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# T-38A CATEGORY II PERFORMANCE TEST

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**TECHNICAL DOCUMENTARY REPORT NO. 63-27**
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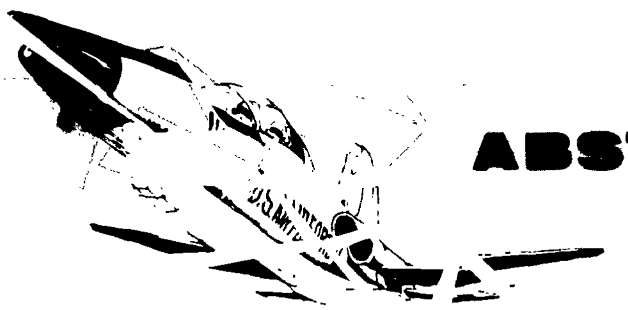
AFFTC PROJECT DIRECTIVE 59-18



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



# ABSTRACT....


This report presents the results of the Category II performance test of the T-38A airplane, Serial Number 58-1194. The test was conducted during the period May 1961 - March 1962 at the Air Force Flight Test Center, Edwards Air Force Base, California, and required 58 flights totaling 53 hours.

The T-38 airplane is a two place tandem supersonic trainer manufactured by the Norair Division of the Northrop Corporation. The mission of the airplane is to accomplish all phases of basic pilot training, including: day and night transition, formation, navigation, instrument and acrobatic flying. The airplane is powered by two General Electric J85-GE-5 engines rated at 3850 pounds of thrust each.

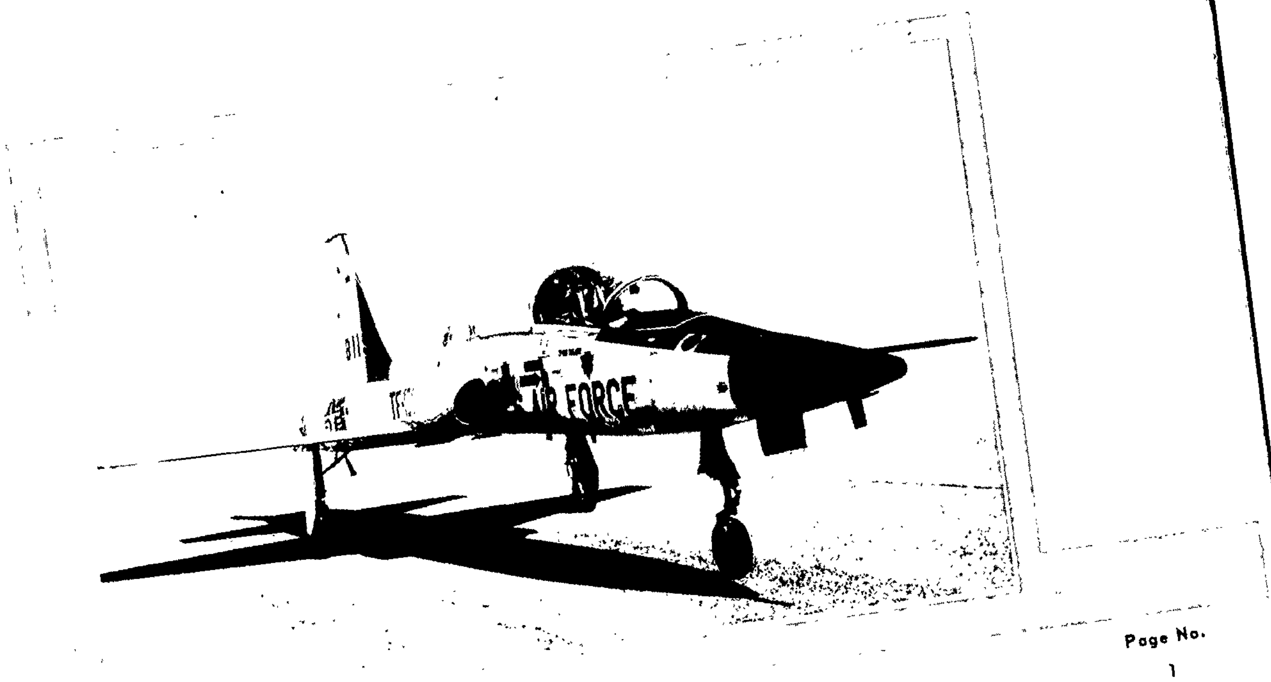
The T-38A is an excellent trainer for transition into high performance aircraft. It has excellent handling characteristics, maximum speed is in the low supersonic range, and takeoff and subsonic climb performance is comparable to the latest Century Series fighters.

With maximum power, the airplane can readily takeoff in 2500 feet and clear a 50 foot obstacle in 3200 feet. With military power, the takeoff roll is 3900 feet and a total distance of 5400 feet is required to clear a 50 foot obstacle. This performance is achieved at a gross weight of 12,000 pounds utilizing 45 percent flaps.

The climb performance of this aircraft is excellent. With maximum power, a sea level rate of climb of 30,000 feet per minute is available at a gross weight of 11,000 pounds and the time to climb to 40,000 feet is only 2.5 minutes. The combat ceiling of the aircraft (500 feet per minute rate of climb) is 51,000 feet.

Single engine performance of the aircraft is very good. With maximum power, a rate of climb of 5500 feet per minute is available at sea level and a combat ceiling of 37,000 feet can be attained.

The maximum calculated ferry range of the aircraft is 954 nautical miles. This range is computed in accordance with landing reserves of MIL-C-5011A from level flight speed-power data. This range is available at the optimum cruise conditions of Mach 0.88 and a weight-pressure ratio of 58,000 pounds.



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*NOTE: Due to the limited requirement for the information contained in this Appendix, it has been published in a separate volume. Copies of this Appendix can be obtained by writing to the Defense Documentation Center (see inside cover).

*This report has been reviewed and approved*

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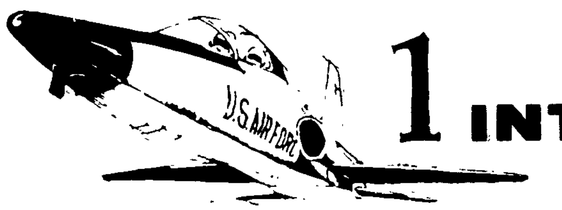



# 1 INTRODUCTION....

This report presents the results of the Air Force Flight Test Center Category II performance tests of T-38A serial number 58-1194. These tests were conducted to obtain quantitative performance data for incorporation into the T-38A Flight Manual.

The test program, which was conducted at the Air Force Flight Test Center, Edwards Air Force Base, California, between 22 May 61 and 12 March 62 consisted of 58 flights totaling 53 flight hours.

The T-38A airplane is a two place tandem trainer manufactured by the Norair Division of the Northrop Corporation. The mission of the airplane is to accomplish all phases of basic pilot training, including: day and night transitions, formation, navigation, instrument and acrobatic flying. The airplane is equipped with J85-GE-5 engines rated at 2500 pounds of thrust at military power and 3850 pounds of thrust with afterburning.

The test aircraft had an average, fully loaded, engine start gross weight of 12,330 pounds. The external configuration of the test aircraft is the same as the production trainer except that the test aircraft was equipped with a test nose boom, interim structural bands in the inlet ducts, and an externally mounted free air temperature probe. As a part of the test program, a production nose boom was installed on the test aircraft and a calibration accomplished. However, it was later determined that, because of manufacturing tolerances, this installation was not representative of the production system; therefore no data for this system is presented in this report.

The test program was conducted under the authority of Headquarters, Air Force Systems Command as directed by AFFTC Project Directive 59-18, dated 18 March 1959. The test data from the program was available to Norair as it was obtained and preliminary copies of the plots contained in Appendix I of this report were sent to Norair in February 1963.

# 2 TEST RESULTS....

## PREFLIGHT, STARTING AND GROUND HANDLING

Discussions of the preflight, starting, taxiing, and ground handling procedures and characteristics, as well as an evaluation of the cockpit have been previously accomplished and results published in AFFTC-TR-61-15, "T-38A Category II Stability and Control Tests," and AFFTC-TR-61-21, "T-38A Category II Systems Evaluation Test."

## TAKE-OFF AND INITIAL CLIMB

The takeoff characteristics of the T-38A aircraft are excellent. At 12,000 pounds gross weight with maximum power and 45 percent flaps, the aircraft will lift-off in 2500 feet at the AFFTC recommended speed of 155 knots IAS[1]. With military power the distance required is 3900 feet for the same conditions. Normal pilot technique is used to obtain the measured performance takeoff. Military power takeoffs are performed by holding the aircraft with the wheel brakes until throttles are advanced to military power. For maximum power takeoffs, military power is set with brakes held, then the afterburners are lit just as the brakes are released. Light differential braking or nose wheel steering is used for directional control up to about 50 knots IAS where the rudder becomes effective.

Stabilizer effectiveness during takeoff is adequate to rotate the aircraft to the takeoff attitude well before takeoff speed is reached. A comfortable speed of 135-140 knots IAS is recommended for rotation.

The initial climb of the aircraft is rapid for the first few seconds after lift-off and a definite push over is necessary to accomplish a shallow climb out.

Takeoff data corrected to sea level, standard day, no wind conditions is presented in Figure 1, Appendix I.

## LEVEL ACCELERATIONS

Level flight accelerations were accomplished at various altitudes from 5,000 to 45,000 feet with maximum power and from 5,000 to 40,000 feet with military power. These data were corrected to standard day conditions at representative climb weights and are presented in Figures 2 and 3, Appendix I, to define optimum climb schedule speeds. Norair's recommended climb schedules are presented as dashed lines on these plots and agree closely with the AFFTC determination of maximum climb potential.

The acceleration data were also corrected to standard day conditions at 10,000 pounds gross weight and presented

[1] Indicated airspeeds in this report are for the test nose boom airspeed system.

in terms of thrust available and thrust required for the various altitudes in Figures 19 through 24, Appendix I.

## CLIMB PERFORMANCE

The maximum power climb performance of the T-38A is impressive for a trainer and is comparable to the "Century Series" fighter aircraft. For example, the T-38A can takeoff and climb to best cruise altitude (40,000 feet) in 4 minutes.

Climb speed schedules recommended by Norair were utilized to obtain climb performance. These schedules, both maximum and military power, were verified by level flight accelerations. Figures 2 and 3, Appendix I, show the climb potential of the aircraft at various speeds.

Because the best climb speed with maximum power is quite high at low altitude (.91 Mach number), a modified schedule was suggested by the Air Training Command. This schedule consists of constant 450 knots CAS to 17,500 feet and .91 Mach number above 17,000 feet altitude.

Although the ATC schedule is lower than the optimum climb schedule at altitudes below 17,500 feet, the time and fuel required to accelerate and climb to higher altitudes are not significantly increased.

The following table presents a summary of the climb performance of the T-38A aircraft.

CLIMB PERFORMANCE SUMMARY

| Power | Schedule | Weight at Which Service Ceiling* is Attained-Lbs | Service Ceiling Ft | Time to 40,000 Ft Min | Fuel Used to 40,000 Ft Lb | Distance Travelled to 40,000 Ft NAM |
|---|---|---|---|---|---|---|
| Max | Norair | 10,900 | 51,500 | 2.5 | 500 | 23 |
| Max | ATC | 10,900 | 51,500 | 2.9 | 612 | 26 |
| Mil | Norair | 11,300 | 43,500 | 10.5 | 545 | 87 |

*Service ceiling is the altitude corresponding to a rate of climb of 100 feet per minute.

Two engine climb performance is presented in Figures 4, 5 and 6, Appendix I.

Single engine climbs were accomplished with both maximum and military power utilizing the Norair recommended climb schedules. Asymmetric power characteristics of the aircraft are excellent. Heading is easily maintained with rudder trim for both maximum and military power. The following table and Figures 7 and 8, Appendix I, present a summary of the single engine climb performance for maximum and military power.

SINGLE ENGINE CLIMB PERFORMANCE SUMMARY

| Power | Service Ceiling-Ft | Time From Sea Level to Service Ceiling-Min | Fuel Used, Sea Level to Service Ceiling-Lb | Distance Travelled from Sea Level to Service Ceiling - NAM |
|---|---|---|---|---|
| Max | 39,750 | 34 | 2420 | 300 |
| Mil | 18,500 | 30 | 900 | 145 |

## LEVEL FLIGHT PERFORMANCE

Maximum ferry range of the T-38A is 954 nautical air miles. This range is achieved by utilizing military power for takeoff and climb to 39,700 feet followed by cruise at a weight-pressure ratio of 58,000 pounds[2] and .88 Mach number.

The following tables present a comparison of a computed ferry range mission for a typical production aircraft (computed on basis of test data in Appendix I) to an actual range mission obtained with the test aircraft.

COMPUTED FERRY RANGE MISSION

| | |
|---|---|
| Gross Weight at Engine Start | 11,755 lbs |
| Specific Weight of Fuel | 6.5 lbs/gal |
| Total Usable Fuel | 583 gal (3790 lbs) |
| Cruise Speed | .88 Mach number |
| Cruise $W/\delta_a$ | 58,000 lb |
| Fuel Reserve (5 percent of initial fuel plus fuel for 20 minutes loiter) | 805 lbs |

| | Fuel Used Lbs | Distance Travelled NAM | Elapsed Time From Brake Release-Min |
|---|---|---|---|
| Takeoff Allowance (fuel for 5 minutes at NRP at sea level | 350 | – | 2.5 |
| Military Power Climb to 39,700 Feet | 490 | 79 | 12.5 |
| Cruise | 2145 | 875 | 116.5 |
| Totals | 2985 | 954 | 131.5 |

[2]Weight-pressure ratio ($W/\delta_a$) is defined as $\frac{\text{Aircraft weight} \times 29.92126}{\text{Ambient pressure}}$.

ACTUAL FERRY RANGE MISSION

| | |
|---|---|
| Gross Weight at Engine Start | 12,330 Lbs |
| Specific Weight of Fuel | 6.26 Lbs/Gal |
| Total Usable Fuel | 583 Gals (3650 Lbs) |
| Cruise Speed | .89 Mach number |
| Fuel Reserve | 938 Lbs |
| Cruise $W/\delta_a$ | 56,000 Lbs |

| | Fuel Used Lbs | Distance Travelled NAM | Elapsed Time From Brake Release-Min |
|---|---|---|---|
| Start, Taxi, Takeoff, and Acceleration to Climb Schedule | 482 | | |
| Military Power Climb to 38,500 Feet | 427 | 68 | 10.45 |
| Cruise | 1803 | 679 | 81.19 |
| Totals (for 938 lbs fuel remaining) | 2712 | 747 | 91.64 |
| Totals (corrected to 3790 lbs of fuel available, a take-off allowance of 350 lbs and a reserve of 805 lbs and corrected for removal of test nose boom and inlet duct bands) | 2985 | 928 | 111.1 |

Significant factors affecting the test range mission were as follows:

1. The gross weight of the test aircraft was 575 pounds heavier than production aircraft.
2. Fuel available was 3650 pounds (583 gallons at 6.26 pounds/gallon). The computed mission was based on 583 gallons at 6.5 pounds/gallon (3790 pounds).
3. At the start of the test mission an abnormally long delay was experienced before takeoff (due to a local emergency) which resulted in more fuel being used than is normally required.
4. The test mission was terminated with 938 pounds of fuel remaining.

When the mission was corrected to 3790 pounds of fuel available, 350 pounds allowance for taxi and takeoff, and 805 pounds allowance for reserve and SFC was corrected 2 percent for removal of test nose boom and inlet duct bands, a range of 928 nautical miles resulted. This range is within 3.0 percent of the range of the production aircraft as computed from specific range data in Appendix I.

Level flight speed-power data to define airplane drag and specific range (nautical air miles traveled per pound of fuel used) is presented in Appendix I for the following configuration:

1. Cruise configuration, Figures 10, 11, and 13.
2. Speed brake extended, Figures 12 and 13.
3. Single-engine cruise configuration, Figures 14 and 15.
4. Landing gear extended, 0, 45, and 100 percent flaps Figures 16 and 17.

Maximum level flight speed with military power varies from approximately .94 Mach number at sea level to approximately .96 Mach number at 36,000 feet.

Maximum level flight speed with maximum power varies from approximately .98 Mach number at sea level to slightly greater than 1.2 Mach number at 36,000 feet. However, acceleration is slow at speeds above 1.1 Mach number and considerable fuel is used to realize the maximum speed capability of the aircraft. This data is presented in Figures 19 through 24, Appendix I.

## DESCENTS

Idle power descents were accomplished at 220, 260 and 300 knots IAS and at .7, .75, .8 and .85 Mach number to define descent characteristics. The effect of speed brake extension is shown for 220 and 260 knots IAS, and .7 and .85 Mach number descents. Figures 27 through 33, Appendix I, present the descent data for the various conditions tested.

## LANDINGS

Landings were accomplished on a dry concrete runway using varying degrees of braking. Data was obtained with flaps retracted and with 45, 60 and 100 percent flaps. All performance landings were accomplished by lowering the nose immediately after touchdown and applying steady braking until the aircraft stopped.

At a gross weight of 9200 pounds, a landing roll of 3000 feet was attained with maximum braking. At this same weight, heavy braking produced ground roll of approximately 3500 feet. This performance was achieved using full flaps and a touchdown speed of 130 knots IAS. Higher touchdown speeds and/or moderate braking resulted in ground roll distances approximately 1000 feet greater than those shown in the Flight Manual.

Test results showed that, in order to achieve stopping distances as shown in the Flight Manual for normal landings, touchdown must be made at the Flight Manual recommended airspeed and heavy braking must be used. The heavy braking required is only slightly less than maximum practical braking and is considerably heavier than would be used for a normal landing. Tests utilizing maximum practical braking resulted in ground roll distances approximately 300 feet shorter than Flight Manual data for normal landings.

Landing performance is presented in the following table.

LANDING PERFORMANCE

**Dry concrete runway, sea level, standard day, no wind conditions.**

| True Airspeed at T.D. Knots | Ground Roll Feet | True Airspeed at 50 Ft Height | Total Distance Feet | Gross Weight Lbs | Flaps % | Braking |
|---|---|---|---|---|---|---|
| 133 | 4,980 | 163 | 10,010 | 9,350 | 0 | Moderate |
| 143 | 4,410 | 167 | 9,290 | 9,200 | 0 | Moderate |
| 129 | 3,650 | 154 | 7,530 | 9,300 | 45 | Heavy |
| 130 | 4,650 | 146 | 9,540 | 9,750 | 45 | Moderate |
| 128 | 4,110 | 151 | 6,980 | 9,400 | 45 | Moderate |
| 130 | 3,440 | 165 | 9,000 | 9,200 | 45 | Heavy |
| 130 | 4,620 | | | 10,350 | 45 | Moderate |
| 133 | 4,510 | | | 9,350 | 45 | Moderate |
| 134 | 3,430 | 169 | 8,440 | 9,250 | 60 | Heavy |
| 135 | 4,550 | 161 | 8,100 | 9,200 | 60 | Moderate |
| 136 | 4,750 | 162 | 9,170 | 9,450 | 60 | Moderate |
| 136 | 3,790 | 168 | 6,980 | 10,800 | 60 | Maximum Practical |
| 122 | 3,100 | 154 | 5,900 | 9,250 | 100 | Maximum Practical |
| 127 | 3,490 | 147 | 5,650 | 9,350 | 100 | Heavy |
| 144 | 5,560 | 147 | 6,230 | 10,300 | 100 | Maximum Practical |
| 128 | 3,900 | 147 | 7,150 | 9,650 | 100 | Moderate |
| 129 | 3,390 | 155 | 5,950 | 9,100 | 100 | Heavy |
| 129 | 3,970 | 146 | 8,050 | 9,600 | 100 | Moderate |
| 128 | 3,000 | 145 | 6,480 | 9,300 | 100 | Maximum Practical |
| 129 | 5,630 | | | 10,550 | 100 | Moderate |

## PERFORMANCE GUARANTEES

The guaranteed performance of the T-38A as specified in Norair Model Specification NS-140C, paragraph 3.1.2 is met in all cases. The performance guarantees and comparable test results are as follows:

| | | Guarantee | Test Results |
|---|---|---|---|
| 1. | Maximum level flight Mach number at 36,089 feet and at the design gross weight of 9,592 pounds. | 1.2 | Approximately 1.21 |
| 2. | Cruising Mach number for long range cruise, not less than | 0.85 | 0.88 |
| 3. | Range at cruise Mach number as specified in item (b), using military thrust takeoff and climb, 20 minute sea level loiter and 5% fuel reserve | 880 NAM | 886 NAM |
| 4. | Maximum thrust takeoff ground run at sea level at gross weight required to perform item(c). | 2,500 Ft | 2,300 Ft* |

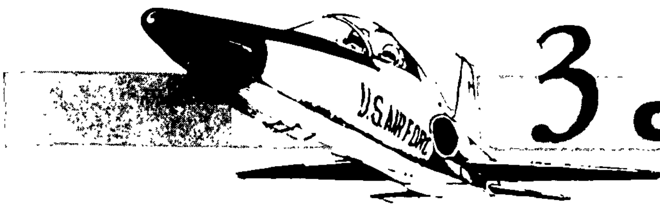
*This ground roll was obtained at a gross weight of 12,000 pounds with a flap setting of 100 % and at the AFFTC recommended takeoff speed of 153 knots TAS.

The above test range includes 5 percent conservatism.

## CALIBRATIONS

Airspeed system calibrations were accomplished with a test nose boom installed. The results of these calibrations are presented in Figures 34 and 35, Appendix I.

Installed static thrust and calibrations were accomplished and the results are presented in Figures 37 and 38, Appendix I.



# 3 CONCLUSIONS...

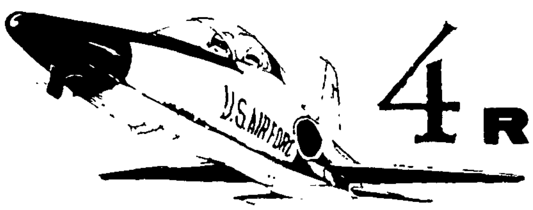
The T-38A is an excellent trainer for transition into high performance aircraft. It has excellent handling characteristics, maximum speed is in the low supersonic range, and takeoff and subsonic climb performance is comparable to the latest Century Series fighters.

With maximum power, the aircraft can readily takeoff in 2,500 feet at 12,000 pounds gross weight, can climb from sea level to 40,000 feet in 2.5 minutes, and can attain a combat ceiling (500 feet per minute rate of climb) of 51,000 feet.

Single engine performance of the aircraft is also good. With maximum power, a rate of climb of 5,500 feet per minute is available at sea level and the combat ceiling is 37,000 feet.

The maximum ferry range of the aircraft is 954 nautical miles. This range is attained when cruising at a $W/\delta_a$ of 58,000 pounds and a Mach number of 0.88[3].

[3] Computed in accordance with landing reserves of MIL-C-5011A from level flight speedpower data.




# 4 RECOMMENDATIONS...

The performance presented in this report should be reflected in the Appendix of the Flight Manual.

This report concurs in the recommendations previously published in AFFTC-TR-61-15 "T-38A Category II Stability and Control Tests" and AFFTC-TR-61-21 "T-38A Category II Systems Evaluation Test." All recommendations generated during the Category II performance tests were included in the above reports and have received System Project Office attention; therefore, they are not repeated in this report.

# APPENDIX I....

## DATA ANALYSIS METHODS

● REFERENCES


1. Air Force Technical Report No. 6273, "Flight Test Engineering Manual."

2. AFFTC-TN-59-22, ASTIA Document No. AD-215865, "AFFTC Performance Flight Test Manual, Part II."

3. AFFTC-TR-59-47, "AFFTC Performance Flight Test Manual I, Part I."

4. "AGARD Flight Test Manual, Volume I, Performance."

5. General Electric S.A.E.D. Report No. 673, J85-5 inflight thrust calculation procedure.

6. Air Force Flight Test Center Technical Note R-12, "Standardization of Take-Off Performance Measurements for Airplanes."

7. FTFE-TM-58-6, AFFTC Technical Memorandum, "Tower Fly-By Facility and Technique for Determining Pitot-Static Position Error."

8. General Electric Company Model Specification E1024-B Engine, Aircraft, Turbojet J85-GE-5.

9. NOR-60-350 "Flight Manual Performance USAF Model T-38A Trainer Airplane with Two J85-GE-5 Engines."

## SPECIFIC ENERGY METHOD

The climb and acceleration data presented in this report were processed in the following manner to obtain rate of change of specific energy under test conditions. Geometric altitude for the pressure altitude increment under consideration was determined utilizing the procedures outlined in Chapter 4, Sections 10 and 11, of Reference 4. The total specific energy based on the first data point of each test was determined by equation 5.702 of Reference I. It should be understood that the energy obtained by this method is not an exact energy as referenced to a fixed earth system, but a specific energy based on the first point of each individual test. This procedure renders the energy levels incorrect but supplies a valid rate of change of specific energy for subsequent calculation.

The variation of specific energy was determined by considering seven consecutive energy values as a function of time and determining a representative quadratic equation by the method of least squares. Special consideration was necessary for end points and points which did not lie within three standard deviations of the quadratic constructed. Rate of change of specific energy was determined by differentiation of the constructed quadratic equation of each of the time values under consideration.

## STANDARD ATMOSPHERE

The data presented in this report has been corrected to the US Standard Atmosphere as defined in Reference 2.

### Takeoff Performance:

The corrections to ground roll and air distance for wind velocity, air density, engine thrust, and aircraft weight were made according to Reference 6.

Because of the fluctuations of nozzle area and related engine parameters immediately after brake release, test thrust for takeoff was arbitrarily measured at 75 knots indicated airspeed. Test thrust/ $\delta_a$ was then plotted against ambient temperature to determine thrust variation with ambient temperature. Standard thrust was then defined as that thrust corresponding to stand- -d temperature at sea level (15 degree ). However, it should be noted that . thrust values indicated on Figure I wer ; static values rather than the above described dynamic thrust values used for correcting the test data to standard conditions.

## CLIMBS

Continuous climbs were accomplished from an altitude of approximately 5,000 feet to as high an altitude as practical. Data was recorded in approximately one second intervals throughout the climb but was transcribed to digital information as a function of altitude to furnish a suitable sample of information for presentation. Basic relations utilized to determine Mach number, altitude, etc., were found in Reference 2. Rate of change of energy at test conditions was determined as outlined previously in this Appendix under "Specific Energy Method." It was apparent that the reduction as outlined included climb potential that would be available only when following an unaccelerated climb schedule. The corrections to climb potential when following the velocity changes indicated by the climb schedule were determined in the following manner:

The velocity gradient as a function of altitude was determined from the climb schedule. This gradient was then multiplied by the time rate of change of altitude at the point concerned. The time rate of change of altitude was obtained by the method of least squares quadratic curve fit as explained under "Specific Energy Method." The two functions, velocity gradient versus altitude and time rate of change of altitude, were multiplied to determine velocity change at the point being considered. The result was multiplied by the velocity of the vehicle at test conditions to obtain the climb potential correction to be added to the unaccelerated rate of climb.

The rate of climb thus obtained was that available under test conditions and can be altered to reflect standard atmospheric conditions by multiplication with $\sqrt{T_{a_s}/T_{a_t}}$.

The thrust increment determined by the difference between test and standard temperature was obtained as outlined in this Appendix, under "Total Net Thrust Measurement." This increment of thrust was converted to rate of climb by multiplication with the ratio of true airspeed divided by aircraft weight. This rate of climb increment was added to the climb potential available in a standard atmosphere while the vehicle is following the velocity gradient as a function of altitude dictated by the climb schedule. This information is presented as rate of climb in the Test Results section of this report.

It should be noted that the climb data has not been corrected for changes in the following variables: aircraft weight, Mach number, normal acceleration and wind gradient effects.

## ● MILITARY POWER ACCELERATIONS

Military power accelerations were accomplished at various altitudes from near minimum flying speed to the thrust limited Mach number under test conditions. The thrust available in military power at these altitudes is presented in Figures 19 through 24. The data was obtained by the methods outlined in "Total Net Thrust Measurment of this Appendix.

## ● MAXIMUM POWER ACCELERATIONS

Maximum power accelerations were accomplished from near minimum speeds to the maximum speed under test conditions. Data was recorded in approximately one second intervals throughout the acceleration but was transcribed to digital information as a function of Mach number to furnish a suitable sample of information for presentation. Basic relations utilized to determine Mach number, altitude, etc., were found in Reference 2. Rate of change of energy at test conditions was determined as outlined previously in this Appendix under "Specific Energy Method" and converted to rate of change of energy in a standard atmosphere by multiplication $\sqrt{T_{a_s}/T_{a_t}}$.

The induced drag correction to rate of climb is:

$$\Delta R/C_3 = \left[\frac{25.33 T_{a_s}}{P_{a_s} M b^2 e}\right]\left[\frac{n\left(W_t^2 \frac{\delta a_s}{\delta a_t} - W_s^2\right)}{W_s}\right]$$

Excess thrust at lest conditions was obtained from the following expression:

$$T_{ex_t} = \frac{R/C_t\, W_t}{101.333\, V_{T_t}}$$

The induced drag was defined as follows:

$$D_i = \frac{(C_{L_t} - \Delta C_L)^2}{\pi AR\, e} \times \frac{M^2 S \delta_{a_t}}{.000675}$$

$\Delta C_L$ is defined in Figure 5, of Reference 10.

The drag correction for change in altitude is expressed utilizing the two previously defined parameters:

$$\Delta R/C = \frac{101.3333 V_{T_s}}{W_s} \times \Delta D_{t-s}$$

The thrust increment determined by the difference in thrust between test and standard atmosphere was obtained as outlined in this Appendix under "Total Net Thrust Measurement." This thrust increment, as were all thrust and drag increments, was converted to rate of climb as follows:

$$R/C_{power} = \frac{\Delta F_{n_{powers-t}} \times V_{T_s} \times 101.333}{W_s}$$

The resultant rate of climb with the above corrections added was then corrected for change of inertia by multiplying the rate of climb by the ratio of test to standard weight.

The final rate of climb was converted to excess thrust as follows:

$$T_{ex_s} = \frac{R/C_s\ W_s}{101.3333\ Vt_s}$$

The thrust available and thrust required is presented at a gross weight of 10,000 pounds.

It should be noted that the acceleration data is presented at the normal load factor obtained during the particular test.

## ● CRUISE

Stabilized speed-power data was obtained at a constant weight-pressure ratio by flying successive points at higher altitudes to compensate for the weight reduction with fuel consumed. This minimized the correction of the data obtained.

Basic relations utilized to determine Mach number, altitude, etc., were found in Reference 2.

Thrust was calculated by the methods outlined in Reference 5.

Corrections for being off the desired weight-pressure ratio were made by an increment of lift coefficient and drag coefficient as defined by the drag polars Figures 13, 15, and 17.

Norair has estminated that the interim structural duct bands in early aircraft decrease the specific range of the aircraft approximately one percent (as compared to production aircraft) and that installation of a test nose boom in place of a production nose boom also decreases the specific range one percent. Based on these estimates, the range mission as shown in the body of this report for a typical production aircraft was computed from specific range data that was two percent higher than the data obtained from the test aircraft.

## ● DESCENTS

Descents were accomplished at several constant calibrated airspeed and constant Mach number schedule from approximately 40,000 feet to 10,000 feet.

The rate of descent data was corrected to standard atmospheric conditions. However, no attempt was made to correct rate of descent for thrust variations from test to standard conditions or for weight variations.

## ● LANDING PERFORMANCE

Landing data was obtained and processed as outlined in Reference 6, The ground roll distance is presented for a test gross weight as shown and is corrected to sea level standard atmosphere as outlined in Reference.

## ● COMPRESSOR INLET PRESSURE ($P_{t_2}$)

Average compressor inlet total pressure data was obtained from a pressure survey rake located immediately ahead of the compressor on the left engine only. $P_{t_2}$ for the right engine was assumed to be the same as the left when engine speeds (air flow) were the same. However, when the right engine was operating at a different power setting than the left, the right $P_{t_2}/P_{t_0}$ was modified by the following procedure: $\left(\frac{W_a \quad \theta_{t_2}}{K_{RN1} \; \delta_{t_2}}\right)_{\text{inlet}}$ was computed for both right and left engines.

$$P_{t_2}/P_{t_0} = f\left(\frac{W_a \sqrt{\theta_{t_2}}}{K_{RNI} \; \delta_{t_2}}\right)_{\text{inlet}}$$

(Figure numbers 42 and 43)

Enter the above curve at the corrected inlet airflow values computed for the left and right engine and extract

$(P_{t_2}/P_{t_0})_{\text{left}}$ and $(P_{t_2}/P_{t_0})_{\text{right}}$

$$\Delta \frac{P_{t_2}}{P_{t_0}}_{L-R} = (P_{t_2}/P_{t_0})_L - (P_{t_2}/P_{t_0})_R$$

$$\Delta P_{t_{2_{L-R}}} = \Delta\left(\frac{P_{t_2}}{P_{t_0}}\right)_{L-R} \times P_{t_0}$$

Then $P_{t_a}$ for the right engine, based on $P_{t_2}$ measured from the inlet rake, is as follows:

$$P_{t_{2_R}} = P_{t_{2_{L(\text{inlet rake})}}} + (\Delta P_{t_2})_{L-R}$$

In some cases, when the $P_{t_2}$ rake was inoperative, $P_{t_2}/P_{t_0}$ was obtained from Figures 42 and 43 as a function of corrected inlet air flow.

## ● TOTAL NET THRUST MEASUREMENT

Total gross and net thrust was obtained by the method outlined in Reference 5.

The nozzle area was measured by instrumenting the nozzle actuating teleflex cable. This area was then corrected for pressure and temperature growth by use of a calibration curve (Appendix III). The resultant area was then assumed to be the true area for subsequent thrust calculations.

Primary and bypass air flow as obtained from curves supplied by the engine and airframe manufacturers which are presented in Figures 41 and 48, respectively.

From the total inlet airflow, total ram drag was computed as follows:

$$F_e = \frac{W_a x V_{t_t} \times 1.6889}{g}$$

Primary net thrust was obtained by:

$$F_{n_p} = F_g - F_e$$

Secondary thrust was obtained from the manufacturer's supplied curves, Figures 43 through 46.

Total net propulsive thrust was then defined as:

$$F_{n_{test}} = F_{n_p} + F_{n_{sec}}$$

Standard net thrust for maximum and military power was obtained by the following procedure:

1. Enter curves from a GE Specification of $F_n$ versus $T_{t_2}$ with test and standard $T_{t_2}$ and extract corresponding values of $F_n$.

2. $\triangle F_{n_{s-t}} = F_{n_{std}} - F_{n_t}$(from curves)

3. Standard $F_{n_s} = F_{n_t}$(computed) $+\triangle F_{n_{s-t}}$

The curves of $F_n$ versus $T_{t_2}$ at various attitudes were obtained from Reference 8.

● FUEL FLOW

Fuel flow data was recorded separately for each engine and afterburner on an oscillograph.

Test fuel flow in gallons per hour was obtained by use of calibration curves of gallons per cycle versus cycles per second.

This test fuel flow was then converted to pounds per hour at the appropriate fuel temperatures as measured at the fuel transmitters.

Standard fuel flow for tests using maximum or military power was obtained by the general method used to obtain thrust as described in the paragraph "Total Net Thrust Measurement."

● AMBIENT TEMPERATURE

The ambient air temperature measuring system in the test aircraft was carefully calibrated and compared to rawinsonde temperature measurements of the upper atmosphere. Results of this comparison were plotted in terms of $(T_{i_c}/T_a)-1$ versus $M^2/5$ (where $T_{i_c}$ is the indicated temperature obtained from the test aircraft and $T_a$ is ambient temperature from rawinsonde). Test ambient temperatures were then extracted from $T_{i_c}$ by the following equation:

$T_a = T_{i_c}(1 + .98 \frac{M^2}{5})$ where .98 was the observed average recovery factor of the test system.

## ● AIRSPEED SYSTEM CALIBRATION

The airspeed system was calibrated using tower fly-by technique as described in Reference 7 and the pacer method as described in Reference 1. Transonic data was obtained by the method described in Section 5.6.3 of Reference 2. Figure 34 presents the results of the test nose boom out-of-ground effect and Figure 35 presents data for the System in ground effect.

## ● STATIC THRUST RUNS

Static thrust calibrations were obtained with the engine installed in the aircraft. The data was obtained primarily to check standard day manufacturer's guaranteed thrust and to check the validity of the thrust measuring procedures used in flight. Engines serial numbers 230184 and 230185 were used throughout the test programs.

The performance of these engines was essentially impossible to generalize, with the test data obtained because of the inability to define standard nozzle area for any given power setting. For this reason, nozzle area was presented for test conditions only (Figures 37 and 38).

