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Investigation of the Effects of Pressure Gradient, Temperature, and Wall Temperature Ratio on the Stagnation Point Heat Transfer for Circular Cylinders and Gas Turbine Vanes

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FOREWORD

This report was prepared under Contract No. NASA Grant No. NAG 3-292 for the NASA Lewis Research Center under the technical direction of Dr. R.W. Graham. The high temperature heat transfer research in the shock tubes was initiated under the sponsorship of National Science Foundation, Dr. W. Aung, and NASA Lewis Research Center, Dr. R.W. Graham, under the NSF Grant No. MEA 80-06806. The work was conducted at the Gas Dynamics Laboratory, Rensselaer Polytechnic Institute in Troy, New York.

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1. INTRODUCTION

The increased cost of fuel for the jet engines on transport aircraft and for generating electrical power with gas turbines has made it necessary to reduce fuel consumption by operating the turbines at higher temperatures and pressures to increase the efficiency [1]. The gas temperatures in the operational jet engines require turbine film cooling and for high gas temperatures more advanced cooling schemes will be required. With a water-cooled gas turbine it seems possible to operate with gas temperatures of 1900°K and higher, and combining this with a steam turbine it appears possible for power generators to have an overall thermal efficiency of over 50% [2].

To achieve the higher efficiencies for the jet engines and gas turbines, it is essential in the design of turbines to have an accurate estimate of the local gas-side heat flux loads on the blades and vanes. For this purpose it is necessary to understand more quantitatively the local heat transfer rates at the stagnation region for the turbine vanes and blades as well as the heat fluxes for laminar, transition, and turbulent boundary layers over the pressure and suction surfaces under the influence of pressure gradients with gas temperatures as high as 2200°K and pressures to 40 atm. Also, it is necessary to know the heat flux distribution at the junction of the vanes with the shroud at these anticipated high temperatures. The available experimental and theoretical literature on the local gas-side heat transfer rates at high temperatures and pressures are very limited for the stagnation and end-wall regions and along the pressure and suction surfaces.

With these realizations, a basic heat transfer investigation to study the heat flux at the stagnation region for circular cylinders and over a flat plate with and without pressure gradient with gas temperature as high as 2200°K was initiated. This research was initially sponsored by NSF and NASA under NSF

Grant No. MEA80-06806, "Investigation of the Effects of Pressure Graident, Temperature and Wall Temperature Ratio on Heat Transfer Over Blunt and Streamlined Bodies," and continued under NASA Grant NAG 3-292. The objectives of the program were to obtain heat transfer data at high temperatures for circular cylinders and flat plates and correlate with the available theories to develop prediction methods for the external heat transfer at the stagnation region for circular cylinder, over a flat plate, and a turbine nozzle airfoil.

To produce shock heated air to 2500°K and to measure the local heat flux over bodies it was necessary to design and construct low and high pressure shock tubes with an internal diameter of 10.16 cm and a length of 21.34 m to cover the range of temperatures and pressures anticipated for the advanced jet engines and gas turbines. A square test section was designed and constructed to produce pressure gradient over a flat plate. Thin-film platinum heat gages were developed and calibrated to measure the local heat flux. The experimental heat transfer and analytical results obtained for the initial phase on the high temperature heat transfer program will be discussed in some detail in the body of this report.

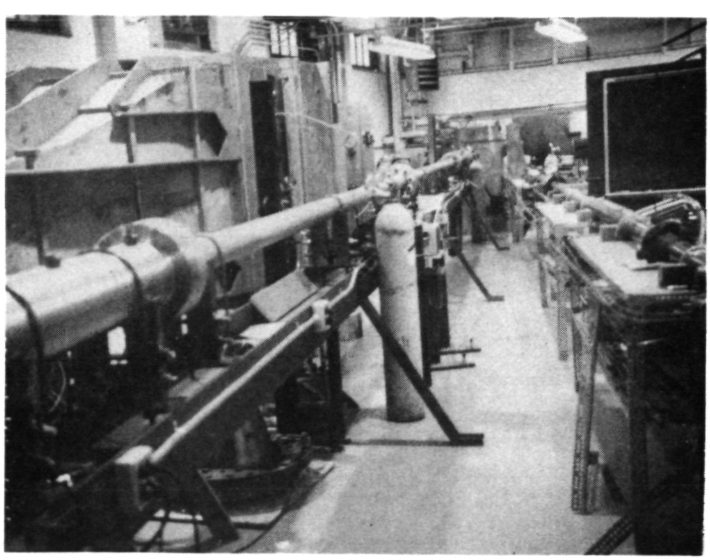
The results from the new shock tubes and instrumentation have been published in two Ph.D. theses [3,4] and seven Masters theses [5-11], and in three publications [12,13,14] which were presented at technical society meetings. A paper [15] on the hot-wire measurements of turbulence level and temperature fluctuations behind incident and reflected shock waves will be presented at the AIAA 13th Aerodynamic Testing Conference, March 5-7, 1984. The stagnation point heat transfer results [16] for circular cylinders at high temperatures and Reynolds number has been submitted to the AIAA 19th Thermophysics Conference, June 16-18, 1984. High temperature heat transfer results [17] for a flat plate with and without pressure for an inlet flow Mach number of 0.45 and over a range of Reynolds numbers has been submitted to the AIAA/SAE/ASME 20th Propulsion Conference, June 11-13, 1984. The heat flux distributions in the junction region between the flat plate and a circular cylinder mounted normal to the plate have been obtained and the paper [18] and has been submitted to the AIAA 19th Thermophysics Conference, June 16-18, 1984. Another paper [19] on the heat transfer distribution over a flat plate with a flow Mach number of 0.12 for laminar, transition, and turbulent boundary layers over a range of Reynolds number and gas temperature has been submitted to the same Conference.

2. EXPERIMENTAL APPARATUS

2.1 Low and High Pressure Shock Tubes

The initial heat transfer investigations for the circular cylinders, flat plates and shock tube wall were conducted in the RPI Low Pressure Shock Tube, and some of the results from these investigations are presented. This shock tube which has an operating pressure limit of 7.8 atm with a diameter of 10.16 cm and a length of 21.34 m is shown in Figure 1 and is described in Refs. [5,6]. In this facility it is possible to study the heat transfer over a temperature range of 350°K to 2600°K, as well as to calibrate the thin film platinum heat gages and piezoelectric pressure transducers [12,14].

High Pressure Shock Tube



Low Pressure Shock Tube

Figure 1 RPI Low and High Pressure Shock Tubes

A high pressure shock tube with a maximum driver pressure of 410 atm and driven tube maximum pressure of 128 atm with a diameter of 10.16 and a length of 21.34 m was designed and constructed, Figure 1. With this facility it is possible to produce shock heated air after reflected shock wave to 2800°K at high pressures and Reynolds numbers. Stagnation point heat transfer data for circular cylinders have been obtained at high Reynolds numbers [4,10,16].

A 5 hp compressor is installed near the 3.05 m long, 4 in. diameter driver sections for the low and high pressure shock tubes to pressurize the air to 205 atm in the storage bottles. This high pressure air is used to charge the driver section for both shock tubes.

A 5.67 m^3 dump tank was obtained from the General Electric Research and Development Center as surplus equipment. The dump tank is installed at the end of the high pressure shock tube as shown in Figure 2, and a 60.96 cm diameter

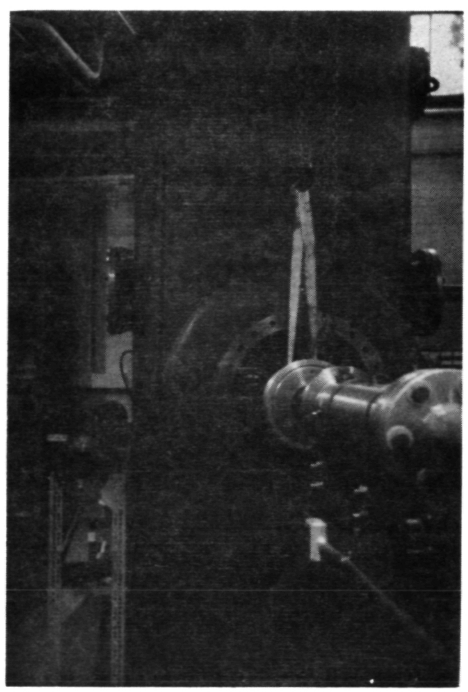


Figure 2 Dump Tank and Nozzle

conical nozzle can be attached to the dump tank for investigating hypersonic flow phenomena. A square test section with adjustable top and bottom walls to produce pressure gradient over a flat plate can be attached to the end of the tube with the exit of this test section located within the dump tank as discussed in Refs. [3,8]. Vacuum pumps are connected to the dump tank for evacuating the tank to the desired pressure.

2.2 Square Test Section With Adjustable Walls and Reflection Wedges

To increase the efficiency of jet engines and gas turbines, the combustion temperature and pressure must be increased to approximately 2200°K and 40 atm, respectively. At present there is no available heat transfer data at these extreme flow conditions in the literature [20-24]. For the vanes the inlet flow Mach number is about 0.15 and the flow accelerates to nearly sonic velocity at the trailing edge. But for the blades the inlet Mach number is approximately 0.45 with an exit Mach number of about 0.85.

