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T-38A CATEGORY II PERFORMANCE TEST

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TECHNICAL DOCUMENTARY REPORT NO. 63-27
NOVEMBER 1963

AFFTC PROJECT DIRECTIVE 59-18

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

<table>
<thead>
<tr>
<th>Power</th>
<th>Schedule</th>
<th>Weight at Which Service Ceiling is Attained—Lbs</th>
<th>Service Ceiling Service Ceiling</th>
<th>Time to 40,000 Ft</th>
<th>Fuel Used to 40,000 Ft</th>
<th>Distance Traveled to 40,000 Ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max</td>
<td>Norair</td>
<td>10,900</td>
<td>51,500</td>
<td>2.5</td>
<td>500</td>
<td>23</td>
</tr>
<tr>
<td>Max</td>
<td>ATC</td>
<td>10,900</td>
<td>51,500</td>
<td>2.9</td>
<td>612</td>
<td>26</td>
</tr>
<tr>
<td>Mil</td>
<td>Norair</td>
<td>11,300</td>
<td>43,500</td>
<td>10.5</td>
<td>545</td>
<td>87</td>
</tr>
</tbody>
</table>

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

<table>
<thead>
<tr>
<th>Power</th>
<th>Sea Level Ceiling-Ft</th>
<th>Sea Level Ceiling-Min</th>
<th>Sea Level to Service Ceiling-Lb</th>
<th>Sea Level to Service Ceiling - NAM</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max</td>
<td>39,750</td>
<td>34</td>
<td>2420</td>
<td>300</td>
</tr>
<tr>
<td>Mil</td>
<td>18,500</td>
<td>30</td>
<td>900</td>
<td>145</td>
</tr>
</tbody>
</table>

**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 and .88 Mach number.

**COMPUTED FERRY RANGE MISSION**

<table>
<thead>
<tr>
<th>Gross Weight at Engine Start</th>
<th>11,755 lbs</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Weight of Fuel</td>
<td>6.5 lbs/gal</td>
</tr>
<tr>
<td>Total Usable Fuel</td>
<td>583 gal (3790 lbs)</td>
</tr>
<tr>
<td>Cruise Speed</td>
<td>.88 Mach number</td>
</tr>
<tr>
<td>Cruise $W/\text{6}_a$</td>
<td>58,000 lb</td>
</tr>
<tr>
<td>Fuel Reserve (5 percent of initial fuel plus fuel for 20 minutes loiter)</td>
<td>805 lbs</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Fuel Used Lbs</th>
<th>Distance Traveled NAM</th>
<th>Elapsed Time From Brake Release Min</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff Allowance (fuel for 5 minutes at NRP at sea level)</td>
<td>350</td>
<td>-</td>
</tr>
<tr>
<td>Military Power Climb to 39,700 Feet</td>
<td>490</td>
<td>79</td>
</tr>
<tr>
<td>Cruise</td>
<td>2145</td>
<td>875</td>
</tr>
<tr>
<td>Totals</td>
<td>2985</td>
<td>954</td>
</tr>
</tbody>
</table>

$^2$Weight-pressure ratio ($W/\text{6}_a$) is defined as $\frac{\text{Aircraft weight x 29.92126}}{\text{Ambient pressure}}$. 
ACTUAL FERRY RANGE MISSION

<table>
<thead>
<tr>
<th>Gross Weight at Engine Start</th>
<th>12,330 Lbs</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specific Weight of Fuel</td>
<td>6.26 Lbs/Cal</td>
</tr>
<tr>
<td>Total Usable Fuel</td>
<td>583 Cals (3650 Lbs)</td>
</tr>
<tr>
<td>Cruise Speed</td>
<td>1.89 Mach number</td>
</tr>
<tr>
<td>Fuel Reserve</td>
<td>938 Lbs</td>
</tr>
<tr>
<td>Cruise W/8a</td>
<td>56,000 Lbs</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Start, Taxi, Takeoff, and Acceleration to Climb Schedule</th>
<th>Fuel Used</th>
<th>Distance Traveled</th>
<th>Elapsed Time From-Brake Release-Min</th>
</tr>
</thead>
<tbody>
<tr>
<td>Military Power Climb to 38,500 Feet</td>
<td>482</td>
<td>68</td>
<td>10.45</td>
</tr>
<tr>
<td>Cruise</td>
<td>1803</td>
<td>679</td>
<td>81.19</td>
</tr>
<tr>
<td>Totals (for 938 lbs fuel remaining)</td>
<td>2712</td>
<td>747</td>
<td>91.64</td>
</tr>
</tbody>
</table>