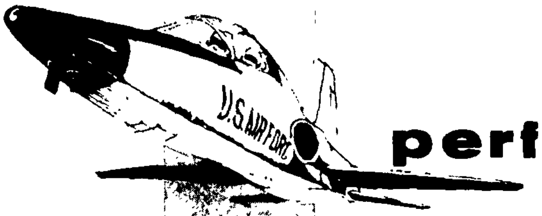
## NOMENCLATURE

| | | |
|---|---|---|
| $A_8$ | exhaust nozzle area | sq in |
| AR | aspect ratio | |
| b | wing span | ft |
| CAS | calibrated airspeed | knots |
| $C_D$ | coefficient of drag | |
| $C_L$ | coefficient of lift | |
| $D_i$ | induced drag | lbs |
| e | airplane efficiency factor | |
| EGT | exhaust gas temperature | °C or °K |
| $F_g$ | gross thrust | lb |
| $F_n$ | net thrust | lb |
| $F_e$ | inlet ram drag | lb |
| $F_{n_{req}}$ | net thrust required for level flight | lb |
| g | acceleration due to gravity | ft/sec/sec |
| $M_{ic}$ | indicated Mach number | |
| M | Mach number, free stream | |
| n | normal load factor | g |
| N | engine speed | rpm |
| NAMPP | nautical air miles travelled per pound of fuel used | mi/lb |
| SFC | specific fuel consumption | $\frac{\text{pounds of fuel/hr}}{\text{pounds of thrust}}$ |
| NAMT | nautical air miles travelled | nautical miles |
| $P_{t_5}$ | turbine exit total pressure | in Hg |
| $P_{t_2}$ | compressor inlet total pressure | in Hg |
| $P_{t_0}$ | free stream total pressure | in Hg |
| $P_a$ | ambient pressure | in Hg |
| $\Delta P/q_{cic}$ | static pressure coefficient for compressible fluid based on q from instrument corrected values only | |
| R/C | rate of climb | ft/min |

| | | |
|---|---|---|
| $S$ | wing area | sq ft |
| $T_{ic}$ | indicated total temperature | °C or °K |
| $T_a$ | ambient temperature | °C or °K |
| $Tt_2$ | compressor inlet total temperature | °C or °K |
| $T_{ex}$ | excess thrust | lb |
| T. O. | take-off | |
| $Tt_5$ | turbine exit total temperature | °C or °K |
| $V_c$(CAS) | calibrated airspeed | knots |
| $V_i$ (IAS) | indicated airspeed | knots |
| $V_t$ | true airspeed | knots |
| W | aircraft gross weight | lb |
| $W/\delta_a$ | weight-pressure ratio | lb |
| $W_f$ | fuel flow | lb/hr |
| $W_a$ | engine airflow | βb/sec |
| $\theta_a$ | ambient pressure ratio, $P_a/29.92$ | |
| $\delta_{t_2}$ | inlet pressure ratio, $P_{t_2}/29.92$ | |
| $\alpha$ | aircraft angle of attack | deg |
| $W_{alB}$ | compressor bleed airflow | lb/sec |
| $\phi$ | Sutherlands Viscosity Index | |
| RNI | Reynolds Number Index | |
| $K_{RNI}$ | Reynolds Number correction factor to engine airflow | |
| $W_{aBP}$ | engine by-pass airflow | lb/sec |
| $\theta_{t_2}$ | $Tt_2/288$ | |

Subscripts s and t refer to standard and test conditions.

Numerical subscripts applied to temperature and pressure data refer to specific locations on the engine where data is obtained.




# performance plots....

FIGURE NO. 1
TAKE-OFF PERFORMANCE
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES
GROSS WEIGHT 12000 LB

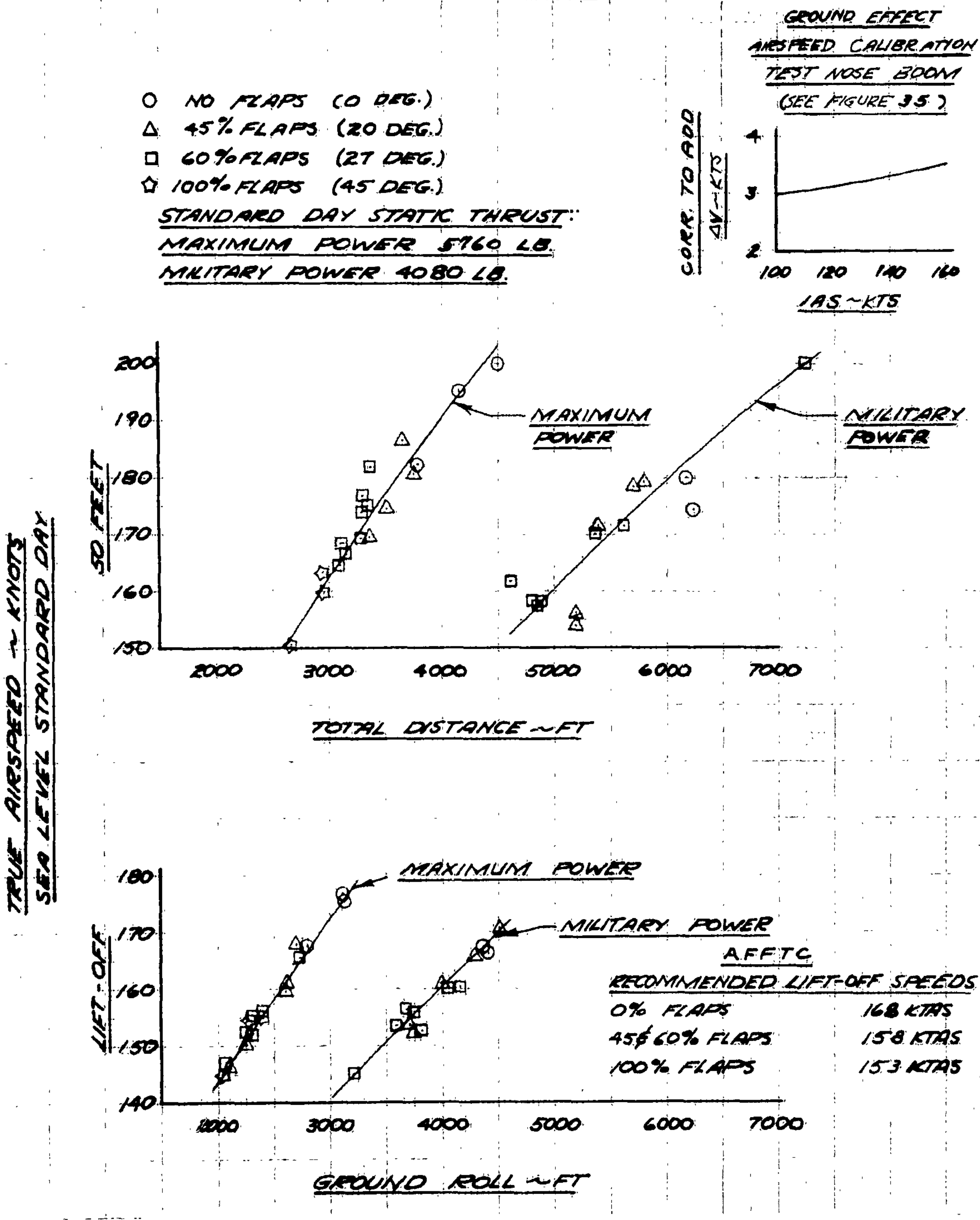


FIGURE NO. 2
MAXIMUM POWER CLIMB POTENTIAL
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230184 AND 230185

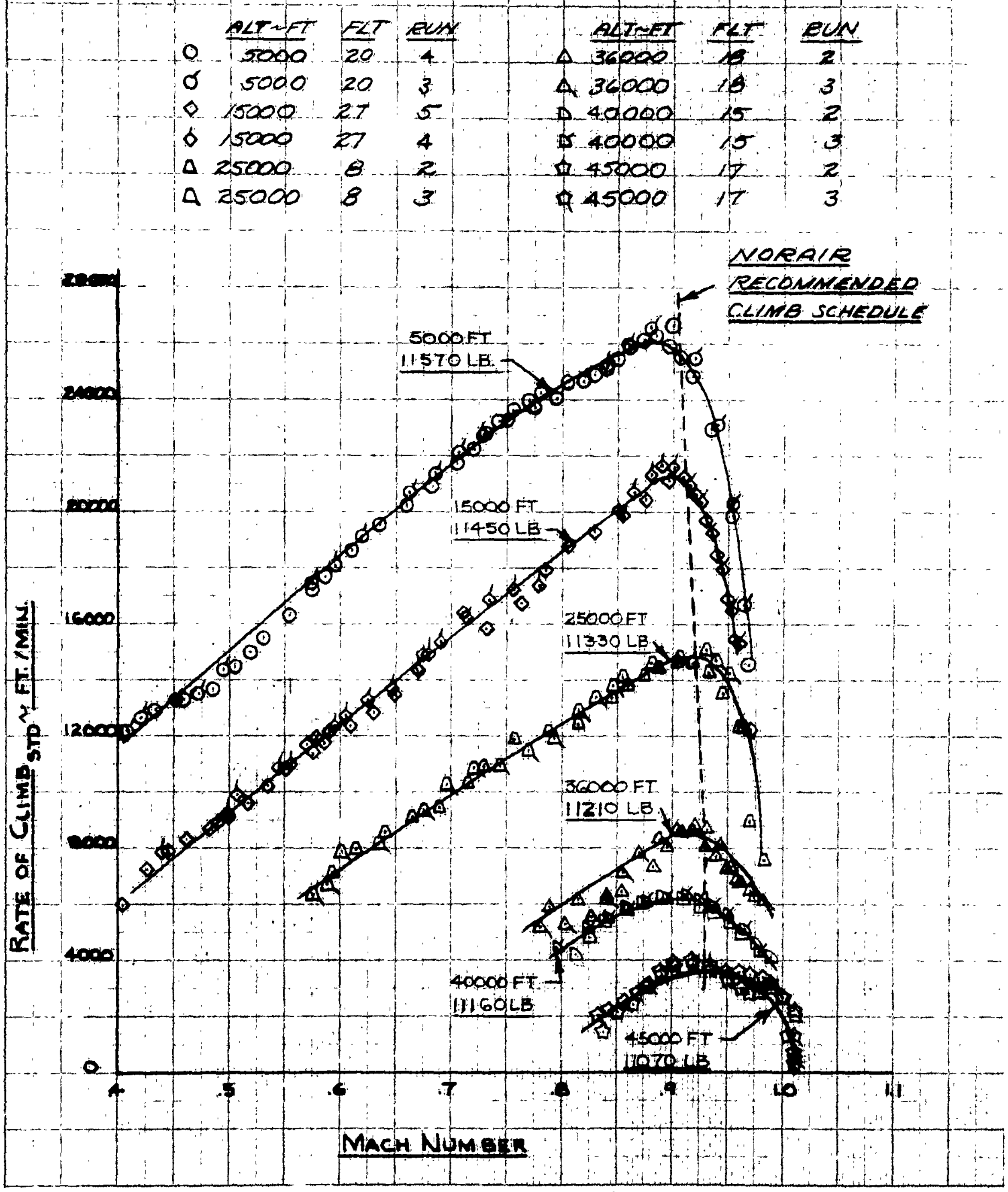


FIGURE NO. 9

MILITARY POWER CLIMB POTENTIAL

T-38A USAF S/N 58-1194

J85-GE-5 ENGINES S/N'S

230184 AND 230185

| | ALT.~FT. | FLT. | RUN | | ALT.~FT. | FLT. | RUN |
|---|---|---|---|---|---|---|---|
| ○ | 5000 | 20 | 5 | △ | 36000 | 24 | 4 |
| Q | 5000 | 20 | 2 | △ | 36000 | 24 | 2 |
| ◇ | 15000 | 23 | 3 | ▽ | 36000 | 24 | 3 |
| ◇ | 15000 | 23 | 4 | D | 40000 | 14 | 4 |
| △ | 25000 | 23 | 6 | D | 40000 | 14 | 5 |
| △ | 25000 | 23 | 5 | ▽ | 40000 | 14 | 3 |

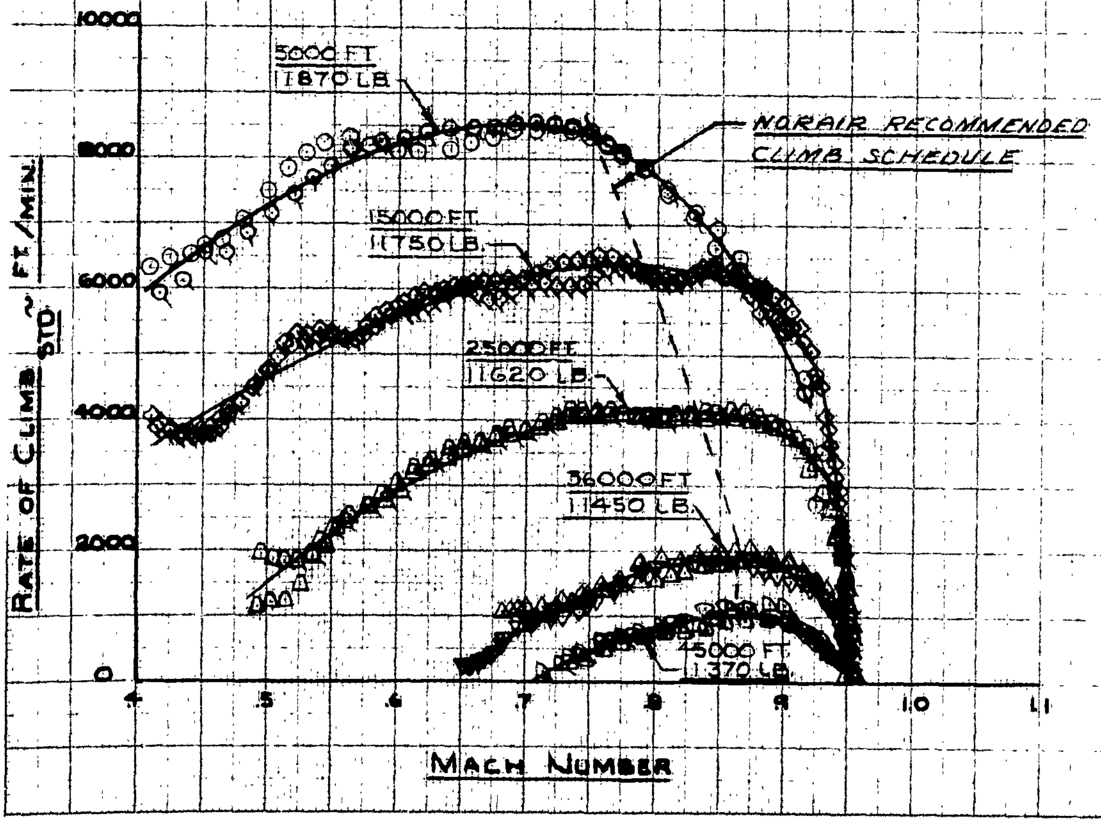


FIGURE NO. 4
MAXIMUM POWER CLIMB PERFORMANCE
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/NS
230184 AND 230185

NOTE:
1. 460 LBS FUEL ARE REQUIRED FOR TAXI, TAKE-OFF AND ACCELERATION TO CLIMB SPEED
2. 1.5 MINUTES ARE REQUIRED TO TAKE-OFF AND ACCELERATE TO CLIMB SPEED

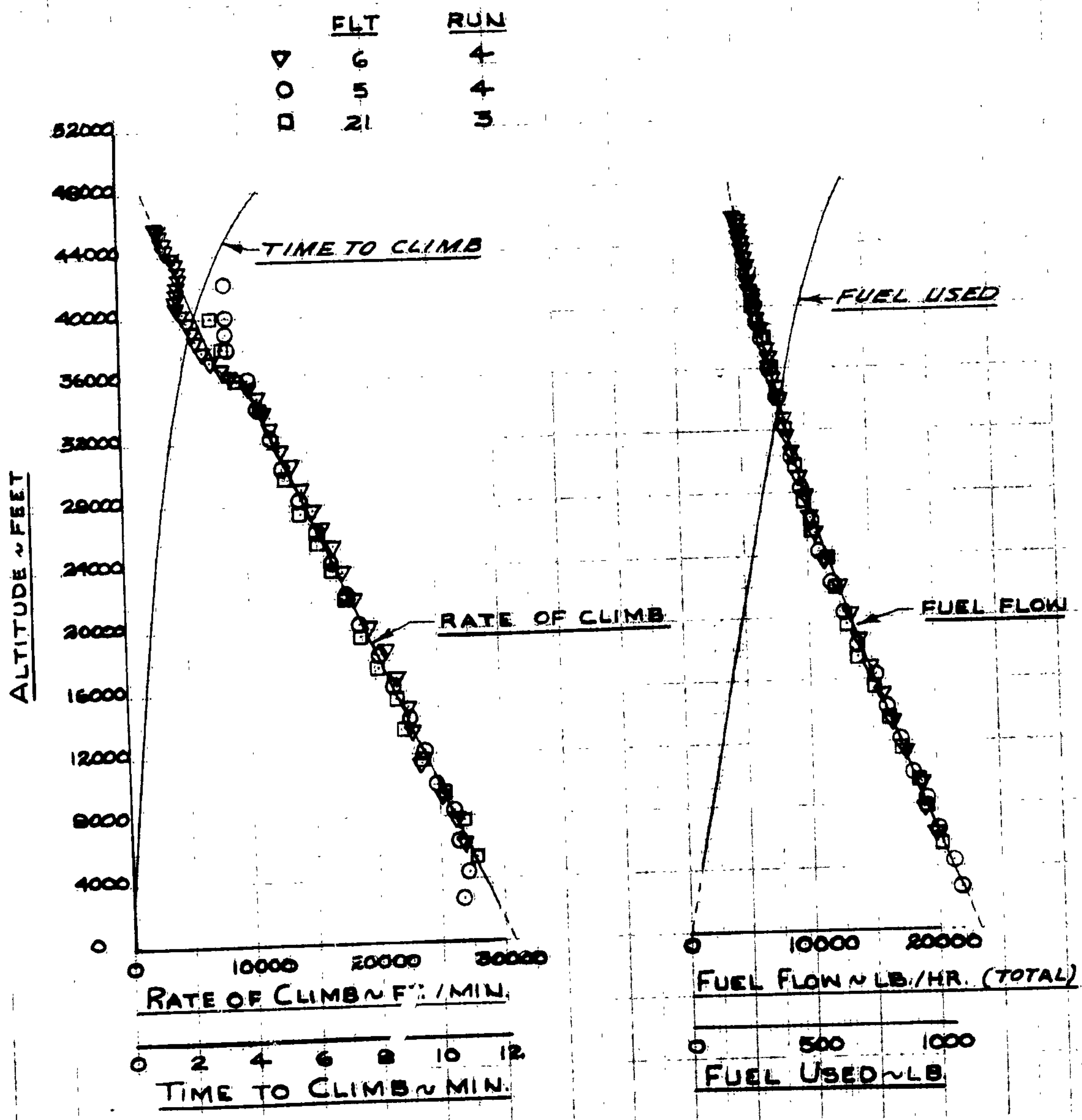


FIGURE No. 4 CONT.

MAXIMUM POWER CLIMB PERFORMANCE

GROSS WEIGHT AT ENGINE START 12330 LB.

NORAIR RECOMMENDED SCHEDULE

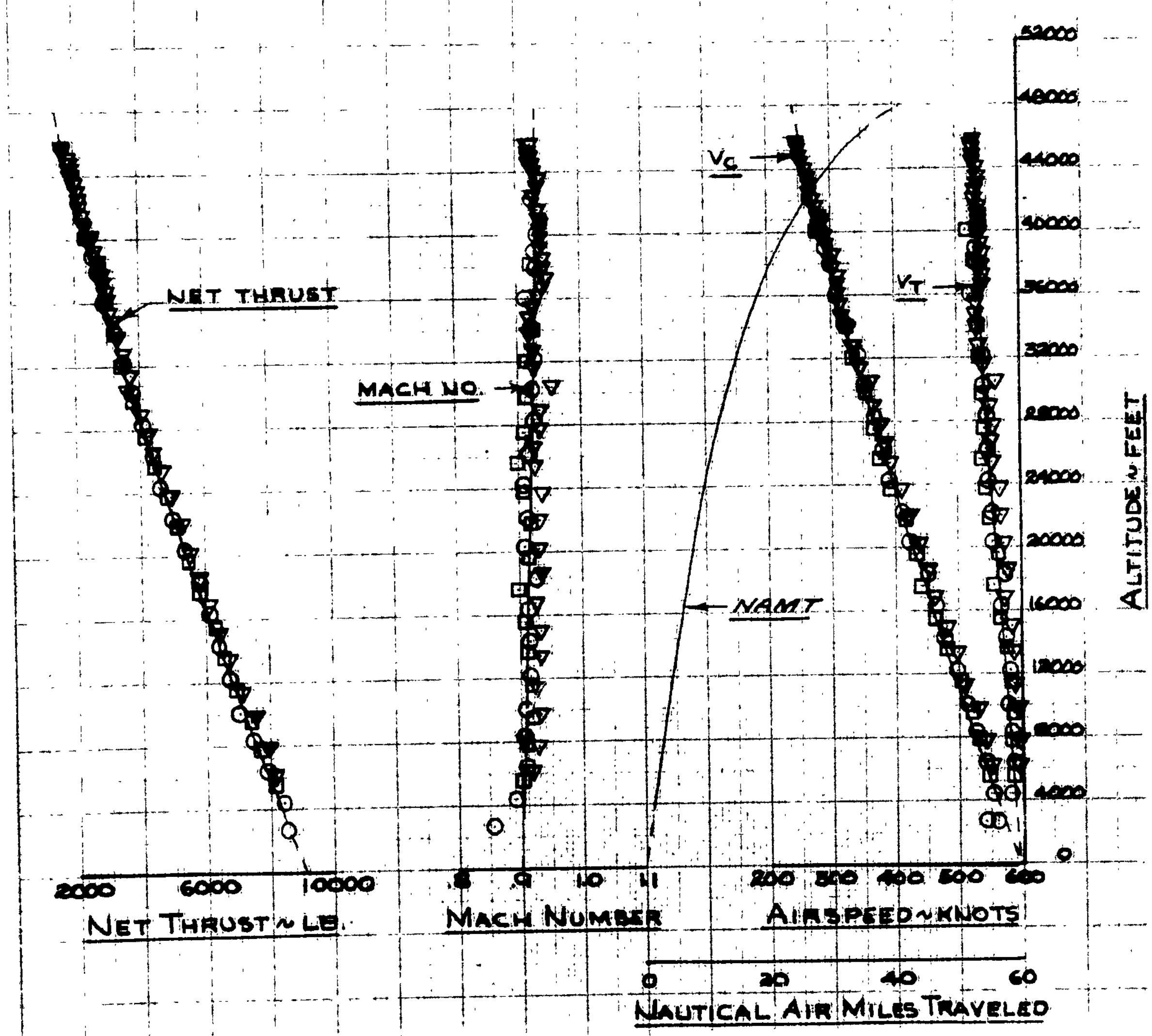


FIGURE NO. 5

MAXIMUM POWER CLIMB PERFORMANCE

T-38A USAF S/N 58-1194

J-85-GE-5 ENGINES S/N'S

230184 AND 230185

NOTE:

1. 360 LBS. OF FUEL ARE REQUIRED FOR TAXI TAKE-OFF AND ACCELERATION TO CLIMB SPEED.
2. 1.3 MINUTES ARE REQUIRED TO TAKE-OFF AND ACCELERATE TO CLIMB SPEED.

| | FLT. | RUN |
|---|---|---|
| ◇ | 24 | 9 |
| ⬡ | 26 | 8 |

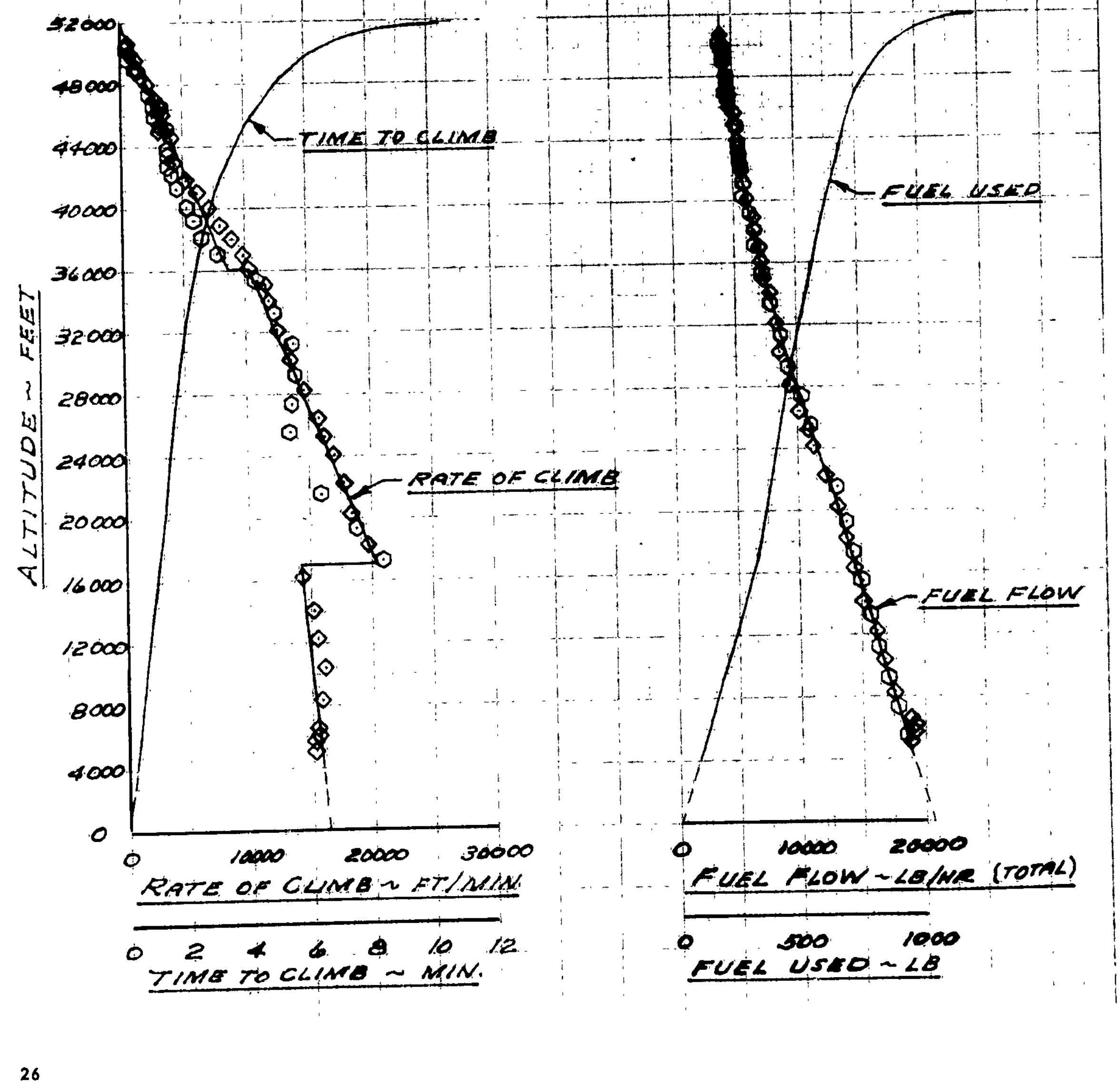
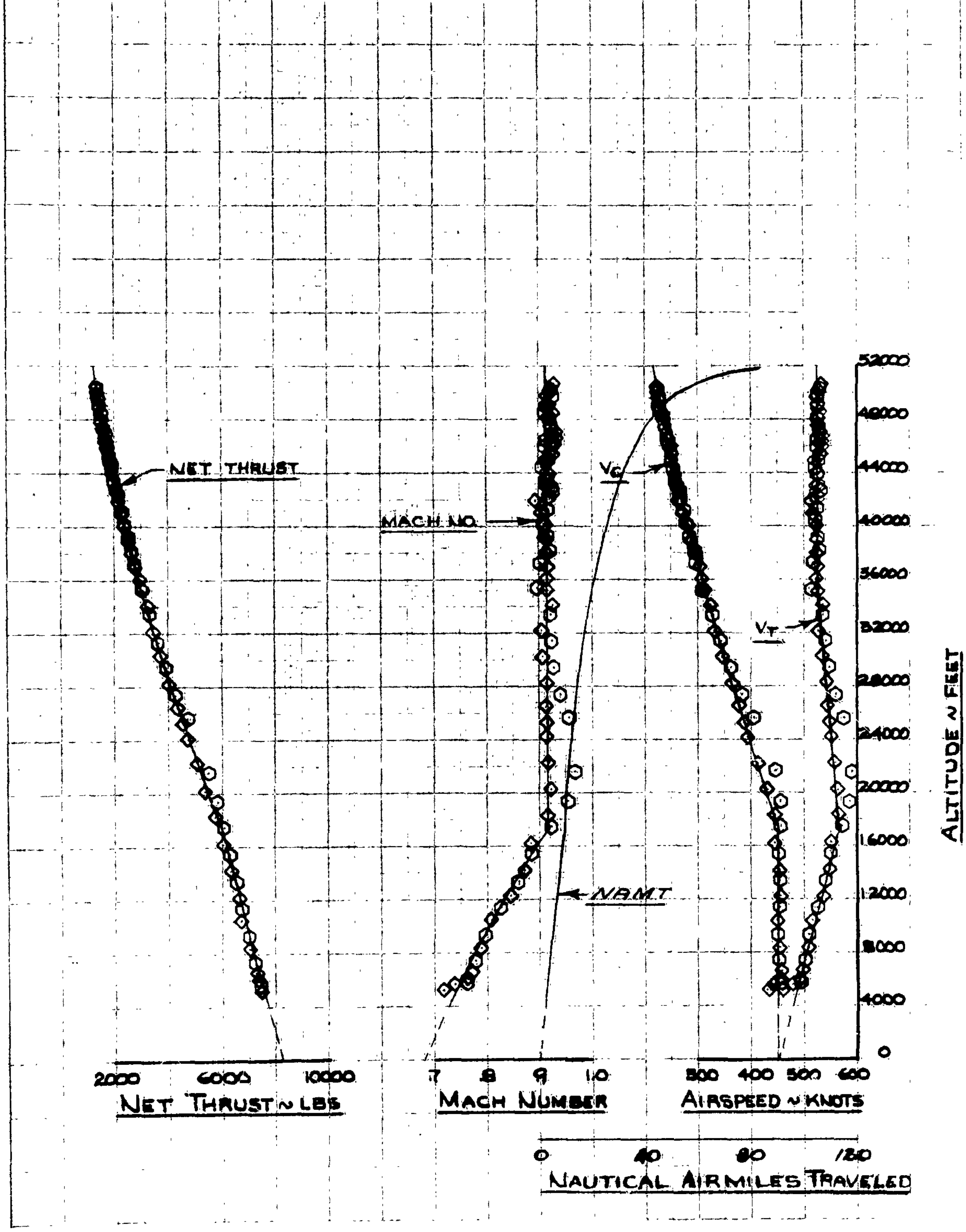


FIGURE NO. 5 CONT.

MAXIMUM POWER CLIMB PERFORMANCE

GROSS WEIGHT AT ENGINE START 12330 LB.

A.T.C. SUGGESTED SCHEDULE



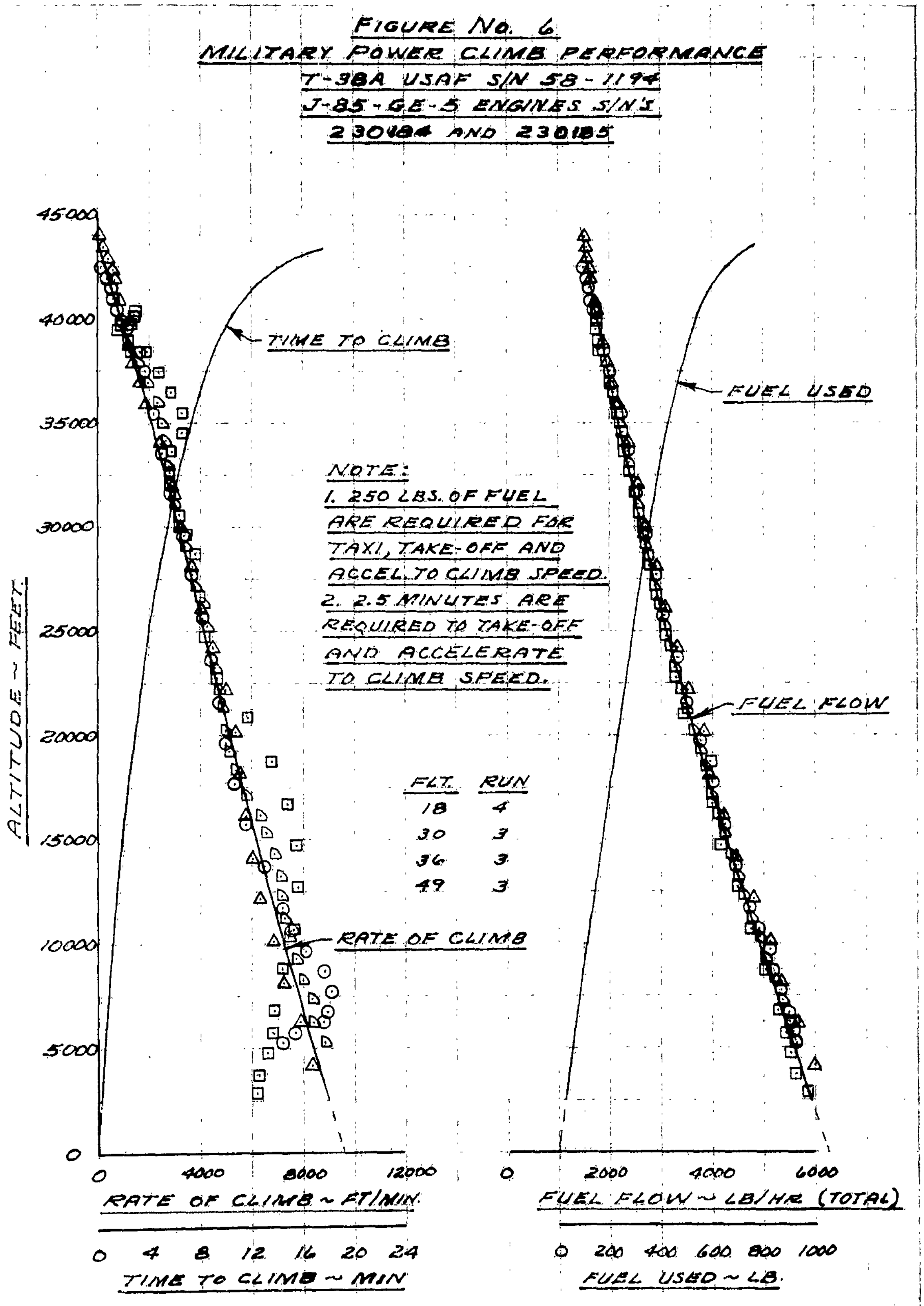
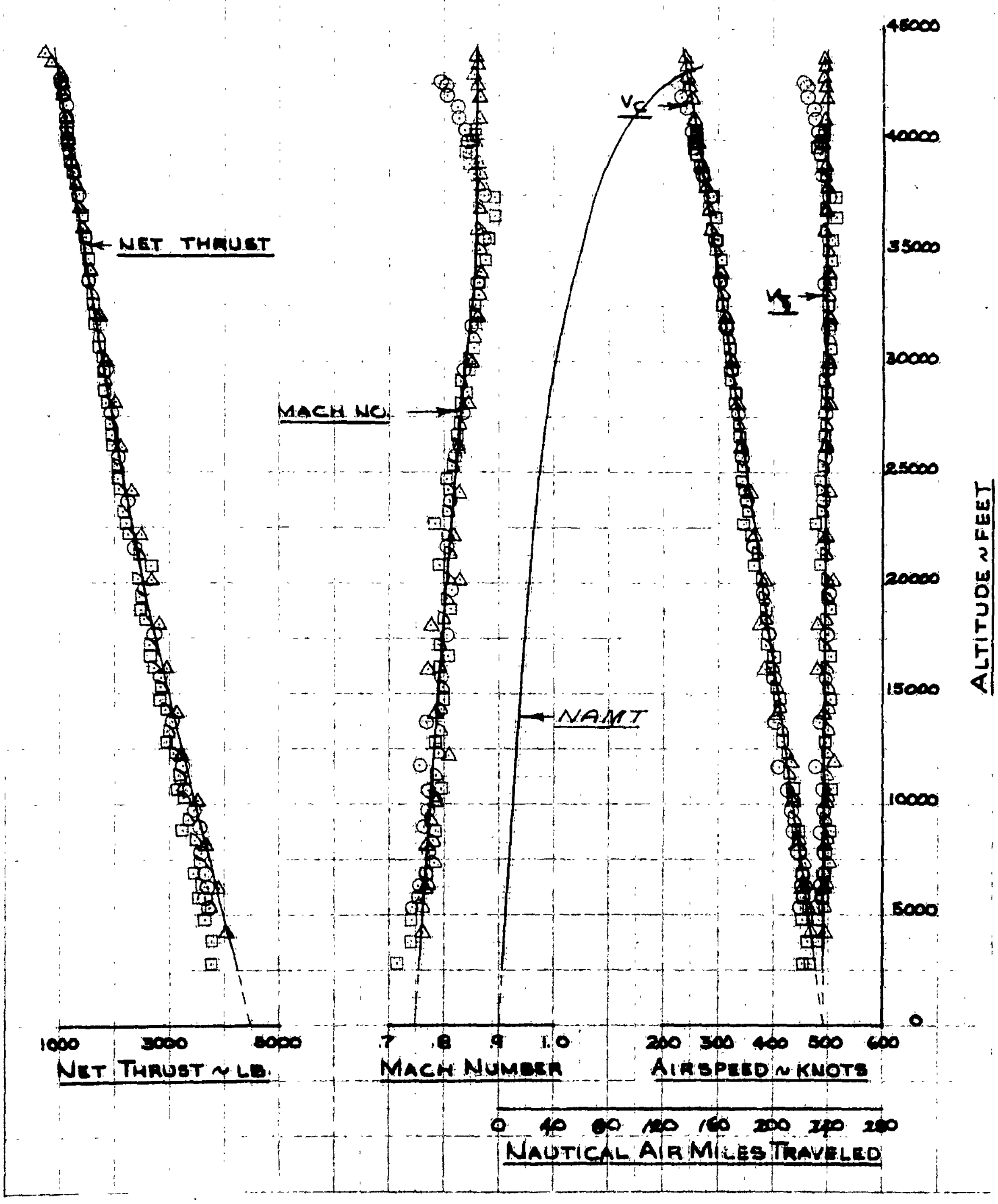


FIGURE No. 6
MILITARY POWER CLIMB PERFORMANCE
T-38A USAF S/N 58-1194
J-85-GE-5 ENGINES S/N'S
230184 AND 230185
ALTITUDE ~ FEET
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
TIME TO CLIMB
RATE OF CLIMB
FUEL USED
FUEL FLOW
NOTE:
1. 250 LBS. OF FUEL ARE REQUIRED FOR TAXI, TAKE-OFF AND ACCEL. TO CLIMB SPEED.
2. 2.5 MINUTES ARE REQUIRED TO TAKE-OFF AND ACCELERATE TO CLIMB SPEED.
FLT. RUN
18 4
30 3
36 3
49 3
0 4000 8000 12000
RATE OF CLIMB ~ FT/MIN.
0 4 8 12 16 20 24
TIME TO CLIMB ~ MIN
0 2000 4000 6000
FUEL FLOW ~ LB/HR (TOTAL)
0 200 400 600 800 1000
FUEL USED ~ LB.