An investigation was initiated in Ref. [8] to design, construct and test a square test section with either a reflection wedge or adjustable top and bottom wall, as shown in Figure 3, for producing high temperatures after the reflected shock waves. The adjustable top and bottom walls reflect the incident shock

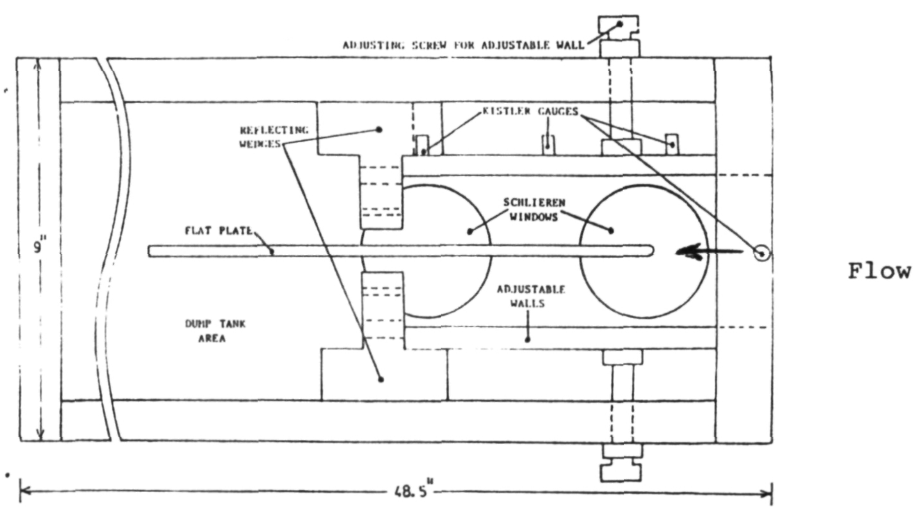


Figure 3 Square Test Section with Flat Plate Model

wave from the trailing edge of the plate to increase the gas temperature [3,8,25], as well as to produce a pressure gradient over the flat plate with the desired inlet flow Mach numbers of 0.15 and 0.45 after the reflected shock wave. The square test section with a dimension of 9.00 cm was connected to a transition section with an inlet diameter of 10.16 cm and square exit cross section, which was machined out of solid aluminum block by a numerically controlled milling machine [8]. The cross-sectional area in the transition section was kept constant.

Reflection wedges with various total open areas to produce reflected flow Mach numbers of 0.15 and 0.45 were constructed and installed in the square test section with a flat plate as shown in Figure 3. The variation of the static pressure with time for the four pressure gages located in the test section and over the flat plate (Figure 3) were measured for the flow Mach number of 0.15 after the reflected shock wave produced by the reflection wedges, Fig. 4a, and

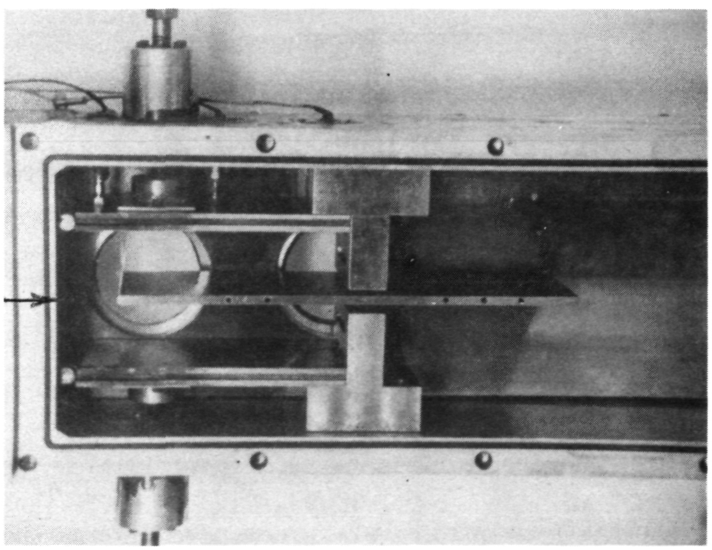


Figure 4a Square Test Section with Wedges

converging walls, Figure 4b, respectively. These results with the converging walls in the test section to simulate the flow through the gas turbine vane section at high temperatures indicate that the technique of adjustable wall will permit the measurement of the heat flux over a flat plate at temperatures up to 2600°K [3,14], with the desired pressure gradients existing in gas turbine vanes and blades.

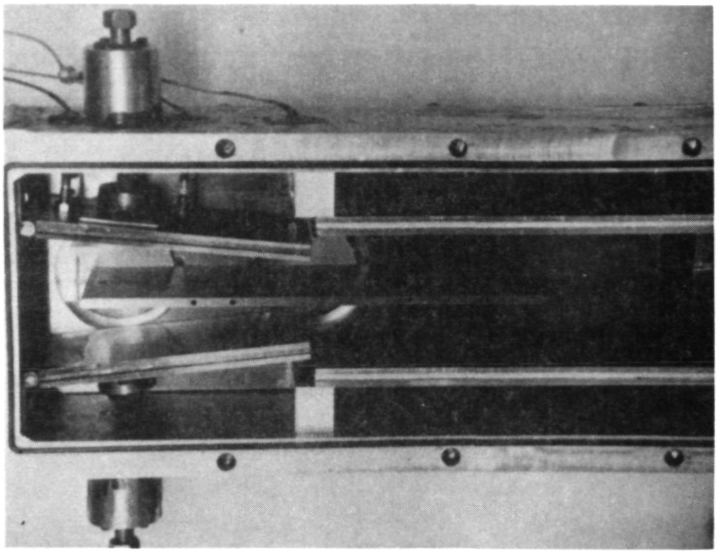


Figure 4b Square Test Section with Converging Walls

2.3 <u>Gas Turbine Nozzle Airfoil Cascade</u>

In Ref. [23] the heat transfer measurements were made on a gas turbine nozzle airfoil in a cascade (Figure 5) over a temperature range of 470° to 1780°K in the hypersonic shock tunnel [25] for the pressure and suction surfaces. Unfortunately, only a limited number of tests were conducted at Reynolds numbers much lower than that for the full-scale water-cooled gas turbine [2]. Thus, the cascade model can be attached to the high pressure shock tube (Figure 1) and heat transfer measurements can be obtained over the pressure and suction surfaces for a temperature range of 540° to 2200°F, various wall-to-gas temperature ratios and a range of Reynolds numbers. The wall temperature ratio on the laminar boundary layer transition Reynolds number for pressure and suction surfaces can be investigated. This cascade tunnel was loaned to RPI by the Department of Energy for use by the graduate students for their heat transfer research projects.

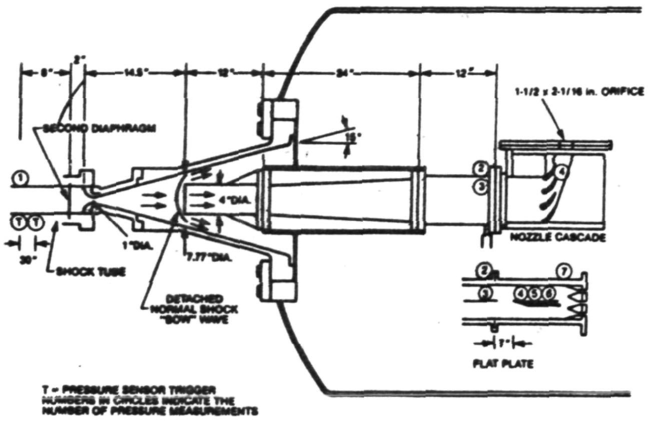


Figure 5 Shock Tunnel Configuration with the Flat Plate and Nozzle Cascade Test Section

2.4 Instrumentation

The General Electric Research and Development Center donated ten Tektronix scopes, scope cameras and peripheral equipment to RPI. These scopes have a response time of 1 μsec, which is adequate to measure the local heat transfer rates with the thin-film heat gages and static pressures with the piezoelectric pressure transducers.

Six Tektronix 122 low-level preamplifiers have been purchased and are being used with the heat gages and four piezoelectric quartz Kistler gages are used to measure the dynamic, static, and reflected pressures in the shock tube. Lead-zirconate-titanate crystals were purchased to construct dynamic pressure gages for use as a time-of-arrival instrumentation to monitor the shock velocity in the shock tube.

Initial static pressure in the driver and the driven tubes have been measured with mercury manometers and Heise pressure gages. Instrument panels to house these pressure gages have been constructed for the low and high pressure shock tubes (Figure 6).

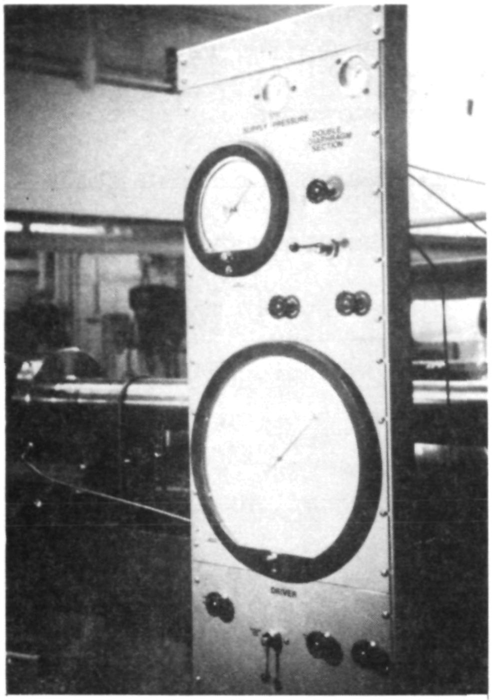


Figure 6a Loading Panel - Driver Tube

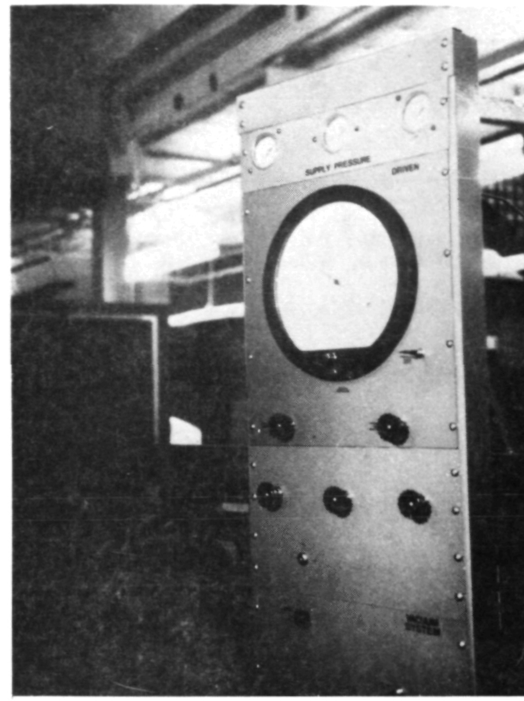


Figure 6b Loading Panel Driven Tube

Painted thin-film platinum heat gages have been fabricated for measuring the local heat transfer rates at the stangation point for circular cylinders [4,10,16] on a shock tube wall [6,12] and to flat plates [3,11,14]. The response time for these heat gages is approximately 1 μsec, and a detailed description for these heat gages is presented in Ref. [10].

