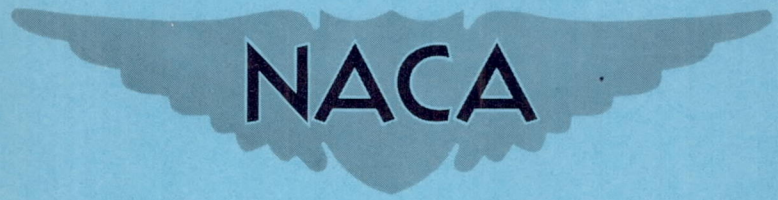


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RESEARCH MEMORANDUM

COMPARISON OF PERFORMANCE AND COMPONENT FRONTAL
AREAS OF HYPOTHETICAL TWO-SPOOL AND
ONE-SPOOL TURBOJET ENGINES

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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RESEARCH MEMORANDUM

COMPARISON OF PERFORMANCE AND COMPONENT FRONTAL AREAS OF HYPOTHETICAL TWO-SPOOL AND ONE-SPOOL TURBOJET ENGINES

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SUMMARY

The performance with afterburning and with afterburner inoperative and the component areas of hypothetical two- and one-spool turbojet engines are compared for two modes of operation over a range of flight conditions. Flight Mach number varies from 0 to 0.9 at sea level and from 0.9 to 2.8 in the stratosphere. Both engines have a design compressor pressure ratio of 7.0. The turbine-inlet temperature is maintained constant at design (2500° R) for modes I and II. For mode I operation the mechanical speeds of the one-spool and of the outer spool of the two-spool engine are assigned constant. For mode II operation the equivalent speeds of these components are assigned to be 110-percent design for all engine-inlet temperatures less than 567° R. For higher inlet temperatures, the mechanical speeds are held constant at 115-percent design. Compressor and turbine frontal areas are calculated from the engine design conditions and assigned aerodynamic limits. Combustor, afterburner, and exhaust-nozzle frontal areas compatible with the assigned limits are calculated over the range of flight conditions.

For mode I operation, the two-spool thrust values are as great as or greater than the one-spool thrust values over the entire flight range considered, while the specific fuel consumption (sfc) for the two engines agrees within 1 percent. The maximum difference in thrust occurs at Mach 2.8, where the two-spool thrust is greater by about 9 percent with the afterburner inoperative and about 6 percent with afterburning. The maximum combustor frontal areas for the two- and one-spool engines are, respectively, 6 and 5 percent larger than the compressor frontal area. The maximum afterburner frontal areas are 19 and 16 percent greater than the compressor frontal area. With afterburner inoperative, the maximum ratio of exhaust-nozzle-exit area for complete expansion to compressor frontal area is 1.92 for the two- and 1.77 for the one-spool engine. With afterburning, these ratios are 2.62 and 2.50.

For mode II operation with afterburning or with the afterburner inoperative, two-spool performance excels over one part of the flight range, while one-spool performance excels over another, the maximum thrust advantage being about 9 percent for the two- and about 7 percent for the

one-spool engine. At sea level, two-spool sfc is up to 5 percent less than one-spool sfc; in the stratosphere, the maximum difference is about 3 percent. The maximum combustor frontal areas for the two- and one-spool engines are, respectively, 11 and 16 percent greater than the compressor frontal area. The maximum afterburner frontal areas are 40 and 37 percent greater than the compressor frontal area. With afterburner inoperative, the maximum ratio of exhaust-nozzle-exit area for complete expansion to compressor frontal area is 2.12 for the two- and 2.20 for the one-spool engine. With afterburning, these ratios are 3.05 and 3.16.

INTRODUCTION

For many turbojet-engine applications, both two- and one-spool configurations are suitable with regard to design performance. The choice for a given application depends on many factors, such as the design values of compressor pressure ratio and turbine-inlet temperature, the range of flight conditions, mode of engine operation, off-design performance requirements, performance requirements during transients, and the size and weight of the engine components.

This report compares the performance and component frontal areas of a hypothetical two-spool turbojet engine with those of a hypothetical one-spool turbojet engine. Both engines have a design over-all compressor pressure ratio of 7.0 and a design turbine-inlet temperature of 2500° R. The design pressure ratio of the outer and inner compressors of the two-spool engine is 2.646.

While the design performance of these two engines is the same, the off-design performance may differ to some extent because of different off-design variations in compressor equivalent weight flow, compressor total-pressure ratio, and compressor and turbine efficiencies. The two-spool engine is expected to pass more weight flow than the one-spool engine at the lower equivalent engine speeds. The performance maps of six compressors having design pressure ratios from 2.08 to 9.20 are generalized in reference 1, where the design pressure ratio corresponds to peak polytropic efficiency. One of the generalized curves shows the variation of compressor equivalent weight flow at maximum efficiency with equivalent speed for different design compressor pressure ratios. At equivalent speeds of design and greater than design, the maximum-efficiency equivalent weight flow is independent of design pressure ratio. At equivalent speeds less than design, the lower-design-pressure-ratio compressors pass more equivalent weight flow than the higher-design-pressure-ratio compressors. For instance, at 70-percent equivalent speed, the maximum-efficiency equivalent weight flow of a compressor having a design pressure ratio of 2.646 is about 17 percent greater than that for a compressor having a design pressure ratio of 7.0.

In an engine, the equivalent weight flow at a particular equivalent speed is not necessarily that for maximum efficiency. Depending on the mode of engine operation, the equivalent weight flow may be greater or less than that for maximum efficiency. The differences in compressor total-pressure ratio and component efficiency between the two- and one-spool engines are expected to have less effect on the engine performance comparison than the difference in equivalent weight flow.

Each of the two hypothetical engines investigated is operated over a range of flight conditions for two modes of operation. At sea level, flight Mach number is varied from 0 to 0.9; in the stratosphere, from 0.9 to 2.8. For both operating modes, the turbine-inlet temperature is maintained constant at 2500° R. For mode I operation, the mechanical speed of the one-spool engine and the outer-spool mechanical speed of the two-spool engine are assigned to be constant. For mode II operation, the equivalent speed of the one-spool engine and the outer-spool equivalent speed of the two-spool engine are assigned to be 110-percent design for all engine-inlet temperatures less than 567° R. For higher inlet temperatures, the mechanical speeds of the one-spool engine and of the outer spool of the two-spool engine are held constant at 115-percent design.

METHOD OF ANALYSIS

After the compressor and turbine performance maps are obtained analytically, the gas-generator pumping characteristics of the two engines are obtained by matching the compressor, combustor, and turbine components. Engine performance with afterburning and with the afterburner inoperative is calculated from the pumping characteristics for two modes of operation over a range of flight conditions. Operating lines are located on the compressor and turbine maps of both engines, and the variation of engine weight flow with flight condition is found.

The compressor and turbine frontal areas are calculated from the engine design conditions and selected aerodynamic limits. Aerodynamic limits are selected for the combustor and afterburner, and the frontal area of these components consistent with the selected limits are calculated over the range of flight conditions. For operation with afterburning and with afterburner inoperative over the complete range of flight conditions, the required variations in exhaust-nozzle-throat and -exit area are calculated for each engine.

Description of Engines

Design conditions. - A cross section of a two-spool turbojet engine with afterburner is shown in figure 1(a) together with the axial stations.

(Stations and other symbols are defined in appendix A.) The two-spool engine design conditions for sea-level operation at a Mach number of zero (identical with those for the engine of ref. 2) are as follows:

Over-all compressor pressure ratio	7.0
Outer-compressor equivalent weight flow, lb/sec	150
Outer- and inner-compressor pressure ratio	2.646
Outer- and inner-compressor polytropic efficiency, percent	90
Inner-turbine-inlet temperature, °R	2500
Inner-turbine equivalent specific work, Btu/lb	12.8
Inner- and outer-turbine adiabatic efficiency, percent	87
Outer-turbine equivalent specific work, Btu/lb	10.3
Afterburner temperature, °R	3500

Figure 1(b) shows a cross section of a one-spool turbojet engine with afterburner. The one-spool engine design conditions for sea-level operation at a Mach number of zero are as follows:

Compressor pressure ratio	7.0
Compressor equivalent weight flow, lb/sec	150
Compressor and turbine adiabatic efficiency, percent	87
Turbine-inlet temperature, °R	2500
Turbine equivalent specific work, Btu/lb	21.9
Afterburner temperature, °R	3500

Component performance. - The compressor maps of the two-spool engine shown in figures 2(a) and (b) were obtained in three steps: (1) the "backbone" or maximum-efficiency line was calculated; (2) the compressor stall-limit line was calculated; and (3) constant-speed lines were constructed. This procedure and the required curves are presented in reference 1. The compressor map of the one-spool engine shown in figure 2(c) was derived from the compressor map presented in reference 3. The design point on figure 2(c) corresponds to a pressure ratio of 7.0 at 90-percent design speed on the map of reference 3.

The inner- and outer-turbine maps of the two-spool engine are shown in figures 2(d) and (e), respectively. Figure 2(f) shows the turbine map of the one-spool engine. All three maps were obtained from the one-stage turbine performance map of reference 4, as described in reference 1.

The inlet performance is shown in figure 2(g), which is a calculated plot of total-pressure-recovery ratio P_1/P_0 against flight Mach number M_0 for a variable-geometry inlet having two adjustable wedges and a bypass duct.

Pumping Characteristics

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The pumping characteristics of a two-spool gas generator, as defined in this report, are the relations among the following quantities: T_4/T_1 , $N_o/\sqrt{\theta_1}$, $N_i\sqrt{\theta_1}$, $w_1\sqrt{\theta_6}/\delta_6$, P_6/P_1 , T_6/T_1 , and f/θ_1 . For the one-spool engine, the quantities are: T_4/T_1 , $N/\sqrt{\theta_1}$, $w_1\sqrt{\theta_6}/\delta_6$, P_6/P_1 , T_6/T_1 , and f/θ_1 . Pumping characteristics are obtained by plotting the compressor and turbine component performance in terms of quantities suitable for matching and then relating the compressor, combustor, and turbine performances so that the matching relations are satisfied. The matching procedures, which are similar to those described in reference 5, are presented in appendix B.

Modes of Engine Operation

Operation is considered for flight Mach numbers from 0 to 0.9 at sea level and from 0.9 to 2.8 in the stratosphere. Before the performance of a turbojet engine at a given flight Mach number and altitude can be calculated, the manner in which the engine is operated must be specified. For equilibrium operation, two engine quantities must be specified. Two modes of operation are considered for the two- and one-spool turbojets of this report.

Two spool. - At a particular outer-spool equivalent speed, engine thrust increases with inner-turbine-inlet temperature. For both operating modes, the inner-turbine-inlet temperature is assigned to be 2500° R, the design value, for all flight conditions. For mode I operation, the second quantity assigned is the mechanical speed of the outer spool. For all flight conditions, the mechanical speed is assigned to be constant at its design value.

At any flight condition, for a fixed turbine to compressor temperature ratio T_4/T_1 , engine thrust generally increases with outer-spool equivalent speed. (At some high value of outer-spool equivalent speed, a further increase in speed results in no increase in equivalent weight flow and a decrease in compressor efficiency, such that engine thrust decreases.) For mode II operation, outer-spool equivalent speed, which is the second engine quantity to be specified, is assigned to be constant at 110-percent design for all values of engine-inlet temperature less than 567° R. At sea level, engine-inlet temperature increases from 518.7° to 567° R as flight Mach number increases from 0 to 0.68. In the stratosphere, engine-inlet temperature is less than 567° R for all flight Mach numbers less than 1.51. For all inlet temperatures greater than 567° R (for flight Mach numbers greater than 0.68 at sea level and 1.51 in the stratosphere), the mechanical speed of the outer spool is held constant

at 115-percent design. At an inlet temperature of 567° R, a mechanical speed of 115-percent design corresponds to an equivalent speed of 110-percent design.

One spool. - The two modes of operation for the one-spool engine are similar to those for the two-spool engine. For both operating modes, the turbine-inlet temperature is assigned to be 2500° R, the design value, for all flight conditions. For mode I operation, the mechanical speed is assigned to be constant at its design value for all flight conditions. For mode II operation, the equivalent speed is assigned to be constant at 110-percent design for all values of engine-inlet temperature less than 567° R. For higher inlet temperatures, the mechanical speed is held constant at 115-percent design.

Component Operating Lines

The variations of T_1 and θ_1 with flight condition were calculated from

$$T_1 = t_0 \left(1 + \frac{\gamma-1}{2} M_0^2 \right) \quad (1)$$

$$\theta_1 = T_1 / 518.7 \quad (2)$$

where t_0 is the ambient temperature at the specified altitude and M_0 is the specified flight Mach number.

For the two modes of operation, the variations of T_4/T_1 and $N/\sqrt{\theta_1}$ or $N_0/\sqrt{\theta_1}$ with flight condition are calculated from the specified values of T_4 ; N , N_0 , or $N_0/\sqrt{\theta_1}$; and the calculated values of T_1 and θ_1 . The operating lines are located on the component maps of the two-spool engine from the variations of $N_0/\sqrt{\theta_1}$ and T_4/T_1 with flight condition and gas-generator plots of P_2/P_1 , $w_1\sqrt{\theta_1}/\delta_1$, P_3/P_2 , $w_2\sqrt{\theta_2}/\delta_2$, $(H_4 - H_5)/\theta_4$, w_4N_1/δ_4 , $(H_5 - H_6)/\theta_5$, and w_5N_0/δ_5 against T_4/T_1 with $N_0/\sqrt{\theta_1}$ as parameter. For the one-spool engine, plots of P_3/P_1 , $w_1\sqrt{\theta_1}/\delta_1$, $(H_4 - H_6)/\theta_4$, and w_4N/δ_4 against T_4/T_1 with $N/\sqrt{\theta_1}$ as parameter were used to locate the operating lines on the compressor and turbine maps.

Engine Performance

Thrust and specific-fuel-consumption values are calculated for operation with afterburning and with afterburner inoperative. The afterburning temperature is assigned to be constant at its design value, 3500° R.

In calculating engine performance, the values of two pumping-characteristic quantities must be found first. For each flight condition, the compressor equivalent speed, $N_o/\sqrt{\theta_1}$ or $N/\sqrt{\theta_1}$, and turbine to compressor temperature ratio T_4/T_1 are found, as previously discussed.

The following procedure is used to calculate thrust and specific-fuel-consumption values:

(1) Values of $w_1\sqrt{\theta_6}/\delta_6$, P_6/P_1 , T_6/T_1 , and f/θ_1 are read from the gas-generator pumping-characteristic curves for the known values of T_4/T_1 and equivalent speed ($N/\sqrt{\theta_1}$ or $N_o/\sqrt{\theta_1}$).

(2) Inlet total-pressure-recovery ratio P_1/P_0 is read from figure 2(g).

(3) Exhaust-nozzle pressure ratio is calculated from

$$\frac{P_7}{P_0} = \frac{P_0}{P_0} \frac{P_1}{P_0} \frac{P_6}{P_1} \frac{P_7}{P_6} \quad (3)$$

where $P_0/P_0 = \left(1 + \frac{\gamma-1}{2} M_0^2\right)^{\frac{\gamma}{\gamma-1}}$, and P_7/P_6 is 0.96 for inoperative-afterburner operation and 0.94 for afterburning operation.

(4) Jet velocity is calculated from

$$V_j = C_V \sqrt{\frac{2gR\gamma_7}{\gamma_7-1} T_7 \left[1 - \left(\frac{P_0}{P_7}\right)^{\frac{\gamma_7-1}{\gamma_7}}\right]} \quad (4)$$

where $C_V = 0.96$, $T_7 = T_6$ for inoperative-afterburner operation, and $T_7 = 3500^\circ \text{R}$ for afterburning operation.

(5) Net thrust is calculated from

$$F_n = \frac{w_1}{g} \left[(1 + f)V_j - V_0 \right] \quad (5)$$

where w_1 is calculated from values of $w_1 \sqrt{\theta_6/\delta_6}$, T_6/T_1 , P_6/P_1 , P_1 , and T_1 ; V_0 is calculated from

$$V_0 = M_0 \sqrt{\gamma g R t_0} \quad (6)$$

and f is calculated from values of f/θ_1 and θ_1 for inoperative-afterburner operation. For afterburning operation, the value of fuel-air ratio for the afterburner is found by using figure 2 of reference 6. In calculating the total fuel-air ratio, combustor efficiency is taken equal to 0.95, and afterburner efficiency equal to 0.90. The fuel is assumed to have a lower heating value of 18,700 Btu per pound and a hydrogen-carbon ratio of 0.175.

(6) Specific fuel consumption is calculated from

$$\text{sfc} = \frac{3600f}{\frac{F_n}{w_1}} \quad (7)$$

Component Areas

Two spool. - The component frontal areas of the two-spool engine were computed for the following design values:

Air flow per unit frontal area at station 1, (lb/sec)/sq ft	35
Outer-compressor tip speed, ft/sec	1100
Outer-compressor-entrance axial Mach number	0.6
Ratio of inner-compressor first-rotor tip relative Mach number to outer-compressor first-rotor tip relative Mach number	1
Ratio of inner-compressor-inlet axial velocity to outer-compressor- inlet axial velocity, $V_{x,2}/V_{x,1}$	1
Inner- and outer-compressor-exit tangential velocity, ft/sec	0
Inner-turbine-exit axial-velocity ratio, $V_{x,5}/a_{cr,5}$	0.5
Number of inner- and outer-turbine stages	1
Inner-turbine loading parameter, $Jg(H_4 - H_5)/U_{h,5}^2$	2.1
Outer-turbine-exit axial-velocity ratio, $V_{x,6}/a_{cr,6}$	0.5
Outer-turbine loading parameter, $Jg(H_5 - H_6)/U_{h,6}^2$	2.1
Primary-combustor reference velocity, ft/sec	200
Primary-combustor hub-tip radius ratio	0.4
Afterburner velocity, ft/sec	550

Outer-compressor frontal area was calculated from the values of air flow and air flow per unit frontal area; inner-compressor frontal area

was assigned equal to outer-compressor frontal area. Values of outer-compressor tip speed and entrance axial Mach number yielded values of rotor tip relative Mach number and hub-tip radius ratio. Guide vanes were not used. Inner-spool tip speed was calculated from the values of temperature, axial velocity, and rotor tip relative Mach number at the entrance of the inner spool.

From cycle calculations and the selected value of turbine-exit axial-velocity ratio, the exit annulus area of each turbine was computed. From values of turbine specific work, stage-loading parameter, and angular velocity, the hub radius of each turbine was calculated. This value, together with the value of annulus area, determined the frontal area of each turbine; constant rotor tip radius was assumed for each turbine. Rotor blade hub centrifugal stresses were calculated for assumed values of blade-taper factor and density of material of 0.7 and 490 pounds per cubic foot, respectively.

The primary-combustor frontal area compatible with the specified values of hub-tip radius ratio and reference velocity was calculated for the full range of flight conditions as described in reference 2. Afterburner frontal area was determined for the specified afterburner velocity. For operation with afterburning and with afterburner inoperative over the full range of flight conditions, exhaust-nozzle-throat and -exit areas for complete expansion were calculated as discussed in reference 2.

One spool. - The component frontal areas of the one-spool engine were computed for the following design values:

Air flow per unit frontal area at station 1, (lb/sec)/sq ft	35
Compressor tip speed, ft/sec	1100
Compressor-entrance axial Mach number	0.6
Number of turbine stages	2
Turbine loading parameter, $Jg(H_4 - H_6)/U_{h,6}^2$	6.3
Turbine-exit axial-velocity ratio, $V_{x,6}/a_{cr,6}$	0.5
Primary-combustor reference velocity, ft/sec	200
Primary-combustor hub-tip radius ratio	0.4
Afterburner velocity, ft/sec	550

The frontal areas of the compressor, turbine, primary combustor, and afterburner, and both the throat and exit areas of the exhaust nozzle were computed as for the two-spool engine.

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RESULTS AND DISCUSSION

Component Operating Lines

The two- and one-spool operating lines for the two modes of operation over the full range of flight conditions considered are shown on the compressor and turbine maps in figure 3.

Mode I operation. - The two-spool operating line on the outer-compressor map (fig. 3(a)) for mode I operation parallels the stall-limit line and passes through the maximum-efficiency region. The slope of the inner-compressor operating line (fig. 3(b)) is greater than the slope of the maximum-efficiency line, so that the inner compressor operates at maximum efficiency only at speeds $N_1/\sqrt{\theta_2}$ near design. Outer-compressor efficiency varies between 0.85 and 0.88, while inner-compressor efficiency varies between 0.77 and 0.88.

The compressor operating line for the one-spool engine parallels the stall-limit line (fig. 3(c)). At each speed $N/\sqrt{\theta_1}$, compressor efficiency is only slightly less than maximum efficiency. Efficiency varies between 0.80 and 0.87.

Along the inner-turbine operating line (fig. 3(d)), equivalent specific work is constant at its design value, and efficiency varies between 0.87 and 0.88. The outer-turbine operating line (fig. 3(e)) lies along the design-speed $N_0/\sqrt{\theta_5}$ line, and efficiency varies between 0.86 and 0.87. The turbine of the one-spool engine operates at design speed $N/\sqrt{\theta_4}$, and efficiency is nearly constant at 0.87 (fig. 3(f)).