FIGURE NO. 6 CONT.
MILITARY POWER CLIMB PERFORMANCE
GROSS WEIGHT AT ENGINE START 12330 LB.
NORAIR RECOMMENDED SCHEDULE



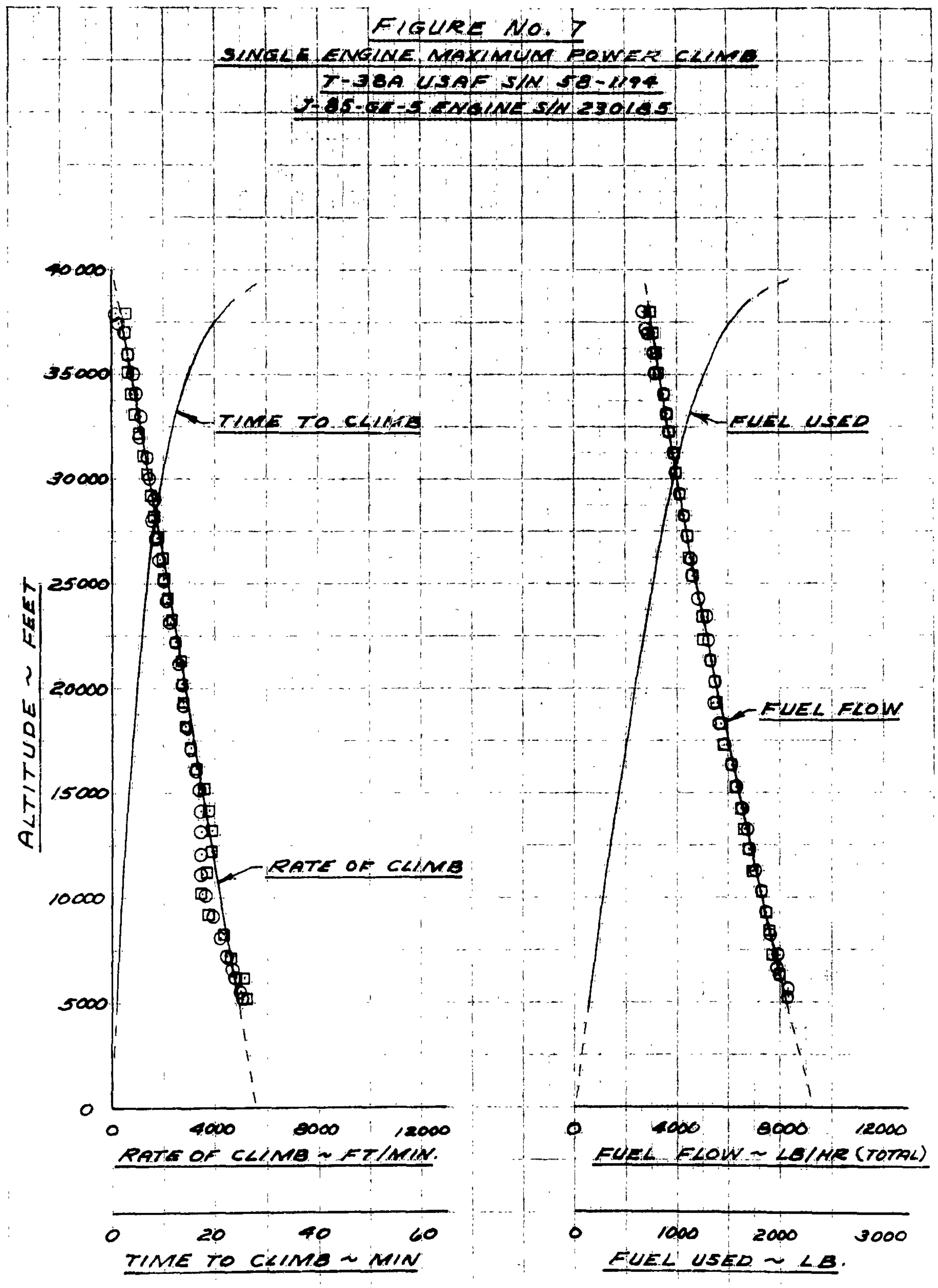


FIGURE No. 7
SINGLE ENGINE MAXIMUM POWER CLIMB
T-38A USAF S/N 58-1194
J-85-GE-5 ENGINE S/N 230185
ALTITUDE ~ FEET
40000
35000
30000
25000
20000
15000
10000
5000
0
TIME TO CLIMB
RATE OF CLIMB
FUEL USED
FUEL FLOW
0
4000
8000
12000
RATE OF CLIMB ~ FT/MIN.
0
20
40
60
TIME TO CLIMB ~ MIN
0
4000
8000
12000
FUEL FLOW ~ LB/HR (TOTAL)
0
1000
2000
3000
FUEL USED ~ LB.

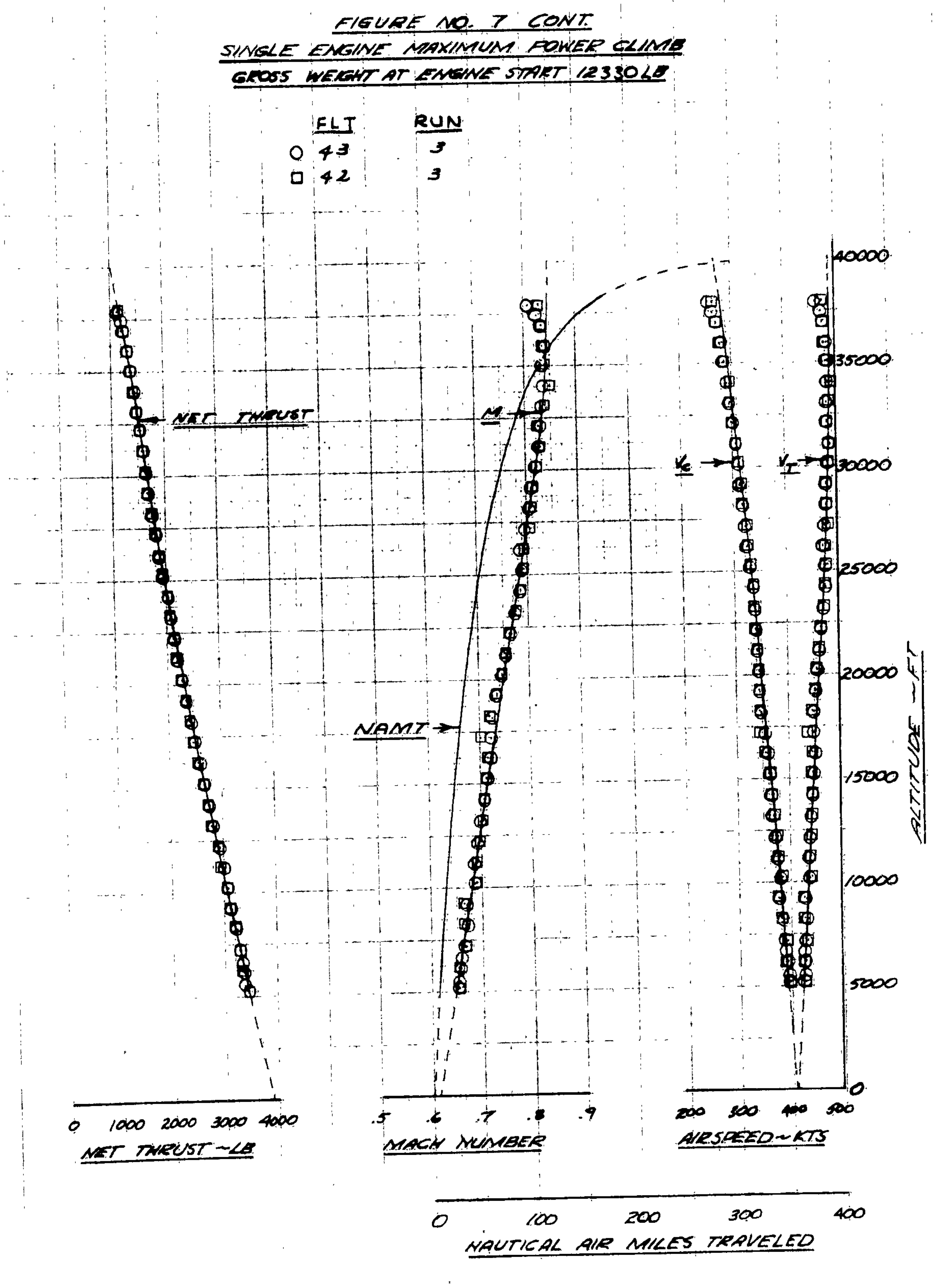


FIGURE NO. 7 CONT.
SINGLE ENGINE MAXIMUM POWER CLIMB
GROSS WEIGHT AT ENGINE START 12330 LB
FLT RUN
○ 43 3
□ 42 3
NET THRUST
M
NAMT
$V_c$
$V_I$
40000
35000
30000
25000
20000
15000
10000
5000
0
ALTITUDE ~ FT
0 1000 2000 3000 4000
NET THRUST ~ LB
.5 .6 .7 .8 .9
MACH NUMBER
200 300 400 500
AIRSPEED ~ KTS
0 100 200 300 400
NAUTICAL AIR MILES TRAVELED

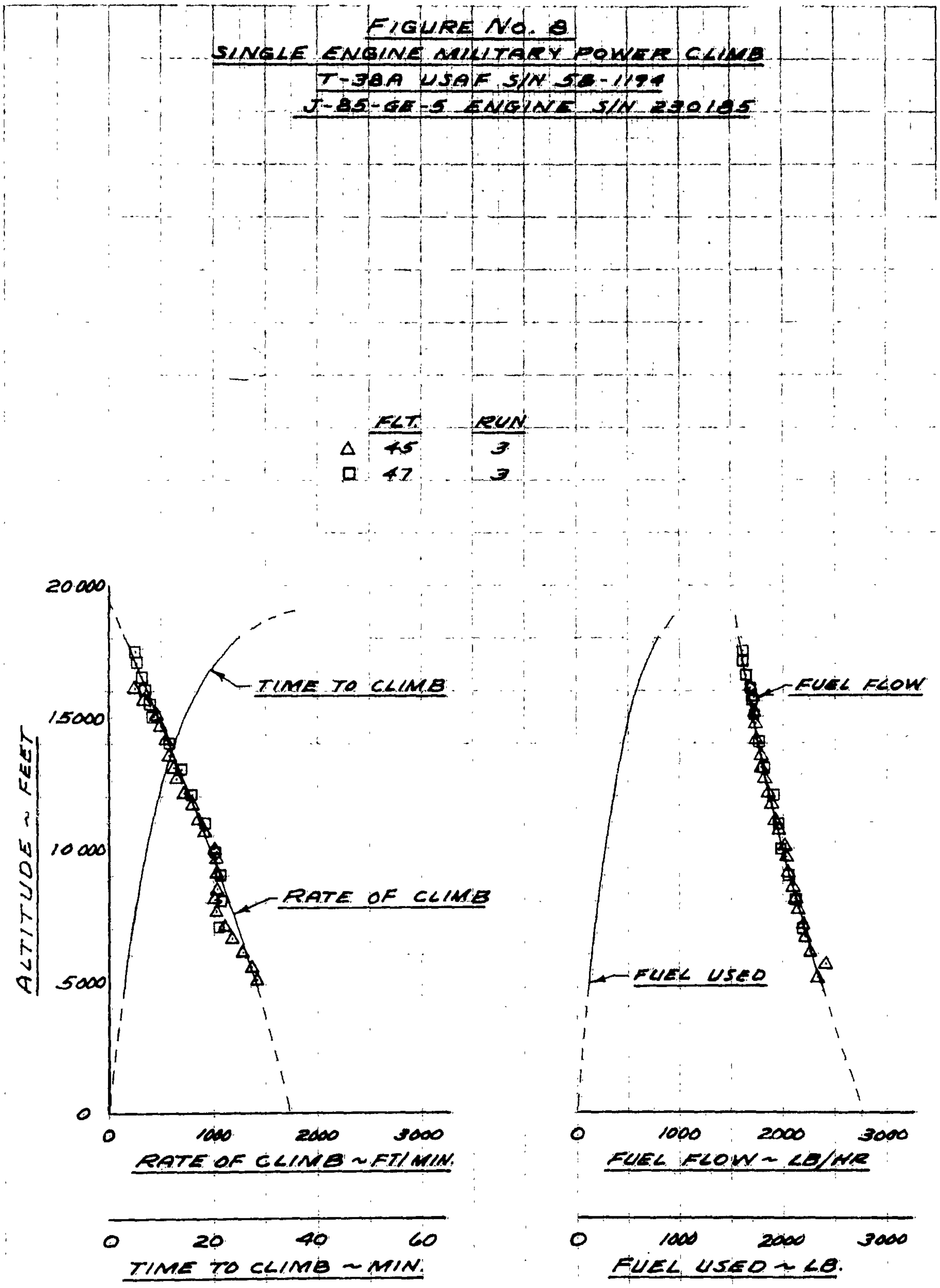


FIGURE No. 8
SINGLE ENGINE MILITARY POWER CLIMB
T-38A USAF S/N 58-1194
J-85-GE-5 ENGINE S/N 230185
FLT. RUN
△ 45 3
□ 47 3
TIME TO CLIMB
RATE OF CLIMB
FUEL FLOW
FUEL USED
ALTITUDE ~ FEET
20 000
15000
10 000
5000
0
0 1000 2000 3000
RATE OF CLIMB ~ FT/MIN.
0 20 40 60
TIME TO CLIMB ~ MIN.
0 1000 2000 3000
FUEL FLOW ~ LB/HR
0 1000 2000 3000
FUEL USED ~ LB.

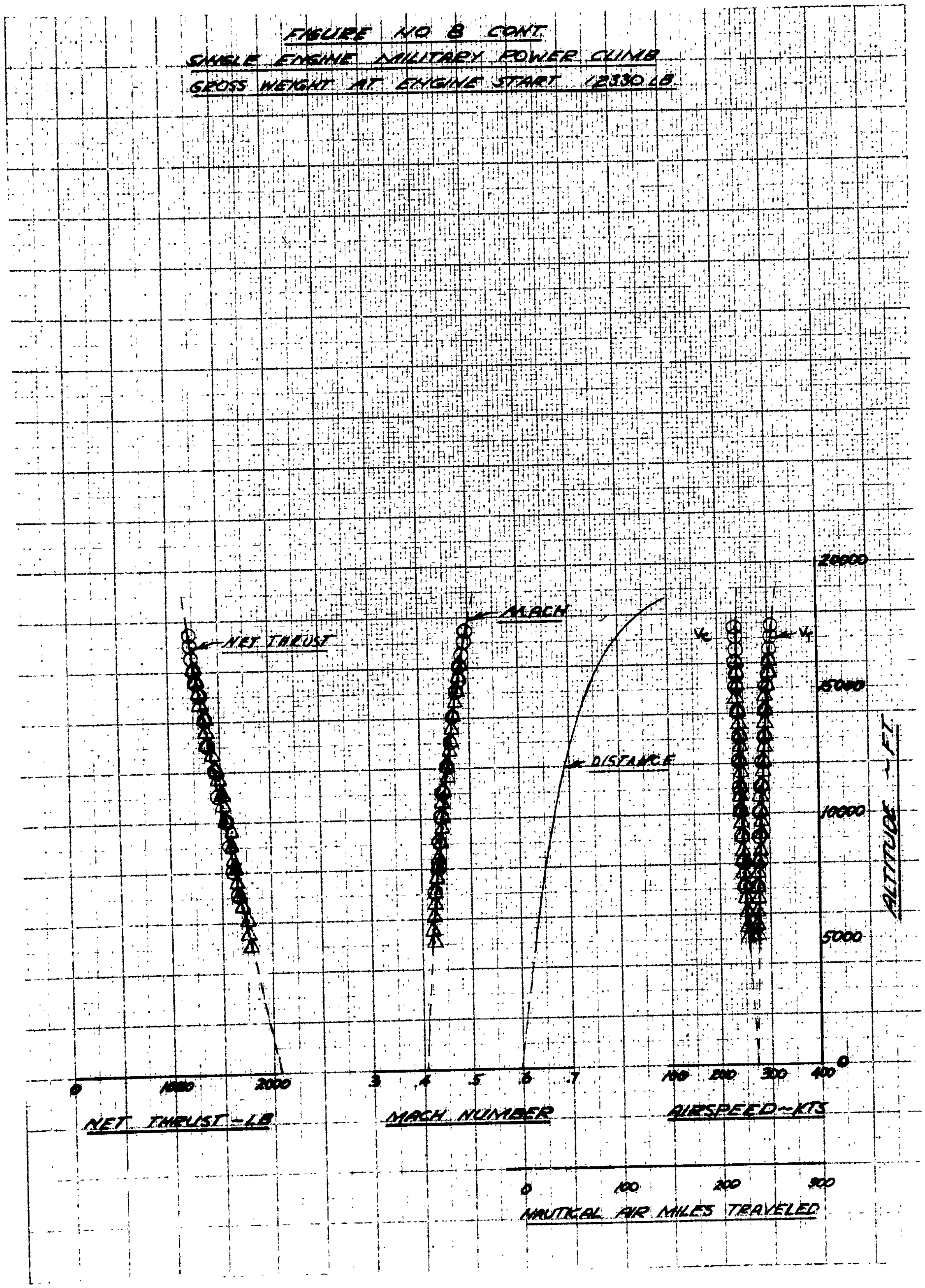


FIGURE NO 8 CONT.
SINGLE ENGINE MILITARY POWER CLIMB
GROSS WEIGHT AT ENGINE START 12330 LB
NET THRUST
MACH
DISTANCE
$V_c$
$V_t$
20000
15000
10000
5000
0
ALTITUDE ~ FT.
0
1000
2000
NET THRUST ~ LB
.3
.4
.5
.6
.7
MACH NUMBER
100
200
300
400
AIRSPEED ~ KTS
0
100
200
300
NAUTICAL AIR MILES TRAVELED

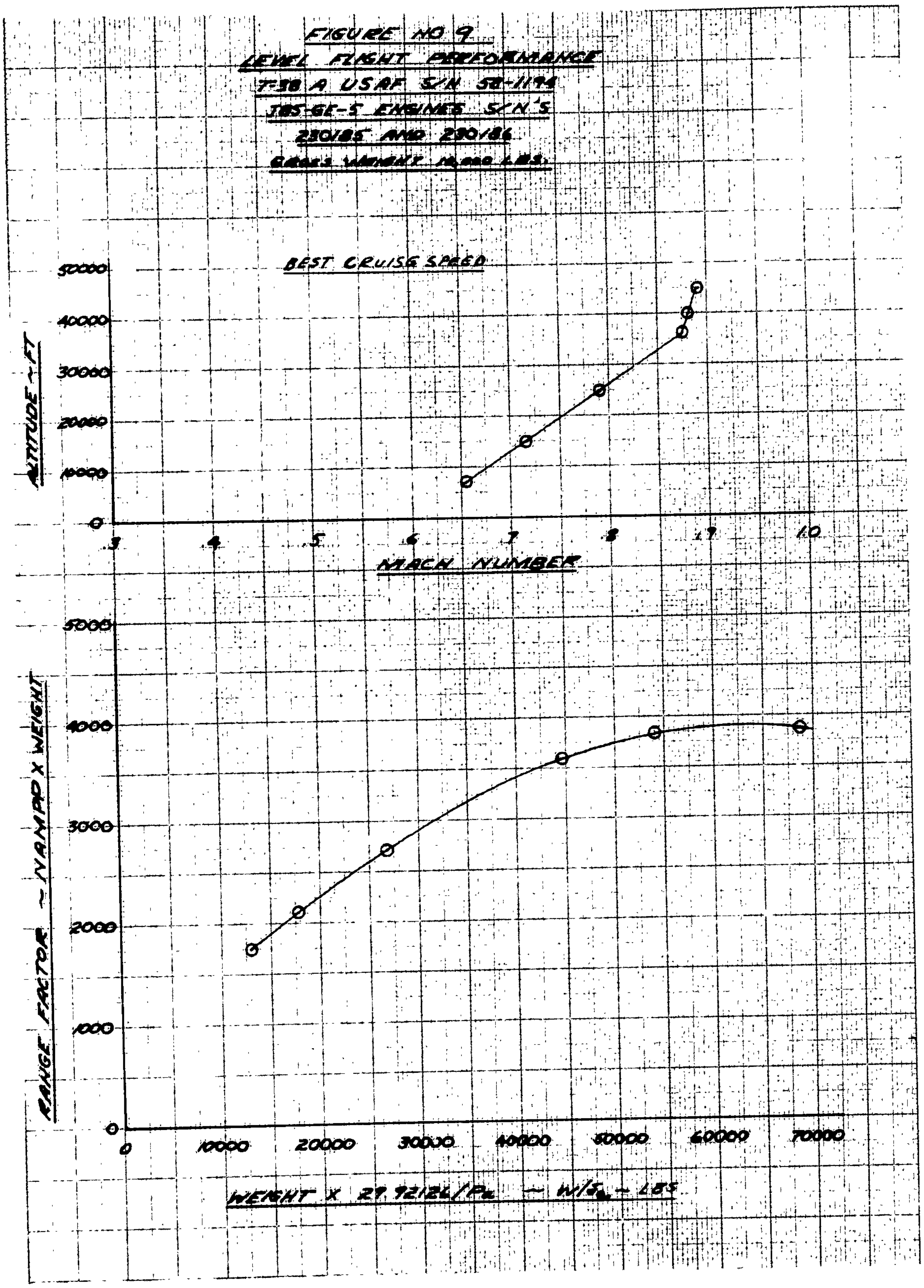


FIGURE NO 9
LEVEL FLIGHT PERFORMANCE
T-38 A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
BEST CRUISE SPEED
ALTITUDE ~ FT
50000
40000
30000
20000
10000
0
.3
.4
.5
.6
.7
.8
.9
1.0
MACH NUMBER
RANGE FACTOR ~ NAMPD X WEIGHT
5000
4000
3000
2000
1000
0
0
10000
20000
30000
40000
50000
60000
70000

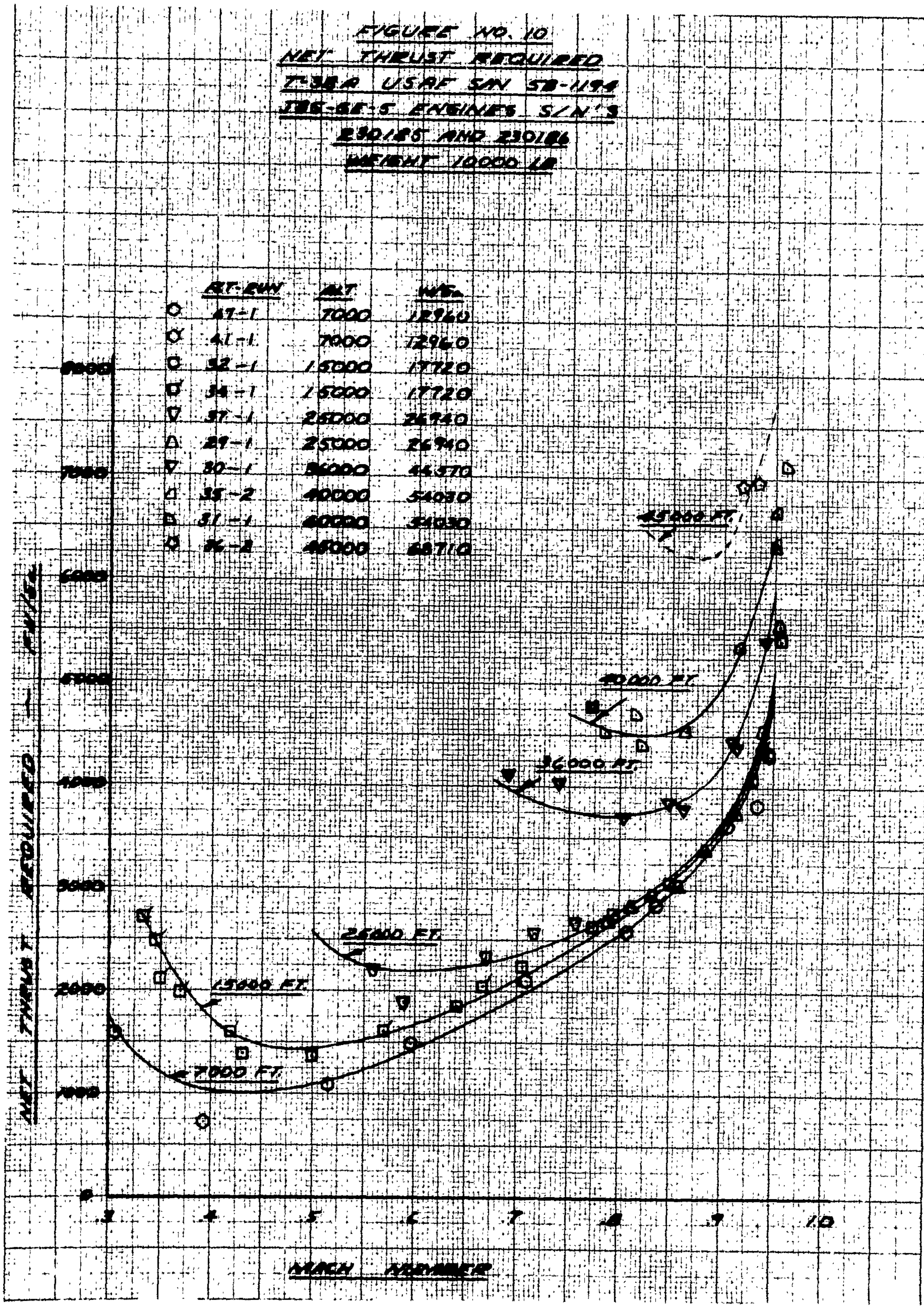


FIGURE NO. 10
NET THRUST REQUIRED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
WEIGHT 10000 LB
FLT-RUN ALT W/δ
41-1 7000 12960
25000 26940
29-1 25000 26940
30-1 36000 44570
35-2 40000 54030
31-1 40000 54030
45000 FT.
40000 FT.
36000 FT.
25000 FT.
15000 FT.
7000 FT.
NET THRUST REQUIRED — FN/δ
MACH NUMBER

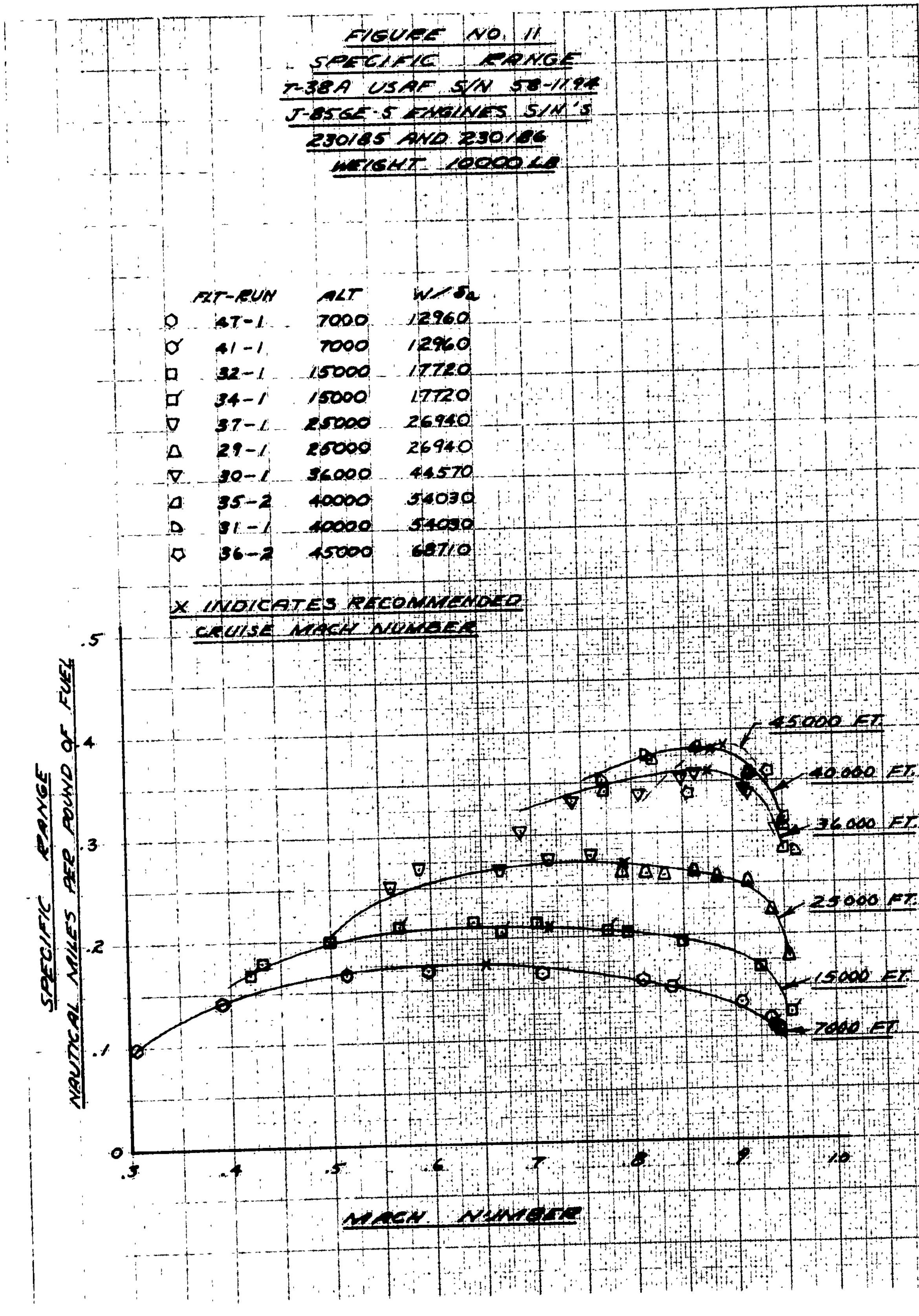


FIGURE NO. 11
SPECIFIC RANGE
T-38A USAF S/N 58-1194
J-85GE-5 ENGINES S/N'S
230185 AND 230186
WEIGHT 10000 LB
FLT-RUN ALT W/δa
47-1 7000 12960
41-1 7000 12960
32-1 15000 17720
34-1 15000 17720
37-1 25000 26940
29-1 25000 26940
30-1 36000 44570
35-2 40000 54030
31-1 40000 54030
36-2 45000 68710
X INDICATES RECOMMENDED CRUISE MACH NUMBER
45000 FT.
40000 FT.
36000 FT.
25000 FT.
15000 FT.
7000 FT.
SPECIFIC RANGE
NAUTICAL MILES PER POUND OF FUEL
MACH NUMBER
0
.1
.2
.3
.4
.5
.3
.4
.5
.6
.7
.8
.9
1.0

FIGURE NO 12

LEVEL FLIGHT PERFORMANCE

SPEED BRAKE EXTENDED

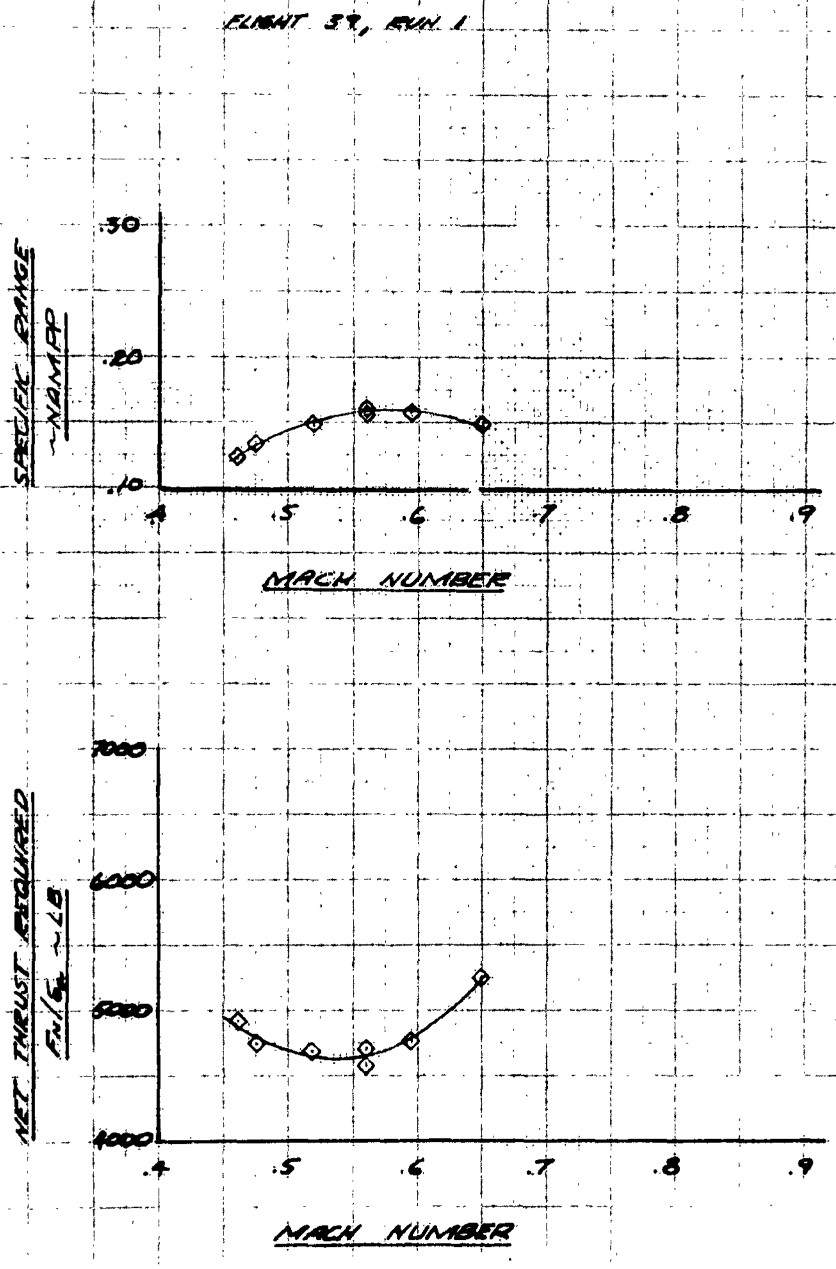
T-38A USAF S/N 58-1194

J85-GE-5 ENGINES S/N'S

230185 AND 230186

WEIGHT 10000 LB

ALTITUDE 25000 FT



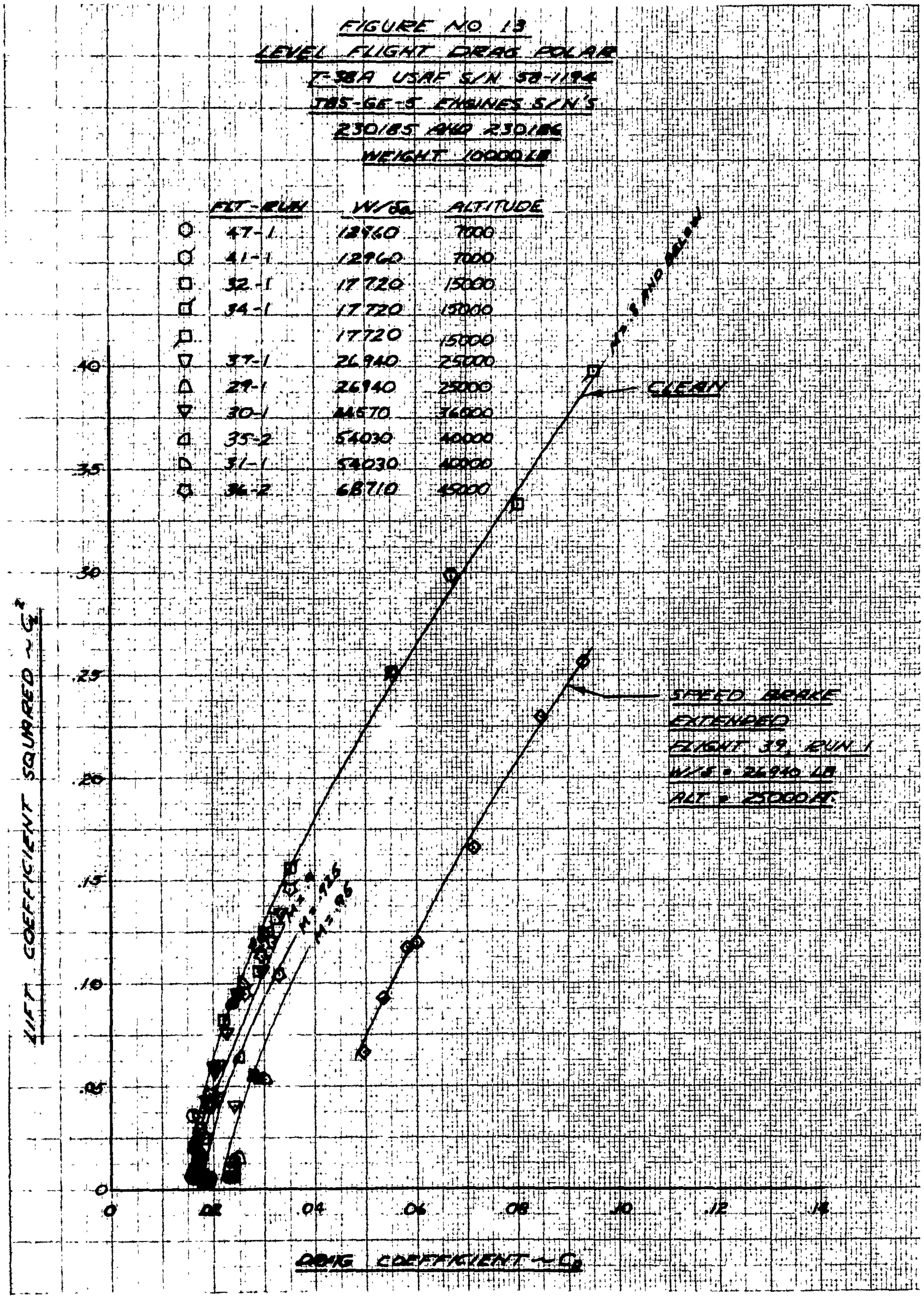
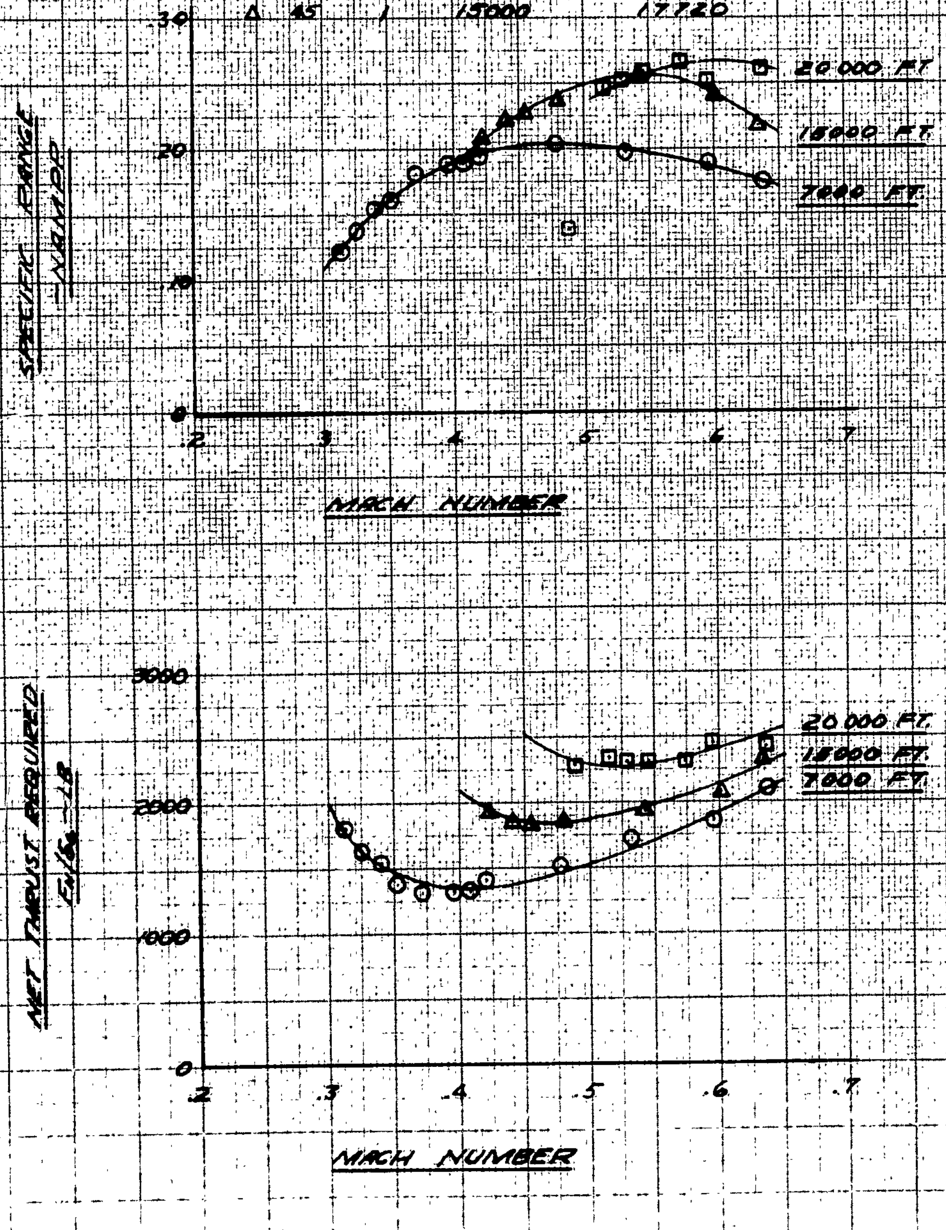