Totals (corrected to 3790 lbs of fuel available, a takeoff 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 1.
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:

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.**

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


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:

<table>
<thead>
<tr>
<th>Test</th>
<th>Guarantee</th>
<th>Results</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>Maximum level flight Mach number at 36,089 feet and at the design gross weight of 9,592 pounds.</td>
<td>1.2</td>
</tr>
<tr>
<td>2.</td>
<td>Cruising Mach number for long range cruise, not less than</td>
<td>0.85</td>
</tr>
<tr>
<td>3.</td>
<td>Range at cruise Mach number as specified in item (b), using military thrust takeoff and climb, 20 minute sea level loiter and 9% fuel reserve</td>
<td>880 NAM</td>
</tr>
<tr>
<td>4.</td>
<td>Maximum thrust takeoff ground run at sea level at gross weight required to perform item(c).</td>
<td>2,500 Ft</td>
</tr>
</tbody>
</table>

*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.
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/5a of 58,000 pounds and a Mach number of 0.883. 

\(^3\) Computed in accordance with landing reserves of MIL-C-5011A from level flight speedpower data.
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

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 1. 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 knot indicated airspeed. Test thrust/δₐ was then plotted against ambient temperature to determine thrust variation with ambient temperature. Standard thrust was then defined as that thrust corresponding to standard temperature at sea level (15 degree C). However, it should be noted that, thrust values indicated on Figure 1 were 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 \( \sqrt{\frac{V_{\text{ts}}}{V_{\text{at}}}} \).

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 obtained from the following expression:

\[
\frac{\Delta V}{\Delta t} = \frac{25.33 T_{\text{as}}}{P_{\text{as}} M_b^2 s} \left[ \frac{n(Lt \frac{V_{\text{as}}^2 - W_s^2}}{\frac{\partial V}{\partial t}} \right]
\]

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

\[
T_{\text{ex}} = \frac{R/C_t W_t}{101.333 V_{\text{as}}^2}
\]

The induced drag was defined as follows:

\[
D_t = \frac{(C_{l\text{at}} - \Delta C_{l})^2 \times M^2 S_{\text{at}}}{\pi A R \ e \ 0.00675}
\]

\( \Delta C_{l} \) is defined in Figure 5, of Reference 10.

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

\[
\Delta R/C = \frac{101.333 \sqrt{V_{\text{as}}^2}}{W_s^2} \times \Delta D_t - s
\]

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 in-
crement, as were all thrust and drag increments, was converted to rate of climb as follows:

\[ \frac{\Delta F_{\text{power}}}{R/C_{\text{power}}} = \frac{\Delta F_{\text{power}}}{R/C_{\text{power}}} = \frac{\Delta F_{\text{power}}}{R/C_{\text{power}}} = \frac{\Delta F_{\text{power}}}{R/C_{\text{power}}} \]

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:

\[ T_{ex} = \frac{R/C_{s} W_{s}}{101.333 V_{ts}} \]

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 estimated 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 ($P_{t2}$)

Average compressor inlet total pressure data was obtained from a pressure survey rake located immediately ahead of the compressor on the left engine only. $P_{t2}$ for the right engine was assumed to be the same as the left when engine speeds (airflow) were the same. However, when the right engine was operating at a different power setting than the left, the right $P_{t2}/P_{t0}$ was modified by the following procedure:

$$W_a \frac{t_{t2}^{0}}{K_RN_1} \frac{t_{t2}}{t_{t2}}$$

for both right and left engines.

$$P_{t2}/P_{t0} = f \left(W_a \frac{t_{t2}^{0}}{K_RN_1} \frac{t_{t2}}{t_{t2}} \right)$$

(Figure numbers 42 and 43)

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

$$(P_{t2}/P_{t0})_{left} \text{ and } (P_{t2}/P_{t0})_{right}$$

$$\frac{\Delta P_{t2}}{P_{t0}L-R} = (P_{t2}/P_{t0})L - (P_{t2}/P_{t0})R$$

$$\Delta P_{t2L-R} = \frac{\Delta P_{t2}}{P_{t0}L-R} \times P_{t0}$$

Then $P_{t3}$ for the right engine, based on $P_{t2}$ measured from the inlet rake, is as follows:

$$P_{t3} = P_{t2} + (\Delta P_{t2})L-R$$

In some cases, when the $P_{t2}$ rake was inoperative, $P_{t2}/P_{t0}$ was obtained from Figures 42 and 43 as a function of corrected inlet airflow.

