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RESEARCH MEMORANDUM

PERFORMANCE OF A TURBOJET ENGINE WITH ADJUSTABLE FIRST-STAGE TURBINE STATOR AND VARIABLE-AREA EXHAUST NOZZLE

By Carl L. Meyer, Ivan D. Smith, and Harry E. Bloomer

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

The performance of a turbojet engine with a two-stage turbine, an adjustable first-stage turbine stator, and a variable-area exhaust nozzle was investigated at selected constant engine speeds and two simulated flight conditions; various fixed settings of the adjustable turbine stator were used. For the particular component characteristics of the engine investigated, little improvement in thrust or specific fuel consumption could be realized at conditions from 75 percent of normal to military thrust by use of an adjustable rather than an optimum fixed first-stage turbine stator.

INTRODUCTION

As part of an experimental evaluation of a full-scale turbojet engine with a two-stage turbine in the NACA Lewis altitude wind tunnel, data were obtained to determine the over-all and component performance of the engine when equipped with both an adjustable first-stage turbine stator and a variable-area exhaust nozzle. The performance of the turbine and a discussion of the design and mechanical reliability of the adjustable first-stage turbine stator are presented in reference 1. The performance of the compressor and over-all engine are presented herein.

For the engine equipped with the adjustable first-stage turbine stator and variable-area exhaust nozzle, it was possible to control the matching between the compressor and the turbine; thus, compressor pressure ratio could be varied independently of turbine-inlet temperature and engine speed within a range limited by the flow-area variation of the adjustable turbine stator, the maximum turbine-inlet temperature, the compressor surge pressure ratio, or the area variation of the exhaust nozzle. Through use of various fixed positions of the adjustable first-stage turbine stator, data were obtained to enable selection of an optimum fixed-stator flow area for the particular engine and to determine whether or not there are performance advantages to be gained by use of adjustable as compared with fixed turbine stators in the given engine. The analysis of reference 2 indicates possible improvements in specific fuel consumption at less than maximum thrust through use of adjustable, rather than fixed, turbine stators in a turbojet engine equipped with a variable-area exhaust nozzle.

At a given simulated flight condition and at various fixed positions of the first-stage turbine stator, data were obtained over approximately the available range of turbine-inlet temperatures at each of various constant engine speeds by varying the exhaust-nozzle area. At simulated conditions corresponding to a flight Mach number of 0.62 at an altitude of 30,000 feet and a flight Mach number of 0.46 at an altitude of 15,000 feet, data were obtained for a range of first-stage turbine-stator positions which correspond to a range of effective stator flow areas from 1.13 to 1.25 square feet.

Compressor performance maps are presented herein for the simulated flight conditions investigated. Composite performance maps are presented for selected engine speeds at the two simulated flight conditions to show engine performance in terms of net thrust, specific fuel consumption, turbine-inlet to engine-inlet temperature ratio, and compressor pressure ratio for the family of first-stage turbine-stator positions investigated. The results of the present investigation are limited, of course, by the component characteristics of the engine.

The adjustable first-stage turbine stator used in the present investigation was not designed as a standard component of the engine but was intended as a means of regulating the engine operating point for other component investigations such as that of compressor surge reported in references 3 and 4; therefore, high performance of the turbine was not a primary consideration in the stator design. In addition, the adjustable stator was the first-stage stator of a two-stage turbine; the turbine rotors and second-stage stator were designed for a fixed-position first-stage stator.

APPARATUS

Engine

A prototype J40-WE-6 turbojet engine was used for the present investigation. Main components of the engine include an 11-stage axial-flow compressor, an annular combustor, a two-stage turbine, an exhaust collector, and a continuously variable clam-shell-type exhaust nozzle. For the present investigation, a mixer-vane assembly was included at the compressor outlet to alleviate turbine-inlet temperature distribution problems; the electronic control was modified to permit independent control of engine speed and exhaust-nozzle area; and the fixed-position first-stage turbine stator was replaced by an adjustable stator.

Approximate sea-level thrust ratings of the J40-WE-6 turbojet engine are as follows:

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Operating condition	Thrust, lb	Engine speed, rpm	Turbine-inlet gas temperature, °F
Take-off and military	7500	7260	1425
Normal	6800	7260	----
90 percent normal	6120	7260	----
75 percent normal	5100	7050	----

Adjustable First-Stage Turbine Stator

The adjustable first-stage turbine stator, which was supplied by the engine manufacturer, is illustrated schematically in figure 1. The stator blades were mounted on shafts and could be turned simultaneously between the inner and outer shrouds through the illustrated actuating mechanism by an externally mounted worm-gear drive. Adjustment of the stator setting varied the flow area of the stator and also the angle through which the gases were turned in passing through the stator. A more detailed description of the turbine and the adjustable stator is given in reference 1. Independent control of the adjustable stator was used in the present investigation to select various fixed stator settings for which the range of effective flow area was from 1.13 to 1.25 square feet. The method of determining effective stator flow area is given in reference 1.

INSTALLATION AND INSTRUMENTATION

The engine was installed on a wing segment that was supported in the 20-foot-diameter test section of the altitude wind tunnel by the tunnel balance frame. Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct connected to the engine inlet. Air flow through the duct was throttled from approximately sea-level pressure to a total pressure at the engine inlet corresponding to the desired flight Mach number and altitude, while the tunnel test-section static pressure was maintained at that corresponding to the desired altitude. Thrust and drag measurements with the tunnel balance scales were made possible by a slip joint with a frictionless seal located in the duct upstream of the engine.

Conventional instrumentation for the measurement of temperatures and pressures was installed at various stations in the engine (fig. 2). Pressures in the inlet-air duct ahead of the engine (station 1) and at the engine inlet (station 2) were measured with water-filled manometers, and those at the compressor outlet (station 4), turbine inlet (station 5), and turbine outlet (station 6) were measured with mercury-filled manometers; all pressures were photographically recorded. Temperatures in the duct ahead of the engine and at the compressor outlet were measured with iron-constantan thermocouples, and those at the turbine-outlet were measured with chromel-alumel thermocouples. All temperatures were automatically recorded by self-balancing potentiometers. Turbine-inlet temperatures were calculated from the turbine-outlet temperatures, with the assumption that the enthalpy drop across the turbine was equal to the enthalpy rise across the compressor. Fuel flow was measured by means of calibrated rotameters, engine speed by means of a stroboscopic tachometer in conjunction with a continuously indicating tachometer, and thrust by means of the tunnel balance scales.

PROCEDURE

Engine speed, exhaust-nozzle area, and first-stage turbine-stator position were independently controlled throughout the investigation. At a given simulated flight condition and at various fixed settings of the adjustable first-stage turbine stator, data were obtained over approximately the available range of turbine-inlet temperatures at each of various constant engine speeds by varying exhaust-nozzle area. Maximum turbine-inlet temperature was limited to 1425° F or to the maximum obtainable without encountering compressor surge. Minimum turbine-inlet temperature was limited by the maximum exhaust-nozzle area.

Data are reported herein for simulated conditions corresponding to a flight Mach number of 0.62 at an altitude of 30,000 feet and a flight Mach number of 0.46 at an altitude of 15,000 feet. Data were obtained with five fixed positions of the first-stage turbine stator at the higher altitude condition and with three at the lower altitude condition; at both flight conditions, the range of stator positions used corresponded to a range of effective stator flow areas from 1.13 to 1.25 square feet. At each turbine-stator position, data were obtained at constant engine speeds within the range of 4720 to 7260 rpm.

RESULTS AND DISCUSSION

Compressor Performance

Compressor performance maps for simulated conditions corresponding to a flight Mach number of 0.62 at an altitude of 30,000 feet and a flight Mach number of 0.46 at an altitude of 15,000 feet are presented in figure 3. On coordinates of compressor pressure ratio and corrected air flow are shown lines of constant corrected engine speed and compressor efficiency. The approximate compressor surge limit is shown for a flight Mach number of 0.62 at an altitude of 30,000 feet; adequate data were not available to determine the compressor surge limit at the other simulated flight condition.

At all corrected engine speeds, compressor efficiency decreased as compressor pressure ratio was increased over the range investigated. In the region of corrected air flows above that for maximum compressor efficiency, the efficiency decreased rapidly as the corrected engine speed was raised. For engine operation from 75 percent of normal thrust to military thrust, the corrected engine speeds are 7630 and 7855 rpm for the conditions of figure 3(a) and 7290 and 7510 for the conditions of figure 3(b); maximum compressor efficiency occurred at lower corrected
2792 engine speeds and compressor pressure ratios. In the range of corrected engine speeds above about 7200 rpm, the corrected air flow changed a comparatively small amount as corrected engine speed was increased, which indicated choking at the compressor inlet.

Engine Performance

The ambient static pressures and temperatures obtained during the investigation deviated somewhat from NACA standard values; therefore all engine performance parameters presented graphically have been adjusted to NACA standard conditions at the respective altitudes by use of the factors δ_a and θ_a (defined in appendix A). All engine performance data obtained at the two simulated flight conditions for the various fixed positions of the adjustable first-stage turbine stator are presented in table I.

Adjustable first-stage turbine stator. - Composite performance plots for engine speeds of 7260, 7050, 6800, 6400, and 5800 rpm are presented in figure 4 for a simulated flight Mach number of 0.62 at an altitude of 30,000 feet, and in figure 5 for a simulated flight Mach number of 0.46 at an altitude of 15,000 feet. The composite performance plots, which were constructed by the method described in appendix B, are presented on coordinates of net thrust against compressor pressure ratio and include curves for the various effective flow areas of the first-stage turbine stator (obtainable by varying exhaust-nozzle area), superimposed lines of constant turbine-inlet to engine-inlet temperature ratio, and contours of specific fuel consumption based on net thrust. Where possible, the approximate compressor surge pressure ratio is indicated. A reliable measurement of exhaust-nozzle area was not available; therefore, the composite plots could not be completed to the extent of superimposing lines of constant exhaust-nozzle area.

By adjusting both the first-stage turbine-stator and the exhaust-nozzle areas at a given engine speed to control the matching between the compressor and turbine, it was possible to obtain either constant turbine-inlet to engine-inlet temperature ratio or constant thrust over a range of compressor pressure ratios (figs. 4 and 5). Similarly, it was possible to obtain a range of turbine-inlet to engine-inlet temperature

ratios and consequently a range of thrusts at a given compressor pressure ratio. Increasing the turbine-stator flow area permitted a given turbine-inlet to engine-inlet temperature ratio to be obtained at lower compressor pressure ratio and also permitted increased turbine-inlet to engine-inlet temperature ratios to be obtained at a given compressor pressure ratio. At a given turbine-inlet temperature, the ratio of compressor-outlet pressures at two different turbine-stator flow areas 2792
should be approximately inversely proportional to the ratio of the turbine-stator flow areas. Small experimental errors in the measurement of effective turbine-stator area, turbine-inlet temperature, and compressor pressure ratio caused some deviation from the aforementioned relation in figures 4 and 5.

Within the range of first-stage turbine-stator areas investigated, variations in compressor pressure ratio at constant turbine-inlet to engine-inlet temperature ratio and engine speed through control of turbine-stator and exhaust-nozzle areas did not have an appreciable effect on thrust (figs. 4 and 5). This was particularly true at the higher engine speeds and turbine-inlet to engine-inlet temperature ratios where the thrust change was generally less than 3 percent; greater thrust changes occurred at the lower temperature ratios. Exhaust-system losses affect the trends of thrust with compressor pressure ratio at constant turbine-inlet to engine-inlet temperature ratio. As the compressor pressure ratio was increased at a given turbine-inlet to engine-inlet temperature ratio, the compressor efficiency decreased somewhat, whereas the turbine efficiency tended to increase slightly. At high turbine-inlet to engine-inlet temperature ratios, the exhaust system losses remained relatively constant over the range of compressor pressure ratios, and at these conditions the thrust was not appreciably affected by variations in compressor pressure ratio. At the low turbine-inlet to engine-inlet temperature ratios, however, the exhaust-system losses tended to decrease with increased compressor pressure ratio and caused larger variations in thrust with compressor pressure ratio.

In general, specific fuel consumption based on net thrust at a given turbine-inlet to engine-inlet temperature ratio decreased as the compressor pressure ratio was increased by adjusting turbine-stator and exhaust-nozzle areas (figs. 4 and 5); in many cases, however, there was an optimum compressor pressure ratio for minimum specific fuel consumption at a given turbine-inlet to engine-inlet temperature ratio within the range of turbine-stator areas investigated. The specific fuel consumption, in general, decreased on the order of 1 to 7 percent as the compressor pressure ratio was increased at the higher turbine-inlet to engine-inlet temperature ratios. The aforementioned trend of decreased specific fuel consumption with increased compressor pressure ratio is attributed to the combined effects of increased thermodynamic efficiency with increased compressor pressure ratio, a trend of increasing turbine efficiency with decreased turbine-stator area, a trend of decreasing exhaust-system losses with decreased turbine-stator area (especially at the lower temperature ratios), and decreased compressor efficiency with increased compressor pressure ratio.

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The composite performance plots of figures 4 and 5 do not show complete agreement with the analysis of reference 2. This analysis, which assumed constant component efficiencies, indicated possible improvement in specific fuel consumption at less than maximum thrust by using an adjustable turbine stator and a variable-area exhaust nozzle to maintain operation at a constant compressor pressure ratio as compared with operation at a constant turbine-stator area. In general, the trends of the specific fuel consumption contours of figure 4 indicate that a fixed first-stage turbine-stator area may be selected to obtain near optimum specific fuel consumption over a range of engine speeds; for example, a first-stage turbine-stator effective area of 1.17 square feet would be near optimum for the conditions of figure 4. The trends of the specific fuel consumption contours of figure 5 indicate that the specific fuel consumption would be somewhat lower for operation at constant compressor pressure ratio as compared with operation at a constant turbine-stator area; however, the gains would be small at thrust levels of interest.

The variation of specific fuel consumption with first-stage turbine-stator effective area at four thrust levels for each of the two simulated flight conditions is shown in figure 6. The four thrust levels were chosen to approximate military, normal, 90 percent of normal, and 75 percent of normal thrust. At the four thrust levels noted, the specific fuel consumption decreased at a decreasing rate as the turbine-stator was closed; total variations of specific fuel consumption of 3 to 12 percent occurred within the range of turbine-stator areas investigated at these thrust levels. The specific fuel consumption was affected only slightly (less than 1 percent) by changes in turbine-stator area between 1.13 and 1.17 square feet. Use of turbine-stator areas less than 1.13 square feet would result in little or no improvement in specific fuel consumption, and the operable range at the smaller areas would be limited by compressor surge.

Fixed first-stage turbine stator. - Performance data are not available for the engine equipped with a standard first-stage turbine stator; however, the effective flow area of the standard first-stage stator was on the order of 1.17 square feet. This area would result in a specific fuel consumption near the optimum values at the thrust levels noted in figure 6. As shown in figure 4, however, the limiting turbine-inlet to engine-inlet temperature ratio at engine speeds of 7260 and 7050 rpm occurred at or near the compressor surge limit for an effective first-stage turbine-stator area of 1.17 square feet.

Preliminary investigation of the engine equipped with the standard first-stage turbine stator revealed a severe compressor surge limitation at high corrected engine speeds (refs. 3 and 4). To make the engine operable without compressor modifications and with a fixed first-stage turbine stator, it would be necessary to increase the flow area of the stator a sufficient amount to reduce the operating compressor pressure ratio below the surge pressure ratio at high corrected engine speeds. Such an increase would result in somewhat higher specific fuel consumption. For example, if a compressor pressure ratio margin of 0.15 between the limiting-temperature operating point and the compressor surge limit were chosen for the conditions of figure 4(a), the effective stator flow area required would be approximately 1.20 square feet; for this area, the specific fuel consumption would be on the order of 1 to 2 percent higher than the minimum obtained at the thrust levels noted in figure 6.