FIGURE NO 13
LEVEL FLIGHT DRAG POLAR
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
WEIGHT 10000 LB
FLT-RUN W/δa ALTITUDE
47-1 12960 7000
41-1 12960 7000
32-1 17720 15000
34-1 17720 15000
17720 15000
37-1 26940 25000
29-1 26940 25000
30-1 36000
35-2 54030 40000
31-1 54030 40000
36-2 68710 45000
CLEAN
SPEED BRAKE EXTENDED
LIFT COEFFICIENT SQUARED ~ $C_L^2$
DRAG COEFFICIENT ~ $C_D$
0 .02 .04 .06 .08 .10 .12 .14
0 .05 .10 .15 .20 .25 .30 .35 .40

FIGURE NO 14
LEVEL FLIGHT PERFORMANCE
SINGLE ENGINE
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
WEIGHT 10000 LB

| | FLT | RUN | ALTITUDE | W/δa |
|---|---|---|---|---|
| □ | 43 | 1 | 20000 | 21760 |
| ○ | 46 | 1 | 7000 | 12960 |
| △ | 45 | 1 | 15000 | 17720 |



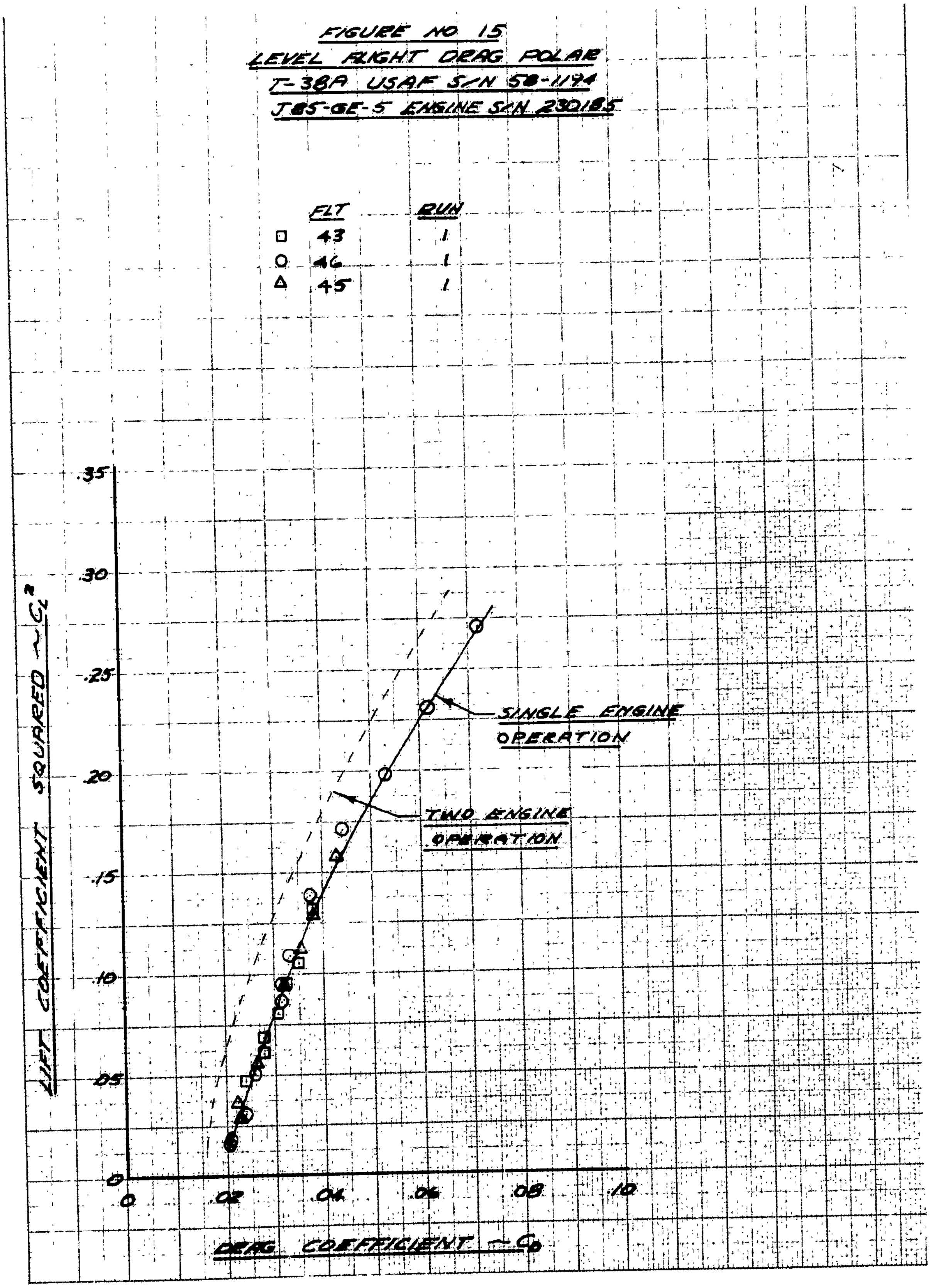


FIGURE NO 15
LEVEL FLIGHT DRAG POLAR
T-38A USAF S/N 58-1194
J85-GE-5 ENGINE S/N 230185
FLT RUN
43 1
46 1
45 1
SINGLE ENGINE OPERATION
TWO ENGINE OPERATION
LIFT COEFFICIENT SQUARED ~ $C_L^2$
DRAG COEFFICIENT ~ $C_D$
.35
.30
.25
.20
.15
.10
.05
0
.02
.04
.06
.08
.10

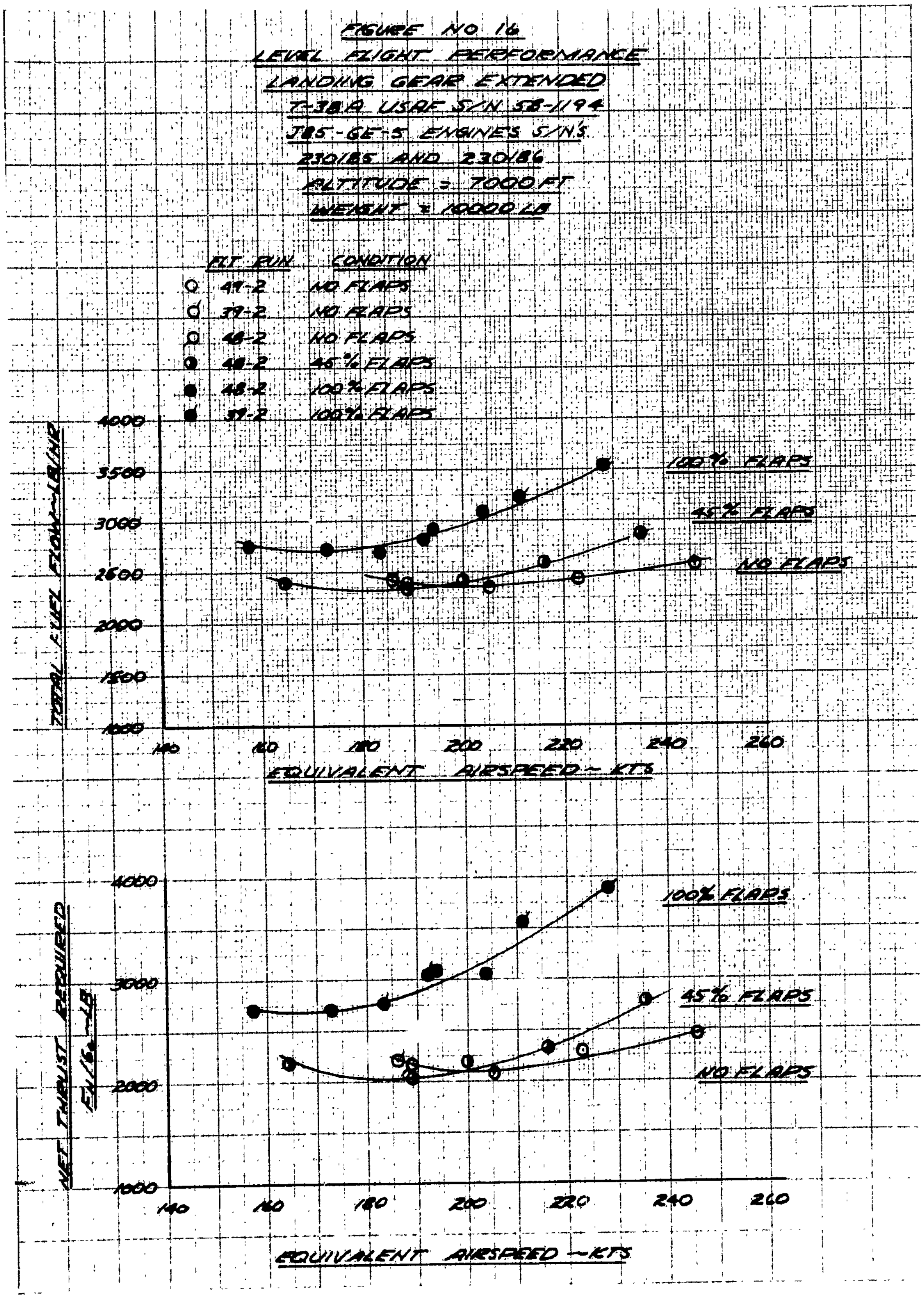


FIGURE NO 16
LEVEL FLIGHT PERFORMANCE
LANDING GEAR EXTENDED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALTITUDE = 7000 FT
WEIGHT = 10000 LB
FLT RUN CONDITION
49-2 NO FLAPS
39-2 NO FLAPS
48-2 NO FLAPS
48-2 45% FLAPS
48-2 100% FLAPS
39-2 100% FLAPS
TOTAL FUEL FLOW ~ LB/HR
100% FLAPS
45% FLAPS
NO FLAPS
EQUIVALENT AIRSPEED ~ KTS
NET THRUST REQUIRED
Fn/δa ~ LB
100% FLAPS
45% FLAPS
NO FLAPS
EQUIVALENT AIRSPEED ~ KTS

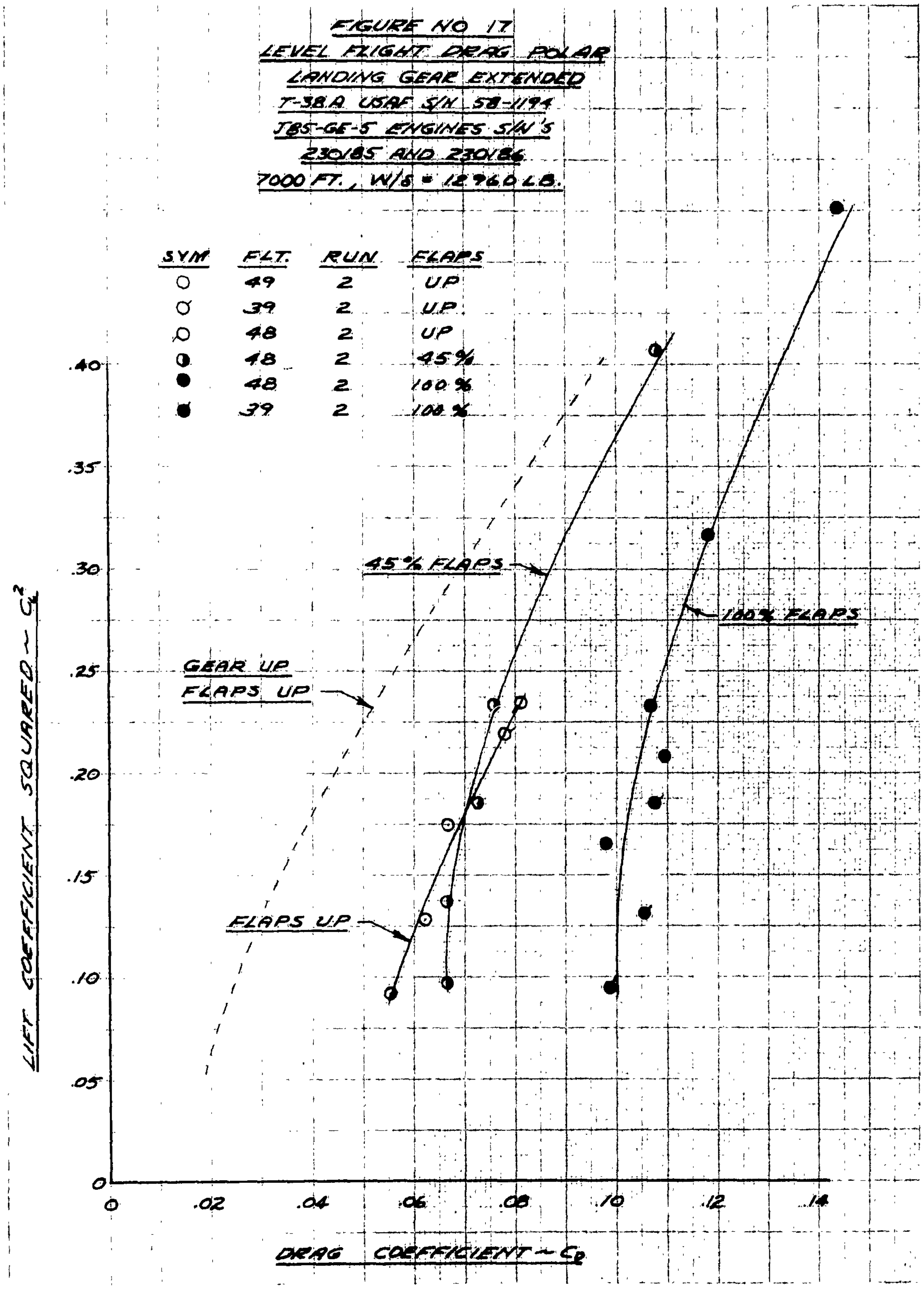


FIGURE NO 17
LEVEL FLIGHT DRAG POLAR
LANDING GEAR EXTENDED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
7000 FT., W/δ = 12,960 LB.
SYM FLT. RUN FLAPS
49 2 UP
39 2 UP
48 2 UP
48 2 45%
48 2 100%
39 2 100%
45% FLAPS
100% FLAPS
GEAR UP
FLAPS UP
FLAPS UP
LIFT COEFFICIENT SQUARED ~ $C_L^2$
DRAG COEFFICIENT ~ $C_D$
0
.05
.10
.15
.20
.25
.30
.35
.40
.02
.04
.06
.08
.10
.12
.14

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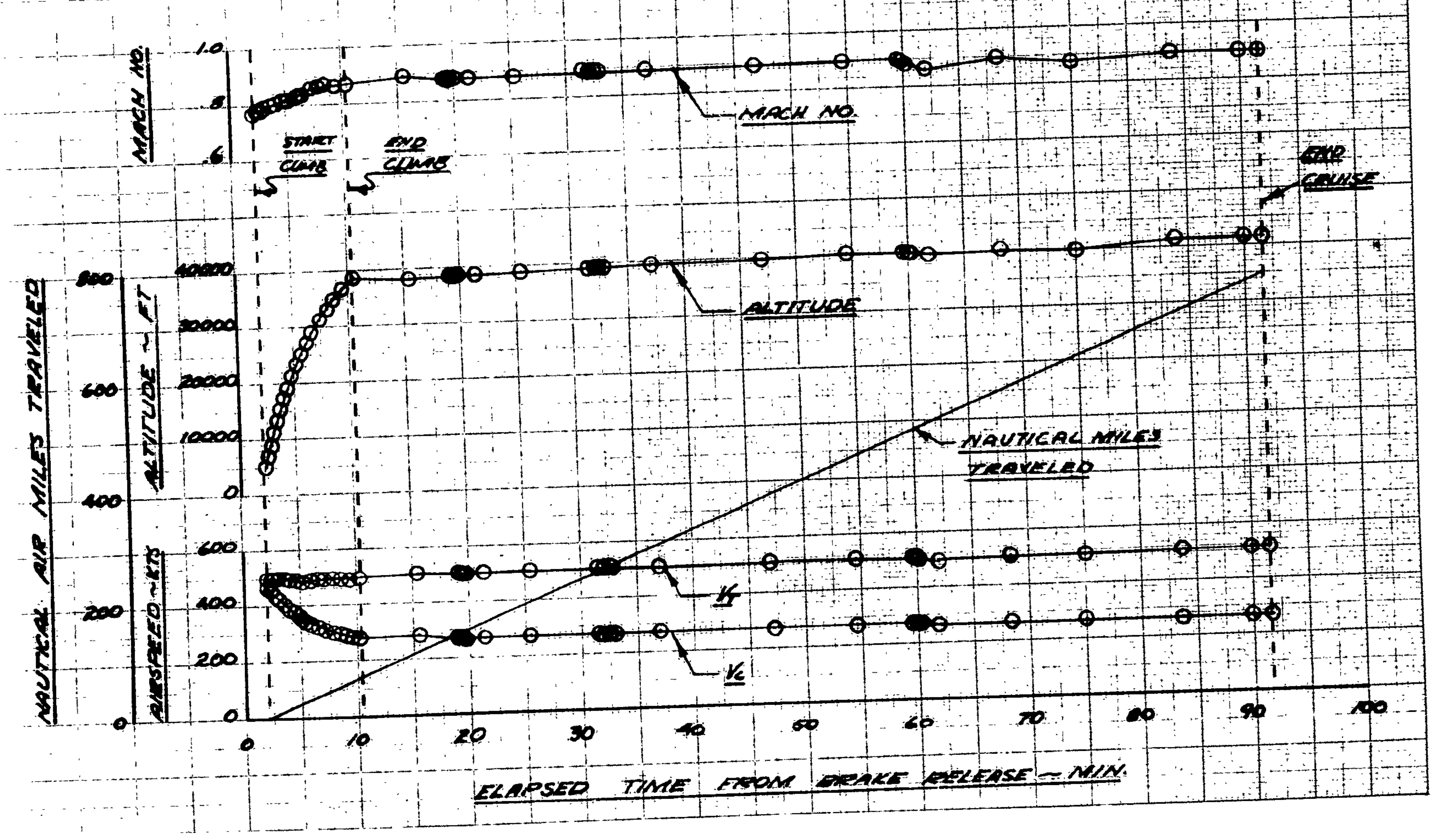
FIGURE NO. 16
FERRY RANGE MISSION
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S 230186 AND 230186
FLIGHT NO. 49



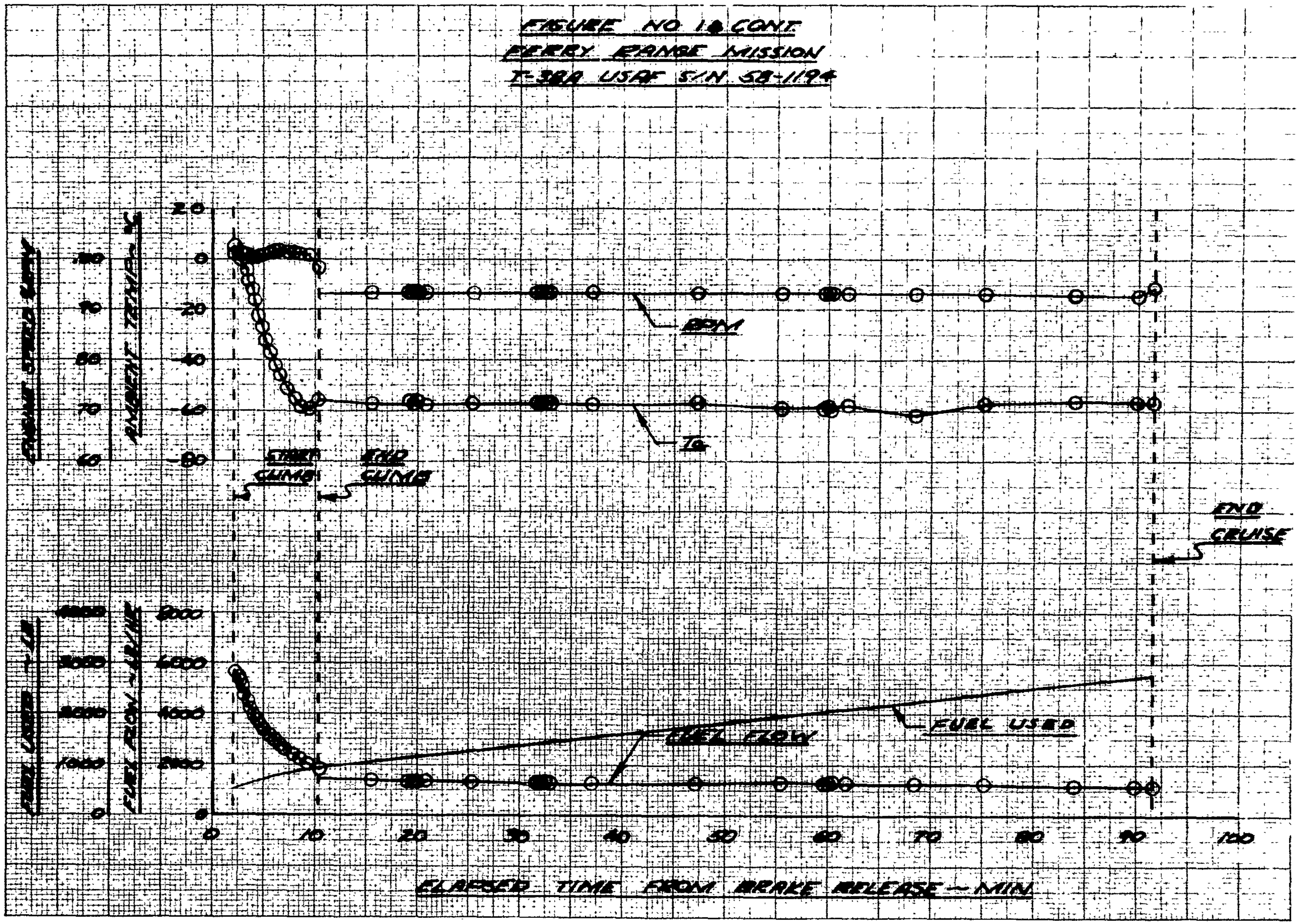


FIGURE NO 18 CONT.
FERRY RANGE MISSION
T-38A USAF S/N 58-1194
ENGINE SPEED %RPM
AMBIENT TEMP ~ °C
FUEL USED ~ LB
FUEL FLOW ~ LB/HR
RPM
$T_a$
START CLIMB
END CLIMB
END CRUISE
FUEL FLOW
FUEL USED
ELAPSED TIME FROM BRAKE RELEASE ~ MIN

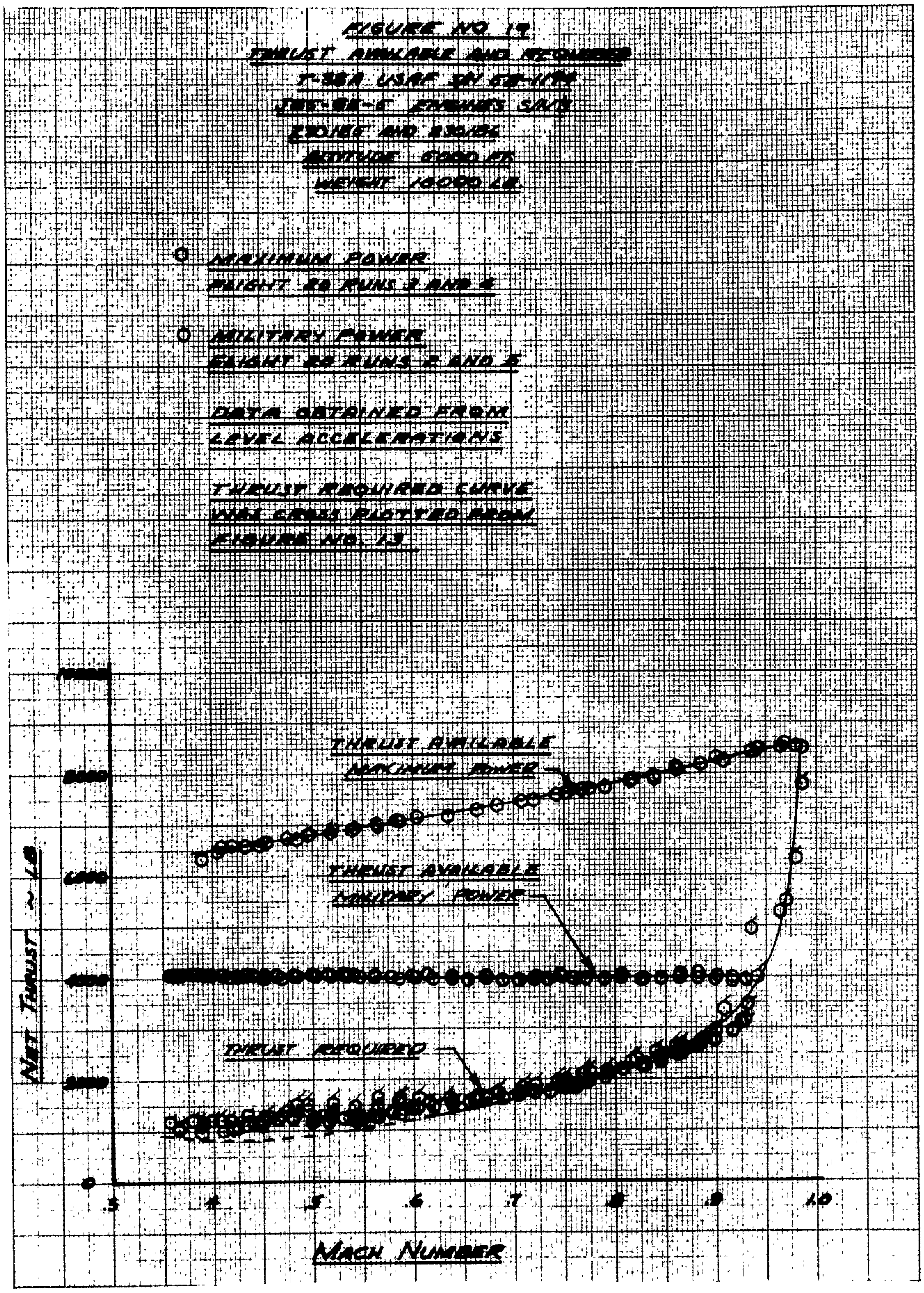


FIGURE NO 19
THRUST AVAILABLE AND REQUIRED
MAXIMUM POWER
MILITARY POWER
DATA OBTAINED FROM
LEVEL ACCELERATIONS
THRUST REQUIRED CURVE
THRUST AVAILABLE
MAXIMUM POWER
THRUST AVAILABLE
MILITARY POWER
THRUST REQUIRED
NET THRUST ~ LB
0
.3
.4
.5
.6
.7
.8
.9
1.0
MACH NUMBER

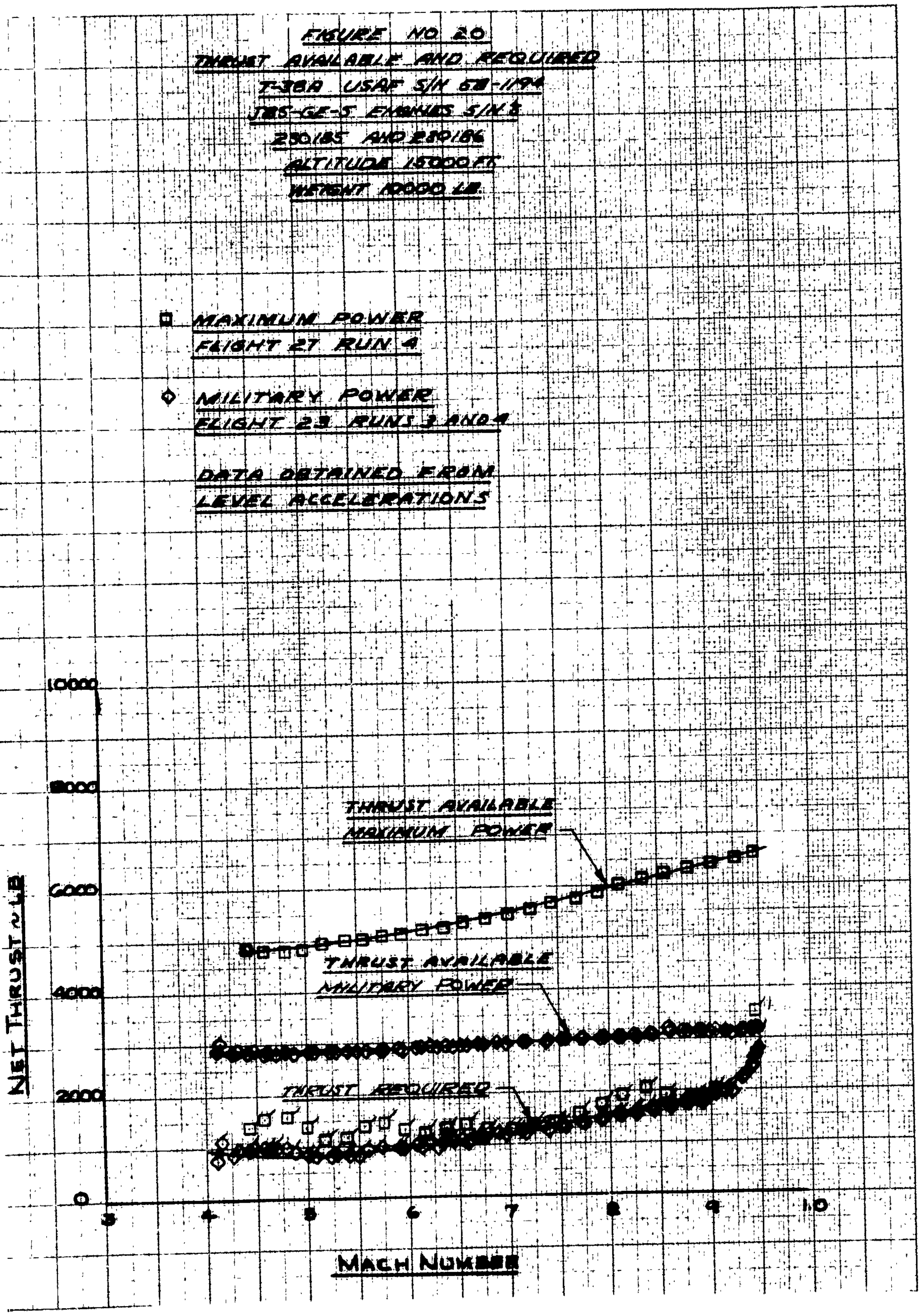


FIGURE NO 20
THRUST AVAILABLE AND REQUIRED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALTITUDE 15000 FT.
WEIGHT 10000 LB.
MAXIMUM POWER
FLIGHT 27 RUN 4
MILITARY POWER
FLIGHT 23 RUNS 3 AND 4
DATA OBTAINED FROM
LEVEL ACCELERATIONS
THRUST AVAILABLE
MAXIMUM POWER
THRUST AVAILABLE
MILITARY POWER
THRUST REQUIRED
NET THRUST ~ LB
MACH NUMBER
0
2000
4000
6000
8000
10000
.3
.4
.5
.6
.7
.8
.9
1.0

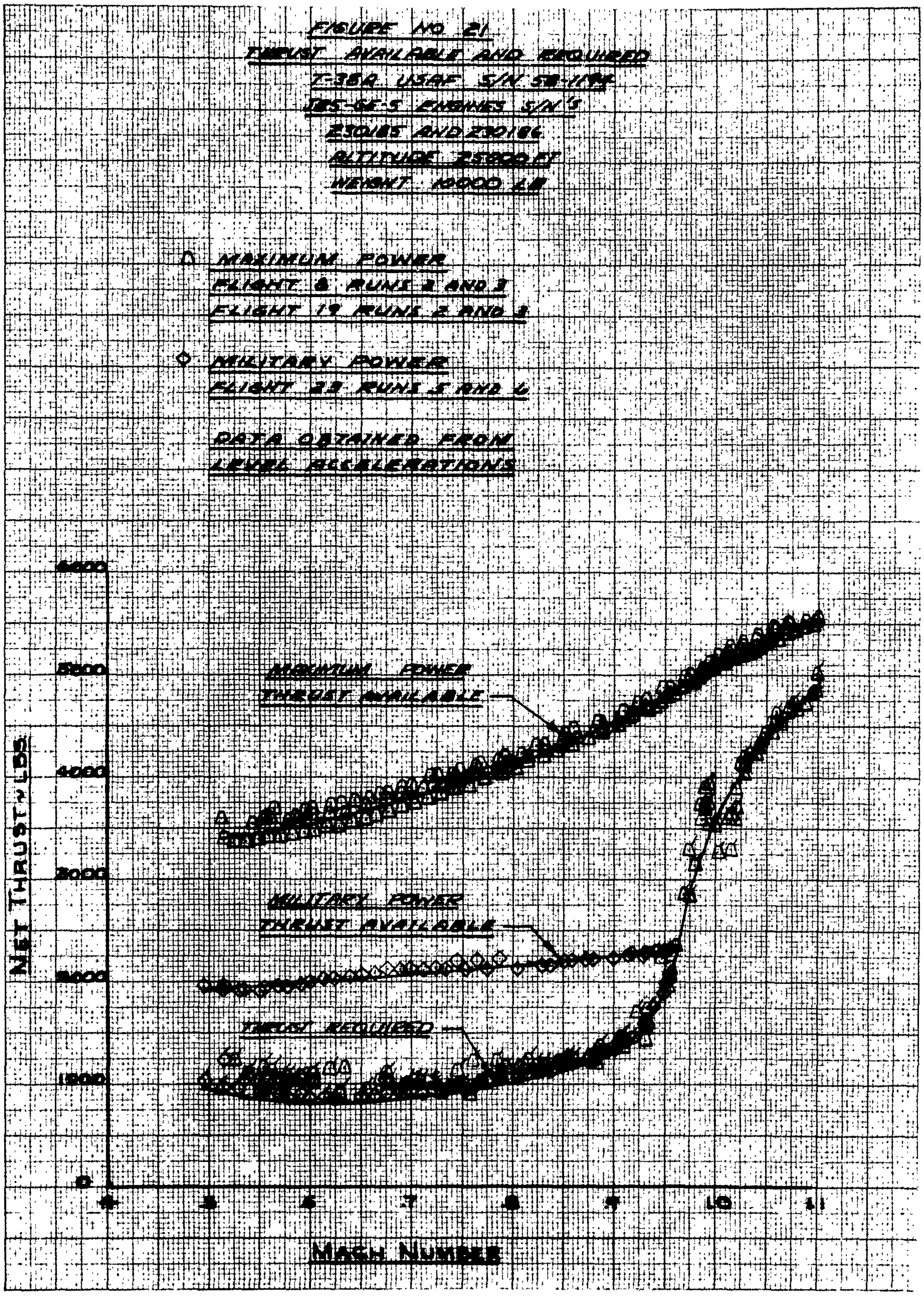


FIGURE NO. 21
THRUST AVAILABLE AND REQUIRED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALTITUDE 25000 FT
WEIGHT 10000 LB
MAXIMUM POWER
FLIGHT 8 RUNS 2 AND 3
FLIGHT 19 RUNS 2 AND 3
MILITARY POWER
FLIGHT 23 RUNS 5 AND 6
DATA OBTAINED FROM
LEVEL ACCELERATIONS
MAXIMUM POWER
THRUST AVAILABLE
MILITARY POWER
THRUST AVAILABLE
THRUST REQUIRED
NET THRUST ~ LBS
0
1000
2000
3000
4000
5000
6000
.4
.5
.6
.7
.8
.9
1.0
1.1
MACH NUMBER

