EXPERIMENTAL OBSERVATION OF TRANSITION BEHAVIOR ON A FLAT PLATE

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INTRODUCTION

In studying transition behavior a shock tube and tunnel were used to produce high temperatures, and thin-film platinum heat gauges were used to measure local heat flux as well as to detect the transition of the laminar boundary layer over a flat plate and a cone. Initial investigations were conducted in the hypersonic shock tunnel to obtain high-temperature information for the development of an ICBM nose cone. Shock Mach numbers as large as 50 with a temperature of 15 000 K after the incident wave were produced in the driven tube (ref. 1). And now we are using shock tubes to investigate the heat transfer over various surfaces to 2500 K for the development of future gas turbines.

The shock tube is an ideal method of producing high-temperature gas for which the equilibrium properties are fairly well known. Thin-film platinum heat gauges are either sputtered (ref. 2) or painted (ref. 3). These gauges, with a response time of 1 μsec are effective in detecting the transition of the laminar boundary layer (refs. 3 to 6) as well as in measuring the local heat flux.

I will describe the shock tubes that were constructed at Rensselaer Polytechnic Institute, under the sponsorship of Dr. R. Graham of the NASA Lewis Research Center and Dr. W. Aung of the National Science Foundation, to obtain heat transfer information over a temperature range 360 to 2500 K for the design of advanced jet engines. I will describe the thin-film heat gauges that were developed and some of the experimental heat transfer results for laminar, transition, and turbulent boundary layers. The transition phenomenon has been investigated with heat gauges since 1960 in shock tubes and tunnels; some of the results are presented in references 2 to 8.

DESCRIPTION OF SHOCK TUBES AND HEAT GAUGES

RPI Low- and High-Pressure Shock Tubes

RPI low- and high-pressure shock tubes (refs. 3 and 7) are shown in figure 1. Both of these shock tubes are 21 m (70 ft) long with a diameter of 10.2 cm (4 in), and the drivers are 3 m (10 ft) long with 18.3 (60-ft) long driven tubes. The low-pressure tube is copper tubing with a maximum pressure of 690 kPa (100 psi); the maximum pressure for the stainless steel tube is 12 400 kPa (1800 psi). In the high-pressure shock tube it is possible to obtain heat transfer data at a pressure of 4150 kPa (600 psi) and temperatures to 2500 K for future water-cooled gas turbines for generating power. At the end of the high-pressure shock tube a nozzle with an exit diameter of 60 cm (24 in) can be attached. With this hypersonic nozzle a flow Mach number of 25 in air has been achieved (ref. 1).

At present we are using both shock tubes to obtain heat transfer over flat plates with and without pressure gradients, stagnation point heat transfer for circular cylinders, and heat transfer in gas turbine vanes. The results obtained in the low-pressure shock tube on the laminar boundary-layer transition and heat transfer for laminar, transition, and turbulent boundary layers will be discussed.

To orient people who are not familiar with the shock tube technique, the operation of the shock tube is briefly described. A shock tube (x-t) diagram is presented in figure 2. The pressures in the driver and driven tubes are selected to produce the desired heated air in the test section (ref. 9). The locations of the shock wave, the contact surface, and the head of the rarefaction wave in the driver after the diaphragm separating the driver and the driven tube is broken and at time $t = t_1$ are shown in figure 2. Across the incident shock wave the gas is heated to temperature T_2, compressed to pressure P_2, and imparted flow velocity V_2. The incident shock wave is reflected from the end of the tube (as shown in fig. 2 for time $t = t_2$). After the reflected shock wave the gas is further heated to temperature T_5 and compressed to pressure P_5. For a solid end wall the flow velocity after the reflected shock wave is zero and is a function of the ratio of the open area in the end wall to the shock tube cross-sectional area. By selecting this area ratio it is possible to produce the desired flow Mach numbers of 0.15 and 0.45 after the reflected shock wave to simulate the inlet flow Mach numbers for gas turbine vanes and blades, respectively.

