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ACTIVE COOLING OF HYPERSONIC AIRPLANES

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ABSTRACT

A combined turbine-compressor system is discussed for application to active cooling of hypersonic airplanes. The basic system analysis is discussed and performance variation analyzed. The various parameters of importance are discussed with their pertinent effects on the system operation displayed. The penalties of the system are shown to be smaller than with other active cooling systems.

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NOMENCLATURE

C_F	Integrated skin friction with injection
C_{F_o}	Integrated skin friction without injection
C_L	Lift coefficient
D	Drag
Δh	Enthalpy difference across rotating machinery
L	Lift
M	Mach number
n	Number of slots
p	Pressure
s	Slot height
S	Wing area
T	Temperature
U	Streamwise velocity; rotational velocity
V	Absolute velocity
w	Mass flow required per unit cooled surface area
W	Relative velocities; total mass flow required
x	Streamwise distance
α_e	Stator turning angle
β	Rotor turning angle
γ	Ratio of specific heats
δ	Boundary layer thickness
ϵ	Thermal effectiveness

NOMENCLATURE

η_c	Compressor efficiency
η_i	Inlet pressure recovery
η_t	Turbine pressure recovery
η_{RO}	Overall pressure recovery
λ	Mass flow ratio
ρ	Density
τ	Stator turning angle
τ_w	Shear stress with injection
τ_{w_o}	Shear stress without injection
φ	Fuel equivalence ratio

SUBSCRIPTS

e	External condition; condition at the entrance to turbocooler
ex	Condition at exit of turbocooler
f	First stage condition
is	Isentropic condition
j	Jet condition
L	Local condition; lower surface condition
u	Upper surface condition
w	Wall condition
o	Stagnation condition
2	Condition at entrance to turbine
3	Free stream condition
∞	Free stream condition

1. Introduction

The development of airplanes capable of flying at hypersonic Mach numbers is of great importance for future commercial, military, and space applications. From the point of view of commercial applications, this increase of airplane speed is attractive provided that the proportion of weight required by the vehicle and the engine structures do not increase substantially with flight Mach numbers and provided that the fuel consumption of the engines increases less than the speed of the vehicle. If these conditions can be met, then the cruise capability of the airplane will increase with increasing flight Mach numbers, thus making such an airplane economically attractive. Additionally, because of the higher velocities the time of flight decreases and since the vehicle can fly at a higher altitude, the strength of the associated sonic boom (measured on the ground) decreases. From the military point of view, the altitude and speed of flight are directly related to the utility of such a vehicle.

The immediate interest in hypersonic airplanes is due to the fact that the development of high-speed airplanes is of direct importance for space investigation. It is quite evident that deep space investigation will require the placement in orbit of large space stations, probably manned by many crew members remaining in space for long periods of time. Such development will require substantial reduction in cost per pound for orbit and a simplification of the operations related to launch and recovery. Safety will require quick response for launch without the need of long range planning. Such progress can be achieved only with the development of reusable first and second stage vehicles, capable of taking off and landing on preselected controlled points in a fashion similar to regular airplanes.

The introduction of vehicles usable for several flights will also decrease substantially the cost of operation. The requirement of landing at controlled points without direct ground assistance introduces the need for development of lifting vehicles capable of maneuvering in the atmosphere. Such additional requirements tend to increase the structural weight of the first and second stage and hence to decrease the actual payload of the last stage unless more efficient propulsion systems are employed. This increase of structural weight can be balanced by the use of more efficient airbreathing engines for the first stage vehicle in place of rocket engines. The main problems related to the development of such stages are the actual cost of their development and the necessity of new technology not yet available, related to advanced structural design. The technology for the design of efficient propulsion airbreathing engines is already substantially advanced so that presently it appears that the development of hypersonic airplanes will depend strongly on the ability of devising new design technologies that permit us to use light and inexpensive structures both for the engine and for the vehicle.

A major step in the direction of obtaining such technology could be achieved if new cooling schemes are developed permitting presently available structural construction techniques to be used. Current supersonic airplanes are constructed either of dural, titanium, or stainless steel. The use of such materials limits the acceptable surface temperature; therefore, cooling methods are required for hypersonic airplane design if these materials are to be used.

The purpose of this report is to outline the possibility of using an active cooling technique based on the following approach: Ram air is to be collected by an inlet and passed through an expansion turbine. The velocity entering the turbine is very large with its axial component supersonic. Therefore, the turbine can absorb, in a single stage, a substantial amount of kinetic energy and transform it into mechanical energy. The air to be discharged from the turbine is then at a sufficiently low stagnation temperature so that it can be used for slot injection on the surface of the airplane to keep the vehicle skin cool. The mechanical energy produced by the turbine may be absorbed by a supersonic compressor that accelerates ram air and produces thrust.

The purpose of this report is as follows:

a. to show that sufficient cold air at pressures suitable for injection can be generated by expansion through such a turbine

b. to show that the amount of air required is within practical limits

c. to suggest a turbine-compressor design that is simple and compact; thus several turbine-compressor units can be distributed around the airplane, reducing substantially the requirement for large ducts for the distribution of the coolant air

d. to indicate that the drag generated by the loss of kinetic energy of the air crossing the turbine is practically balanced by the reduction of skin friction drag and the thrust of the air crossing the compressor

e. to analyze briefly the turbine-compressor design, installation, and the cooling problems related to the turbine design.

This report deals mainly with the system analysis of the problem; other reports will discuss the detail of these various topics*.

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* Besides the authors, Mr. R. Sadala and Mr. F. Mendoza have contributed to some of the results presented here.

2. Production of Low Stagnation Temperature Air

The possibility of supplying cold air for the slot cooling of hypersonic airplane surfaces has been suggested by the senior author and investigated in detail in the past by the senior author and by Mr. A. Marino of General Applied Science Laboratories, Inc. (GASL)* The basic principle applied in this scheme is a supersonic turbine of impulse type placed downstream of a fixed geometry inlet as shown schematically in Fig. 1. The inlet decreases the free stream velocity to a lower supersonic speed by increasing the static pressure, decreasing the Mach number,and decreasing the stream tube area. Then the flow enters a supersonic turbine which changes its direction without major changes in the static pressure. Because of the high value of the entrance velocity and large turning in the passage, the turbine substantially decreases the total enthalpy of the flow in a single stage. Observe that this process can take place with zero static pressure rise across the rotor. Therefore, the turning can be performed efficiently since the variation of the static pressure required in the turning is due to the centrifugal forces in the passage and depends mainly on the length of the blades; consequently the pressure gradients in the boundary layer can be kept below separation limits. In addition, it may be observed that the larger the contraction ratio of the inlet, the smaller the turbine required for a given amount of mass flow and the smaller the amount of the total enthalpy decrease, Δh, across the turbine. However the variation of Δh as a function of the entrance Mach number M_e, is gradual while the decrease of frontal area of the turbine as a function of M_e is very rapid. The possibility of producing large enthalpy decreases per stage permits the design of a turbine capable of producing large amounts of cool air without the requirement of installing large size rotating components.

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* A patent application for a possible design has been filed through NYU under the senior author's name.

Typical performance of such a scheme is presented in Figs. 2 and 3. The case of fixed flight Mach number $M_\infty = 6$, but variable Mach number at the entrance of the turbine and thus variable contraction ratio of the inlet is considered. The turbine consists of a single stage of impulse type turning the direction of the flow 90° with a rotational velocity of 1800 ft/sec. As a function of the entrance Mach number, values of Δh produced by the turbine, values of a scaled area $A_e \rho_\infty$, for a unit mass flow of air of one pound per second at the entrance of the turbine, and values of exit stagnation temperature are shown in Fig. 2. The associated velocity diagrams used for this analysis are shown in Fig. 3 for several selected values of entrance Mach numbers.

In Fig. 4, values of exit stagnation temperature again for a flight Mach number of 6 over a range of M_e with the same value of turning angle and rotational speed are compared for one and two stage turbines. Figure 5 compares values of exit stagnation temperature as a function of flight Mach number for two values of turbine entrance Mach number M_e for one stage and two stage turbines having the same turning angle and rotational speed.

The data of these figures gives some important indications of the potentiality of such a device for airplane cooling. The values of the downstream stagnation temperature $T_{o_{ex}}$ so obtained indicate that sources of air usable for structural cooling can be made available by using a relatively simple one stage supersonic turbine of impulse type turning the flow 90° in the passage for Mach numbers below M = 6 and two stages for Mach numbers between 6 and 10. The value of $P_{o_{ex}}$, the exit stagnation pressure, which depends on the efficiency of the turbine and on the inlet pressure recovery, can be sufficiently high to permit the use of small ducts for the local distribution of the cooling air on the surface of the airplane.

Usually the efficiency of the turbine is expressed in terms of the adiabatic efficiency. Such efficiency is defined by the ratio between the actual variation of total enthalpy, Δh, and an ideal isentropic variation of total enthalpy corresponding to the same stagnation pressure ratio across the turbine.

In order to obtain some general information as to the possibility of such schemes, the efficiency of the turbine can be expressed in terms of an equivalent pressure recovery in place of the adiabatic efficiency. Then, as will be shown, the value of the exit stagnation pressure and exit stagnation temperature will become independent of flight Mach number.

This equivalent indicator of efficiency can be developed by defining a pressure recovery of the turbine given by the ratio between the actual stagnation pressure of the air downstream of the turbine to the isentropic stagnation pressure corresponding to the same Δh produced across the turbine. This concept was first introduced by A. Marino of GASL and permits relations to be obtained that are independent of flight Mach numbers. Thus a pressure recovery of the system can be derived as a function of the isentropic pressure ratio and the true pressure losses in the inlet and in the turbine. By using this notion of pressure recovery, the relation between stagnation temperature and pressure downstream of the turbine will be given below for the case of constant γ. Here $P_{o_{ex}}$ and $T_{o_{ex}}$ are the stagnation pressure and temperature of the air at the exit of the turbine, P_{o_∞} and T_{o_∞} are the stagnation quantities corresponding to free stream conditions, P_∞ and T_∞ the static free stream temperature and pressure, P_{o_2} is the stagnation pressure at the entrance of the turbine, and $(P_o)_{is}$ is the ideal isentropic stagnation pressure downstream of the turbine corresponding to zero losses in the turbine. Then

$$P_{o_{ex}}/P_\infty = \left(P_{o_{ex}}/P_{o_{is}}\right)\left(P_{o_{is}}/P_{o_2}\right)\left(P_{o_2}/P_{o_\infty}\right)\left(P_{o_\infty}/P_\infty\right)$$

Now define

$$P_{o_{ex}}/P_{o_{is}} \equiv \eta_t \quad \text{as the pressure recovery of the turbine}$$

and

$$P_{o_2}/P_{o_\infty} \equiv \eta_i \quad \text{as the pressure recovery of the inlet}$$

and since

$$P_{o_{is}}/P_{o_2} = \left(T_{o_3}/T_{o_2}\right)^{\gamma/\gamma-1}$$

and

$$T_{o_2} = T_{o_\infty}$$

simple algebra leads to

$$\left(T_{o_3}/T_\infty\right)^{\gamma/\gamma-1} = 1/\eta_{R_o} \left(P_{o_{ex}}/P_\infty\right)$$

where $\eta_{R_o} = \eta_t \times \eta_i$ and may be thought of as the overall pressure recovery of the system.

Possible values of η_i vary between 0.95 and 0.80 depending on the design and on the values of M_e and M_∞. The higher values corresponding to a configuration where the boundary layer is removed or energized by injection. Experimental data from turning passages with supersonic velocities indicate that pressure recoveries between 0.80 and 0.90 are feasible for the turning on the order of 90° envisioned here with entrance Mach numbers between 2 and 3 and no boundary layer control (Ref. 1). Then, combining these recoveries yields values of $\eta_{R_o} = \eta_t \times \eta_i$ conservatively between 0.7 or 0.6 for feasible turbine performance.

In Fig. 6 values of T_{o_∞} as a function of $P_{o_{ex}}/P_\infty$ are presented for $T_\infty = 400°R$. Note that the value of $P_{o_{ex}}$ is important in several respects. The cooling air as derived from the turbocooler is to be injected on the surface of the vehicle where the local pressure P_L can be higher than its free stream static value; as a consequence the losses leading to $P_{o_{ex}}$ must be such that we can still expel the fluid or $P_{o_{ex}} >> P_L$. For hypersonic vehicles in the general class under discussion a typical value of pressure coefficient on the airplane is 0.10 to produce sufficient lift for cruise. Then for free stream Mach numbers between 6 and 8, P_L/P_∞ varies between 3.5 - 5.5; then from Fig.6 cooling temperatures on the order of 1000°R are feasible. Naturally, lower temperatures are possible for still lower values of M_∞. It may be emphasized that such temperatures are already sufficiently low for available advanced structural design already developed for existing supersonic airplane configurations.

3. Turbocooler Design and Performances

The analysis of rotating machines having supersonic axial velocity has been performed in the past, (Ref. 2). It has been shown that turbines of impulse type having large entrance velocities can produce significant work per stage. However, the designs considered before were directed toward turbojet engine applications and therefore considered only small values of axial velocity. The senior author of this report, when associated with GASL, had developed the concept of a turbocooler, using the aforementioned turbines, with high supersonic axial velocity, for the production of cold air and subsequently using this air for slot cooling of aircraft structures. In this concept a turbine and a compressor having supersonic axial velocity are connected mechanically by a shaft, with the turbine and the compressor operating on different air streams; the compressor has the function of absorbing the shaft power of the turbine. The air entering the turbine is used for cooling; a different stream tube enters the compressor and this air is used for propulsion. At hypersonic speeds, the compressed air could be used in a combustor having supersonic combustion or could be discharged directly. A schematic design of such a device is shown in Fig. 7.

The thermodynamics and cooling capabilities of such a device appear promising for practical applications; several problems must be investigated. The shaft power produced by the turbine is transmitted to a compressor that operates a separate air stream. If the compressor is placed behind the turbine as indicated in Fig. 7, the air entering the compressor must undergo several turns in the inlet; as a consequence, at supersonic speeds, the inlet is inefficient and difficult to design. The design of the rotating components requires substantial development. The turbine and compressor must have high tip speed and must be cooled. An additional problem is related to the distribution of the generated coolant air on the surface of the airplane; note, however, that this is general to the installation of any active cooling scheme. Possible preliminary solutions of these problems will be presented here.

Many of the detailed thermodynamic aspects of the turbocooler will be discussed in another report; here a few of the main characteristics will be presented. First several specific design examples will be given. This will be followed by some results obtained by variation of the critical parameters so that complete definition of the system can be obtained.

A typical turbine for such an application can be composed of a set of guide vanes and one or two counterrotating stages. In order to have a feeling for the order of magnitude of the parameters involved, consider for example a flight Mach number of 6. Assume that the turbocooler is placed under the wing in a region where the local pressure coefficient is of the order of 0.10; then the local Mach number corresponds to about 5. (Only small losses are assumed.) An inlet is used in front of the turbine which reduces the entrance Mach number to a value of 4. Assuming that the total pressure recovery of the inlet is 0.90, including the effect of the wing, the stream tube at the entrance of the turbine has been contracted 4.45 times with respect to the axial direction. The axial velocity decreases and the cross section increases. The rotor then turns the flow 100°. A second rotor that rotates in the opposite direction is placed next to the first. The turbine tip velocity diagram for a rotational speed of 1600 ft/sec is shown in Fig. 8a; the velocity diagram for a compressor capable of absorbing the energy output of the turbine is shown in Fig. 8b. The discharged air stagnation temperature is equal to

1043°R. Such air has a sufficiently low temperature to be used for cooling and employing standard materials (e.g., Titanium) for the fabrication of the airplane.

In Fig. 9, turbine and compressor velocity diagrams are shown for a single stage device with $M_e = 4$ and $U = 1800$ ft/sec. The figures show that temperatures of the order of 1100° can be obtained at $M = 6$ with a single stage rotor at tip speeds of the order of 1800 ft/sec. The stagnation pressure of the air is of the order of 2.5 times its free stream value. It is important to note that the rotational velocity assumed for the compressor is smaller than the velocity assumed for the turbine, so that the compressor and turbine could be placed in a single wheel. Further consideration of this point will be made later.

The general thermodynamic characteristics of these systems can be defined by the values of the stagnation temperature and stagnation pressure of the coolant air as a function of free stream Mach number, and are affected by the turbine entrance Mach numbers and rotor tip speed. These effects will be discussed now. An acceptable value of the stagnation pressure should be of the order of twice to three times $(0.07\ M^2 + 1)P_\infty$. (The values of $(0.07\ M^2 + 1)P_\infty$ correspond to a pressure coefficient of $c_p = 0.10$.) Typical values of $T_{o_{ex}}$ obtainable are given in the curve of Fig. 10a as a function of flight Mach numbers for values of $P_{o_{ex}}$ equal to $3p_\infty$ and $4p_\infty$. The variation of $T_{o_{ex}}$ depends on the variation of losses in the inlet and rotors of the turbines. Typical values of inlet pressure recovery as a function of flight Mach number and rotor pressure recovery as a function of the entrance Mach number to the rotors are given in Fig. 10b.

The effect of rotational speed of the turbine is shown in Figs. 11a and 11b for $M_\infty = 6$. Fig. 11a corresponds to a single stage; Fig. 11b to two counter-rotating stages. Although not shown, the corresponding turbine velocity diagrams assume constant velocity turning. The entrance Mach number in Figs. 11a and 11b is kept constant and equal to 4. Note that the relative entrance Mach number is equal to 3 for a rotational speed of 1600 ft/sec, and even lower for higher rotational speeds. Experiments at M = 2.5 show that the pressure recovery of a turning passage at these Mach numbers can be as high as 0.93 (Ref. 2). Here it has been assumed to be of the order of 0.90; in addition, in the present design, boundary layer control must be introduced at the surfaces of the blades. Boundary layer control is required because slot cooling is planned for the blades of the turbine. The total pressure recovery of the system is given by the product of the inlet and rotor pressure recoveries in view of the fact that the turning does not change velocity and therefore Mach number of the relative coordinate system. Hence the pressure recovery in the passage changes only the value of the exit static pressure.

The idea that a single wheel could be employed was observed earlier and is based on the fact that the rotational velocity of the compressor is smaller than that of the turbine. Figure 12 shows a schematic design of such a single wheel installation. The design presented in Fig. 12 avoids the necessity of large turning in the compressor inlet. The unit is small and therefore several can be mounted on a vehicle in order to produce the coolant air close to the region of utilization, and therefore the problem of ducting the coolant air is minimized.

The turbocooler concept can be utilized in a large range of Mach numbers. At Mach numbers of the order of 3 to 4, the design of the turbocooler is simpler because the structural problems are similar to the problems encountered in present turbojet turbine design. This concept can lead to a direct advance to a $M_\infty = 4$ vehicle with little change in present structural design. In particular, Fig. 13 gives a typical velocity diagram for $M_\infty = 4$ and a one stage turbine. The figure shows that a one stage turbine can produce air at a stagnation temperature of 800°R with a turbine rotational speed of 1600 ft/sec. Then Dural construction can surely be used for a $M = 4$ airplane if slot cooling is applied.

Finally, the actual efficiency of the machine is a very important parameter because it fixes the minimum temperature selected for the coolant air, and in addition fixes losses in the momentum of the coolant air, which in turn is related to the drag of the system. The efficiency of the turbine cannot be determined accurately until a substantial amount of data on cascade and rotating component experiments become available. Presently, the only such information at hand is at Mach numbers in the range of 2 to 2.5 (Ref.1). Such information can be extrapolated to Mach numbers of 3 to 3.5 and used in the turbine design for flight Mach numbers of the order of 3 to 5.

Consider the flow in the passage next; a typical turbine passage design for a 90° turning and entrance Mach number of 3 is shown in Fig. 14. The flow is compressed at the leading surface of the blade and expanded on the trailing surface. The shock produced at the leading edge is contained inside the passage, because the axial velocity is supersonic. However, such a shock is cancelled at the opposite side. The flow at the end of the passage can be expanded to the static pressure of the trailing surface or can be recompressed to the entry static pressure. Usually the recompression produces local separation that should be avoided in order to obtain good efficiency. In an actual blade design for flight Mach numbers of the order of 6 or higher, cooling is required at the leading edge of the blade, and along the curved surface.

The aerodynamic heating of the turbine and compressor blades is a function of the stagnation temperature of the stream entering the wheel corresponding to the properties in relative coordinates. This quantity is affected by the free stream Mach number, velocity diagram at the entrance of the wheel, and the rotational speed. Fig. 15 indicates the values of the stagnation temperature of the air in relative coordinates, as a function of free stream Mach numbers for several rotational speeds and entrance conditions corresponding to $M_e = 4$ and $\beta = 60°$ (of the stator). Fig. 16 gives the value of stagnation conditions in rotating coordinates for a value of U = 1800 ft/sec, and several values of M_e and α_e.

To prevent separation and cool the blades high pressure, high Mach number coolant air can be injected near the blade rather than employing boundary layer scoops. Figure 17 indicates the design of a passage with such injection. High pressure air at high Mach numbers is injected tangentially to the blade at given stations along the chord with the injected air having lower stagnation temperature than free stream. The Mach number of the injected air is higher than the outside Mach number; however, its velocity is lower. The boundary layer is then energized because it is cooled by the mixing and the wall temperature is kept close to the stagnation temperature of the injected air.

In Fig. 18 the characteristics of the air after mixing are shown for a typical case. The injected air has a stagnation temperature of $1000^{o}R$ and a Mach number of 6 at the injection station. The boundary layer existing ahead of the slot is accelerated close to the free stream Mach number. The amount of injected air considered in the case shown in the figure is equal to twice the mass flow of air in the boundary layer.