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:

\[ F_e = \frac{W_a \times V_{t2} \times 1.6889}{g} \]

Primary net thrust was obtained by:

\[ F_{np} = F_g - F_e \]

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

Total net propulsive thrust was then defined as:

\[ F_{test} = F_{np} + F_{ns} \]

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 \( T_{t2} \) with test and standard \( T_{t2} \) and extract corresponding values of \( F_n \).

2. \( \Delta F_{ns-t} = F_{nstd} - F_{ntest} \) (from curves)

3. Standard \( F_{ns} = F_{ntest} + \Delta F_{ns-t} \)

The curves of \( F_n \) versus \( T_{t2} \) 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 \( (T_{lc}/T_a) - 1 \) versus \( M^2/5 \) (where \( T_{lc} \) is the indicated temperature obtained from the test aircraft and \( T_a \) is ambient temperature from rawinsonde). Test ambient temperatures were then extracted from \( T_{lc} \) by the following equation:

\[ T_a = T_{lc}(1 + 0.98 \frac{M^2}{5}) \]

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

A₈  exhaust nozzle area                        sq in
AR  aspect ratio                              ft
b   wing span                                 ft
CAS calibrated airspeed                       knots
CD  coefficient of drag                        
CL  coefficient of lift                        
Dᵢ  induced drag                              lbs
E  airplane efficiency factor                 
EGT exhaust gas temperature                  °C or °K
F₉  gross thrust                              lb
Fₙ  net thrust                                lb
Fₑ  inlet ram drag                            lb
Fₑreq net thrust required for level flight    lb
G  acceleration due to gravity                ft/sec/sec
Mᵢ  indicated Mach number                    
M  Mach number, free stream                  
 thì  normal load factor                       g
N  engine speed                               rpm
NAMPP nautical air miles travelled per pound of fuel used ml/lb
SFC specific fuel consumption                pounds of fuel/hr
P₄  nautical air miles travelled              nautical miles
Pt₅ compressor exit total pressure            in Hg
Pt₂ compressor inlet total pressure          in Hg
Pt₀ free stream total pressure               in Hg
Pₐ  ambient pressure                          in Hg
ΔP/dᶜⁱᶜ static pressure coefficient for compressible fluid based on q from instrument corrected values only
R/C rate of climb                             ft/min
\begin{itemize}
\item[$S$] wing area \quad \text{sq ft}
\item[$T_{1c}$] indicated total temperature \quad \degree C or \degree K
\item[$T_a$] ambient temperature \quad \degree C or \degree K
\item[$T_{t2}$] compressor inlet total temperature \quad \degree C or \degree K
\item[$T_{ex}$] excess thrust \quad \text{lb}
\item[T. O.] take-off
\item[$T_{t5}$] turbine exit total temperature \quad \degree C or \degree K
\item[$V_{c(CAS)}$] calibrated airspeed \quad \text{knots}
\item[$V_i (IAS)$] indicated airspeed \quad \text{knots}
\item[$V_t$] true airspeed \quad \text{knots}
\item[W] aircraft gross weight \quad \text{lb}
\item[$W/\delta_a$] weight-pressure ratio \quad \text{lb}
\item[$W_f$] fuel flow \quad \text{lb/hr}
\item[$W_a$] engine airflow \quad \text{lb/sec}
\item[$\theta_a$] ambient pressure ratio, $P_a/29.92$ \quad \text{deg}
\item[$\delta_{t2}$] inlet pressure ratio, $P_{t2}/29.92$ \quad \text{deg}
\item[$\alpha$] aircraft angle of attack \quad \text{deg}
\item[$W_{aIB}$] compressor bleed airflow \quad \text{lb/sec}
\item[$\Phi$] Sutherland's Viscosity Index
\item[RNI] Reynolds Number Index
\item[K\textsuperscript{RNI}] Reynolds Number correction factor to engine airflow
\item[$W_{aBP}$] engine by-pass airflow \quad \text{lb/sec}
\item[$\theta_{t2}$] $T_{t2}/288$
\end{itemize}

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 52-1194
J85-GE-5 ENGINES
GROSS WEIGHT 18000 LB

○ NO FLAPS (0 DEG.)
△ 45% FLAPS (20 DEG.)
□ 60% FLAPS (27 DEG.)
♦ 100% FLAPS (45 DEG.)