Heat transfer distribution over bodies can be determined in region 2 after the incident shock wave or in region 5 after the reflected shock wave from the reflection plate (refs. 3, 7, and 8). The duration of the test time in the test section is limited by the arrival of the expansion wave from the driver end (fig. 2). The shock tube is a very effective method for producing high-temperature and high-pressure gas for which the thermodynamic properties are well defined for the equilibrium condition.

A test section with a square cross section with area equal to that of the 10.2-cm (4-in) diameter driven tube was constructed and attached to the end of the low-pressure shock tube. This section was constructed with adjustable top and bottom walls to permit testing with a pressure gradient over the flat plate (fig. 3) for simulating the accelerating flow over vanes and blades. Windows are mounted to side walls for taking schlieren photographs of the boundary layer over the flat plate.

Piezoelectric pressure transducers (e.g., fig. 4) are located in the test section to measure the pressure after the incident and reflected shock waves. The response time of the quartz pressure gauges is approximately 10 μsec.

Thin-film platinum heat gauges were constructed by painting the platinum on the Pyrex substrate, which was placed in an oven to evaporate the solvent and to cause the platinum to adhere to the Pyrex. The platinum is approximately 104 Å thick and has a resistance of about 15 Ω. At the General Electric Research and Development Center the heat gauges were fabricated by sputtering platinum to the glass to a thickness of approximately 300 Å (refs. 2 and 4 to 6). These platinum heat gauges have a response time of a few microseconds and a current of approximately 30 mA. The heat gauge power supply schematic is shown in figure 5. A change in the surface temperature due to the heat transfer increases the resistance of the platinum and causes a voltage change that is recorded on the oscilloscope. A computer is used to calculate the heat flux from the voltage trace (refs. 2, 3, and 7).

A flat plate with a span of 9 cm (3.55 in) and with thin-film platinum heat gauges is shown in figure 6. The platinum film is about 0.305 cm (0.12 in) long and 0.15 cm (0.06 in) wide and the diameter of the glass disk is 0.13 cm (0.05 in). With these heat gauges it is possible to measure the local heat flux as well as to detect the transition of the laminar boundary layer.

EXPERIMENTAL RESULTS AND DISCUSSION

For evaluating the local heat transfer rate it is necessary to determine the heat gauge constant β, which is defined as

$$\beta = \frac{(\rho C_p k)_b}{\alpha} \tag{1}$$

where ρ, C_p, and k are the density, specific heat, and thermal conductivity of the backing material and α is the thermal resistivity of platinum. The gauge constants β are used in calculating the heat transfer for each gauge. The gauge constant was obtained by accurately evaluating the initial voltage jump that is displaced on an oscilloscope trace when the incident shock wave passes over the heat gauge. The expression for the heat gauge constant is

$$\beta = \frac{I_0 R_0}{\Delta E} k_w [T_w - T_{wi}] S'(0) \left[\frac{\pi}{2} \frac{1}{\nu_w} \frac{U_2}{U_1}\right]^{\frac{1}{2}} \tag{2}$$

where I_0 and R_0 are initial gauge current and resistance, ΔE the step change in voltage, k_w the thermal conductivity at the wall, T_w the wall temperature, T_{wi} the insulated wall temperature, ν_w the kinematic viscosity at the wall, U_2 the velocity behind the normal stationary shock, U_1 the free-stream velocity, and $S'(0)$ a coefficient tabulated in reference 10.

