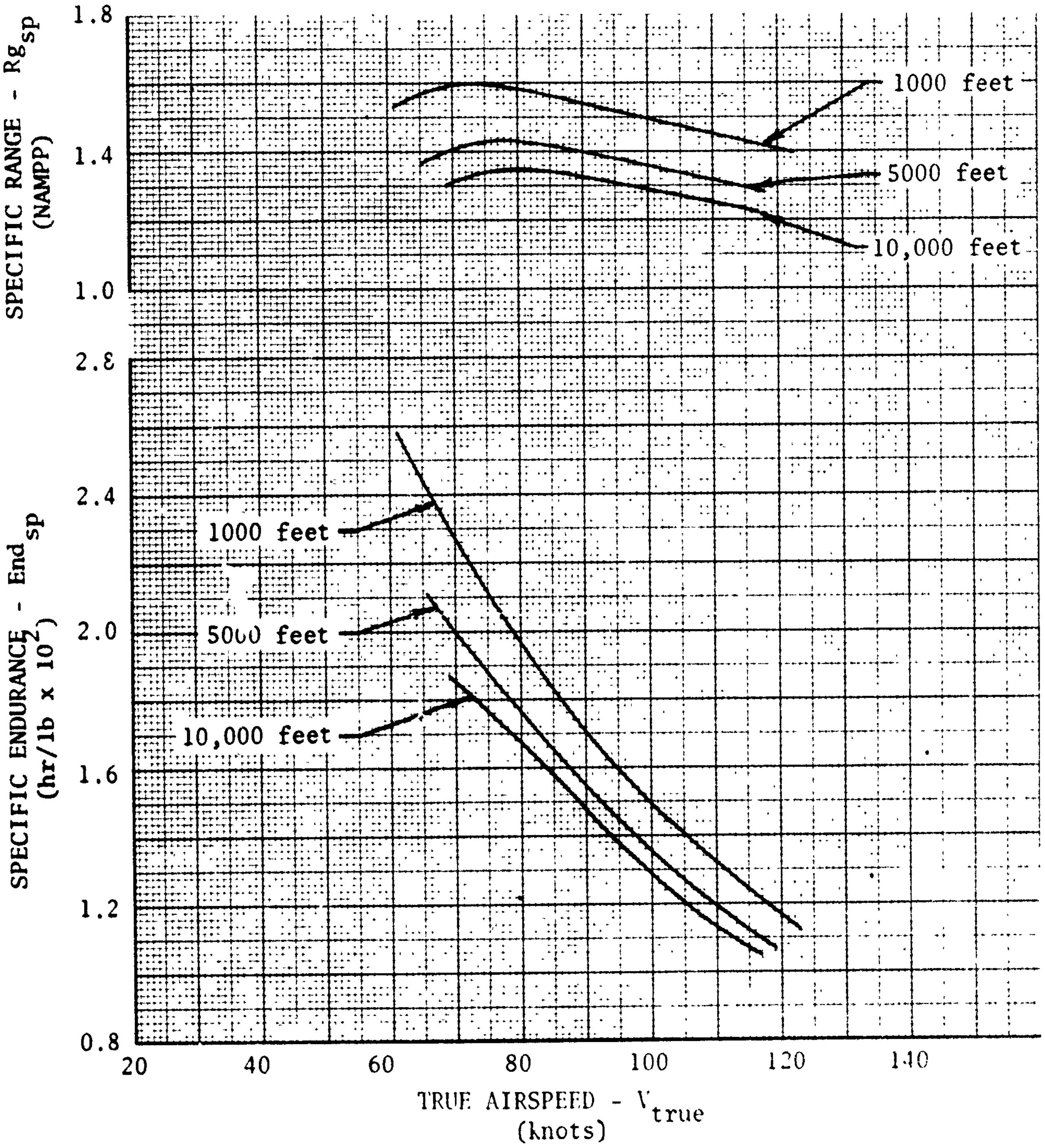


FIGURE 14

SPECIFIC RANGE AND ENDURANCE
DERIVED FROM 3500 POUND DATA
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)
CR	1000	3381	85.34

Note: Fuel mixture: full rich.

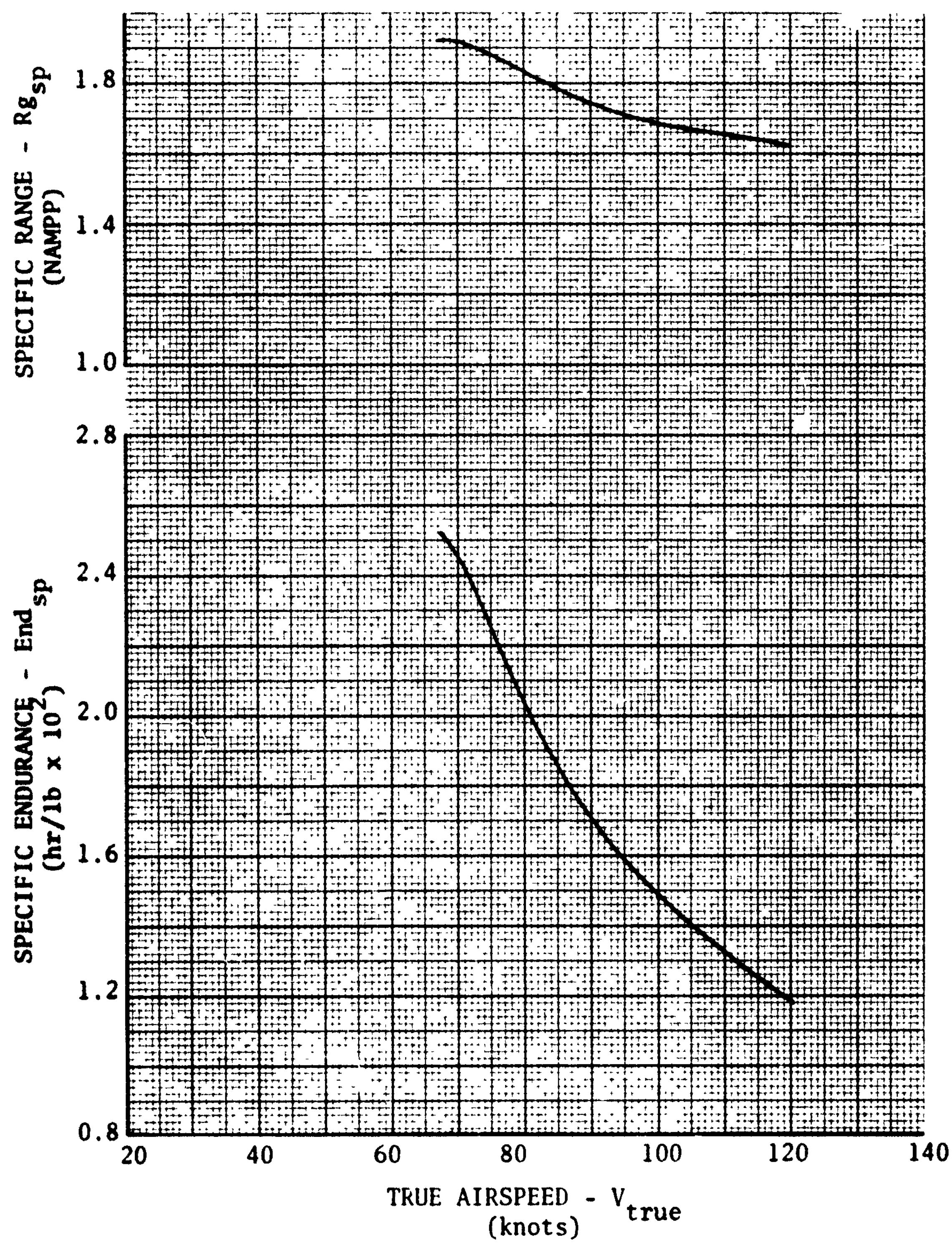


FIGURE 15

SAWTOOTH CLIMB AND DESCENT PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	PRESSURE ALTITUDE (ft)	GROSS WEIGHT (lb)
□	1000	3500
△	3000	3500
○	5000	3500

Notes:
1. Mixture setting: full rich.
2. Climb: full throttle.
3. Descent: idle power.
4. Cowl flaps closed.

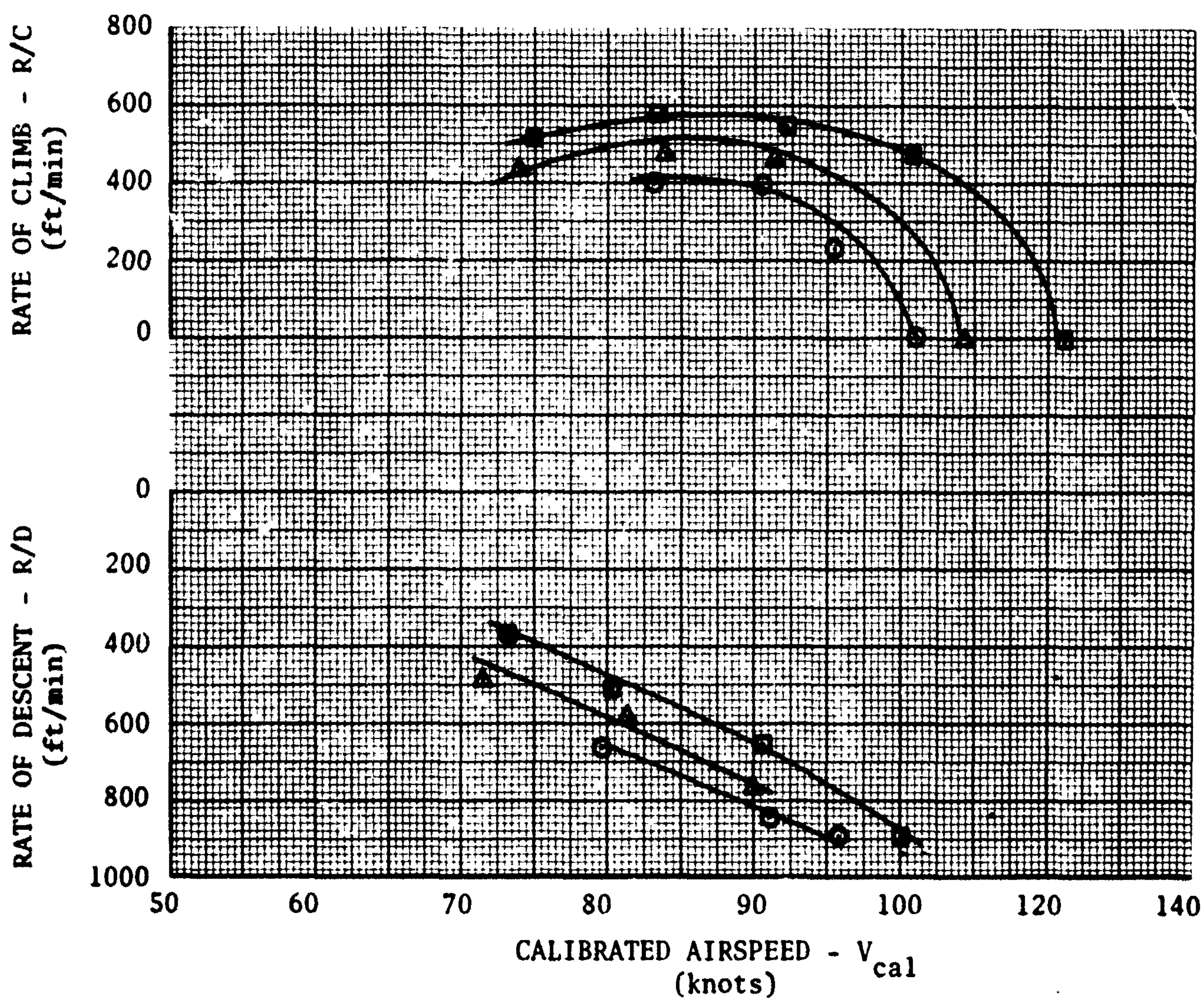


FIGURE 16

SAWTOOTH CLIMB PERFORMANCE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

Notes:
1. Derived from figure 15 at airspeed for maximum rate of climb.
2. Mixture setting: full rich.
3. Gross weight: 3500 pounds.
4. Full throttle.
5. Cowl flaps closed.

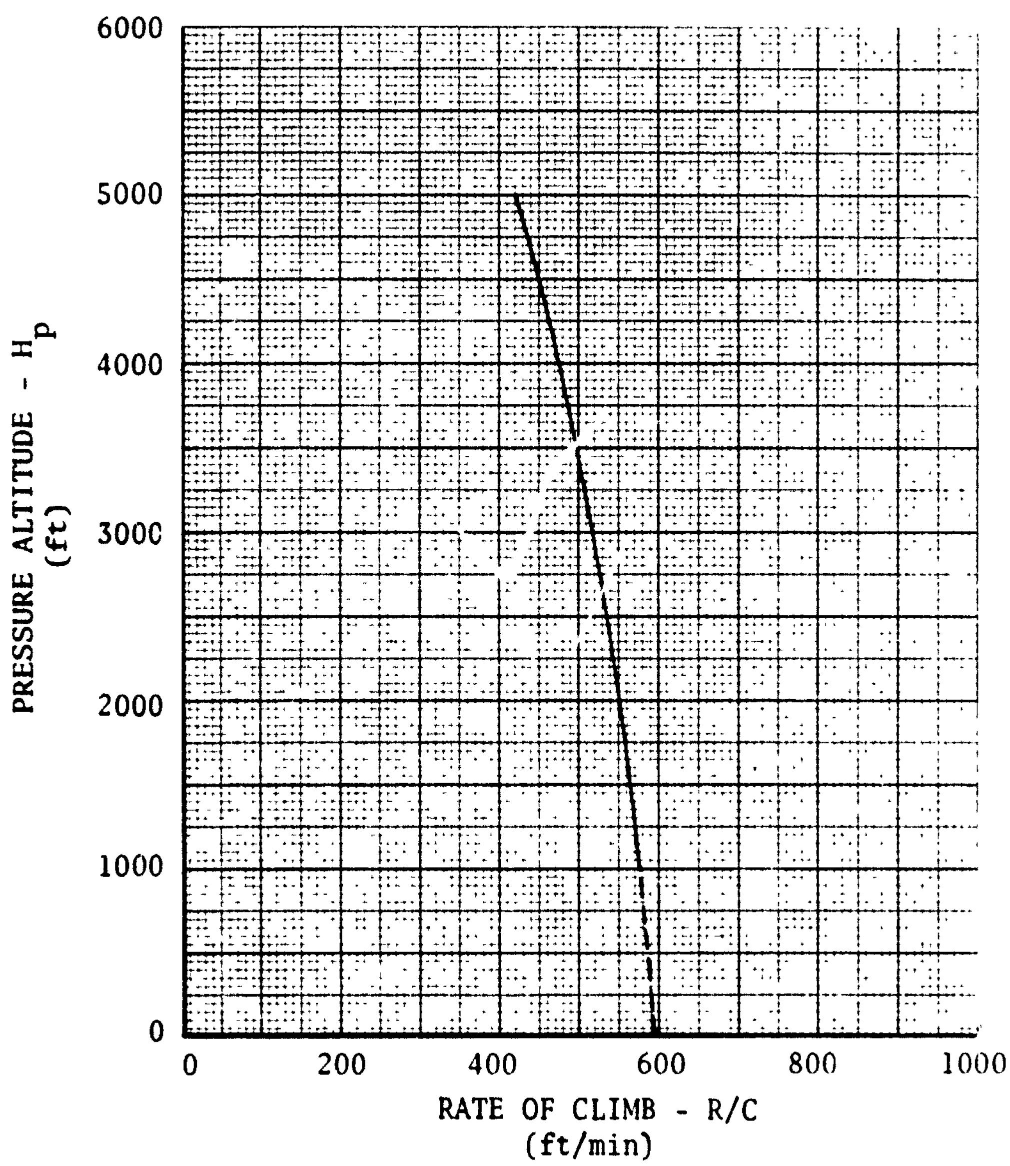


FIGURE 17

STATIC LONGITUDINAL STABILITY
YO-3A S/N 69-18000

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)	MISSION EQUIPMENT
○	P-CLIMB	6710	3510	86.08	81.7	Installed
□	CR	5000	3500	85.35	75.0	Not Installed
◇	CR	5000	3500	84.66	75.0	Installed
△	CR	6980	3450	86.08	93.0	Installed
▽	CR	5730	3500	84.66	104.7	Installed
⬡	PA	5878	3415	86.08	68.5	Installed

Note: Shaded symbol denotes trim.

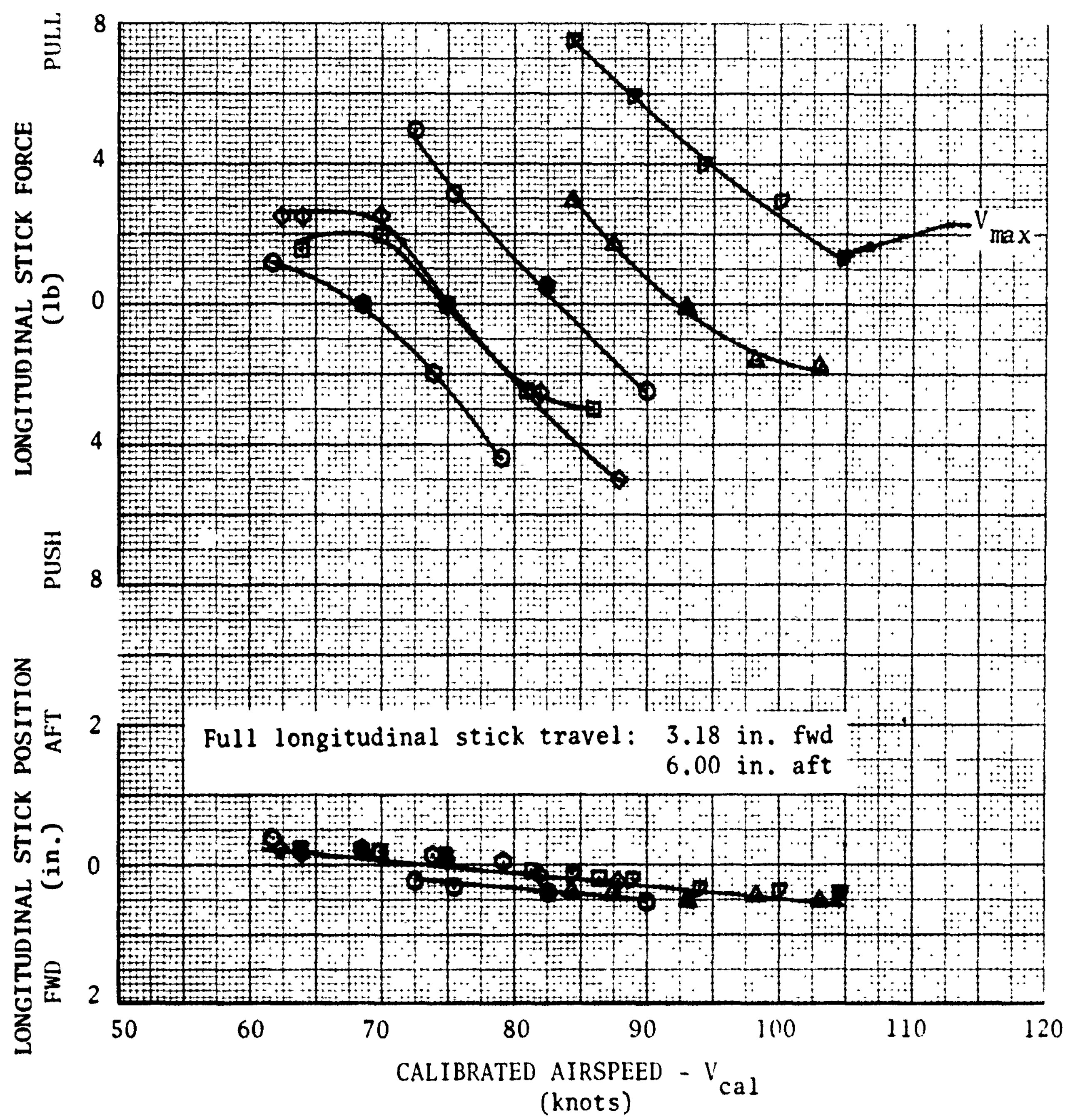


FIGURE 17

STATIC LONGITUDINAL STABILITY
YO-3A S/N 69-18000

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)	MISSION EQUIPMENT
○	P-CLIMB	6710	3510	86.08	81.7	Installed
□	CR	5000	3500	85.35	75.0	Not Installed
◇	CR	5000	3500	84.66	75.0	Installed
△	CR	6980	3450	86.08	93.0	Installed
◻	CR	5730	3500	84.66	104.7	Installed
⬡	PA	5878	3415	86.08	68.5	Installed

Note: Shaded symbol denotes trim.

