INSTRUMENTATION AND CALIBRATION OF UTIAS
11 in. x 15 in. HYPERSONIC SHOCK TUNNEL

by

Y. Y. Chan
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Fig. 21  Abscissa should be $M_St$

Fig. 23  Abscissa should be $M_S$

Fig. 37  Abscissa should be $Re_b = \frac{P_\infty U_\infty R_b}{\mu_e}$

Fig. 46  (a) Initial Tunnel Pressure: 5 $\mu\text{Hg.}$
(b) Initial Tunnel Pressure: 100 $\mu\text{Hg.}$
ACKNOWLEDGEMENT

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SUMMARY

An outline of the problems encountered in the calibration of the UTIAS 11 in. x 15 in. Hypersonic Shock Tunnel and some of the solutions are presented.

An impulsively heated tungsten-wire technique was successfully developed to obtain detonation-free, smooth, and very repeatable combustion by burning stoichiometric mixtures of oxygen and hydrogen, diluted with helium. It was found that tailored shock Mach numbers as high as $M_s = 9.5$ were achieved.

A test section flow calibration was done by measuring wall static pressures, pitot pressures and stagnation point heat-transfer rates. Qualitative pressure-time histories of static and pitot pressures obtained from theoretical considerations were used to interpret the experimental data as well as the nozzle starting time and testing times. Schlieren pictures of the flows over a flat plate and a right circular cylinder are also presented.

Methods of predicting the test-section flow quantities in the different flow regimes are indicated. Nozzle starting processes are explained by utilizing available theories. The reservoir and test section conditions were usually $P_o = 30$ atmospheres to 90 atmospheres; $T_o = 3000^\circ K$ to $4500^\circ K$; $\gamma_e = 1.00$ to 1.35; $\rho_\infty = 0.003$ to 0.02 $\text{lbm}/\text{ft}^3$, $h_e = 0.1$ to 0.8 $\text{psi}$, $M_\infty = 8.0$ to 13.8, $Re/ft = 1.5 \times 10^4$ to $3.82 \times 10^3$ (based on flow field behind the shock).
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NOTATIONS

\( \alpha_{41} \)
Ratio of Speed of Sound of driver gas to driven gas

A
Cross sectional area of the nozzle at any location

\( A_s \)
Shock tube cross sectional area

\( A_1, B_1, C_1, D_1 \)
See Fig. 54

C
Contact Surface

\( C_v \)
Specific heat at constant volume

h
Static enthalpy

\( h_0 \)
Stagnation enthalpy (Reservoir enthalpy)

K
Amplification factor of the preamplifier

l
See Fig. 27

M
Flow Mach number at any point in the nozzle

\( M_s \)
Shock Mach number

\( M_{st} \)
Tailored Shock Mach number

\( \vec{M_w} \)
Mach wave

\( M_\infty \)
Test section Mach number

\( N_i \)
No. of degrees of freedom in the active mode

p
Static pressure at any point in the nozzle

\( P, P^* \)
Gauge surface temperature outputs when the gauge is pulsed in air and in water respectively

\( P_i \)
Initial pressure in the test section before a run

\( P_{i1}, P_{i2}, P_{i3} \)
Different initial pressures in the test section

\( P_t \)
Pitot pressure in the test section

\( P_o \)
Reservoir pressure (\( \equiv P_5 \))
\( p_1 \) Initial channel pressure

\( p_{4i} \) Initial charging pressure

\( p_5 \) Pressure at the end of shock tube after shock reflection

\( p_w \) Free stream test section pressure

\( q \) Heat transfer rate

\( R \) Gas constant \((1715 \text{ ft}^2/\text{sec}^2 ^{0}\text{F})\)

\( \overline{R} \) Rarefaction wave

\( RC \) Resistance - capacitance constant for the analogue network

\( R \) Universal Gas constant

\( S_0 \) Entropy

\( S_1 \) Defined in Fig. 39

\( S \) Shock

\( T \) Static temperature at any point in the test section

\( T_5 \) Temperature behind the reflected shock

\( t_1, t_2, t_3, t_4 \) See Fig. 42

\( t_5 \) See Fig. 41

\( u \) Voltage output from the analogue network

\( U_o \) Initial voltage across the gauge

\( V \) Velocity at any point in the nozzle

\( W \) Mass flow through the nozzle \((\text{lb/sec})\)