The heat gauge traces for laminar, transition, and turbulent boundary layers are presented in figure 7 for the flow over a flat plate (fig. 3), with a flow Mach number after a reflected shock wave of 0.12 and a gas temperature of 405 K. The laminar trace for a low Reynolds number (fig. 7(a)) is smooth after the passage of the reflected shock wave. As the Reynolds number increases, the transition trace (fig. 7(b)) shows oscillations associated with the passage of turbulent bursts over the heat gauge. For high Reynolds number with a turbulent boundary layer the heat gauge trace (fig. 7(c)) has a relatively smooth parabolic shape with a superimposed high-frequency oscillation caused by the presence of turbulent eddies. Similar heat gauge traces were observed for laminar, transition, and turbulent boundary layers over a 10° cone of 1.2-m (4-ft) length at Mach 10 (refs. 4 and 5) and of 2.4-m (8-ft) length for Mach 16 (ref. 6) and on the shock tube wall (ref. 3). The turbulent trace indicates the decrease in voltage output from the initiation of the flow over the flat plate. This decrease is due to the expansion wave from the shock tube driver arriving in the test section (fig. 2).

The experimental heat transfer rate was reduced from the voltage-time oscilloscope traces presented in figure 7. The voltage traces were then digitized by using a Talos digitizer to evaluate the corresponding local heat flux as a function of time and the following equation derived by Vidal (ref. 11) was used:

$$q(t) = \frac{\beta\sqrt{\pi}}{2I_0R_0}\left\{\frac{E(t)}{\sqrt{t}} + \frac{1}{\pi\sqrt{t}}\int_0^t \frac{\left[\sqrt{t}\cdot E(t)\right] - \left[\sqrt{t}\cdot E(\tau)\right]}{(t-\tau)^{3/2}}\,d\tau\right\} \tag{3}$$

This equation is solved by numerical integration through a computer program on the IBM 3033 and by using the digitized voltage data to obtain the experimental value for the local heat transfer rate, Stanton, and Nusselt numbers as functions of time.

The experimental results for the laminar boundary-layer heat transfer to the shock tube wall (ref. 3) and flat plate (refs. 7 and 8) agreed well with Mirels' (ref. 10) prediction for the variation with time of the heat flux after the passage of the incident shock wave (as shown in fig. 8 for the shock tube wall). Similar laminar heat transfer variation with time after the incident shock wave was observed for the flat plate (refs. 7 and 8) at low Reynolds numbers.

The digitized voltage trace and the corresponding local heat flux for laminar, transition, and turbulent boundary layers for the heat gauge located 7.95 cm (5.16 in) from the leading edge of the flat plate are presented in figures 9(a) to (c), respectively, for a gas temperature of 416 K, Mach 0.12, and a wall-to-gas temperature 0.71 (ref. 12). A laminar boundary layer was observed for a Reynolds number of 9.96×10^3, and the digitized voltage trace and the heat flux are presented in figure 9(a). The laminar boundary layer over the plate after the passage of a reflected shock wave is established in approximately 1 msec. There is a scatter in the heat flux result due to the digitizing of the heat gauge voltage trace. For a Reynolds number of 4.72×10^4 the boundary layer over the plate is in the transition regime with large variations in the local heat flux (fig. 9(b)). These large fluctuations for the transition boundary layer are caused by the passage of turbulent bursts over the heat gauge (as observed previously in refs. 3 to 8). The fast response time of the thin-film platinum heat gauge detects the variation in the heat flux to the surface. A fully developed turbulent boundary layer was observed for a Reynolds number of 1.32×10^6 (fig. 9(c)). The digitized voltage trace is relatively smooth and decreases after 12 msec when the expansion wave arrives from the driver section (fig. 2). The corresponding heat flux is nearly constant for the duration of the test gas over the heat gauge. Once the boundary layer is fully turbulent over the plate, a sublayer exists near the surface of the plate, and consequently the voltage trace is smooth with small-amplitude, high-frequency fluctuations (refs. 3 to 8).

CONCLUSIONS

A shock tube is a useful device to produce shock-heated air over a temperature range of 350 to 2500 K. By using a reflected shock wave technique it is possible to produce flow Mach numbers of 0.12 and 0.45 to simulate the flow

Mach numbers for future gas turbine vanes and blades, respectively. And it is possible to produce pressures as high as 40 atm after the reflected shock wave.