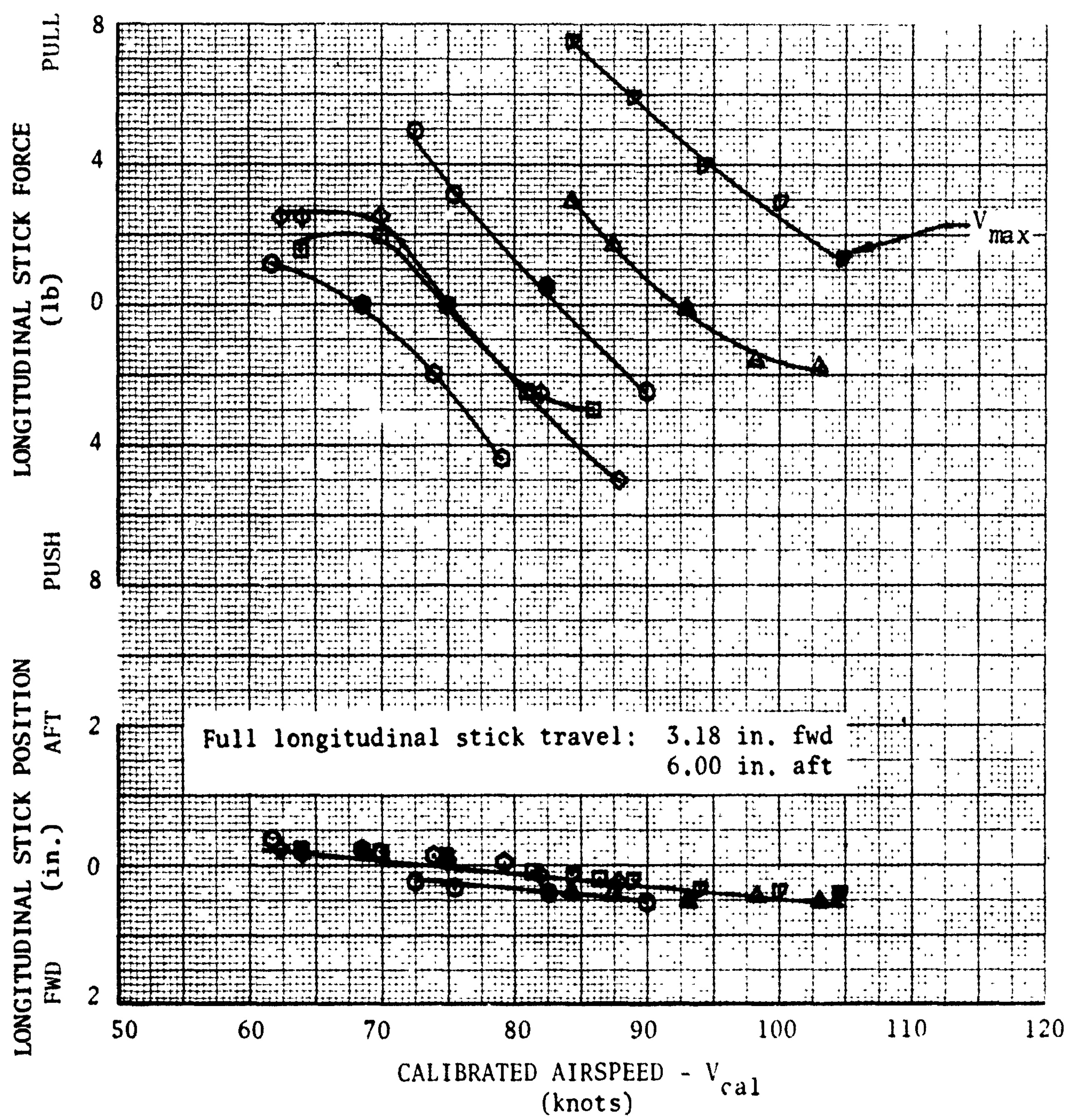


FIGURE 18

LONGITUDINAL DYNAMIC STABILITY
LONG PERIOD STICK FREE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	5000	3500	85.37	63

ω_n = 0.358 rad/sec ζ = -0.221 C_2 = 0.486 cycles T = 18 sec

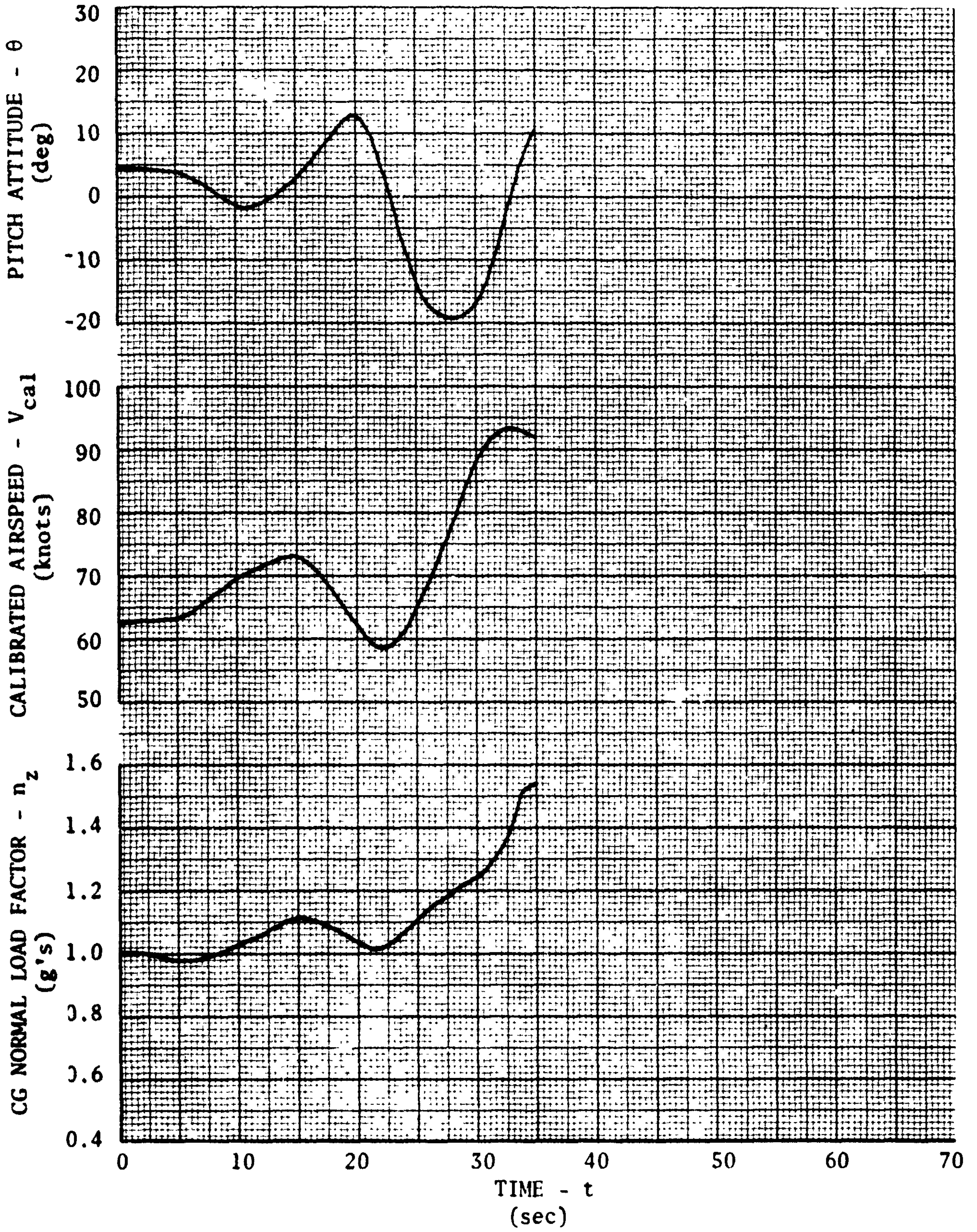


FIGURE 19

LONGITUDINAL DYNAMIC STABILITY
LONG PERIOD STICK FREE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
WO	6460	3500	86.08	68

ω_n = 0.210 rad/sec ζ =-0.083 C_2 = 1.320 cycles T = 30 sec

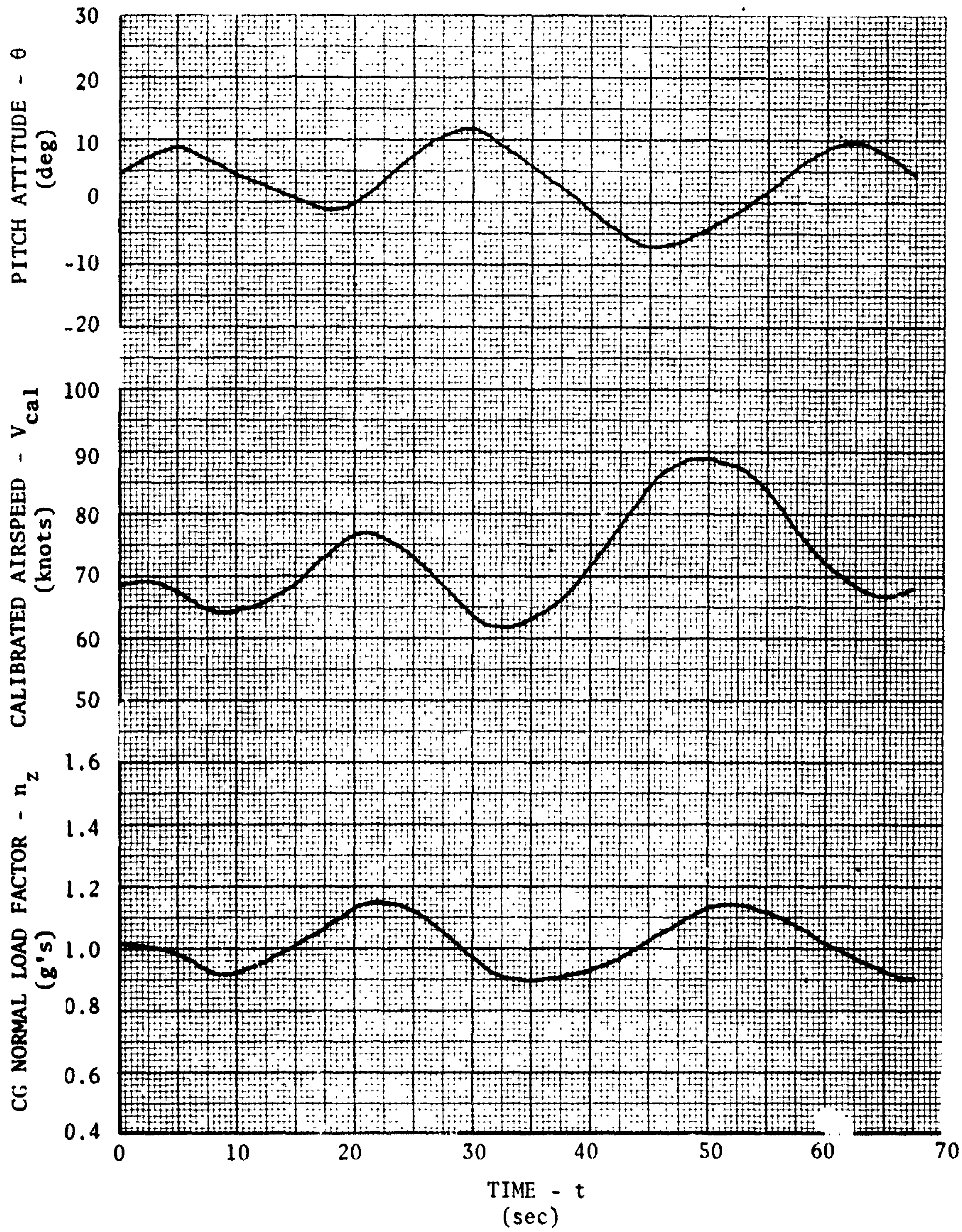


FIGURE 20

LONGITUDINAL DYNAMIC STABILITY
LONG PERIOD STICK FREE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CLIMB	6460	3500	86.08	82

ω_n = 0.178 rad/sec ζ = +0.010 $C_{1/2}$ = 20 cycles T = 35 sec

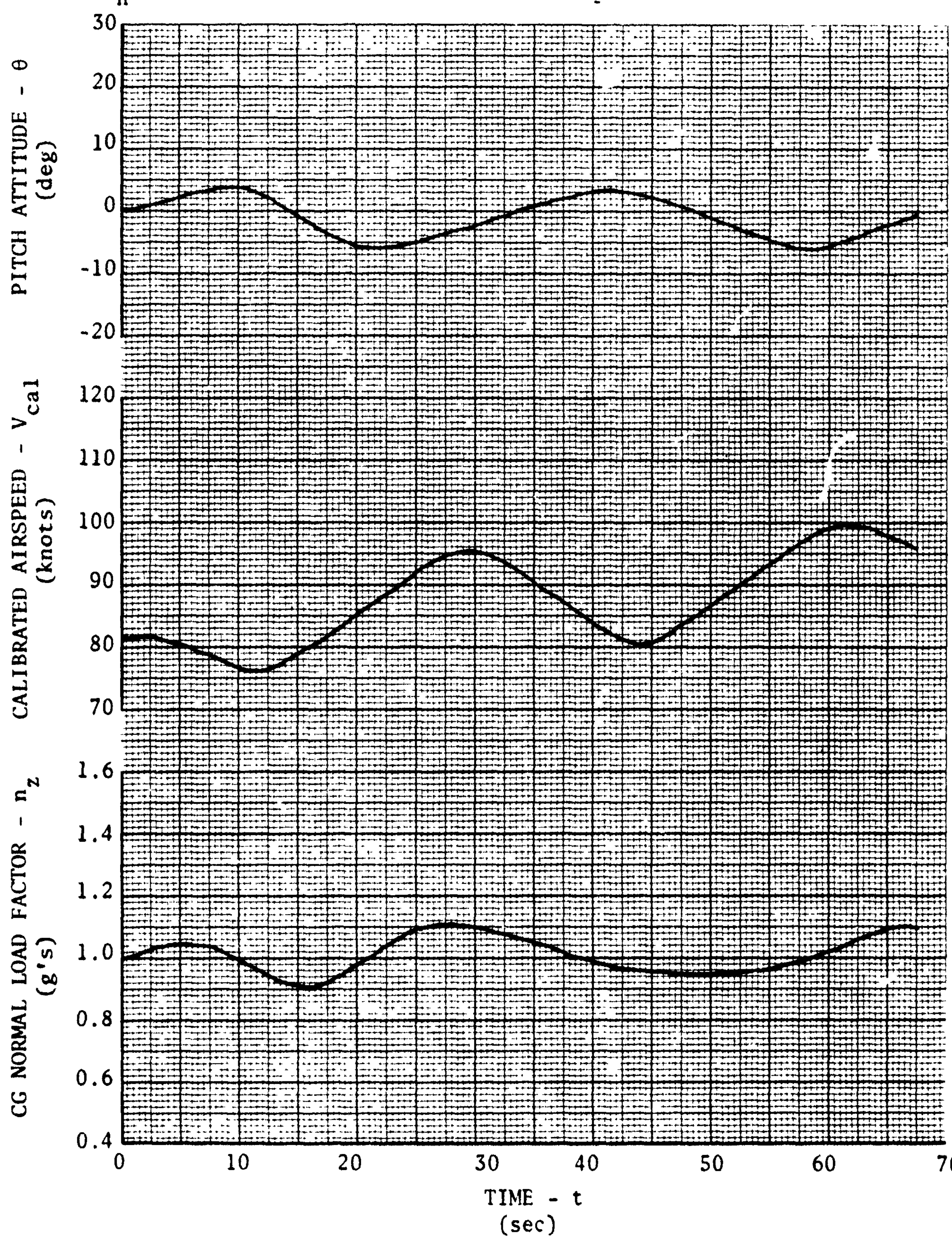


FIGURE 21

LONGITUDINAL DYNAMIC STABILITY
LONG PERIOD STICK FREE
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	6460	3500	86.08	92.5

ω_n = 0.165 rad/sec ζ = -0.070 C_2 = 1.57 cycles T = 38 sec

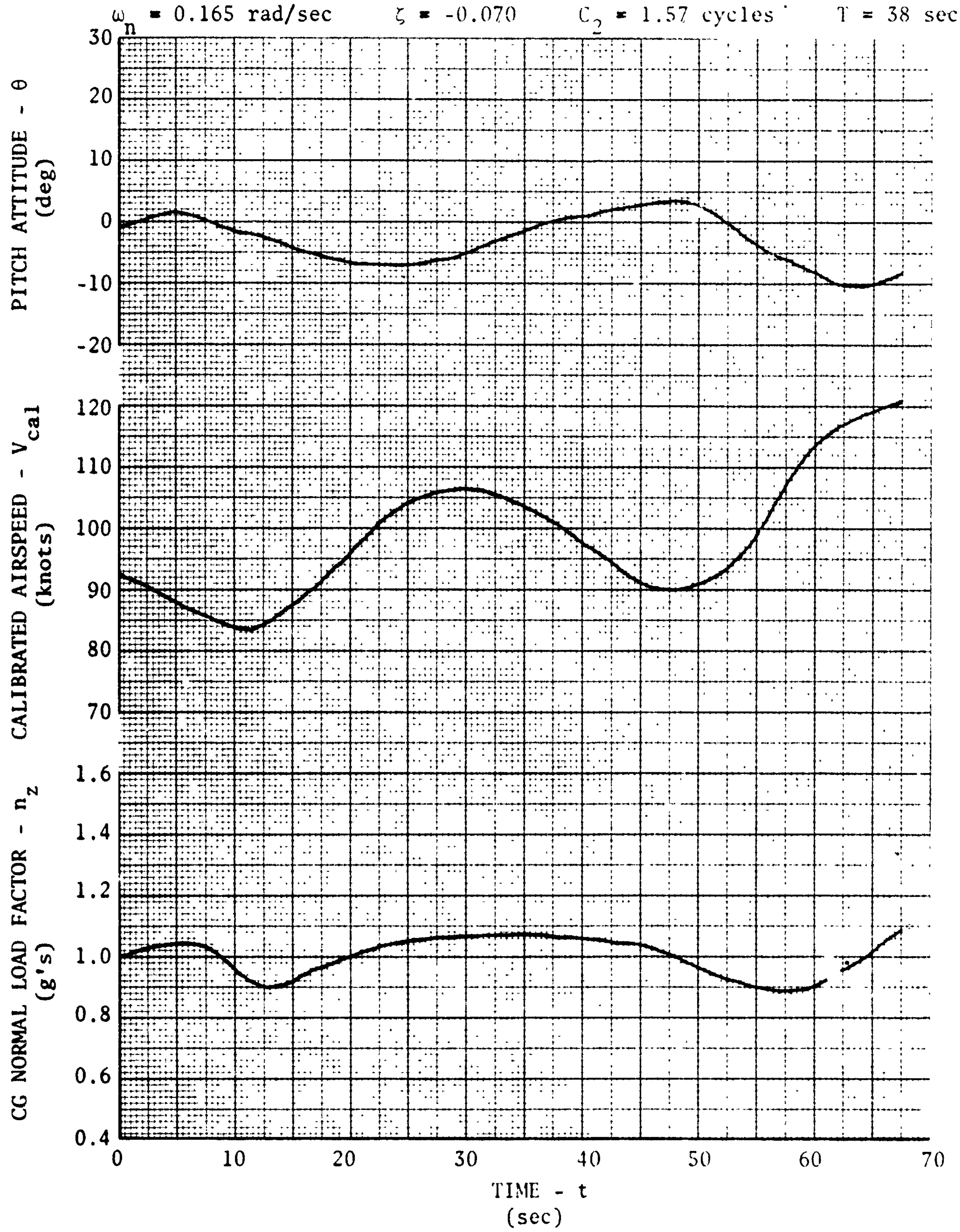


FIGURE 22

STATIC LATERAL DIRECTIONAL STABILITY
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	5000	3500	86.08	66.6

Note: Shaded symbol denotes trim.

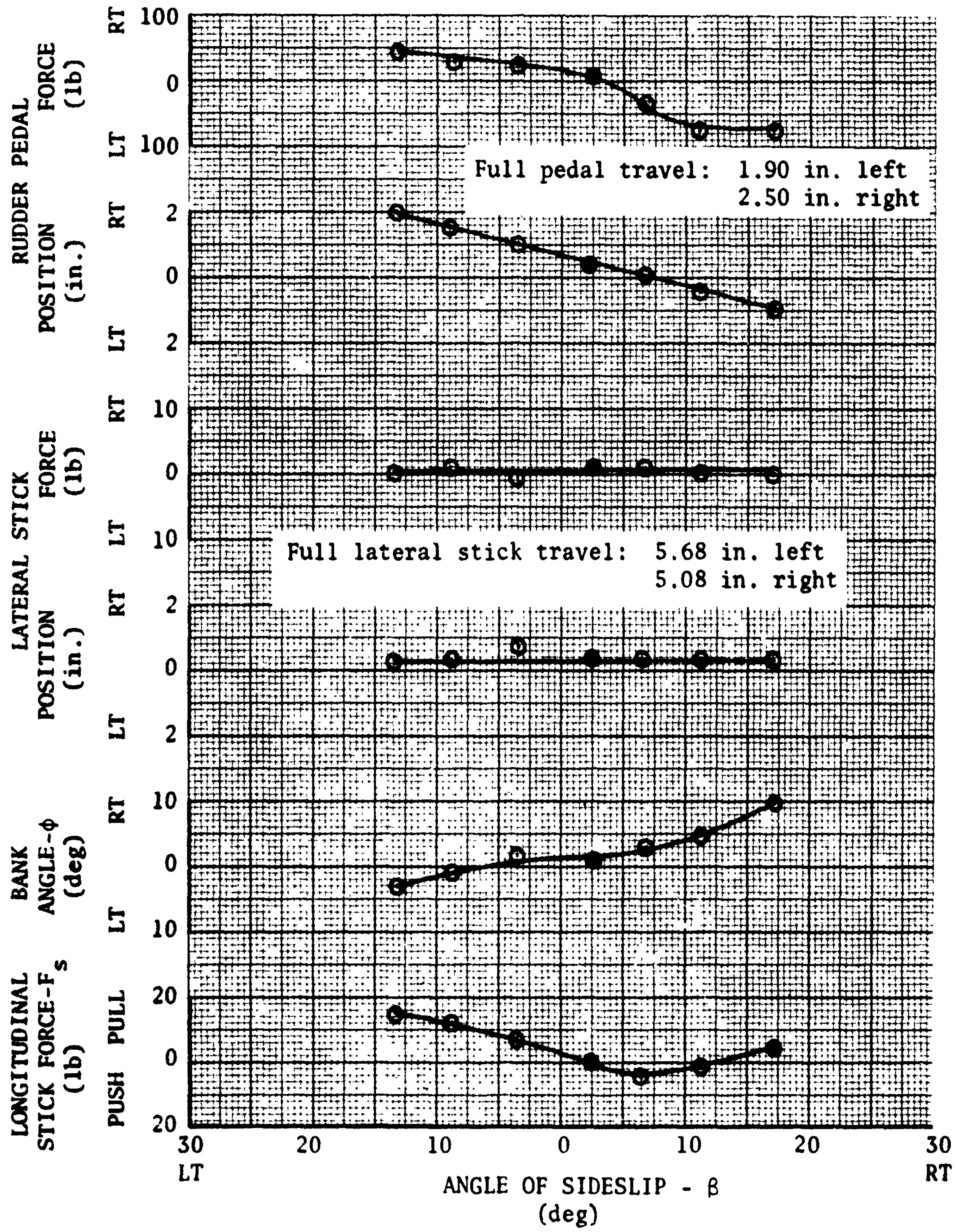


FIGURE 23

STATIC LATERAL DIRECTIONAL STABILITY
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	5000	3500	85.37	73.0

Note: Shaded symbol denotes trim.

