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RESEARCH MEMORANDUM

PERFORMANCE INVESTIGATION AND HIGH-FLIGHT-SPEED APPLICATION OF A TURBINE WITH A VARIABLE-AREA STATOR

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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PERFORMANCE INVESTIGATION AND HIGH-FLIGHT-SPEED APPLICATION OF A TURBINE WITH A VARIABLE-AREA STATOR

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SUMMARY

An investigation was conducted in an NACA Lewis altitude test chamber to determine the performance characteristics of a turbine which employed a variable-area stator. Turbine-stator area was varied from 86 to 181 percent of standard area. For each stator area and a given engine speed, a range of turbine-inlet temperatures was obtained by use of a variable-area exhaust nozzle.

For the particular turbine used in this investigation, increasing the turbine-stator area from 89 to 181 percent of standard area increased the corrected turbine-gas flow 59 percent. The decrease in corrected turbine-gas flow required in a turbojet engine for operation over a range of flight Mach numbers from 0.9 to 2.2 in the stratosphere for constant-compressor-point operation (constant aerodynamic or corrected speed) is only 32 percent. Over this range of flight operation at constant turbine-inlet temperature the turbine efficiency varied approximately 5 points. The variation in turbine efficiency as turbine nozzle area was varied to maintain constant-compressor-point operation did not negate the gains in engine thrust expected from increased mass flow and improved compressor performance. Analysis indicates that a given percentage variation in compressor efficiency had a greater effect on thrust than an equal percentage variation in turbine efficiency.

Engine performance calculations for constant-compressor-point operation showed a 13 percent thrust increase over those for constant-rotational-speed operation at a flight Mach number of 2.2 for engines having equal take-off thrust. This 13 percent thrust advantage is mainly a result of the increase in air flow for the constant-compressor-point operation over that of the fixed-rotational-speed operation.

In addition to the performance investigation, a brief investigation of the effect of the variable-area turbine stator on acceleration characteristics was also conducted. Increasing the turbine-stator area from 86 to 134 percent of standard area approximately tripled the maximum acceleration of the particular engine investigated when only compressor stall was limiting the acceleration.

INTRODUCTION

The increasing engine-inlet temperatures accompanying flight into higher supersonic speeds reduce the corrected compressor rotational speed of the conventional turbojet engine and thereby reduce the corrected engine air flow, pressure ratio, and thrust. This reduction in engine performance, in the high-flight-speed region, greatly reduces the top speed potential of supersonic airplanes. However, analyses such as that of reference 1 have shown that this limitation of the conventional turbojet engine can be largely overcome by the increased operational flexibility provided by an adjustable-area turbine stator. One way these performance gains can be realized is by using a variable-area turbine stator, variable engine speed, and a variable-area exhaust nozzle to keep the compressor operating at its aerodynamic design point regardless of the flight speed or altitude of operation. With constant-compressor-point operation, the corrected engine speed, corrected air flow, compressor pressure ratio, and actual turbine-inlet temperature are held constant.

Additional work in this field (ref. 2) has included an experimental investigation of a two-stage turbine employing adjustment of the first-stage stator. This turbine stator provided a variation in turbine-inlet gas flow of about 10 percent, which, however, was not adequate to maintain constant-compressor-point operation over the wide range of flight conditions presently of interest. An analytical study (ref. 3) has shown that turbine design will have considerable effect on the ability of a variable-stator turbine to meet the requirements for constant-compressor-point operation without exceeding operating temperature limits or encountering limiting turbine loading. In addition to the foregoing operational requirements and turbine design considerations, a discussion of the aerodynamic design considerations for a variable turbine stator can be found in reference 4.

With the aforementioned background of analytical and test rig data for guidance, the turbine stator of a current production turbojet engine was made adjustable to determine (1) whether a single-stage turbine could operate over the required range of effective nozzle area for supersonic flight, and (2) the losses in turbine efficiency resulting from varying the turbine-nozzle area. These data were used to compute an example of engine performance with constant-compressor-point operation at flight Mach numbers up to 2.2. The constant-compressor-point operation was then compared with fixed-rotational-speed operation. The results of this analytical comparison together with the supporting experimental data are presented herein.

The turbine used in this investigation was a single-stage turbine which had a relatively high blade loading. It was designed by the free-vortex method and is, in a general way, representative of present-day turbines. However, as indicated in reference 3, the design of a turbine for constant-compressor-point operation requires that the high-flight-speed design point be somewhat compromised in order that the blade-loading limit will not be exceeded during off-design operation. Because this compromise has not been made in the turbine tested, the results obtained with this turbine may be considered conservative as compared with an optimum turbine design for an engine operating at a constant compressor point.

In addition to the possible advantages of a variable-area turbine nozzle for supersonic flight, there is, as discussed in reference 1, the possibility of improving the acceleration characteristics of a turbojet engine by employing adjustable turbine stators. Therefore, the acceleration characteristics for the particular engine configuration used in the investigation reported herein were determined for several turbine-stator areas and engine speeds.

APPARATUS

Engine and Installation

The current production-model axial-flow turbojet engine used in this investigation has a 12-stage compressor, eight can-type combustors, and a single-stage turbine. A schematic sketch of the engine showing instrumentation stations is presented in figure 1. Remotely actuated tabs were used to adjust the effective area of the exhaust nozzle. During the acceleration phase of the investigation, an exhaust nozzle having an area equal to approximately 130 percent of the turbine-exit area was used.

The engine was installed in an altitude test chamber 10 feet in diameter and 60 feet in length. A front bulkhead, which incorporated a labyrinth seal around the forward end of the engine, separated the inlet and exhaust sections of the chamber. A rear bulkhead was installed to act as a radiation shield and to prevent recirculation of the hot exhaust gases about the engine.

Turbine and Variable-Area Turbine Stator

The single-stage turbine was of free-vortex design with a hub-tip ratio of 0.7833 and at the design point had a corrected tip speed of 603 feet per second, a pressure ratio of 2.65, and a turbine efficiency of about 81 percent.

A sketch of the turbine stator used in this investigation is shown in figure 2. The stator blades and shrouds were production parts. Pins were welded into both ends of the blades and the shrouds were drilled for the pins. The actuating ring was located axially by means of the aft-frame flange and was rotated about the engine center line by means of screw-jack actuators. Welded to each blade outer pin was an arm and ball which fitted into a slot in the actuating ring. When the actuating ring was rotated, the motion was transmitted to the stator blades, causing each blade to rotate about its axis and change the flow area. The stator was designed to have a total chord angle variation of 22°, but because of the short actuating arms and the peening of the brass ring by the steel balls, there was a hysteresis, or lost motion, of approximately 5° in the indicated chord angle. The stator-blade tip clearance was about 3.5 percent of the blade height.

Instrumentation

The locations of the instrumentation stations at the inlet and outlet of each engine component are indicated in the sketch of the engine (fig. 1). The 24 total-pressure tubes at station 1 were located at the centers of 24 equal areas, and the 18 total-pressure tubes at station 9 were located 3 on each of 6 equal areas. The thermocouples at stations 1, 3, 4, and 6 and the total-pressure tubes at stations 3, 4, and 5 were located on approximately equal spacings. The altitude pressure surrounding the jet nozzle was measured by four lip static probes located in the exhaust portion of the chamber. The air flow through the engine was measured at station 1, the engine inlet, and such air flow was corrected for any leakages existing in the compressor. Steady-state fuel flow was measured by means of calibrated rotameters.

During the transient phase of the investigation, eight channels of a strip recorder were used to record tail-pipe total temperature, tail-pipe total pressure, engine speed, engine-inlet total pressure, engine-inlet dynamic pressure, compressor-outlet total pressure, altitude exhaust pressure, and fuel flow. The signal for each pressure was obtained from a strain-gage-type pressure transducer connected to a single probe and correlated with the other probes at that station during steady-state operation. Tail-pipe temperature was measured by means of a thermocouple which was electronically compensated for time lag and was correlated with the temperature measured at survey station 9 during steady-state operation. Engine speed signal was obtained from an engine driven generator and the fuel flow signal, from a turbine-type mass-flow meter.