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An adjustable first-stage turbine stator could be used to avoid the compressor surge limitation of the particular engine at high turbine-inlet temperatures and corrected engine speeds, and also to permit operation at or near optimum compressor pressure ratios for minimum specific fuel consumption at reduced thrusts. The available improvement in specific fuel consumption by use of the adjustable stator as compared with use of a fixed turbine-stator effective area of 1.20 square feet, selected to give an arbitrary margin between the operating and compressor surge pressure ratios, would not warrant the complications of the adjustable stator. An increase in the flow area of the fixed first-stage turbine stator may be considered a temporary method of increasing the margin between the operating and surge compressor pressure ratios of the particular engine; modifications to the compressor to improve the surge limit, such as those discussed in references 3 and 4, would permit use of a fixed first-stage turbine-stator-flow area which would result in near optimum specific fuel consumption at thrust levels of interest.

Operational Characteristics

It is possible that an adjustable first-stage turbine stator could be used to improve the acceleration characteristics of the engine. Figures 4 and 5 illustrate that the compressor pressure ratio may be decreased at a given steady-state turbine-inlet temperature by increasing the stator flow area; thus the margin between the operating compressor pressure ratio and the compressor surge pressure ratio could be increased for acceleration purposes. The present program, however, did not include an investigation of the engine acceleration characteristics.

SUMMARY OF RESULTS

The performance of a turbojet engine with a two-stage turbine, an adjustable first-stage turbine stator, and a variable-area exhaust nozzle was investigated at selected constant engine speeds and two simulated flight conditions; various fixed settings of the adjustable stator were used.

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For the particular component characteristics of the engine investigated, little improvement in thrust or specific fuel consumption could be realized at conditions from 75 percent of normal to military thrust by use of an adjustable, rather than a fixed, first-stage turbine stator. In general, the thrust available at a given turbine-inlet to engine-inlet temperature ratio was not appreciably affected by variations in compressor pressure ratio through control of first-stage turbine-stator and exhaust-nozzle areas. The flexibility provided by an adjustable first-stage turbine stator and a variable-area exhaust nozzle may be used to obtain minimum specific fuel consumption at a given thrust; however, the available improvement in specific fuel consumption by use of an adjustable as compared with an optimum fixed turbine stator was small (less than 1 percent) at thrust levels of interest.

Because of compressor surge limitations of the particular engine investigated, it would be necessary to use a fixed turbine-stator flow area larger than that for minimum specific fuel consumption if the engine were to be made operable without compressor modifications and with fixed first-stage turbine stator. The specific fuel consumption obtainable by use of a given setting of the first-stage turbine stator, selected to give an arbitrary compressor pressure ratio margin of 0.15 between the limiting temperature operating point and the compressor-surge limit for the most critical condition investigated, would be on the order of 1 to 2 percent higher than the minimum obtained by use of the adjustable stator at thrusts from 75 percent of normal to military. This penalty in specific fuel consumption would not warrant the complications of the adjustable stator.

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National Advisory Committee for Aeronautics
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APPENDIX A

SYMBOLS

The following symbols were used in this report:

A_n effective area of first-stage turbine stator, sq ft

F_n net thrust, lb

M Mach number

N engine speed, rpm

P total pressure, lb/sq ft

p static pressure, lb/sq ft

T total temperature, $^{\circ}R$

W_a air flow, lb/sec

W_f fuel flow, lb/hr

δ ratio of total pressure at engine inlet to absolute static pressure of NACA standard atmosphere at sea level

δ_a ratio of ambient static pressure p_0 to the absolute static pressure of NACA standard atmosphere at altitude

θ ratio of absolute total temperature at engine inlet to absolute static temperature of NACA standard atmosphere at sea level

θ_a ratio of absolute ambient static temperature to absolute static temperature of NACA standard atmosphere at altitude

Subscripts:

0 free-stream conditions

1 cowl inlet

2 engine inlet

4 compressor outlet

5 turbine inlet

APPENDIX B

METHOD OF CONSTRUCTING COMPOSITE PERFORMANCE PLOTS

Comparable engine performance data adjusted to standard altitude conditions were not directly available for the various positions of the adjustable first-stage turbine stator at constant engine speeds, including rated engine speed, because of ambient-air temperature differences and because the ambient-air temperatures were higher than standard at the respective altitudes (table I). It was therefore necessary to use a method of cross-plotting and extrapolation of the data to obtain comparable engine performance for the various positions of the first-stage turbine stator at constant engine speeds and standard altitude conditions.

Typical engine performance data for a fixed first-stage turbine-stator position for which the effective stator flow area was 1.13 square feet are presented in figure 7 for a simulated flight Mach number of 0.62 at an altitude of 30,000 feet, and in figure 8 for a simulated flight Mach number of 0.46 at an altitude of 15,000 feet. These figures show compressor pressure ratio against turbine-inlet to engine-inlet temperature ratio, and net thrust and fuel flow against compressor pressure ratio for four engine speeds; at each engine speed, exhaust-nozzle area was varied to obtain the range of compressor pressure ratios. The constant engine speed curves of these figures were extrapolated to the limiting turbine-inlet to engine-inlet temperature ratio or to the compressor surge pressure ratio.

The data of figures 7 and 8 and similar data for the other turbine-stator positions were cross-plotted to obtain the performance at selected engine speeds for which composite performance plots were to be made. The cross plots used the coordinates of compressor pressure ratio against engine speed and were made for various constant turbine-inlet to engine-inlet temperature ratios, net thrusts, and fuel flows. Extrapolation of these cross plots was necessary to obtain performance at the rated engine speed of 7260 rpm. Thus, the variation of turbine-inlet to engine-inlet temperature ratio, net thrust, and fuel flow with compressor pressure ratio was determined for each first-stage turbine-stator position at the selected engine speeds.

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TABLE I. - ENGINE PERFORMANCE DATA

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Adjusted engine speed $N/\sqrt{\theta_a}$ (rpm)	Effective area of first-stage turbine stator A_n (sq ft)	Fuel flow W_f $\left(\frac{lb}{hr}\right)$	Adjusted fuel flow $W_f/\delta_a\sqrt{\theta_a}$ (lb/hr)	Engine-inlet total pressure P_2 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Cowl-inlet total temperature T_1 (°R)	Compressor-outlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Compressor-outlet total temperature T_4 (°R)	Turbine-inlet total temperature T_5 (°R)	Engine inlet-air flow $W_{a,1}$ (lb/sec)	Net thrust F_n (lb)	Adjusted net thrust F_n/δ_a (lb)
1	30,000	1.301	0.625	608	7260	7165	1.25	2245	2289	791	455	3673	783	1550	57.52	1616	1669
2		1.292	.616	612	7260	7131	1.25	2575	2394	791	459	3752	787	1590	57.23	1785	1831
3		1.299	.623	610	7260	7139	1.25	2500	2533	793	459	3786	790	1650	57.35	1984	2044
4		1.303	.627	608	7260	7148	1.25	2625	2670	792	456	3850	792	1650	57.39	2075	2141
5		1.296	.621	610	7260	7148	1.25	2765	2824	791	456	3890	796	1720	57.16	2216	2282
6		1.299	.623	607	7260	7131	1.25	3080	3131	789	460	3953	806	1810	56.81	2441	2526
7		1.299	.623	608	6897	6776	1.25	1995	2025	790	460	----	758	1443	56.40	1419	1466
8		1.299	.623	608	6897	6776	1.25	2270	2304	790	460	3601	768	1548	56.38	1792	1851
9		1.300	.624	608	6897	6765	1.25	2560	2601	790	459	3717	773	1643	56.50	2068	2126
10		1.294	.619	614	6897	6782	1.25	2848	2865	795	458	3823	780	1730	56.75	2265	2317
11		1.301	.625	608	6897	6785	1.25	3015	3063	791	459	3917	787	1793	56.28	2422	2502
12		1.297	.621	610	6353	6248	1.25	1625	1646	791	459	3121	721	1270	52.86	1093	1126
13		1.301	.625	610	6353	6233	1.25	1860	1899	794	462	3221	753	1428	52.64	1457	1501
14		1.296	.621	608	6353	6248	1.25	2100	2133	788	459	3303	736	1500	52.26	1676	1733
15		1.301	.625	608	6353	6246	1.25	2220	2255	791	458	3344	739	1543	52.42	1764	1822
16		1.296	.621	609	6353	6248	1.25	2410	2443	789	459	3433	746	1620	52.16	1913	1972
17		1.301	.625	610	5808	5699	1.25	1209	1222	794	462	2585	681	1150	46.70	649	668
18		1.301	.625	608	5808	5699	1.25	1301	1316	792	461	2627	684	1200	45.56	806	831
19		1.300	.624	609	5808	5699	1.25	1427	1443	792	461	2681	686	1287	46.45	952	982
20		1.300	.624	610	5808	5706	1.25	1550	1589	794	460	2724	690	1330	46.52	1053	1085
21		1.298	.622	610	5808	5706	1.25	1661	1701	792	460	2775	695	1390	46.22	1182	1217
22		1.298	.622	610	4719	4662	1.25	790	802	792	457	1750	607	996	33.61	140	144
23		1.303	.627	606	4719	4662	1.25	800	819	788	457	1726	608	1010	33.35	135	140
24		1.304	.628	601	4719	4652	1.25	860	875	784	457	1729	610	1043	33.15	250	261
25		1.293	.618	604	4719	4641	1.25	960	982	781	458	1765	615	1130	32.22	363	378
26		1.319	.642	603	4719	4663	1.25	1160	1193	795	457	1803	623	1265	31.65	559	582
27		1.274	.599	625	7260	7049	1.23	2130	2085	794	468	3694	798	1627	56.96	1562	1574
28		1.295	.619	608	7260	7057	1.23	2315	2324	787	469	----	809	1607	56.44	1803	1862
29		1.284	.609	618	7260	7049	1.23	2615	2580	794	469	3885	816	1700	56.75	2115	2149
30		1.291	.616	604	7260	7065	1.23	2760	2813	780	468	3905	820	1767	56.75	2244	2334
31		1.255	.579	622	7260	7026	1.23	3015	2947	781	469	4003	824	1840	55.57	2440	2464
32		1.294	.619	616	6897	6720	1.23	1890	1876	797	467	3524	769	1430	56.49	1378	1404
33		1.298	.621	607	6897	6720	1.23	2350	2370	787	467	3666	788	1604	55.68	1925	1992
34		1.304	.628	604	6897	6727	1.23	2590	2627	788	467	3792	795	1685	55.67	2142	2226
35		1.306	.629	610	6897	6738	1.23	2875	2892	796	466	3917	801	1777	56.27	2342	2412
36		1.293	.618	620	6897	6727	1.23	3295	3256	802	466	4102	811	1897	56.39	2583	2627
37		1.292	.616	612	6353	6197	1.23	1535	1536	791	466	3126	730	1300	52.44	1020	1047
38		1.298	.622	605	6353	6190	1.23	1695	1714	786	466	3191	741	1377	51.70	1285	1334
39		1.305	.629	605	6353	6197	1.23	1890	1914	789	467	3285	747	1465	51.91	1484	1540
40		1.310	.634	606	6353	6204	1.23	2080	2108	793	467	3360	751	1533	52.09	1670	1733
41		1.305	.627	612	6353	6204	1.23	2215	2219	798	466	3431	750	1570	52.43	1758	1804
42		1.300	.624	606	5808	5678	1.23	1136	1155	787	464	2614	666	1163	46.09	599	622
43		1.306	.629	603	5808	5672	1.23	1305	1326	786	466	2672	695	1255	45.92	840	874
44		1.301	.625	605	5808	5672	1.23	1479	1499	787	466	2756	702	1325	46.66	1023	1062
45		1.310	.634	609	5808	5685	1.23	1602	1617	798	464	2807	704	1380	46.31	1089	1125
46		1.306	.630	605	5808	5692	1.23	1720	1750	790	463	2846	709	1440	45.69	1252	1300
47		1.294	.619	608	4719	4619	1.23	717	726	787	463	1733	614	967	33.07	61	63
48		1.301	.625	603	4719	4625	1.23	743	751	798	465	1741	614	1003	33.51	108	111
49		1.324	.647	606	4719	4650	1.23	851	867	801	464	1800	619	1067	33.24	226	235
50		1.302	.626	603	4719	4614	1.23	917	934	785	465	1798	625	1150	32.14	352	366

TABLE I. - Continued. ENGINE PERFORMANCE DATA

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Adjusted engine speed $N/\sqrt{\theta_a}$ (rpm)	Effective area of first-stage turbine stator A_n (sq ft)	Fuel flow W_f $\left(\frac{lb}{hr}\right)$	Adjusted fuel flow $W_f/\delta_a\sqrt{\theta_a}$ (lb/hr)	Engine-inlet total pressure P_2 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Cowl-inlet total temperature T_1 (°R)	Compressor-outlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Compressor-outlet total temperature T_4 (°R)	Turbine-inlet total temperature T_5 (°R)	Engine inlet-air flow $W_{a,1}$ (lb/sec)	Net thrust F_n (lb)	Adjusted net thrust F_n/δ_a (lb)
51	30,000	1.326	0.650	603	4719	4641	1.23	1060	1076	801	462	1884	627	1200	32.09	478	498
52		1.292	.616	611	7260	7182	1.21	2020	2054	789	453	3716	778	1473	57.49	1467	1508
53		1.298	.622	607	7260	7182	1.21	2350	2408	788	454	3840	794	1593	57.27	1879	1945
54		1.301	.625	605	7260	7174	1.21	2645	2713	787	456	3966	806	1703	56.97	2155	2216
55		1.299	.623	607	7260	7165	1.21	2975	3039	789	456	4099	817	1810	56.90	2378	2461
56		1.268	.590	622	7260	7131	1.21	3245	3220	787	457	4179	821	1890	56.68	2580	2606
57		1.289	.613	611	6897	6775	1.21	1605	1625	787	459	3555	765	1403	56.25	1327	1364
58		1.294	.619	616	6897	6775	1.21	2050	2066	796	460	3679	775	1493	56.85	1653	1688
59		1.288	.613	621	6897	6767	1.21	2390	2371	800	460	3828	785	1610	57.03	1961	1983
60		1.293	.616	620	6897	6767	1.21	2775	2758	802	461	3979	796	1743	56.94	2278	2308
61		1.293	.618	620	6897	6783	1.21	3126	3113	802	459	4136	804	1853	56.81	2504	2537
62		1.297	.621	620	6353	6218	1.21	1460	1468	804	463	3212	731	1285	53.48	1024	1057
63		1.298	.622	617	6353	6226	1.21	1685	1681	801	462	3278	738	1367	53.05	1318	1342
64		1.302	.626	617	6353	6226	1.21	1905	1901	805	463	3599	746	1483	53.01	1551	1579
65		1.304	.628	605	6353	6226	1.21	2090	2126	788	463	3418	753	1545	51.88	1682	1746
66		1.303	.627	616	6353	6226	1.21	2390	2391	801	465	3550	759	1643	52.53	1860	1919
67		1.282	.607	613	5808	5698	1.21	1063	1069	786	460	2810	685	1130	46.15	584	600
68		1.296	.621	607	5808	5692	1.21	1480	1501	787	462	2785	705	1347	45.71	1050	1087
69		1.290	.614	613	5808	5685	1.21	1690	1694	791	462	2860	712	1443	45.62	1200	1229
70		1.282	.616	608	5808	5692	1.21	1675	1699	788	462	2880	720	1567	46.18	1376	1420
71		1.285	.619	619	5808	5692	1.21	2060	2069	801	462	2987	724	1643	45.75	1369	1390
72		1.311	.634	615	4719	4635	1.21	700	702	808	461	1779	613	----	33.97	35	36
73		1.291	.616	614	4719	4625	1.21	791	793	793	461	1795	616	1043	32.99	202	207
74		1.317	.640	610	4719	4646	1.21	850	862	803	460	1834	619	1070	33.06	277	285
75		1.299	.623	612	4719	4619	1.21	951	955	798	463	1857	631	----	32.21	382	392
76		1.291	.618	612	4719	4609	1.21	1009	1011	790	465	1862	634	1223	31.56	443	455
77		1.292	.616	612	7260	7123	1.17	2060	2074	791	460	3820	793	1470	57.19	1542	1582
78		1.296	.621	611	7260	7131	1.17	2258	2272	792	460	3946	805	1580	57.12	1817	1868
79		1.293	.618	614	7260	7115	1.17	2390	2398	794	462	4016	809	1610	57.02	1961	2008
80		1.290	.614	614	7260	7131	1.17	2590	2508	792	459	4098	812	1673	57.02	2152	2181
81		1.290	.614	614	7260	7139	1.17	2765	2782	792	458	4171	815	1733	56.99	2284	2318
82		1.291	.618	612	6897	6752	1.17	1770	1777	790	463	3629	774	1367	56.23	1299	1333
83		1.302	.628	612	6897	6768	1.17	2020	2034	797	462	3769	785	1480	56.66	1667	1710
84		1.297	.621	617	6897	6760	1.17	2308	2300	801	462	3916	795	1590	56.86	1824	1959
85		1.286	.611	614	6897	6768	1.17	2686	2605	789	460	3988	804	1713	55.98	2185	2233
86		1.291	.616	615	6897	6657	1.17	2965	2913	794	479	4070	838	1880	54.71	2357	2406
87		1.310	.634	605	6897	6783	1.17	3050	3093	792	460	4163	813	1847	55.93	2410	2502
88		1.300	.624	612	6353	6226	1.17	1375	1383	795	462	3224	736	1177	52.79	922	946
89		1.289	.614	608	6353	6233	1.17	1515	1535	784	460	3273	740	1277	51.99	1136	1172
90		1.285	.619	612	6353	6241	1.17	1700	1713	792	460	3379	746	1356	52.43	1352	1387
91		1.311	.634	607	6353	6241	1.17	1910	1942	798	461	3478	753	1440	52.37	1531	1565
92		1.304	.628	615	6353	6241	1.17	2085	2101	802	461	3565	758	1530	52.70	1659	1694
93		1.300	.624	612	5808	5705	1.17	1030	1038	798	460	2690	686	1060	46.77	512	525
94		-----	----	---	5808	----	1.17	1112	----	---	457	2735	659	----	-----	---	----
95		1.298	.621	608	5808	5725	1.17	1245	1271	785	457	2778	696	1200	45.87	788	816
96		1.289	.614	610	5808	5688	1.17	1334	1348	787	460	2805	701	1243	45.77	890	917
97		1.309	.633	612	5808	5712	1.17	1436	1449	801	460	2902	705	1280	46.38	989	1015
98		1.300	.624	606	4719	4641	1.17	700	718	786	459	1803	614	940	32.40	19	20
99		1.300	.624	609	4719	4646	1.17	710	721	792	458	1784	613	960	32.87	90	93
100		1.306	.629	609	4719	4657	1.17	738	748	795	456	1801	612	973	32.85	141	145