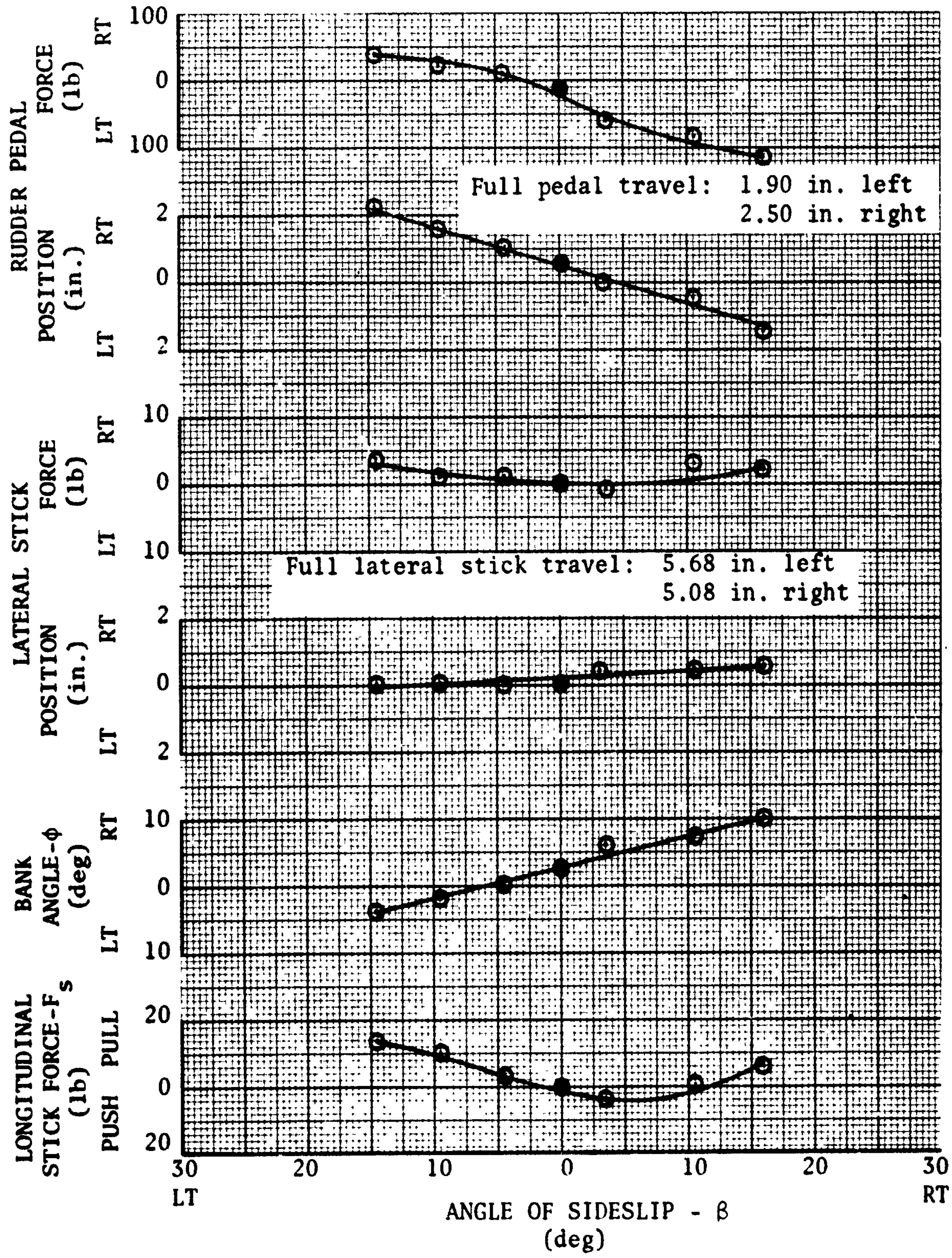


FIGURE 24

STATIC LATERAL DIRECTIONAL STABILITY
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
PA	5000	3500	86.08	68.3

Notes: 1. Shaded symbol denotes trim.
2. Rate of descent = 500 ft/min.

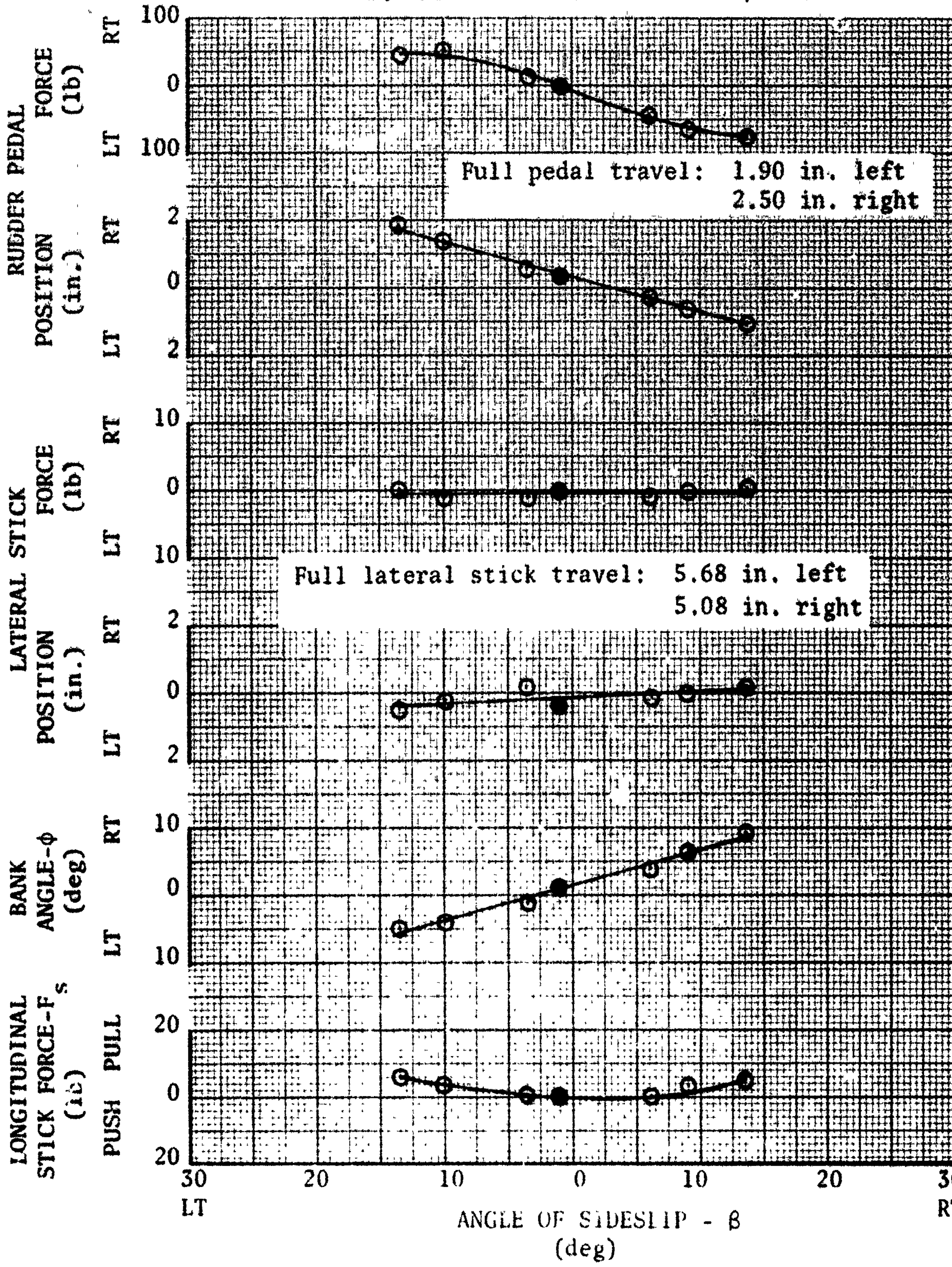


FIGURE 25

STATIC LATERAL DIRECTIONAL STABILITY
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
WO-CLIMB	5000	3500	86.08	68.3

Notes: 1. Shaded symbol denotes trim.
2. Cowl flaps closed.

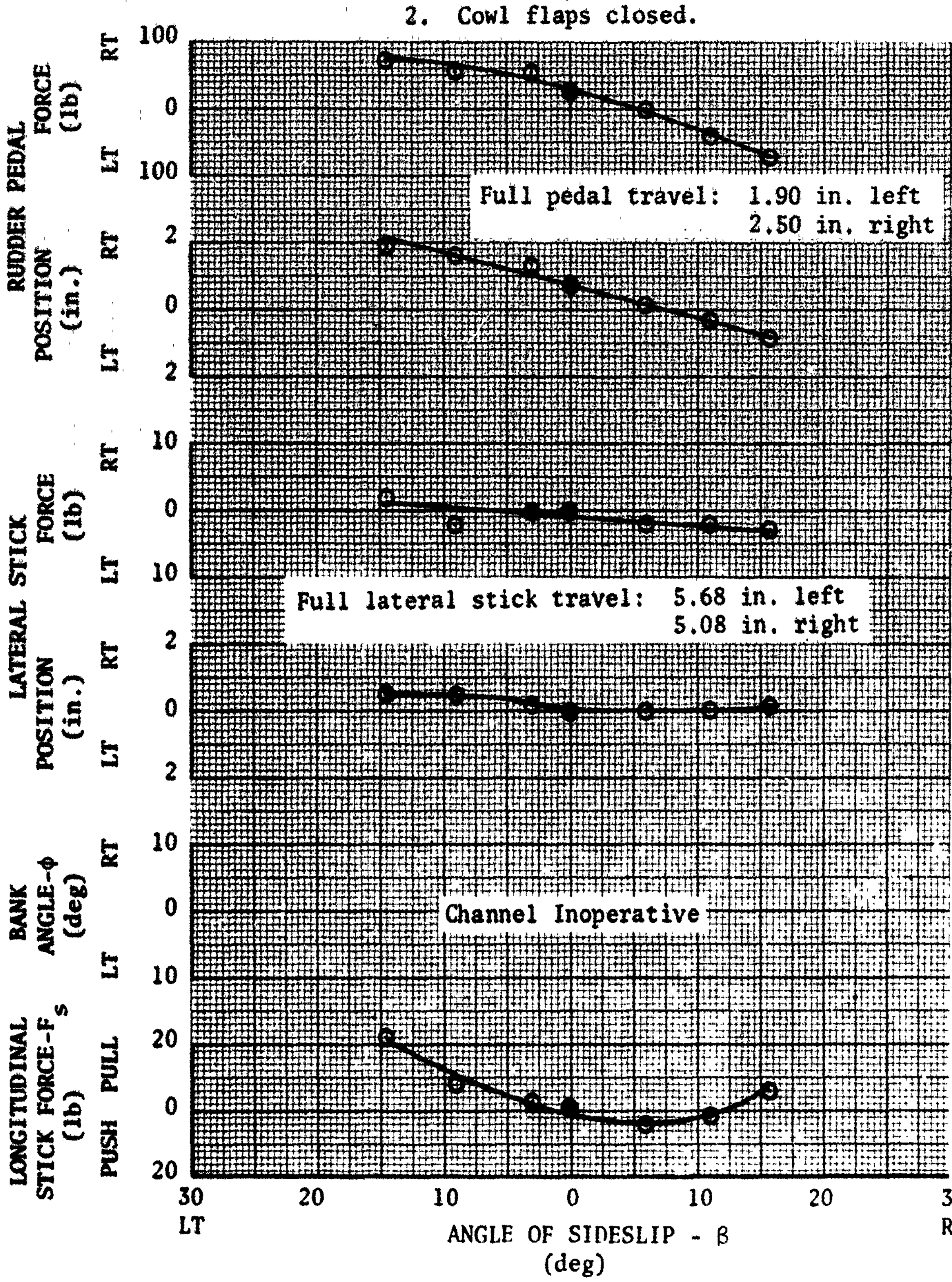


FIGURE 26

STATIC LATERAL DIRECTIONAL STABILITY
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
P-CLIMB	5000	3500	86.08	81.5

Note: No trim point.

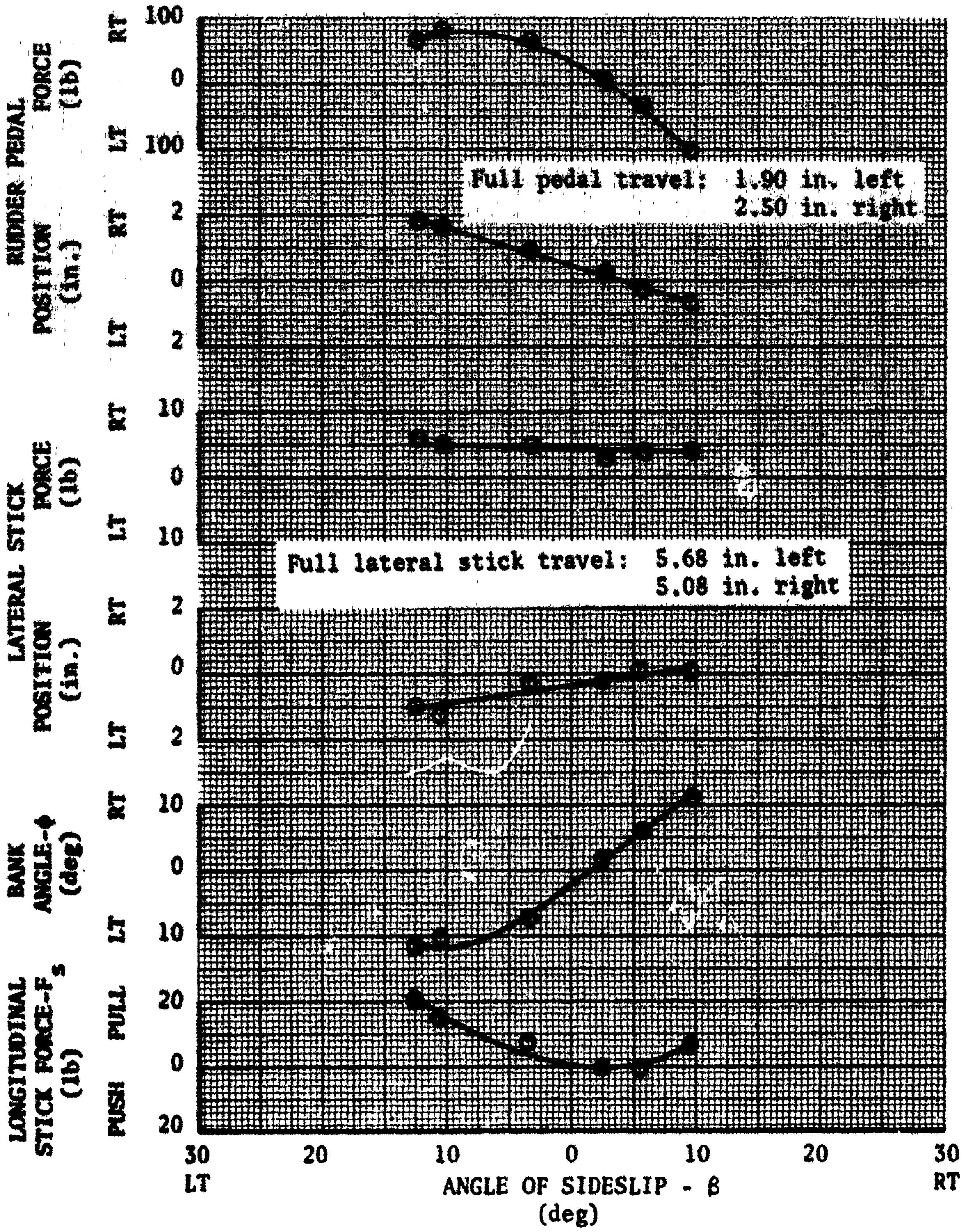


FIGURE 27

DUTCH ROLL OSCILLATION
STICK FREE PEDAL FIXED
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	8300	3500	86.08	73.0

ω_n = 1.64 rad/sec ζ = 0.71 $C_{1/2}$ = 0.109 cycles ϕ/β = 1:3.8

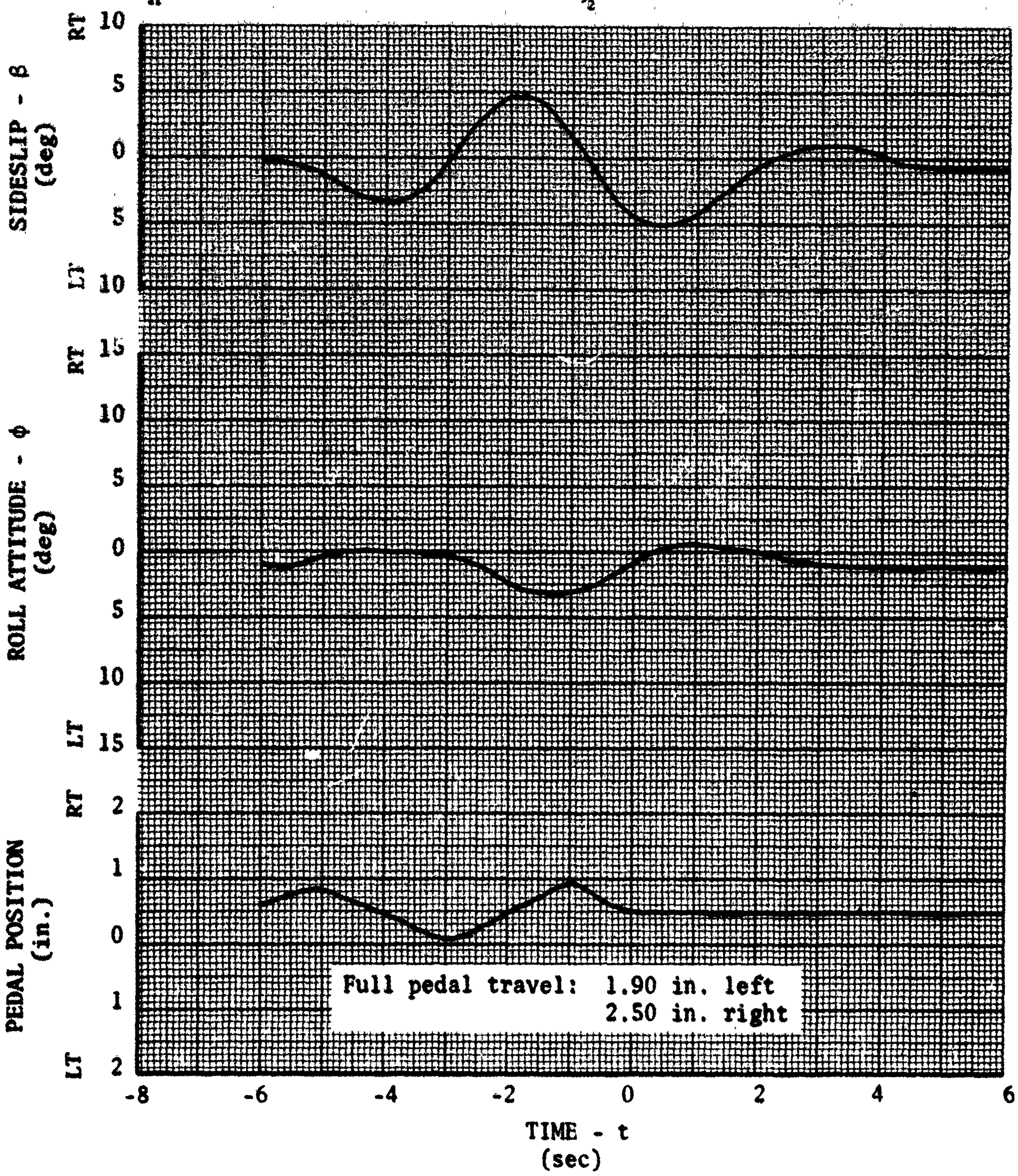


FIGURE 28

DUTCH ROLL OSCILLATION
STICK FREE PEDAL FIXED
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	7350	3500	86.08	104.0

