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INVESTIGATION OF A HIGH-TEMPERATURE SINGLE-STAGE TURBINE SUITABLE FOR AIR COOLING AND TURBINE STATOR ADJUSTMENT

I - DESIGN OF VORTEX TURBINE AND PERFORMANCE WITH STATOR AT DESIGN SETTING

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SUMMARY

The turbojet engine for the supersonic airplane must operate over a wide range of temperature at the compressor inlet. This variation in compressor inlet temperature may impose severe engine operational problems over the range of flight conditions, depending on the particular mode of engine operation. An investigation was conducted at the NACA Lewis laboratory to study the aerodynamic problems associated with a turbine design for a mode of engine operation which utilizes turbine stator adjustment to maintain a fixed compressor operating point.

An analysis of the turbine requirements for a particular compressor operating point over a range of engine temperature ratio from approximately 6.25 to 3.31 indicated that a feasible single-stage air-cooled turbine design could be obtained within reasonable aerodynamic limits. This range of engine temperature ratio can represent engine operation from sea-level take-off to a flight Mach number of 2.12 in the stratosphere and requires that the turbine operate over a range of inlet equivalent weight flow, equivalent work output, and equivalent tip speed.

A single-stage air-cooled turbine was designed to meet these requirements, and the performance of a scale model was obtained with the stator at the design take-off setting. At the design equivalent shaft work of 19.05 Btu per pound and design equivalent tip speed of 662.6 feet per second, a brake internal efficiency of 0.87 was obtained.

INTRODUCTION

The turbojet engine for the supersonic airplane must operate over a wide range of compressor inlet temperature, from the take-off condition to that of a supersonic flight Mach number in the stratosphere. The compressor inlet temperature varies as a result of the ambient temperature variation at different flight altitudes and of the variation of ram temperature rise at different flight Mach numbers. As an example, the total temperature at the compressor inlet is 745° R at a flight Mach number of 2.12 in the stratosphere, and at the sea-level take-off condition the compressor inlet temperature is 518.4° R. This variation of compressor inlet temperature may impose severe problems in engine operation over a range of flight conditions.

The engine of the supersonic airplane must have high thrust per unit engine weight. One way this may be accomplished is by the use of higher turbine inlet temperatures than are in current practice. At supersonic speeds these high temperatures are also required in order to obtain an engine temperature ratio (ratio of turbine inlet to compressor inlet temperature) that is sufficient for good cycle efficiency. Reference 1 shows that the performance of a supersonic fighter airplane is significantly increased when turbine inlet temperatures are increased from approximately 2000° to 2500° R. However, if high turbine inlet temperatures are used, it will be necessary to cool both the turbine stator and rotor blades or to use a high-temperature material capable of withstanding the required stresses at these elevated temperatures.

An investigation was conducted at the NACA Lewis laboratory to study the aerodynamic problems associated with the design of single-stage turbines for a turbojet engine for the supersonic airplane. The variation in compressor inlet temperature may require that either the compressor or turbine, or both, be capable of operating over a wide range of flow conditions. Whether this flexibility is required of the compressor or turbine, or both, is dependent on the specified mode of engine operation. Four different modes of operation were considered, and one mode, that which uses turbine stator adjustment and exhaust nozzle area adjustment to keep the compressor operating point fixed, has certain characteristics which are advantageous for supersonic flight. However, this mode of operation shifts the problem of operational flexibility to the turbine and requires that the turbine operate over an extreme range of equivalent weight flows and equivalent speeds. No flexibility is required for the compressor, as its operating point is fixed. The specific problem to be considered in this report is the design of an air-cooled single-stage turbine which uses turbine stator adjustment to operate over a wide range of equivalent weight flow and equivalent speed.

Reference 2 presents an analysis of the aerodynamic problems encountered in the turbine when the turbojet engine is operated such that the compressor operating point remains fixed. This report shows, in general, that the change in the turbine flow conditions depends on the particular turbine application and the design conditions of required compressor work, turbine inlet temperature, and range of engine temperature ratio.

For this investigation, particular compressor operating characteristics were chosen, and the range of engine temperature ratio considered is approximately 3.31 to 6.25. An engine temperature ratio of 3.31 corresponds to a turbine inlet temperature of 2460° R at a flight Mach number of 2.12 in the stratosphere. If the turbine inlet temperature is held constant at 2460° R, the engine temperature ratio at sea-level take-off is 4.75. However, if turbine inlet temperatures greater than 2460° R are employed to obtain additional thrust for take-off, or if the engine is operated with a turbine inlet temperature of 2460° R below a flight Mach number of 1.25 in the stratosphere, higher engine temperature ratios will result.

The purpose of this investigation is to determine the effect of the changes of flow conditions due to stator adjustment for a particular turbine as the engine temperature ratio varies from approximately 3.31 to 6.25.

Because of the high turbine inlet temperatures, the turbine stator and rotor blades will be air-cooled by means of air passing through internal cooling passages. The turbine design therefore represents a compromise of the aerodynamic design for minimum blade loss to satisfy the cooling requirements. A scale model of this turbine was fabricated and its performance was obtained as a single-stage turbine having a tip diameter of 15 inches and with entrance conditions slightly below atmospheric pressure and 685° R temperature. In this report, only the performance with the stator set at its design position, sea-level take-off with a turbine inlet temperature of 2460° R, is presented. The performance data were obtained over a range of equivalent tip speed from 45 to 121 percent of design equivalent tip speed and of total pressure ratio from 1.3 to 2.4.

SYMBOLS

A	annulus area, sq ft
a	local speed of sound, ft/sec
a'_{cr}	critical velocity, $\left(\frac{2\gamma}{\gamma+1} g RT'\right)^{1/2}$, ft/sec
b	ratio of bleed air flow to compressor weight flow
E	specific work, Btu/lb
f	ratio of fuel flow to compressor exit flow
g	acceleration due to gravity, ft/sec^2
h	specific enthalpy, Btu/lb

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J mechanical equivalent of heat, ft-lb/Btu

l ratio of leakage flow to compressor inlet flow

M Mach number

N turbine speed, rpm

p absolute pressure, lb/sq ft

R gas constant, ft-lb/(lb)($^{\circ}$R)

r radius, in.

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

U blade velocity, ft/sec

V absolute flow velocity, ft/sec

W relative flow velocity, ft/sec

w weight flow, lb/sec

α angle of absolute flow velocity with tangential direction, deg

β angle of relative flow velocity with tangential direction, deg

γ ratio of specific heats

δ ratio of gas pressure to NACA standard sea-level pressure, p'/p_0

η_C compressor adiabatic efficiency

η_T turbine brake internal efficiency based on stagnation condition

ε $\dfrac{\gamma_0}{\gamma_e}\left[\left(\dfrac{\gamma_e+1}{2}\right)^{\frac{\gamma_e}{\gamma_e-1}} \middle/ \left(\dfrac{\gamma_0+1}{2}\right)^{\frac{\gamma_0}{\gamma_0-1}}\right]$, specific heat correction factor

θ^* square of ratio of critical velocity to critical velocity at NACA standard sea-level temperature, $(a'_{cr}/a_{cr,0})^2$

τ torque, lb-ft

Subscripts:

0	NACA standard sea-level conditions
1	compressor inlet
2	compressor exit
3	turbine stator inlet
4	turbine stator exit, rotor inlet
5	turbine rotor exit
C	compressor
cr	critical
e	engine operating condition
h	turbine hub
m	turbine mean
s	isentropic
T	turbine
t	turbine tip
u	tangential
v	absolute
w	relative
x	axial

Superscript:

'	total state

ENGINE OPERATION FOR SUPERSONIC FLIGHT

The mode of engine operation determines the range of flow conditions over which either the compressor or turbine must operate because of the variation of compressor inlet temperature. Four possible modes of operation are presented in the following table; their advantages and disadvantages are briefly discussed:

Mode of operation	Geometric adjustment
Fixed engine speed	None
Fixed compressor equivalent speed and turbine inlet temperature	Exhaust nozzle area
Fixed compressor equivalent speed and engine temperature ratio	None
Fixed compressor equivalent speed, compressor pressure ratio, and turbine inlet temperature	Turbine stator flow area and exhaust nozzle area

Fixed engine speed. - This is the manner in which some current turbojet engines operate and is adequate for subsonic flight in the stratosphere, as the variation of the compressor inlet temperature from that at the take-off condition is relatively small. This mode of operation can be obtained for a fixed turbine stator area and a fixed exhaust nozzle area; thus, it is mechanically simple because no change in engine geometry is required as engine temperature ratio varies. However, for supersonic flight, this mode of operation has several disadvantages.

At the take-off condition approximately a 20 percent increase in compressor equivalent speed over that at a flight Mach number of 2.12 in the stratosphere is required. Accompanying this increase in compressor equivalent speed should be increases in compressor pressure ratio and compressor equivalent weight flow. However, a compressor which has this flexibility in flow conditions will be large and heavy, since the operational flexibility would normally be obtained by additional compressor stages. At the supersonic flight condition, only a fraction of the potential output of this large and heavy compressor is being used, and consequently, the performance of the supersonic airplane may decrease.

Fixed compressor equivalent speed and turbine inlet temperature. - This mode of engine operation is one which will circumvent the disadvantage resulting from the required increase in compressor equivalent speed. The engine speed is allowed to vary as the compressor inlet temperature varies such that the compressor equivalent speed remains constant. This mode of engine operation is accomplished through the use of a variable-area exhaust nozzle. The compressor operating point moves toward the compressor stall limit as the engine temperature ratio increases. Thus, if the compressor stall limit is not to be encountered as the flight condition changes from that at a Mach number of 2.12 in the stratosphere to take-off, the compressor must again have the required flexibility in pressure ratio. The compressor which has this flexibility is again going to be large and heavy and may adversely affect the performance of the supersonic airplane.

Fixed compressor equivalent speed and engine temperature ratio. - Using this mode of engine operation is one way of circumventing the disadvantages associated with the required compressor flexibility that prevailed with the two previous modes of engine operation. In this mode, the engine speed and turbine inlet temperature are so varied that the compressor equivalent speed and engine temperature ratio remain constant as the compressor inlet temperature changes. An advantage of this mode of operation is that the compressor and turbine continue to operate at their design points at all conditions, and thus good component performance can be achieved. However, this mode of operation has a serious disadvantage in that for engine operation within a specified maximum turbine inlet temperature, low thrust will be available for take-off. For instance, operating at a Mach number of 2.12 in the stratosphere with a turbine inlet temperature of 2460° R would require that the turbine inlet temperature be 1715° R for take-off, resulting in a low take-off thrust.

Fixed compressor equivalent speed, compressor pressure ratio, and turbine inlet temperature. - This mode of engine operation keeps the compressor operating point fixed, thus not requiring flexibility of the compressor, and yet will allow high turbine inlet temperatures to be used at all flight conditions. A high thrust is therefore available for take-off, and the disadvantages of the previous mode of engine operation are offset. This mode of operation requires adjustment of the turbine stator area and of the exhaust nozzle area. Since the stator flow area and the engine speed vary, the turbine must be capable of operating over a wide range of equivalent weight flow and equivalent speed. For instance, for take-off, the turbine equivalent speed must decrease 16.5 percent from that at a flight Mach number of 2.12 in the stratosphere. The turbine inlet equivalent weight flow will increase approximately 20 percent for the same change in flight condition. The actual amount of this change in turbine equivalent weight flow is a function of the change in fuel flow, engine leakage, and bleed for cooling from one flight condition to the other. Thus, this mode of operation transfers the engine flexibility requirement from the compressor to the turbine.

From the previous discussion of the different modes of engine operation for supersonic flight, it becomes apparent that the last mode, which uses turbine stator and exhaust nozzle area adjustment to keep the compressor operating point fixed, does not have the disadvantages associated with the compressor flexibility requirement of the first two modes of engine operation, or a low turbine inlet temperature for take-off that was present in the third mode of engine operation. However, the turbine must be capable of satisfactory operation over a wide range of flow conditions, and this problem requires further study.

Probably none of these possible modes of engine operation will be used exclusively, but some combination will be used. However, fixed-point compressor operation has certain advantages for supersonic flight and requires maximum flexibility of the turbine. Consequently, this mode of engine operation was chosen for this investigation as representing the most extreme turbine problem.

TURBINE ANALYSIS AND DISCUSSION

Turbine analysis. - The ratio of the turbine inlet to compressor inlet temperature (engine temperature ratio) is the basic engine parameter which controls, to a large extent, the range of flow conditions over which the adjustable-stator turbine must operate. In order to study the specific variation of flow conditions for a particular turbine, it is necessary to specify the range of engine temperature ratio and the compressor design point operating characteristics. Also necessary are certain assumptions about factors which affect the matching of the compressor and turbine and the flow conditions within the turbine.

A turbine inlet temperature of 2460° R was specified at a flight Mach number of 2.12 in the stratosphere. For this constant turbine inlet temperature, figure 1 presents the range of engine temperature ratio that will be encountered as the flight Mach number is decreased both in the stratosphere and at sea level. Thus the range of engine temperature ratio to be considered in this analysis is approximately 3.31 to 6.25. A supersonic airplane will probably not fly at low subsonic Mach numbers in the stratosphere. Consequently, the engine temperature ratios greater than 4.75 (sea-level take-off with turbine inlet temperature of 2460° R) were assumed to correspond to higher turbine inlet temperatures for take-off, since the actual engine rpm and corresponding centrifugal blade stress are lower than those for a flight Mach number of 2.12 in the stratosphere. In order to maintain constant equivalent compressor speed, the actual engine rpm is proportional to the square root of the compressor inlet temperature. Therefore the actual engine rpm is reduced 16.5 percent as the flight condition changes from 2.12 in the stratosphere to sea-level take-off. For a given engine geometry there is a corresponding 30 percent decrease in centrifugal blade stress.

The design point compressor operating characteristics selected for the analysis are as follows:

Compressor pressure ratio, p_2'/p_1'	3.96
Compressor equivalent speed, $N/\sqrt{\theta_1^*}$, rpm	11,500
Compressor adiabatic efficiency, η_C	0.70
Compressor equivalent weight flow, $w\sqrt{\theta_1^*}/\delta_1$, lb/sec	75.00

These compressor operating characteristics were assumed to remain constant for all engine operating conditions.

Because of the high turbine inlet temperatures, it is necessary to estimate the blade cooling air flow variation with engine temperature ratio. Figure 2 presents the variation of the blade cooling air (bled at the compressor exit) with engine temperature ratio that is specified in this analysis. At the low engine temperature ratio, which corresponds to a flight Mach number of 2.12 in the stratosphere, zero cooling air bleed was specified because ram air could possibly be used for turbine blade cooling provided that the cooling air was discharged at ambient pressure. At a temperature ratio of 4.75 or above (sea-level take-off), the values specified should be capable of satisfactorily cooling the turbine blades. Other factors which affect the matching of the compressor and turbine were specified as follows:

Engine leakage, l 0.02
Burner pressure ratio, p_3'/p_2' 0.95
Fuel air ratio, f (approximate) . . . . 0.013 to 0.040

With these specifications and assumptions, the turbine variables of equivalent speed, equivalent work, inlet equivalent flow, and exit equivalent flow were determined as described in the following paragraphs.

