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RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE OF J35-A-17 TURBOJET ENGINE IN AN ALTITUDE CHAMBER

By K. R. Vincent and B. M. Gale

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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SUMMARY

An investigation of the altitude performance characteristics of an Allison J35-A-17 turbojet engine has been conducted in an altitude chamber at the NACA Lewis laboratory. Engine performance was obtained over a range of altitudes from 20,000 to 60,000 feet at a flight Mach number of 0.62 and a range of flight Mach numbers from 0.42 to 1.22 at an altitude of 30,000 feet.

The performance of the engine over the range investigated could be generalized up to an altitude of 30,000 feet. Performance of the engine at any flight Mach number in the range investigated can be predicted for those operating conditions at which critical flow exists in the exhaust nozzle with the exception of the variables corrected net thrust, and net-thrust specific fuel consumption.

INTRODUCTION

An investigation was conducted in an NACA Lewis altitude chamber to determine the altitude performance characteristics of an Allison J35-A-17 axial-flow turbojet engine. The engine used for this investigation was a standard production engine. The range of simulated flight conditions extended from flight Mach numbers of 0.42 to 1.22, assuming 100-percent ram recovery, at an altitude of 30,000 feet, and from altitudes of 20,000 to 60,000 feet at a flight Mach number of 0.62.

Altitude-performance data for the engine are presented in both graphical and tabular form. In addition, the performance data have been generalized to standard sea-level conditions to determine the extent to which altitude performance may be predicted from sea-level operation of the engine.

APPARATUS AND PROCEDURE

Engine

The Allison J35-A-17 axial-flow turbojet engine (fig. 1) has eleven stages of compression giving a compression ratio of approximately 5 to 1 at the rated engine speed of 7800 rpm, eight com- 1420 bustion chambers of the through-flow type, and a single-stage axial-flow turbine. A fixed conical exhaust nozzle having a diameter of 17.69 inches was installed on the engine. Operation of the engine with this nozzle produced a tail-pipe gas temperature of 1265° F (1725° R) at rated engine speed, at static sea level conditions, for the particular tail-pipe configuration used. At these operating conditions the guaranteed rated thrust is 4900 pounds with a specific fuel consumption of 1.08 pounds per hour per pound of thrust and an air flow of 85 pounds per second. The maximum dimensions of the engine are a 40-inch diameter and a 146-inch length. The dry engine weight without starter generator and tachometer generator is 2220 pounds.

For this investigation the standard engine fuel-control system was removed and a J33-type fuel pump, barometric, and throttle were installed. These alterations were made in order that the performance of the engine could be obtained outside the limits imposed by the fuel-control system. An aluminum accessory cover and bell cowl (fig. 1) were installed at the compressor inlet to obtain a smooth air flow into the compressor.

Altitude Chamber

The altitude chamber in which the engine was installed is 10 feet in diameter and 60 feet long (fig. 2). The chamber is divided into an inlet section and an exhaust section separated by a steel bulkhead. The engine was mounted on a thrust measuring bed in the inlet section, and the tail pipe of the engine extended through the bulkhead into the exhaust section. Freedom of movement for the tail pipe was provided in both axial and radial directions by means of a seal composed of three asbestos board rings. A calibration of the thrust indicators was made.

Air is supplied to the inlet section of the chamber through a supply line from the laboratory air system. Combustion air can be obtained from this system over a range of temperatures from -40° to 85° F. Fine control of the inlet-air temperature is obtained by the use of electric heaters installed in a bypass line upstream of the chamber. The inlet and exhaust pressures are controlled by means of remote-controlled valves in the supply lines and the exhaust lines, respectively. A set of straightening vanes is installed in the chamber upstream of the engine to provide a uniform air-flow distribution at the engine inlet.

The hot gases discharged from the exhaust nozzle of the engine are removed from the exhaust section of the altitude chamber through a diffusing elbow and a dry-type primary cooler. A dry-type secondary cooler downstream of the exhaust valves further cools the hot gases before passing them into the laboratory exhaust system.

Instrumentation

The locations of the instrumentation stations for the engine are shown in figure 3. The detailed location of the separate temperature and pressure probes within a given station is shown in figure 4 for those stations at which data are presented herein. Pressure and temperature probes at each station were so located that a mean value of temperature or pressure could be obtained by mechanical, electrical, or arthmetical averaging of the individual readings.

Engine-inlet pressure and temperature were set, for a given run, using the readings of the instrumentation at station 1. The instrumentation at station 2 was used in calculating the altitude correction factors θ and δ and the compressor pressure ratio. Air flow and jet thrust were computed using the arithmetical average of the total temperatures and pressures, and the mechanical average of the four wall static pressures at station 8 (appendix). The thermocouples at station 9 were provided by the manufacturer and used in determining the size of the exhaust nozzle to be installed on the engine for the purposes of this investigation. The atmospheric pressure surrounding the jet nozzle was sensed by a tube installed on the downstream side of the bulkhead dividing the inlet and exhaust sections of the altitude chamber.

Fuel flow was measured by two rotameters connected in series; two rotameters were necessary to cover the entire range of flows and to keep the physical size at a minimum. Calibration of the rotameters was made with the type of fuel used in the investigation (AN-F-58).

Procedure

Inlet and exhaust pressures and inlet temperature were set to correspond to the desired flight conditions in the NACA standard atmosphere assuming 100-percent ram recovery. The pressures were set to within ±0.02 inch of mercury and, in general, the inlet temperature to within ±5° F. The range of altitudes covered was from 20,000 to 60,000 feet at a flight Mach number of 0.62, and the range of flight Mach numbers covered was from 0.42 to 1.22 at an altitude of 30,000 feet. At each flight condition, the engine speed was varied over a range from approximately 5500 to 7800 rpm.

The additional instrumentation at station 8 used to measure the air flow through the engine dictated the use of a longer tail pipe to obtain a smooth velocity profile. A tail-pipe length of 65 inches was therefore used in place of the 30-inch tail pipe supplied by the manufacturer. The alterations necessitated a static sea-level calibration of the engine before the simulated altitude performance was determined in order to compare the calibration with that obtained by the manufacturer during the engine acceptance tests.

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RESULTS AND DISCUSSION

Sea-Level Calibration

The installation of the long tail pipe and more extensive instrumentation increased the back pressure in the tail pipe and reduced the engine speed at which rated tail-pipe gas temperature was obtained. In an effort to more nearly realize rated engine speed and rated tail-pipe gas temperature simultaneously, the exhaust-nozzle diameter was increased from 17.47 to 17.69 inches. Engine performance at static sea-level pressure with both the original 17.47-inch and 17.69-inch exhaust nozzles is shown in figure 5, along with the manufacturer's acceptance test with the original tail pipe and exhaust nozzle. The addition of the long tail pipe and extra instrumentation raised the corrected tail-pipe gas temperature approximately 70° for a given corrected engine speed (fig. 5(c)). When the original nozzle was replaced by the larger nozzle, the corrected tail-pipe gas temperatures dropped approximately 30° below those encountered with the manufacturer's original tail pipe and nozzle above an engine speed of 6500 rpm. The change in nozzle diameter reduced the corrected jet thrust about 3 percent (fig. 5(a)) and the corrected net-thrust specific fuel consumption by 1.5 percent (fig. 5(b)). At the conclusion of the investigation reported herein, a calibration run was made and the results obtained after 52 hours of engine operation are included in figure 5. At the end of the 52-hour period, neither the corrected jet thrust nor the corrected net-thrust specific fuel consumption had changed. Above a corrected engine speed of 6000 rpm, the corrected tail-pipe gas temperature decreased about 1 percent for a given corrected engine speed. The data presented in figure 5 are also tabulated in table I.

Simulated Flight Performance

Simulated altitude performance data obtained in this investigation are presented in table II. The data were adjusted to correct for small variations in compressor-inlet pressure and temperature from the desired standard flight conditions and are presented in table III and in figures 6 and 7.

Effect of altitude. - The effect of altitude and engine speed on jet thrust, net thrust, air flow, fuel flow, net-thrust specific fuel consumption, and tail-pipe gas total temperature are shown in figure 6 for a flight Mach number of 0.62. The reduction of jet thrust, net thrust, air flow, and fuel flow with increasing altitude is shown in figures 6(a) to 6(d). The net-thrust specific fuel consumption decreased as the altitude was increased from 20,000 to 40,000 feet below rated engine speed (fig. 6(e)) and then increased with further increases in altitude. The decrease, in part, was caused by an increase in compressor Mach number and compressor pressure ratio associated with the lower temperatures at the higher altitudes; the adverse trend came from the effects of Reynolds number (reference 1) on the compressor efficiency and the decrease in combustion efficiency. For a flight Mach number of 0.62, a minimum net-thrust specific fuel consumption of 1.21 pounds per hour per pound of thrust occurred at an altitude of 40,000 feet at an engine speed of 7000 rpm, 800 rpm below rated speed.

The tail-pipe total gas temperature (fig. 6(f)) was unaffected by changes in altitude up to an altitude of 40,000 feet. The increase in tail-pipe total gas temperature above an altitude of 40,000 feet is attributed to the decrease in compressor efficiency at the lower Reynolds numbers, which requires more work from the turbine to maintain a given rotational speed and hence a higher turbine-inlet temperature.

Effect of flight Mach number. - The effect of flight Mach number on engine performance at an altitude of 30,000 feet is shown in figure 7, in which the performance variables are shown as functions of engine speed for a range of flight Mach numbers. The jet-thrust and air-flow curves (figs. 7(a) and 7(c)) show the typical increases encountered with increasing flight Mach number. The net-thrust and fuel-flow curves (figs. 7(b) and 7(d)) show the typical crossover point between engine speeds of 6200 and 6800 rpm. 1420

The net-thrust specific fuel consumption shown in figure 7(e) increased for a given engine speed as the flight Mach number increased. A large portion of the increase occurred at flight Mach numbers between 0.42 and 0.62. A minimum net-thrust specific fuel consumption of 1.15 pounds per hour per pound of thrust occurred at an engine speed of 6800 rpm, a flight Mach number of 0.42, and an altitude of 30,000 feet. At rated engine speed, increasing the flight Mach number from 0.42 to 1.05 increased the specific fuel consumption from 1.22 to 1.32 pounds per hour per pound of thrust.

The tail-pipe total gas temperature (fig. 7(f)) decreased with increasing flight Mach number below an engine speed of 7000 rpm, and increased with increasing flight Mach number near rated engine speed. At rated engine speed, increasing the flight Mach number from 0.42 to 1.05 raised the tail-pipe total gas temperature from 1600° to 1650° R.

Generalized Performance

The performance data shown in figures 6 and 7 have been generalized by means of the altitude correction factors θ and δ. The generalized data are tabulated in table III and presented in figures 8 and 9. The correction factors were derived on the assumption that the effect of compressor Reynolds number would be negligible (reference 2). The performance data will thus generalize as a function of corrected engine speed if this assumption is correct.

Effect of altitude. - The effects of altitude on the generalized variables corrected jet thrust, corrected net thrust, corrected air flow, corrected fuel flow, corrected net-thrust specific fuel consumption, corrected tail-pipe total gas temperature, and compressor pressure ratio are shown in figure 8(a) to 8(g). Generalization of these variables was not obtained above an altitude of 30,000 feet over the complete speed range.

The effect of decreasing compressor Reynolds number on the compressor is to decrease the air flow and the compressor efficiency (reference 1), which results in a change in the engine operating line, giving rise to an increased compressor pressure ratio (fig. 8(g)), and consequently the nongeneralization of the performance data above an altitude of 30,000 feet.

Lack of a continued decrease in air flow (fig. 8(c)) with the change in altitude from 50,000 to 60,000 feet is attributed to instrumentation errors.

<u>Effect of flight Mach number</u>. - The effect of flight Mach number on corrected engine-performance data at an altitude of 30,000 feet is shown in figure 9. When the exhaust nozzle becomes choked the Mach number of the flow at any point within the engine is dependent only on the compressor Mach number and, therefore, for a given compressor Mach number the conditions of flow similarity are satisfied and the performance is independent of the flight Mach number. The relation between corrected engine speed and flight Mach number for exhaust-nozzle choking is shown in figure 10.

For the range of corrected engine speeds where the exhaust nozzle is choked, the performance variables air flow, fuel flow, and tail-pipe total gas temperature generalized (figs. 9(d), 9(e), and 9(g)). Although corrected jet thrust can not be expected to generalize because of changes in the pressure ratio across the exhaust nozzle, the corrected jet-thrust parameter, developed in reference 3, was independent of flight Mach number when the exhaust nozzle was choked (appendix), (fig. 9(b)). The corrected net thrust and net thrust specific fuel consumption were also independent of flight Mach number above a flight Mach number of 0.62 (figs. 9(c) and 9(f)). At higher altitudes, where the effect of compressor Reynolds number on engine performance is more pronounced, nongeneralization of the data might be expected.

SUMMARY OF RESULTS

Altitude performance of a J35-A-17 turbojet engine was investigated in an altitude chamber over a range of altitudes from 20,000 to 60,000 feet at a flight Mach number of 0.62 and a range of flight Mach numbers from 0.42 to 1.22 at an altitude of 30,000 feet. The following results were obtained.

1. At a flight Mach number of 0.62 the generalized performance variables corrected jet thrust, corrected net thrust, corrected air flow, corrected fuel flow, corrected net-thrust specific fuel consumption, and corrected tail-pipe total gas temperature generalized up to an altitude of 30,000 feet.

2. For corrected engine speeds at which critical flow existed in the exhaust nozzle, the generalized performance variables corrected jet thrust parameter, corrected air flow, corrected fuel flow, and corrected tail-pipe total gas temperature were independent of flight Mach number at an altitude of 30,000 feet.

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3. For a flight Mach number of 0.62, a minimum net-thrust specific fuel consumption of 1.21 pounds per hour per pound of thrust occurred at an altitude of 40,000 feet at an engine speed of 7000 rpm.

4. At rated engine speed, tail-pipe total gas temperature increased with increasing altitude above 40,000 feet. At 60,000 feet, the tail-pipe total gas temperature at rated engine speed exceeded the limit of 1725° R of the engine. No significant variation in tail-pipe total gas temperature with flight Mach number occurred at rated engine speed. Thus, operation of the engine was not temperature limited by variations in flight Mach number.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio.

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APPENDIX - CALCULATIONS

Symbols

The following symbols are used in the calculations and figures:

A area, sq ft

C_v velocity coefficient, $C_v = \frac{V_t}{V_t'}$

F thrust, lb

g acceleration of gravity, 32.2 ft/sec^2

H enthalpy of gas based on total temperature, Btu/lb

h enthalpy of gas based on static temperature, Btu/lb

ΔH difference between total and static enthalpy, Btu/lb

J mechanical equivalent of heat, 778 ft-lb/Btu

N engine speed, rpm

P total pressure, in. Hg absolute

p static pressure, in. Hg absolute

R universal gas constant, 53.34 ft-lb/(lb)(^{o}R)

T total temperature, ^{o}R

t static temperature, ^{o}R

V velocity, ft/sec

W_a air flow, lb/sec

W_f fuel flow, lb/hr

W_g gas flow, lb/sec

γ ratio of specific heats

δ ratio of compressor-inlet total pressure to standard sea-level pressure, 29.92 in. Hg absolute

θ ratio of compressor-inlet total temperature to standard sea-level temperature, 518.4 °R

Subscripts:

i indicated 1420

j jet

n net

s seal

t exhaust-nozzle throat

0 free-stream conditions

Numbered subscripts refer to instrumentation stations within the engine (fig. 3). Prime superscripts refer to calculations assuming an isentropic process.

Methods of Calculation

Temperature. - The static temperature was calculated from the indicated temperature by the following relation:

$$t_8 = \frac{T_{8,i}}{1 + 0.96\left(\frac{T_8}{t_8}\right) - 1}$$

where the temperature ratio was determined by using reference 4 and assuming:

$$\frac{T_{8,i}}{t_{8,i}} = \left(\frac{P_8}{p_8}\right)^{\frac{\gamma-1}{\gamma}}$$

and

$$T_8 - t_8 = T_{8,i} - t_{8,i}$$

Reference 4 accounts for the variation in γ (for air) due to temperature. The factor 0.96 is an average recovery factor for the thermocouples.

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Air flow. - Air flow was determined from the measurements of indicated temperature and total and static pressures in the tail pipe at station 8 as follows:

$$W_g = \frac{70.42\ p_8\ A_8}{R t_8} \sqrt{2gJ\Delta H_8}$$

where reference 4 was used to determine ΔH_8. The air flow is then

$$W_{a,8} = W_g - \frac{W_f}{3600}$$

Added to these computed values of air flow were a midframe leakage correction determined from a calibration supplied by the manufacturer and a cooling air flow measured by means of a flat plate orifice.

Flight Mach number. - The flight Mach number was determined from the equation

$$M_0 = \sqrt{\frac{2}{\gamma-1}\left[\left(\frac{p_1}{p_0}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}$$

Flight velocity. - The flight velocity was determined from the equation

$$V_0 = \sqrt{2gJH_0}$$

Jet thrust. - The jet thrust was determined from the pressure and temperature readings taken at stations 8 and 10 and calculated from the following equations:

$$F_j = \frac{W_g C_v V_t'}{g} + 70.42\ A_t (p_t - p_0)$$

where $C_v = 0.975$, $A_t = 1.7083$ square feet, and

$$V_t' = \sqrt{2gJ(H_8-h_t')}\ .$$

For critical flow in the exhaust nozzle,

$$p_t = P_8 \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$$

For subcritical flow,

$$p_t = p_0 = p_{10}$$

Net thrust. - Net thrust was determined by subtracting the inlet momentum of the air, at the particular simulated flight speed, from the jet thrust. Thus

$$F_n = F_j - \frac{W_a V_0}{g}$$

Jet-thrust parameter. - The jet-thrust parameter can be developed from the following equation for jet thrust:

$$F_j = \frac{W_g V_t}{g} + 70.42\ A_t\ (p_t - p_0)$$

If two different flight Mach numbers x and y are applied to the engine, both at the same corrected engine speed and both causing critical flow in the exhaust nozzle, then

$$\left(\frac{W_g V_t}{g\delta}\right)_x = \left(\frac{F_j}{\delta}\right)_x - 70.42\ A_t \left(\frac{p_t - p_0}{\delta}\right)_x$$

$$\left(\frac{W_g V_t}{g\delta}\right)_y = \left(\frac{F_j}{\delta}\right)_y - 70.42\ A_t \left(\frac{p_t - p_0}{\delta}\right)_y$$

Because it is assumed that the gas flow and throat velocity will generalize, these two equations are equal and

$$\left(\frac{F_j}{\delta}\right)_x - \left(\frac{F_j}{\delta}\right)_y = 70.42\ A_t\left[\left(\frac{p_t}{\delta}\right)_x - \left(\frac{p_0}{\delta}\right)_x - \left(\frac{p_t}{\delta}\right)_y + \left(\frac{p_0}{\delta}\right)_y\right]$$

Also

$$\left(\frac{p_t}{\delta}\right)_x = \left(\frac{p_t}{\delta}\right)_y$$

Therefore

$$\left(\frac{F_j + 70.42\ p_0\ A_t}{\delta}\right)_x = \left(\frac{F_j + 70.42\ p_0\ A_t}{\delta}\right)_y$$

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TABLE I - STATIC SEA-LEVEL CALIBRATIONS

[Altitude, 0; average flight Mach number, 0]

Point	Compressor-inlet total pressure, P_2 (in. Hg abs.)	Exhaust static pressure, p_0 (in. Hg abs.)	Compressor-inlet total temperature T_2 (°R)	Engine speed N (rpm)	Jet thrust F_j (lb)	Tail-pipe total gas temperature T_8 (°R)	Net-thrust specific fuel consumption W_f/F_n lb/(lb)(hr)	Accumulative engine time (hr)
1	29.40	29.39	573	2970	254	1634	3.063	4.25
2	29.40	29.38	577	4088	533	1545	2.253	4.25
3	29.40	29.37	579	5567	1287	1450	1.474	4.25
4	29.40	29.38	579	6122	1811	1461	1.250	4.25
5	29.40	29.39	582	6678	2433	1550	1.164	4.25
6	29.40	29.39	577	7058	3045	1643	1.133	4.25
7	29.40	29.36	577	7422	3646	1750	1.119	4.25
8	29.33	29.30	570	7793	4269	1735	1.107	6
9	29.33	29.35	571	7420	3633	1633	1.100	6
10	29.32	29.30	569	7053	3066	1535	1.099	6
11	29.32	29.30	567	6682	2513	1457	1.130	6
12	29.34	29.35	571	6131	1819	1383	1.226	6
13	29.32	29.33	574	5574	1129	(a)	1.694	6
14	29.33	29.35	579	4096	533	1486	2.429	6
15	29.31	29.30	572	2972	252	1574	3.733	6
16	29.35	29.37	549	2972	265	1516	(a)	52
17	29.35	29.35	555	4085	556	1445	(a)	52
18	29.35	29.35	556	5567	1327	1347	1.334	52
19	29.28	29.28	555	6123	1877	1346	1.183	52
20	29.35	29.35	557	6680	2543	1433	1.113	52
21	29.35	29.35	557	7040	3132	1515	1.082	52
22	29.35	29.35	562	7426	(a)	(a)	(a)	52
23	29.32	29.32	561	7784	4337	1704	1.104	52

[a]Data unavailable

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TABLE II - PERFORMANCE AND OPERATIONAL DATA OBTAINED AT SIMULATED ALTITUDE CONDITIONS