ω_n = 2.17 rad/sec ζ = 0.65 $C_{\frac{1}{2}}$ = 0.128 cycles ϕ/β = 1:5.3

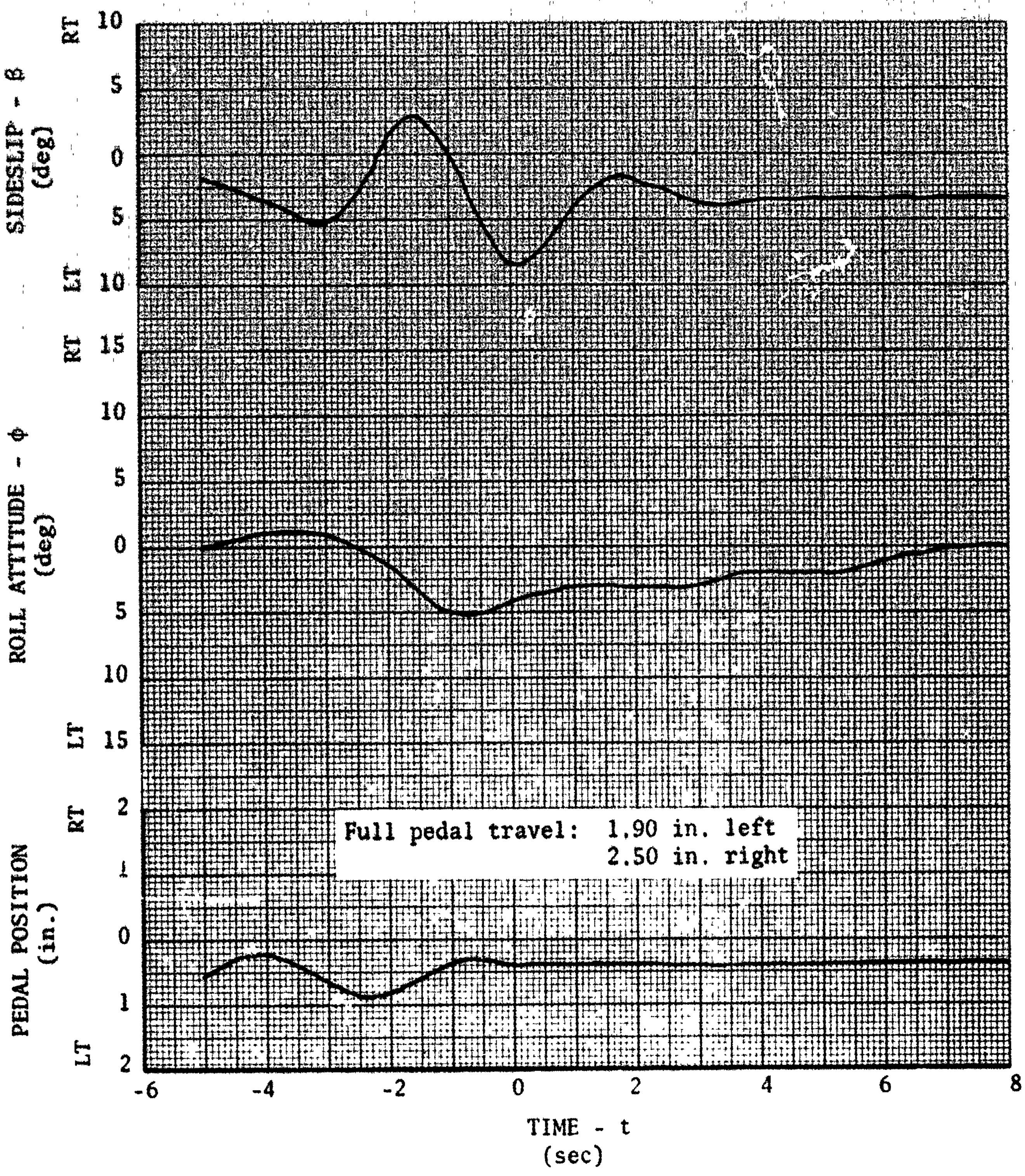


FIGURE 29

DUTCH ROLL OSCILLATION
STICK FREE PEDAL FIXED
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
P-CLIMB	5500	3500	86.08	81.5

ω_n = 1.75 rad/sec ζ = 0.77 $C_{1/2}$ = 0.0914 cycles ϕ/β = 1:3.0

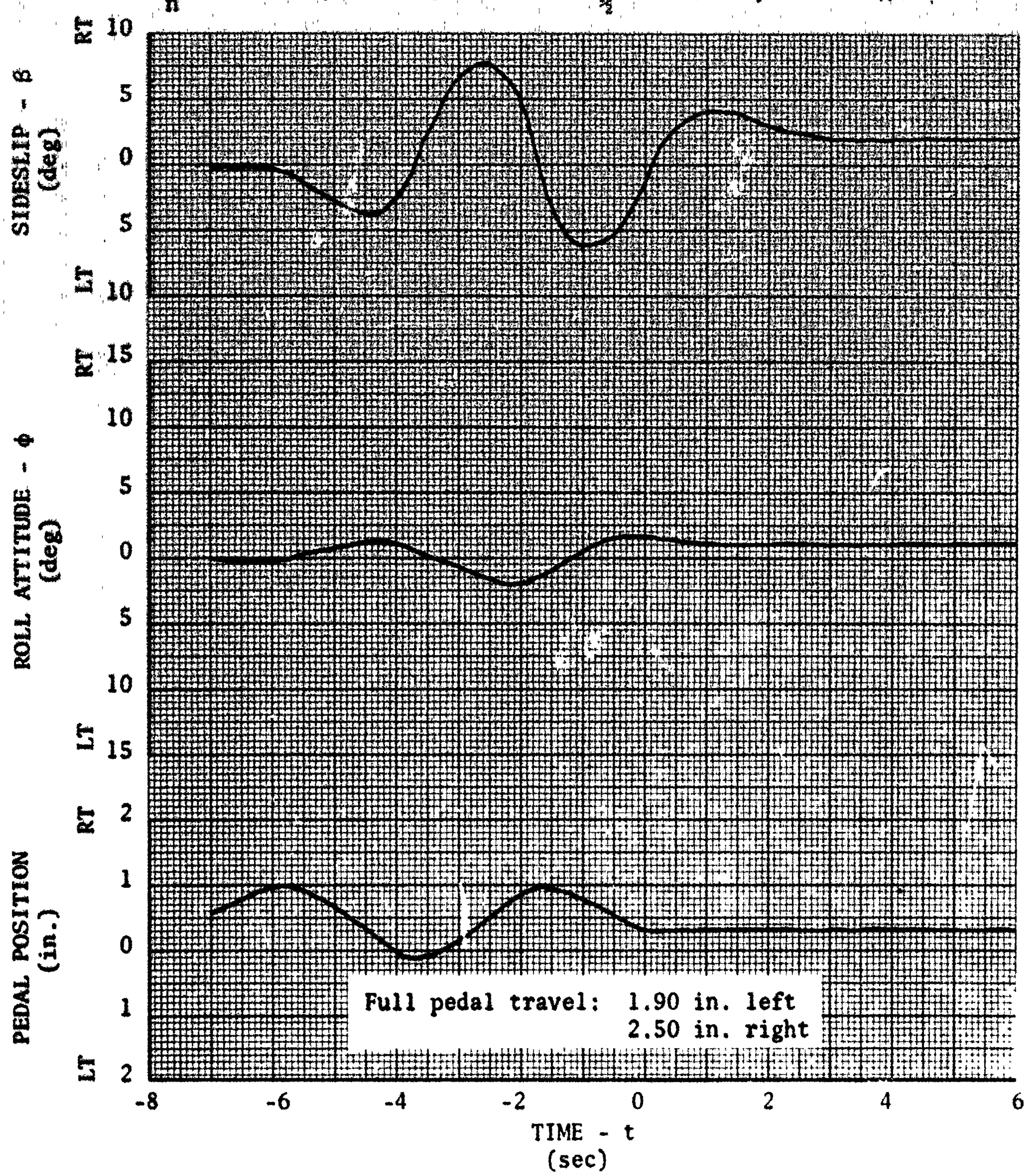


FIGURE 30

ADVERSE SIDESLIP
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
PA	6200	3500	86.08	68.0

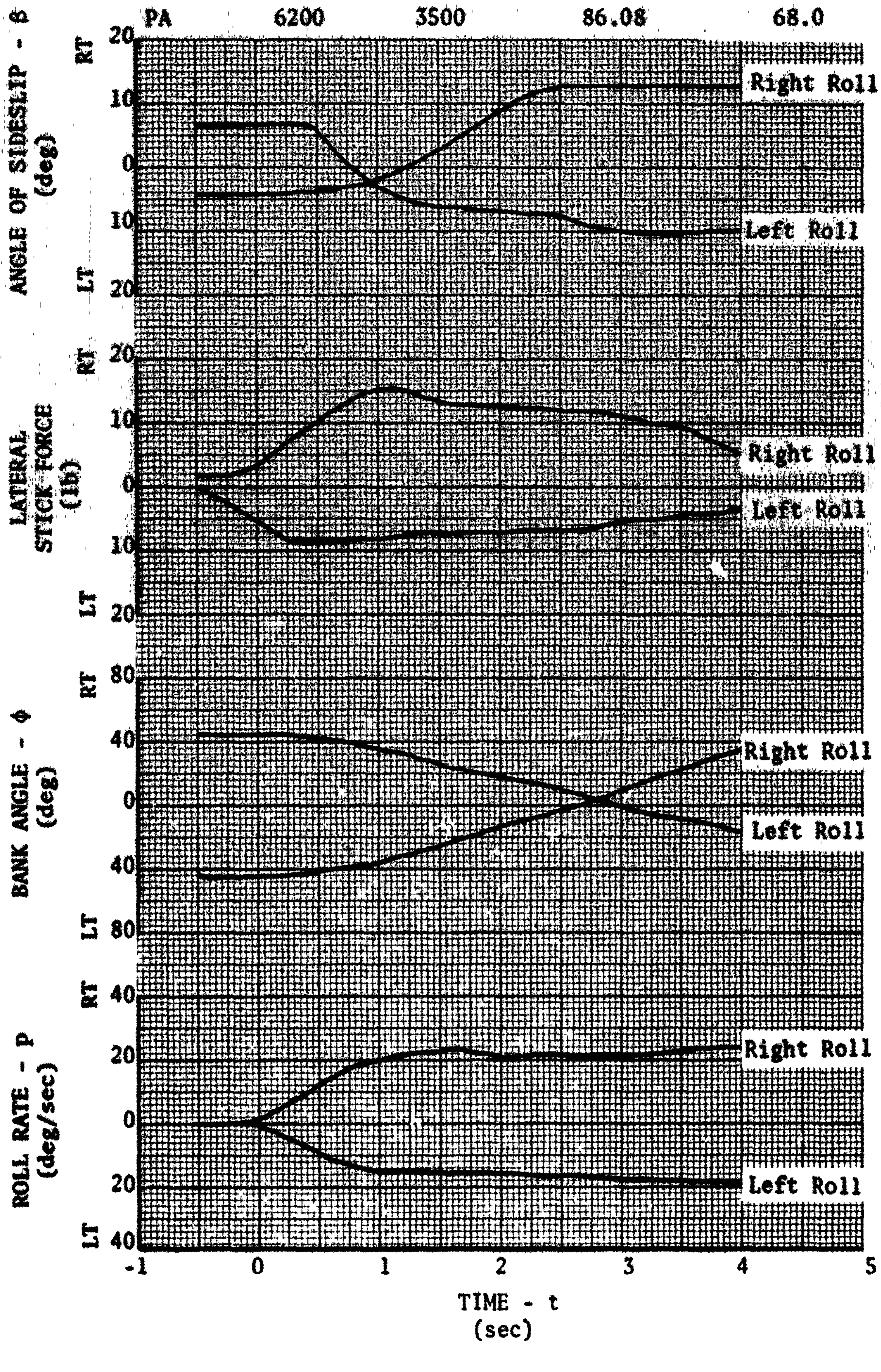


FIGURE 31

ADVERSE SIDESLIP
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	6200	3500	86.08	66.6

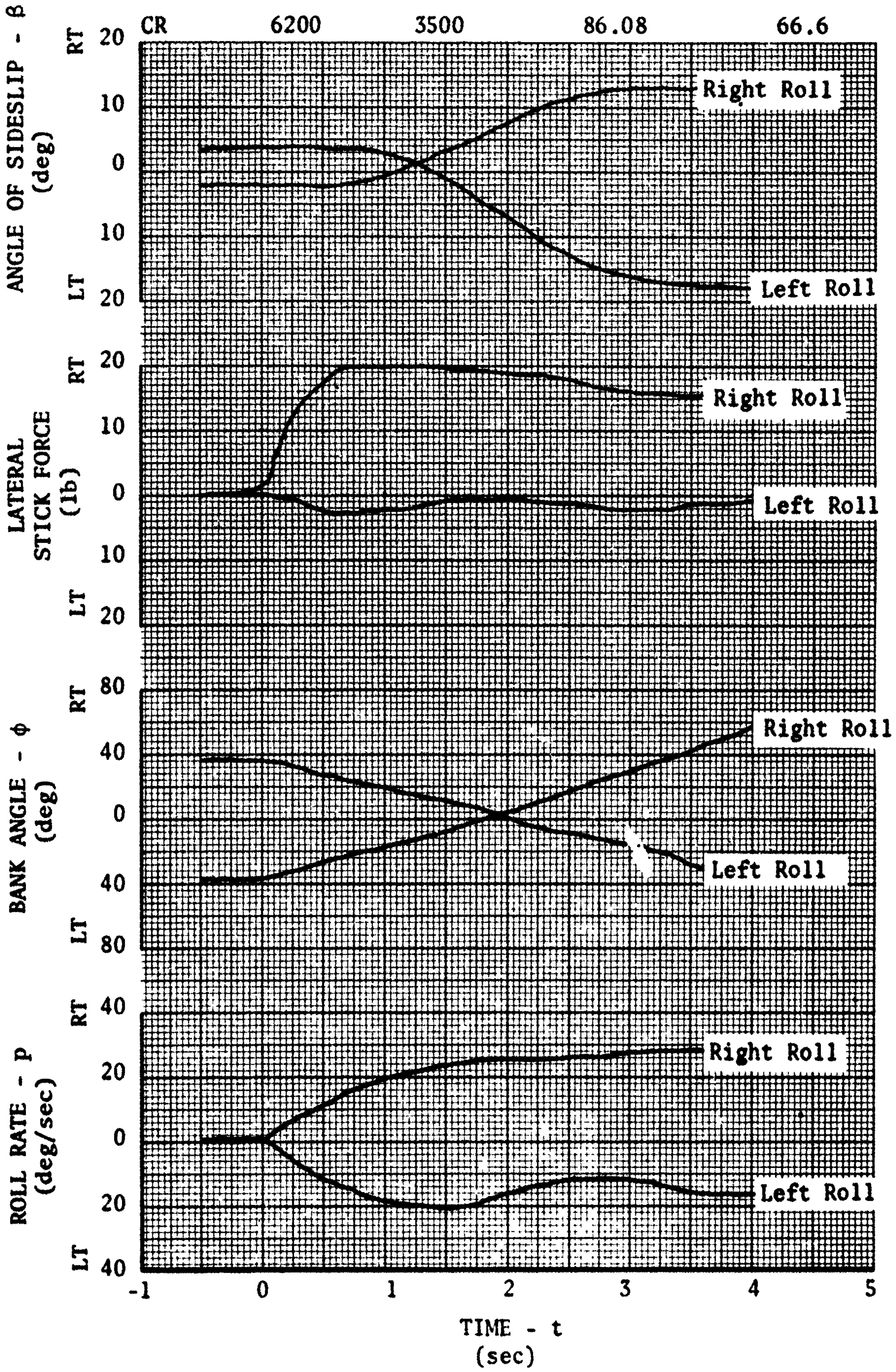


FIGURE 32

ADVERSE SIDESLIP
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	6200	3500	86.08	73.0

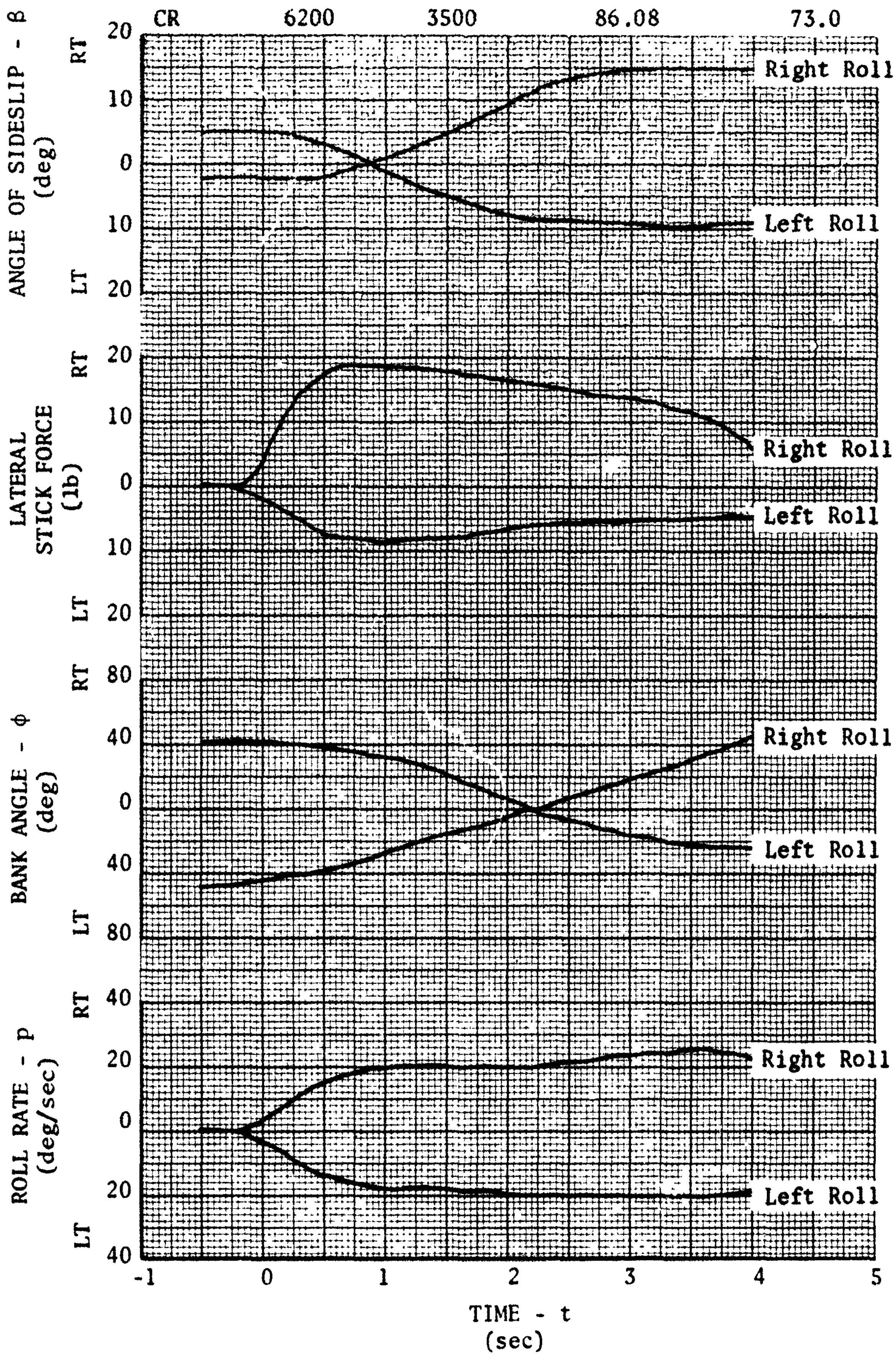


FIGURE 33

ROLL CONTROL EFFECTIVENESS
PEDAL FIXED
YO-3A S/N 69-18000
MISSION EQUIPMENT INSTALLED

SYM	CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	FLIGHT CONDITION
○	CR	6200	3500	86.08	Left Roll
□	CR	6200	3500	86.08	Right Roll
●	PA	6200	3500	86.08	Left Roll
■	PA	6200	3500	86.08	Right Roll

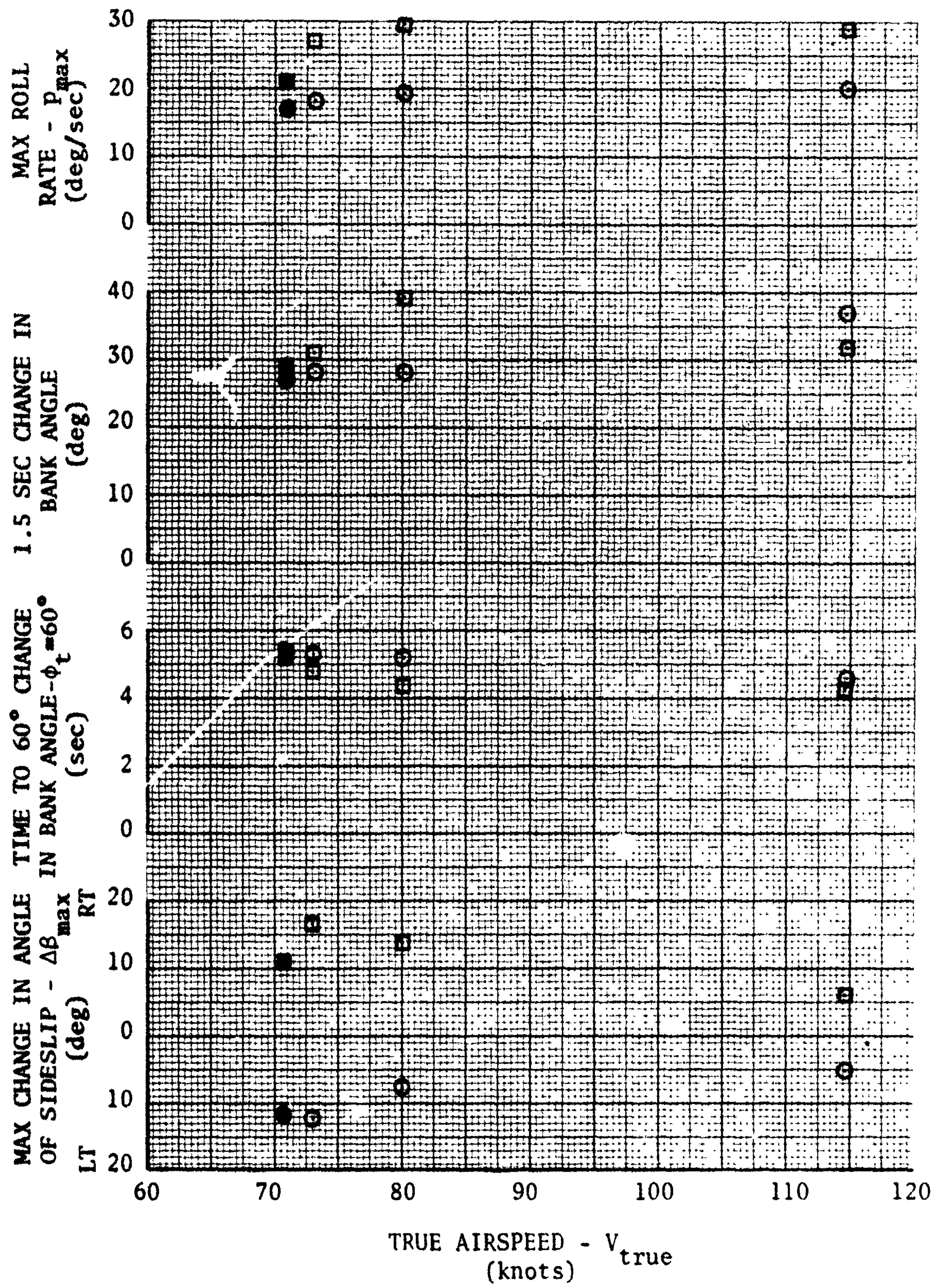


FIGURE 34

LONGITUDINAL CONTROL FORCE CHARACTERISTICS
YO-3A S/N 69-18000
MISSION EQUIPMENT NOT INSTALLED

CONFIG	DENSITY ALTITUDE (ft)	GROSS WEIGHT (lb)	CG LOCATION (FS) (in.)	TRIM AIRSPEED (KCAS)
CR	5000	3500	85.35	70

Notes: 1. Shaded symbol denotes trim.
2. Breakout plus friction = 2 lb.
3. Friction band = 2½ lb.