The most difficult cooling problem is related to the blade leading edge A large leading edge radius would produce strong detached shocks with large losses in the region of such strong shocks. However, active cooling of the leading edge appears promising for such applications. High pressure air is injected upstream at the leading edge probably at an angle with respect to the leading edge. Then the radius of the blade can be kept to a minimum radius, (less than 1 mm.) with a shock corresponding to a leading edge radius without injection of the order of 1.5 mm. Then the high entropy losses affect only the flow which is energized by downstream injection under the boundary layer.

Fig. 19 schematically shows the shock and flow near the stagnation point for these conditions. This analysis will be discussed in a separate report. With this approach the pressure recovery of the passage can be very high. The pressure recovery in the turning passage shown in Fig. 14 is equal to 0.93 (including viscous losses).

It is important that the flow leaving the passages be as uniform as possible; a non-uniform distribution of the velocity of the exit of the guide vanes or at the exit of the first rotor produces additional losses due to non-steady wake effects. In the proposed design, the introduction of boundary layer bleeds and slot injection gives a much more uniform velocity distribution at the exit minimizing such losses.

The cooling of the turbine and compressor blades requires availability of high stagnation pressured-low stagnation temperature air. Such a requirement exists already for the hot regions of the engine. The turbocooler concept combined with the use of fuel cooling can be used for generating such sources of air that are required. A possible typical configuration for producing this coolant air is illustrated in Fig. 20. The turbine passage is followed by a heat exchanger with the medium for the heat exchange taken to be the fuel. For the example presented here hydrogen is assumed for the fuel. After the heat exchange process the air is then recompressed (utilizing some of the energy from the turbine) to high stagnation pressure values. For the example shown, the coolant air can have temperatures as low as 500°R and high stagnation pressure. This value is certainly acceptable for blade cooling.

4. Basic Characteristics of Slot Cooling

(a) Thermal Effectiveness

Relatively few experiments on slot and porous cooling have been performed at the high Mach number and high Reynolds number of interest in the present application. Recently extensive experimental results have been performed at New York University by K. Parthasarathy and V. Zakkay (Ref. 3). The injection from a single slot was investigated at Mach 6 in a natural transition turbulent boundary layer. Predominantly the measurements were performed for cases where the slot to boundary layer thickness ratio was extremely small * and included detailed heat transfer results. From the heat transfer data the adiabatic wall temperatures could be determined. It was found that such results could be correlated in terms of a cooling effectiveness, ϵ, defined by

$$\epsilon = (T_{o_w} - T_{o_e}) / (T_{o_j} - T_{o_e})$$

as a function of mass flow weighted distance, $(x/s)\,\lambda^{-.8}$, where x/s is the downstream distance measured in slot heights and where

$$\lambda = \rho_j U_j / \rho_e U_e$$

Representative results for this single slot experiment are shown in Fig. 21.

* In these experiments $\delta/s \approx 0(100)$ while in most previous work this ratio was $\delta/s \approx 0(1)$.

The results indicate that up to a value of $(x/s)\ \lambda^{-.8}$ of the order of 150 the temperature of the wall downstream of the slot remains roughly equal to the temperature of the air injected in the slot. If a second slot is then placed at the actual physical distance, x corresponding to this value, and additional coolant is injected, the surface downstream of this point remains, for some additional distance, again at the temperature of the coolant air.

At the point of injection of the second slot, the boundary layer profile is quite different from the boundary layer profile ahead of the first slot; therefore the efficiency of the second slot can be expected to be greater than the efficiency of the first slot. At the second slot the upstream flow from the first slot is far from being completely mixed. The boundary layer profiles at this station are qualitatively displayed in Fig. 22; therefore the mixing between the boundary layer air and the coolant downstream of the second slot is slower and consequently the efficiency of the second injection must be higher.

In order to obtain additional information on multiple slot effectiveness a preliminary experimental investigation has been performed on a three slot configuration at New York University. These results will be discussed in detail in a separate report by H. Fox and W. Hoydysh (Ref. 4). Additionally an analysis has been performed based on mixing-type calculations for treatment of more than one slot. This analysis has been tested against available experimental results both for the new experiment at M = 6 and for lower Mach numbers. The results of this analysis are discussed in detail in a separate report (Ref. 4) and are summarized in Fig. 23.

The results indicate that for multiple slots an average value of $(x/s)\lambda^{-8}$ of the order of 250 can be used for an effectiveness down to 0.9. This value then permits determination of the amount of coolant required for any given configuration. For a given value of $(x/s)\lambda^{-0.8}$ the value of x/s is a function of the particular value of λ. From Fig. 23 it can be seen that lengths 12 ft. long can be cooled with slots 2" high and mass flow characterized by $\lambda = 0.2$. If λ becomes of the order of 0.6 than a slot with height 2" can cool 29 ft. lengths. Thus a wing having a 100 ft. chord would require 4 to 8 slots depending on the mass flow available.

(b) The drag of the slot cooling system

The turbine cycle decreases the kinetic energy of the air and therefore produces a drag. At the same time, the mechanical energy transmitted to the compressor produces a thrust; therefore, the difference between drag and thrust is a contribution to the net drag of the airplane. However, the injection of low velocity air at the surface of the body decreases the overall skin friction of the airplane. In order to evaluate the complete contribution of the cooling to the drag of the airplane all of these quantities need to be evaluated. The drag and thrust of the turbine and compressor flows can be evaluated by assuming reasonable values for losses of stagnation pressure in the passages. The effect on skin friction is more difficult to determine. Very little experimental data is available, and such data as are reported have been obtained for regions very far downstream of the slot where the mixing between coolant air and external air has been extensive and essentially complete. In the scheme proposed here, for optimum cooling effectiveness the air near the wall is always at the stagnation temperature of the injected air so that a condition of zero heat transfer is obtained. For this condition, the mixing has not yet reached the wall and the properties of the air outside the boundary layer correspond to the coolant air. As a consequence the skin friction can be determined by assuming that the boundary layer has, as effective external conditions, the properties of this injected air. When this observation is utilized and it is assumed that the boundary layer initiates somewhat upstream of the slot inside the channel that carries the air to the slot, the skin friction can be determined. Note that the boundary layer is very probably turbulent because the turbulence of the external flow will generate fluctuations that propagate inside the injected air. When this observation is utilized in the region where $\epsilon \sim 1$ and before the mixing reaches the wall, an approximate relation between shear stress with slot injection and shear stress without injection, for the same wall temperature can be obtained as a function of λ. The results of such an analysis are shown in Fig. 24. The mentioned observation is sufficient for the determination of the skin friction for the applications discussed in this report. However, it can be useful to test such an approach by comparing calculated and experimental results, available for $\epsilon < 1$. To complete the distribution of shear stress in the region where $\epsilon < 1$ it is assumed that the relation between skin friction and measured heat transfer can be determined by implementation of the Reynolds' analogy. This assumption has been used in Ref. 3 and has given satisfactory results. If this analogy is considered valid in the region when the mixing between the outer boundary layer and inner jet flows takes place, then it is possible to obtain the skin friction from the measured heat transfer and then compare the results of such an analysis with available experimental measurements of skin friction. The local skin friction has been determined from the heat transfer data at M = 6 and then has been integrated along the length. In Fig. 25, the results of the integration for any given length are shown for 3 values of λ. The data are presented as C_F/C_{F_o} where C_{F_o} is the integrated skin friction in absence of injection. An experimental data point for $\lambda = 0.10$ obtained at M = 3 is also shown in the figure. The difference between the calculated values for M = 6 and the experimental value at M = 3 is small, giving confidence to this approximate computational scheme. On the basis of this information the penalty of the system can be determined to first approximation.

5. Turbocooler Design for Engine and Turbomachinery Cooling

The static pressure on the external surface of the airplane is close to atmospheric pressure for all high-speed flight conditions; therefore the coolant air needs a minimum stagnation pressure equal to 3 or 4 times free stream stagnation pressure. Then an efficient turbine is sufficient for generating such a stream. However, the situation is different when we apply slot cooling to the components of an engine. Here the pressure is several times free stream pressure. At high Mach numbers, on the order of 5 to 12, the simplest engine to be considered is the scramjet, where combustion takes place at low supersonic speed. For this system the static pressure in the region of the combustion reaches values of the order of 200 times free stream pressure, while at the walls the pressure is equal to 50 to 60 times the free stream pressure, provided that three-dimensional effects are utilized in the engine design. Therefore, slot cooling utilizing the air from an expansion turbine cannot be employed.

For the regions of high static pressure a different turbocooler scheme can be envisioned. As noted in Section 3 the amount of coolant air required for the high pressure regions of an efficient engine design is small; therefore the amount of high pressure air required is also small. In this case the fuel can be utilized to cool the injected air in place of using the fuel to locally cool the engine structure. The main advantage of such a scheme is that it does not require distributing the fuel all along the structure with carefully controlled mass flow rates as a function of the local heat transfer rates and therefore permits a much simpler and lighter engine structure. The scheme proposed has been shown in Fig. 20. A supersonic turbine cools the air to a stagnation temperature of $2500^{o}R$. The air then enters a heat exchanger where fuel is used to further decrease the temperature of the air. If the heat exchanger uses hydrogen fuel to absorb the heat, the heat exchanger may be small and the final temperature of the air can reach values as low as $250^{o}R$. The air, after cooling, is recompressed back to high stagnation pressure by a compressor that utilizes some of the shaft power of the turbine.

The amount of mechanical work generated by the turbine is absorbed by the compressor after cooling. Such work can produce a pressure ratio through the system given by $p_{o_{exit,c}} / p_{o_\infty} = \eta_i \eta_t \left[(1 - \frac{\Delta h}{h_{o_\infty}}) \; (1 + \eta_c \; \frac{\Delta h}{h_{o_3}}) \right]^{\frac{\gamma}{\gamma-1}}$ where η_i is the inlet pressure recovery, η_t is the turbine pressure recovery (loss of stagnation pressure with respect to an isentropic transformation), Δh is the enthalpy drop across the turbine, η_c the compressor adiabatic efficiency (defined in the standard way), h_{o_∞} the free stream stagnation enthalpy and h_{o_3} the enthalpy of the air at the exit of the heat exchanger. The value of $\frac{\Delta h}{h_{o_3}}$ is larger than one and can be as high as 10,12 while $\frac{\Delta h}{h_{o_\infty}}$ is much less than one; therefore $p_{o_{exit,c}}$ can be much larger than p_{o_∞}. The availability of high pressure cold air can simplify strongly the inlet combustor design because it permits energizing the boundary layer and so avoids separation without the necessity of bleed.

An alternate possibility to generate coolant air could be to use only the heat exchanger; then the air is decelerated in an inlet and a high-pressure high-temperature heat exchanger is inserted at the end of the inlet. The second approach requires the design of a very efficient variable inlet that makes design of the system difficult. In the first approach the main structural problem to be solved is to cool the rotating turbine leading edges. Such a problem exists also for the turbocooler and can be solved by upstream injection as discussed before.

The turbine and heat exchangers required are relatively small. Consider, for example, a scramjet engine for a M =8 vehicle weighing 10^6 lb. The thrust required for cruise is on the order of 250,000 lbs. to 300,000 lbs. Then if we assume that at cruise $\varphi < 1$ is utilized in the engine, a mass flow of air on the order of 7,500 lbs./sec must cross the engine. If the airplane flies at 100,000 ft. then a possible practical engine will require on the order of 900 $ft.^2$ of capture area. If the inlet is 50 ft. long and 18 ft. high, then the high pressure region to be cooled will have an area on the order of 50 x 3 ft. x 2. (The length of the inlet burner nozzle high pressure region is estimated conservatively to be 3 ft. long.) The amount of air required to keep this region at 1200^oR can be calculated as follows. Assume single slot injection, and coolant air at a stagnation temperature of 500^oR then $(T_{o_e} - T_{o_w})/(T_{o_e} - T_{o_j}) = 0.920$. Assume an inlet pressure recovery of 0.80, then the system of Fig. 20 will give a $p_{o_4} / p_{o_\infty} = 0.476$. If the burner static pressure is 60 p_∞, then the discharge Mach number of the coolant air is 3.51, the ρ_j is 0.174 $\frac{lb}{ft^3}$, and the velocity is 2070 ft/sec. The external conditions, if the engine inlet pressure recovery is assured to be 0.75, are $M_e = 3.84$, $\rho_e = 0.0175$ lb/ft^3 and $V_e = 7150$ ft/sec. Then for x= 2ft., s is 0.425×10^{-2} ft and the total mass flow required is 75 lb/sec. Such mass flow can be generated by a turbine having an entrance Mach number of 4, and 0.6 ft^2 passage area. The air can be slightly overexpanded in order to increase the dimensions of s or the flow can be injected upstream where the local pressure is lower. This concept appears to be simpler than a structure having hydrogen internal cooling, distributed over an area of 1200 ft.^{2}, and much less dangerous from an operational point of view.

A similar approach can be used to cool the turbocooler components. Present turbine technology indicates that rotating machines can be constructed operating at high static pressures and high stagnation temperatures, provided that the blades are cooled by injection or by a circulation of coolant in the blades. A stagnation gas temperature on the order of 3200°R is considered possible with internal cooling. If the same stagnation temperature can be accepted for the turbine-compressor, then the turbocoolers for flight Mach numbers on the order of 6 to 8 could be developed by using internal cooling and existing technology. However, in the present design, the flow is entirely supersonic; therefore, the leading edge of the blade should have a small radius otherwise severe losses would result at the leading edges. For this application upstream injection and slot injection of the coolant appear more attractive. A significant amount of experimental and analytical work is required to develop the design of slot and upstream injection; however, the amount of air required for each blade is small and therefore the system is feasible.

6. Contribution of the Cooling System to the Drag of the Vehicle

In order to evaluate the total contribution to the drag of the vehicle due to the loss of momentum of the injected air three quantities must be evaluated: (1) the loss of momentum of air used as coolant; (2) the thrust imparted to the air by the compressor; and (3) the decrease in skin friction drag due to slot injection. The first two quantities depend on the performance of the rotating components, and the third on the characteristics of the turbulent boundary layer on the vehicle. In this section, the drag penalties are estimated for two vehicles in order to show that such penalities are minimal provided that efficient turbocoolers can be developed:

(a) Vehicle 1 is taken to be a commercial transport with a weight of 0.5×10^6 lbs flying at a Mach number of 4 at 90,000 ft and having a lift to drag ratio, L/D, equal to 7. The lift coefficient, C_L, is assumed to be 0.11.

(b) Vehicle 2 is taken to be a space launcher also with a weight of 0.5×10^6 lbs but flying at $M_\infty = 6$ at 110,000 ft with an L/D = 4 and with $C_L = 0.14$.

For both vehicles the total wetted area is assumed twice the wing area plus 0.4 times the wing area (for the other components). The average pressures on the upper and lower surfaces of the wing are determined so that sufficient lift is obtained; the average pressure on the remaining surfaces is then taken to be equal to the local ambient pressure. With these assumptions and vehicle definition, the cooling requirements, the thrust produced by the compressor, and the reduction in skin friction drag may be determined.

Consider first the M =4 vehicle. The coolant air temperature is taken equal to $720^{o}R$. An inlet pressure recovery of 0.92 is assumed; the rotating machinery is assumed to consist of a single rotor with an inlet pressure recovery (in the guide vanes and turning passage) of 0.90. Computation yields the result that the exit stagnation pressure is equal to 6.4 P_{∞}. Further, taking into account losses in the ducts carrying air to the slots, the stagnation pressure at the injection surface can be assumed to be $P_{o_j} = 5\ P_{\infty}$.

The vehicle coolant requirements can then be estimated for the three surfaces as follows:*

(i) High pressure surface (lower surface) (average quantities)

local static pressure: $P_L = 1.8\ P_{\infty}$

local static temperature: $T_L = 1.18\ T_{\infty}$

local Mach number: $M_L = 3.57$

local density: $\rho_L = 1.52\ \rho_{\infty}$

local velocity: $V_L = 3820$ ft/sec

The turbine output is assumed to expand to the local atmospheric pressure $P = P_L$; the coolant properties are then

$P_j = P_L$

$\rho_j = 1.35\rho_{\infty}$

$u_j = 1480$ ft/sec

and the required $\lambda_{jL} = \rho_j u_j / \rho_L V_L = 0.342$

*Details will be presented for the high pressure surface only. Since similar techniques can be applied to the other surfaces as well, only the important computed data will be presented for these.

Since multiple slots are required for vehicle cooling, a value of $(x/s)\lambda^{-.8} = 250$ is assumed and may be recognized as typical of effectiveness for a multiple slot configuration where $\varepsilon = 0.9$. Then for a distance between slots of L = 10ft, s = 0.0945 ft. A convenient parameter for defining coolant requirements is the mass flow required per unit cooled surface area:

$$w_L = (\rho_j u_j s) / L$$

and for the high pressure side

$$w_L = 0.0316 \text{ lb/sec-ft}^2$$

The total mass flow required for the lower surface is then given by:

$$W_L = w_L S_L$$

where the area, S_L, may be defined by

$$S_L = 2W / (\gamma P_\infty M_\infty^2 C_L)$$

For the condition here $S_L = 11{,}250 \text{ ft}^2$ and $W_L = 356$ lb/sec.

(ii) Low pressure surface (vehicle upper surface):

local static pressure:	$P_u = 0.524\ P_\infty$
local Mach number:	$M_u = 4.5$
mass flow ratio:	$\lambda_u = 0.44$
total mass flow required	$W_u = 153$ lb/sec

(iii) Remaining surfaces:

local static pressure:	$P_f = P_\infty$
local Mach number:	$M_f = 4$
total mass flow	$W_f = 102$ lbs/sec

Therefore, the total coolant requirement can be estimated as:

$$W_T = W_L + W_u + W_f = 611 \text{ lb/sec}$$

Assuming that 12 turbocoolers are used, then each must handle 100 lb/sec which is to be apportioned into 50 lb/sec for the turbine and 50 lb/sec for the compressor. Based on the velocity diagram of Fig. 13 if the Mach number at the turbocooler face is of the order of 3, then the wheel diameter is roughly 3 ft. The exhaust area is of the same order. The 12 units can be conveniently placed around the vehicle to reduce to a minimum the ducting losses. A schematic diagram of a possible installation is shown in Fig. 26.

Consider now the performance of the compressor. The work output of the turbine is 233 Btu/lb. If this power is absorbed by a compressor having a pressure recovery of 0.85 and processing the same mass flow of air on the turbine, the exit stagnation temperature of the compressor is 2660°R. If the flow is then expanded to P_{∞} the exit Mach number is 5.08*. The velocity at the exit of the compressor bozzle is 5200 ft/sec and corresponds to a thrust of 39.1 lbs/lb/sec of air.

The net change in drag can now be evaluated by considering the various contributions. The drag of the turbine section can be determined quite simply by investigating the velocity of the coolant air when expanded to free stream conditions. By doing so, the drag of the coolant device becomes 66.3 lbs/lb/sec of air. To this and to the thrust of the compressor must be added the decrease in drag due to the reduction in skin friction produced by the coolant air. For

*In view of the relatively low temperature then γ = constant = 1.4 is employed.

each of the surfaces this may be written as:

$$D_{red} = (1/2)\, \rho_{local}\, U^2_{local}\, A_{local}\, C_{f_o}\, [1 - \tau w/\tau w_o]$$

where C_{f_o} is the skin friction coefficient with no injection, and $\tau w/\tau w_o$ is the ratio of shear stresses with and without injection respectively. The appropriate value of C_{f_o} may be determined by the classical techniques while the value of $\tau w/\tau w_o$ may be determined as in Section 5. For vehicle 1, the total reduction in skin friction drag becomes

$$D_{red} = 14140 \text{ lbs.}$$

then the penalty is simply:

$$D_{penalty} = (39.1 + 66.3)\, 611 - 14140 = +2460 \text{ lbs.}$$

which for a vehicle in the present class with an L/D = 7 really does not amount to a substantial change in fuel requirements. Comparison of the penalty should be made with similar penalties incurred when internally cooled structures are employed. The other possible comparison is with a similar airplane cruising at lower Mach number and zero cooling, so that the structure is at the same temperature. Then the fuel consumption for cruising varies with the ratio of the drag to the flight Mach number. Then an increase of M from 2.7 to 4 would justify a 47% increase of fuel for cruising.

A similar analysis can be performed for the second vehicle operating at 110,000 ft and $M_\infty = 6$. For this case the coolant temperature is 1090°R and with a pressure recovery for the inlet of 0.9 and for the turbine of 0.7, the exit stagnation pressure is $P_{o_{ex}} = 16P_\infty$. Taking the duct losses the same as before, the injection stagnation is $P_{o_j} = 12.5P_\infty$.

The coolant requirements can be evaluated in a manner exactly analogous to the $M_\infty = 4$ case. Summarized below are the pertinent aerodynamic characteristics:

(i) High pressure surface (lower surface)

local static pressure:	$P_L = 4.7P_\infty$
local Mach number:	$M_L = 4.62$
mass flow ratio:	$\lambda_L = 0.248$

In this case L is taken to be 16.5 ft and the corresponding wing area $S = 10100\ ft^2$. Then

$$w_L = 0.0348\ lb/ft^2 sec$$

$$W_L = 352\ lb/sec$$

(ii) Low pressure surface (upper surface)

local static pressure:	$P_u = 0.1275\ P_\infty$
local Mach number:	$M_u = 8.30$
mass flow ratio:	$\lambda_u = 0.4$
total mass flow required:	$W_u = 31.2\ lb/sec$

(iii) Remaining surfaces:

local static pressure:	$P_f = P_\infty$
local Mach number:	$M_f = 6$
total mass flow required:	$W_f = 153\ lb/sec$

Therefore, the total coolant required is:

$$W_T = W_L + W_u + W_f = 536\ lb/sec$$

The thrust produced by the compressor is now determined. The work output of the turbine is 595 Btu/lb. This is absorbed by a compressor having a pressure recovery of 0.7 and processing the same mass flow of air as the turbine; the exit stagnation temperature of the compressor is then 6050°R. This flow is expanded to P_∞ and the exit Mach number is 7.48 with an exit velocity of 8200 ft/sec. This leads to a thrust per lb of air processed of 63.6 lbs/lb of air.