Point	Altitude (ft)	Compressor-inlet total pressure, P_1 (in. Hg abs.)	Exhaust static pressure, p_0 (in. Hg abs.)	Ram pressure ratio, P_1/p_0	Compressor-inlet total temperature, T_2 (°R)	Engine speed, N (rpm)	Jet thrust, F_j, (lb)	Air flow, W_a (lb/sec).	Fuel flow, W_f (lb/hr)	Tail-pipe indicated gas temperature, $T_{8,i}$ (°R)	Jet-nozzle indicated gas temperature, $T_{9,i}$ (°R)	Compressor pressure ratio, P_4/P_2	Fuel-pump discharge pressure (lb/sq in. gage)	Main fuel manifold pressure (lb/sq in. gage)	Rear bearing oil pressure (lb/sq in. gage)	Rear bearing temperature (°F)	Oil tank temperature (°F)	Accumulative engine time (hr)
1	20,000	17.90	13.77	1.300	474	5565	1121	37.22	818	926	925	2.499	240	67	6	155	(a)	45
2	20,000	17.89	13.75	1.301	477	6123	1615	42.49	1173	1049	1049	3.096	240	97	7	166	(a)	45
3	20,000	17.89	13.75	1.301	478	6680	2334	48.24	1780	1238	1233	3.814	235	120	8	183	(a)	45
4	20,000	17.90	13.77	1.300	480	7045	2779	50.61	2250	1379	1374	4.228	225	140	9	205	(a)	45
5	20,000	17.88	13.75	1.300	483	7414	3201	52.95	2700	1504	1501	4.644	220	159	9	236	(a)	45
6	20,000	17.87	13.74	1.301	482	7756	3516	54.38	3110	1612	1610	4.905	220	173	8	255	(a)	45
7	30,000	10.00	8.85	1.131	424	5569	725	23.03	666	992	993	2.949	120	38	4	153	(a)	36
8	30,000	10.00	8.89	1.124	422	6120	1064	26.96	906	1104	1102	3.700	125	77	6	154	(a)	36
9	30,000	10.00	8.88	1.126	428	6676	1447	29.87	1242	1270	1267	4.376	130	93	7	166	(a)	36
10	30,000	10.02	8.89	1.127	422	7053	1712	31.68	1500	1378	1368	4.843	135	103	13	184	(a)	36
11	30,000	10.02	8.88	1.128	427	7422	1840	31.90	1720	1488	1489	5.155	140	110	13	205	(a)	36
12	30,000	10.00	8.88	1.126	428	7791	2007	32.38	1960	1609	1615	5.416	140	120	13	238	(a)	36
13	30,000	11.54	8.89	1.298	447	5567	791	25.36	608	914	918	2.691	145	34	11	116	63	45
14	30,000	11.56	8.89	1.300	450	6125	1181	29.36	878	1053	1053	3.369	145	76	9	148	55	45
15	30,000	11.55	8.88	1.301	444	6680	1695	33.20	1345	1250	1248	4.213	145	97	9	167	50	45
16	30,000	11.54	8.88	1.300	445	7042	2014	35.13	1640	1378	1379	4.660	145	109	8	214	(a)	45
17	30,000	11.57	8.89	1.301	446	7409	2237	36.15	1910	1497	1502	5.024	150	122	8	238	(a)	45
18	30,000	11.51	8.88	1.296	447	7778	2390	36.28	2200	1616	1617	5.272	150	130	7	258	(a)	45
19	30,000	13.31	8.87	1.501	467	5565	848	27.49	542	846	843	2.438	170	26	5	168	(a)	28
20	30,000	13.35	8.89	1.502	466	6122	1310	32.09	841	1002	1001	3.148	175	70	7	181	(a)	28
21	30,000	13.32	8.88	1.500	466	6678	1945	36.51	1385	1232	1226	3.952	175	101	8	199	(a)	28
22	30,000	13.30	8.87	1.499	462	7051	2342	38.87	1775	1380	1377	4.483	175	116	9	222	(a)	28
23	30,000	13.34	8.89	1.501	465	7426	2665	40.48	2140	1510	1508	4.890	175	130	8	237	(a)	28
24	30,000	13.34	8.89	1.501	465	7793	2861	41.16	2440	1621	1620	5.145	175	142	9	261	(a)	28
25	30,000	15.11	8.88	1.702	476	5565	994	31.11	514	797	794	2.304	200	24	3	213	55	28
26	30,000	15.10	8.87	1.702	478	6127	1547	35.97	862	974	.974	2.976	205	74	4	195	55	28
27	30,000	15.12	8.88	1.705	476	6682	(a)	(a)	1480	1219	1214	(a)	205	107	6	202	54	28
28	30,000	15.12	8.88	1.703	478	7049	2734	43.32	1930	1379	1377	4.360	200	125	8	222	51	28
29	30,000	15.11	8.88	1.702	478	7422	3104	45.22	2350	1513	1508	4.721	200	142	7	238	50	28
30	30,000	15.11	8.89	1.700	478	7787	3336	46.09	2695	1626	1625	5.108	200	155	6	257	50	28
31	30,000	17.76	8.88	2.000	491	5565	1178	35.03	483	757	755	2.164	250	24	5	170	61	28
32	30,000	17.89	8.89	2.012	492	6120	1843	40.80	896	948	947	2.835	250	83	7	178	60	28
33	30,000	17.76	8.87	2.002	492	6678	2703	46.21	1640	1212	1208	3.664	245	115	8	198	60	28
34	30,000	17.76	8.88	2.000	491	7049	3256	49.42	2170	1380	1375	4.189	240	132	8	218	60	28
35	30,000	17.74	8.87	2.000	495	7424	3729	52.02	2670	1518	1515	4.665	235	156	8	248	57	28
36	30,000	17.72	8.85	2.002	494	7791	4077	53.61	3100	1635	1631	4.982	235	172	7	277	55	28
37	30,000	22.20	8.89	2.497	533	6123	2104	46.03	800	894	891	2.484	320	73	5	204	91	49
38	30,000	22.20	8.88	2.500	545	6127	1993	44.65	745	889	886	2.413	320	62	4	202	89	49
39	30,000	22.19	8.87	2.502	534	6678	3051	52.14	1570	1150	1153	3.206	315	117	4	225	86	49
40	30,000	22.20	8.87	2.503	534	6680	3014	51.65	1570	1151	1151	3.203	310	118	5	217	91	49
41	30,000	22.19	8.87	2.502	535	7049	3742	55.93	2240	1342	1337	3.730	305	144	6	237	90	49
42	30,000	22.20	8.88	2.500	534	7420	4391	59.59	2920	1506	1501	4.198	300	170	8	250	86	49
43	40,000	7.21	5.54	1.301	420	5582	560	16.97	455	917	918	2.897	90	13	3	227	(a)	43
44	40,000	7.21	5.54	1.301	425	6101	816	19.39	627	1060	1058	3.625	100	32	7	205	(a)	43
45	40,000	7.22	5.54	1.303	424	6689	1152	21.70	913	1268	1267	4.457	100	75	8	208	(a)	43
46	40,000	7.21	5.54	1.301	424	7058	1350	22.85	1115	1395	1395	4.948	102	84	8	222	(a)	43
47	40,000	7.22	5.55	1.301	424	7420	1481	23.43	1290	1506	1507	5.245	105	92	9	241	(a)	43
48	40,000	7.20	5.53	1.302	429	7782	1569	23.44	1450	1626	1631	5.470	110	97	7	263	(a)	43
49	50,000	4.48	3.43	1.306	425	6118	500	11.70	435	1100	1095	3.690	85	9	2	211	(a)	43
50	50,000	4.45	3.42	1.301	423	6695	706	13.05	622	1324	1328	4.594	95	29	3	230	(a)	43
51	50,000	4.46	3.42	1.304	426	7060	820	13.69	734	1446	1450	5.060	100	40	0	250	(a)	43
52	50,000	4.47	3.43	1.303	427	7429	915	14.21	845	1560	1570	5.416	100	63	2	268	(a)	43
53	50,000	4.48	3.44	1.302	426	7780	999	14.57	965	1676	1686	5.675	105	75	0	257	(a)	43
54	60,000	2.77	2.14	1.294	428	6672	434	7.94	415	1369	1377	4.521	96	6	3	294	(a)	43
55	60,000	2.78	2.14	1.299	430	7053	526	8.50	510	1507	1519	5.027	100	8	8	250	(a)	43
56	60,000	2.78	2.12	1.311	429	7401	576	8.77	575	1601	1618	5.398	105	22	1	234	(a)	43
57	60,000	2.76	2.11	1.308	434	7772	636	8.88	673	1780	1804	5.856	110	34	0	276	(a)	43

[a]Data unavailable

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TABLE III - PERFORMANCE DATA ADJUSTED TO STANDARD ALTITUDE AND CORRECTED TO STANDARD SEA-LEVEL ATMOSPHERIC CONDITIONS

Point	Altitude (ft)	Average flight Mach number	Engine speed (rpm) Altitude N	Engine speed (rpm) Sea level $N/\sqrt{\theta}$	Jet thrust (lb) Altitude F_j	Jet thrust (lb) Sea level F_j/δ	Jet thrust (lb) $\frac{F_j+p_0A_t}{\delta}$	Net thrust (lb) Altitude F_n	Net thrust (lb) Sea level F_n/δ	Air flow (lb/sec) Altitude W_a	Air flow (lb/sec) Sea level $W_a\sqrt{\theta}/\delta$	Fuel flow (lb/hr) Altitude W_f	Fuel flow (lb/hr) Sea level $W_f/\delta\sqrt{\theta}$	Net-thrust specific fuel consumption lb/(hr)(lb thrust) Altitude W_f/F_n	Net-thrust specific fuel consumption lb/(hr)(lb thrust) Sea level $W_f/F_n\sqrt{\theta}$	Exhaust-gas temperature (°R) Tail pipe Altitude T_8	Exhaust-gas temperature (°R) Tail pipe Sea level T_8/θ	Exhaust-gas temperature (°R) Jet nozzle Sea level $T_{9,1}/\theta$
1	20,000	0.62	5610	5818	1120	1874	4678	378	633	36.88	59.50	824	1429	2.179	2.259	943	1015	1012
2	20,000		6155	6383	1615	2699	5469	762	1275	42.23	68.13	1178	2044	1.545	1.603	1061	1141	1141
3	20,000		6705	6953	2332	3901	6670	1366	2285	48.01	77.45	1786	3097	1.307	1.355	1250	1345	1337
4	20,000		7062	7323	2776	4644	7419	1760	2944	50.44	81.37	2255	3909	1.280	1.328	1387	1493	1486
5	20,000		7407	7681	3201	5355	8127	2135	3572	53.00	85.51	2697	4679	1.263	1.310	1503	1617	1612
6	20,000		7756	8043	3518	5886	8658	2420	4049	54.41	87.78	3112	5399	1.286	1.333	1612	1735	1731
7	30,000	0.42	5558	6153	706	2170	5360	416	1275	22.54	62.38	649	2201	1.560	1.726	989	1212	1213
8	30,000		6125	6778	1040	3186	6395	707	2165	26.34	72.88	886	3004	1.254	1.388	1107	1357	1352
9	30,000		6636	7346	1414	4331	7536	1041	3189	29.37	81.26	1207	4093	1.159	1.283	1257	1540	1535
10	30,000		7058	7813	1670	5115	8319	1274	3902	30.89	85.48	1465	4967	1.150	1.273	1381	1694	1680
11	30,000		7385	8175	1795	5498	8696	1393	4267	31.28	86.54	1670	5663	1.199	1.327	1474	1807	1806
12	30,000		7745	8574	1962	6010	9213	1557	4769	31.84	88.09	1905	6459	1.224	1.354	1592	1952	1957
13	30,000	0.62	5545	5994	791	2050	4828	302	783	25.46	61.06	606	1697	2.007	2.168	907	1061	1063
14	30,000		6082	6575	1179	3057	5830	607	1574	29.51	70.79	870	2439	1.433	1.550	1040	1215	1214
15	30,000		6680	7221	1693	4390	7162	1064	2733	33.17	79.57	1344	3767	1.275	1.378	1252	1463	1459
16	30,000		7035	7605	2014	5222	8067	1329	3446	35.17	84.37	1638	4591	1.233	1.332	1376	1609	1607
17	30,000		7394	7993	2230	5782	8555	1534	3978	36.11	86.61	1900	5326	1.239	1.339	1491	1743	1747
18	30,000		7747	8375	2396	6213	8995	1696	4398	36.52	87.59	2197	6158	1.295	1.400	1604	1875	1875
19	30,000	0.78	5537	5864	848	1905	4510	178	401	27.66	58.66	540	1284	3.029	3.206	838	941	937
20	30,000		6098	6458	1307	2936	5539	525	1176	32.15	68.18	836	1988	1.597	1.690	995	1116	1113
21	30,000		6651	7043	1945	4368	6772	1054	2367	36.66	77.75	1379	3280	1.308	1.386	1224	1374	1365
22	30,000		7051	7462	2347	5271	7676	1405	3156	38.95	82.60	1779	4231	1.266	1.341	1382	1551	1545
23	30,000		7404	7841	2660	5974	8377	1677	3767	40.52	85.94	2129	5064	1.270	1.344	1502	1685	1680
24	30,000		7770	8228	2855	6412	8816	1856	4169	41.20	87.38	2428	5775	1.308	1.385	1612	1809	1805
25	30,000	0.91	5583	5806	993	1967	4085	125	248	30.98	59.01	515	1061	4.107	4.271	803	869	864
26	30,000		6136	6381	1547	3065	5182	542	1074	35.92	68.42	863	1778	1.592	1.655	979	1059	1056
27	30,000		6701	6969	(a)	(a)	(a)	(a)	(a)	(a)	(a)	1483	3055	(a)	(a)	(a)	(a)	1321
28	30,000		7059	7341	2731	5410	7528	1521	3013	43.22	82.33	1931	3978	1.270	1.320	1385	1499	1494
29	30,000		7432	7729	3101	6143	8260	1839	3643	45.11	85.93	2351	4844	1.278	1.330	1520	1644	1636
30	30,000		7793	8105	3333	6603	8724	2046	4057	46.01	87.64	2694	5550	1.315	1.368	1631	1765	1761
31	30,000	1.05	5623	5719	1178	1985	3788	57	96	34.67	57.45	488	836			775	800	797
32	30,000		6176	6281	1830	3084	4878	526	887	40.15	66.53	898	1539	1.706	1.735	968	1000	997
33	30,000		6741	6856	2703	4555	6356	1222	2059	45.78	75.85	1656	2838	1.355	1.378	1257	1279	1273
34	30,000		7121	7242	3256	5486	7289	1673	2819	48.92	81.06	2192	3756	1.310	1.332	1410	1458	1451
35	30,000		7470	7597	3733	6290	8093	2060	3471	51.75	85.74	2689	4608	1.305	1.328	1539	1592	1586
36	30,000		7850	7983	4085	6883	8685	2360	3977	53.31	88.32	3130	5364	1.326	1.349	1662	1719	1712
37	30,000	1.22	6135	6041	2104	2836	4281	369	497	46.94	62.89	802	1064	2.174	2.141	899	872	867
38	30,000		6072	5978	1993	2687	4129	288	389	46.06	61.67	738	980	2.561	2.522	874	847	843
39	30,000		6685	6582	3063	4115	5557	1083	1460	52.11	71.34	1573	2088	1.452	1.430	1155	1120	1120
40	30,000		6683	6580	3014	4063	5504	1062	1432	51.63	70.68	1571	2085	1.479	1.456	1154	1119	1117
41	30,000		7049	6940	3744	5047	6488	1628	2195	55.98	76.62	2241	2974	1.377	1.355	1344	1303	1296
42	30,000		7422	7308	4391	5919	7361	2137	2881	59.57	81.55	2921	3877	1.367	1.346	1509	1463	1456
43	40,000	0.62	5602	6201	560	2523	5095	243	1010	16.91	63.42	487	2099	1.877	2.078	925	1134	1133
44	40,000		6089	6741	816	3387	6162	451	1872	19.43	72.88	626	2876	1.388	1.536	1058	1297	1292
45	40,000		6679	7394	1150	4775	7542	742	3079	21.69	81.34	910	4182	1.227	1.358	1266	1552	1549
46	40,000		7051	7805	1350	5605	8379	920	3820	22.87	85.76	1114	5120	1.211	1.340	1394	1709	1707
47	40,000		7416	8210	1478	6137	8909	1039	4314	23.39	87.71	1287	5915	1.239	1.371	1506	1847	1846
48	40,000		7736	8564	1571	6523	9293	1124	4667	23.60	88.50	1443	6632	1.284	1.421	1606	1971	1974
49	50,000	0.62	6106	6759	499	3339	6102	277	1857	11.70	70.79	433	3211	1.563	1.729	1097	1346	1336
50	50,000		6692	7408	709	4746	7517	464	3103	13.11	79.30	624	4628	1.347	1.491	1325	1624	1627
51	50,000		7032	7784	822	5502	8268	561	3758	13.77	83.27	733	5430	1.306	1.445	1437	1761	1754
52	50,000		7392	8183	915	6123	8891	645	4318	14.28	86.36	841	6231	1.304	1.443	1547	1896	1906
53	50,000		7757	8587	997	6672	9441	724	4845	14.58	88.15	960	7116	1.327	1.469	1668	2045	2055
54	60,000	0.62	6632	7342	435	4690	7475	285	3077	7.99	77.91	415	4931	1.447	1.603	1355	1661	1668
55	60,000		6997	7746	524	5653	8429	363	3921	8.54	83.27	504	6026	1.388	1.537	1484	1819	1831
56	60,000		7357	8144	574	6196	8944	406	4384	8.79	85.70	570	6807	1.405	1.553	1582	1940	1958
57	60,000		7672	8493	639	6896	9209	467	5042	9.04	88.10	667	7969	1.428	1.581	1738	2130	2156

[a]Data unavailable

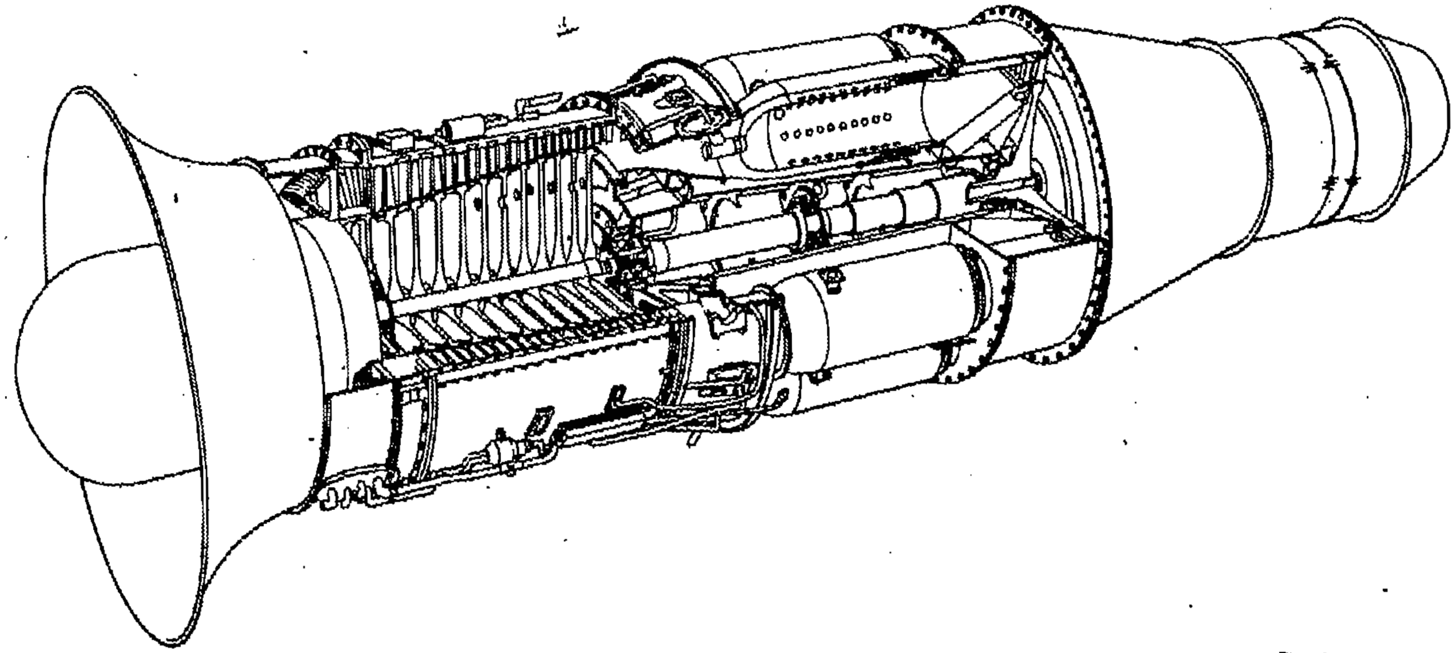


Figure 1. - Cutaway view of engine.