Turbine equivalent speed variation. - Since the compressor and turbine are coupled together, the following equation may be written:

$$\frac{N}{\sqrt{\theta_3^*}} = \frac{N}{\sqrt{\theta_1^*}}\sqrt{\frac{\theta_1^*}{\theta_3^*}} \tag{1}$$

This equation can be transformed to

$$\frac{N}{\sqrt{\theta_3^*}} = \frac{N}{\sqrt{\theta_1^*}}\sqrt{\left(\frac{\gamma}{\gamma+1}\right)_C \left(\frac{\gamma+1}{\gamma}\right)_T \frac{T_1'}{T_3'}} \tag{2}$$

The known value of the compressor equivalent speed $N/\sqrt{\theta_1^*}$ can be substituted in equation (2), and the variation of the turbine equivalent speed with the engine temperature ratio can be calculated. The results of the calculations are presented in figure 3, which shows that the turbine equivalent speed must decrease as the engine temperature ratio increases.

Turbine equivalent work variation. - Since the compressor and turbine power are equal if the turbine work done on the cooling air as it flows through the turbine blades is neglected, it follows that

$$\frac{E_T}{\theta_3^*} = \left(\frac{E_C}{\theta_1^*}\right)\left[\frac{1}{(1-b-l)(1+f)}\right]\left(\frac{\theta_1^*}{\theta_3^*}\right) \tag{3}$$

This equation can be transformed to

$$\frac{E_T}{\theta_3^*} = \left(\frac{E_C}{\theta_1^*}\right)\left[\frac{1}{(1-b-l)(1+f)}\right]\left[\left(\frac{\gamma}{\gamma+1}\right)_C \left(\frac{\gamma+1}{\gamma}\right)_T\right]\left(\frac{T_1'}{T_3'}\right) \tag{4}$$

The value of the compressor equivalent work E_C/θ_1^*, which is constant, can be substituted in equation (4). Using the specified compressor bleed b, the fuel-air ratio f, and the leakage l for each engine temperature ratio, the variation of the turbine equivalent work with the engine temperature ratio can be found. The results of these calculations are presented in figure 4, which shows that the turbine equivalent work must decrease as the engine temperature ratio increases.

Variation of turbine inlet equivalent weight flow. - Again using continuity between the compressor and the turbine, the following equation can be written:

$$\frac{w_T\sqrt{\theta_3^*}}{\delta_3}\left(\frac{\delta_3}{\delta_2}\right)\left(\frac{\delta_2}{\delta_1}\right)\left(\sqrt{\frac{\theta_1^*}{\theta_3^*}}\right) = \frac{w_C\sqrt{\theta_1^*}}{\delta_1}\left[(1-b-l)(1+f)\right] \tag{5}$$

This equation can be transformed to

$$\frac{w_T\sqrt{\theta_3^*}}{\delta_3} = \left(\frac{w_C\sqrt{\theta_1^*}}{\delta_1}\right)\left[(1-b-l)(1+f)\right]\left(\frac{p_2'}{p_3'}\right)\left(\frac{p_1'}{p_2'}\right)\sqrt{\left(\frac{\gamma}{\gamma+1}\right)_T\left(\frac{\gamma+1}{\gamma}\right)_C\left(\frac{T_3'}{T_1'}\right)} \tag{6}$$

in which the compressor equivalent weight flow $w_C\sqrt{\theta_1^*}/\delta_1$, the compressor pressure ratio p_2'/p_1', and the burner pressure drop p_3'/p_2' remain constant. The results of these calculations are presented in figure 5, and show that the turbine equivalent weight flow at the stator inlet, or the "swallowing capacity" of the turbine, must increase as the engine temperature ratio increases. This is accomplished by increasing the stator flow area as the engine temperature ratio increases.

Variation of turbine exit equivalent weight flow. - Again using continuity, the turbine exit equivalent flow is obtained from the following equation:

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$$\frac{w_T\sqrt{\theta_5^*}}{\delta_5}\left(\frac{\delta_5}{\delta_3}\right)\left(\frac{\delta_3}{\delta_2}\right)\left(\frac{\delta_2}{\delta_1}\right)\left(\sqrt{\frac{\theta_1^*}{\theta_3^*}}\right)\left(\sqrt{\frac{\theta_3^*}{\theta_5^*}}\right) = \frac{w_C\sqrt{\theta_1^*}}{\delta_1}(1-b-l)(1+f) \qquad (7)$$

This equation can be transformed to

$$\frac{w_T\sqrt{\theta_5^*}}{\delta_5} = \frac{w_C\sqrt{\theta_1^*}}{\delta_1}(1-b-l)(1+f)\left(\frac{p_1'}{p_2'}\right)\left(\frac{p_2'}{p_3'}\right)\left(\frac{p_3'}{p_5'}\right)\left(\sqrt{\frac{T_3'}{T_1'}}\right)\left(\sqrt{\frac{T_5'}{T_3'}\left[\left(\frac{\gamma}{\gamma+1}\right)_T\left(\frac{\gamma+1}{\gamma}\right)_C\right]}\right) \qquad (8)$$

In equation (8), the compressor equivalent weight flow $w_C\sqrt{\theta_1^*}/\delta_1$, the compressor pressure ratio p_2'/p_1', and the burner pressure drop p_3'/p_2' are constant. The compressor bleed and fuel-air ratio for each engine temperature ratio may be substituted in the equation to determine the variation of the turbine exit equivalent weight flow with engine temperature ratio. The turbine temperature ratio T_5'/T_3' and turbine pressure ratio p_3'/p_5' were determined by using the charts of reference 3 and the required turbine work, and assuming an internal efficiency of 0.94 for all engine temperature ratios. The results of these calculations are shown in figure 6. As the engine temperature ratio decreases from 6.25 to 3.31 (Mach 2.12 in the stratosphere), the turbine exit equivalent weight flow increases. Therefore, for a constant annulus area at the turbine exit, the turbine exit Mach numbers are increasing and the turbine exit flow conditions become more critical. If the engine temperature ratio continues to decrease below 3.31, the turbine will ultimately reach the condition of limiting blade loading. Thus, if the turbine is to be designed for the take-off condition ($T_3' = 2460^\circ$ R), the exit annulus area must be selected so as to avoid limiting blade loading at the supersonic flight condition.

TURBINE DESIGN

Design Velocity Diagrams

Take-off with a turbine inlet temperature of 2460° R was selected as the turbine design condition, and the resulting turbine design was investigated over the range of required flow conditions to determine whether reasonable aerodynamic limits are exceeded. The design take-off condition results in the following specifications for the turbine:

Turbine inlet temperature, T_3', $^{\circ}R$ 2460
Turbine inlet pressure, p_3', lb/sq ft abs 7963
Turbine equivalent speed, $N/\sqrt{\theta_3^*}$, rpm 5347
Turbine equivalent weight flow, $w_T \sqrt{\theta_3^*}/\delta_3$, lb/sec 41.83
Turbine equivalent shaft work, E_T/θ_3^*, Btu/lb 19.05

The first physical dimension selected for this turbine was the annulus area. This area is critical as it controls the flow Mach numbers throughout the turbine, especially at the rotor exit. If the annulus area were too small, the rotor blade exit Mach numbers would be high and the turbine might reach limiting blade loading before attaining the design work output. For this particular turbine, the selection of the annulus is especially critical in that blade limiting loading must be avoided not only at the take-off condition but also at the supersonic flight condition where the rotor exit Mach numbers are highest. For this reason, the rotor blade exit Mach number must be conservative at the take-off condition. However, the blade centrifugal stress is a function of the turbine annulus area and the square of the rotative speed. Thus, the annulus area cannot be so large that the turbine blade stress limit is exceeded. For the design condition, an absolute rotor exit Mach number of 0.48 was specified. Using the turbine exit equivalent weight flow and the specified rotor Mach number results in a calculated annulus area of approximately 2.06 square feet and an untapered centrifugal blade stress of approximately 47,900 pounds per square inch. If a taper factor of 0.7 were specified, the take-off centrifugal stress would be reduced to 33,500 pounds per square inch and the stress at Mach number 2.12 in the stratosphere would be 48,200 pounds per square inch. It is believed that operation at this blade stress would be feasible if the blades were adequately cooled. Consequently, an annulus area of 2.06 square feet was specified at the turbine exit and was assumed to remain constant through the turbine.

In addition, a turbine tip radius of 14.2 inches and a negative exit whirl of 10 percent of the mean rotor speed were specified together with the following assumptions about the flow through the turbine:

1. The absolute total pressure and temperature are uniform over the blade height at the entrance to the stator and the rotor.

2. Free-vortex, simplified radial equilibrium conditions exist at the stator and rotor exits.

3. The expansion in the turbine is adiabatic.

4. The total-pressure ratio across the stator p_4'/p_3' is 0.98.

5. The ratio of the effective annulus area to the actual annulus area at the stator exit is 0.98.

6. The effective annulus area equals the actual annulus area at the rotor exit.

7. The design internal efficiency is 0.94. A high efficiency is assumed because the velocity diagrams at the rotor exit will be used to determine the turbine rotor blade throat area and it is assumed that no losses occur between the rotor throat and the turbine exit.

The detail design velocity diagrams were then calculated and the results are summarized in figure 7.

Off-Design Operation

The estimation of the off-design velocity diagrams is based on the use of continuity and the required turbine work to determine the velocity diagrams at the mean section of the turbine for a specified off-design operating condition. The same assumptions with respect to flow coefficients and internal efficiency as those used to calculate the design velocity diagrams were used. The rotor throat area at the mean section was determined from the design velocity diagrams. The ratio of this throat area to the rotor pitch at the mean section was computed and used to estimate the rotor exit relative flow angle through the use of references 4 and 5. This flow angle was assumed to vary with the rotor exit relative Mach number in the same manner as recommended in reference 5. The velocity diagrams at the rotor exit were estimated through the use of the required specific weight flow, wheel speed, and assumed angle at the mean section. The rotor exit absolute whirl was determined at which the rotor exit relative flow angle calculated from the velocity diagram equals the rotor exit relative flow angle specified by the geometric considerations previously discussed. This rotor exit absolute whirl and the required turbine work are employed in calculating the stator exit absolute whirl at the mean section. The velocity diagrams for the stator exit and rotor inlet at the mean section are then calculated by use of the stator exit absolute whirl, the specific weight flow, and the wheel speed. The resulting velocity diagrams at an engine temperature ratio of 3.31 are summarized in figure 8. This velocity diagram was then used to calculate the stator throat area at the mean section. The stator throat area at the supersonic flight condition is 17 percent less than the stator throat area at the design take-off condition.

The preceding calculations were also carried out at several other engine temperature ratios corresponding to different flight conditions at which this engine might be required to operate, and the more important of the velocity diagram parameters are plotted against the engine temperature ratio in figure 9. As the flight condition changes from take-off (engine temperature ratio of 4.75) to flight at a Mach number of 2.12 in the stratosphere (engine temperature ratio of 3.31), the turbine inlet equivalent weight flow decreases and the turbine equivalent work and equivalent speed increase. This results in the following changes in the turbine flow conditions between take-off and the supersonic flight condition:

1. The stator exit Mach number at the mean section increases from 0.83 to 1.06.

2. The rotor inlet relative Mach number at the rotor hub increases from 0.57 to 0.74.

3. The rotor exit absolute Mach number at the mean section increases from 0.48 to 0.53.

4. The rotor exit relative Mach number at the tip increases from 0.85 to 0.97. This indicates that the rotor exit flow conditions are becoming critical at the supersonic flight condition, especially at the tip radius.

5. The static-pressure ratio across the rotor hub decreases from 1.13 to 1.015 at an engine temperature ratio of 3.68 and then increases to 1.03 at the supersonic flight condition.

6. The leaving loss $(V_{u,5}^2/2gJE_T)_m$ at the mean section decreases from 0.003 to zero.

7. The turning across the rotor hub increases from 96° to 104°.

8. The relative flow angle with respect to the rotor at the mean blade section decreases from 62.8° to 52.8°. This corresponds to a 10° increase in angle of incidence relative to the rotor blades.

The results of the calculation of the estimated velocity diagrams show that reasonable turbine aerodynamic limits are not exceeded for the range of engine temperature ratio over which this engine might operate. However, the turbine flow conditions are more critical at the supersonic flight condition than at the take-off condition.

Blade Profile Design

The final step in the turbine design is the geometric design of the stator and the rotor blade profiles. The velocity diagrams for the take-off condition, which were summarized in figure 7, were used for the profile design. A problem in the aerodynamic design of an air cooled turbine blade, whether a stator or a rotor blade, is the problem of designing the trailing edge section so that adequate cooling can be provided. Adequate cooling of the trailing edge is dependent upon proportioning the trailing edge cooling air passage with the other cooling air passages such that the trailing edge of the blade will be sufficiently cooled. The use of as small trailing edge radii as possible is consistent with the aerodynamic requirements for minimum blade loss. Reference 6 indicates that the blade profile loss is a function of the ratio of blade trailing edge thickness to blade pitch. Thus, the aerodynamic design of these blades is a compromise between the cooling requirements and the aerodynamic requirements for minimum loss at the blade trailing edge.

Before the blade profile can be designed, the axial chord must be determined. The solidity (based on axial chord) depends on choosing a suitable loading for each blade. The blade loading that is specified at the tip section of the stator and the hub section of the rotor is based on the results of reference 7. With the use of 48 blades in the stator and 56 blades in the rotor, the specified blade loading results in an axial chord for the stator of 1.890 inches and for the rotor, 2.270 inches.

Next, the throat area at each section of the stator and rotor blade was calculated. The method used to calculate the throat area at each blade section was the same and will be briefly described. Through the use of the velocity diagram, a specific weight flow based on the actual flow area within the blade passage immediately upstream of the trailing edge is calculated. The blade trailing edge blockage is accounted for in this manner. Constant tangential velocity is assumed between the point downstream of the blade at which the velocity diagram is calculated and the point within the blade passage. The flow Mach number within the blade passage is then calculated and, assuming the same Mach number at the throat, continuity is used to calculate the required throat area. If the calculated flow Mach number within the blade passage is greater than unity, the flow Mach number at the blade throat is assumed to be unity and the throat area is then calculated.