The total drag penalty can be evaluated in a manner analogous to the $M_\infty = 4$ vehicle. In this case the turbine (coolant) section produces a drag (per unit mass flow) of 109.5 lbs/lb/sec of air. The total reduction in skin friction drag can be evaluated and is:

$$D_{red} = 15250 \text{ lbs}$$

The drag penalty is then:

$$D_{penalty} = - 536 \left[63.6 - 109.5\right] - 15250 = +9350 \text{ lbs}$$

which again is not very severe and only amounts to less than 8% of the total drag of the vehicle.

Finally, it is of interest to consider some of the effects of variations in performance of the rotating components. The most important parameter is clearly the pressure recovery of the turbine; its effect is two-fold: an increase of pressure recovery decreases the momentum losses of the injected air; however, for a given coolant temperature the value of $\rho_j u_j$ increases and as a consequence λ increases. With λ determined, the above analysis can be applied and the required mass flow and drag penalties assessed. For Vehicle 1, o.e., $M_\infty = 4$, $C_L = 0.11$, these calculations have been performed as a function of η_{RO}. The variation of λ are shown in Fig. 27 for Vehicle 1 at the three surfaces noted above. The total mass flow requirements and the drag penalty (ratioed to the total drag of the vehicle) is presented in Fig. 28.

An alternative approach is to select an effectiveness and a maximum wall temperature as the basic parameters. These in turn lead to the nondimensionalized cooling length $(x/s)\lambda^{-0.8}$ and to the required coolant temperature. This approach is quite reasonable since now we are essentially closing the loop and asking for basic requirements, i.e., output, of the turbocooler for a specific design condition. Additionally we will investigate here again a range of efficiencies so that sensitivity to this parameter can be obtained.

Proceeding in general, as before, but now with $T_{ow} = 1000°R$ the variation of mass flux ratio, $\lambda = \rho_j u_j / \rho_e u_e$, with effectiveness, ϵ, and with overall pressure recovery, η_{RO} is shown in Fig. 29. Displayed thereon are the high pressure surface (lower surface) and the low pressure surface (upper surface) values. An additional scale is also shown indicating the variation of coolant temperature, T_{oj}, with ϵ.

The total mass flow requirement, W_T, is shown in Fig. 30. The most interesting feature here is the relative insensitivity to the substantial variations in pressure recovery.

Of primary interest are the associated drag penalties under such conditions; these are displayed in Fig. 31. We observe for a reasonable effectiveness, i.e., $\epsilon \gtrsim 0.8$, the drag penalty even for a very inefficient device, is not more than roughly 10%. Further when the effectiveness is low corresponding to large cooling lengths and when the efficiency is high the entire device can in fact produce a net thrust.

7. Conclusions

An analysis of active cooling systems has been performed. The system, based on using supersonic expansion turbines that drive compressors, appears attractive from the viewpoint of producing coolant air to be used for slot cooling. The drag generated by the cooling mechanism is not high, and does not penalize the system. The main problem of mechanical design appears to be solvable and the development of hardware can be performed in already existing facilities for flight Mach numbers as high as 10.

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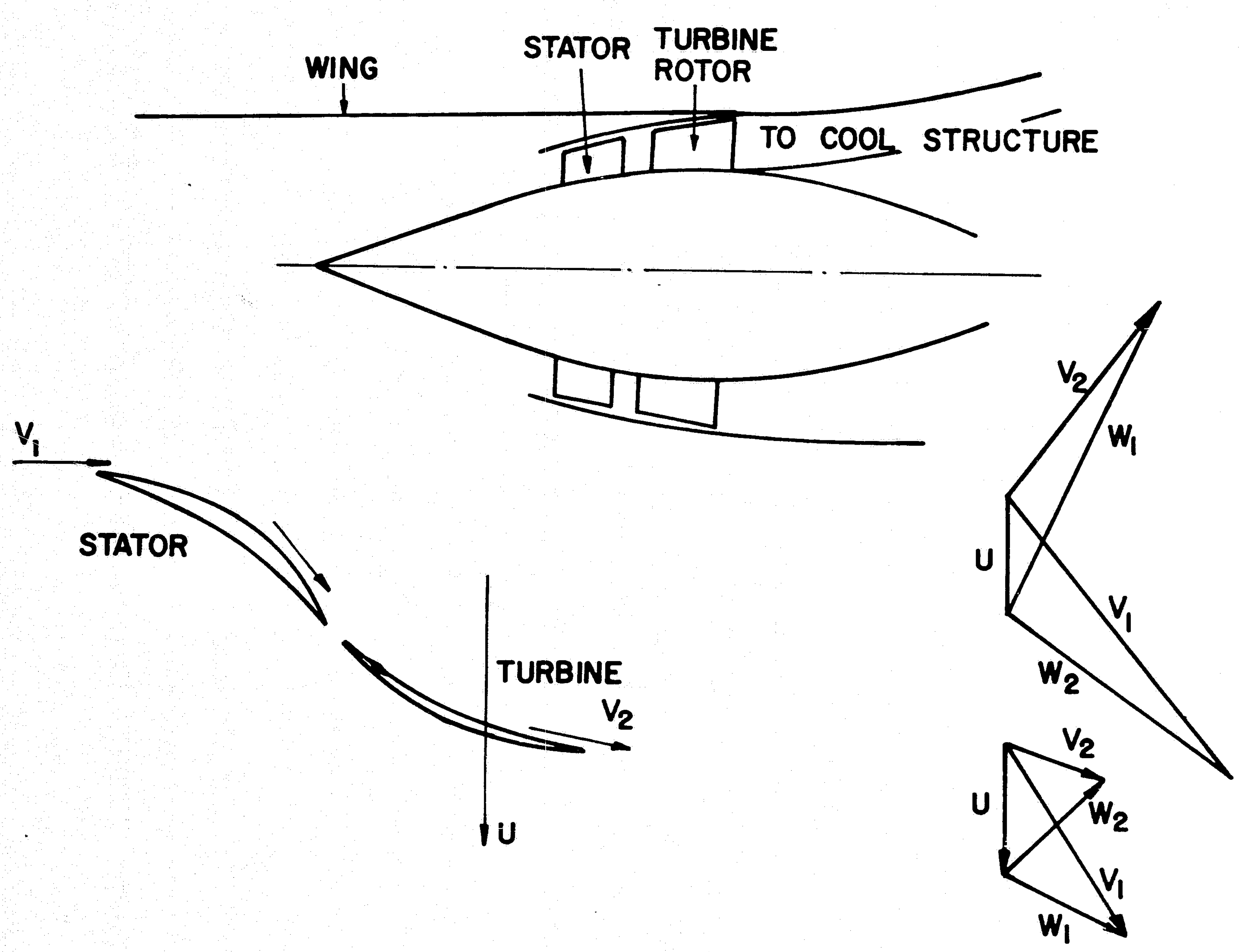

Fig. 1 Schematic diagram of turbine cooling device with typical velocity diagrams

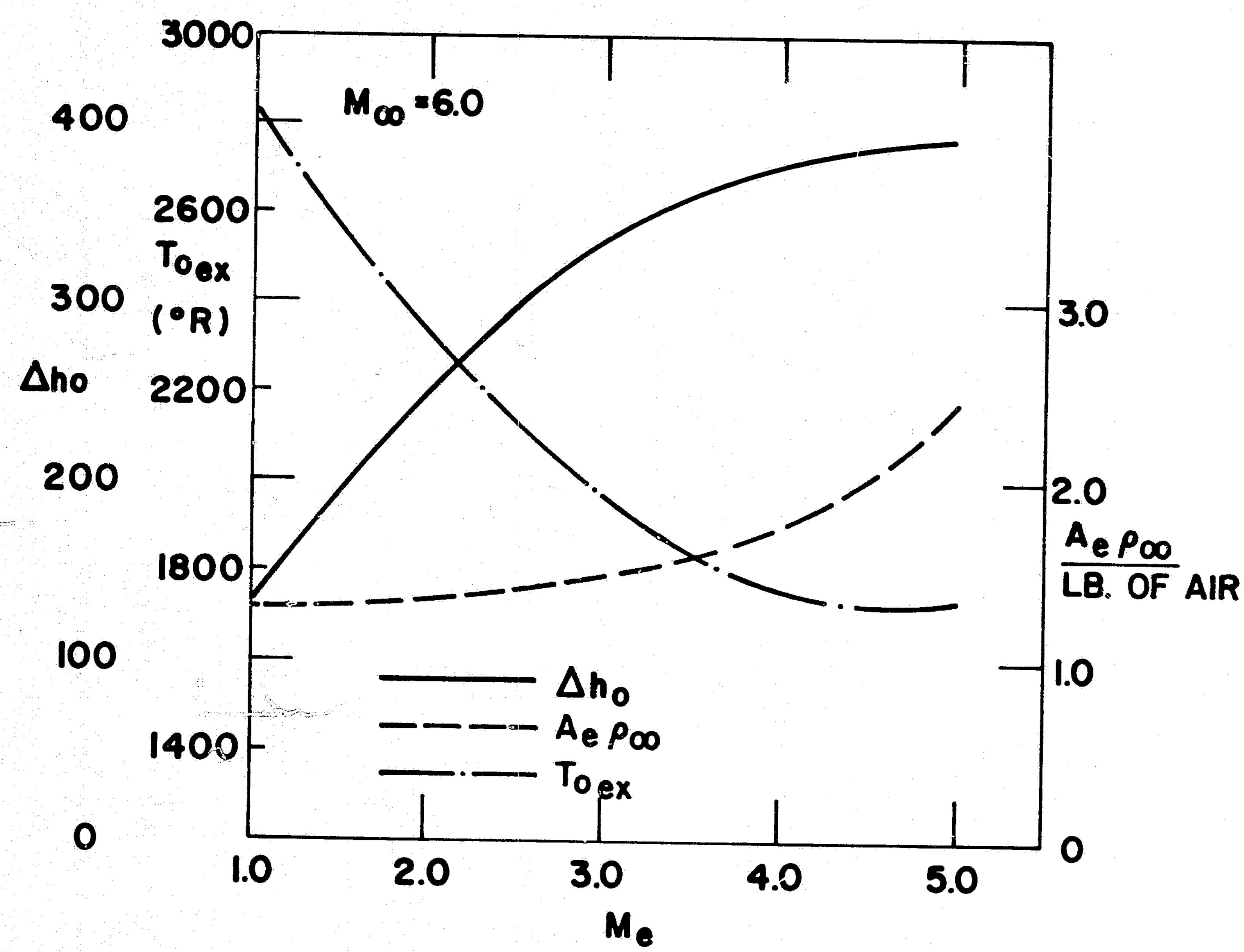

Fig. 2 Effect of entrance Mach number on performance and required contraction

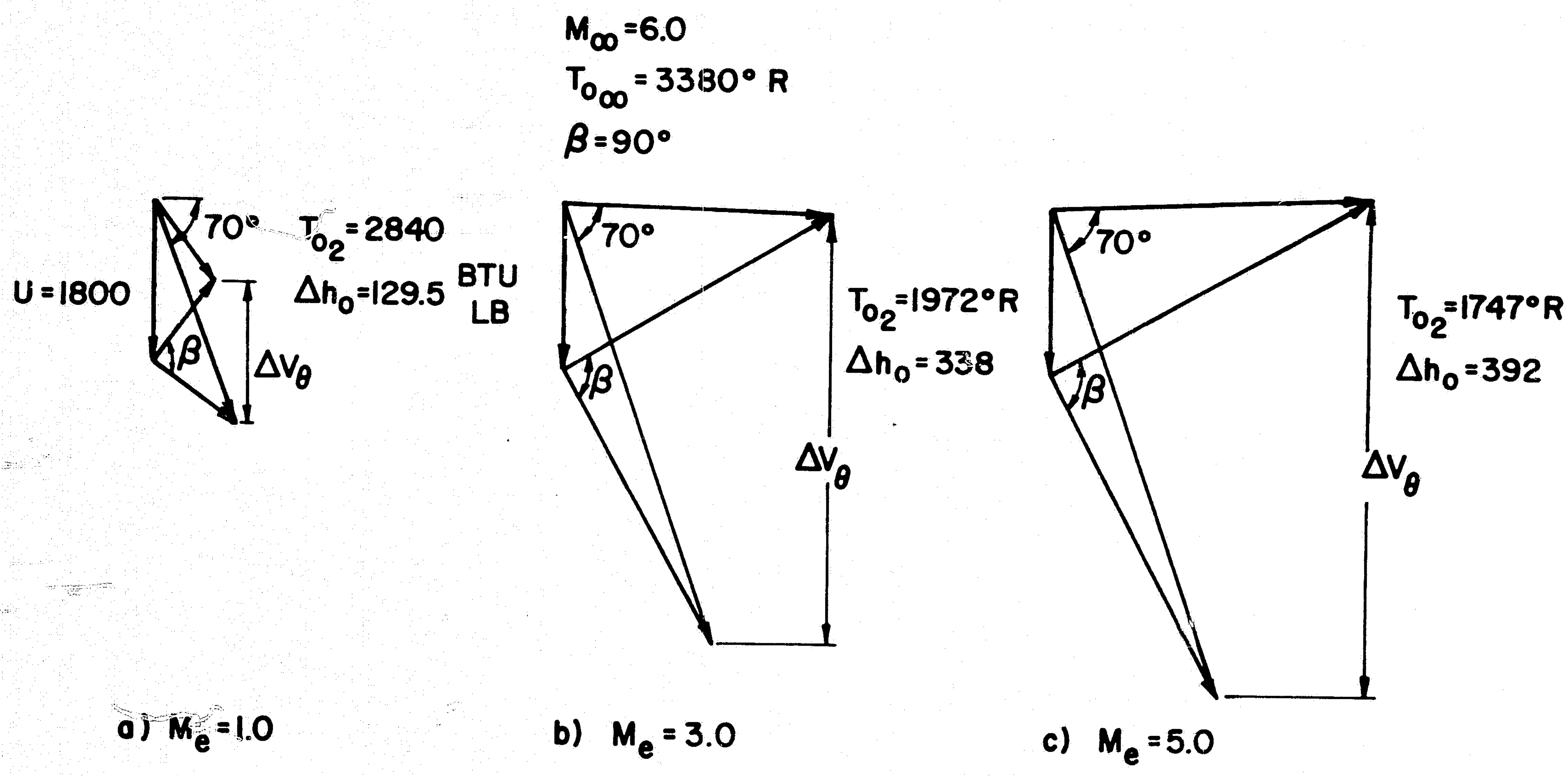

Fig. 3 Sample turbine velocity diagrams showing effect of entrance Mach number

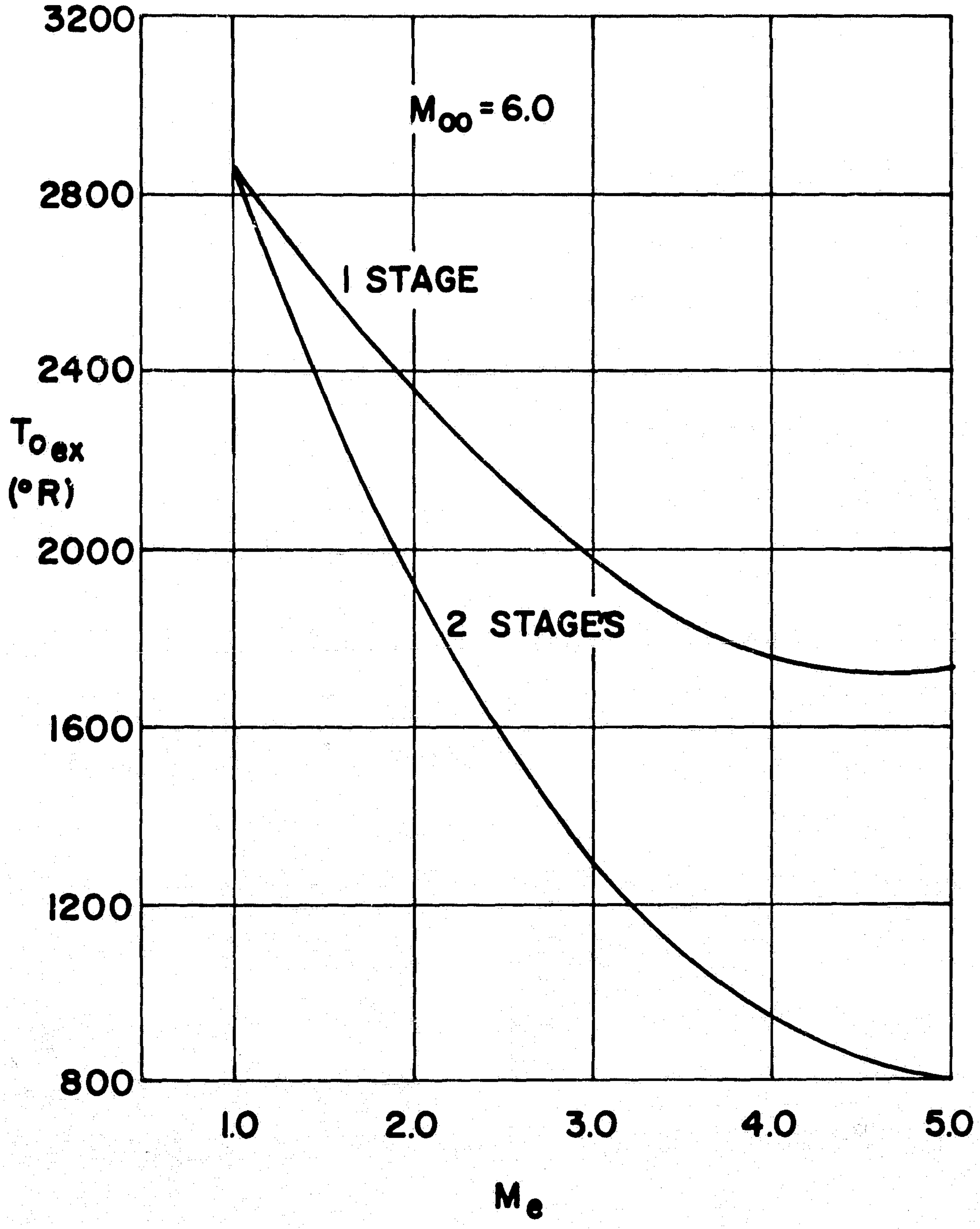

Fig. 4 Turbine exit temperature for one and two stage turbines - Effect of entrance Mach number

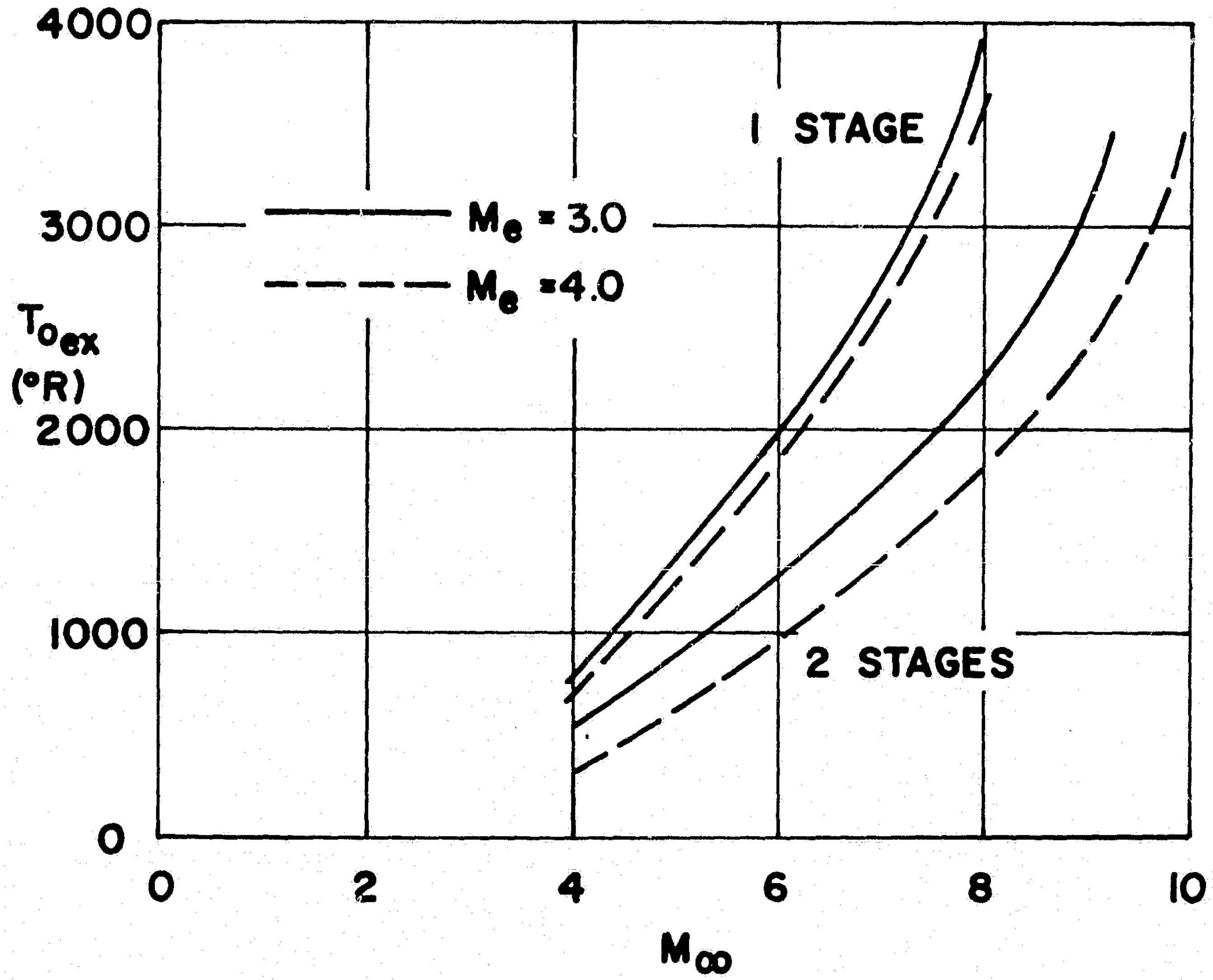

Fig. 5 Turbine exit temperature for one and two stage turbines - Effect of free stream Mach number