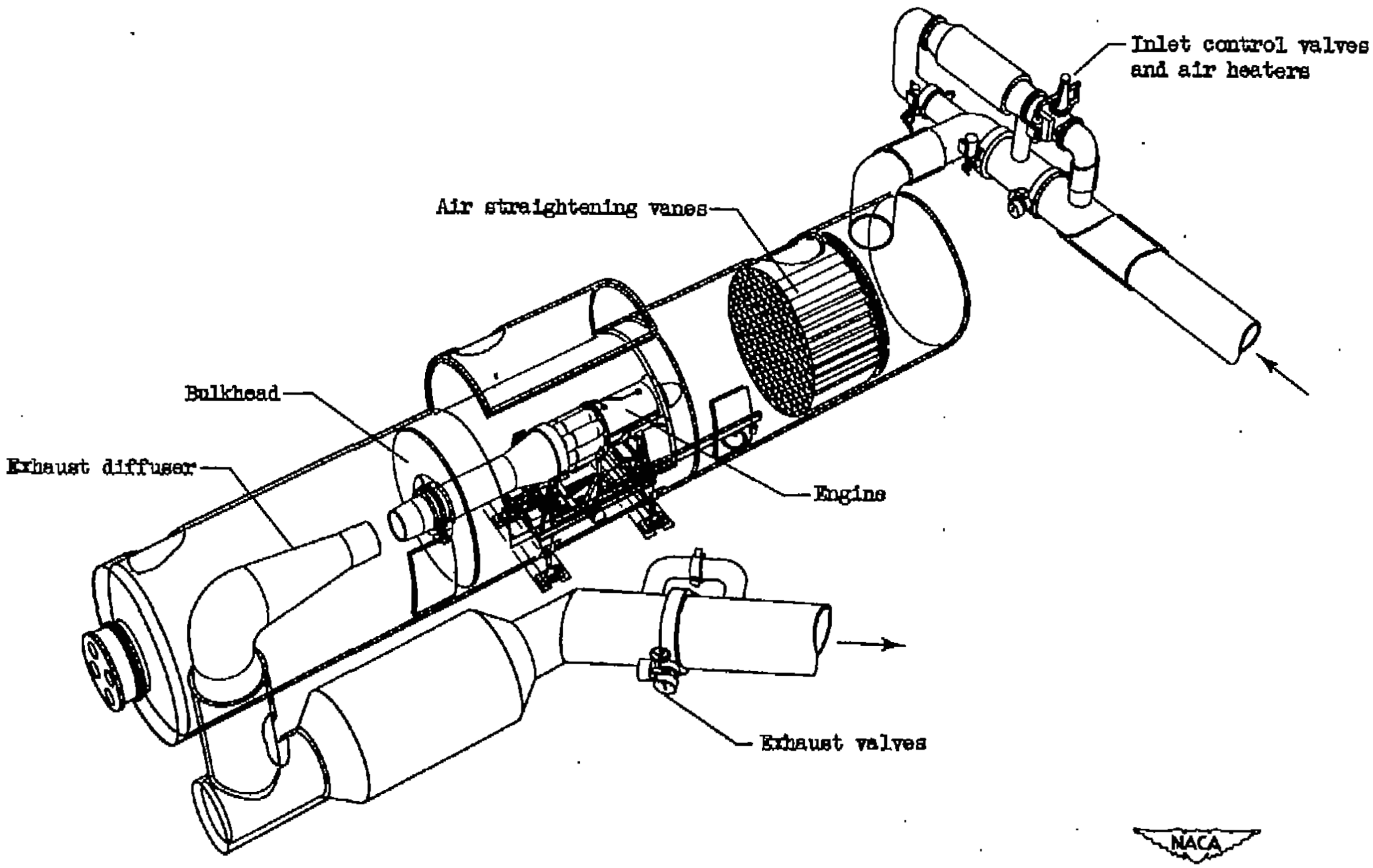


Figure 2. - Altitude chamber with J35-A17 engine installed in test section.

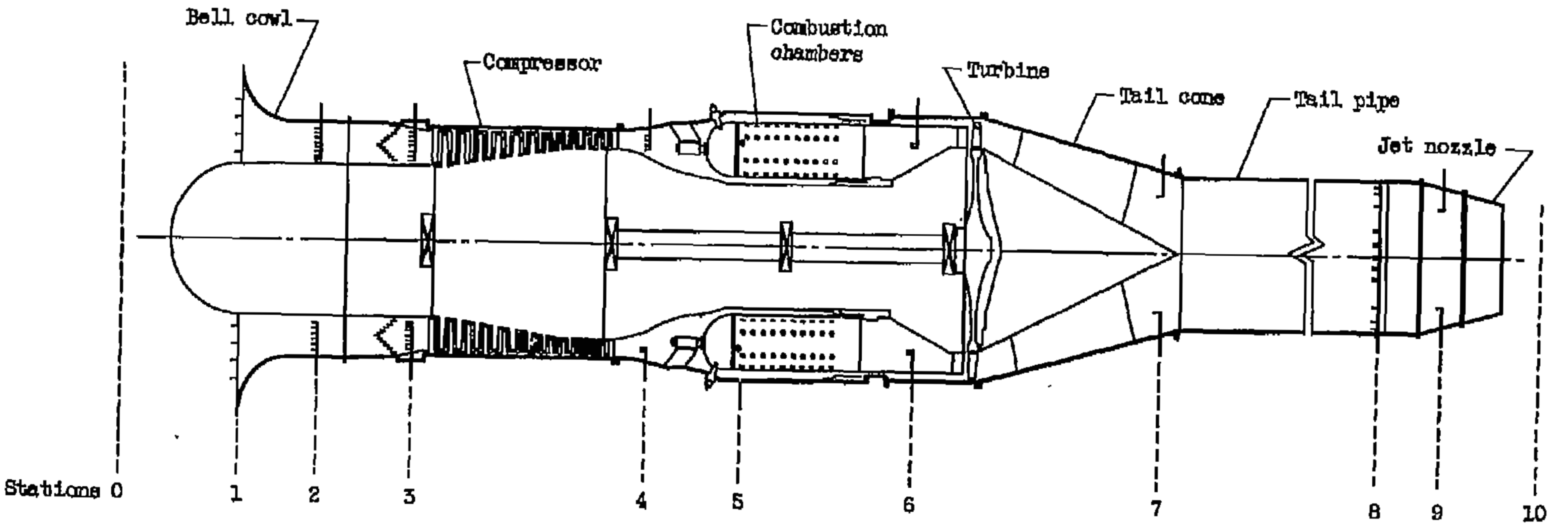
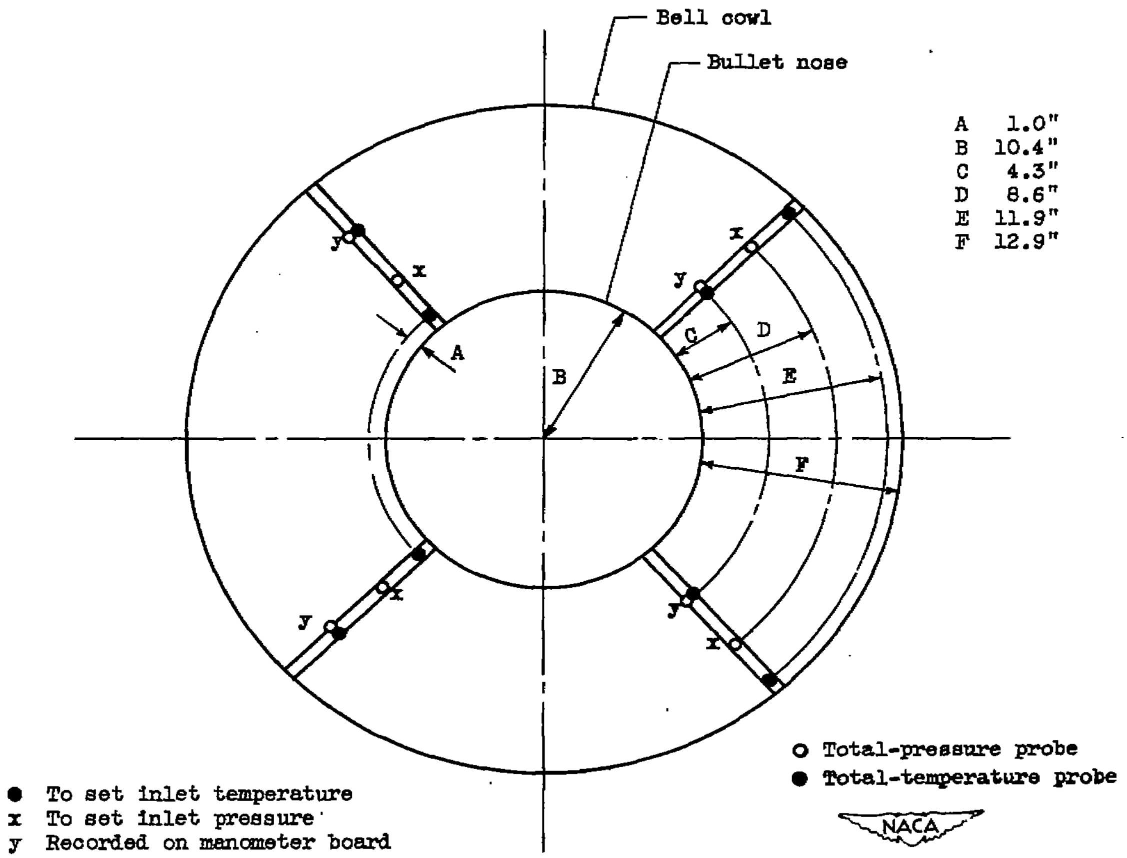


Figure 3. - Sectional view of engine showing instrumentation stations.

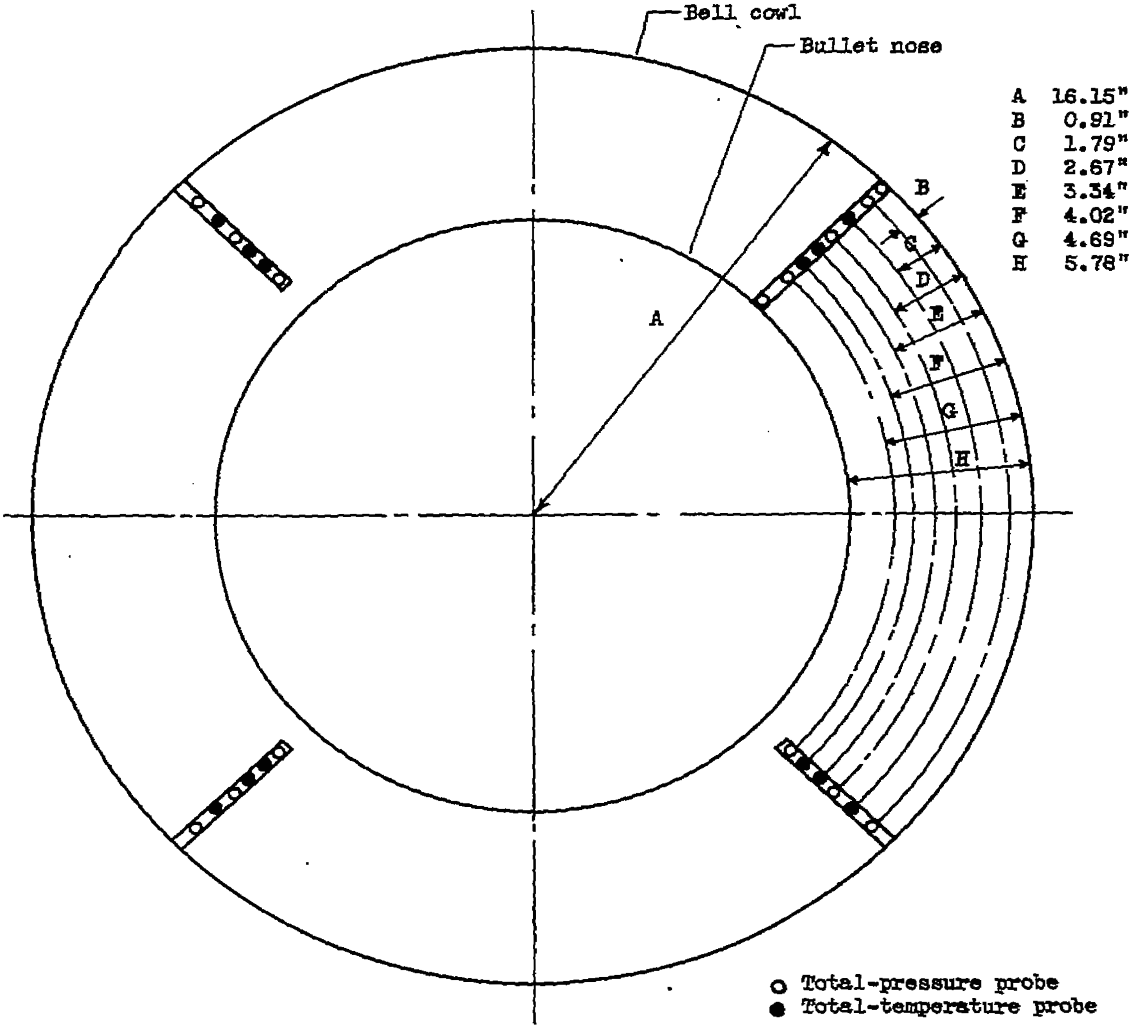
1420



(a) Station 1.

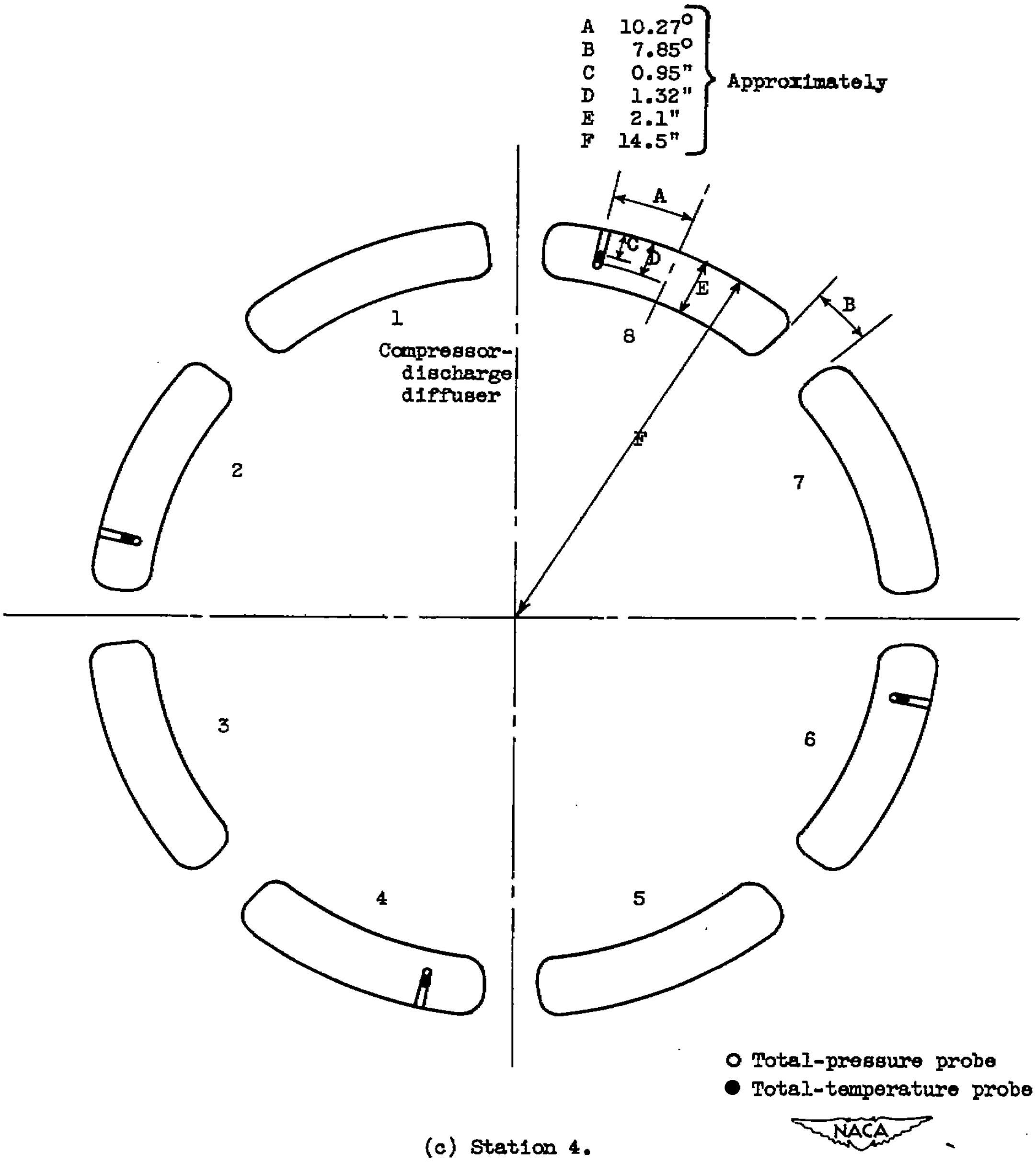
Figure 4. - Instrumentation details.

1420



(b) Station 2.

Figure 4. - Instrumentation details.



(c) Station 4.

Figure 4. - Instrumentation details.

1420

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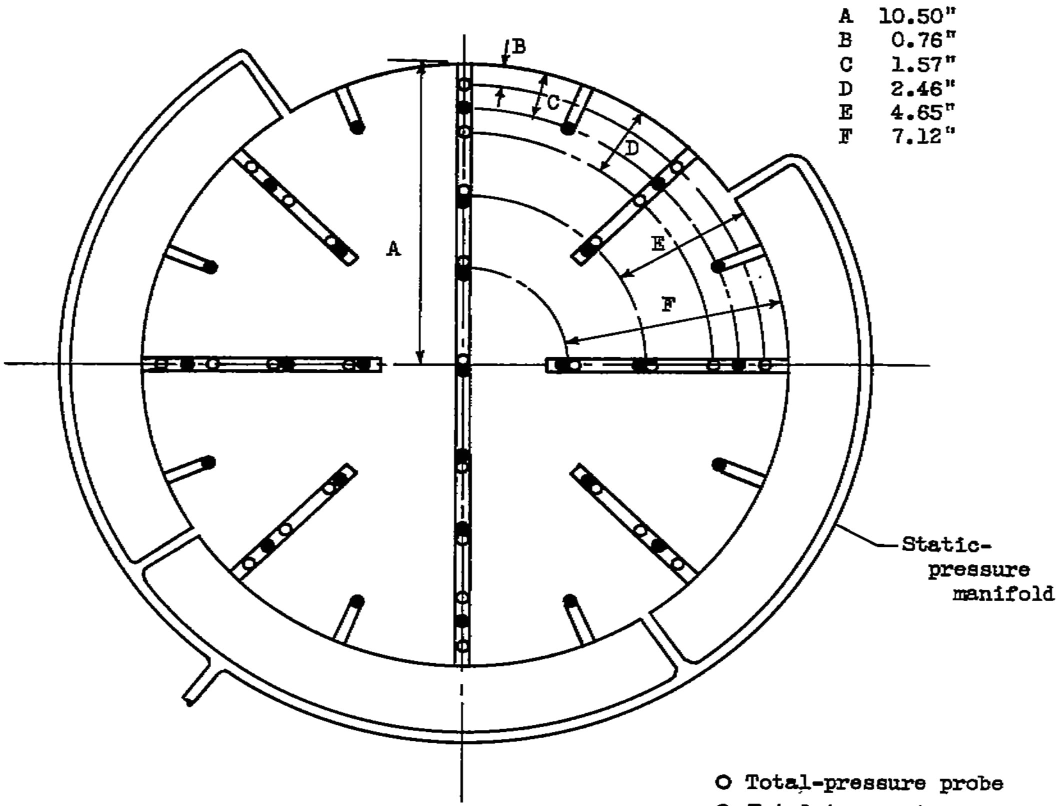
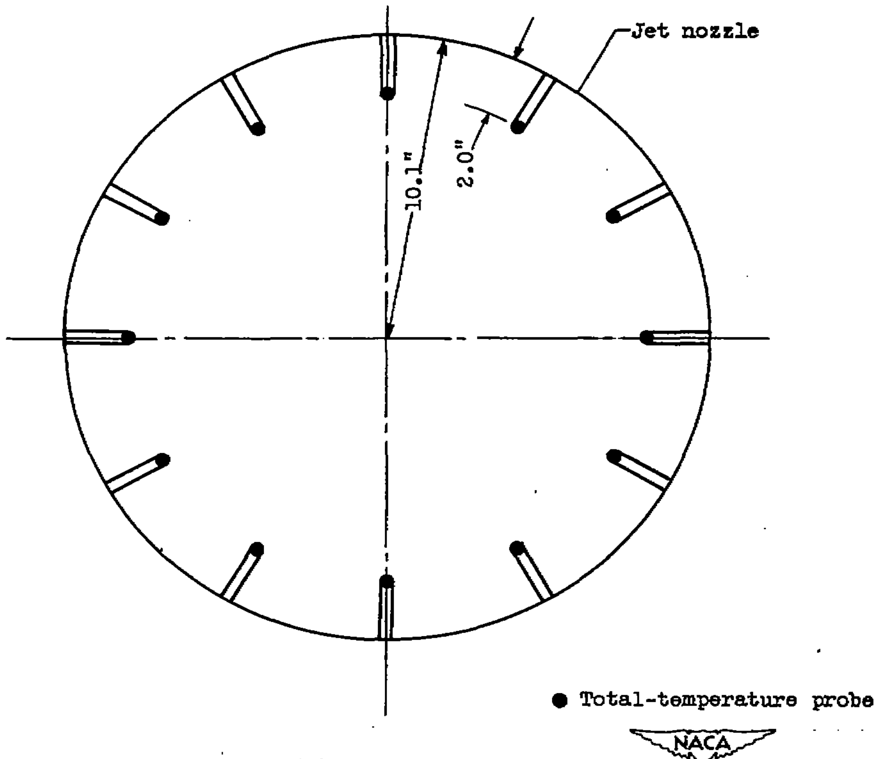


(d) Station 8.

Figure 4. - Instrumentation details.