Mode II operation. - The outer-compressor operating line for mode II operation (fig. 3(a)) lies closer to the stall-limit line than does the line of maximum efficiency, because the compressor and turbine components were sized for the zero Mach number sea-level conditions of mode I. The components could be designed for other conditions so that the mode II operating line would pass through the maximum-efficiency region. If this were done, the engine equivalent weight flow at each speed $N_0/\sqrt{\theta_1}$ would be about 4 to 8 percent higher. The inner-compressor operating line for mode II operation (fig. 3(b)) is nearly coincident with that for mode I operation. Inner-compressor efficiency varies between 0.71 and 0.88. For the one-spool engine, the compressor operating line lies farther from the stall-limit line than does the line of maximum efficiency (fig. 3(c)). If the compressor and turbine were sized so that the mode II operating line would pass through the maximum-efficiency region, the engine equivalent weight flow at speeds $N/\sqrt{\theta_1}$ below design would decrease slightly.

Along the inner-turbine operating line (fig. 3(d)), equivalent specific work is constant at its design value, and efficiency varies between 0.87 and 0.88. The outer-turbine operating line has two branches (fig. 3(e)). For operation at constant outer-compressor equivalent speed, outer-turbine equivalent speed varies, and efficiency varies between 0.85 and 0.87. For operation at constant outer-compressor mechanical speed, outer-turbine equivalent speed is constant, and efficiency varies between 0.85 and 0.80. The end of the outer-turbine operating line lies very close to the limiting-loading line. If more margin is desired, the outer-turbine annulus area would have to be increased.

The one-spool turbine operating line also has two branches (fig. 3(f)). For constant compressor equivalent speed operation, turbine equivalent speed varies, and efficiency is nearly constant at 0.87. For constant compressor mechanical speed operation, turbine equivalent speed is constant, and efficiency varies between 0.87 and 0.80. The end of the turbine operating line lies very close to the limiting-loading line.

Weight-Flow Variations

Mode I operation. - The variation of engine equivalent weight flow in percent design with flight condition for mode I operation is shown in figure 4(a). These weight-flow variations influence the engine thrust variations, as thrust is proportional to weight flow. The weight-flow variations for the two engines are very similar. The maximum difference occurs at the highest flight Mach number, where the weight flow for the two-spool engine is 6 percent greater than that for the one-spool engine.

If the outer compressor of the two-spool engine and the one-spool compressor each operated at maximum efficiency, the two-spool equivalent weight flow at Mach 2.8 (72-percent equivalent speed) would be 15 percent higher than for the one-spool engine. The difference of 6 percent results because the outer compressor of the two-spool engine operates at an equivalent weight flow lower than that for maximum efficiency, while the one-spool compressor operates at an equivalent weight flow higher than that for maximum efficiency.

Mode II operation. - The variation of engine equivalent weight flow in percent design with flight condition for mode II operation is shown in figure 4(b). During that part of the flight range for which equivalent speed is held constant at 110-percent design, the weight flow for the two-spool engine exceeds that for the one-spool engine by 1 to 7 percent. During most of the flight range for which mechanical speed is held constant at 115-percent design, the weight flow for the one-spool engine exceeds that for the two-spool. The maximum difference is about 8 percent.

If, for mode II operation, the compressor and turbine components were sized so that the operating line of the one-spool engine would pass through the compressor design point and the operating line of the two-spool engine would pass through the outer-compressor design point, the weight-flow variations for the two engines would be altered. Over the flight range, the one-spool weight flow would decrease less than 2 percent, while the two-spool weight flow would increase 4 to 8 percent. The effect on the weight-flow comparison between the two engines would be to increase the difference during constant-equivalent-speed operation and decrease the difference during constant-mechanical-speed operation over most of the flight range.

Engine Performance

Two-spool performance with afterburning and with the afterburner inoperative is compared with one-spool performance for the two modes of operation in figure 5. In this figure, equivalent net thrust and specific fuel consumption, both as percent design, are plotted against flight Mach number. The design value of equivalent net thrust is 11,300 pounds, and the design value of sfc is 1.166 pounds of fuel per hour per pound thrust.

Mode I operation. - For mode I operation with the afterburner inoperative (fig. 5(a)), the two-spool thrust values are as great as or greater than those for the one-spool engine over the entire flight range considered. The maximum difference occurs at Mach 2.8 in the stratosphere, where the two-spool thrust is about 9 percent greater than the one-spool thrust. The sfc values for the two engines are practically the same over the entire flight range, the difference being 1 percent or less.

The comparison of two- and one-spool performance for mode I operation with afterburning (fig. 5(b)) is similar to the comparison for mode I operation with the afterburner inoperative. Two-spool thrust is as great as or greater than one-spool thrust over the entire flight range. The maximum difference in thrust occurs at Mach 2.8 in the stratosphere, where two-spool thrust is about 6 percent greater than one-spool thrust. The sfc values for the two engines are practically the same over the entire flight range with a difference of 1 percent or less.

Mode II operation. - For mode II operation with the afterburner inoperative (fig. 5(c)), two-spool thrust at zero Mach number and sea level is about 8 percent greater than one-spool thrust. At Mach number 0.9 and sea level, two-spool thrust is equal to one-spool thrust. In the stratosphere, two-spool thrust excels below Mach 1.6, while one-spool thrust excels above this Mach number. At Mach 0.9, two-spool thrust is greater by about 8 percent, while at Mach 2.8 one-spool thrust is higher by about 7 percent. Two-spool sfc is less than one-spool sfc over most of the

flight range. The exception is for flight between Mach 1.8 and 2.3, when one-spool sfc is slightly less than two-spool sfc. At sea level, two-spool sfc is 1 to 5 percent less than one-spool sfc. In the stratosphere the maximum difference is about 3 percent.

The comparison of two- and one-spool performance for mode II operation with afterburning (fig. 5(d)) is similar to the comparison for mode II operation with the afterburner inoperative. For mode II operation with afterburning, two-spool thrust at zero Mach number and sea level is about 7 percent greater than one-spool thrust. At Mach 0.9 and sea level, two-spool and one-spool thrust are about equal. In the stratosphere, two-spool thrust exceeds below Mach 1.6, while one-spool thrust excels above this Mach number. At 0.9 Mach number, two-spool thrust excels by about 9 percent, while at Mach 2.8 one-spool thrust is greater by about 6 percent. Two-spool sfc is less than or equal to one-spool sfc over the entire flight range. At sea level, two-spool sfc is 2 to 5 percent less than one-spool sfc. In the stratosphere, the maximum difference is about 2 percent.

Component Areas

The following compressor and turbine design values were calculated from the assigned design conditions and limits:

Component value	Engine	
	Two spool	One spool
(Outer-) compressor frontal area, sq ft	4.286	4.286
(Outer-) compressor first-rotor tip relative Mach number	1.184	1.184
(Outer-) compressor first-rotor hub-tip radius ratio	0.397	0.397
Inner-compressor frontal area, sq ft	4.286	
Inner-compressor tip speed, ft/sec	1357	
Inner-compressor first-rotor hub-tip radius ratio	0.766	
Inner-turbine frontal area, sq ft	3.788	
Inner-turbine hub-tip radius ratio	0.671	
Inner-turbine-exit centrifugal stress at rotor hub, psi	33,136	
(Outer-) turbine frontal area, sq ft	4.739	4.350
(Outer-) turbine-exit hub-tip radius ratio	0.634	0.587
(Outer-) turbine centrifugal stress at rotor hub, psi	29,566	29,780

The frontal areas of the inner and outer turbines are, respectively, about 12 percent smaller and 11 percent larger than the frontal areas of the outer and inner compressors. Outer- and inner-turbine frontal areas equal to the compressor frontal area result from altering the pressure-ratio division between the outer and inner compressors and increasing the inner-turbine loading parameter by employing exit whirl at the inner-turbine exit. The new design values of outer- and inner-compressor pressure ratio and inner-turbine loading parameter would be 2.15, 3.26, and 2.15, respectively. For the one-spool engine, the ratio of turbine to compressor frontal area is 1.015.

Variations of combustor, afterburner, and exhaust-nozzle areas with flight condition are shown in figure 6 for mode I operation and in figure 7 for mode II operation.

Mode I operation. - For mode I operation, the maximum combustor frontal areas for the two- and one-spool engines are, respectively, 6 and 5 percent greater than the compressor frontal area (fig. 6(a)). Combustor frontal area exceeds compressor frontal area only above flight Mach numbers of about 2.6 in the stratosphere.

The maximum afterburner frontal areas for the two-spool and the one-spool engines are about 19 and 16 percent greater, respectively, than the compressor frontal area (fig. 6(b)).

The required exhaust-nozzle-throat area variation (fig. 6(c)) is small for the two engines. For operation both with the afterburner inoperative and with afterburning, the required variation in exhaust-nozzle-throat area for either engine is only 3 to 4 percent.

To obtain complete expansion in the exhaust nozzle for engine operation with the afterburner inoperative, nozzle-exit areas greater than compressor frontal area are required for flight Mach numbers greater than 1.5 (fig. 6(d)). At Mach 2.8, the ratio of nozzle-exit area to compressor frontal area is 1.92 for the two-spool and 1.77 for the one-spool engine. If the exhaust-nozzle area is limited to the compressor frontal area, net thrust at a Mach number of 2.8 is decreased 10 percent for the two-spool and 8 percent for the one-spool engine. For engine operation with afterburning (fig. 6(e)), nozzle-exit areas greater than compressor area are required for all flight in the stratosphere. At Mach 2.8, the ratio of exhaust-nozzle-exit area to compressor frontal area is 2.62 for the two-spool engine and 2.50 for the one-spool engine. If the exhaust-nozzle area is limited to the compressor frontal area, net thrust at Mach 2.8 is decreased 15 percent for the two-spool and 14 percent for the one-spool engine.

Mode II operation. - For mode II operation, the maximum combustor frontal areas for the two-spool and the one-spool engines are, respectively, 11 and 16 percent greater than the compressor frontal area (fig. 7(a)). Combustor frontal area exceeds compressor frontal area only above a flight Mach number of 2.4 for the two-spool engine and 2.25 for the one-spool engine.

The maximum afterburner frontal areas (fig. 7(b)) for the two- and one-spool engines are, respectively, 40 and 37 percent greater than the compressor frontal area.

The required exhaust-nozzle-throat area variation is about the same for the two engines (fig. 7(c)). For operation with afterburner inoperative, an 18-percent variation in throat area is required for both engines. For operation with afterburning, a 19-percent variation is required for the two-spool engine and a 20-percent variation for the one-spool engine.

To obtain complete expansion in the exhaust nozzle for engine operation with the afterburner inoperative (fig. 7(d)), nozzle-exit areas greater than compressor frontal area are required for flight Mach numbers greater than about 1.3. At Mach 2.8, the ratio of exhaust-nozzle-exit area to compressor frontal area is 2.12 for the two-spool and 2.20 for the one-spool engine. If the exhaust-nozzle area is limited to the compressor frontal area, net thrust at Mach 2.8 is reduced 15 percent for the two-spool and 16 percent for the one-spool engine. For engine operation with afterburning, nozzle-exit areas greater than compressor area are required for all flight in the stratosphere (fig. 7(e)). At Mach 2.8, the ratio of exhaust-nozzle-exit area to compressor frontal area is 3.05 for the two-spool and 3.16 for the one-spool engine. If the exhaust-nozzle area is limited to the compressor frontal area, net thrust at Mach 2.8 is decreased 21 percent for the two-spool and 22 percent for the one-spool engine.

SUMMARY OF RESULTS

An analytical investigation was made to compare the engine performance and component areas of hypothetical two- and one-spool turbojet engines for two modes of operation. For both modes, the turbine-inlet temperature was assigned to be 2500° R for all operating conditions. The mechanical speeds of the one-spool engine and the outer spool of the two-spool engine were assigned to be constant for mode I operation. For mode II operation, the equivalent speeds of the one-spool engine and the outer spool of the two-spool engine were assigned to be 110-percent design for all inlet temperatures less than 567° R. For all inlet temperatures greater than 567° R, the mechanical speeds of the one-spool engine and of the outer spool of the two-spool engine were held constant at 115-percent design.

The following results were obtained:

1. For mode I operation, the two-spool thrust values are as great as or greater than the one-spool thrust values over the entire flight range considered. The specific-fuel-consumption values for the two engines agree within 1 percent over the entire flight range. The maximum difference in thrust occurs at Mach 2.8 in the stratosphere, where the two-spool thrust is greater than the one-spool thrust by about 9 percent for operation with the afterburner inoperative and by about 6 percent for operation with afterburning.

2. For mode II operation with afterburning or with the afterburner inoperative, two-spool performance excels over one part of the flight range, while one-spool performance excels over another. The maximum two-spool thrust advantage is about 9 percent, while the maximum one-spool thrust advantage is about 7 percent. At sea level, two-spool specific fuel consumption is up to 5 percent less than one-spool specific fuel consumption. In the stratosphere, the maximum difference is about 3 percent.

3. For mode I operation, the maximum combustor frontal areas for the two- and the one-spool engines are, respectively, 6 and 5 percent larger than the compressor frontal area. The maximum afterburner frontal areas for the two- and the one-spool engine are, respectively, 19 and 16 percent greater than the compressor frontal area. The required variation in exhaust-nozzle-throat area for either engine is only 3 to 4 percent. For operation with the afterburner inoperative, the maximum ratio of exhaust-nozzle-exit area for complete expansion to compressor frontal area is 1.92 for the two-spool and 1.77 for the one-spool engine. For operation with afterburning, the ratios are 2.62 for the two-spool and 2.50 for the one-spool engine.

4. For mode II operation, the maximum combustor frontal areas for the two- and the one-spool engines are, respectively, 11 and 16 percent greater than the compressor frontal area. The maximum afterburner frontal areas for the two- and one-spool engines are, respectively, 40 and 37 percent greater than the compressor frontal area. The required exhaust-nozzle-throat area variation, which is about the same for the two engines, is 18 to 20 percent. For operation with the afterburner inoperative, the maximum ratio of exhaust-nozzle-exit area for complete expansion to compressor frontal area is 2.12 for the two-spool and 2.20 for the one-spool engine. For operation with afterburning, the ratios are 3.05 for the two-spool and 3.16 for the one-spool engine.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, October 14, 1955

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APPENDIX A

SYMBOLS

The following symbols are used in this report:

a_{cr}	critical velocity of sound, ft/sec
C_V	velocity coefficient
F	thrust, lb
f	fuel-air ratio
g	standard gravitational acceleration, 32.2 ft/sec ²
H	stagnation enthalpy, Btu/lb
J	mechanical equivalent of heat, 778.2 ft-lb/Btu
M	Mach number
N	rotational speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, 53.3 ft-lb/(lb)(°R)
sfc	specific fuel consumption, lb fuel/(hr)(lb thrust)
T	total temperature, °R
t	static temperature, °R
U	wheel speed, ft/sec
V	velocity, ft/sec
w	weight flow, lb/sec
γ	ratio of specific heat at constant pressure to specific heat at constant volume

- δ ratio of total pressure to NACA standard sea-level pressure of 2116 lb/sq ft
- η adiabatic efficiency
- θ ratio of total temperature to NACA standard sea-level temperature of 518.7° R

Subscripts:

- h hub
- i inner spool
- j jet
- n net
- o outer spool
- x axial
- 0 ambient condition

Two-spool notation

One-spool notation

- | | | |
|----|------------------------|-----------------------|
| 1 | outer-compressor inlet | compressor inlet |
| 2 | inner-compressor inlet | |
| 3 | combustor inlet | combustor inlet |
| 4 | inner-turbine inlet | turbine inlet |
| 5 | outer-turbine inlet | |
| 6 | outer-turbine exit | turbine exit |
| 6a | afterburner inlet | afterburner inlet |
| 7 | exhaust-nozzle inlet | exhaust-nozzle inlet |
| 7a | exhaust-nozzle throat | exhaust-nozzle throat |
| 8 | exhaust-nozzle exit | exhaust-nozzle exit |

APPENDIX B

MATCHING PROCEDURES

Two Spool

The pumping characteristics of the two-spool gas generator are obtained by first matching the inner compressor, combustor, and inner turbine, and then matching a resultant inner-spool performance map with performance maps of the outer compressor and outer turbine. Similar procedures are discussed in reference 5. The performance of the inner spool was found as follows:

Inner-compressor performance was plotted as $(H_3 - H_2) (N_i / \sqrt{\theta_2})_d^2 / N_i^2$ against $w_2 N_i / (N_i / \sqrt{\theta_2})_d (\delta_3) (P_4 / P_3)$ for constant values of $(N_i / \sqrt{\theta_2}) / (N_i / \sqrt{\theta_2})_d$. Inner-turbine performance was plotted as $(H_4 - H_5) (N_i / \sqrt{\theta_2})_d^2 / N_i^2$ against $w_4 N_i / (N_i / \sqrt{\theta_2})_d \delta_4$ for constant values of $(N_i / \sqrt{\theta_4}) / (N_i / \sqrt{\theta_2})_d$. The maps were superimposed so as to satisfy the matching relations:

$$\frac{w_2}{w_4} \frac{H_3 - H_2}{N_i^2} = \frac{H_4 - H_5}{N_i^2} \tag{B1}$$

$$\frac{w_2 N_i}{P_4} = \frac{w_4 N_i}{\delta_3 P_3} \frac{w_2}{w_4} \tag{B2}$$

The fuel added in the combustor was assumed to compensate for the air bled from the compressor exit to cool the turbines, so that w_2/w_4 was assumed equal to 1.0. Accessory power was assumed to be negligible for matching purposes. The pressure ratio across the combustor was taken equal to 0.97.

At each match point common to the superimposed maps, the turbine to compressor temperature ratio was calculated from

$$\frac{T_4}{T_2} = \left[\frac{\frac{N_i / \sqrt{\theta_2}}{(N_i / \sqrt{\theta_2})_d}}{\frac{N_i / \sqrt{\theta_4}}{(N_i / \sqrt{\theta_2})_d}} \right]^2 \tag{B3}$$

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CW-3 back

Other inner-spool quantities were calculated or read from appropriate plots, so that inner-spool performance could be plotted as $w_2\sqrt{\theta_2}/\delta_2$ against $w_5\sqrt{\theta_5}/\delta_5$ for constant values of T_5/T_2 .

The following iteration procedure was used to match the inner spool with the outer compressor and outer turbine:

(1) Outer-compressor performance was plotted as $(H_2-H_1)(N_o/\sqrt{\theta_1})_d^2/N_o^2$ and T_2/T_1 against $w_2\sqrt{\theta_2}/\delta_2$ for constant $(N_o/\sqrt{\theta_1})/(N_o/\sqrt{\theta_1})_d$.

(2) Outer-turbine performance was plotted as $(H_5-H_6)(N_o/\sqrt{\theta_1})_d^2/N_o^2$ against $w_5N_o/(N_o/\sqrt{\theta_1})_d\delta_5$ for constant $(N_o/\sqrt{\theta_5})/(N_o/\sqrt{\theta_1})_d$.

(3) An outer-compressor operating point was assigned. This gave values of $w_2\sqrt{\theta_2}/\delta_2$, $(N_o/\sqrt{\theta_1})/(N_o/\sqrt{\theta_1})_d$, $(H_2-H_1)(N_o/\sqrt{\theta_1})_d^2/N_o^2$, and T_2/T_1 .

(4) A trial value of $(N_o/\sqrt{\theta_5})/(N_o/\sqrt{\theta_1})_d$ was assigned.

(5) A value of $w_5N_o/(N_o/\sqrt{\theta_1})_d\delta_5$ was read from the curve of step (2). Because $w_1 = w_2 = w_4 = w_5$, the value of $(H_2-H_1)(N_o/\sqrt{\theta_1})_d^2/N_o^2$ from step (3) equals $(H_5-H_6)(N_o/\sqrt{\theta_1})_d^2/N_o^2$.

(6) A value of $w_5\sqrt{\theta_5}/\delta_5$ was calculated from $w_5N_o/(N_o/\sqrt{\theta_1})_d\delta_5$ and $(N_o/\sqrt{\theta_5})/(N_o/\sqrt{\theta_1})_d$.

(7) A value of T_5/T_2 was read from the inner-spool performance map for known values of $w_2\sqrt{\theta_2}/\delta_2$ and $w_5\sqrt{\theta_5}/\delta_5$.

(8) A value of T_5/T_2 was calculated by making use of the relation between the outer-spool component equivalent speeds:

$$\frac{T_5}{T_2} = \frac{\left[\frac{N_o/\sqrt{\theta_1}}{(N_o/\sqrt{\theta_1})_d} \right]^2}{\frac{T_2}{T_1}} \quad (B4)$$

If this did not equal the value from step (7), steps (4) to (8) were repeated until agreement was reached.

All the two-spool gas-generator quantities required to compute the pumping characteristics may now be calculated or read from appropriate plots. The two-spool pumping characteristics were plotted as T_4/T_1 , T_6/T_1 , P_6/P_1 , f/θ_1 , and $N_i/\sqrt{\theta_1}$ against $w_1\sqrt{\theta_6}/\delta_6$ for constant $N_o/\sqrt{\theta_1}$.