At this point in the design procedure, the actual blade profile is determined. The same procedure is used for the stator and rotor blades. A minimum blade trailing edge diameter is specified by the cooling requirements. For this turbine the ratio of the trailing edge diameter to the blade pitch at the mean section was 0.04 for the stator and 0.06 for the rotor. A straight line tangent to the trailing edge circle is drawn at an angle such that the correct throat area exists between adjacent blades. This straight line is the suction surface of the blade in the unguided portion of the flow passage. Next, the remaining part of the suction surface and the entire pressure surface are drawn such that the cooling requirements are satisfied. The velocity distribution of the flow on these surfaces is then determined according to the method presented in reference 8 to find whether any adverse pressure gradients exist which might cause flow separation. If such is the case, the blade profiles are altered to avoid this trouble. The resulting blade profiles are then stacked radially such that the coolant passage can be incorporated within the blade.

The resulting blade profiles and flow passage shapes of this turbine are shown in figure 10 for the hub, mean, and tip sections of the stator and rotor blades.

EQUIPMENT AND INSTRUMENTATION

The turbine used in this investigation was a 0.528 scale model which could be accommodated in an existing turbine test facility. The arrangement of the experimental equipment is shown diagrammatically in figure 11. Ambient air was drawn from the test cell through an electrostatic precipitator to remove most of the dust particles. The air was then heated in passage through a steam heater. This was necessary to avoid water condensation when the air was expanded through the turbine. Constant turbine inlet temperature was maintained by an automatic control which regulates the amount of air bypassing the steam heater. After passing through the turbine, the air was exhausted by the laboratory low pressure exhaust system. An automatically controlled butterfly valve downstream of the surge tank in the low pressure exhaust line maintained the desired pressure ratio across the turbine.

The stator blade ring assembly of the model turbine is shown in figure 12. Figure 13 shows the rotor of the scale model turbine mounted in the rotor-blade wheel assembly. The base of the rotor blades is cylindrical and the blades are mounted in a ring as shown in figure 14.

The power output was absorbed by a water brake that was cradle-mounted for torque measurements which were made with a commercial springless scale. The turbine speed was indicated by a calibrated electric tachometer. Air flow was measured by means of a standard flat plate orifice located between the precipitator and the steam heater and installed in conformance with A.S.M.E. specifications.

A cross-sectional view of the turbine showing the location of the instrumentation is presented in figure 15. The inlet total pressures and temperatures were measured by eight stagnation-type probes (four for total pressure and four for total temperature) located at equal circumferential stations 0.52 inch upstream of the stator blade leading edge. These probes were so located radially as to be at the area centers of equal annular areas. The stator exit static pressures were measured with 12 static wall taps located 0.25 inch downstream of the stator blade trailing edge, six on the outer wall and six on the inner wall. The rotor exit static pressures were measured with 14 static wall taps located 0.41 inch downstream of the rotor blade trailing edge, six on the outer and eight on the inner wall. The rotor exit total temperature was measured with four stagnation-type temperature probes located at the downstream end of the exhaust housing. The probes were so located radially as to be at the area centers of equal annular areas. Although radial variations of the temperature may have prevented the thermocouples from indicating an accurate mass-averaged temperature, the effect on turbine performance is small because this temperature was used only to compute the available rotor exit total pressure. An error of 5° in temperature would change the computed efficiency less than 0.3 percent. This method of obtaining rotor exit total pressure is described in the section PROCEDURE AND PERFORMANCE CALCULATIONS.

The instruments used in determining the turbine performance were read with the following precision:

Absolute pressure, in. tetrabromoethane	±0.5
Orifice pressure drop, in. water	±0.5
Temperature, deg	±1
Torque load, lb	±0.2
Rotational speed, rpm	±10

For a total-pressure ratio of 2.00 or greater, the probable error in reproducing the turbine efficiency was 0.5 percent.

PROCEDURE AND PERFORMANCE CALCULATIONS

Data were taken at nominal values of total pressure ratios from 1.3 to 2.4. At each of the pressure ratios, the turbine was operated at constant equivalent tip speeds ranging from 300 to 800 feet per second, which correspond to turbine tip speeds of 645 to 1721 feet per second for a turbine inlet temperature of 2460° R. This is a range of 45 to 121 percent of design speed for the turbine. For all tests points, the inlet total temperature was maintained between 683° and 687° R. The inlet total pressure varied between 24 and 27 inches of mercury absolute, depending on the air flow and the ambient air pressure.

The brake internal efficiency, which is based on the expansion between the inlet and exit total pressure, was used to express the performance of the turbine. This efficiency is defined as

$$\eta = E_T/(h_3' - h_5')_s$$

where E_T is the turbine shaft work. This shaft work includes the losses of the antifriction bearings which were used in the test setup. This manner of measuring the work output is indicative of the power available to the compressor of a jet engine. The ideal drop in enthalpy $(h_3' - h_5')_s$ was computed by using the inlet total pressure and temperature and the available exit total pressure. The available exit total pressure was computed on the assumption that one-dimensional flow exists and occupies the full annulus area and that the tangential velocity is zero at this station by using the following formula in which all quantities are known except $p_{x,5}'$:

$$\frac{w\sqrt{T_5'}}{p_5 A_5} = \frac{p_{x,5}'}{p_5}\sqrt{\left[\left(\frac{2\gamma}{\gamma-1}\right)\left(\frac{g}{R}\right)\right]\left[\left(\frac{p_5}{p_{x,5}'}\right)^{\frac{2}{\gamma}} - \left(\frac{p_5}{p_{x,5}'}\right)^{\frac{\gamma+1}{\gamma}}\right]} \qquad (9)$$

Because of the assumption of zero exit tangential velocity, this method of computing the available exit total pressure gives a conservative value of turbine efficiency. The weight flow of air was determined from the orifice measurements and the data of reference 9.

All the turbine performance data were reduced to NACA standard sea-level conditions at the turbine stator entrance. The performance was expressed in terms of the following variables: brake internal efficiency η, total-pressure ratio p_3'/p_5', equivalent tip speed $U_t/\sqrt{\theta_3^*}$, equivalent shaft work E_T/θ_3^*, and product of equivalent weight flow and equivalent tip speed $(wU_t/\delta_3)\,\varepsilon$.

However, the turbine was designed for an inlet temperature of 2460° R and was tested as a scale model at an inlet temperature of 685° R. Thus, it is necessary to correct some of the design parameters presented in the TURBINE DESIGN for the change in temperature and turbine size. According to an analysis presented in reference 10, the design parameters are as follows for the scale model turbine:

Equivalent tip speed, $U_t/\sqrt{\theta_3^*}$, ft/sec	662.6
Equivalent weight flow, $(w\sqrt{\theta_3^*}/\delta_3)\,\varepsilon$, lb/sec	12.18
Equivalent shaft work, E_T/θ_3^*, Btu/lb	19.05
Turbine tip diameter, in.	15.00

EXPERIMENTAL RESULTS AND DISCUSSION

The performance of the scale model of this turbine with the stator set for the design take-off condition is presented in figures 16 and 17. Figure 16 is a plot of the equivalent torque and equivalent weight flow against the total-pressure ratio for constant values of equivalent tip speed. The data in figure 16 were used to obtain the data for figure 17, which gives the over-all performance of this turbine. This figure is in the form of a composite plot where parameters of brake internal efficiency, equivalent tip speed, and total-pressure ratio are plotted against the equivalent shaft work as the ordinate and the product of the equivalent weight flow and equivalent tip speed as the abscissa.

In figure 16(a) the equivalent torque is plotted against the total-pressure ratio for constant values of equivalent tip speed. For each equivalent tip speed, the equivalent torque increases as the total-pressure ratio increases, but the slope of the curve is decreasing. At the high equivalent tip speeds, it can be noted that the equivalent torque no longer increases, but remains constant after a total-pressure ratio of 2.46. At this pressure ratio, the turbine has reached limiting loading, and the equivalent work remains constant for each speed when the pressure ratio is increased. It is these equivalent works which are labeled "limiting loading" on figure 17. Figure 16(b) is a plot of the equivalent weight flow against the total-pressure ratio for constant values of equivalent tip speed. At all total-pressure ratios above 1.9, the value of the equivalent weight flow does not change as the total-pressure ratio increases for each value of equivalent tip speed. This indicates that choking is occurring in the turbine. The value of choking equivalent weight flow is constant for equivalent tip speeds from 300 to 450 feet per second, which indicates that choking in the turbine stator is limiting the equivalent weight flow in this speed range. For equivalent tip speeds from 500 to 800 feet per second, the value of choking weight flow decreases as the speed increases. This indicates that choking in the turbine rotor is now limiting the equivalent weight flow in this speed range. Figure 18 shows the effect of the turbine total-pressure ratio on the total-to-static pressure ratio across the stator at the design value of the equivalent tip speed, 662.6 feet per second. Above a total-pressure ratio of 2.0, the total-to-static pressure ratio across the turbine stator is constant at both the hub and tip radii. This is another indication that choking occurs in the turbine rotor and limits the equivalent weight flow at the design equivalent tip speed for total-pressure ratios above 2.0. The value of the equivalent weight flow at the design values of equivalent shaft work and equivalent tip speed, 19.05 Btu per pound and 662.6 feet per second, respectively, is 11.96 pounds per second. This is 1.8 percent less than design equivalent flow.

The over-all performance of the scale model turbine is shown in figure 17. A maximum brake internal efficiency of 0.89 was obtained in the performance of this turbine with the stator at the design take-off position. At the design equivalent shaft work, 19.05 Btu per pound, and design equivalent speed, 662.6 feet per second, a brake internal efficiency of 0.87 was obtained. It should be noted that because of the specified mode of operation for the turbojet engine, the only equilibrium operating point of the engine on this turbine map is the design point. At other operating conditions the stator chord angle is changed and other performance maps are necessary to determine the off-design operation of the turbine.

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SUMMARY OF RESULTS

From an analytical investigation of the operational requirements of a particular turbojet engine suitable for supersonic flight which utilizes turbine stator adjustment to maintain a fixed compressor operating point, it was found that a single-stage air-cooled turbine could be designed without exceeding reasonable aerodynamic limits. The range of engine operating conditions considered was for sea-level take-off to a flight Mach number of 2.12 in the stratosphere and for a range of engine temperature ratio from approximately 6.25 to 3.31. This range of engine temperature ratio required a variation in turbine inlet equivalent flow from 110 to 85 percent of the design take-off value and a corresponding variation in equivalent turbine work from 80 to 141 percent. The results of the analysis indicate that the internal flow conditions within the turbine are most critical and the centrifugal blade stress is highest at the high flight Mach number.

An adjustable stator turbine was designed with these operational requirements in view, and the experimental performance of a scale model with the stator set at the design take-off value was obtained in a cold-air test facility. At the design equivalent work and tip speed, 19.05 Btu per pound and 662.6 feet per second, respectively, a brake internal efficiency of 0.87 was obtained. The maximum brake internal efficiency was 0.89 and the over-all performance was good for a wide range of equivalent shaft work and tip speeds.

National Advisory Committee for Aeronautics
Lewis Flight Propulsion Laboratory
Cleveland, Ohio, March 17, 1954

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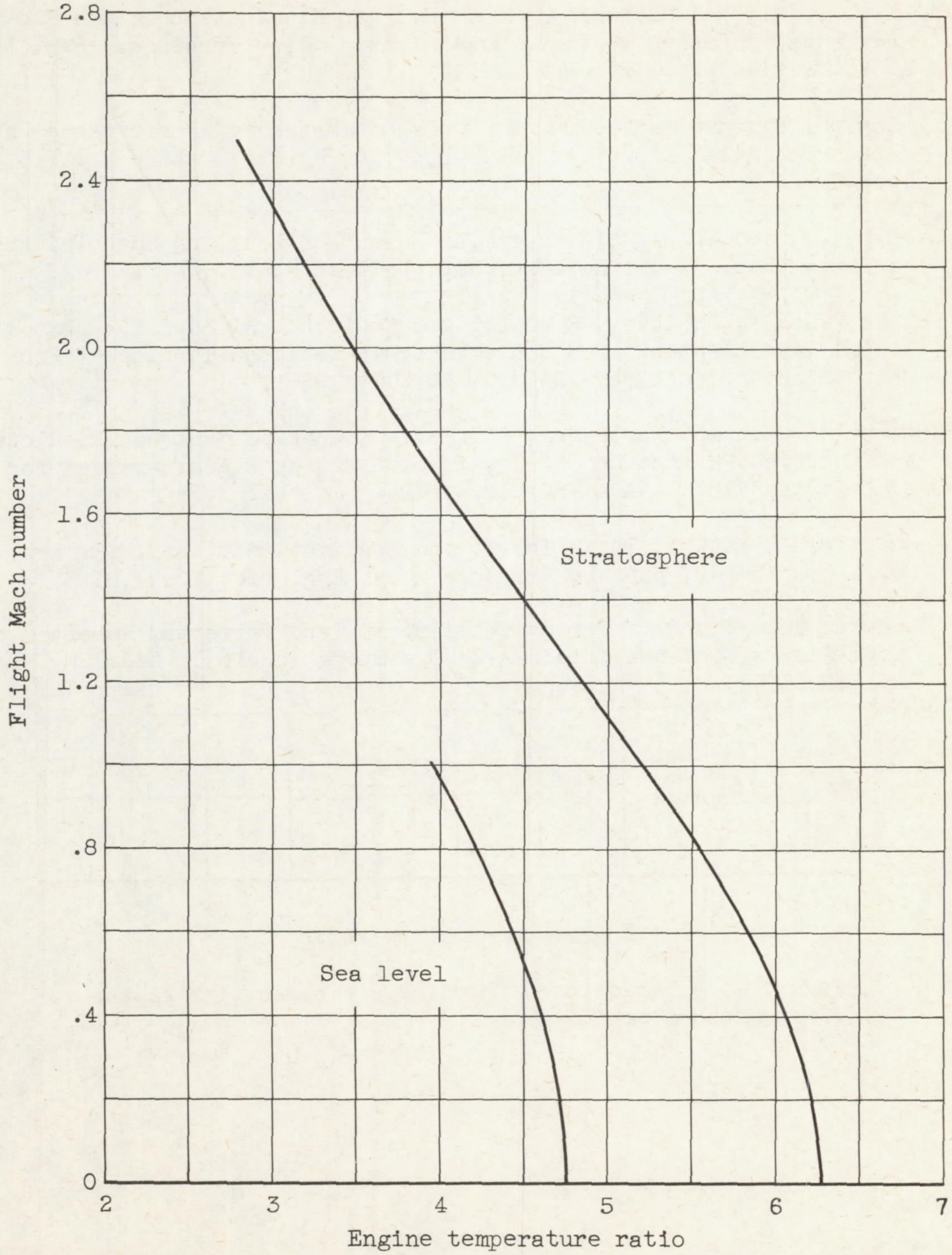


Figure 1. - Variation of flight Mach number with engine temperature ratio. Turbine inlet temperature, 2460° R.