Going from a circular driven tube to a square test section and deflecting the top and bottom walls permitted the investigation of the effects of pressure gradient on the heat transfer to a flat plate. This method simulates the accelerating flow through vanes and blades.

Thin-film platinum heat gauges with a response time of approximately 1 μsec can be used to detect the transition phenomena for the laminar boundary layer in subsonic to hypersonic flows. A computer is used to calculate from the heat gauge voltage output the local heat flux for laminar, transition, and turbulent boundary layers for gas temperatures as high as 2500 K.

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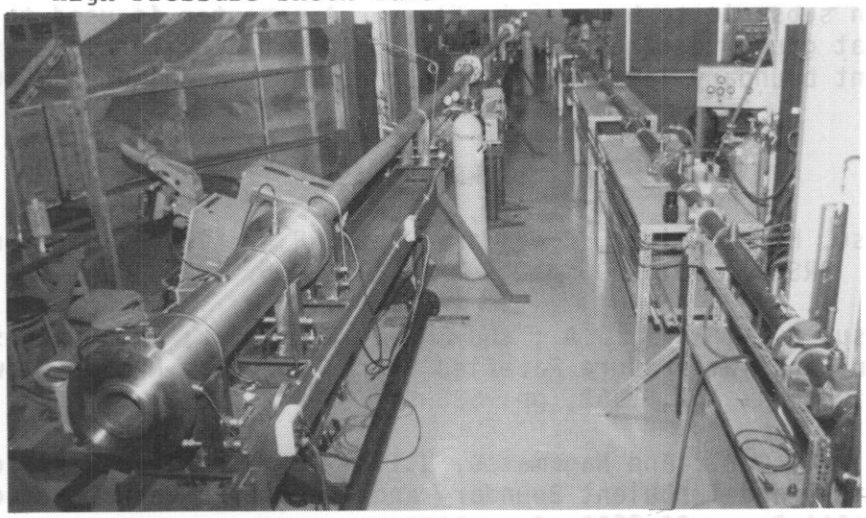
High Pressure Shock Tube



Low Pressure Shock Tube

Figure 1. - RPI shock tube.

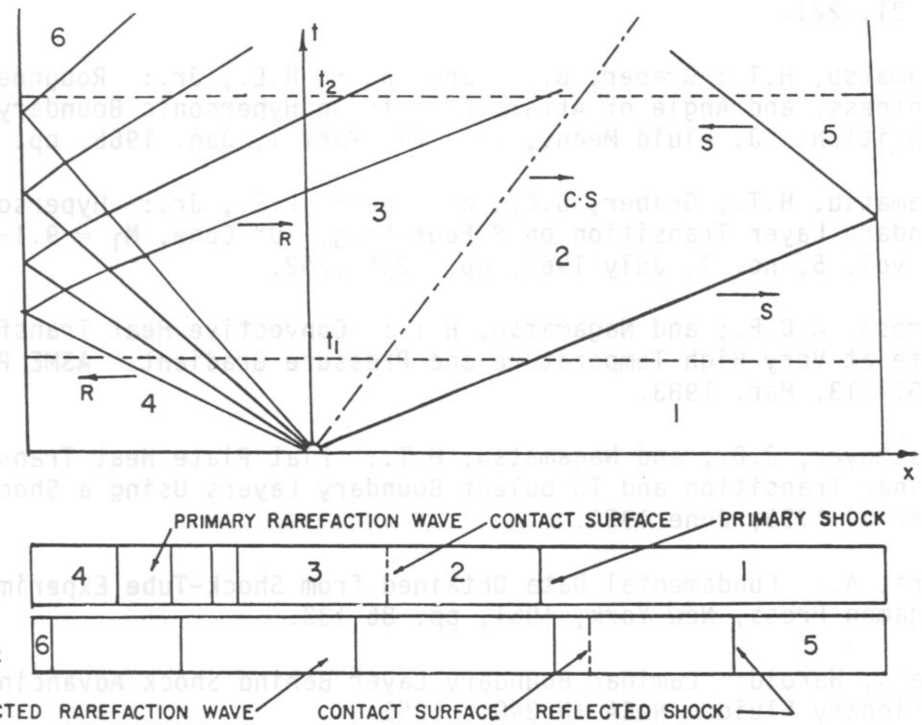


Figure 2. - Ideal wave system in a shock tube.