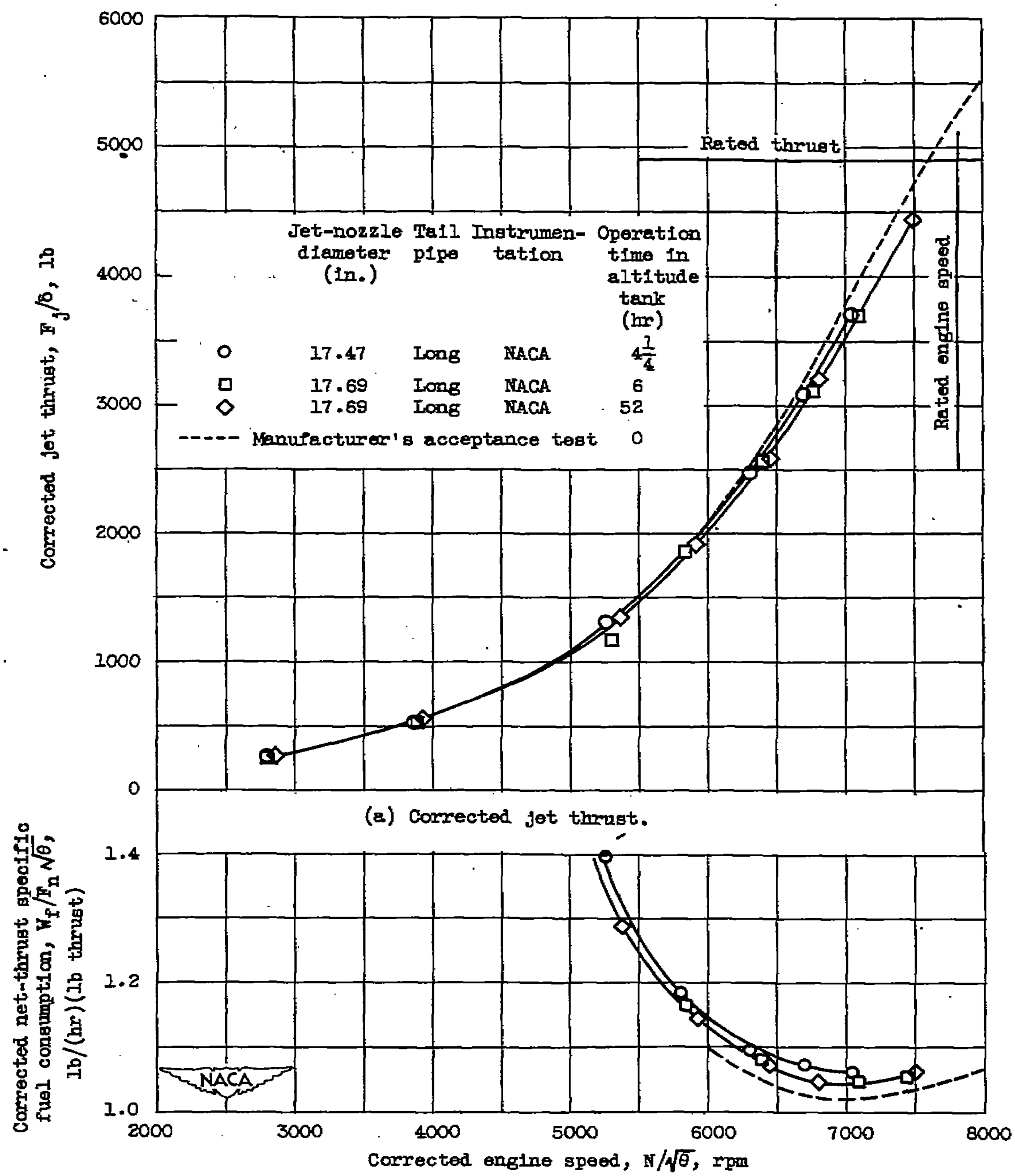
1420



(e) Station 9.

Figure 4. - Instrumentation details.

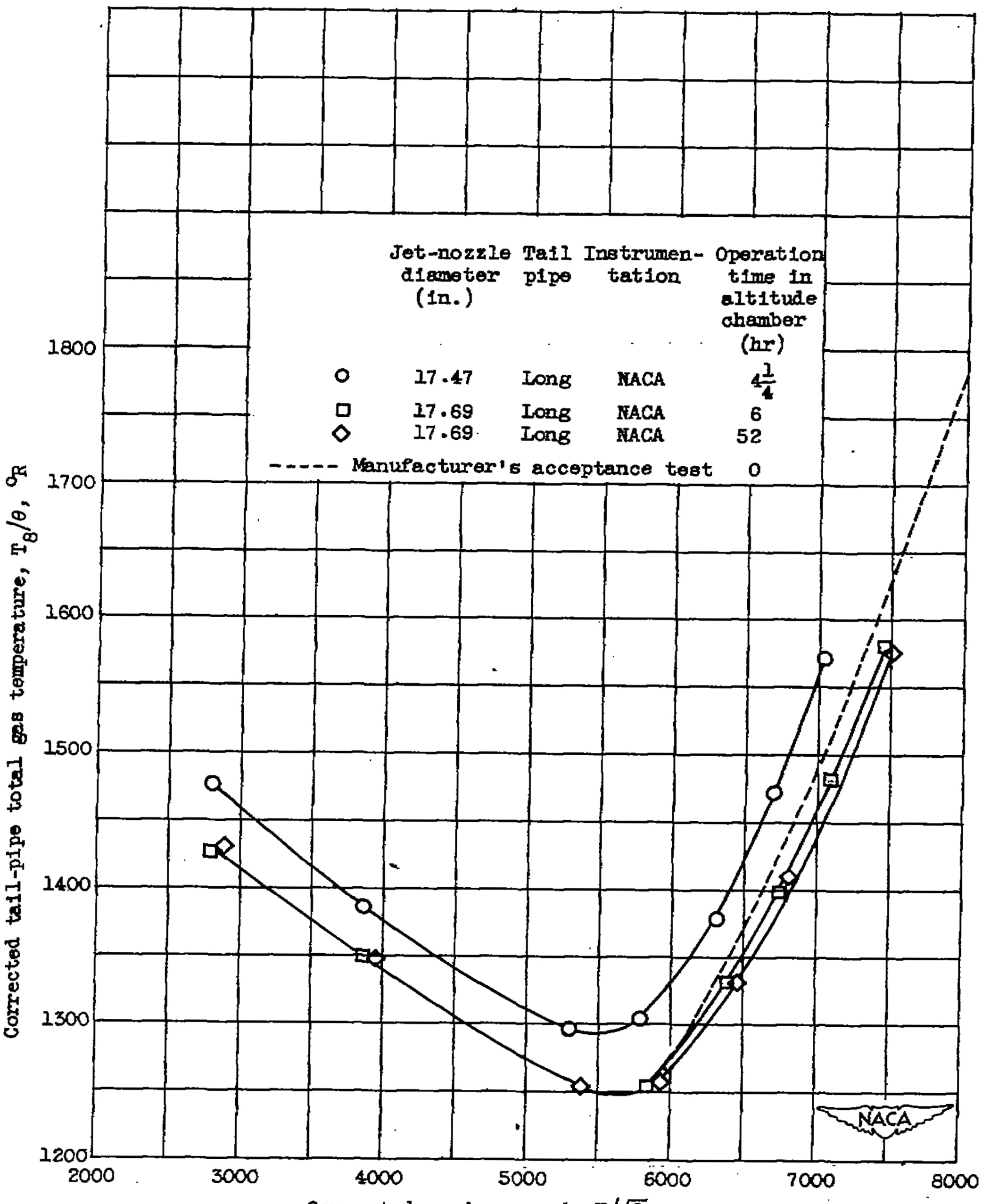
1420



(b) Corrected net-thrust specific fuel consumption.

Figure 5. - Effect of exhaust configuration and engine time on performance of engine at static sea-level pressures.

1420

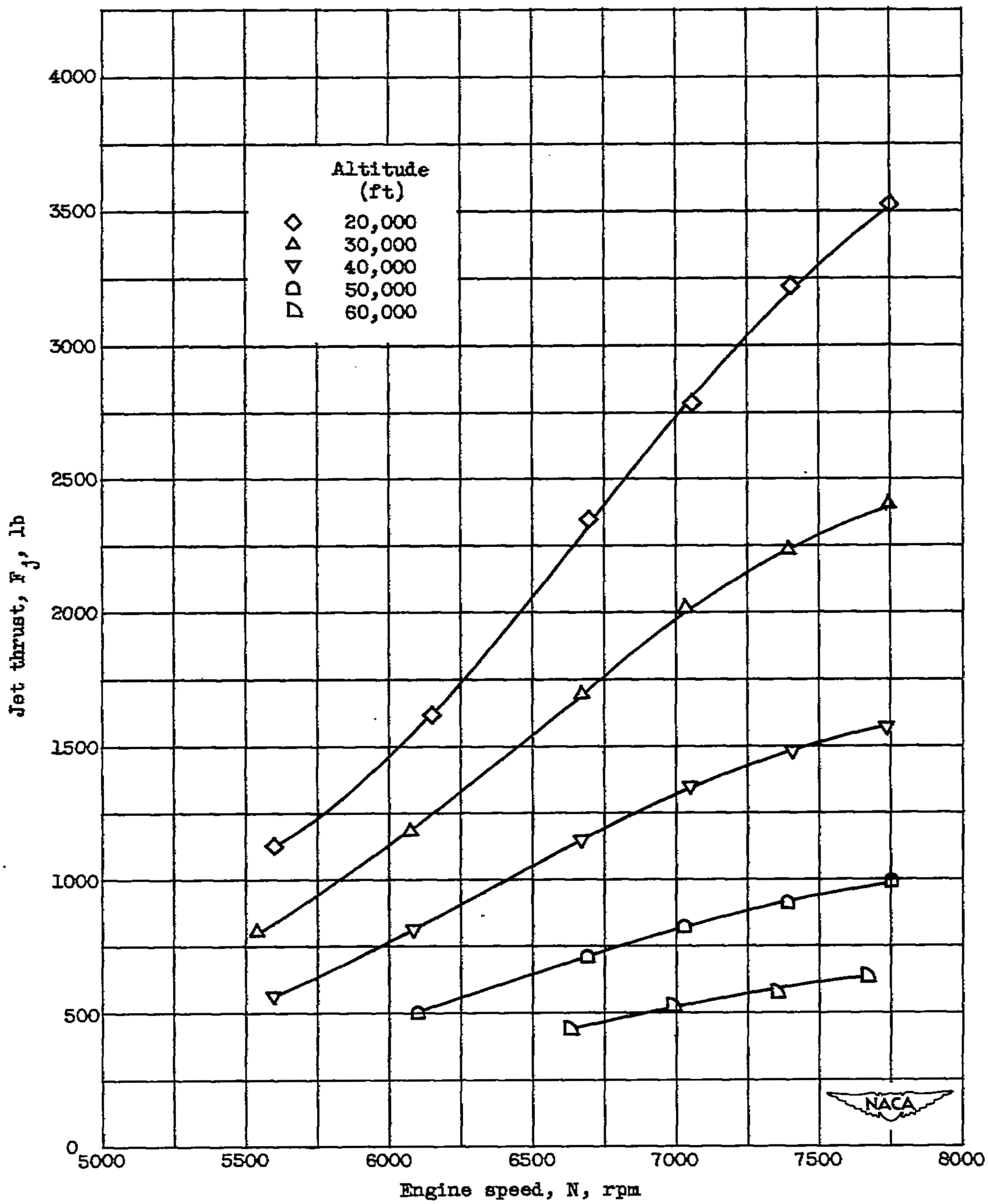


Corrected engine speed, $N/\sqrt{\theta}$, rpm

(c) Corrected tail-pipe gas temperature.

Figure 5. - Concluded. Effect of exhaust configuration and engine time on performance of engine at static sea-level pressure.

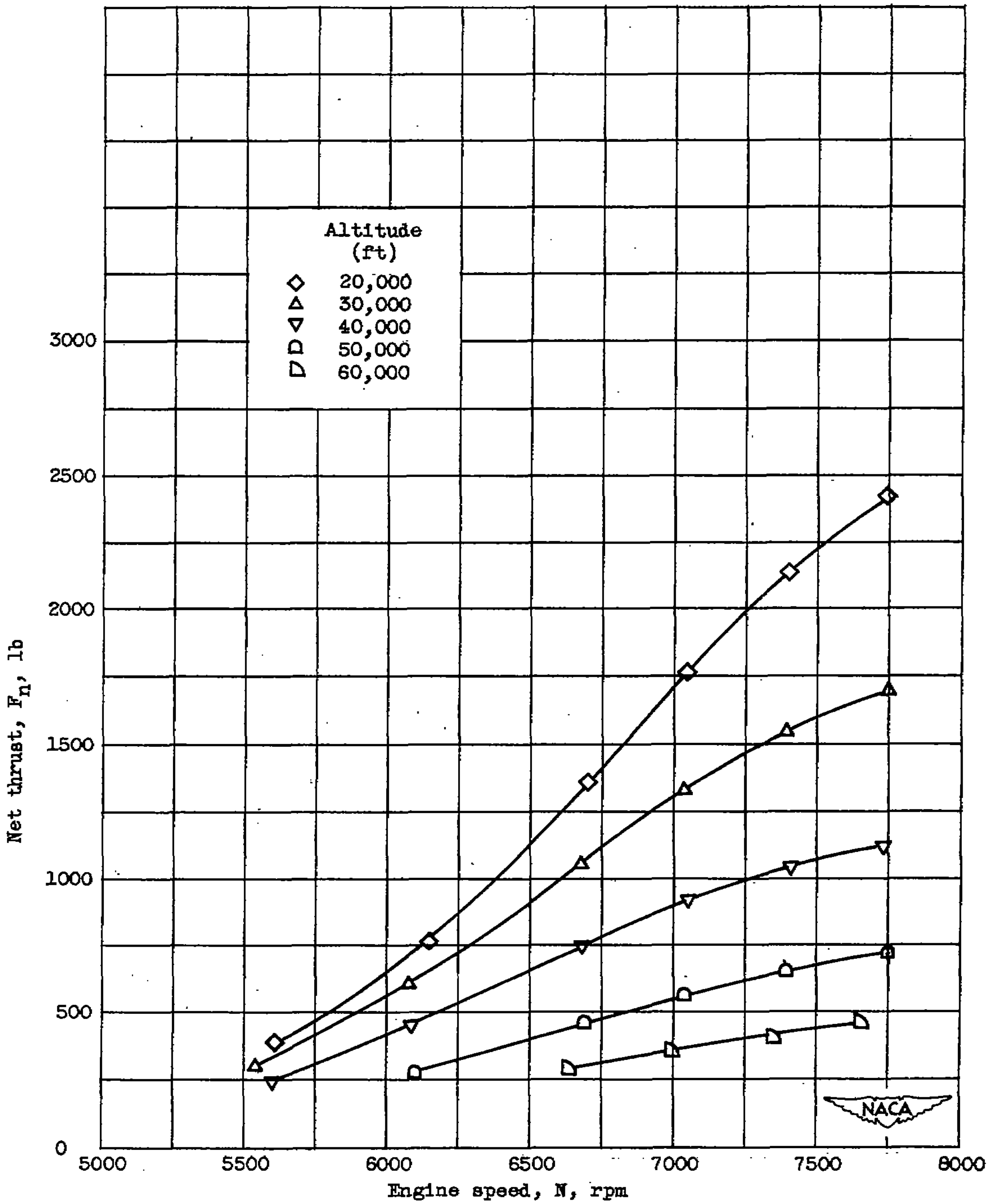
1420



(a) Jet thrust.

Figure 6. - Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

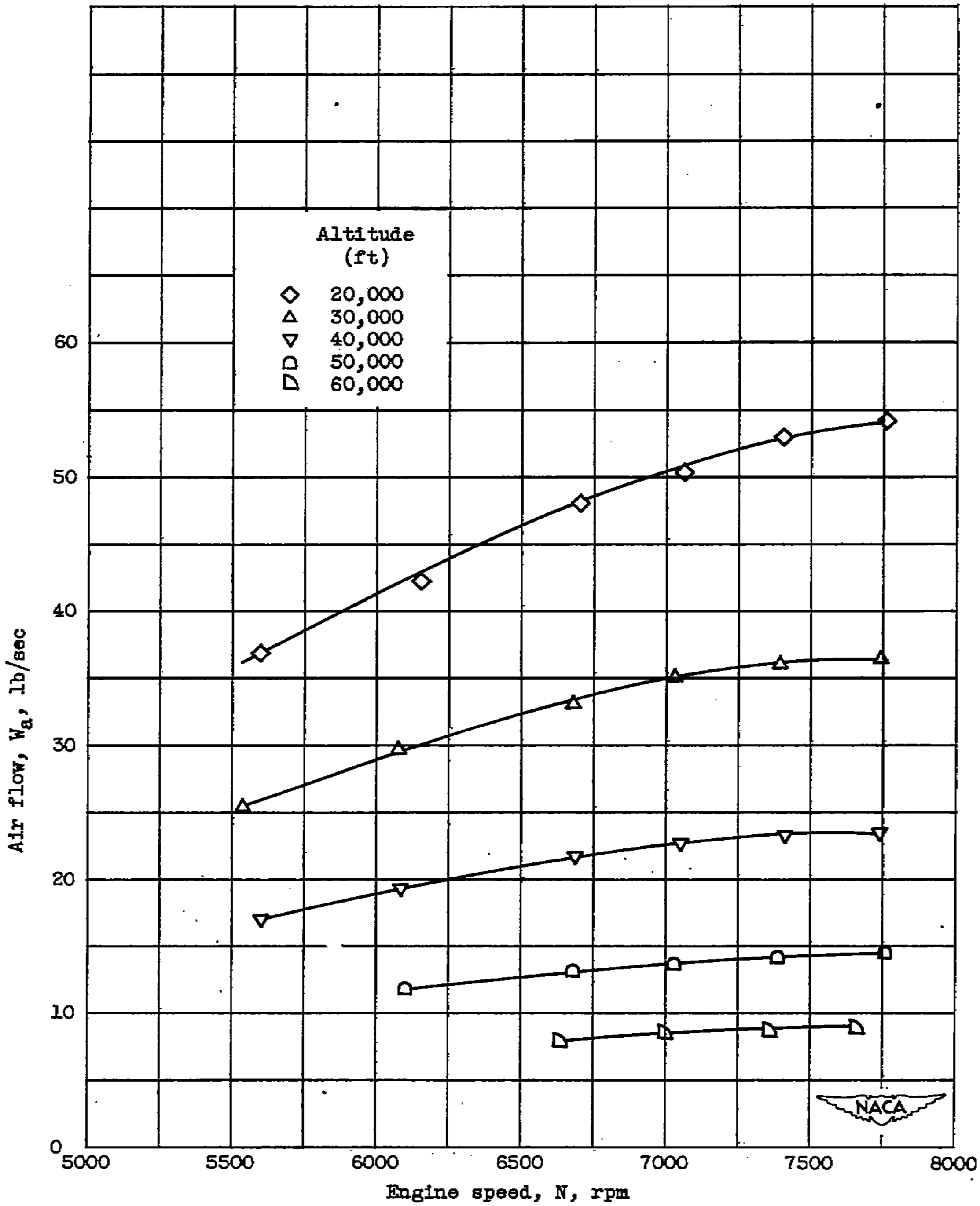
1420



(b) Net thrust.

Figure 6. - Continued. Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

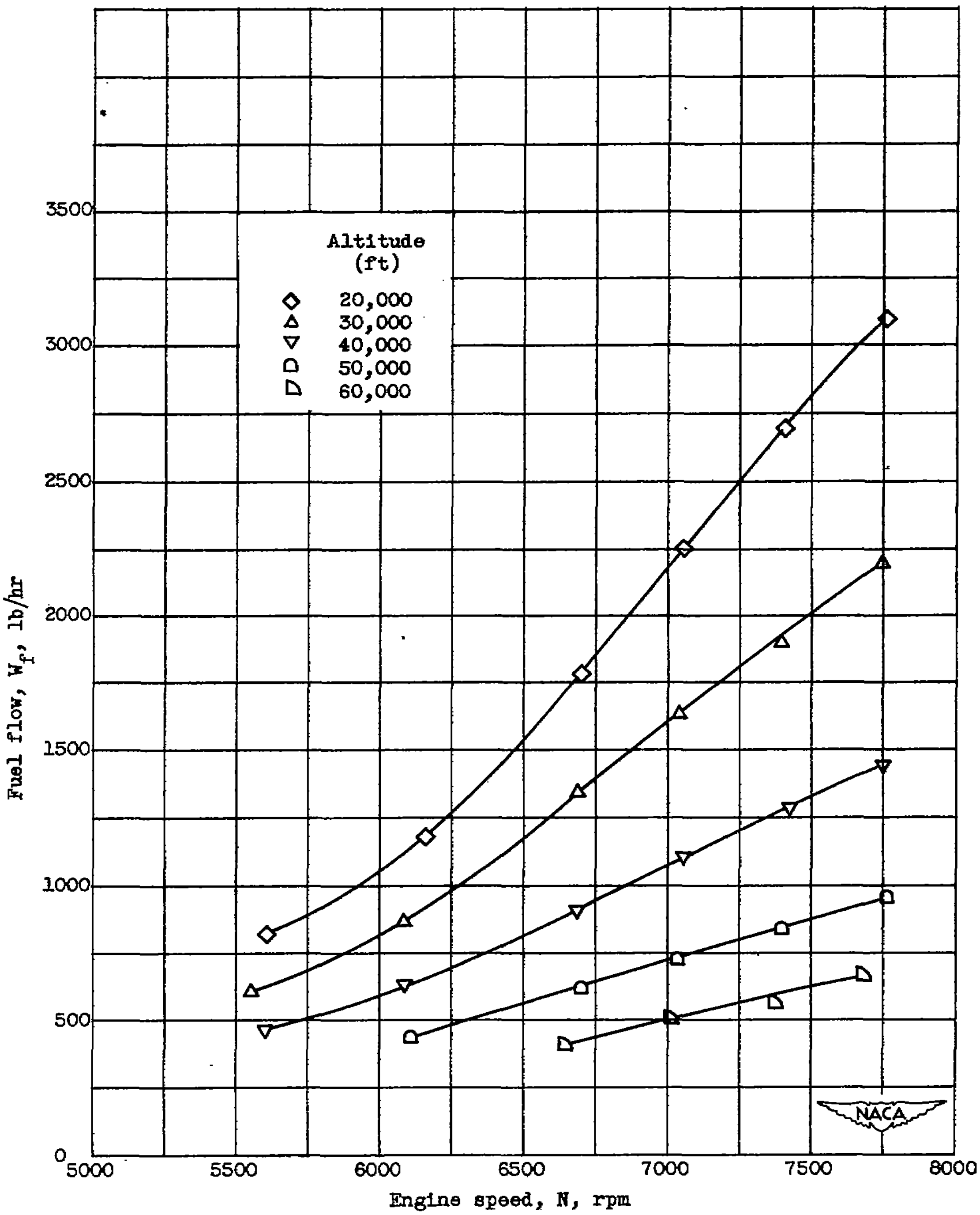
1420



(c) Air flow.