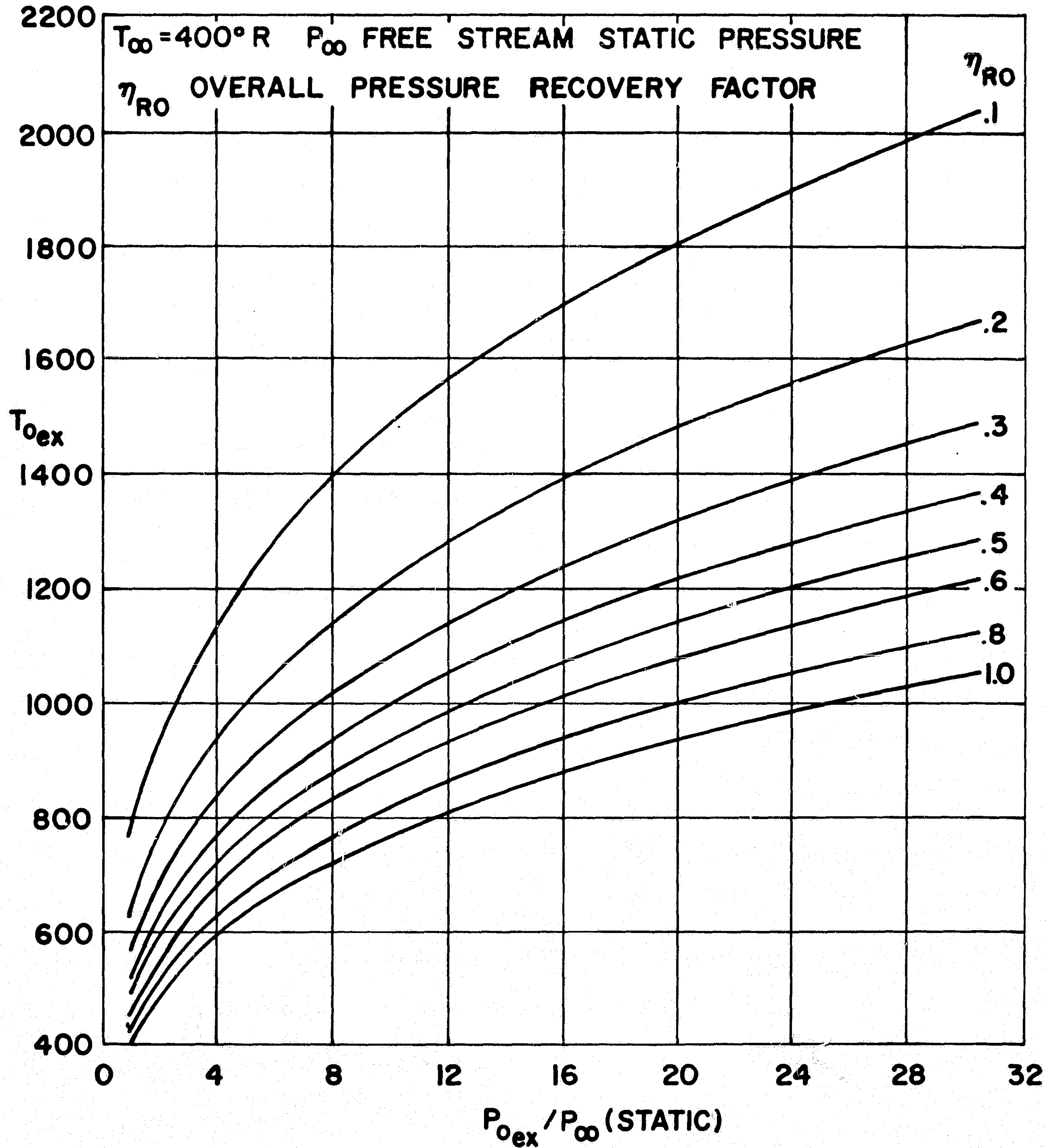

Fig. 6 Turbine discharge stagnation temperature as a function of stagnation pressure discharge

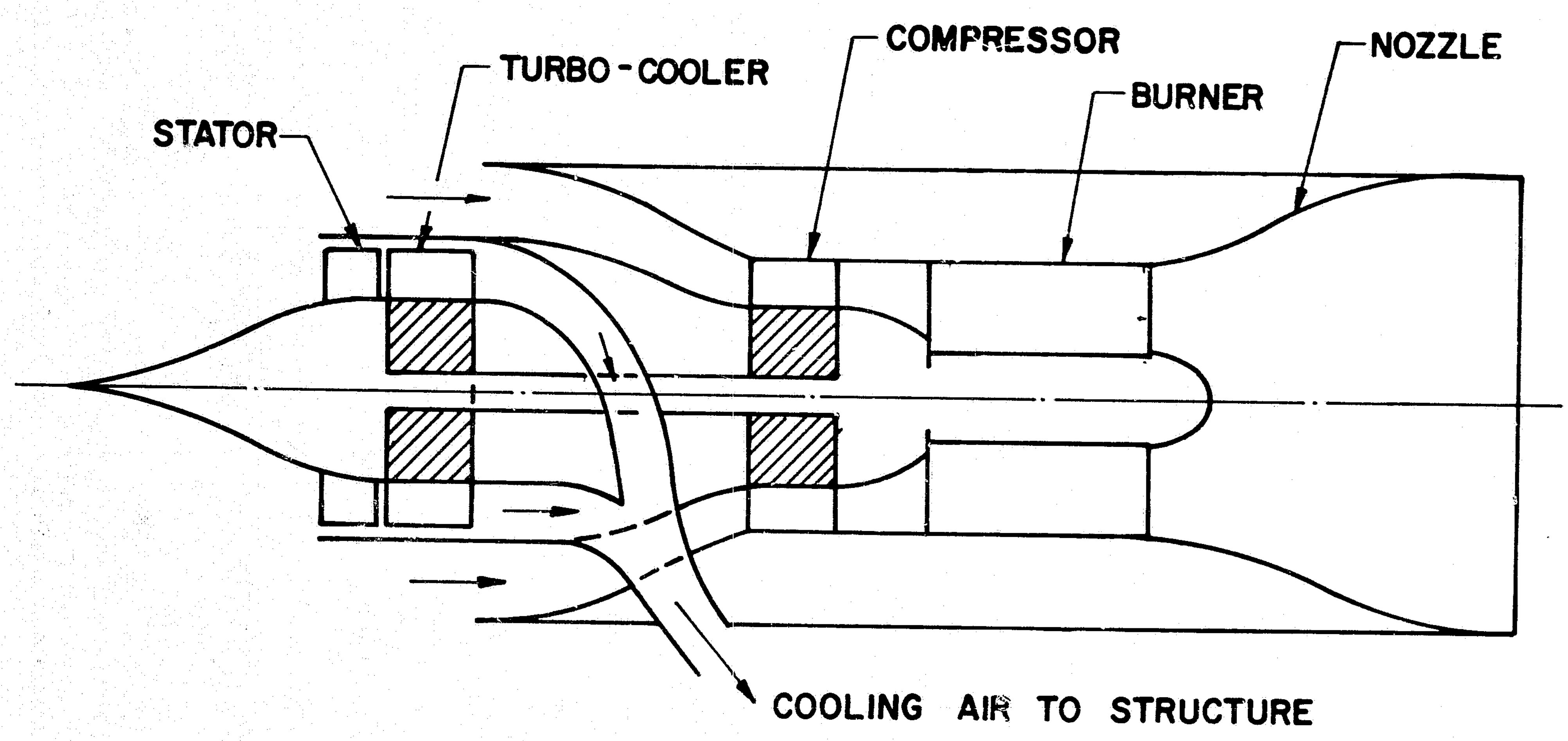

Fig. 7 Possible turbo-cooler arrangement (Ref. 1)

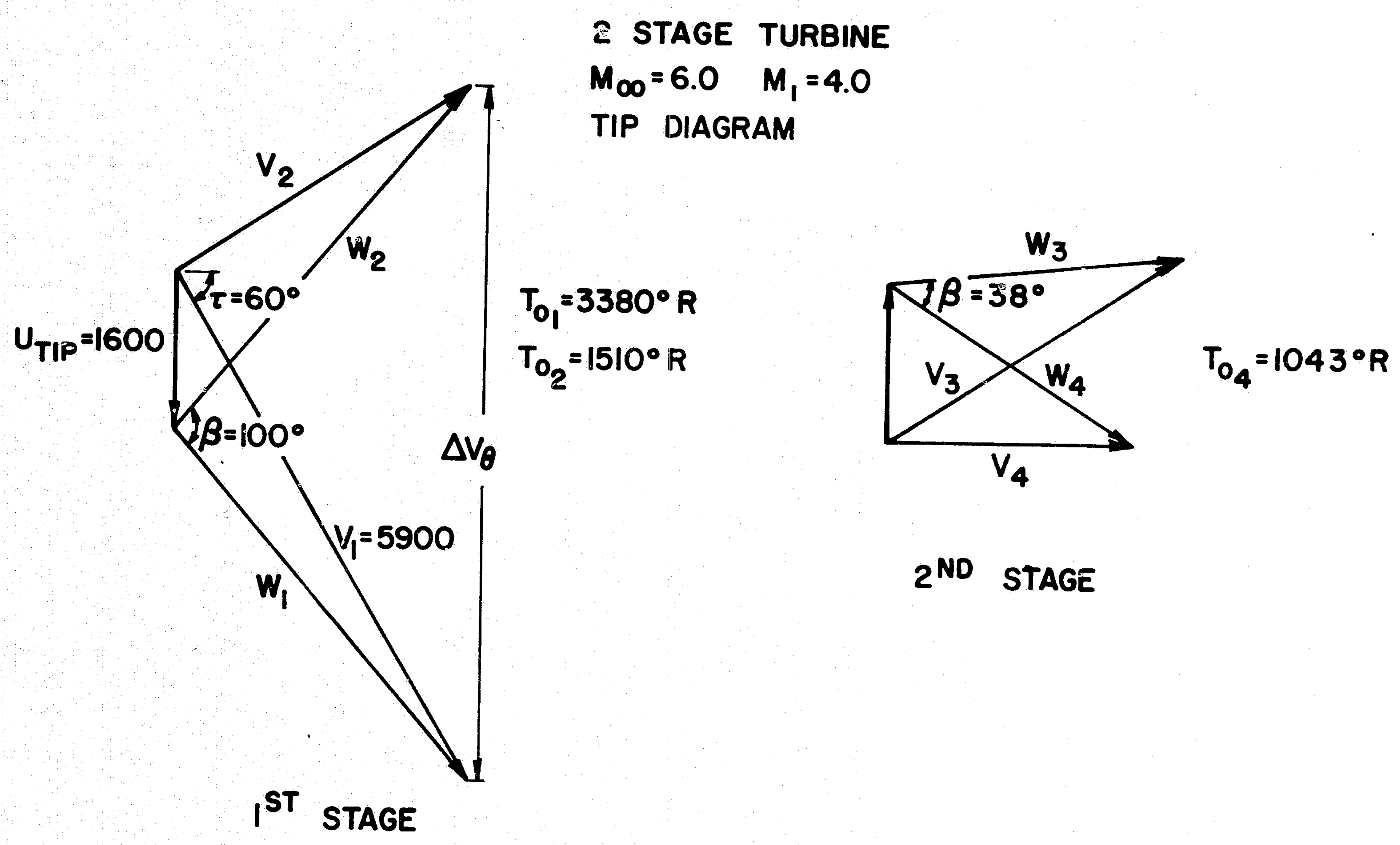

Fig. 8a Typical two stage turbine velocity diagram

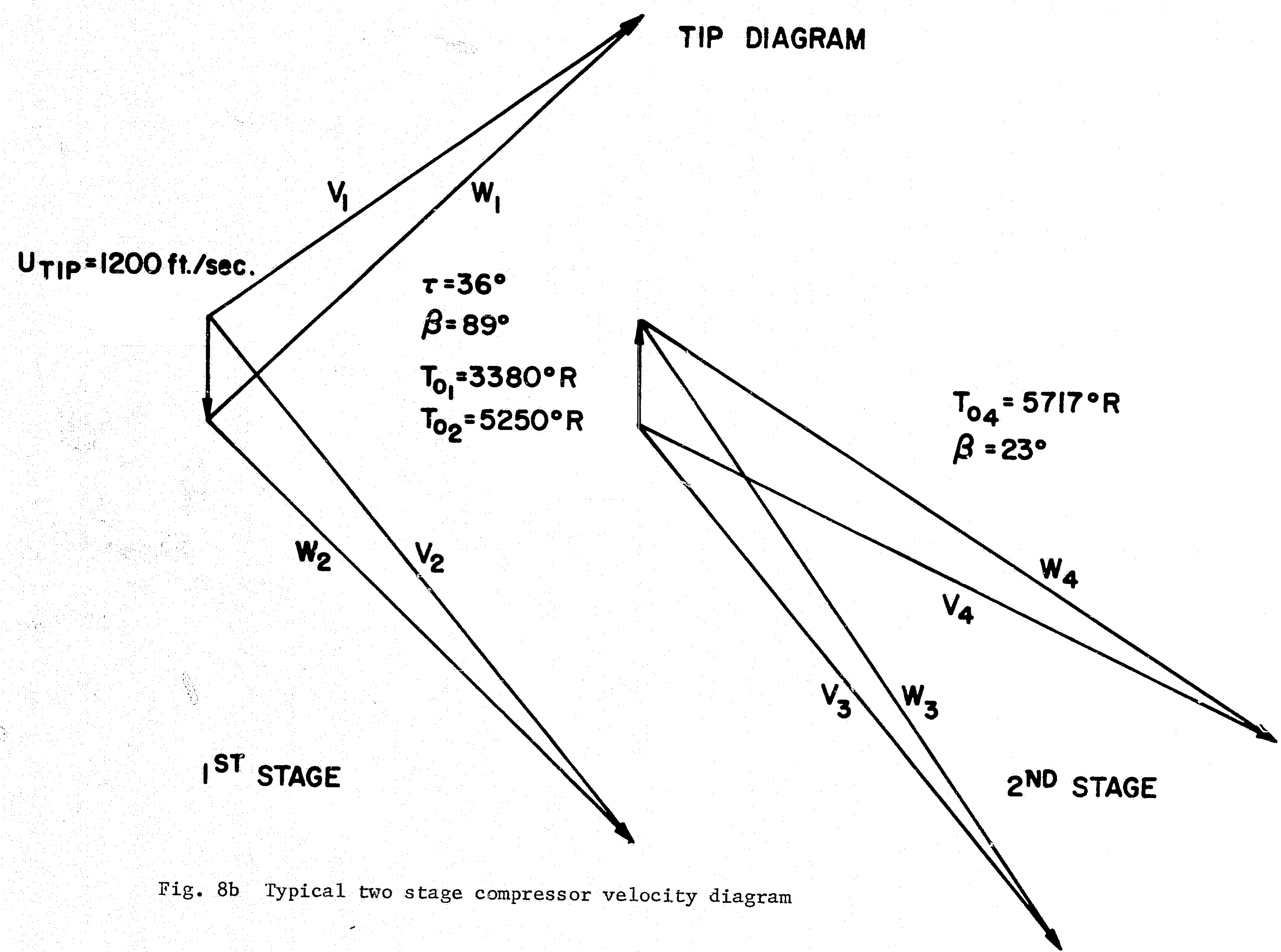

Fig. 8b Typical two stage compressor velocity diagram

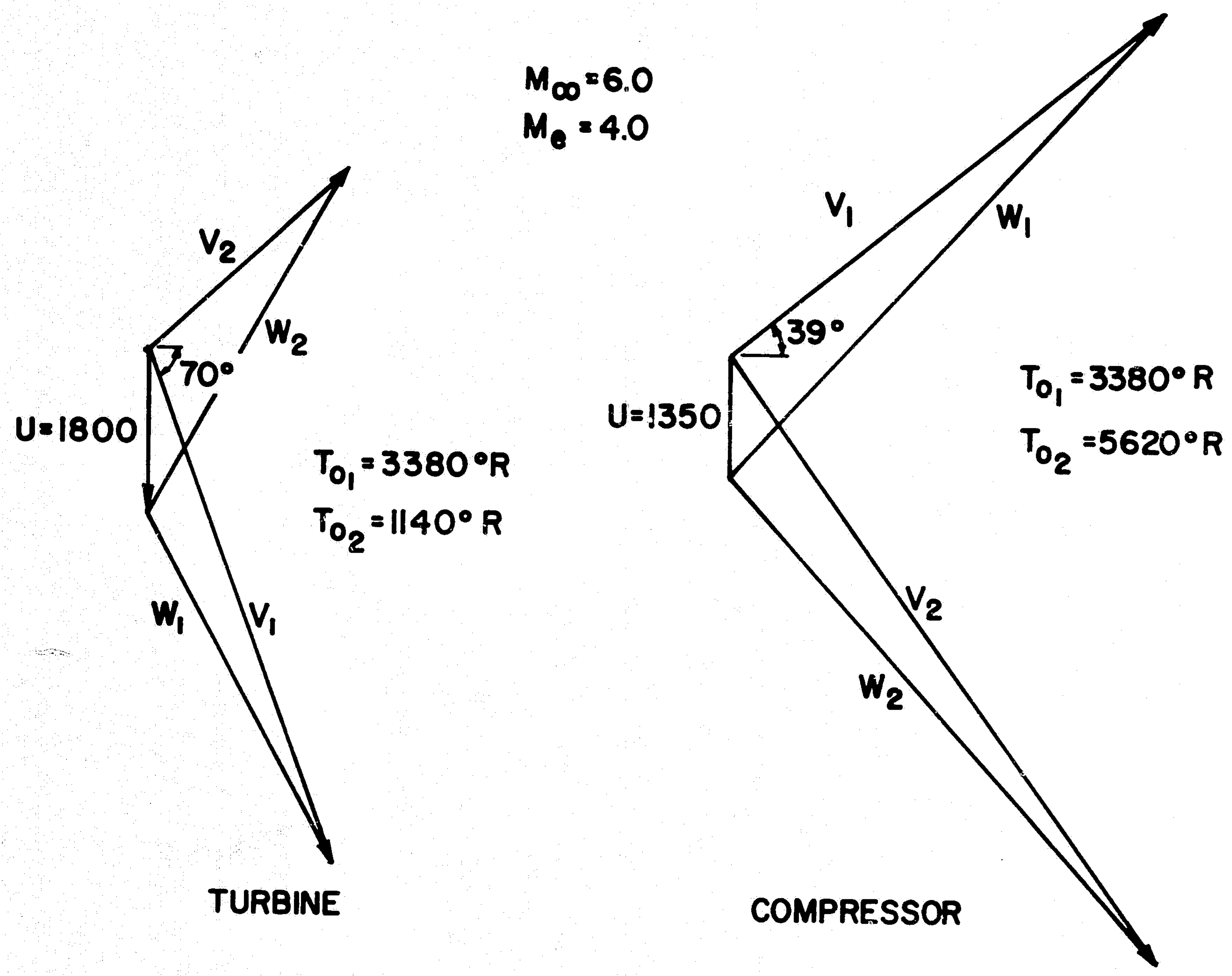

Fig. 9 Typical one stage turbine and compressor velocity diagrams

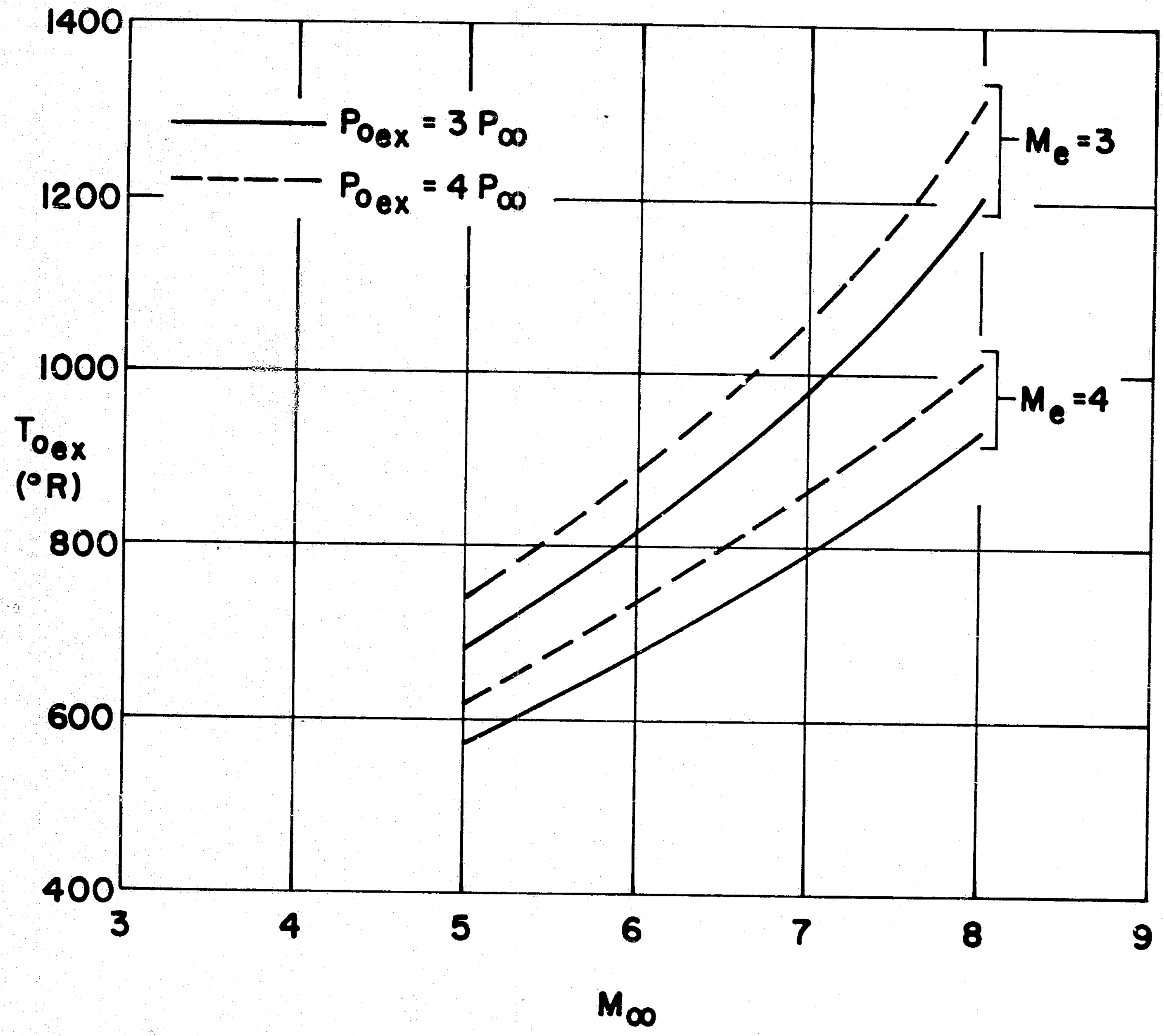

Fig. 10a Effects of free stream Mach number, entrance Mach number, exit pressure on turbine exit temperature