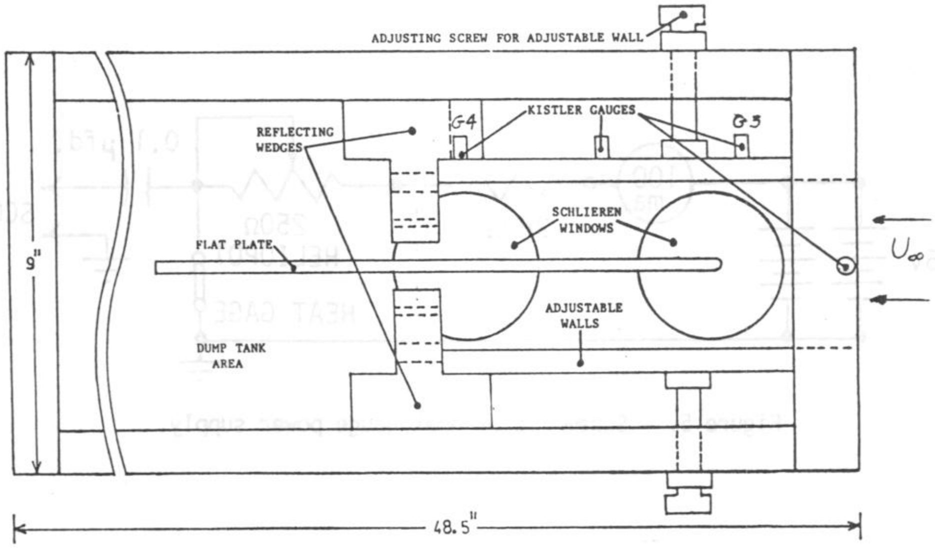


Figure 3. - Schematic of square test section.

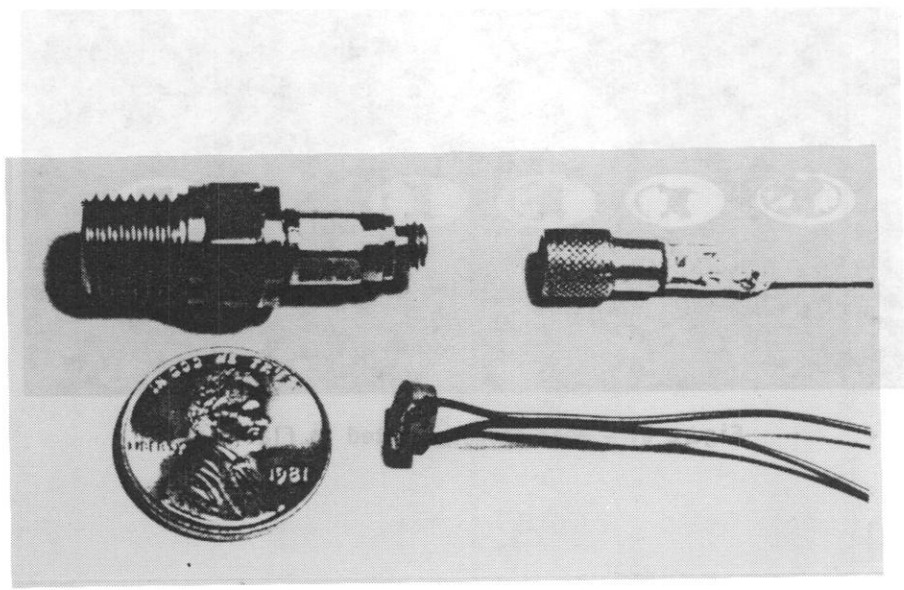


Figure 4. - Kistler type-603 quartz pressure gauge and lead-zirconate-titanate piezoelectric crystal.