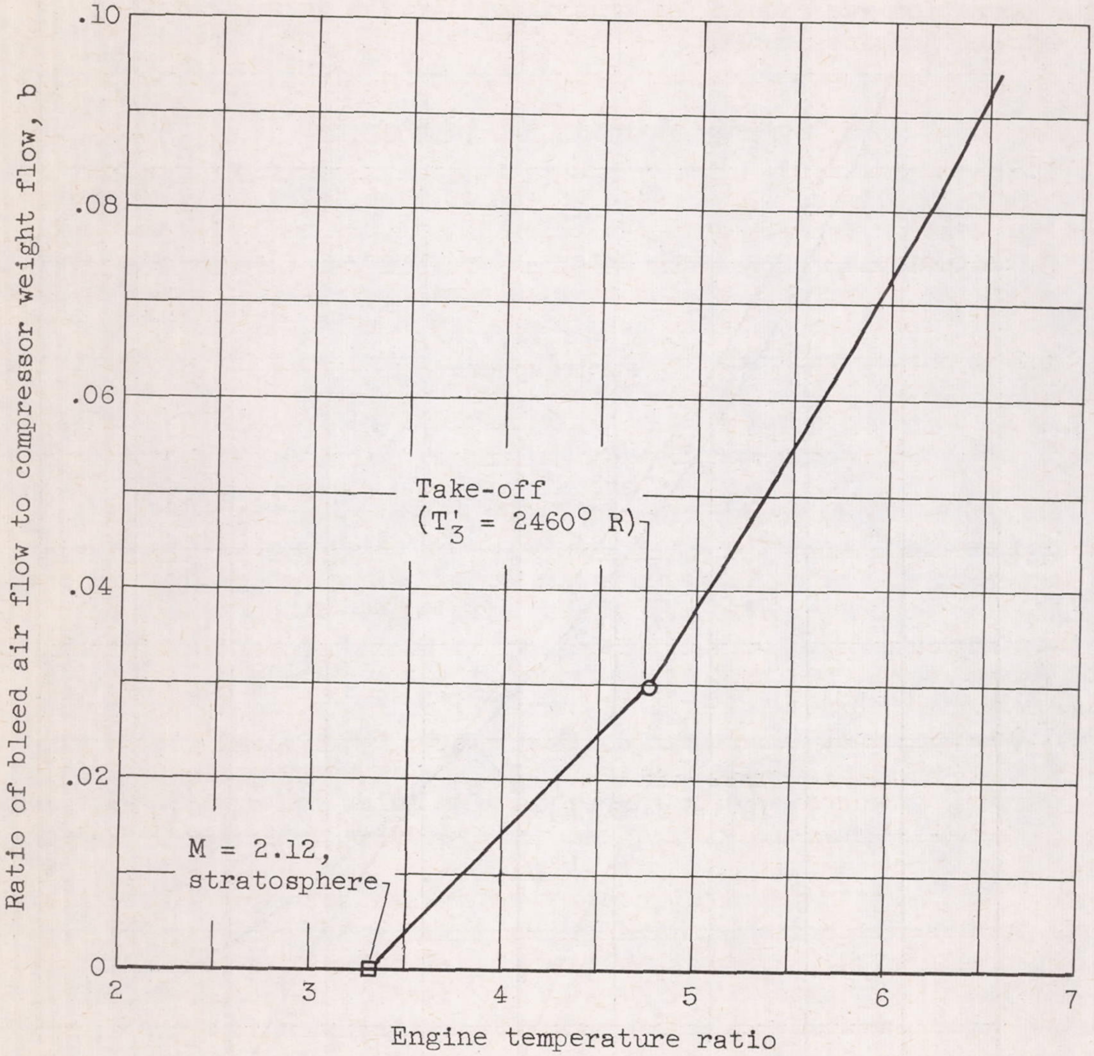


Figure 2. - Variation of cooling air bleed with engine temperature ratio.

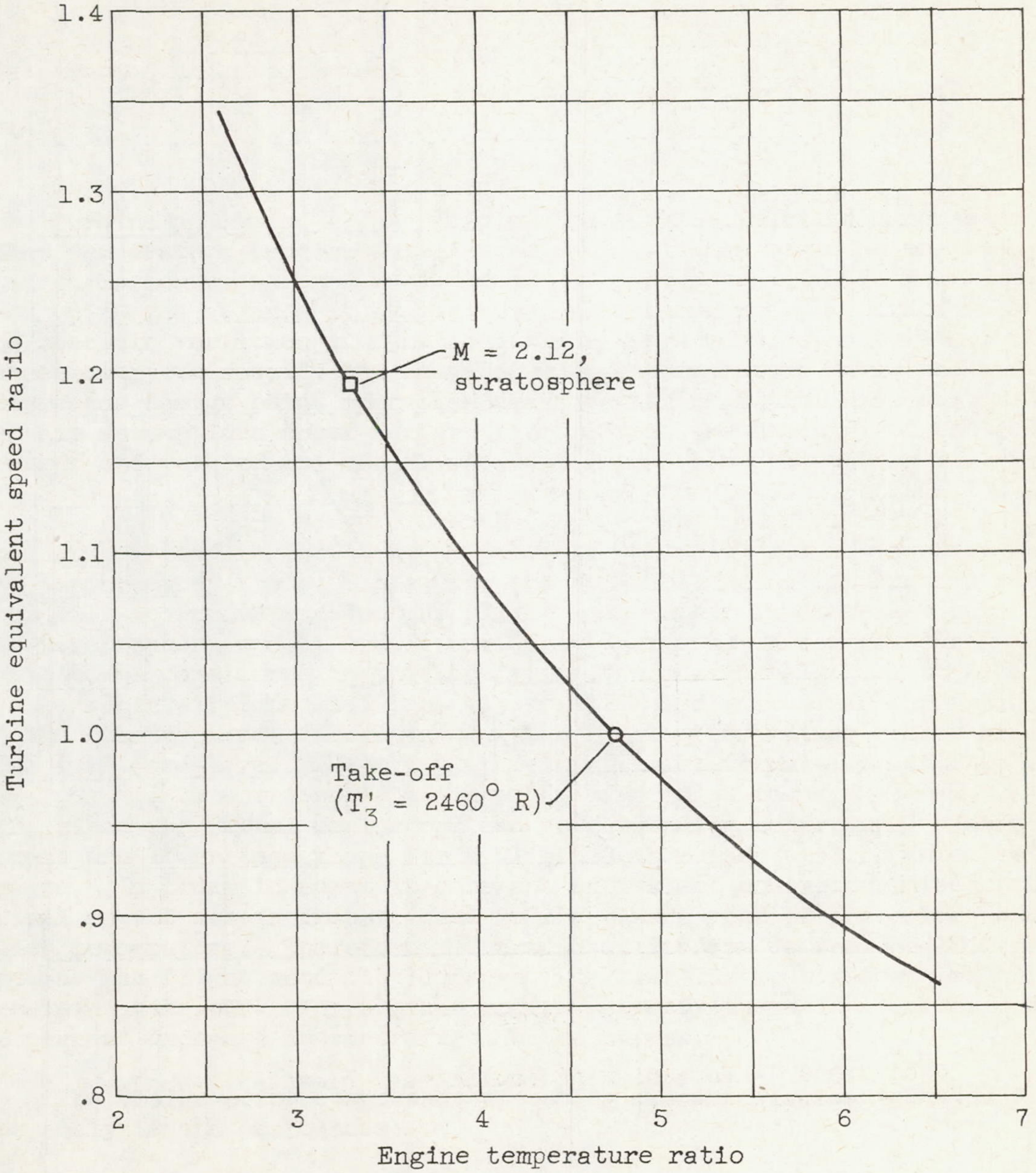


Figure 3. - Variation of turbine equivalent speed ratio with engine temperature ratio.

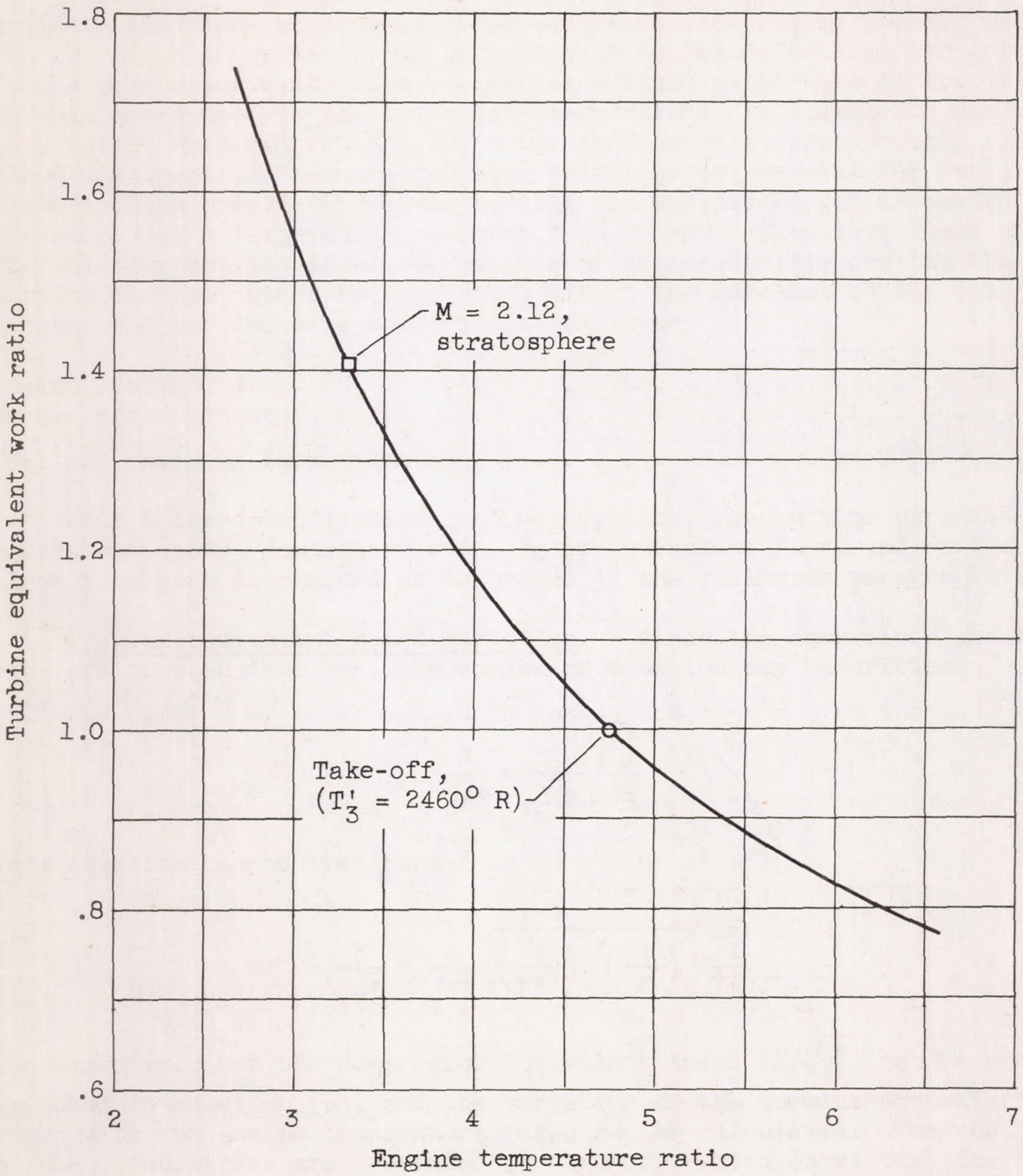


Figure 4. - Variation of required turbine work ratio with engine temperature ratio.

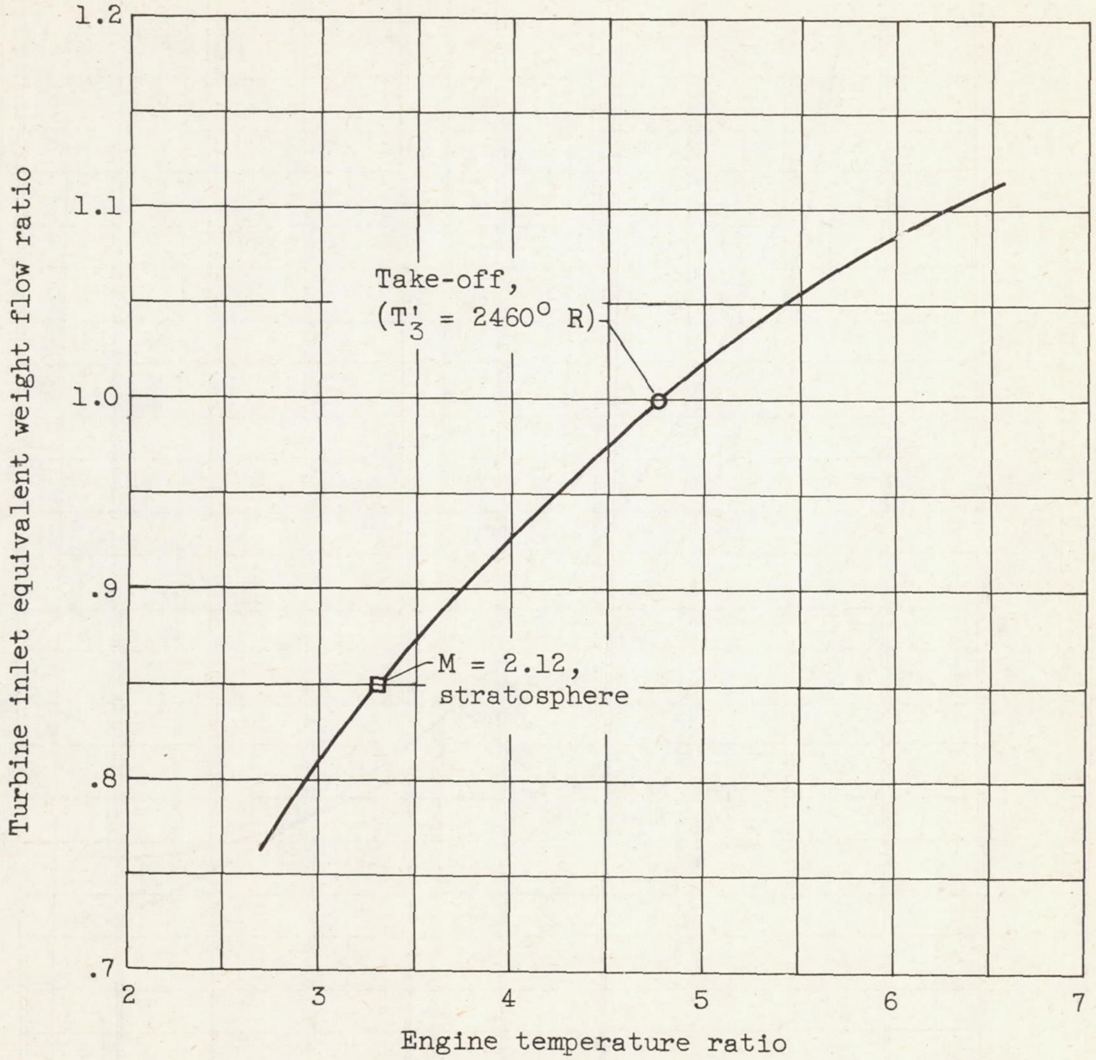


Figure 5. - Variation of turbine inlet equivalent weight flow ratio with engine temperature ratio.

