USAAEFA PROJECT NO. 81-01-3

FUEL CONSERVATION EVALUATION OF U.S. ARMY HELICOPTERS, PART 3 UH-1H FLIGHT TESTING

FINAL REPORT

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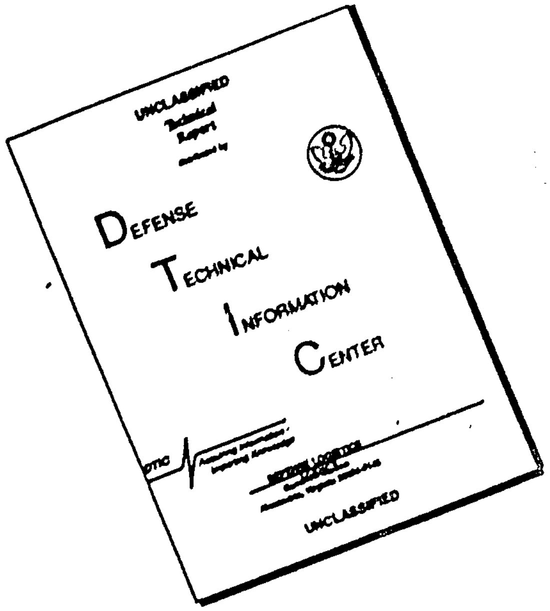

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The United States Army Aviation Engineering Flight Activity conducted level flight performance tests of the UH-1H helicopter to provide data to determine the most fuel efficient operating characteristics. Hot and cold weather test sites were used to extend the advancing tip Mach number data range to supplement existing YUH-1H performance data. Based on preliminary analysis, range can be increased significantly at optimum altitude and rotor speed compared to sea level and normal rotor speed (324 RPM). Rotor compressibility effects were significant at all test conditions.

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DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
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DRDAV-D

SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 81-01-3, Fuel Conservation Evaluation of US Army Helicopters, Part 3,UH-1H Flight Testing

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1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. The report documents part 3 of a 5 part effort which involves performance flight testing of the UH-1H to obtain performance data and determine the most efficient operating characteristics. Part 1 involved conducting a flight operation improvement analysis. Part 2 was initiated to develop and evaluate flight manual data designed for optimizing fuel conservation. Parts 3, 4 and 5 entail flight testing of the UH-1H, OH-58C, and AH-1S which is specifically oriented towards obtaining performance data applicable to fuel conservation. The part 3 evaluation conducted by the US Army Aviation Engineering Flight Activity (USAAEFA) consisted of obtaining detailed comprehensive performance data for the UH-1H in both hot and cold temperatures. The UH-1H Operator's Manual is currently being revised to increase the basic drag level defined in this report. A future change will incorporate performance data to emphasize optimum cruise techniques.

2. This Directorate agrees with the report conclusions and recommendations.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

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INTRODUCTION

BACKGROUND

1. The US Army is emphasizing fuel conservation in the operation of aircraft. The Deputy Chief of Staff for Logistics (DCSLOG), Aviation Logistics Office/Special Assistant (DALO-AV) requested the US Army Aviation Engineering Flight Activity (USAAEFA) provide data concerning the potential fuel savings of flying helicopters with doors closed. Analysis of existing data indicated the effect of door position on fuel consumption was negligible. However, it was suggested (ref 1, app A) that cruising at optimum rotor speed and altitude could reduce fuel consumption on specific flights up to 50 percent and overall Army Aviation fuel consumption by 5 percent or more. The US Army Aviation Research and Development Command (AVRADCOM) requested USAAEFA conduct the fuel conservation program on the UH-1H (refs 2, 3, and 4, app A). A project proposal (ref 5) and test plan (ref 6) were prepared in response to these requests.

TEST OBJECTIVES

2. The overall objective was to obtain flight test data to determine the most fuel efficient operating characteristics.

3. Specific test objectives were as follows:

a. Provide quantitative and qualitative flight test data for determining optimum cruise techniques.

b. Evaluate the production test aircraft performance and update the performance baseline on the 1960 model prototype aircraft (ref 7, app A).

c. Gather additional data to investigate rotor blade compressibility and its effect on fuel consumption.

DESCRIPTION

4. The UH-1H is a thirteen-place single engine helicopter using a single two-bladed teetering main rotor and a two-bladed pusher tail rotor. The maximum gross weight is 9500 pounds. Power is provided by a Lycoming T53-L-13B free turbine engine rated at 1400 shaft horsepower (SHP) at sea level standard day conditions. The main rotor transmission is limited to 1100 SHP for continuous operation. The test aircraft, US Army serial number 69-15532, is a standard production UH-1H. A more complete description of

the aircraft is presented in appendix B, the detail specification (ref 8, app A) and the operator's manual (ref 9, app A).

TEST SCOPE

5. During this evaluation, level flight performance tests were conducted. A total of 31 flights yielded 28.5 productive test hours. Eleven cold weather (-7°C to -16°C) flights were conducted in the vicinity of St. Paul, Minnesota from 18 February through 13 March 1981. Ten hot weather (16°C to 33°C) flights were conducted in the vicinity of El Centro, California from 11 August through 17 August 1981, and ten test flights were performed at Edwards Air Force Base, California from 25 August through 18 December 1981. Some test conditions of the previously tested YUH-1H (ref 7, app A) were duplicated. Flight restrictions in the airworthiness release issued by AVRADCOM and operating limitations contained in the operator's manual (ref 9, app A) were observed during the evaluation.

6. These tests were conducted at various referred rotor speeds ($N_R/\sqrt{\theta}$) over an actual rotor speed range from minimum (294 rpm) to maximum (324 rpm). To allow maximum possible variation of airspeed and rotor tip Mach number, tests were conducted at low to moderate thrust coefficients (C_T) of 26.0 to 35.5 x 10^{-4}. Most of the tests were conducted in a clean configuration (windows and doors closed), mid average longitudinal and lateral center of gravity (cg) (FS 137.0, BL 0.0), with engine bleed air OFF, and zero sideslip. One flight was conducted with an Infrared (IR) Suppressor and IR jammer installed. Hot weather tests were flown with cockpit windows and vents open.

TEST METHODOLOGY

7. Established engineering flight test techniques and data reduction procedures were used (ref 10, app A). Test methods are also briefly discussed in the Results and Discussion section of this report. A Vibration Rating Scale (VRS) (fig. D-1, app D) was used to augment crew comments relative to aircraft vibration levels. Ratings of handling qualities were based on a Handling Qualities Rating Scale (HQRS) (fig. D-2, app D). Flight test data were obtained from calibrated test instrumentation and were recorded on magnetic tape. A detailed listing of the test instrumentation is contained in appendix C. Data analysis methods are described in appendix D.

RESULTS AND DISCUSSION

GENERAL

8. Level flight performance tests were conducted to provide data to determine the most fuel efficient operating characteristics. Emphasis was placed on expanding the Mach number range of data available to aid the analysis of compressibility effects and supplement data available for the prototype YUH-1H. The basic test configuration was clean with doors and windows closed, mid lateral and longitudinal center of gravity, and zero sideslip. The production UH-1H test aircraft showed a drag increase of approximately one ft^2 (~5%) equivalent flat plate area over the prototype. The referred rotor speed technique of flying level flight performance test was found to be valid for the UH-1H and should be used for most of future level flight performance tests. The constant rotor speed technique should still be used to achieve maximum Mach number range at the extreme temperature conditions. The validity of the C_T and $N/\sqrt{\theta}$ parameters should be checked near their extreme test values. Analysis indicates that a potential range improvement of more than 100% and fuel saving as much as 50% exists for certain conditions. Optimum cruise data should be made available to operational units as early as practical.

LEVEL FLIGHT PERFORMANCE

General

9. Level flight performance tests were conducted to determine the power required for level flight and specific range versus airspeed. Emphasis was placed on expanding the Mach number range to the maximum extent practical. The thrust coefficient range was from 26 to 35.5 x 10^{-4}. This allowed for maximum possible airspeed (and Mach number) variation. An advancing tip Mach number range from .69 to .96 was achieved using the full allowable rotor speed range of 294 to 324 rpm. The power available from the test engine was approximately 13% lower than that available for the reference 7 tests which precluded expanding the thrust coefficient range to investigate blade stall (as requested in ref 3). One test was conducted to determine the effect of an infrared (IR) Suppressor and Jammer installation. Constant C_T was maintained for each test by using either the constant rotor speed or the constant referred rotor speed technique. Test results are shown on figures 1 through 31, appendix E.

Comparison with YUH-1H

10. The power required for the YUH-1H is compared to the UH-1H test data on figures 1, 10 through 16, and 25 through 30 in appendix E for those conditions within the Mach number range of the reference 7 tests. At typical cruise speeds the UH-1H shows slightly higher power required than the YUH-1H. Converted to equivalent flat plate area, the power increase varies from 0 to 2.1 square feet with an average of approximately 1.0 square feet. Several external equipment changes from the prototype to the production airframe might account for the drag difference. These include: roof mounted pitot-static probe, roof mounted FM homing antenna, and tailboom mounted anticollision light. A more complete description of the differences is included in appendix B. The baseline data from reference 7, appendix A, should be corrected by the apparent one square foot drag difference for operator's manual use.

11. The low speed range also shows a difference in power required. Although there is substantial variation, the general trend at the lowest speeds tested is less power required for the production aircraft than the prototype. The reason for this has not been determined.

Configuration Variation

12. One level flight performance test was conducted with the Garrett Hot Metal Plus Plume IR Suppressor and AN/ALQ-144 IR Jammer installed. Results are compared to the clean YUH-1H data (ref 7) on figure 30, appendix E. The data indicates an increase in drag of approximately 2.2 square feet at cruise speed. The hot weather flights were conducted with cockpit windows and vents open. Figures 22 and 23, appendix E at similar conditions show no apparent drag change for this configuration variation.

Engineering Parameter Validation

13. This test program was flown using both the constant referred rotor speed ($N/\sqrt{\theta}$) and constant actual rotor speed (W/σ) methods. Those tests using the $N/\sqrt{\theta}$ method are designated on the appendix E data by noting the average $N/\sqrt{\theta}$ value. From the test results it appears the best procedure for most conditions is to use the $N/\sqrt{\theta}$ technique. The W/σ technique should be used with the highest rotor speed at the coldest temperature, and the lowest rotor speed at the warmest temperature to obtain the maximum possible variation of blade tip Mach number. The validity of the engineering parameters, C_T and $N/\sqrt{\theta}$, was checked by repeating some tests at the

same C_T and $N/\sqrt{\theta}$ but differing conditions of pressure altitude, ambient temperature, rotor speed, and gross weight. An example of such a comparison is shown on figure 31. Within the range of data the $N/\sqrt{\theta}$ and C_T parameters appear valid for the UH-1H. The validity of the C_T and $N/\sqrt{\theta}$ parameters should be checked near the extreme values by repeating them at as widely different dimensional conditions as possible.

Rotor Speed Effects

14. The effect of rotor speed on UH-1H cruise power and fuel flow at fairly typical conditions was directly demonstrated on one flight. The purpose of the test flight was to define the level flight power required from 20 knots to maximum airspeed at an aim gross weight to air density ratio of 7400 lb, for rotor speeds of 294 and 324 rpm. Temperature varied from -11.4°C to -6.0°C over the flight. Results are shown on figures 1 and 6, appendix E. For the 324 rpm test, average fuel flow was 77 gal/hr. Maximum speed achieved was 123 knots true airspeed (KTAS) and this required 1140 SHP. Fuel flow at this point was 105 gal/hr or 714 lb/hr. For the 294 rpm test, average fuel flow was 45 gal/hr. Maximum speed achieved was 126 KTAS, which required 824 SHP. Fuel flow at this point was 82 gal/hr or 564 lb/hr. Comparison of maximum and average fuel consumption rates provides a graphic example of the effects of rotor speed.

Optimum Cruise Mission

15. Operational type missions were flown to provide a more realistic comparison of potential fuel savings than derived from engineering data. No corrections were made for winds or route of flight. On the ferry flight from Edwards AFB, to El Centro, California, (198 nautical miles) approximate optimum profiles were flown. Takeoff weight was 7900 lb. A maximum power cruise climb was made to an enroute altitude of 9500 feet at a free air temperature of 16°C using 314 rotor rpm. Wind was forcast as a 15 knot quartering tailwind. Optimum rotor speed (below 7500 lb) would have been 294 rpm at an optimum altitude of approximately 7000 feet. This rotor speed was undesirable because of degraded flying qualities in the existing moderate turbulence (para 22). The 314 rotor rpm used, required the higher optimum altitude. The direct flight took 1.5 hours and consumed 122 gallons. On the return flight to Edwards AFB "normal" low level (1000-1500 ft above ground level (AGL)) flight procedures were used at a rotor speed of 324 rpm. Air temperature was approximately 25°C and wind was forcast as calm. Takeoff weight was again 7900 lbs. A fuel stop was made at Palm Springs, California. to insure adequate reserves. The flight took 2.1 hours (total of 3.5 hours

enroute time), consuming a total of 207 gallons. Distance was 201 nautical miles. The comparison flights illustrate the general benefit of increased range which in this case reduced both flight time and total trip time in addition to saving fuel by eliminating a fuel stop.

ENGINE PERFORMANCE

16. Test engine performance data were obtained throughout these tests. The test engine (T53-L-13B S/N LE 20825B) was calibrated by the Corpus Christi Army Depot prior to the tests. This engine had two major overhauls and at the start of the test had 2343 total hours and 623 hours since overhaul. Relatively low maximum power (as limited by maximum N_1 setting) was available, compared to those engines used for the reference 7 tests. Fuel consumption, however, was still low compared to the model specification (ref 11) applicable at the time of the reference 7 tests. Accessory bleed air useage has changed since the prototype tests. The prototype used a bleed air driven fuel boost pump, while the production boost pumps are electrical. The production aircraft uses bleed air to scavenge the particle separator, while the prototype did not. Referred engine characteristics are shown in figures 32 through 34, appendix E.

Power Available

17. The test engine just met the field acceptance criteria for military power available (3% less than specification power). To achieve this power, the engine was set at 101.1% gas producer speed (with 101.5% maximum allowable). By comparison, the test engines used during earlier tests (ref 7) produced approximately 10% more than specification power at gas producer speed trim values of approximately 96%.

Fuel Consumption

18. Although power available is significantly degraded, the test engine fuel consumption is comparable to the early T53-L-13 engines. Specific fuel consumption data from the engine calibration are compared to the model specification (ref 11) at 6600 output shaft rpm on figure A. Measured fuel flow rate was less than the model specification curve.

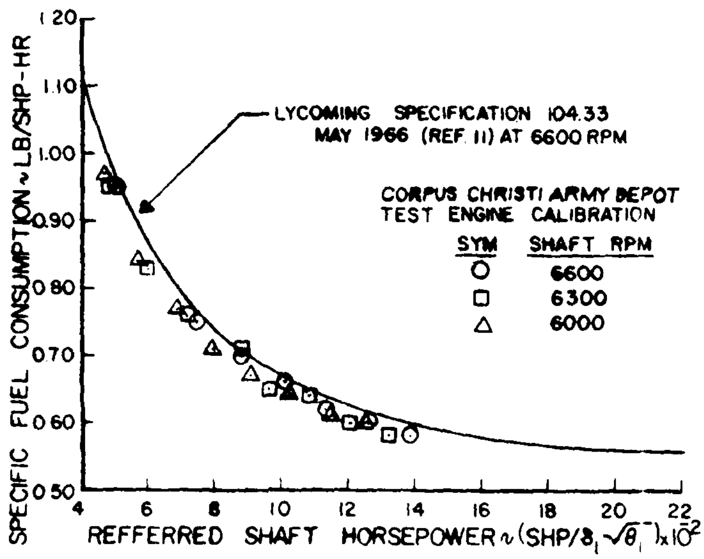


Figure A. T53-L-13 Engine Specific Fuel Consumption

At a given power the engine calibration data shows a fuel flow decrease of 15 to 25 lb/hr as output shaft speed is reduced from 6600 to 6000 rpm. The specific fuel consumption (SFC) curve (ref 11 for 6600 rpm) on figure A can be closely approximated by:

$$SFC = 0.4933 + 2791/(SHP/\delta_1 \sqrt{\theta_1})^{1.4}$$

In the absence of a current model specification computer program, this was used to define the specific range fairings on figures 1 through 30, appendix E.

HANDLING QUALITIES

General

19. A limited qualitative handling qualities evaluation was conducted throughout these tests. Particular emphasis was placed on operations at the lower rotor speeds (below 3 4 rpm) for which little recent experience exists. The following paragraphs discuss simulated engine failures, controllability and vibration.

Simulated Engine Failures

20. Autorotational entry characteristics of the UH-1H helicopter to simulated power failure were evaluated by rapidly turning the throttle twist grip to the flight idle position, with the collective control position held fixed for a period of 2 seconds or

until pilot corrective action was required, whichever came first. Tests were conducted in level flight at 86 KTAS at an average gross weight of 7320 pounds, a density altitude of 5000 feet, and rotor speeds of 324 and 300 rpm. Main rotor speed was the determining factor in initiating corrective action. Rotor speed dropped rapidly following throttle chop to a minimum of 267 rpm. Corrective action was initiated as rotor speed passed through 294 rpm by lowering the collective, lowering the nose, rolling level and correcting for heading. To regain rotor speed, the collective was lowered in a smooth continuous motion. During all tests, rotor speed was re-established between 314 and 324 rpm within 2-3 seconds after collective was lowered. No unusual handling qualities or vibrations were experienced during the test and the autorotational entry characteristics were satisfactory.

21. An Army study (ref 12) indicated mast separation accidents might be caused by excessive flapping at low rotor speeds following sudden engine failures. This question should be resolved. To insure an operational low rotor speed warning, the threshold was reset to 293 rpm for this project. This was an easily accomplished organizational maintenance procedure. The low rotor speed warning threshold is currently set at 300 to 310 rpm. The normal setting should be lowered to 290 to 294 rpm so the warning system will be active at the proposed lower cruise rotor speeds.

Controllability

22. The handling qualities of the UH-1H were qualitatively evaluated and compared with previous UH-1H tests to determine if the handling characteristics were significantly affected by operation at lower rotor speeds. The only appreciable difference in handling qualities noted at lower rotor speeds was reduced controllability in all three axes at rotor speeds below 314 rpm. The reduced controllability characteristics were perceived as aircraft response becoming more sluggish with decreasing rotor speed during maneuvering flight, and increasing yaw and roll gust response during flight in turbulent conditions. The overall effect was an increase in pilot workload to maintain trim flight while maneuvering or operating in light to moderate turbulence (HQRS 4). Rotor speeds below 314 rpm should be used only for cruise flight in less than moderate turbulence and aggressive maneuvering should be avoided. No low airspeed tests were accomplished. However, based on the directional control difficulties reported in reference 7, all low airspeed operations should be conducted with maximum rotor speed (324 rpm).

Vibration

23. Vibration characteristics were evaluated qualitatively during all tests. The vibration characteristics did not appear to change significantly with changes in density altitude at the limit airspeeds. At the colder temperatures (below 0°C), pilot station 2/rev vibration levels at 324 and 314 rpm were generally low (VRS 3) at minimum airspeeds and increased continuously to severe at maximum airspeed (123 KTAS) (VRS 7). At rotor speeds from 314 rpm to the minimum tested (294 rpm) vibration levels were very low (VRS 2) throughout the airspeed range, except at maximum airspeed (above 116 KTAS) where they abruptly changed to very high amplitudes (VRS 7). The high vibration levels at 324 rpm are thought to be due to compressibility effects (advancing tip Mach number above 0.95 at the maximum airspeeds) which is well beyond wind tunnel test drag and moment divergence for the rotor airfoil section. The sudden increase in vibration levels at the maximum airspeed at 294 rpm may have been due to blade stall or Mach divergence on the retreating blade. Although not necessarily related to the high vibration levels, accelerated pitch-change-link rod-end-bearing wear became apparent during preflight inspections. This may have been caused by the high Mach numbers, lower rotor speeds or some other factor. The bearings were replaced during the cold weather tests. They were approaching maximum wear tolerance at the end of the project. Loads associated with the lower (294) rotor rpm should be reviewed. The impact of increased use of low rotor speed on component lives should be determined.

OPTIMUM CRUISE ANALYSIS

24. The majority of the information in this section is based on data derived from the level flight summary data in reference 7 and the engine model specification (ref 11) available during the reference 7 tests. The derivation methods are described in paragraph 17, appendix D.

Engine Effects

25. A large part of the gain in cruise performance comes from operating the engine at more efficient conditions. This is best illustrated by the engine specific fuel consumption curve in figure A (para 18). Examining the referred power parameter, $SHP/\delta_1\sqrt{\theta_1}$, implies that higher power, lower inlet pressure (higher altitude) and lower inlet temperature improve engine fuel efficiency.

Rotor Speed Effects

26. Primary factors influencing optimum cruise rotor speed are advancing blade drag effects, retreating blade drag effects and optimum engine power turbine speeds relative to rotor operating speeds. If retreating blade effects dominate, optimum rotor speed will tend to be high. If advancing blade effects dominate, optimum rotor speed will tend to be low. Optimum turbine speed may shift optimum rotor speed either way or have no effect. The UH-1H optimum rotor speed is always at or below minimum allowable rotor speed. Optimum turbine speed for the T53-L-13 engine is generally below allowable equivalent rotor speed. From the test engine calibration data, fuel flow at equivalent power decrease 15 to 25 lb/hr with reduced output shaft speed from 6600 to 6000 rpm (324 to 294 rotor rpm). Sufficient data were not available to develop a reliable relationship of fuel flow variation with turbine speed. This relationship is normally obtained from the engine model specification computer program. The predominant effect of rotor speed for the UH-1H is variation of power required with advancing rotor tip Mach number (compressibility) because of the relatively high rotational tip speed. This effect is shown for typical cruise conditions on figure B over the possible operational ambient temperature range as power required versus ambient temperature for three rotor speeds with advancing rotor tip Mach number noted. Compressibility effects for the UH-1H were significant at all test conditions and considerable fuel savings resulted from operating at minimum rotor speeds.

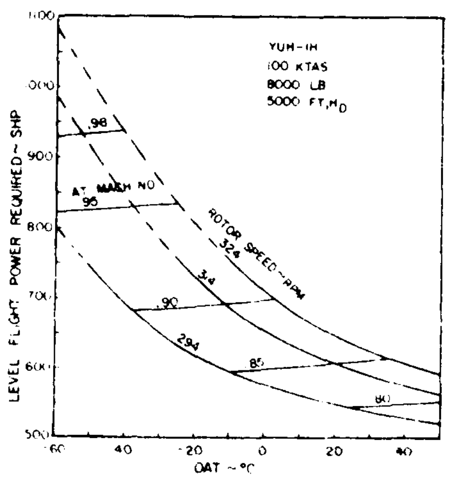


Figure B. Effect of Rotor Speed and Temperature on YUH-1H Power Required for Level Flight

Altitude and Temperature Effects

27. The primary effect of altitude is the improvement in engine fuel efficiency through the pressure ratio parameter δ_1. An additional effect is through the $\sqrt{\theta_1}$ parameter since temperature normally decreases with increasing altitude. The effect of altitude on fuel flow at a constant power is shown on figure C for three temperature conditions.

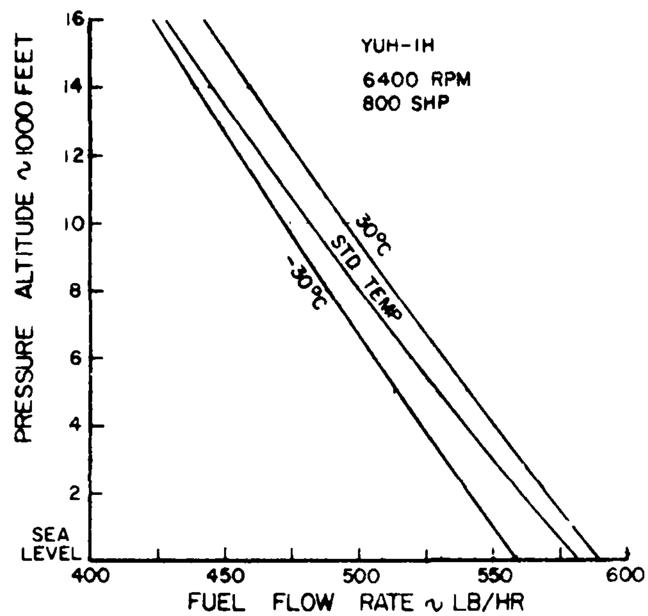


Figure C. Effect of Altitude and Temperature on T53-L-13 Fuel Flow

The variation of power required for level flight with altitude depends on the particular helicopter design and conditions of weight, altitude, temperature, rotor speed and airspeed. The general trend, for airspeeds near cruise speed, is approximately constant power (slight increase or decrease depending on conditions) up to some altitude above which a sharp increase in power required for level flight occurs. The altitude at which this increase occurs also depends on flight conditions. An example of the variation in power required with altitude at typical cruise conditions is shown on figure D.

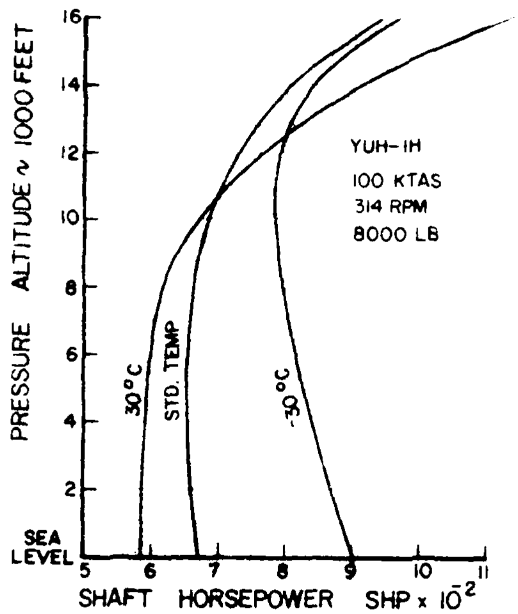


Figure D. Effect of Altitude and Temperature on YUH-1H Level Flight Power

The combined effects on cruise performance of the variation of the power required for level flight and the improvement in engine fuel efficiency with increased altitude are shown on figure E as specific range versus pressure altitude. Optimum altitude occurs where the increase in fuel flow caused by increasing power with altitude equals the decrease in fuel flow caused by the improving engine fuel efficiency with increased altitude.