A Schlieren system consists of two 20.32 cm diameter parabolic mirrors and two flat mirrors with a 0.4 μsec spark light source. This system can be used to take Schlieren and shadowgraph photographs of the boundary layer flow over a flat plate and at stagnation region for circular cylinders.

3. DATA REDUCTION

3.1 Determination of Test Conditions

The flow conditions in the test section are determined from the measured incident shock Mach number, initial pressure, P_1, and temperature, T_1, in the shock tube and the ratio of the area between the test section and total open area in the reflection wedges [26]. The pressure, P_5, behind the reflected shock wave from the wedges was measured with a Kistler quartz gage, and the corresponding reflected temperature, T_5, was calculated using the known shock velocity at the end of the driven tube and the equilibrium thermodynamic properties for air [27,28]. For the straight-through flow condition in the test section without the reflection wedges, the pressure, P_2, after the incident shock wave and the shock wave velocity were measured and the temperature, T_2, of the shock heated gas was calculated by using the normal shock wave equations as discussed in Ref. [26].

Stagnation point heat transfer investigations [4,10,16] were conducted in the low and high pressure shock tubes, Figure 1, over a range of pressure, P_5, from 5 to 250 psia and corresponding temperature, T_5, from 390 to 1000°K. The heat flux distribution over flat plates with and without pressure gradient [3,11,14], over shock tube wall [6,12] and in the junction region between a flat plate and a normal circular cylinder [18] were determined for laminar, transition, and turbulent boundary layers in the low pressure shock tube over a range of pressure, P_5, from 5 to 150 psia and corresponding temperature, T_5, from 670°K to 2800°K. At high temperatures the real gas effects were significant and the gas properties were calculated using the equilibrium thermodynamic properties presented in [27] and [28].

3.2 Heat Transfer Gage Data Analysis

The platinum film on the backing material of the heat transfer gage is thin enough such that the thermal effects of the platinum can be neglected in comparison to the backing material. Furthermore, the backing material can be treated as a semi-infinite solid having the property $\beta_b = (\rho C_p k)_b$, where ρ,

C_p, and k are the density, specific heat, and thermal conductivity of the backing material respectively.

The change in gage resistance, ΔR, due to a surface temperature change, $\Delta T = T_f - T_o$, is

$$\Delta R = R_f - R_o = R_o \alpha \; \Delta T, \tag{1}$$

where R_o and R_f are the gage resistances at temperatures T_o and T_f, and α is the resistivity. The gage is operated at a constant current I and hence the voltage change ΔE across the gage for a temperature change ΔT is

$$\Delta E = I \; \Delta R = I \; R_o \alpha \; \Delta T \tag{2}$$

When $\Delta T = \Delta T(t) = \Delta E(t)/\alpha I \; R_o$ is a function of time, the heat transfer q(t) to the gage is given in Ref. [29] as

$$q(t) = \frac{\sqrt{\pi \beta_b}}{2\alpha \; IR_o} \left[\frac{\Delta E(t)}{\sqrt{t}} + \frac{1}{\pi} \int_0^t \frac{\sqrt{\tau/t} \; \Delta E(t) - \Delta E(\tau)}{(t - \tau)^{3/2}} \right] d\tau \tag{3}$$

The heat transfer rate can be calculated from Eq.(3) using the voltage-time trace gage when the values of β_b and α are known. The constant coefficient $\sqrt{\beta_b}/\alpha$ for each gage can be obtained by a dynamic calibration.

To find the coefficient, the gages were mounted in the flat plate and placed in the center of a 10.16 cm diameter low pressure shock tube [3]. A piece of plexiglass which extended forward from the leading edge and from the plate edges to the tube walls was used to produce surface conditions for the gages which simulate a plate of infinite extent. When the shock passes over the plate a boundary layer forms behind the shock and there is a small step rise in the gage surface temperature evidenced by the voltage-time traces of the gage. The change in temperature is generally .1 to .2°C which is small compared to the plate surface temperature of approximately 300°K but adequate enough to give a voltage change of approximately .1 to .2 millivolts. For a step rise in temperature, the heat transfer rate $q_b(t)$ for assumed semi-infinite solid backing material is

$$q_b(t) = \left(\frac{\beta_b}{\pi}\right)^{1/2} \frac{\Delta T}{\sqrt{t}} = \left(\frac{\beta_b}{\pi}\right)^{1/2} \frac{\Delta E}{\alpha\, I\, R_o \sqrt{t}} \tag{4}$$

For a constant wall temperature, T_w, the local heat transfer rate q(t) at the wall from the gas in the boundary layer formed by a normal shock passage over a plate is given by [30]

$$q(t) = -k_w \sqrt{\frac{u_a}{2u_b t \nu_w}}\ (T_w - T_{w,i})\ S'(0) \tag{5}$$

where k_w and ν_w are the gas conductivity and kinematic viscosity at the wall temperature, $T_{w,i}$ is the insulated wall temperature, u_b and u_a are the free stream velocities before and after the normal shock, and S' (0) is a coefficient tabulated in Ref. [30].

Since at the gage surface $\Delta T \ll T_w$, the gage surface temperature changes have a negligible effect on the heat trasnfer rate q(t). Furthermore, since the effect of the shock passage on the gage is to give a step rise in the surface temperature, Eq.(4) and (5) can be equated and the gage coefficient $\sqrt{\beta_b/\alpha}$ can be found by

$$\frac{\alpha}{\sqrt{\beta_b}} = \frac{\Delta E}{IR_o k_w (T_w - T_{w,i}) \sqrt{\frac{\pi u_a}{2u_b \nu w}}\ S'(0)} \tag{6}$$

The constants associated with the gas are accurately known and depend only on the wall temperature.

The heat gages were calibrated in this manner over a shock Mach number range of 1.5 to 3, with the actual pressure rises on the surface of the plate corresponding to the estimated pressure rises that the plate would experience in the actual test section flows. This permitted the gage calibration to absorb any strain gage effects that the thin platinum film might exhibit under pressure loading.

4. EXPERIMENTAL AND ANALYTICAL RESULTS

4.1 Heat Flux for Laminar, Transition, and Turbulent Boundary Layers and Transition Phenomenon on a Shock Tube Wall

Dillon [6,12] conducted an investigation in the low pressure shock tube (Figure 1) over an incident shock Mach number range of 1.16 to 3.0, with a corresponding total temperature range of 331°K to 909°K, to study the local heat flux to the shock tube wall for laminar, transition and turbulent boundary layers and to observe the transition phenomenon and the transition Reynolds number

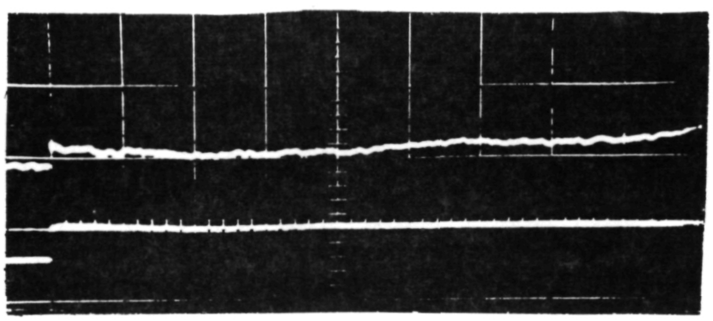


Figure 7 Heat and Pressure Gage Output Against Time Fully Laminar Boundary Layer, M_2 = .70, T_2 = 410°K, Re/cm = 2481

in the shock induced flow. Representative heat gage oscilloscope traces for a laminar boundary layer are presented in Figure 7. In this figure P is the shock tube wall pressure and H's are the heat gage located along the wall. For a laminar boundary layer the heat gage traces are smooth without fluctuations.

The heat gage traces were integrated on a computer to determine the local heat flux as a function of time. For a laminar boundary layer the local heat flux variation with time is presented in Figure 8 together with Mirels' [30] theoretical prediction for a laminar boundary layer. The agreement between the experimental data and the theory is very close.

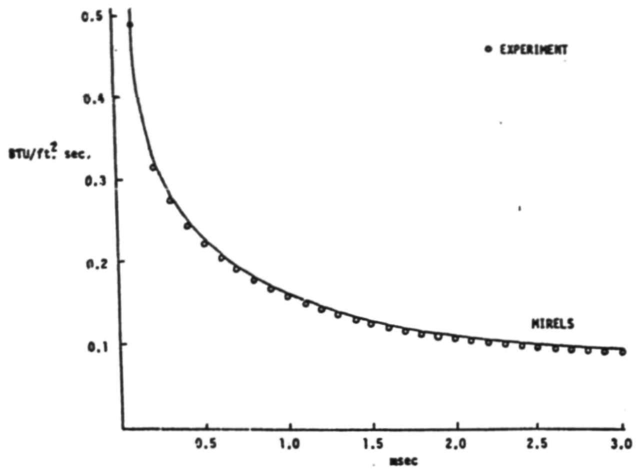


Figure 8 Local Heat Transfer Rate Against Time, Laminar Boundary Layer, M_2 = .70, T_2 = 410°K, Re/cm = 2481

For the case of fully developed turbulent boundary layer following the shock wave, the local heat flux variation with time is presented in Figure 9.

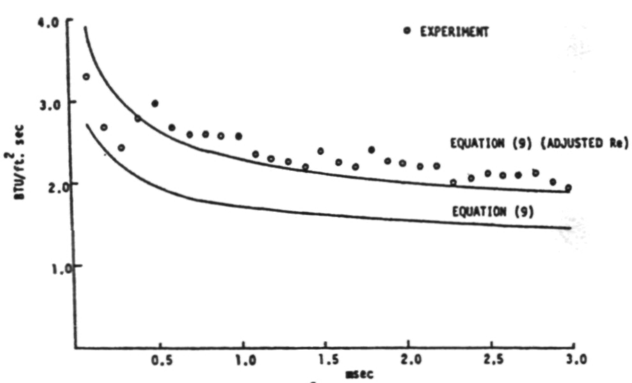


Figure 9 Local Heat Transfer Rate Against Time, Turbulent Boundary Layer, M_2 = .72, T_2 = 415°K, Re/cm = 30,000

In this figure the heat fluxes calculated by the modified von Kármán equation [31] for compressible turbulent boundary layer are presented for the cases with and without adjusted flow Reynolds number. The agreement between the experimental and the theoretical heat transfer rates are reasonable.