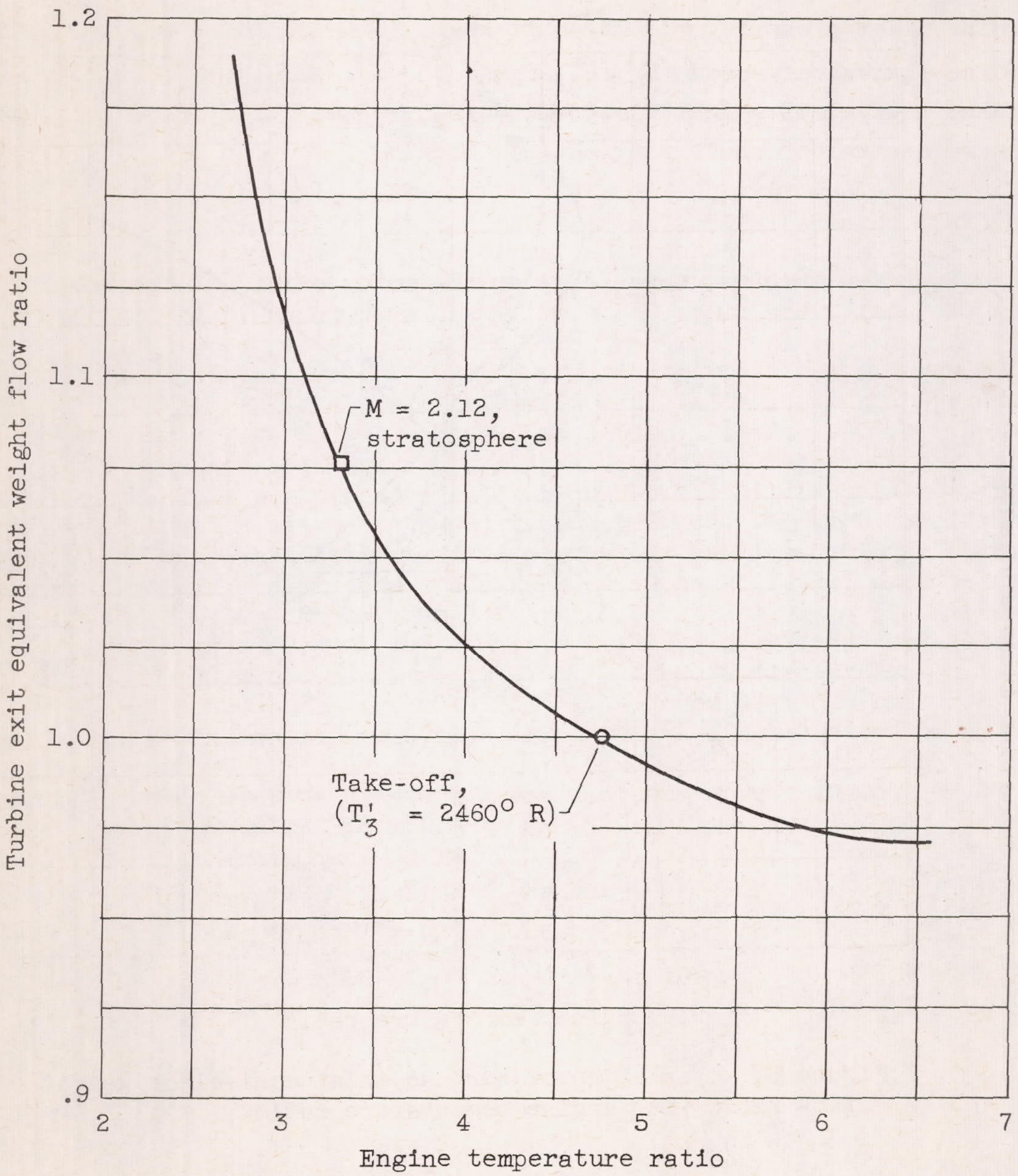
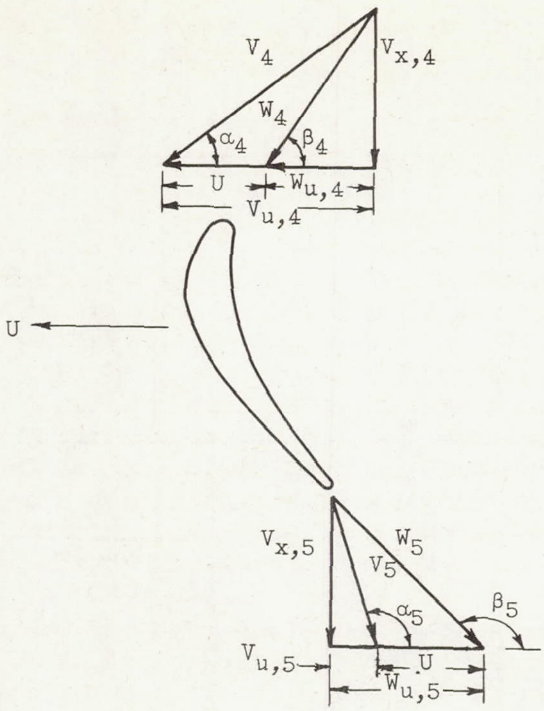
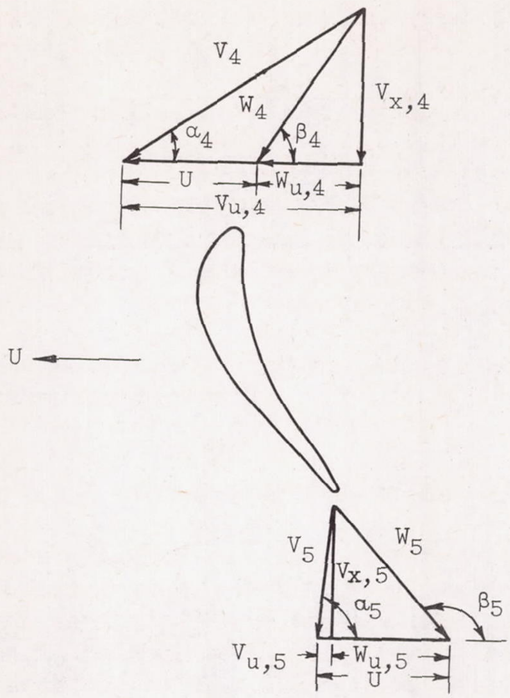


Figure 6. - Variation of turbine exit equivalent weight flow ratio with engine temperature ratio.



Parameter	Hub	Mean	Tip
$M_{x,4}$	0.385	0.378	0.374
$M_{V,4}$	.976	.834	.737
$M_{W,4}$	.574	.425	.374
U_4/a_4	.471	.549	.628
$W_{u,4}/a_4$	.426	.194	.008
$V_{u,4}/a_4$	.897	.743	.635
α_4	23.24°	26.98°	30.50°
β_4	42.12°	62.83°	88.89°
$M_{x,5}$	.482	.482	.482
$M_{V,5}$	.486	.485	.484
$M_{W,5}$	.727	.788	.853
U_5/a_5	.478	.567	.655
$W_{u,5}/a_5$	-.545	-.623	-.704
$V_{u,5}/a_5$	-.067	-.057	-.049
α_5	97.93°	96.70°	95.79°
β_5	138.52°	142.29°	145.63°
$\beta_5-\beta_4$	96.40°	79.46°	56.74°

Figure 7. - Design velocity diagram for turbine. Inlet temperature, 2460° R; engine temperature ratio, 4.75.



Parameter	Hub	Mean	Tip
$M_{x,4}$	0.413	0.399	0.391
$M_{V,4}$	1.277	1.062	920
$M_{W,4}$	.740	.501	.396
U_4/a_4	.595	.681	.772
$W_{u,4}/a_4$	.613	.303	.061
$V_{u,4}/a_4$	1.208	.984	.833
α_4	18.89°	22.09°	25.15°
β_4	33.97°	52.82°	81.07°
$M_{x,5}$	.534	.534	.534
$M_{V,5}$	.534	.534	.534
$M_{W,5}$	.791	.878	.970
U_5/a_5	.598	.709	.821
$W_{u,5}/a_5$	-.584	-.698	-.810
$V_{u,5}/a_5$	.014	.012	.010
α_5	88.52°	88.75°	88.92°
β_5	137.57°	142.57°	146.62°
$\beta_5-\beta_4$	103.60°	89.75°	65.55°

Figure 8. - Estimated velocity diagrams for off-design operation of turbine. Mach number 2.12 in stratosphere; engine temperature ratio, 3.31.

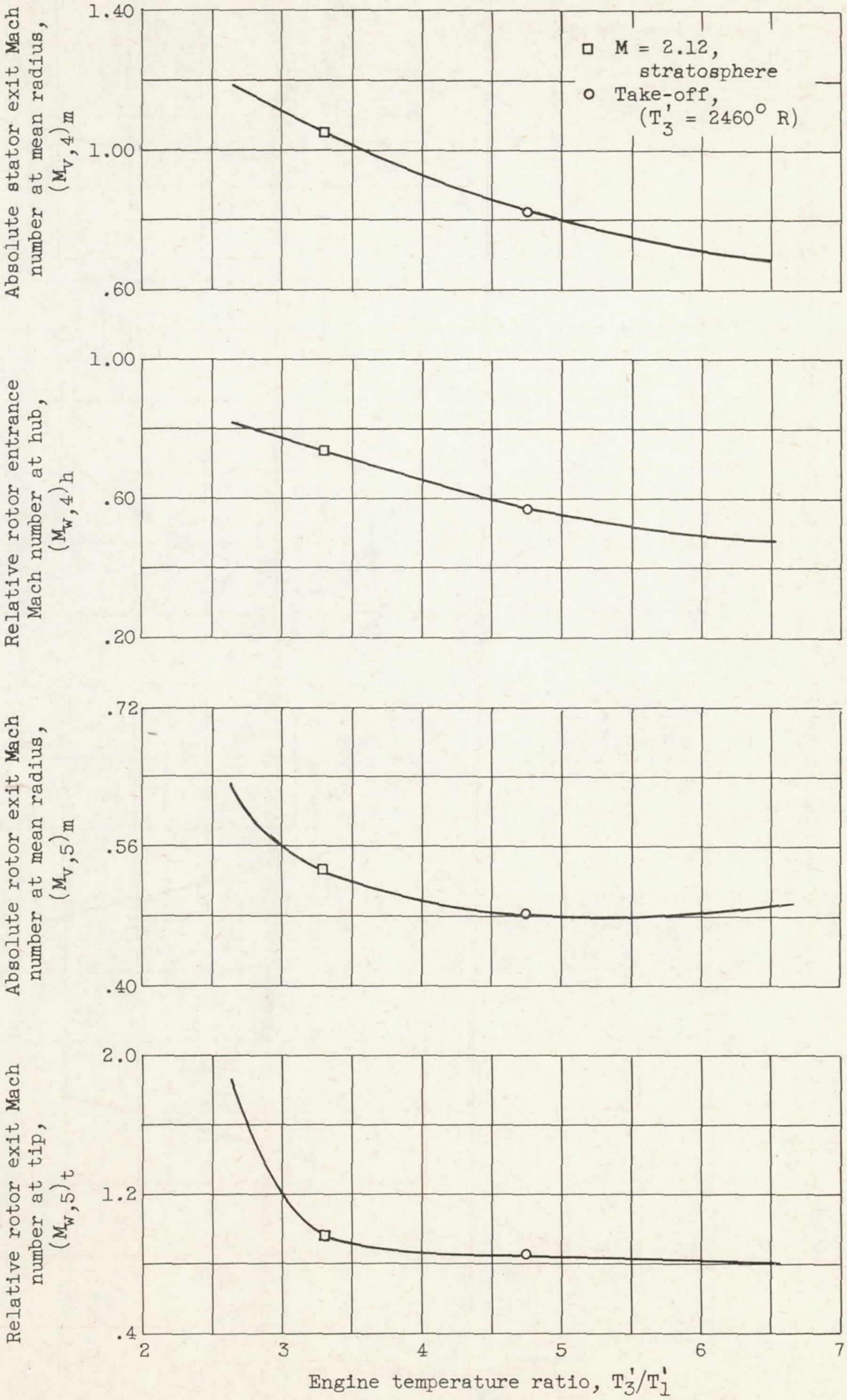


Figure 9. - Variation of turbine velocity diagram parameters with engine temperature ratio.