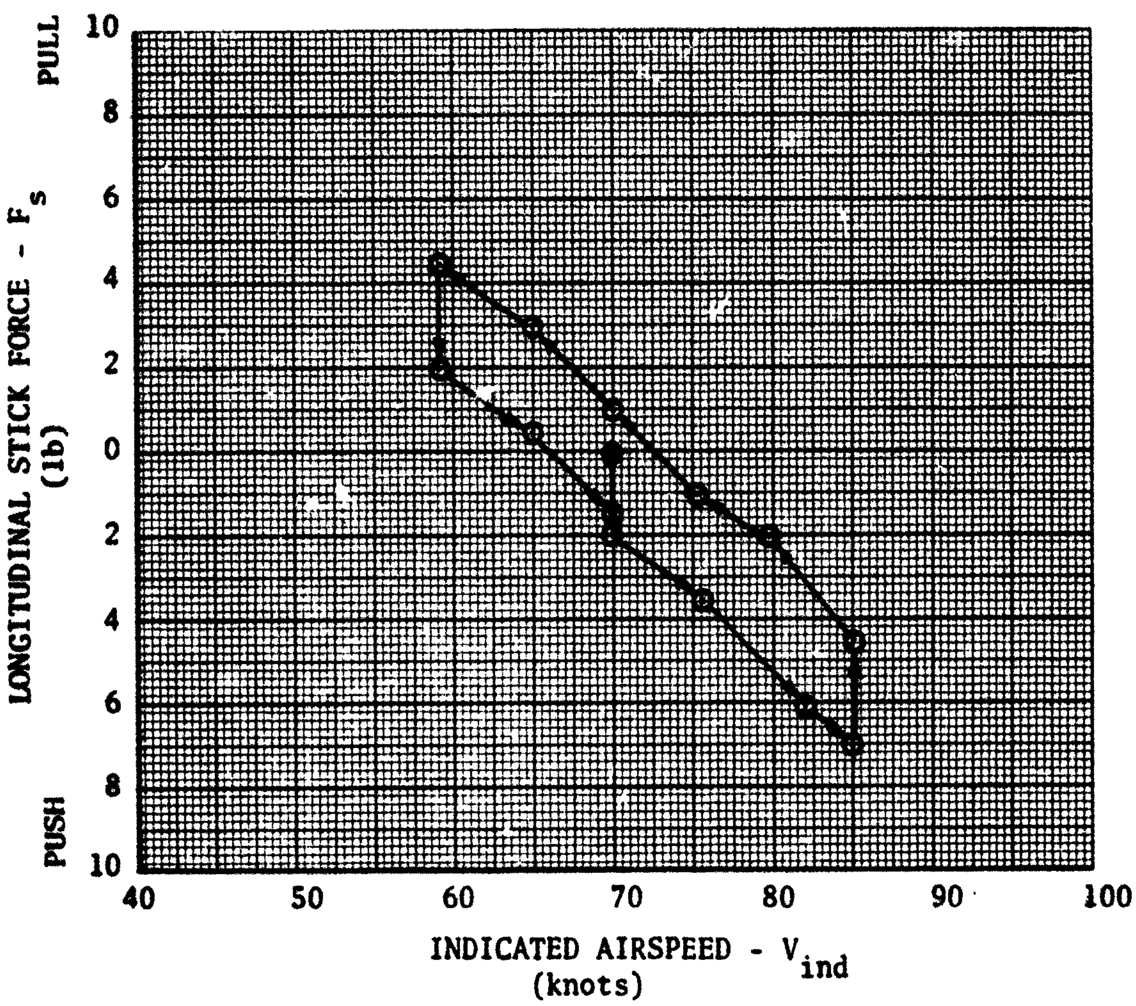


FIGURE 35

POWER LOSS CORRECTION CURVE
DERIVED FROM LMSC DATA

Note: Based on a 2 HP altenator loss.

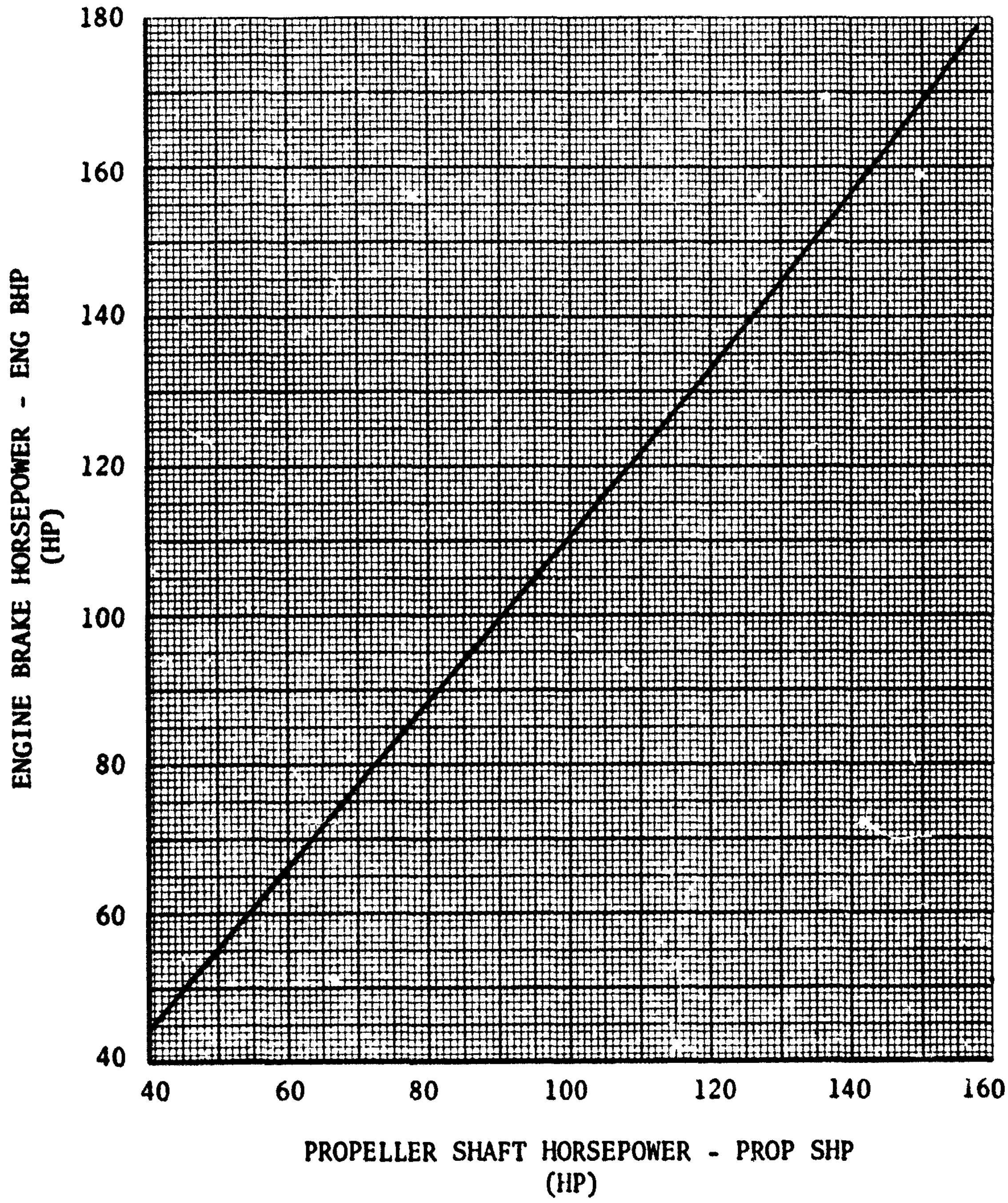


FIGURE 36

PROPELLER EFFICIENCY
PROVIDED BY LMSC

Note: Blade pitch at 0.75 radius equals 37°.

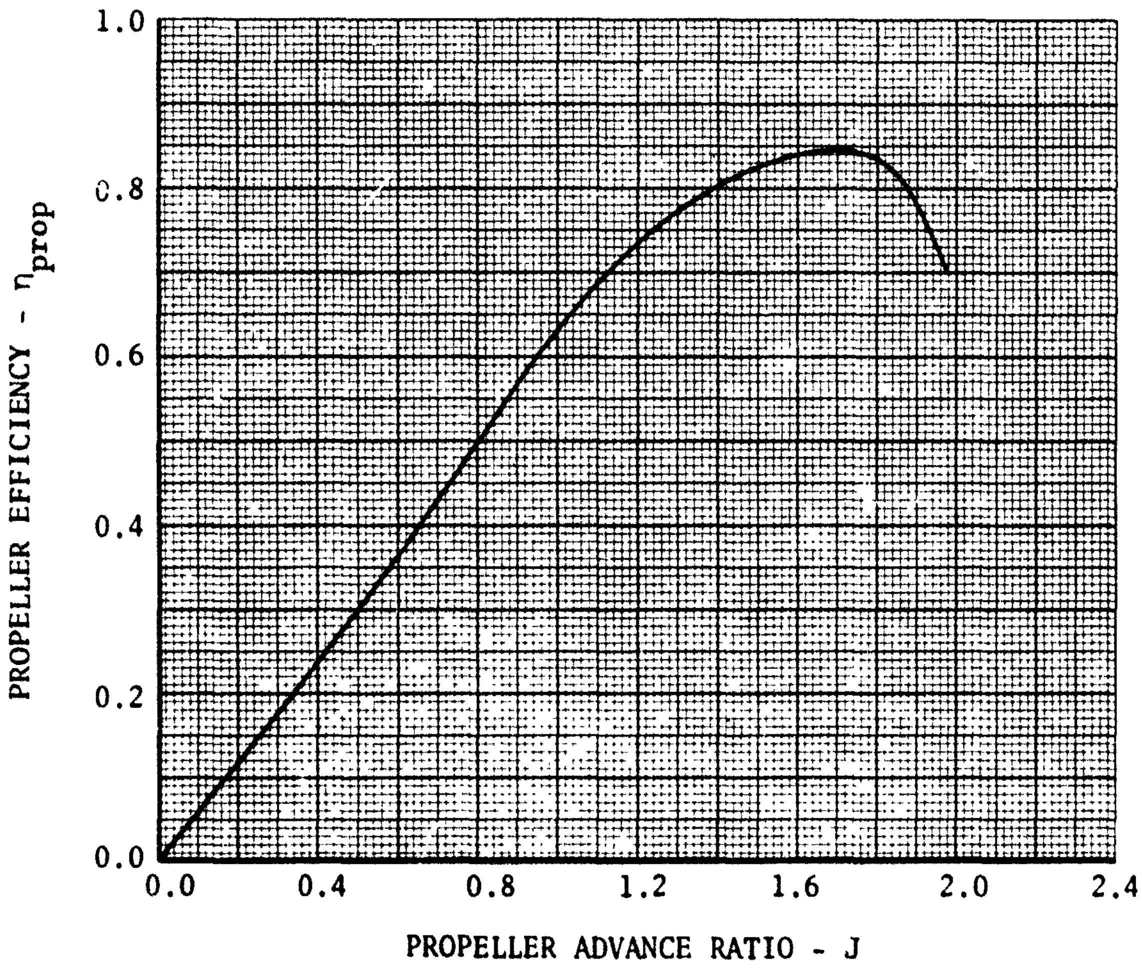


Table A. YO-3A, S/N 69-18000, STALL CHARACTERISTICS.

Configuration	Normal Acceleration (g's)	Gross Weight (lb)	Pressure Altitude (ft)	CG Location (FS)(in.)	Altitude Lost in Recovery (ft)	Trim Airspeed (KIAS)	Buffet Airspeed (KIAS)	Stall Airspeed (KIAS)	Remarks
Power Approach	1.00	3500	5220	86.08	195	69.5	55.0	53.9	
	1.00	3486	5100	86.08	130	66.0	55.5	53.8	
	1.23	3465	4530	86.08	340	63.0	56.2	54.1	Wings level pullup
	1.16	3453	4290	86.08	155	65.0	56.2	54.0	30-degree left bank
	1.13	3452	3740	86.08	No record	66.5	59.4	57.6	30-degree right bank
Glide	1.00	3490	4440	85.37	230	60.5	58.6	56.7	
	1.00	3491	5050	85.37	390	60.5	59.6	56.7	
Takeoff	1.00	3473	4740	86.08	110	67.5	54.8	53.2	
	1.11	3441	4760	86.08	90	65.0	54.8	53.8	
	1.00	3481	4460	86.08	110	67.5	55.7	54.3	
Dive	1.00	3489	5300	86.05	280	65.5	60.5	58.0	
	1.00	3486	4760	86.05	290	64.0	60.5	59.0	
	1.35	3462	5550	86.05	340	71.5	70.0	69.0	Wings level pullup
Cruise	1.00	3495	5520	85.33	240	79.0	54.0	53.5	
	1.00	3492	5680	85.33	130	59.0	55.0	54.0	
	1.00	3510	4840	86.05	100	60.5	55.5	54.0	
	1.35	3479	5740	86.05	220	68.5	59.5	56.5	30-degree left bank
	1.30	3470	5370	86.05	200	68.5	69.0	68.0	30-degree left bank
	1.20	3473	5460	86.05	100	68.0	70.0	68.0	30-degree right bank

APPENDIX III. TEST INSTRUMENTATION

Parameter	Cockpit	Photopanel	Oscillograph
Airspeed (boom system)	X	X	
Airspeed (standard system)		X	
Altitude (boom system)	X	X	
Altitude (standard system)	X		
Outside air temperature (remote)		X	
Exhaust gas temperature	X		
Cylinder head temperatures	X		
Oil temperature		X	
Oil pressure		X	
Fuel pressure		X	
Fuel flow rate		X	
Fuel used	X	X	
Engine lower plenum pressure		X	
Manifold pressure		X	
RPM		X	
Time		X	
Photopanel frame light	X		
Photopanel frame counter	X	X	X
Oscillograph burst counter	X	X	X
Oscillograph ON light	X	X	
Oscillograph (no record light)		X	
Instrumentation power on	X		
Pilot event switch	X		
Event marker		X	X
Correlation counter	X		
Sideslip angle - yaw angle	X		X
Angle of attack	X		X
Aileron position	X		
Rudder position	X		
Stabilator position	X		
Stabilator trim position		X	
Yaw rate			X
Pitch rate			X
Roll rate			X
Longitudinal stick force			X
Lateral stick force			X
Rudder pedal force			X
Longitudinal stick position			X
Lateral stick position			X
Rudder pedal position			X
Pitch attitude			X

Parameter	Cockpit	Photopanel	Oscillograph
Roll attitude			X
Differential pressures ΔP for fuel tank		X	
CG lateral acceleration		X	X
CG Normal acceleration	X		X
Pressure differential - engine plenum top to bottom		X	
Attitude gyro ungage light	X		

APPENDIX IV. DETAILED FLIGHT CONTROL SYSTEM DESCRIPTION

General

1. The YO-3A airplane reversible control system is composed of push/pull tubes, steel cables, pulleys, brackets, fabric covered aileron and rudder control surfaces, an all-metal stabilator, down spring, fixed aileron and rudder trim tabs, and a spoiler control. The aileron and stabilator control motion is produced by mechanically interconnected control sticks which permit control from either cockpit position. The observer's control stick can be telescoped upward when required. The observer and pilot's rudder pedals are mechanically interconnected and control the rudder steerable tail wheel. Table I summarizes the control surface and flight control motion.

Table I. Summary of Flight Control System Motion.

Rudder	22-24 degrees left	28-30 degrees right
Rudder Pedals		
Left pedals	1.90 inches forward	2.50 inches aft
Right pedals	2.50 inches forward	1.90 inches aft
Stabilator	19.5(+0,-3) degrees up	9.5(+0,-3) degrees down
Stabilator Cont		
Pilot	3.18 inches forward(8°)	6.00 inches aft(16°)
Observer	2.40 inches forward(8°)	4.72 inches aft(16°)
Stabilator Tab	6(±2) degrees up	7(±2) degrees down
Stabilator Tab control	One turn for 3 degrees of tab movement	
Ailerons	29(+0,-3) degrees up	13(+0,-3) degrees down
Ailerons Cont		
Pilot	5.68 inches left(13.5°)	5.08 inches right(12°)
Observer	4.05 inches left(13.5°)	3.60 inches right(12°)
Spoiler		
Top	0 to 73(±5) degrees	
Bottom	0 to 42(±5) degrees	
Spoiler Control	Electrical actuator, 6.45 seconds to full open	

Longitudinal

2. The longitudinal control system is activated by a conventional metal control stick which pivots about its base and is connected to the stabilator control by a three-sixteenth inch steel cable through a system of pulleys and brackets. The steel control cable is preloaded to 40 pounds (tension). The down springs are installed with a tension preload of 62.5 (±0.5) pounds. A stabilato trim tab control wheel is mounted to the right bulk-head in the pilot's compartment. The trim tab wheel is mechanically connected to the stabilator by a one-eighth inch steel cable and screw jack tab actuator. A trim tab position indicator is located forward of and adjacent to the trim tab wheel. A symmetrical airfoil stabilator, hinged along the quarter-chord line, produces a center of pressure about the hinge line. The stabilator trim tab is geared to lead the motion of the stabilator and produces the moment required for elevator-free stability. Trimming is accomplished by reducing the relative angle between the stabilator and trim tab until the moment about the hinge line is nulled. The stabilator operates through an arc of 19.5 degrees up and 9.5 degrees down from neutral. The pilot's control stick displacement is 3.18 inches forward from neutral and 6.0 inches aft from neutral (fig. a). The stabilator trim tab travels through an arc of 6 (±2) degrees up and 7 (±2) degrees down from trim. One complete cycle of the stabilator trim tab control wheel is equivalent to 3 degrees of tab motion (fig. b).

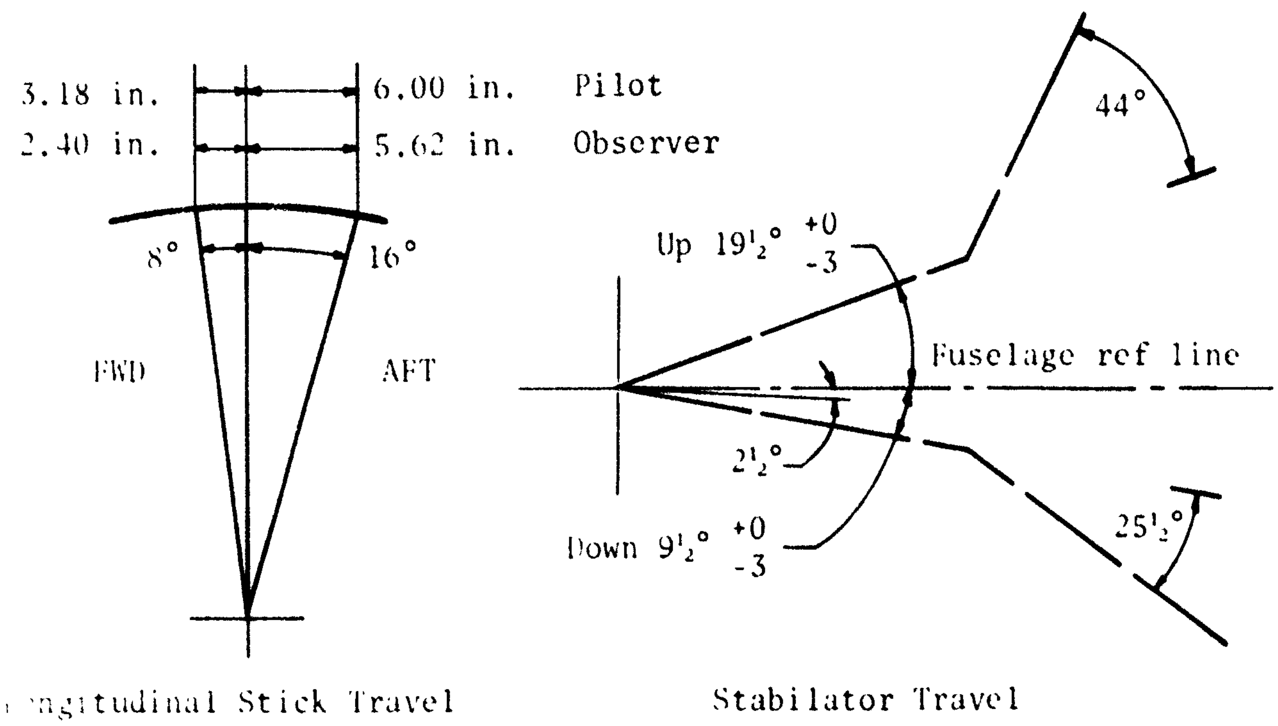
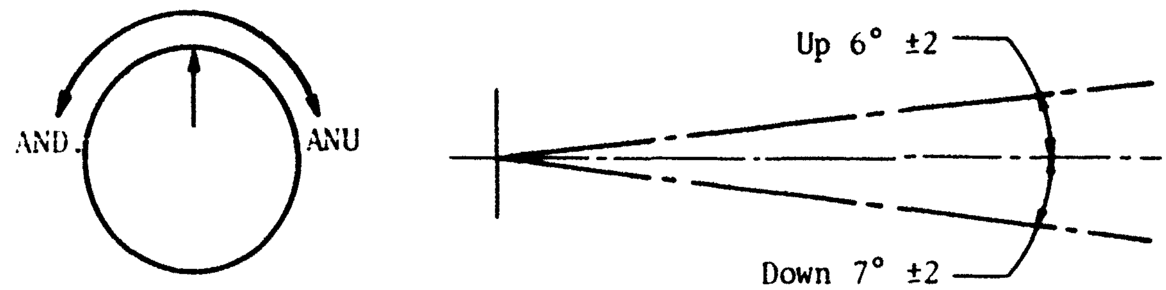


Figure a. Longitudinal Flight Control System.