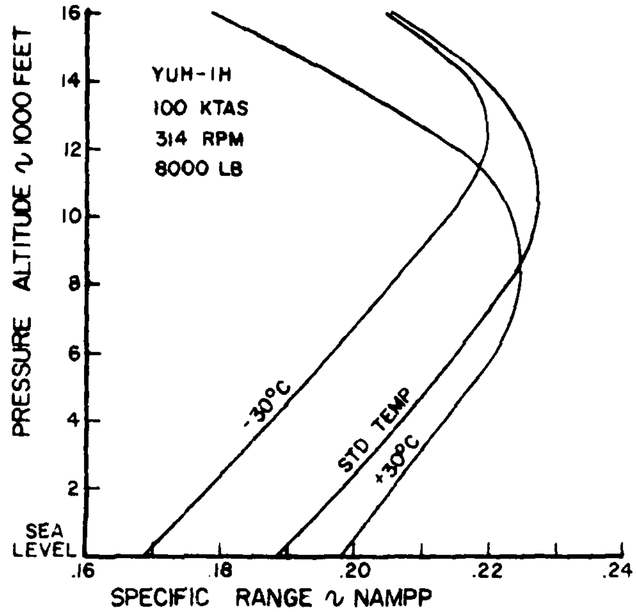


Figure E. Effect of Altitude and Temperature on YUH-1H Specific Range

Airspeed - Windspeed Effects

28. Wind speed can have a significant effect on optimum cruise airspeed. To obtain maximum range in terms of ground distance cruise airspeed for the UH-1H must be varied approximately 40% of the effective wind speed (difference between airspeed and ground speed). Because of the possible large cruise airspeed variation and therefore cruise power variation, optimum altitude will also be a function of wind speed. This effect has not been fully analyzed.

Optimum Climb and Descent Schedules

29. Optimum climb and descent profiles for helicopters have not yet been thoroughly investigated. Preliminary analysis shows the best climb schedule is a constant indicated airspeed 10 knots below optimum altitude maximum range cruise speed and maximum available power. Optimum descent schedule is to maintain optimum altitude cruise power and increase airspeed to obtain a normal descent rate (~500 ft/min). Where airspeed limits or turbulence preclude this schedule, airspeed should be the maximum practical with power reduced to achieve normal descent rates.

Optimum Cruise Summary

30. The optimum cruise results at rotor speeds of 294 and 314 rpm are summarized in figure F for the YUH-1H.

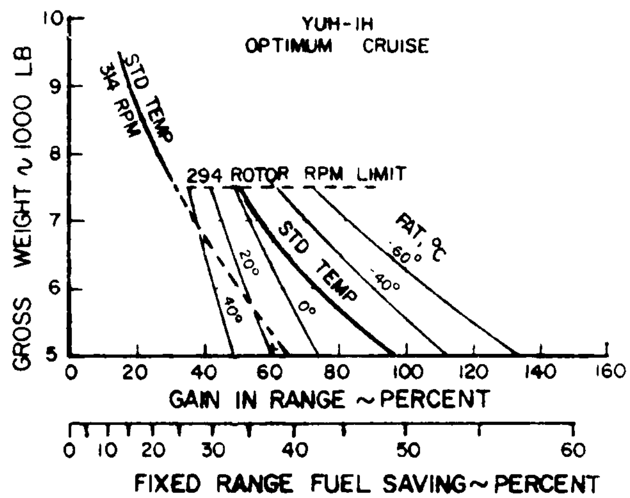


Figure F. YUH-1H Optimum Cruise Gains Compared to Sea Level and 324 Rotor rpm

Potential fuel savings for a given mission vary from 20% to more than 50% and range can be increased more than 100% depending on mission weights and ambient temperatures. Overall fuel savings will depend on operational requirements which may limit optimum technique use. Low level tactics and training preclude optimizing altitude and airspeed. Lack of oxygen systems and psychological factors may reduce the use of optimum altitudes. Even with these limitations actual fuel savings are estimated to be on the order of 10%. This estimate is based on the following assumptions: reduction of normal rotor speed from 324 to 314 rpm for cruise, climb and descent; full use of optimum cruise procedures 10% of the time; partial use (for 50% of potential savings) 25% of the time; average mission conditions of 8600 lb takeoff weight (full fuel, two crew members and 1500 lb payload); and standard temperatures. Another benefit from optimum cruise will be the extended range or endurance. Optimum endurance charts are yet to be developed but endurance gains will be even greater than the range benefits. Optimum cruise data should be made available to operational units as soon as practical.

CONCLUSIONS

31. The following conclusions in order of importance were reached as a result of these tests and the optimum cruise analysis.

a. Optimum cruise techniques can increase range more than 100% depending on conditions. Fuel savings for a given mission can be as much as 50% (para 30).

b. Compressibility effects for the UH-1H were significant at all test conditions and considerable fuel savings resulted from operating at minimum rotor speeds (para 26).

c. The production UH-1H has approximately one ft^2 (~5%) equivalent flat plate area drag increase compared to the prototype YUH-1H (para 10).

d. Handling qualities in cruise flight were degraded but acceptable in less than moderate turbulence at rotor speeds below 314 rpm (para 22).

e. The thrust coefficient and referred rotor speed parameters were determined to be valid for the UH-1H within the range of conditions of these tests (para 13).

f. Two-per-rotor-revolution vibration levels at cold temperatures could be reduced by decreasing rotor speed (para 23).

RECOMMENDATIONS

32. The following recommendations are made with regard to the operator's manual.

a. Optimum cruise data should be included in the operator's manual as soon as practical (para 30).

b. UH-1H data should be corrected based on the increased drag determined from these tests (para 10).

c. For the UH-1H, rotor speeds below 314 rpm should be used only in cruise flight in less than moderate turbulence. Aggressive maneuvering should be avoided at low rotor speeds (para 22).

33. The following additional studies should be conducted:

a. The question of excessive rotor flapping at low rotor speeds should be resolved (para 22).

b. Flight loads data at low rotor speeds should be reviewed. The impact of increased use of low rotor speeds on component lives should be determined (para 23).

34. The following recommendations are made with regard to future level flight performance:

a. The referred rotor speed technique should be used for most tests. The constant rotor speed technique should be used at extreme temperatures to achieve the maximum practical Mach number range (para 13).

b. The validity of thrust coefficient and referred rotor speed should be checked (para 13).

APPENDIX A. REFERENCES

1. USAAEFA Message to DALO-AV, subject: Aviation Fuel Conservation, 18 August 1980.

2. Letter, AVRADCOM, DRDAV-DI, 22 December 1980, subject: Test Request No. 80-22, Fuel Conservation Evaluation of US Army Helicopters.

3. Letter, AVRADCOM, DRDAV-DI, 27 March 1981, subject: Test Request No. 81-01-3, Fuel Conservation Evaluation of US Army Helicopters, Part 3, UH-1H Flight Testing.

4. Letter, AVRADCOM, DRDAV-DI, 27 March 1981, subject: Test Request No. 81-01-2, Fuel Conservation Evaluation of US Army Helicopters, Part 2, Development and Evaluation of Fuel Conservation Formatted Data.

5. USAAEFA Project Proposal: Blade Stall/Compressibility Investigation, submitted 9 January 1981.

6. Letter, DAVTE-M, 4 September 1981, subject: Test Plan, Fuel Conservation Evaluation (UH-1H Flight Testing), USAAEFA Project No. 81-01-3.

7. Final Report, USAAEFA Project No. 66-04, *Engineering Flight Test of the YUH-1H Helicopter (Phase D Limited)* November 1970.

8. Detail Specification, Bell Helicopter Textron, No. 205-947-177, UH-1H Utility Helicopter FY-74 Procurement, 15 May 1973.

9. Technical Manual, TM 55-1520-210-10, *Operator's Manual, US Army Models UH-1D/H and EH-1H Helicopters* 18 May 1979, change 15, 25 February 1982.

10. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Helicopter Stability and Control* 10 June 1968.

11. Report, Lycoming, *Model Specification T53-L-13, Shaft Turbine Engine*. September 1964, revised May 1966.

12. Summary Report, Air Mobility R&D Laboratory AVSCOM, UH-1H/AH-1 Rotor System Technical Risk Assessment, March 1974.

APPENDIX B. DETAILED HELICOPTER DESCRIPTION

SOURCES OF INFORMATION

1. The information contained in this appendix was obtained from the operator's manual (ref 9), the airframe model specification (ref 8), the engine model specification (ref 11), the airframe manufacturer, the engine manufacturer or directly by measurement on the test aircraft.

COMPARISON WITH PROTOTYPE AIRCRAFT

2. Data and other comparisons of the production test aircraft used for these tests with the prototype aircraft used for the reference 7 tests are made throughout this report. The current test aircraft, UH-1H S/N 69-15532 is shown in photo A. The prototype aircraft YUH-1H S/N 60-6029 is shown in photo B. The only significant external instrumentation was the test airspeed boom that included swiveling pitot-static source, ram air temperature probe, angle of attack vane and sideslip vane. The boom installation was similar on both aircraft. Boom drag was not determined for either aircraft. There was a difference in engine bleed air driven accessories. The prototype had an engine bleed air driven fuel boost pump. The boost pump and oil cooler fan used 1.15% of engine airflow. The production aircraft has electrically powered boost pumps. The production aircraft has a bleed air scavenged inlet particle separator (bleed air amount unknown). The prototype inlet particle separator was manually cleaned and did not require engine bleed air. Apparent external differences of the two aircraft are listed below.

Item	Production	Prototype
Cargo Hook	Not Installed	Installed
Pitot-Static Probe	Roof Mounted	Nose Mounted
FM Homing Antenna	Roof Mounted	Nose Mounted
Anti-Collision Lights(s)	Engine Cowl and Tailboom	Engine Cowl Only

DESIGN DATA

Overall Dimensions

Length (rotor turning)	57 ft, 1.1 in.
Length (nose to tail)	41 ft, 11.1 in.
Width of skids (maximum width except rotor)	9 ft, 6.6 in.

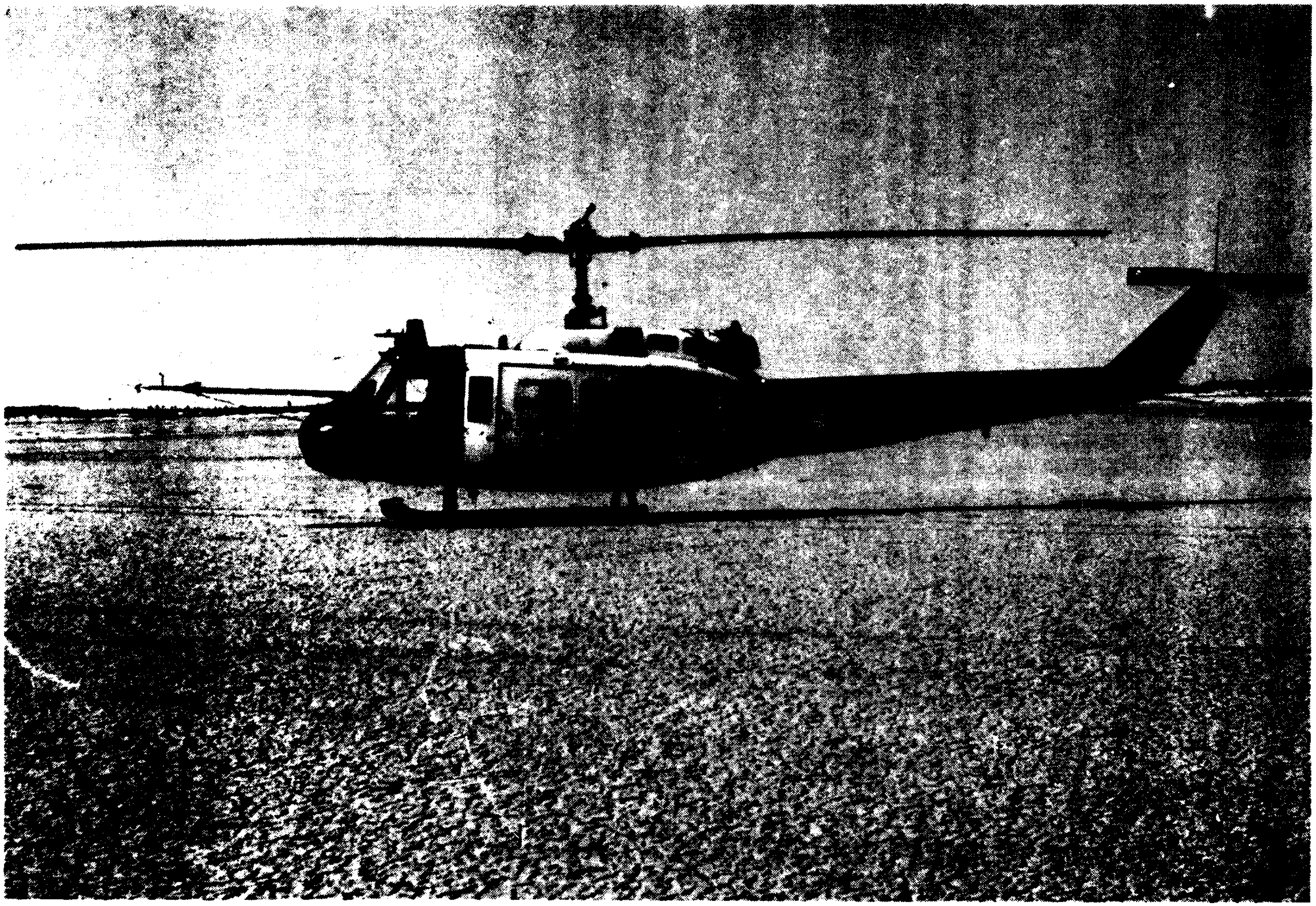


Photo A. Test Aircraft, Production UH-1H S/N 69-15532

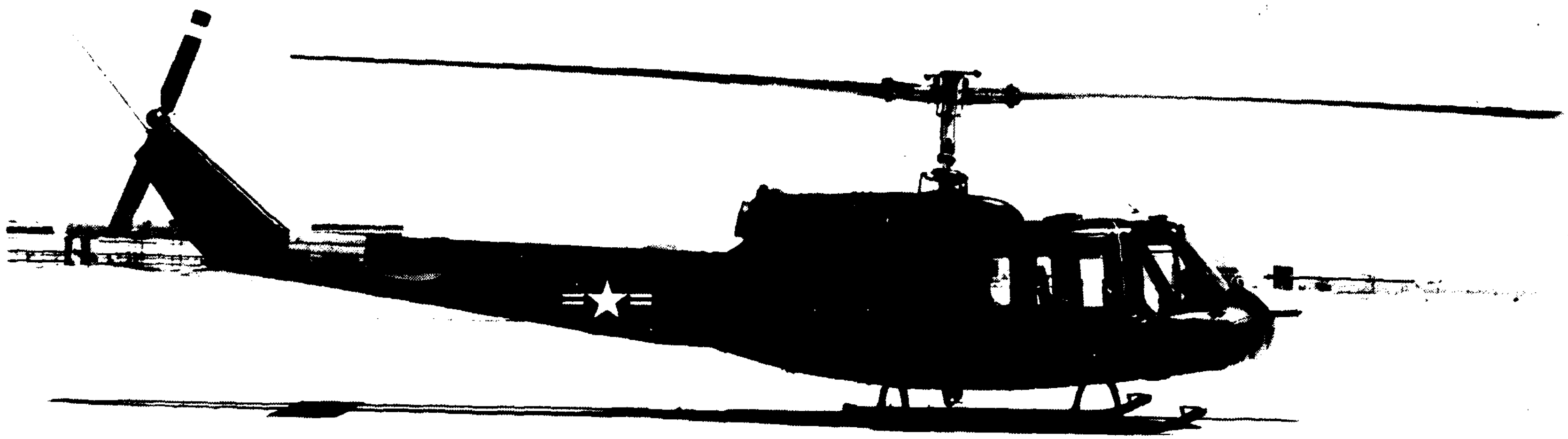


Photo B. Prototype YUH-1H S/N 60-6029

Height (to top of turning tail rotor)	14 ft, 5.5 in.
Height (to top of rotor mast)	14 ft, 0.7 in.
Fuselage around clearance (at design weight)	1 ft, 3.0 in.
Main rotor clearance (rotor tip to tail boom, static)	1 ft, 10.7 in.

Weights

Manufacturer's empty weight	4973 lb
User's empty weight	5350 lb
Design gross weight	6600 lb
Maximum gross weight	9500 lb

Main rotor

Number of blades	2
Rotor diameter (blades)	48 ft
Rotor diameter (including tracking tips)	48 ft, 3.2 in.
Blade chord (root to tip)	21 in.
Blade twist (root to tip)	-10 deg
Preconing angle	2.75 deg
Mast angle (relative to horizontal reference)	5 deg forward tilt
Control travel: (measured at center of grip)	
Collective	10.75 in. (27 deg)
Longitudinal cyclic	12.2 in. (30 deg)
Lateral cyclic	12.3 in. (30 deg)
Blade travel:	
Flapping (any direction)	± 11 deg
Collective (measured at 75% radius)	0 to 15 deg
Longitudinal cyclic	± 12 deg
Lateral cyclic	± 10 deg

Tail Rotor

Number of blades	2
Rotor diameter	8 ft, 6 in.
Blade chord (root to tip)	8.41 in.
Blade twist	0 deg
Blade airfoil (root to tip)	NACA 0015
Pedal travel	6.8 in.
Blade travel:	
Thrust to right (left yaw)	+19 deg
Thrust to left (right yaw)	-7 deg

DERIVED DATA

Main Rotor

Disc area (total swept area)	1809 ft^2
Blade area (including hub)	82 ft^2
Solidity	0.0464
Disc loading:	
6600 lb	3.65 lb/ft^2
9500 lb	5.25 lb/ft^2
Blade loading:	
6600 lb	80.5 lb/ft^2
9500 lb	115.9 lb/ft^2
Power loading (1137 shp)	
6600 lb	5.80 lb/shp
9500 lb	8.36 lb/shp
Tip speed in a hover:	
324 rotor rpm (maximum)	814.3 fps (482.1 kt)
294 rotor rpm (minimum)	738.9 fps (437.5 kt)
Maximum tip speed in forward flgiht: (V_T = 123.6 kt)	
Power on (324 rotor rpm)	1023.0 fps (605.7 kt)
Power off (339 rotor rpm)	1068.0 fps (628.1 kt)

Tail Rotor

Disk area (total swept area)	56.7 ft^2
Blade area (including hub)	5.96 ft^2
Solidity	0.105
Tip speed in a hover:	
324 rotor rpm	736 fps (436 kt)
294 rotor rpm	668 fps (395 kt)

Gear Ratios

	Ratio	Teeth
Power turbine to output shaft	3.2105:1	
Output shaft to main rotor	20.38306:1	$\frac{62}{29} \times \left(\frac{57+119}{57}\right)^2$
Output shaft to tail rotor	3.990229:1	$\frac{62}{29} \times \frac{41}{55} \times \frac{26}{27} \times \frac{39}{15}$
Tail rotor to main rotor	5.108239:1	$\frac{15}{39} \times \frac{27}{26} \times \frac{55}{41} \times \left(\frac{57+119}{57}\right)^2$
Gas producer turbine to tach pad (100% = 25,150 rpm)	5.9863:1	
Output shaft to tach pad	1.5627:1	
Tach pad to main rotor	13.28143	$\frac{27}{26} \times \frac{55}{41} \times \left(\frac{57+119}{57}\right)^2$

FLIGHT LIMITATIONS

Engine and Drive Train

Power ratings:	
Military power (30-minute limit)	1400 shp derated to 1100 shp
Maximum continuous power	1250 shp derated to 1100 shp
Torque limits:	
Maximum continuous	50 psi
Transient overtorque (not to be used intentionally) (no maintenance required)	50 to 54 psi

Transient overtorque (inspect drive train)	54 to 61 psi
Transient overtorque (replace all drive train and rotor components)	Over 61 psi
Output shaft speed:	
Maximum steady state	6600 rpm
Minimum steady state	6400 rpm
Minimum steady state below 7500 lb	6000 rpm
Maximum transient (below 91% N_1) (not to be used intentionally)	6750 rpm

Exhaust Gas Temperature

Maximum continuous	625°C
30-min limit	625° to 645°C
5-second limit for starting and acceleration	675°C
Maximum for starting and acceleration	760°C

Gas Producer

Maximum speed	25,600 rpm (101.8%)
Flight idle speed	15,900 to 17,00 rpm (63 to 68%)
Ground Idle/start speed	12,100 to 13,100 rpm (48 to 52%)

Rotor Speed

Maximum power on	324 rpm
Power on transient	331 rpm
Power off	339 rpm
Minimum power on	314 rpm
Power on less than 7500 lb	294 rpm
Power off	294 rpm

Airframe

Loading:

Design weight	6600 lb
Maximum overload weight	9500 lb
Maximum floor loading	300 lb/ft^2
Maximum cargo hook capacity	4000 lb
Maximum forward cg	Sta 130
Maximum aft cg	Sta 144
Maximum lateral cg (see ref 9 for complete cg envelope)	± 5 in.

Limit load factors:

Positive 6600 lb	+3.0 g
9500 lb	+2.1 g
Negative 6600 lb	-0.5 g
9500 lb	-0.35 g

Airspeed:

Forward flight	
Maximum (see ref 9, for complete airspeed envelope)	123.6 KTAS at 2000 ft
Sideward and rearward flight	
Maximum	30 kt

APPENDIX C. TEST INSTRUMENTATION

1. All instrumentation was calibrated and installed prior to commencing the test program and periodically recalibrated. All quantitative data obtained during this flight test program were derived from special instrumentation. A boom, mounted on and extending 92 inches forward from the nose of the aircraft, equipped with a swiveling pitot-static tube provided airspeed, altitude, angle of attack, and sideslip information. A detailed tabulation of calibrated instrumentation, equipment, and recorded data is listed below.

Pilot Station

Event switch

Copilot Station

Instrumentation controls and displays
Event switch

Instrument Panel

Airspeed
Pressure altitude
Radar altitude
Angle of sideslip
Free air temperature
Dew point temperature
Control positions
    Longitudinal
    Lateral
    Directional
    Collective
Rotor speed
Engine torque pressure
Gas generator speed
Measured gas temperature
Fuel used
Time
Record counter

Recorded data

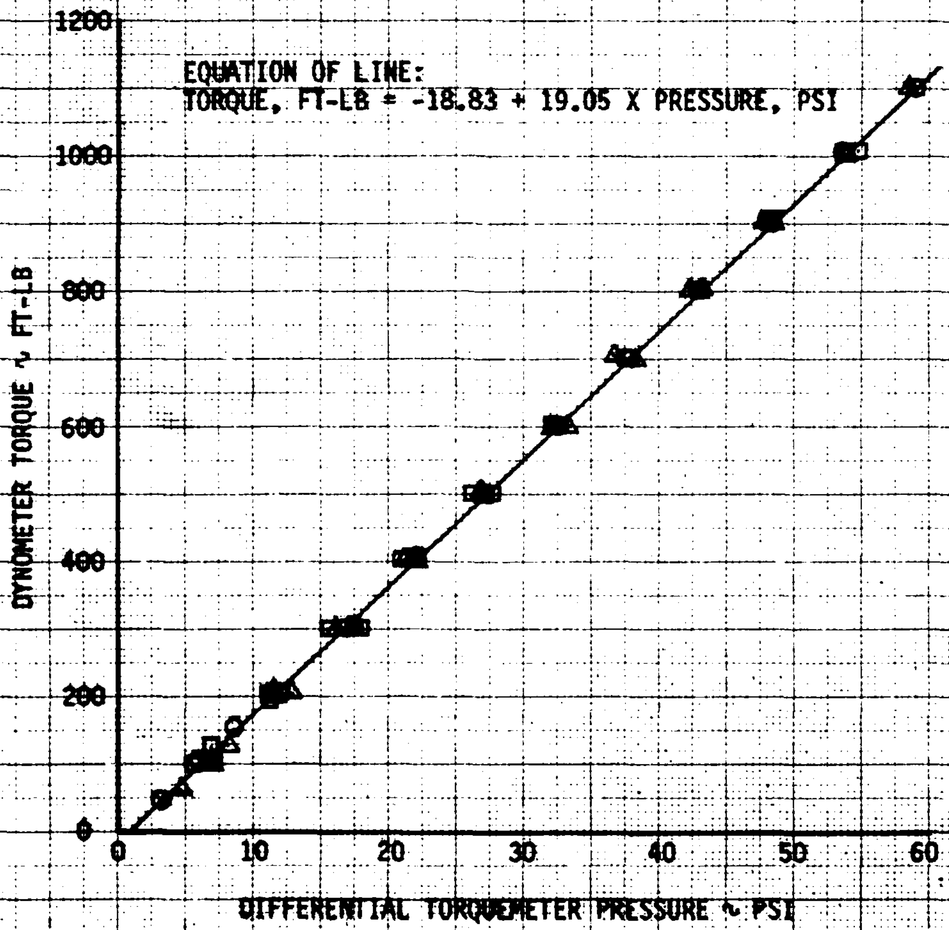
Airspeed (boom system)
Altitude (boom system)
Angle of sideslip
Angle of attack
Free air temperature
Dew point temperature

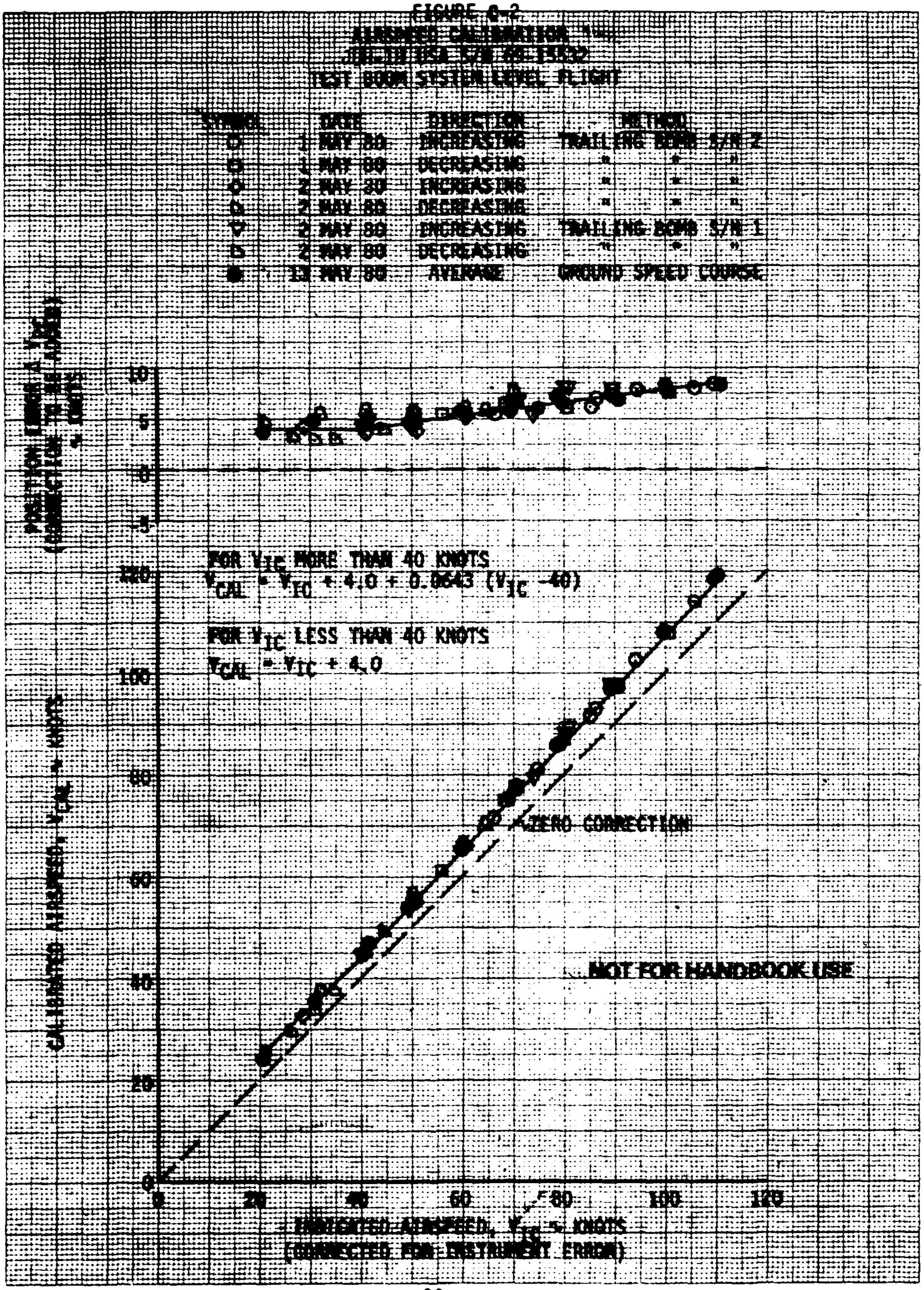
Control positions
    Longitudinal
    Lateral
    Directional
    Collective
    Engine condition (twist grip)
Stabilator Positon
Tail rotor collective blade angle
Rotor speed
Engine torque pressure
Gas generator speed
Measured gas temperature
Fuel flow
Fuel used
Fuel temperature (at flowmeter)
Pitch attitude
Pitch rate
Roll Attitude
Roll rate
Aircraft heading
Yaw rate
Center of gravity acceleration
    Vertical
    Longitudinal
    Lateral
Pilot's event
Copilot's event
Time

FIGURE D-1
ENGINE TORQUEMETER CALIBRATION
T53-L-13B S/N LE20825B

SYMBOL	OUTPUT SHAFT SPEED
△	6600 RPM
□	6300 RPM
○	6000 RPM

NOTE: DATA OBTAINED FROM CORPUS CHRISTI ARMY DEPOT TEST CELL NO. 9 LOG DATED 6 FEB 81




TEST BOOM SYSTEM LEVEL FLIGHT
SYMBOL DATE DIRECTION METHOD
1 MAY 80 INCREASING TRAILING BOMB S/N 2
1 MAY 80 DECREASING " " "
2 MAY 80 INCREASING " " "
2 MAY 80 DECREASING " " "
2 MAY 80 INCREASING TRAILING BOMB S/N 1
2 MAY 80 DECREASING " " "
13 MAY 80 AVERAGE GROUND SPEED COURSE
FOR V_{IC} MORE THAN 40 KNOTS
$V_{CAL} = V_{IC} + 4.0 + 0.0643\ (V_{IC} - 40)$
FOR V_{IC} LESS THAN 40 KNOTS
$V_{CAL} = V_{IC} + 4.0$
ZERO CORRECTION
NOT FOR HANDBOOK USE
CALIBRATED AIRSPEED, V_{CAL} ~ KNOTS
INDICATED AIRSPEED, V_{IC} ~ KNOTS
(CORRECTED FOR INSTRUMENT ERROR)

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

General

1. Conventional level flight performance test techniques were used to conduct this evaluation. Two techniques were used to achieve constant thrust coefficient throughout each test. The constant rotor speed technique required maintaining the gross weight to air density ratio constant. The referred rotor speed technique required maintaining constant the ratio of rotor speed to square root of static air temperature ratio, and the ratio of gross weight to static air pressure. Both techniques required that altitude be increased for each data point as fuel was consummed. Those tests using the referred rotor speed technique are indicated on the appendix E data figures by noting the average referred rotor speed. All tests were conducted in nonturbulent atmospheric conditions to preclude uncontrolled disturbances influencing the results. Ten second records were taken at the beginning and end of approximately 1 minute stable points.