One Spool

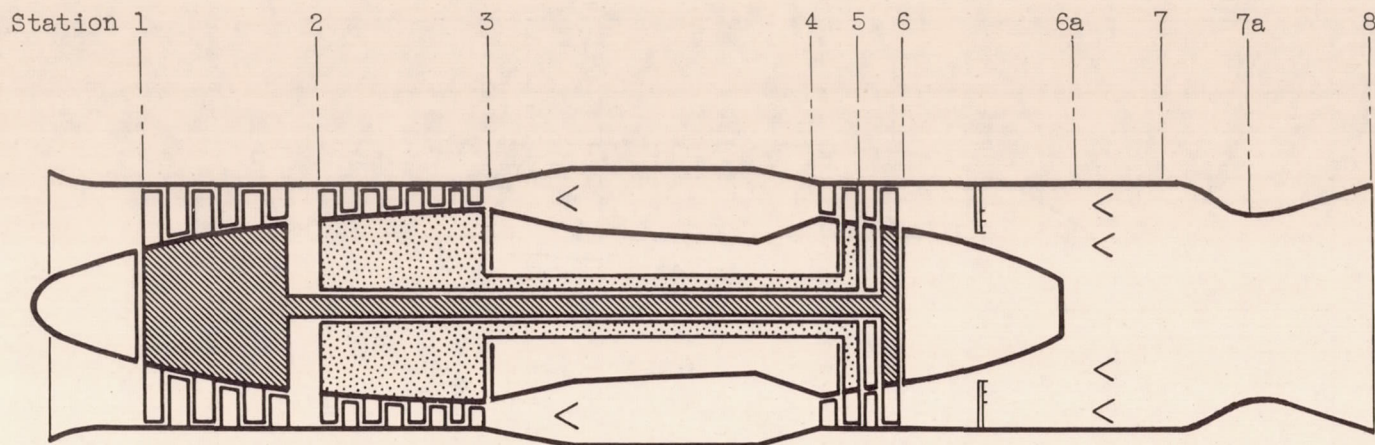
The one-spool gas-generator pumping characteristics were obtained by following the procedures outlined for the inner spool of the two-spool gas generator. Axial stations 1, 3, 4, and 6 for the one-spool gas generator replace stations 2, 3, 4, and 5 for the inner spool of the two-spool gas generator. The one-spool pumping characteristics were plotted as T_4/T_1 , T_6/T_1 , P_6/P_1 , and f/θ_1 against $w_1\sqrt{\theta_6}/\delta_6$ for constant $N/\sqrt{\theta_1}$.

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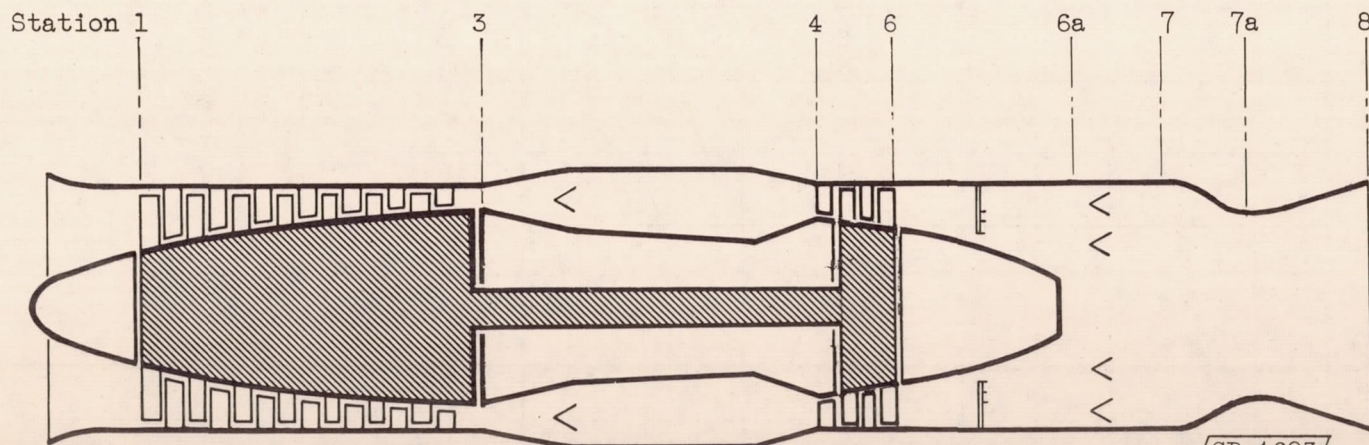
1. Dugan, James F., Jr.: Effect of Design Over-All Compressor Pressure Ratio Division on Two-Spool Turbojet-Engine Performance and Geometry. NACA RM E54F24a, 1954.
2. Dugan, James F., Jr.: Performance and Component Frontal Areas of a Hypothetical Two-Spool Turbojet Engine for Three Modes of Operation. NACA RM E55H31, 1955.
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6. Turner, L. Richard, and Bogart, Donald: Constant-Pressure Combustion Charts Including Effects of Diluent Addition. NACA Rep. 937, 1949. (Supersedes NACA TN's 1086 and 1655.)

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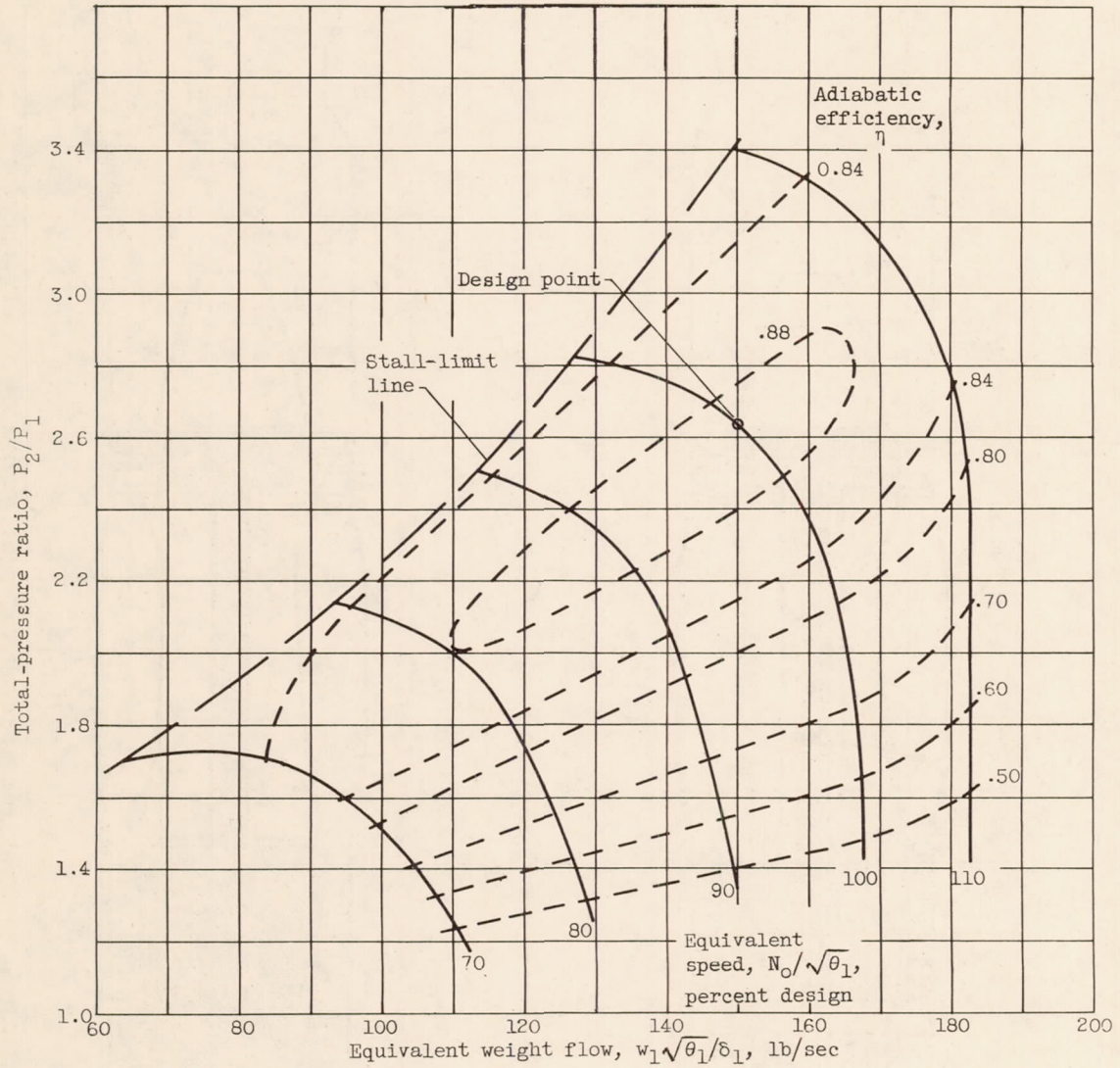
(a) Two spool.



(b) One spool.

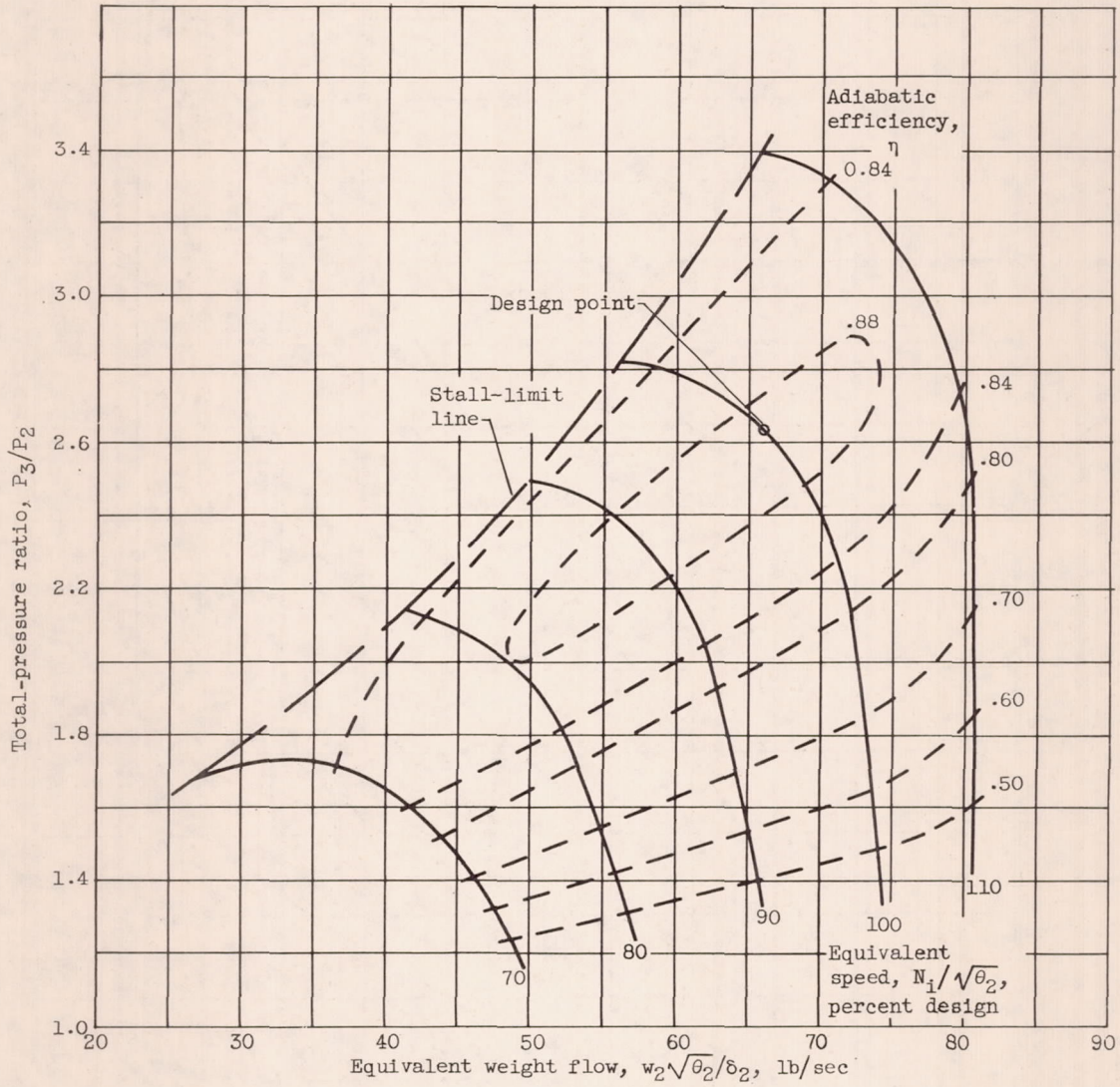
Figure 1. - Cross sections of turbojet engines with afterburners.

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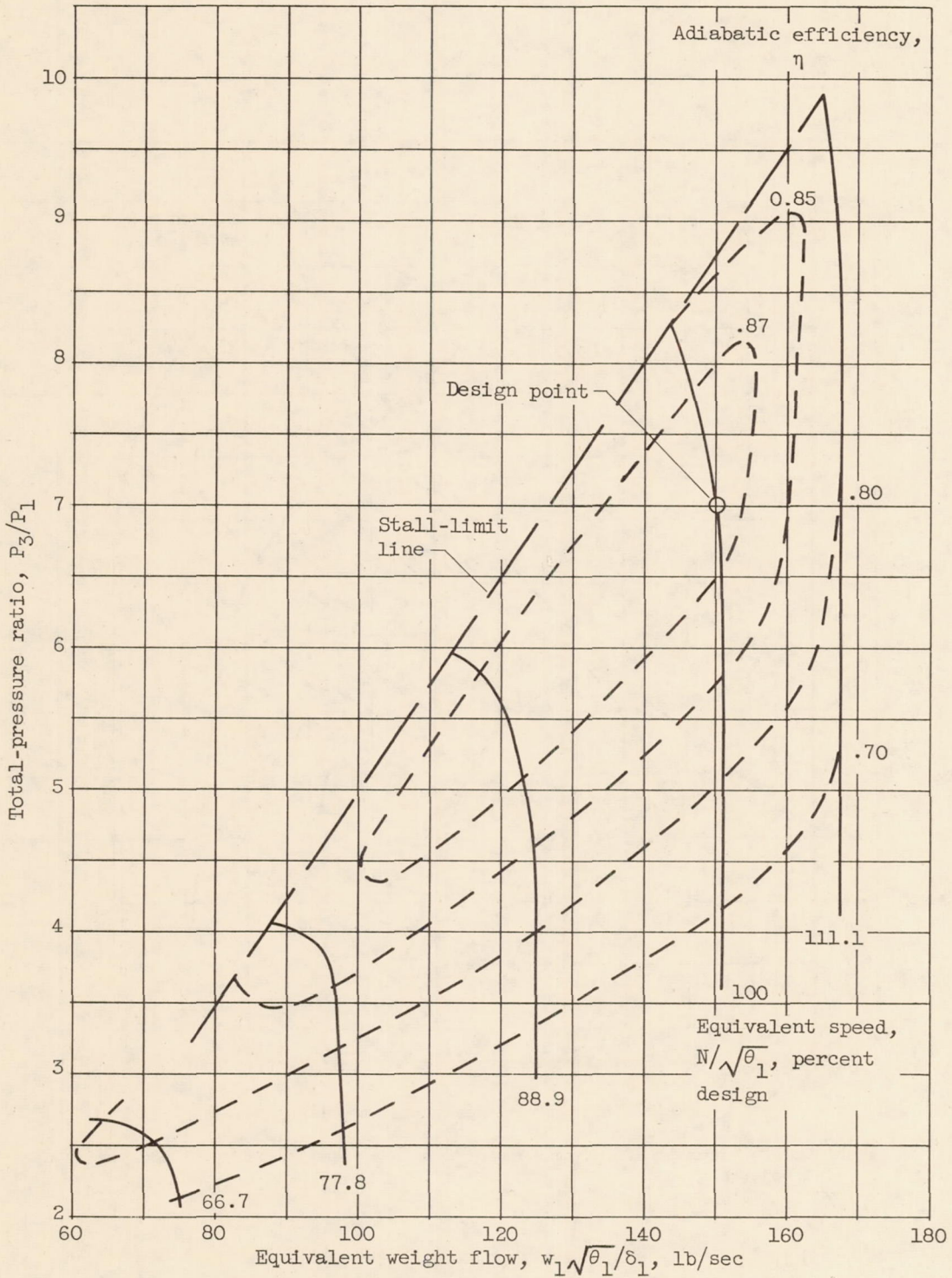
(a) Outer compressor of two-spool engine.

Figure 2. - Component performance maps of two-spool and one-spool turbojet engines.



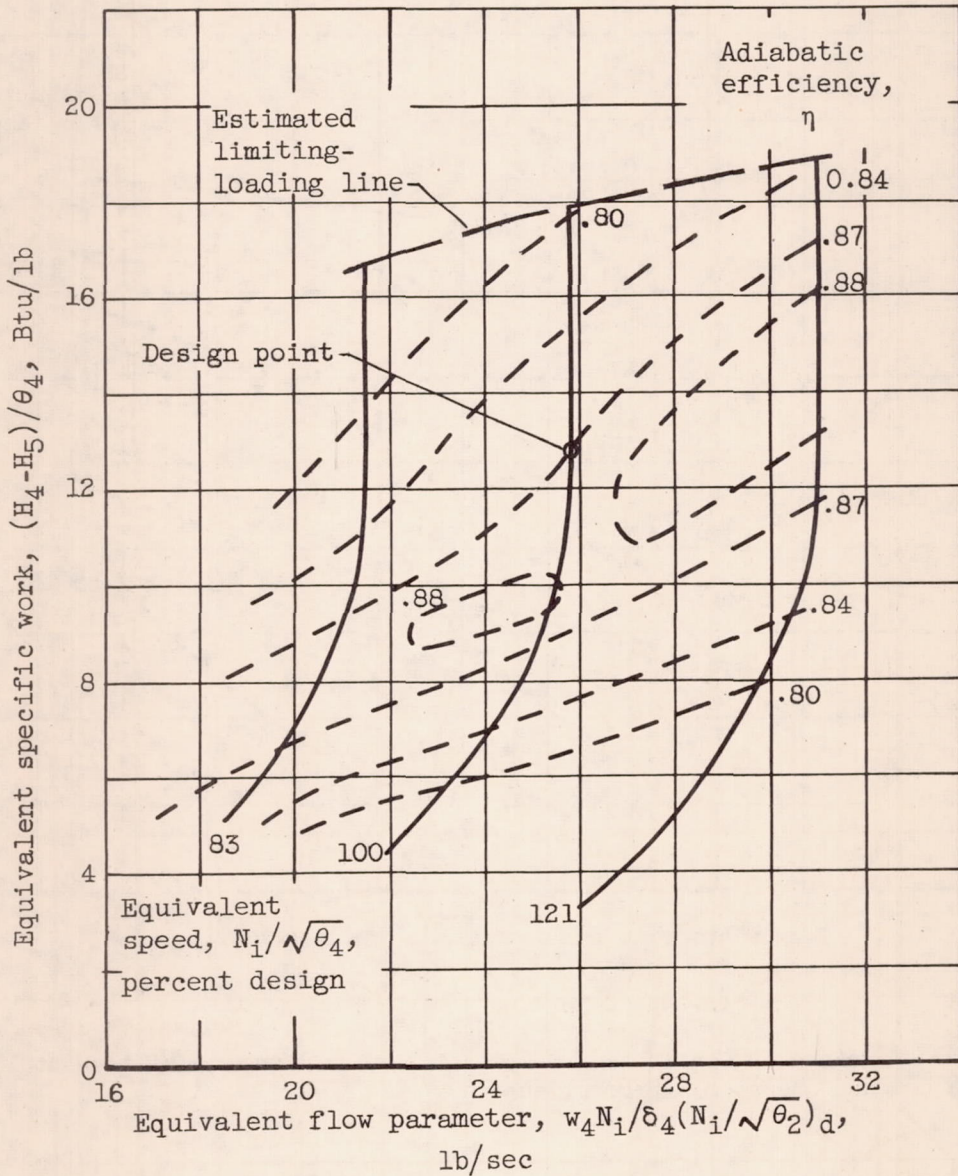
(b) Inner compressor of two-spool engine.

Figure 2. - Continued. Component performance maps of two-spool and one-spool turbojet engines.



(c) Compressor of one-spool engine.

Figure 2. - Continued. Component performance maps of two-spool and one-spool turbojet engines.