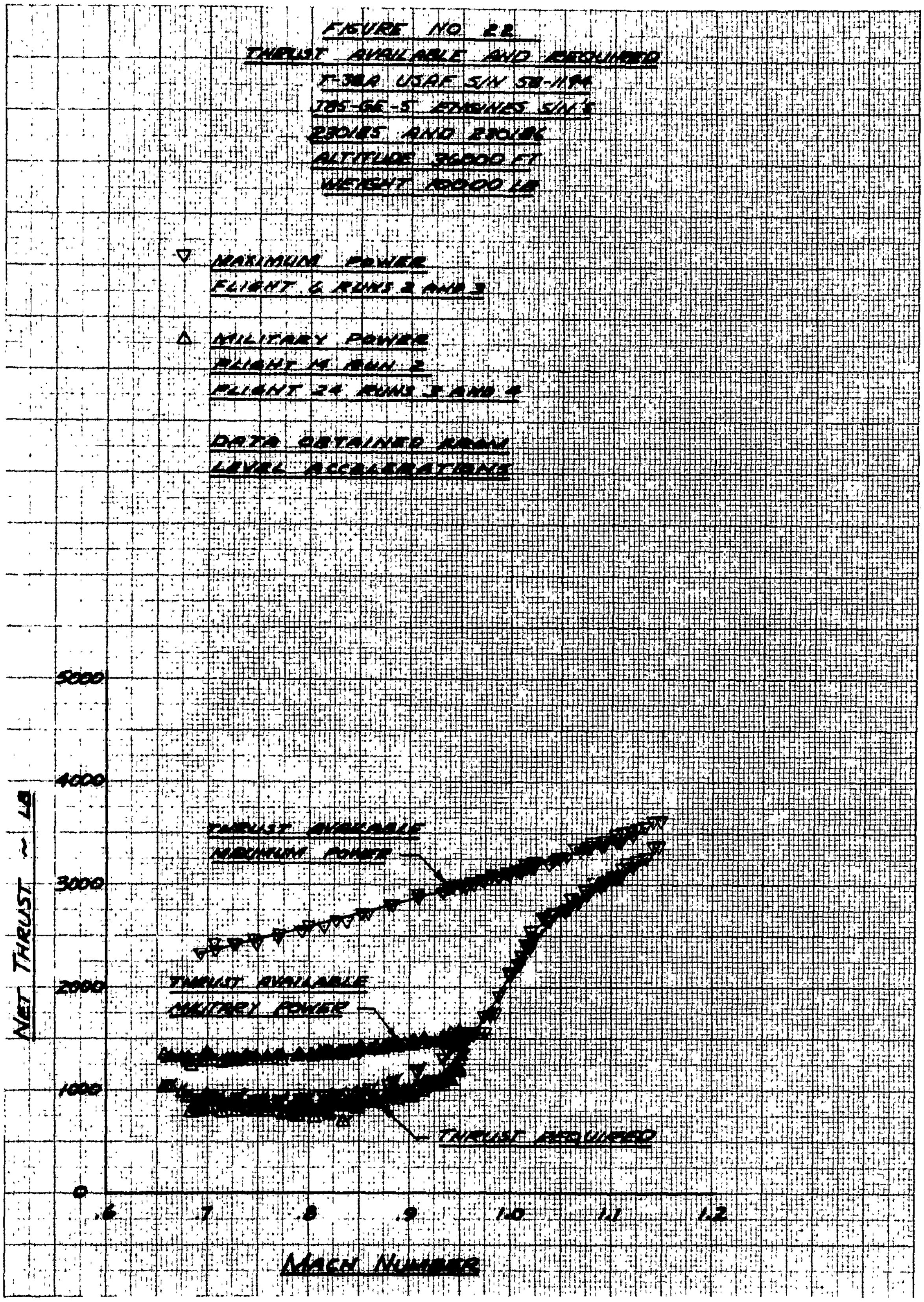
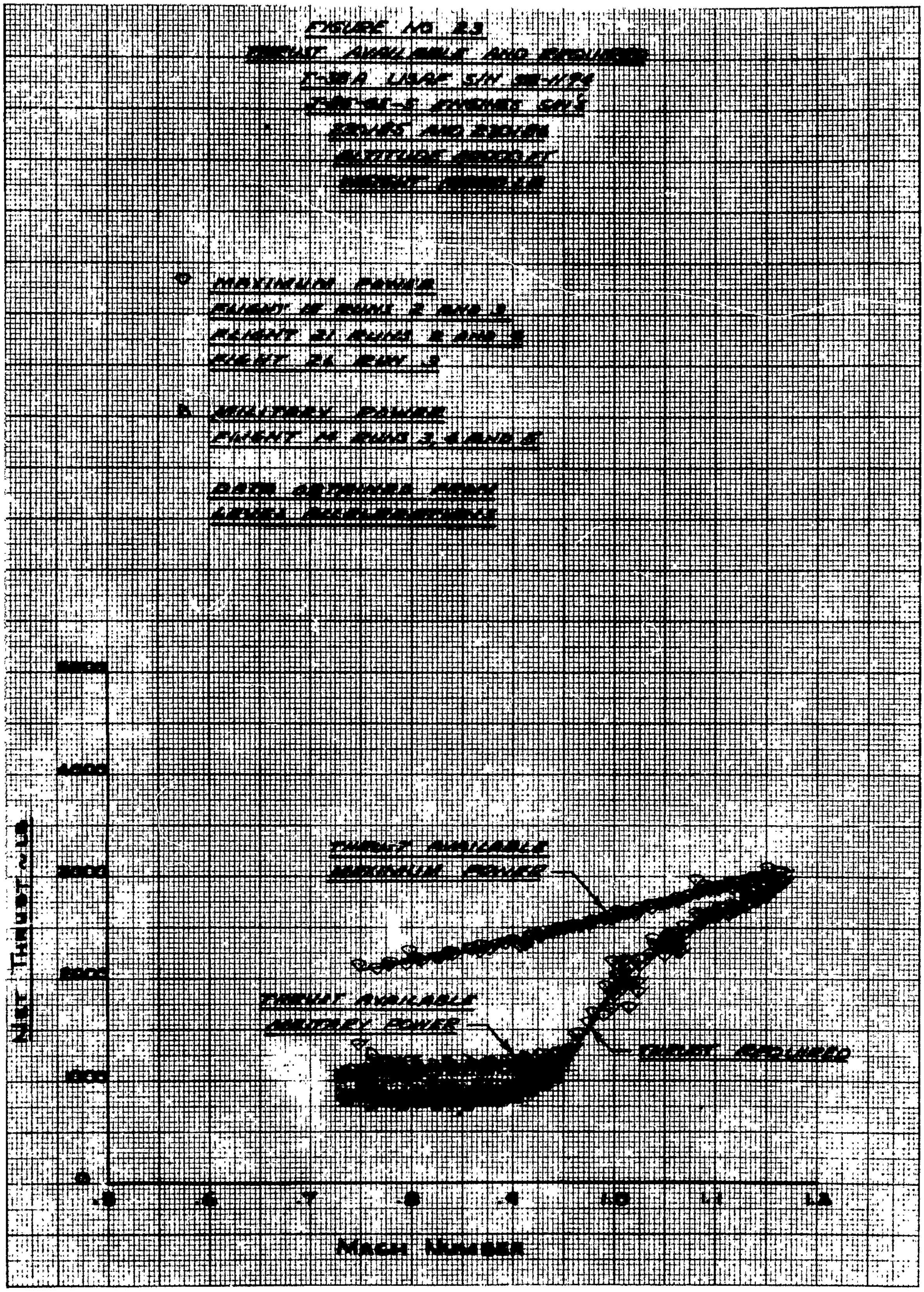
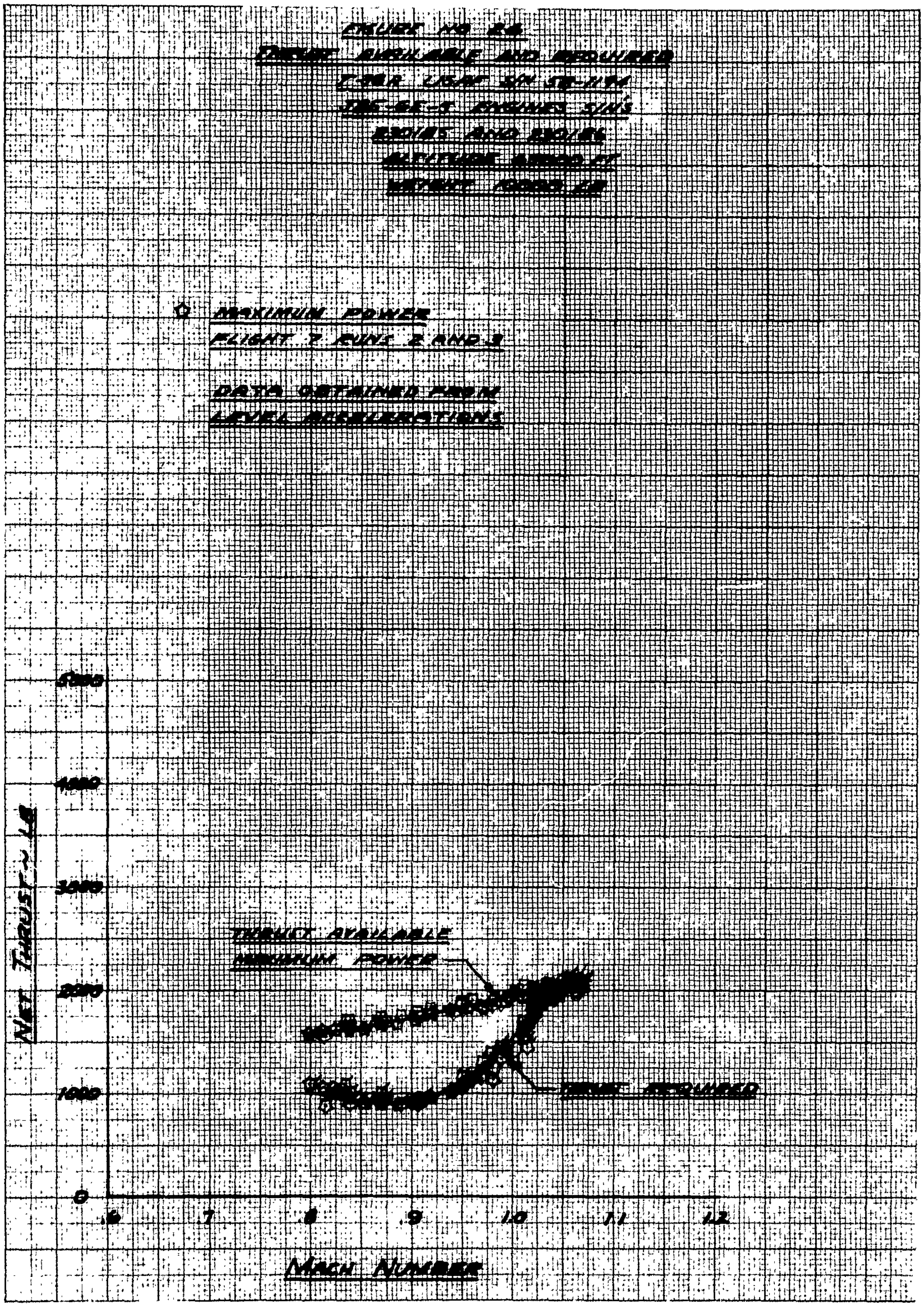


FIGURE NO 22
THRUST AVAILABLE AND REQUIRED
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALTITUDE 36000 FT
WEIGHT 10000 LB
MAXIMUM POWER
FLIGHT 6 RUNS 2 AND 3
MILITARY POWER
FLIGHT 14 RUN 2
FLIGHT 24 RUNS 3 AND 4
DATA OBTAINED FROM
LEVEL ACCELERATIONS
THRUST AVAILABLE
MAXIMUM POWER
THRUST AVAILABLE
MILITARY POWER
THRUST REQUIRED
NET THRUST ~ LB
5000
4000
3000
2000
1000
0
.6
.7
.8
.9
1.0
1.1
1.2
MACH NUMBER



MAXIMUM POWER
MILITARY POWER
THRUST AVAILABLE MAXIMUM POWER
THRUST AVAILABLE MILITARY POWER
THRUST REQUIRED
NET THRUST ~ LB
0
.5
.6
.7
.8
.9
1.0
1.1
1.2
MACH NUMBER


THRUST AVAILABLE AND REQUIRED
MAXIMUM POWER
FLIGHT 7 RUNS 2 AND 3
DATA OBTAINED FROM
LEVEL ACCELERATIONS
THRUST AVAILABLE
MAXIMUM POWER
THRUST REQUIRED
NET THRUST ~ LB
5000
4000
3000
2000
1000
0
.6
.7
.8
.9
1.0
1.1
1.2
MACH NUMBER

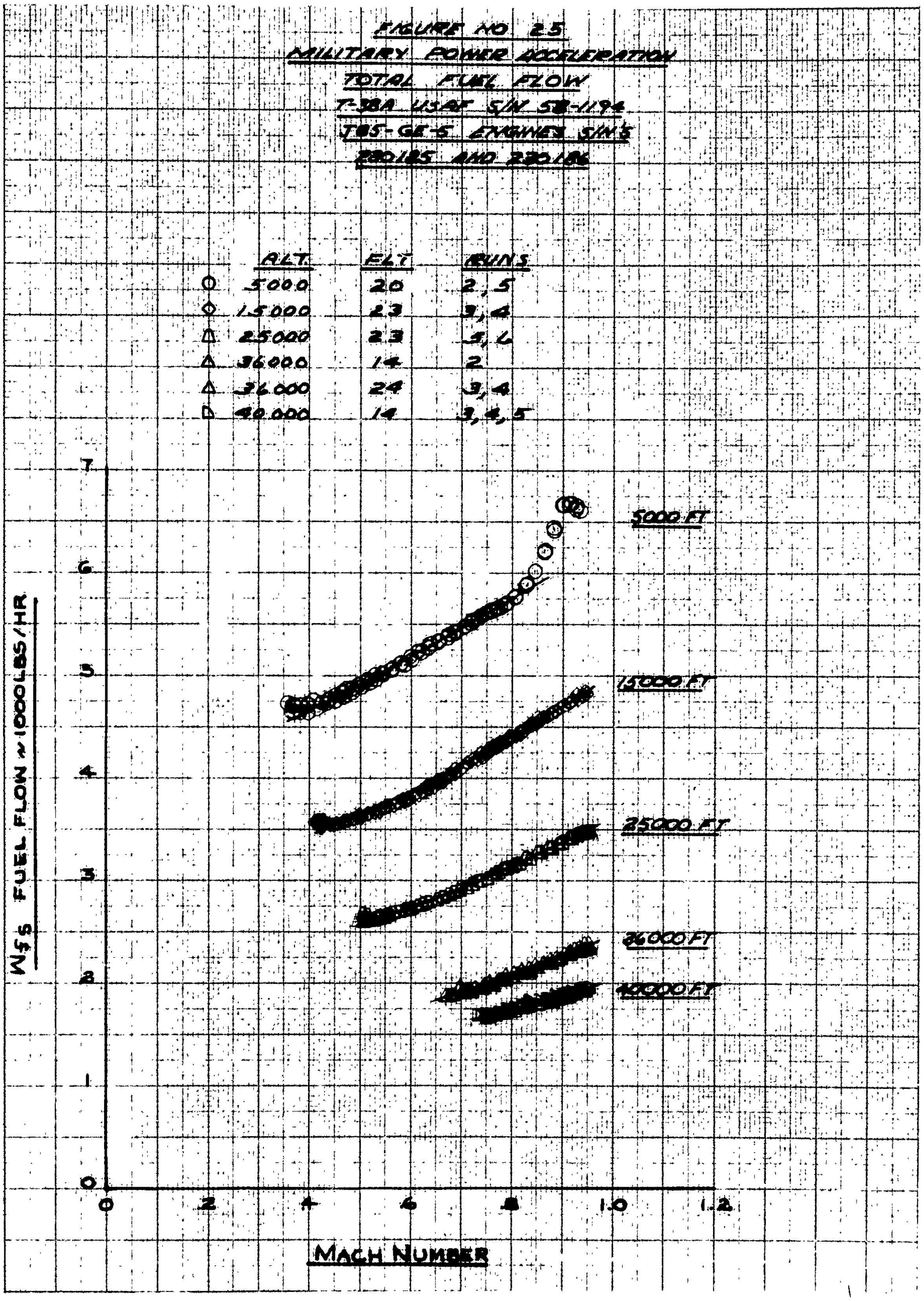


FIGURE NO 25
MILITARY POWER ACCELERATION
TOTAL FUEL FLOW
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALT. FLT RUNS
5000 20 2,5
15000 23 3,4
25000 23 5,6
36000 14 2
36000 24 3,4
40000 14 3,4,5
5000 FT
15000 FT
25000 FT
36000 FT
40000 FT
$W_{fS}$ FUEL FLOW ~ 1000 LBS/HR
MACH NUMBER
0
.2
.4
.6
.8
1.0
1.2
1
2
3
4
5
6
7

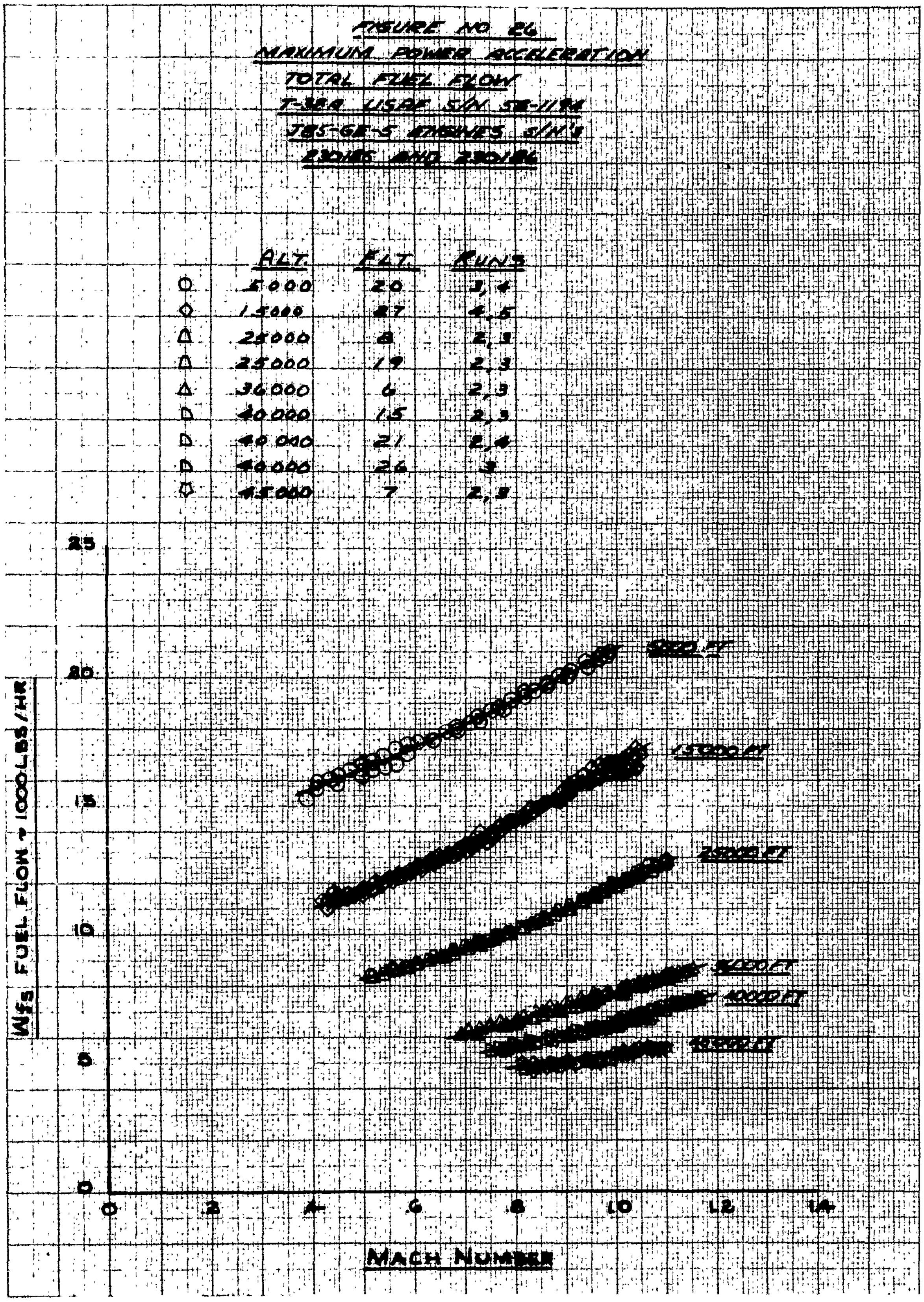


FIGURE NO 26
MAXIMUM POWER ACCELERATION
TOTAL FUEL FLOW
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
ALT. FLT. RUNS
5000 20 3,4
15000 27 4,5
25000 8 2,3
25000 19 2,3
36000 6 2,3
40000 15 2,3
40000 21 2,4
40000 26 3
45000 7 2,3
15000 FT
25000 FT
40000 FT
Wfs FUEL FLOW ÷ 1000 LBS/HR
25
20
15
10
5
0
0
.2
.4
.6
.8
1.0
1.2
1.4
MACH NUMBER

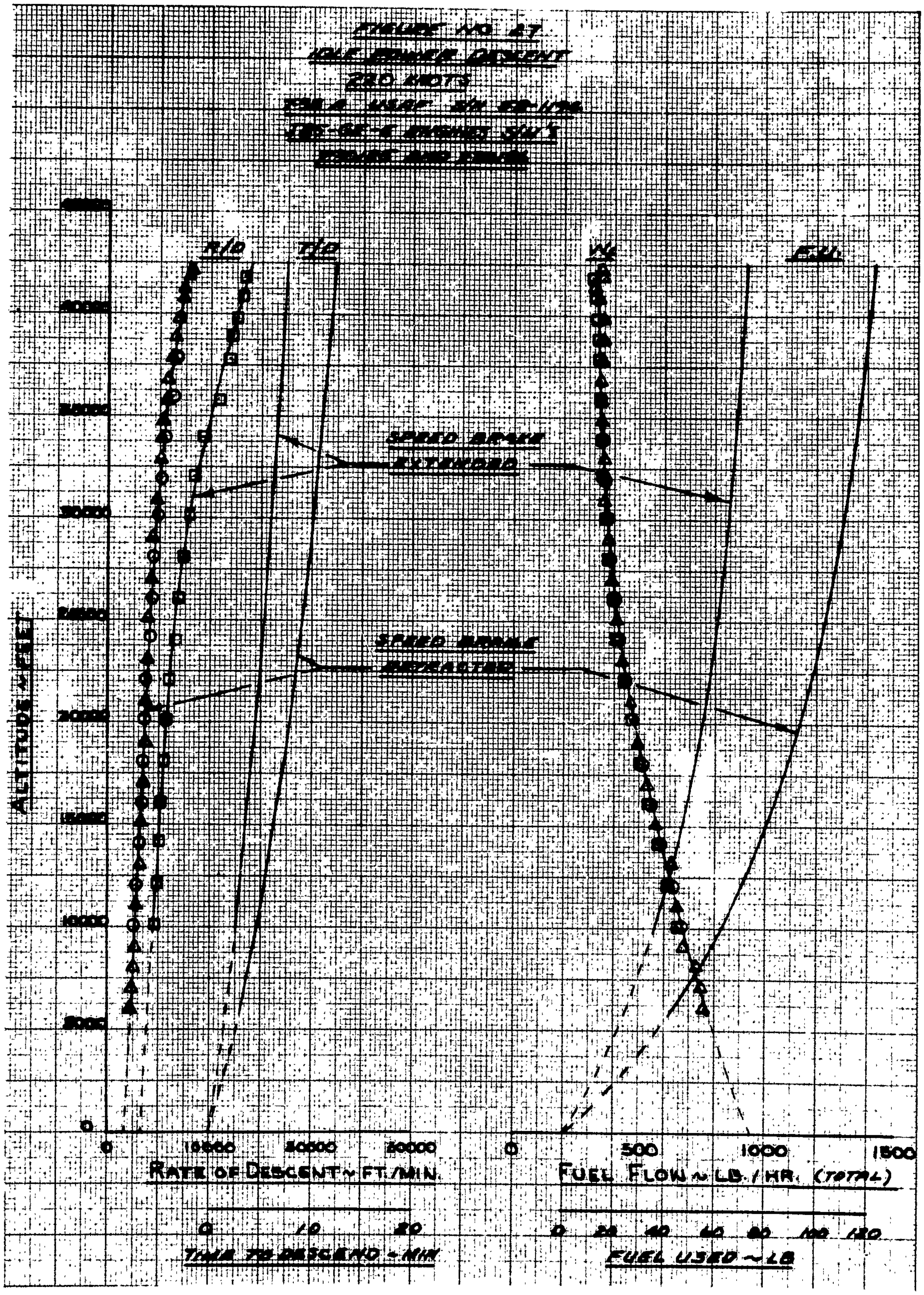


FIGURE NO. 27
IDLE POWER DESCENT
220 KNOTS
R/D
T/D
Wf
F.U.
SPEED BRAKE EXTENDED
SPEED BRAKE RETRACTED
ALTITUDE ~ FEET
0
10000
0
10000
20000
30000
RATE OF DESCENT ~ FT./MIN.
0
500
1000
1500
FUEL FLOW ~ LB./HR. (TOTAL)
0
10
20
TIME TO DESCEND ~ MIN
0
20
40
60
80
100
120
FUEL USED ~ LB

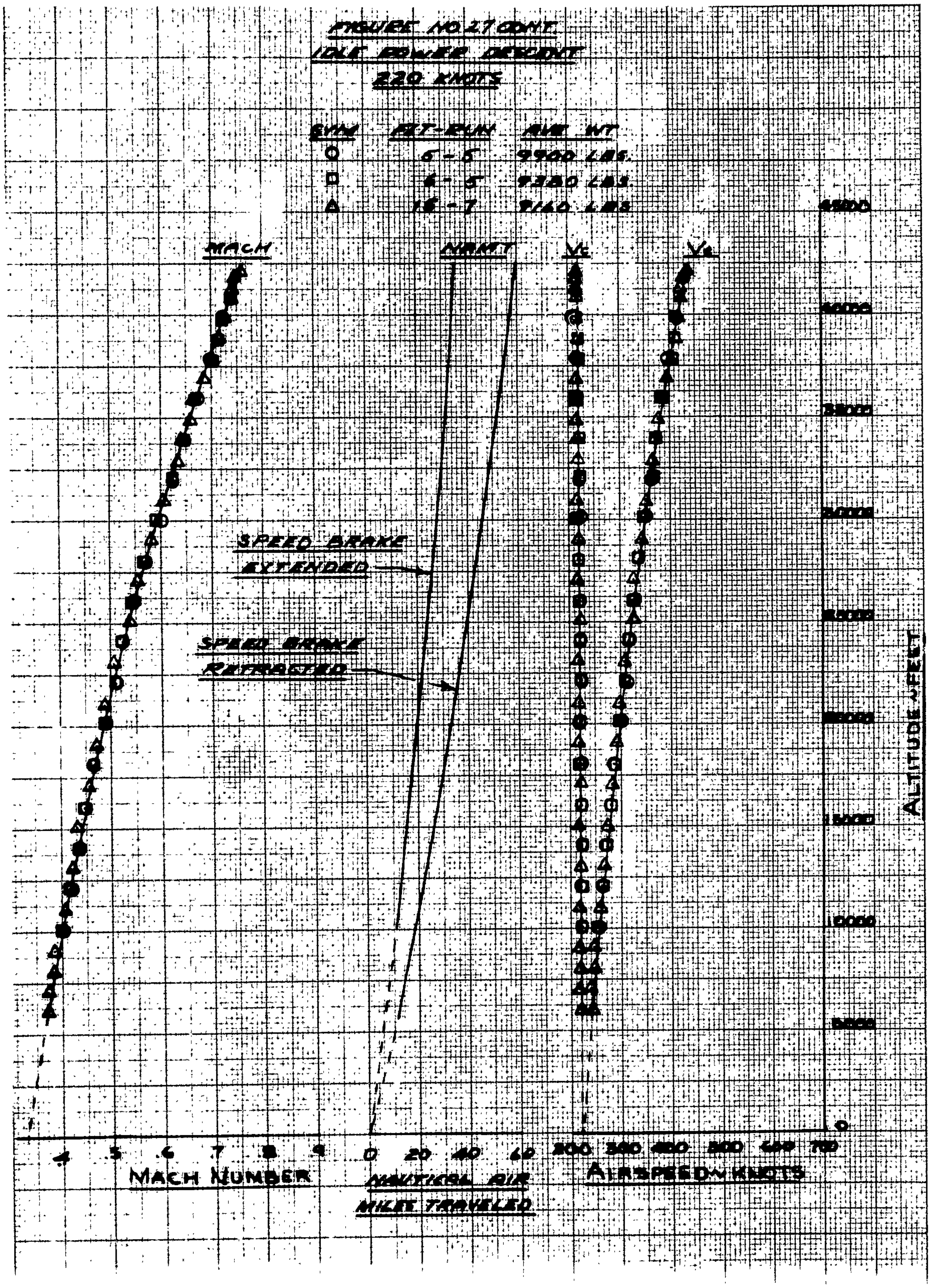
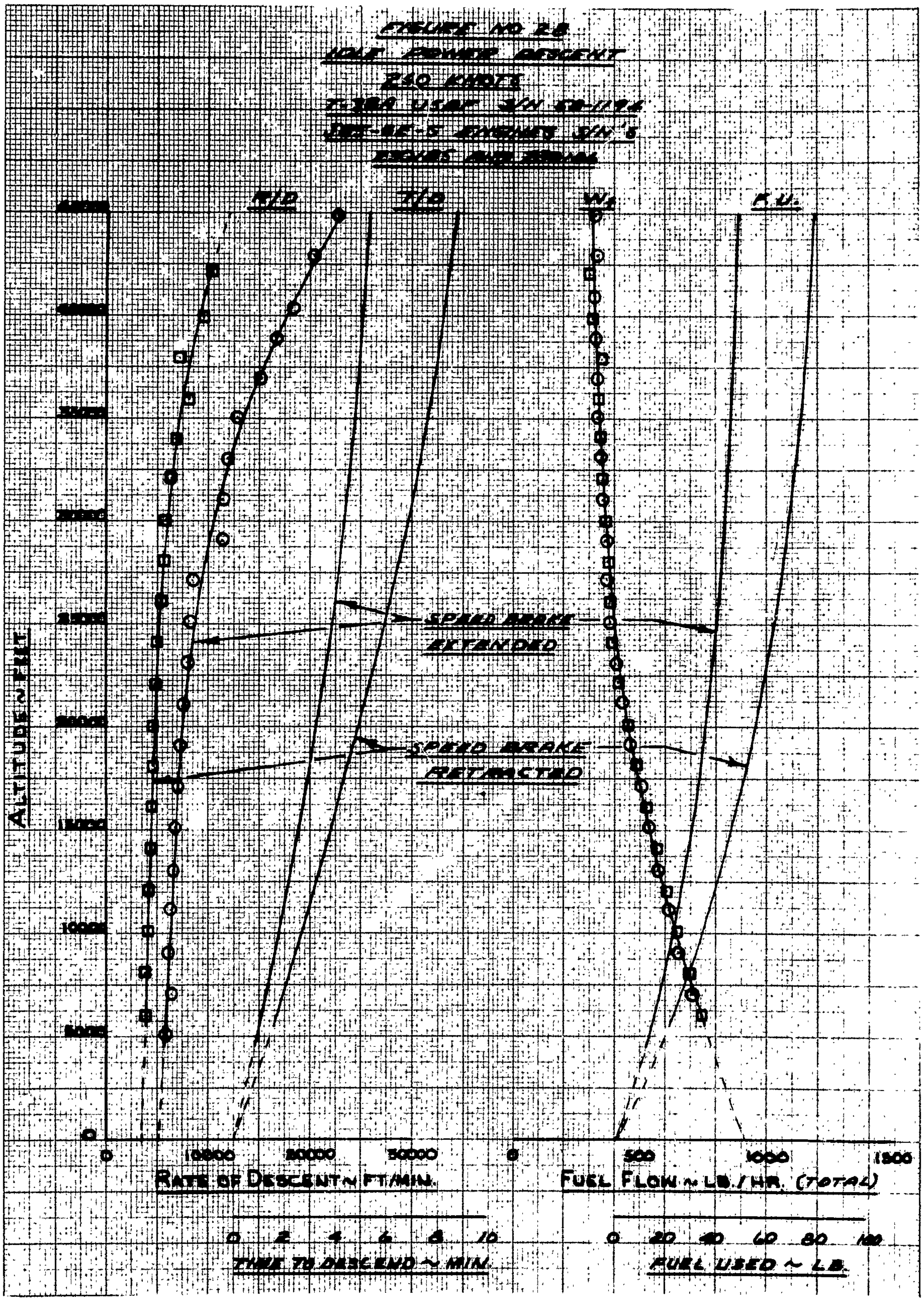


FIGURE NO. 27 CONT.
IDLE POWER DESCENT
220 KNOTS
SYM
FLT-RUN
AVE WT
6 - 5
9900 LBS.
MACH
NAMT
SPEED BRAKE EXTENDED
SPEED BRAKE RETRACTED
MACH NUMBER
NAUTICAL AIR MILES TRAVELED
AIRSPEED ~ KNOTS
ALTITUDE ~ FEET



IDLE POWER DESCENT
240 KNOTS
R/D
T/D
F.U.
SPEED BRAKE EXTENDED
SPEED BRAKE RETRACTED
ALTITUDE ~ FEET
0
10000
20000
30000
RATE OF DESCENT ~ FT/MIN.
500
1000
1500
FUEL FLOW ~ LB./HR. (TOTAL)
0 2 4 6 8 10
TIME TO DESCEND ~ MIN.
0 20 40 60 80 100
FUEL USED ~ LB.