\( x, y, z \) See Figs. 42 and 45

\( y_i \) Mole fraction of \( i \)th species

\( z_0 \) Compressibility factor

\( \lambda \) Mass fraction
\( \gamma_f \)  
Frozen specific heat ratio

\( \alpha_p \)  
Coefficient for resistance of platinum film

\( \varepsilon \)  
Density ratio across the shock \( \left( \frac{\rho_o}{\rho_e} \right) \)

\( \rho \)  
Density at any point in the nozzle

\( \rho_e \)  
Density behind the shock

\( \left( \rho_b c_{p_b} k_b \right)^{\frac{1}{2}} \)  
Calibration constant for the gauge backing material

\( \rho_\infty \)  
Free stream test section density
INTRODUCTION

During recent years, the shock tube and its modifications have been widely exploited as a device for the generation of hypersonic flows of short duration. These modifications of the shock tube called "shock tunnels" are able to simulate the temperatures and velocities encountered in hypersonic flight.

A basic type of shock tube consists of a constant-area duct separated by a diaphragm into regions of high and low pressure. The wave process begins with the destruction of the diaphragm, allowing a sudden expansion of the driver gas into the low pressure section. The gas in the low pressure chamber is compressed by a shock wave which, ideally, propagates through the low pressure chamber at constant velocity. Between the gas interface and the shock wave, there exists a region of steady flow which is at high temperature and pressure. At low diaphragm pressure ratios the Mach number of this flow is subsonic and at high pressure ratios the flow Mach number is supersonic.

The extension of this shock tube technique to hypersonic flows is limited by the maximum flow Mach number which can be attained between the shock and the interface. For example, the maximum flow Mach number that can be obtained in this manner in an ideal diatomic gas is 1.89, while the conventional shock tube is not capable of simulating hypersonic flows, the ability of the shock tube to heat gases to the conditions encountered in hypersonic flight makes the shock tube an attractive tool for hypersonic flow research.

Modification of the shock tube to escape the limitation imposed by the rather low Mach number obtainable increases the value of the shock tube as a tool for hypersonic research. At Cornell Aeronautical Laboratory, Buffalo, New York, it was observed that, if a diverging nozzle is placed at the end of the shock tube, the high stagnation temperature flow behind the shock wave can, in principle, be expanded to the desired test section Mach number and temperature. In this type of modification it was observed that the flow in the nozzle was not established instantly and the motion of the primary shock wave in passing through the nozzle sets up a system of waves that delayed the establishment of steady flow in the nozzle. The system of waves that are generated tend to move upstream against the flow and reduce the available testing time. By properly lowering the initial pressure in the nozzle, only weak disturbances will be generated. Therefore, at the end of the low pressure section, it is necessary to insert a weak secondary diaphragm between the end of the shock tube and the nozzle, shattering upon impact with the shock wave. However, in this type of modification the testing time is very short, limited by the arrival of the interface in the test section.
A modification called the "reflected shock tunnel" offers the possibility of increasing the testing time at least by one order of magnitude. In the reflected shock tunnel the downstream end of the shock tube is not opened directly, but is instead terminated by a conventional converging-diverging hypersonic nozzle. The ratio of the nozzle throat area to the shock tube cross section area is such that the primary shock wave is nearly completely reflected, creating a region of stagnant compressed air at the end of the low pressure section in the shock tube. This air is then allowed to expand through the hypersonic nozzle to the desired test section flow Mach number. The flow entering the nozzle may be considered to be steady until the reflection of the shock wave from the gas interface returns and strikes the nozzle entrance.

The testing time of the "reflected shock tunnel" is limited to the time interval between the arrival of the incident shock wave at the nozzle and the arrival of the wave generated by interaction of the reflected shock wave with the gas interface. However, at large contraction ratios all the air between the nozzle and the interface is not devoured by the nozzle during the test interval.