TABLE I. - Continued. ENGINE PERFORMANCE DATA

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Adjusted engine speed $N/\sqrt{\theta_a}$ (rpm)	Effective area of first-stage turbine stator A_n (sq ft)	Fuel flow W_f $\left(\frac{lb}{hr}\right)$	Adjusted fuel flow $W_f/\delta_a\sqrt{\theta_a}$ (lb/hr)	Engine-inlet total pressure P_2 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Cowl-inlet total temperature T_1 (°R)	Compressor-outlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Compressor-outlet total temperature T_4 (°R)	Turbine-inlet total temperature T_5 (°R)	Engine inlet-air flow $W_{a,1}$ (lb/sec)	Net thrust F_n (lb)	Adjusted net thrust F_n/δ_a (lb)
101	30,000	1.306	0.630	607	4719	4646	1.17	800	815	795	459	1833	619	1043	35.27	209	216
102		1.306	.630	609	4719	4652	1.17	910	925	796	458	1878	622	1123	35.00	324	334
103		1.300	.624	605	4719	4652	1.17	985	1008	786	457	----	630	----	-----	----	----
104		1.308	.632	605	7260	7148	1.13	1979	2025	791	459	3940	809	1480	57.07	1456	1510
105		1.294	.619	616	7260	7033	1.13	2255	2227	797	473	4099	837	1618	56.49	1816	1851
106		1.282	.607	621	7260	7033	1.13	2480	2429	798	471	4202	844	1693	56.50	2026	2048
107		1.297	.621	614	7260	7090	1.13	2810	2807	797	465	4342	845	1803	56.66	2304	2357
108		1.285	.610	620	7260	----	1.13	3020	----	787	463	4414	844	----	-----	----	----
109		1.297	.621	626	6897	6666	1.13	1710	1658	812	476	3785	801	1393	57.00	1187	1191
110		1.297	.621	619	6897	6666	1.13	1806	1669	803	475	3846	810	1477	56.33	1485	1507
111		1.286	.611	622	6897	6689	1.13	2115	2072	800	471	3927	816	1573	56.26	1758	1755
112		1.285	.610	620	6897	6697	1.13	2490	2449	797	469	4072	826	1710	55.90	2011	2037
113		1.293	.618	615	6897	6727	1.13	2935	2923	795	466	4230	832	1857	55.57	2508	2356
114		1.304	.628	615	6353	6148	1.13	1323	1307	802	475	3276	762	1270	51.17	832	849
115		1.285	.619	619	6353	6113	1.13	1445	1411	801	479	3317	770	1537	50.87	997	1012
116		1.300	.624	618	6353	6127	1.13	1567	1535	804	477	3392	772	1370	51.07	1161	1169
117		1.292	.616	618	6353	6113	1.13	1688	1650	798	479	3422	779	1430	50.49	1288	1307
118		1.300	.624	617	6353	6120	1.13	1786	1751	802	479	3452	781	1455	50.69	1352	1376
119		1.289	.623	615	5808	5601	1.13	997	982	799	477	2742	720	1150	46.04	486	476
120		1.286	.611	624	5808	5601	1.13	1169	1134	802	476	2787	724	1240	46.14	658	662
121		1.292	.619	622	5808	5620	1.13	1333	1303	805	474	2856	730	1318	46.18	811	819
122		1.292	.616	621	5808	5613	1.13	1498	1464	802	475	2889	737	1400	45.72	956	967
123		1.298	.622	616	5808	5620	1.13	1614	1592	800	474	2912	745	1477	45.47	1050	1070
124		1.305	.627	621	4719	4561	1.13	641	627	808	476	1802	636	963	32.34	-26	-26
125		1.297	.621	621	4719	4582	1.13	708	696	806	471	1837	638	1015	31.16	132	133
126		1.300	.624	618	4719	4572	1.13	773	761	804	475	1860	643	1070	31.03	220	224
127		1.288	.613	622	4719	4576	1.13	887	849	801	471	1890	647	1157	30.60	305	308
128		1.298	.622	619	4719	4582	1.13	964	950	804	471	1950	654	1230	29.90	409	415
129	15,000	1.158	0.483	1190	7260	7260	1.25	3840	3852	1378	485	6108	813	1573	96.36	2791	2799
130		1.158	.460	1180	7260	7168	1.25	4040	4035	1384	497	6130	827	1643	94.20	3125	3159
131		1.158	.463	1186	7260	7214	1.25	4310	4308	1375	491	6268	824	1680	96.40	3428	3449
132		1.154	.457	1183	7260	7198	1.25	4720	4717	1385	493	6404	837	1780	94.64	3821	3852
133		1.151	.453	1188	7260	7237	1.25	5030	5034	1386	487	6525	830	1830	95.40	4076	4092
134		1.152	.455	1183	6897	6780	1.25	3370	3329	1362	504	5680	807	1507	90.85	2404	2423
135		1.156	.460	1186	6897	6809	1.25	3765	3759	1371	497	5875	805	1580	92.06	2969	2977
136		1.159	.464	1188	6897	6798	1.25	4065	4029	1375	500	5980	813	1650	91.99	3256	3276
137		1.161	.467	1186	6897	6824	1.25	4480	4460	1377	496	6176	817	1730	92.48	3650	3682
138		1.158	.460	1181	6897	6798	1.25	4890	4868	1385	499	6291	827	1825	91.48	3910	3949
139		1.159	.464	1186	6353	6266	1.25	2695	2674	1374	499	5071	762	1370	85.43	1797	1808
140		1.159	.464	1191	6353	6279	1.25	3160	3129	1381	487	5294	789	1475	85.35	2393	2398
141		1.154	.457	1183	6353	6279	1.25	3645	3632	1385	498	5406	777	1600	84.49	2896	2919
142		1.155	.459	1186	6353	6279	1.25	4075	4052	1370	495	5580	789	1703	85.50	3248	3267
143		1.157	.462	1187	6353	6266	1.25	4450	4411	1374	498	5690	792	1793	82.84	3438	3455
144		1.157	.462	1181	5808	5878	1.25	2090	2136	1366	475	4332	674	1215	77.08	1279	1292
145		1.157	.462	1182	5808	5795	1.25	2500	2517	1366	487	4438	722	1377	75.38	1733	1749
146		1.155	.459	1190	5808	5802	1.25	2965	2971	1376	486	4585	735	1520	74.55	2106	2112
147		1.156	.460	1188	5808	5820	1.25	3250	3249	1375	483	4651	734	1600	73.30	2281	2290
148		-----	----	1184	5808	----	1.25	3595	----	----	484	4720	676	----	-----	----	----
149		1.164	.471	1181	4719	4607	1.25	1260	1232	1575	510	2821	660	1143	49.15	254	257
150		1.161	.467	1195	4719	4645	1.25	1445	1420	1387	501	2942	657	1197	50.73	517	516

TABLE I. - Concluded. ENGINE PERFORMANCE DATA

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Adjusted engine speed $N/\sqrt{\theta_a}$ (rpm)	Effective area of first-stage turbine stator A_n (sq ft)	Fuel flow W_f $\left(\frac{lb}{hr}\right)$	Adjusted fuel flow $W_f/\delta_a\sqrt{\theta_a}$ (lb/hr)	Engine-inlet total pressure P_2 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Cowl-inlet total temperature T_1 (°R)	Compressor-outlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Compressor-outlet total temperature T_4 (°R)	Turbine-inlet total temperature T_5 (°R)	Engine inlet-air flow $W_{a,1}$ (lb/sec)	Net thrust F_n (lb)	Adjusted net thrust F_n/δ_a (lb)
151	15,000	1.159	0.464	1188	4719	4654	1.25	1585	1579	1377	499	2969	658	1255	49.10	646	649
152		1.158	.460	1188	4719	4635	1.25	1705	1682	1374	502	3011	664	1325	48.84	777	780
153		1.165	.472	1184	4719	4684	1.25	1910	1911	1379	493	3084	662	1400	51.51	1073	1082
154		1.156	.460	1185	7260	7275	1.21	3370	3377	1379	483	6252	812	1480	96.96	2510	2510
155		1.158	.463	1183	7260	7198	1.21	3785	3783	1369	493	----	836	1595	95.29	3102	3127
156		1.159	.459	1192	7260	7229	1.21	4493	4480	1377	489	6677	844	1743	96.18	3708	3712
157		1.159	.464	1187	6897	6824	1.21	3005	2988	1376	495	5857	803	1430	92.54	2269	2280
158		1.158	.463	1187	6897	6824	1.21	3495	3475	1374	495	6067	815	1540	92.42	2891	2905
159		1.167	.462	1190	6897	6831	1.21	3865	3829	1377	494	6241	823	1627	92.73	3273	3285
160		1.158	.463	1184	6897	6817	1.21	4450	4433	1371	498	6429	851	1748	92.11	3722	3782
161		1.159	.464	1186	6353	6299	1.21	2400	2394	1375	493	5197	764	1327	86.00	1707	1717
162		1.159	.464	1185	6353	6306	1.21	2890	2889	1374	492	5396	775	1433	85.39	2282	2288
163		1.157	.462	1188	6353	6312	1.21	3390	3381	1374	491	5580	785	1563	84.46	2752	2783
164		1.158	.463	1188	6353	6319	1.21	3868	3879	1375	490	5768	794	1680	83.61	3091	3103
165		1.156	.460	1188	6353	6326	1.21	4390	4388	1374	489	5879	799	1800	82.61	3409	3423
166		1.159	.464	1186	5808	5771	1.21	1865	1864	1374	491	4381	724	1280	75.77	1109	1116
167		1.159	.464	1187	5808	5795	1.21	2250	2238	1376	487	4541	727	1323	75.17	1600	1608
168		1.158	.463	1185	5808	5789	1.21	2500	2509	1372	488	4584	733	1410	73.53	1806	1819
169		1.161	.467	1184	5808	5789	1.21	2890	2904	1374	488	4698	743	1540	72.34	2075	2092
170		1.155	.459	1188	5808	5795	1.21	3295	3301	1572	487	4778	753	1675	70.28	2381	2370
171		1.159	.464	1187	4719	4714	1.21	1176	1180	1376	486	2975	644	1083	52.38	299	300
172		1.161	.467	1185	4719	4728	1.21	1295	1307	1375	483	----	643	1147	52.53	437	440
173		1.164	.471	1184	4719	4738	1.21	1530	1548	1378	481	3095	646	1243	52.06	717	725
174		1.159	.464	1186	4719	4735	1.21	1655	1670	1374	482	3151	653	1293	51.21	827	832
175		1.160	.466	1184	4719	4743	1.21	1740	1763	1374	480	----	653	1330	51.32	885	892
176		1.131	.424	1185	7260	7131	1.13	3140	3106	1540	499	----	885	----	------	----	----
177		1.159	.464	1189	7260	7183	1.13	3625	3496	1378	485	6722	858	1563	96.40	2911	2920
178		1.159	.464	1189	7260	7191	1.13	3955	3929	1379	494	6939	866	1660	95.48	3357	3367
179		1.156	.460	1196	7260	7191	1.13	4340	4288	1382	494	7118	871	1720	95.72	3640	3651
180		1.161	.467	1188	7260	7198	1.13	4798	4773	1379	494	7294	880	1850	95.46	3993	4009
181		1.155	.459	1199	6897	6824	1.13	2855	2811	1365	495	6276	824	1410	93.25	2248	2257
182		1.152	.455	1191	6897	6853	1.13	3515	3500	1372	491	6518	837	1590	92.67	3012	3018
183		1.154	.457	1200	6897	6846	1.13	3765	3715	1384	490	6664	839	1600	93.54	3214	3195
184		1.151	.453	1196	6897	6853	1.13	4195	4161	1375	490	6809	849	1704	92.98	3576	3570
185		1.156	.460	1198	6897	6817	1.13	4610	4538	1385	496	7001	862	1810	93.04	3779	3765
186		1.159	.464	1188	6353	6308	1.13	2238	2227	1377	492	5477	761	1300	84.84	1616	1622
187		1.153	.456	1191	6353	6312	1.13	2590	2578	1374	491	5616	789	1394	84.11	2093	2097
188		1.153	.456	1192	6353	6312	1.13	3000	2984	1375	491	5722	801	1485	83.50	2520	2523
189		1.161	.467	1186	6353	6308	1.13	3250	3245	1377	490	5873	802	1555	82.72	2651	2667
190		1.154	.457	1197	6353	6312	1.13	3615	3580	1381	491	5983	809	1650	81.88	2942	2932
191		1.162	.469	1183	5808	5785	1.13	1800	1810	1375	488	4589	678	1150	73.73	1050	1058
192		1.164	.471	1186	5808	5802	1.13	2115	2126	1381	487	4700	736	1310	73.40	1460	1469
193		1.165	.472	1176	5808	5808	1.13	2455	2489	1370	486	4759	743	1420	70.28	1796	1821
194		1.156	.460	1186	5808	5837	1.13	2795	2826	1371	480	4838	748	1530	65.98	2103	2116
195		1.152	.455	1188	5808	5783	1.13	3015	3014	1368	488	4798	763	1630	68.71	2047	2065
196		1.167	.475	1176	4719	4704	1.13	1098	1106	1372	489	3033	650	1060	52.10	199	202
197		1.174	.485	1162	4719	4724	1.13	1229	1241	1368	486	3131	651	1098	52.17	392	396
198		1.157	.462	1186	4719	4709	1.13	1365	1372	1371	487	3151	657	1166	50.80	572	576
199		1.165	.472	1162	4719	4714	1.13	1511	1523	1377	487	3231	663	1230	49.70	690	695
200		1.166	.474	1186	4719	4709	1.13	1639	1645	1383	488	3269	668	1290	49.19	809	814