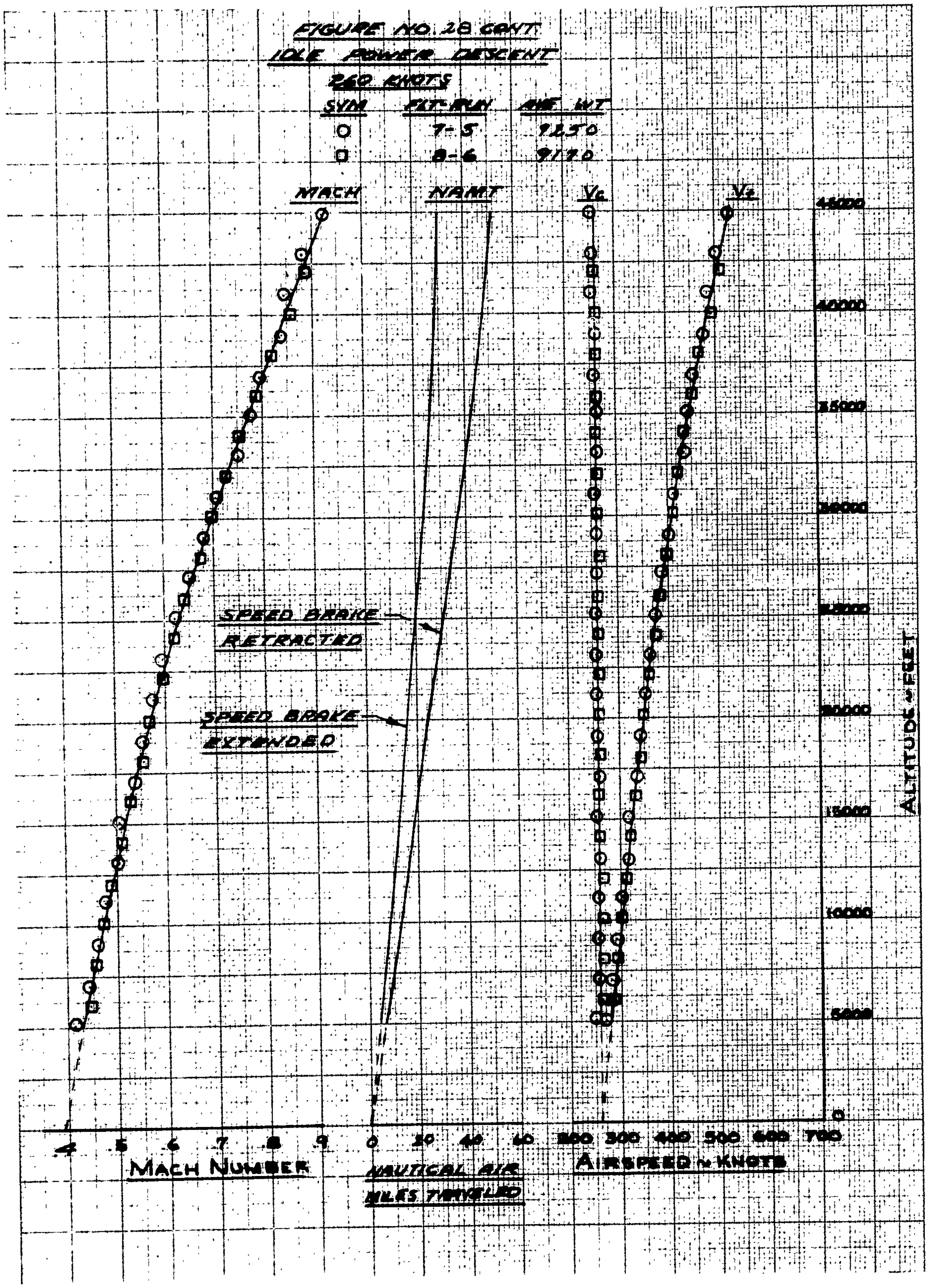
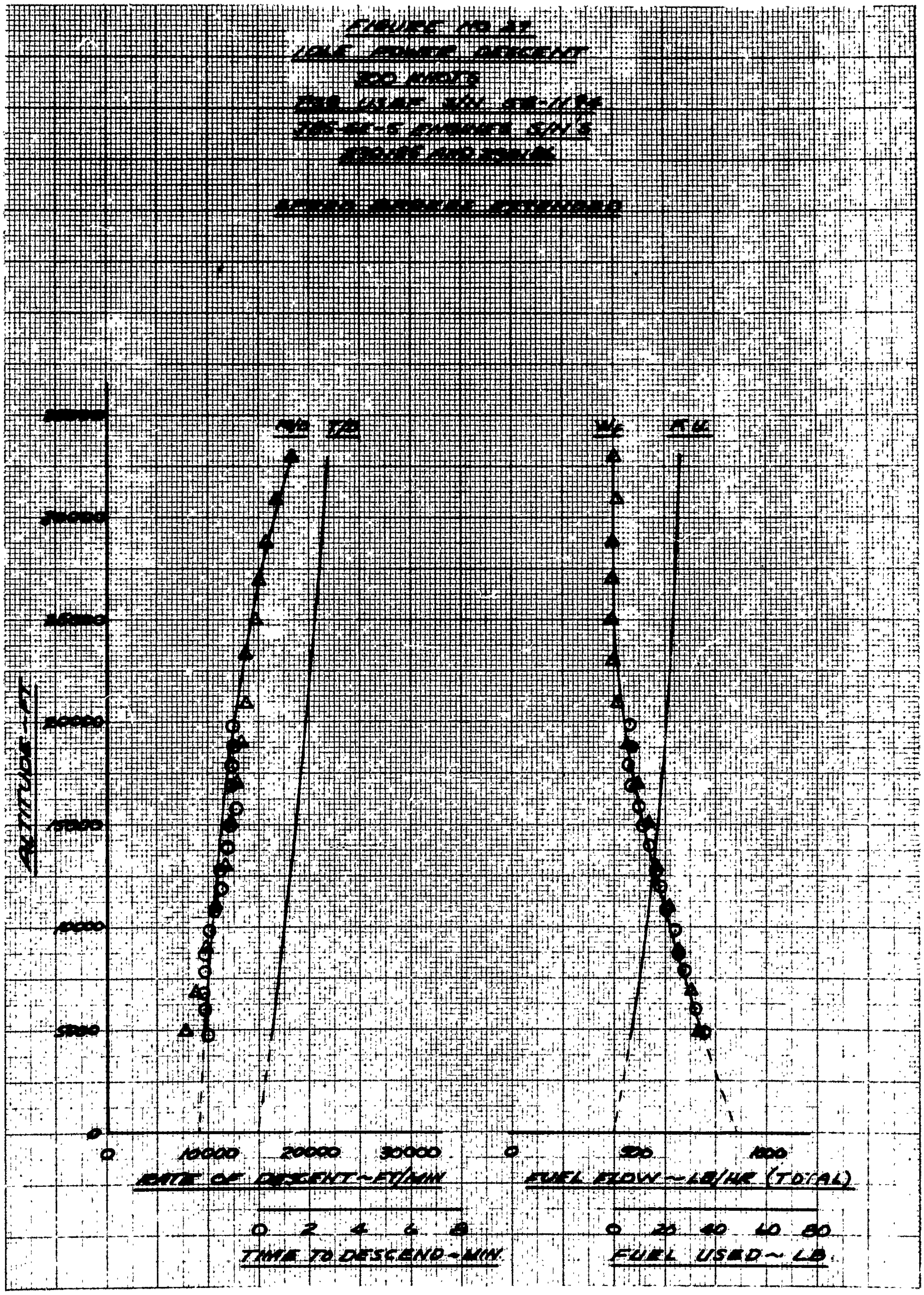


FIGURE NO. 28 CONT
IDLE POWER DESCENT
260 KNOTS
SYM FLT-RUN A/C WT
O 7-5 9250
□ 8-6 9170
MACH
NAMT
$V_c$
$V_t$
SPEED BRAKE RETRACTED
SPEED BRAKE EXTENDED
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
ALTITUDE ~ FEET
.4 .5 .6 .7 .8 .9
MACH NUMBER
0 20 40 60
NAUTICAL AIR MILES TRAVELED
200 300 400 500 600 700
AIRSPEED ~ KNOTS


IDLE POWER DESCENT
ALTITUDE ~ FT
R/D
T/D
RATE OF DESCENT ~ FT/MIN
0
10000
20000
30000
TIME TO DESCEND ~ MIN.
0
2
4
6
8
FUEL FLOW ~ LB/HR (TOTAL)
0
500
1000
FUEL USED ~ LB.
0
20
40
60
80

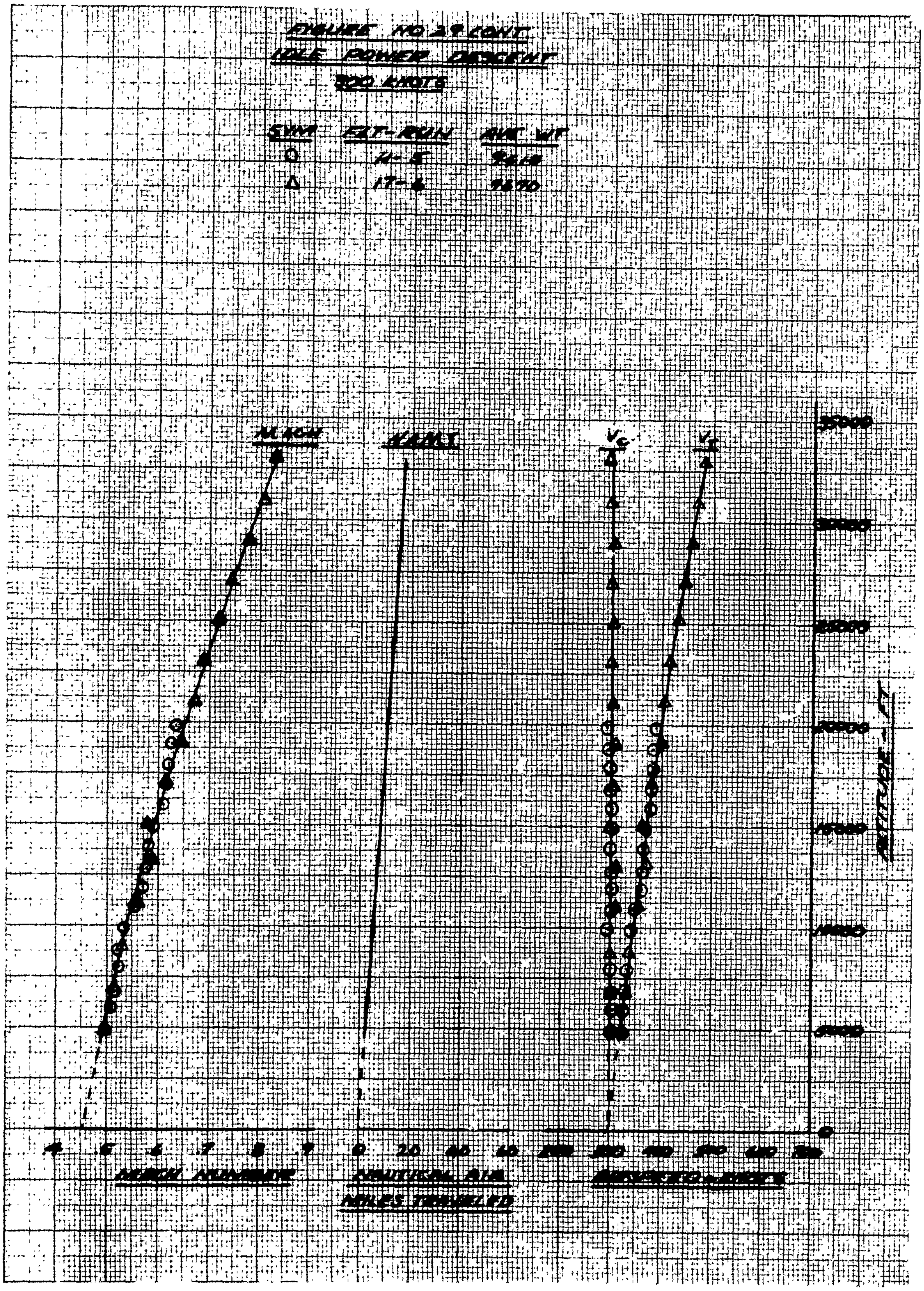


FIGURE NO 29 CONT.
IDLE POWER DESCENT
300 KNOTS
SYM
FLT-RUN
AVE WT
MACH
NAMT
MACH NUMBER
NAUTICAL AIR MILES TRAVELED
AIRSPEED - KNOTS
ALTITUDE - FT

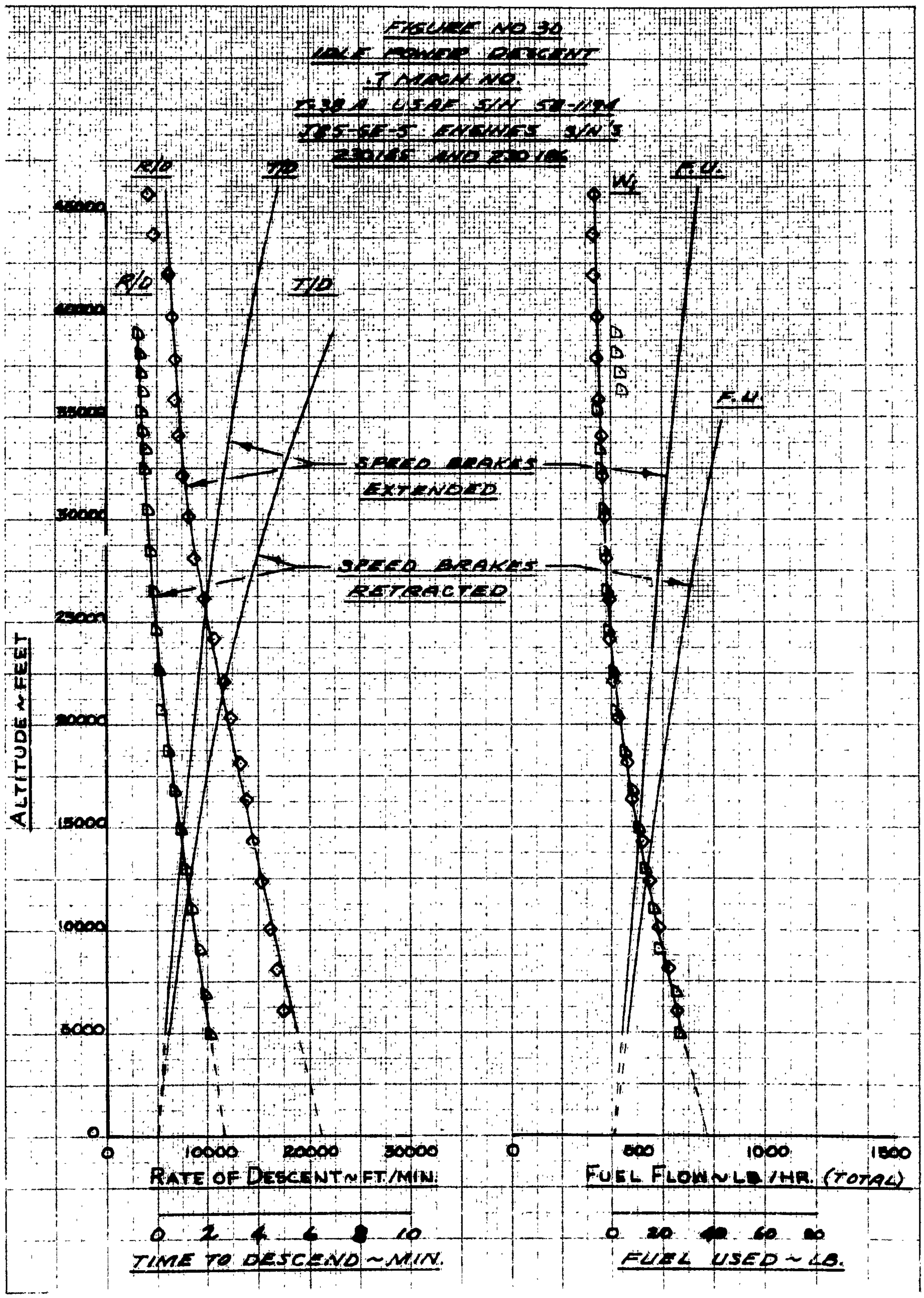


FIGURE NO 30
IDLE POWER DESCENT
.7 MACH NO.
T-38 A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230166 AND 230186
R/D
T/D
W_f
F.U.
SPEED BRAKES EXTENDED
SPEED BRAKES RETRACTED
ALTITUDE ~ FEET
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
0
10000
20000
30000
RATE OF DESCENT ~ FT/MIN.
0
500
1000
1500
FUEL FLOW ~ LB./HR. (TOTAL)
0 2 4 6 8 10
TIME TO DESCEND ~ MIN.
0 20 40 60 80
FUEL USED ~ LB.

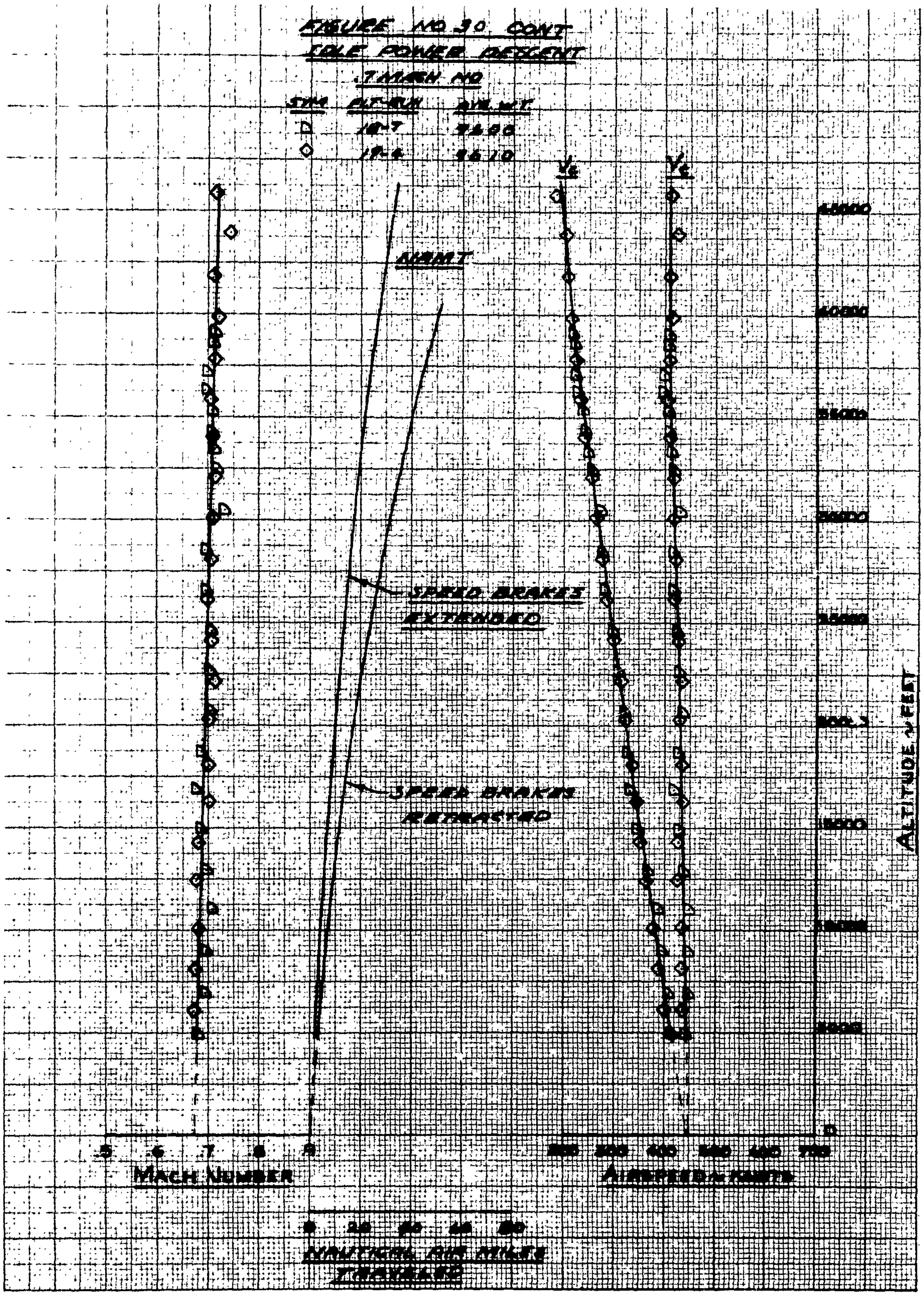


FIGURE NO 30 CONT
IDLE POWER DESCENT
.7 MACH NO
SYM
FLT-RUN
AVE. WT.
SPEED BRAKES EXTENDED
SPEED BRAKES RETRACTED
.5
.6
.7
.8
.9
MACH NUMBER
0
20
40
60
80
NAUTICAL AIR MILES TRAVELED
ALTITUDE ~ FEET

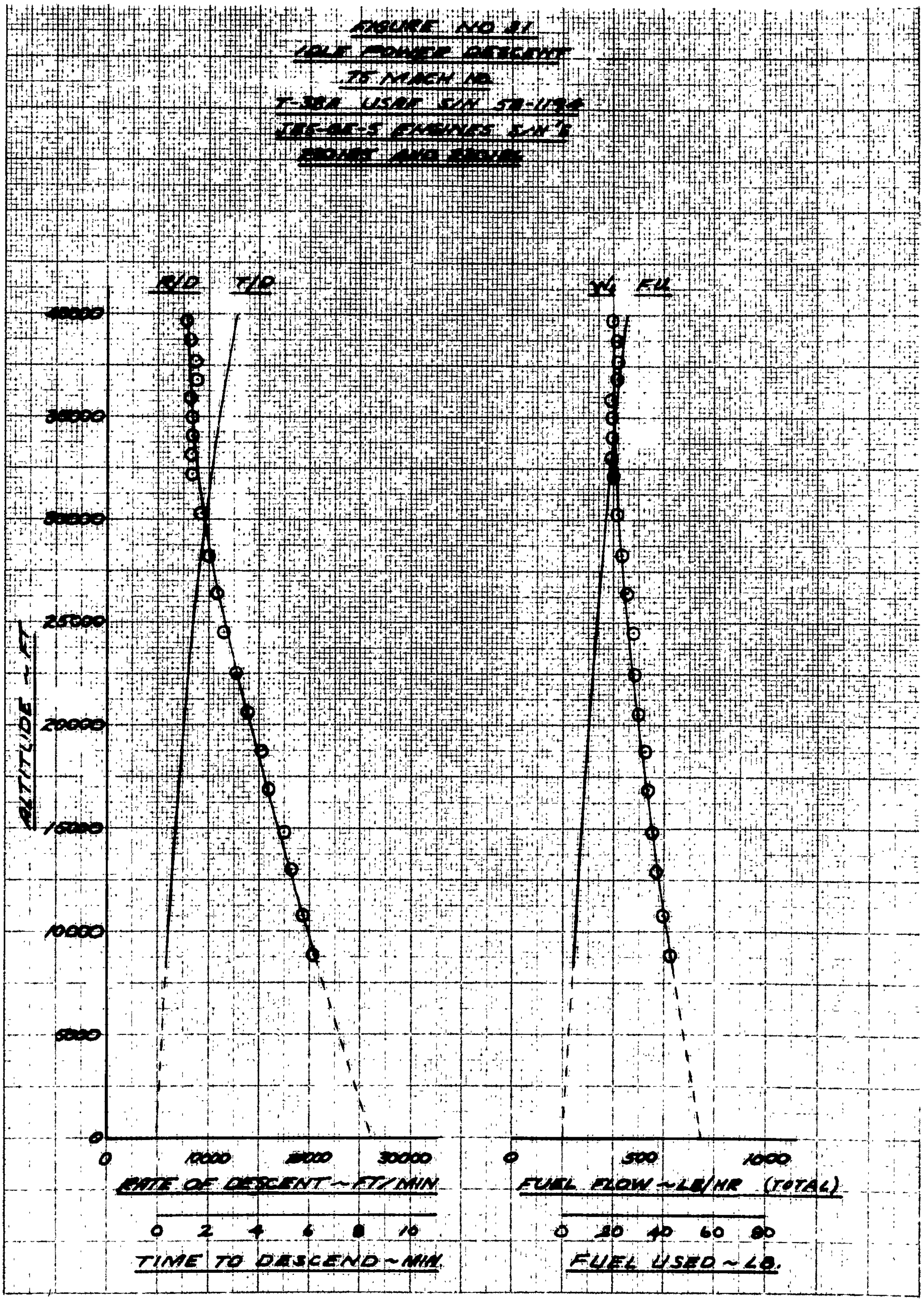


FIGURE NO 31
IDLE POWER DESCENT
.75 MACH NO.
T-38A USAF S/N
J85-GE-5 ENGINES S/N'S
R/D
T/D
Wf
F.U.
ALTITUDE ~ FT
40000
35000
30000
25000
20000
15000
10000
5000
0
0
10000
20000
30000
RATE OF DESCENT ~ FT/MIN
0 2 4 6 8 10
TIME TO DESCEND ~ MIN.
0
500
1000
FUEL FLOW ~ LB/HR (TOTAL)
0 20 40 60 80
FUEL USED ~ LB.

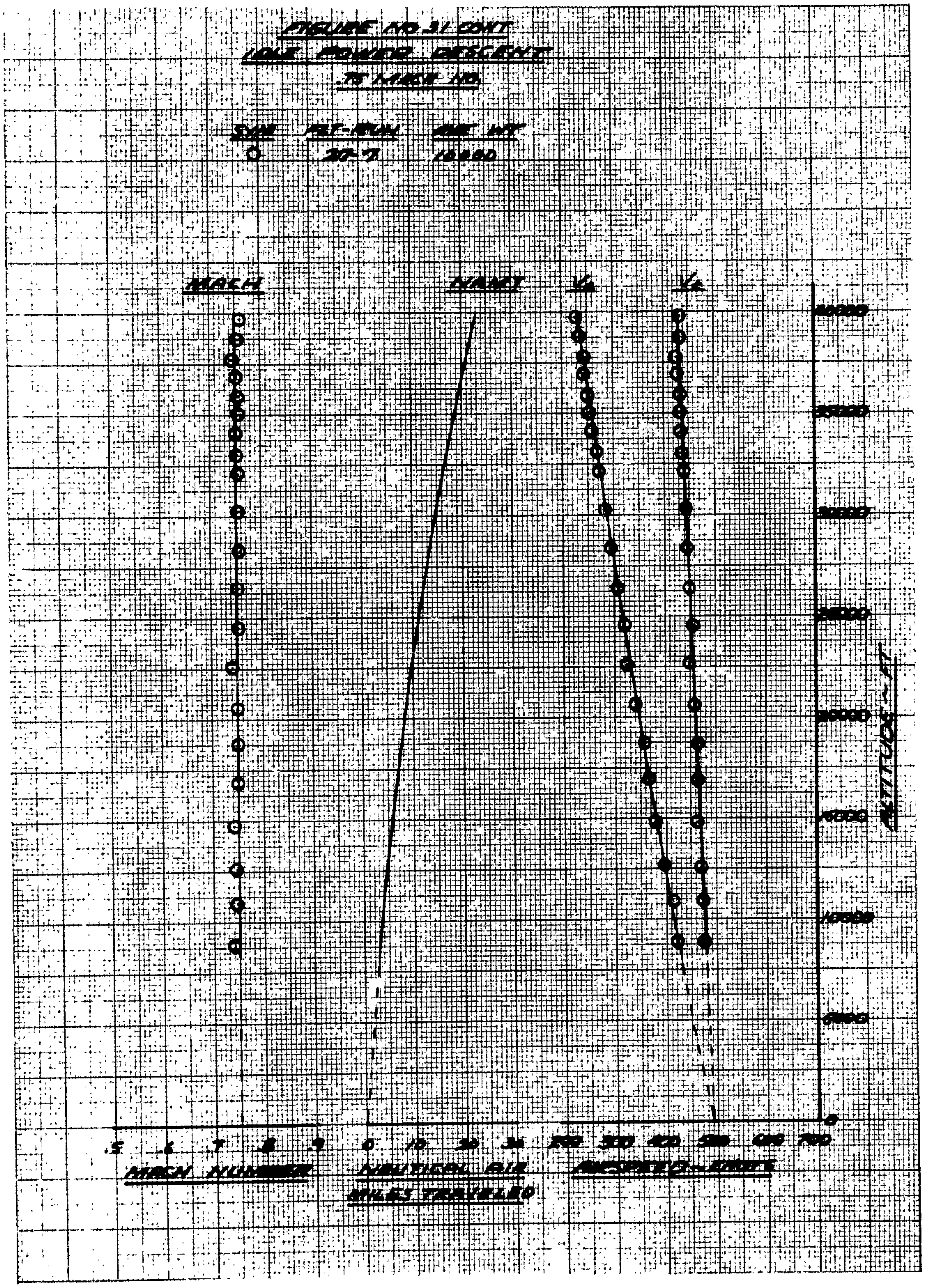
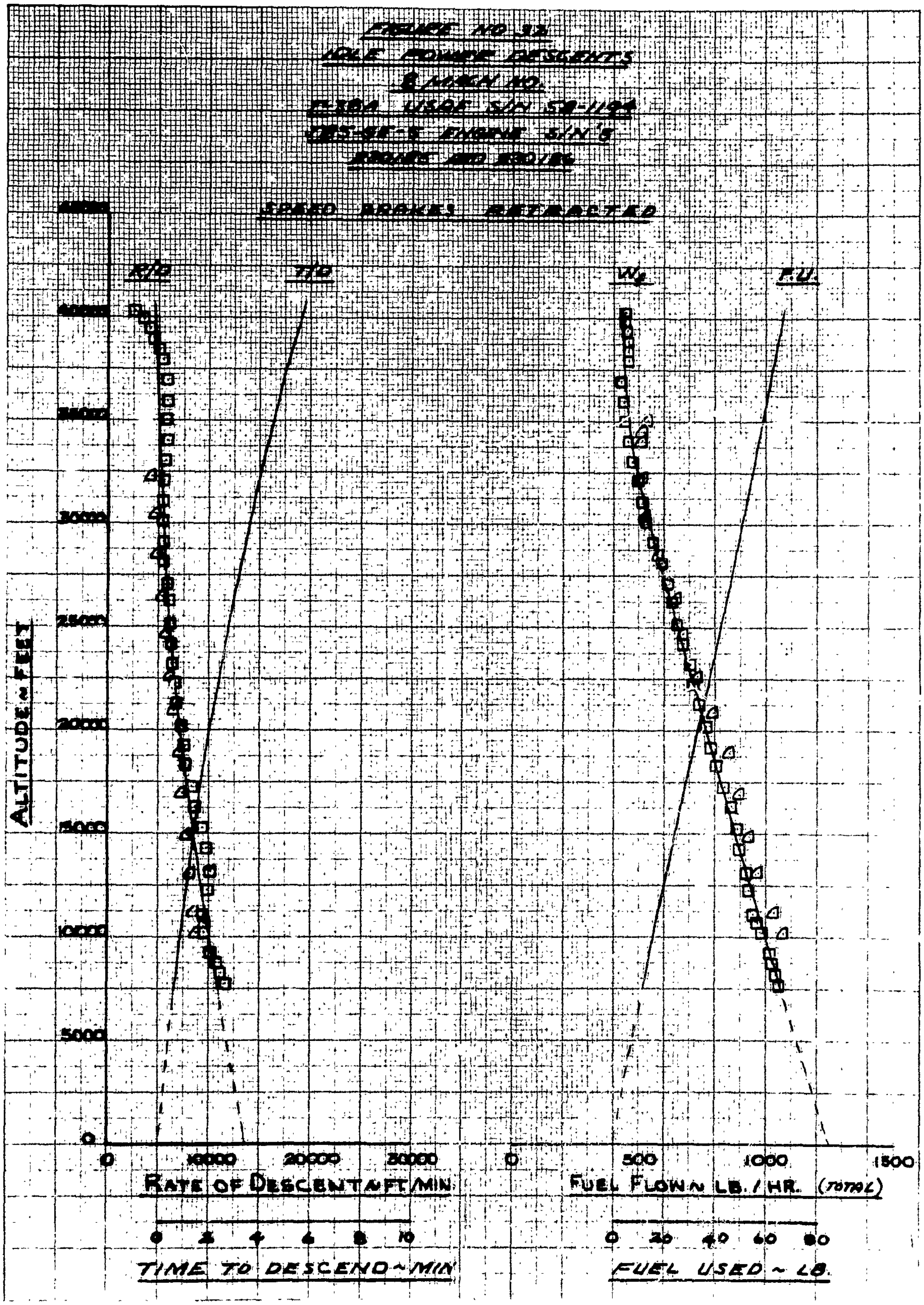



FIGURE NO 31 CONT
IDLE POWER DESCENT
VS MACH NO.



IDLE POWER DESCENTS
SPEED BRAKES RETRACTED
R/D
T/D
$W_f$
F.U.
ALTITUDE ~ FEET
0
5000
10000
15000
20000
25000
30000
35000
40000
0
10000
20000
30000
RATE OF DESCENT ~ FT/MIN
0
500
1000
1500
FUEL FLOW ~ LB./HR. (TOTAL)
0
2
4
6
8
10
TIME TO DESCEND ~ MIN
0
20
40
60
80
FUEL USED ~ LB.

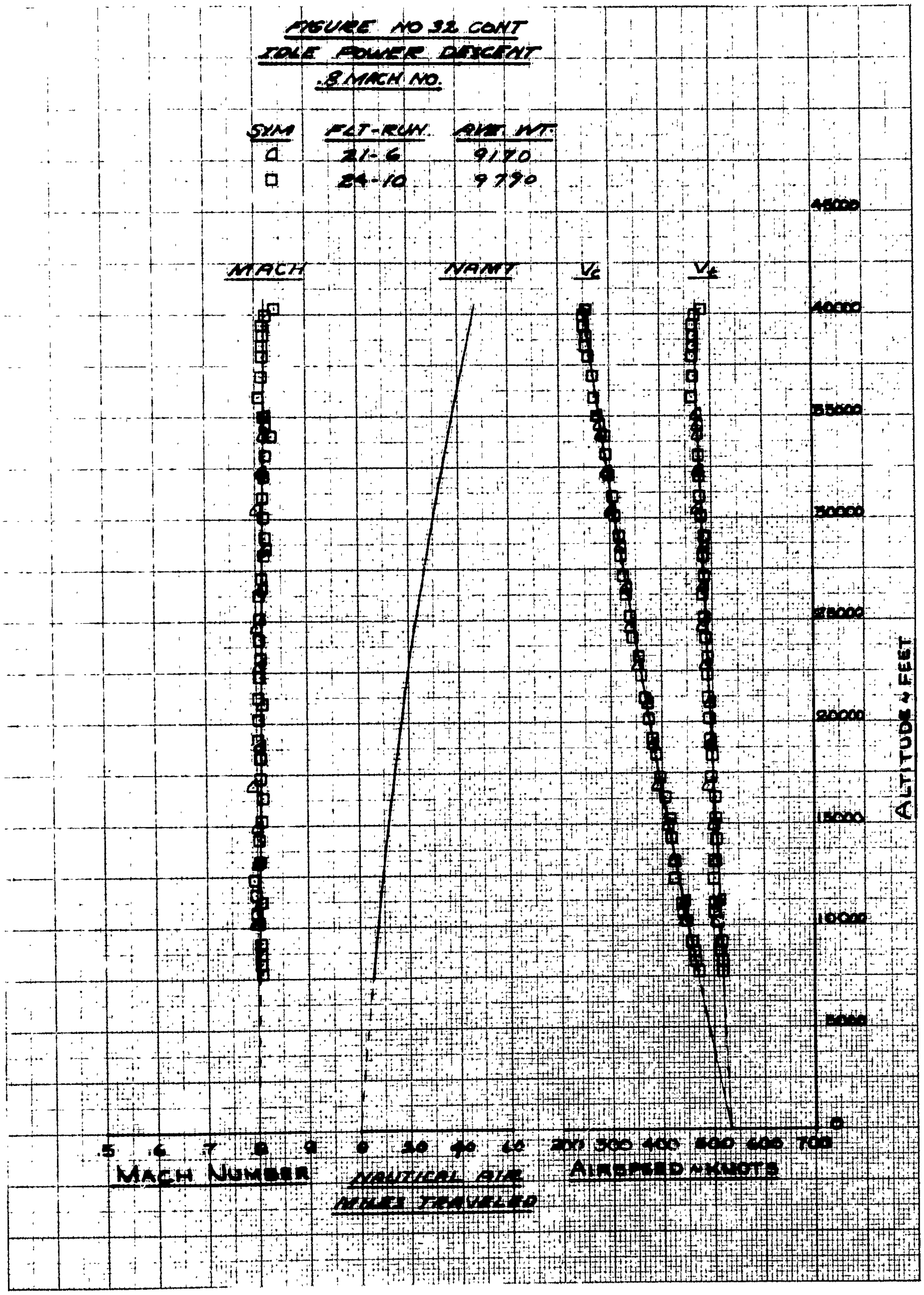
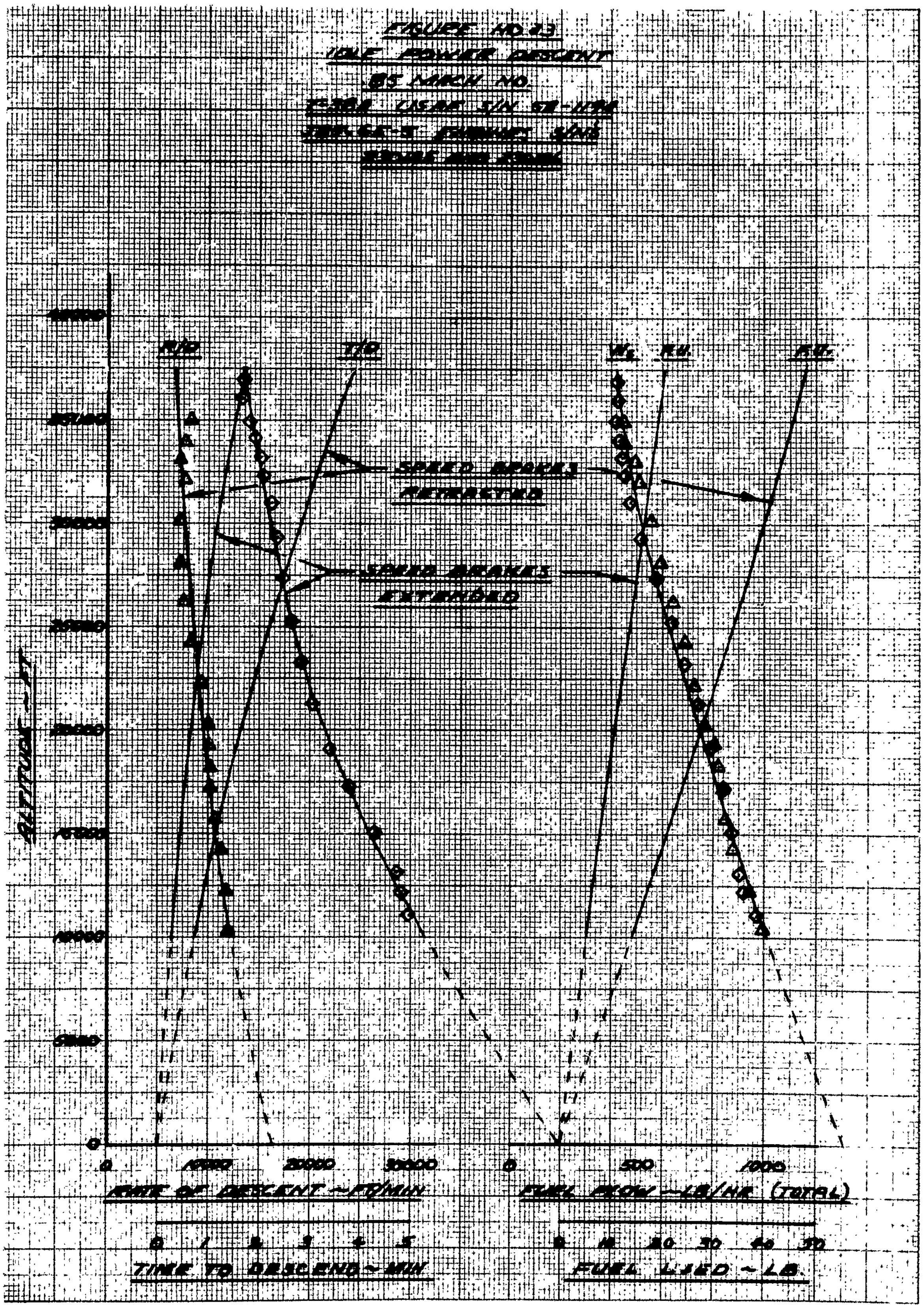


FIGURE NO 32 CONT
IDLE POWER DESCENT
.8 MACH NO.
SYM FLT-RUN AVE WT
△ 21-6 9170
□ 24-10 9790
MACH
NAMT
$V_c$
$V_t$
ALTITUDE ~ FEET
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
.5 .6 .7 .8 .9
0 20 40 60
200 300 400 500 600 700
MACH NUMBER
NAUTICAL AIR MILES TRAVELED
AIRSPEED ~ KNOTS


IDLE POWER DESCENT
.85 MACH NO.
R/D
T/D
F.U.
SPEED BRAKES RETRACTED
SPEED BRAKES EXTENDED
ALTITUDE ~ FT
0
5000
10000
15000
20000
25000
30000
35000
40000
45000
0
10000
20000
30000
RATE OF DESCENT ~ FT/MIN
0
500
1000
FUEL FLOW ~ LB/HR (TOTAL)
0
1
2
3
4
5
TIME TO DESCEND ~ MIN
0
10
20
30
40
50
FUEL USED ~ LB.