The available testing time of the reflected method could be markedly increased if the total amount of air processed by the reflected shock wave could be utilized. By properly controlling the initial conditions in the driver and driven sections, the state of gas behind the interface can be made such that the reflected shock will pass through the gas interface without reflection. Since the shock impedance on both sides of the gas interface must be clearly matched, this method is called the "tailored interface technique". The limiting wave which then determines the maximum available testing time is the downstream propagating wave that is generated by the bursting of the shock tube diaphragm. Analysis of the testing time of the shock tube based on the wave diagrams indicates testing times approximately 10 times greater than those which could be achieved with a conventional non-reflected shock tunnel. This method was first suggested by Hertzberg, Cornell Aeronautical Laboratory, Buffalo, New York.

The brief testing times make the problem of instrumentation extremely difficult. Shock tunnel measurements demand extremely rapid rise times, in addition to the high sensitivity normally required in hypersonic nozzle testing at rather low densities.

1. GENERAL DESCRIPTION OF THE SHOCK TUNNEL

The UTIAS 11 in. x 15 in. Hypersonic Shock Tunnel consists of two main parts, the shock tube driver and driven section and the wind
tunnel section. The general layout of the shock tunnel is shown in Fig. 1, and photographs are seen in Figs. 2 and 3. The combustion driver of the shock tube is 7' long and is installed in a concrete blockhouse for the safety of the operators. The channel or driven section is 15' 5" long and is coupled to the nozzle section of the wind tunnel by a coupling nut.

The wind-tunnel section consists of the expansion-nozzle system, the working section, and the receiving tanks, a total length of 22 ft. The control console and the measuring instruments are installed beside the shock tunnel as seen in Figs. 2 and 3.

1.1 Shock Tube Driver and Driven Sections

The driver section of the shock tube is 7 ft. long with a 2 in. x 2 in. internal cross section. The driven section has the same cross section and a length of 15 ft. 5 in. The cross section is formed from ½ inch thick steel plates encased in a steel pipe with Woods metal. The structure is shown in Fig. 4. Both sections are designed for a working pressure of 3,000 psi. with a factor of safety of six for the tube. (Ref. 1)

There are five ports in the driver section, seven in the driven section and two in the coupling section, for instrumentation and gas inlet and outlet. The two sections are mounted on rails and may be moved independently by compressed air actuators.

The ignition wire is held at one end by a modified automotive spark plug mounted in the end plate and at the other end by a support just behind the diaphragm. The diaphragm is made of stainless steel of thickness between 0.018" and 0.047", and has two grooves scribed at right angles at a depth which controls the bursting pressure. The details of diaphragm calibration are given in Section 4.4.

The end of the driven section is joined to the tunnel section by means of a coupling nut which has two instrument ports 1 ½ in. from the entrance to the primary nozzle. A Mylar diaphragm 0.002 in. thick is installed at this point in order that the tunnel section may be evacuated to low pressures.

1.2 Nozzle System

The nozzle system is employed in order to expand the gas at the end of the shock tube which is at high temperature and pressure to high Mach numbers required in the test section.

The nozzle system consists of three sections, as illustrated in Fig. 5. The first expansion is provided by a two dimensional convergent-divergent nozzle with a contraction area ratio of 10.7 to 1, and an expansion ratio of 49 to 1. The height of the nozzle is 1 ½ in. throughout its length, the
width increasing from 0.25 in. to 12.25 in. The second expansion is provided by a deflection plate at an angle of attack of -100°, and in the plane of the primary nozzle. The supersonic flow from the nozzle is turned by a corner expansion (Prandtl-Meyer expansion for a perfect gas). This plate serves the important purpose of preventing any particles such as diaphragm fragments from entering the terminal nozzle. While the supersonic flow is turned, the particles go straight through without going into the terminal nozzle and fall into the receiving tank. The terminal nozzle, also at an angle of 100°, is mounted behind the deflection plate and is simply a straight-sided two-dimensional nozzle with an included angle of 150°. The terminal flow Mach number is controlled by changing the height of the nozzle throat, the width remaining constant at 11 inches. Interchangeable throats of fixed geometry are available for changing the test section Mach number. The final test section size is 11 inches wide by 15 inches high, which is just at the end of terminal nozzle. Testing is usually done at this station. Model angle of attack can be changed by moving the model support assembly over an arch shaped sting.

1.3 Test Section and Receiving Tanks

The test section has rectangular observation windows 10 inches by 24 inches at either side, which may be replaced by steel windows, when taking static pressure measurements.