4.2 Stagnation Point Heat Transfer

Experiments have been conducted in the low pressure and high pressure shock tubes at low gas temperatures to investigate the stagnation point heat transfer for circular cylinders of various diameters (Figure 10) which simulate the leading edge of turbine vanes and blades. By using cylinders of different diameters, the effects of the pressure gradient on the stagnation heat transfer were investigated. The results for the low temperature heat transfer data are presented in Figure 11 as a function of Reynolds number [4,10,16] for flow Mach numbers of 0.15 and 0.45.

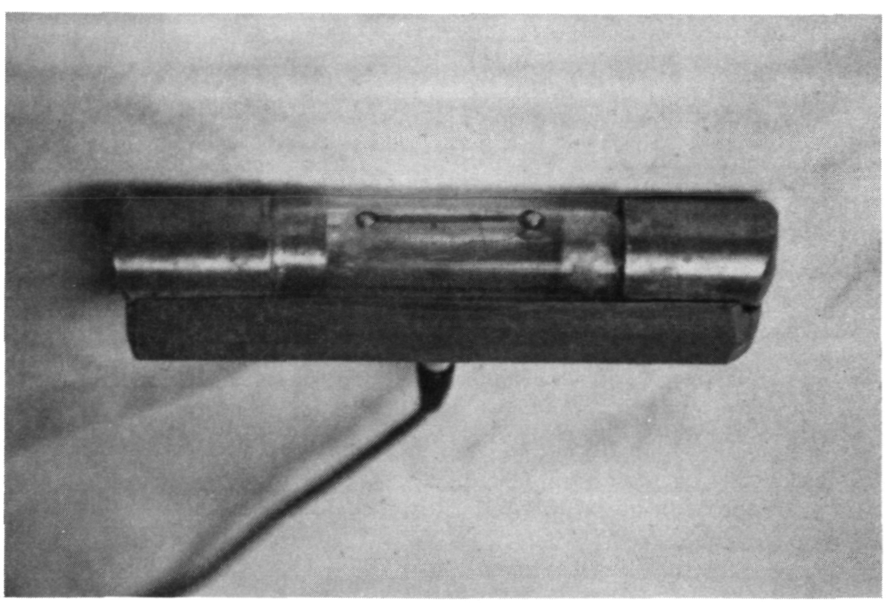


Figure 10 Different Heat Gage Units

Two known theories (laminar by Eckert [32] and turbulent by Van Driest [33]) respectively are plotted on Figures 11 and 12 with some typical test results obtained from the four different diameter heat gages. All data have been corrected for the effects of tunnel blockage according to the methodology of Ref. 34.

For low temperatures of 422°K at both flow Mach numbers of 0.15 and 0.45 the stagnation point heat transfer measurements were in very good agreement with Eckert's predicted laminar stagnation point heat transfer on a cylindrical capped flat plate for the Reynolds number range of $10^3 < RN < 10^6$, as shown in Figures 11 and 12.

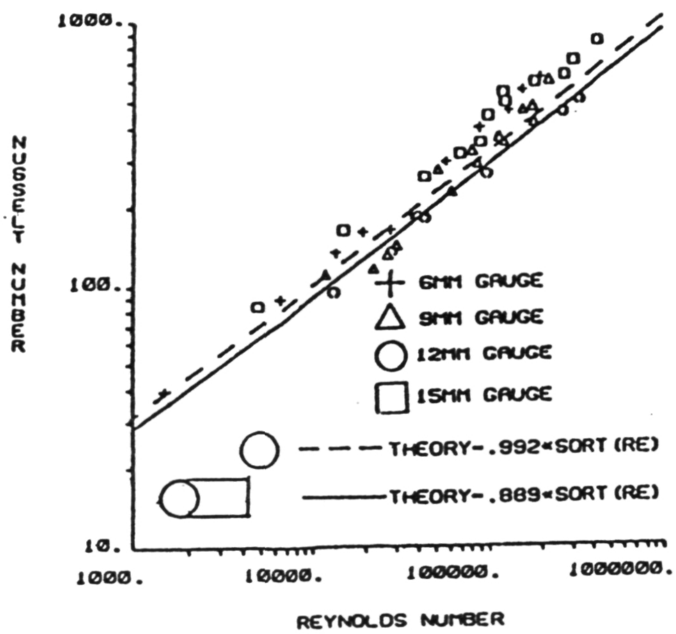


Figure 11a Stangation Point Heat Transfer Results for $M_{FLOW} = .15$, $T_{FLOW} = 422°K$

At higher flow temperature of 922°K the experimental heat flux measurements agree with Eckerts theory at low Reynolds. However, at Reynolds number of approximately 5 x 10^4 a marked increase in the heat transfer exists. The data show a trend predicted by Van Driest's turbulent stagnation point heat

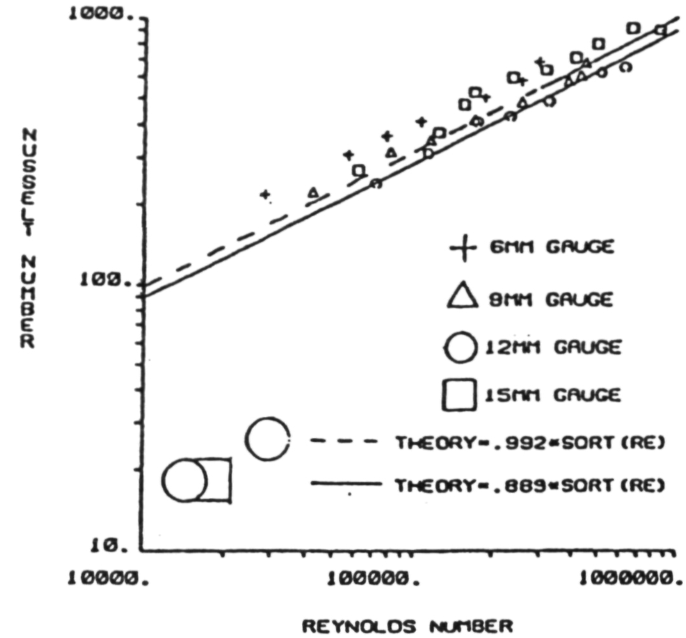


Figure 11b Stagnation Point Heat Transfer Results for $M_{FLOW} = 0.45$, $T_{FLOW} = 422°K$

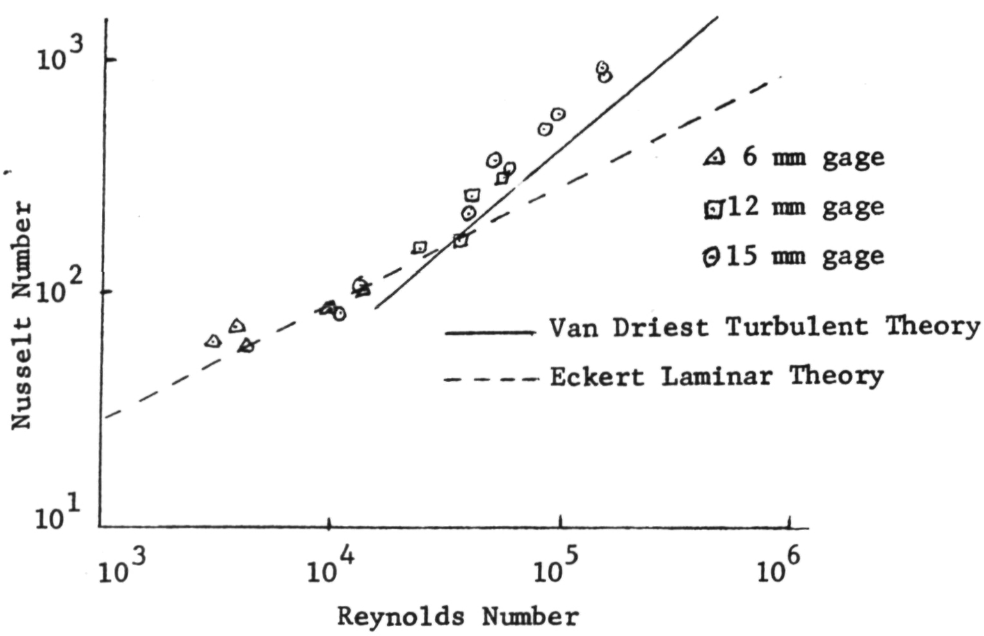


Figure 12 Stagnation Point Heat Transfer Nusselts Number vs Reynolds Number, $M_{FLOW} = 0.15$, $T_{FLOW} = 922°K$

transfer theory although approximately twenty percent higher than theory. Typical measurements illustrating this effect at a flow Mach number and temperature of 0.15 and 922°K respectively are shown in Figure 12. Some preliminary measurements have been made to access the turbulence level in the shock induced flow [9]. Indications are that the level of observed turbulence intensity is not sufficient to explain the high measured heat transfer.

The data gathered will contribute to a fundamental heat transfer data file which will be useful in the design of future jet engines and gas turbines.

4.3 Flat Plate Heat Transfer

Two flat plate models were constructed to measure the heat flux for laminar, transition, and turbulent boundary layers as fucntions of Mach number, Reynolds number, gas temperature, and wall temperature ratio. The first model was 10.16 cm wide, 60.96 cm long, and 6.35 mm thick with eight heat gages located along the surface for testing in the 10.16 mm diameter low pressure shock tube (Figure 13). The second plate model was 30.48 cm long, 9.02 cm wide and 6.35 mm thick for use in the square test section shown in Figure 3. The leading edge was machined round and the plate housed five thin-film platinum heat gages spaced along the plate ahead of the reflection wedges [3,11].