Figure 6. - Continued. Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

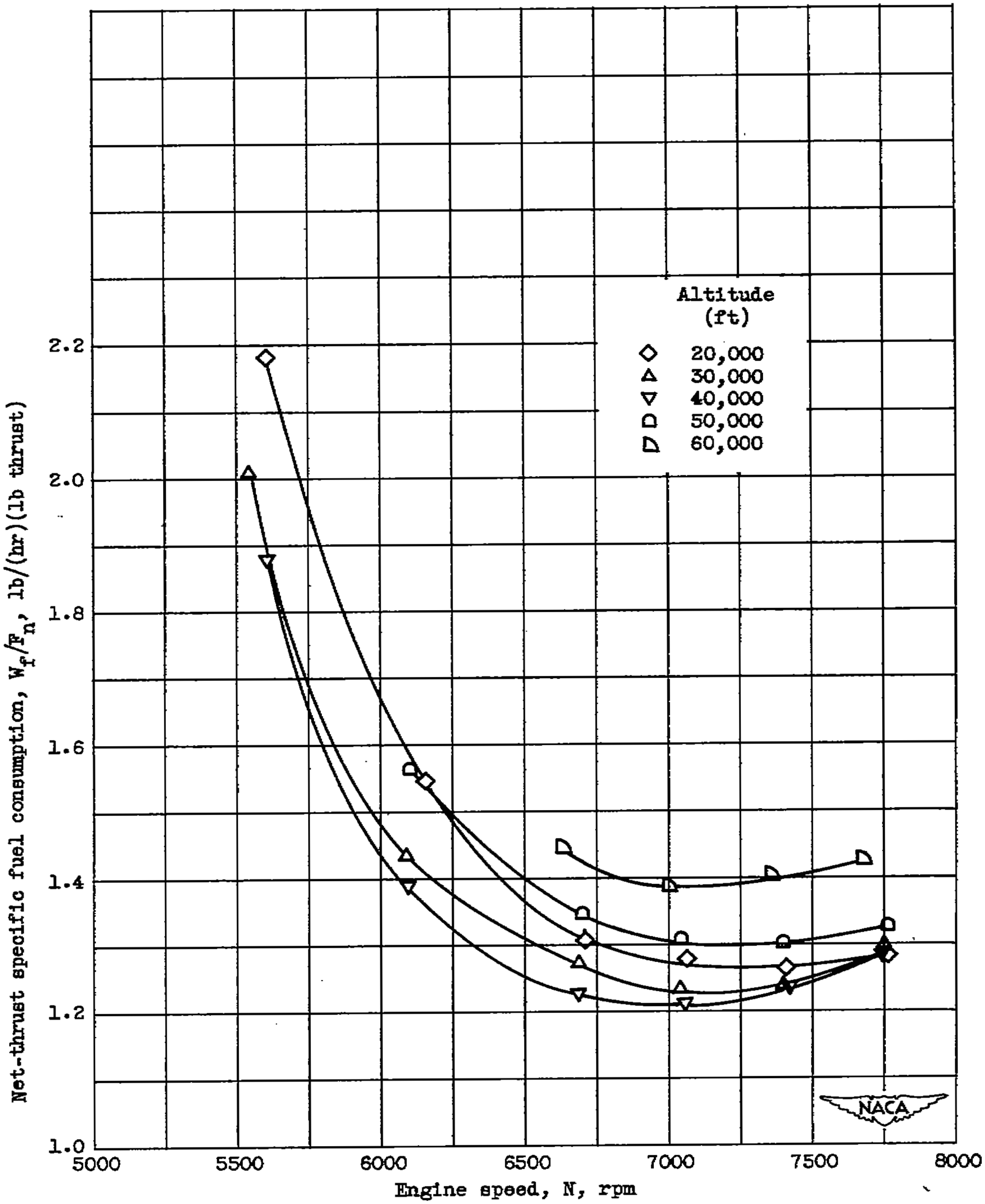
1420



(d) Fuel flow.

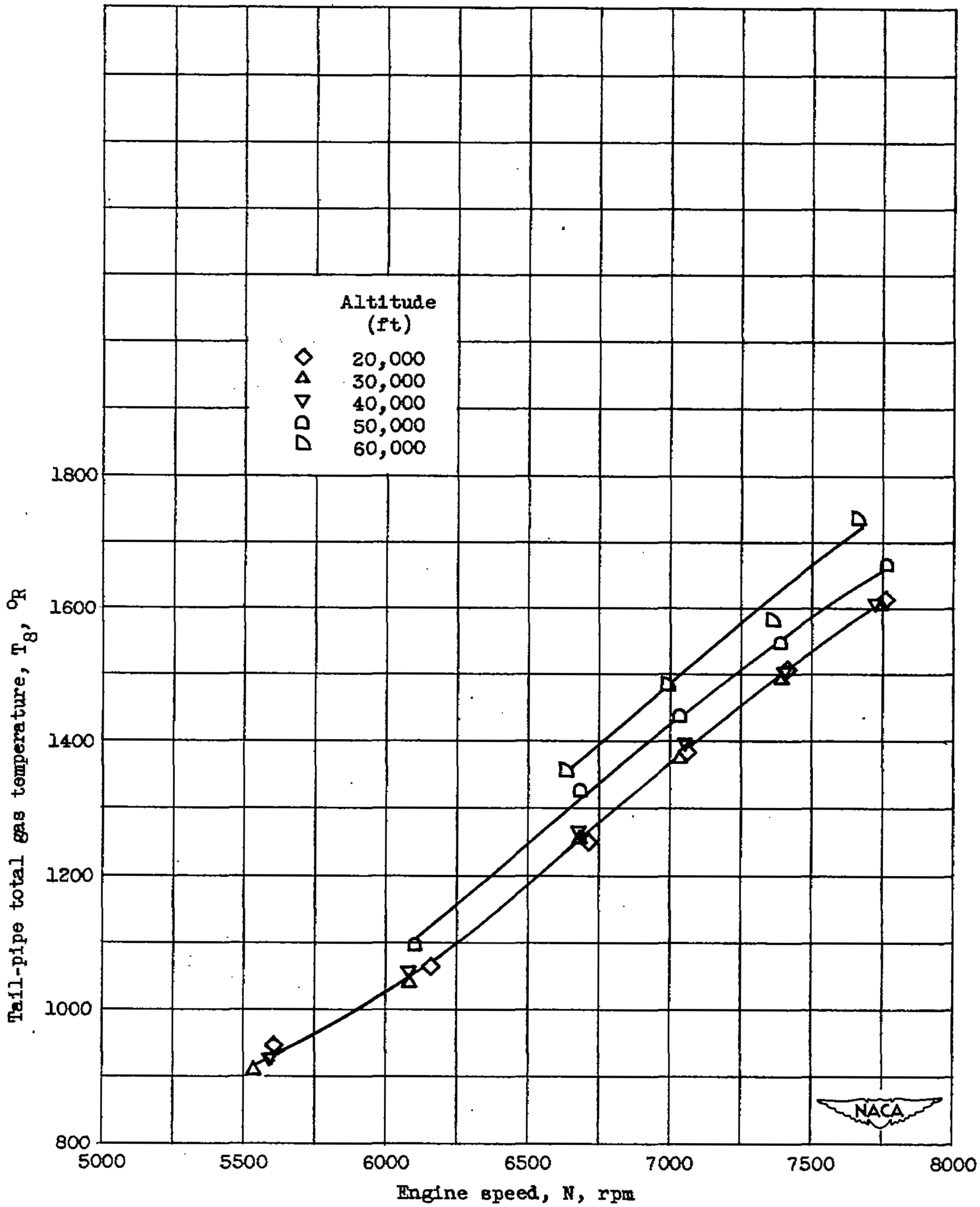
Figure 6. - Continued. Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

1420



(e) Net-thrust specific fuel consumption.

Figure 6. - Continued. Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

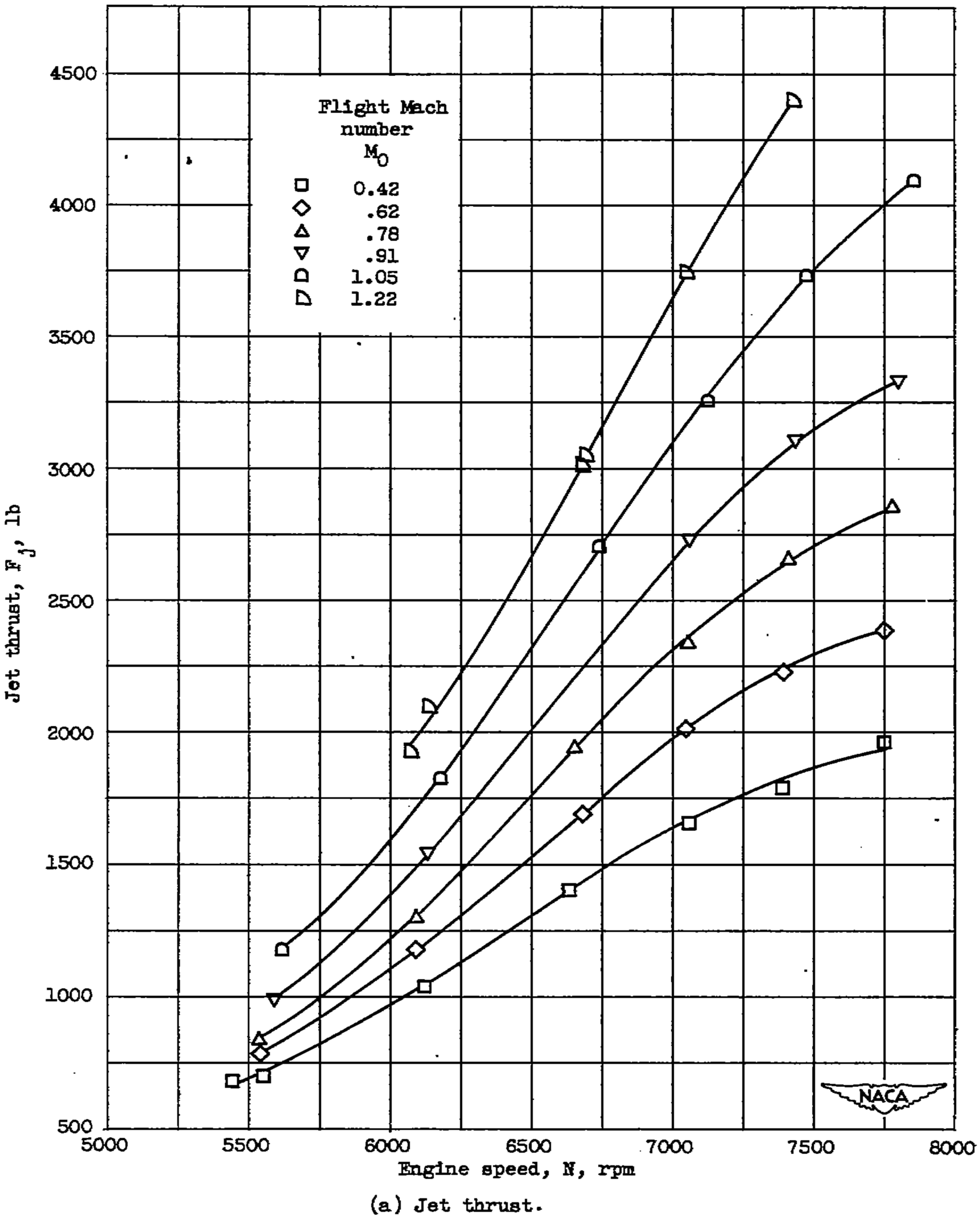


(f) Tail-pipe total gas temperature.

Figure 6. - Concluded. Effect of altitude on engine performance. Flight Mach number M_0, 0.62.

1420

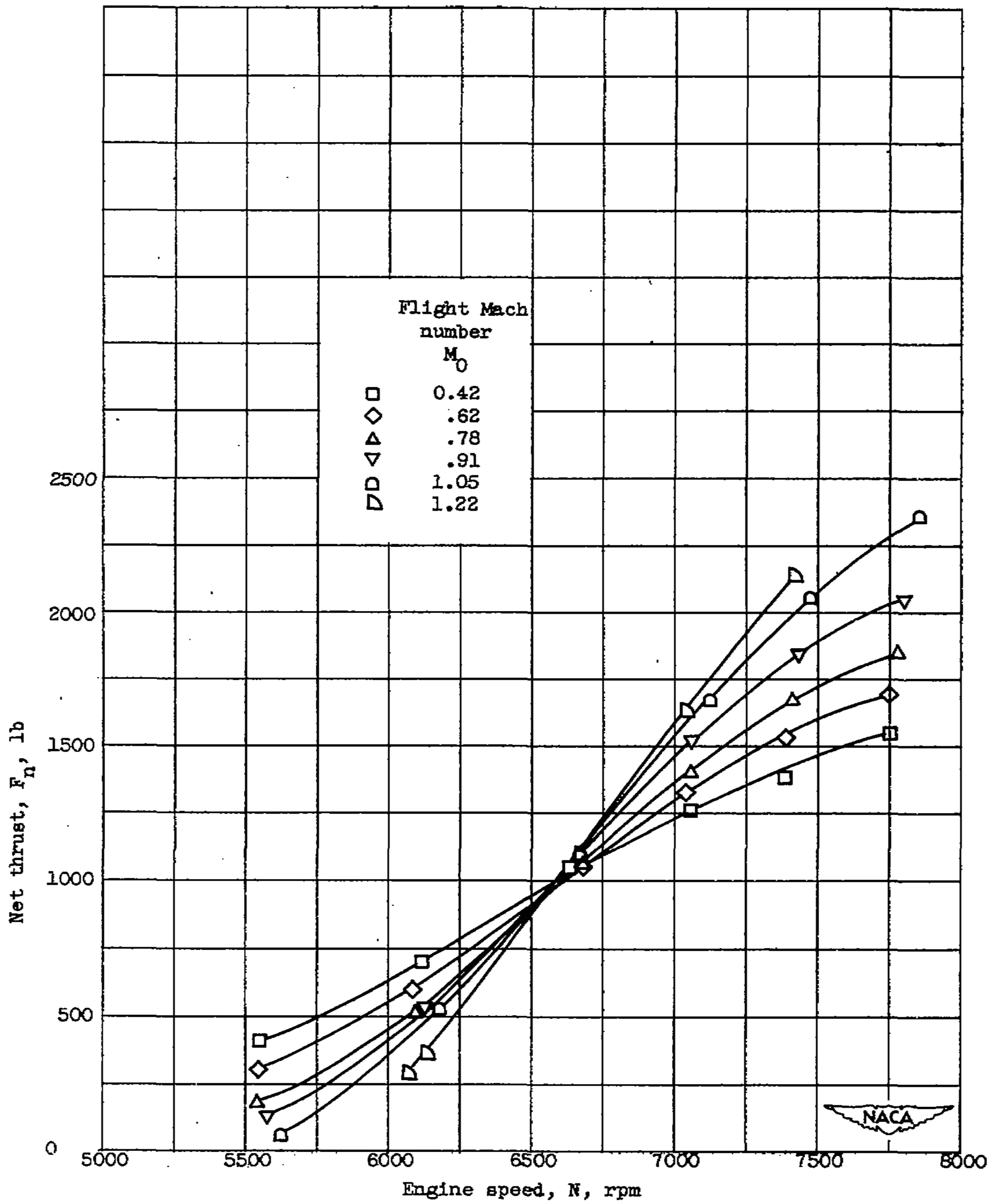
1420



(a) Jet thrust.

Figure 7.- Effect of flight Mach number on engine performance Altitude, 30,000 feet.

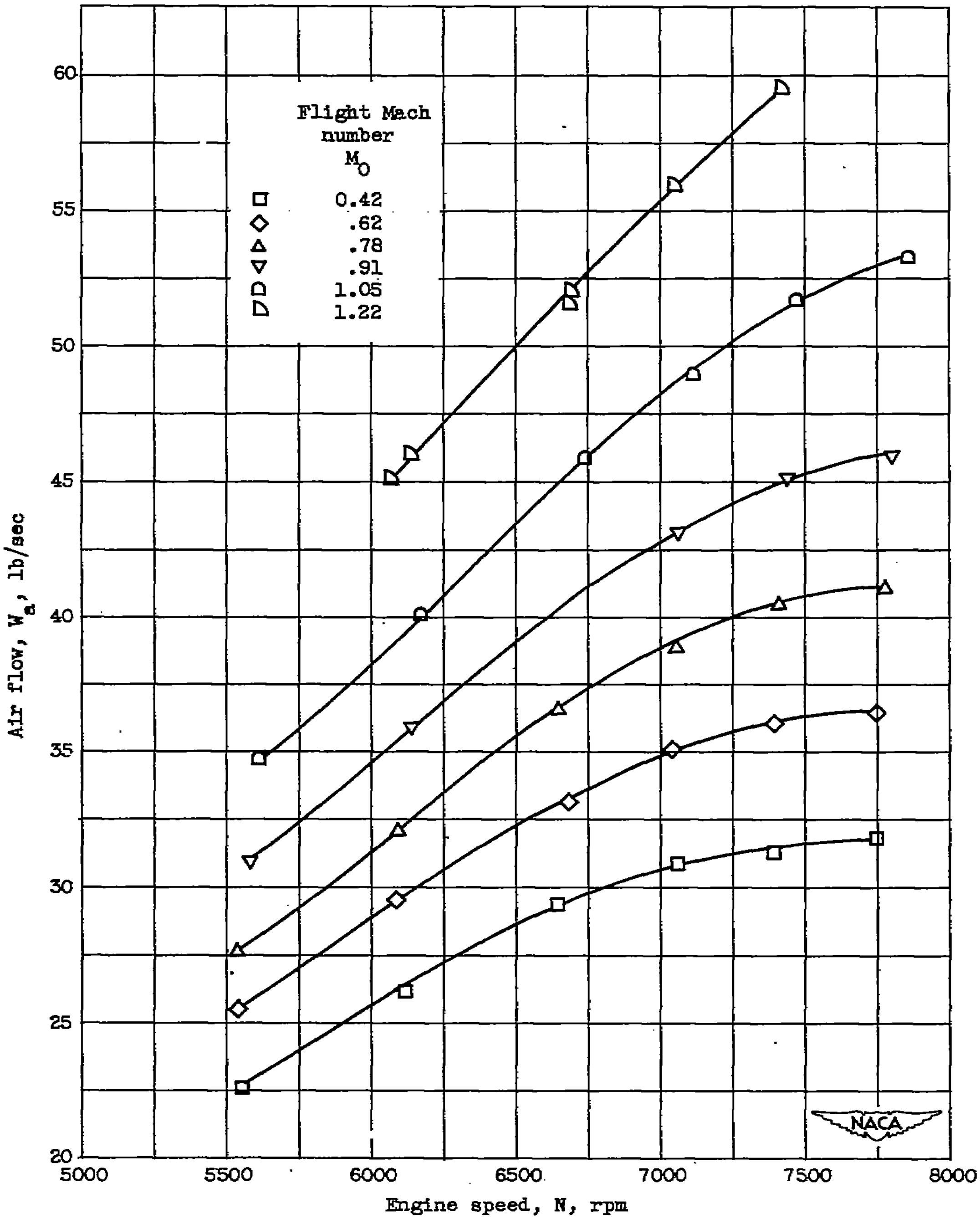
1420



(b) Net thrust.

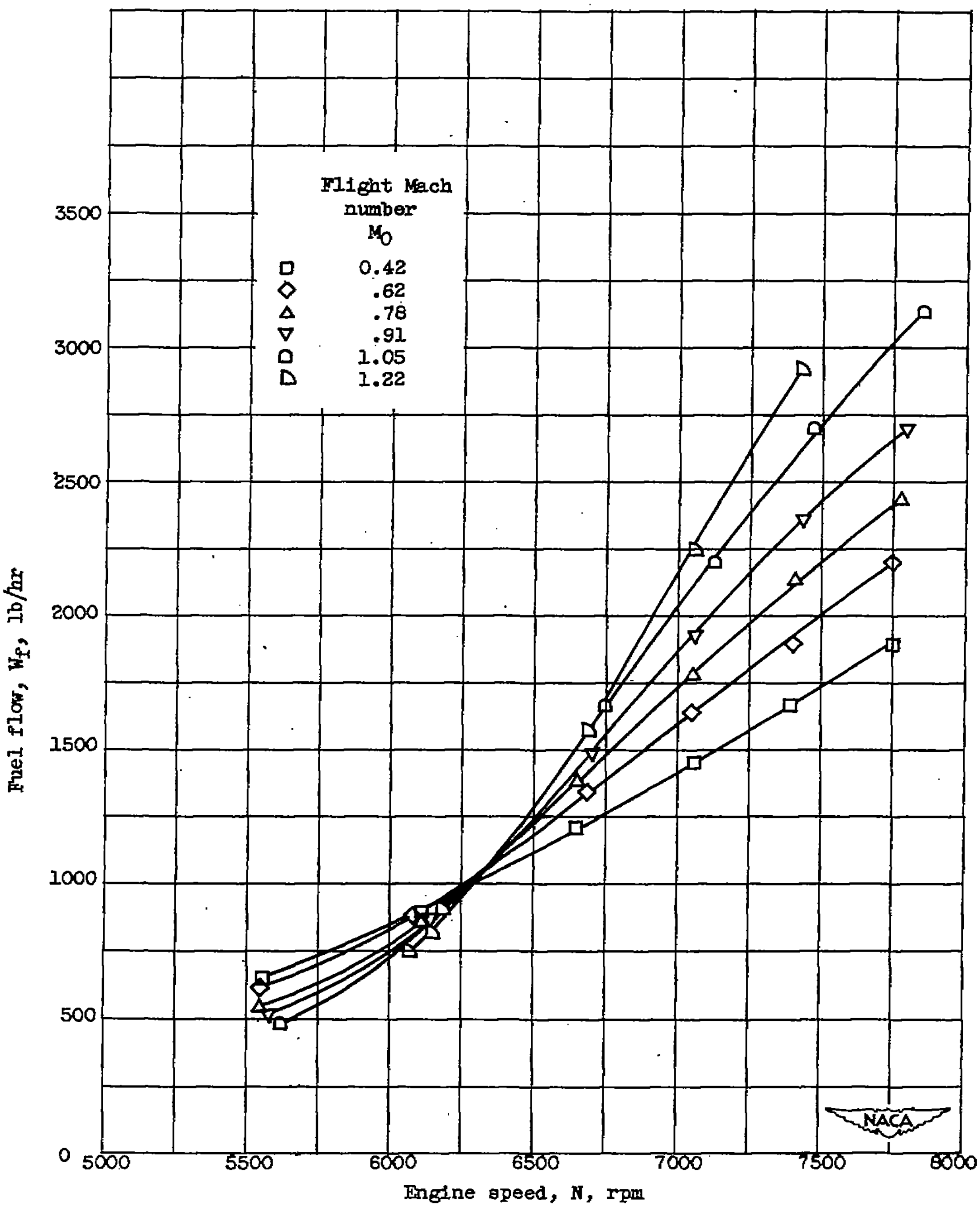
Figure 7. - Continued. Effect of flight Mach number on engine performance. Altitude, 30,000 feet.