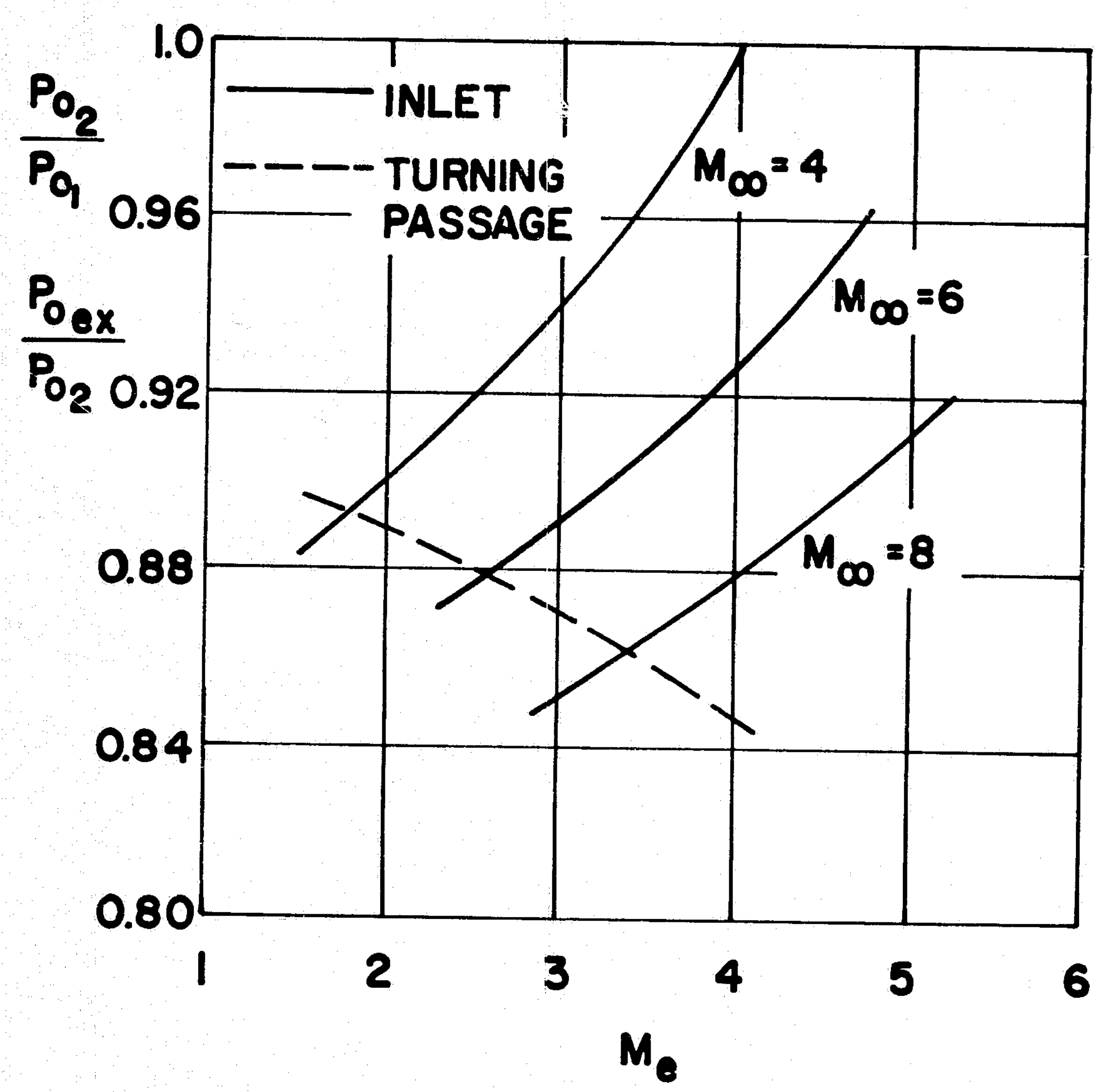

Fig. 10b Typical pressure recoveries currently available

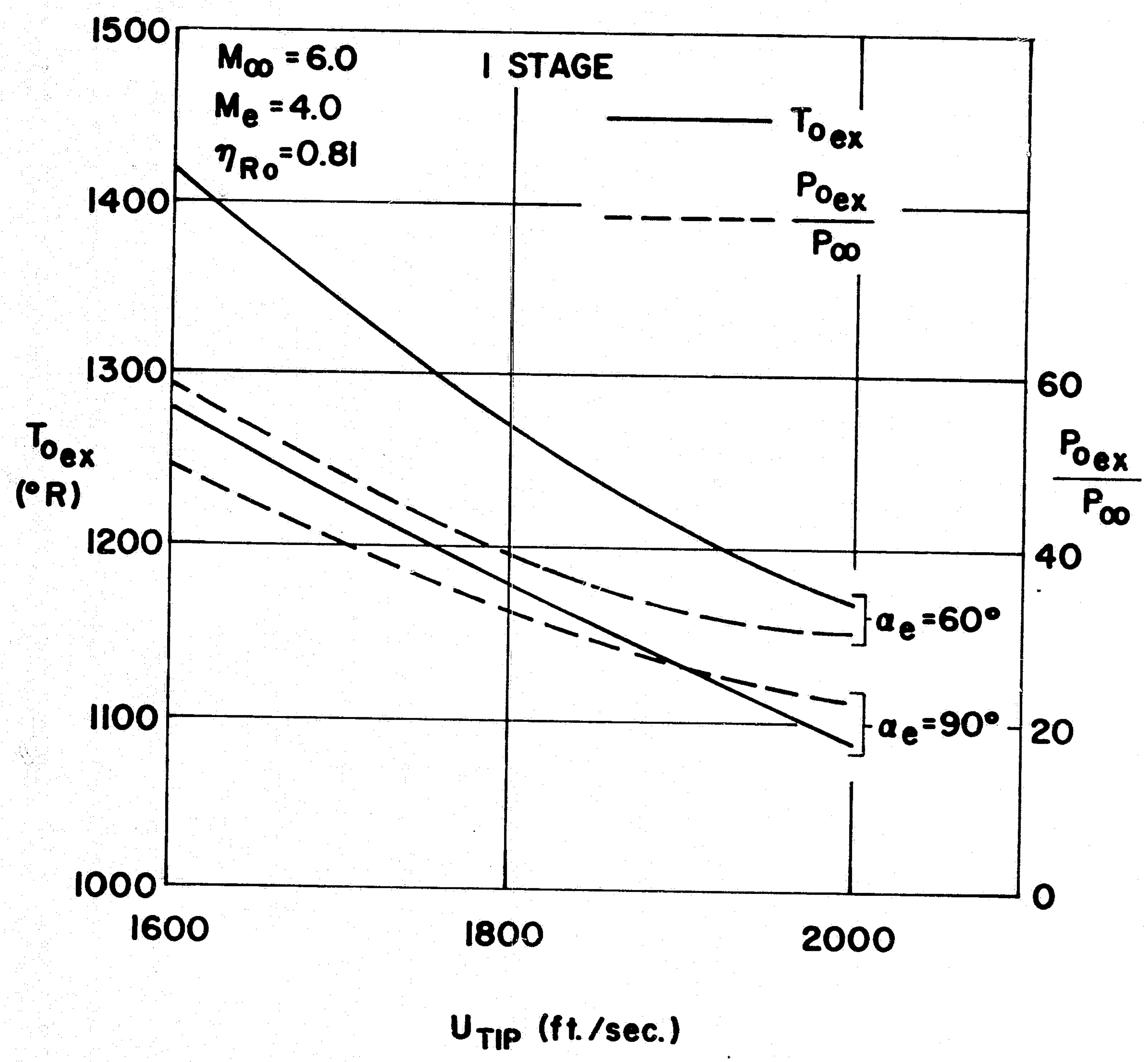

Fig. 11a Effect of tip speed on turbine exit temperature - single stage

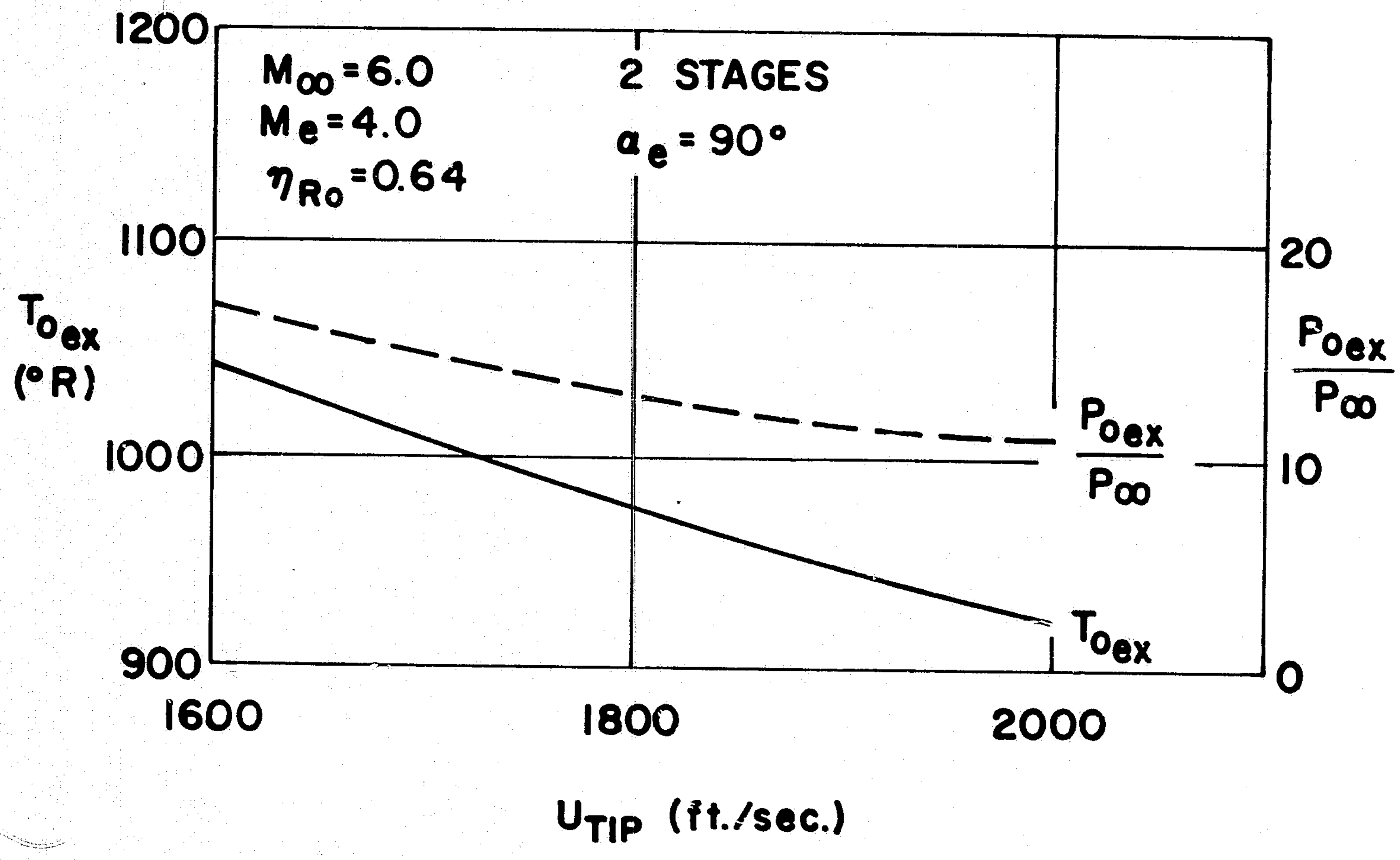

Fig. 11b Effect of tip speed on turbine exit temperature - two stages

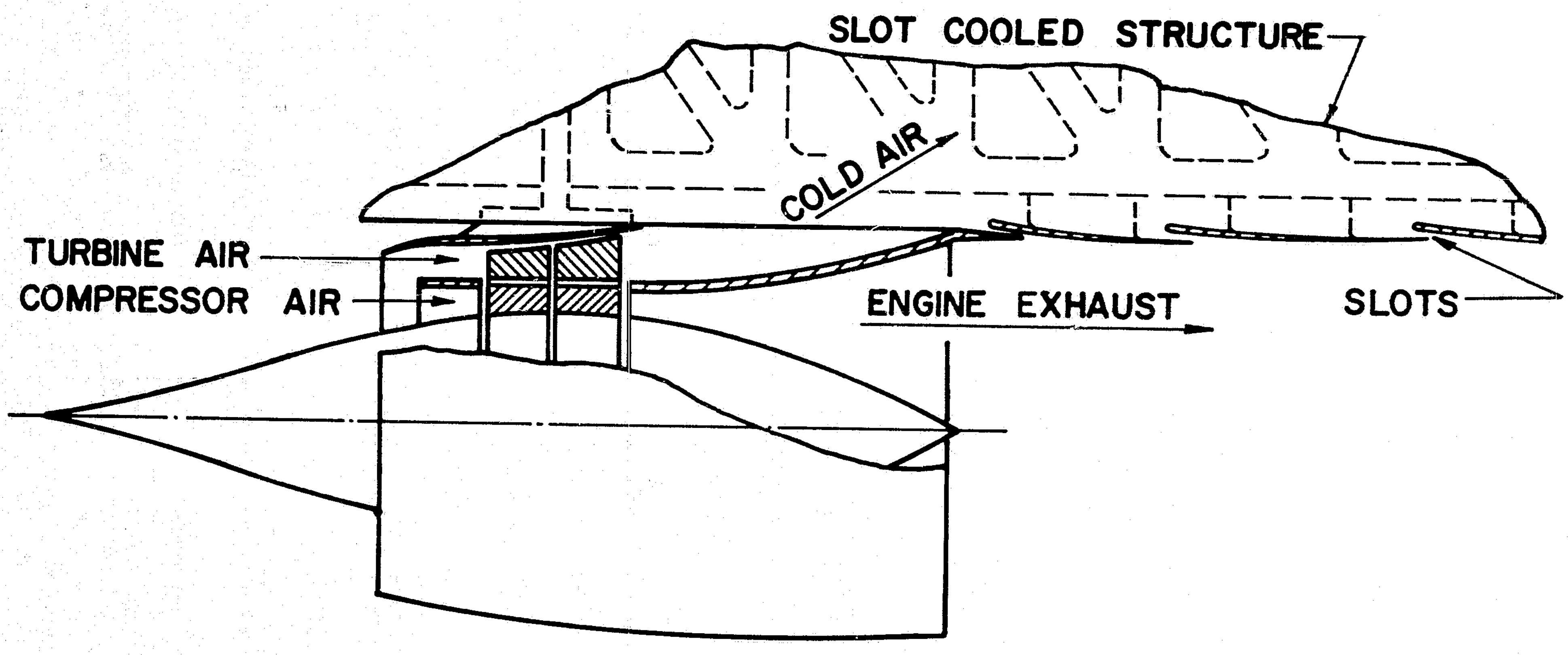

Fig. 12 Proposed turbocooler design

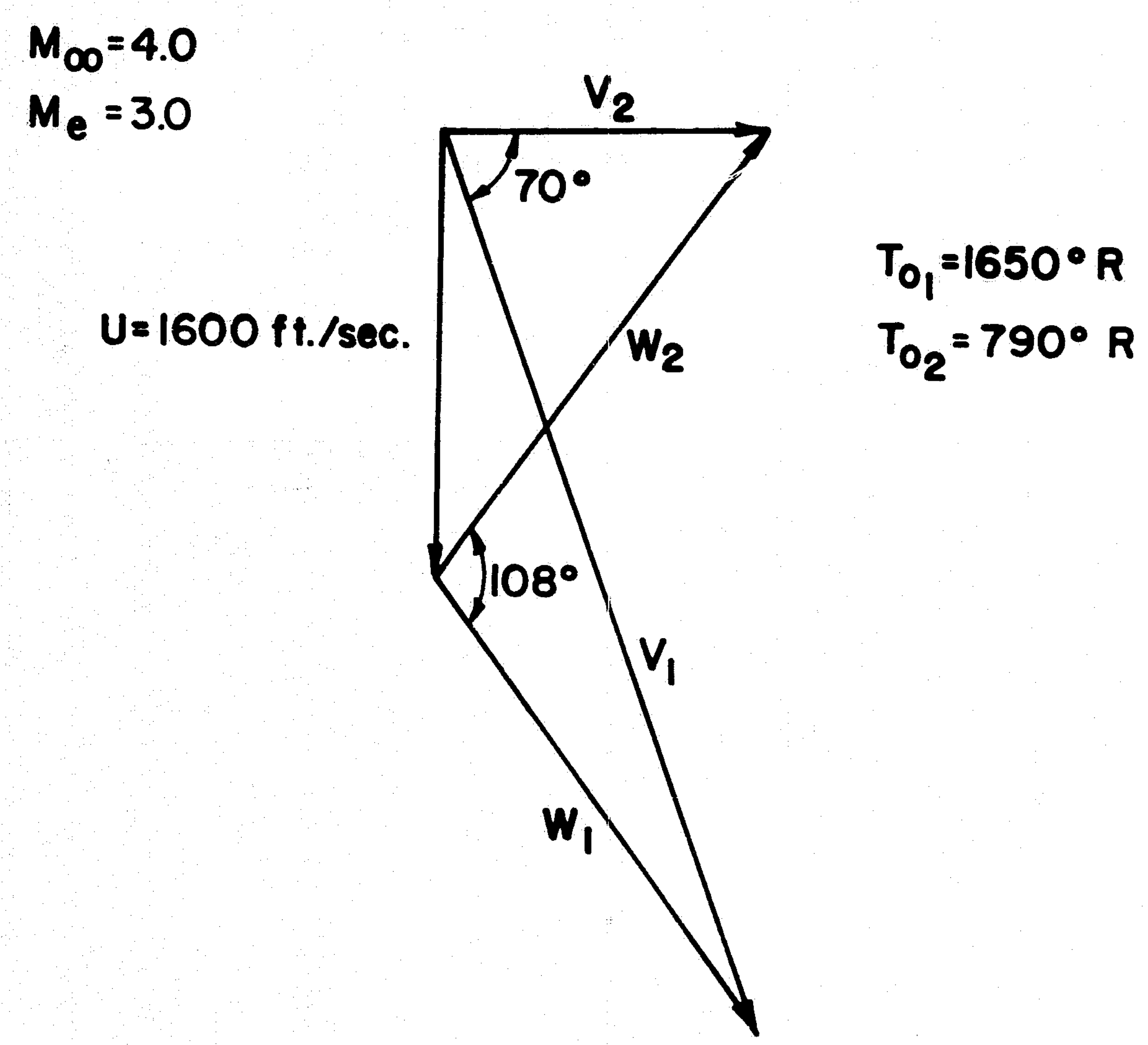

Fig. 13 Typical velocity diagram for $M_{\infty} = 4$ application

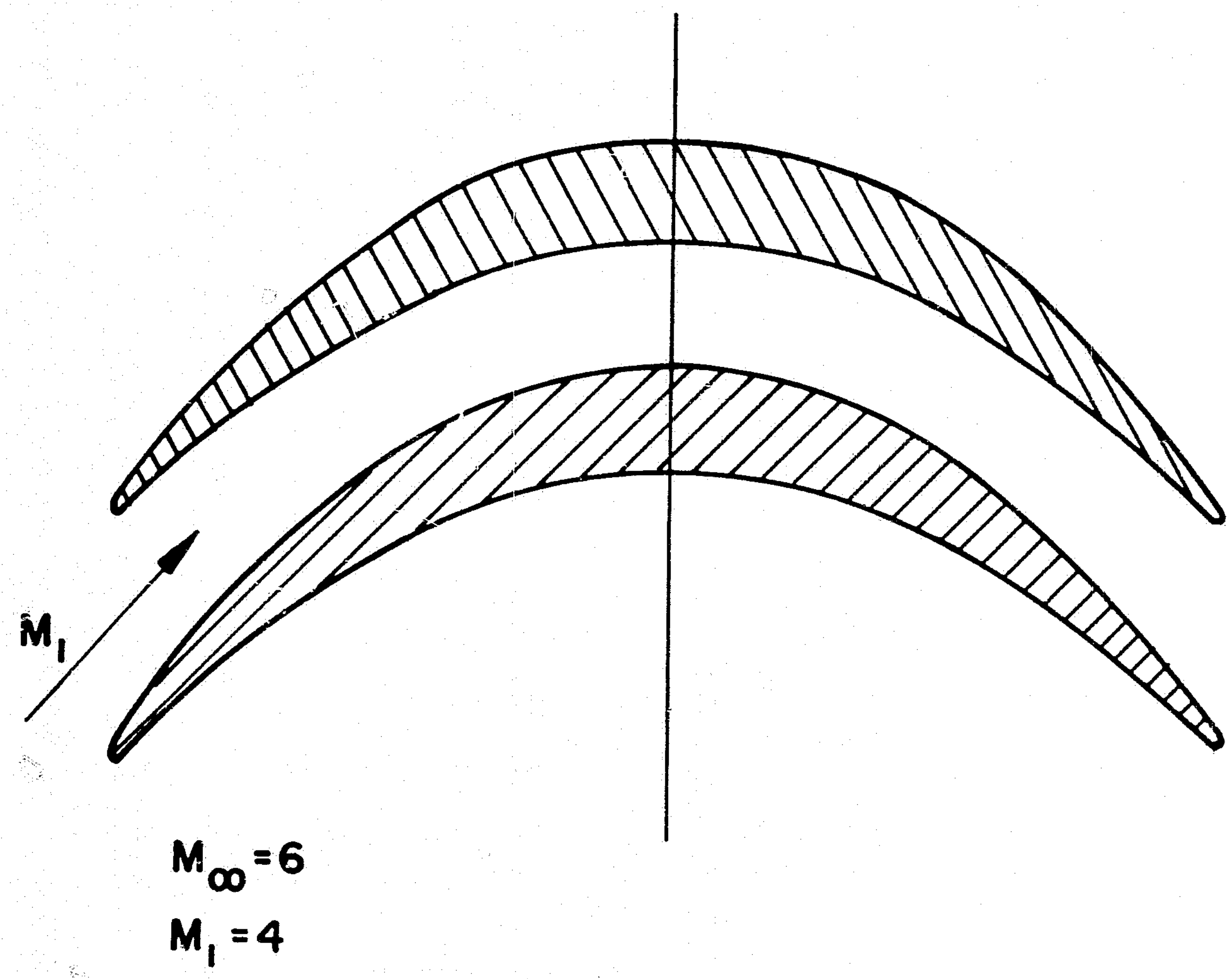

Fig. 14 Blade turning passage