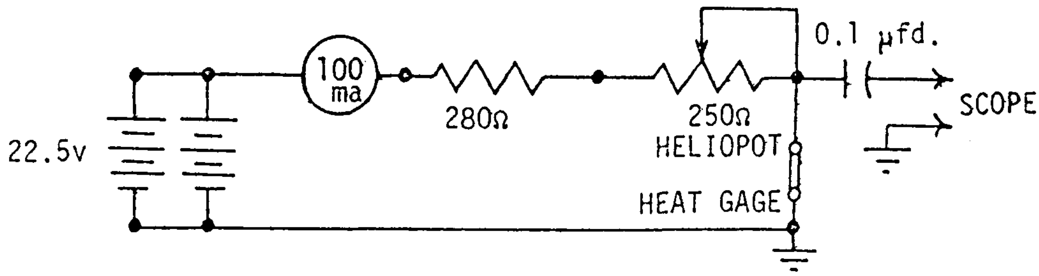


Figure 5. - Schematic of heat gauge power supply.

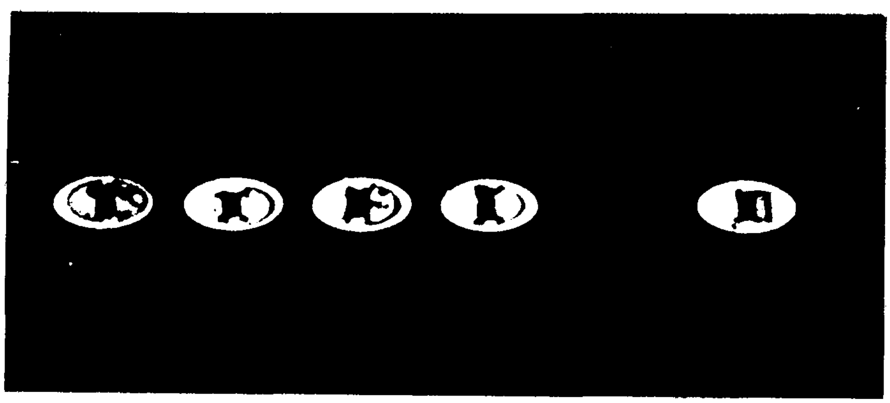
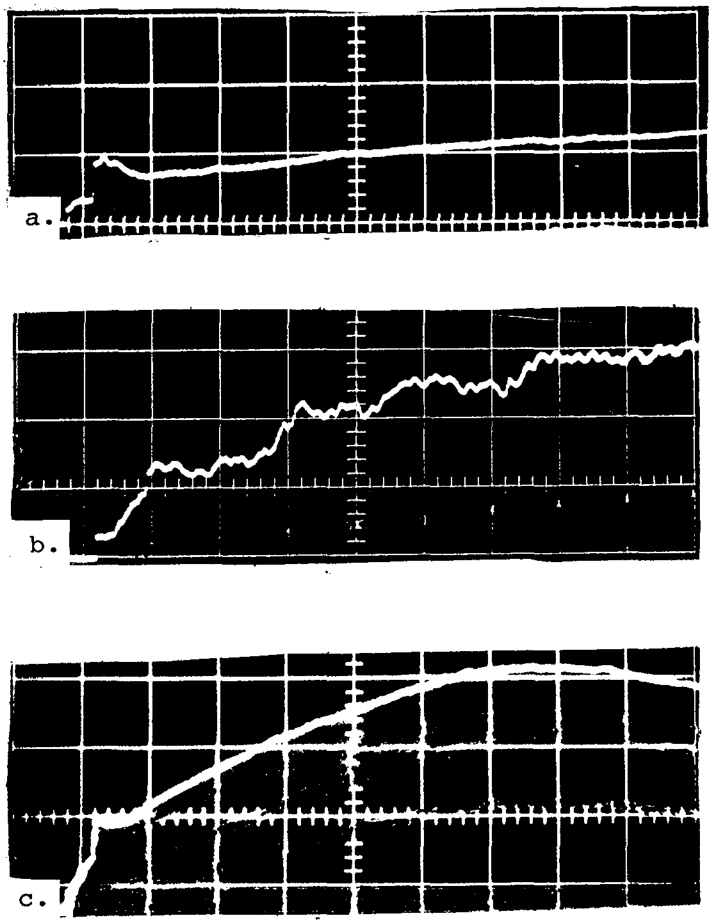