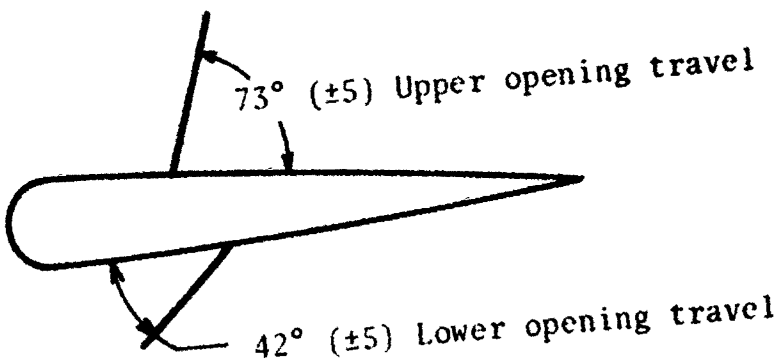
One complete revolution for 3° of tab travel.

Trim Tab Wheel Travel.

Trim Tab Surface Travel.

Figure b. Stabilator Trim Tab Travel.

3. The spoiler controls are rigid surfaces attached to the upper and lower surfaces of each wing just aft of the main wing span. Both spoiler surfaces, above and below each wing, operate simultaneously through bell cranks attached to a single torque tube. The spoilers are operated by an electric motor which is activated by a switch located on top of the pilot's control stick, or by the observer's switch, attached to the throttle control quadrant. In the event of an activator failure with the spoilers in the open position, an emergency release cable can be operated manually by either pilot or observer. The emergency release disengages the power controls from the torque tube and the spoilers close by air pressure on the larger upper doors, together with a force applied by an emergency return spring on the lower doors. Spoiler opening rates and travel motion are shown in figure c.



Spoiler rates: Upper 11.3°/sec
Lower 6.5°/sec

Spoilers operate from a common torque tube which rotates through 110 degrees in 6.45 seconds to open fully.

Figure c. Spoiler Travel and Rates.

Lateral

4. Rolling motion about the longitudinal axis is derived from up and down movements of the ailerons resulting from left and right motions of the control stick transmitted through push/pull tubes. The ailerons are fabric-covered and an aerodynamic balancing fabric seal is located between the wing and aileron. Aileron surface travel and stick travel dimensions are presented in figure d.

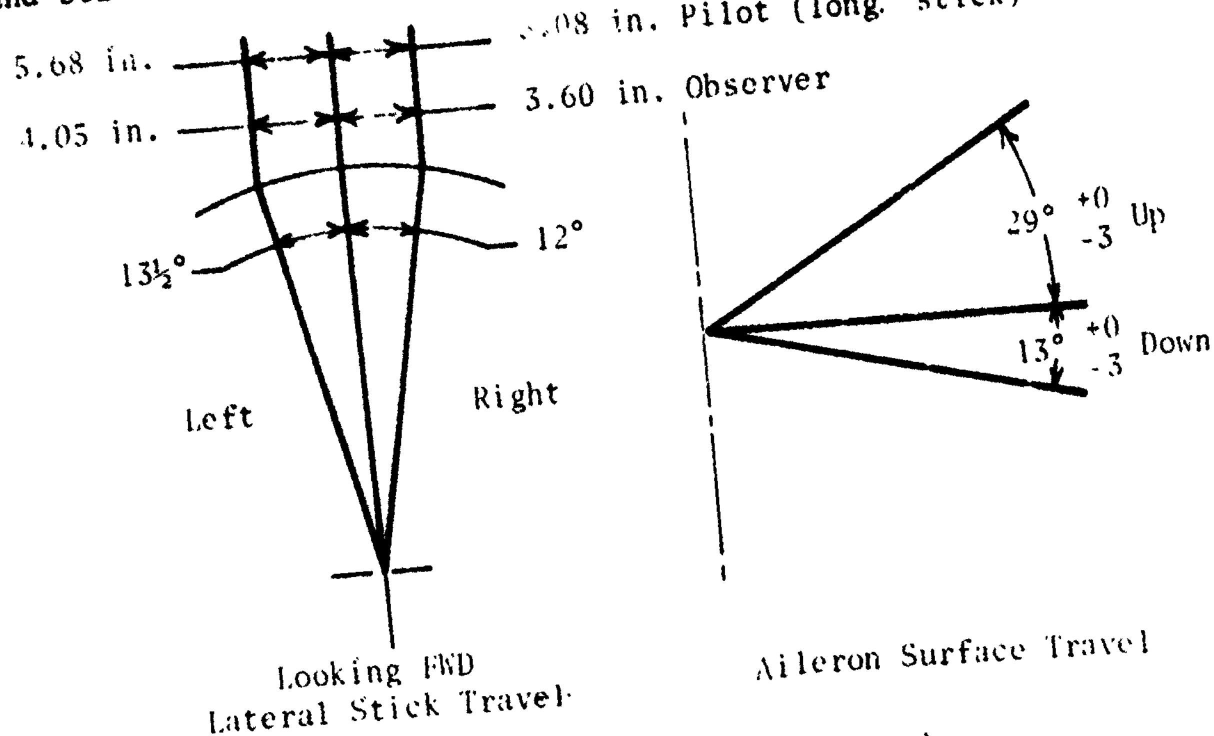


Figure d. Aileron Stick Motion and Surface travel.

Directional

5. Motion about the vertical axis is derived from left and right movements of the fabric-covered rudder surface. Two sets of interconnected rudder pedals are provided to mechanically operate the rudder and steerable tail wheel. The rudder pedals are adjustable to 1 inch on both sides of nominal. Rudder pedal and rudder surface travel characteristics are presented in figure e

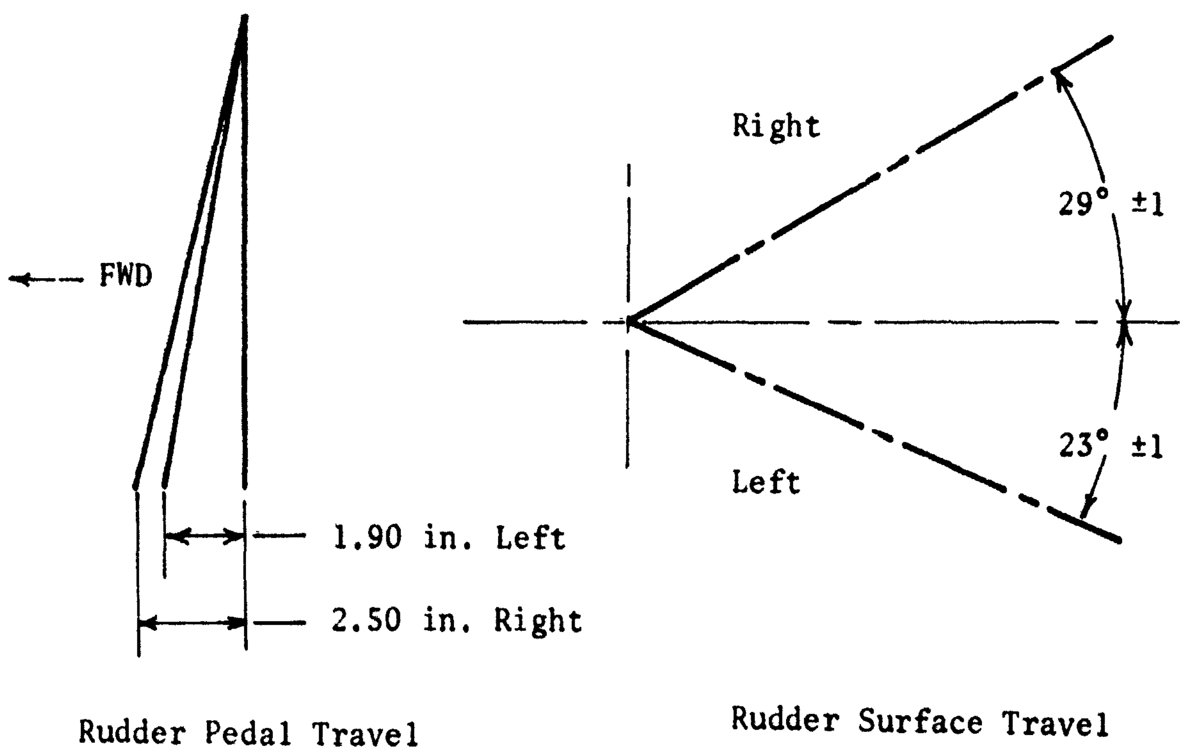


Figure e. Directional Control Motion Characteristics.

APPENDIX V. TEST CONDITIONS

Table a. Performance Test Conditions.

Test Description	Density Altitude (ft)	Engine Start Gross Weight (lb)	CG Start Location (FS) (in.)	Configuration
Sawtooth climbs	[1]1,000 [1]3,000 [1]5,000	3523 3523 3523	85.37 85.37 85.37	Power Power Power
Level flight	1,000 1,000 1,000 1,000 5,000 5,000 10,000 10,000	3523 3523 3523 3491 3523 3491 3523 3491	85.37 85.37 85.37 85.34 85.37 85.34 85.37 85.34	Power Approach Cruise (cowl open) Cruise (cowl closed) Cruise[2] Cruise Cruise[2] Cruise Cruise[2]
Stall performance	5,000 5,000 5,000 5,000	3523 3537 3523 3523	85.37 86.08 85.37 85.37	Cruise Power Approach Glide Dive
Takeoffs and landings	200	3491	85.34	Cruise[2]
V_{max} at sea level	300	3491	85.34	Cruise[2]

[1]Pressure altitude.
[2]Mission equipment not included.

Table b. Stability and Control Test Conditions.

Test Description	Density Altitude (ft)	Gross Weight (lb)	CG Location (FS) (in.)	Trim Airspeed (KCAS)	Configuration
Longitudinal static stability	5000	3500	84.66	66	Cruise[1]
	5000	3500	85.37	67	Cruise
	5000	3500	84.66	73	Cruise[1]
	5000	3500	85.37	73	Cruise
	5000	3500	86.08	93	Cruise
	5000	3500	84.66	105	Cruise[1]
	5000	3500	86.08	69	Power Approach
	5000	3500	86.08	82	Power
Longitudinal dynamic stability	6500	3500	86.08	93	Cruise
	6500	3500	86.08	68	Waveoff
	6500	3500	86.08	68	Power Approach
	6500	3500	86.08	82	Power
Lateral-directional static stability	5000	3500	85.37	73	Cruise
	6200	3500	86.08	73	Cruise
	8300	3500	86.08	73	Cruise
	[2]3500	3500	86.08	67	Cruise
	5000	3500	86.08	68	Waveoff
	5000	3500	86.08	68	Power Approach
	5000	3500	86.08	82	Power
Lateral-directional dynamic stability	5000	3500	85.37	73	Cruise
	7400	3500	86.08	V_{max}	Cruise
	6900	3500	86.08	V_{max}	Cruise
	8300	3500	86.08	73	Cruise
	6200	3500	86.08	67	Cruise
	[2]4000	3500	86.08	68	Power Approach
	[2]4000	3500	86.08	82	Power
	5600	3500	86.08	82	Power
Roll control effectiveness	6200	3500	86.08		Cruise
Spiral stability	6900	3500	86.08	V_{max}	Cruise
	6200	3500	86.08	67	Cruise

Table b. Continued.

Test Description	Density Altitude (ft)	Gross Weight (lb)	CG Location (FS) (in.)	Trim Airspeed (KCAS)	Configuration
Adverse yaw	6200	3500	86.08	73	Cruise
	6200	3500	86.08	67	Cruise
Stall characteristics	[2]5000	3500	86.05	60	Cruise
	[2]5000	3500	85.33	59	Cruise[1]
	[2]5000	3500	86.08	65	Power Approach
	[2]5000	3500	86.05	64	Dive
	[2]5000	3500	85.33	63	Dive[1]
	[2]5000	3500	85.37	60	Glide
	[2]5000	3500	86.05	59	Glide
	[2]5000	3500	86.08	67	Takeoff
	[2]5000	3500	86.08	65	Land
Maneuvering stability (wings level pull-ups	5000	3500	85.37	75	Cruise
	5000	3500	85.35	75	Cruise[1]
	5000	3500	85.37	94	Power
	5000	3500	85.35	75	Power[1] (spoilers extended)
	5000	3500	85.35	93	Power[1]
	5000	3500	85.37	126	Dive
Maneuvering stability in turns	5000	3500	85.35	75	Cruise[1]
	5000	3500	85.37	91	Power (spoilers extended)
	5000	3500	85.35	91	Power[1]
	5000	3500	85.35	65	Power Approach[1]
Trimmability	Derived from previous testing.				

[1]Mission equipment not included.
[2]Pressure altitude.

Table c. Airplane Test Configurations[1]

Configuration[2]	Symbol	Power	Gear Position	Spoiler Position
Takeoff	TO	Takeoff	Down	Closed
Power	P	Maximum continuous	Up	Closed
Cruise	CR	Power for level flight (PFLF)	Up	Closed
Dive	D	25% Normal rated power (NRP)	Up	Extended
Glide	G	Idle power	Up	Closed
Power approach	PA	PFLF at $1.15V_{S_L}$[3] or normal approach speed whichever is lower	Down	Closed
Waveoff	WO	Takeoff	Down	Closed
Landing	L	Idle power	Down	Closed

[1]Items of configuration not specified, such as canopy position, cowl flaps, oil cooler or payload bay doors shall be in their normal settings for the particular configuration.
[2]Configurations are defined in LMSC Document 681689, 9 Sep 68, *Design Requirements for YO-3A Airplane, Flying Qualities* (ref 13, app I), hereafter referred to as the design specification.
[3]V_{S_L} equals stall speed in the landing configuration.

APPENDIX VI. PHOTOGRAPHS

Photo 1. Control restricted by pilot's left leg.

Photo 2. Throttle quadrant contacted by pilot's left leg.

photo 3. Longitudinal trim tab control wheel mounting (right side of the airplane).

APPENDIX VII. SYMBOLS AND ABBREVIATIONS

1. The following symbols and abbreviations are used both in the data analysis section (app VIII) and in the Results and Discussion section of this report.

LETTER SYMBOLS FOR PRIMARY CONCEPTS

BHP	Brake horsepower	HP
b	Wing span	ft
C	Coefficient; cycles	
D	Drag	lb
End	Endurance	hr
g	Acceleration of gravity at SL	ft/sec^2
H	Altitude	ft
HP	Horsepower	HP
J	Propeller advance ratio	
L	Lift	lb
N	Rotational frequency	RPM
n	Load factor	
P	Power	ft-lb/sec
p	Pressure, Roll rate	lb/in.^{2}, rad/sec or deg/sec
Q	Torque	ft-lb
Rg	Range	Nautical air miles
R/C	Rate of Climb	ft/min
R/D	Rate of Decent	ft/min

S	Surface Area	ft^2
SHP	Shaft horsepower	HP
s	Horizontal distance	ft
T	Temperature, Period	^oC, sec
THP	Thrust horsepower	HP
V	Airspeed	knots
W	Weight flow rate	lb/hr
WT	Weight	lb
α	Angle of attack	deg or rad
β	Angle of sideslip	deg or rad
ζ	Damping ratio	
η	Efficiency	%
θ	Attitude	deg or rad
Θ	Temperature ratio	
ρ	Mass density	$lb\text{-}sec^2/ft^4$
σ	Density ratio	
τ	Time constant	sec
ϕ	Angle of bank or roll	deg or rad
ω	Angular frequency	rad/sec

LETTER SYMBOLS FOR SECONDARY CONCEPTS

For Use as Subscripts

a	air
abs	absolute

avail	available
cal	calibrated
climb	climb
corr	corrected
CR	cruise configuration
D	drag; dive configuration
DR	dutch roll
d	density
eng	engine
FLF	for level flight
f	fuel
gnd	ground
G	glide configuration
ic	instrument corrected
ind	indicated
iw	generalized
L	lift; landing configuration
LO	lift-off
max	maximum
n	undamped
p	power; power configuration
P	pressure

pc	position corrected
prop	propeller
r	roll
rnwy	runway
rqrd	required
SL	sea level
s	stick
sp	specific
stall	stall
std	standard
TO	takeoff; takeoff configuration
test	test
tot	total
true	true
w	wing
w/	with
w/o	without
WO	wave-off configuration
wind	wind

For Use as Prefixes

GR	gross
PROP	prop

For Use as Operators

-	Average	Superscript over symbol
.	Differentation with respect to time	Superscript over symbol
Δ	Change	Prefix

ABBREVIATIONS (U)

AND.	Airplane nose down
ANU	Airplane nose up
APE	Army preliminary evaluation
CG	Center of gravity
CONFIG	Configuration
CR	Cruise
D	Dive
DEG	Degrees
DEV	Deviation
DN	Down
FPM	Feet per minute
FS	Fuselage station
FT	Feet
FWD	Forward
G	Glide
HP	Horsepower
HQRS	Handling Qualities Rating Scale
IN.	Inches

KCAS	Knots calibrated airspeed
KIAS	Knots indicated airspeed
KT	Knots
KTAS	Knots true airspeed
L	Landing
LB	Pounds
LMSC	Lockheed Missiles and Space Company
LT	Left
MAC	Mean aerodynamic chord
MIN	Minutes
MRP	Military rated power
NM	Nautical miles
NAMPP	Nautical air miles per pound of fuel
NRP	Normal rated power
P	Power
PA	Power approach
PFLF	Power for level flight
PSI	Pounds per square inch
RAD	Radians
RPM	Revolutions per minute
RT	Right
SEC	Seconds
TO	Takeoff

USAASTA	US Army Aviation Systems Test Activity
USAAVSCOM	US Army Aviation Systems Command
USAMC	US Army Materiel Command
WO	Wave off

APPENDIX VIII. DATA ANALYSIS METHODS

PITOT-STATIC SYSTEM CALIBRATION

1. To calibrate the boom and standard pitot-static systems of the YO-3A, a calibrated trailing bomb towed by a pace UH-1C helicopter was used to determine the airspeed and altitude position errors. The instrument corrected airspeeds (V_{ic}) and instrument corrected altitudes (H_{ic}) indicated on the boom and standard systems were subtracted from the instrument corrected airspeed and altitude indicated on the trailing bomb (which has zero position errors) to yield the airspeed position error (ΔV_{pc}) and the altimeter position error (ΔH_{pc}) of the boom and standard pitot-static systems.

AIRSPEED DETERMINATION

2. Airspeed test results were determined from indicated airspeeds (V_{ind}) of the calibrated test boom system of the YO-3A and were instrument corrected. To determine calibrated airspeed (V_{cal}) the test boom system airspeed position error (ΔV_{pc}), corresponding to the instrument corrected airspeed (V_{ic}), was added to the instrument corrected airspeed.

$$V_{cal} = V_{ic} + \Delta V_{pc} \tag{1}$$

3. True airspeed (V_{true}) was determined by dividing calibrated airspeed (V_{cal}) by the square root of the test day density ratio (σ_{test}).

$$V_{true} = \frac{V_{cal}}{\sqrt{\sigma_{test}}} \tag{2}$$

ATMOSPHERIC PARAMETERS DETERMINATION

4. Altitude test results were determined from indicated pressure altitudes ($H_{P_{ind}}$) of the calibrated test boom system of the

YO-3A and were instrument corrected. To determine calibrated pressure altitude ($H_{P_{cal}}$) the test boom system altitude position error (ΔH_{pc}), corresponding to the instrument corrected airspeed (V_{ic}), was added to the instrument corrected pressure altitude ($H_{P_{ic}}$).

$$H_{P_{cal}} = H_{P_{ic}} + \Delta H_{pc} \tag{3}$$

5. Density altitude was determined from calibrated pressure altitude ($H_{P_{cal}}$) and instrument corrected outside air temperature ($T_{ic_{test}}$) by use of equations used to establish the *US Standard Atmosphere*, 1962. The test values of the density (ρ_{test}) and the standard values of the density (ρ_{std}) and density ratio (σ_{std}) were also determined by using the appropriate density ratio (σ_{test}) equations from *US Standard Atmosphere*, 1962.