The test gas which by-passes the nozzle system flows into two receiving tanks on either side of the tunnel with volumes of about 12 cubic feet each. The test section is followed by a receiving tank of a volume of about 25 cubic feet. (see Fig. 2 and 3)

2. VACUUM SYSTEM

In order to minimize the nozzle starting time, the tunnel section downstream from the second diaphragm has to be evacuated to very low pressure. The vacuum system consists of a mechanical pump and a diffusion pump, illustrated in Fig. 6. The mechanical pump is a CVC type E-135 with a pumping speed of 80 CFM, and is used to lower the pressure of the tunnel to about 100 μHg. As the diffusion pump takes over the evacuation at low pressure, the mechanical pump is then used as the backing pump for the diffusion pump. The diffusion pump is CVC type PMC-1440, and a CVC chevron ring baffle, type BCR is installed at the inlet of the diffusion pump to prevent diffusion of pump oil vapour back into the tunnel. A manually operated quick-acting gate valve is installed between the tunnel and the pumping system. The gate valve is closed before the operation of the tunnel to protect the vacuum system from the high pressure blast generated during operation. The pressure inside the tunnel is measured by a Pirani gauge, type GP-145, or Stokes McLeod gauge. The complete system
can be evacuated to the order of 1 µHg. in 30 minutes.

The channel section of the shock tube is also evacuated by the same vacuum system. The driver section is evacuated by a small mechanical pump located inside the block house.

3. CONTROL SYSTEM

3.1 Valve System

The valve system for the driver and the channel is shown schematically in Fig. 7. The gases used in the combustion driver are led through the control console. Each gas line is controlled independently by a high pressure needle valve and the charging pressure is indicated by a Heise type pressure gauge. A vent valve is connected to the charging line in order to release the high pressure gas to atmosphere if necessary. The charging line leads to a system of remotely controlled valves which are operated by compressed air and are controlled electrically from the console. The main valve is used to cut off the charging line from the distribution line; the pump valve is used to protect the evacuation pump from the high pressure lines, while the vent valve, connected directly to the driver, releases the gases, if necessary, from the driver.

A compressed-air operated valve, which is controlled from the console, separates the channel from the evacuation line. The pressure, which is usually lower than one atmosphere, is indicated by two Wallace & Tiernan type absolute pressure gauges. If a working gas other than air is used it can be fed into the channel through the charging valve and filter after the evacuation is completed.

The complete operation of the shock tube, i.e. the evacuation, charging and firing can be controlled from the console outside the block house.

3.2 Remote Control and Interlocking Safety System

The electrical circuit diagram for the remotely controlled valves is shown in Fig. 8. The switches for the valves are mounted on the console. To ensure the positive opening or closing of the valves, the movements of the valve stems are sensed by micro-switches located at the valves. The indication of the micro-switches are displayed on the panel by neon lights of different colours.

In order to ensure that all valves are closed before the firing of the shock tube, an interlocking safety system is installed. The signal which triggers the firing can only be transmitted if all valves are closed as shown in Fig. 8. In order to prevent accidental firing, a 'ready' switch with an indicator light is installed in the firing signal line. The 'ready' switch has
to be ON before the firing signal can be transmitted. An alternative firing button is installed in parallel with the main one and is situated near the diffusion pump gate valve.

If an optical study of the flow field is required, the firing signal is then used to trigger the schlieren camera shutter. When the shutter is fully opened, it in turn triggers the firing circuit. This control switch is also mounted on the display-panel of the console.

3.3 Ignition System

To produce uniform combustion and to minimize the possibility of detonation in the driver, the stoichiometric mixture of hydrogen and oxygen diluted with helium is ignited by instantaneous heating of a tungsten wire (which is mounted at the centre of the driver) over its complete length. (The wire is a General Electric tungsten wire of 0.010 inches diameter.) A discharge from the capacitor bank (50μ, 3000v) passing through the wire with an energy of about 225 joules or 32 joules/ft. of wire, heats the wire to a bright red and is sufficient to ignite the mixture. The time taken to discharge the voltage to \( \frac{1}{2} \) of its initial value is 150 μsec and to 99% of its initial value is 700 μsecs. The electronic circuit diagram is shown in Fig. 9. The system can be fired by grounding the grid of the first thyratron 2P21. This firing system is connected to the interlocking valve control system and finally to the firing control on the control console. A more detailed description of the investigation of the ignition system is found in section 4.2 and 4.3.