2792

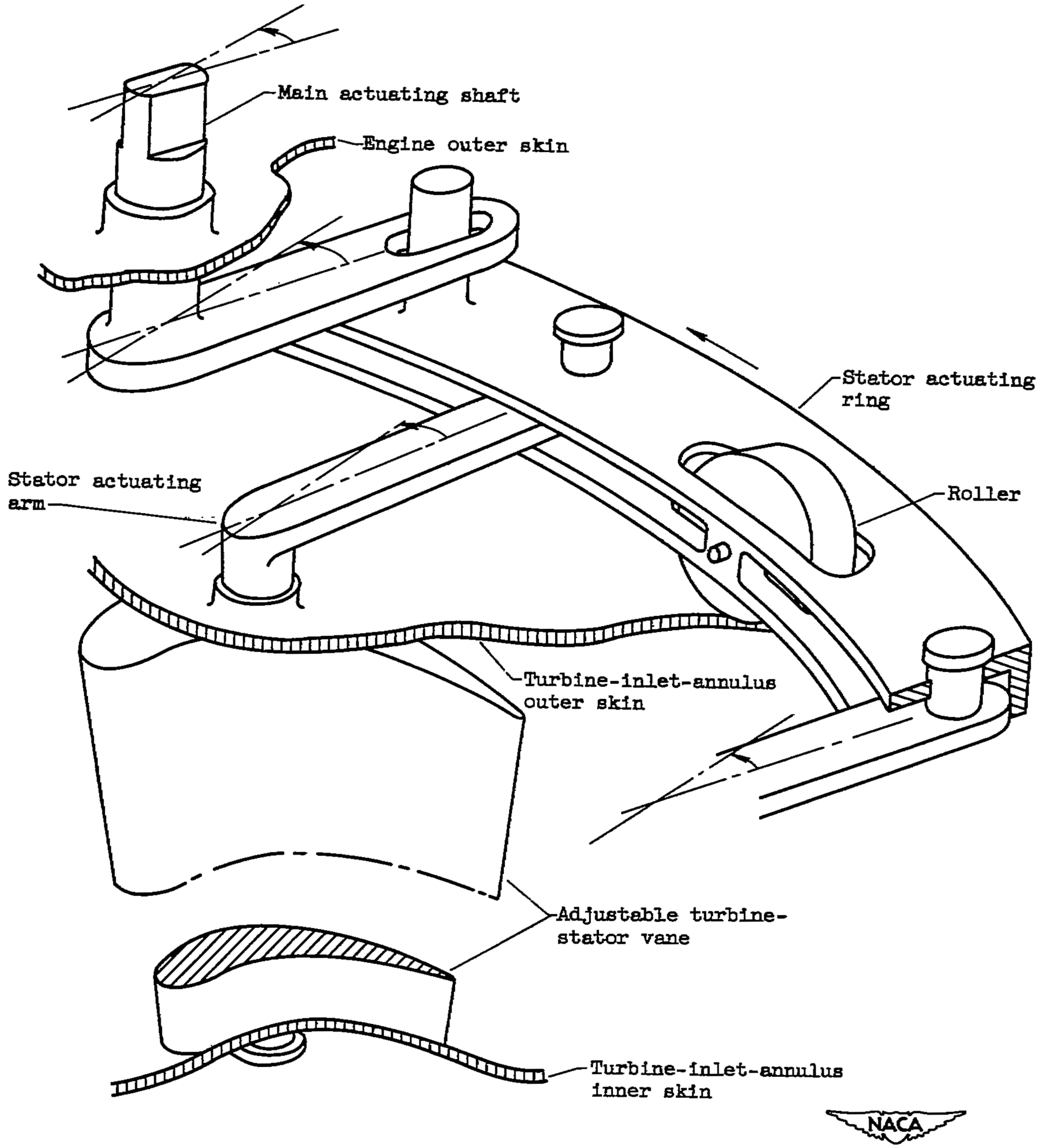


Figure 1. - Schematic sketch of adjustable turbine-stator actuating mechanism.

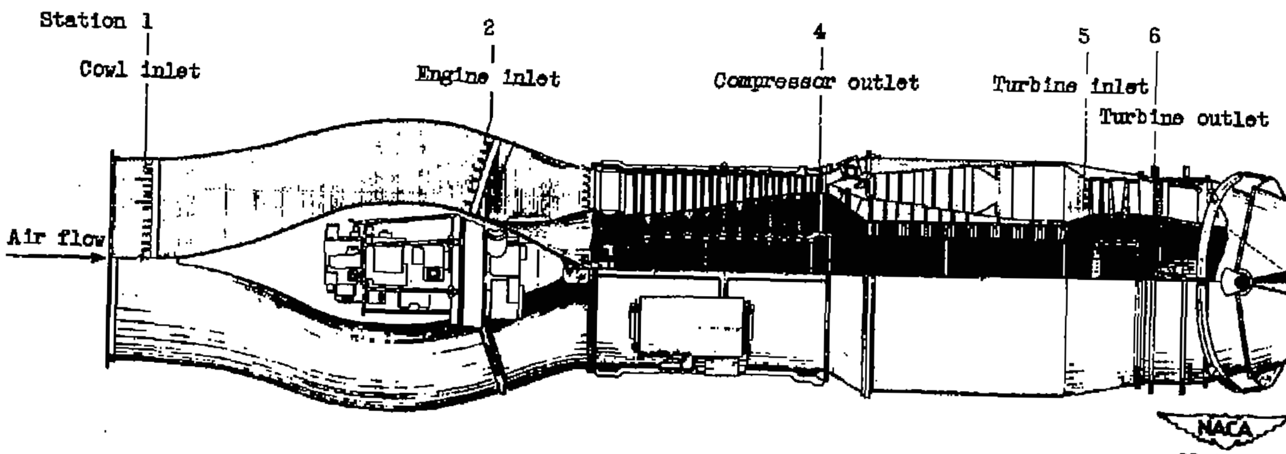
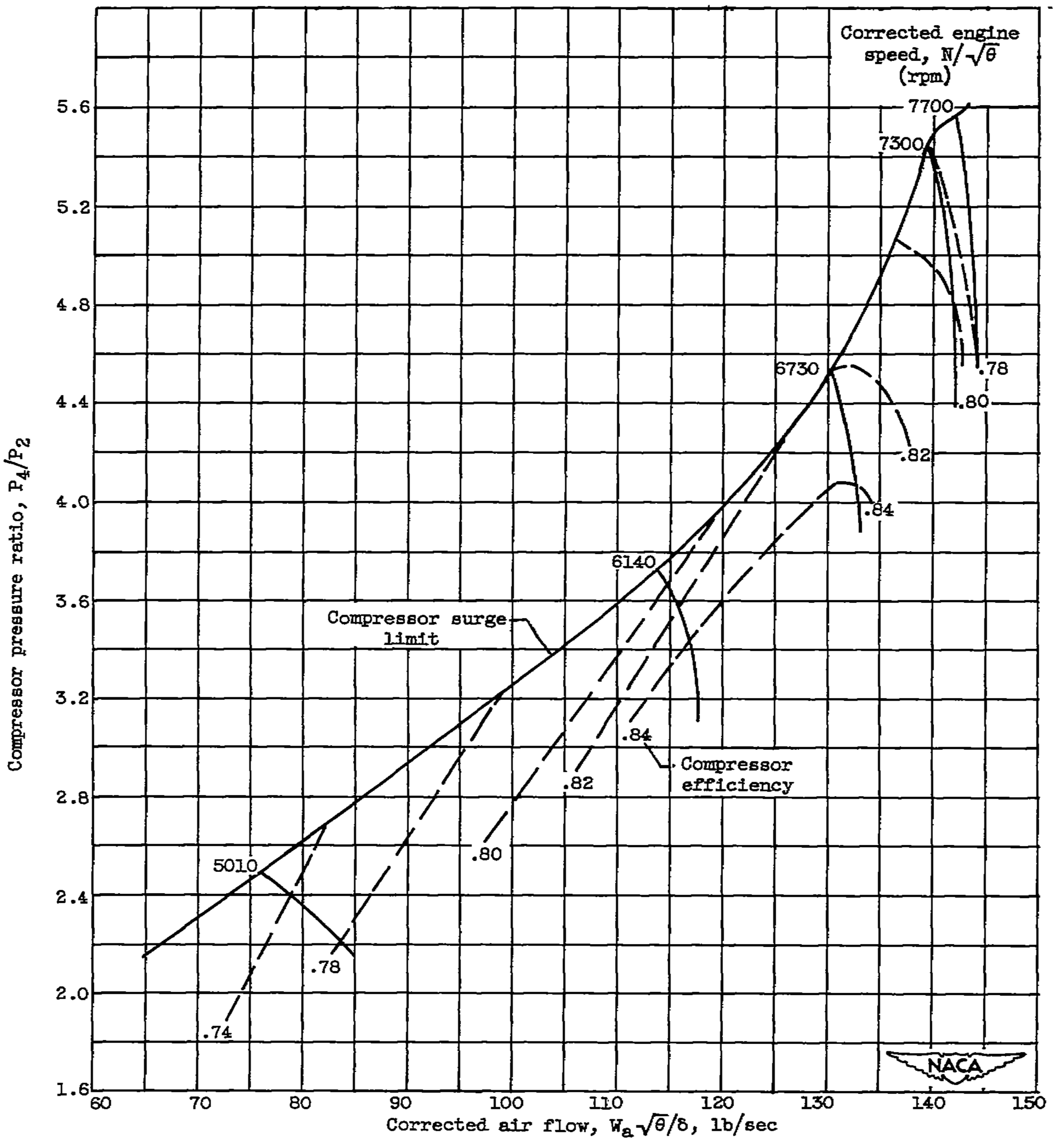


Figure 2. - Cross section of engine showing instrumentation stations.

2792

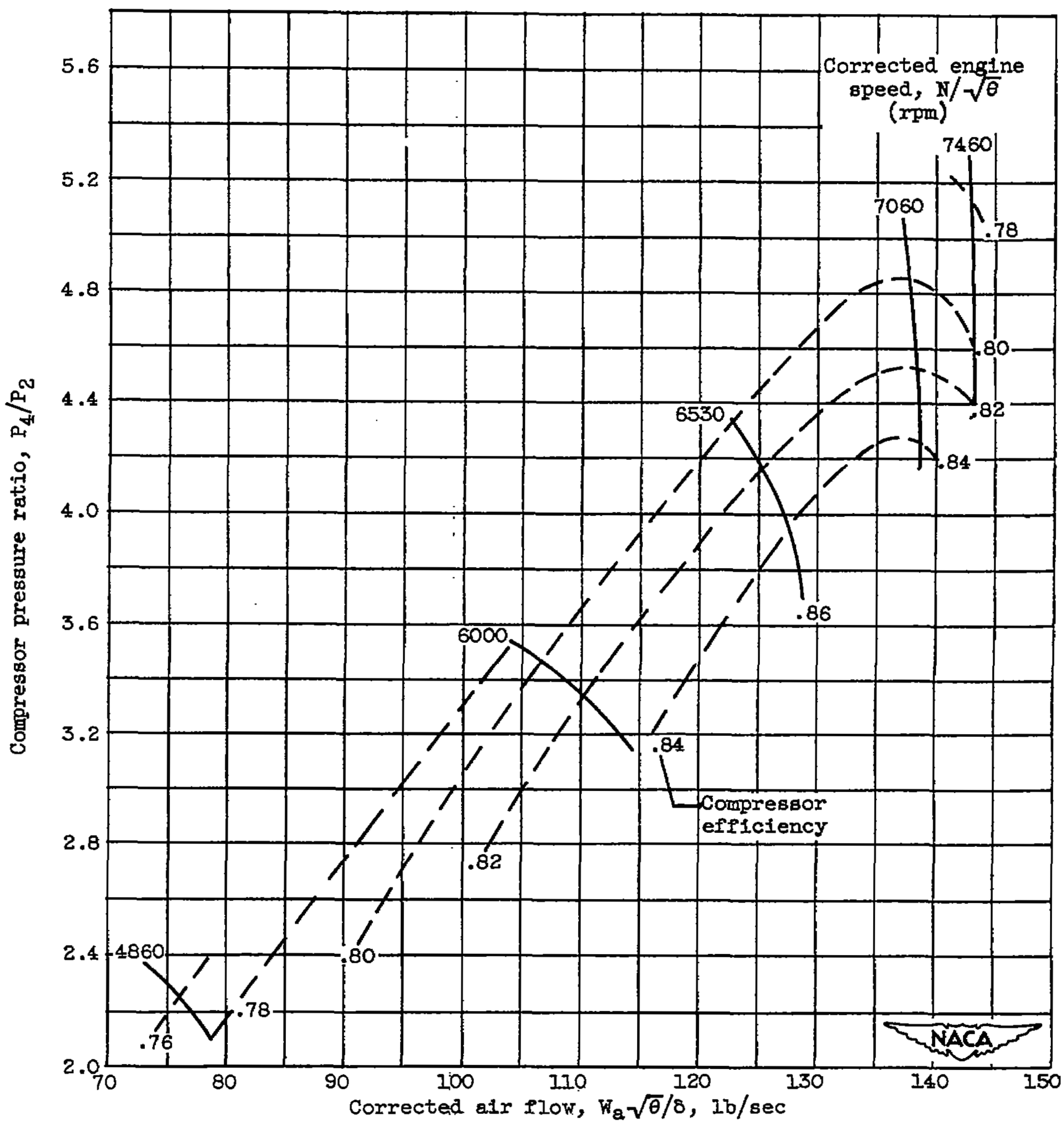


(a) Altitude, 30,000 feet; flight Mach number, 0.62.

Figure 3. - Compressor performance maps.

[redacted]

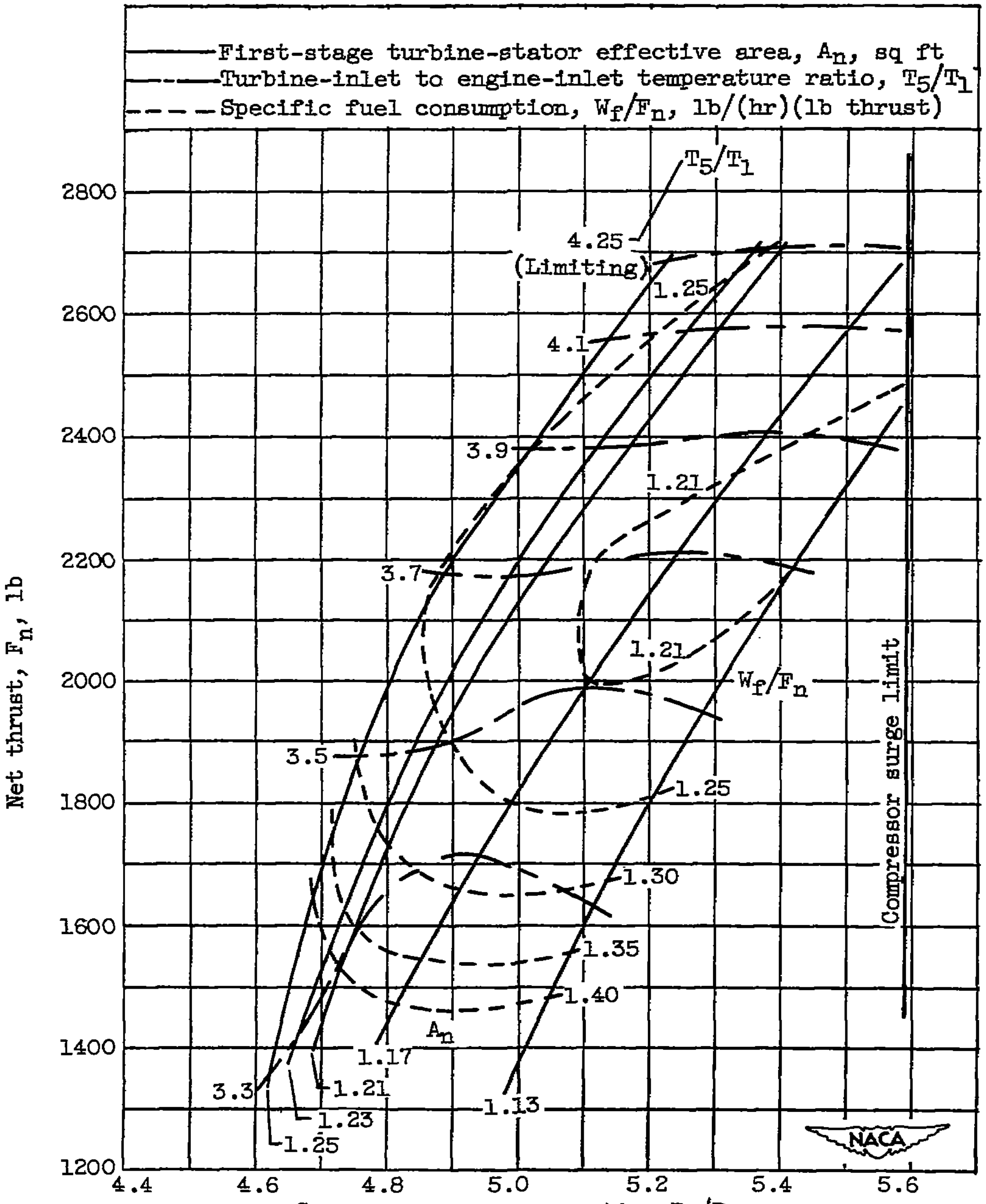
2792



(b) Altitude, 15,000 feet; flight Mach number, 0.46.