GROSS WEIGHT DETERMINATION

6. The airplane in-flight gross weight (GRWT) was computed from fuel counter readings, preflight gross weight and the average fuel density from preflight and postflight readings.

CENTER OF GRAVITY DETERMINATION

7. As shown in the Weight and Balance section of the Results and Discussion (para 12), a shift of 0.28 inch in the cg location existed between the empty fuel airplane and the full fuel airplane. Due to the small magnitude of this shift and due to fuel being drawn from one tank at a time while in flight, the cg location at takeoff was used as the in-flight cg location.

POWER REQUIRED DETERMINATION

8. Since the YO-3A test airplane, S/N 69-18000, did not have a calibrated engine, it was necessary to measure horsepower directly. This was done from propeller shaft horsepower (PROP SHP) as determined from the equation:

$$\text{PROP SHP} = \frac{2\Pi\, Q_{prop}\, N_{prop}}{33{,}000} \sim \text{HP} \tag{4}$$

9. Propeller shaft torque (Q_{prop}) was measured by a strain gage torque meter and recorded on the oscillograph. Propeller shaft speed (N_{prop}) was computed by counting the number of revolutions of the shaft over a time period, as recorded on the oscillograph.

10. Thrust horsepower (THP) was computed from propeller shaft horsepower and propeller efficiency (η_{prop}) according to the equation:

$$\text{THP} = \eta_{prop} \times \text{PROP SHP} \sim \text{HP} \tag{5}$$

Figure 36, appendix II shows the LMSC supplied curve of propeller efficiency as a function of propeller advance ratio (J) where:

$$J = \frac{12.15\, V_{true}}{N_{prop}} \tag{6}$$

11. Engine brake horsepower required (ENG BHP_{rqrd}) was calculated by adding the losses due to the drive system, exhaust system, induction system and alternator to the propeller shaft horsepower.

$$\text{ENG BHP}_{rqrd} = \text{PROP SHP} + \Delta HP_{drive} + \Delta HP_{exhaust} + \Delta HP_{induction} + \Delta HP_{alternator} \tag{7}$$

These losses were furnished by LSMC and agreed upon by USAAVSCOM. The loss due to the alternator during level flight testing was a constant 2 HP. Taking this into account, the losses were combined to give one curve of ENG BHP_{rqrd} versus PROP SHP as shown in figure 35, appendix II.

12. It should be noted that substantial discrepancies exist between the engine brake horsepower required, calculated in the above manner and the engine brake horsepower required, calculated from the engine specification chart. No attempt has been made in this report to reconcile the two values.

LEVEL FLIGHT PERFORMANCE

13. Level flight performance tests were conducted at various altitudes to determine, primarily, engine brake horsepower required and fuel flow data. The power required data was generalized, to one curve P_{iw} -V_{iw}, for altitude and standard gross weight (3500 pounds) according to the following equations:

$$P_{iw} = \text{ENG BHP}_{rqrd} \sqrt{\sigma} \left[\frac{\text{GRWT}_{std}}{\text{GRWT}_{test}}\right]^{3/2} \sim \text{HP} \qquad (8)$$

$$V_{iw} = V_{true} \sqrt{\sigma} \left[\frac{\text{GRWT}_{std}}{\text{GRWT}_{test}}\right]^{1/2} \sim \text{knots} \qquad (9)$$

14. The fairing of data, defining the power-airspeed relationship, was made on the P_{iw}-V_{iw} plot. The fairings for BHP_{rqrd} versus V_{true} were then determined by solving equations 8 and 9 for ENG BHP_{rqrd} and V_{true}, using the values of P_{iw} and V_{iw} derived from the fairing, 3500 pounds as GRWT_{test} and $\sqrt{\sigma}$ corresponding to the desired altitude. Plots of coefficient of lift (C_L) versus coefficient of drag (C_D) were also presented for power required versus airspeed where:

$$C_L = \frac{0.702 \text{ GRWT}}{\rho_{test} S_w \left[V_{true}\right]^2} = \frac{0.702 \text{ GRWT}}{\rho_{SL} S_w \left[V_{cal}\right]^2} \qquad (10)$$

$$C_D = \frac{228.7 \text{ THP}}{\rho_{test} S_w \left[V_{true}\right]^3} = \frac{228.7 \text{ THP}}{\rho_{SL} S_w \left[V_{cal}\right]^3} \qquad (11)$$

These curves are configuration dependent. The fairings were made to best represent the data.

15. The fuel flow requirements were presented as specific endurance (End_{sp}) versus true airspeed (V_{true}) where:

$$End_{sp} = \frac{1}{W_f} \sim hr/lb_f \qquad (12)$$

16. Endurance (End) of the airplane was calculated using specific range and the weight of fuel available ($^{WT}f_{avail}$) (para 20, app VIII) where:

$$End = End_{sp} (WT_{f_{avail}}) \sim hr \qquad (13)$$

17. Specific range (Rg_{sp}) versus true airspeed (V_{true}) was presented where:

$$Rg_{sp} = End_{sp} (V_{true}) \sim NAMPP \qquad (14)$$

18. Range (Rg) of the airplane was calculated from:

$$Rg = Rg_{sp} (WT_{f_{avail}}) \sim \text{nautical air miles} \qquad (15)$$

19. The fairing of data, defining the fuel flow requirements, was made on the specific range curve and used as a basis for fairing the specific endurance curve. The level flight performance tests, used for determining specific range and specific endurance, were performed both with and without mission equipment installed.

20. In the calculation of endurance and range, from equations 13 and 14 respectively, $^{WT}f_{avail}$ (ref 11, app I) was taken as the weight of useable fuel (189 pounds) minus a takeoff-and-climb allowance (10.5 pounds) for six minutes at maximum continuous power at sea level and minus a 10-percent reserve (18.9 pounds) of the initial useable fuel. For the calculation of the endurance, equation 13 was used with the maximum value of specific endurance at a true airspeed greater than 1.15 $^{V}stall_G$ as End_{sp} and 152.4 pounds as $^{WT}f_{avail}$. For the calculation of the range, equation 15 was used with 152.4 pounds as ($^{WT}f_{avail}$). For Rg_{sp}, 0.99 the maximum value of the specific range was used, if this point occurred above 80 KTAS: if it didn't, the value of the specific range at 80 KTAS was used.

21. Sea level maximum level-flight airspeed was not recorded directly since test site conditions prohibited flying at sea level density altitudes. Since power required equals power available at maximum airspeed, knowledge of the power available at sea level would allow use of the P_{iw}-V_{iw} curve to determine the maximum level-flight airspeed at sea level and at standard gross weight (3500 pounds). The YO-3A did not have a calibrated engine. Therefore, sea level maximum brake horsepower available had to be approximated. To accomplish this, a maximum airspeed level flight was flown as close to sea level as possible (300 ft H_d) and the engine brake horsepower required was recorded. To determine the sea level brake horsepower available, the test value, corrected for temperature, was multiplied by a percentage increase in horsepower proportional to that in the engine specification chart for a 300 foot variation in altitude from sea level. This value of engine brake horsepower was used as P_{iw}, the corresponding V_{iw} being the sea level maximum level-flight airspeed at a standard gross weight of 3500 pounds.

22. This method of correction was also used to determine the sea level maximum level flight airspeed for the airplane with mission equipment installed, since engine brake horsepower available is not a function of configuration. Therefore, the same P_{iw} was used to find V_{iw} but from the curve with mission equipment installed.

23. To correct the level flight performance test results to a different gross weight ($GRWT_2$) than the original standard gross weight ($GRWT_{std_1}$) of 3500 pounds, P_{iw}-V_{iw} was again used. At maximum sea level airspeed ($V_{max_{SL}}$) engine brake horsepower available is constant. P_{iw} was determined from equation 8 using the previously determined ENG $BHP_{avail_{SL}}$, $\sqrt{\sigma}$ at sea level, the new gross weight ($GRWT_2$) as the test gross weight ($GRWT_{test}$) and the original standard gross weight ($GRWT_{std_1}$) as the standard gross weight. The V_{iw} corresponding to this P_{iw} determined the sea level level-flight airspeed when equation 9 was solved for V_{true} using $\sqrt{\sigma}$ at sea level and the original gross weight ($GRWT_{std_1}$).

24. To correct fuel flow to a new gross weight, it was necessary to generate a curve of specific endurance ($1/W_f$) versus true airspeed for the new gross weight. This was accomplished by cross plotting from curves of ENG BHP_{rqrd} versus V_{true} and ENG BHP_{rqrd} versus W_f to eliminate ENG BHP_{rqrd} (ref fig. I).

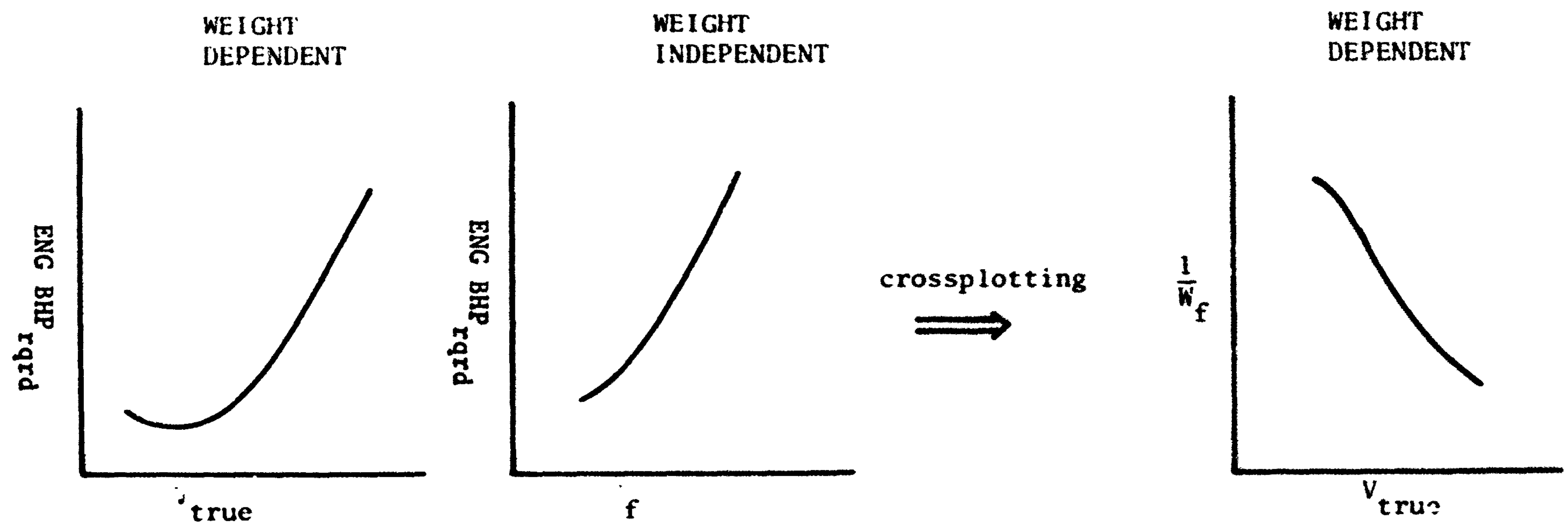


Figure II. Fuel Correction.

ENG BHP_{rqrd} versus W_f is not weight dependent and was calculated from the fairings of $1/W_f$ versus V_{true} and ENG BHP_{rqrd} versus V_{true} for the 3500-pound gross weight (fig. II). The ENG BHP_{rqrd} versus V_{true} is weight dependent and was calculated from the fairing for P_{iw} - V_{iw} using the test gross weight ($GRWT_{test}$) as the new gross weight ($GRWT_2$) and the sea level value of $\sqrt{\sigma}$.

STALL PERFORMANCE

25. Stall occurs when the coefficient of lift is a maximum. Therefore, stall speed can be corrected from test conditions to standard conditions merely by equating maximum coefficients of lift ($C_{L_{max}}$) where:

$$C_{L_{max}} = \frac{0.702\ GRWT_{test}}{\rho_{SL}\ S_w \left[V_{cal_{stall\ test}}\right]^2} = \frac{0.702\ GRWT_{std}}{\rho_{SL}\ S_w \left[V_{cal_{stall\ std}}\right]^2} \qquad (15)$$

Solving equation 15 for the corrected stall speed shows calibrated stall speed to be solely a function of gross weight - not altitude - where:

$$V_{cal_{stall\ std}} = V_{cal_{stall\ test}} \sqrt{\frac{GRWT_{std}}{GRWT_{test}}} \qquad (16)$$

26. Testing was conducted on the YO-3A airplane with mission equipment installed. Since mission equipment affects drag and not lift, as seen by comparison of the C_L - C_D curves with and without mission equipment installed, the maximum value of the coefficient of lift was the same with and without mission equipment installed. Therefore the stall speed was the same with and without mission equipment installed.

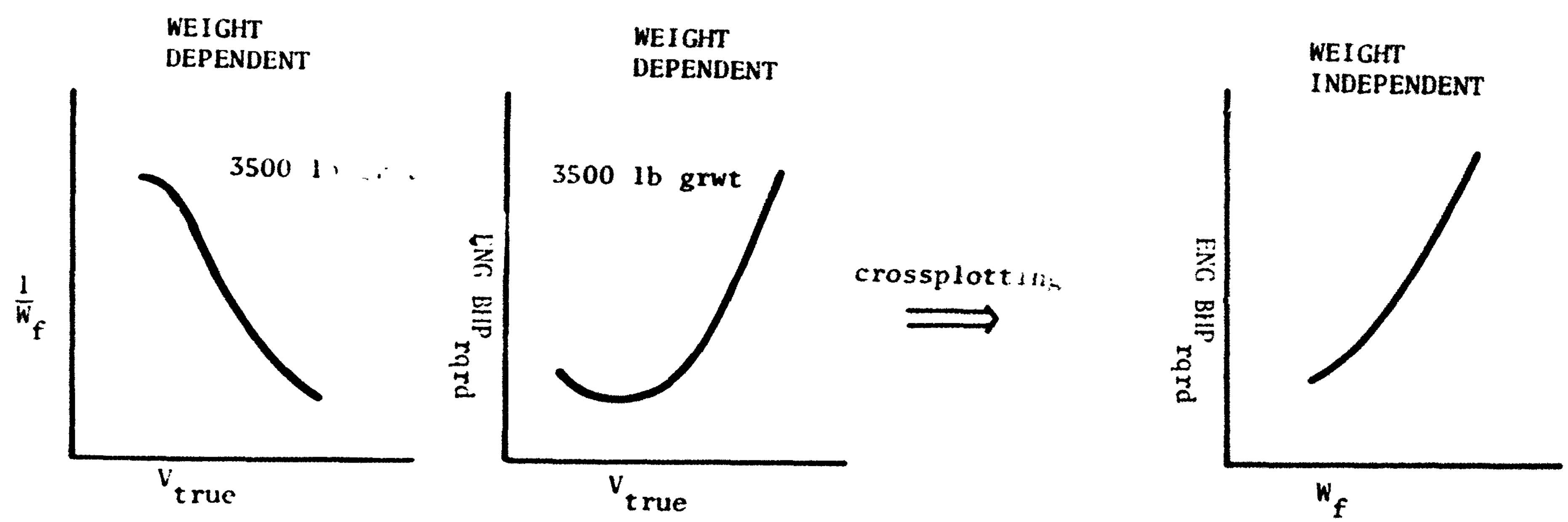


Figure II. Fuel Correction.

27. To correct stall speed at the standard gross weight ($GRWT_{std_1}$) to a new standard gross weight ($GRWT_{std_2}$), equation 16 was used with:

$$GRWT_{std} = GRWT_{std_2}$$

$$GRWT_{test} = GRWT_{std_1}$$

CLIMB PERFORMANCE

28. Sawtooth climb tests were conducted at several pressure altitudes to determine maximum rate of climb as a function of pressure altitude and, specifically, to check the sea level rate of climb guarantee. The test rates of climb (R/C_{test}) were determined by dividing the change in calibrated altitude ($\Delta H_{cal_{test}}$) by the change in time (Δt), where this data was recorded manually.

$$R/C_{test} = \frac{\Delta H_{cal_{test}}}{\Delta t} \sim \text{ft/min} \qquad (17)$$

29. Climb data has been standardized by correcting for differences in free air temperature, horsepower and gross weight according to the following equation:

$$R/C_{std} = \left\{ R/C_{test} \sqrt{\frac{\bar{T}_{abs_{test}}}{\bar{T}_{abs_{std}}}} + \frac{33000 \eta_{prop} \text{ PROP SHP}}{GRWT_{test}} \times \left[\sqrt{\frac{\bar{T}_{abs_{test}}}{\bar{T}_{abs_{std}}}} - \sqrt{\frac{\bar{T}_{abs_{std}}}{\bar{T}_{abs_{test}}}} \right] \right\} \times \left\{ 1 + \frac{GRWT_{test} - GRWT_{std}}{GRWT_{test}} \right\} \sim \text{ft/min} \qquad (18)$$

30. Climb testing was conducted on the YO-3A airplane with mission equipment installed. To determine rates of climb of the aircraft with the mission equipment removed, it was necessary to use the relationships between rate of climb (R/C), thrust horsepower available (THP_{avail}), thrust horsepower required for level flight ($THP_{rqrd\ FLF}$) and thrust horsepower available for climbing (THP_{climb}) - all of which are airspeed dependent.

31. Thrust horsepower available for climb, at a particular airspeed, is the difference between thrust horsepower available and thrust horsepower required for level flight at that particular airspeed (fig. III). THP_{avail} is altitude dependent but not gross weight or configuration dependent.

$$THP_{climb} = THP_{avail} - THP_{rqrd\ FLF} \tag{19}$$

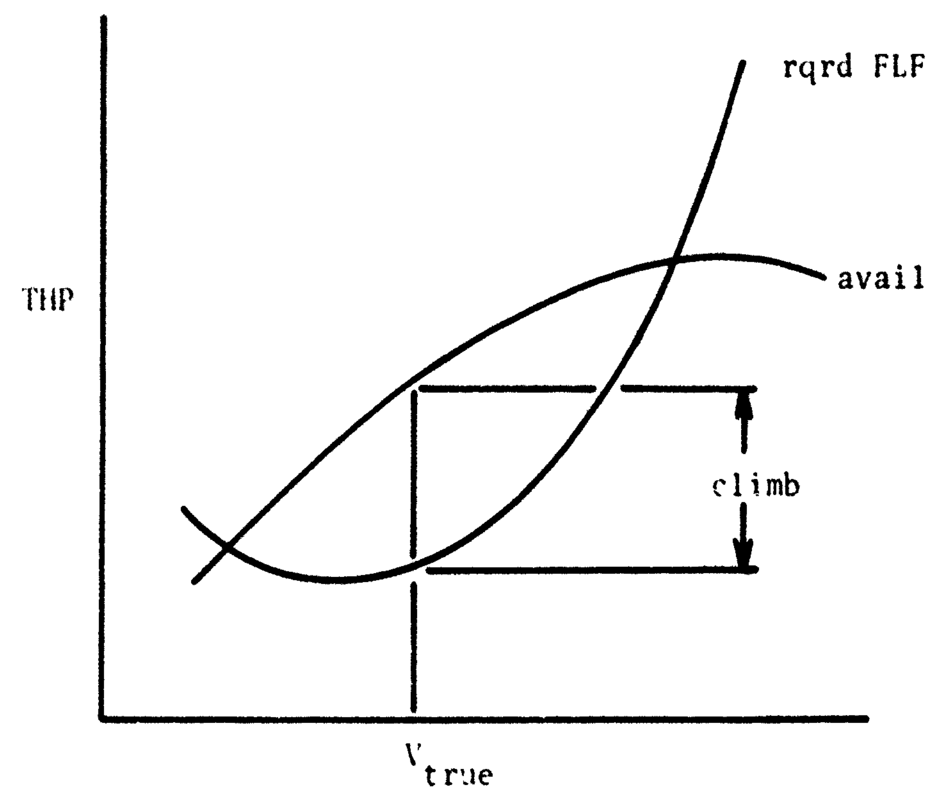


Figure III. Thrust Horsepower.

32. $THP_{rqrd\ FLF}$ is determined from a C_L versus C_D plot using equations 20 and 21, and is altitude, gross weight and configuration dependent.

$$C_L = \frac{0.702\ GRWT}{\rho_{test}\ S_w\left[V_{true}\right]^2} \tag{20}$$

$$C_D = \frac{228.7\ THP}{\rho_{test}\ S_w\left[V_{true}\right]^3} \tag{21}$$

Thrust horsepower available for climbing is related to rate of climb by equation 22.

$$R/C = \frac{33,000}{GRWT}\ THP_{climb} \sim ft/min \tag{22}$$

33. For a certain gross weight and configuration, $THP_{rqrd\ FLF}$ at sea level can be determined as a function of V_{true} from equations 20 and 21 and the corresponding C_L - C_D plot. If THP_{avail} is known, then THP_{climb} can be determined from equation 19 and R/C can then be determined from equation 22. A schematic of this process is shown in figure IV. Note that the necessary parameters are gross weight, true airspeed and configuration.

34. To determine the rate of climb without mission equipment installed, the rate of climb, from flight testing with mission equipment installed, was used. This rate of climb with mission equipment installed ($R/C_{w/}$) was used according to the rate of climb schematic, with the mission equipment installed C_L - C_D plot to determine the thrust horsepower available - not a function of configuration or gross weight. This thrust horsepower available and the mission equipment not installed C_L - C_D plot were then used, according to the rate of climb schematic, to determine the sea level rate of climb without mission equipment installed ($R/C_{w/o}$).