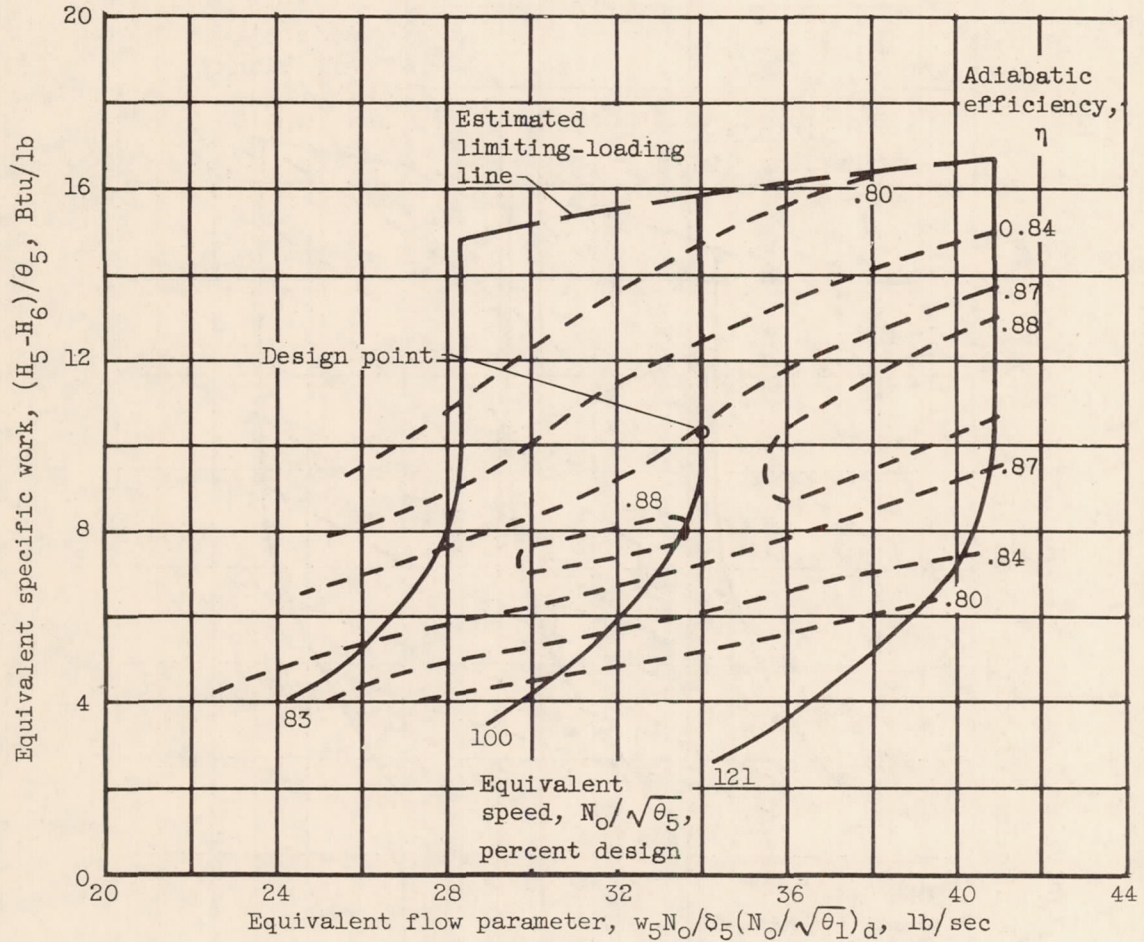


(d) Inner turbine of two-spool engine.

Figure 2. - Continued. Component performance maps of two-spool and one-spool turbojet engines.

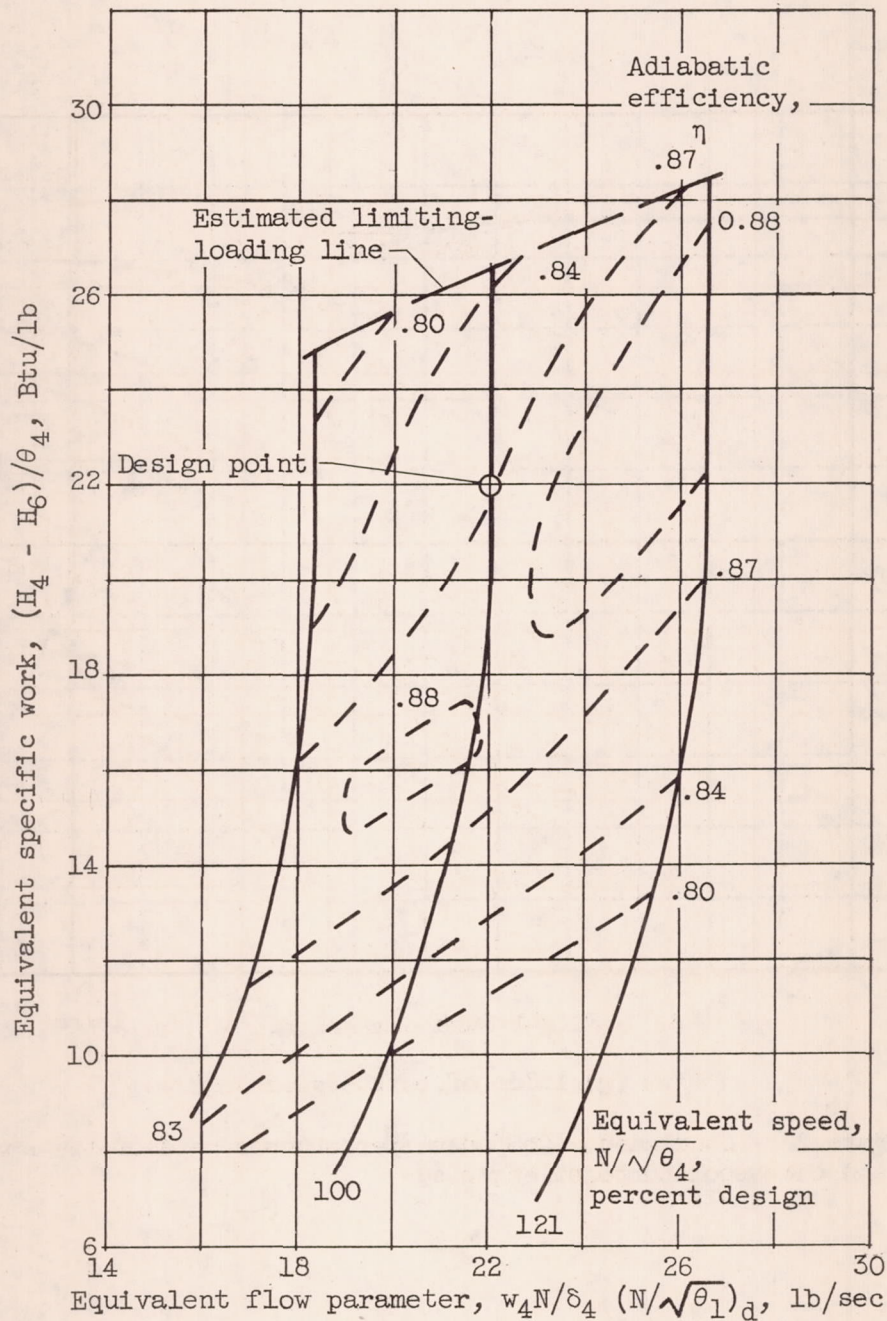
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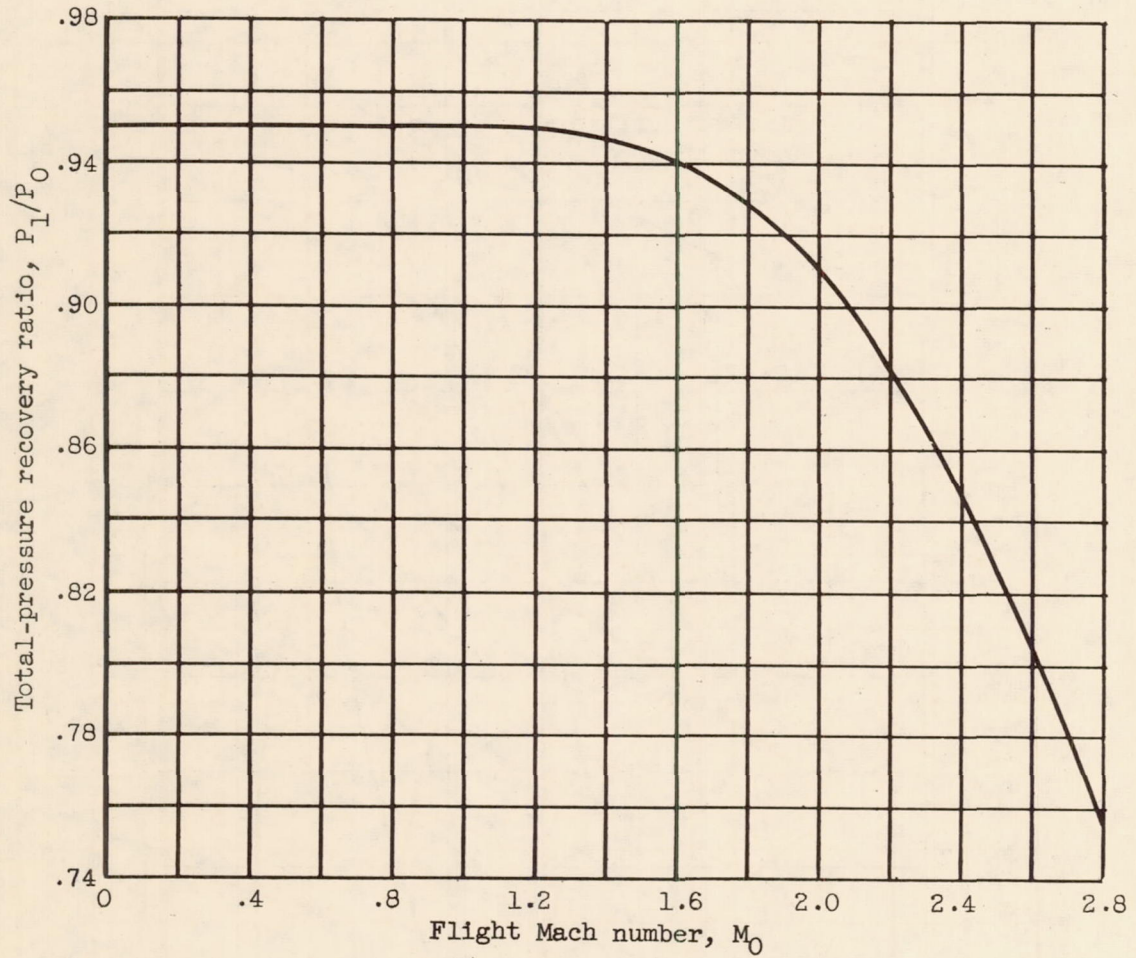
(e) Outer turbine of two-spool engine.

Figure 2. - Continued. Component performance maps of two-spool and one-spool turbojet engines.



(f) Turbine of one-spool engine.

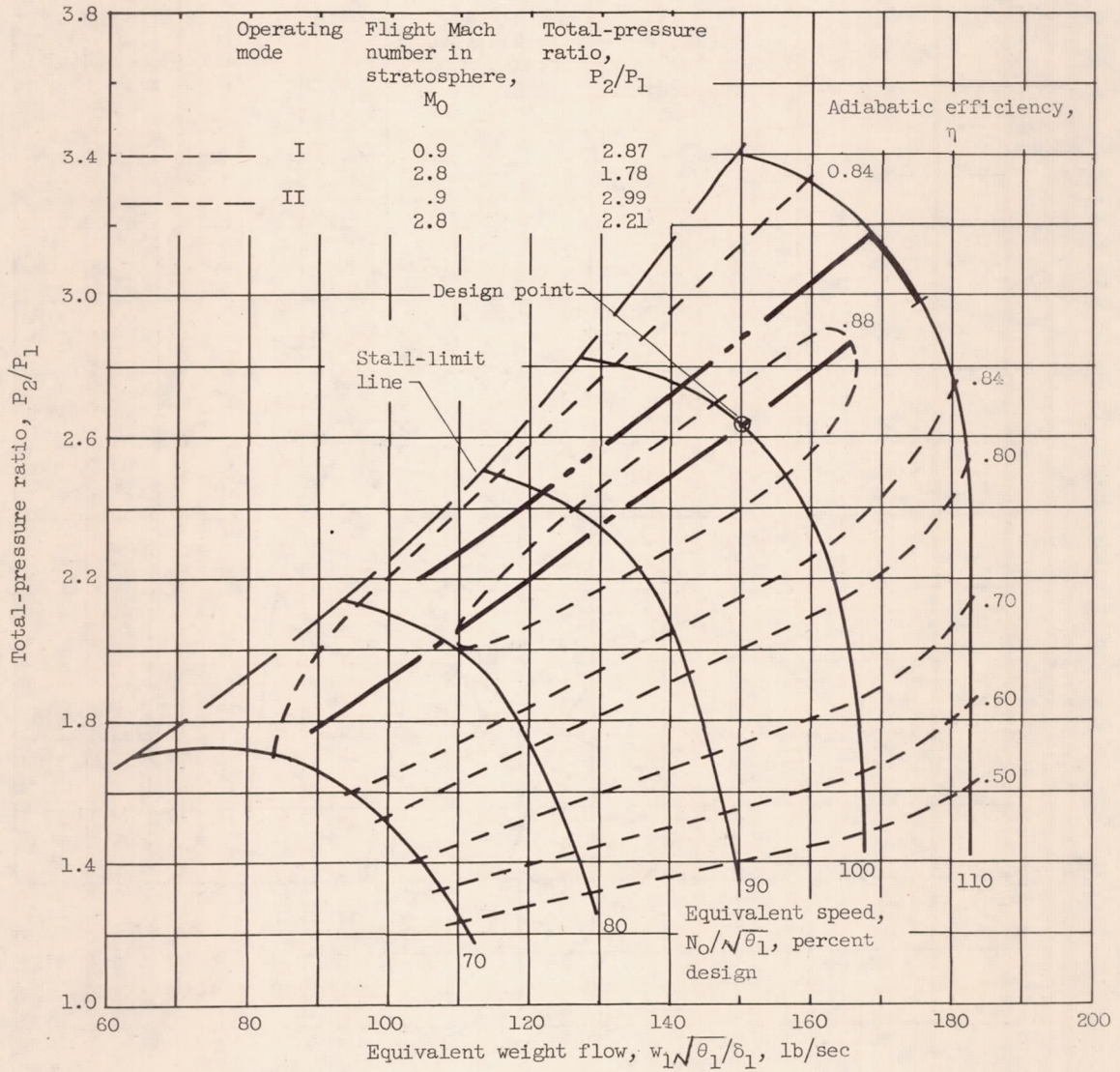
Figure 2. - Continued. Component performance maps of two-spool and one-spool turbojet engines.



(g) Inlet of both engines.

Figure 2. - Concluded. Component performance maps of two-spool and one-spool turbojet engines.

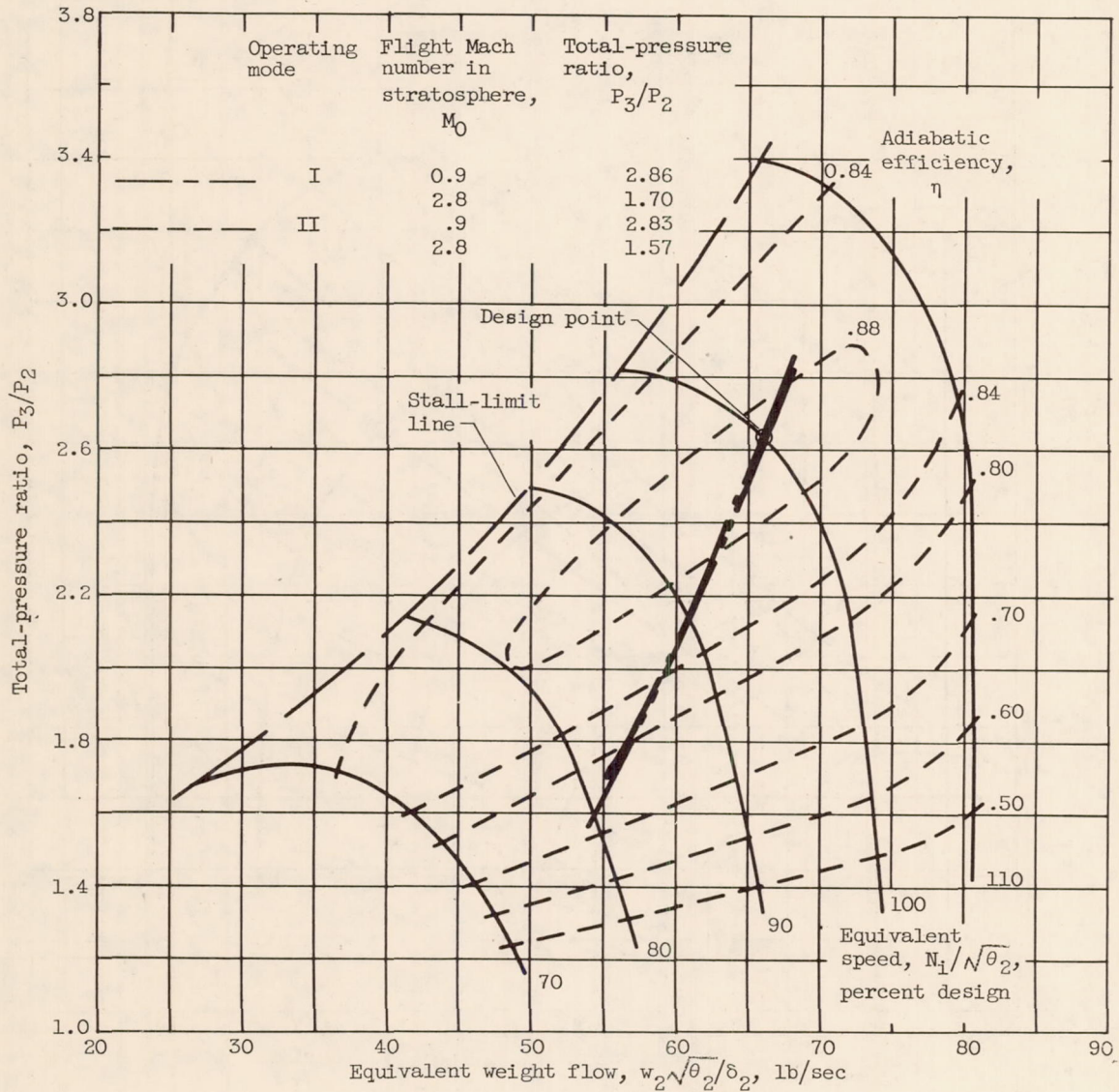
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(a) Outer compressor of two-spool engine.

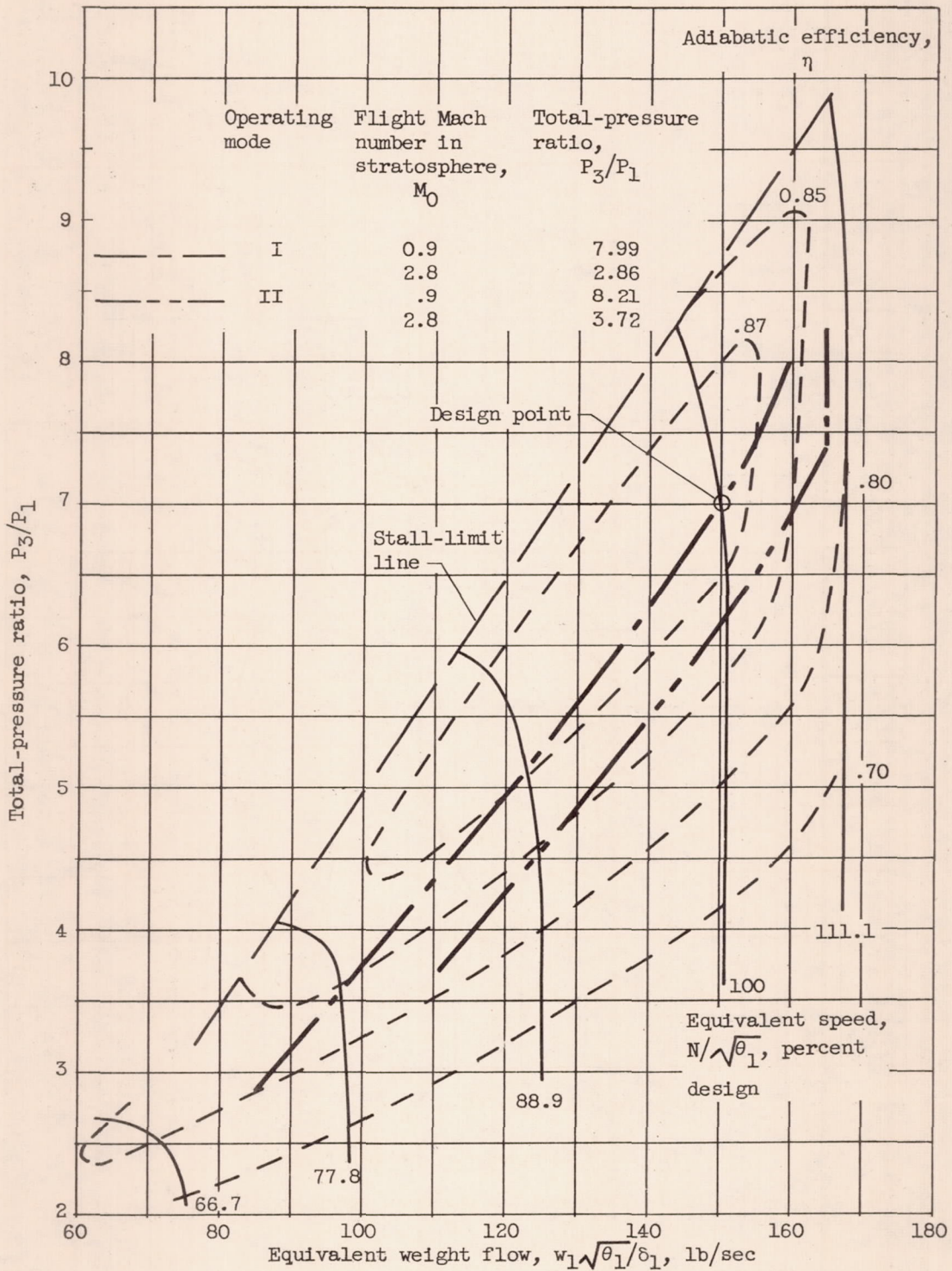
Figure 3. - Two-spool and one-spool operating lines for two modes of operation.

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(b) Inner compressor of two-spool engine.