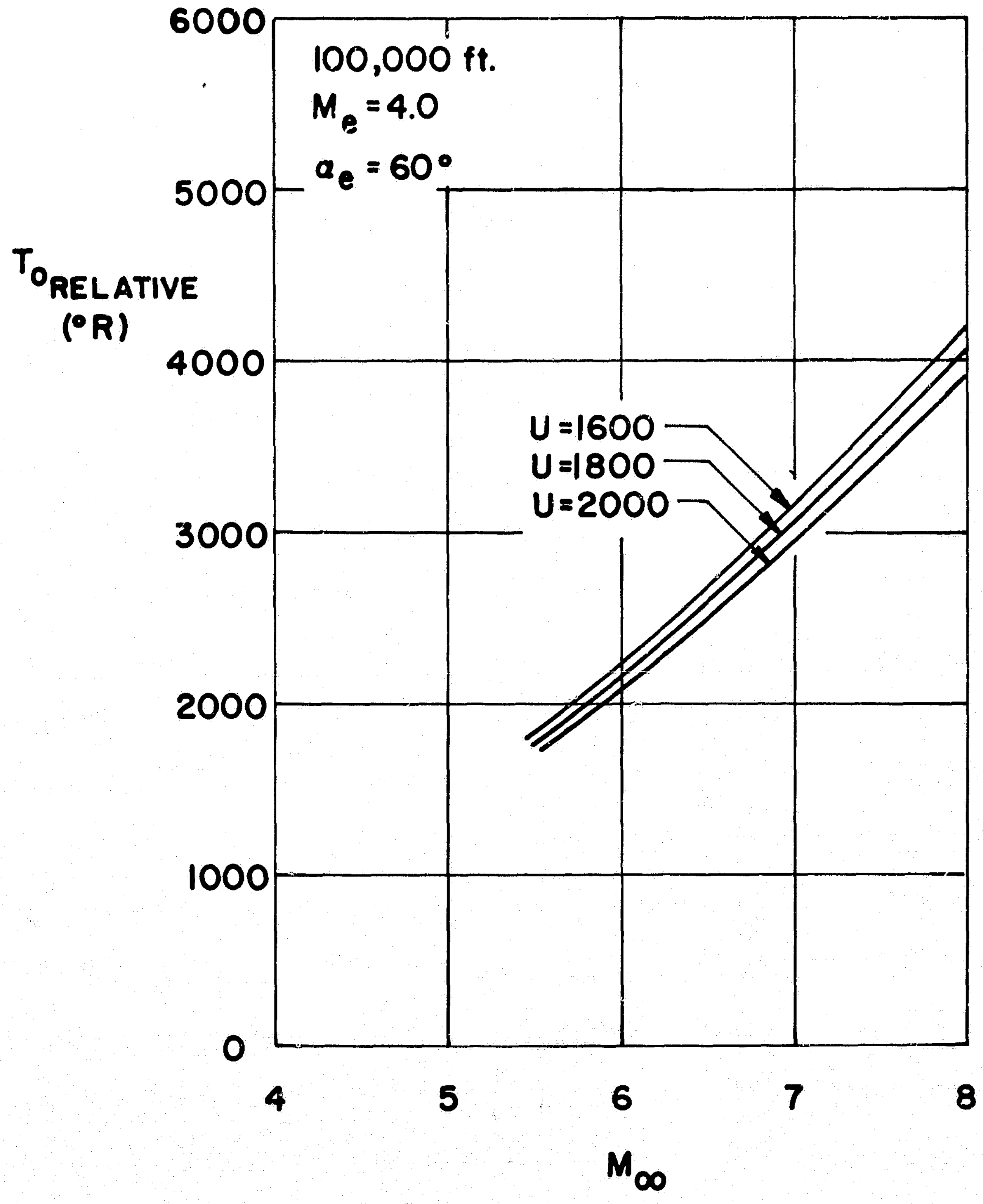

Fig. 15 Relative stagnation temperature-effect of tip speed

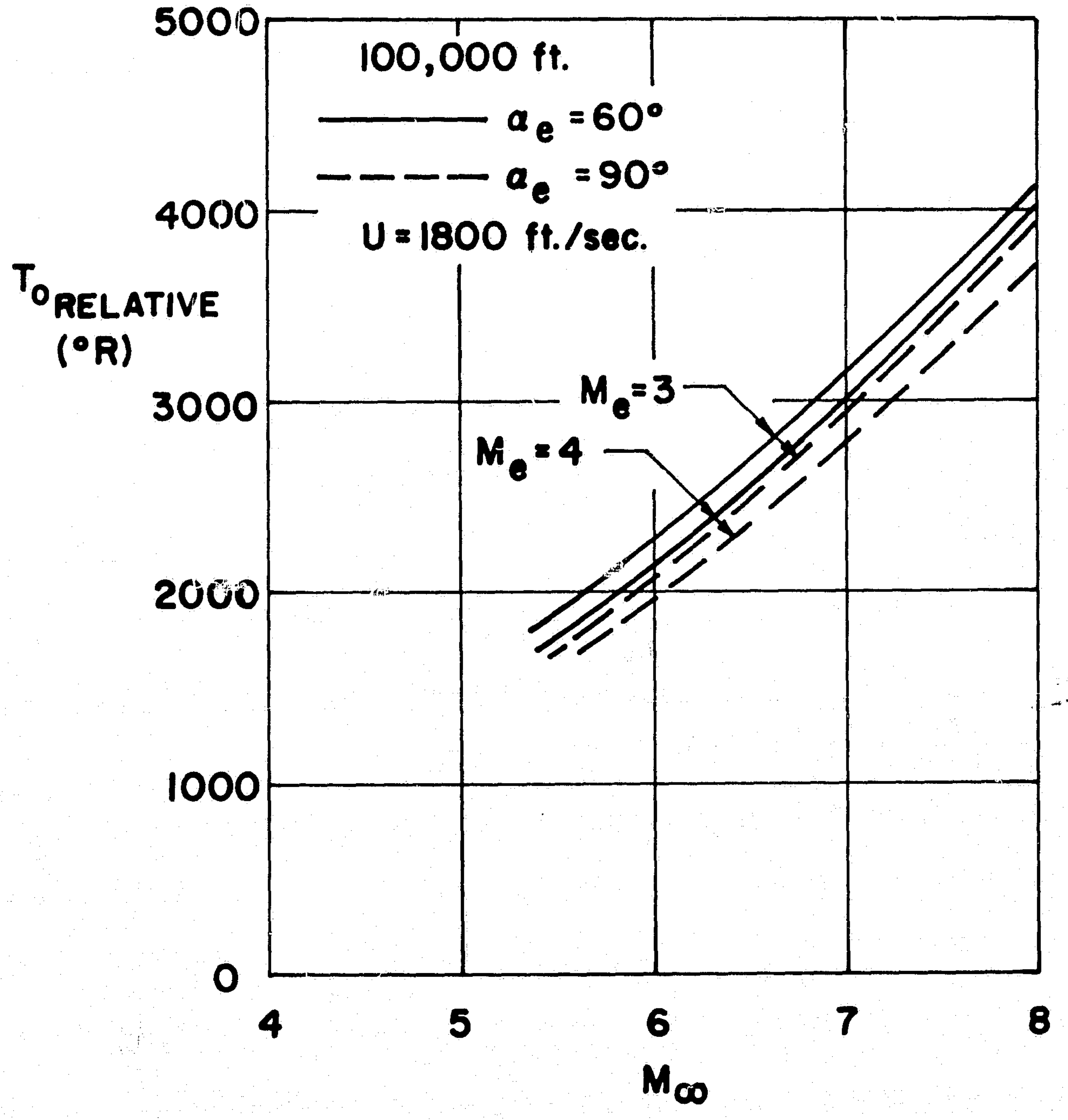

Fig. 16 Relative stagnation temperature-effect of upstream turning

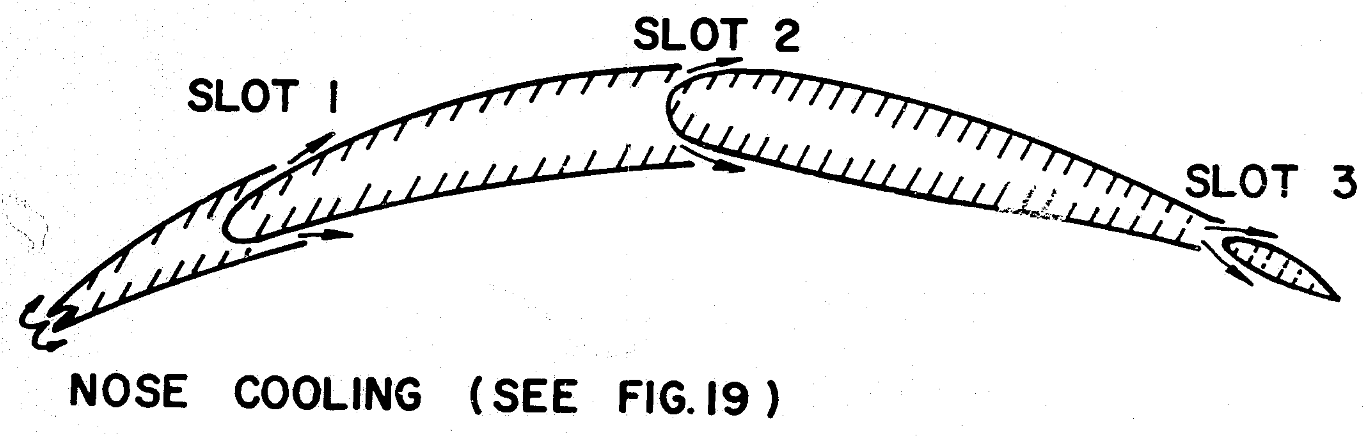

Fig. 17 Schematic of slot-cooled blades

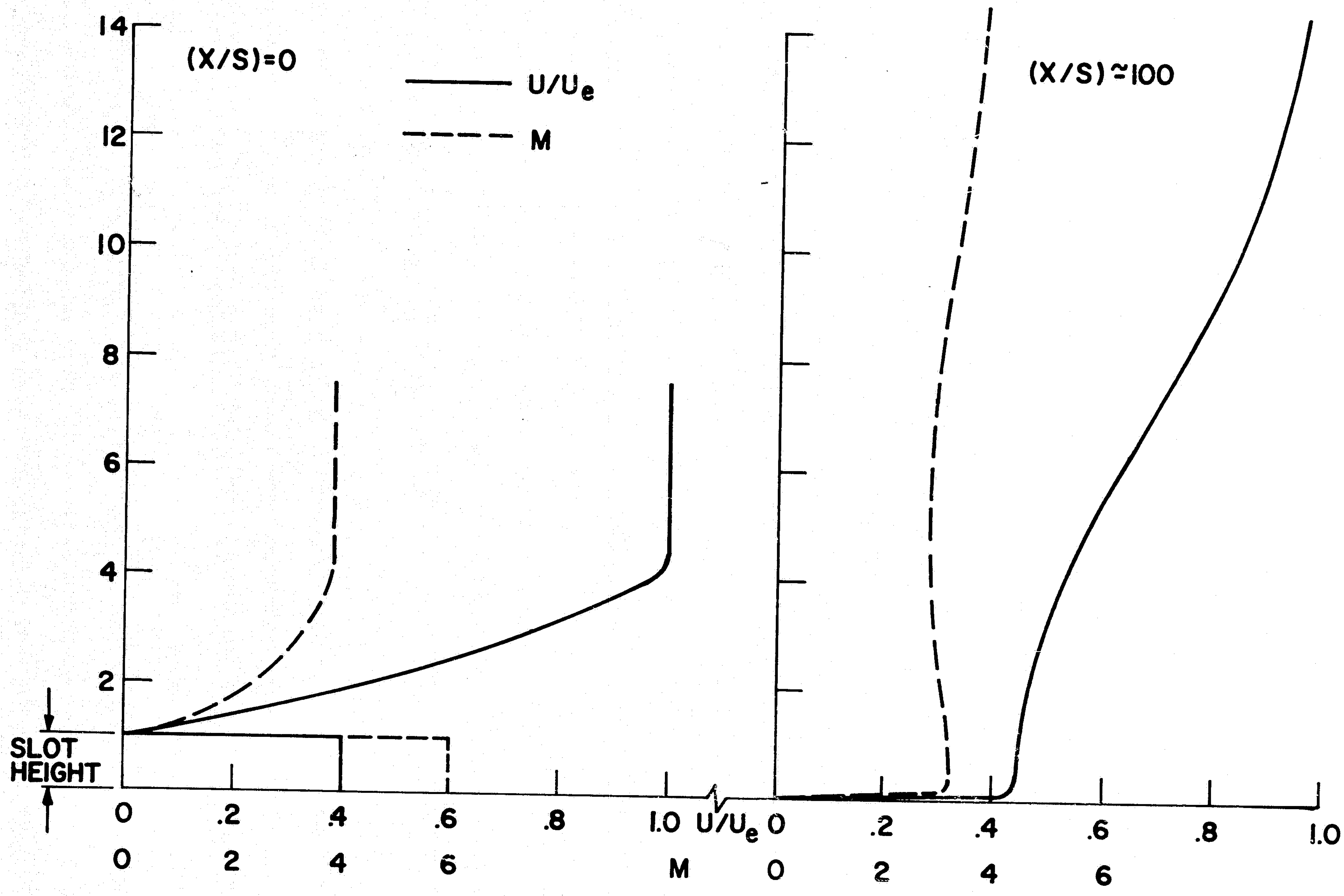

Fig. 18 Typical profiles for energized turbulent boundary layer

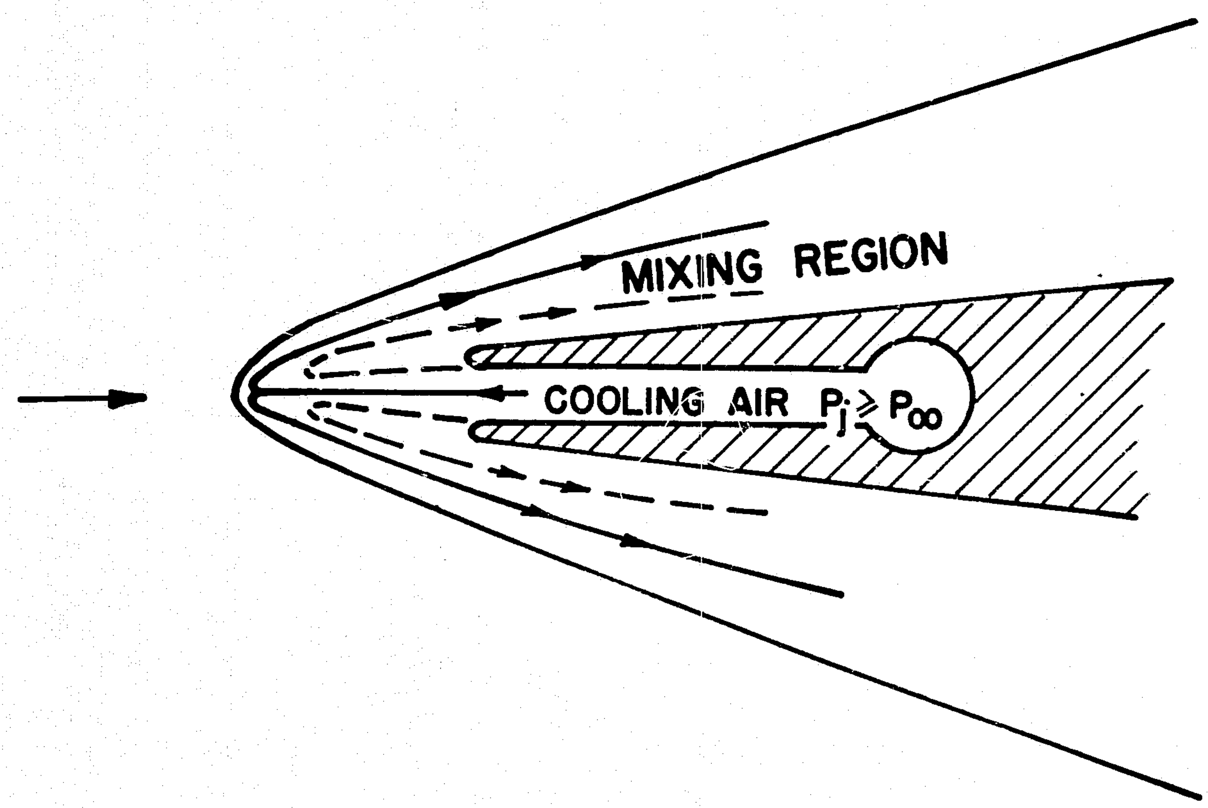

Fig. 19 Schematic of nose cooling

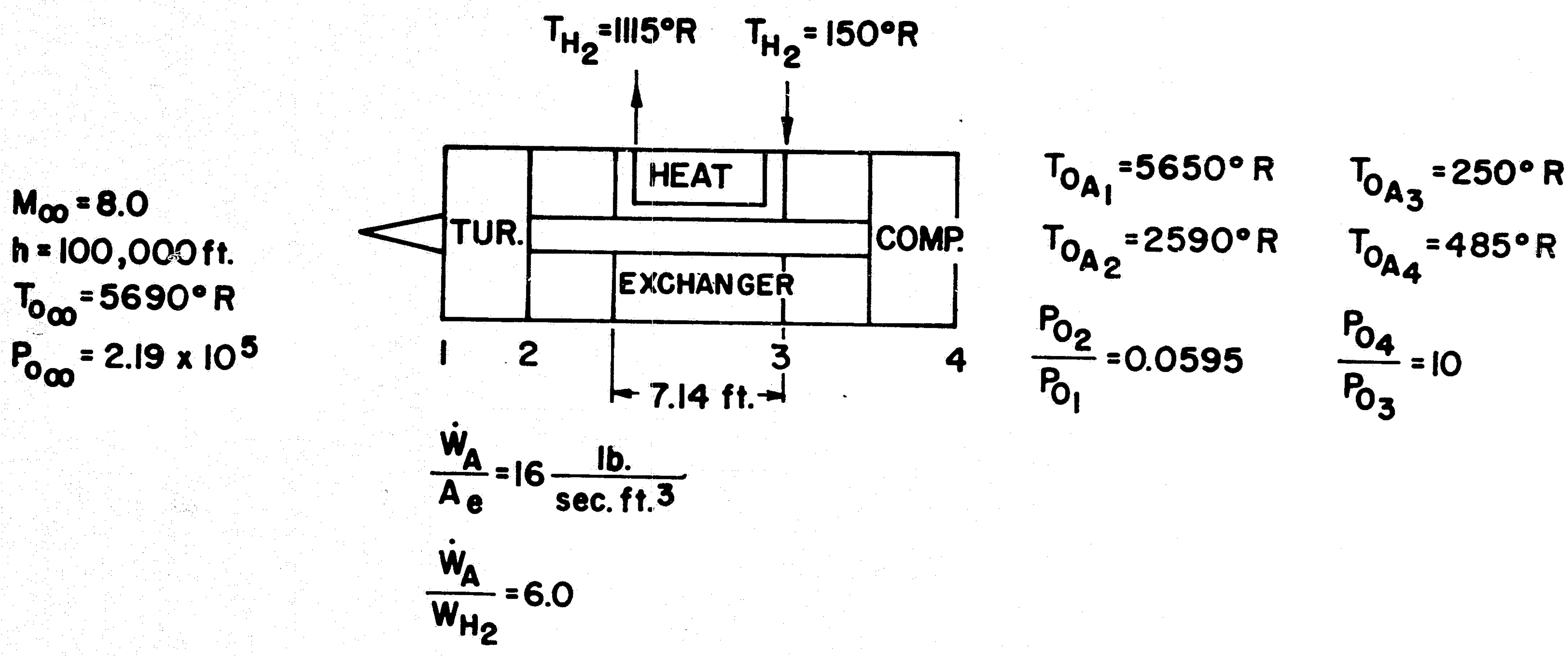

Fig. 20 Schematic drawing of heat exchanger system for cooling high pressure surfaces