3.4 Hydrogen Detectors

"Davis Diffusion Head Combustible Gas Alarm" system has been installed to detect excessive amounts of hydrogen in the laboratory during tunnel operation. This alarm system is made by Davis Instruments, Newark 4, New Jersey. There are three detection heads, two installed inside the block house and one near the tunnel nozzles. All the three detection heads are connected in series. The gas alarm is used to detect combustible gases up to "Lower Explosive Limit". When the combustible gases reach a concentration in air where a hazard is being formed, the Analyzer operates a relay which sounds an alarm. An exhaust fan installed in the block house is kept running during tunnel operation to avoid any build up of hydrogen in the block house.

4. SHOCK TUBE CALIBRATION

The shock tube consists of a 7' driver and a 15'5" channel and both channel and driver are of 2in. x 2in. internal cross section. The construction details have already been reported in Ref. 1. In calibrating the above shock tube, only the essential quantities for the operation of a reflected-type shock tunnel are measured.

4.1 Shock Tube Instrumentation

Kistler #603 and #605 pressure transducers are used to
measure the pressure immediately after shock reflection. The transducer is mounted in the coupling nut 1\(\frac{3}{4}\) inches upstream of the nozzle entrance (see Fig. 10a). These gauges are calibrated statically using a dead weight pressure tester made by Mansfield and Green Inc., Cleveland, Ohio. A typical calibration curve is given in Fig. 11. Another Kistler #603 pressure transducer is mounted at the driver wall to monitor the combustion pressure history. These transducers are used in conjunction with Kistler charge amplifiers and a Tektronix 565 oscilloscope to record the pressure histories.

Two barium titanate pressure transducers (BD-25) made by Atlantic Research Corporation are mounted on the shock tube walls one foot apart, the second gauge being 11 inches upstream of the nozzle entrance (Fig. 10a). These gauges have a response time of less than 1\(\mu\)sec. and have a quoted output of 28\(\text{mV/psi}\). A photograph of these sensors is shown in Fig. 10b. The outputs from these gauges are amplified in a dual channel pulse amplifier and used to start and stop a 'RACAL' time interval counter which has a counting accuracy of \(\pm 1\mu\)sec. Since the shock speed is measured over a small distance at the very end of the shock tube, the attenuation effects over this interval are negligible. The outputs from the pressure transducers have also been displayed on an oscilloscope without using the pulse amplifier and the shock speed calculated from this time interval. The difference between these two methods is less than the 3% accuracy of the oscilloscope. The time interval counter is therefore used for this measurement.

4.2 Ignition Systems for Combustion Drivers

The success of combustion drivers mainly depends on the way in which combustion is initiated. Point ignition was used in almost all the early methods, so that detonation was frequently experienced. The possibility of point ignition is completely eliminated in the heated wire technique by heating the wire throughout its length just enough to initiate burning of the combustible mixture. The wire contamination problem, which frequently accompanied the early methods, is entirely avoided by retaining the wire after combustion. The various ignition methods employed and their relative advantages and disadvantages are outlined in Table I.

It can be seen from Table I that the method employing a heated tungsten wire such as used at UTIAS has many advantages, whereas no disadvantages have been found to date.

4.3 Constant Volume Combustion Studies (Hangfires)

The combustion processes in the driver have been investigated by closing off the driver section by means of a thick plate and igniting various mixtures. Initially, the source of ignition consisted of an aluminium wire crimped every three inches and stretched along the centre line of the driver. The release of the energy from the capacitors (Sec. 3.3) exploded the wire at
the crimping points. The advantage of this technique is that ignition occurs at many points simultaneously, instead of at one point. If the latter occurs, the single deflagration wave can, under certain conditions, form into a detonation wave. Such a wave caused a rapid rise in pressure to a very high value, which could in some cases be sufficient to damage the driver tube. Detonation must therefore be avoided. Even if a diaphragm is present (which would fail before the driver tube) the sharp rise in pressure could cause a local failure or cause the diaphragm to be ruptured in such a way that pieces of it are torn off and projected along the shock tube causing damage to the tube walls, the model under test, and the nozzle throat.