Weight and Balance

2. Prior to testing, the aircraft empty weight (including full oil and trapped fuel) and horizontal center-of-gravity location were determined with calibrated scales. Vertical cg was determined by suspending the helicopter from the top of the rotor mast and measuring the resulting attitude. Vertical cg was then calculated from the intersection of the suspension point with the horizontal CG. The empty weight was 5818 lb. The center-of-gravity was: FS 143.36, BL - 0.02, WL 64.0.

3. A manometer-type external sight gauge was calibrated and used to determine fuel volume. Fuel specific gravity was measured with a hydrometer. The fuel loading for each test flight was determined both prior to engine start and following engine shutdown. Fuel used in flight was recorded by a test fuel-used system and compared with the pre- and post flight sight gauge reading. Fuel cg versus fuel volume contained in the fuel cell (208.5 gallon capacity) had been previously determined. This calibration was used to calculate aircraft cg for each test point. Aircraft gross weight and cg were also controlled by ballast installed at various locations in the aircraft.

Level Flight Performance and Specific Range

4. The helicopter level flight performance data were generalized by the following nondimensional coefficients:

a. Coefficient of power (C_P):

$$C_P = \frac{SHP\ (550)}{\rho\ A(\Omega R)^3} = 8.05518\ SHP/\sigma N^3 = 8.05518\ (SHP/\delta\sqrt{\theta})/(N/\sqrt{\theta})^3 \qquad (1)$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{W}{\rho A(\Omega R)^2} = 0.0368089\ W/\sigma N^2 = 0.0368089\ (W/\delta)/(N/\sqrt{\theta})^2 \qquad (2)$$

c. Advance ratio (μ):

$$\mu = \frac{1.68781\ V_T}{\Omega R} = 0.671558\ V_T/N = 0.671558\ (V_T/\sqrt{\theta})/(N/\sqrt{\theta}) \qquad (3)$$

d. Advancing blade tip Mach number (M_{tip}):

$$M_{tip} = \frac{1.68781\ V_T + (\Omega R)}{a} = 0.00225113\ (N + 0.671558 V_T)/\sqrt{\theta} = 0.00225113\ (1+\mu)\ N/\sqrt{\theta} \qquad (4)$$

Where:

SHP = Engine output shaft horsepower
550 = Conversion factor (ft-lb/sec/shp)
ρ = Air density (slug/ft^3)
ρ_o = Standard day sea level density (.00237689 slugs/ft^3)
σ = Air density ratio = ρ/ρ_o
δ = Ambient pressure ratio (test point to sea level standard)
A = Main rotor disc area (ft^2) = 1809.5
Ω = Main rotor angular velocity (radian/sec) = $\frac{2\pi}{60}$ x N
N = Main rotor angular velocity (rpm)
R = Main rotor radius (ft) = 24.0
W = Gross weight (lb)
$\theta = (T_a + 273.15)/288.15$
T_a = Ambient air temperature (°C)
1.68781 = Conversion factor (ft/sec/knot)
V_T = True airspeed (knot)
a = Speed of sound (ft/sec) = 1116.45 $\sqrt{\theta}$
a_o = Speed of sound at sea level standard (ft/sec) = 1116.45

5. Test-day (measured) level flight power was corrected to average flight conditions for each set of speed-power data by

assuming the test-day dimensionless parameters C_{P_t}, C_{T_t}, and μ_t are identical to $C_{P_{avg}}$, $C_{T_{avg}}$, and μ_{avg}, respectively. From equation 1, the following relationship can be derived:

$$SHP_{avg} = SHP_t \left(\frac{\rho_{avg}}{\rho_t}\right) \qquad (5)$$

Where:

Subscript t = test day (measured for each data point)
Subscript avg = average over each set of speed power data

6. Test specific range was calculated using level flight performance data and the measured fuel flow.

$$SR = \frac{V_T}{W_f} \qquad (6)$$

Where:

SR = Specific range (nautical air miles per pound of fuel)
V_T = True airspeed (knot)
W_f = Fuel flow (lb/hr)

Shaft Horsepower Required

7. The engine output shaft torque was determined from the engine manufacturer's torque system. The relationship of measured torque pressure (psi) to engine output shaft torque (ft-lb) was determined from the engine test cell calibration is shown in figure C-1, appendix C. The output shp was determined from the engine output shaft torque and rotational speed by equation (7).

$$SHP = \frac{2\pi \times N_P \times Q}{33,000} = \frac{N_P \times Q}{5252.113} \qquad (7)$$

Where :

N_P = Engine output shaft rotational speed (rpm)
Q = Engine output shaft torque (ft-lb)
33,000 = Conversion factor (ft-lb/min/shp)

Indicated Airspeed and Pressure Altitude

8. Total pressure, static pressure, and total temperature were measured from sensors mounted on a flight test boom installed on the nose of the aircraft. The output signals were recorded on magnetic tape, and the following expressions were used to calculate the parameters:

a. Indicated airspeed corrected for instrument error (V_{ic}):

$$V_{ic} = a_o \{5 [(qc_{ic}/Pa_o + 1)^{2/7} - 1]\}^{1/2} \qquad (8)$$

b. Indicated pressure altitude corrected for instrument error (HP_{ic}):

$$HP_{ic} = 145,442.2 [1-(Pa_{ic}/Pa_o)^{0.1902632}] \qquad (9)$$

Where:

V_{ic} = Indicated airspeed corrected for instrument error (kt)
a_o = Speed of sound at standard day, sea level = 661.479 kt
qc_{ic} = Indicated differential pressure corrected for instrument error (in. Hg)
Pa_o = Atmospheric pressure at standard day, sea level = 29.92125 in. Hg
HP_{ic} = Indicated pressure altitude corrected for instrument error (ft)
(Pa_{ic}) = Indicated static pressure corrected for instrument error (in. Hg)

Airspeed Calibration

9. The boom pitot-static system was calibrated using the trailing bomb method to determine the airspeed position error. This calibration is shown in figure C-2, appendix C. Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) using instrument (ΔV_{ic}) and position (ΔV_{pc}) error corrections.

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \qquad (10)$$

True Airspeed

10. True airspeed was computed using the following relationship:

$$V_T = a \{ 5 [(qc/Pa + 1)^{2/7} - 1]\}^{.5} \qquad (11)$$

Where:

a = Speed of sound (knots)

qc = Corrected differential pressure (in.Hg)

Pa = Corrected static pressure (in.Hg)

Corrected Pressure Altitude and Altitude Position Error

11. HP_{ic} was corrected for position error by using ΔV_{pc}. The assumption was made that position error (ΔP_p) was produced entirely at the static source. Since both airspeed and altitude systems utilize the same static source, the following relationships were used:

$$qc = Pa_o \{[.2(V_{cal}/a_o)^2 + 1]^{3.5} - 1\} \quad (12)$$

$$\Delta P_p = qc - qc_{ic} \quad (13)$$

$$Pa = Pa_{ic} - \Delta P_p \quad (14)$$

$$H_p = 145{,}442.2\ [1-(Pa/Pa_o)^{.1902632}] \quad (15)$$

Where:

qc = Differential pressure corrected for position and instrument error (in. Hg)
qc_{ic} = Indicated differential pressure corrected for instrument error (in. Hg)
V_{cal} = Calibrated airspeed (knots)
a_o = Speed of sound at standard day sea level = 661.479 knots
Pa_o = Atmospheric pressure at standard day, sea level = 29.92125 in. Hg
ΔP_p = Pressure position error (in. Hg)
Pa = Atmospheric pressure at corrected altitude (in. Hg)
Pa_{ic} = Indicated static pressure corrected for instrument error (in. Hg)
H_p = Corrected pressure altitude (ft)

Static Temperature

12. Static temperature was obtained by correcting the measured total temperature for temperature rise due to compressibility. The following relationships were used:

$$T_{tic} = OAT_{ic} + 273.15 \quad (16)$$

$$Ta = T_{tic}/\{1 + K_t\,[(qc/P_A + 1)^{2/7} - 1]\} \quad (17)$$

$$SAT = Ta - 273.15$$

Where:

T_{tic} = Instrument corrected measured air temperature (°K)

Ta = Static air temperature (°K)

K_T = Temperature probe recovery factor = 0.97 (from previous tests)

P_A = Corrected static air pressure (in.Hg)

SAT = Corrected static air temperature (°C)

<u>Humidity</u>

13. For tests above 0°C where humidity effects could have a significant effect on air density and the speed of sound, dew point temperature was measured and humidity correct ons made. The following relationships were used:

$$P_V = e^{(69.5137 - 7246.6/T + .0057449T - 8.247 \ln T)} \quad (19)$$

Where:

P_V is the vapor pressure (in.Hg)

e is the base for Napierian logarithm = 2.71828···

T is dew point temperature (T_d, °K) for existing vapor pressure, P_{VD} (in.Hg) or T is static air temperature (T_a, °K) for saturation vapor pressure P_{VS} (in.Hg).

The density correction factor, K_d, is:

$$K_d = 1 - 0.3779\, P_{VD}/P_A \quad (20)$$

The mixing ratio, M_R is:

$$M_R = 0.62201\, P_{VD}/(P_A - P_{VD}) \quad (21)$$

And the sound speed correction factor, K_a is:

$$K_a = [(1 + 1.8375\ M_R)/K_d\ (1 + 1.9357\ M_R)]^{.5} \qquad (22)$$

Air Density

14. Air density, ρ, was computed as follows:

$$\rho = 0.0228901\ K_d\ P_a/T_a\ (\text{slugs/ft}^3) \qquad (23)$$

for cold temperatures or where dew point was not measured $K_d = K_a = 1.0$.

Sound Speed

15. Sound speed, a, was computed as follows:

$$a(\text{Kts}) = 38.96785\ K_a\ \sqrt{T_a},\ ^\circ K \qquad (24)$$

Drag

16. The following relationships were used to compute differential drag in terms of equivalent flat plate area (EFPA):

$$\Delta F_e = \frac{228.782\ \Delta SHP}{\rho V^3} = 2\ A\ \Delta C_p/\mu^3$$

Where:

ΔFe = differential equivalent flat plate area (ft^2)
ΔSHP = differential engine shaft horsepower (horsepower) (note: drag area based on wind tunnel tests (thrust horsepower) would be smaller).
ρ = air density ($slugs/ft^3$)
V = true airspeed (knots)
A = main rotor disk area (ft^2)
ΔCp = differential power coefficient (based on engine power)
μ = advance ratio

Drag comparisons were made to reference 7 data. No corrections were made for airspeed boom drag (similar booms on both aircraft). Past attempts to determine boom drag have been unsuccessful because of the loss of sideslip reference.

17. The optimum cruise analysis used the following procedure:

a. Derive speed-powers from level flight summary data for the full range of weights, altitudes, temperatures, and rotor speeds.

b. Derive specific range for a, using engine model specification.

c. Find optimum airspeed and maximum specific range for each b.

d. Repeat c for limit airspeed (V_{NE}).

e. Repeat c for continuous power limited airspeed (V_H).

f. Determine regions of weight, altitude, temperature, and rotor speed for applicable (lowest) airspeed (c, d, or e).

g. For each weight, temperature, and rotor speed; plot specific range versus altitude. Fair and determine altitude for maximum specific range (optimum altitude).

h. For each rotor speed, fair optimum altitude versus weight at each temperature.

i. For each rotor speed, fair optimum altitude versus temperature at each weight.

j. Crossplot h and i for continuous variation with weight and temperature.

k. Repeat h through j for specific range at optimum altitude.

l. For each weight-temperature condition plot specific range and optimum altitude versus rotor speed.

m. Check j, k, and l against test data.

n. Iterate to smooth and minimize errors.

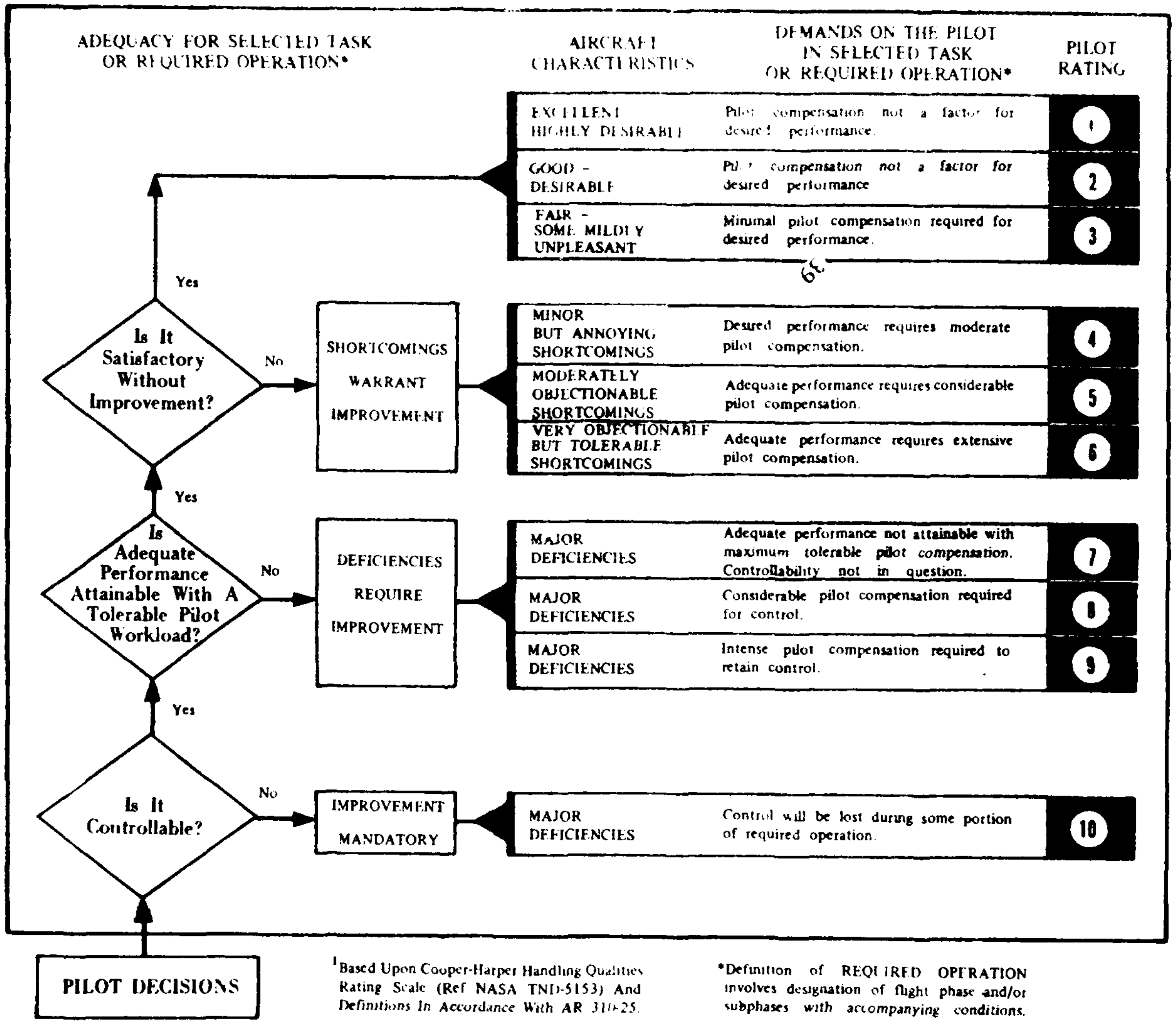
o. Convert specific range to fuel flow.

This process was very lengthy and complex. An effort is being made to develop a simplified helicopter power required analysis methodology.

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1, 2, 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4, 5, 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7, 8, 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure D-1. Vibration Rating Scale



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure D-2. Handling Qualities Rating Scale

APPENDIX E. GRAPHICAL TEST DATA

	Figure
Level Flight Performance $C_T = 26 \times 10^{-4}$	1
Level Flight Performance $C_T = 30$ to 32×10^{-4}	2 - 16
Level Flight Performance $C_T = 34$ to 36×10^{-4}	17 - 29
Level Flight Performance IR Suppressor Installed	30
Level Flight Performance Referred Rotor Speed Comparison	31
Referred Engine Characteristics	32 - 34

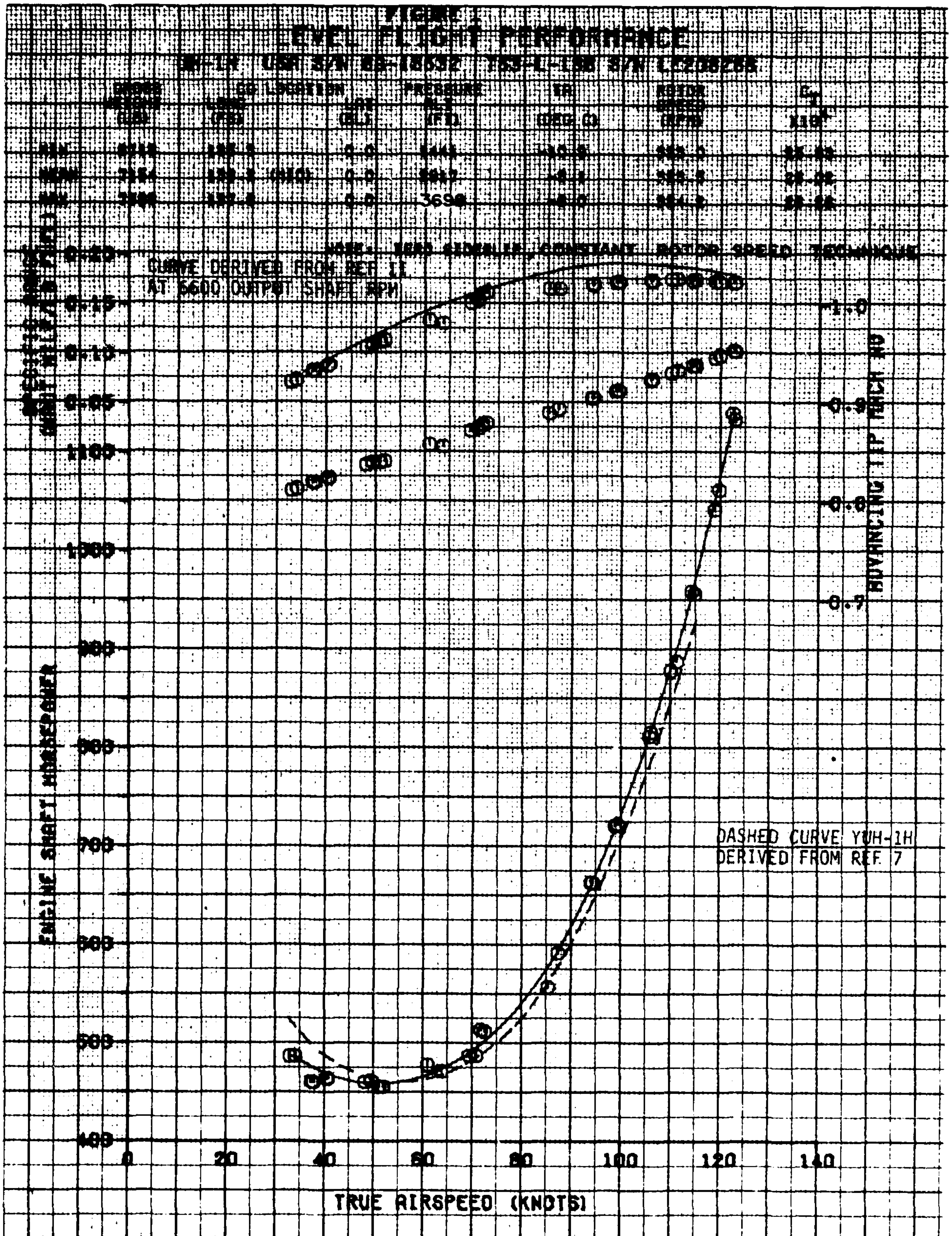
Abbreviations Used on Figures

C_T: Thrust coefficient (non-dimensional)

T_A: Air temperature (°C)

T_D: Dew point temperature (°C)

Data are arranged in increasing order of referred rotor speed within each thrust coefficient group. Average referred rotor speed is noted on each figure where the referred rotor speed test technique was used. For those not so noted, the constant rotor speed technique was used. The minimum or maximum tabulated values did not necessarily occur on the same data point. They are included only to indicate the maximum range of the parameter for the entire test. Tabulated data are presented to the full precision of the measurement.