The heat gage constants were determined in the low pressure shock tube for Reynolds number per inch range of 0.20×10^3 to 2.3×10^5. The flow Mach number after the incident shock wave varied from 0.45 to 0.87. The response time of these gages is a few microseconds [12,23,25]. Heat gage traces on the oscilloscope for laminar, transition, and turbulent boundary layers are presented in Figure 14. For the laminar boundary layer the trace is relatively smooth,

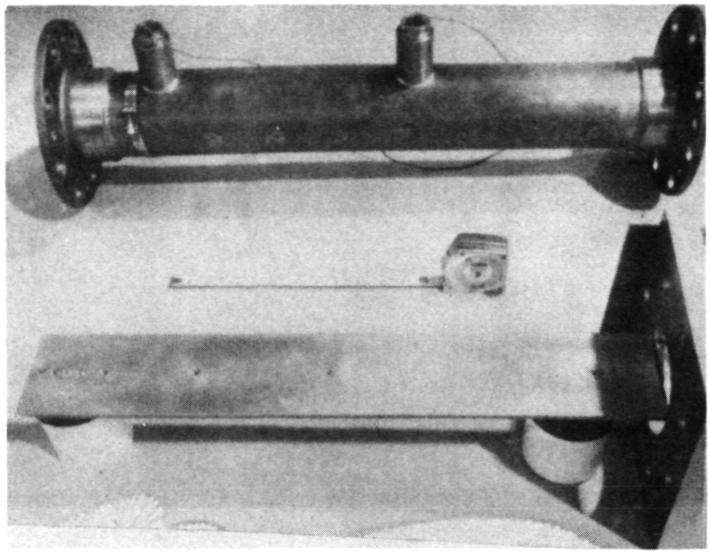


Figure 13 Circular Test Section and Flow Plate

but for the transition flow the fluctuation due to the turbulent bursts as they traverse across the heat gage produce relatively large oscillations. For nearly fully turbulent flow, the heat gage trace returned to reasonably smooth parabolic variation with a superimposed high frequency oscillation due to the turbulent eddies. Similar results were obtained by Nagamatsu, Graber and Sheer [35] for hypersonic flow over a 10° cone.

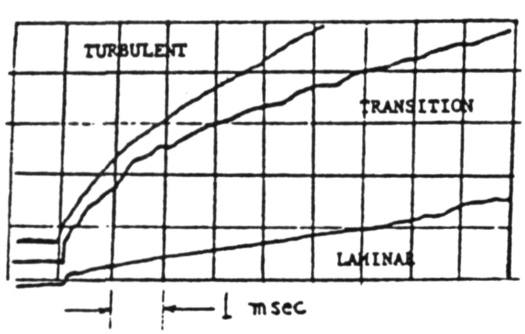


Figure 14 Heat Gage Traces for Laminar, Transition and Turbulent Boundary Layers

The local heat flux distributions over the 60.96 cm long flat plate, with cylindrical leading edge (Figure 13) using the wedges to produce a flow Mach number after the reflected shock wave of 0.15, for laminar, transition, and turbulent boundary layers were determined in Refs. [3,14], and the results for laminar and turbulent boundary layers are presented in Figures 15a and b, respectively. The Reynolds numbers per inch for these flow conditions were 8.05 x 10^2 and 7.6 x 10^4 for a gas temperature of approximately 440°K. In these

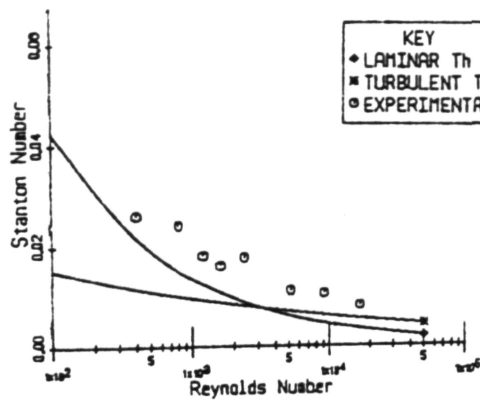


Figure 15a Heat Transfer for Partial Reflection - Laminar Flow.
Re/cm = 199, T_∞ = 517°K, M_∞ = .15

figures the theoretical and experimental laminar and turbulent boundary layer heat flux variations with Reynolds number [36] are presented. The experimental Stanton numbers are slightly greater than the laminar boundary layer prediction but the variation of the Stanton number with Reynolds number agrees reasonably well with the theoretical variation. The higher heat flux at a given gage location from the leading edge is due mainly to the blunt leading edge. In Figure 15b the experimental Stanton numbers are presented for a Reynolds number range of 3.84 x 10^4 to 1.65 x 10^6 and a temperature of 420°K. The variation of Stanton number with Reynolds number is close to the turbulent boundary layer prediction which indicates that the boundary layer is turbulent. Higher experimental heat flux at a given location is due primarily to the blunt leading edge which makes the boundary layer thinner at a fixed distance from the leading edge than the flow over a plate with a sharp leading edge.

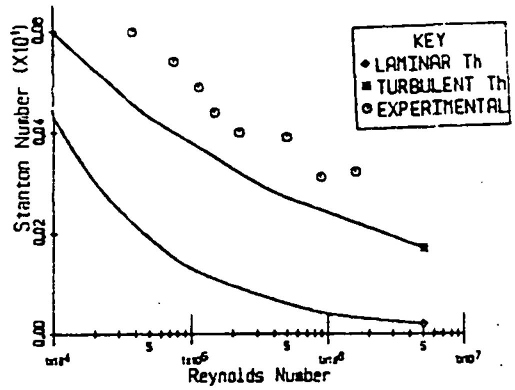


Figure 15b Heat Transfer for Partial REflection - Turbulent Flow. Re/cm = 30,197, T_∞ = 416°K, M_∞ = .15

The local heat transfer rates over a flat plate were measured for various Reynolds numbers with the top and bottom walls converged (Figures 3, 4b) to produce pressure gradient and flow Mach number after the reflected shock wave of 0.10 and 0.15 and temperature of 400°K and 2570°K. The variations of Nusselt number with Reynolds number for these representative flow conditions were reported in Refs. [3,14] and are presented in Figures 16a and b. The Nusselt numbers for a Reynolds number of 7.84 x 10^4 over the initial one inch of the plate at a temperature of 406°K are close to the turbulent theory as shown in Figure 16a. For this flow conditions the Reynolds number varied from 7.84 x 10^4 to 2.35 x 10^5 over the heat gages. Preliminary high temperature heat transfer measurements were conducted for a temperature of 2575°K with the converging walls and the results are presented in Figure 16b. The Reynolds number over the initial one inch of the plate was 2.93 x 10^4 and a wall temperature ratio of 0.11. For this flow condition after the reflected shock wave, the Nusselt numbers and the variation with Reynolds number agree reasonably well with the turbulent theory. Additional heat transfer investigations will be conducted to determine the effects of high temperatures and wall temperature ratios with

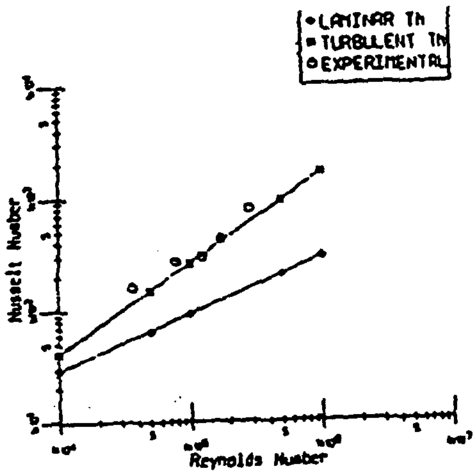


Figure 16a Converging Wall - Turbulent Flow Heat Transfer.
Re/cm = 30,806, T_∞ = 406°K, M_∞ = .15

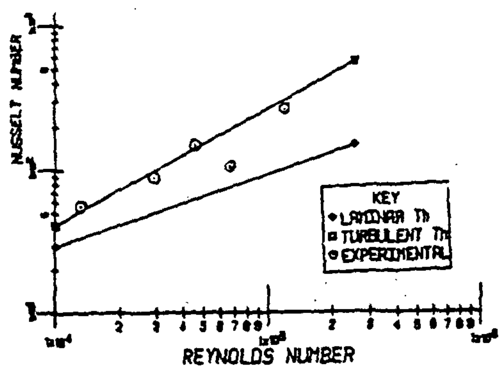


Figure 16b High Temperature - Converging Walls Heat Transfer.
Re/cm = 11,535, T_∞ = 2575°k, M_∞ = .15

and without pressure gradient on the heat transfer.

An investigation was initiated to study the effects of large temperature gradient and corresponding density fluctuations on the turbulent boundary layer heat transfer [37,38] for flow Mach number of 0.12, which is the flow Mach number after the combustor in gas turbines. For water cooled gas turbines to generate power [2] the gas temperature may be as high as 2220°K with the vanes and blades surface temperature of approximately 830°K. Under these conditions there will be large temperature gradient in the laminar and turbulent boundary layers.

The square test section was used with the top and bottom walls in the horizontal position to measure the heat flux distribution over the flat plate, as shown in Figure 3. Reflection wedges were used to reflect the incident shock wave to produce flow Mach after the shock wave of 0.12. This flow Mach number was produced by selecting the ratio of the throat area in the reflecting wedges to the test section area ahead of the flat plate. Six thin-film platinum heat gages were mounted flush on the plate surface to measure the local heat flux.

The heat transfer results for laminar, turbulent and transition boundary layers are presented in Figures 17a and b, respectively for a gas temperature

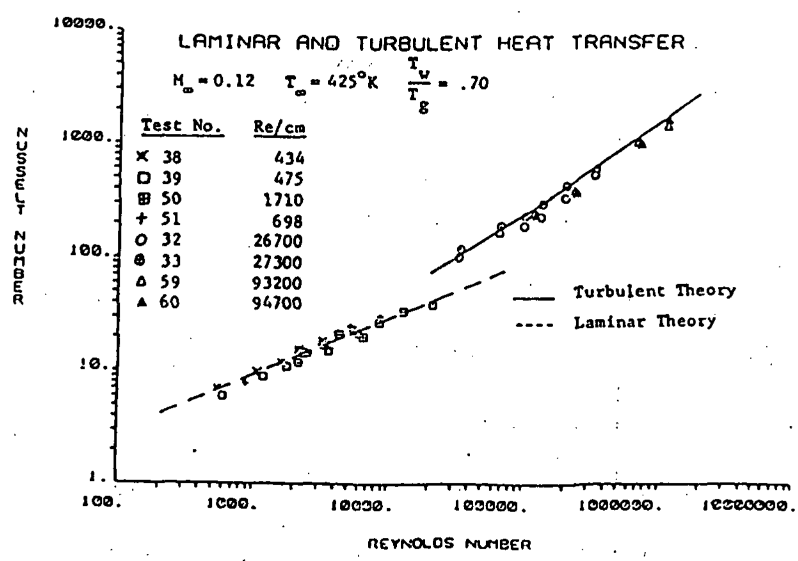


Figure 17a Heat Transfer for Laminar and Turbulent Boundary Layer
$T = 425°K$, $M = .12$

of 370°K, Mach number of 0.12, and wall to gas temperature ratio of 0.70 [11,19]. For unit Reynolds number, Re/cm of 434 to 1710, the Nusselt numbers agree with laminar boundary layer prediction, but for unit Reynolds number greater than 2.67×10^4, the heat flux rates are close to the turbulent prediction.