Figure 3. - Continued. Two-spool and one-spool operating lines for two modes of operation.

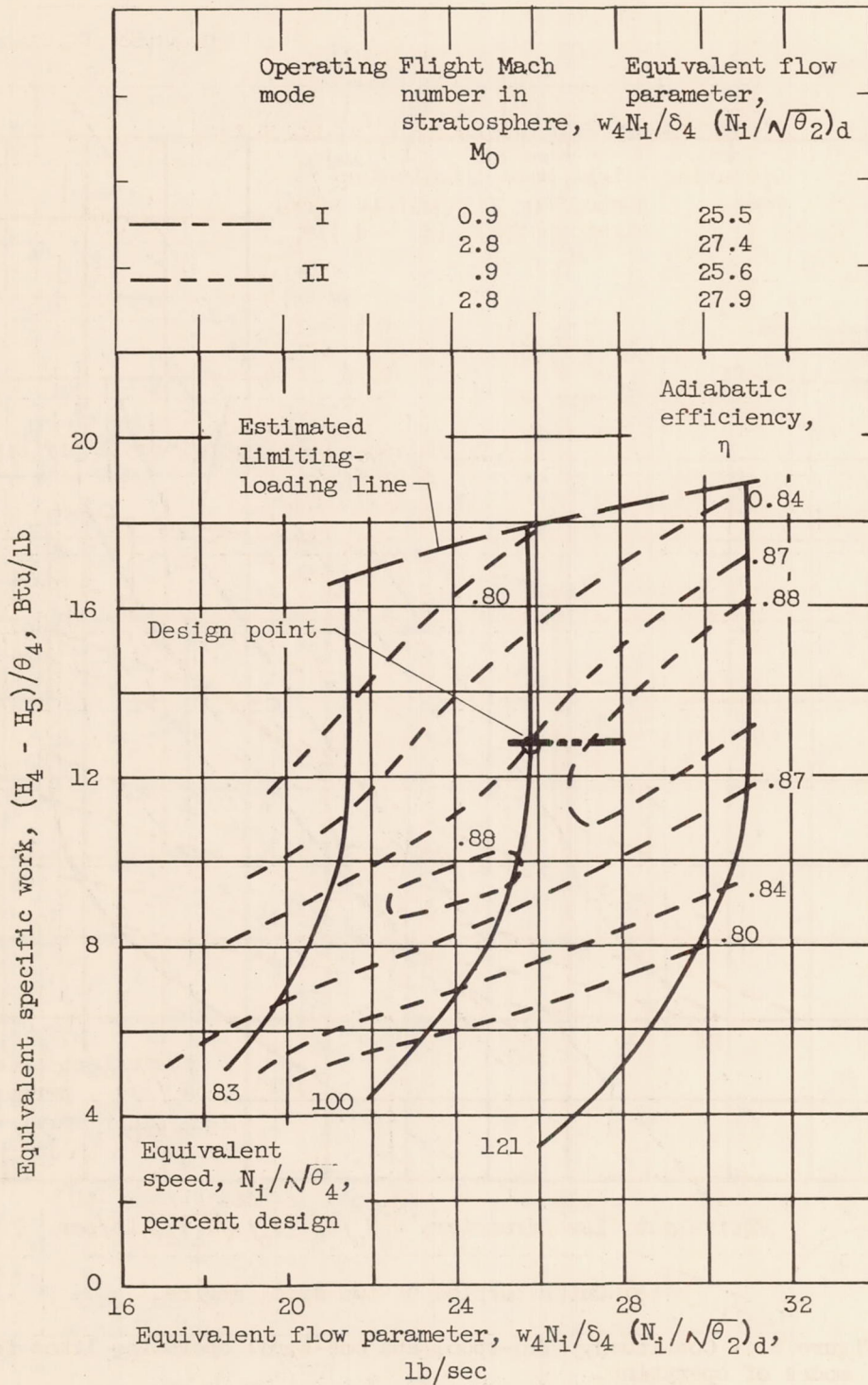


(c) Compressor of one-spool engine.

Figure 3. - Continued. Two-spool and one-spool operating lines for two modes of operation.

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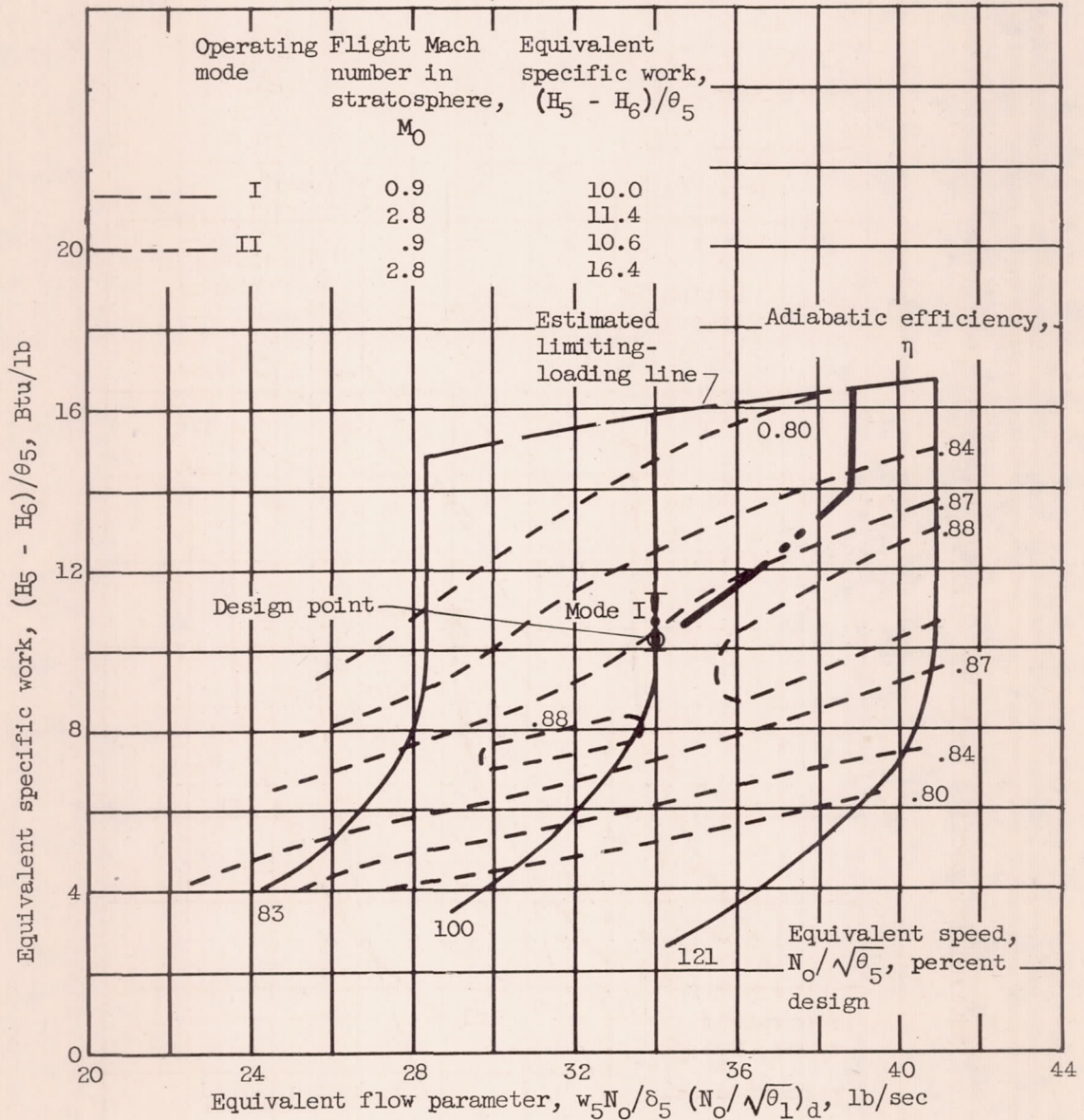
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(d) Inner turbine of two-spool engine.

Figure 3. - Continued. Two-spool and one-spool operating lines for two modes of operation.

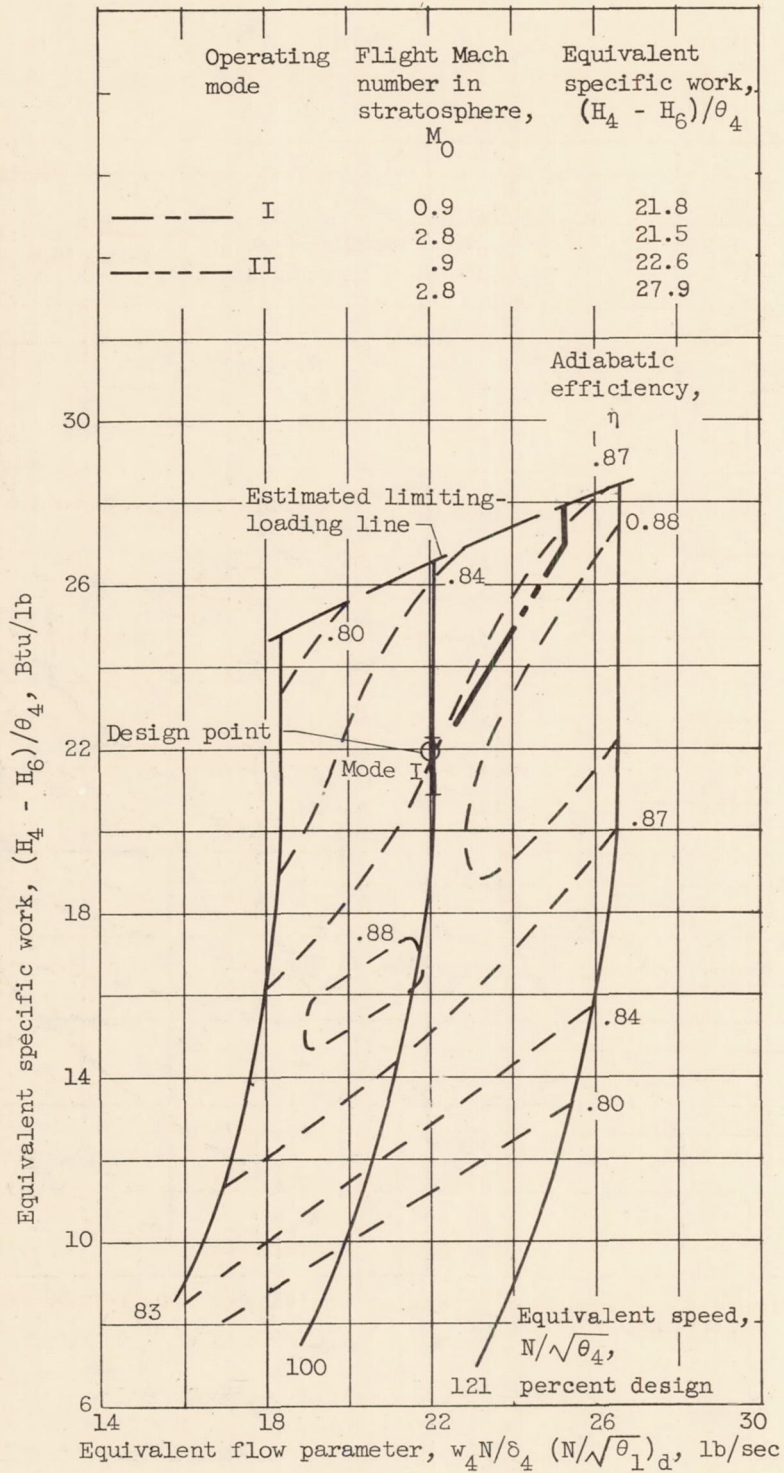
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(e) Outer turbine of two-spool engine.

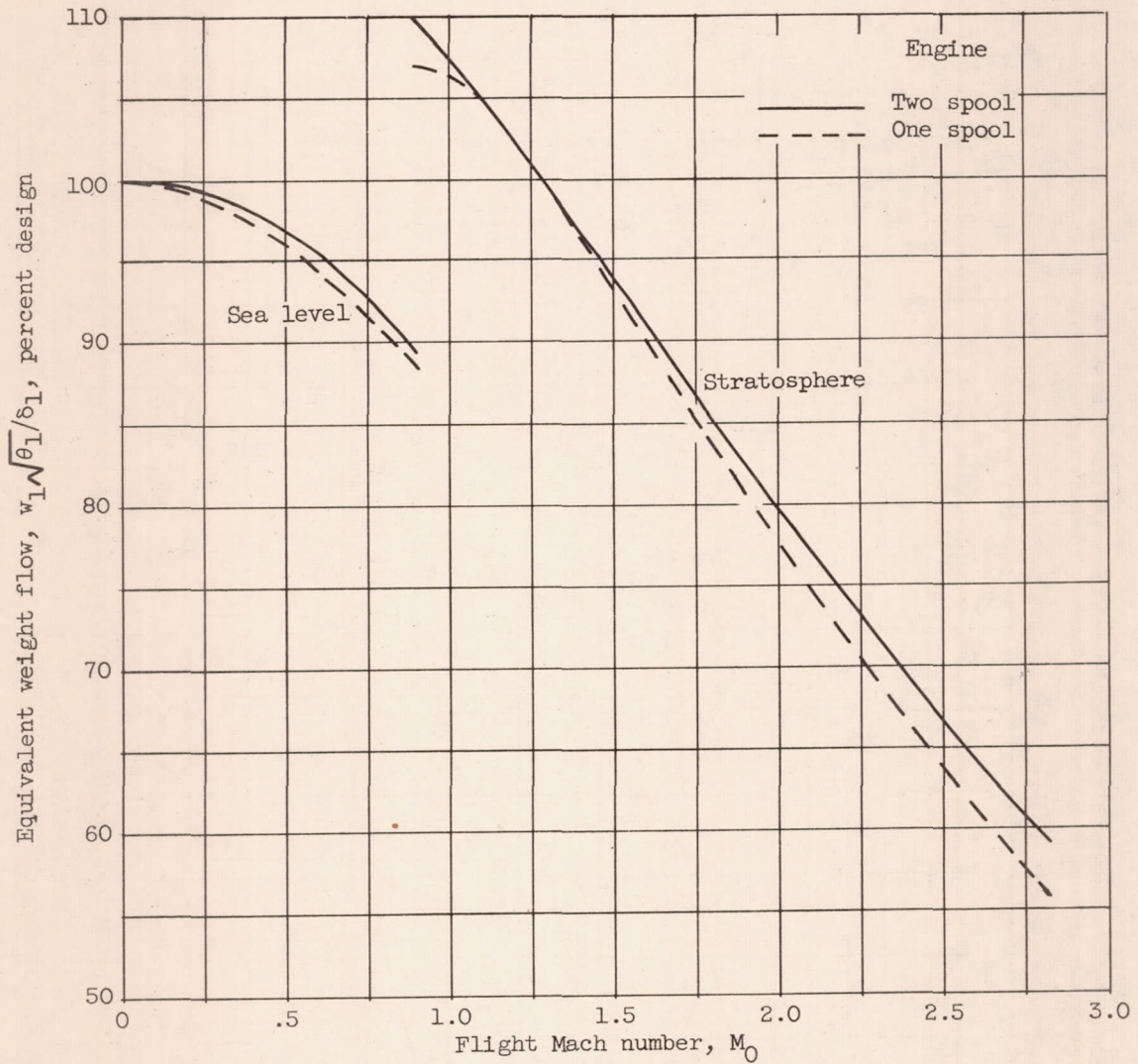
Figure 3. - Continued. Two-spool and one-spool operating lines for two modes of operation.

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(f) Turbine of one-spool engine.

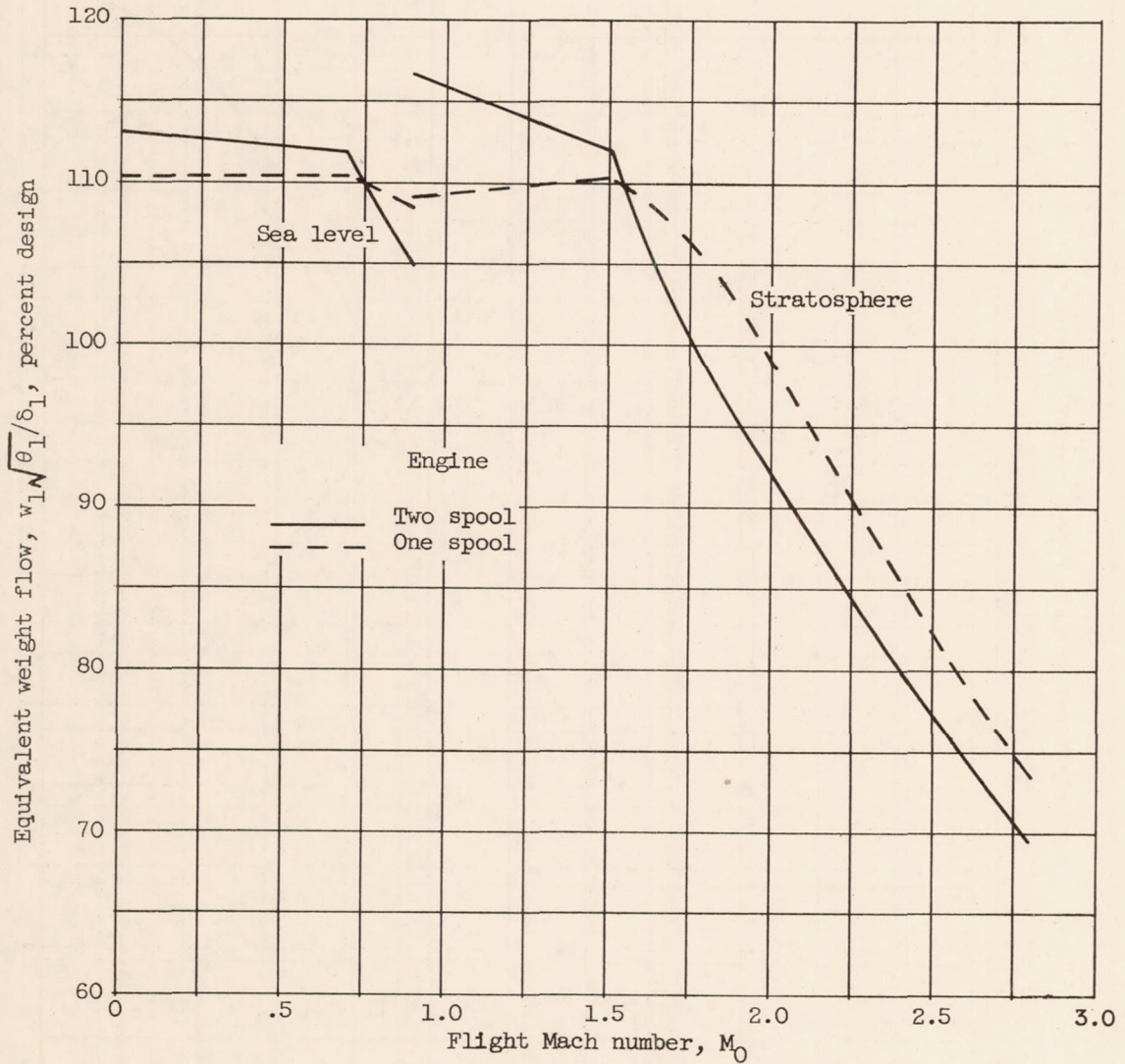
Figure 3. - Concluded. Two-spool and one-spool operating lines for two modes of operation.



(a) Mode I operation.

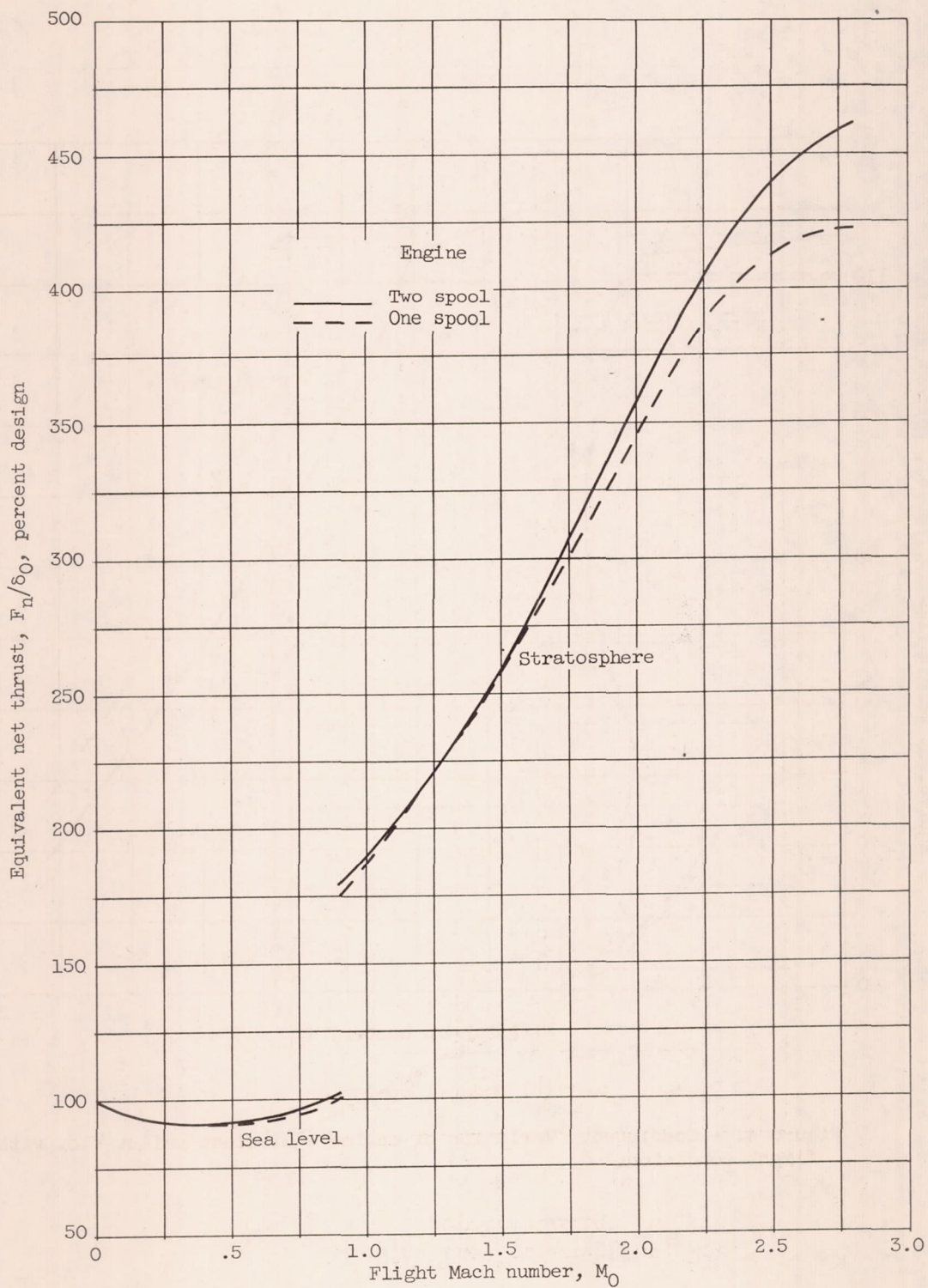
Figure 4. - Variation of engine equivalent weight flow with flight condition.