CURVE DERIVED FROM REF 11
AT 5600 OUTPUT SHAFT RPM
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)
0
20
40
60
80
100
120
140
1.0
0.9
0.8
0.7

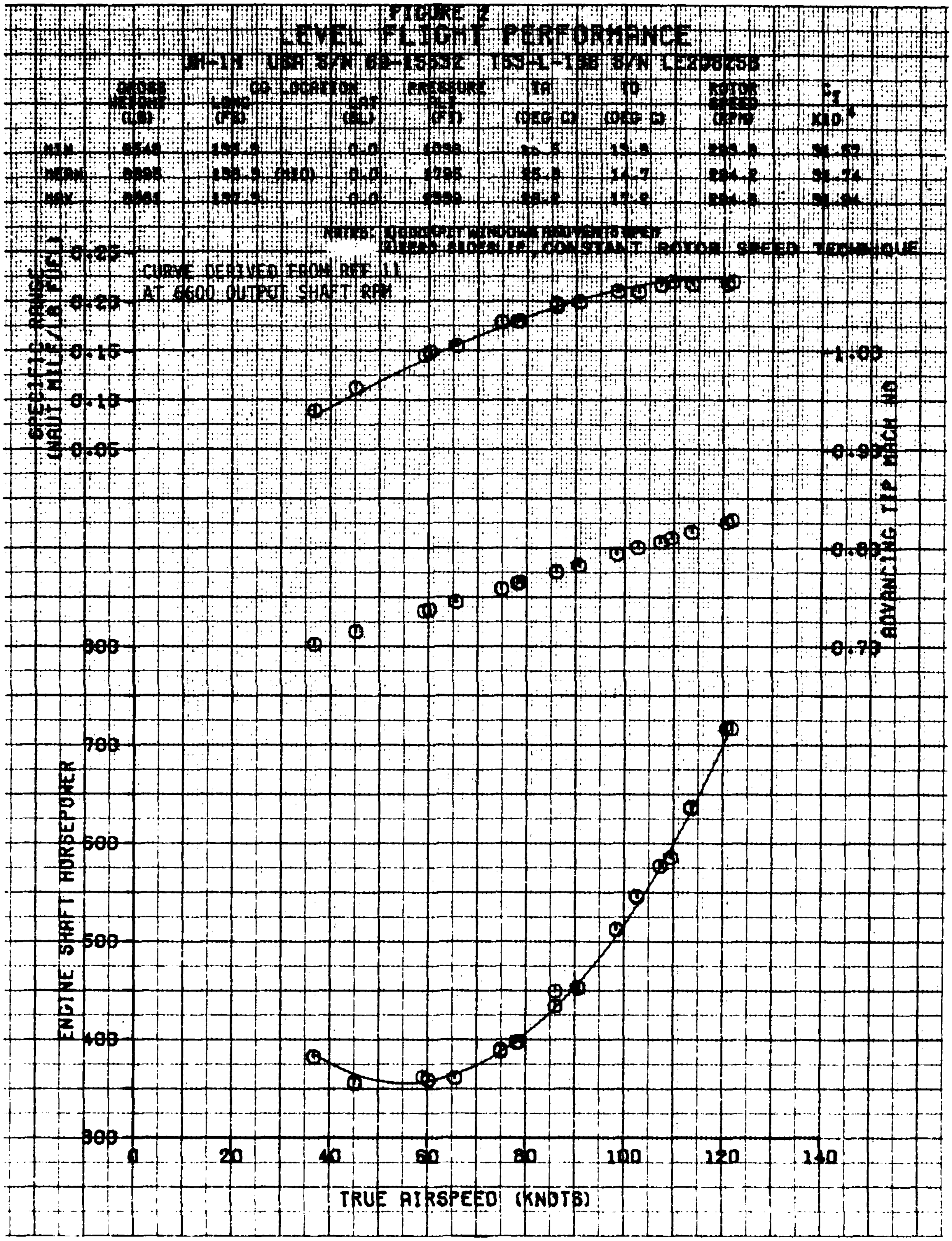


FIGURE 2
LEVEL FLIGHT PERFORMANCE
GROSS WEIGHT (LB)
CG LOCATION
LONG (FS)
LAT (BL)
PRESSURE ALT (FT)
ROTOR SPEED (RPM)
MIN
MEAN
MAX
0.0
CONSTANT ROTOR SPEED TECHNIQUE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE (NAUT MILE/LB FUEL)
0.25
0.20
0.15
0.10
0.05
ADVANCING TIP MACH NO
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER
800
700
600
500
400
300
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

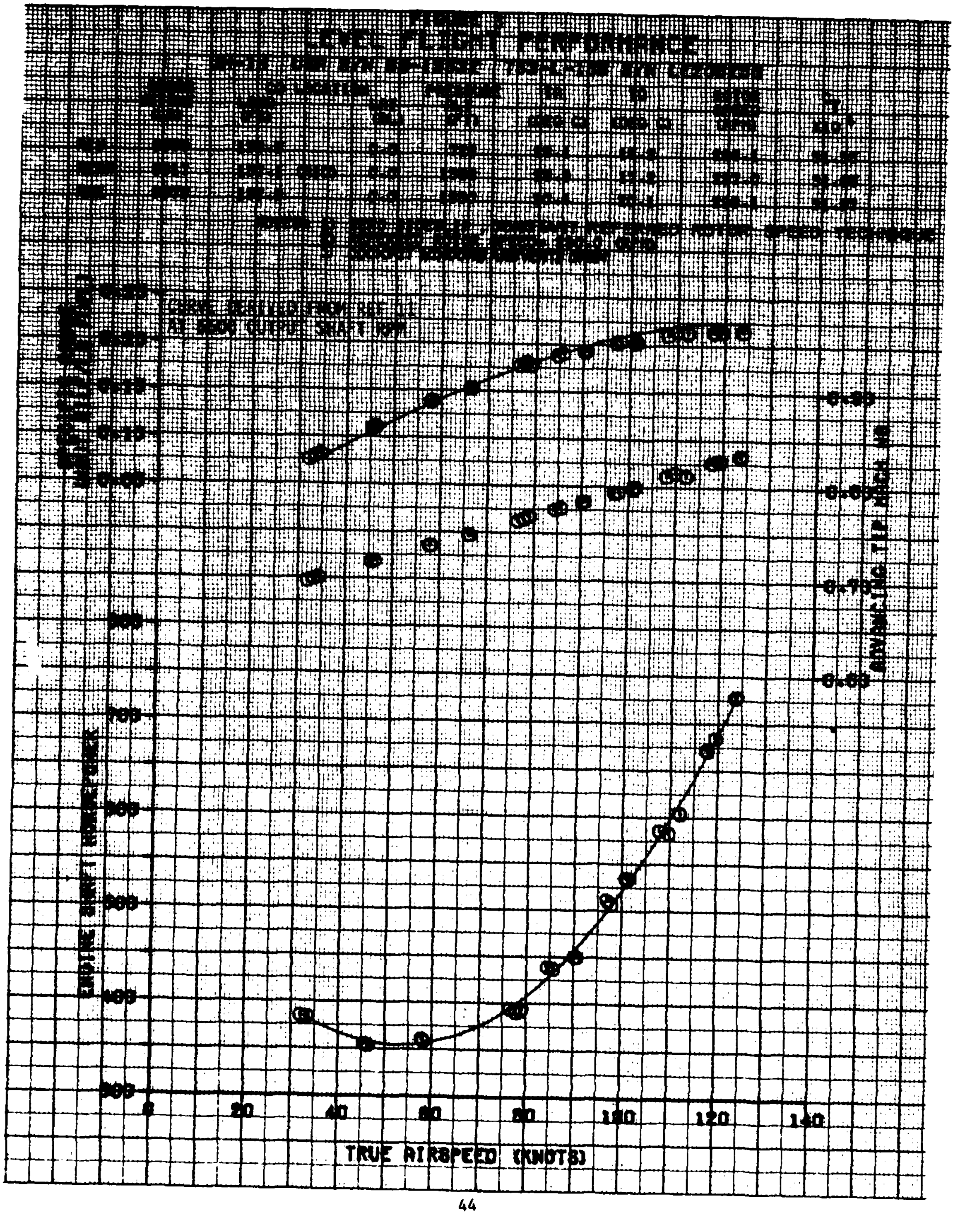

LEVEL FLIGHT PERFORMANCE
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

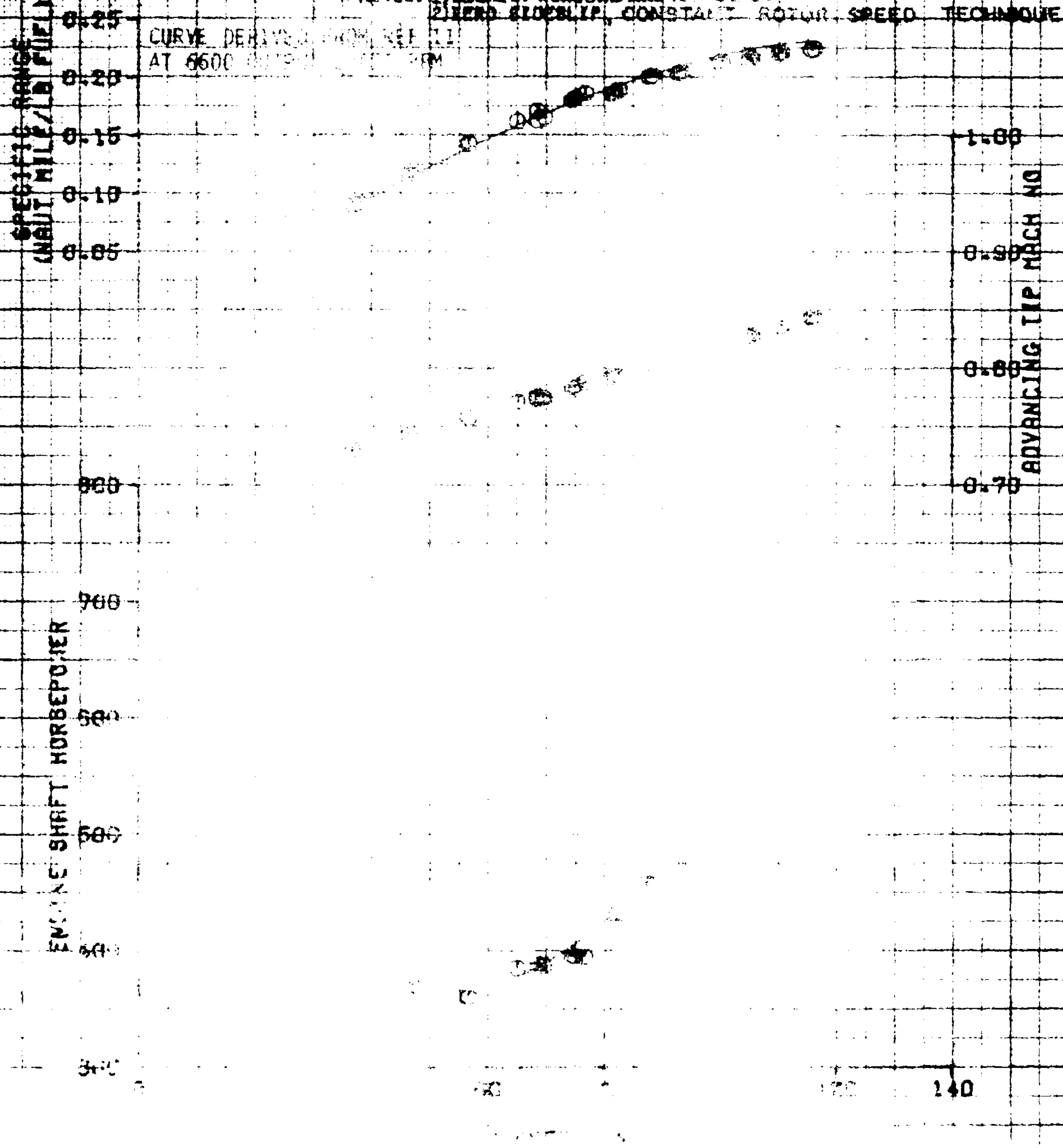
FIGURE 8

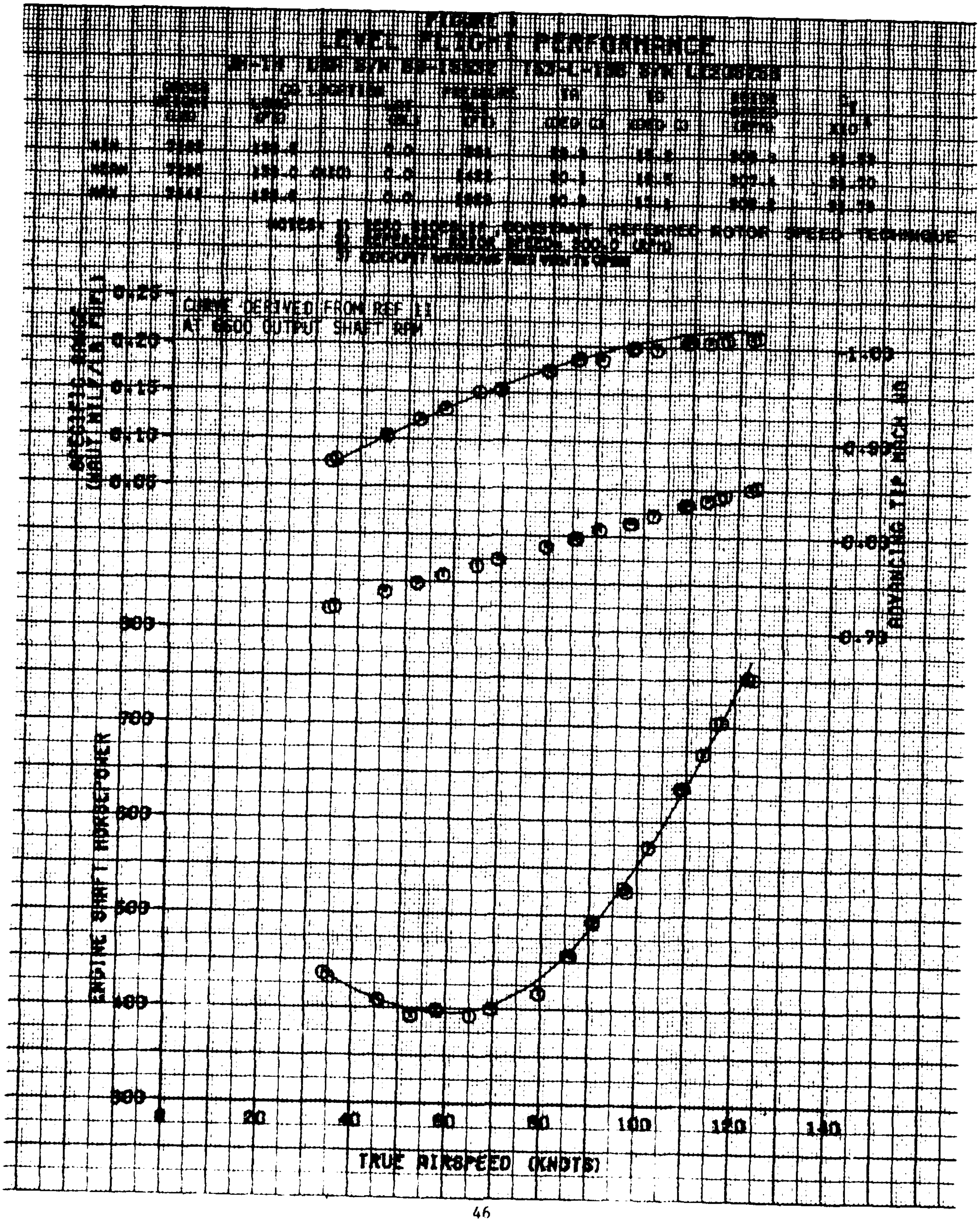
LEVEL FLIGHT PERFORMANCE

UH-1N USA S/N 69-15532 T53-L-13B S/N LE20625B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TC (DEG C)	ROTOR SPEED (RPM)	C_T $\times 10^4$
MIN	[illegible]	138.5		0.0	[illegible]	[illegible]	[illegible]	303.3	[illegible]
AVG	[illegible]	137.[illegible]	(MID)	0.0	[illegible]	[illegible]	11.8	304.1	[illegible]
MAX	[illegible]	13[illegible]		0.0	[illegible]	[illegible]	[illegible]	304.5	[illegible]

NOTES: 1) COCKPIT WINDOWS AND VENTS OPEN
2) ZERO SIDESLIP, CONSTANT ROTOR SPEED TECHNIQUE




LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)
0.25
0.20
0.15
0.10
0.05
1.00
0.90
0.80
0.70
800
700
600
500
400
300
0
20
40
60
80
100
120
140

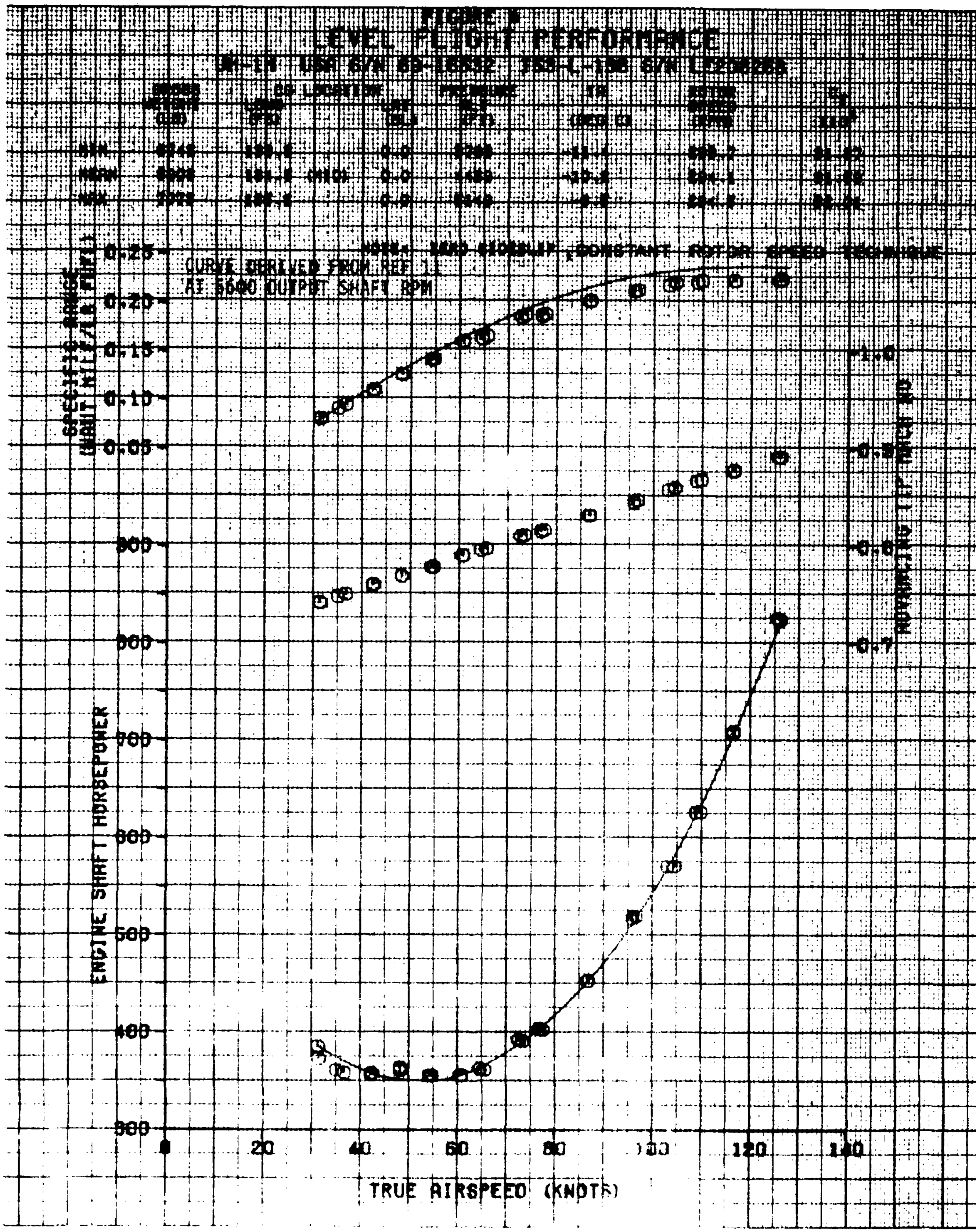
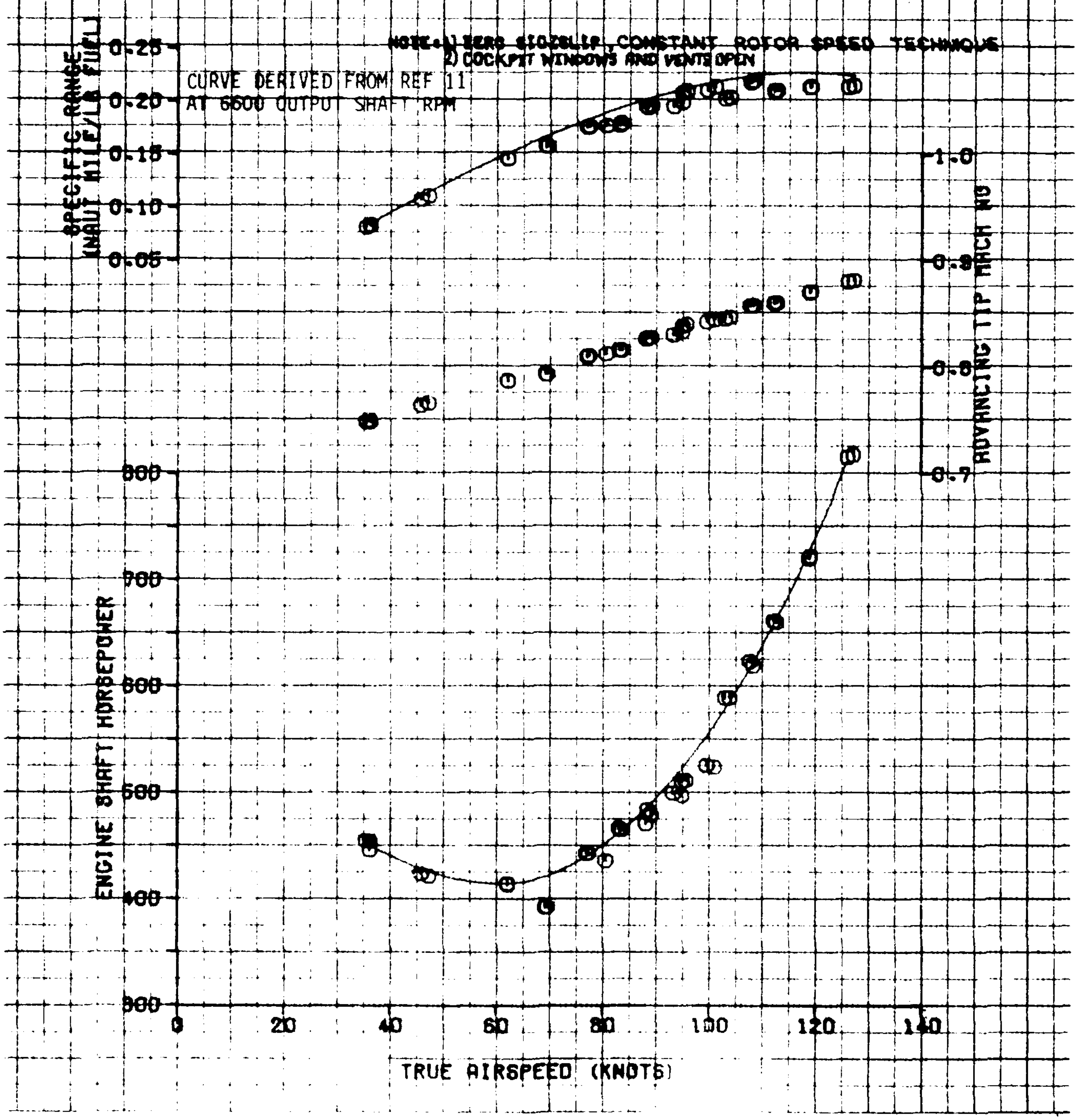
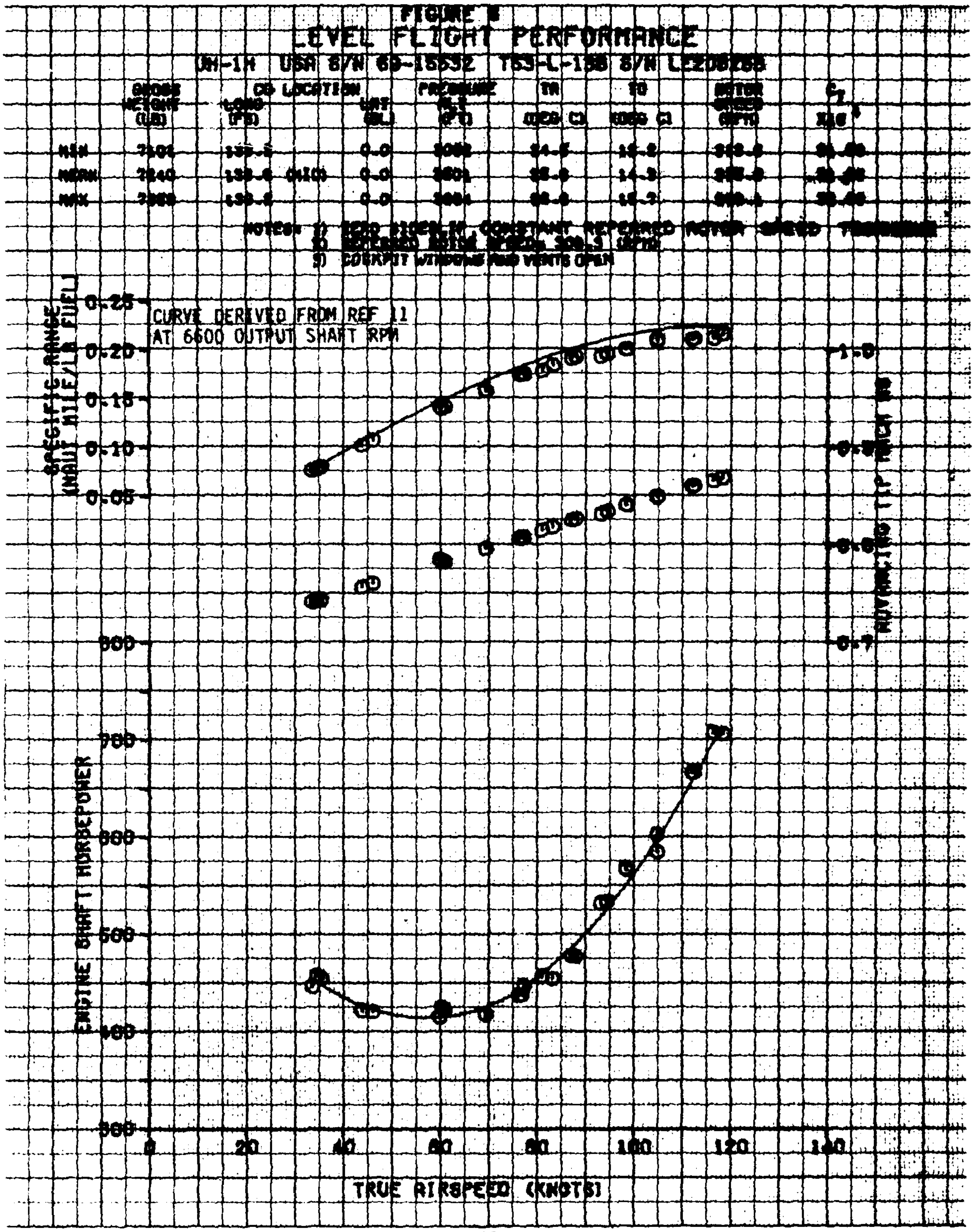
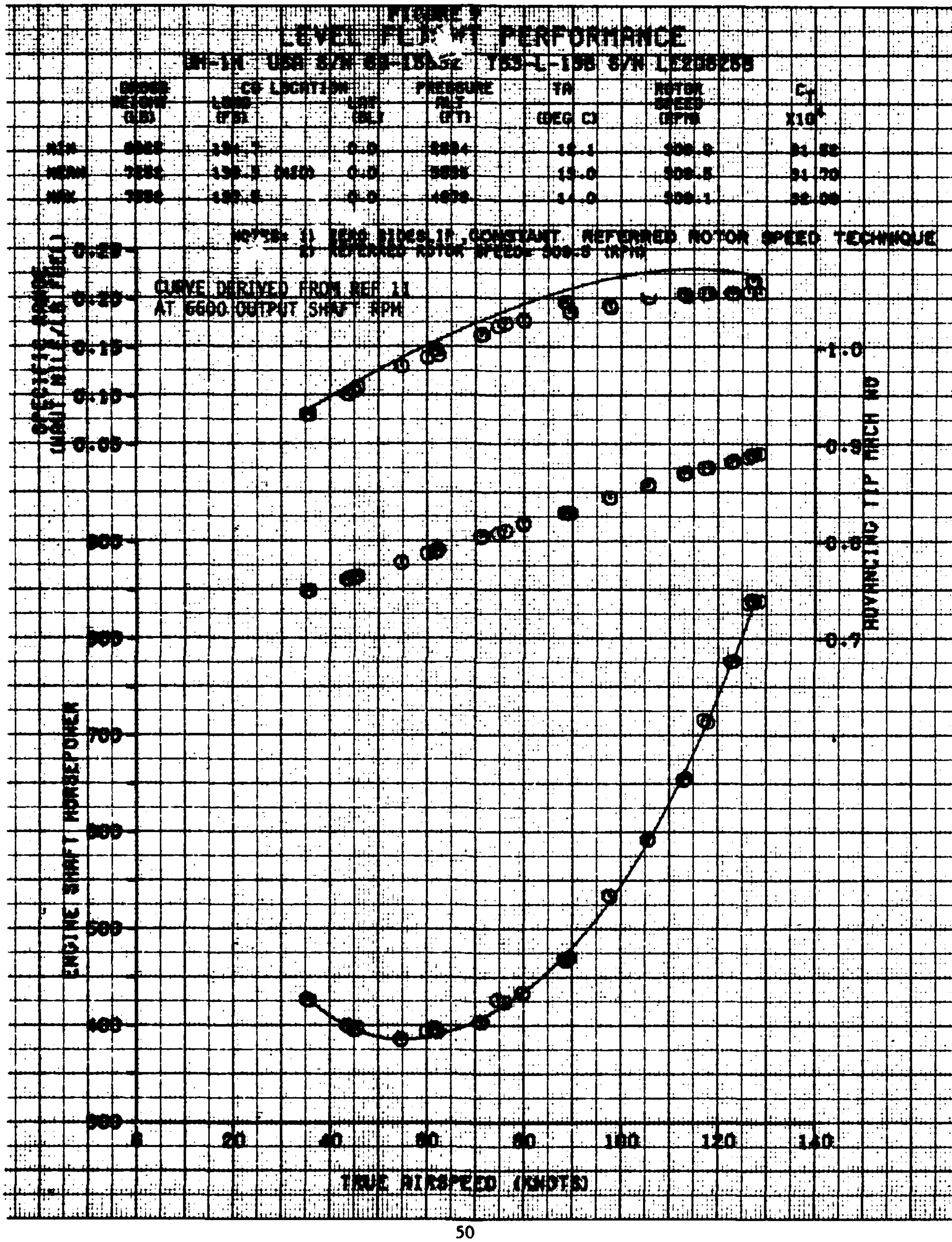