In Figure 17b the heat transfer rates are for the transition region from laminar to turbulent boundary layer and the Nusselt numbers lie between the laminar and turbulent predictions. The unit Reynolds numbers for these tests ranged from 3.0×10^3 to 1.36×10^4.

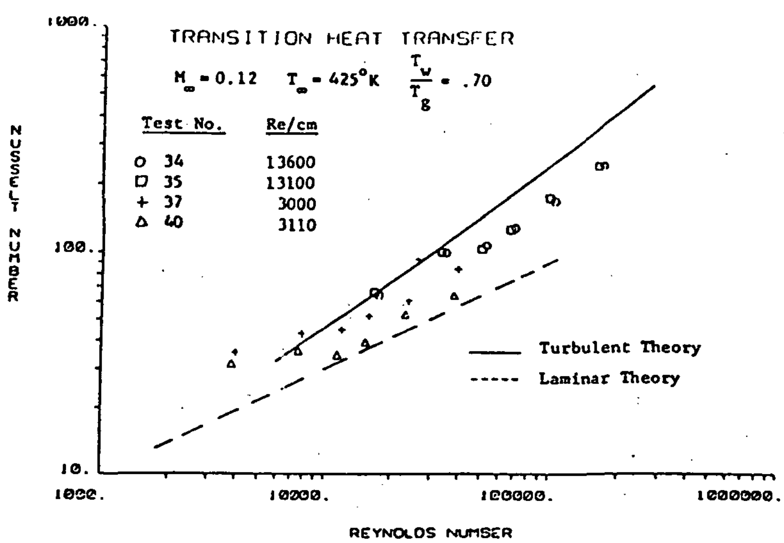


Figure 17b Heat Transfer for Transition Boundary Layer
$T = 425°K$, $M = .12$

The Nusselt numbers over the flat plate for a gas temperature of 1000°K, flow Mach number of 0.12, and wall to gas temperature ratio of 0.33 are presented in Figure 18. At low unit Reynolds number the Nusselt numbers are close to the laminar predictions, but at higher unit Reynolds number of 2.09×10^4 the Nusselt numbers are greater than for the turbulent prediction which indicate that the turbulent boundary layer was not fully developed.

Additional experiments will be conducted at higher unit Reynolds numbers and temperatures to achieve fully developed turbulent boundary layer and to determine the effects of large density gradient in the boundary layer on the heat transfer at low Mach numbers.

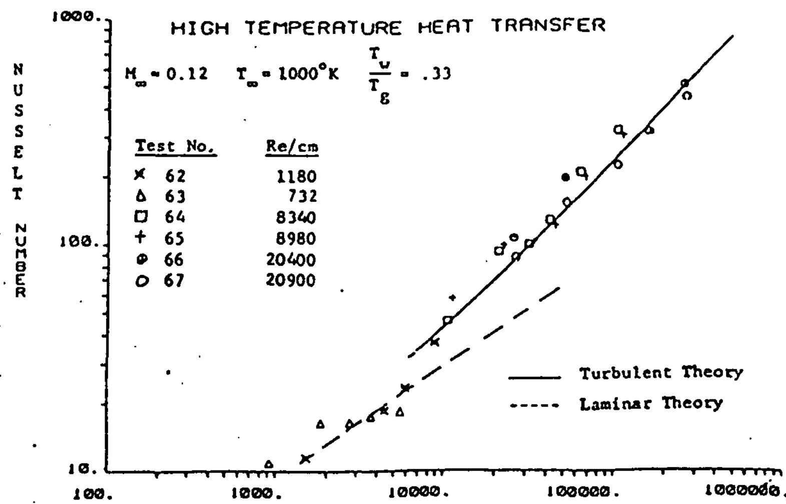


Figure 18 Heat Transfer for Laminar and Turbulent Boundary Layers
$T = 1000°K$, $M = .12$

4.4 Heat Transfer in the Junction Region of a Circular Cylinder Normal to a Flat Plate

An experimental investigation was conducted to measure the heat transfer rate in the junction region of a circular cylinder mounted normal to a flat plate [18]. This geometrical configuration was selected to simulate the endwall of a gas turbine vane. To achieve higher efficiency for jet engines and gas turbines, higher temperatures and pressures are required. Because of the smaller span for the vanes at high pressures, the interaction between vanes and the endwall on the heat transfer becomes increasingly important. For adequate and efficient cooling of the endwall and the vanes, the heat flux distribution in the junction region is required. Only limited experimental heat transfer data are available and an analytical treatment of the problem is difficult due to the nonlinearity of the viscous interaction phenomena.

A low pressure 21.34 m long shock tube with a diameter of 10.16 cm, shown in Figure 1, was used to generate the shock heated air. Partial reflection of the incident shock wave by a slotted plate shown in Figure 19 was used to obtain the desired flow Mach number of 0.14 over the flat plate, which is approximately the inlet flow Mach number for the first stage vanes in the jet engines. Since the flow Mach number after the reflected shock wave is determined by the ratio of the slotted throat area to the test section area, the flow Mach number is very consistent. About 10 to 15 ms of steady flow was observed after the reflected shock wave [18].

The sketch of the flat plate with a circular cylinder mounted normal to the surface is presented in Figures 19. A slotted wall is placed downstream of the 1.59 cm inch diameter cylinder and the total opening in the plate is determined to achieve flow Mach number after the reflected shock wave of approximately 0.14. Two rows of heat gages are mounted on a rotatable circular plate inserted

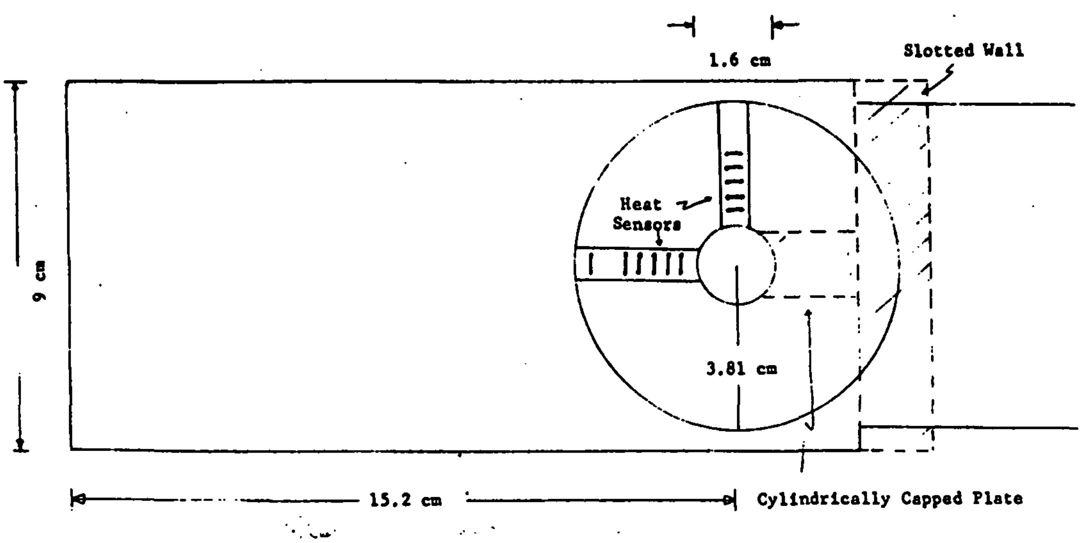


Figure 19 Flat Plate with Heat Gages (Side View)

flush with the flat plate with the cylinder placed on the axis as shown in Figure 19. The center of the cylinder was located 15.2 cm from the leading edge of the flat plate and the heat gages were located on the horizontal plane 0.2, 0.4, 0.6, 0.8, 1.0 and 1.5 diameters from the leading edge of the cylinder. In the vertical plane the heat gages are located at similar distances from the edge of the cylinder. Fast response thin-film platinum heat sensors were painted on pyrex glass and baked in an oven to get good bonding to the glass. The change in the surface heat temperature produces a change in the voltage which was recorded on an oscilloscope. A computer program was used to calculate the heat flux from the voltage variation with time as discussed in Refs. 6 and 14. The response time of the platinum heat gages is fast enough to resolve the passage of turbulent eddies over the plate.

To determine the effects of the circular cylinder on the flat plate heat transfer rate in the junction region, Figure 19, experiments were conducted for the same flow conditions for both the flat plate with and without the circular cylinder of 1.59 cm diameter. The heat flux distributions on the flat plate were measured in the horizontal plane ahead of the cylinder and perpendicular to the flow as shown in the figure. For the present tests the temperature after the reflected shock wave was approximately 450°K and a flow Mach number of about 0.14.

The Stanton number variations with Reynolds number for the heat gages located in the horizontal plate ahead of the cylinder are presented in Figures 20a to c for gages located at 0.2, 0.6, and 1.5 diameter from the cylinder, cf. Figure 19. In these figures the theoretical heat flux variation with Reynolds number for laminar and turbulent boundary layers are presented to correlate the experimental data.

For the heat gage located 0.2 diameter ahead of the cylinder the Stanton number at low Reynolds number for the flat plate only is close to the laminar theory and for Reynolds number greater than 10^5 the Stanton numbers are close to the turbulent prediction as shown in Figure 20a. But with the cylinder the Stanton number is increased nearly 300% for a Reynolds number of 3.0×10^4.