Figure 3. - Concluded. Compressor performance maps.

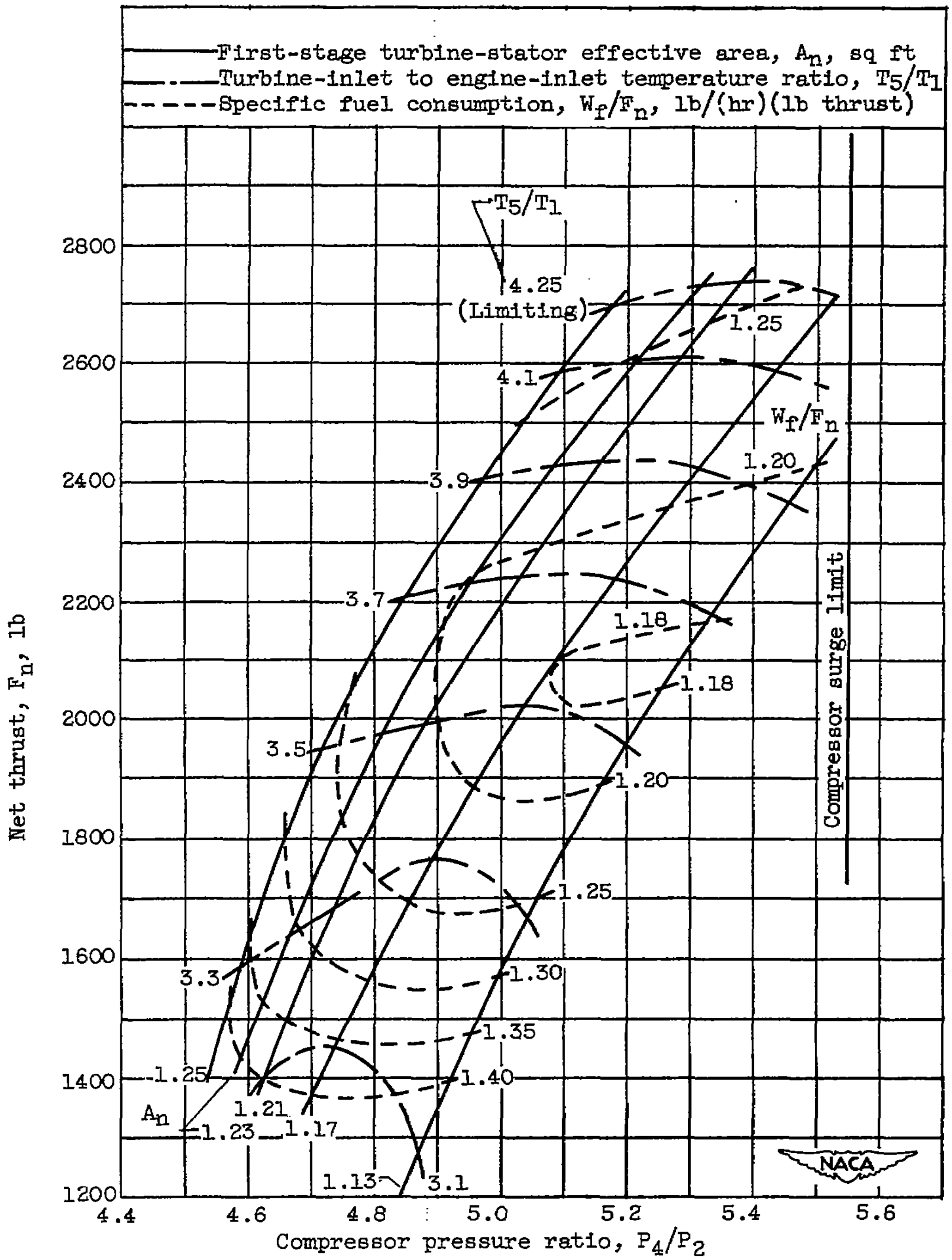
2792



(a) Engine speed, 7260 rpm; corrected engine speed, 7855 rpm.

Figure 4. - Composite performance plots. Altitude, 30,000 feet; flight Mach number, 0.62.

2792



(b) Engine speed, 7050 rpm; corrected engine speed, 7630 rpm.

Figure 4. - Continued. Composite performance plots. Altitude, 30,000 feet; flight Mach number, 0.62.

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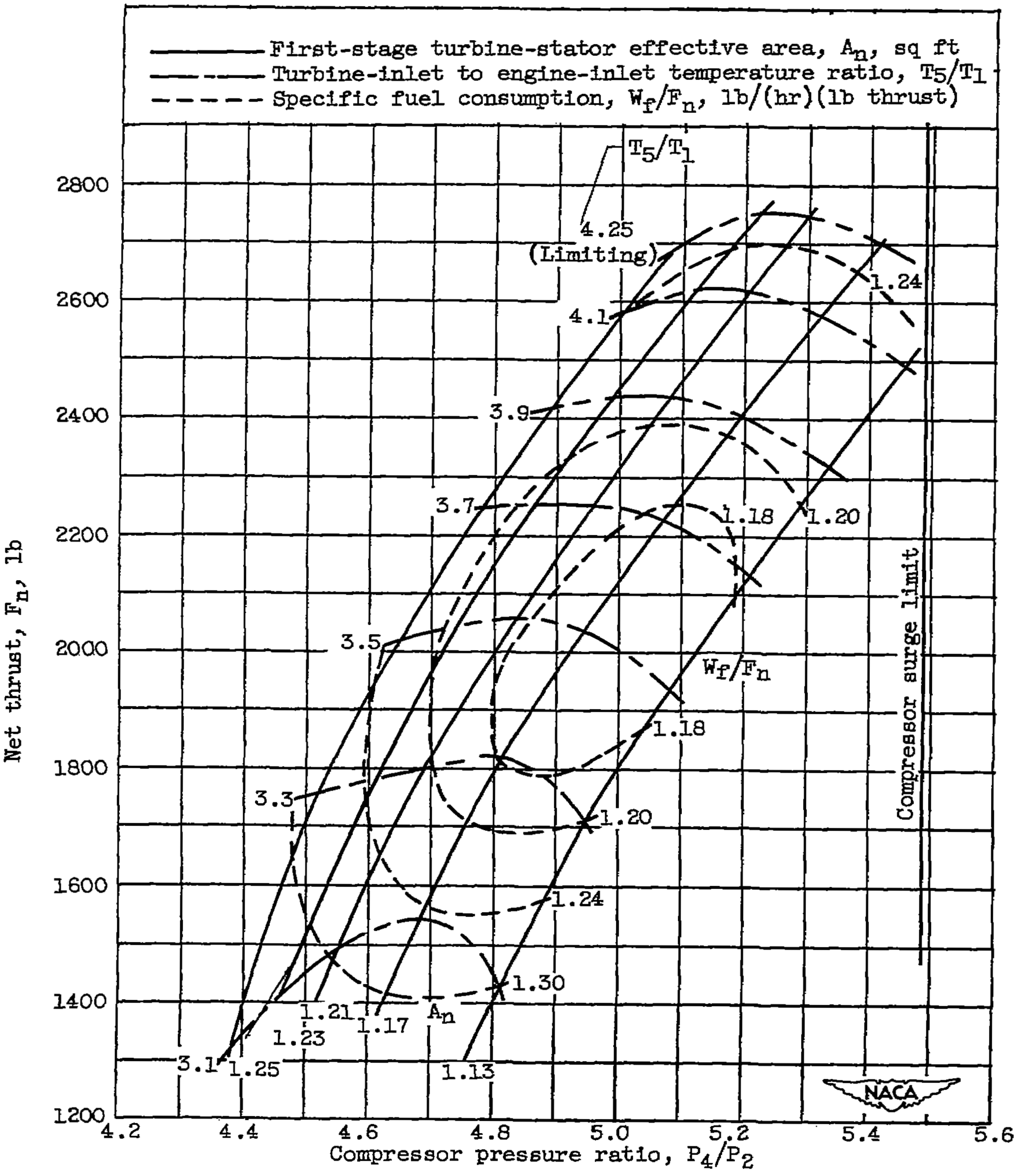
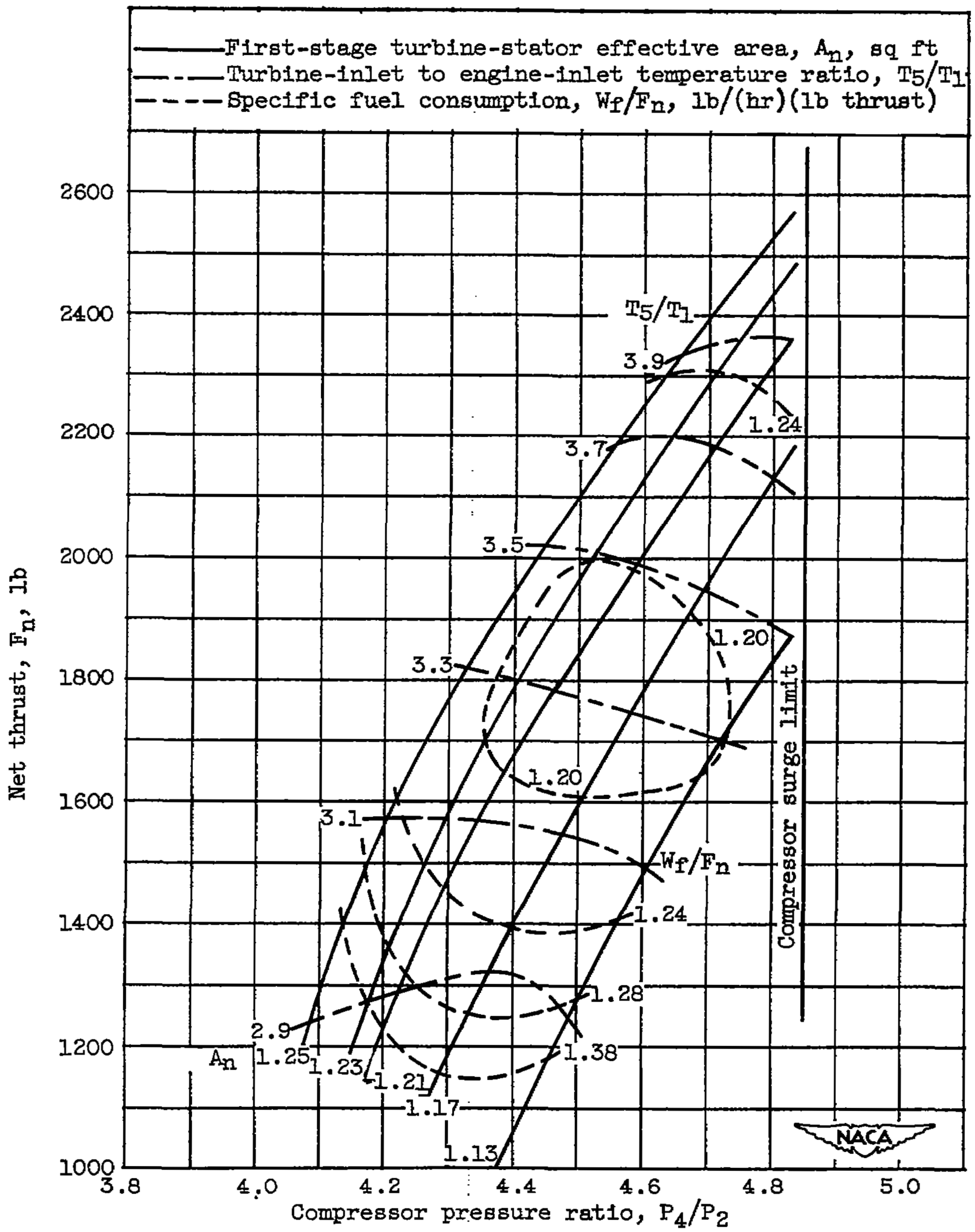
2792



(c) Engine speed, 6800 rpm; corrected engine speed, 7360 rpm.

Figure 4. - Continued. Composite performance plots. Altitude, 30,000 feet; flight Mach number, 0.62.

2792



(d) Engine speed, 6400 rpm; corrected engine speed, 6925 rpm.

Figure 4. - Continued. Composite performance plots. Altitude, 30,000 feet; flight Mach number, 0.62.