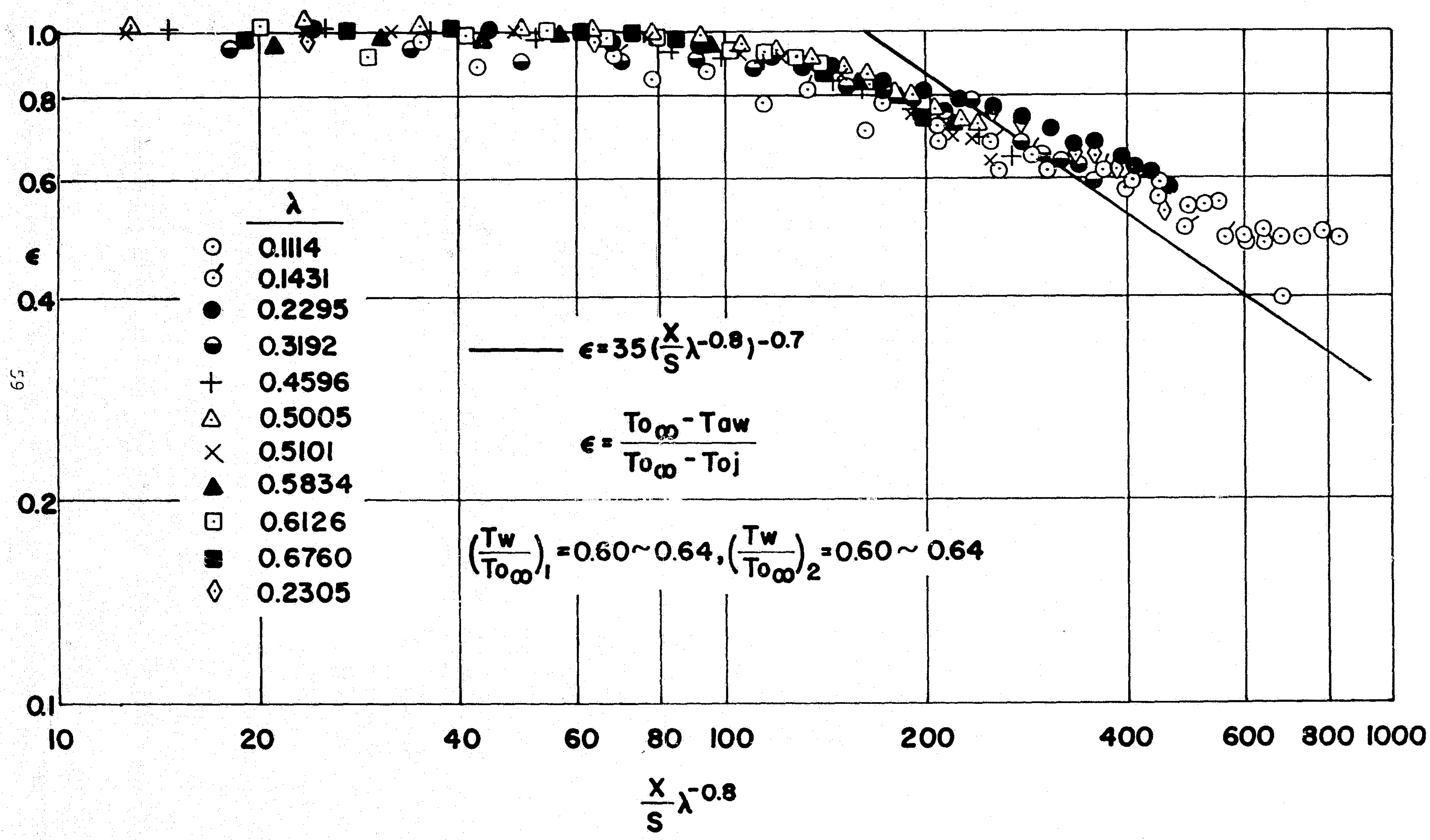

Fig. 21 Film Cooling Effectiveness ε (Without Heating Model)