PROCEDURE

For the steady-state data, engine-inlet ram-pressure ratio was maintained at a value which gave critical flow through the exhaust nozzle. Facility limitations prevented simulation of the actual inlet temperatures associated with high flight Mach number operation. Inasmuch as this was true, the engine-inlet-air temperature was held constant at approximately 410° R for all data. The low engine-inlet-air temperature also allowed the turbine-inlet temperature to be maintained at a minimum and thus prolonged the life of the turbine stator. In order to simulate engine operation at constant turbine-inlet temperature, each condition of simulated flight Mach number required a different actual turbine-inlet temperature. To ensure that the correct engine operating condition was obtained for each flight condition, the data were obtained by selecting several turbine-stator settings, and for each setting the exhaust-nozzle area was varied so as to cover a range of turbine-inlet temperatures at each of several engine speeds. This procedure in data taking permitted cross-plotting of the performance results in order to obtain the desired operating condition of the engine for the range of flight conditions considered.

The acceleration characteristics of the engine with a fixed-area exhaust nozzle (130 percent of turbine-outlet area) were determined for a range of turbine-stator areas at a flight Mach number of 0.4. At each stator area and over a range of engine speeds, the engine was subjected to step changes in fuel flow of increasing magnitude until either compressor stall or maximum allowable exhaust-gas temperature was encountered.

A list of the symbols used in this report appears in appendix A. Appendix B contains a detailed discussion of the methods of calculations.

RESULTS AND DISCUSSION

Turbine Performance

Turbine equivalent weight flow. - The effect of turbine-stator area on corrected turbine-gas flow is presented in figure 3(a) for a range of corrected turbine speeds. For a given turbine-stator area, the maximum corrected turbine speed that could be obtained in the engine configuration investigated was limited by the maximum exhaust-nozzle area or turbine limiting loading (ref. 3), while the minimum corrected turbine speed was limited by compressor surge or limiting exhaust-gas temperature. These limits converged as the turbine-stator area was increased. The stator was choked over the entire range investigated, so that for any given stator setting the corrected gas flow was constant. The greater part of the scatter in the data presented on this figure is attributed to the difficulty in setting or holding a desired stator area, as previously discussed.

As shown in figure 3(a), turbine speed had no effect on corrected turbine-gas flow; therefore the corrected turbine-gas flow may be presented as a function of turbine-stator area only (fig. 3(b)). Increasing the turbine-stator area from 89 to 181 percent of standard area resulted in a 59 percent increase in corrected gas flow. The decreasing slope of the curve with increasing stator area is an indication of decreasing flow coefficient. The corrected turbine-gas flow that could be expected if the flow coefficient were constant is represented by the dashed line.

Turbine performance maps. - Turbine performance maps obtained for three stator settings are presented in figure 4. For the range of data presented in each map, corrected gas flow is constant (fig. 3). These maps represent actual data taken from the engine and therefore have been corrected for variable specific heats. For a given corrected turbine speed, an increase in the turbine-nozzle area provided an increase in air flow but a decrease in turbine work and turbine pressure ratio (comparison of figs. 4(a), (b), and (c)). The variation in turbine efficiency level between turbine performance maps is primarily a result of the variation in turbine-inlet Mach number and angle of incidence as the stator vanes are rotated to change the stator area. Therefore, the turbine efficiency trends with increasing nozzle area cannot be generally defined until a particular mode of operation (which defines pressure ratio, speed, and area) has been selected. An illustrative example was computed to compare performance of the constant-compressor-point and constant-rotational-speed modes of operation and so to determine the amount which turbine efficiency would vary if the variable-area turbine were used in an engine designed for constant-compressor-point operation and whether or not these changes in efficiency would negate the engine performance gains to be expected from constant-compressor-point operation. On the basis of the results of this analysis, turbine efficiency for the constant-compressor-point mode of operation will be discussed in one of the subsequent sections of this report.

Constant-Compressor-Point Operation

Computed example of constant-compressor-point operation. - If constant-compressor-point operation is to be maintained as the flight speed is increased, the actual rotational speed of the engine must be increased in proportion to the square root of the engine-inlet-temperature increase. Increasing the actual rotational engine speed requires that the turbine-stator effective flow area be decreased in proportion to the square root of the reciprocal of the increase in engine-inlet temperature. In addition, applying the turbine to constant-compressor-point operation requires that the turbine and compressor be matched for work and air flow at the selected design point on the compressor map. The matching of these components must also preclude exceeding turbine limiting loading at high-flight-speed conditions. The air flow can be matched by adjusting the relative size of the compressor with respect to the turbine. In the analysis, both size and stress limitations have been ignored.

The compressor characteristics employed were those obtained experimentally from the engine used to determine the variable-area turbine performance. In order that the matching requirements previously discussed would be satisfied, the compressor operating point was selected at a corrected compressor speed of 75 percent of rated speed, compressor pressure ratio of 3.3, and compressor efficiency of approximately 0.785.

Although the pressure ratio is low compared with those of common practice, it is believed that the trends in turbine and engine performance shown by the performance of engines with this pressure ratio for constant-rotational-speed and constant-compressor-point operation will be similar to the trends for engines with higher pressure ratios.

When the compressor operating point has been selected and the experimental relation between flight Mach number and turbine-stator area is known, the turbine operating line may be determined. The turbine performance is presented in figure 5 as a function of both turbine-stator area and flight Mach number. As turbine-stator area was increased, turbine efficiency increased until approximately standard area was reached, and then decreased. Incidence angle and rotor-inlet relative Mach number at the mean section generally decreased as stator area was increased. The decrease in turbine efficiency as the stator area was increased above standard area was caused by the rapid decrease in blade incidence angle to negative values. The decrease in efficiency at the closed stator settings is a result of the increase in rotor-inlet Mach number above values of approximately 0.8. As indicated in reference 3, a rotor-inlet Mach number of about 0.8 is a limit for good turbine efficiency.

Figure 6 indicates the thrust gains computed for constant-compressor-point operation as compared with fixed-rotational-speed operation. The compressor had an efficiency at design point for constant-compressor-point operation of 0.785 and an efficiency variation of 0.10 for the corrected speed range covered in the fixed-rotational-speed operation. The turbine efficiency for constant-compressor-point operation varied from 0.72 to 0.78, while the turbine efficiency for fixed-rotational-speed was essentially constant at 0.77. The intersection of the corrected weight flow and compressor efficiency curves which occurs at a flight Mach number of 1.27 in the stratosphere is where the design point for constant-compressor-point operation corresponds to the fixed-rotational-speed operation. Being matched at the altitude flight condition, the two modes of operation will also be matched at standard sea-level take-off conditions.

Below a flight Mach number of 1.27 in the stratosphere the thrusts for the two modes of operation were approximately equal for the particular engine configuration. However, at a flight Mach number of 2.2, a 13 percent increase in thrust was obtained for constant-compressor-point operation. At the low flight speeds the thrusts for the two modes of operation were approximately equal because the effect of the gain in air flow and loss in compressor efficiency on thrust for fixed-rotational-speed operation is equal to the effect of the loss in turbine efficiency for constant-compressor-point operation. At the high flight speeds, the thrust advantage of the constant-compressor-point operation results from the severe fall-off in air flow for fixed-rotational-speed operation. Therefore, computations have been made assuming variations in these efficiencies in order to generalize the performance trends for the particular compressor-turbine combination and to determine the sensitivity of the performance with constant-compressor-point operation to various levels of compressor and turbine efficiency.

Effect of turbine efficiency on constant-compressor-point engine performance. - As an illustration of the effect of turbine efficiency on constant-compressor-point engine performance, curves of thrust coefficient are presented in figure 7 for turbine efficiencies of 0.77 and 0.87 and constant compressor efficiency of 0.785. These curves indicate that the 10 point increase in turbine efficiency resulted in an increase in thrust of about 3 to 5 percent for flight Mach numbers from 0.9 to 2.2.

Effect of compressor efficiency on constant-compressor-point engine performance. - As a further study of the effect of component efficiency on this type of operation, an analysis was made of the engine performance with compressor efficiencies of 0.785 and 0.885 and constant turbine efficiency of 0.77. The results of this analysis are also presented in figure 7 and indicate that the 10 point increase resulted in about a 5 to 7 percent increase in thrust, the larger effect being at the high flight speed. The effect of compressor efficiency on engine performance is greater because the effect of losses in the compressor is compounded by a reduction in allowable energy to be added in the combustor (temperature limit) and an increase in work required by the turbine, while a reduction in turbine efficiency results only in an increase in pressure energy taken from the gases by the turbine.