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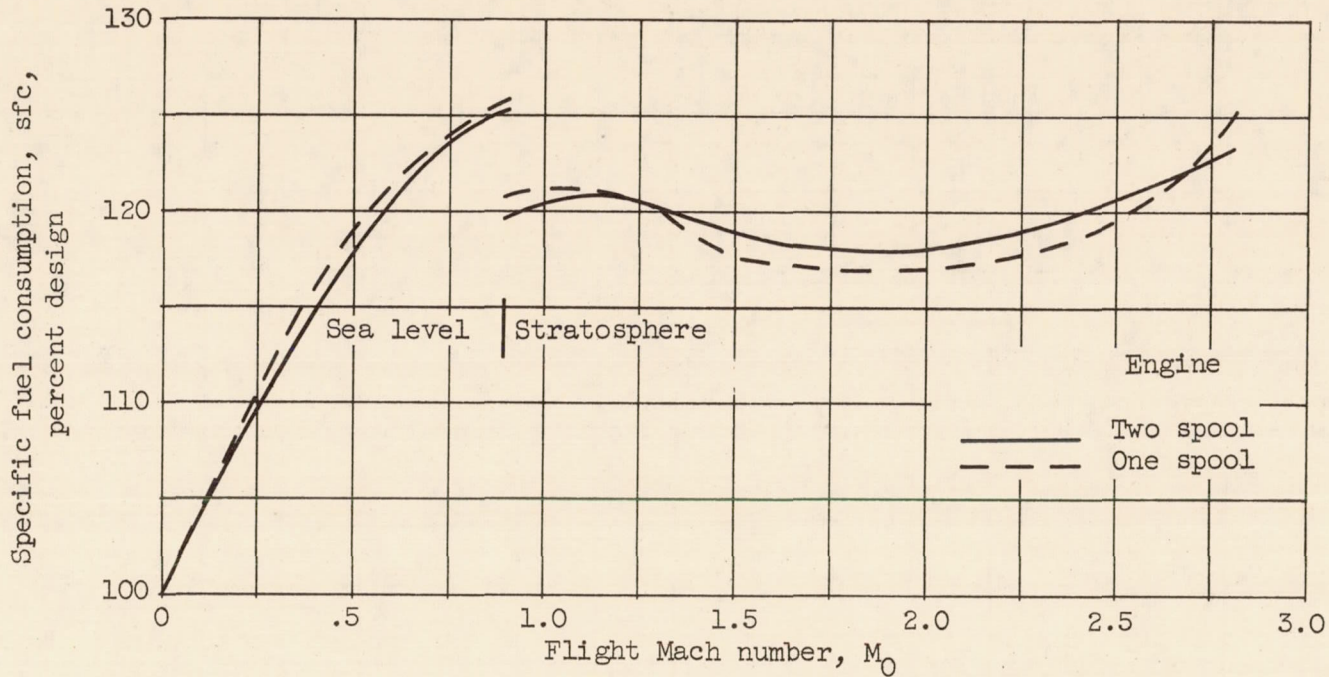
(b) Mode II operation.

Figure 4. - Concluded. Variation of engine equivalent weight flow with flight condition.



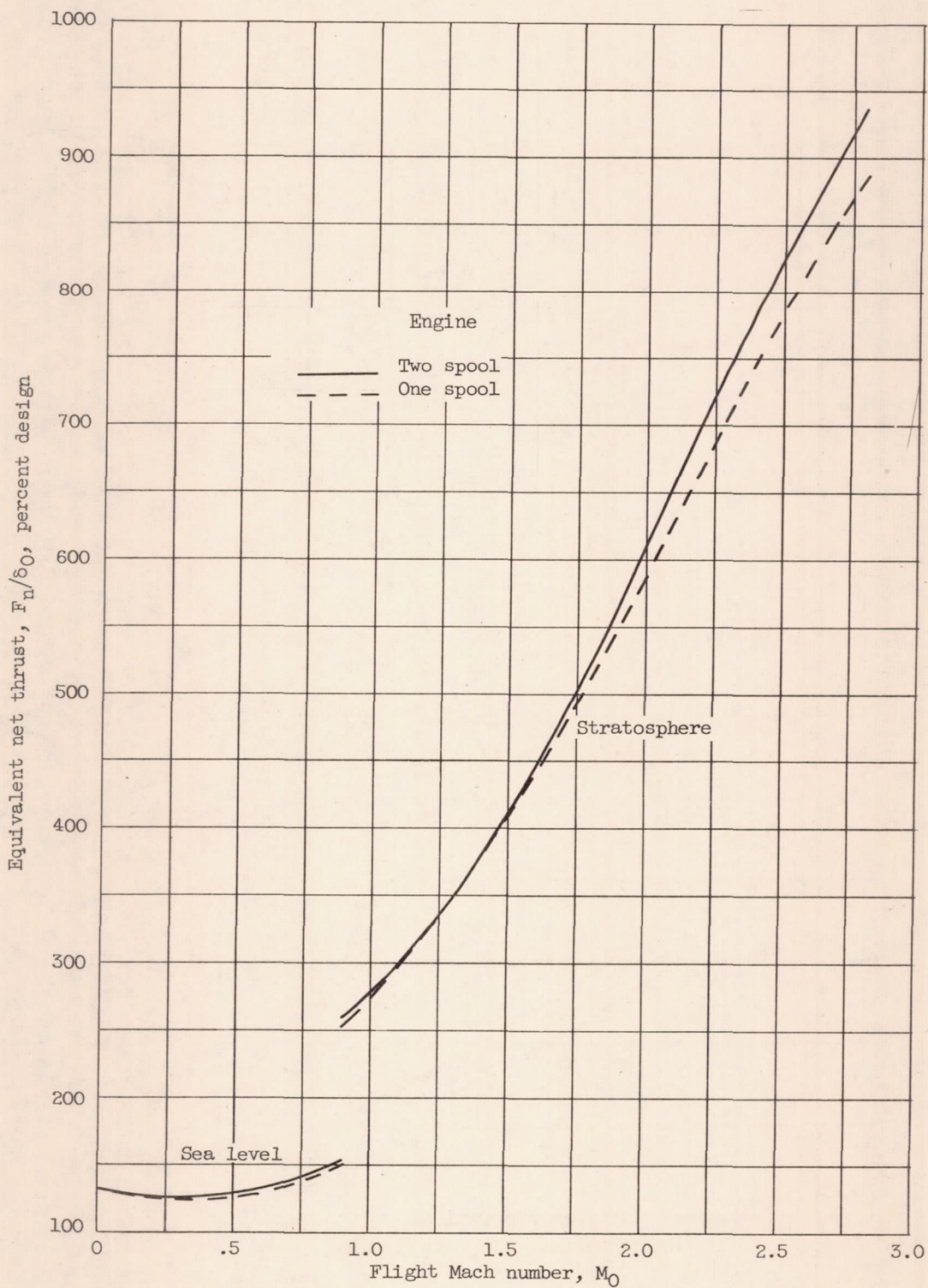
(a) Mode I operation with afterburner inoperative.

Figure 5. - Comparison of two- and one-spool performance.



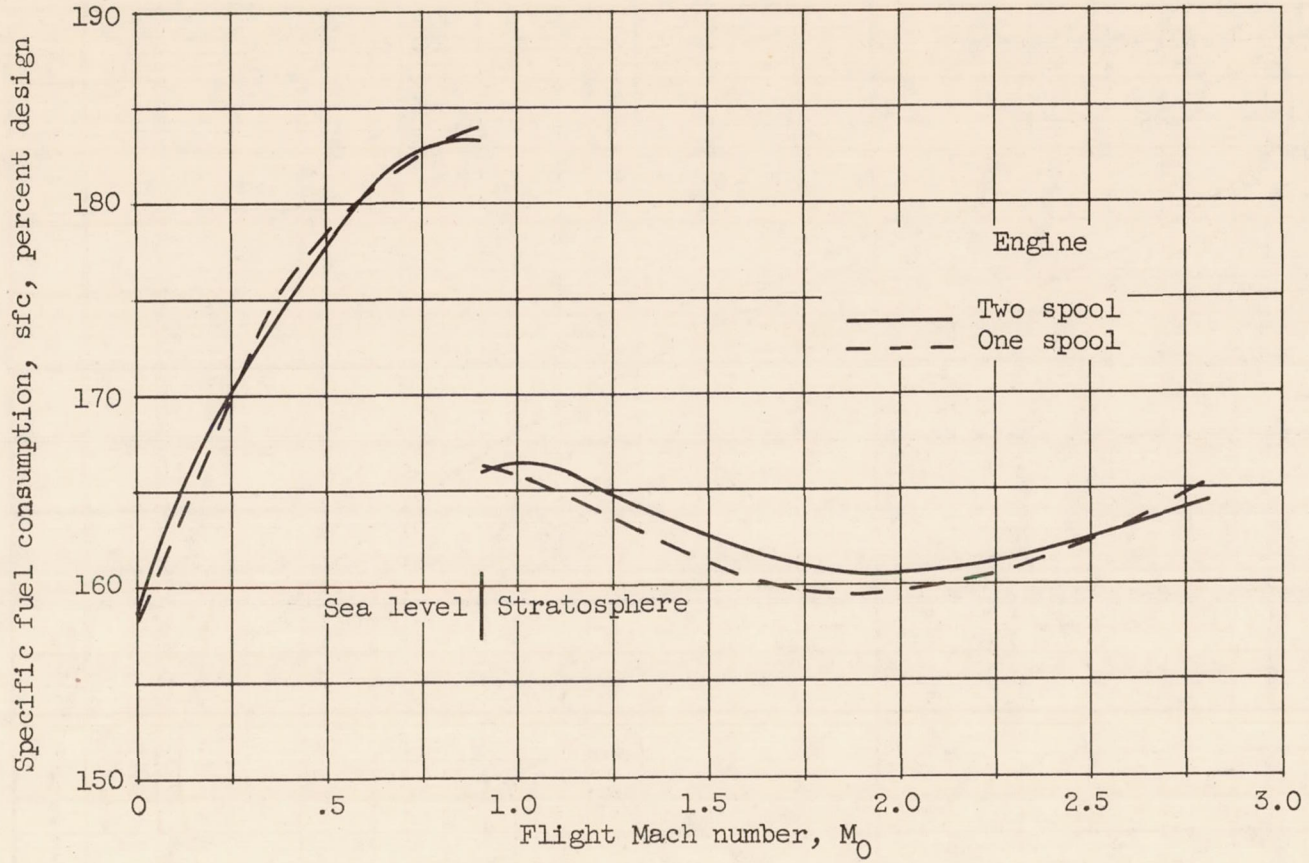
(a) Concluded. Mode I operation with afterburner inoperative.

Figure 5. - Continued. Comparison of two- and one-spool performance.



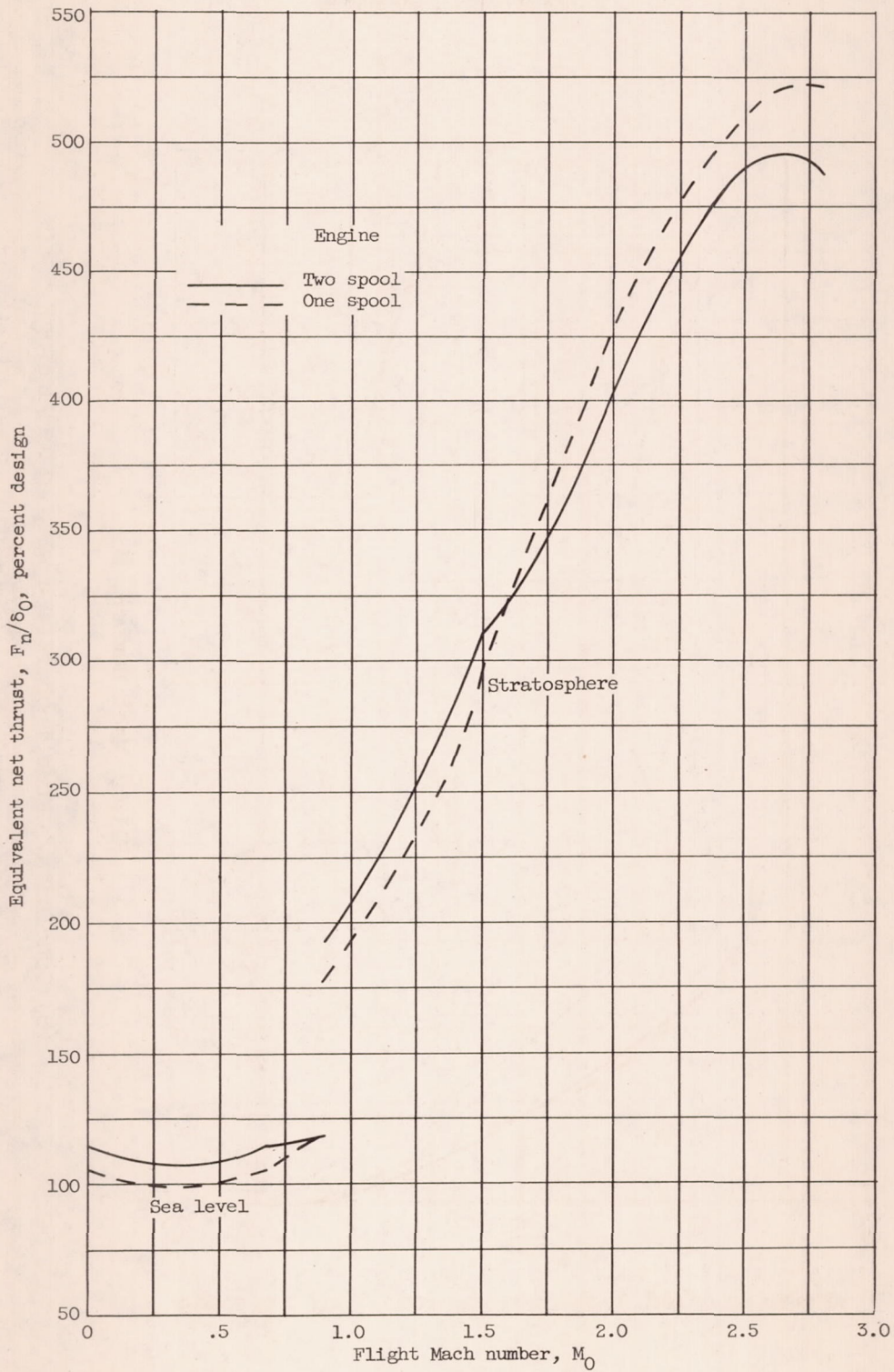
(b) Mode I operation with afterburning.

Figure 5. - Continued. Comparison of two- and one-spool performance.



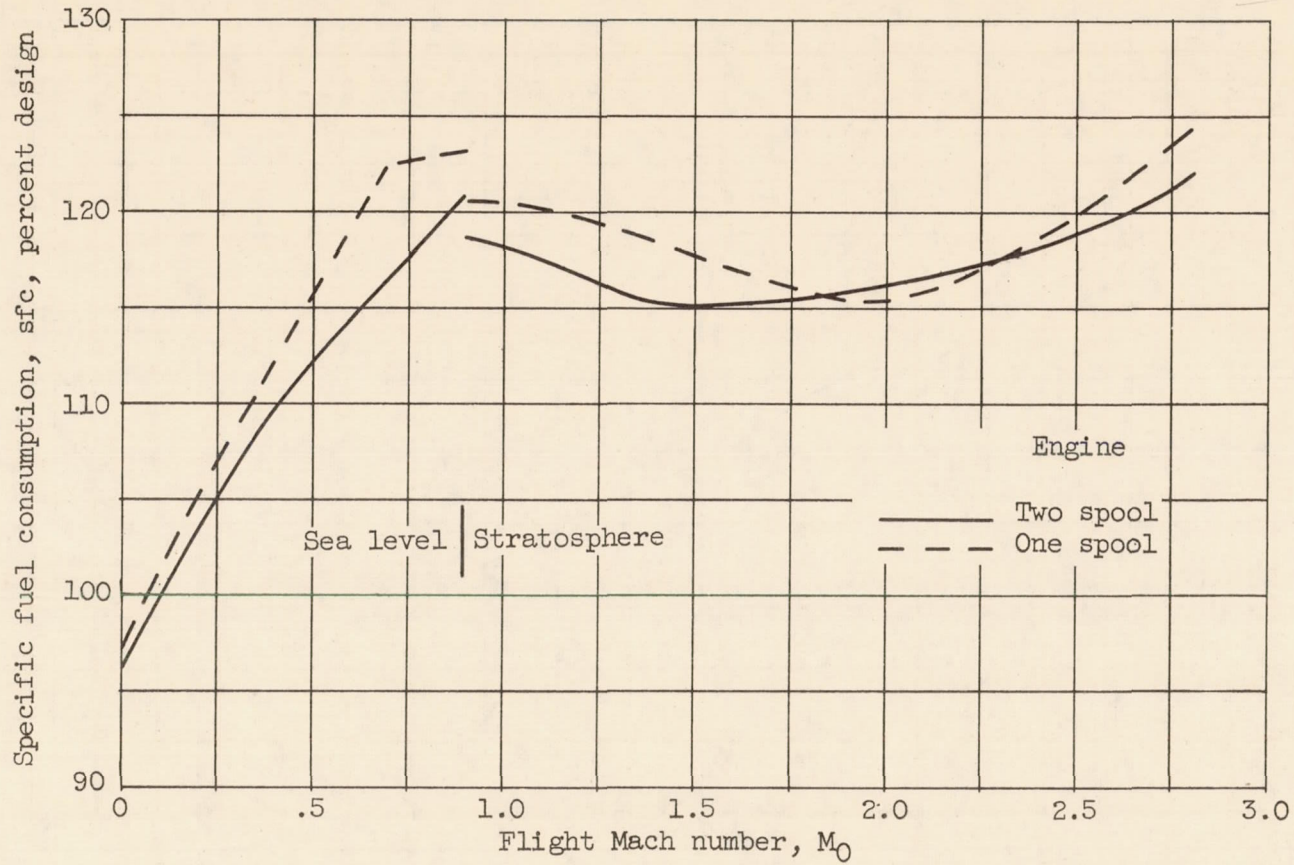
(b) Concluded. Mode I operation with afterburning.

Figure 5. - Continued. Comparison of two- and one-spool performance.