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE
ENGINE SHAFT HORSEPOWER
TRUE AIRSPEED (KNOTS)
0.25
0.20
0.15
0.10
0.05
800
900
800
700
600
500
400
300
0
20
40
60
80
100
120
140
-1.0
-0.5
-0.6
-0.7

FIGURE 7
LEVEL FLIGHT PERFORMANCE
UH-1H USA S/N 68-15532 T53-L-13B S/N LE208258

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TD (DEG C)	ROTOR SPEED (RPM)	C_T X10[4]
MIN	7030	130.0		0.0	3410	11.1	10.5	313.3	31.57
MEAN	7230	130.7	(MID)	0.0	3900	13.7	13.1	313.9	31.71
MAX	7430	131.5		0.0	4410	15.9	15.3	314.4	31.82




LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
0.25
0.20
0.15
0.10
0.05
800
700
600
500
400
300
0
20
40
60
80
100
120
140
1.0
0.7
ENGINE SHAFT HORSEPOWER
TRUE AIRSPEED (KNOTS)



PERFORMANCE
CG LOCATION
PRESSURE
(DEG C)
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
REFERRED ROTOR SPEED TECHNIQUE
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)
0
20
40
60
80
100
120
140
700
600
500
400
1.0
0.9
0.8
0.7

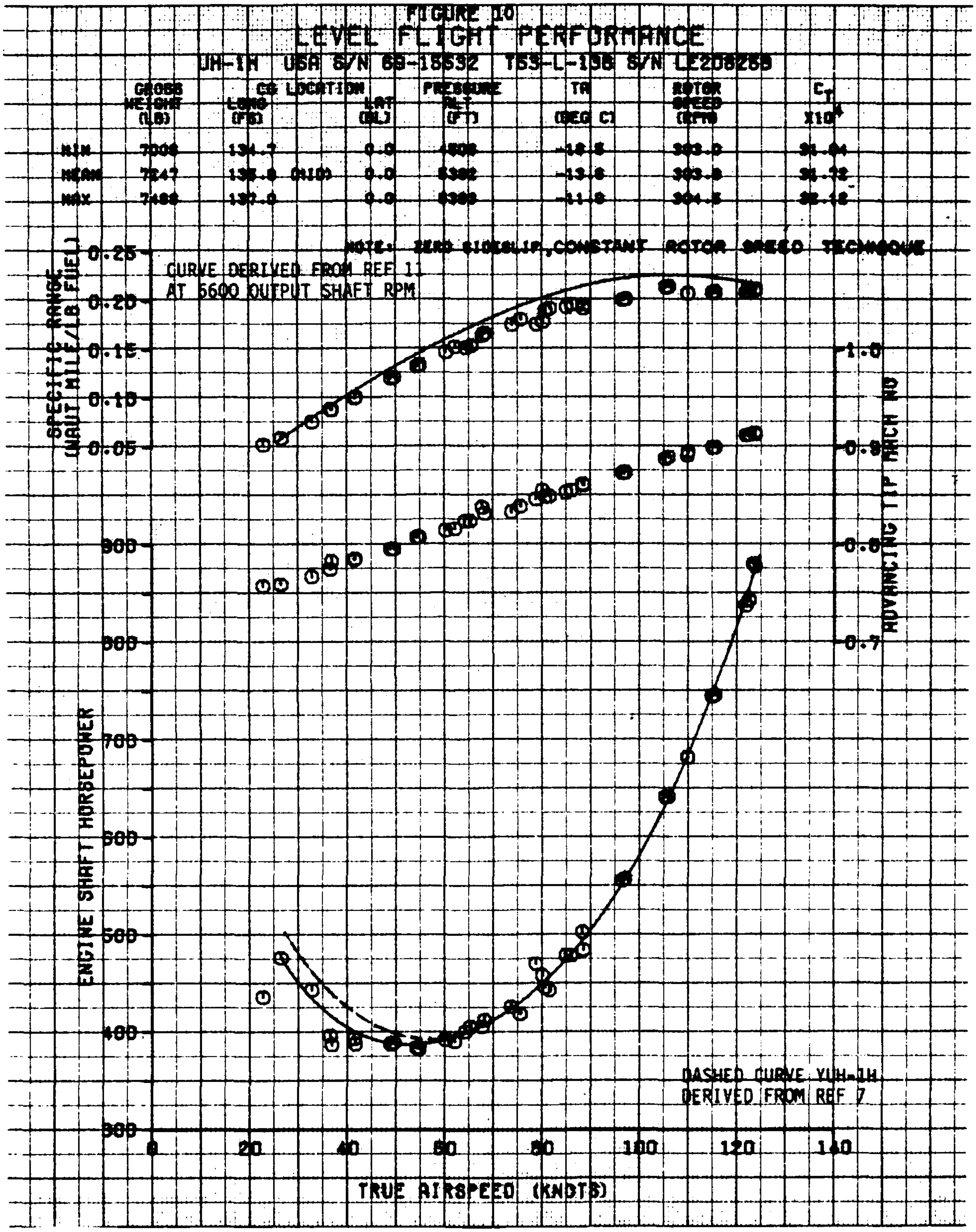
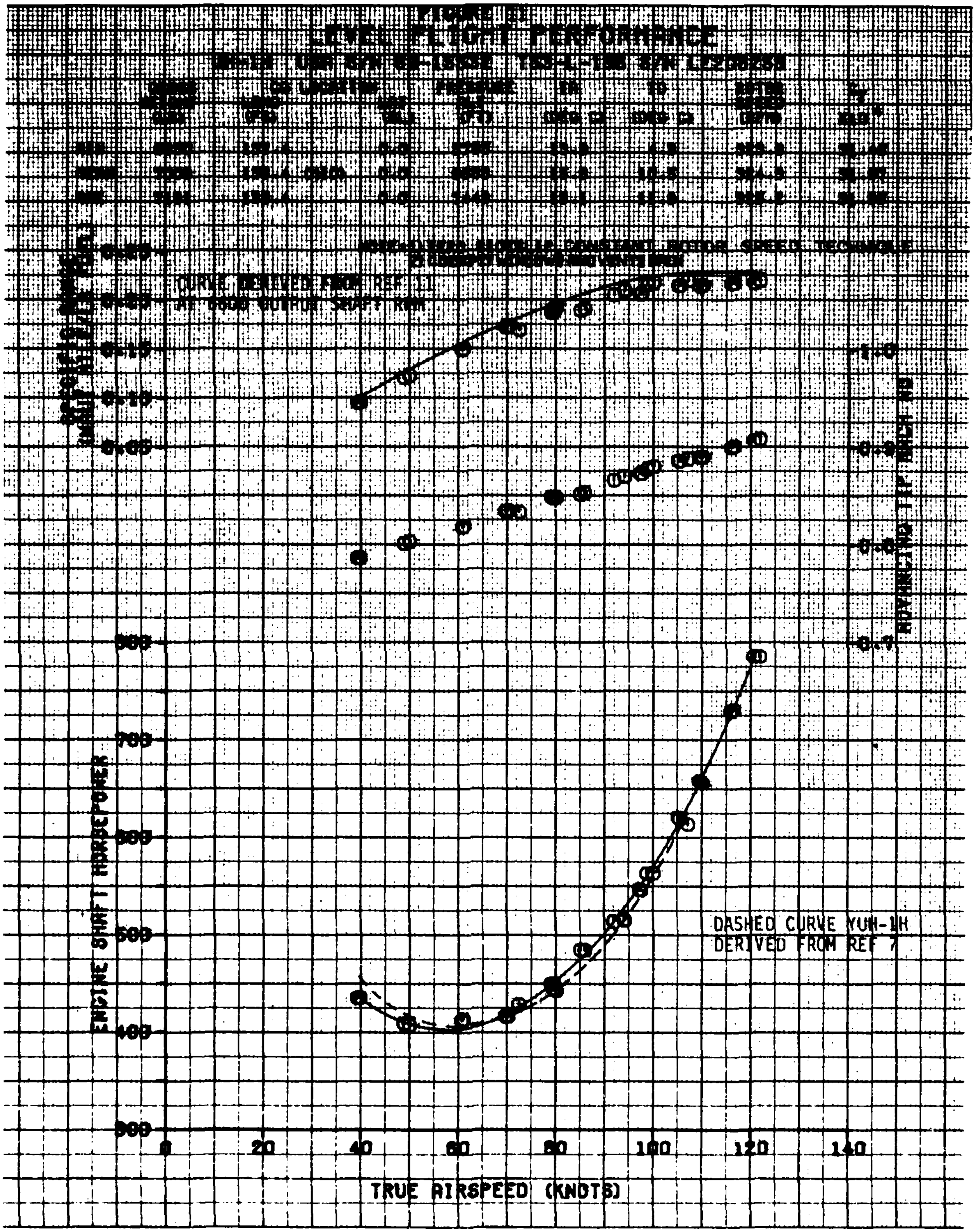


FIGURE 10
LEVEL FLIGHT PERFORMANCE
UH-1H USA S/N 68-15532 T53-L-13B S/N LE208288
GROSS WEIGHT (LB)
CG LOCATION
LONG (FS)
LAT (BL)
PRESSURE ALT (FT)
TR (DEG C)
ROTOR SPEED (RPM)
C_T X10[4]
MIN
MEAN
MAX
NOTE: ZERO SIDESLIP, CONSTANT ROTOR SPEED TECHNIQUE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE (NAUT MILE/LB FUEL)
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
DASHED CURVE YUH-1H
DERIVED FROM REF 7
TRUE AIRSPEED (KNOTS)



LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
800
700
600
500
400
300
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

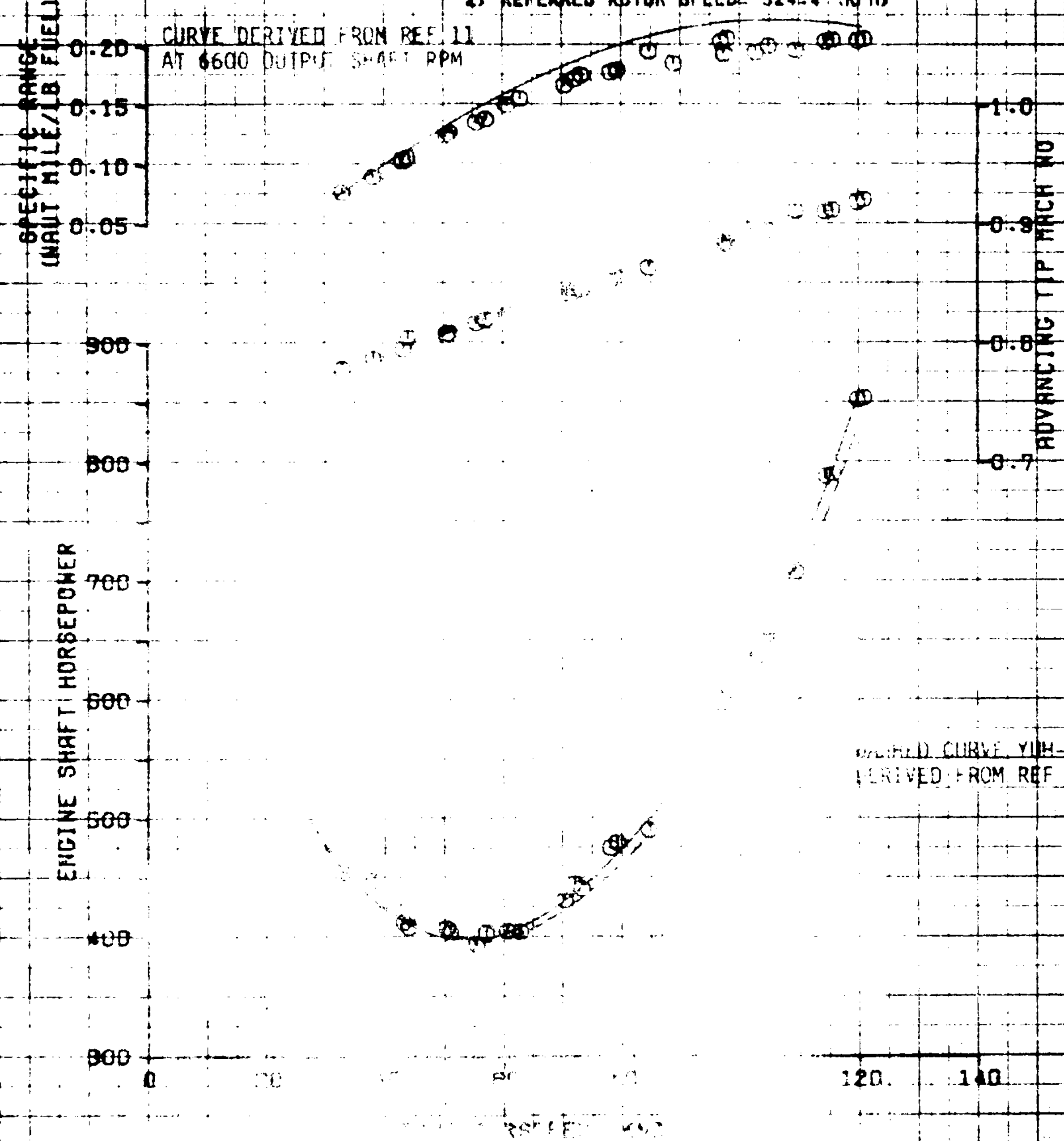
FIGURE 12

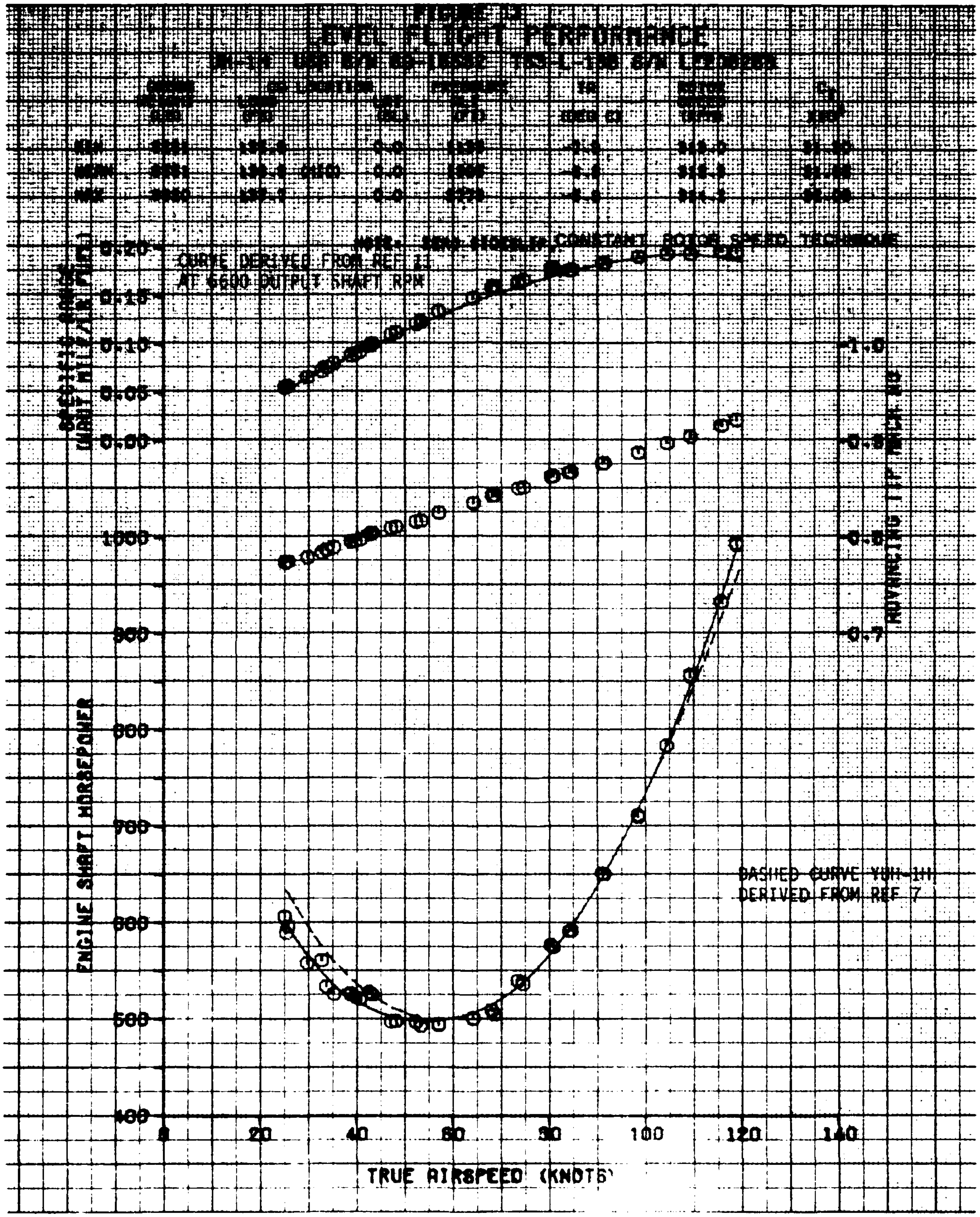
LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 69-15532 T53-L-13B S/N LE20828B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	ROTOR SPEED (RPM)	C_T $\times 10^4$
MIN	[illegible]	134.7		0.0	[illegible]	-17.0	307.8	30.28
MEAN	7245	135.8	(MID)	0.0	5337	-12.0	308.8	30.64
MAX	7500	137.2		0.0	[illegible]	-10.2	310.0	31.20

NOTES: 1) ZERO SIDESLIP, CONSTANT REFERRED ROTOR SPEED TECH
2) REFERRED ROTOR SPEED= 324.4 (RPM)





LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
CONSTANT ROTOR SPEED TECHNIQUE
0.20
0.15
0.10
0.05
0.00
1000
900
800
700
600
500
400
1.0
0.9
0.8
0.7
ENGINE SHAFT HORSEPOWER
DASHED CURVE YUH-1H
DERIVED FROM REF 7
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

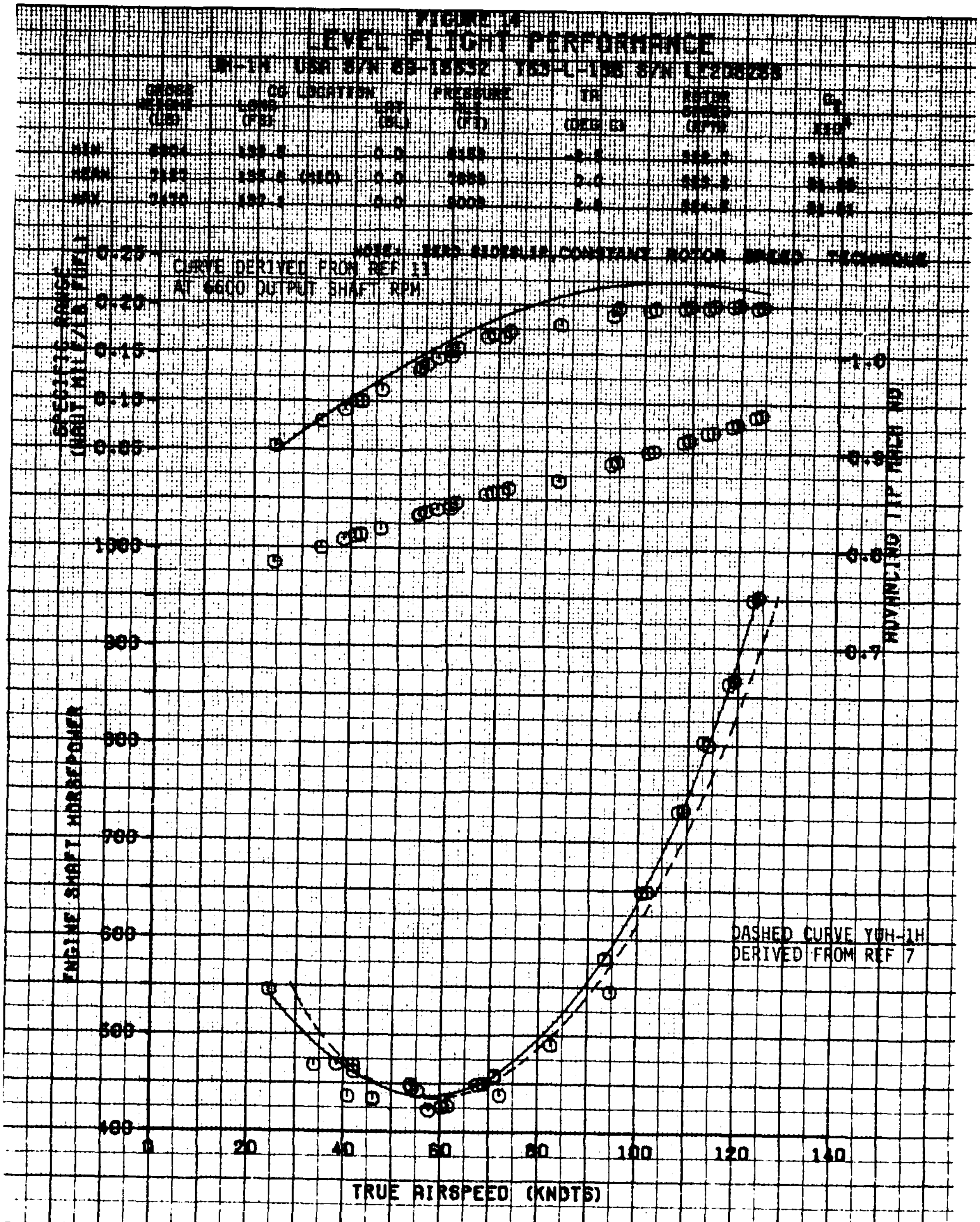
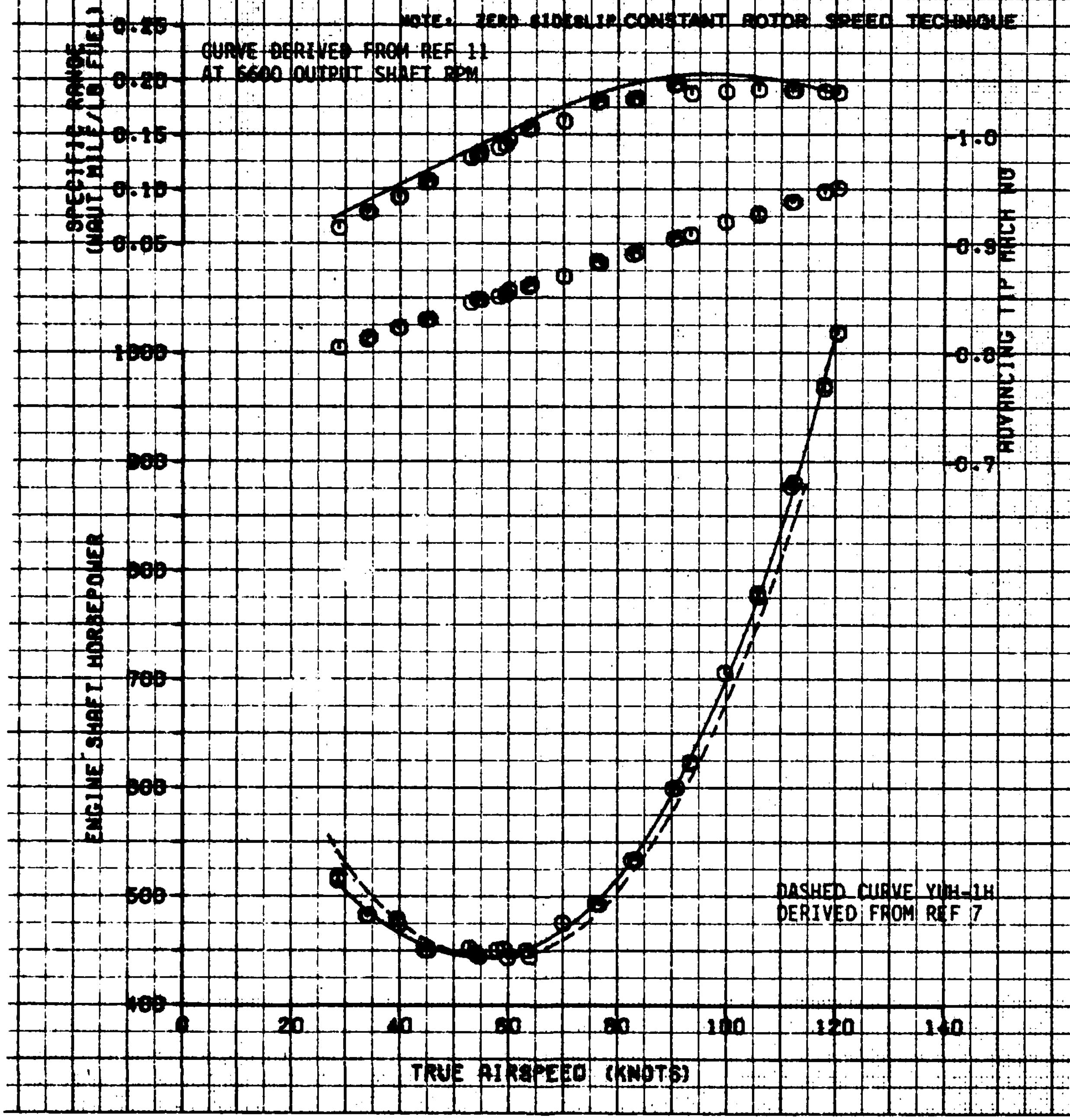