It soon became clear that this ignition system, although utilised successfully by some investigators (Ref. 2) was not giving multi-point ignition at all times. Erratic combustion and detonations were being experienced even when the helium dilution was as high as 80%. The latter is an important parameter influencing the burning processes. Examples of erratic combustions and detonations are seen in Fig. 12.

The reasons for this may be either that the crimps are not sufficiently uniform along the wire or the wire was slightly damaged during installation causing the wire to explode at a smaller number of points. The close confines of the 2 in. x 2 in. driver makes the securing of the wire a difficult operation. Improvements were also sought in the gas mixing process by providing three inlets along the length of the driver.

Experiments with an impulsively heated tungsten wire (Ref. 3) were immediately more successful. In this method the energy used is chosen to be just sufficient to cause the wire to glow a bright red, uniformly along the wire. It can be seen that this method ensures ignition along a line, making detonations rare, provided that care is taken to ensure that no other path is available for the discharge. A more complete discussion is found in Ref. 4.

The combustion pressure histories obtained are shown in Figs. 13. It can be seen that quite smooth combustion is achieved with helium dilution as low as 65%. There are two successive runs on Fig. 13a, which were so similar that the two traces are almost completely superimposed. An added advantage was that the wire can be reused, as many as five times without encountering combustion problems. The pressure rises obtained at different initial pressures and dilutions are plotted in Fig. 14 and compared with the analysis of Ref. 47. The time to reach peak pressures is also plotted in Fig. 15. With dilutions of 60 and 65%, this time is difficult to ascertain, however, the general trend is clear and shows similar forms to that reported in Ref. 5.
4.4 Diaphragm Calibration

4.4.1 Brass Diaphragms

The diaphragms used are discs of 6 inches in diameter. Initially brass diaphragms 0.035 in. in thickness were used, these having been found to be satisfactory for earlier work (Ref. 1). However, in order to achieve consistent tailoring it was found necessary to maintain a higher order of accuracy of bursting pressure than was possible with the hand scribing technique employed at that time. Consequently, a programme of tight quality control was initiated. The diaphragms were scribed by machine using a milling cutter with a 'V' angle of 90 degrees and a tip radius of 0.020 inches. The brass diaphragms were scribed to various depths and then tested to bursting by means of the Diaphragm Hydraulic Test Rig constructed for the UTIAS 4 in. x 7 in. Hypersonic Shock Tube (Ref. 6). The bursting pressures obtained in these tests are shown in Fig. 16. Various depths were chosen and diaphragms then produced in quantity for use in the shock tube, random samples being taken and tested in the Hydraulic Test Rig. It was found that the diaphragms ruptured in the shock tube at pressures about 400 psi greater than in the test rig. However, good consistency was achieved. Typically, random samples from quantity production show standard deviations of about ± 10%. The reason for the greater apparent strength exhibited by the diaphragms when used in the shock tube may be attributed to increased moduli during impulsive loading. However, in the case of the brass diaphragms, which were several years old, there may have been an effect due to age hardening or work hardening caused by the machining. It is perhaps significant that no similar difference was noted in the case of the stainless steel diaphragms, discussed below.

4.4.2 Stainless Steel Diaphragms

At this time, problems in igniting the driver gases were causing occasional detonations. (See Sec. 4.3). These were sometimes sufficiently severe to cause the petals of the diaphragms to be torn off, causing considerable damage to the walls of the driven section and the primary nozzle. However at this stage stainless steel (Type #303) diaphragms were tried and found to burst much more cleanly, less metal being lost even when poor combustion was obtained.

These diaphragms were also tested in the hydraulic test rig, the results for a typical thickness are shown in Fig. 17.

A vacuum chuck is now used to hold the diaphragms flat whilst being machined and cutting depths are now being held to better than 0.0002 in. (see Ref. 6 for further details).