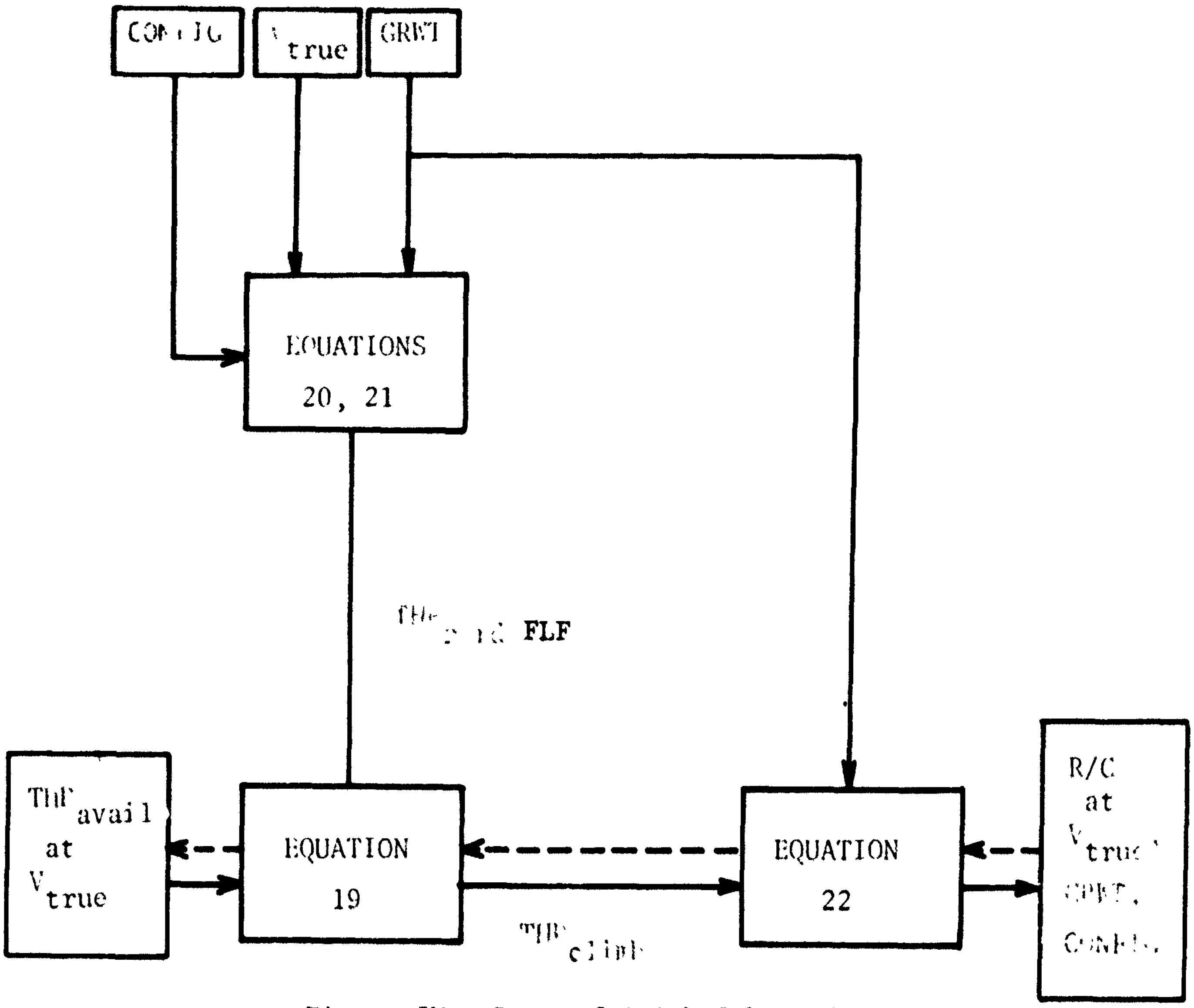


Figure IV. Rate of Climb Schematic.

35. To determine the rate of climb at 3381 pounds with mission equipment not installed, the thrust horsepower available, the new gross weight and the mission equipment not installed C_L - C_D plot were used according to the rate of climb schematic.

TAKEOFF PERFORMANCE

36. The total horizontal distance required to clear a 50-foot obstacle (s_{tot}) is composed of the ground roll distance (s_{gnd}) and the airborne horizontal distance (s_a).

$$s_{tot} = s_{gnd} + s_a \sim \text{feet} \qquad (23)$$

37. By use of a Fairchild Flight Analyzer Camera, test values of ground roll distance ($s_{gnd_{test}}$) and total horizontal distance to clear a 50-foot obstacle ($s_{tot_{test}}$) were determined. The airborne horizontal distance ($s_{a_{test}}$) was then determined from equation 23.

38. Test values were corrected to sea level standard day, zero runway slope and standard gross weight (3500 pounds). Wind conditions during testing were less than 1 knot and variable; therefore, no wind corrections were used in the calculations. Corrections to standard were made according to the following equations.

$$s_{std} = s_{a_{test}} + \Delta s_{a_{corr}} + s_{gnd_{test}} + \Delta s_{gnd_{corr}} \sim \text{feet} \qquad (24)$$

where:
$$\Delta s_{gnd_{corr}} = \frac{s_{gnd_{test}}}{1 - \frac{2g}{\left[V_{gnd_{LO}}\right]^2} s_{gnd_{test}} \text{ SIN } \theta_{rnwy}}$$

$$\times \left[2.4 \times \frac{GRWT_{std} - GRWT_{test}}{GRWT_{test}} - 2.4 \times \frac{\sigma_{SL} - \sigma_{test}}{\sigma_{test}} \right.$$

$$\left. + 0.5 \frac{T_{abs_{SL}} - T_{abs_{test}}}{T_{abs_{test}}} \right] \sim \text{feet} \qquad (25)$$

$$\Delta s_{a_{corr}} = s_a \left[2.2 \times \frac{GRWT_{std} - GRWT_{test}}{GRWT_{test}} - 2.2 \times \frac{\sigma_{SL} - \sigma_{test}}{\sigma_{test}} \right.$$

$$\left. + 0.6 \frac{T_{abs_{SL}} - T_{abs_{test}}}{T_{abs_{test}}} \right] \sim \text{feet} \qquad (26)$$

39. Testing was conducted on the airplane without mission equipment installed. To correct the takeoff distance to a different standard gross weight, the complete correction process must be repeated using the new standard gross weight in place of the original standard gross weight ($GRWT_{std}$).

STABILITY AND CONTROL

40. The data recorded from dynamic stability testing are presented as time histories of the pertinent parameters that describe the motion of the aircraft. Analysis of these time histories is necessary to determine the stability of the aircraft. An oscillating motion is defined by a time parameter and a damping parameter. For lightly damped systems, time is represented by the undamped natural frequency (ω_n) or the time period (T), and damping is represented by the damping factor (ζ) or the cycles-to-half ($C_{\frac{1}{2}}$) amplitude if convergent motion or the cycles-to-double amplitude (C_2) if divergent motion.

41. A typical time history is shown in figure V, where X_n/X_{n+1} is the half-cycle amplitude ratio and $T_{\frac{1}{2}}$ is the half-cycle period. To determine ζ, X_n/X_{n+1} was calculated from measurements of the actual time history data, and the damping factor graph shown as figure VI was used (ref 6, app I). If the motion was divergent, the abscissa was changed to X_{n+1}/X_n, and the sign of the corresponding ζ was changed.

42. The undamped natural frequency (ω_n) of the motion, in radians/sec was calculated from equation 27.

$$\omega_n = \frac{\pi}{T_{\frac{1}{2}} \sqrt{1 - \zeta^2}} \sim \text{rad/sec} \qquad (27)$$

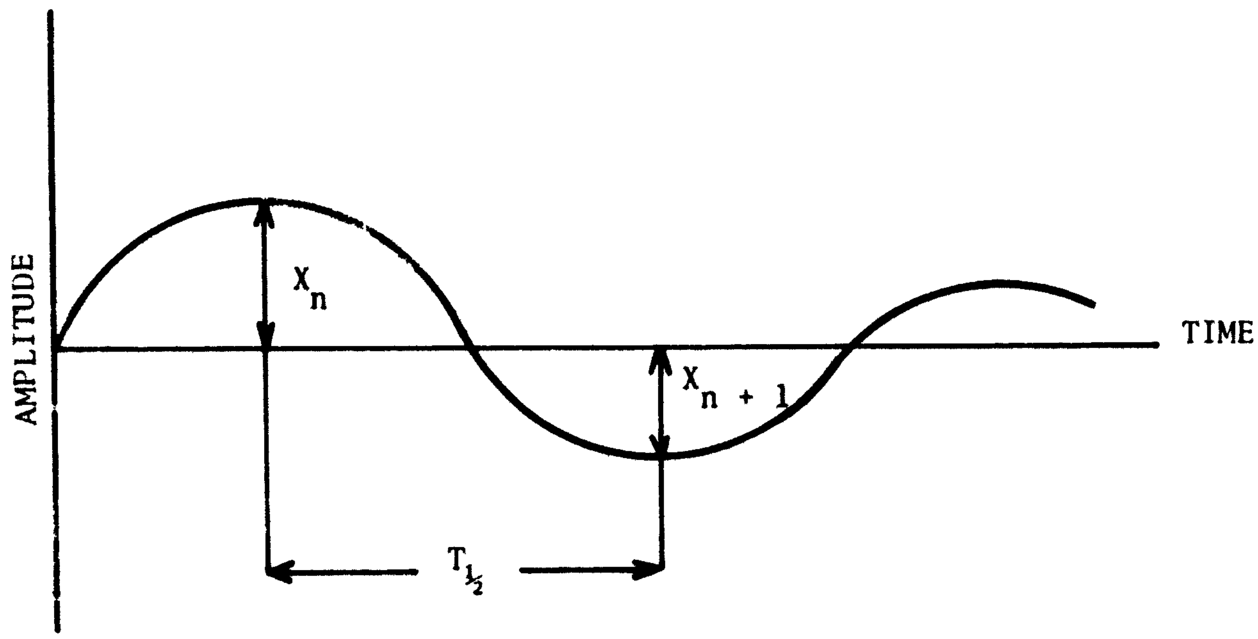


Figure V. Typical Time History.

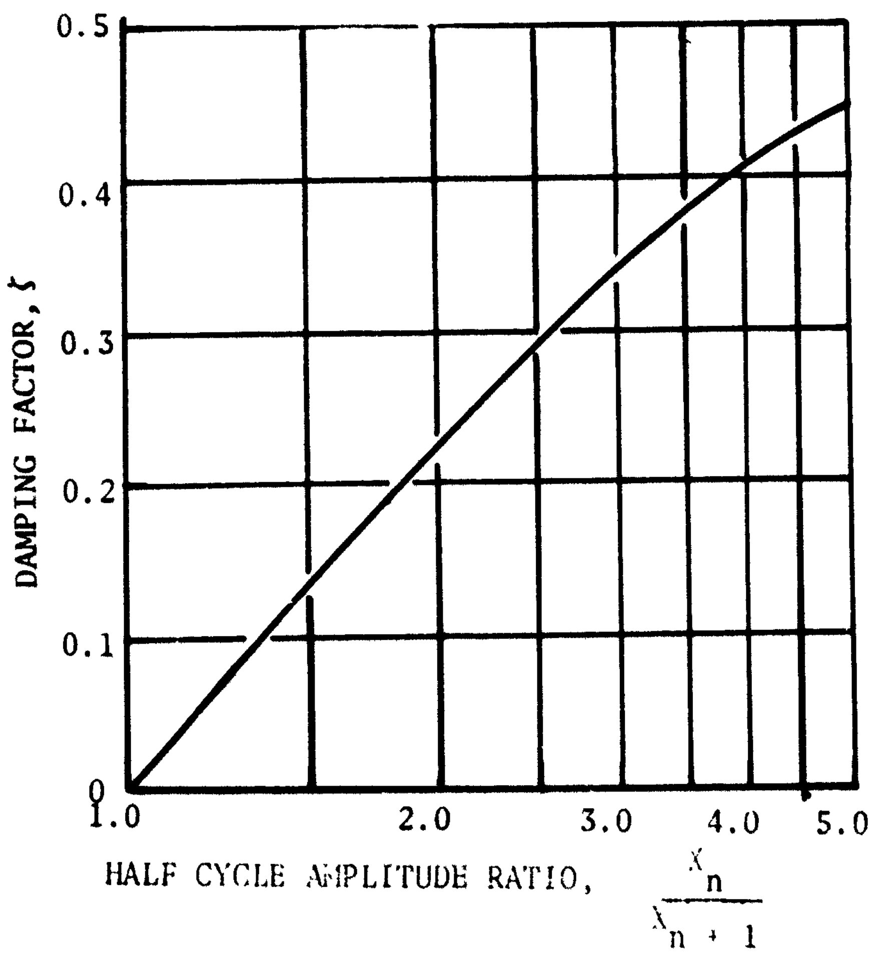


Figure VI. Damping Factor.

43. If the motion was convergent ($\zeta > 0$), then the cycles-to-half amplitude ($C_{\frac{1}{2}}$) was determined from X_n/X_{n+1} by using figure VII (ref 6, app I). If the motion was divergent ($\zeta < 0$), then the abscissa was changed to (X_{n+1}/X_n) and the ordinate to $1/C_2$.

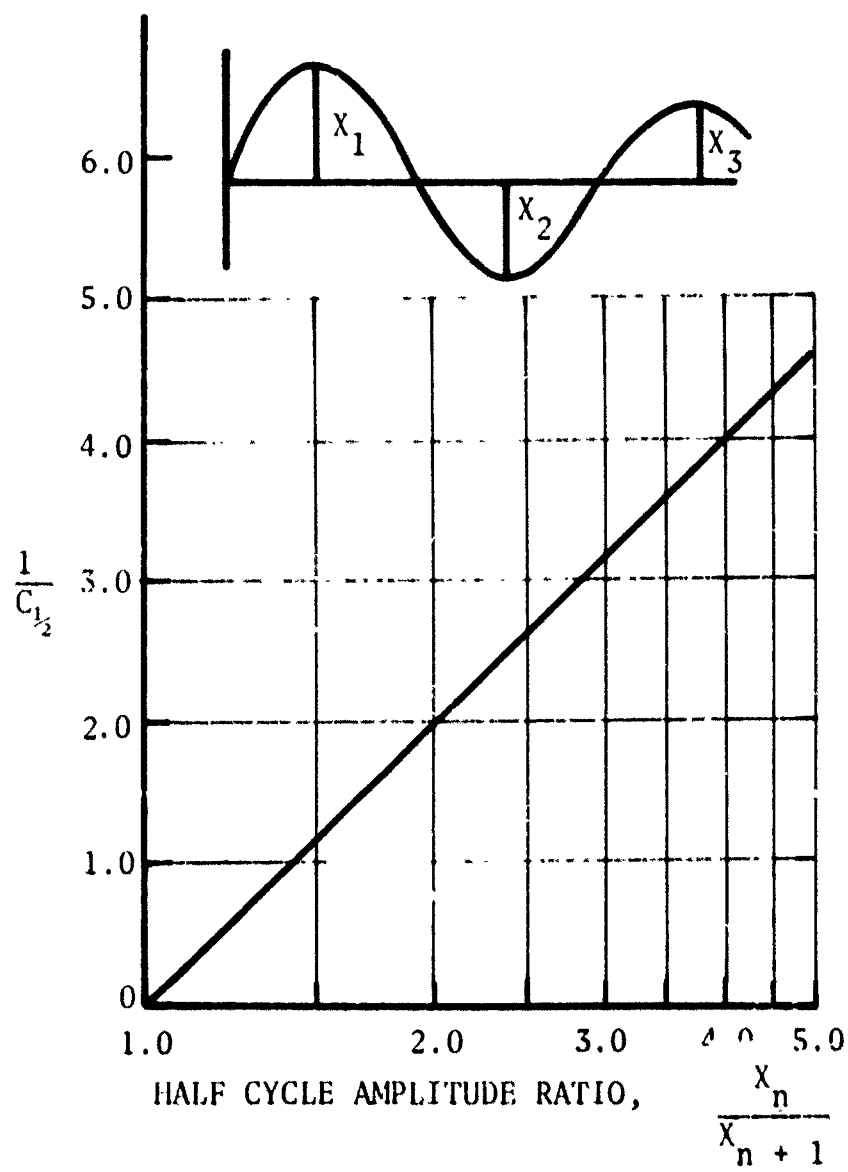
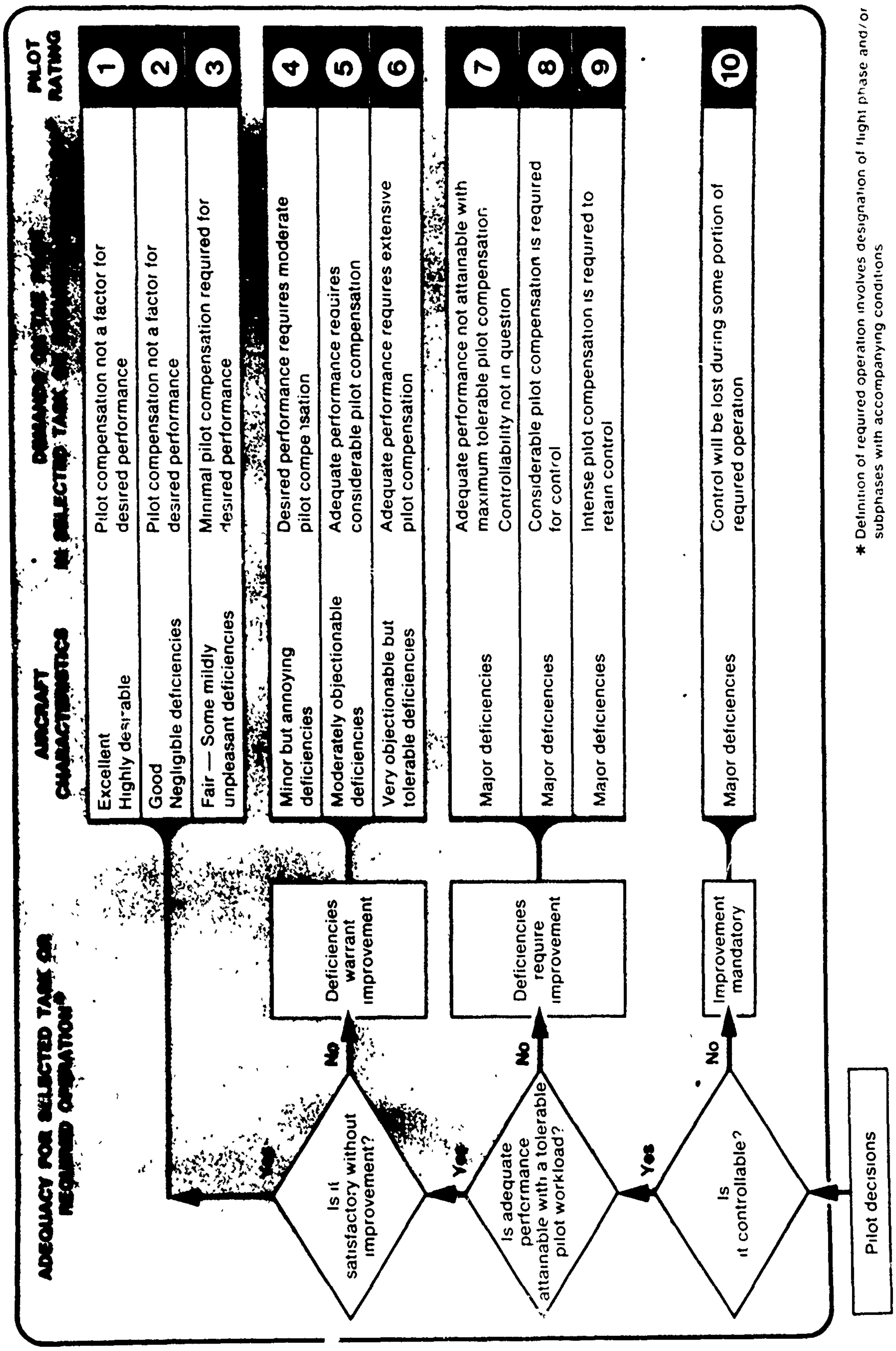


Figure VII. Cycles to Half Amplitude.

APPENDIX IX. HANDLING QUALITIES RATING SCALE



* Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions

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13. ABSTRACT

The Army Preliminary Evaluation I (APE I) of the YO-3A airplane, manufactured by the Lockheed Missiles and Space Company of Sunnyvale, California, was conducted from 18 June through 9 July 1969. Twenty-seven test flights encompassing 36.0 flight hours were flown from Moffet Field Naval Air Station and Crows Landing Naval Auxiliary Landing Field, California. Performance and stability and control tests were conducted to determine the suitability of the airplane to perform a classified mission. The test airplane exhibited an excellent mission concept with potential for the satisfactory accomplishment of the intended mission. However, the test airplane as evaluated was not sufficiently developed for operational deployment. The test program revealed 8 safety-of-flight deficiencies, 28 deficiencies for which correction is mandatory, and 23 shortcomings for which correction is desirable. The engine and fixed-pitch propeller combination results in reduced performance which compromises the takeoff, climb and maximum speed capabilities of the YO-3A airplane. With the exception of the stall speed and maximum level-flight airspeed guarantees, the test airplane did not meet the performance guarantees listed in the Detail Specification. The performance capability of the test airplane was marginal; however, an acceptable airplane capable of performing the intended mission would result with the correction of the following deficiencies: engine overheating, excessive directional control forces, excessive control system breakout plus friction force, unintentional braking during the takeoff ground roll, lack of an acceptable mixture leaning indicator and lack of static power rpm limits.

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13. ABSTRACT (continued)

Stall characteristics were acceptable with all controls effective throughout the stall sequence. An adverse sideslip characteristic combined with high rudder force requirements compromises the satisfactory accomplishment of the intended mission.

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