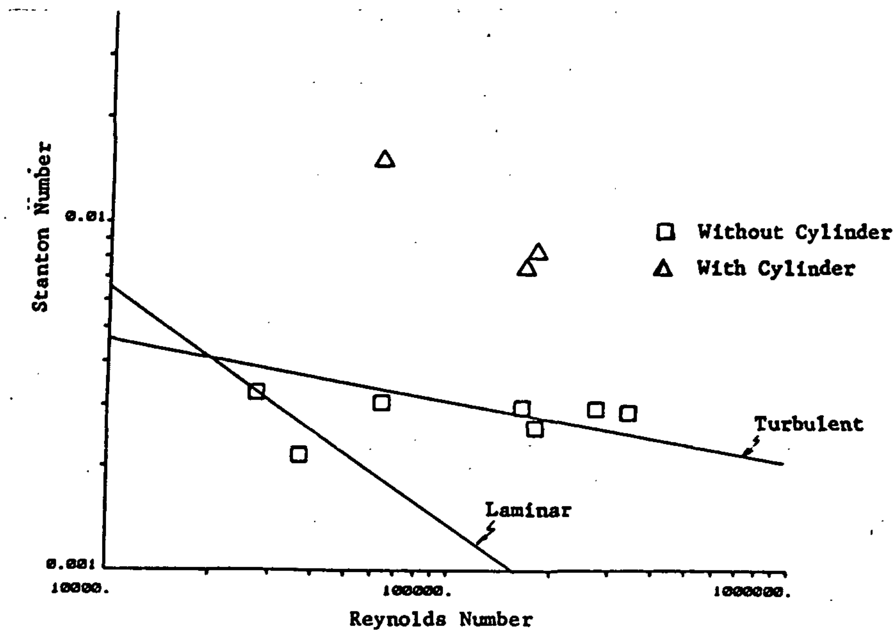


Figure 20a Heat Transfer for Flat Plate with and without Cylinder for Heat Gage at x = .2D

Similar Stanton number variation with Reynolds number is presented in Figure 20b for the heat gage located 0.6 diameter from the cylinder. With the cylinder the heat flux is increased approximately 200% for a Reynolds number of 3.5×10^4. For the heat gage located 1.5 diameter from the cylinder, the Stanton numbers with and without the cylinder are presented in Figure 20c, and the heat flux is increased approximately 15% by the cylinder.

The Stanton numbers at the three heat gage locations are normalized by the Stanton number for a turbulent flow over a flat plate and the results are presented in Figure 21. For the present test conditions, the heat flux increase by the circular cylinder is appreciable near the stagnation region of the cylinder and the increase in the heat flux decreases for the gages located farther from the leading edge. A new set of experiments has just been completed to extend the range of Reynolds number. The heat transfer data are being reduced and will be correlated with the existing results.

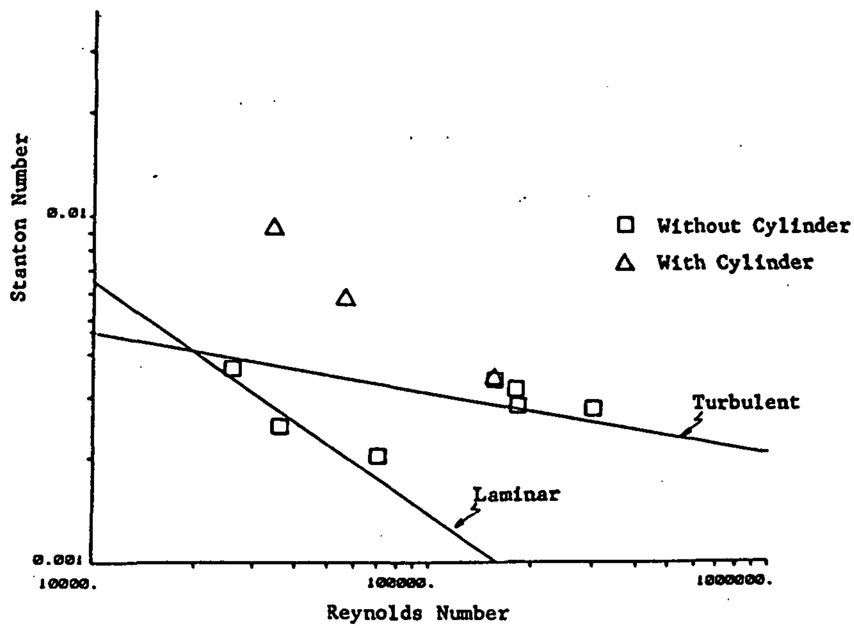


Figure 20b Heat Transfer for Flat Plate with and without Cylinder of $x^{*} = .6D$

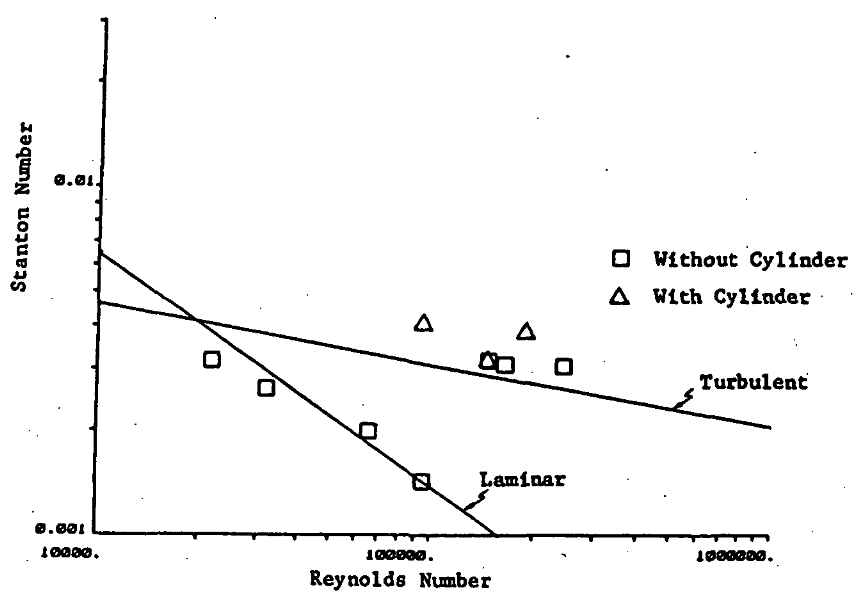


Figure 20c Heat Transfer for Flat Plate with and without Cylinder at $x^{*} = 1.5D$

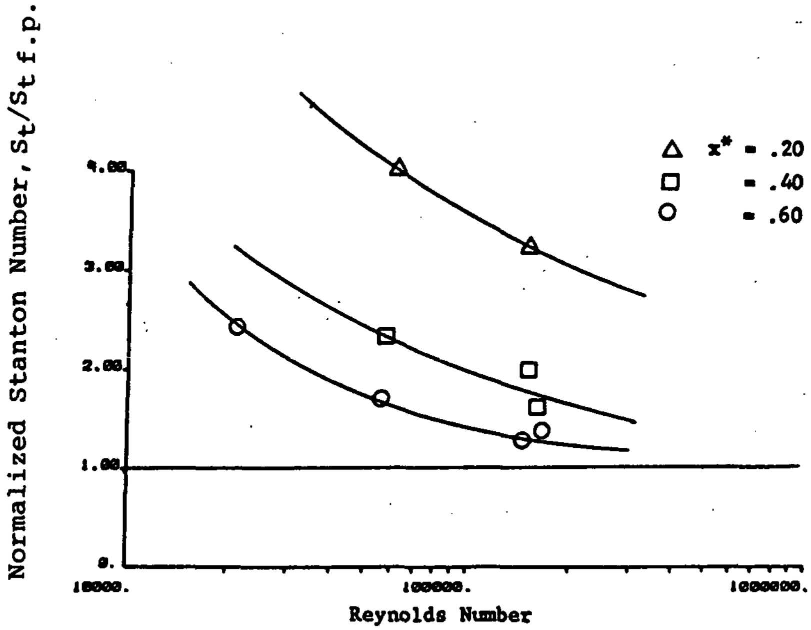


Figure 21 Normalized Heat Transfer Rate for Heat Gages Located at Various Distances from the Circular Cylinder Leading Edge

4.5 Shock Induced Turbulence

The purpose of this investigation is to determine the free stream turbulence intensity generated by shock induced flows. Inasmuch as heat transfer measurements are strongly influenced by free stream turbulence level, an accurate determination of the turbulence present in the test environment must be made to satisfactorily interpret heat transfer data.

The experimental measurements of shock induced turbulence utilizes a hot wire anemometer. Two tungsten hot wires with a diameter of 0.0038 mm are being used as sensors with two TSl Model 1050 anemometer modules. By operating these wires at different temperatures it is possible to obtain two separate time histories of density, velocity, and temperature, from which the turbulence intensities can be calculated.

Initial hot wire results have been obtained for a shock Mach number range of 1.1 and 1.6 which corresponds to induced flow Mach numbers between 0.15 and 0.70, static temperatures between 310° and 415°K, and static pressures between 0.014 and 1.70 atm. The unit Reynolds number range for these conditions span between 7800 and 78,000. Tests are being conducted with tungsten sensors in nitrogen to permit the operation of the wire to 1370°K. Since air is composed largely of nitrogen and has a ratio of specific heat very close to that of nitrogen, turbulence tests in nitrogen are anticipated to closely resemble turbulence tests in air. Preliminary hot-wire results indicate low turbulence intensity after the incident shock wave and the turbulence intensity increased after the reflected shock wave [9,15].

4.6 Investigation of Steady-Flow Establishment Time in a 15° Half-Angle Concial Nozzle

The shock tunnel facility [25] was modified to obtain local heat flux distribution over gas turbine nozzles [23]. One of the problems associated with facilities of this type is the loss of the testing time due to the transient flow produced by the passage of the starting shock wave system through the nozzle, as shown in Figure 22.

In an effort to clarify the nature of the flow starting process and to obtain experimental values of the starting times, an investigation was conducted and the results are presented in Refs. [7,13]. A computer model similar to the finite difference scheme used by Takayama [39] was developed and extended in Refs. [40,41] to include effects of variable geometry and shock reflecting diaphragm. The computer program was adopted for the particular geometry of the test section used in the investigation (Figure 22) and a range of shock Mach numbers and nozzle pressure ratios were computed and tests were then conducted using this test section. The passage times of the primary and secondary shock waves were measured with a thin-film platinum heat gage probe.