STANDARD DAY STATIC THRUST:
MAXIMUM POWER 5760 LB.
MILITARY POWER 4080 LB.

GROUND EFFECT
AIRSPEED CALIBRATION
TEST NOSE BOOM
(SEE FIGURE 35)

185-173

TAKE-OFF SPEED STANDARD DAY
SEA LEVEL STANDARD DAY

TOTAL DISTANCE ~ FT

LIFT-OFF ~ FT

GROUND ROLL ~ FT

RECOMMENDED LIFT-OFF SPEECS
0% FLAPS 168 KTAS
45% 60% FLAPS 158 KTAS
100% FLAPS 153 KTAS
Figure No. 2

Maximum Power Climb Potential

T38A USAF S/N 58-1194
J85-GE-5 Engines S/N's
230 84 and 230 85

<table>
<thead>
<tr>
<th>ALT-FT</th>
<th>FLT M.S.P</th>
<th>R.U.</th>
<th>ALT-FT</th>
<th>FLT M.S.P</th>
<th>R.U.</th>
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<tr>
<td>5000</td>
<td>3</td>
<td></td>
<td>36000</td>
<td>10</td>
<td>3</td>
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<tr>
<td>15000</td>
<td>27</td>
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<td>40000</td>
<td>15</td>
<td>2</td>
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<tr>
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<td>40000</td>
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<td>8</td>
<td>2</td>
<td>45000</td>
<td>17</td>
<td>2</td>
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<tr>
<td>25000</td>
<td>8</td>
<td>3</td>
<td>45000</td>
<td>17</td>
<td>3</td>
</tr>
</tbody>
</table>

NORAIR RECOMMENDED CLIMB SCHEDULE

5000 FT
11570 LB

15000 FT
11450 LB

25000 FT
11330 LB

35000 FT
11210 LB

45000 FT
11090 LB

50000 FT
10700 LB

Rate of Climb Std. *Ft./Min.*

Mach Number
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
Maximum Power Climb Performance

Gross Weight at Engine Start: 12,330 lb

U.S. Air Recommended Schedule

Figure No. 4 cont.

Net Thrust - lb

Mach Number

Airspeed - knots

Nautical Air Miles Traveled

25
Figure No. 5
Maximum Power Climb Performance
T-39A USAF 515th E-1145
J-85-GE-F Engines 1/45
2501/4 and 2501/8

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.

Flight Run

<table>
<thead>
<tr>
<th>FLI</th>
<th>RUN</th>
</tr>
</thead>
<tbody>
<tr>
<td>24</td>
<td>9</td>
</tr>
<tr>
<td>26</td>
<td>8</td>
</tr>
</tbody>
</table>

Time to Climb

Rate of Climb

Fuel Used

Fuel Flow

Rate of Climb - ft/min

Time to Climb - min.

Fuel Flow - lb/hr (total)

Fuel Used - lb
Figure No. 5 Cont.

Maximum Power Climb Performance
Gross Weight at Engine Start 12500 lbs.
A.T.C. Suggested Schedule

NET THRUST - lbs
MACH NO.

Vc - ...

Vt - ...

N/AFT

2000 6000 10000
Net Thrust - lbs

7 8 9 10
Mach Number

0 40 80 120
Nautical Airmiles Traveled

ALTITUDE - FEET

0 2000 4000 6000 8000 10000

27
FIGURE No. 6
MILITARY POWER CLimb PERFORMANCE
T-38A USAF S/N 58-1194
J-85-GE-5 ENGINES S/N's
230484 AND 238185

NOTE:
1. 250 LBS. OF FUEL
   ARE REQUIRED FOR
   TAXI, TAKE-OFF AND
   ACCELERATE TO CLimb SPEED.
2. 2.5 MINUTES ARE
   REQUIRED TO TAKE-OFF
   AND ACCELERATE
   TO CLimb SPEED.

FLIGHT RUN
18  4  
32  3  
34  3  
49  3  

RATE OF CLIMB ~ FT/Min

FUEL FLOW ~ LB/Hr (TOTAL)

TIME TO CLIMB ~ MIN

FUEL USED ~ LB.
Figure No. 6 cont.