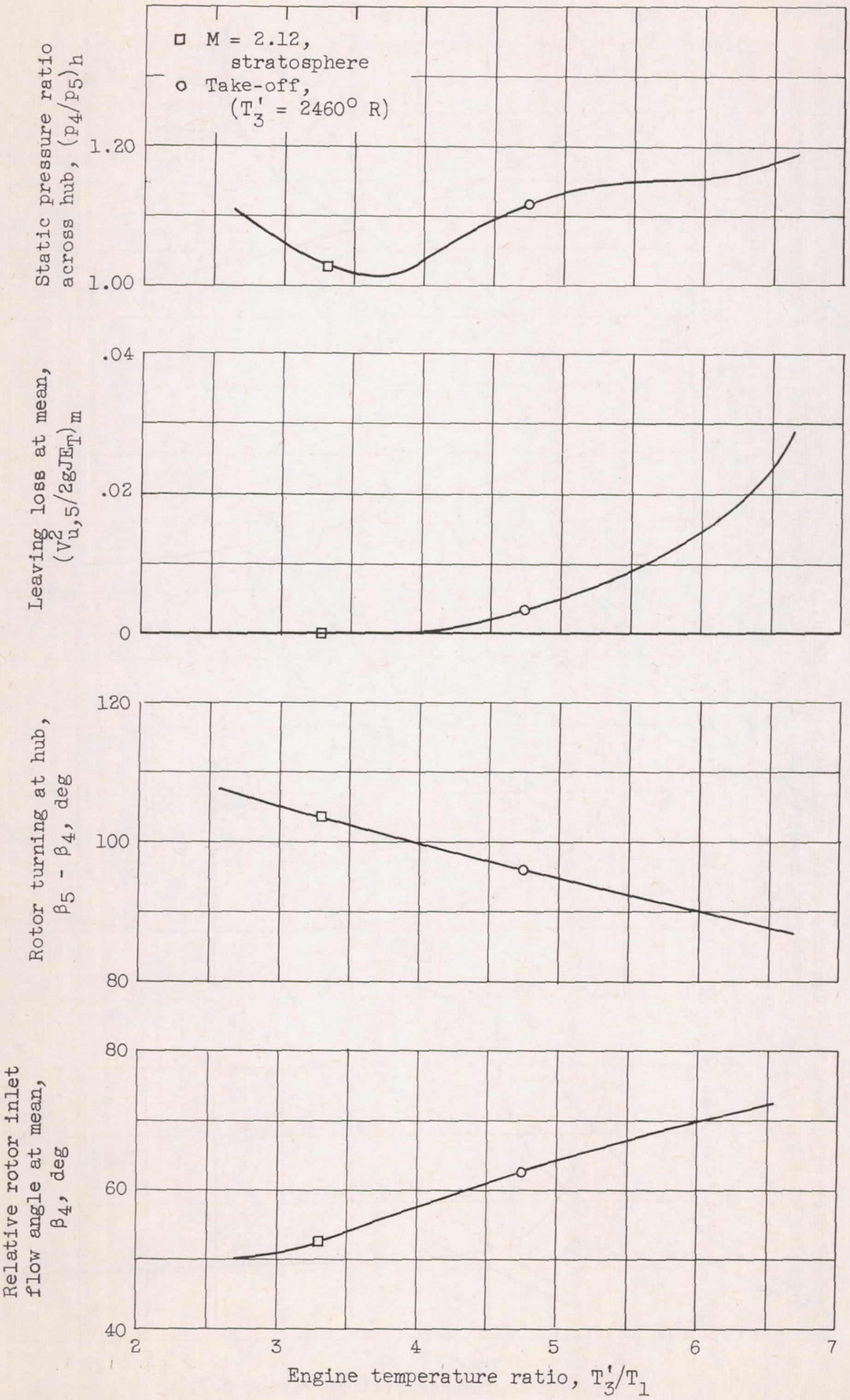
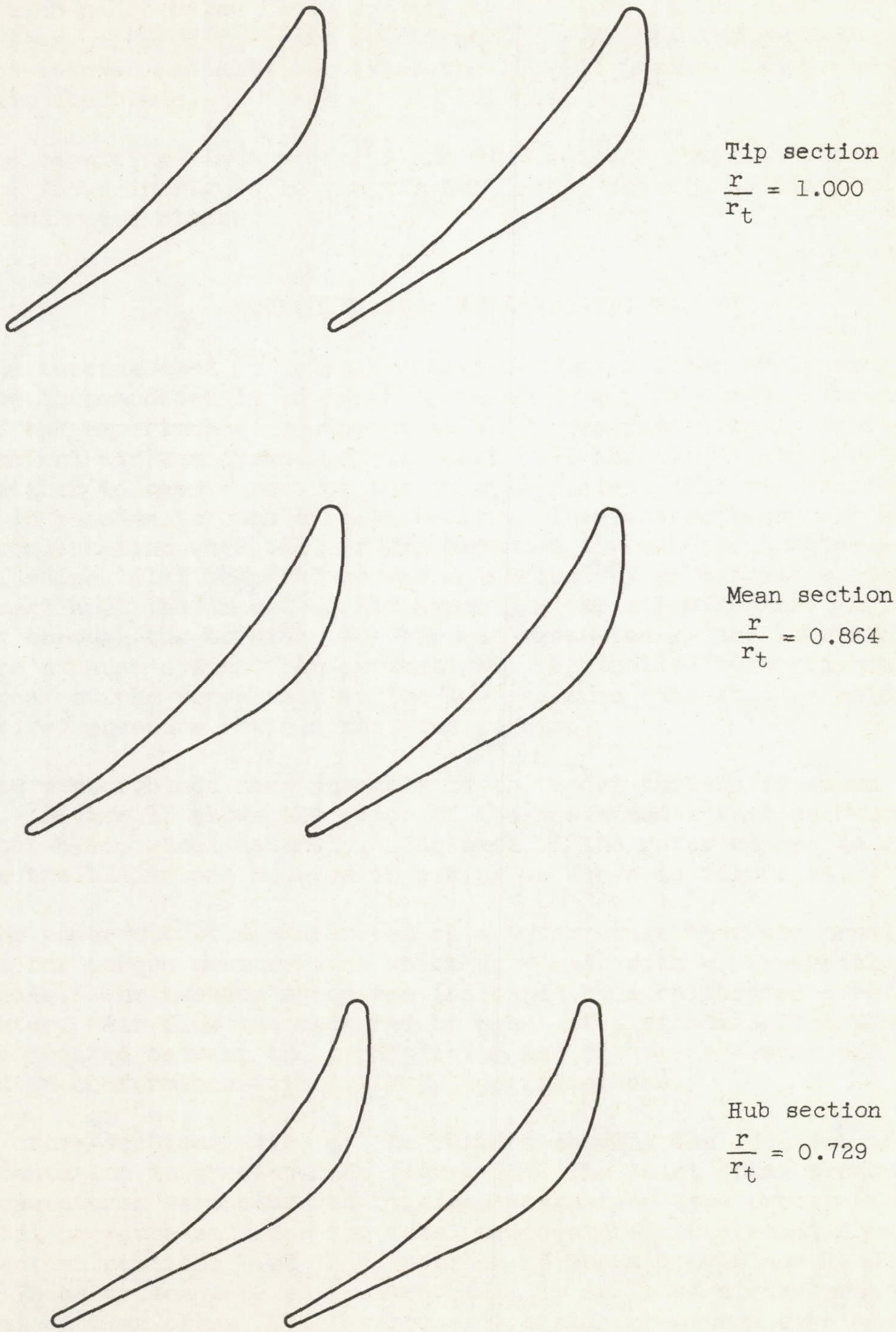


Figure 9. - Concluded. Variation of turbine velocity diagram parameters with engine temperature ratio.



(a) Stator.

Figure 10. - Turbine blade and passage shapes.

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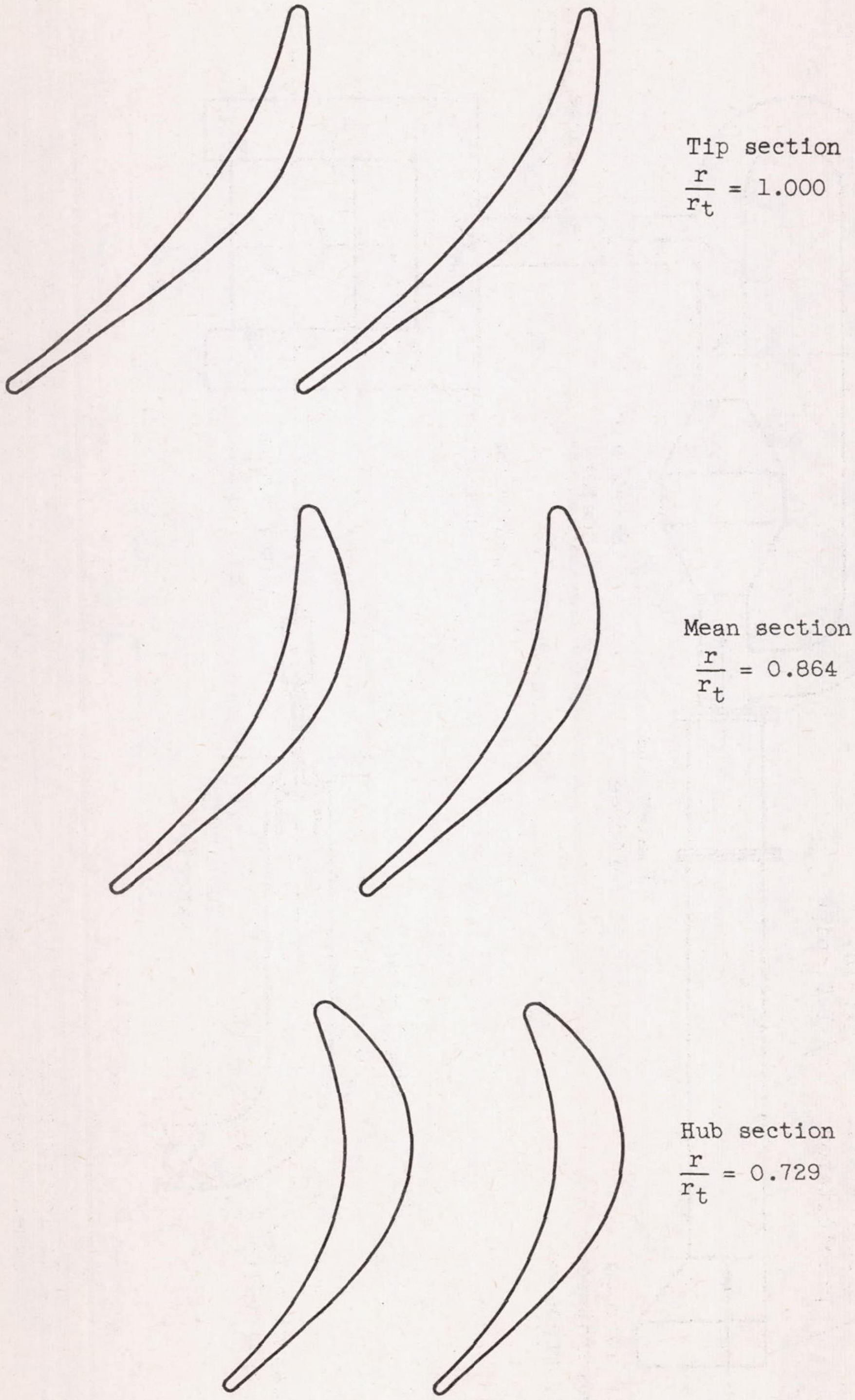

(b) Rotor.

Figure 10. - Concluded. Turbine blade and passage shapes.

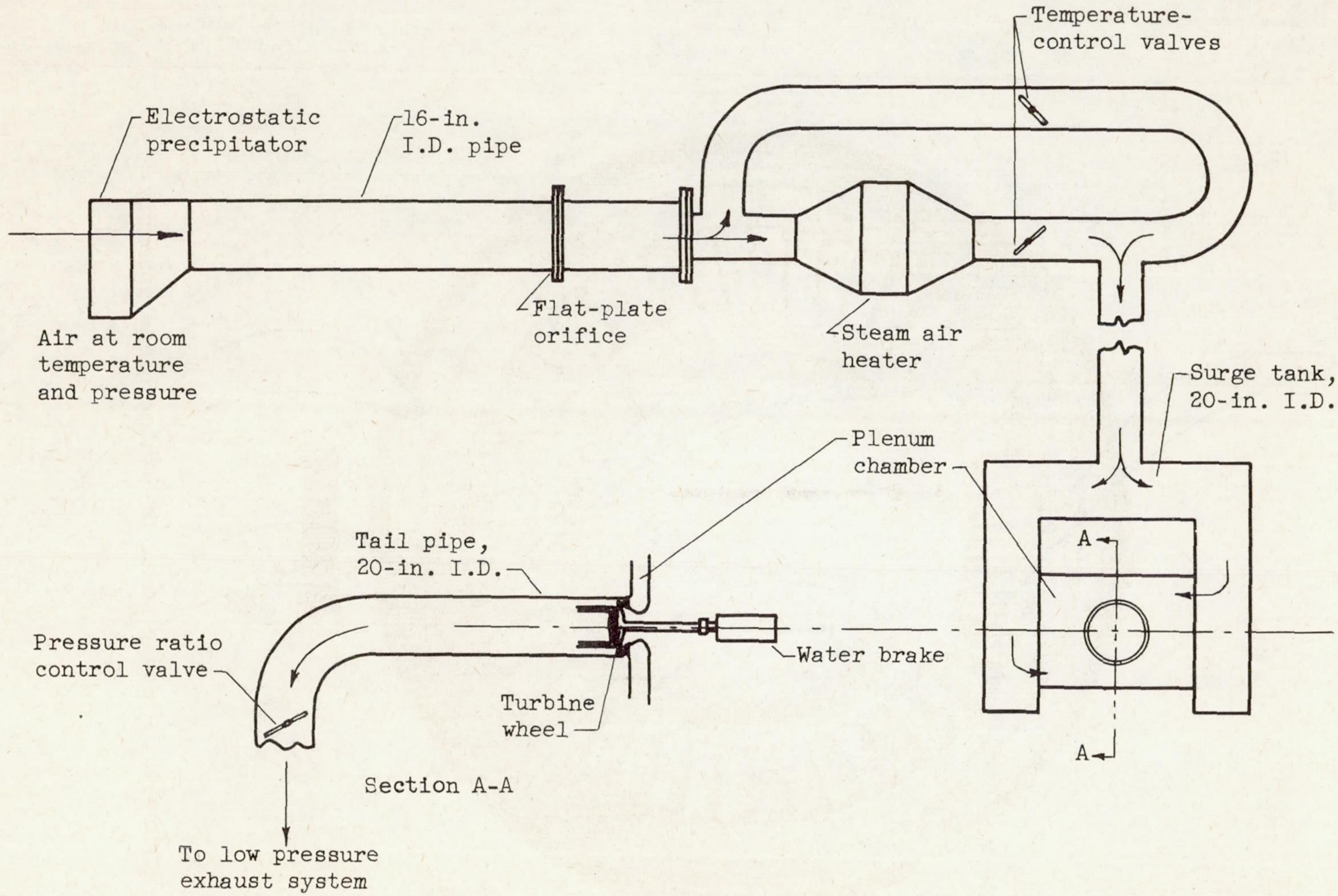

Figure 11. - Diagrammatic sketch of arrangement of experimental equipment showing pressure ratio control valve, air-flow path, and auxiliary equipment.