Engine Acceleration Characteristics

The acceleration characteristics for the particular engine investigated were determined over a range of corrected engine speeds and turbine-stator areas for a flight Mach number of 0.4 and a fixed-area exhaust nozzle (130 percent of turbine-outlet area). Figure 8 is a typical curve showing the corrected acceleration as a function of turbine-stator area for a corrected engine speed of 50 percent of rated speed. Any point under the compressor stall limit is an attainable condition, with the maximum acceleration being the compressor stall limit unless some limiting gas temperature is reached. At a given engine speed, increasing the stator area increases the maximum acceleration; but if a temperature limit is encountered before compressor stall, increasing the stator area will decrease the acceleration.

The maximum acceleration is shown as a function of corrected engine speed in figure 9, where the data are cross plotted for four turbine-stator areas. The acceleration is limited by the initiation of compressor surge or stall. As pointed out in reference 1, increasing the turbine-stator area increased the surge margin by reducing the compressor pressure ratio and permitted the addition of more fuel to the combustors, thus providing faster acceleration. When only the acceleration and not the transient temperature limit was considered, increasing the stator area from 86 to 134 percent approximately tripled the maximum acceleration over the range of corrected speeds investigated. As turbine-stator area or corrected engine speed is increased, limiting transient exhaust-gas temperature may be reached before compressor surge, thus preventing maximum acceleration from being realized. The maximum acceleration for a particular turbine-stator area is typical for a conventional turbojet engine in that as speed is increased, maximum acceleration increases and occurs at higher exhaust-gas temperatures.

CONCLUDING REMARKS

For the particular turbine used in this investigation, increasing the turbine-stator area from 89 to 181 percent of standard area increased the corrected turbine-gas flow 59 percent. The variations in turbine-stator area and corrected turbine-gas flow required for a range of flight Mach numbers from 0.9 to 2.2 in the stratosphere for constant-compressor-point operation were only 48 and 32 percent, respectively. Over this range of flight operation at constant turbine-inlet temperature, the turbine efficiency varied approximately 5 points.

A sample computation using experimental component performance maps, but ignoring any size or stress limitations, showed no thrust increase at low flight Mach numbers and a 13 percent increase in thrust for constant-compressor-point operation over fixed-rotational-speed operation at a flight Mach number of 2.2 in the stratosphere when the two engines were sized for equal thrust at take-off conditions. The 13 percent increase in thrust at the high flight Mach number is mainly a result of the higher air flow for the constant-compressor-point operation.

The variation in turbine efficiency as turbine nozzle area was varied to maintain constant-compressor-point operation did not negate the gains in engine thrust expected from increased mass flow and improved compressor performance. Compressor efficiency has a greater effect on thrust than an equal variation in turbine efficiency. An analysis indicates that a 10 percent variation in turbine efficiency resulted in about a 3 to 5 percent variation in thrust, while a 10 percent variation in compressor efficiency resulted in a 5 to 7 percent increase in thrust over the flight speed range investigated.

Increasing the turbine-stator area from 86 to 134 percent of standard area tripled the maximum acceleration of the particular engine investigated as limited by compressor stall. However, as the engine speed is increased the maximum acceleration occurs at higher exhaust-gas temperatures, so that a temperature limitation may restrict the allowable acceleration.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 28, 1954.

APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
C_T	thrust coefficient
C_V	velocity coefficient
c_p	specific heat at constant pressure, Btu/(lb)($^{\circ}$R)
F	thrust, lb
g	acceleration due to gravity, 32.2 ft/sec^2
h	enthalpy of gas, Btu/lb
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, 53.4 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
V	velocity, ft/sec
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr
α_i	turbine rotor inlet incidence angle, deg
γ	ratio of specific heat for gases
δ	pressure correction factor, P/2116 (total pressure divided by NACA standard sea-level pressure)
η	efficiency

θ temperature correction factor, $\gamma T/(1.4)(519)$ (product of γ and total temperature divided by product of γ and temperature for air at NACA standard sea-level conditions)

ρ density, slugs/cu ft

Subscripts:

c compressor

e engine

g gas

t turbine

0 free-stream conditions

1 engine or compressor inlet

3 compressor outlet

4 turbine inlet

5 turbine outlet

6 exhaust-nozzle inlet

APPENDIX B

METHODS OF CALCULATION

Flight Mach number and airspeed. - Flight Mach number and velocity were calculated from free-stream conditions.

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1}\left[\left(\frac{P_0}{p_0}\right)^{\frac{\gamma_1-1}{\gamma_1}} - 1\right]}$$

and

$$V_0 = M_0\sqrt{\gamma_1 gRT_1\left(\frac{p_0}{P_0}\right)^{\frac{\gamma_1-1}{\gamma_1}}}$$

Weight flow. - Air flow was determined from pressures and temperatures measured at the engine inlet by the equation

$$W_{a,1} = p_1A_1\sqrt{\frac{2\gamma_1 g}{(\gamma_1 - 1)RT}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1-1}{\gamma_1}} - 1\right]}$$

Turbine-inlet weight flow was calculated taking into account compressor-air-flow leakage and engine fuel flow.

$$W_{g,4} = W_{a,1} - W_c + \frac{W_{f,e}}{3600}$$

Compressor efficiency. - The compressor adiabatic efficiency is defined as

$$\eta_c = \frac{(P_3/P_1)^{\frac{\gamma_c-1}{\gamma_c}} - 1}{\frac{T_3}{T_1} - 1}$$

Turbine efficiency. - The turbine adiabatic efficiency is defined as

$$\eta_t = \frac{1 - T_5/T_4}{1 - (P_5/P_4)^{\frac{\gamma_t - 1}{\gamma_t}}}$$

Incidence angle and exit Mach number. - The turbine-rotor-inlet incidence angle and rotor-inlet Mach number were determined by the method described in reference 5.

Thrust. - The net thrust for the turbojet engine with a convergent-divergent exhaust nozzle is

$$F = \frac{W_{g,6}}{g} V_j - \frac{W_{a,1}}{g} V_0$$

where V_j is the effective velocity of the jet at the exhaust-nozzle exit; or

$$F = \frac{W_{g,6}}{g} C_V \sqrt{\frac{2\gamma_6}{(\gamma_6 - 1)} \frac{R}{g} T_6 \left[1 - \left(\frac{p_0}{P_6}\right)^{\frac{\gamma_6 - 1}{\gamma_6}}\right]} - \frac{W_{a,1}}{g} V_0$$

Calculation procedure for engine performance. - For fixed-rotational-speed operation, each flight Mach number will have a corresponding corrected engine speed. At each corrected engine speed values of corrected air flow, compressor pressure ratio, and compressor efficiency may be obtained from the compressor characteristics. Assuming a constant turbine efficiency and setting a value for turbine-inlet temperature allow calculation of the turbine pressure ratio and turbine-outlet temperature.

$$\frac{P_5}{P_4} = \left[1 - \frac{\Delta h_c}{c_{p,t} \eta_t T_4}\right]^{\frac{\gamma_t}{\gamma_t - 1}}$$

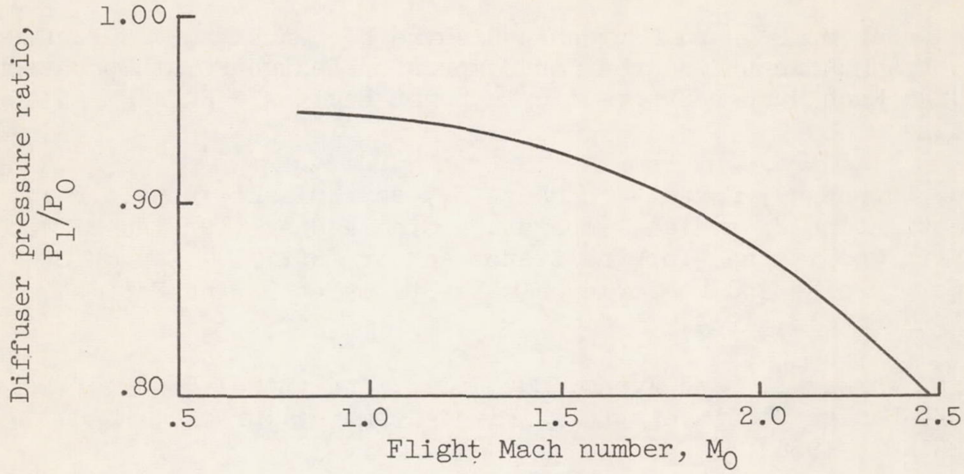
and

$$T_6 = T_4 - \frac{\Delta h_c}{c_{p,t}}$$

where

$$\Delta h_c = \Delta h_t = \frac{c_{p,c} T_1}{\eta_c} \left[\left(\frac{P_3}{P_1} \right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1 \right]$$

The engine-inlet diffusion pressure loss was obtained from the following curve:



The exhaust-nozzle pressure ratio may be determined by assuming a combustor pressure loss of 0.96 and a tail-pipe pressure loss of 0.96.

$$P_6/p_0 = (P_0/p_0)(P_1/P_0)(P_3/P_1)(P_4/P_3)(P_5/P_4)(P_6/P_5)$$

With the exhaust-nozzle pressure ratio, gas flow, and temperature, the thrust parameter of the engine may be calculated (engine fuel flow assumed equal to compressor leakage air flow).

$$C_T\, M_0^2 = \frac{F}{\frac{\rho_0}{2g} V_0^2 A_c} M_0^2$$

where A_c is the frontal area of the compressor.