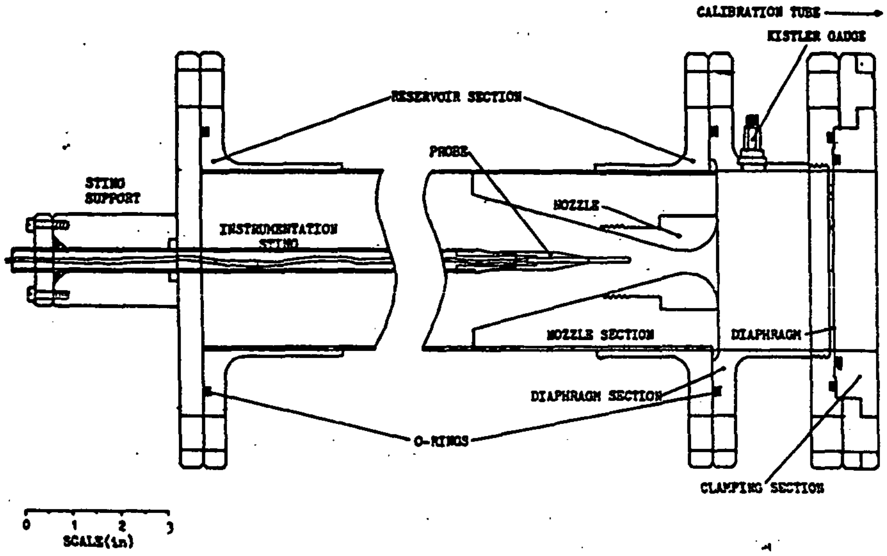


Figure 22 Test Section With Conical Nozzle

Tests were conducted in the RPI Low Pressure Shock Tube (Figure 1), and the test section is shown in Figure 22. Tests were made with and without the reflection diaphragm for an incident shock Mach number range of 1.25 to 3.0, and the heat gage probe was located at various distances from the nozzle throat.

The computer results [40,41] for the flow starting process without the diaphragm and for an incident shock Mach number of 2.5 are presented in Figure 23. The variation of pressure with axial distance and time in this figure indicates the movement of the primary and secondary shock waves down the conical nozzle with time.

The passage times for the primary and secondary shock waves at various distances from the nozzle throat were determined for an incident shock Mach number of 2 and without a reflection diaphragm [39,40], and the results are presented in Figure 24. The agreement of the experimental data with the computer

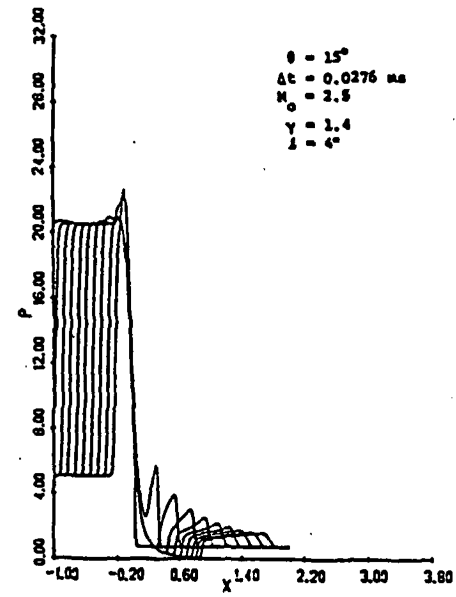


Figure 23 Nondimensional Pressure Distribution for Various Times Following Deblocking

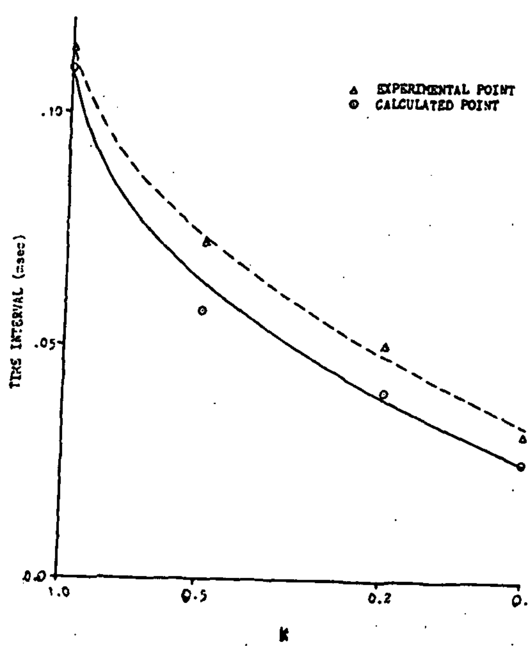


Figure 24 Interval Between Shock Waves - Nozzle Pressure Survey

results for the time interval between the shock wave is quite close, as well as the predictions of the shock wave location. Thus, it can be seen that the computer program [40,41] can predict the primary shock wave velocities through the nozzle.

5. CONCLUSIONS

Low Pressure (7.80 atm) and high pressure (124 atm) shock tubes with an inside diameter of 10.16 cm and a length of 21.34 m were designed and constructed for measuring heat transfer over a temperature range of 390° to 2500°K and pressures of 1.68 to 42 atm in the test section. Heat transfer data for circular cylinders and flat plate with and without pressure gradient have been obtained and heat flux data over flat plate have been measured up to 2580°K.

A square test section with adjustable top and bottom walls was constructed and the experimental pressure data indicated that the converging top and bottom walls produced the desired favorable pressure gradient over the flat plate. Flow Mach numbers of 0.15 and 0.45 after the reflected shock wave were produced by varying the deflection of the walls.

Painted thin-film platinum heat gages with response time of a few microseconds were fabricated and used to measure the local heat transfer rate and to determine the type of boundary layer existing over the surface. A computer program was developed to determine the heat flux from the heat gage output.

Department of Energy loaned to RPI a water cooled gas turbine nozzle cascade which is attached to the high pressure shock tube within the dump tank. Heat transfer distributions over the pressure and suction surfaces for the nozzle airfoil can be determined at high temperatures and pressures.

Laminar, transition, and turbulent boundary layer heat fluxes were determined on the low pressure shock tube wall with the platinum heat gages. The variation with time of the heat flux for laminar boundary layer after the shock agreed with Mirel's prediction.

Stagnation point heat flux was measured for various diameter circular cylinders in the low and high pressure shock tubes for a range of Reynolds numbers and free stream Mach numbers of 0.15 and 0.45 at temperatures of 420°K and 920°K temperatures. The heat flux data indicated no significant pressure gradient effect at low temperatures and agreed reasonably well with Eckerts laminar theory. But at high temperature the stagnation heat fluxes were greater then Van Driest's turbulent theory.

Flat plate heat transfer measurements were made for free-stream temperatures of 350° to 2580°K, pressure gradients, Reynolds numbers, and Mach numbers anticipated in advanced gas turbines. Preliminary local fluxes for laminar, transition and turubulent boundary layers were determined for various flow conditions and correlated with the laminar and turbulent theories.

Heat fluxes over a flat plate without pressure gradient were determined for laminar, transition, and turbulent boundary layers for a flow Mach number of 0.12 and temperatures of 370° and 1000°K. At the low temperature the heat fluxes agreed well with the laminar and turbulent theories. For the high temperature the heat fluxes were greater than the turbulent theory but this difference may be due to the fact that the Reynolds numbers were too low to produce fully developed turbulent boundary layer. Additional tests are necessary to determine the contribution of the turbulent density fluctuations to the heat transfer at high temperatures.

The heat flux distributions were determined in the junction region of a circular cylinder mounted normal to a flat plate for various Reynolds numbers and a flow Mach number of 0.14, which corresponds to the vane inlet Mach number for jet engines. Large effects of the cylinder on the flat plate heat transfer were felt in the region 1/4 to 3/4 diameter ahead of the cylinder which depended on the Reynolds number.

A probe with two hot wires was developed and used to measure the turbulence intensity behind the incident and reflected shock waves. Turbulence level is necessary to correlate the heat transfer data for circular cylinders and flat plates with existing theories. Preliminary hot wire results indicated low turbulence intensity behind the incident shock wave and higher intensity after the reflected shock wave.

Measurements of the time intervals of passage of the primary and secondary shock waves in the convergent-divergent nozzle of a shock tunnel were made and the data correlated well with the finite difference computer program predictions. The flow starting process in the nozzle depended on the pressure ratio across the nozzle.

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Final report. Project Manager, Robert W. Graham, Aerothermodynamics and Fuels Division. NASA Lewis Research Center, Cleveland, Ohio 44135.

16. Abstract

Low and high pressure shock tubes were designed and constructed for the purpose of obtaining heat transfer data over a temperature range of 390 to 2500 K, pressures of 0.3 to 42 atm, and Mach numbers of 0.15 to 1.5 with and without pressure gradient. A square test section with adjustable top and bottom walls was constructed to produce the favorable and adverse pressure gradient over the flate plate with heat gages. A water cooled gas turbine nozzle cascade which is attached to the high pressure shock tube was obtained to measure the heat flux over pressure and suction surfaces. Thin-film platinum heat gages with a response time of a few microseconds were developed and used to measure the heat flux for laminar, transition, and turbulent boundary layers. The laminar boundary heat flux on the shock tube wall agreed with Mirel's flat plate theory. Stagnation point heat transfer for circular cylinders at low temperature compared with the theoretical prediction, but for a gas temperature of 922 K the heat fluxes were higher than the predicted values. Preliminary flat plate heat transfer data were measured for laminar, transition, and turbulent boundary layers with and without pressure gradients for free-stream temperatures of 350 to 2575 K and flow Mach numbers of 0.11 to 1.9. The experimental heat flux data were correlated with the laminar and turbulent theories and the agreement was good at low temperatures which was not the case for higher temperatures. The heat flux distributions were determined in the junction region of a circular cylinder mounted normal to a flat plate for various Reynolds numbers and a flow Mach number of 0.15. Large effects of the cylinder on the flate plate heat transfer were felt in the region close to the cylinder. A hot-wire probe was used to measure the turbulence level after the incident and reflected shock wave and after the reflected shock wave the turbulence increased to six percent. Measurements of the time intervals of passage of primary and secondary shock waves in the convergent-divergent nozzle were made and the results agreed with a finite difference computer program prediction.

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