1420



(c) Air flow.

Figure 7. - Continued. Effect of flight Mach number on engine performance. Altitude, 30,000 feet.

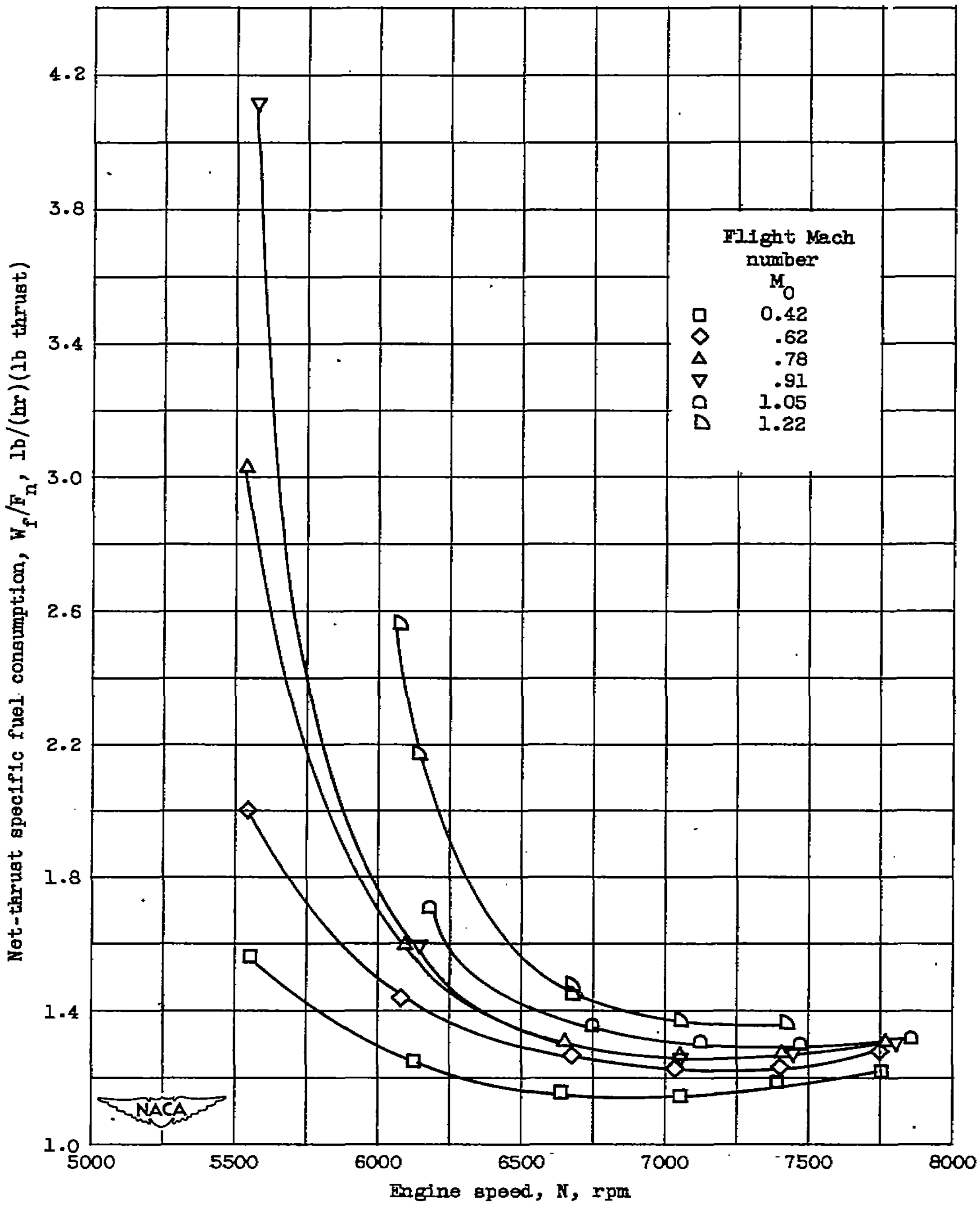


(d) Fuel flow.

Figure 7. - Continued. Effect of flight Mach number on engine performance. Altitude, 30,000 feet.

1420

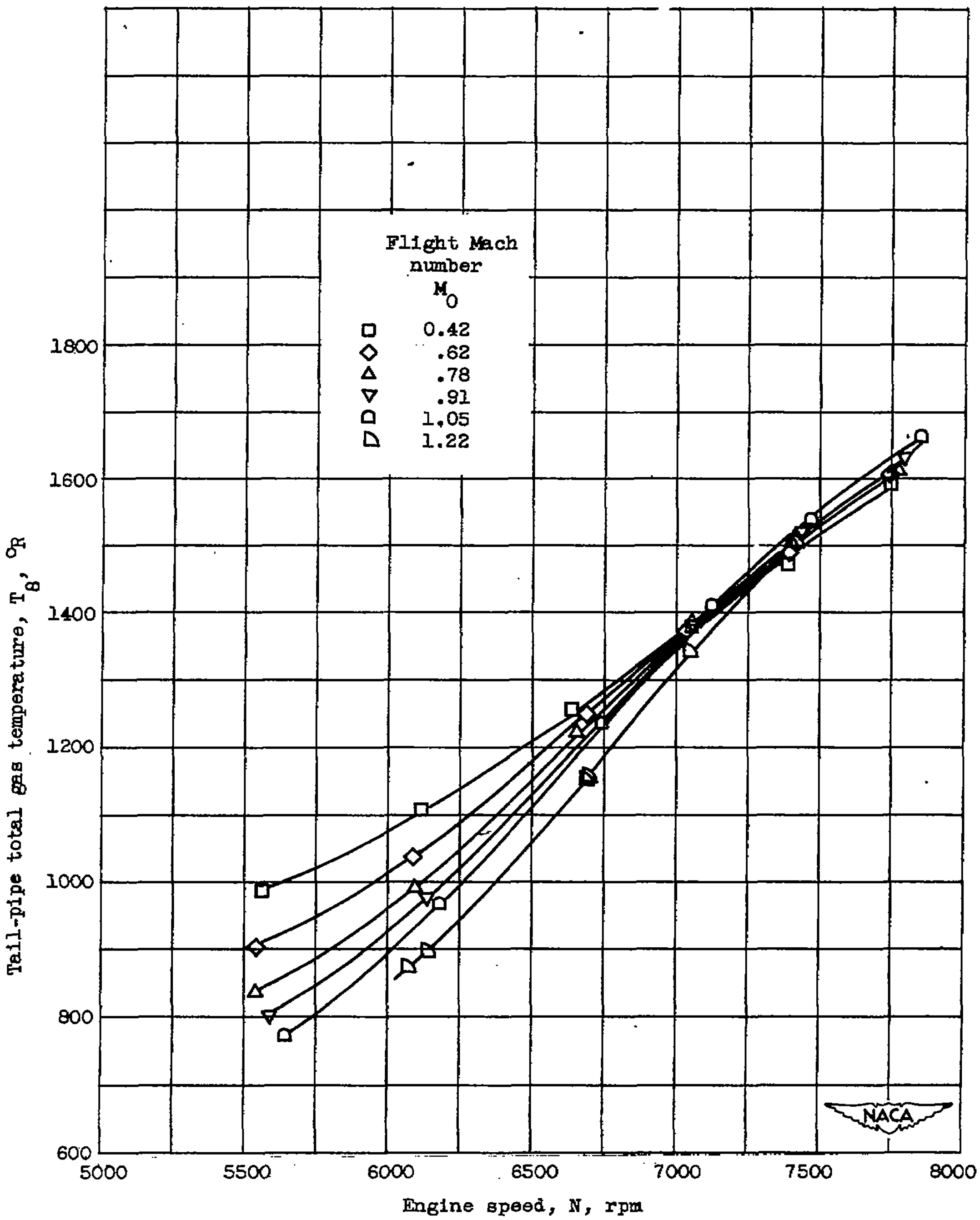
1420



(e) Net-thrust specific fuel consumption.

Figure 7. - Continued. Effect of flight Mach number on engine performance. Altitude, 30,000 feet.

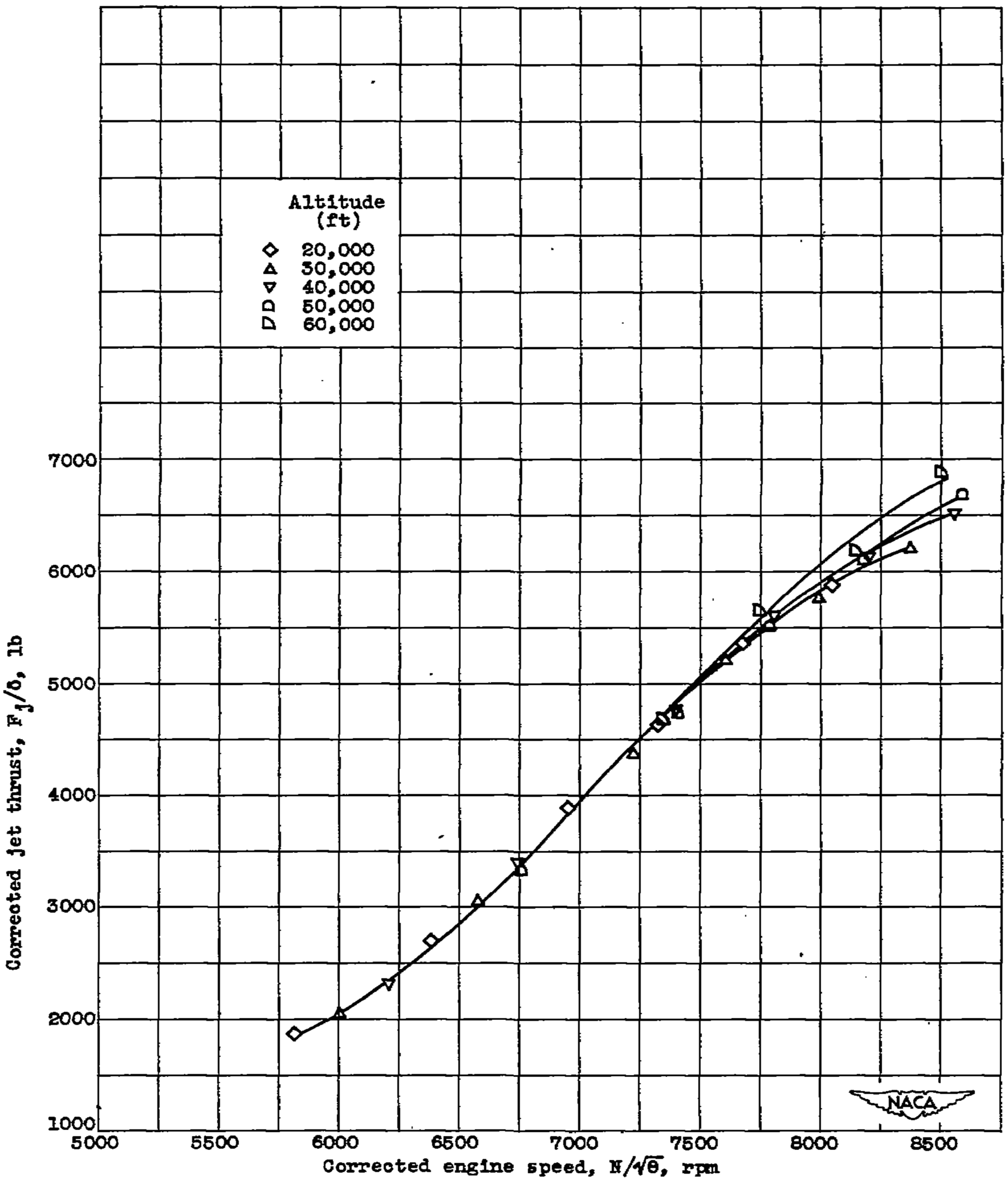
1420



(f) Tail-pipe total gas temperature.

Figure 7. - Concluded. Effect of flight Mach number on engine performance. Altitude, 30,000 feet.

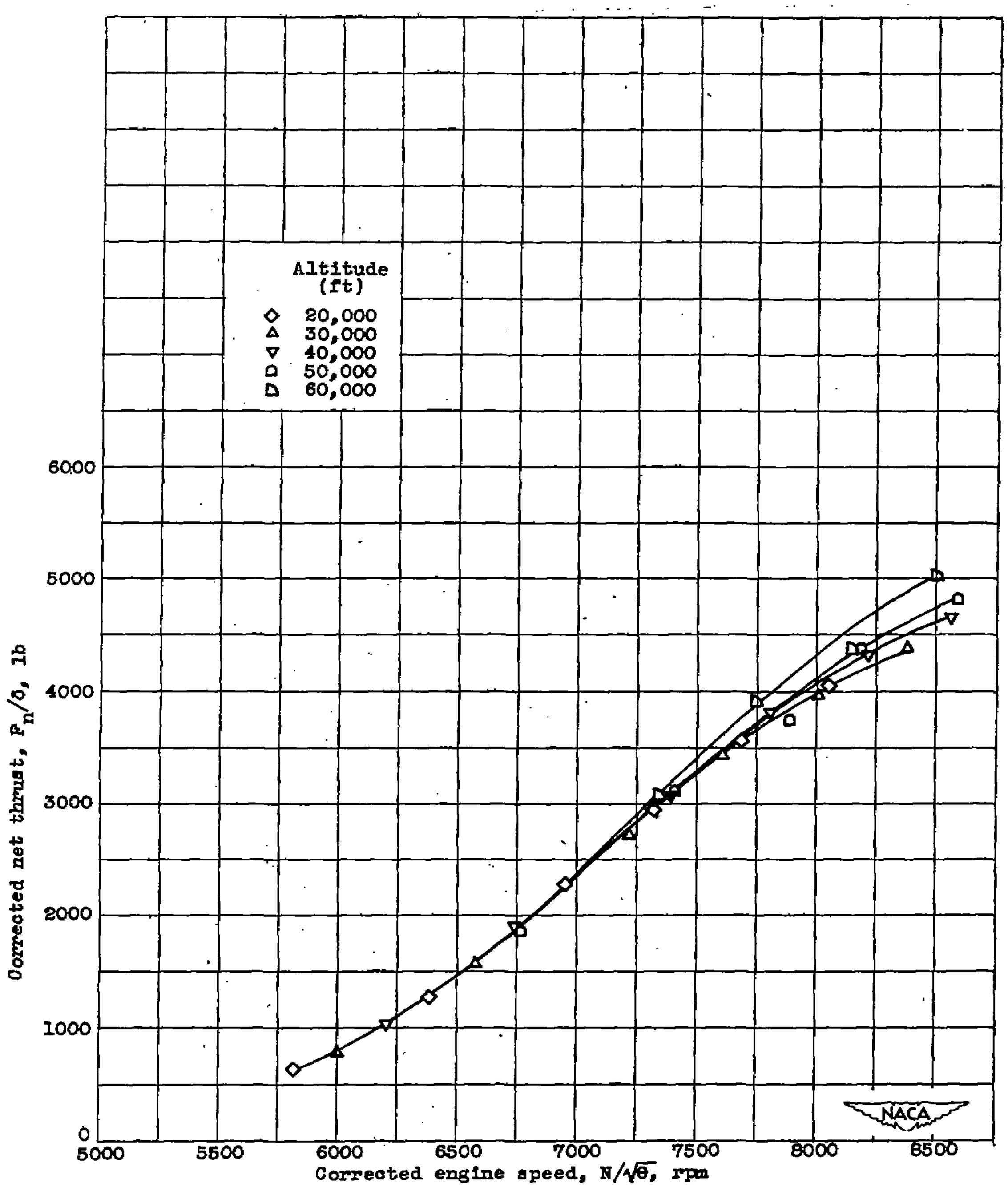
1420



(a) Corrected jet thrust.

Figure 8. - Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

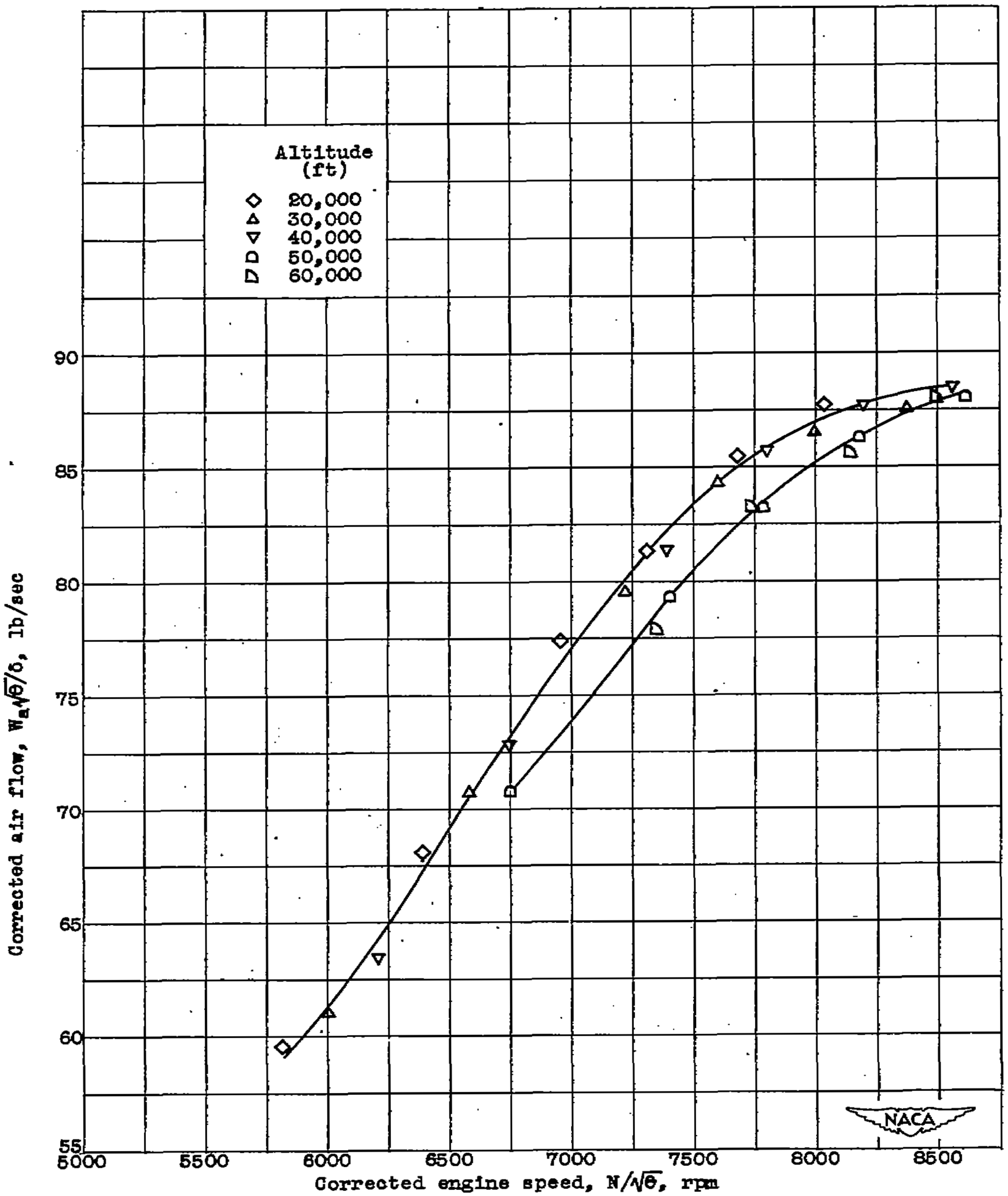
1420



(b) Corrected net thrust.

Figure 8. - Continued. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

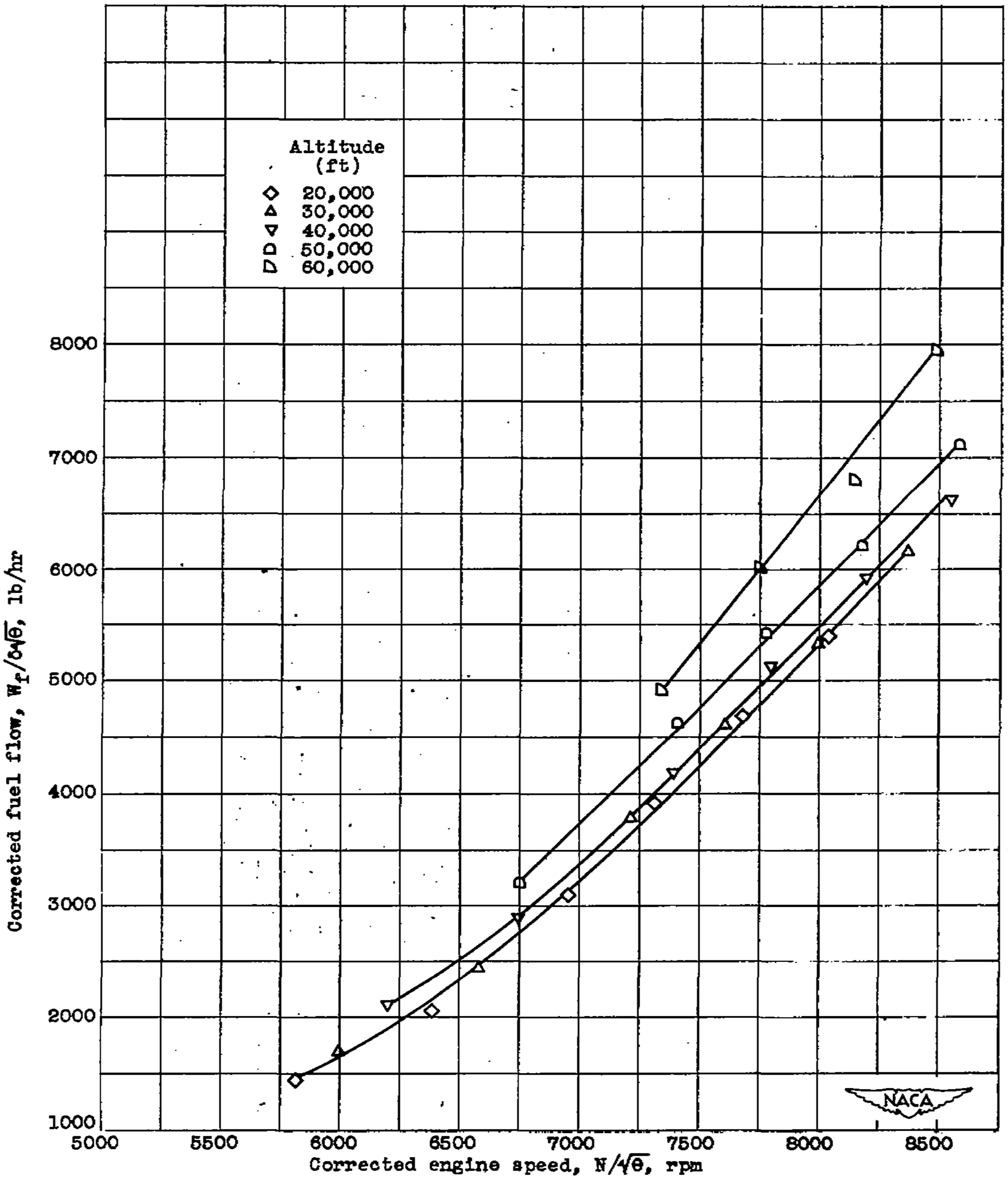
1420



(c) Corrected air flow.