MILITARY POWER CLIMB PERFORMANCE

GROSS WEIGHT AT ENGINE START 12350 LBS.

LORAIR RECOMMENDED SCHEDULE

NET THRUST

MACH NO.

N.A.M.T.

1000 3000 6000 NET THRUST = LB.

1.0 2.0 3.0 4.0 MACH NUMBER

200 300 400 500 600 AIRSPEED = KNOTS

0 40 80 120 160 200 240 280 NAUTICAL AIR MILES TRAVELED

ALTITUDE = FEET
Figure No. 7
SINGLE ENGINE MAXIMUM POWER CLimb
J-38A USAF SIN 58-1144
J-85 GE-4 ENGINE SIN 230145
FIGURE NO. 7 CONT.
SINGLE ENGINE MAXIMUM POWER CLIMB
GROSS WEIGHT AT ENGINE START 12300 LBS

FLIGHT RUN
- 43
- 42
- 3
- 3

NET THRUST

MACH NUMBER

ALTITUDE - FT

AVSPEED-KTS

0 1000 2000 3000 4000

NET THRUST - LB

0 .5 .6 .7 .8 .9

AIRCRAFT TRAVELLED

0 100 200 300 400

NAUTICAL AIR MILES TRAVELED
FIGURE NO. 12
LEVEL FLIGHT PERFORMANCE
SPEED BRAKE EXTENDED
T-33A USAF S/N 52-1194
J85 GE-5 ENGINES S/N 5
230185 AND 230186
WEIGHT 10000 LB
ALTITUDE 25000 FT

FLIGHT 54, RUN 1
FIGURE NO 14

LEVEL FLIGHT PERFORMANCE

SINGLE ENGINE

F-5A, USAF, S/N 58-1134
J57-GE-5 ENGINES S/N 9's
EQUIPS AND PROB
WEIGHT 10,000 LB

FIT RAR ALTITUDE W/S
A 43 1 20,000 21,740
B 46 1 10,000 16,720
C 65 1 15,000 17,280

MACH NUMBER

NET IMPULSE REQUIRED

1000

2000

3000

4000

5000

6000

7000

MACH NUMBER

20,000 FT

18,000 FT

16,000 FT

12,000 FT

10,000 FT
FIGURE NO. 15
LEVEL RIGHT DRAG POLAR
T-38A USAF S/N 58-1134
T35-GE-5 ENGINE S/N 230185

FIT
RUN

□ 43 1
△ 46 1
▲ 45 1
FIGURE NO 17
LEVEL FLIGHT DURG POLAR
LANDING GEAR EXTENDED
T-38A USAF S/N 58-W-1194
185-GE-5 ENGINES S/N'S 238/85 AND 238/86
7000 FT., W/G = 18760 LB.

<table>
<thead>
<tr>
<th>SYM</th>
<th>FLT</th>
<th>RUN</th>
<th>FLAPS</th>
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<td>49</td>
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<td>2</td>
<td>UP</td>
</tr>
<tr>
<td></td>
<td>48</td>
<td>2</td>
<td>UP</td>
</tr>
<tr>
<td></td>
<td>48</td>
<td>2</td>
<td>45%</td>
</tr>
<tr>
<td></td>
<td>48</td>
<td>2</td>
<td>100%</td>
</tr>
<tr>
<td></td>
<td>39</td>
<td>2</td>
<td>100%</td>
</tr>
</tbody>
</table>

FLAPS UP

GERAR UP

45% FLAPS

100% FLAPS

LIFT COEFFICIENT SQUARED - C^2

DRAG COEFFICIENT - C^
FIGURE NO. 19

PLANT AVAILABLE AND REQUIRED
5-6A VARIOUS SYSTEMS
PLANT-CYCLE UNIT
COST AND ECONOMY
REFERENCES
WEIGHT ISOPROGRESS

1. MAXIMUM POWER
   WEIGHT TO PUMP 2 AND 8

2. MILITARY POWER
   WEIGHT TO PUMP 2 AND 8

DATA OBTAINED FROM
LEVEL ACCELERATIONS

THRUST REQUIRED CURVE
WITH VARIOUS FLIGHTS

FIGURE NO. 19

THRUST AVAILABLE
MAXIMUM POWER

THRUST AVAILABLE
HYDROGEN POWER

THRUST REQUIRED
FIGURE NO 20
THrust AVAILABLE AND REQUIRED
T-33A USAF S/N ET-1794
TRE-10-F-1 ENGINE S/N 16
250185 AND 290186
ALTITUDE 18500 FT
WEIGHT 16000 LB