Figure 6. - Heat gauges mounted in flat plate.



(a) Laminar, Re/cm = 698.
(b) Transition, Re/cm = 3000.
(c) Turbulent, Re/cm = 93 200.

Figure 7. - Heat gauge traces for laminar, transition, and turbulent boundary layers. $M = 0.12$; $T_g = 405$ K.

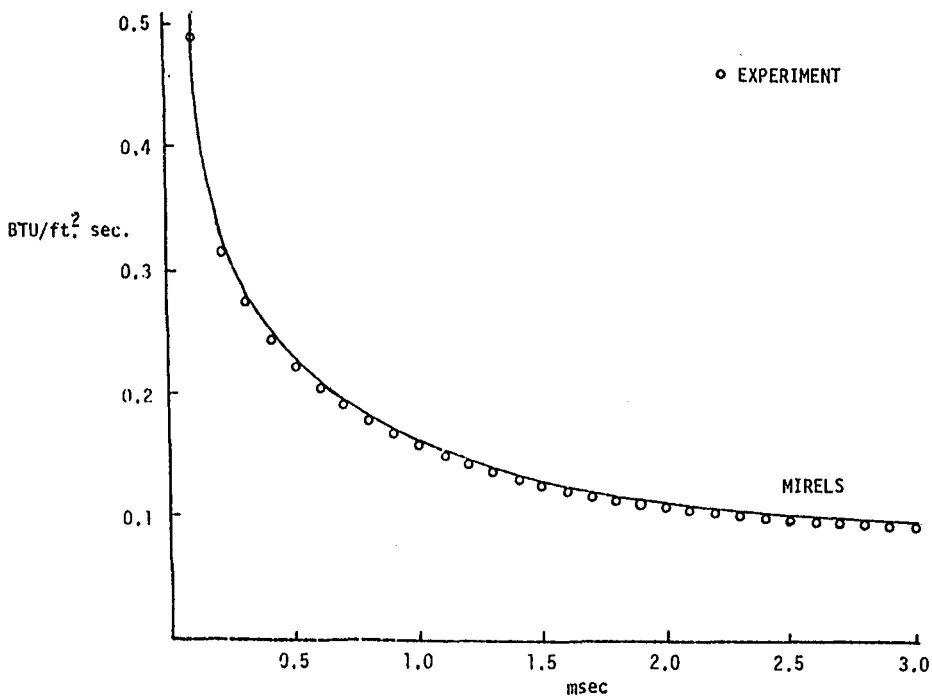
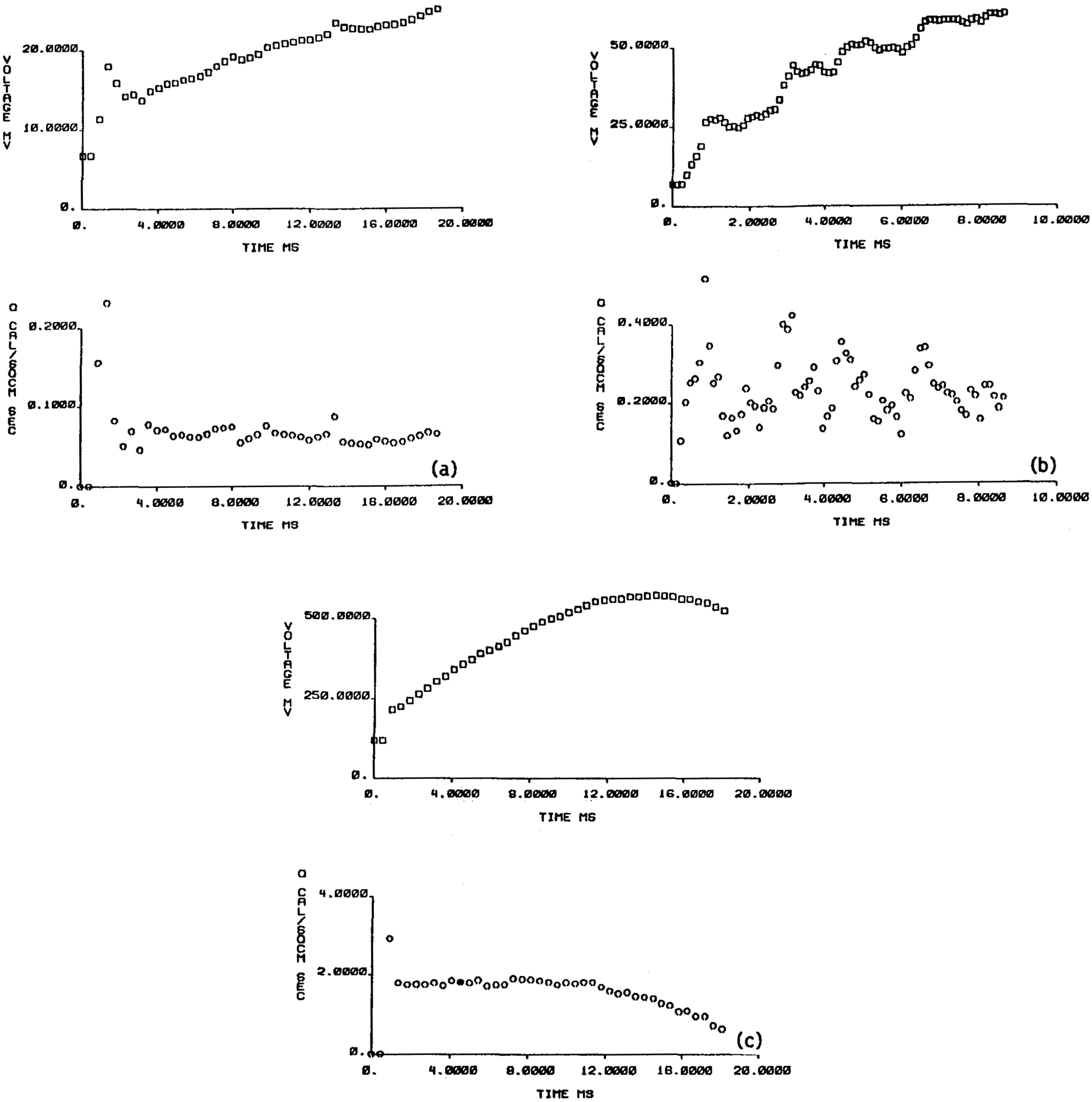


Figure 8. - Local heat transfer rate for laminar boundary layer. $M_2 = 0.697$; $T_2 = 409$ K; Re/cm = 2.48×10^3.



(a) T_g = 416 K; Re = 9.96×10^6.
(b) T_g = 405 K; Re = 4.72×10^4.
(c) T_g = 405 K; Re = 1.32×10^6.

Figure 9. - Digitized voltage traces and heat transfer for laminar boundary layer. M = 0.12.