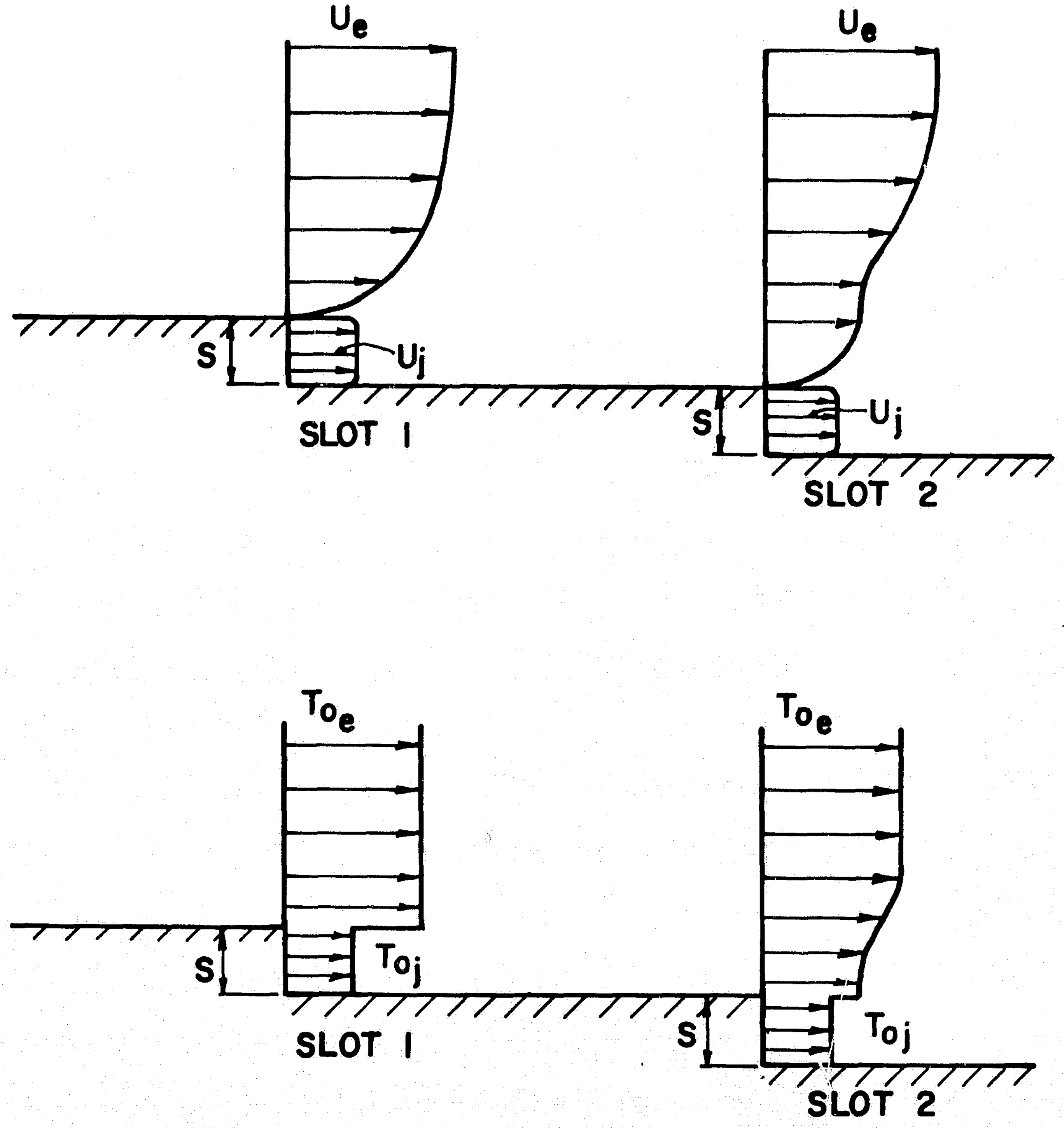

Fig. 22 Effect of slot injection on boundary profiles in effective region

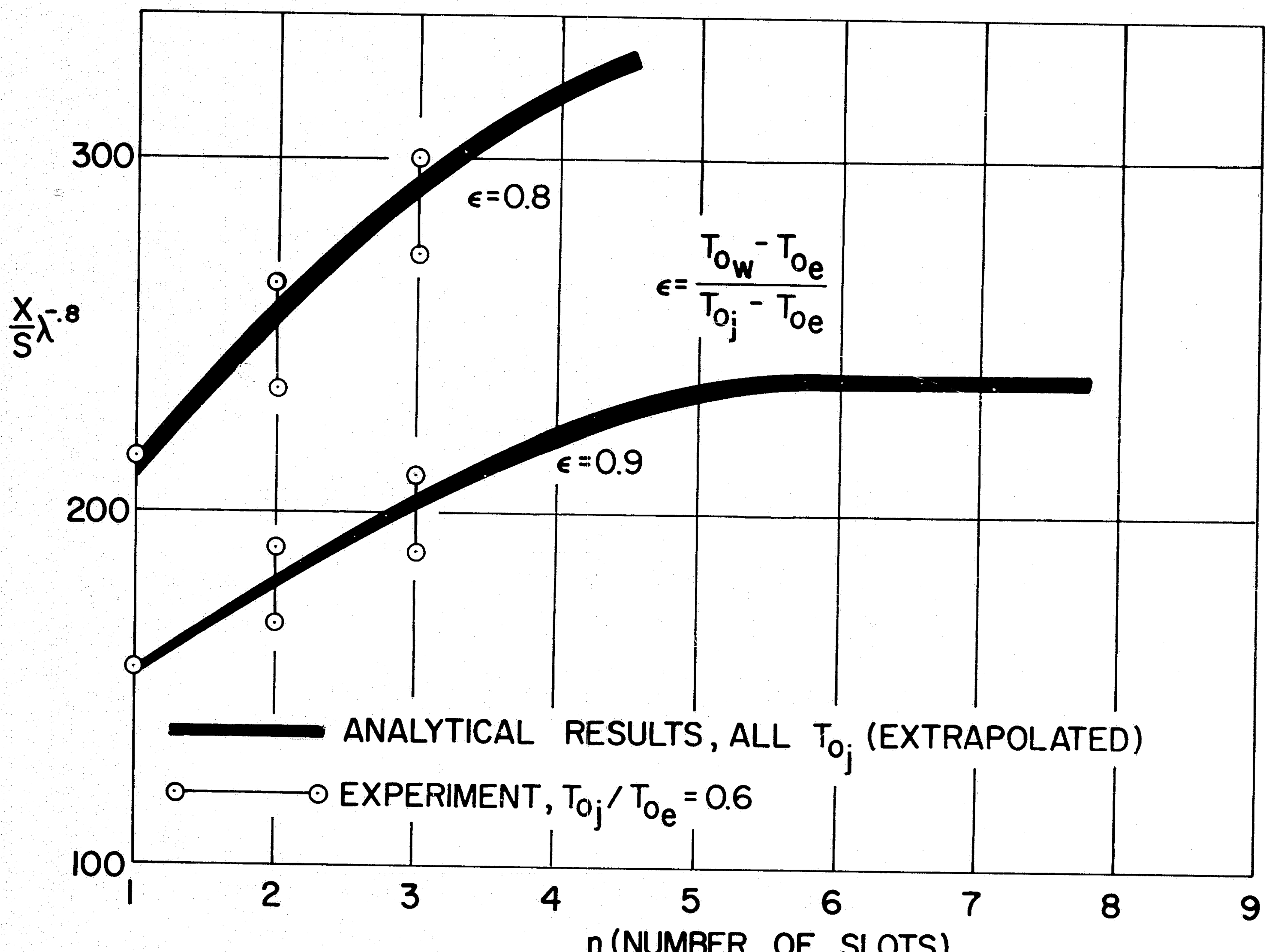

Fig. 23 INCREASING EFFECTIVENESS WITH NUMBER OF SLOTS

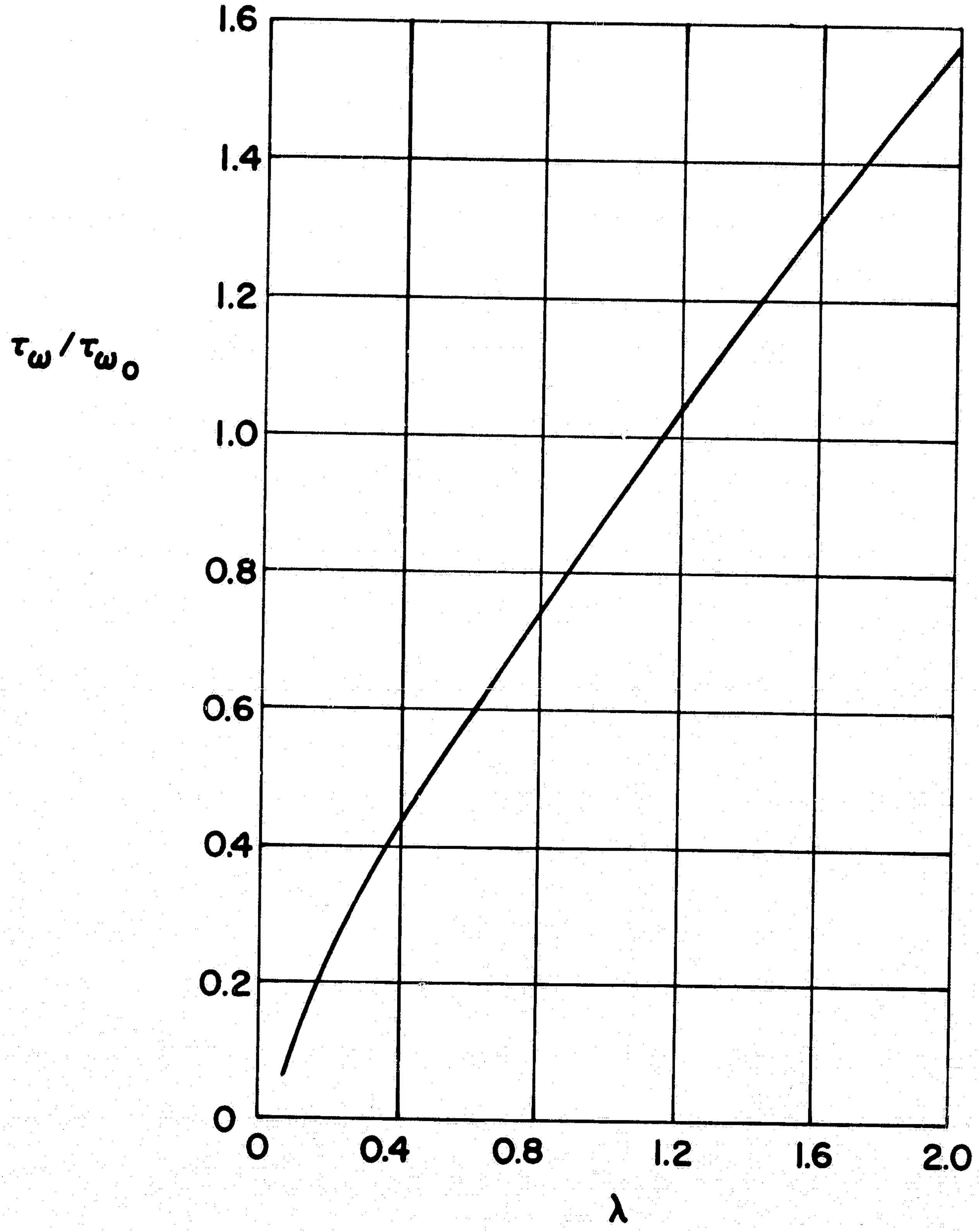

Fig. 24 Shear stress in effectiveness region as a function of injection rate

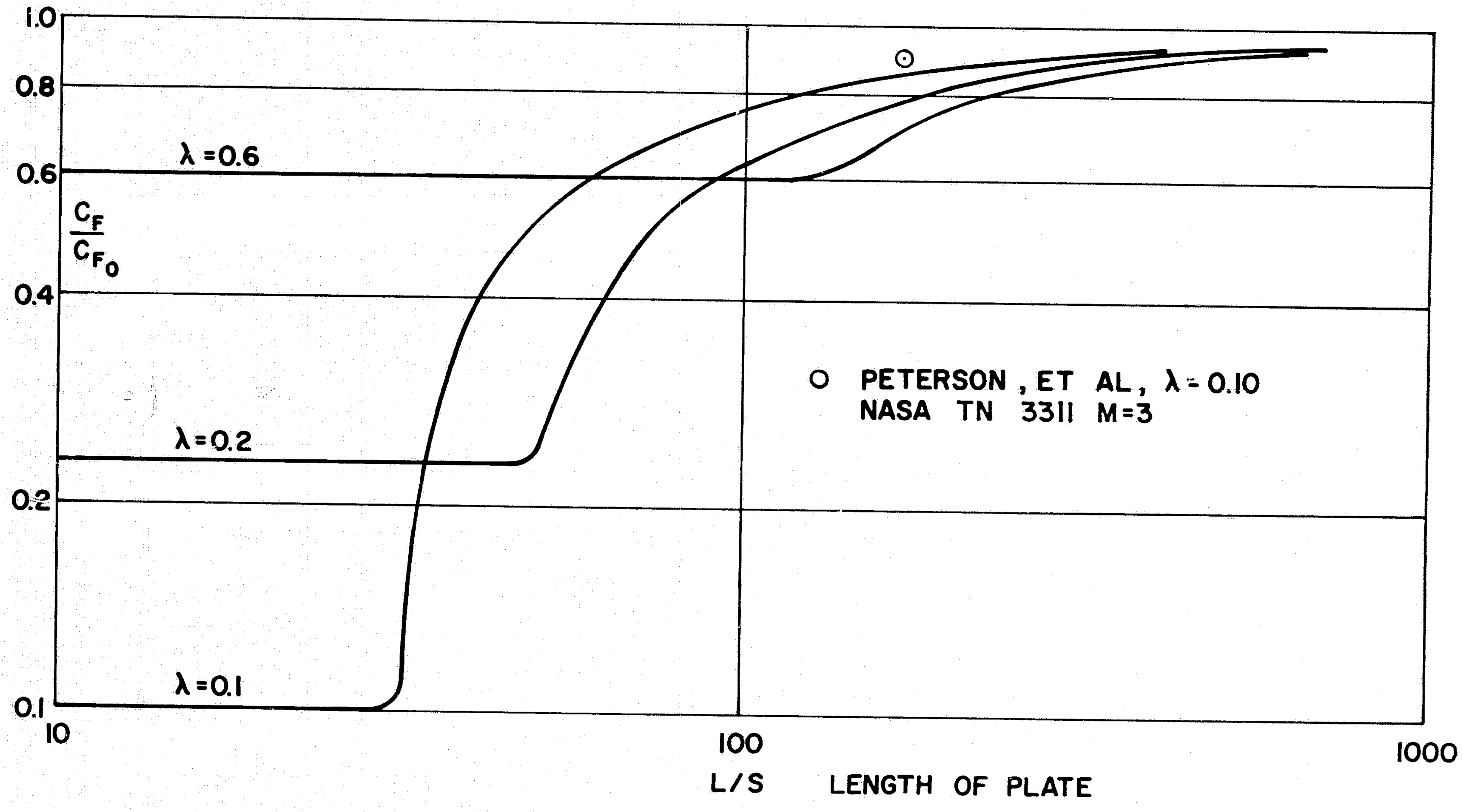

Fig. 25 Integrated skin friction distributions

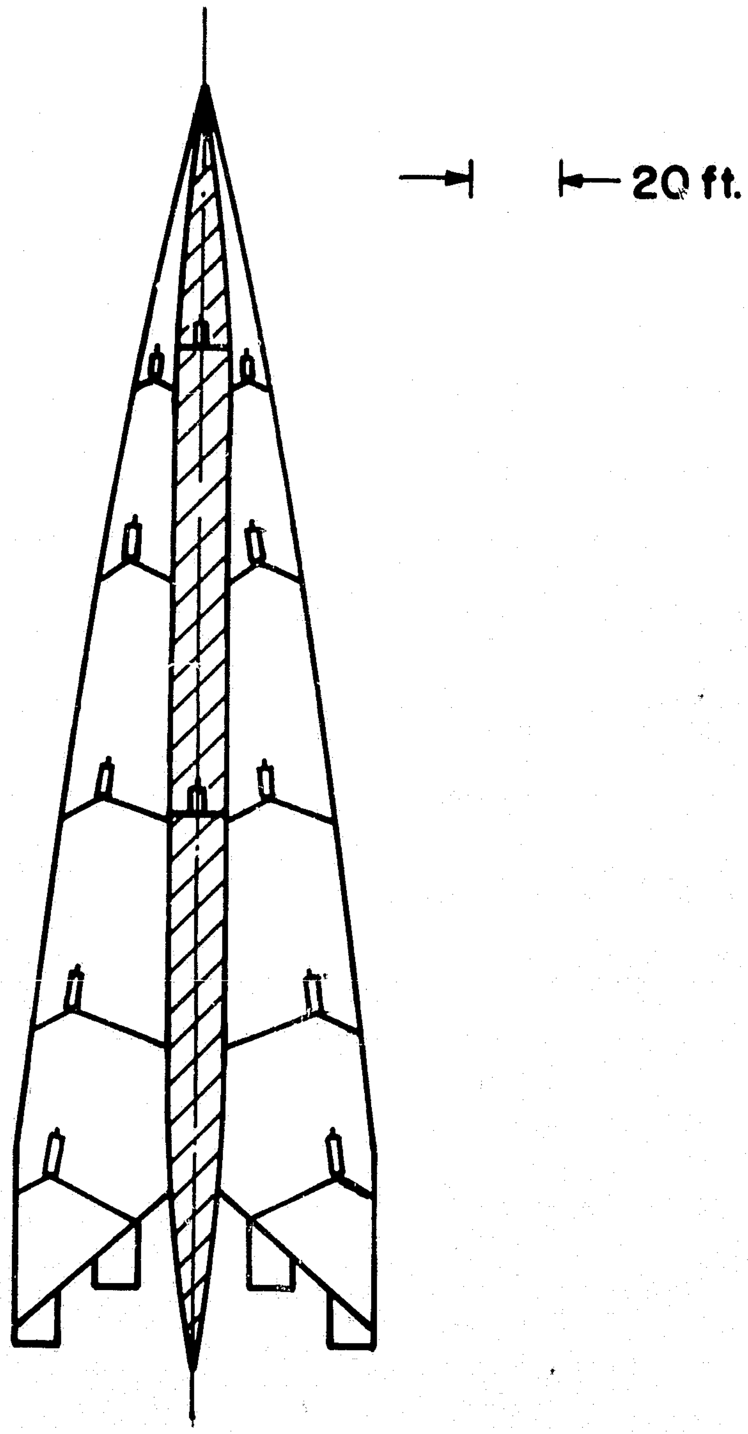

Fig. 26 Typical airplane - turbocooler arrangement

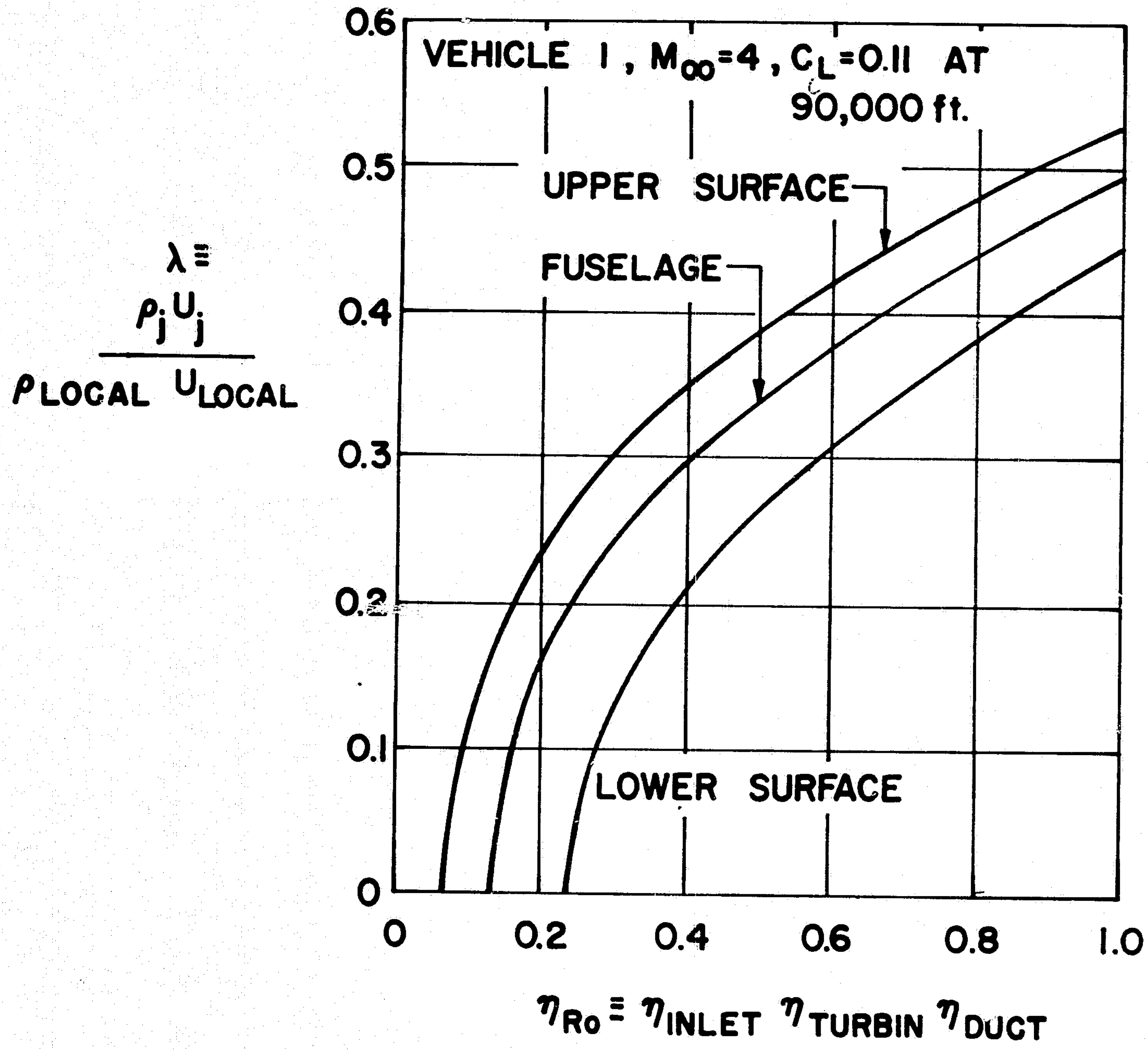

Fig. 27 Injection rate variations with efficiency

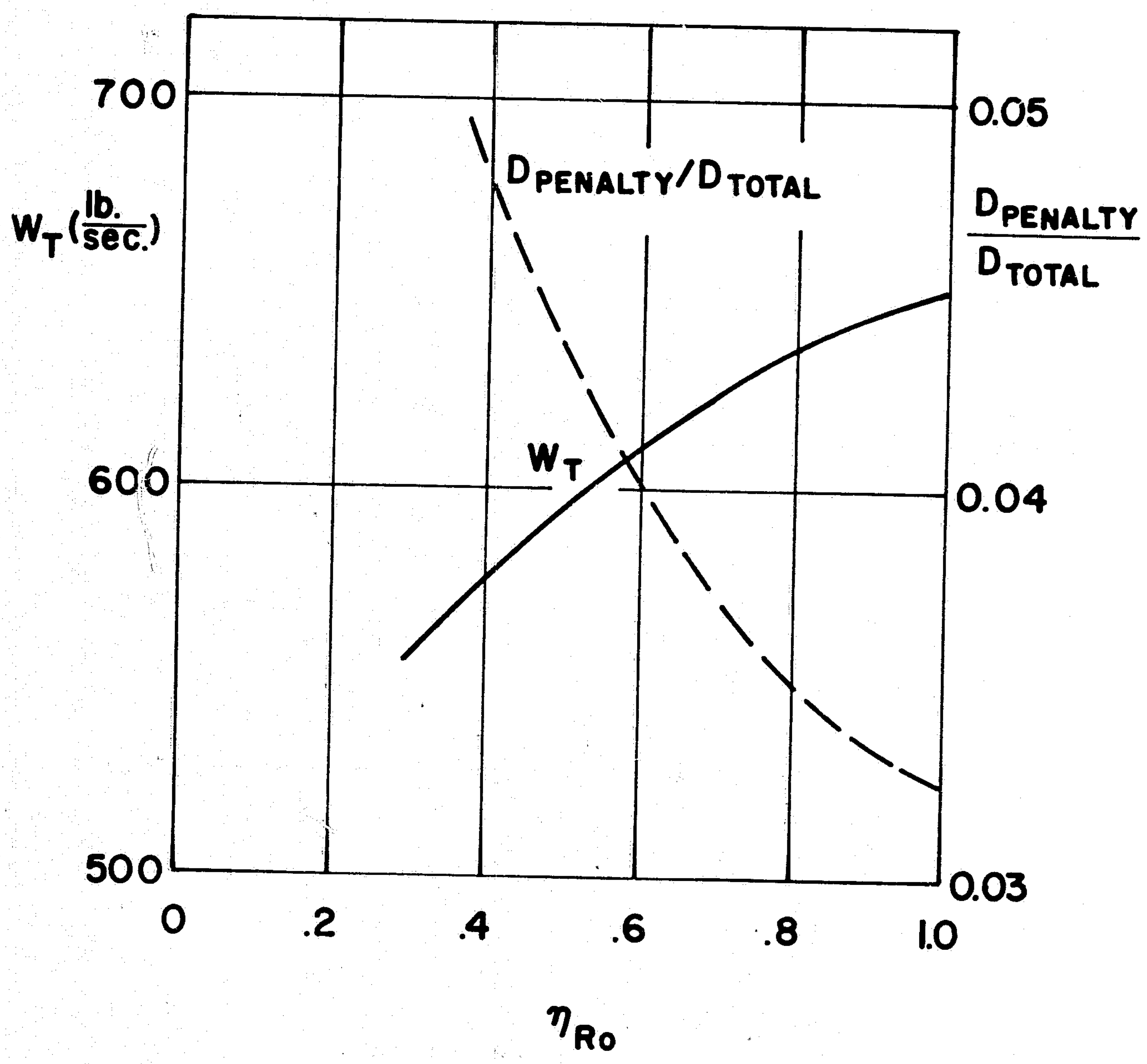

Fig. 28 Total mass flow requirement and drag penalty variations with efficiency

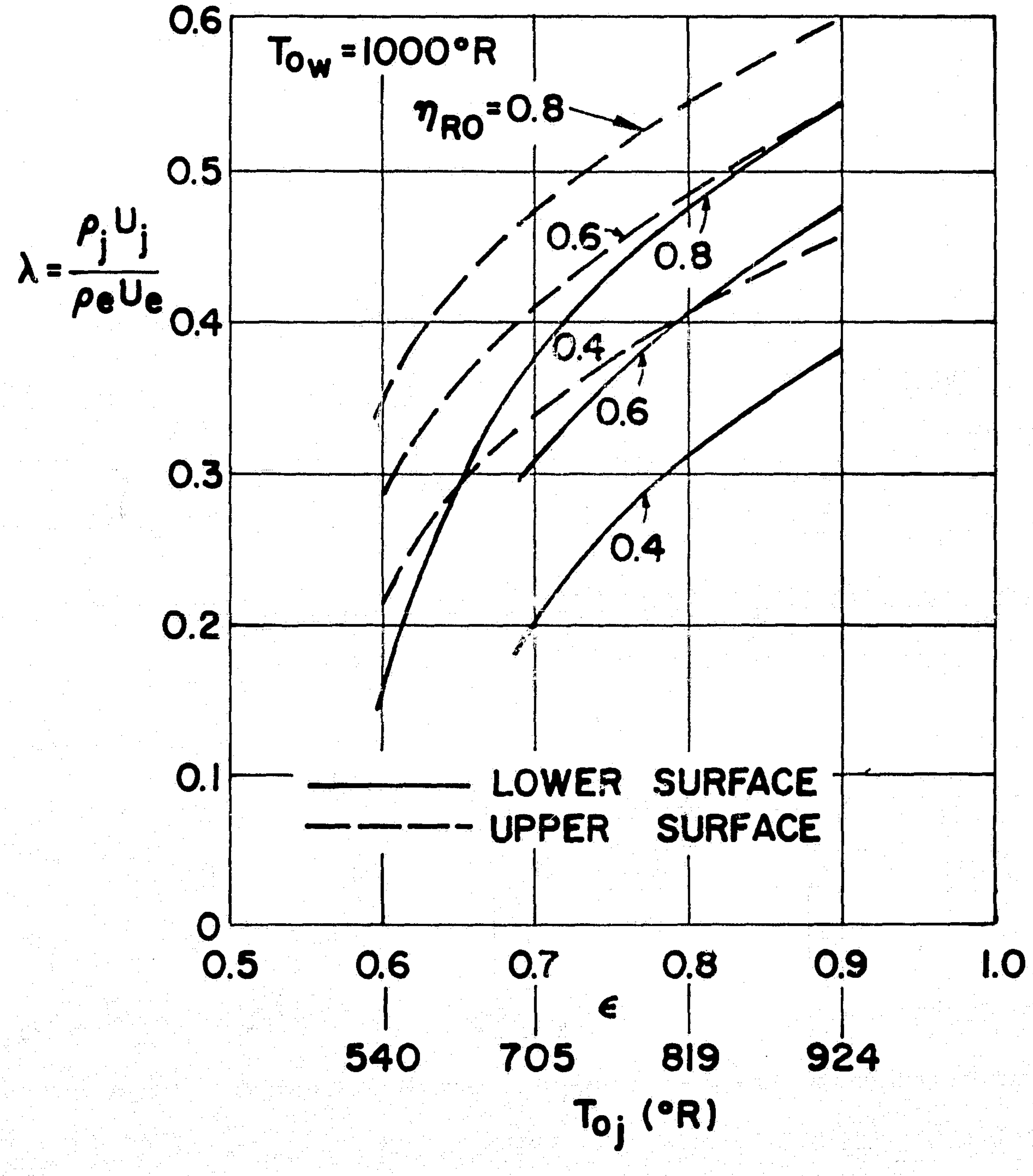

Fig. 29 Variations of Mass Flux Ratio

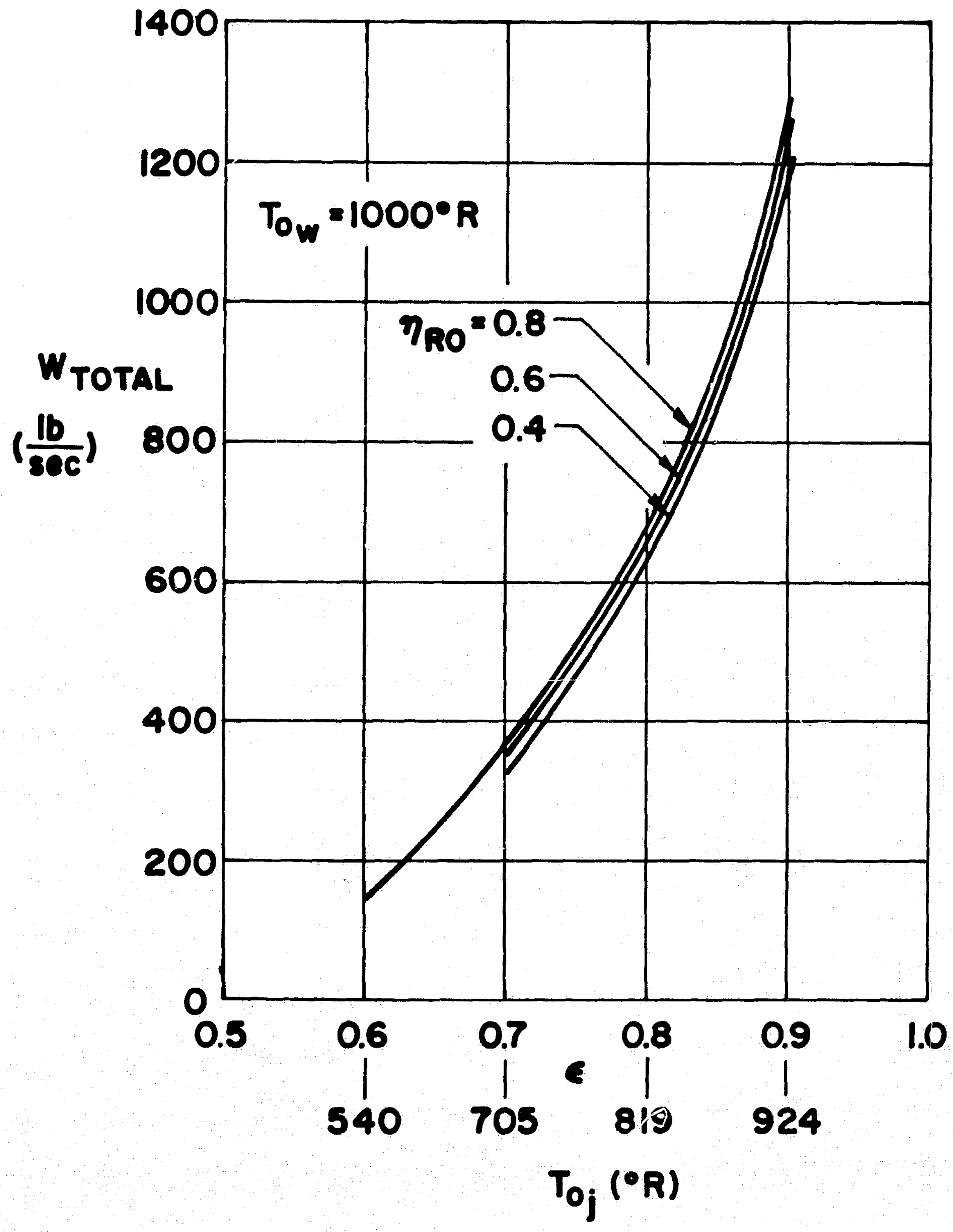

Fig. 30 Total Cooling Requirement Variations with Effectiveness

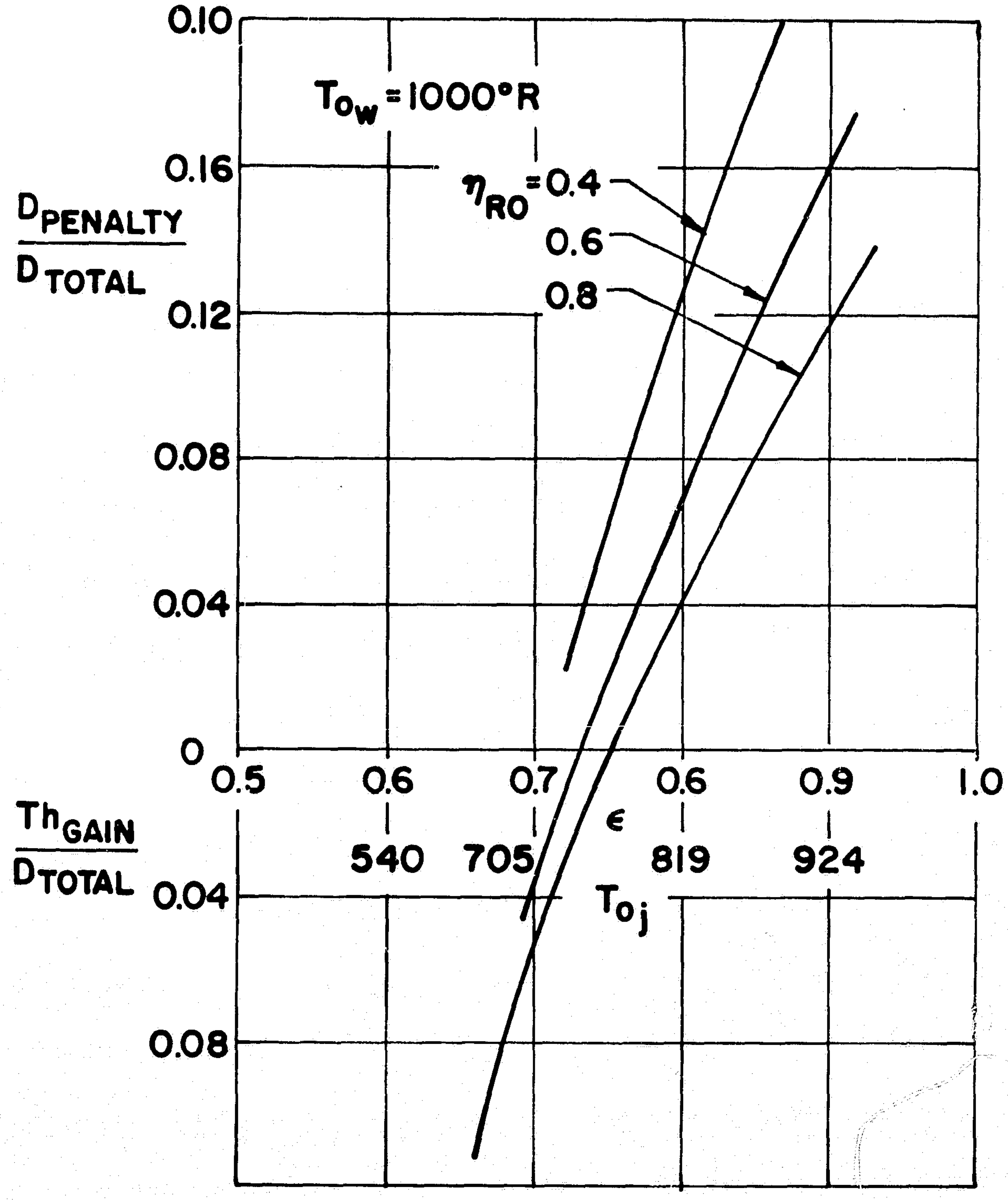

Fig. 31 Drag Penalty/Thrust Gain Variations with Effectiveness