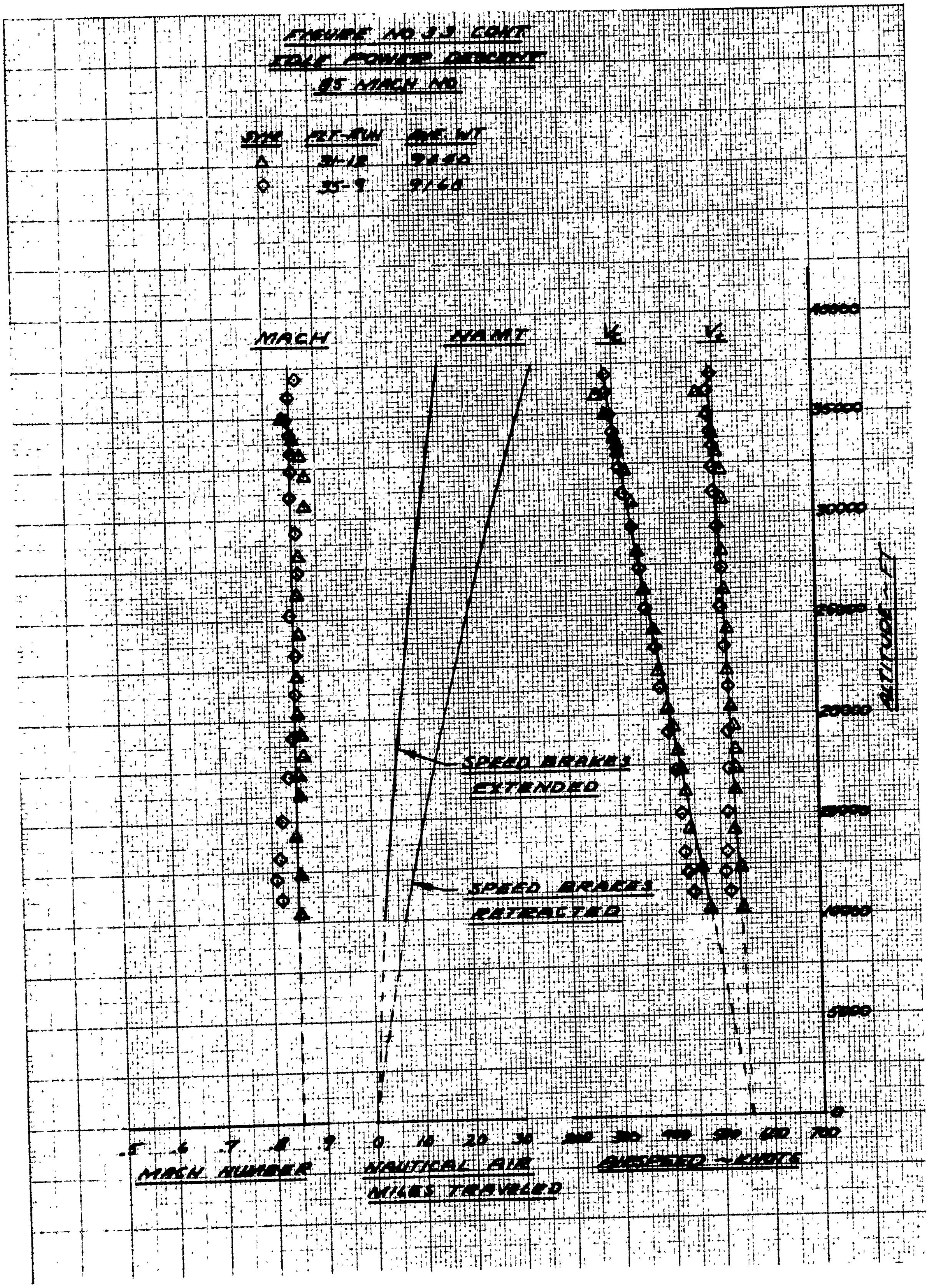


FIGURE NO 33 CONT
IDLE POWER DESCENT
.85 MACH NO
SYM
FLT-RUN
AVE. WT
31-12
35-9
9160
MACH
NAMT
SPEED BRAKES EXTENDED
SPEED BRAKES RETRACTED
40000
35000
30000
25000
20000
15000
10000
5000
0
ALTITUDE ~ FT
.5
.6
.7
.8
.9
0
10
20
30
200
300
400
500
600
700
MACH NUMBER
NAUTICAL AIR MILES TRAVELED
AIRSPEED ~ KNOTS

FIGURE NO 34

AIRSPEED SYSTEM CALIBRATION

T-38A USAF S/N 58-1194

TEST NOSE BOOM

| SYM | FLT NO | METHOD | DATE | CONFIGURATION |
|---|---|---|---|---|
| ○ | 9 | TOWER FLY-BY | 20 JUNE 1961 | CLEAN |
| □ | 10 | TOWER FLY-BY | 21 JUNE 1961 | CLEAN |
| △▽ | 13 | SMOKE TRAIL ACCEL. | 27 JUNE 1961 | CLEAN |
| ● | 22 | STABILIZED PACE | 8 SEPT 1961 | GEAR DOWN, 60% FLAPS |

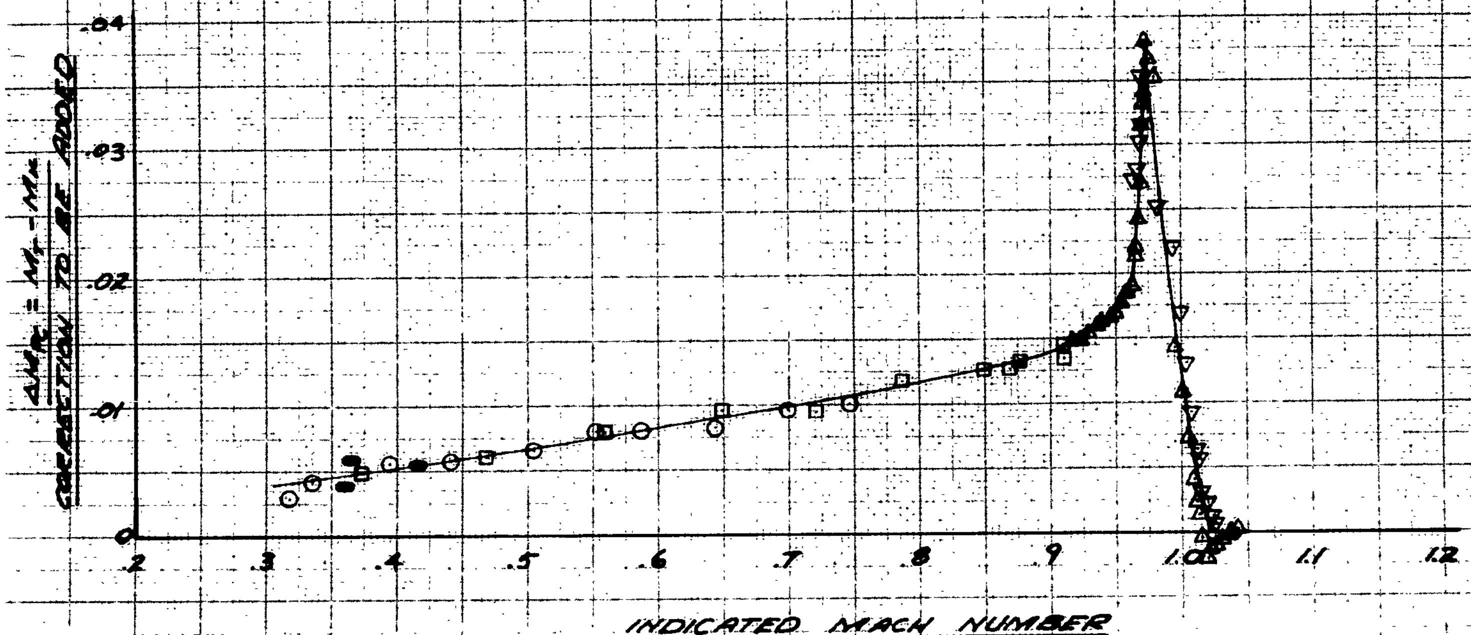
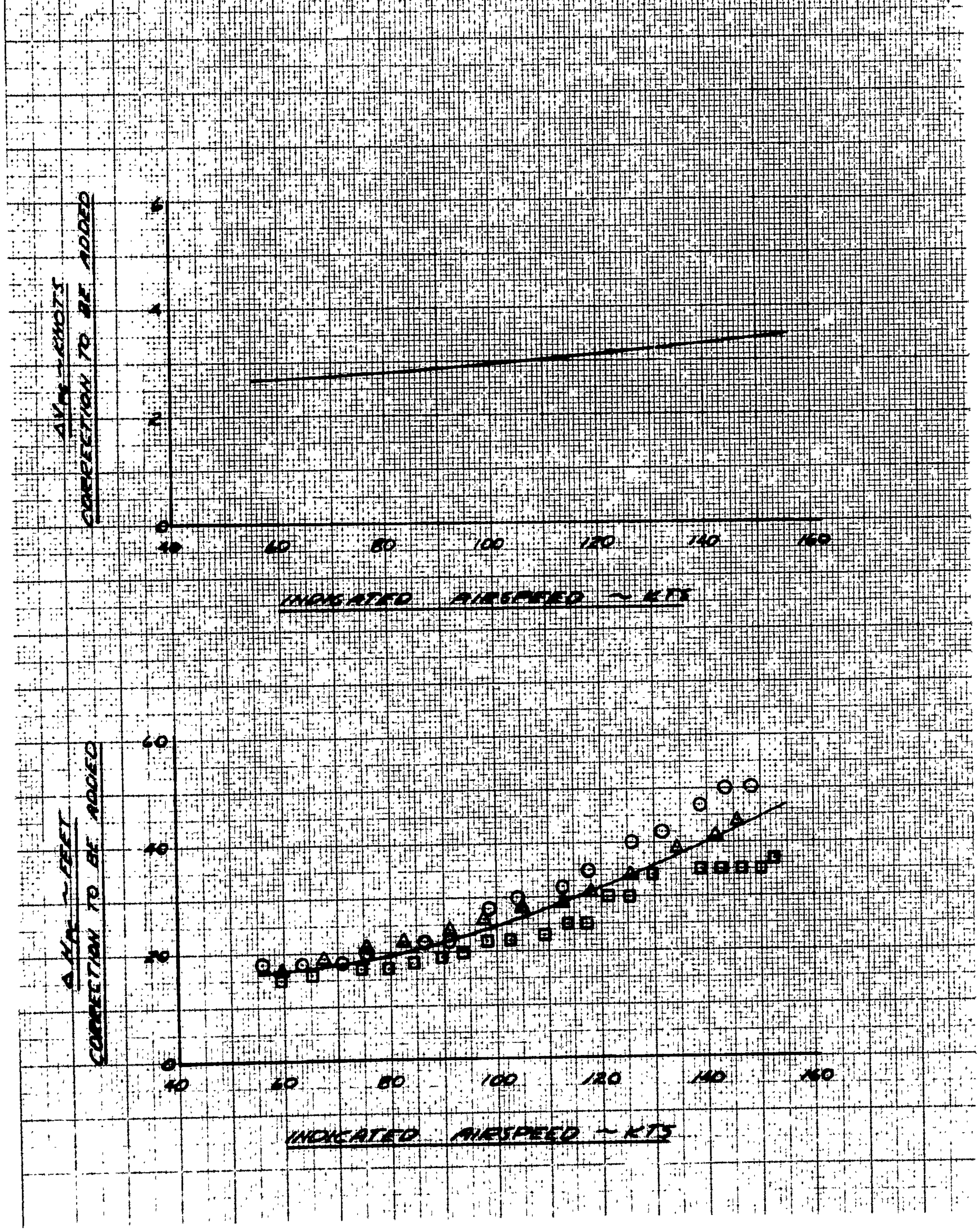


FIGURE NO 35

AIRSPEED CALIBRATION IN GROUND EFFECT

T-28A USAF S/N 51-1194

TEST NOSE BOOM



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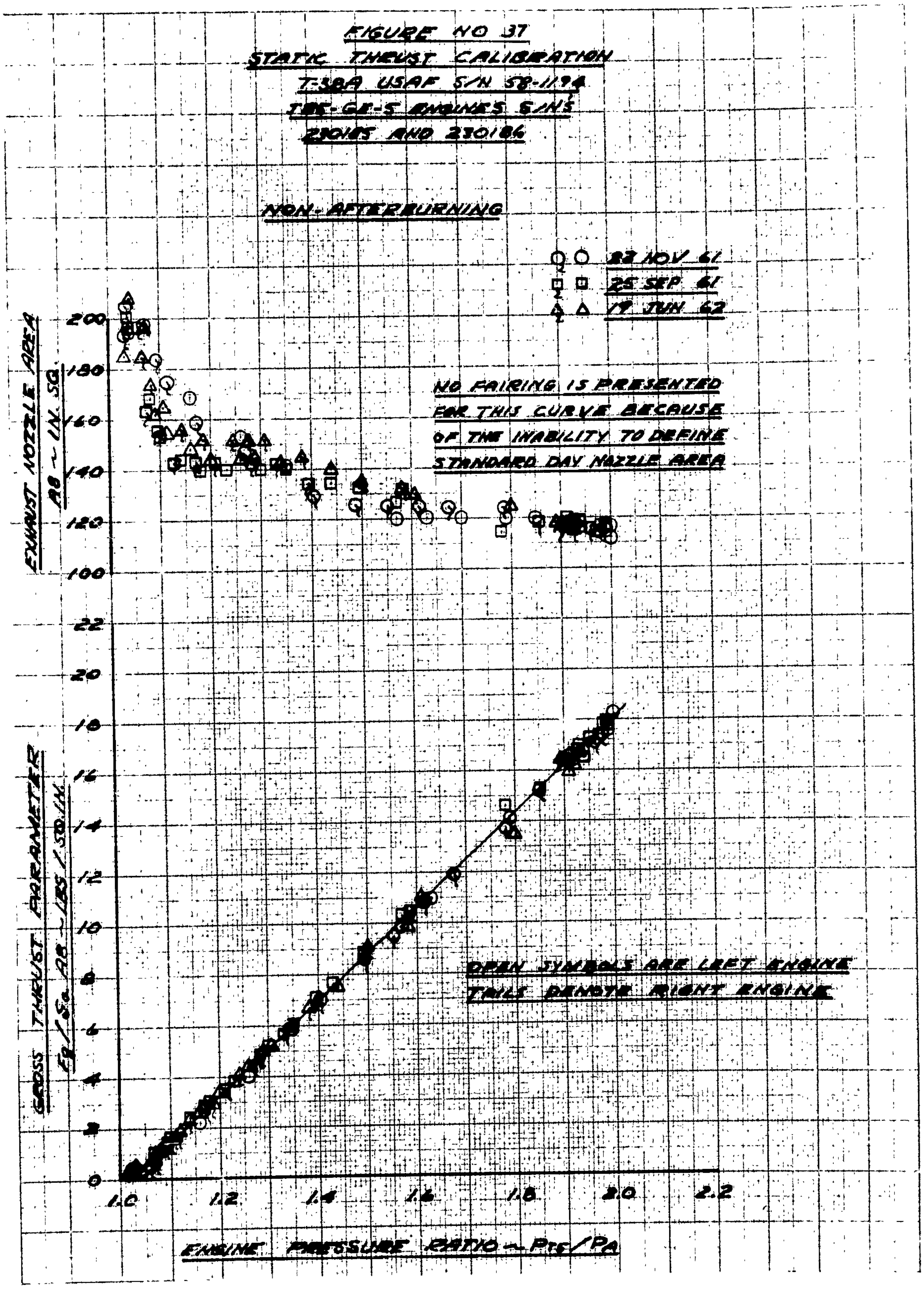


FIGURE NO 37
STATIC THRUST CALIBRATION
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
NON-AFTERBURNING
23 NOV 61
25 SEP 61
19 JUN 62
NO FAIRING IS PRESENTED FOR THIS CURVE BECAUSE OF THE INABILITY TO DEFINE STANDARD DAY NOZZLE AREA
EXHAUST NOZZLE AREA
A8 ~ IN. SQ.
GROSS THRUST PARAMETER
Fg/δa A8 ~ LBS / SQ. IN.
OPEN SYMBOLS ARE LEFT ENGINE
TAILS DENOTE RIGHT ENGINE
ENGINE PRESSURE RATIO ~ Pt5/Pa
200
180
160
140
120
100
22
20
18
16
14
12
10
8
6
4
2
0
1.0
1.2
1.4
1.6
1.8
2.0
2.2

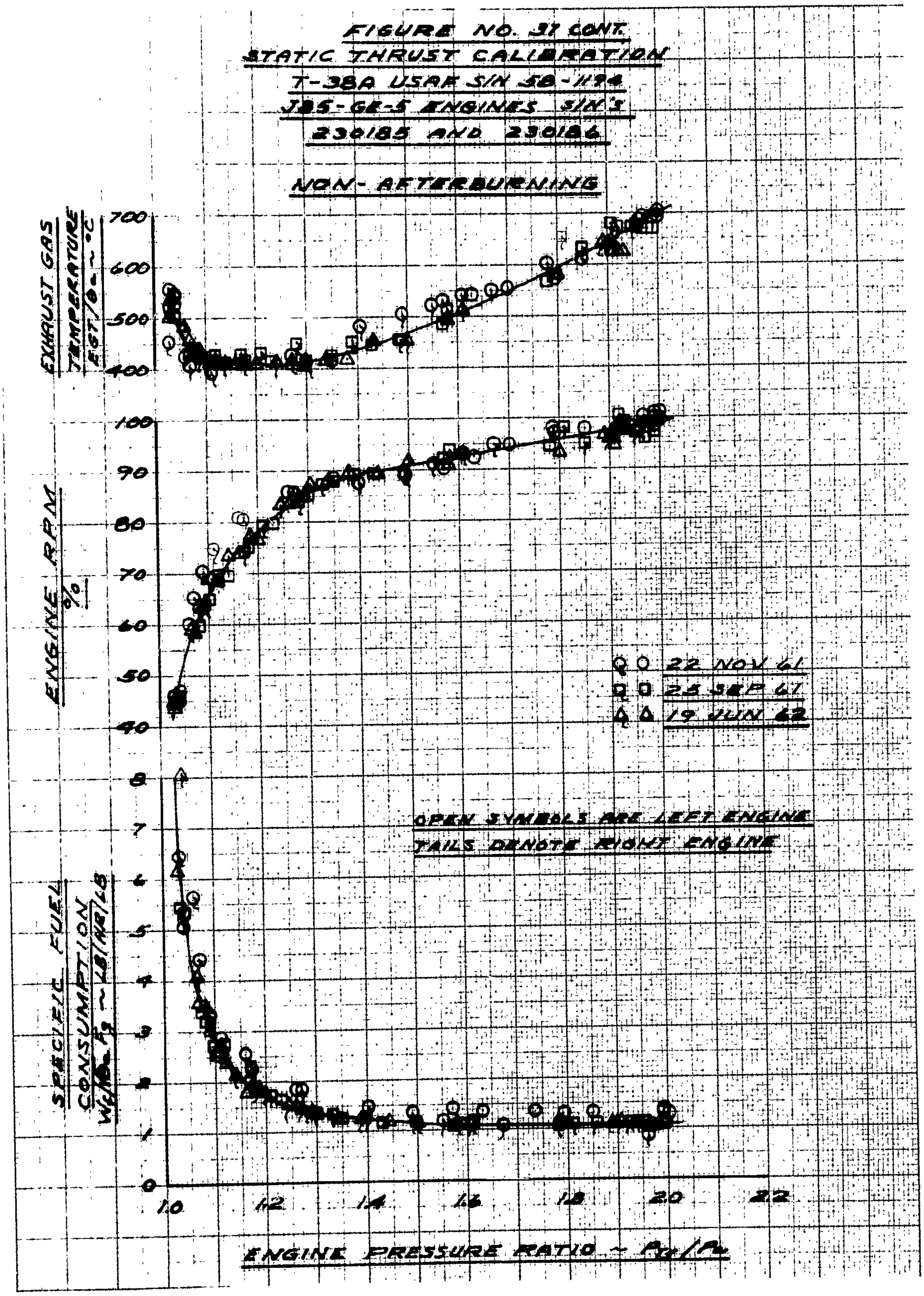


FIGURE NO. 37 CONT.
STATIC THRUST CALIBRATION
T-38A USAF S/N 58-1194
J85-GE-5 ENGINES S/N'S
230185 AND 230186
NON-AFTERBURNING
EXHAUST GAS TEMPERATURE
ENGINE RPM
%
SPECIFIC FUEL CONSUMPTION
22 NOV 61
25 SEP 61
19 JUN 62
OPEN SYMBOLS ARE LEFT ENGINE
TAILS DENOTE RIGHT ENGINE
ENGINE PRESSURE RATIO

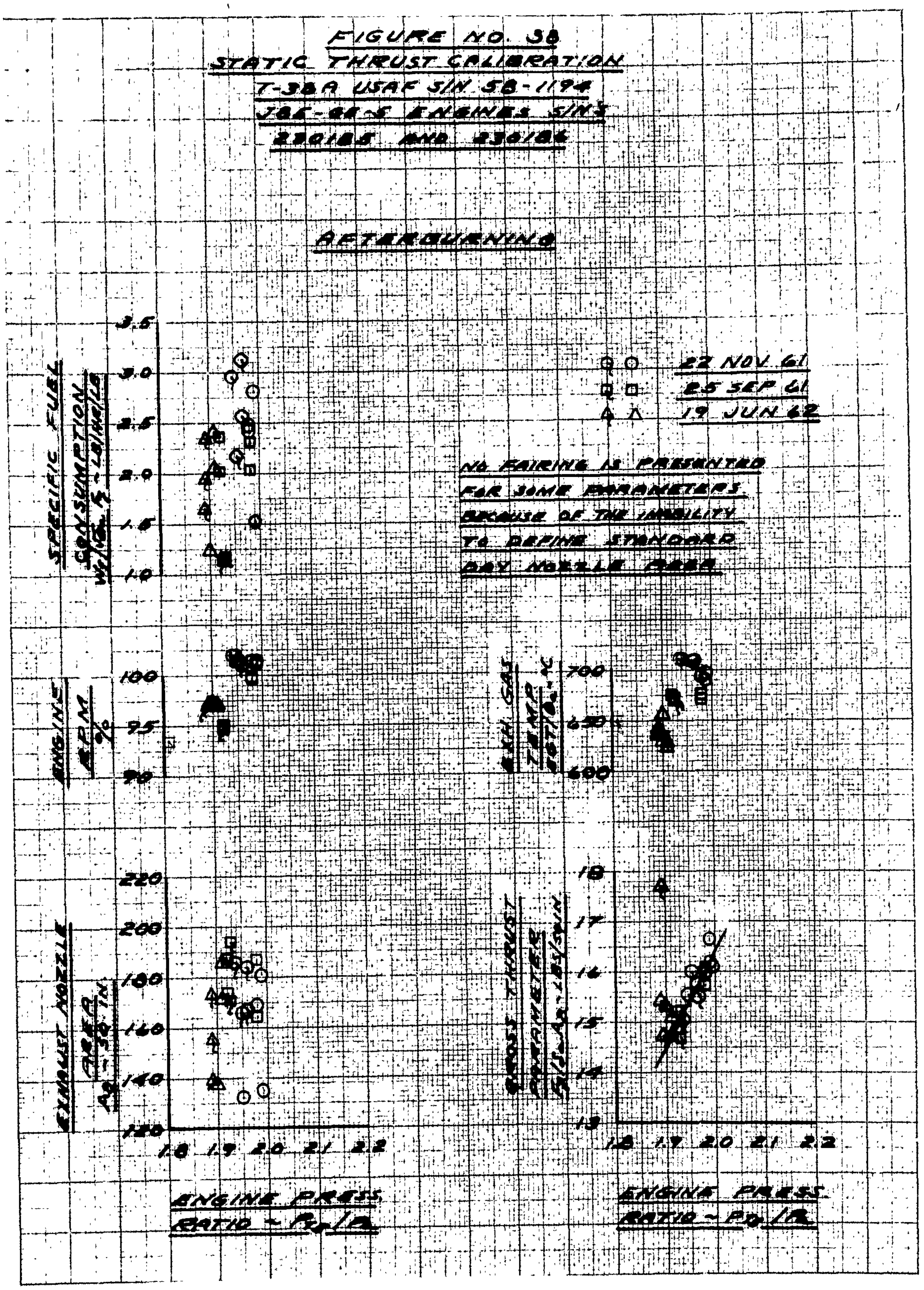


FIGURE NO. 38
STATIC THRUST CALIBRATION
T-38A USAF S/N 58-1194
22 NOV 61
25 SEP 61
19 JUN 62
SPECIFIC FUEL CONSUMPTION
3.5
3.0
2.5
2.0
1.5
1.0
ENGINE RPM %
100
95
90
EXH. GAS TEMP
700
650
600
EXHAUST NOZZLE AREA ~ SQ. IN.
220
200
180
160
140
120
GROSS THRUST PARAMETER
18
17
16
15
14
13
1.8 1.9 2.0 2.1 2.2
ENGINE PRESS. RATIO
1.8 1.9 2.0 2.1 2.2
ENGINE PRESS. RATIO

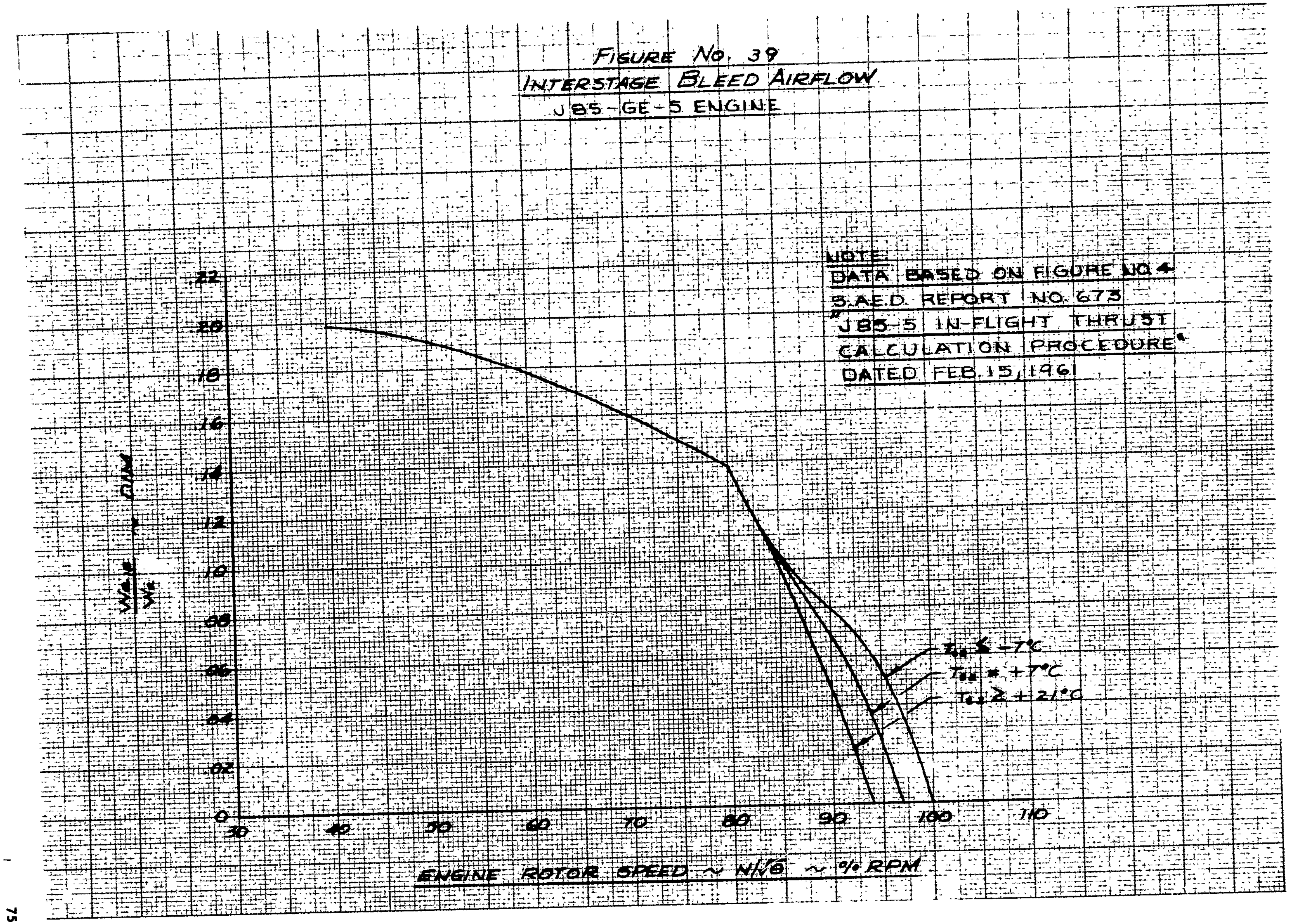


FIGURE NO. 39
INTERSTAGE BLEED AIRFLOW
J85-GE-5 ENGINE
NOTE:
DATA BASED ON FIGURE NO 4
S.A.E.D. REPORT NO. 673
"J85-5 IN-FLIGHT THRUST
CALCULATION PROCEDURE"
DATED FEB. 15, 1961
.22
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0
30
40
50
60
70
80
90
100
110
ENGINE ROTOR SPEED ~ N/√θ ~ % RPM

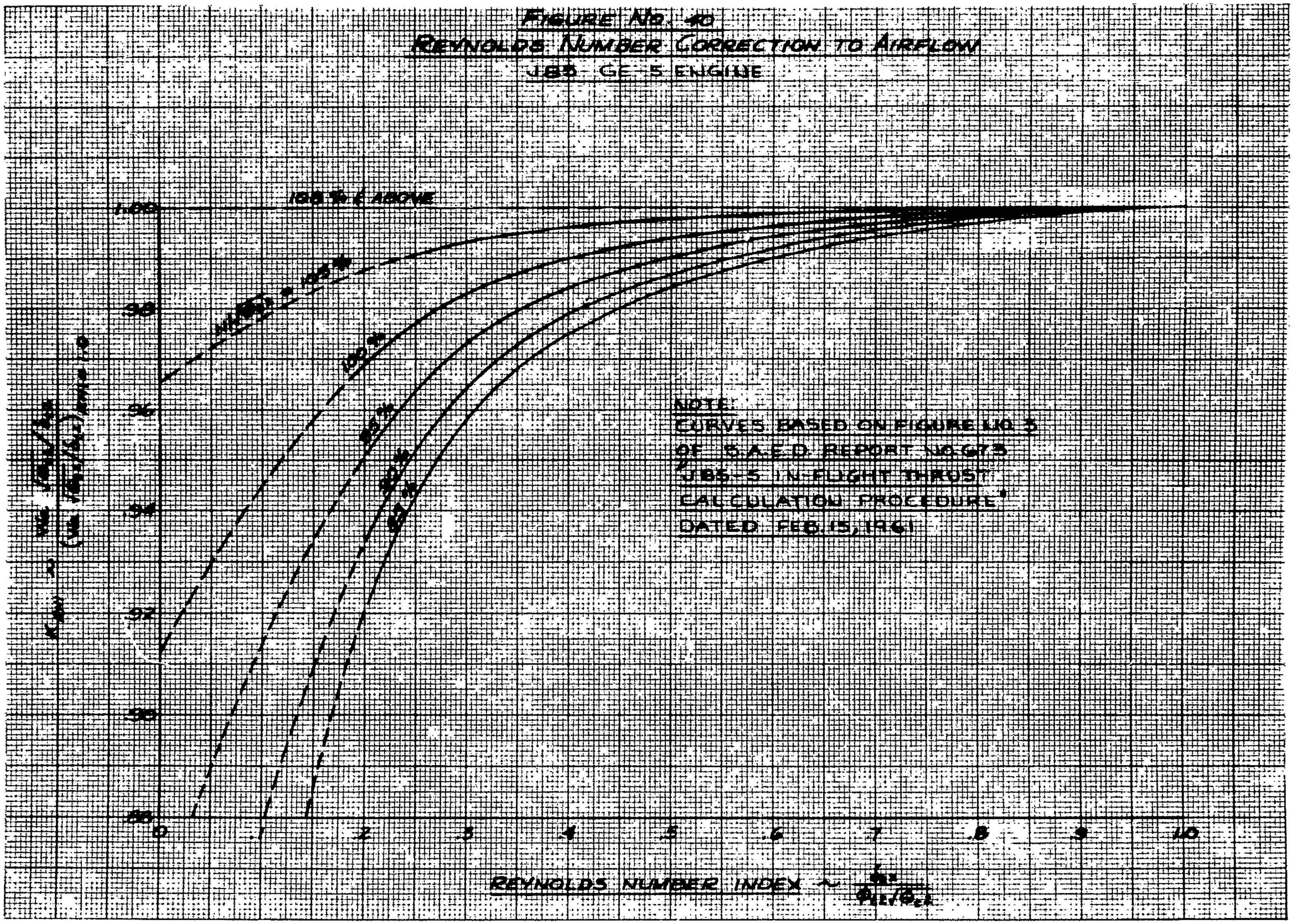
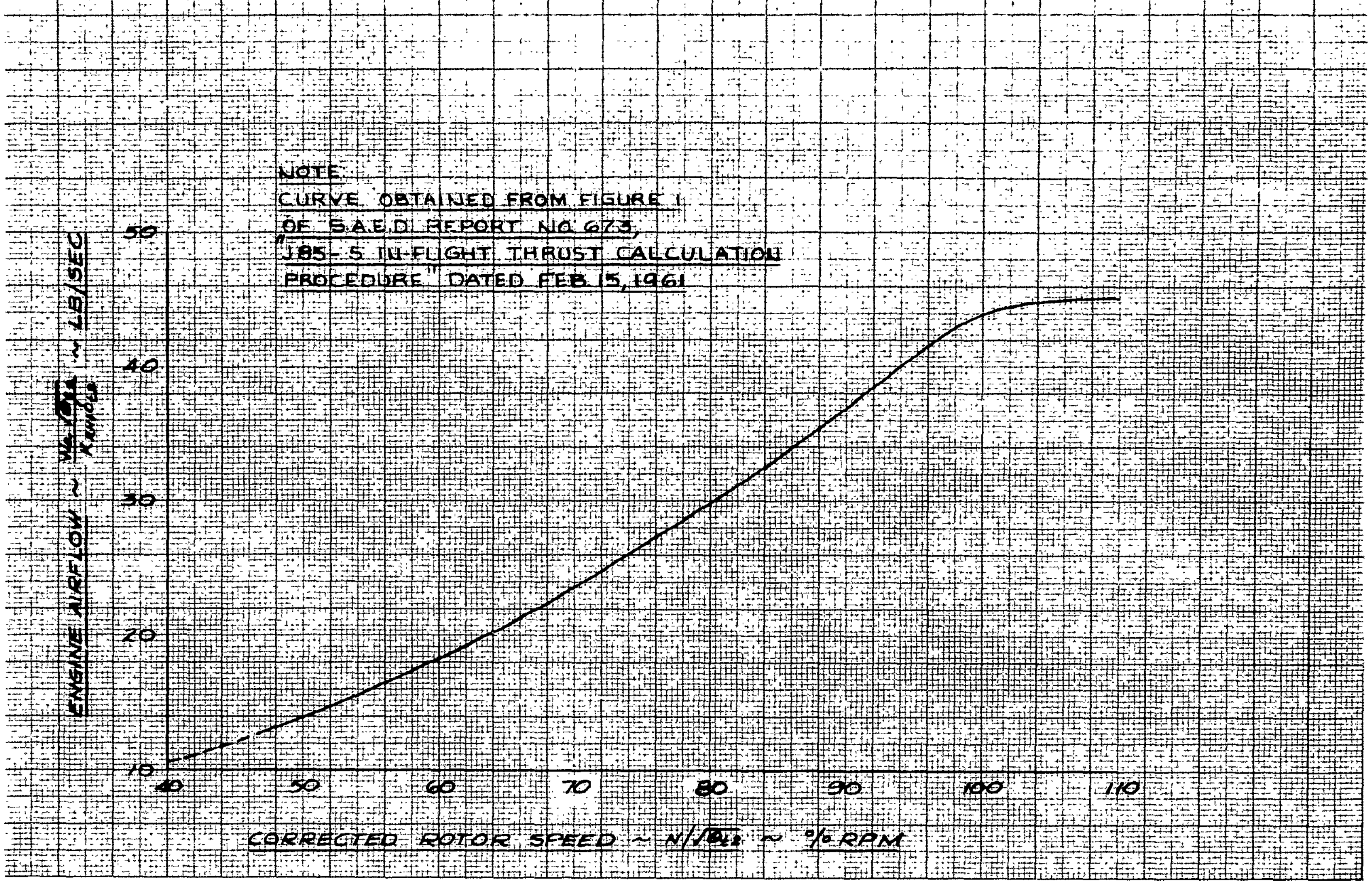