- MAXIMUM POWER
  FLIGHT 27 RUN 4

- MILITARY POWER
  FLIGHT 23 RUN 3 AND 4

DATA OBTAINED FROM
LEVEL ACCELERATIONS

THRUST AVAILABLE
MILITARY POWER

THRUST AVAILABLE
MAXIMUM POWER

THRUST REQUIRED

MACH NUMBER
FIGURE 10 EL
FAVOR AVAILABILITY AND INFLUENCE
CET TRIAL S/N 51-614357
12-6-40 INETERS 576
PERTIES AND PROB.
ALTITUDE, DEGREES
WEIGHT, 10,000 LBS.

b. MAXIMUM POWER
FLIGHT 8, RUNS 2 AND 4
FLIGHT 19, RUNS 2 AND 4

C. MILITARY POWER
FLIGHT 28, RUNS 2 AND 4

DATA OBTAINED FROM
LEVEL ACCELERATIONS

MINIMUM POWER
THROTTLE AVAILABLE

MILITARY POWER
THROTTLE AVAILABLE

THROTTLE REQUIRED

MACH NUMBER
Figure 12.6
Thrust Available and Required

- 7.5:4b USAA SW-US
- USS-5 SW-US
- USS-5 SW-US

Flight and Events
- Altitude: 5000 ft
- Weight: 10000 lb

Maximum Power
- Flight 6, Runs 1 and 2

Military Power
- Flight 14, Runs 1 and 2
- Flight 24, Runs 3 and 4

Data Obtained From
- Level Accelerations

Net Thrust - lb

Mach Number

Thrust Available
- Medium Power

Thrust Required
<table>
<thead>
<tr>
<th>STATEMENT</th>
<th>PAGE 3473</th>
</tr>
</thead>
</table>

### ENGAGE NO. 15101

### LOAD POWER DEMAND

#### POWER LIMITS

<table>
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<tr>
<th>STATE</th>
<th>FIT. BOLN</th>
<th>TOTAL WT</th>
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<tbody>
<tr>
<td>A</td>
<td>40-5</td>
<td>3400</td>
</tr>
<tr>
<td>B</td>
<td>17-3</td>
<td>6400</td>
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</table>

---

### Diagram Description:

- **Axes:**
  - x-axis: Load Power Demand
  - y-axis: Engage No.

- **Legend:**
  - Critical Value
  - Stable Tolerable
  - Critical Limit

---

59
Figure No. 30
Half Power Descent

图表 No.
燃油使用情况

飞机发动机

速度刹车

延伸

速度刹车

拉起

高度（英尺）

0 10000 20000 30000

速率（下降）英尺/分钟

0 2 4 6 8 10

时间下降（分钟）

0 20 40 60 80

燃油流量（磅/小时）

0 500 1000 1500

燃油使用量（磅）

60
**FIGURE NO 34**

**AIRSPEED SYSTEM CALIBRATION**

T-38A USAF S/N 58-1194

<table>
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<tr>
<th>REF NO</th>
<th>METHOD</th>
<th>DATE</th>
<th>CONFIGURATION</th>
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</thead>
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<tr>
<td>9</td>
<td>TOWER FLY-BY</td>
<td>20 JUNE 1961</td>
<td>CLEAN</td>
</tr>
<tr>
<td>10</td>
<td>TOWER FLY-BY</td>
<td>21 JUNE 1961</td>
<td>CLEAN</td>
</tr>
<tr>
<td>△▽</td>
<td>SMOKE TRAIL ACCEL</td>
<td>27 JUNE 1961</td>
<td>CLEAN</td>
</tr>
<tr>
<td>○</td>
<td>STABILIZED FLAP</td>
<td>8 SEPT 1961</td>
<td>GEAR DOWN, 60% FLAPS</td>
</tr>
</tbody>
</table>

**Diagram:**
- Indicates Mach Number vs. Indicated Mach Number.
- Grid lines for data points.
- Data points marked with symbols for different conditions.

**Indicated Mach Number**

**Mach Number**
Figure No. 38

Static Thrust Calibration

T-38A USAF SN 58-1176
J85-GE-6 Engine SN 33685

Regime and Date

Aftburning

No fairing is presented for these parameters, because of the inability to define standard dry nozzle mass flow.