Figure 8. - Continued. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

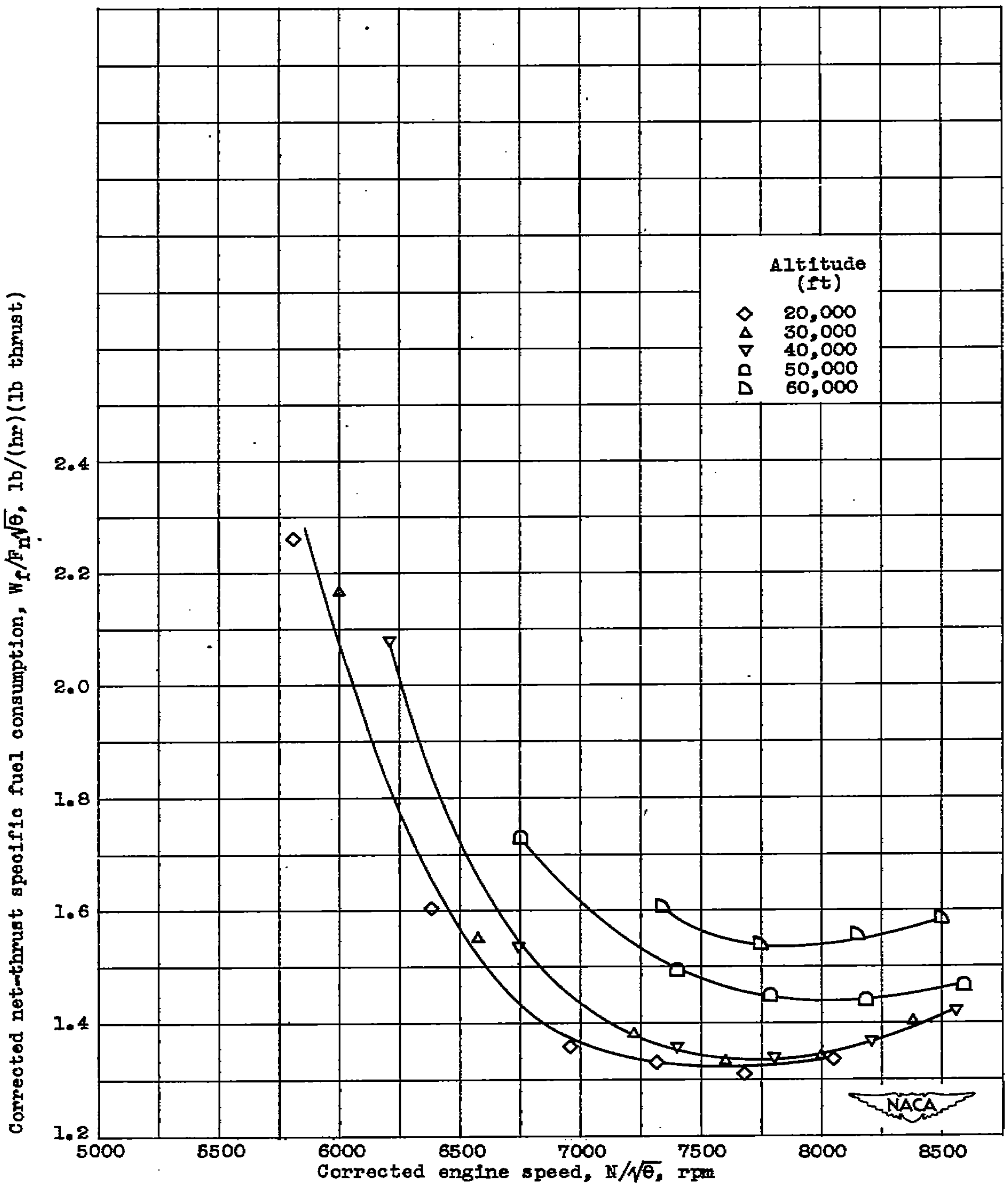
1420



(d) Corrected fuel flow.

Figure 8. - Continued. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

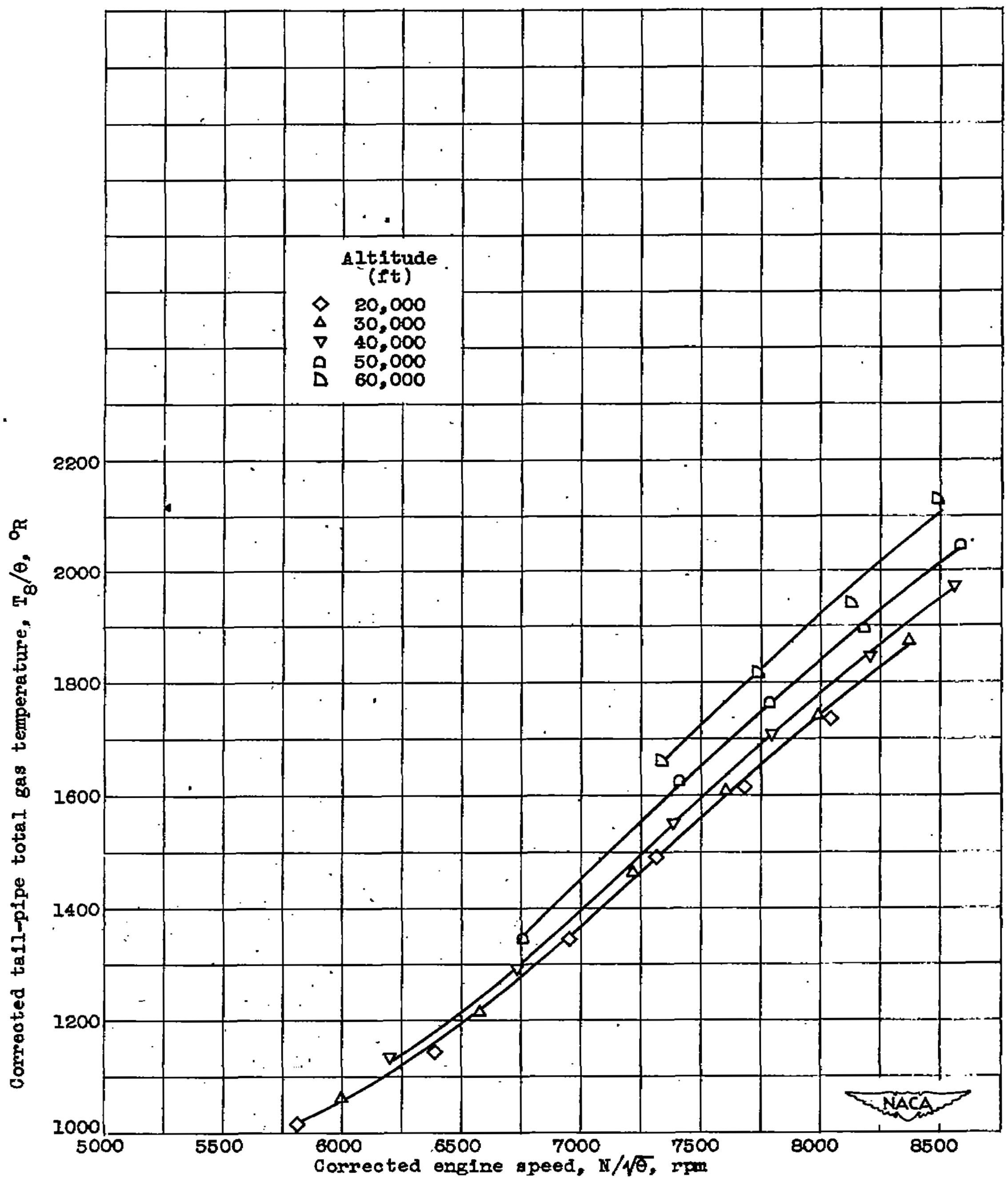
1420



(e) Corrected net-thrust specific fuel consumption.

Figure 8. - Continued. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

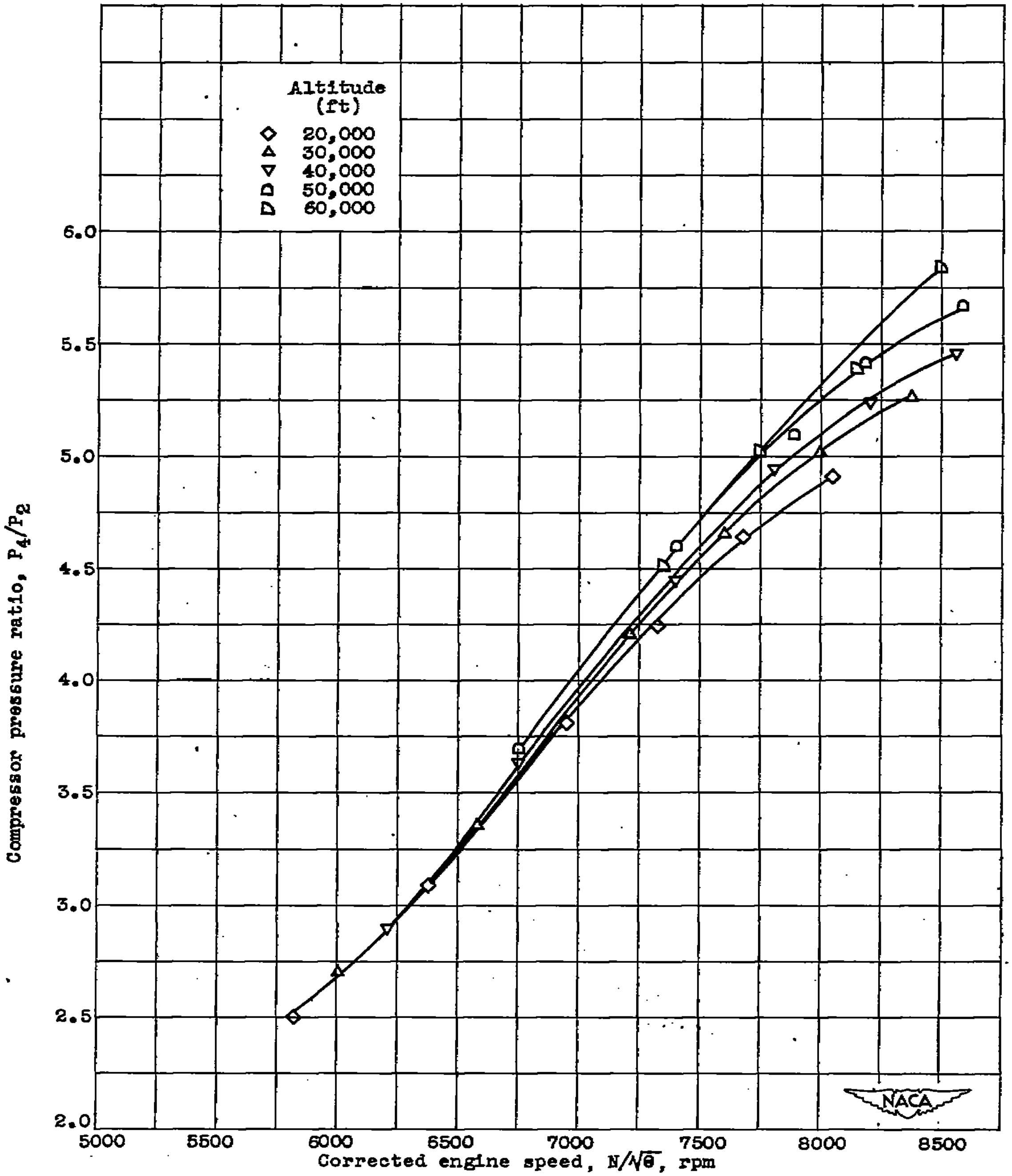
1420



(f) Corrected tail-pipe total gas temperature.

Figure 8. - Continued. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

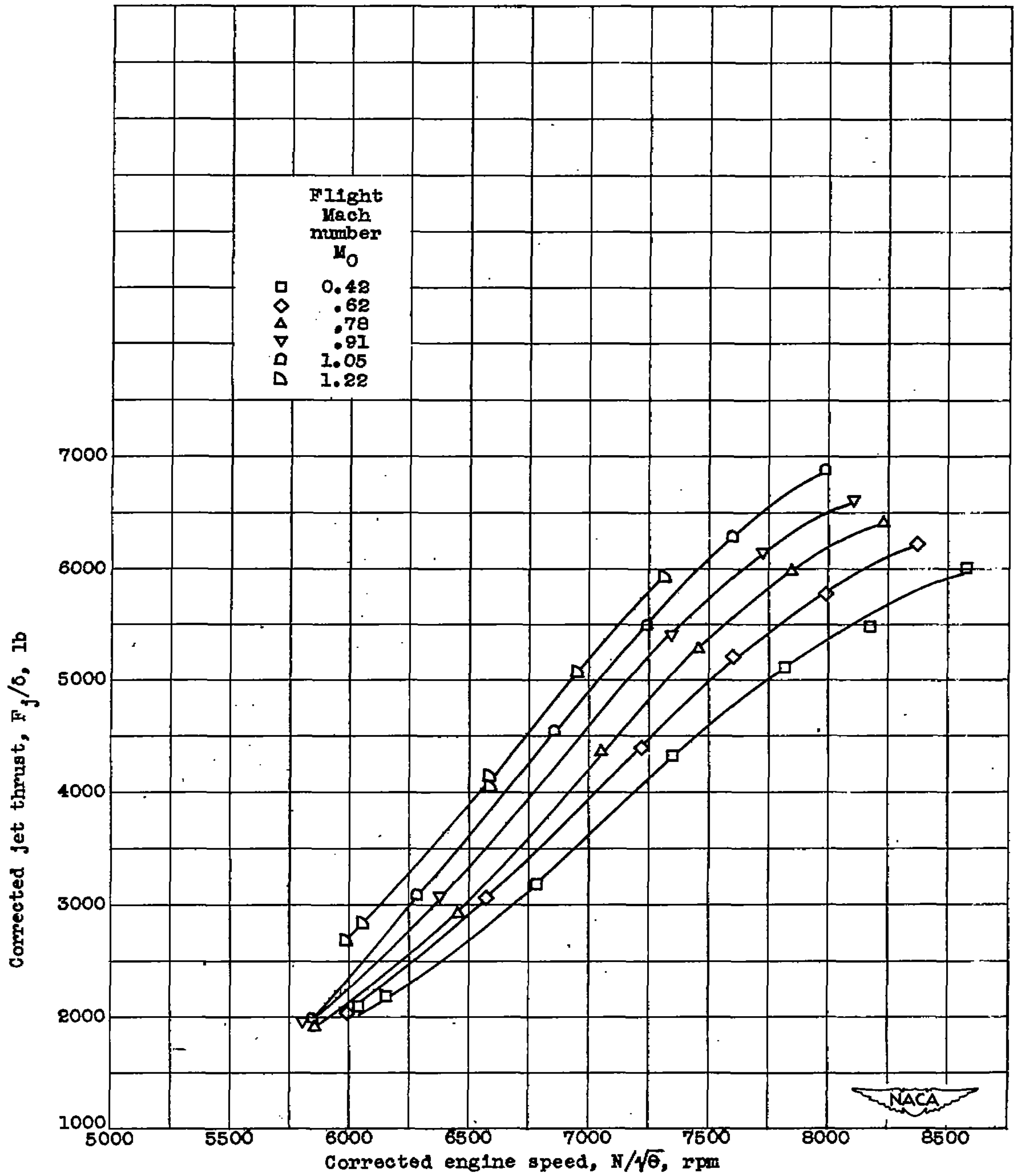
1420



(g) Compressor pressure ratio.

Figure 8. - Concluded. Effect of altitude on corrected engine performance. Flight Mach number M_0, 0.62.

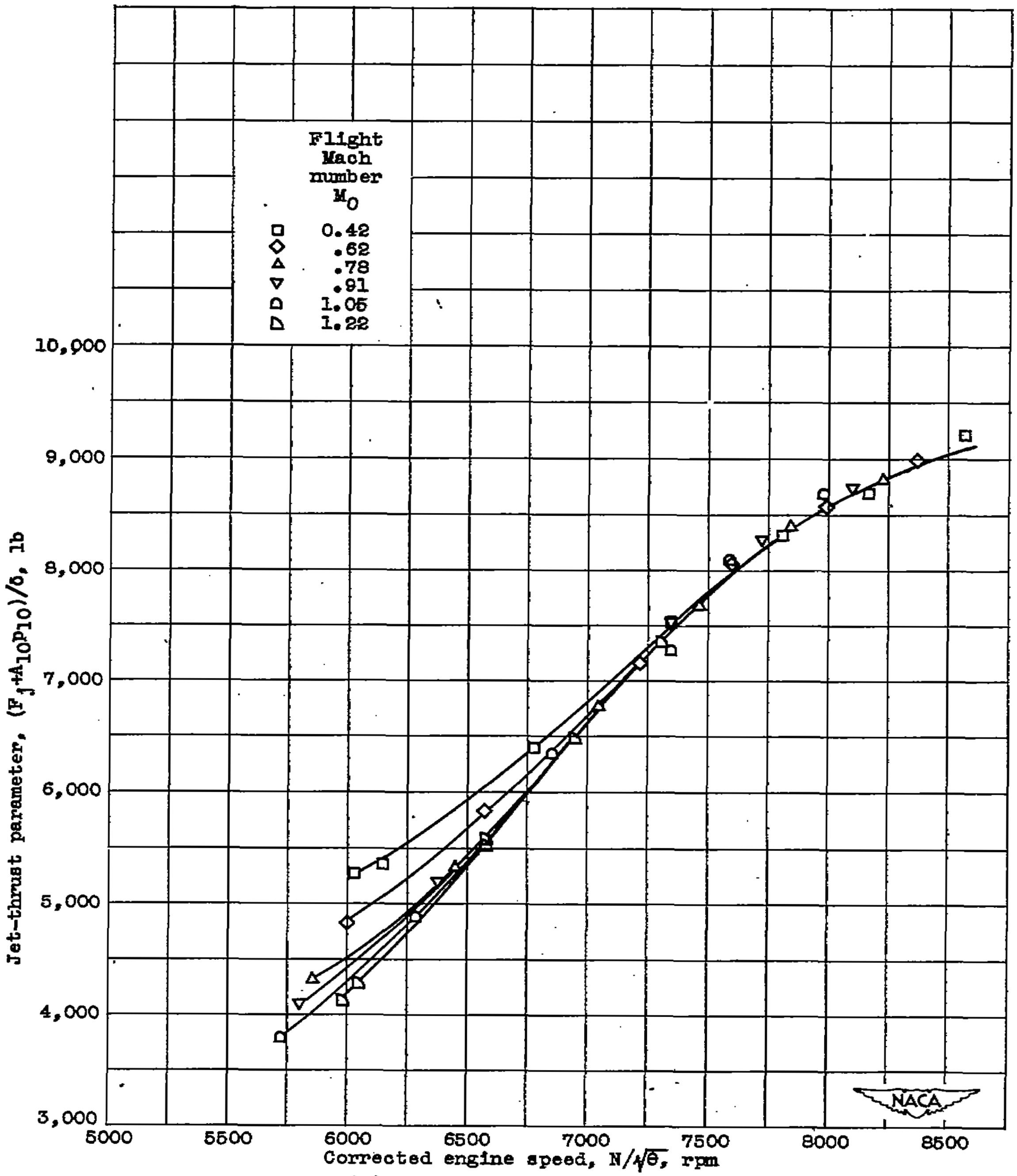
1420



(a) Corrected jet thrust.

Figure 9. - Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.