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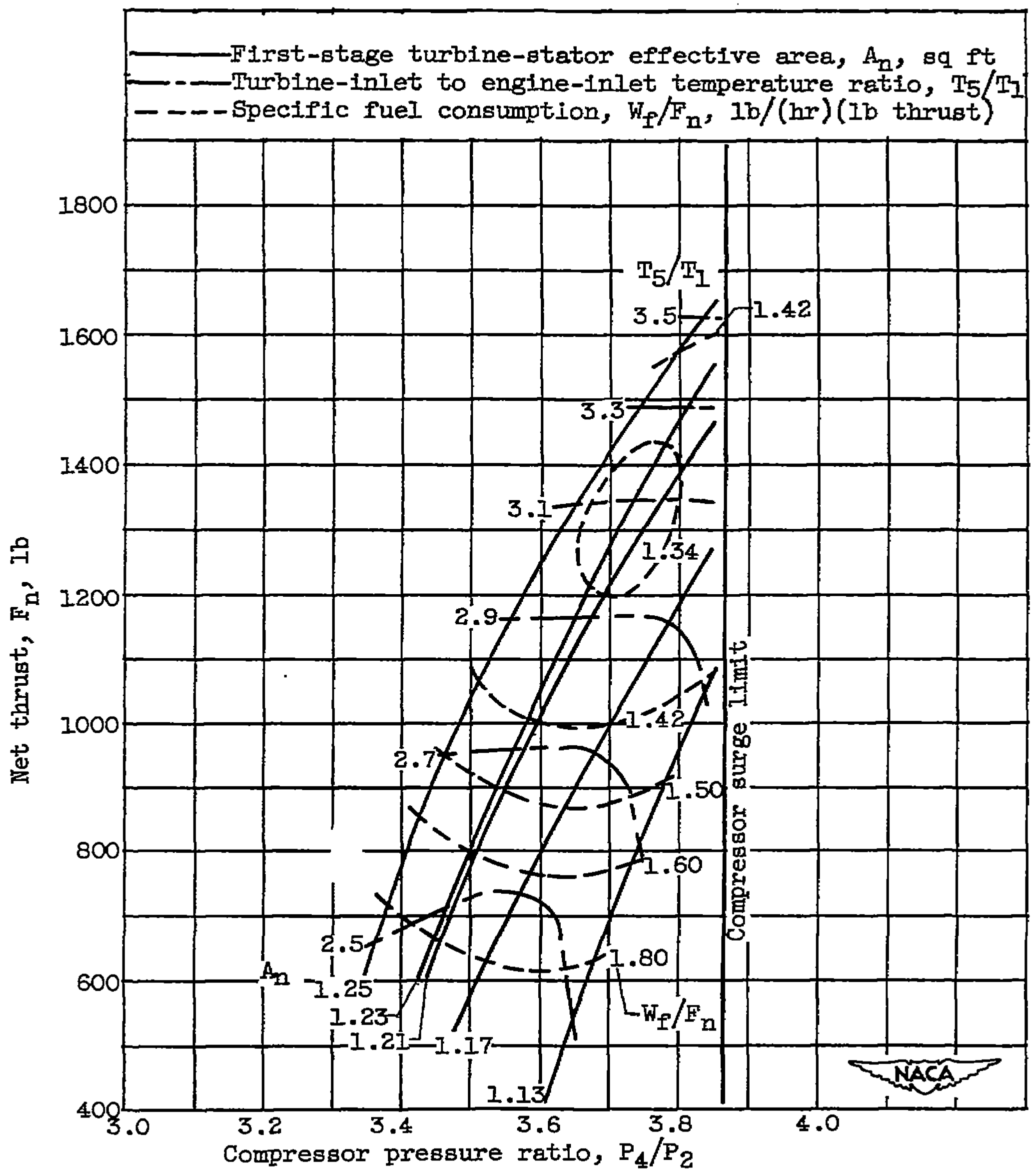
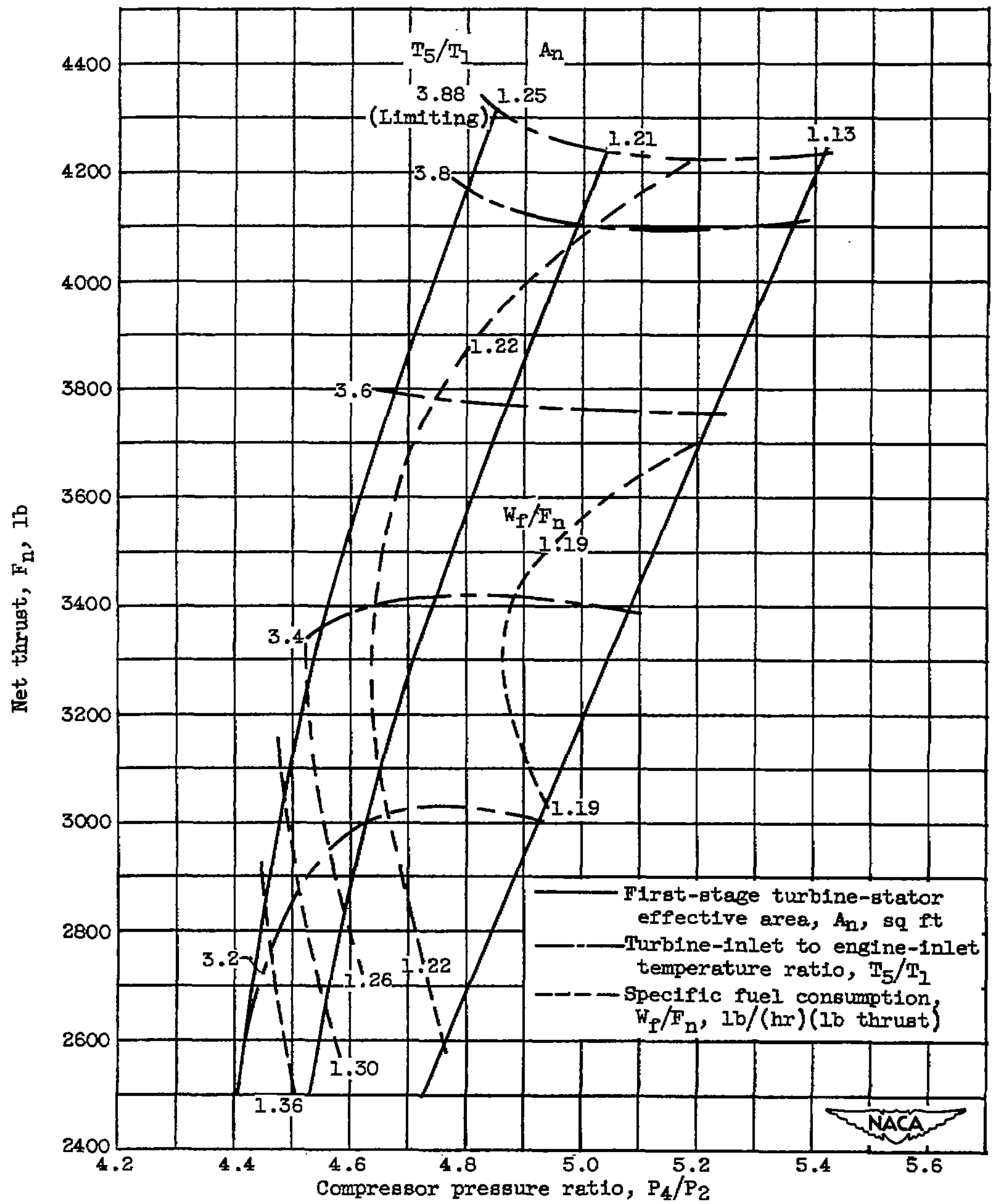
2792



(e) Engine speed, 5800 rpm; corrected engine speed, 6275 rpm.

Figure 4. - Concluded. Composite performance plots. Altitude, 30,000 feet; flight Mach number, 0.62.

2792



(a) Engine speed, 7260 rpm; corrected engine speed, 7510 rpm.

Figure 5. - Composite performance plots. Altitude, 15,000 feet; flight Mach number, 0.46.

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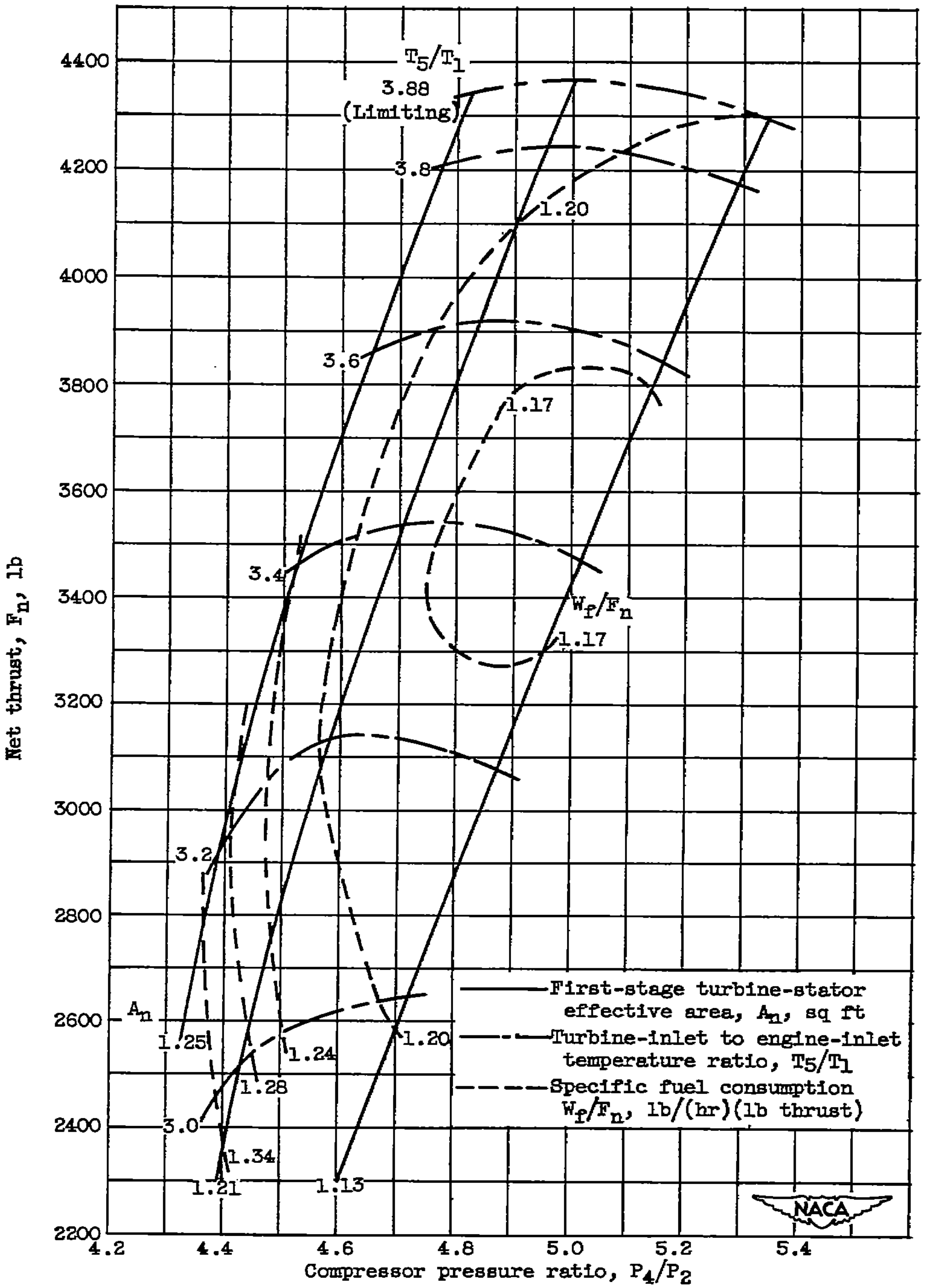
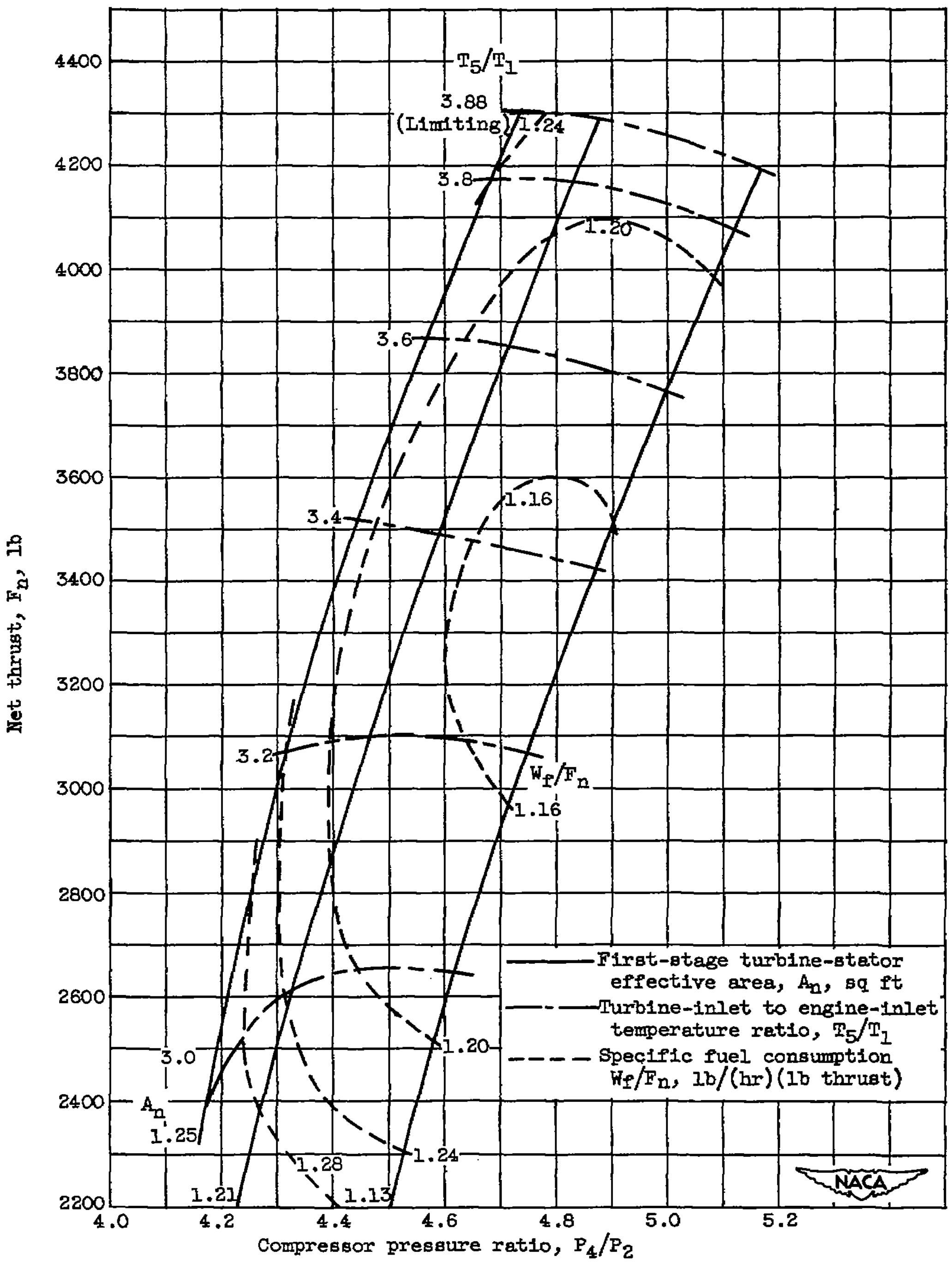


(b) Engine speed, 7050 rpm; corrected engine speed, 7290 rpm.

Figure 5. - Continued. Composite performance plots. Altitude, 15,000 feet; flight Mach number, 0.46.

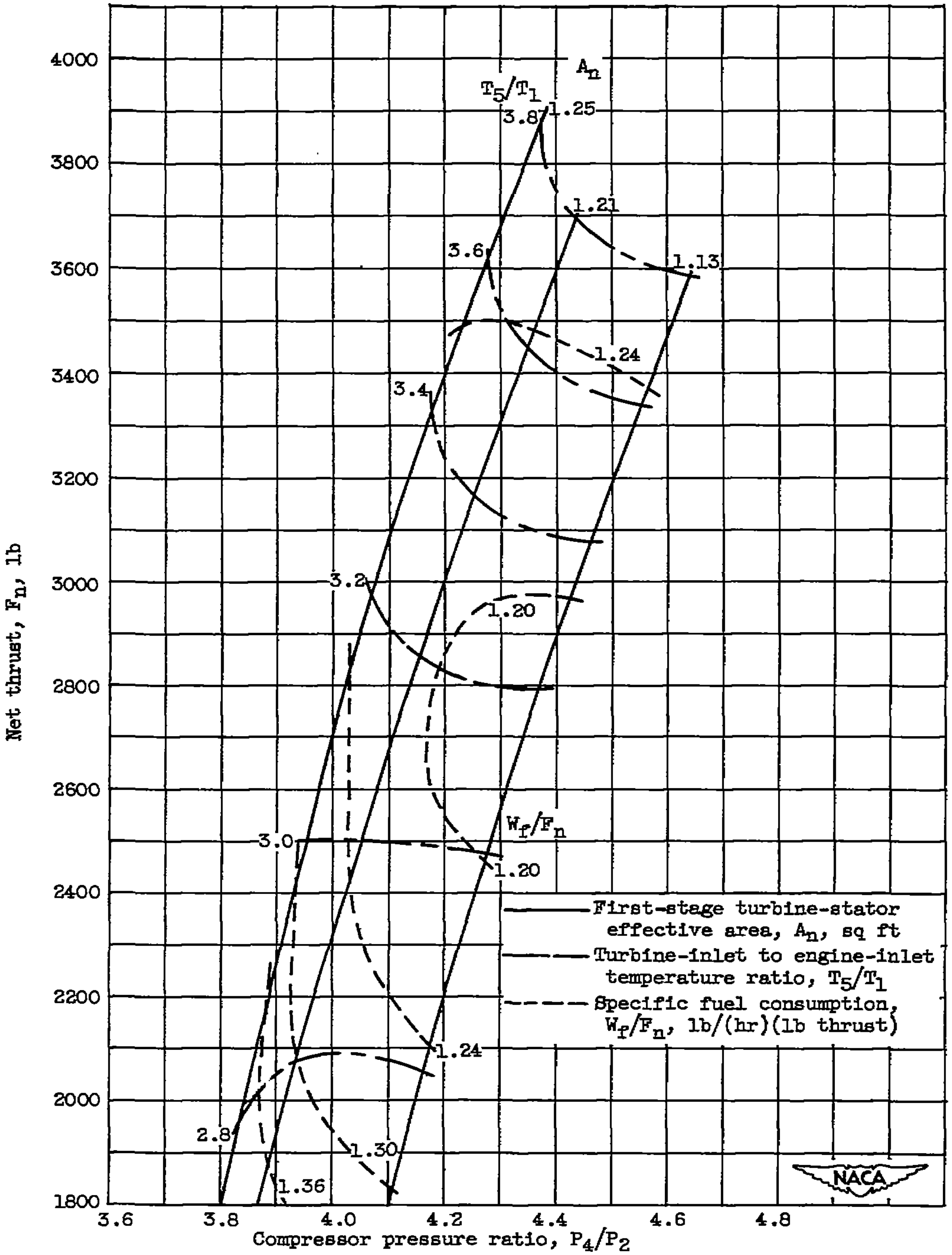
2792



(c) Engine speed, 6800 rpm; corrected engine speed, 7035 rpm.