Photographs of scribed and burst diaphragms are seen in Figs. 18 and 19. The loss of portions of petals referred to above is clearly
4.5 Investigation of Tailoring Conditions

The concept of tailored operation was first suggested by Hertzberg in Ref. 7 and described in detail in Ref. 8. The main principle of 'Tailored-interface' operation is that the incident shock wave is reflected at the entrance to the convergent-divergent nozzle and the conditions of the driver and driven gases are matched (or 'tailored') so that no additional waves are created by the interaction of the reflected shock and interface (Ref. 9). Following the passage of the incident shock wave, both the driver and driven gases will be at equal pressure and velocity. The interface between the two gases is a discontinuity in temperature and density. If the reflected shock wave is to pass through the interface without creating any additional waves, the driver and driven gases must be at equal pressure and velocity behind the reflected shock. Therefore the condition for tailoring is that the pressure ratio and velocity change across the reflected shock be the same in both driver and driven gases. That is, only a reflected Mach wave is permissible after the shock refraction at the contact surface.

Recently, a thorough investigation of tailoring conditions for different gas combinations and geometries at the diaphragm station, has been reported in Refs. 10 and 11. The theoretical values from Refs. 10 and 11 were used to compare the experimental results in Sec. 4.5.1.

4.5.1 Constant-Volume Combustion

This type of combustion was used to obtain a range of tailoring shock Mach numbers in air. Constant volume combustion is ensured in every run by carefully calibrating the diaphragms and determining the actual pressure rise (see Secs. 4.3 and 4.4), for any given initial charging pressure and helium dilution, in a separate contained combustion run. Initial charging pressures of 100, 200, 300 and 450 psi. were used and helium dilution was varied from 80% to 65%, in steps of 5% at each initial charging pressure ($P_{di}$.)

The tailoring shock Mach number is that which gives a reflected pressure ($P_{r}$) history constant with time. This condition is achieved experimentally by varying channel pressure ($P_c$) for a given combination of charging pressure ($P_{di}$) and helium dilution. The initial channel pressures were varied from 180 mm to 40 mm of Hg. Some of the typical tailoring pressure histories as well as combustion pressure histories are given in Fig. 20. Experimental tailored shock Mach numbers are compared with theory (Ref. 11) in Fig. 21. It is shown in Ref. 11 that by using a strong shock approximation, the diaphragm speed of sound ratio varies almost linearly with the tailored shock Mach number ($MST$). Also, the curve
for \( \gamma_2 = \gamma_3 = 1.4 \) agrees very well with the one calculated by taking into account real gas effects (see Ref. 11, Fig. 4). The ratio of specific heats \( (\gamma_4) \) of the driver gas after combustion was computed from Ref. 47 for the range of initial charging pressures and helium dilutions that were used and an average value of 1.432 was obtained. These considerations motivated to take the theoretical curve from Ref. 11 of \( \alpha_{41} \) versus \( M_{ST} \) corresponding to \( \gamma_2 = \gamma_3 = 1.4 \), as given in Fig. 21. Since the attenuation of incident shock was not measured, the experimental points are plotted uncorrected for attenuation. The theoretical curve, shifted by an average value of 32%, is plotted as dotted line. This curve is in good agreement with the experimental results. Thus the average attenuation in this tube may be taken as about 32%, (i.e. 2% per foot of channel), which is the usual order of magnitude for constant-volume combustion driver (Ref. 12). For any given helium dilution the tailoring shock Mach number should be unique but there is a maximum variation of about \( \pm 0.50 \) in Mach number in the experimental results. This is probably due to real gas effects.

### 4.5.2 Constant-Volume, Constant-Pressure Combustion

Tailoring shock Mach numbers attainable from constant volume burning is limited by attenuation effects as shown in Fig. 21. In Ref. 11, it has been demonstrated that for the case of equal specific heat ratios of driver and driven gases, for perfect, inviscid gases, the interface, if tailored initially, remains tailored, whether the shock is attenuated or accelerated. (see also Plate 17, Ref. 13; this result has not been shown for imperfect and viscous, heat conducting flows.) Therefore, tailoring Mach numbers greater than those attainable by constant-volume combustion are possible, if the incident shock Mach number that is tailored initially could be accelerated by some technique.

It has been established in Refs. 14 and 15 that partial constant-pressure burning in the driver of the shock tube produces higher shock Mach numbers than those obtained by constant-volume burning. The technique of constant-pressure burning is quite simple. The driver volume is filled with the stoichiometric oxygen-hydrogen mixture diluted with helium but to a pressure such that the diaphragm will burst before the peak pressure is reached. The pressure which is increasing in the 'quasi-steady' region of the expanded driver accelerates the shock as it moves down the tube until combustion is complete. Afterwards, the shock stops accelerating and begins to decelerate (or attenuate) in the usual manner.