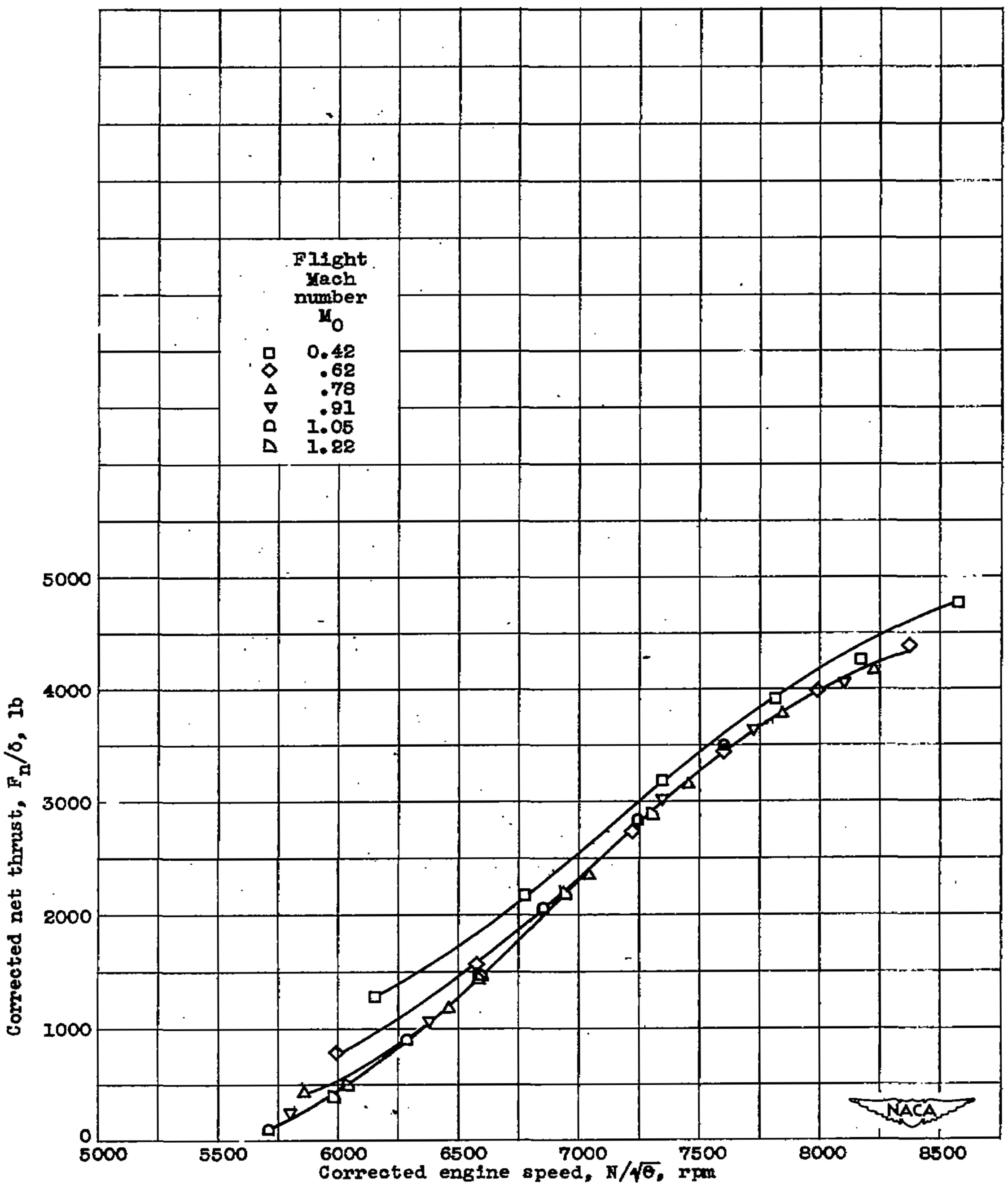
1420



(b) Jet-thrust parameter.

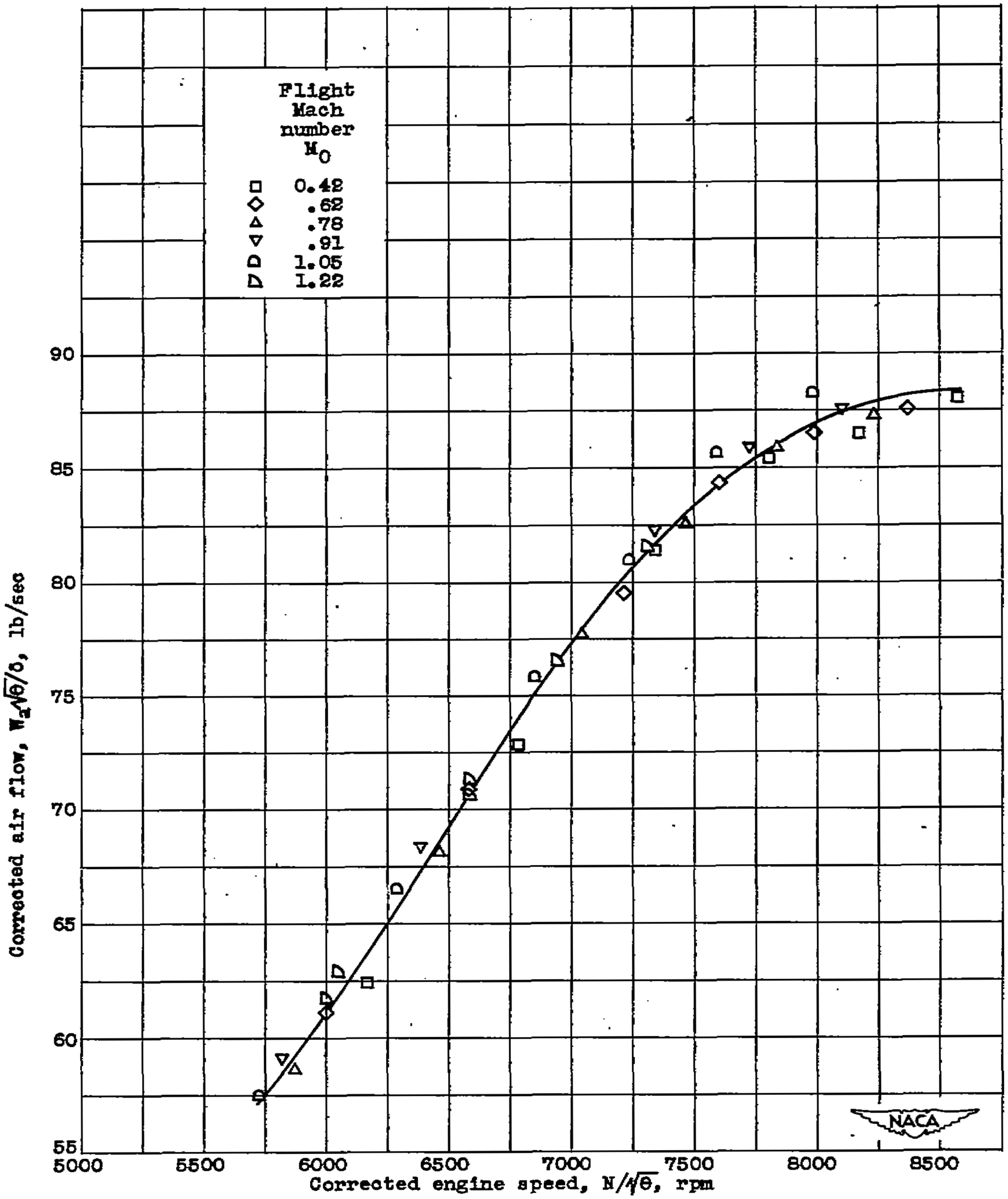
Figure 9. - Continued. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.

1420



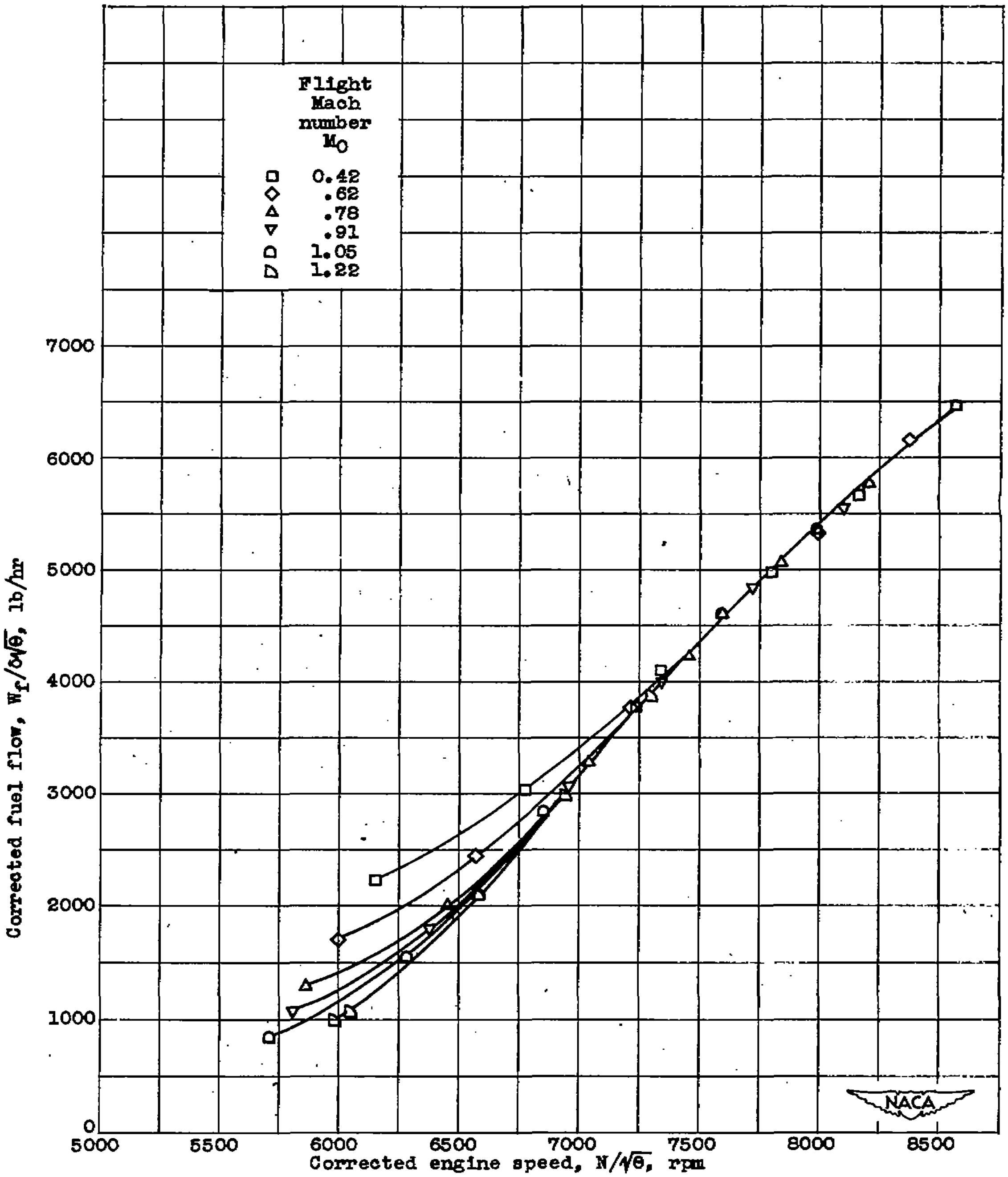
(c) Corrected net thrust.

Figure 9. - Continued. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.



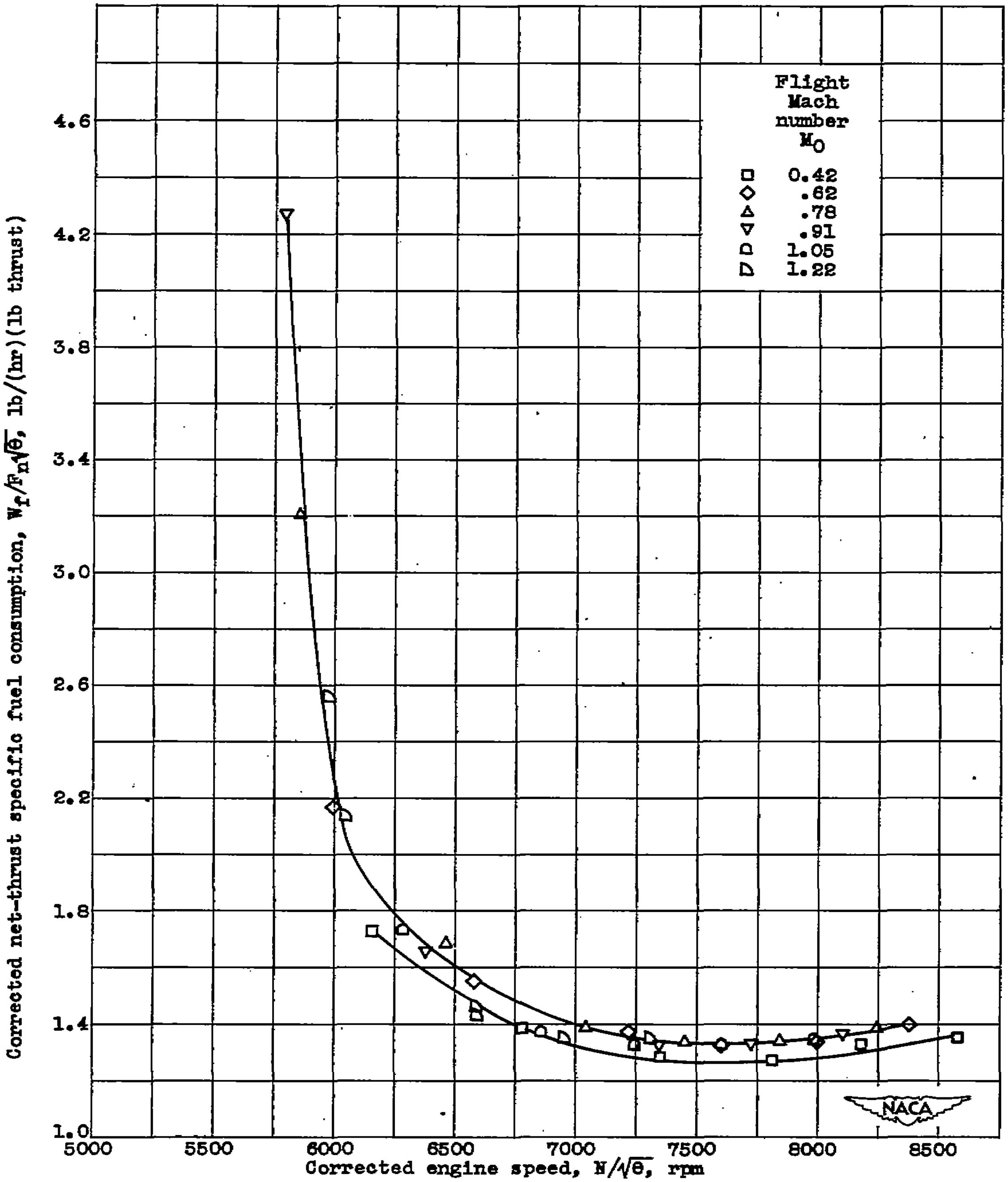
(d) Corrected air flow.

Figure 9. - Continued. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.



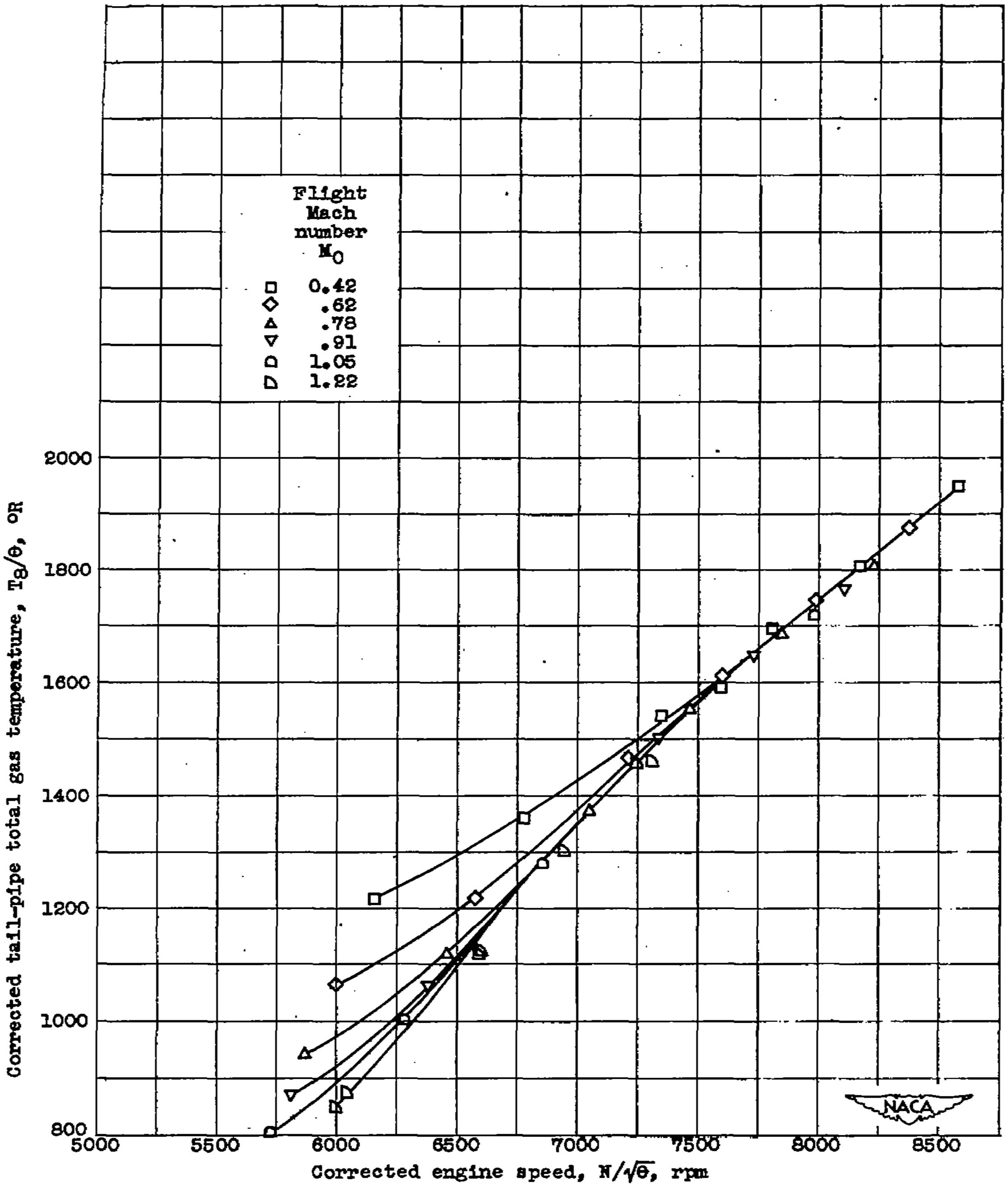
(e) Corrected fuel flow.

Figure 9. - Continued. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.



(f) Corrected net-thrust specific fuel consumption.

Figure 9. - Continued. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.



(g) Corrected tail-pipe total gas temperature.

Figure 9. - Concluded. Effect of flight Mach number on corrected engine performance. Altitude, 30,000 feet.

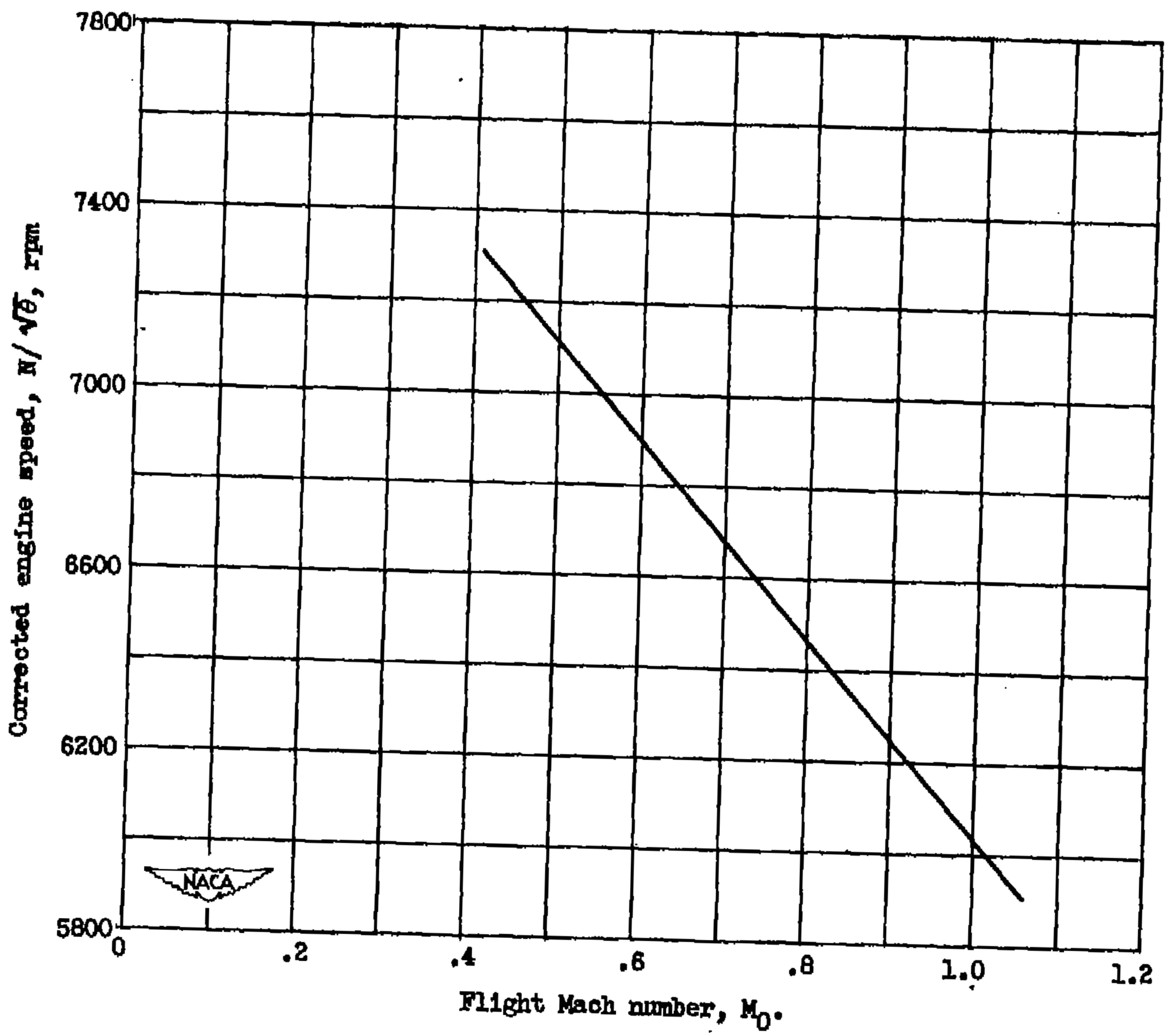


Figure 10. - Corrected engine speed for choking in jet nozzle as function of flight Mach number.