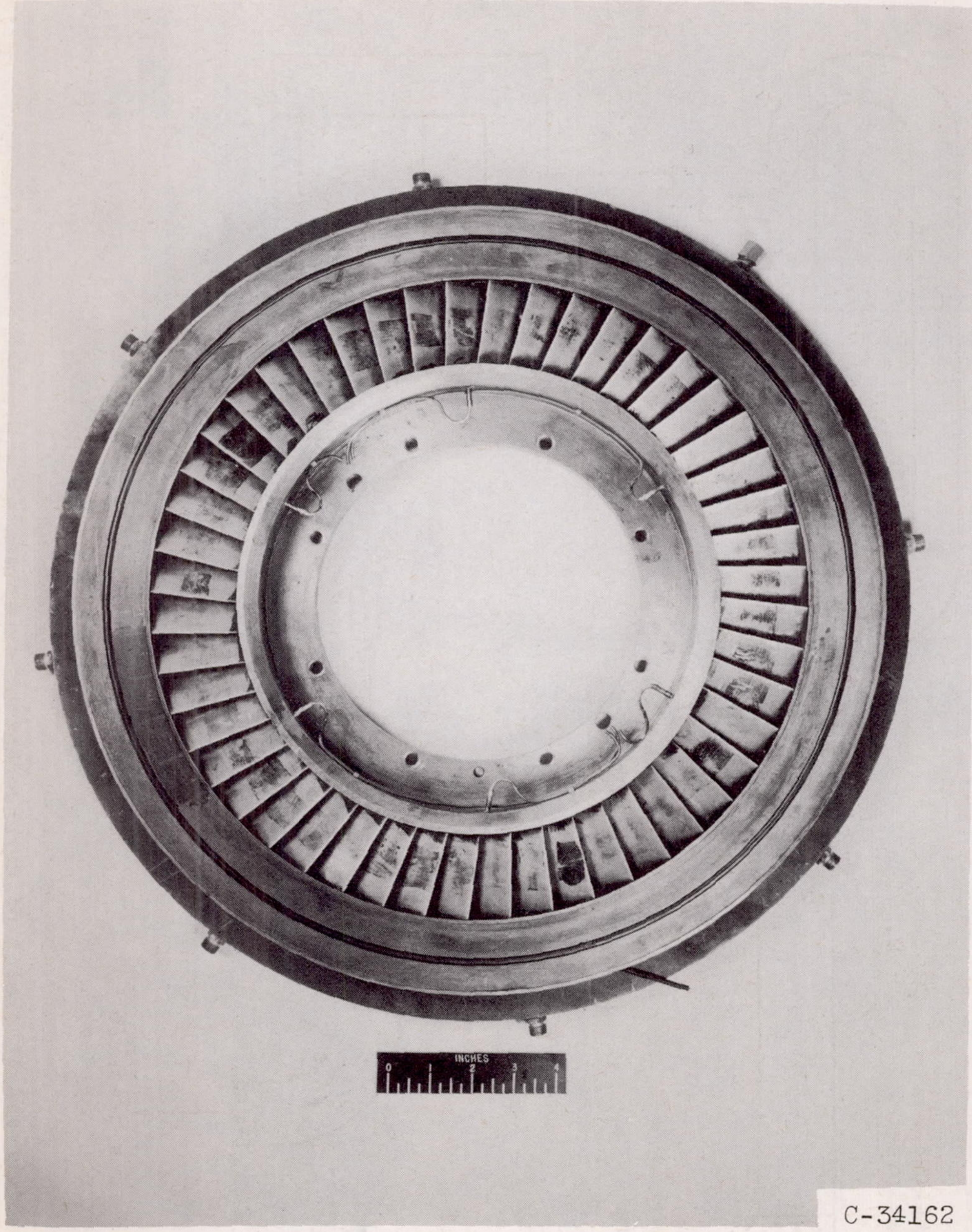

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Figure 12. - Scale model of stator blade assembly.

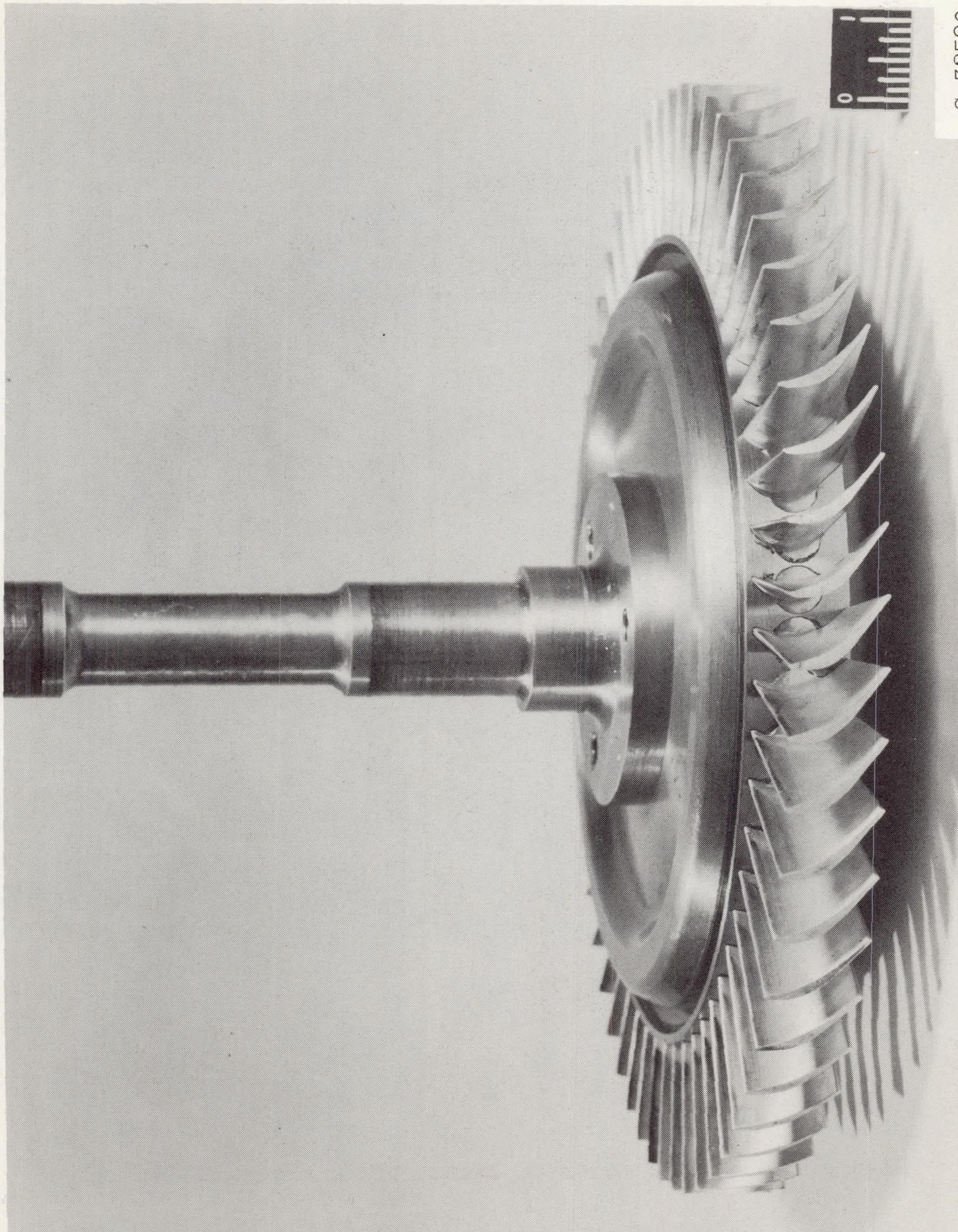


Figure 13. - Scale model of rotor blade wheel assembly.

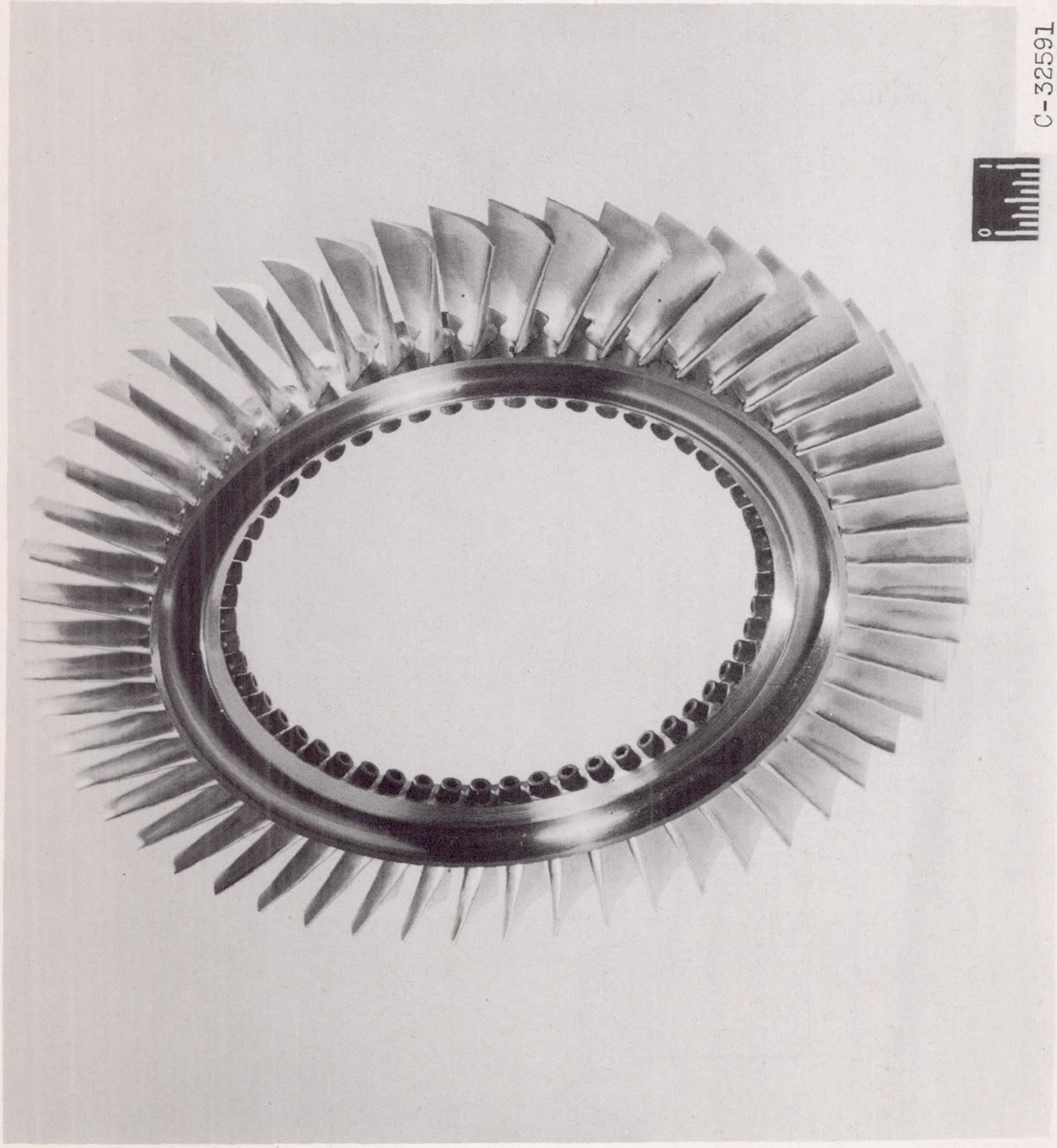


Figure 14. - Scale model of rotor blade ring assembly.