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)

FIGURE 15

LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 68-15532 T53-L-13B S/N LE208258

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TR (DEG C)	ROTOR SPEED (RPM)	C_T $X10^4$
MIN	7143	135.6	0.0	7050	-9.4	323.5	[illegible]
MEAN	7372	136.5 (MID)	0.0	7808	-8.5	323.9	[illegible]
MAX	7600	137.5	0.0	8132	-7.1	324.3	[illegible]



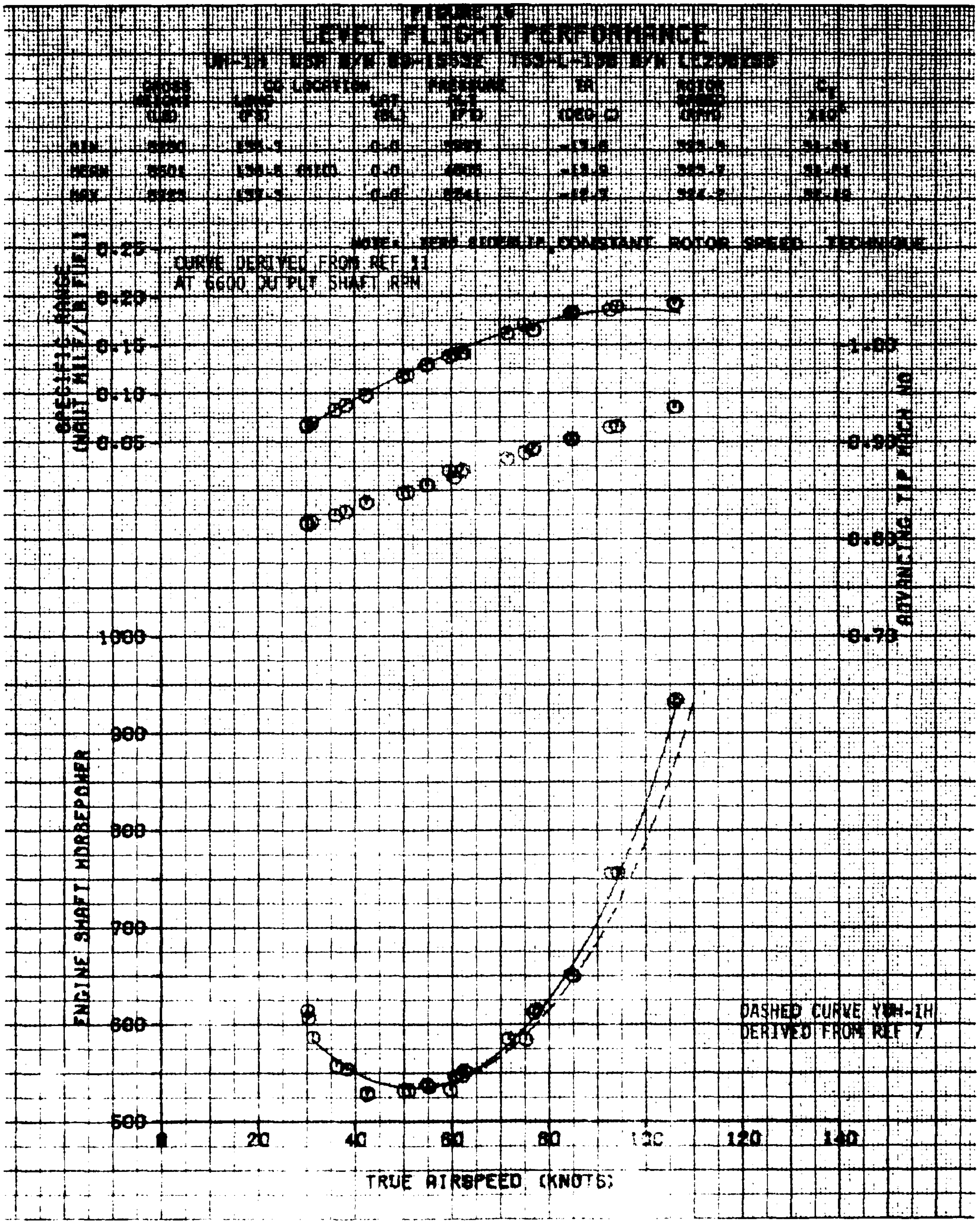

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)

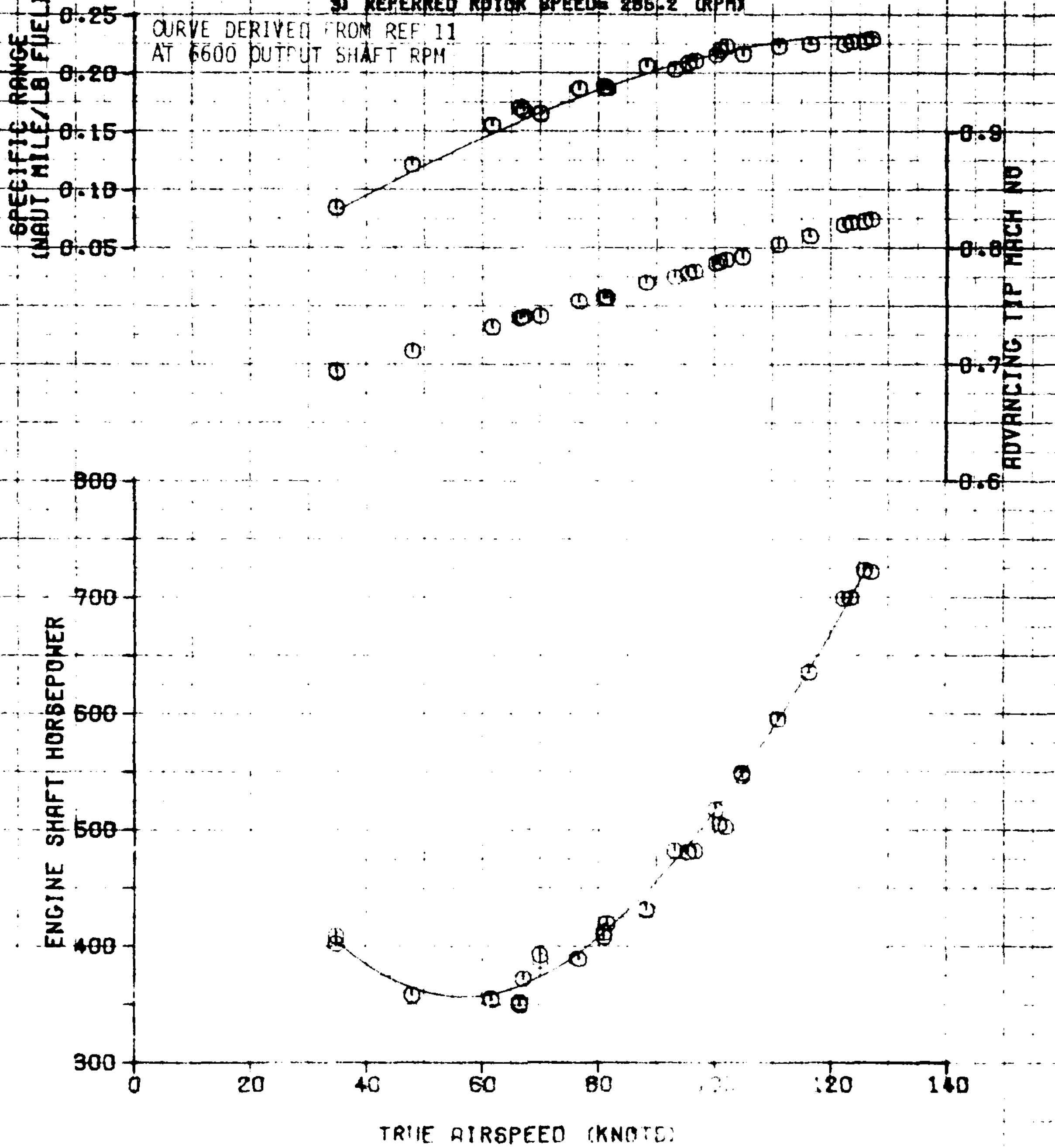
FIGURE 17

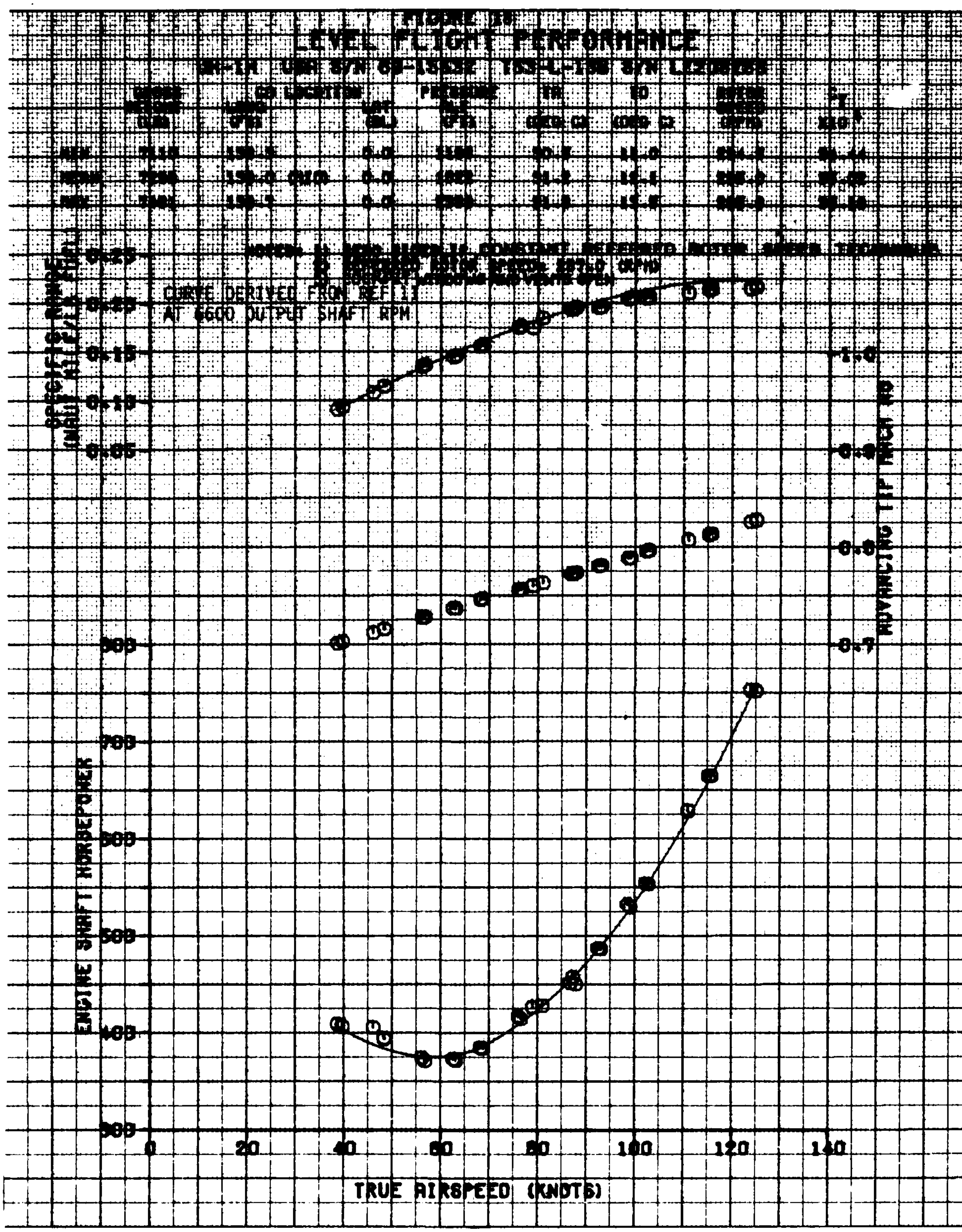
LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 69-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TD (DEG C)	ROTOR SPEED (RPM)	C_T X10[4]
MIN	6790	136.7		0.0	1882	31.4	12.3	293.6	54.46
MEAN	6957	137.6	(MID)	0.0	2571	33.5	13.0	294.1	54.77
MAX	7135	138.4		0.0	3168	35.5	14.2	294.7	54.99

NOTES: 1) COCKPIT WINDOWS AND VENTS OPEN
2) ZERO SIDESLIP CONSTANT REFERRED ROTOR SPEED TECHNIQUE
3) REFERRED ROTOR SPEED= 285.2 (RPM)




LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
0.25
0.20
0.15
0.10
0.05
1.0
0.9
0.8
0.7
ADVANCING TIP MACH NO
800
700
600
500
400
300
ENGINE SHAFT HORSEPOWER
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

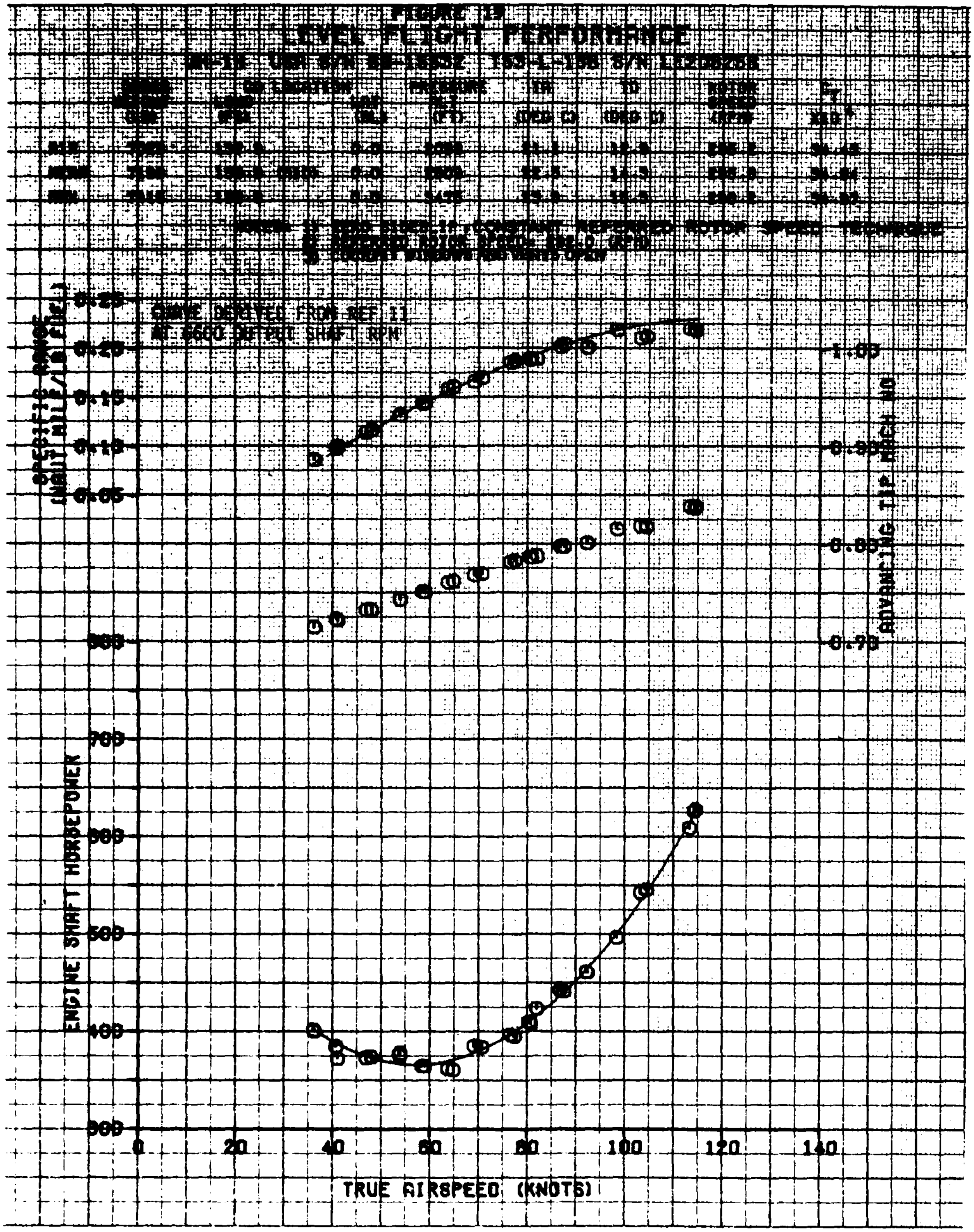
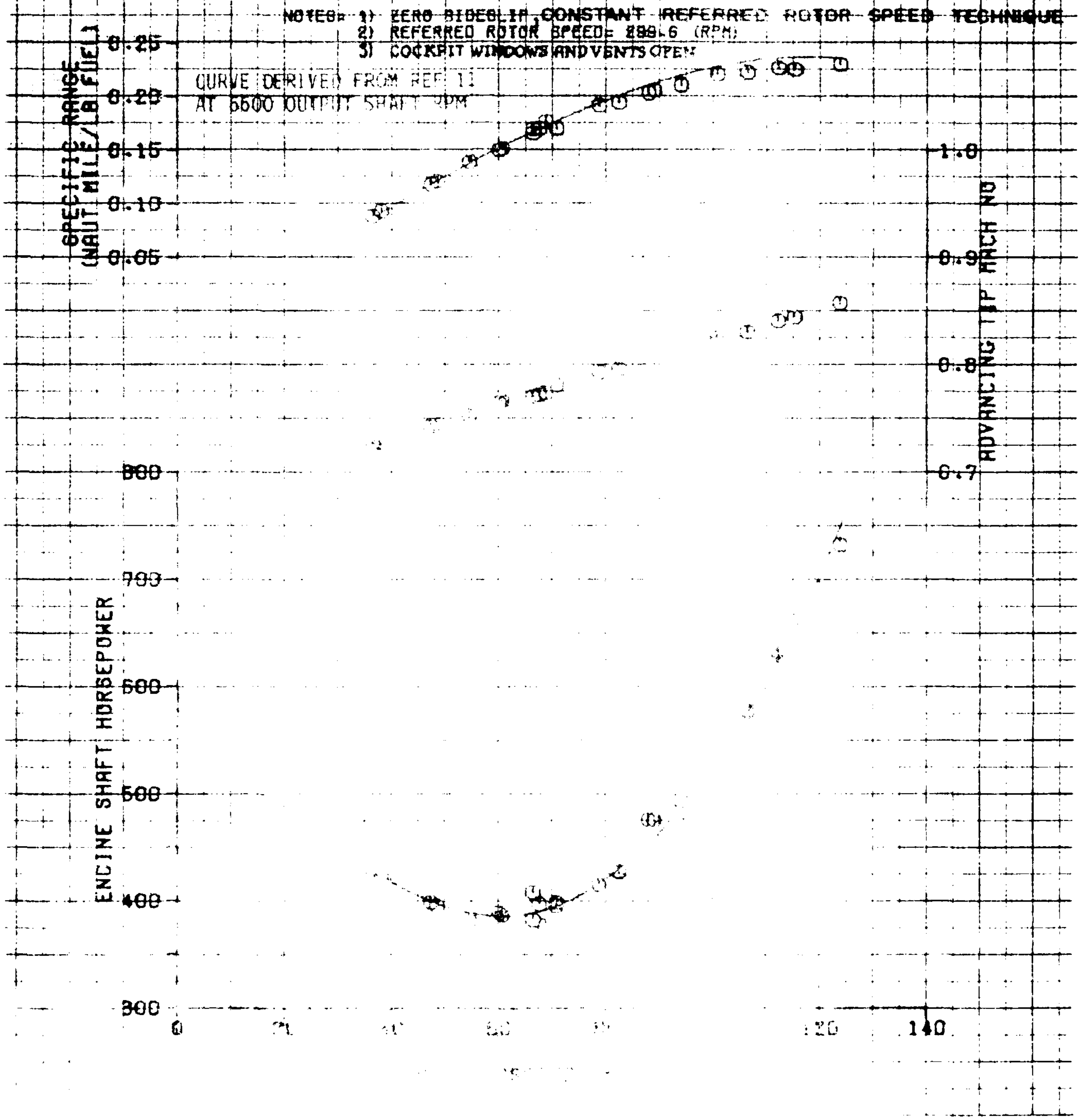

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE (NAUT MILE/LB FUEL)
0.25
0.20
0.15
0.10
0.05
ADVANCING TIP MACH NO
1.00
0.90
0.85
0.70
ENGINE SHAFT HORSEPOWER
800
700
600
500
400
300
0
20
40
60
80
100
120
140
TRUE AIRSPEED (KNOTS)

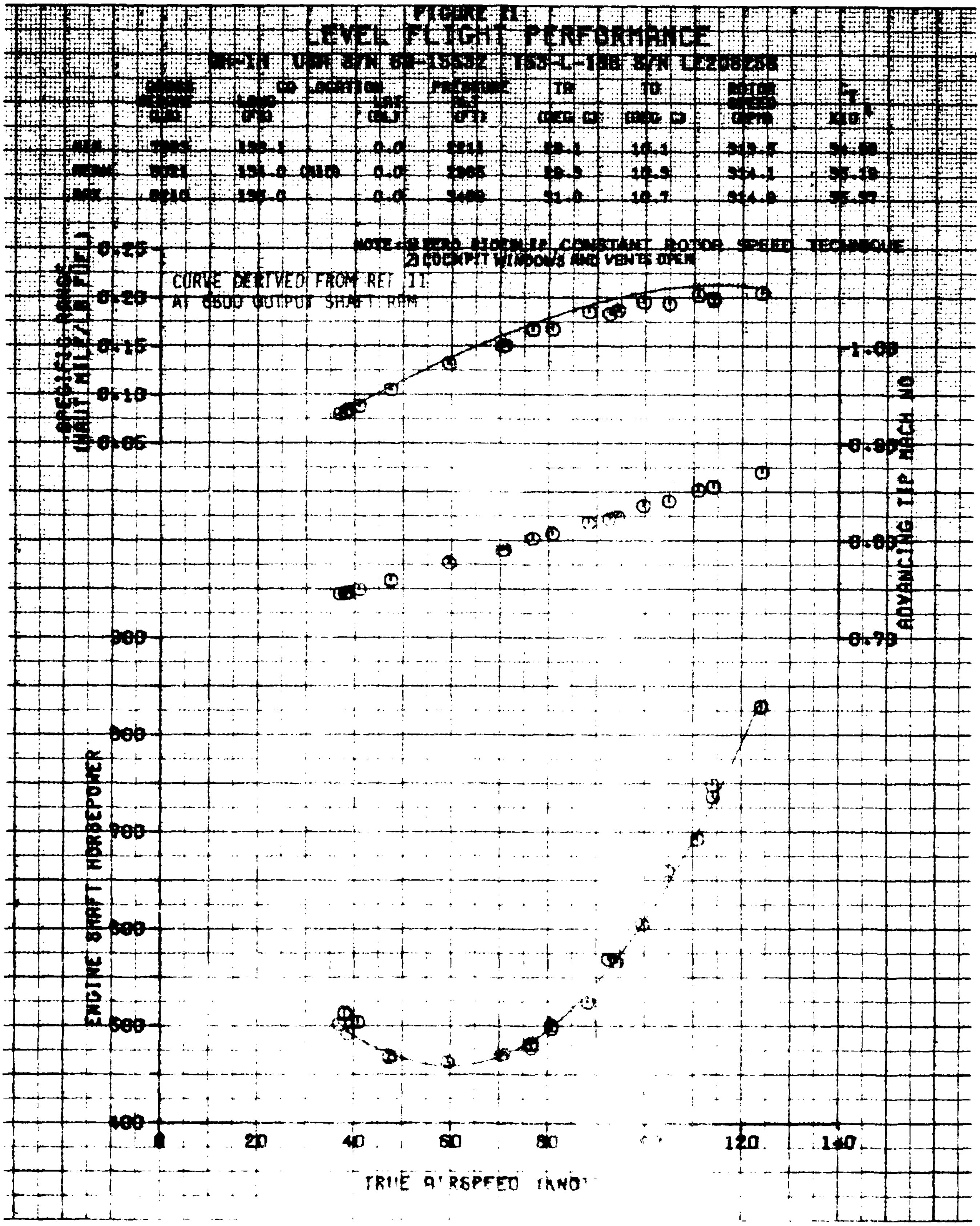
FIGURE 20

LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 69-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TD (DEG C)	ROTOR SPEED (RPM)	$C_T \times 10^4$
MIN	7104	138.2	0.0	3767	19.9	9.3	301.7	54.88
MEAN	7259	138.8 (MID)	0.0	4417	20.7	12.8	302.5	55.21
MAX	7428	139.5	0.0	4970	21.5	15.5	304.0	55.45





LEVEL FLIGHT PERFORMANCE
CONSTANT ROTOR SPEED TECHNIQUE
2) COCKPIT WINDOWS AND VENTS OPEN
CURVE DERIVED FROM REF 11
0.25
0.20
0.15
0.10
0.05
1.00
0.90
0.80
0.70
ADVANCING TIP MACH NO
ENGINE SHAFT HORSEPOWER
800
700
600
500
400
20
40
60
80
120
140