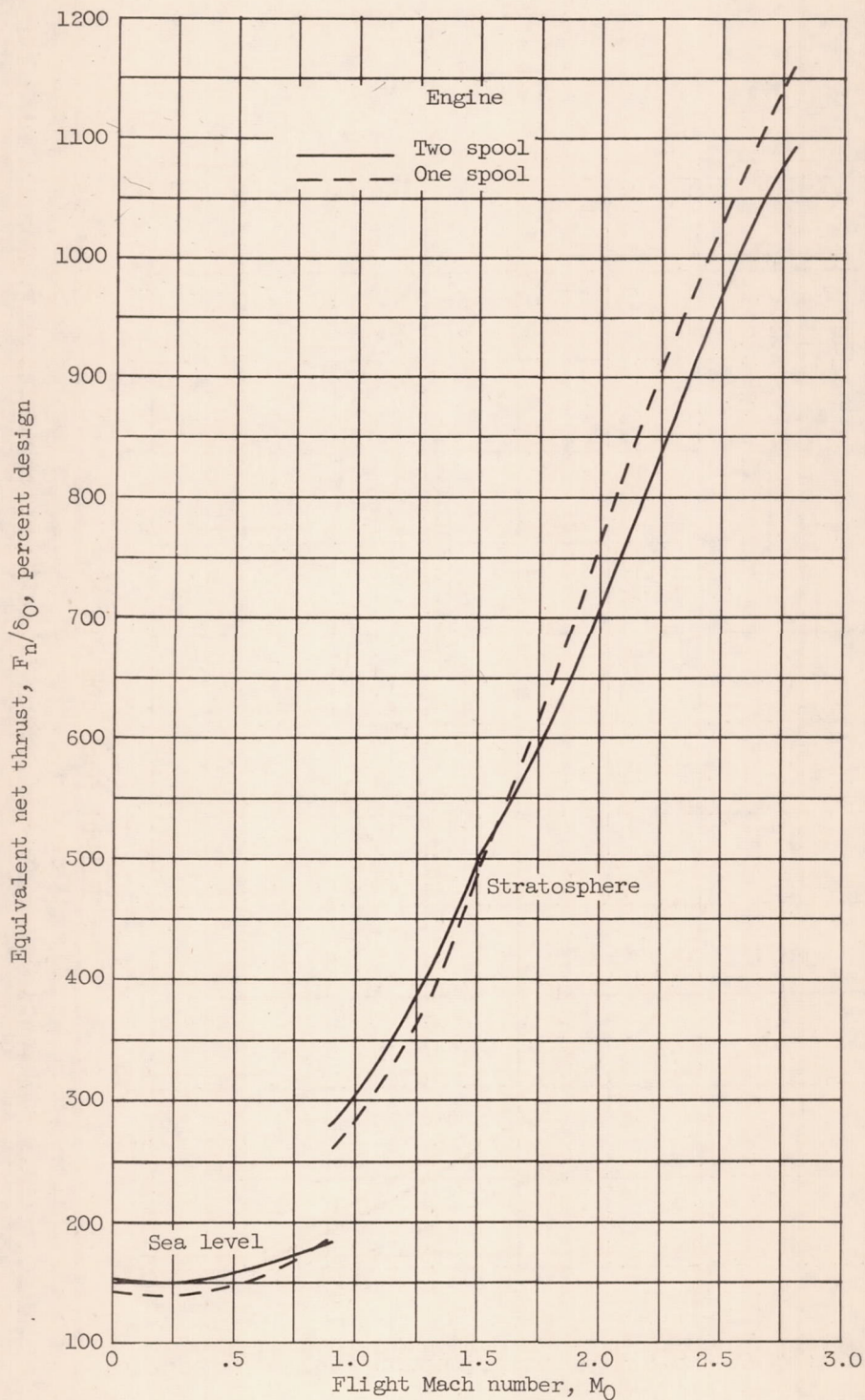
(c) Mode II operation with afterburner inoperative.

Figure 5. - Continued. Comparison of two- and one-spool performance.



(c) Concluded. Mode II operation with afterburner inoperative.

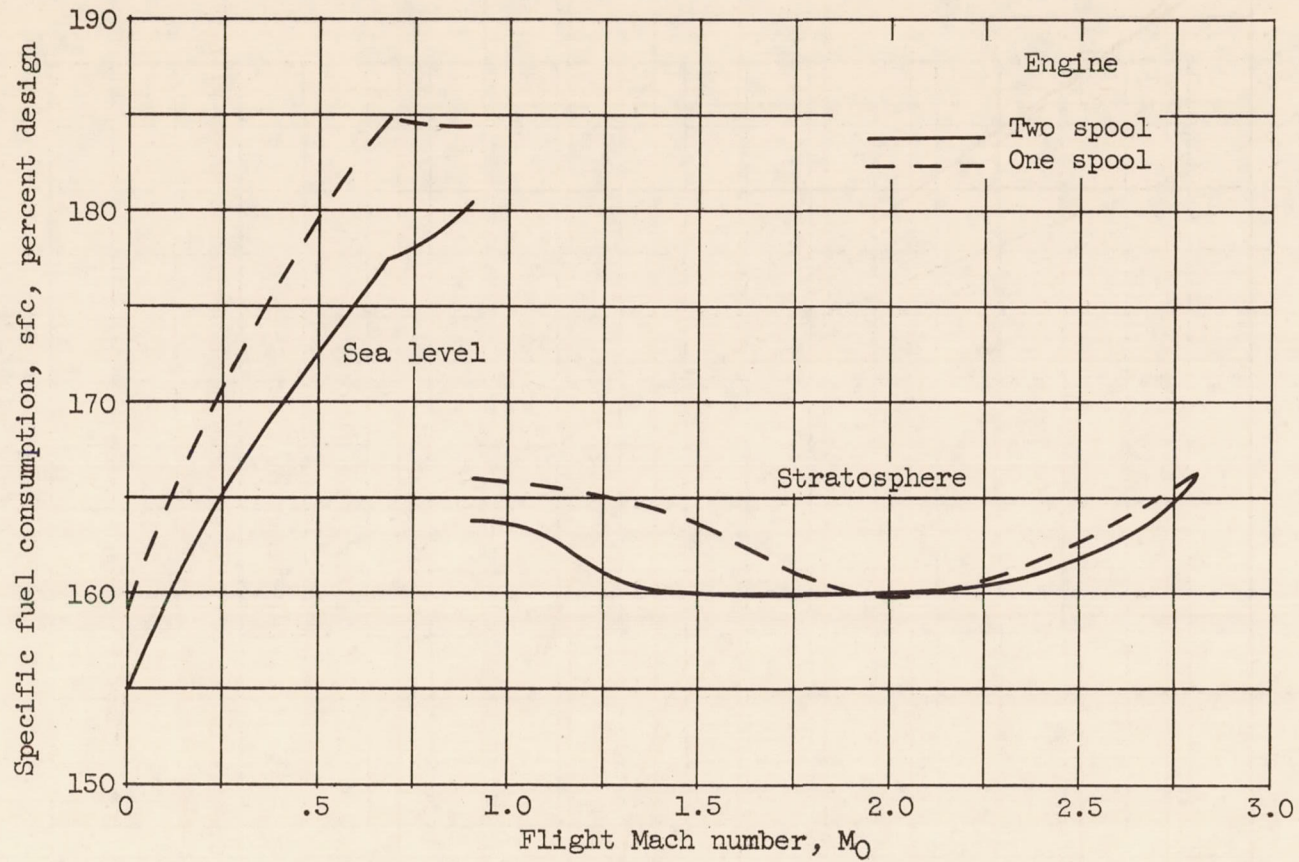
Figure 5. - Continued. Comparison of two- and one-spool performance.



(d) Mode II operation with afterburning.

Figure 5. - Continued. Comparison of two- and one-spool performance.

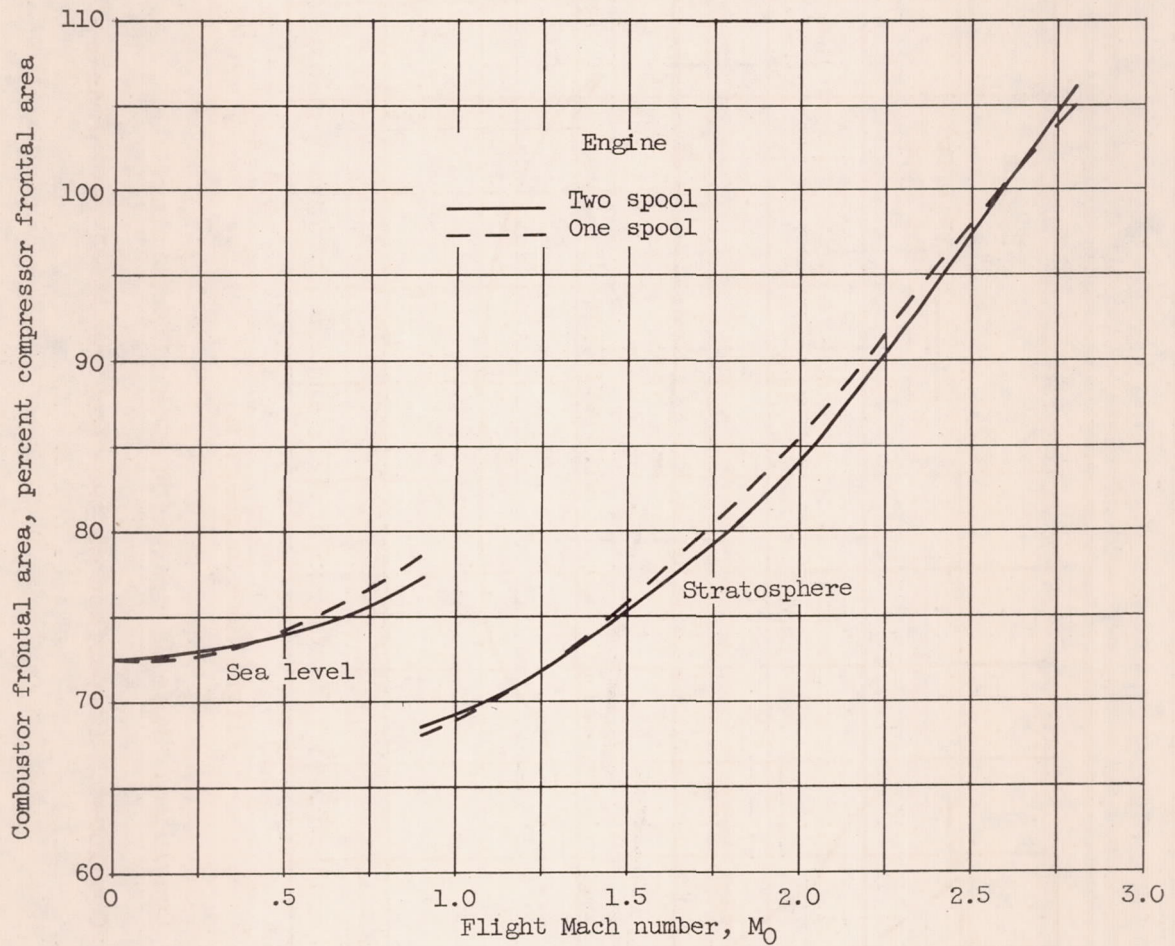
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(d) Concluded. Mode II operation with afterburning.

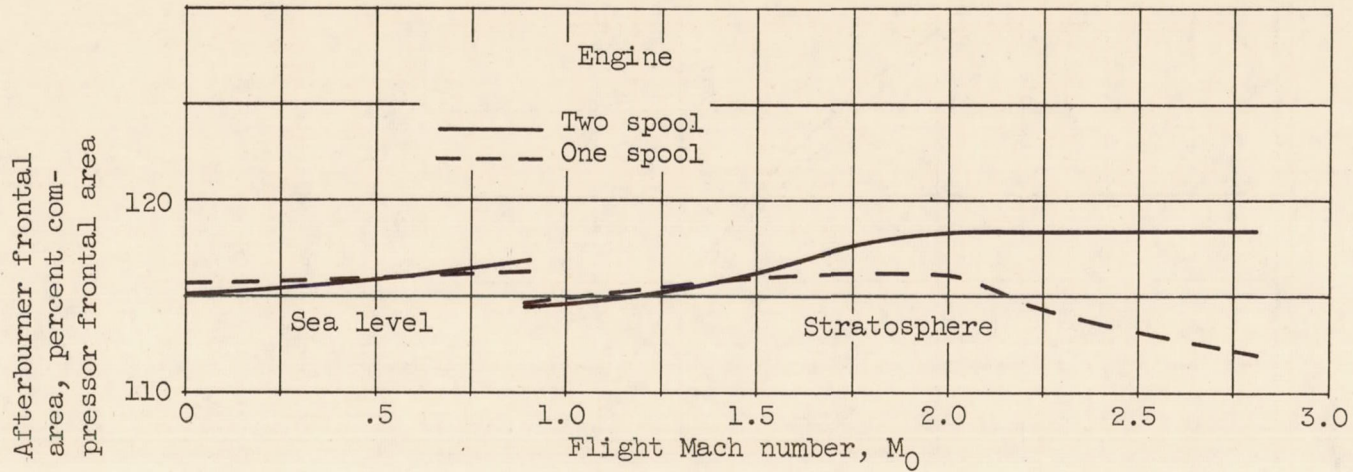
Figure 5. - Concluded. Comparison of two- and one-spool performance.

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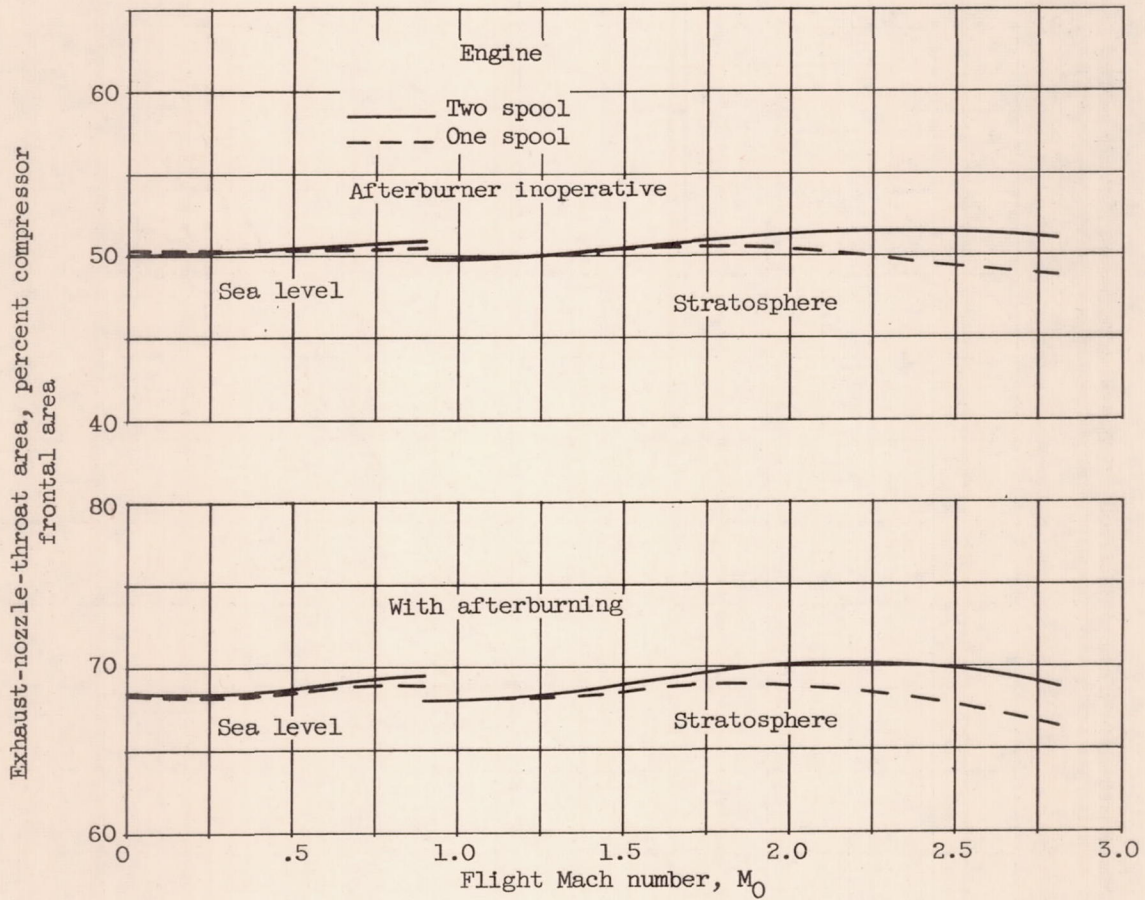
(a) Combustor frontal area.

Figure 6. - Variation of two- and one-spool component areas with flight Mach number for mode I operation.



(b) Afterburner frontal area.

Figure 6. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode I operation.

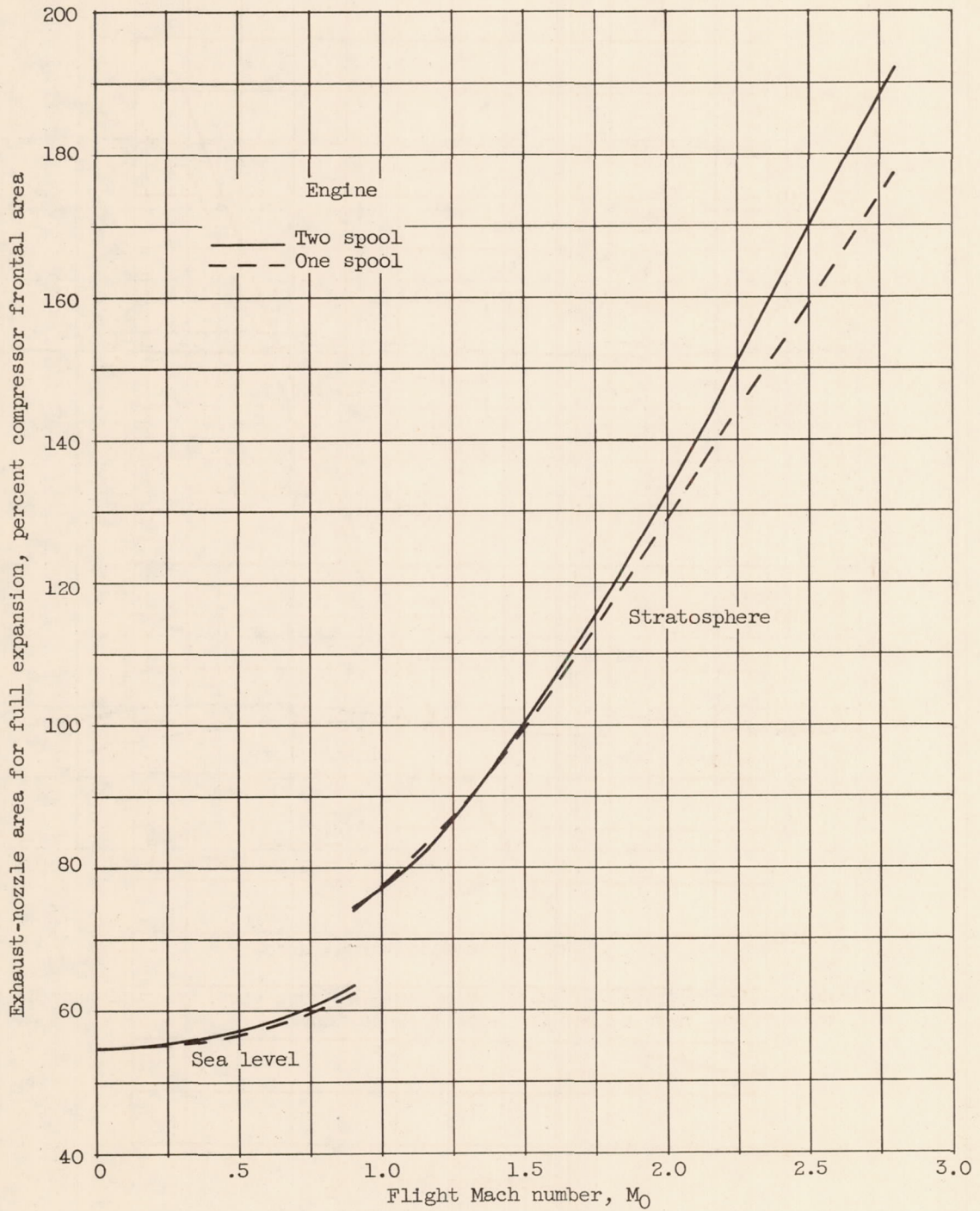


(c) Exhaust-nozzle-throat area.

Figure 6. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode I operation.

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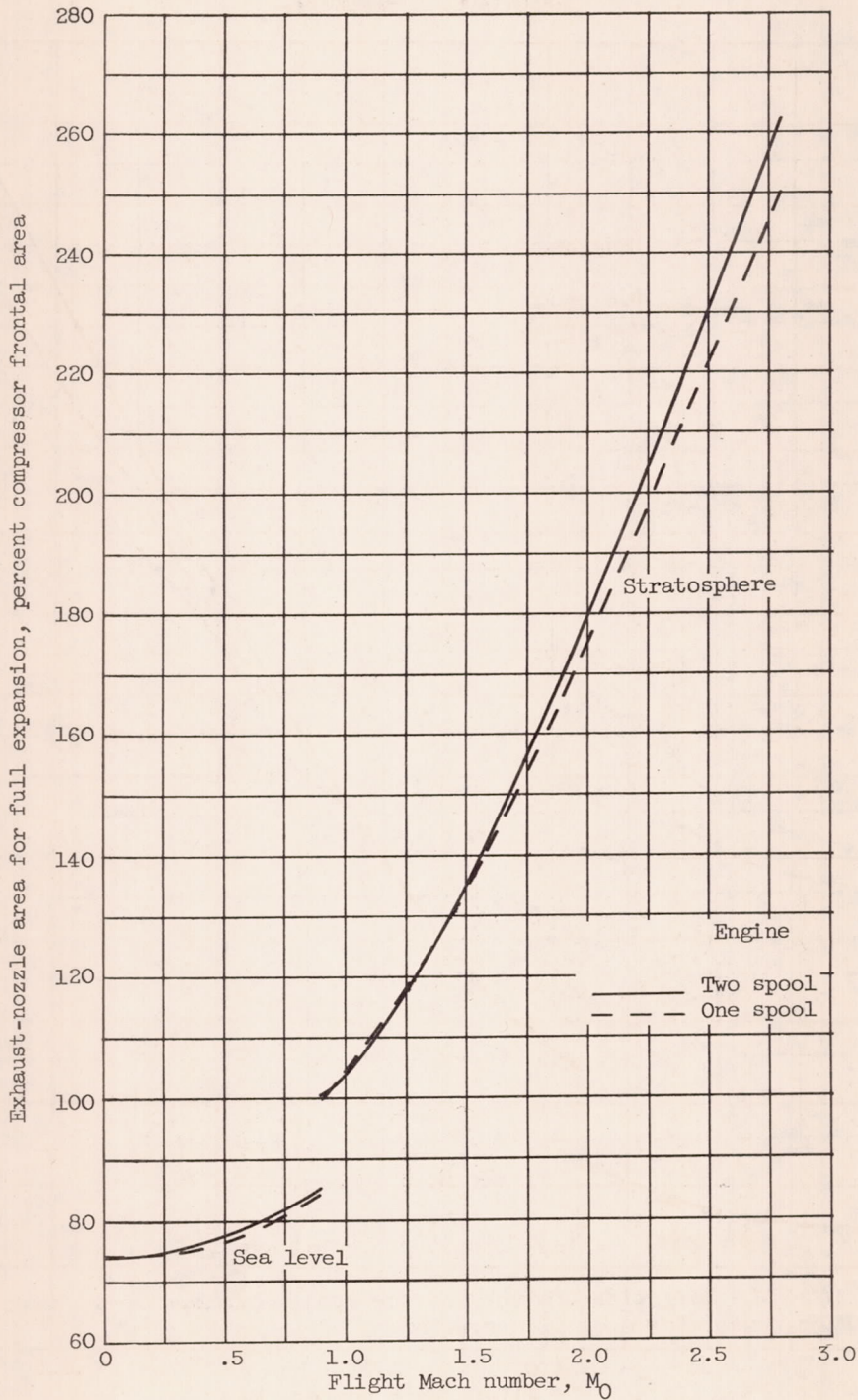


(d) Exhaust-nozzle area for full expansion. Afterburner inoperative.