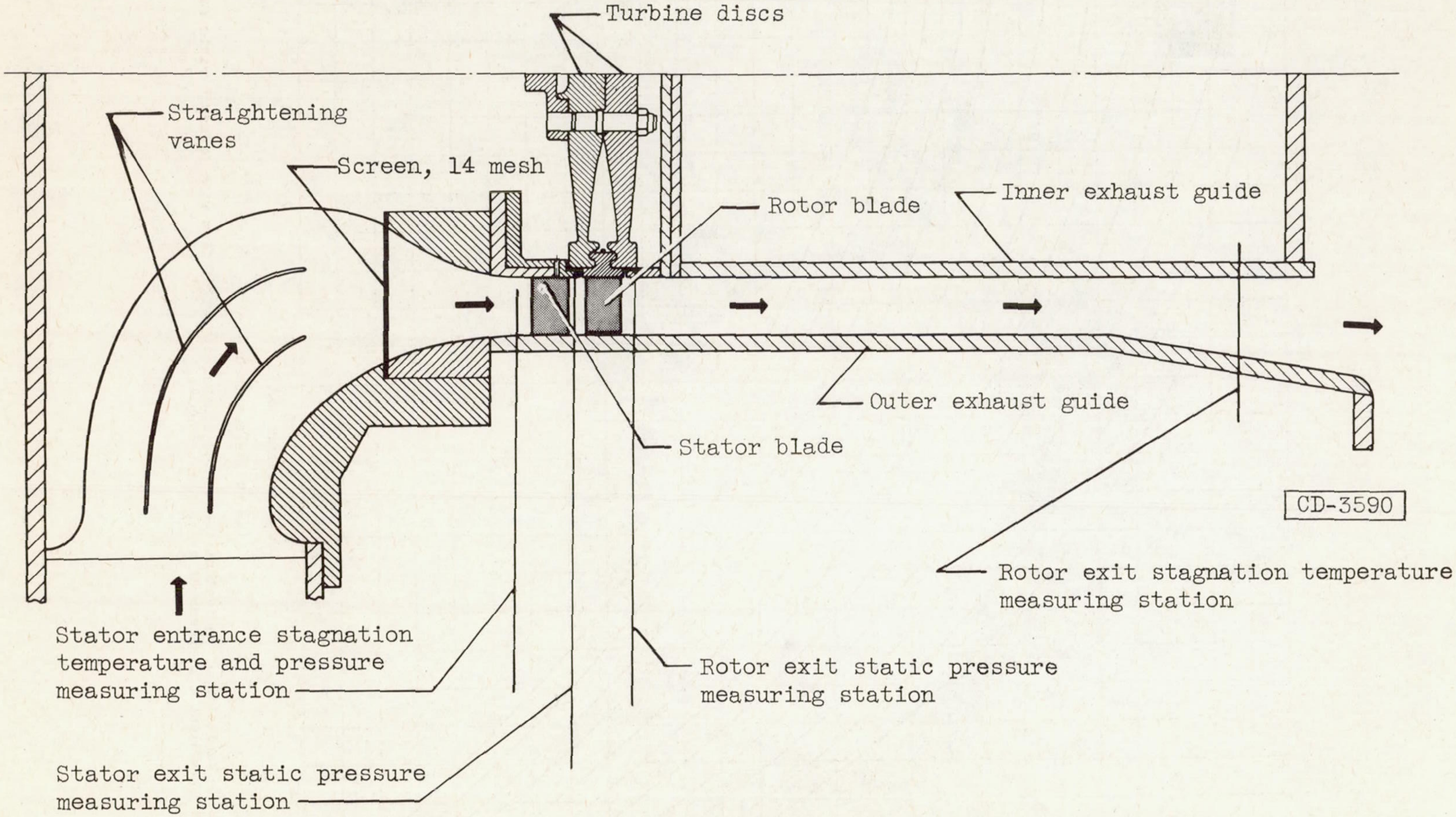
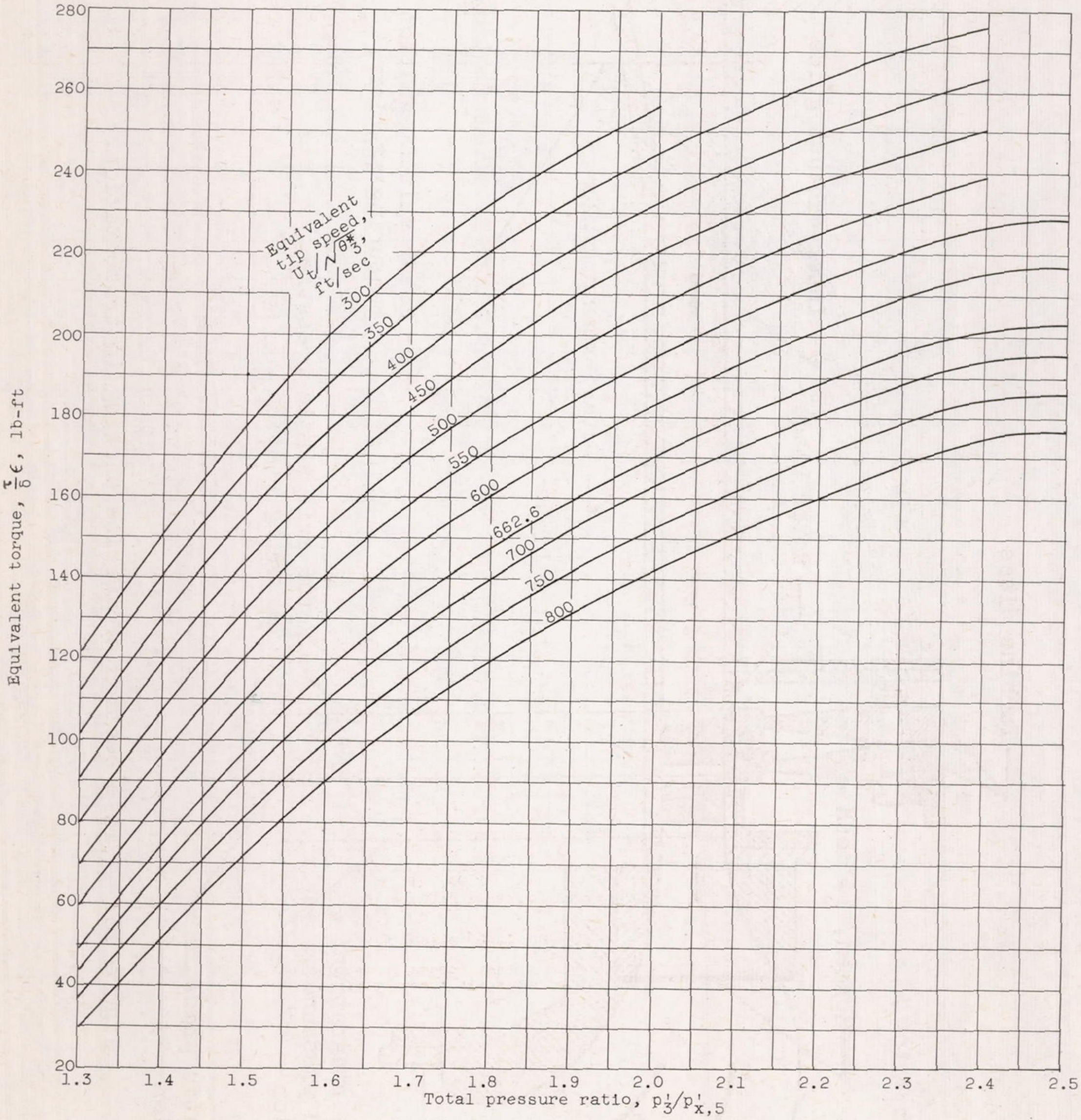
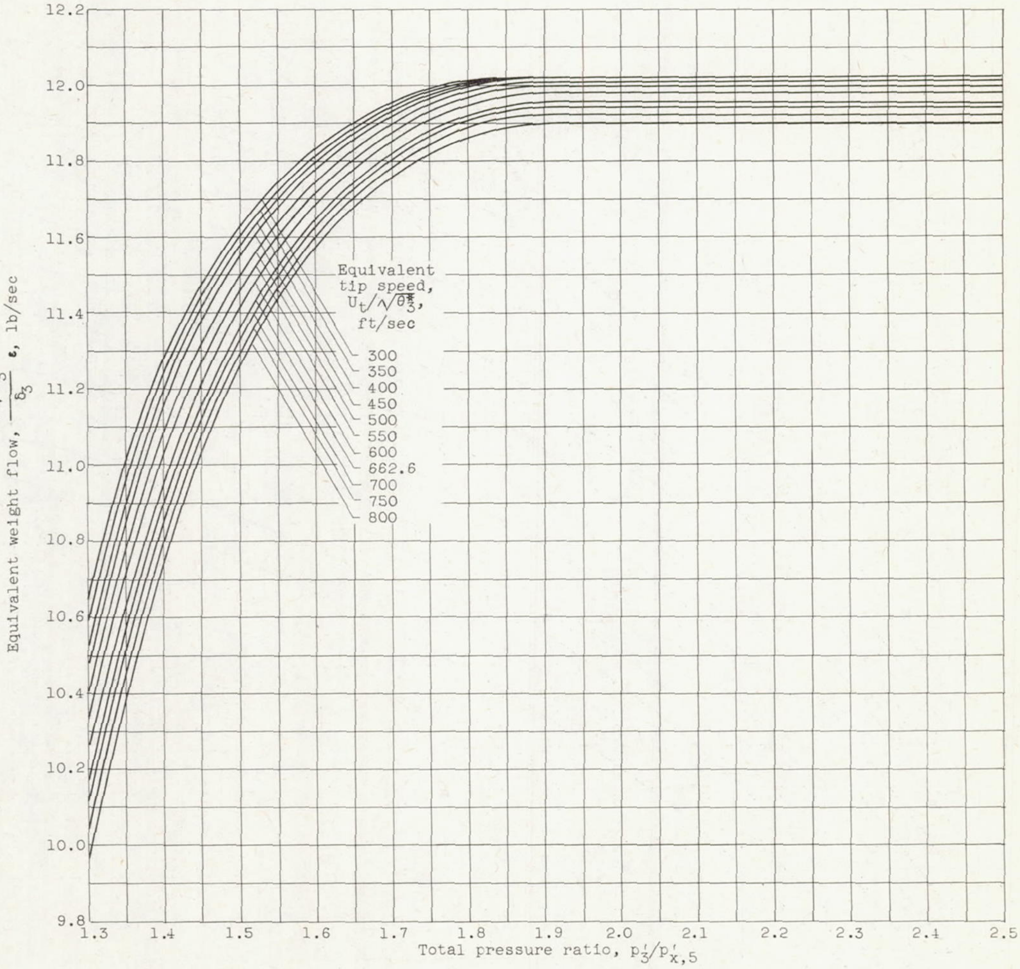


Figure 15. - Cross section of turbine showing instrumentation.



(a) Equivalent torque.

Figure 16. - Variation of equivalent torque and equivalent weight flow with pressure ratio for various equivalent tip speeds of scale model turbine.



(b) Equivalent weight flow.

Figure 16. - Concluded. Variation of equivalent torque and equivalent weight flow with pressure ratio for various equivalent tip speeds of scale model turbine.

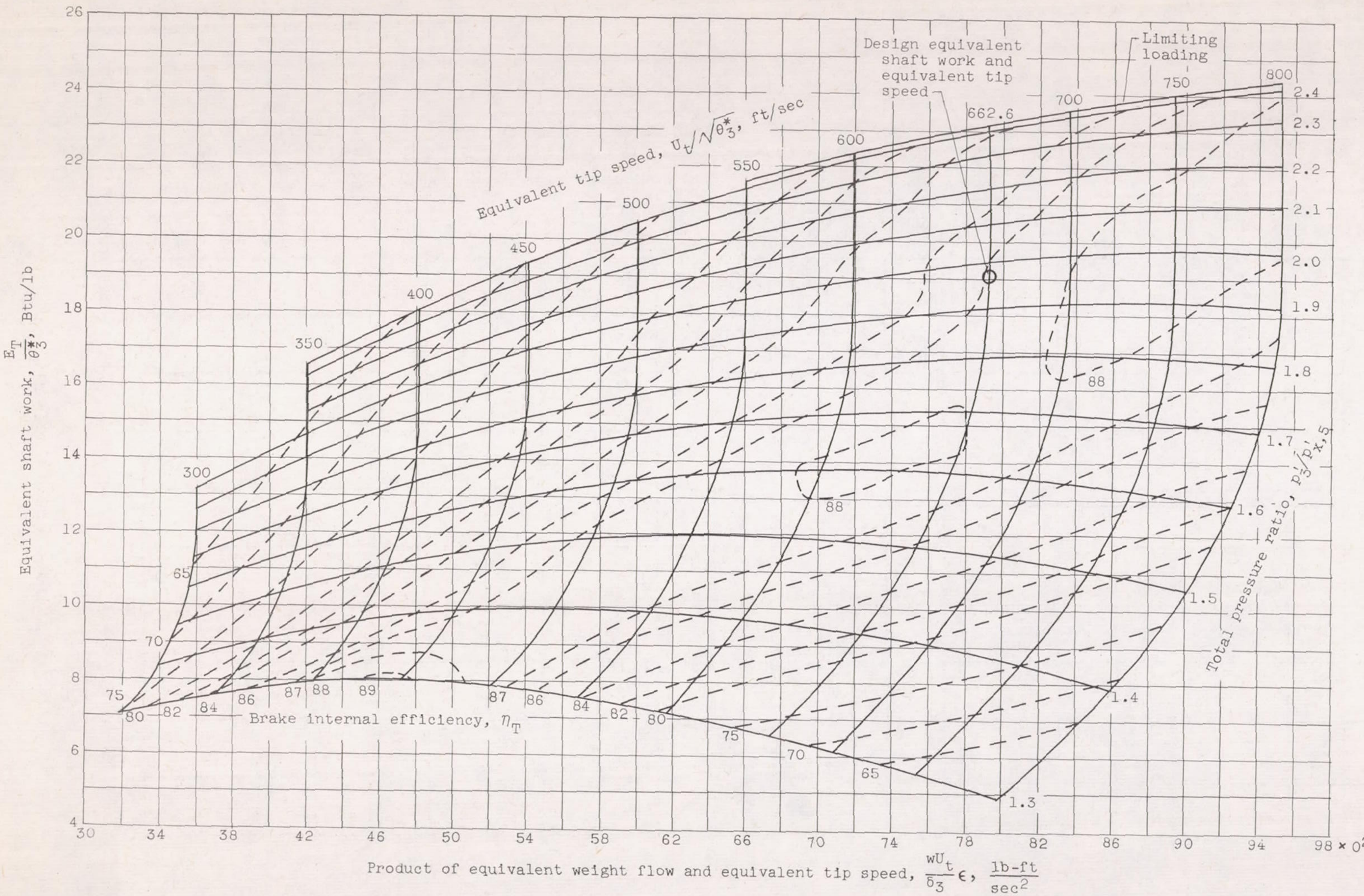


Figure 17. - Over-all performance of scale model turbine at engine temperature ratio of 4.75.

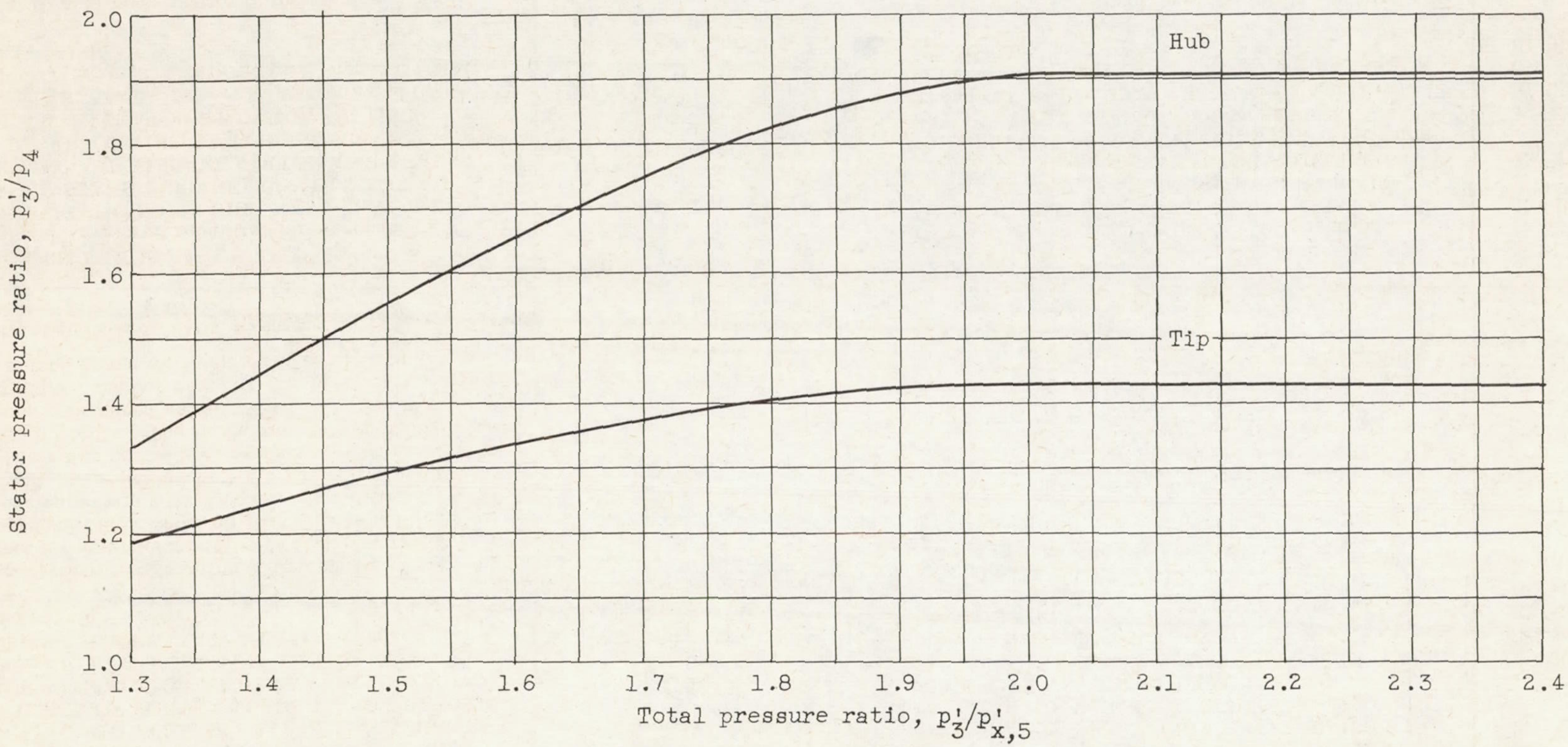


Figure 18. - Effect of total pressure ratio across turbine on total-to-static pressure ratio across stator. Design equivalent tip speed, 662.6 feet per second.