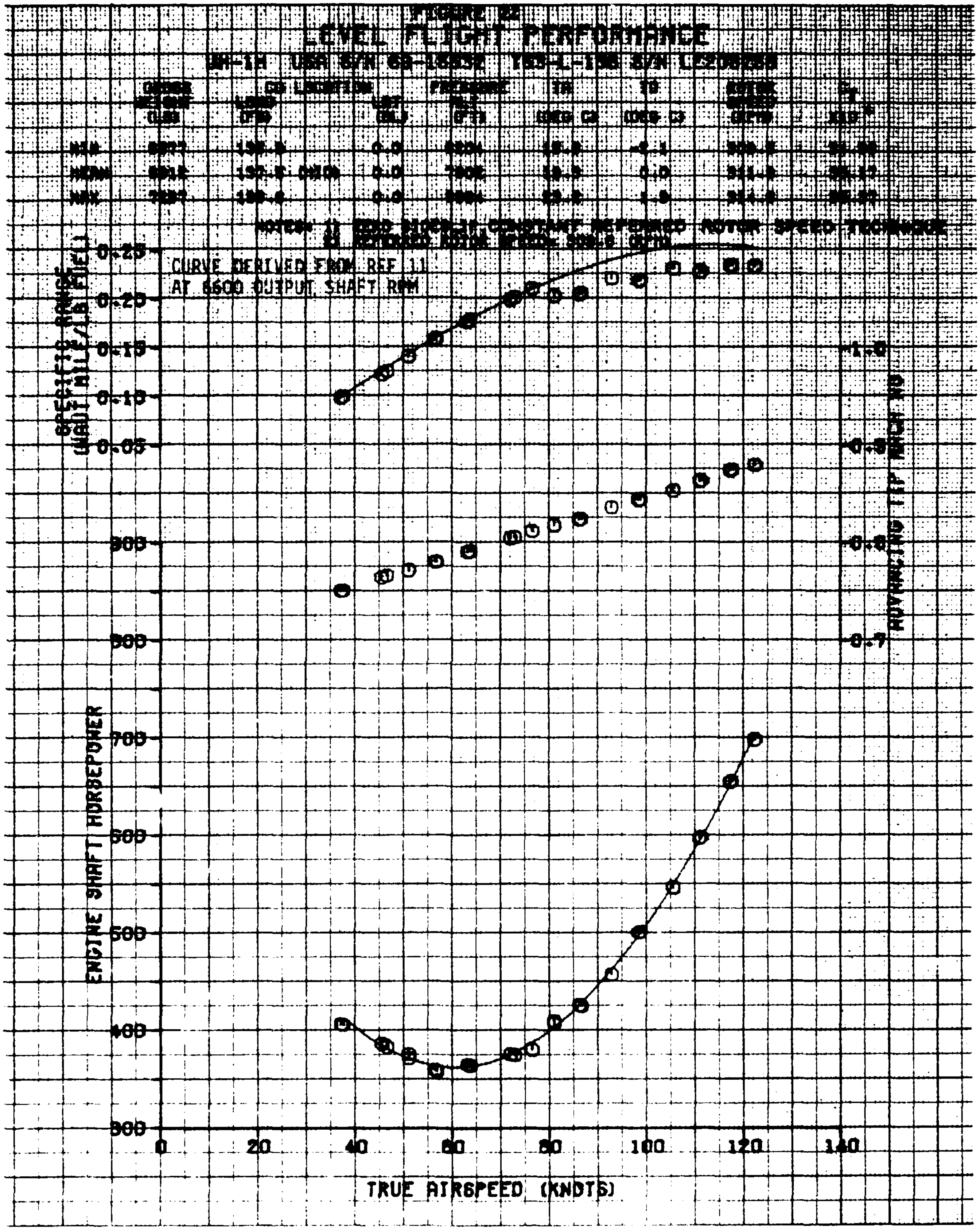

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
0.25
0.20
0.15
0.10
0.05
1.0
0.9
0.8
0.7
700
600
500
400
300
0
20
40
60
80
100
120
140
ENGINE SHAFT HORSEPOWER
TRUE AIRSPEED (KNOTS)

FIGURE 25

LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 69-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TO (DEG C)	ROTOR SPEED (RPM)	C_T X10[4]
MIN	8010	138.3	0.0	6970	17.2	-4.5	310.2	34.87
MEAN	8088	137.0	0.0	7774	18.7	0.0	311.7	35.18
MAX	7059	137.7	[illegible]	8320	21.0	1.5	313.8	35.58

NOTES: 1) [illegible]RO SIDESLIP CONSTANT REFERRED ROTOR SPEED TECHNIQUE
2) [illegible]FERRED ROTOR SPEED= 309.7 (RPM)
3) [illegible]CKPIT WINDOWS AND VENTS OPEN

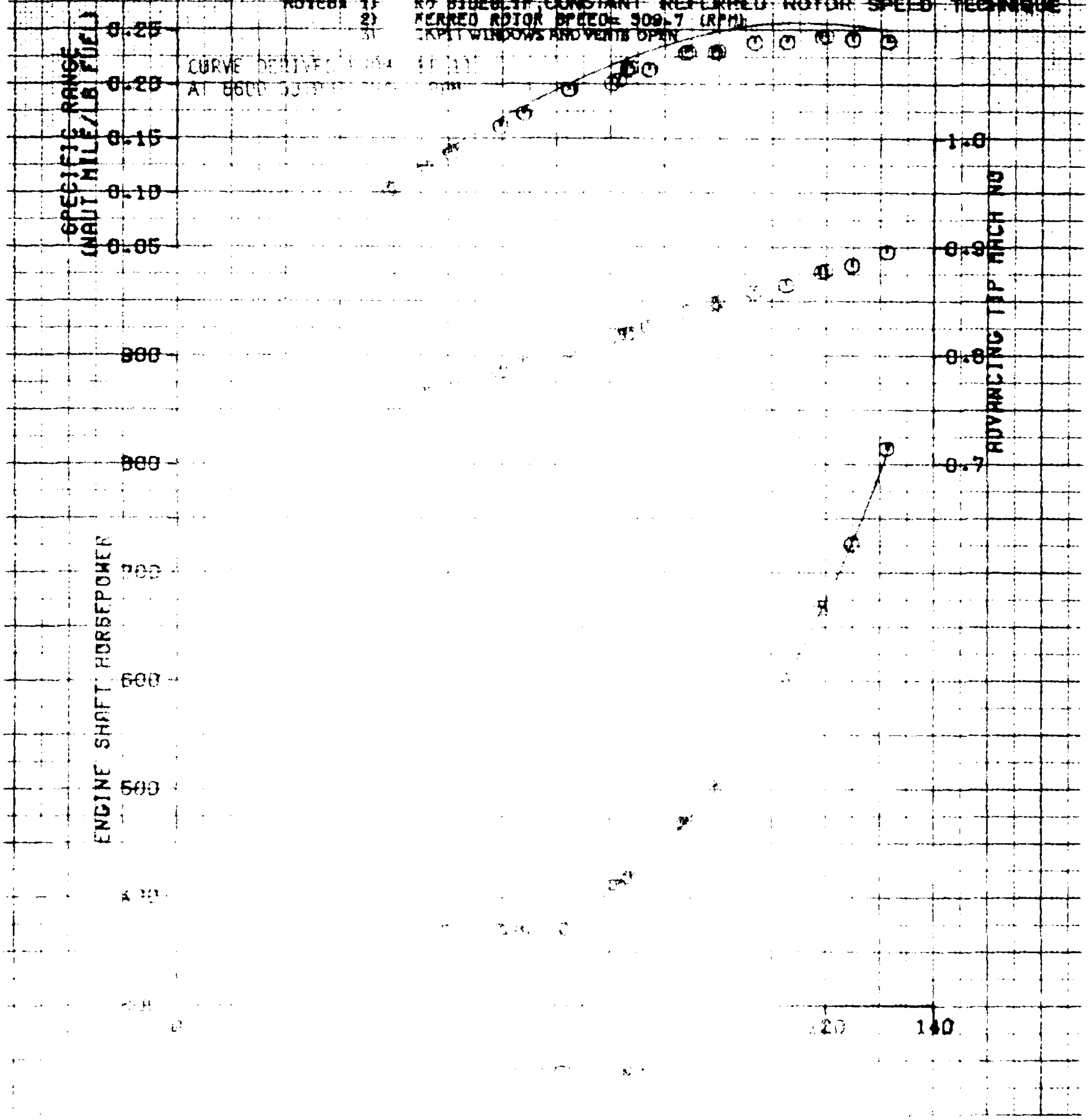
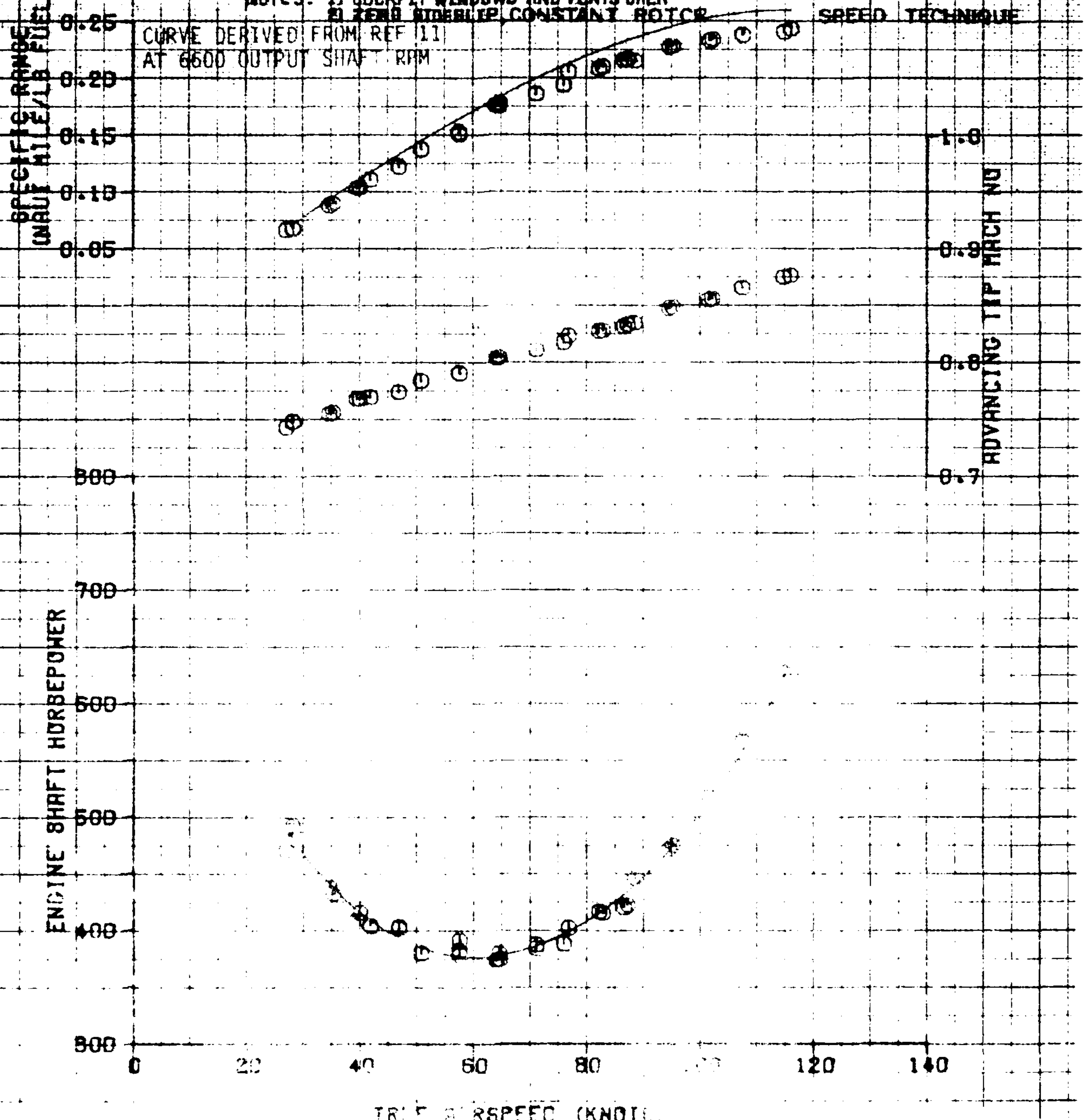
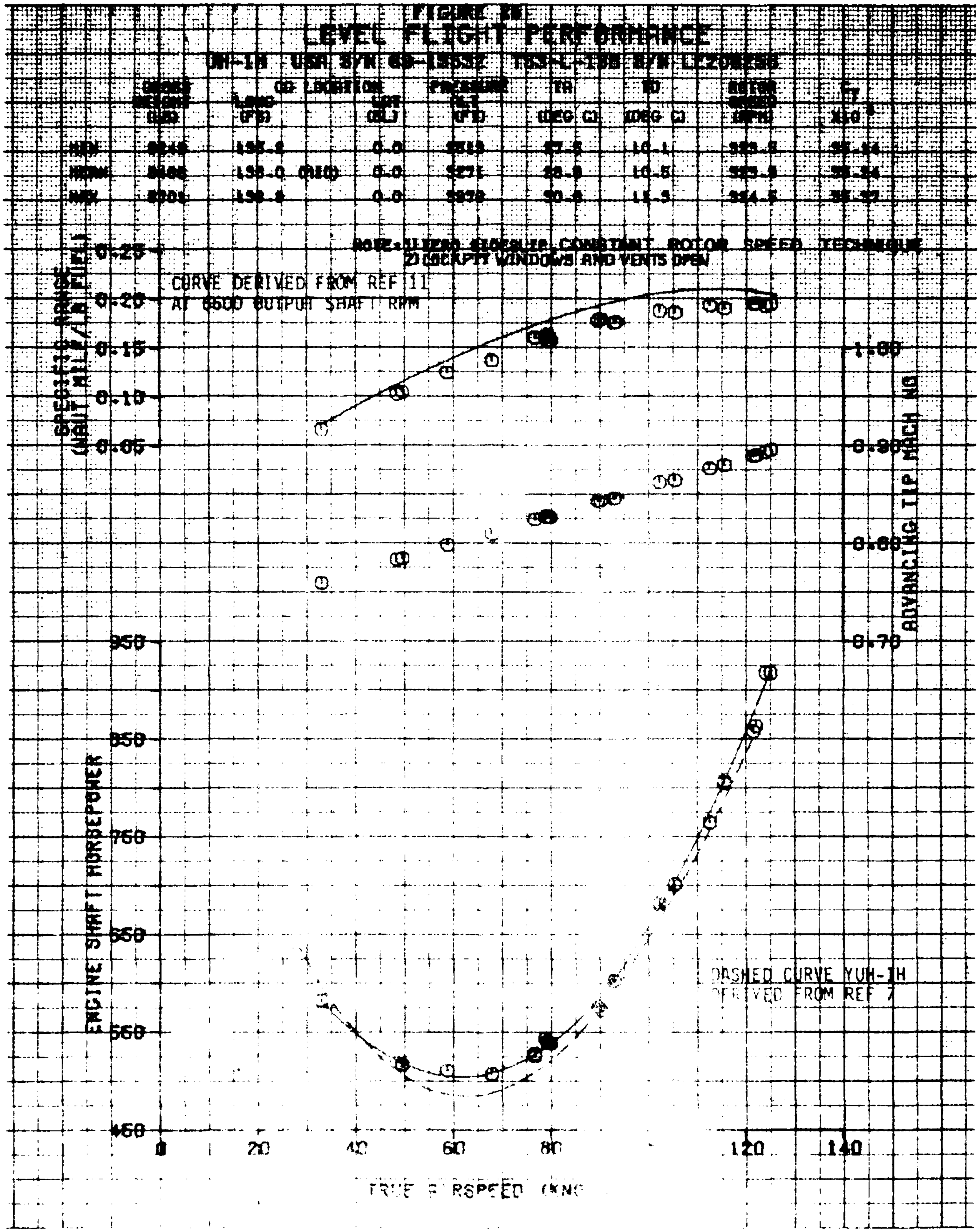


FIGURE 24
LEVEL FLIGHT PERFORMANCE
UH-1H USA S/N 69-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	TD (DEG C)	ROTOR SPEED (RPM)	C_T X10^4
MIN	6853	155.7	0.0	7554	12.0	5.2	312.0	54.07
MEAN	6914	157.6 (MID)	0.0	8230	15.0	6.1	314.2	55.51
MAX	7153	158.6	0.0	9506	18.5	9.3	315.0	55.84

NOTES: 1) COCKPIT WINDOWS AND VENTS OPEN
2) ZERO SIDESLIP, CONSTANT ROTOR SPEED TECHNIQUE





LEVEL FLIGHT PERFORMANCE
CG LOCATION
ROTOR SPEED
(DEG C)
NOTE: 1) ZERO SIDESLIP CONSTANT ROTOR SPEED TECHNIQUE
2) COCKPIT WINDOWS AND VENTS OPEN
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE
0.25
0.20
0.15
0.10
0.05
ADVANCING TIP MACH NO
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER
950
850
750
650
550
450
DASHED CURVE YUH-1H
DERIVED FROM REF 7
0
20
40
60
80
120
140
TRUE AIRSPEED

FIGURE 26
LEVEL FLIGHT PERFORMANCE
UH-1H USA S/N 68-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	ROTOR SPEED (RPM)	C_T $X10^4$
MIN	[illegible]	134.7		0.0	7705	-13.7	303.3	34.64
MEAN	7150	135.6	(MID)	0.0	8425	-13.4	304.0	35.18
MAX	7875	136.6		0.0	[illegible]	-12.7	304.7	35.55

NOTE: ZERO SIDESLIP, CONSTANT ROTOR SPEED TECHNIQUE

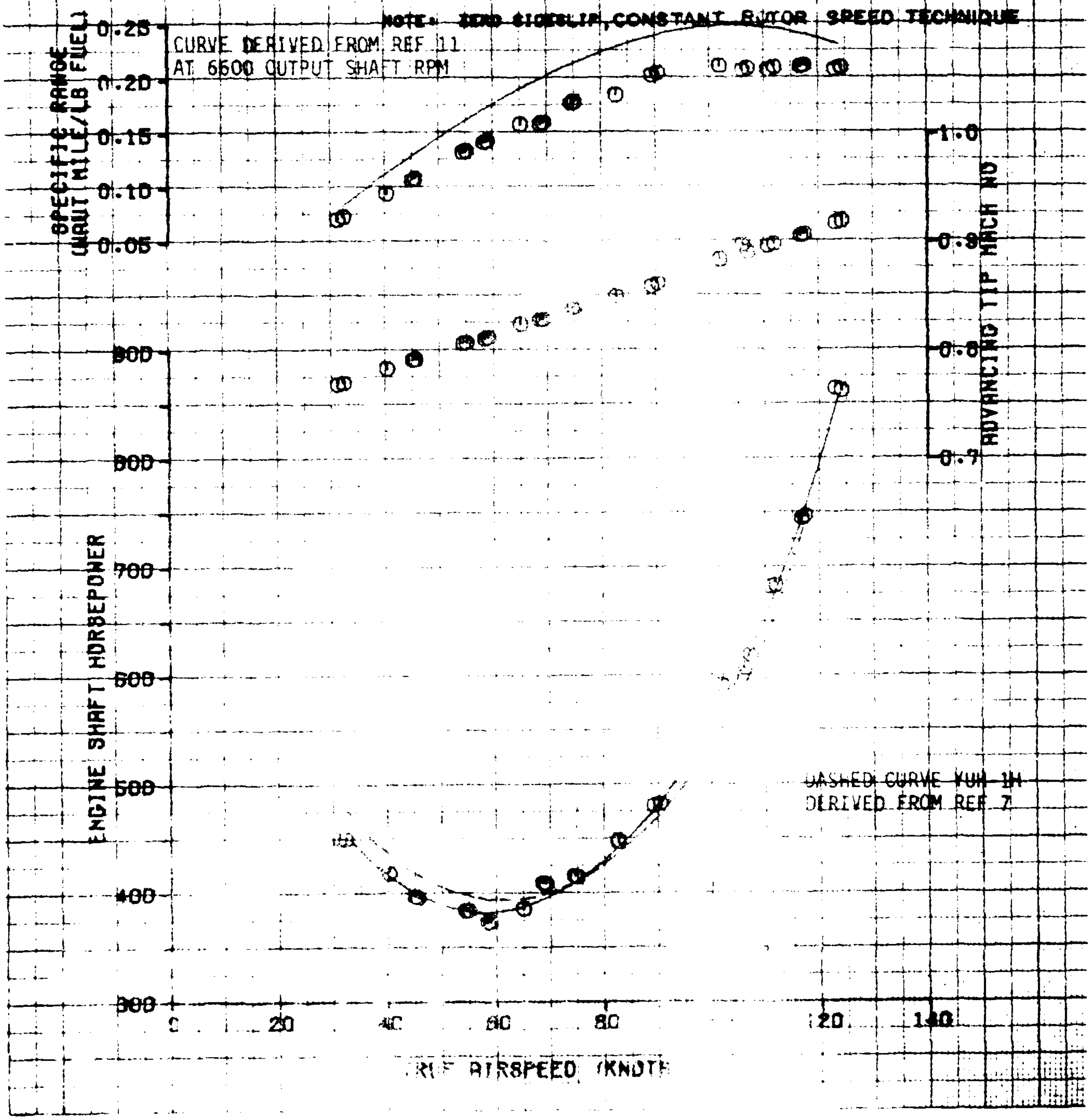
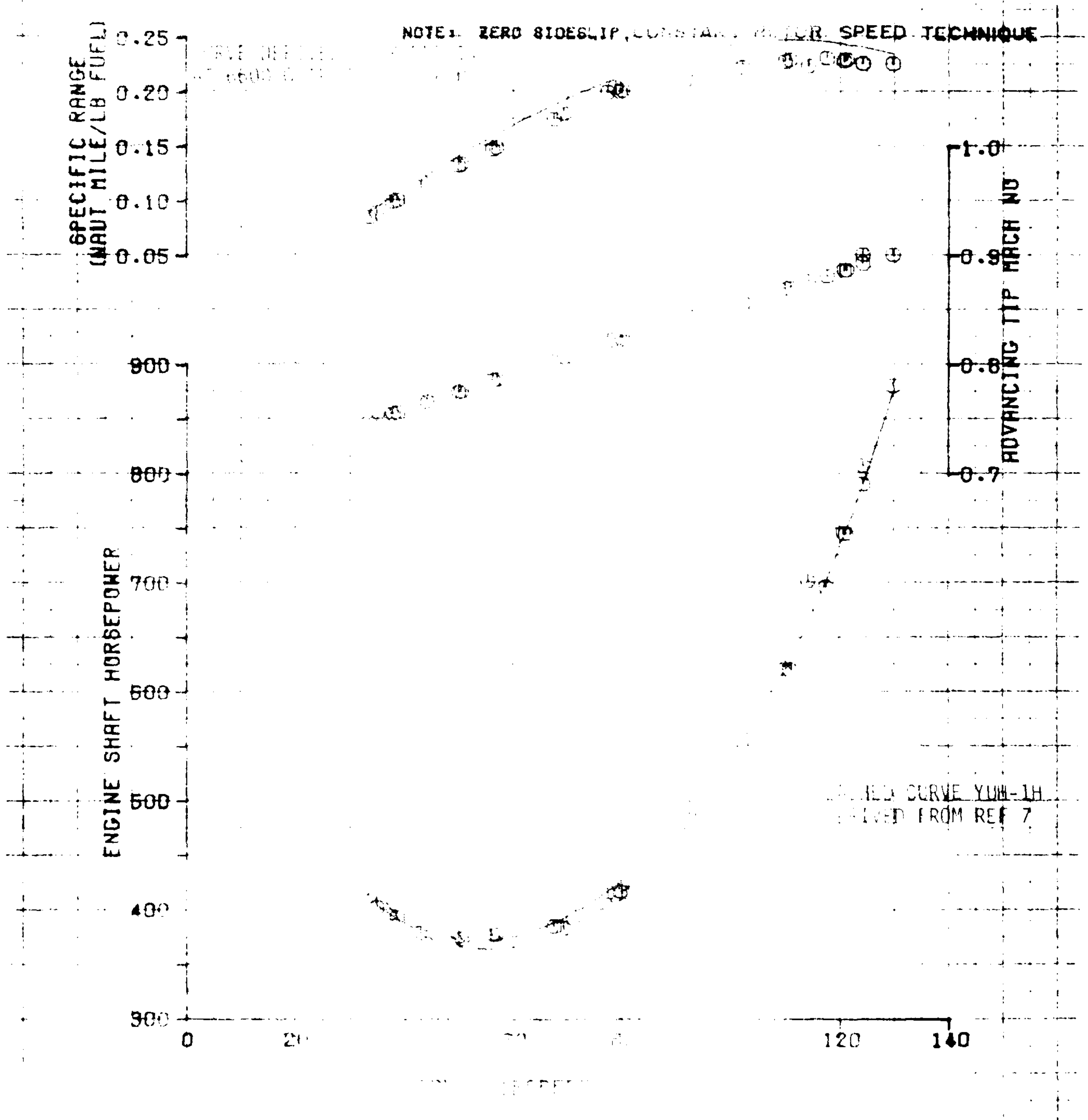


FIGURE 27

LEVEL FLIGHT PERFORMANCE

UH-1H USA S/N 68-15532 T53-L-13B S/N LE20825B

	GROSS WEIGHT (LB)	CG LOCATION LONG (FS)		CG LOCATION LAT (BL)	PRESSURE ALT (FT)	TA (DEG C)	ROTOR SPEED (RPM)	C_T X10[4]
MIN	6984	134.7		0.0	5899	-17.8	293.2	34.85
MEAN	7159	135.8	(MID)	0.0	6535	-13.0	293.9	35.07
MAX	7354	136.8		0.0	7439	[illegible]	294.8	35.56