For constant-compressor-point operation, the calculation procedure is identical to the preceding discussion except that the corrected engine speed, corrected air flow, compressor pressure ratio, and compressor efficiency are held constant and turbine efficiency is allowed to vary.

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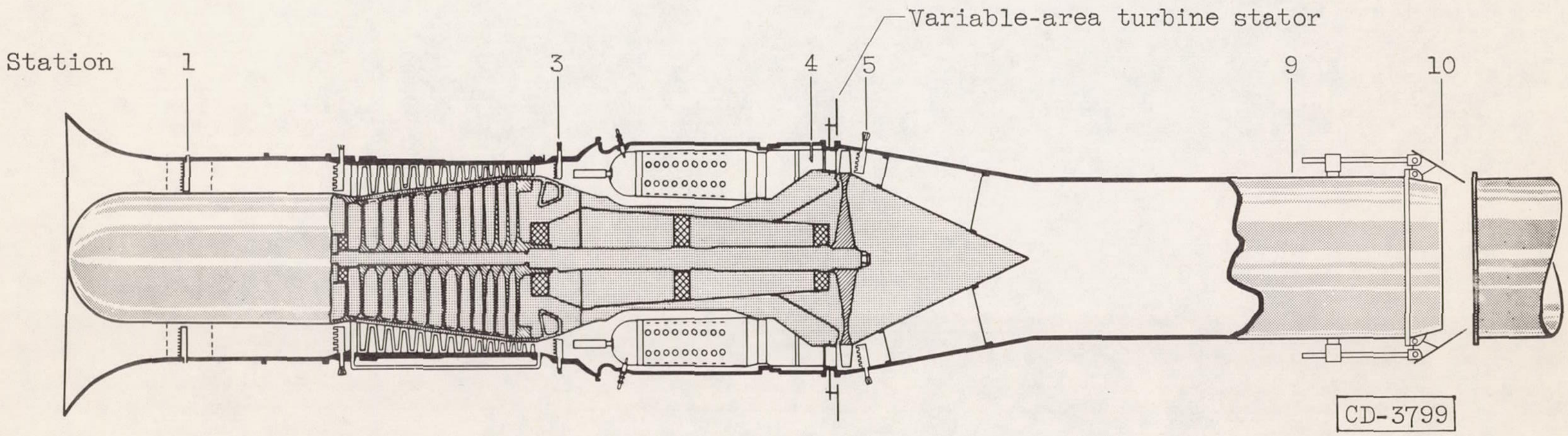


Figure 1. - Sketch of engine showing instrumentation station locations.