Figure 5. - Continued. Composite performance plots. Altitude, 15,000 feet; flight Mach number, 0.46.

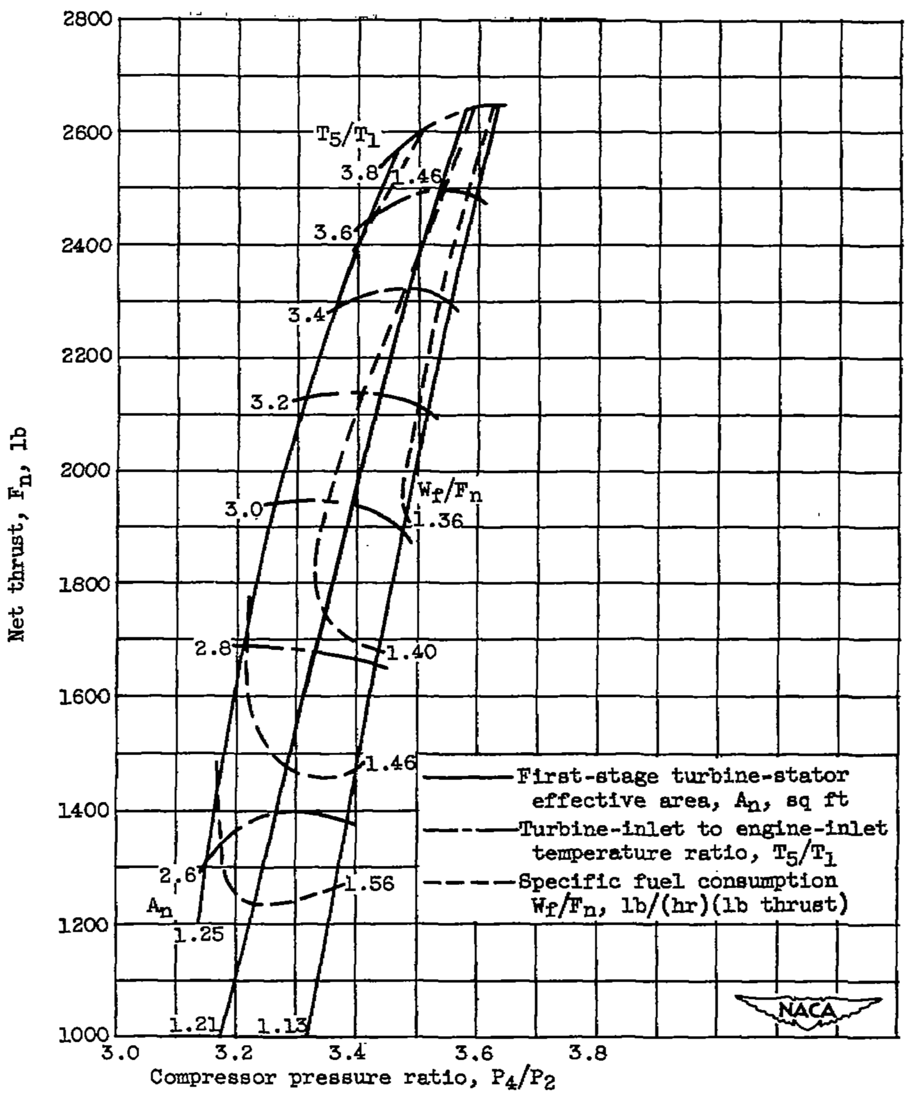
2792



(d) Engine speed, 6400 rpm; corrected engine speed, 6620 rpm.

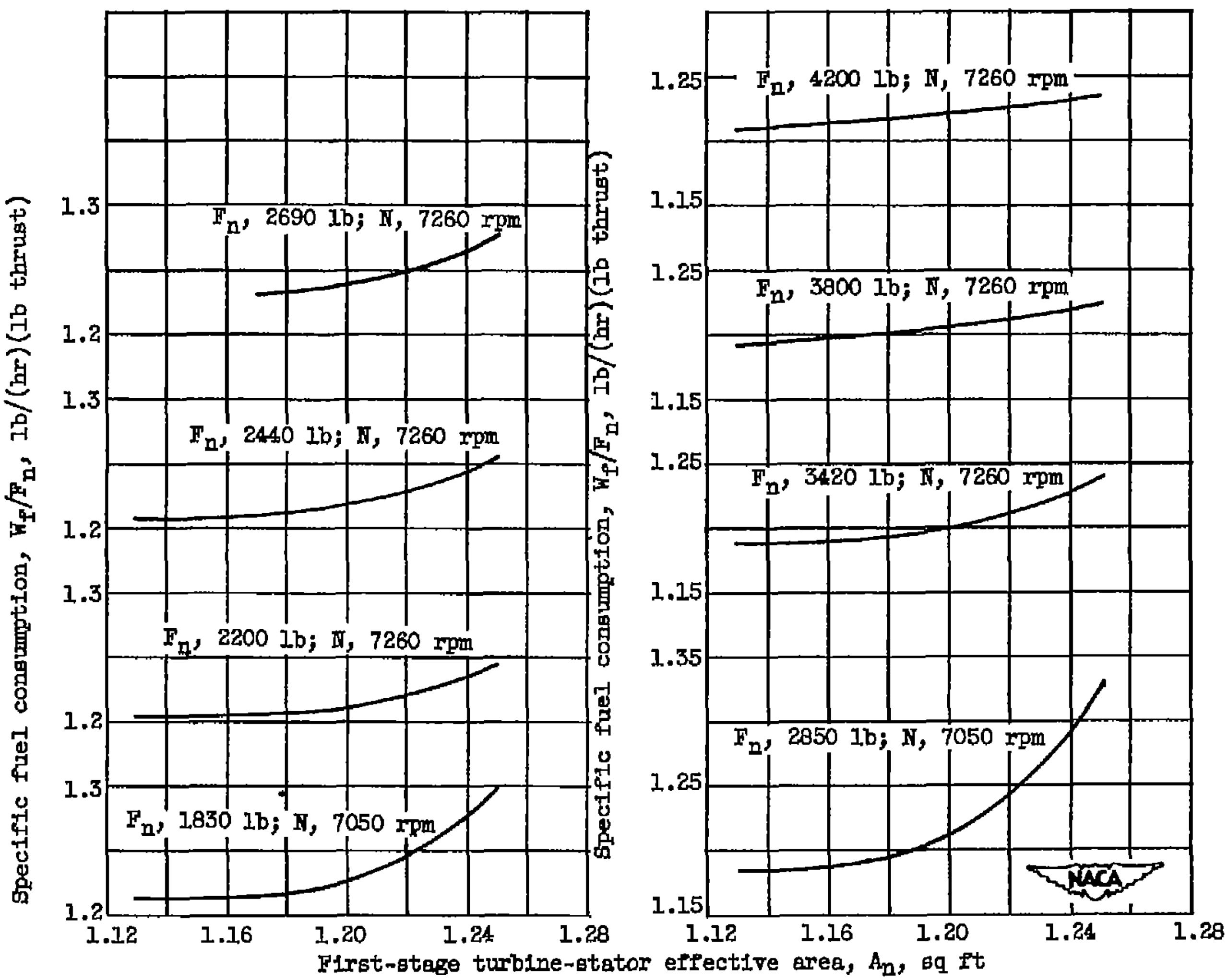
Figure 5. - Continued. Composite performance plots. Altitude, 15,000 feet; flight Mach number, 0.46.

2792



(e) Engine speed, 5800 rpm; corrected engine speed, 6000 rpm.

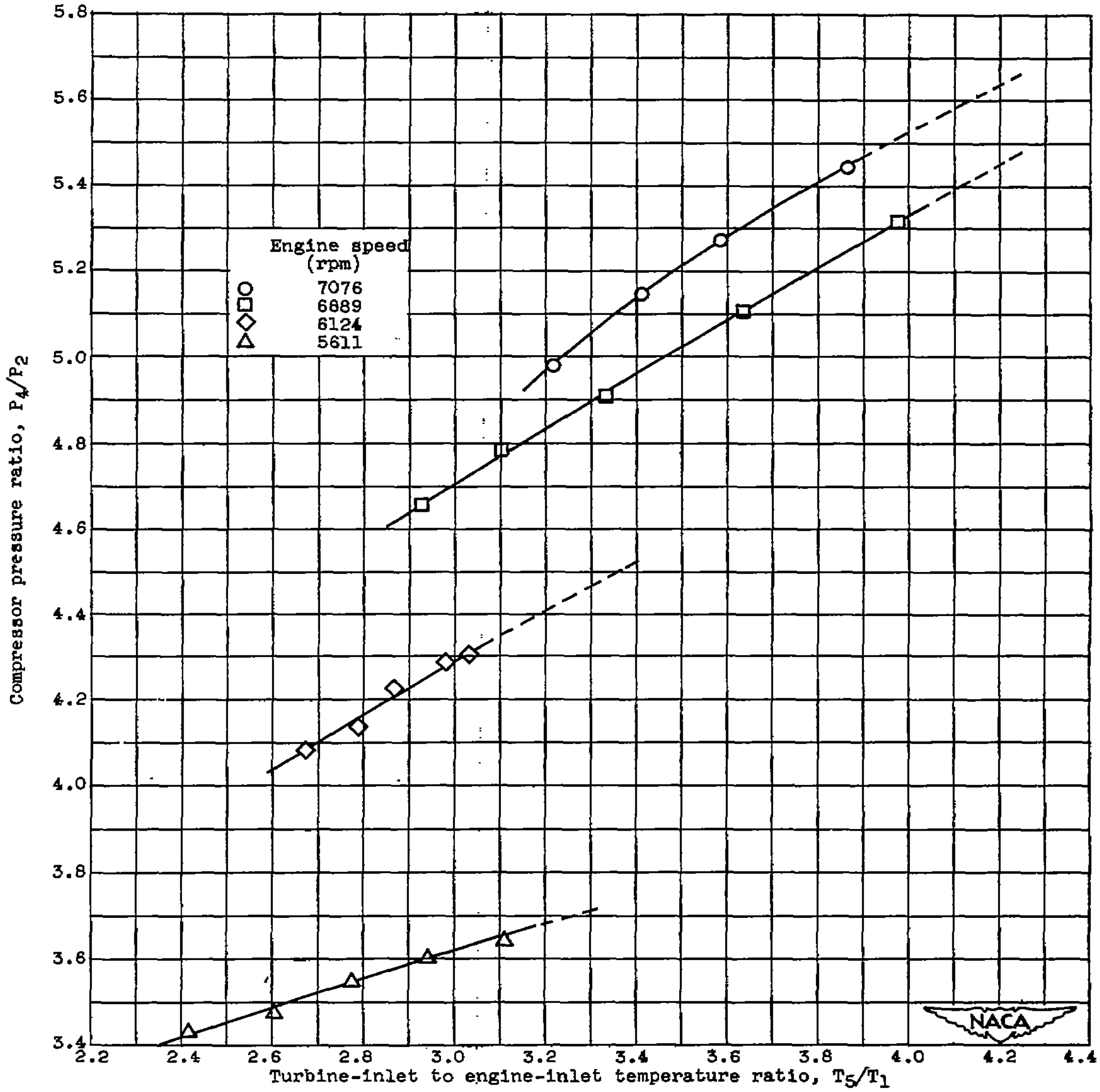
Figure 5. - Concluded. Composite performance plots. Altitude, 15,000 feet; flight Mach number, 0.46.



(a) Altitude, 30,000 feet; flight Mach number, 0.62.

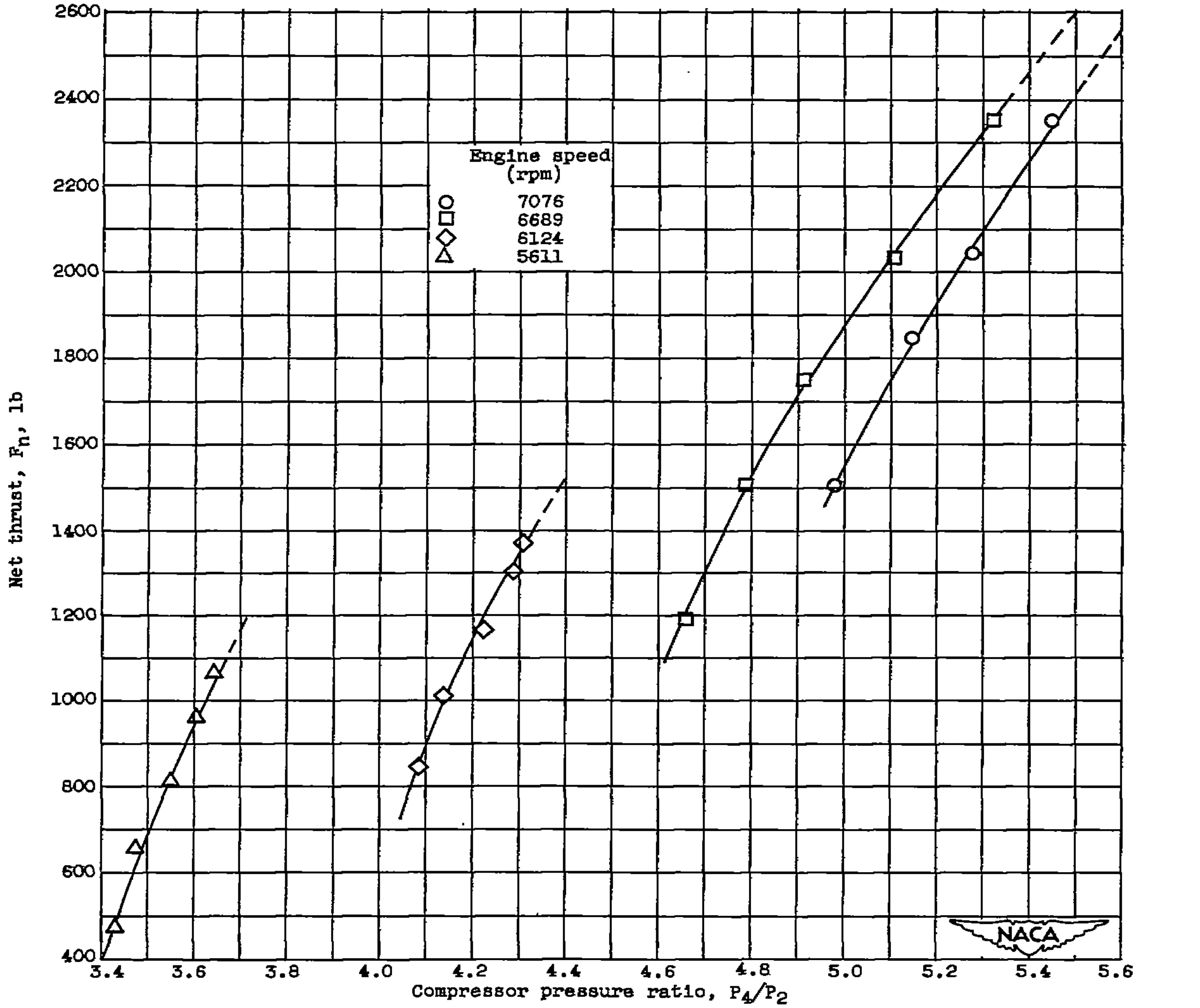
(b) Altitude, 15,000 feet; flight Mach number, 0.46.