Engine Press Ratio - Pn/Pa

16 17 18 19 20 21 22
Figure No. 41
J-85-5 AIRFLOW

Note:
Curve obtained from Figure 1
of S.A.E. Report No. 673,
J85-5 In-Flight Thrust Calculation
Procedure dated Feb. 15, 1961

Engine Airflow - CFM - Lb/sec

Corrected Rotor Speed - N/1000 × % RPM
FIGURE NO. 45
SECONDARY SYSTEM NET THRUST
MILITARY POWER
T-38A AIRCRAFT
J85-GE-5 ENGINES

NOTE:
DATA OBTAINED FROM NORDAIR
$F_n \text{ sec} = F_n \text{ ram sec} + F_n \text{ ejection} + F_n \text{ additive}$

SECONDARY SYSTEM NET THRUST
ENGINE: LBS

ALTITUDE FEET
60000
55000
50000
45000
40000
35000
30000
25000
20000
15000
10000
5000
SEA LEVEL

SECONDS
0 1 2 3 4 5 6 7 8 9 10 11

MACH NUMBER
FIGURE NO. 46
SECONDARY SYSTEM NET THRUST
MAXIMUM POWER
T.SEA AIRCRAFT
J.E.S.E.S. ENGINES

NOTE
DATA OBTAINED FROM UNRAIR
FUS. S.E.W. NAM. SEC. 5. INJECTION FIN ADJUTIVE

ALTITUDE
FEET
50000
45000
40000
35000
30000
25000
20000
15000
10000
5000
SEA LEVEL

SECONDARY SYSTEM NET THRUST
PER ENGINE, lb
0 5 10 15 20 25 30 35 40 45 50

MACH NUMBER
Figure No. 47
SUTHERLAND VELOCITY INDEX

NOTE:
DATA BASED ON FIG. 3 OF S.A.I.D. REPORT NO. 573,
JBS-5 N-FLIGHT THRUST
CALCULATION PROCEDURE
DATED FEB. 5, 1951

[Graph showing data points and trend line]

SUTHERLAND VELOCITY INDEX = C x "C"
Figure No. 49
Engine Nozzle Performance
J85-GE-5 Engine

Exit Nozzle Pressure Ratio $\frac{P_t}{P_0}$

NOTE:
DATA FROM FIGURE NO. 5
FL 2 OF S.A.E.D REPORT
NO. 673, J85-5 IN-FLIGHT
THrust Calculation
PROCEDURE:
DATED FEB 15, 196
### GENERAL AIRCRAFT INFORMATION

#### DIMENSION AND DESIGN DATA

**Wing:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Area, total</td>
<td>170.00 sq ft</td>
</tr>
<tr>
<td>Taper ratio</td>
<td>0.20</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>3.75</td>
</tr>
<tr>
<td>Span/thickness</td>
<td>51.1</td>
</tr>
<tr>
<td>Airfoil</td>
<td>NACA 65A 004.8 modified (.65) 50 camber</td>
</tr>
<tr>
<td>Span</td>
<td>25.25 ft</td>
</tr>
</tbody>
</table>

**Horizontal Tail:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Area, total</td>
<td>59.00 sq ft</td>
</tr>
<tr>
<td>Area, exposed</td>
<td>33.34 sq ft</td>
</tr>
<tr>
<td>Taper ratio, exposed</td>
<td>0.33</td>
</tr>
<tr>
<td>Aspect ratio, exposed</td>
<td>2.82</td>
</tr>
<tr>
<td>Span/thickness, exposed</td>
<td>58.5</td>
</tr>
<tr>
<td>Airfoil</td>
<td>NACA 65A 004</td>
</tr>
</tbody>
</table>

**Vertical Tail:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Area, total</td>
<td>41.42 sq ft</td>
</tr>
<tr>
<td>Area, exposed</td>
<td>41.07 sq ft</td>
</tr>
<tr>
<td>Taper ratio, exposed</td>
<td>0.25</td>
</tr>
<tr>
<td>Aspect ratio, exposed</td>
<td>1.21</td>
</tr>
<tr>
<td>Span/thickness</td>
<td>42.2</td>
</tr>
<tr>
<td>Airfoil</td>
<td>NACA 65A 004 modified</td>
</tr>
</tbody>
</table>
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 ($P_{S2}$)
10. Left hand duct total pressure ($P_{T2}$)
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
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