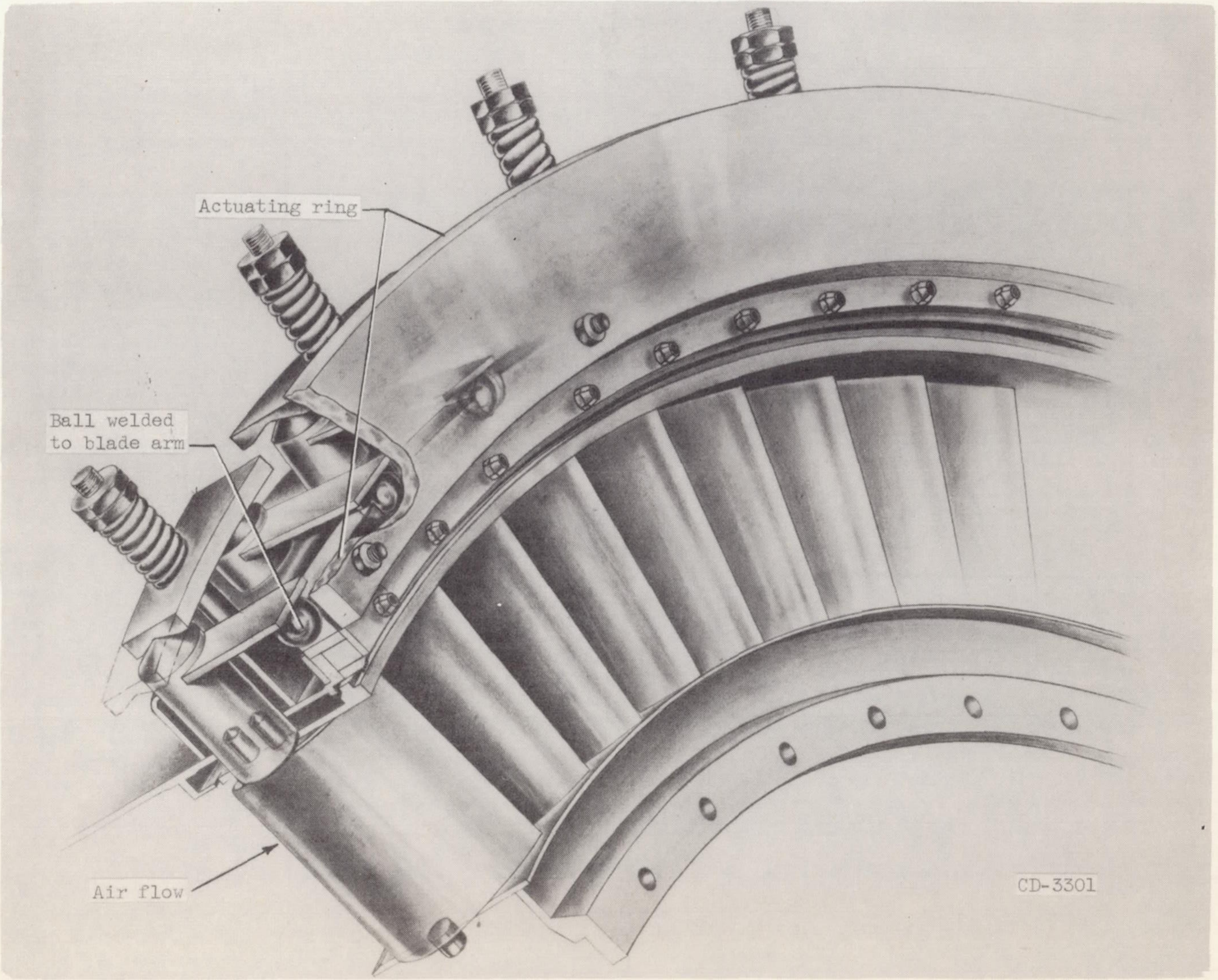
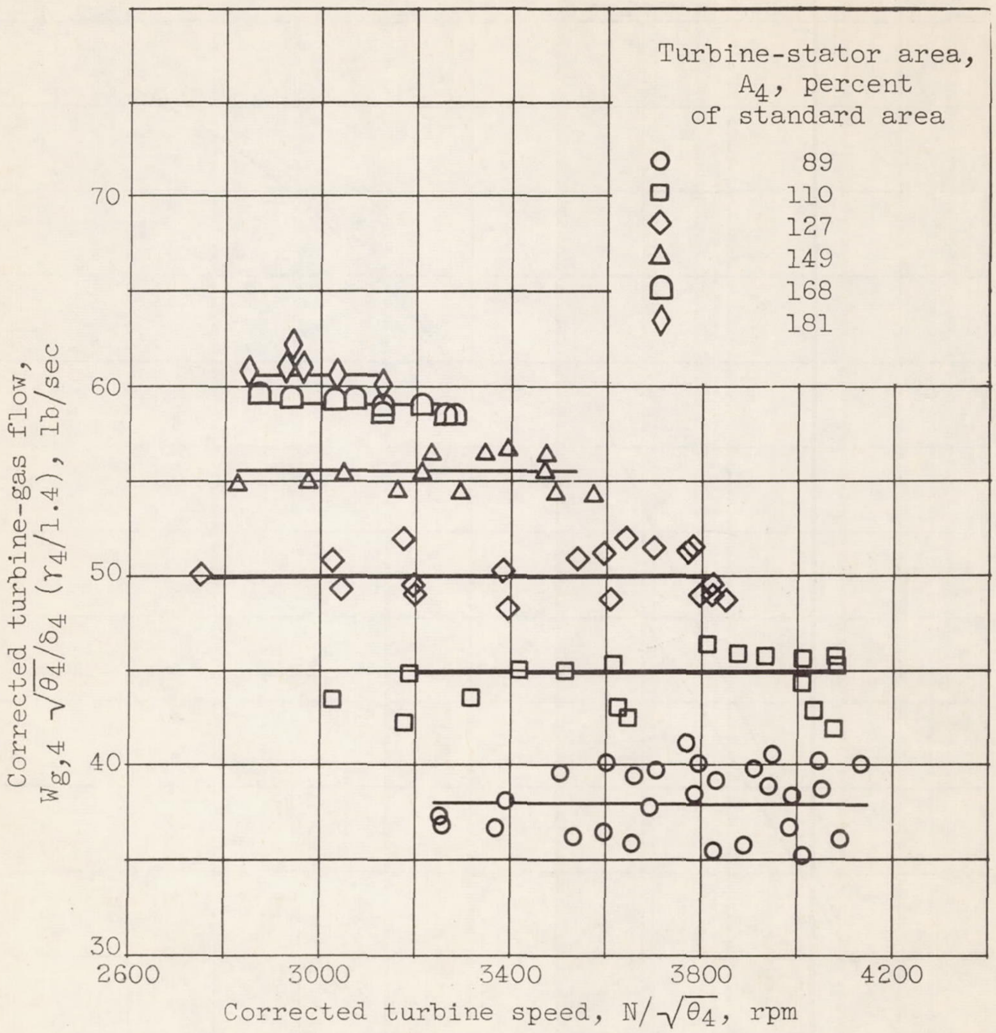
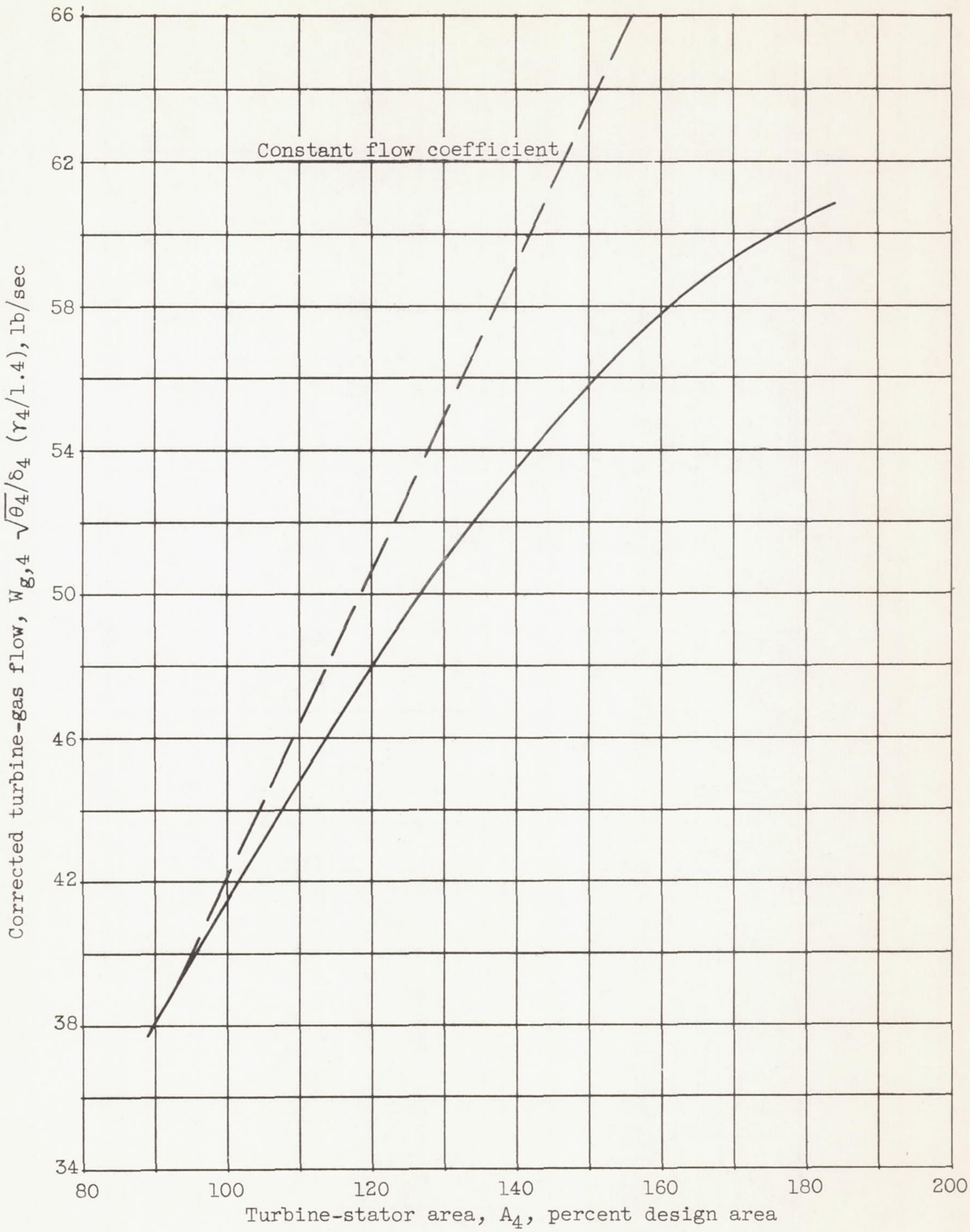


Figure 2. - Sketch of variable-area turbine stator showing blade actuating mechanism.



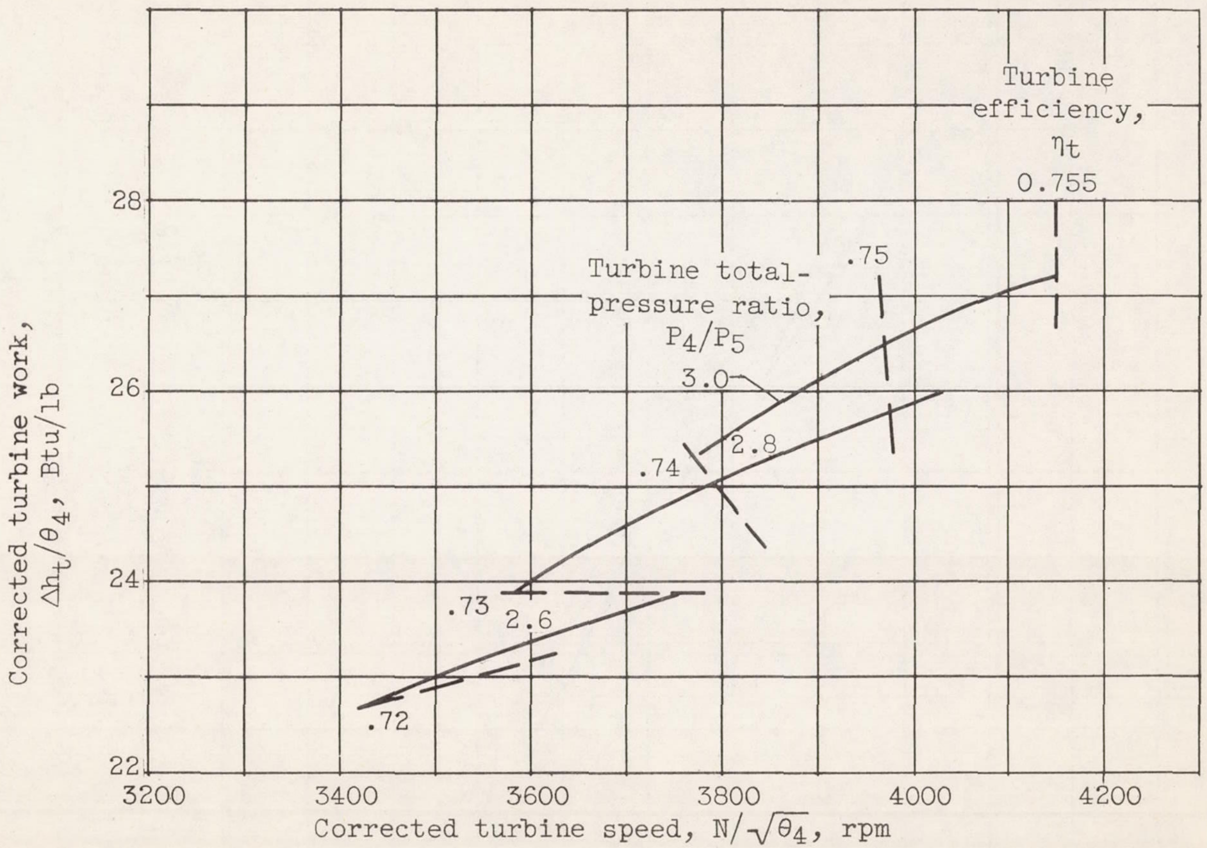
(a) Variation with turbine speed.