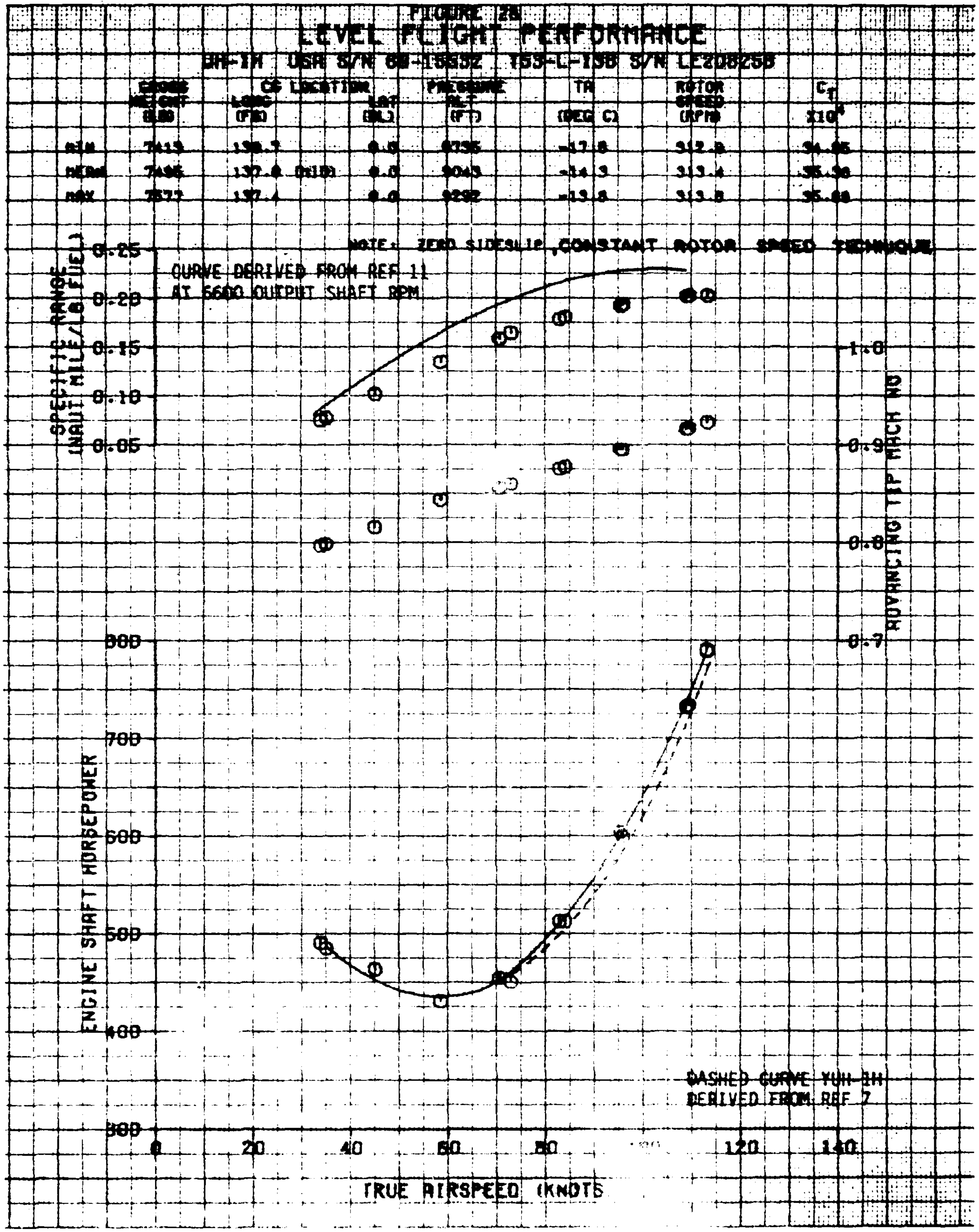
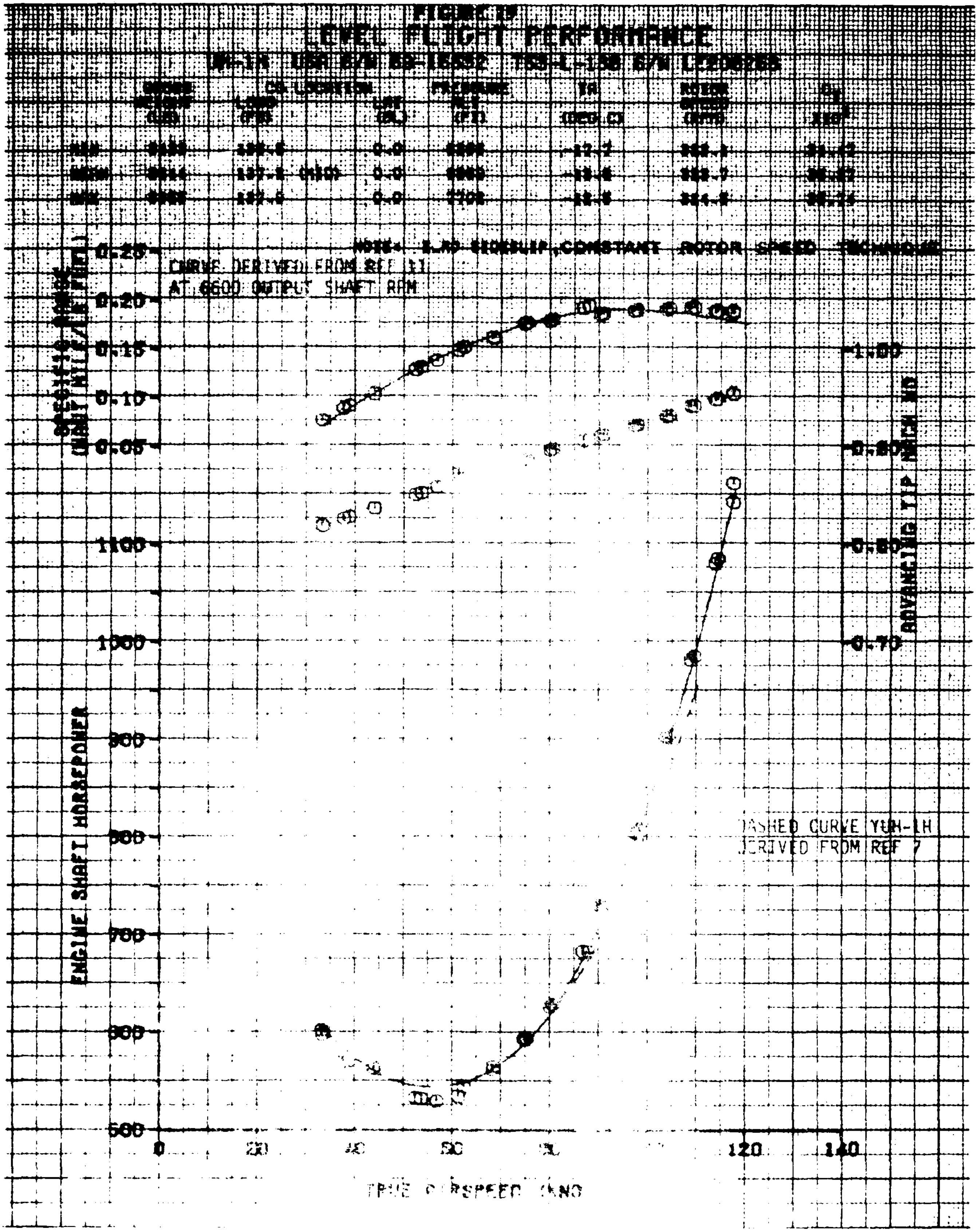


FIGURE 28
LEVEL FLIGHT PERFORMANCE
CG LOCATION
TA
(DEG C)
ROTOR
SPEED
(RPM)
NOTE: ZERO SIDESLIP, CONSTANT ROTOR SPEED TECHNIQUE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
SPECIFIC RANGE (NAUT MILE/LB FUEL)
ADVANCING TIP MACH NO
ENGINE SHAFT HORSEPOWER
DASHED CURVE YUH-1H
DERIVED FROM REF 7
TRUE AIRSPEED (KNOTS)


LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)
0.20
0.15
0.10
0.05
1100
1000
900
800
700
600
500
1.00
0.80
0.70
0
20
40
60
80
120
140

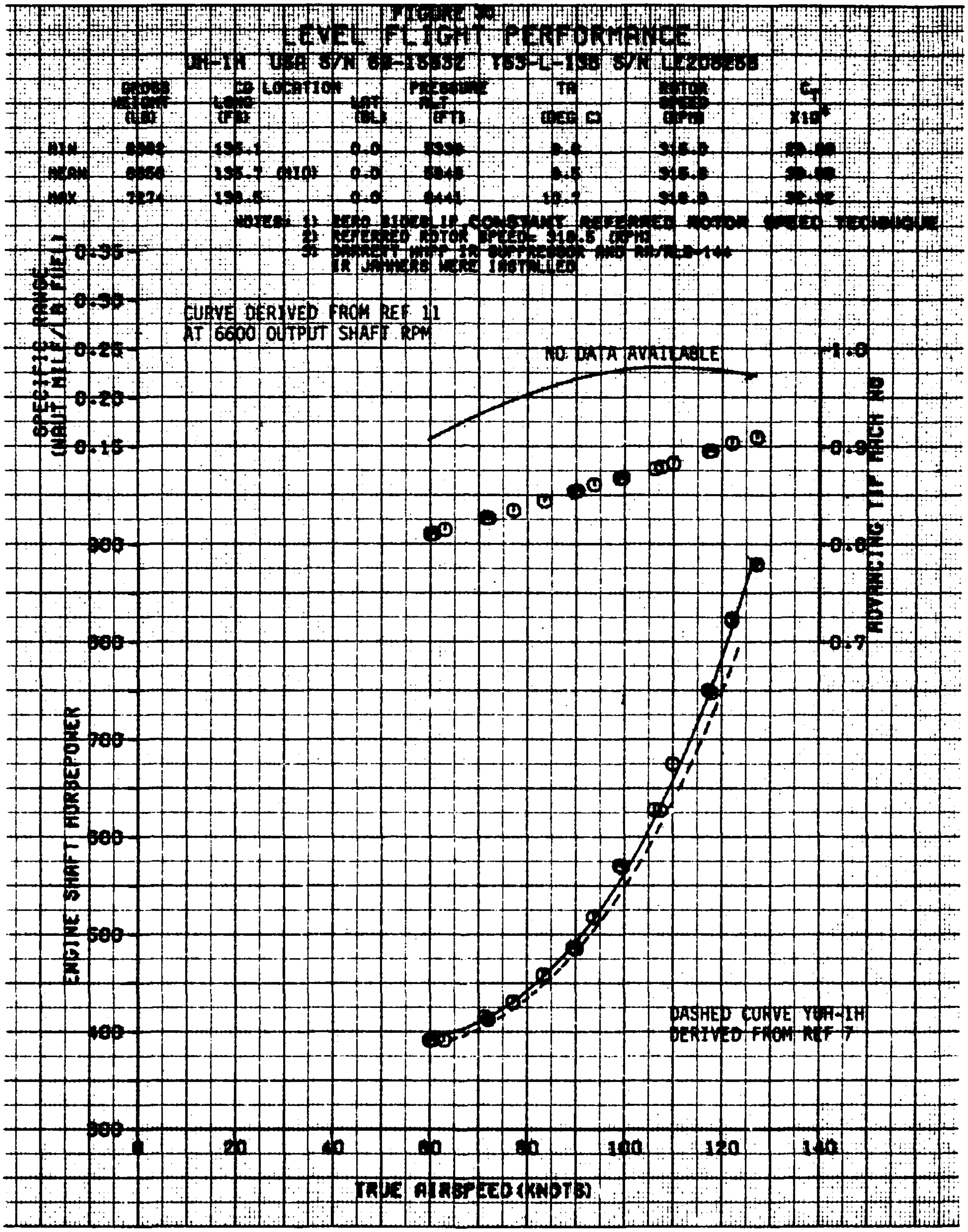
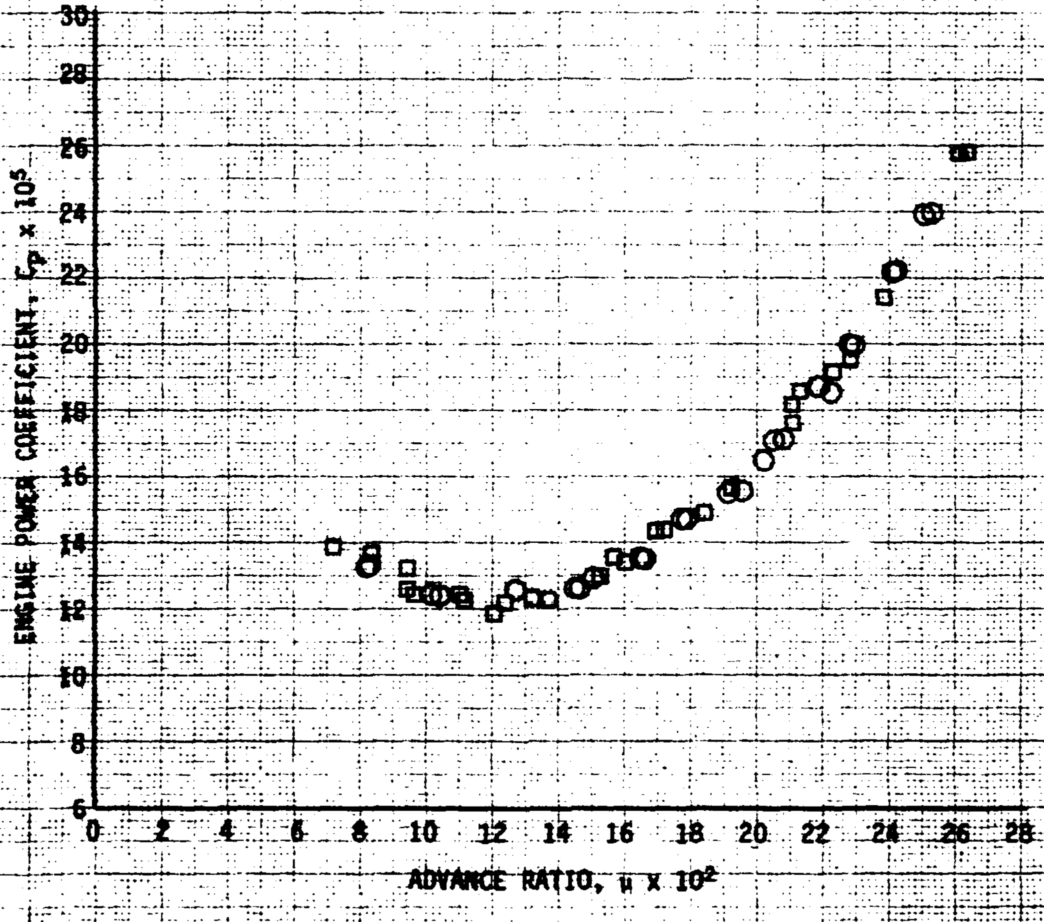

LEVEL FLIGHT PERFORMANCE
CURVE DERIVED FROM REF 11
AT 6600 OUTPUT SHAFT RPM
NO DATA AVAILABLE
DASHED CURVE YUH-1H
DERIVED FROM REF 7
ENGINE SHAFT HORSEPOWER
ADVANCING TIP MACH NO
TRUE AIRSPEED (KNOTS)
0
20
40
60
80
100
120
140
300
400
500
600
700
800
900
0.15
0.20
0.25
0.30
0.35
0.7
0.8
0.9
1.0

FIGURE 31
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE REFERRED ROTOR SPEED COMPARISON
UH-1H S/N 69-15532 T53-L-13B S/N LE20825B

SYMBOL	MEAN C_T $\times 10^4$	MEAN TA (°C)	MEAN ROTOR SPEED (RPM)	MEAN $N/\sqrt{\theta}$ (RPM)	MEAN W/δ (LB)
○	31.67	+15.6	324.3	324.0	9032
□	31.84	-12.0	308.8	324.4	9103



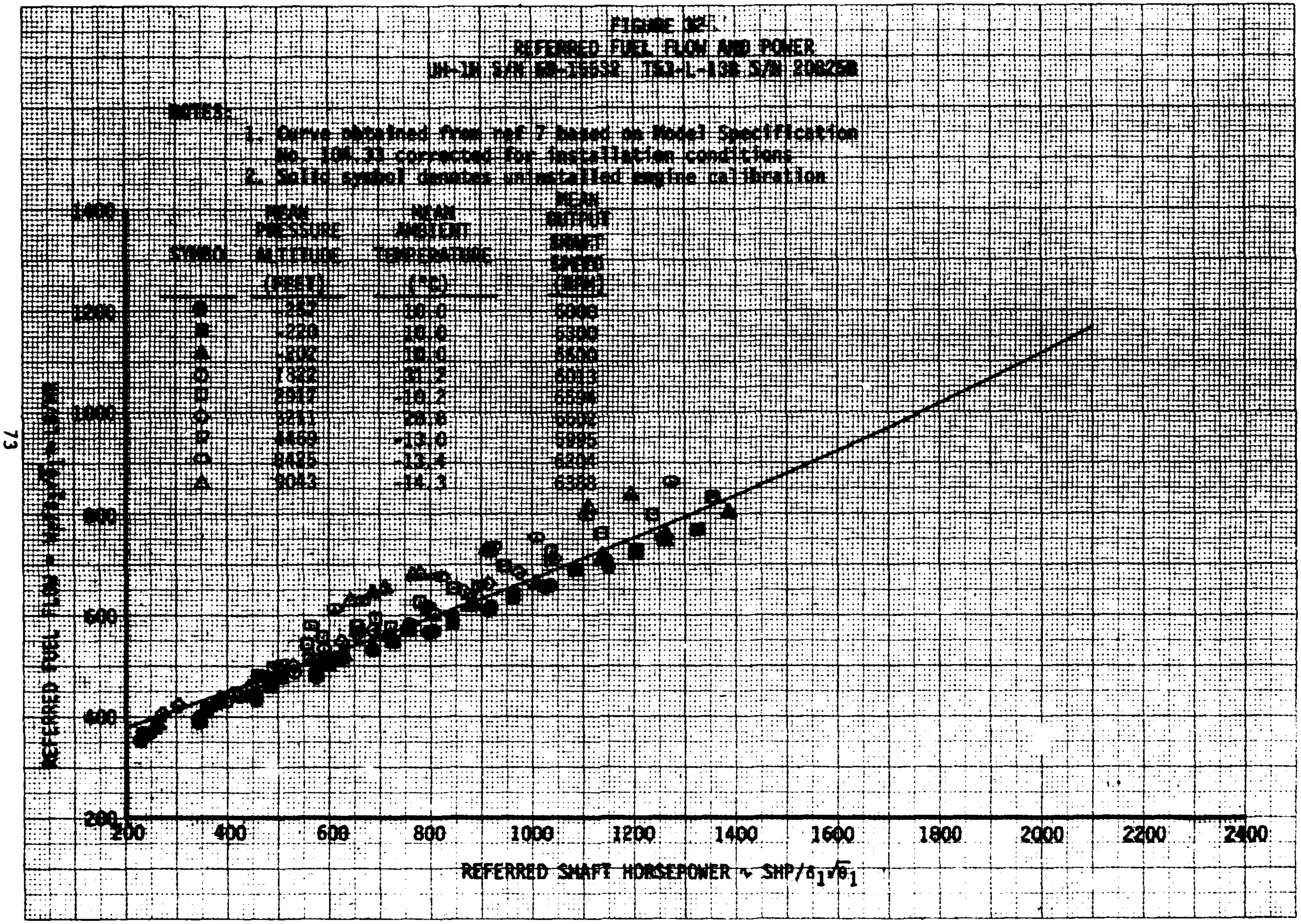


FIGURE 32
REFERRED FUEL FLOW AND POWER
NOTES:
1. Curve obtained from ref 7 based on Model Specification No. 104.33 corrected for installation conditions
2. Solid symbol denotes uninstalled engine calibration
SYMBOL
MEAN PRESSURE ALTITUDE (FEET)
MEAN AMBIENT TEMPERATURE (°C)
MEAN OUTPUT SHAFT SPEED (RPM)
-220
1822
2917
4459
9043
REFERRED SHAFT HORSEPOWER ~ SHP/δ1√θ1
REFERRED FUEL FLOW
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400

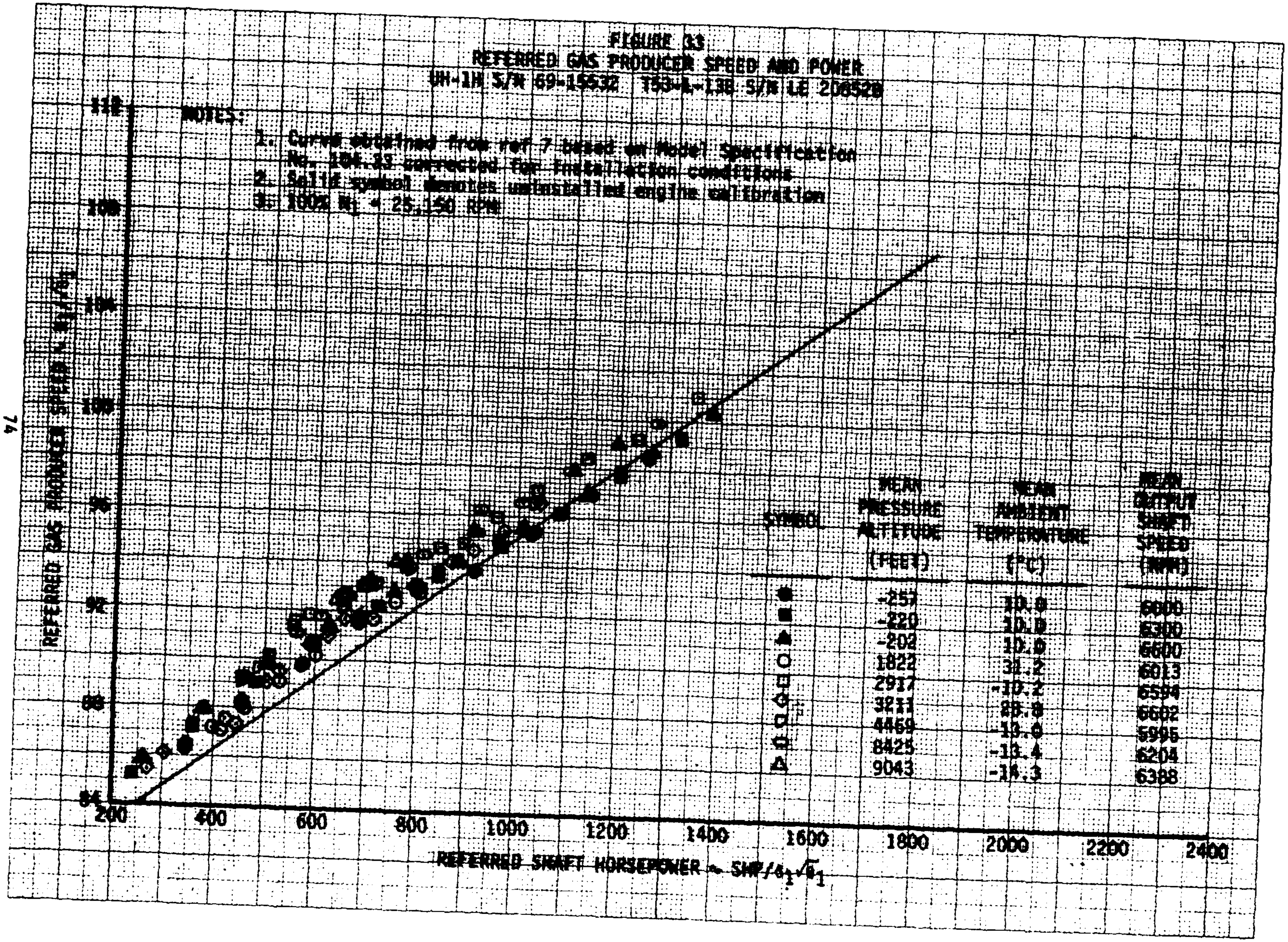


FIGURE 33
REFERRED GAS PRODUCER SPEED AND POWER
UH-1H S/N 69-15532 T53-L-13B S/N LE 20852B
NOTES:
1. Curve obtained from ref 7 based on Model Specification No. 104.33 corrected for installation conditions
2. Solid symbol denotes uninstalled engine calibration
3. 100% N_1 = 25,150 RPM
SYMBOL
MEAN PRESSURE ALTITUDE (FEET)
MEAN AMBIENT TEMPERATURE (°C)
MEAN OUTPUT SHAFT SPEED (RPM)
-257 10.0 6000
-220 10.0 6300
-202 10.0 6600
1822 31.2 6013
2917 -10.2 6594
3211 28.8 6602
4469 -13.0 5996
8425 -13.4 6204
9043 -14.3 6388
REFERRED GAS PRODUCER SPEED ~ $N_1/\sqrt{\theta_1}$
REFERRED SHAFT HORSEPOWER ~ $SHP/\delta_1\sqrt{\theta_1}$
84
88
92
96
100
104
108
112
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400

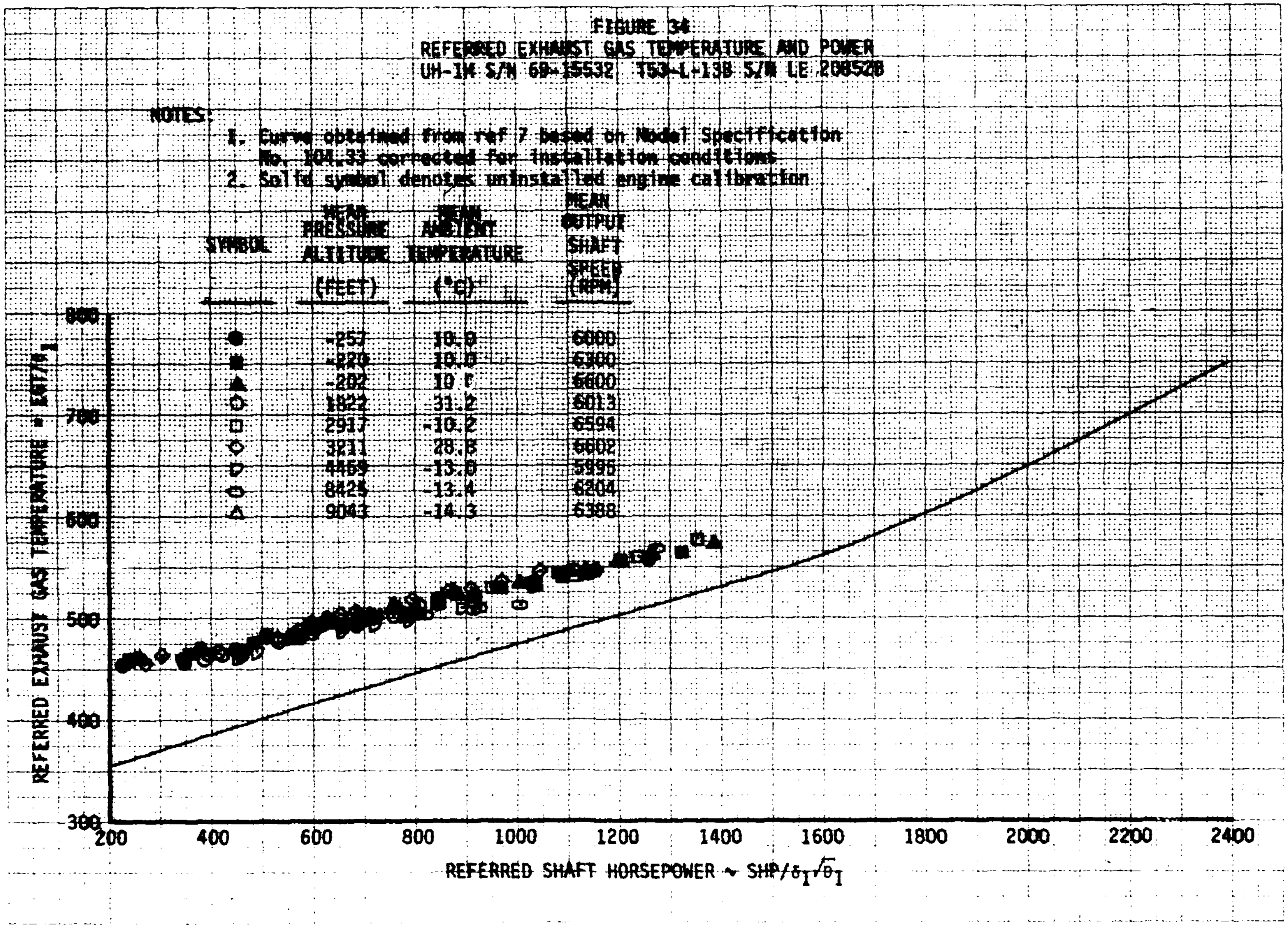


FIGURE 34
REFERRED EXHAUST GAS TEMPERATURE AND POWER
UH-1M S/N 69-15532 T53-L-13B S/N LE 208528
NOTES:
1. Curve obtained from ref 7 based on Model Specification No. 104.33 corrected for installation conditions
2. Solid symbol denotes uninstalled engine calibration
SYMBOL
MEAN PRESSURE ALTITUDE (FEET)
MEAN AMBIENT TEMPERATURE (°C)
MEAN OUTPUT SHAFT SPEED (RPM)
-257 10.0 6000
-229 10.0 6300
-202 10.0 6600
1822 -31.2 6013
2917 -10.2 6594
3211 28.8 6602
4469 -13.0 5995
8425 -13.4 6204
9043 -14.3 6388
REFERRED EXHAUST GAS TEMPERATURE ~ EGT/θ_I
300
400
500
600
700
800
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400
REFERRED SHAFT HORSEPOWER ~ SHP/$\delta_I\sqrt{\theta_I}$

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