Figure 6. - Variation of specific fuel consumption with first-stage turbine-stator effective area at constant thrusts.



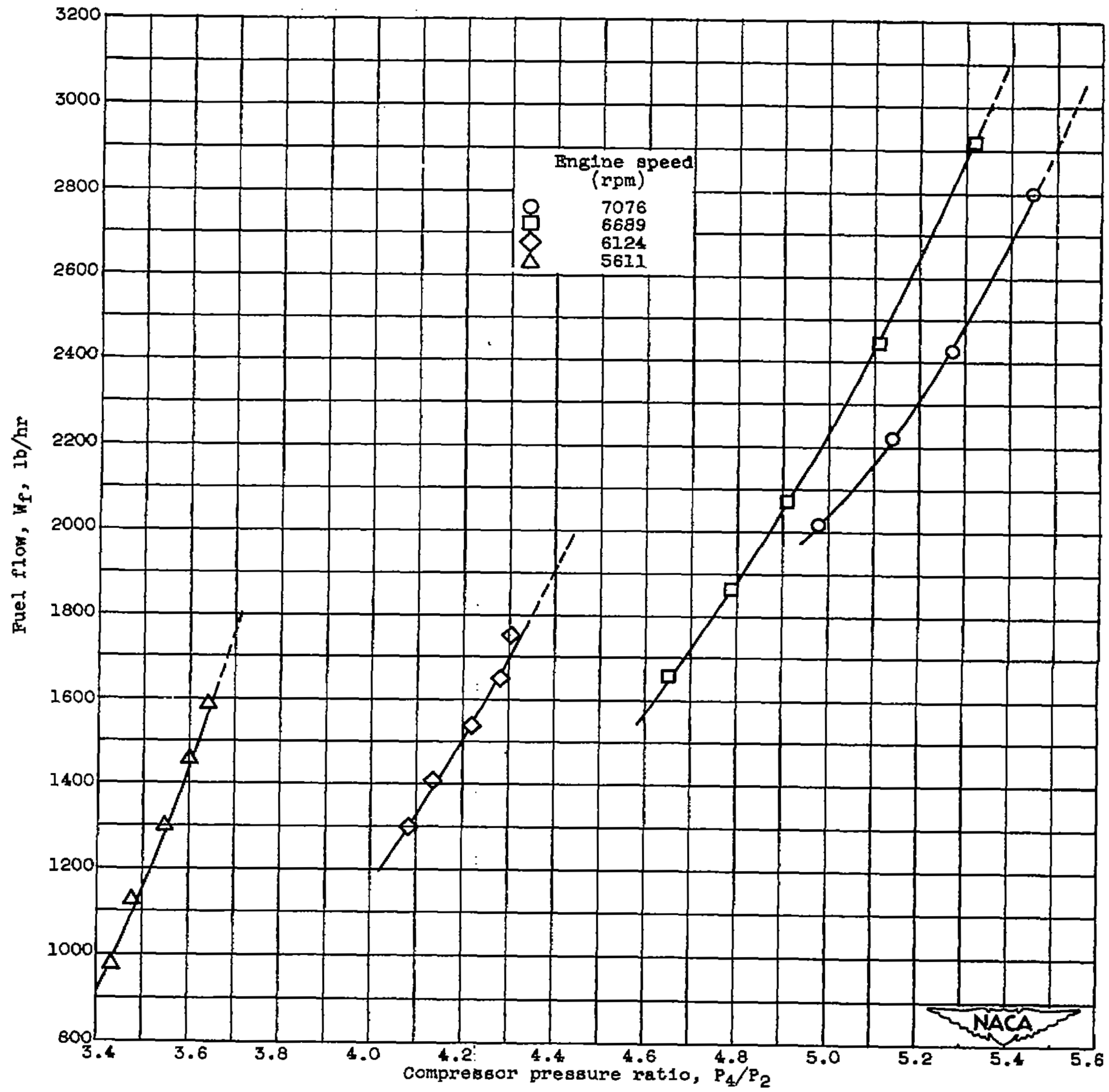
(a) Variation of compressor pressure ratio with turbine-inlet to engine-inlet temperature ratio.

Figure 7. - Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 30,000 feet; flight Mach number, 0.62.



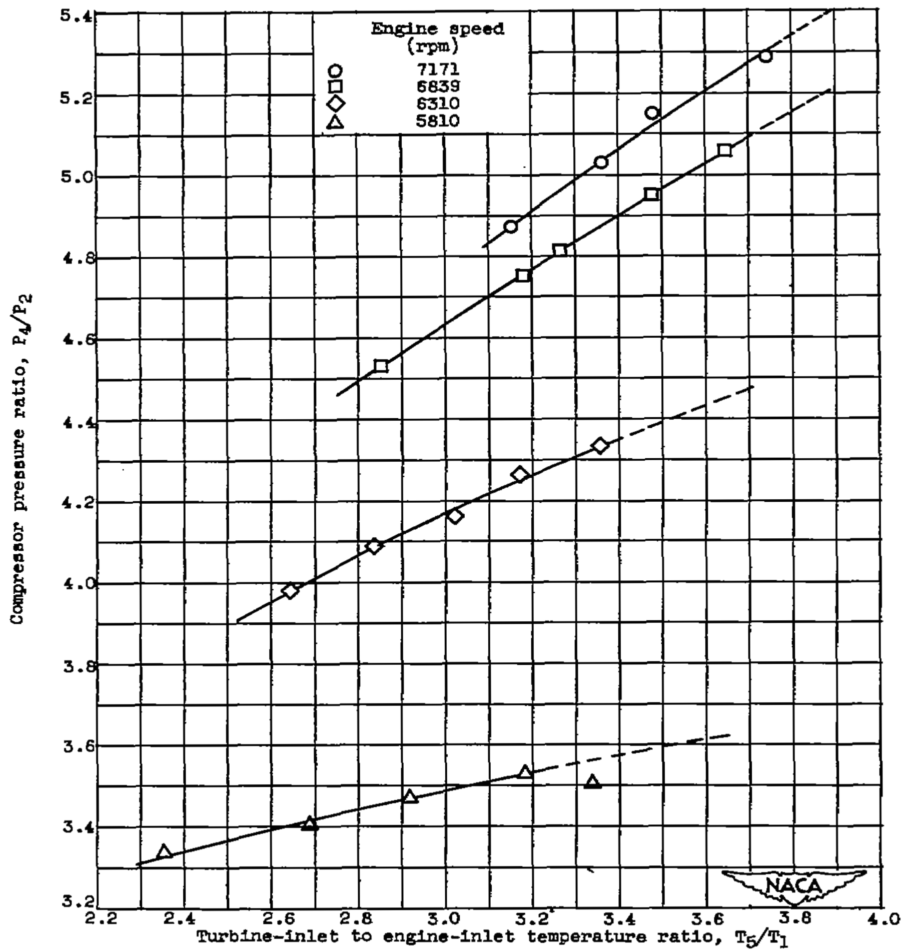
(b) Variation of net thrust with compressor pressure ratio.

Figure 7. - Continued. Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 30,000 feet; flight Mach number, 0.62.



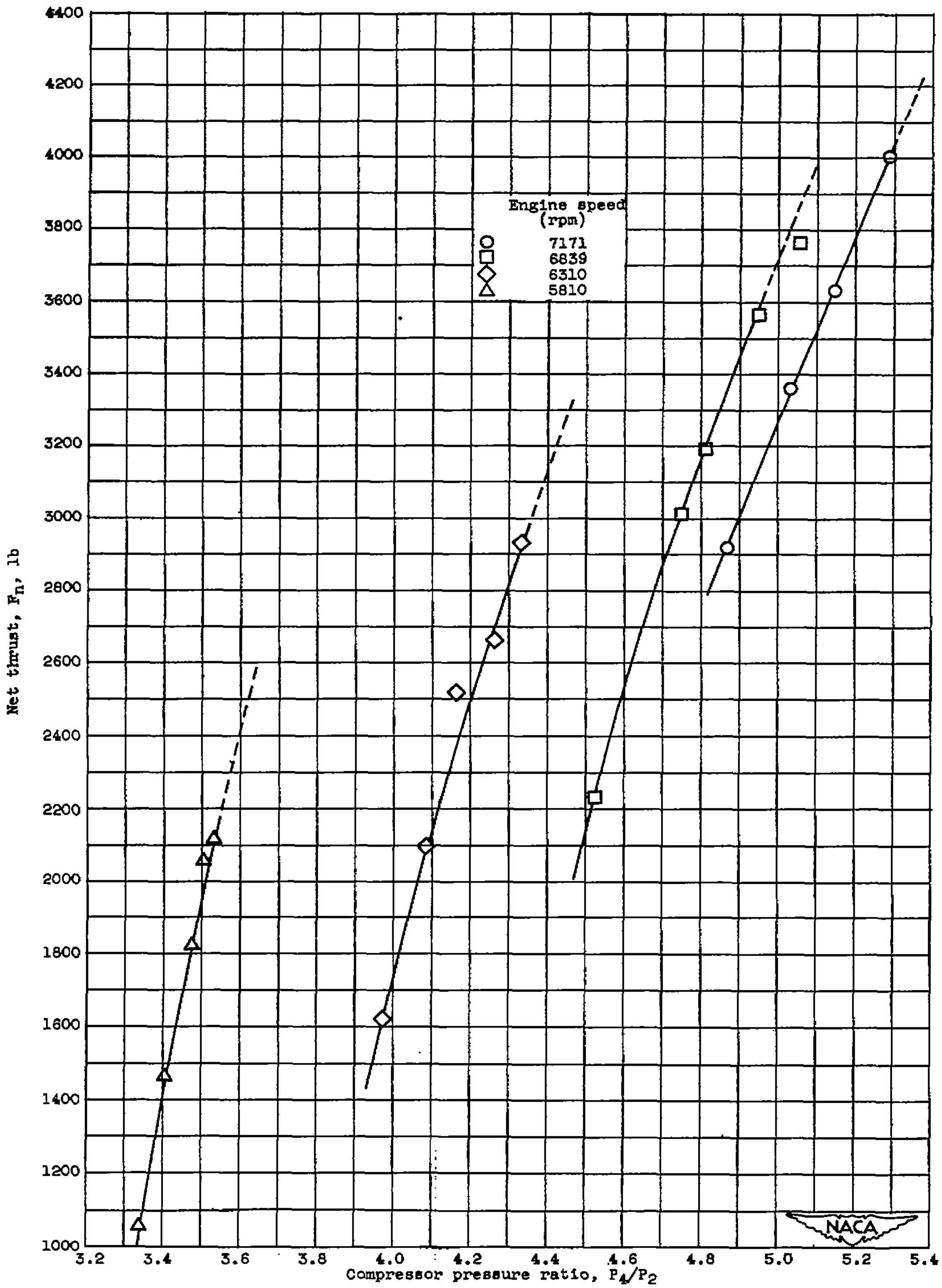
(c) Variation of fuel flow with compressor pressure ratio.

Figure 7. - Concluded. Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 30,000 feet; flight Mach number, 0.62.



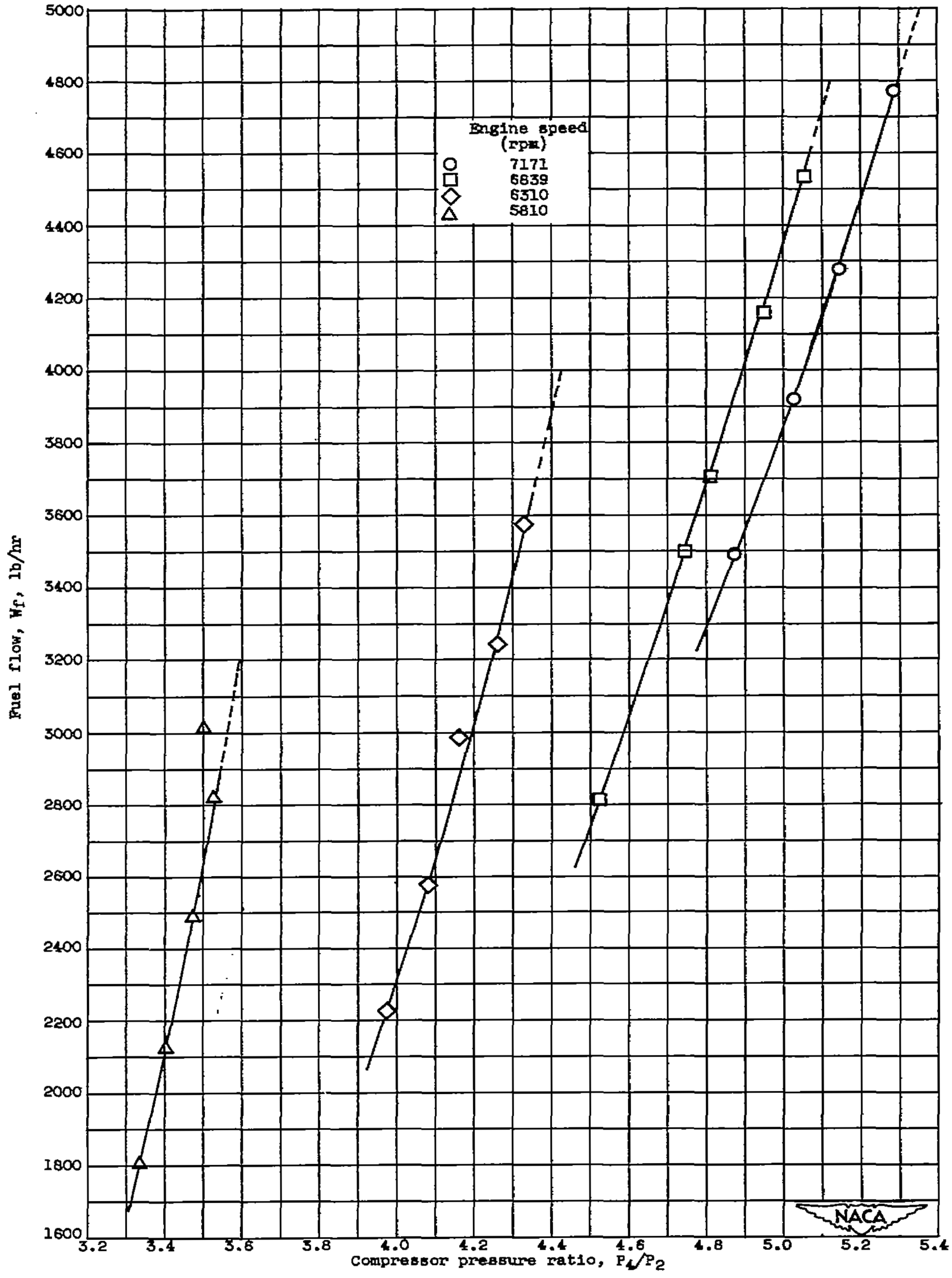
(a) Variation of compressor pressure ratio with turbine-inlet to engine-inlet temperature ratio.

Figure 8. - Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 15,000 feet; flight Mach number, 0.46.



(b) Variation of net thrust with compressor pressure ratio.

Figure 8. - Continued. Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 15,000 feet; flight Mach number, 0.46.



(c) Variation of fuel flow with compressor pressure ratio.

Figure 8. - Concluded. Effect of engine speed on engine performance parameters. First-stage turbine-stator effective area, 1.13 square feet; altitude, 15,000 feet; flight Mach number, 0.46.

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