Figure 3. - Effect of turbine-stator area on corrected turbine-gas flow.



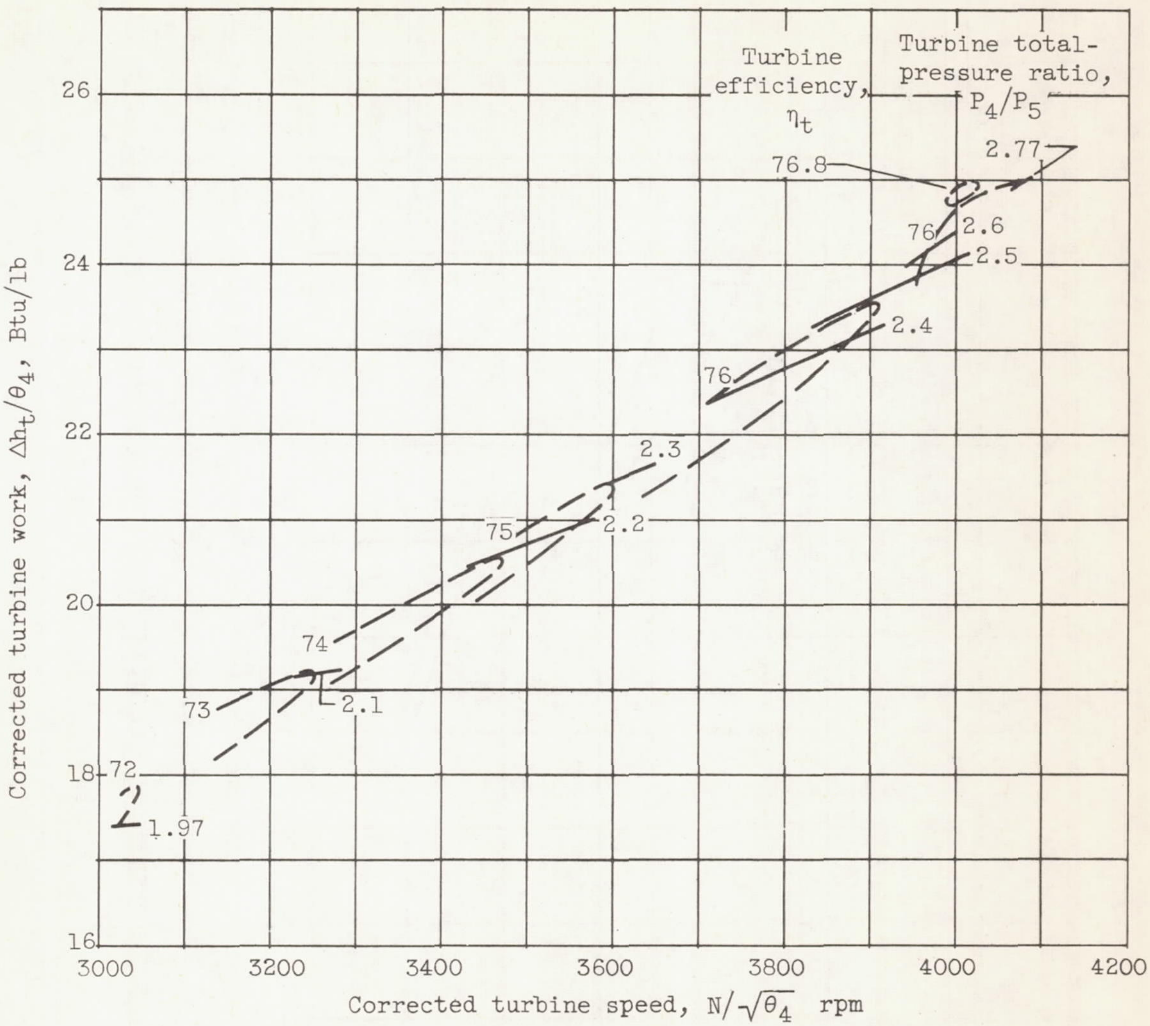
(b) Average variation with area.

Figure 3. - Concluded. Effect of turbine-stator area on corrected turbine-gas flow.



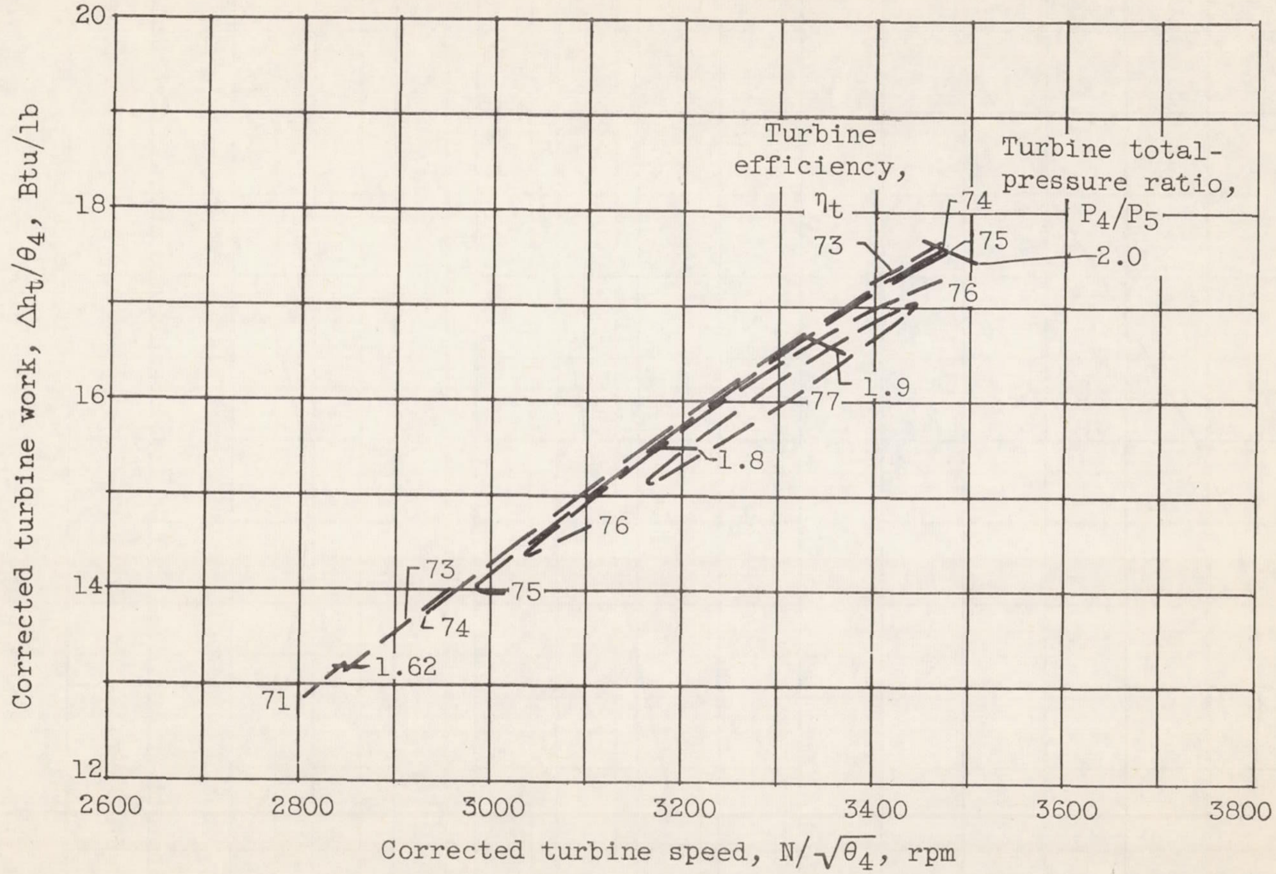
(a) Turbine-stator area A_4, 89 percent of standard area. Corrected turbine-gas flow, $W_{g,4} \sqrt{\theta_4}/\delta_4 \ (\gamma_4/1.4)$, 38 pounds per second.