FIGURE No. 40
REYNOLDS NUMBER CORRECTION TO AIRFLOW
J85 GE 5 ENGINE
100% & ABOVE
NOTE:
CURVES BASED ON FIGURE NO 3
OF S.A.E.D. REPORT NO.673
"J85-5 IN-FLIGHT THRUST
CALCULATION PROCEDURE"
DATED FEB.15, 1961
1.00
.98
.96
.94
.92
.90
.88
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
REYNOLDS NUMBER INDEX

FIGURE NO. 41
J-85-5 AIRFLOW



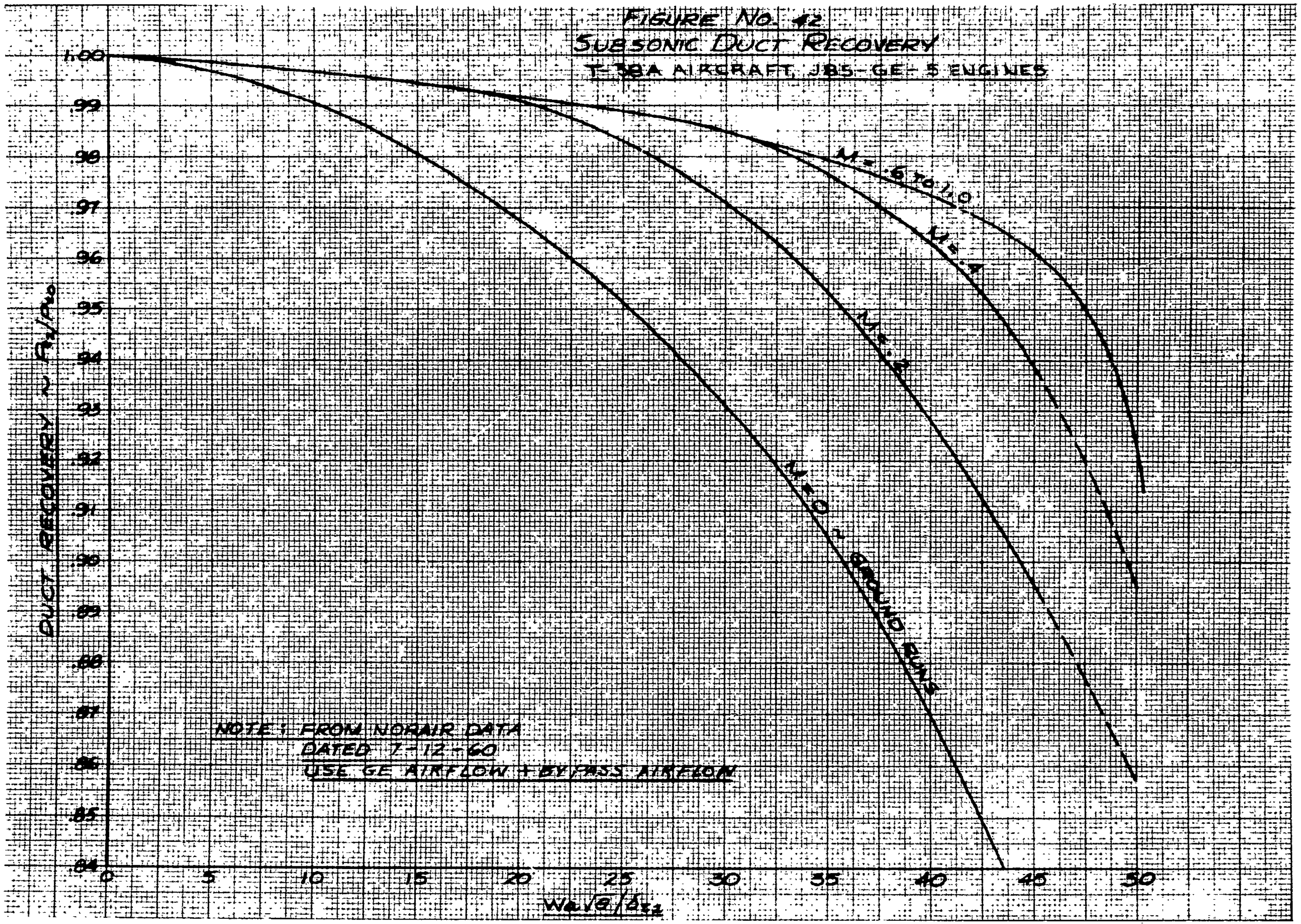


FIGURE NO. 42
SUBSONIC DUCT RECOVERY
T-38A AIRCRAFT, J85-GE-5 ENGINES
M = .6 TO 1.0
M = .4
M = .2
M = 0 ~ GROUND RUNS
NOTE: FROM NORAIR DATA
DATED 7-12-60
USE GE AIRFLOW + BYPASS AIRFLOW
DUCT RECOVERY ~ $P_{t2}/P_{t0}$
1.00
.99
.98
.97
.96
.95
.94
.93
.92
.91
.90
.89
.88
.87
.86
.85
.84
0
5
10
15
20
25
30
35
40
45
50
$W_a\sqrt{\theta}/\delta_{t2}$

FIGURE No. 43
SUPERSONIC DUCT RECOVERY
T-38A AIRCRAFT
J85-GE-5 ENGINES

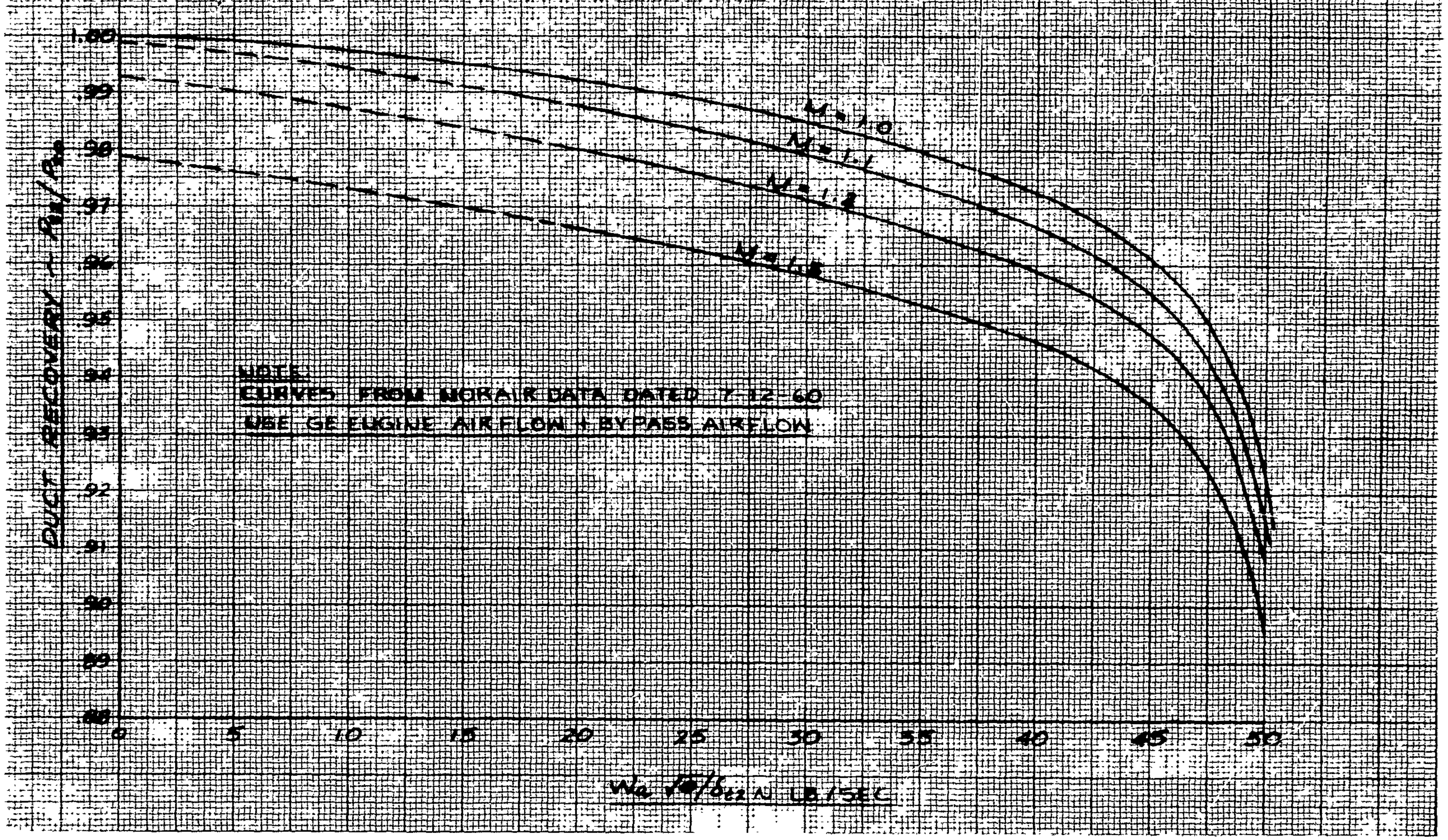


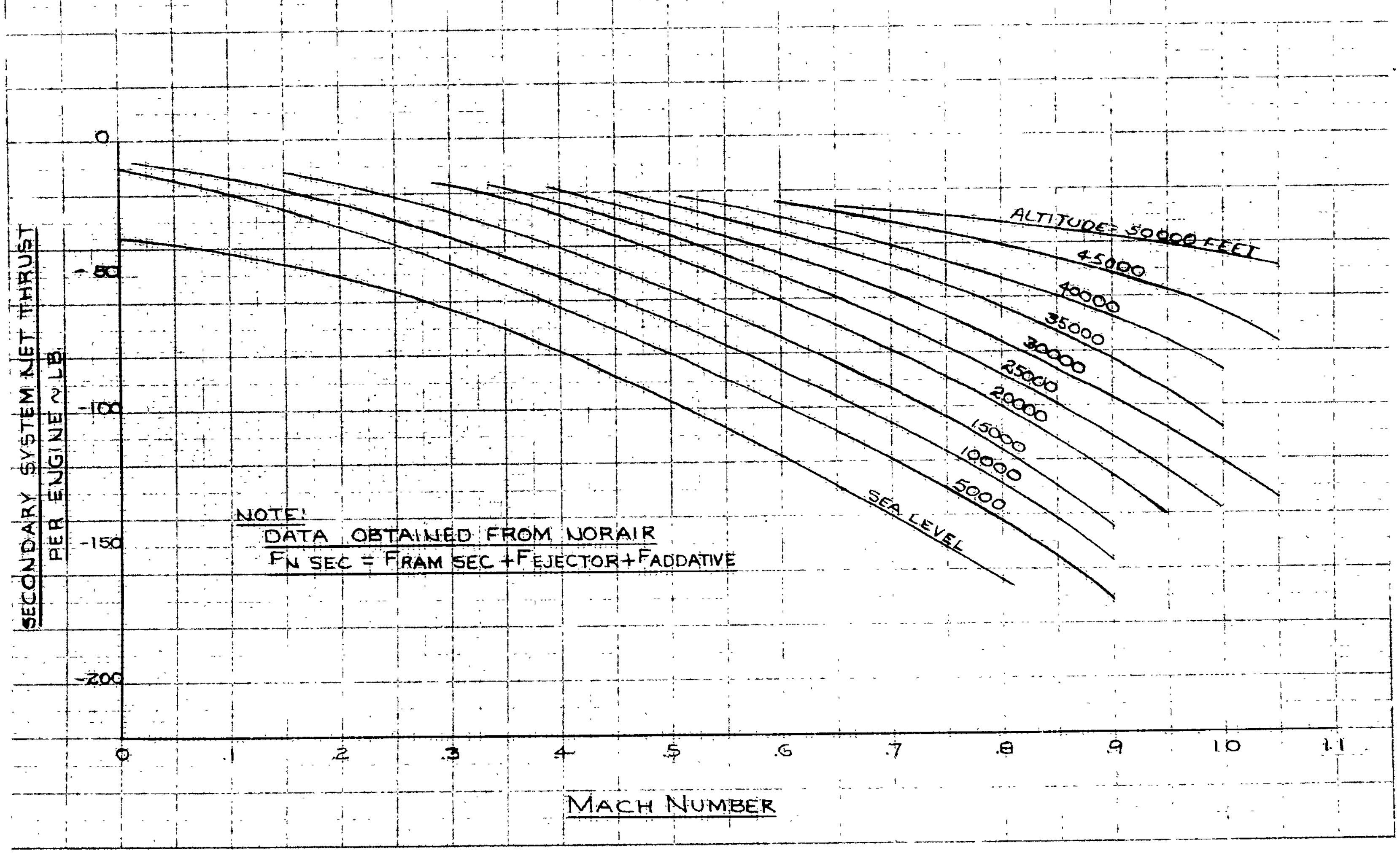
FIGURE NO. 44

SECONDARY SYSTEM NET THRUST

REDUCED POWER

T-38A AIRCRAFT

J85-GE-5 ENGINES



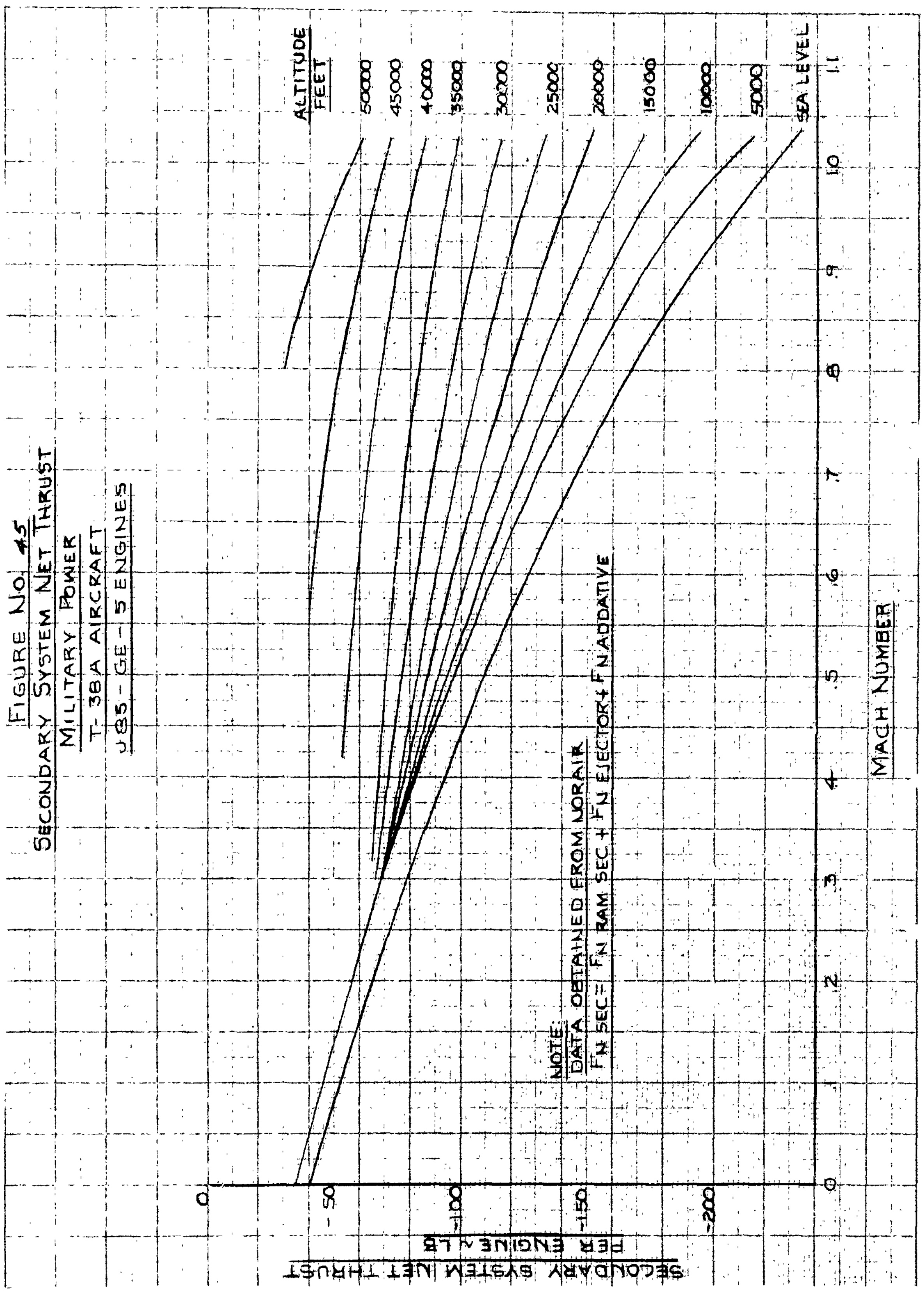


FIGURE NO. 45
SECONDARY SYSTEM NET THRUST
MILITARY POWER
T-38A AIRCRAFT
J85-GE-5 ENGINES
ALTITUDE FEET
50000
45000
40000
35000
30000
25000
20000
15000
10000
5000
SEA LEVEL
NOTE:
DATA OBTAINED FROM NORAIR
$F_N$ SEC = $F_N$ RAM SEC + $F_N$ EJECTOR + $F_N$ ADDATIVE
SECONDARY SYSTEM NET THRUST PER ENGINE ~ LB
0
-50
-100
-150
-200
MACH NUMBER
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1

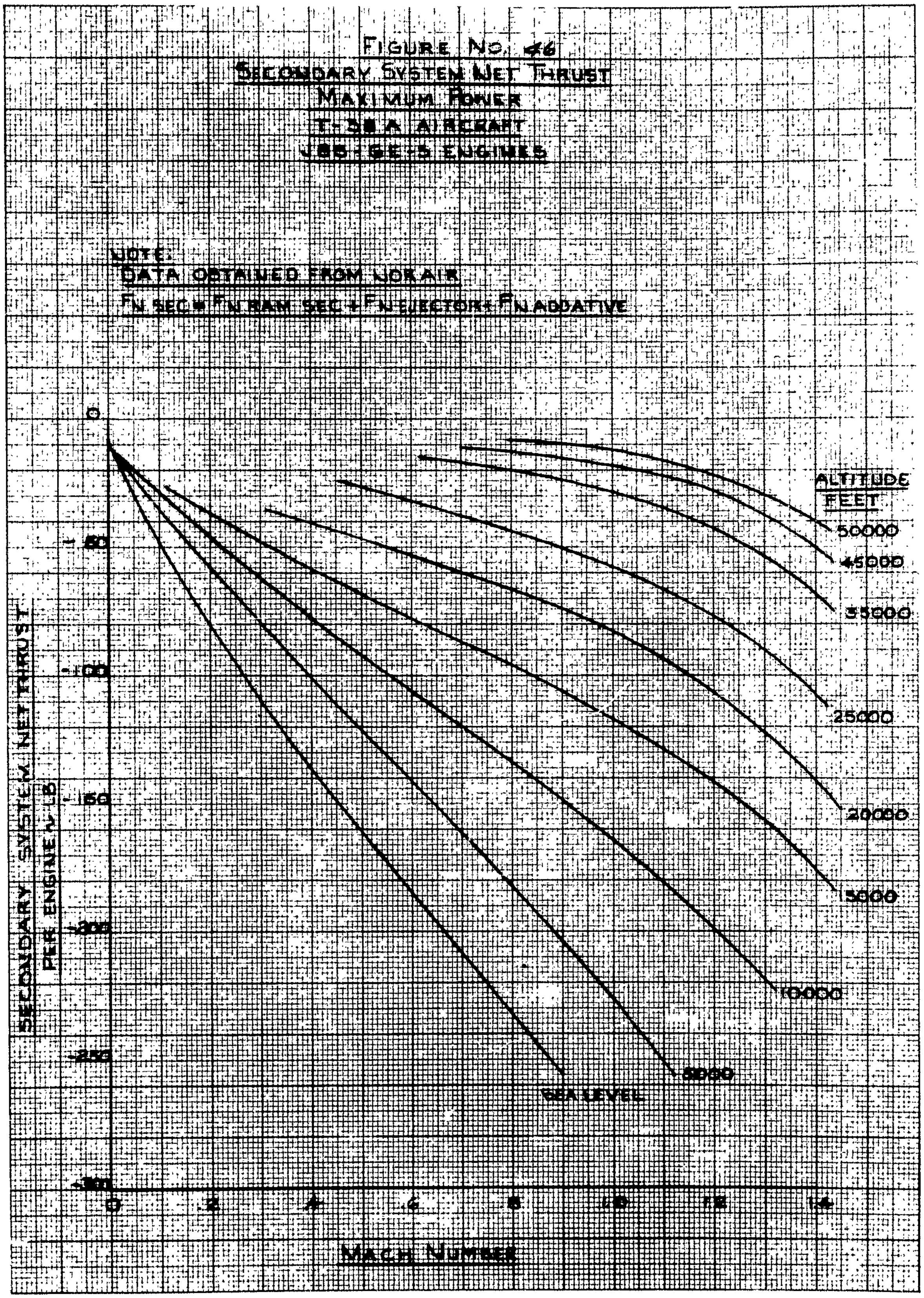


FIGURE NO. 46
SECONDARY SYSTEM NET THRUST
MAXIMUM POWER
T-38A AIRCRAFT
J85-GE-5 ENGINES
NOTE:
DATA OBTAINED FROM NORAIR
FN SEC = FN RAM SEC + FN EJECTOR + FN ADDATIVE
ALTITUDE FEET
50000
45000
35000
25000
20000
15000
10000
5000
SEA LEVEL
SECONDARY SYSTEM NET THRUST PER ENGINE ~ LB
0
-50
-100
-150
-200
-250
-300
0
.2
.4
.6
.8
1.0
1.2
1.4
MACH NUMBER

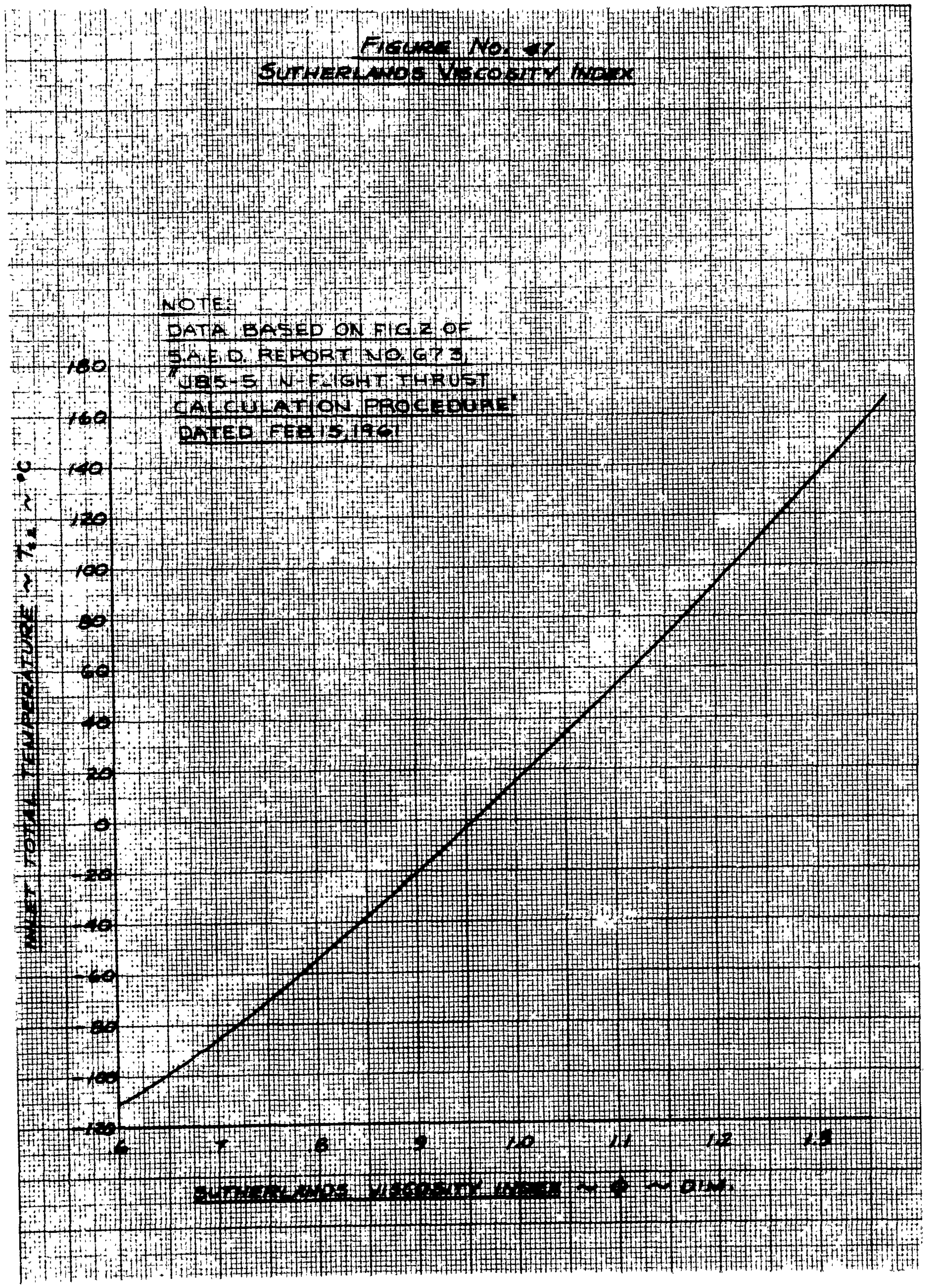


FIGURE NO. 47
SUTHERLANDS VISCOSITY INDEX
NOTE:
DATA BASED ON FIG.2 OF
S.A.E.D. REPORT NO. G73,
"JB5-5 IN-FLIGHT THRUST
CALCULATION PROCEDURE"
DATED FEB 15, 1961
INLET TOTAL TEMPERATURE ~ $T_{t2}$ ~ °C
180
160
140
120
100
80
60
40
20
0
-20
-40
-60
-80
-100
-120
.6
.7
.8
.9
1.0
1.1
1.2
1.3
SUTHERLANDS VISCOSITY INDEX ~ Φ ~ DIM.

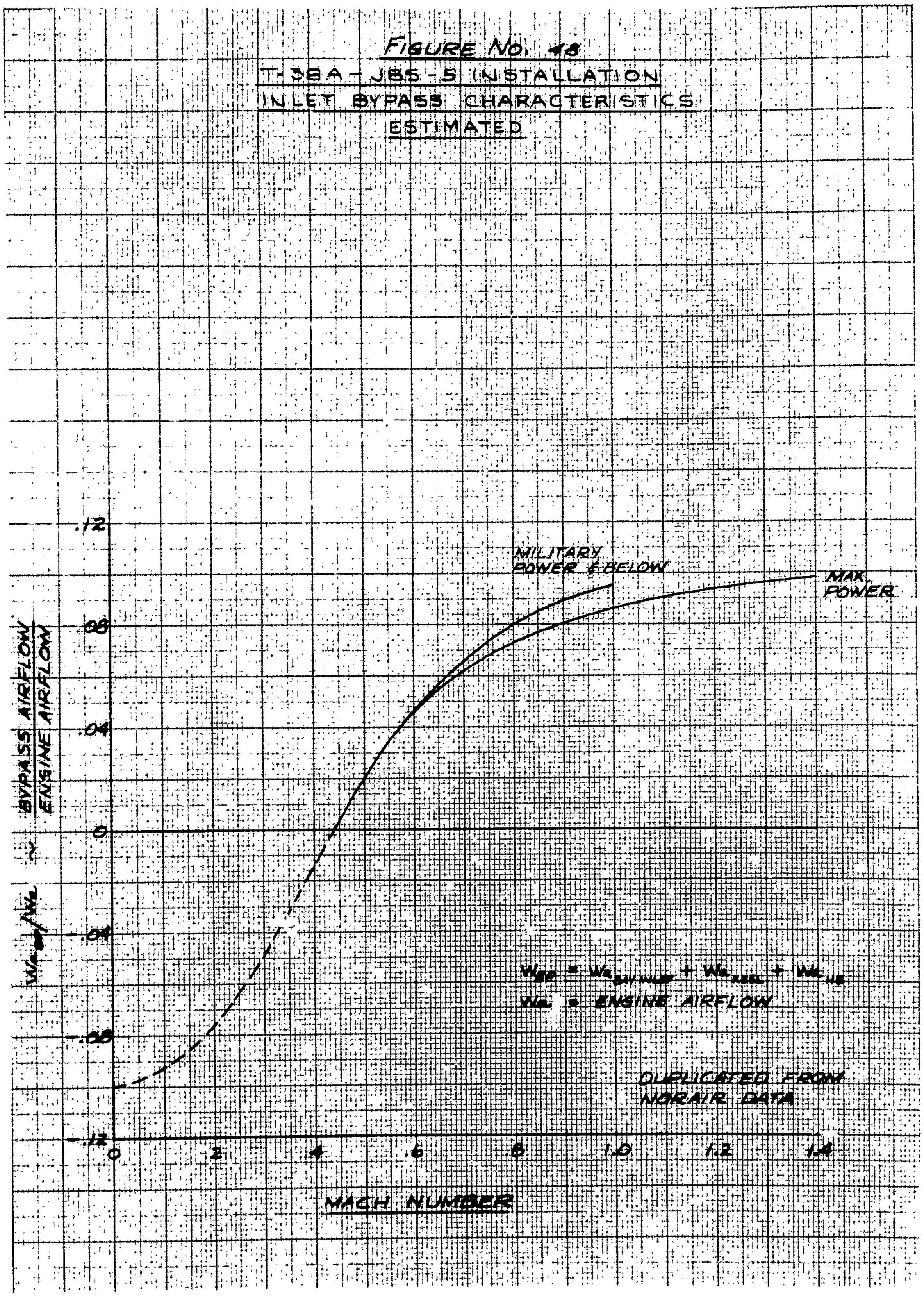


FIGURE No. 48
T-38A - J85-5 INSTALLATION
INLET BYPASS CHARACTERISTICS
ESTIMATED
MILITARY POWER & BELOW
MAX. POWER
.12
.08
.04
0
-.04
-.08
-.12
0
.2
.4
.6
.8
1.0
1.2
1.4
BYPASS AIRFLOW / ENGINE AIRFLOW
MACH NUMBER
DUPLICATED FROM NORAIR DATA

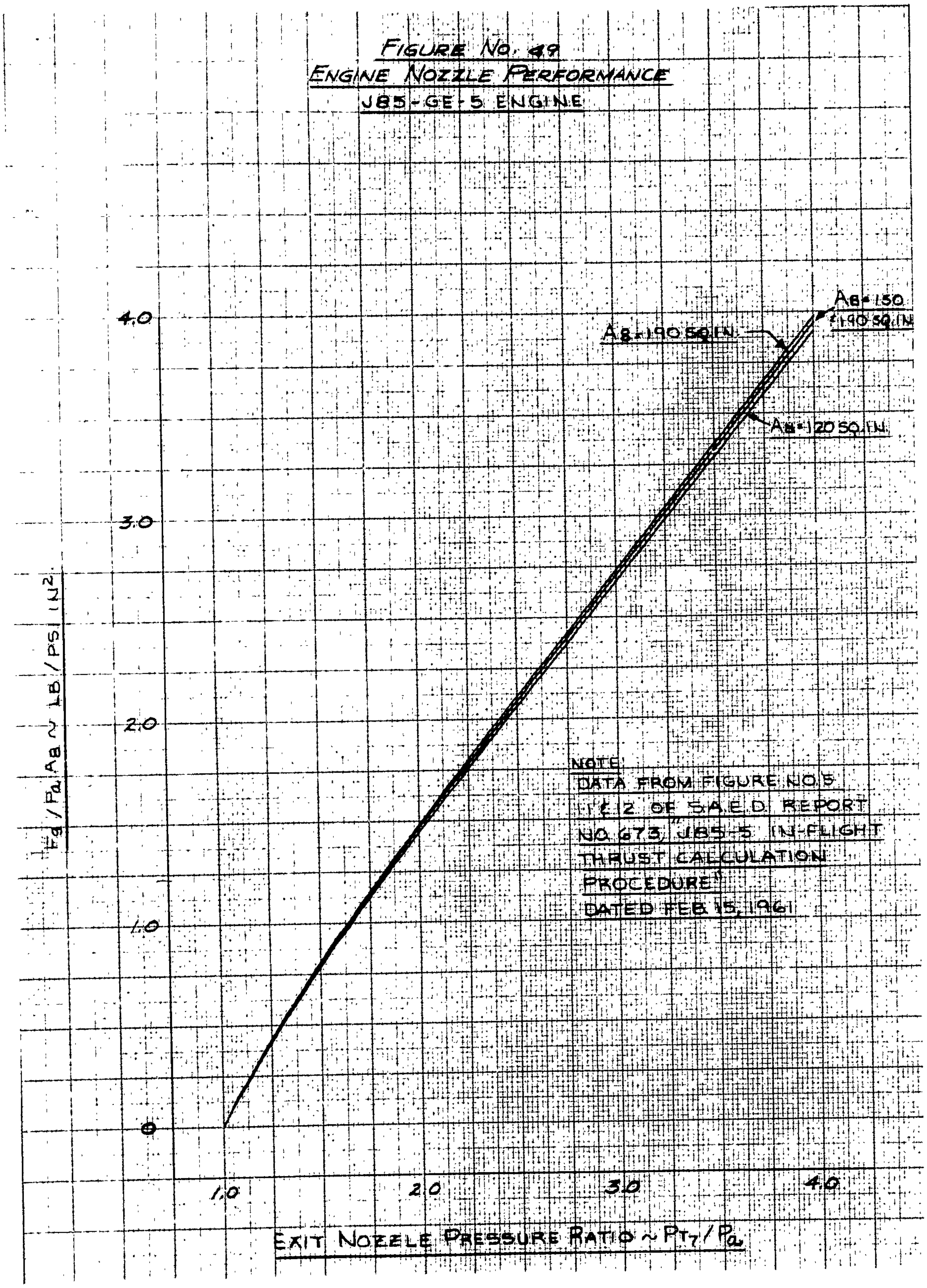
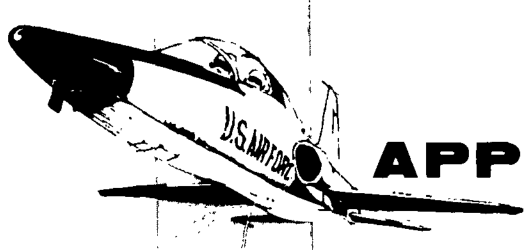


FIGURE NO. 49
ENGINE NOZZLE PERFORMANCE
J85-GE-5 ENGINE
A8 = 150
= 190 SQ.IN
A8 = 190 SQ.IN.
A8 = 120 SQ.IN.
NOTE
DATA FROM FIGURE NO. 5
1 & 2 OF S.A.E.D. REPORT
NO. 673 "J85-5 IN-FLIGHT
THRUST CALCULATION
PROCEDURE"
DATED FEB 15, 1961
4.0
3.0
2.0
1.0
0
1.0
2.0
3.0
4.0
Fg / Pa A8 ~ LB / PSI IN²
EXIT NOZZLE PRESSURE RATIO ~ PT7 / Pa




# APPENDIX II....

## GENERAL AIRCRAFT INFORMATION

● DIMENSION AND DESIGN DATA

| | |
|---|---|
| Wing: | |
| Area, total | 170.00 sq ft |
| Taper ratio | .20 |
| Aspect ratio | 3.75 |
| Span/thickness | 51.1 |
| Airfoil | NACA 65A 004.8 modified (.65) 50 camber |
| Span | 25.25 ft |
| Horizontal Tail: | |
| Area, total | 59.00 sq ft |
| Area, exposed | 33.34 sq ft |
| Taper ratio, exposed | .33 |
| Aspect ratio, exposed | 2.82 |
| Span/thickness, exposed | 58.6 |
| Airfoil | NACA 65A 004 |
| Vertical Tail: | |
| Area, total | 41.42 sq ft |
| Area, exposed | 41.07 sq ft |
| Taper ratio, exposed | .25 |
| Aspect ratio, exposed | 1.21 |
| Span/thickness | 42.2 |
| Airfoil | NACA 65A 004 modified |

## ● INSTRUMENTATION

The test instrumentation was installed by the contractor and maintained by the AFFTC Instrumentation Branch.

The following parameters were recorded on a photo panel at a rate of one frame per second.

1. Time correlation
2. Time of day
3. Altitude
4. Airspeed
5. Outside air temperature
6. Fuel used totalizer, left engine plus left afterburner
7. Fuel used totalizer, right engine plus right afterburner
8. Fuel used totalizer, left and right afterburner pilot line
9. Left and right engine fuel flow
10. Left and right afterburner fuel flow
11. Left and right compressor discharge pressure
12. Left and right turbine discharge pressure
13. Left and right engine fuel temperature
14. Left and right afterburner fuel temperature
15. Left and right turbine discharge temperature
16. Left and right engine rpm.
17. Photo panel environment temperature
18. Lift-off and pilot event light

The following parameters were recorded on a 50 channel Consolidated Engineering Corporation oscillograph recorder:

1. Pilot lift-off and event correlation
2. Left and right engine inlet guide vane position
3. Left and right wheel brake hydraulic pressure
4. Left and right throttle position
5. Left and right engine nozzle position
6. Longitudinal acceleration at the A/C cg
7. Normal acceleration at the A/C cg
8. Lateral acceleration at the A/C cg
9. Left hand duct static pressure ($P_{s_2}$)
10. Left hand duct total pressure ($P_{t_2}$)
11. Angle of attack
12. Angle of sideslip
13. Left and right engine fuel flow
14. Left and right afterburner fuel flow
15. Speed brake position
16. Flap position
17. Swinging rake position
18. Swinging rake total pressure
19. Swinging rake static pressure
20. Ejector static pressure

continued overleaf ....

Test instrumentation available to the pilot included the following:

1. Altimeter
2. Airspeed
3. Left fuel totalizer, engine plus after burner
4. Right fuel totalizer, engine plus after-burner
5. Time correlation
6. Photo panel and oscillograph controls