Figure 6. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode I operation.

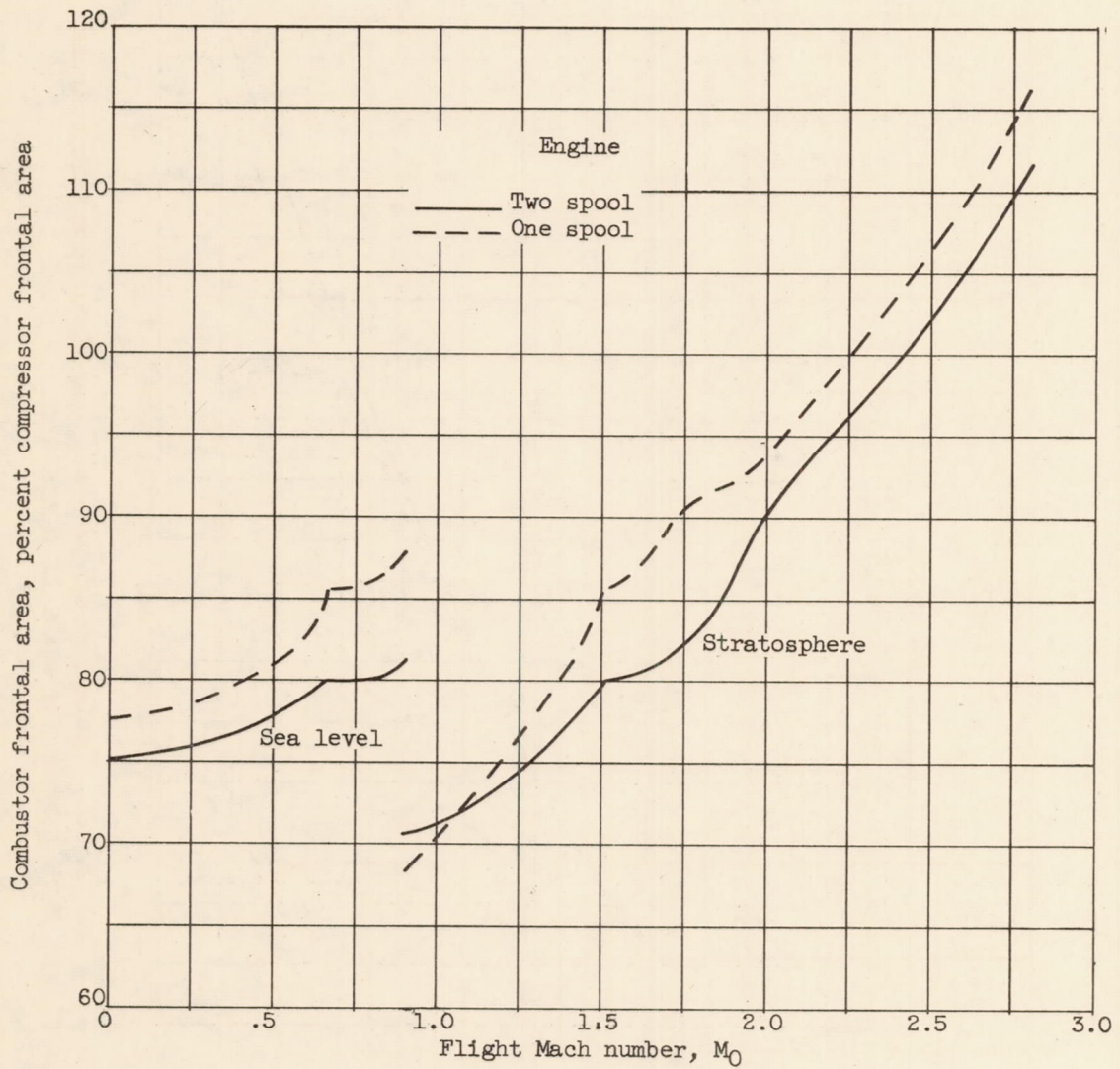
3819

3819
*CW-7 back



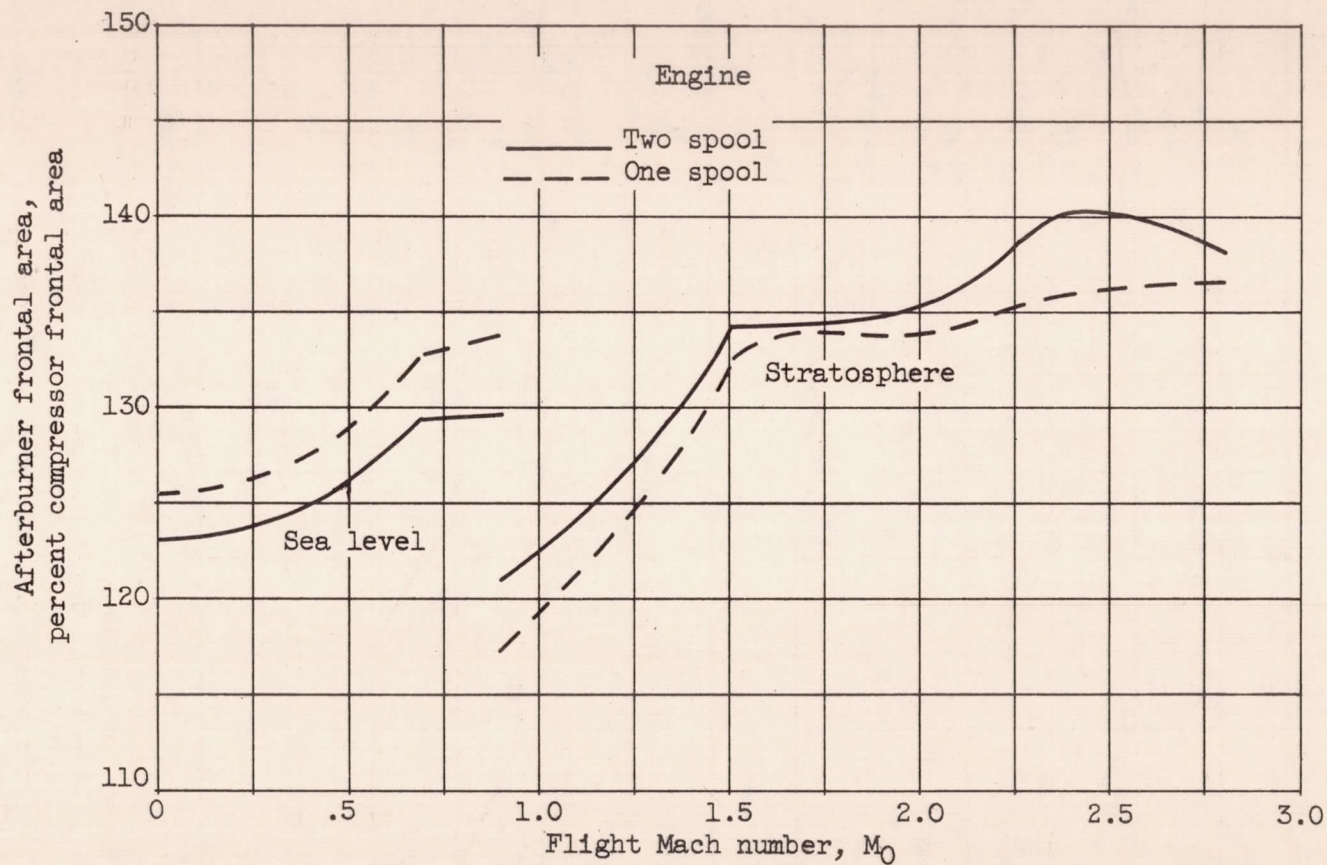
(e) Exhaust-nozzle area for full expansion. With afterburning.

Figure 6. - Concluded. Variation of two- and one-spool component areas with flight Mach number for mode I operation.



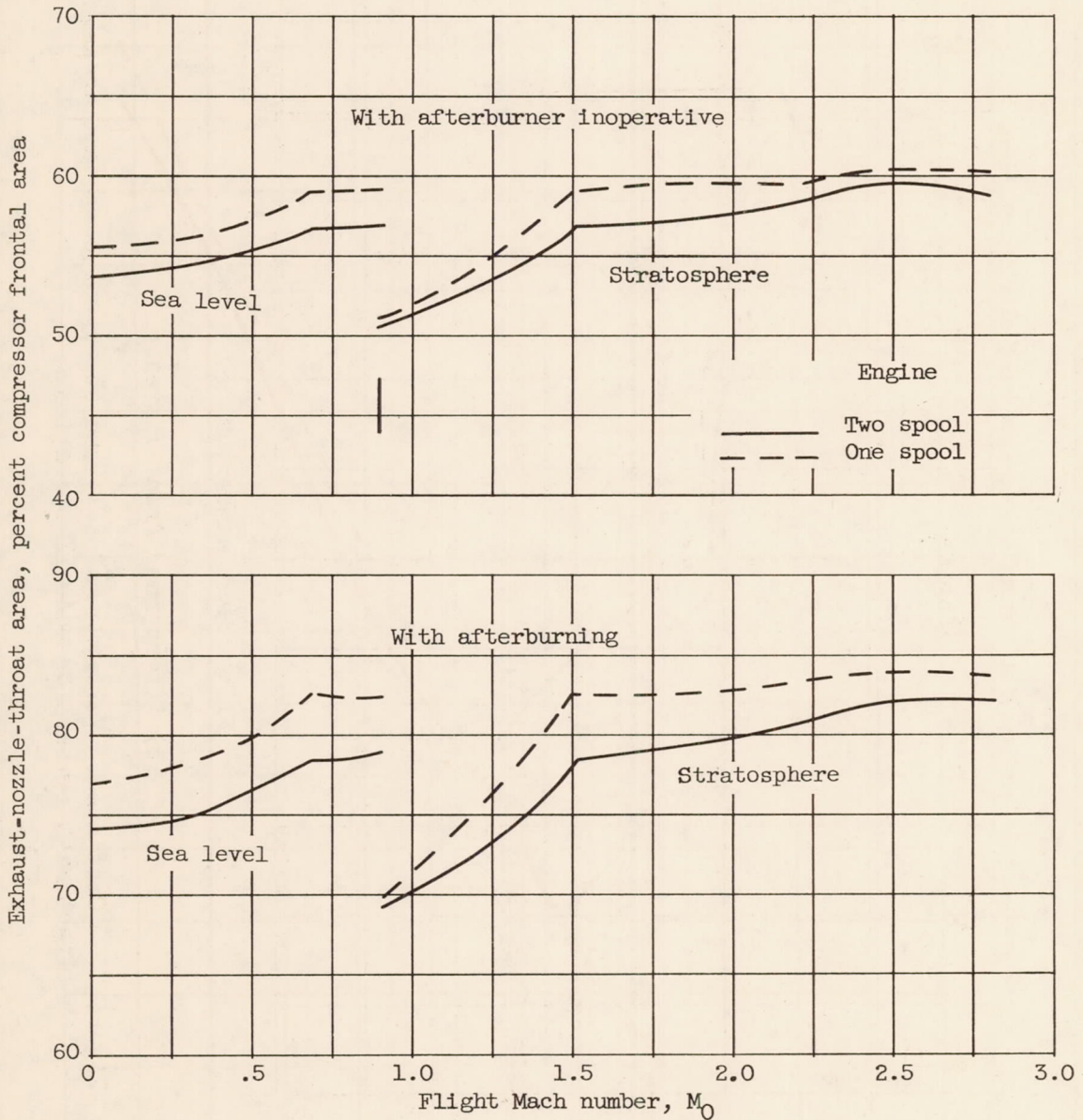
(a) Combustor frontal area.

Figure 7. - Variation of two- and one-spool component areas with flight Mach number for mode II operation.



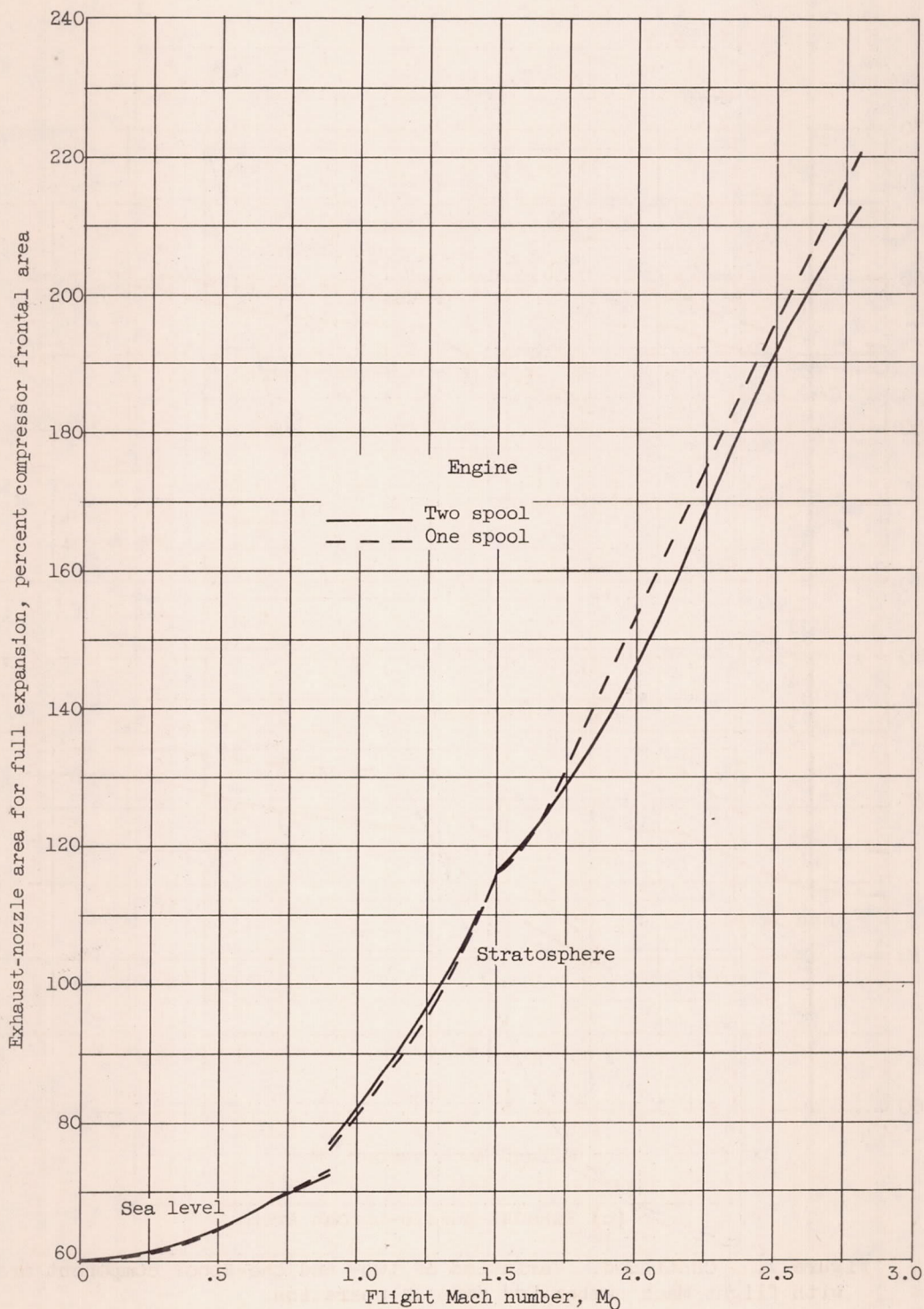
(b) Afterburner frontal area.

Figure 7. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode II operation.



(c) Exhaust-nozzle-throat area.

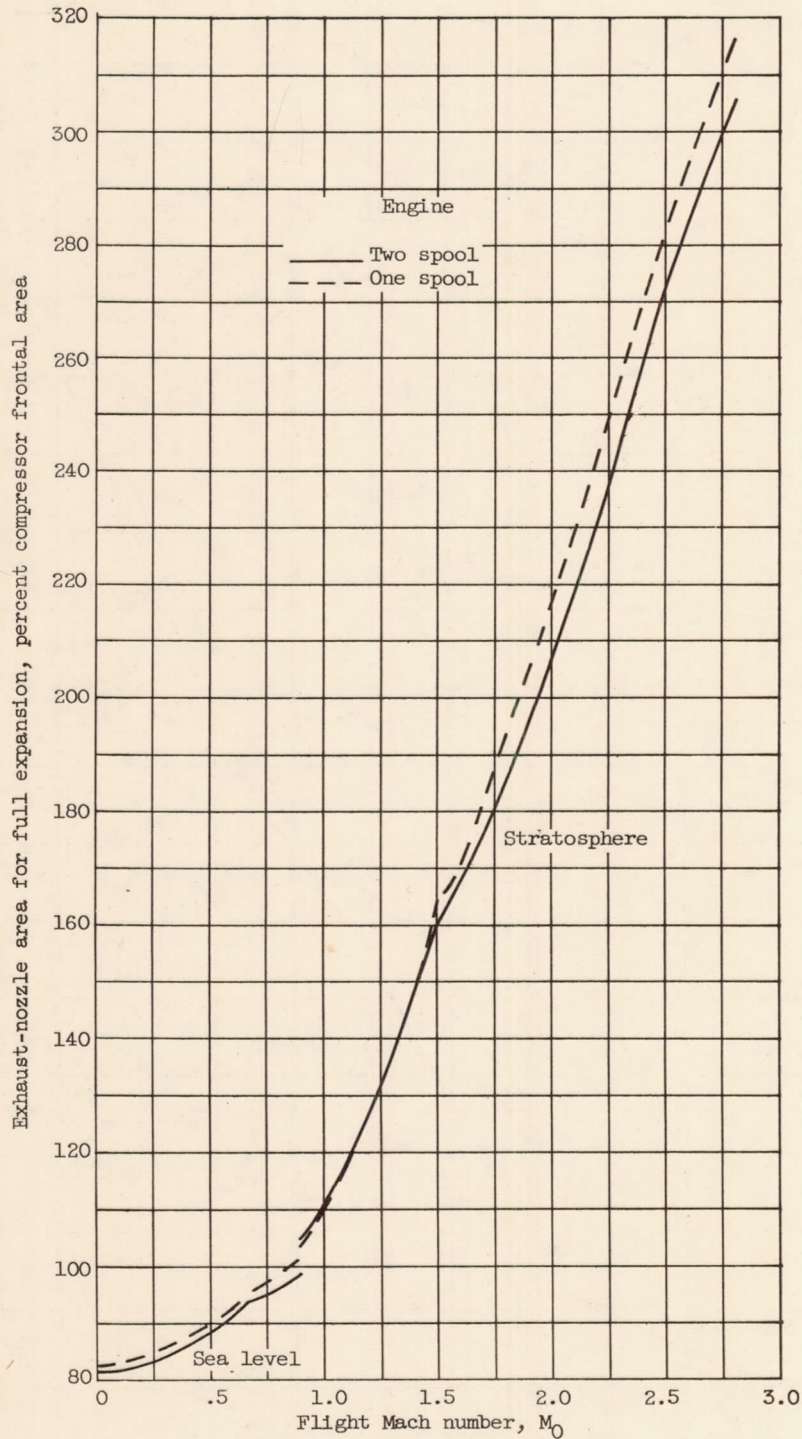
Figure 7. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode II operation.



(d) Exhaust-nozzle area for full expansion. Afterburner inoperative.

Figure 7. - Continued. Variation of two- and one-spool component areas with flight Mach number for mode II operation.

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(e) Exhaust-nozzle area for full expansion. With afterburning.

Figure 7. - Concluded. Variation of two- and one-spool component areas with flight Mach number for mode II operation.

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