Figure 4. - Turbine performance map.



(b) Turbine-stator area A_4, 110 percent of standard area. Corrected turbine-gas flow, $W_{g,4}\ \sqrt{\theta_4}/\delta_4\ (\gamma_4/1.4)$, 45 pounds per second.

Figure 4. - Continued. Turbine performance map.



(c) Turbine-stator area A_4, 149 percent of standard area. Corrected turbine-gas flow, $W_{g,4} \sqrt{\theta_4}/\delta_4$ $(\gamma_4/1.4)$, 55.5 pounds per second.

Figure 4. - Concluded. Turbine performance map.

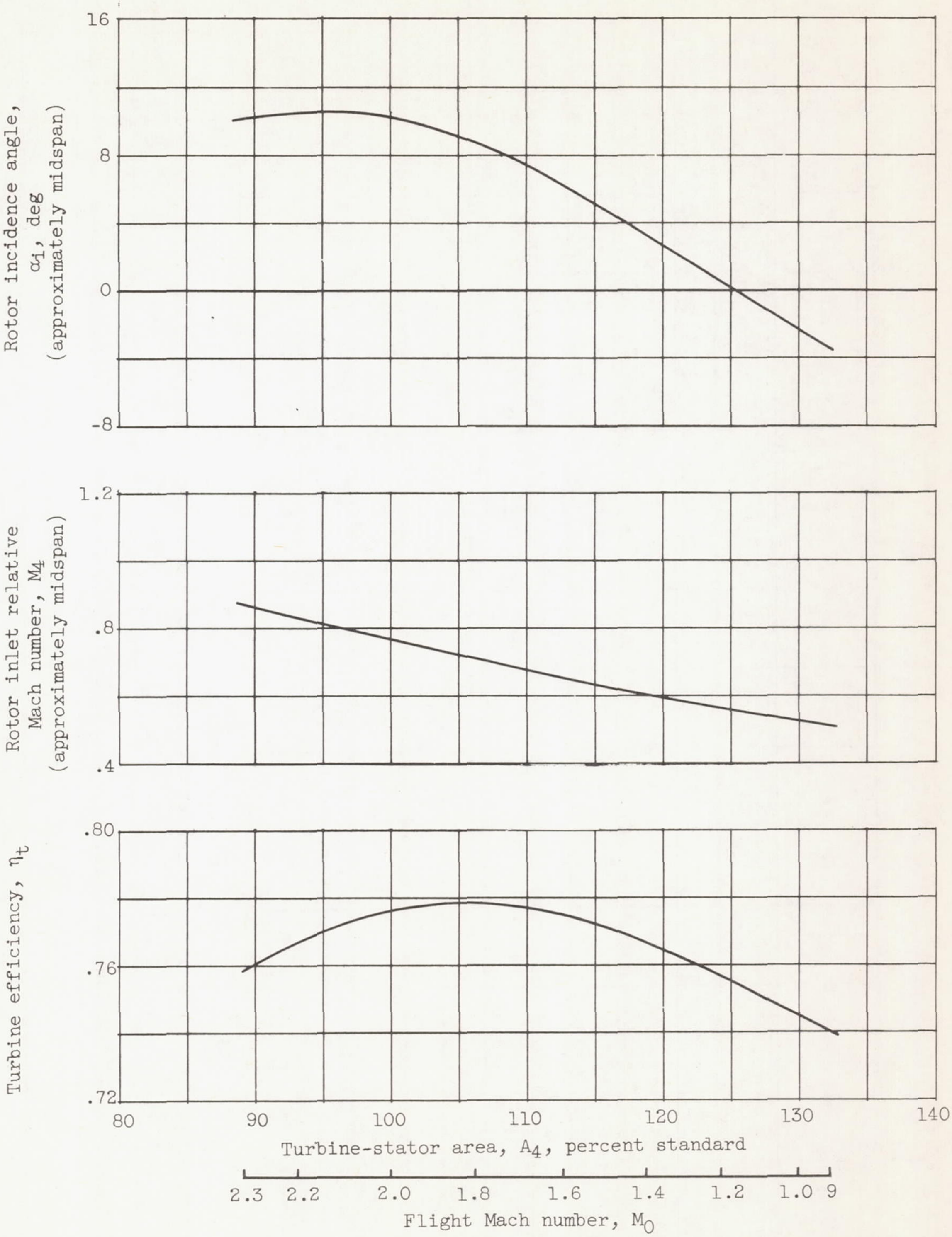


Figure 5. - Turbine performance along operating line for constant-compressor-point operation.

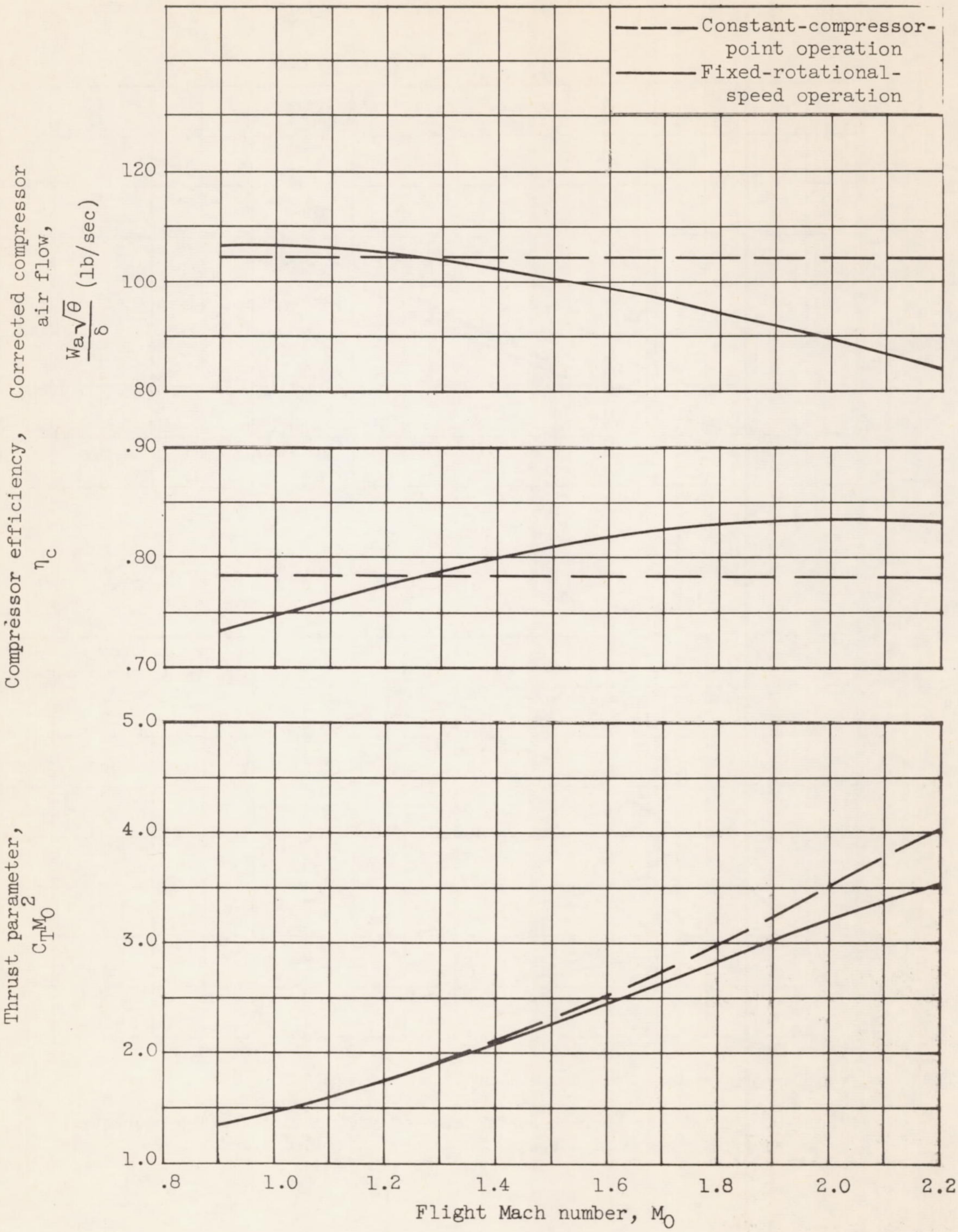


Figure 6. - Comparison of thrust for fixed-rotational-speed and constant-compressor-point operation.

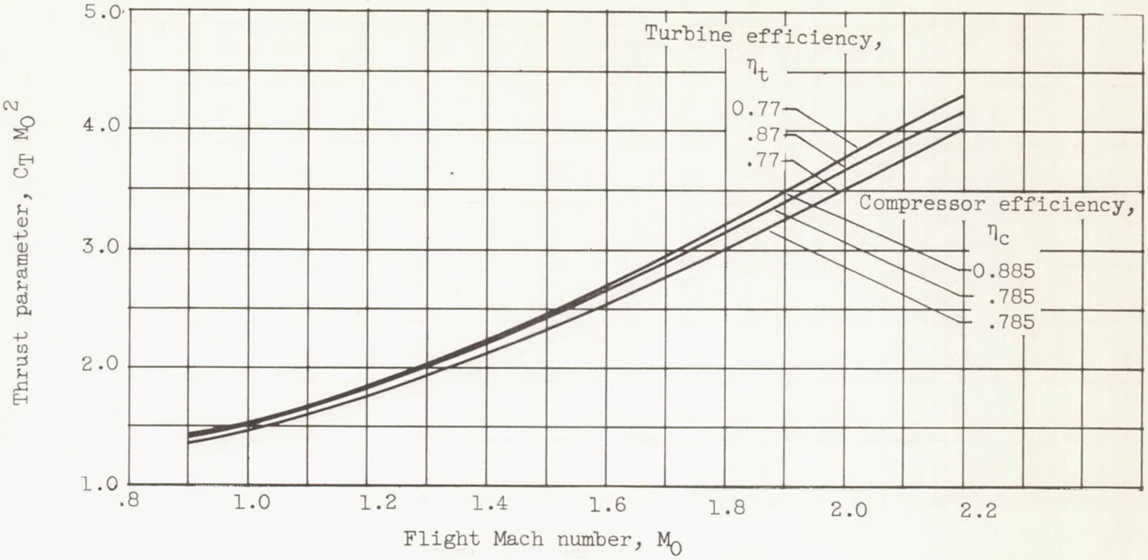


Figure 7. - Effect of compressor and turbine efficiency on thrust for constant-compressor-point operation.

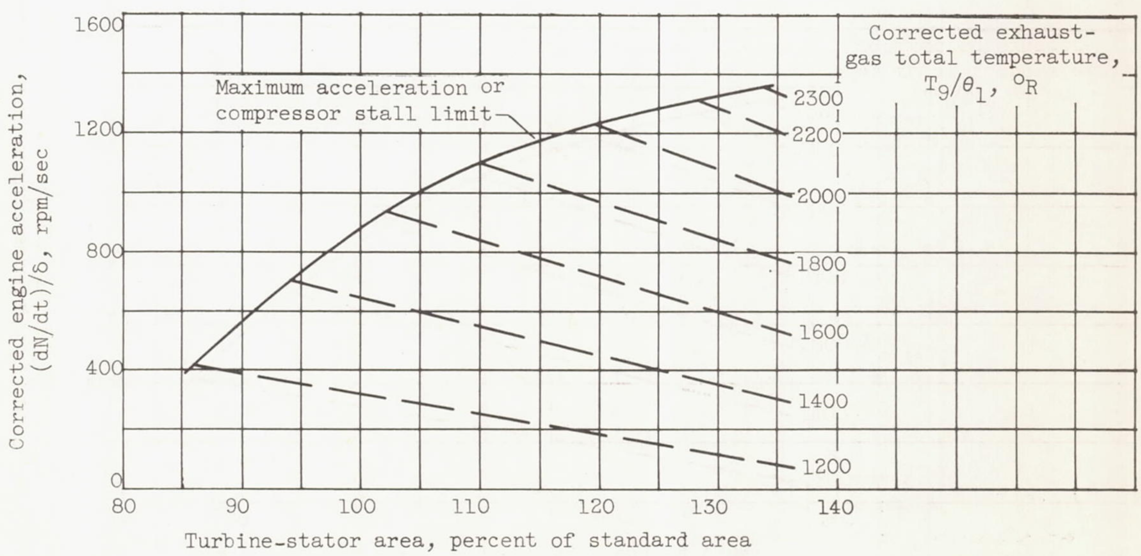


Figure 8. - Effect of turbine-stator area on engine acceleration characteristics. Corrected engine speed, 50 percent of rated speed.

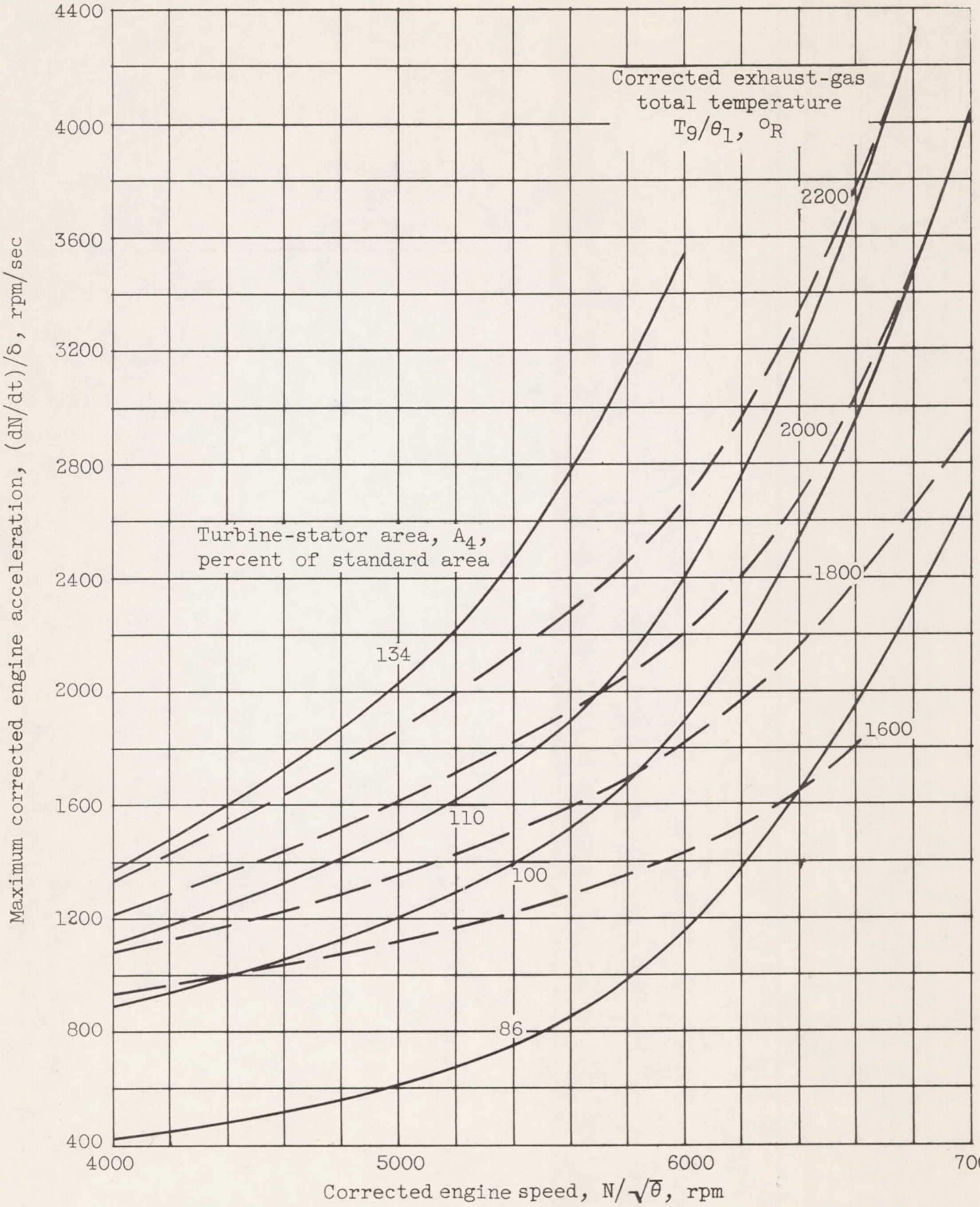


Figure 9. - Effect of engine speed and turbine-stator area on maximum engine acceleration.