The above principle was used in the 2 in. x 2 in. shock tube to extend the range of tailored shock Mach numbers. For this mode of operation the following conditions were used:
1) A stainless steel diaphragm with 750 psi. rupture strength (in static tests)
2) 70% Helium dilution
3) Initial channel pressure of 25 mmHg.
4) Oxygen as test gas
5) Initial charging pressures of 80, 120, 160 and 200 psi.

During this series of tests reflected pressure ($P_5$), combustion pressure history ($P_4$) and shock Mach numbers were measured. Four typical oscilloscope traces corresponding to each initial charging pressure are shown in Fig. 22. There are two traces of $P_5$, one is with Kistler #603 and the other with #605, mounted in the opposite sides of the shock tube walls.

The following definition is useful to explain the effect of constant pressure burning: The ratio of pressure rise in "contained" constant volume combustion to the pressure rise at the instant of diaphragm opening can be called the "over-Pressure Combustion Ratio". Even though the same strength diaphragm is used in all the runs, the actual pressure at which the diaphragm opens in a given run seems to increase slightly with initial charging pressure. In calculating the constant pressure combustion ratio, the actual pressure from the oscilloscope traces ($P_4$) is used in each run. This ratio increases from 1.27 to 1.67 corresponding to the initial charging pressure from 80 to 120 psi and the shock Mach number changes from 8.18 to 9.85 (Fig. 22). The first three traces can be considered as almost tailored. There is a slight rise in the reflected pressure histories ($P_5$), which may be due to the arrival of the tail of the rarefaction wave from the diaphragm station. The last trace is slightly over-tailored, that is, a reflected shock wave results from the refraction process. Therefore, there seems to be a limit after which constant pressure burning is ineffective and the tailored shock Mach number can only be extended up to $M_s = 9.4$ by this technique. The corresponding theoretical value is $M_s = 10.75$ (Ref. 11). The difference between the theoretical and experimental tailored shock Mach numbers is not unreasonable, in view that the calculations were done for a perfect inviscid gas and heat loss to the driver walls during combustion is not taken into account in the theoretical analysis.

4.6 Comparison of Reflected Pressure and Diaphragm Pressure Ratio With Theory

During the investigation of tailoring conditions using constant volume combustion as well as constant-volume, constant-pressure combustion, reflected and combustion pressure histories were measured using Kistler #605 and #603 pressure transducers. Air was used as a test gas in the constant-volume combustion investigation. In Fig. 23 experimental reflected pressures are compared with theoretical values taken from Ref. 16.
Theoretical values of $P_5$ are obtained from charts corresponding to the measured shock Mach numbers and compared with the measured $P_5$ values. Since reflected pressure as well as the shock speed are measured near the end of the channel, there is no need to consider the attenuation of the incident shock. The reason for the discrepancy between the theory and experiment is given in the following section. In Fig. 24 diaphragm pressure ratios are compared with theoretical values taken from Ref. 11. Experimental points are plotted uncorrected for attenuation since no attenuation measurements were made.

Reflected pressure histories were also measured during investigation of tailoring conditions using constant-volume, constant-pressure combustion. For these tests oxygen was used as a test gas. In Fig. 25 experimental values are compared with theoretical values taken from Ref. 17. As found in air, the theoretical values are consistently larger than the experimental values when the transducer is mounted at the side wall of the shock tube. To explain this discrepancy the effect of boundary layer on the one-dimensional inviscid theory was examined. Mark (Ref. 18) and Rudinger (Ref. 19) investigated the interaction of the reflected shock wave with the boundary layer. More recently, Woods (Ref. 20) has re-examined this problem. He considers two distinct boundary layer effects, unlike previous works:

1. At any instant the fluxes of mass, momentum and energy into the plane of the shock, averaged over the whole width of the tube, are less than those in the inviscid core by amounts which depend on the thickness and state of the boundary layer. These defects must be taken into account in obtaining shock relations for the average flow quantities.

2. The inviscid core flow is itself nonuniform in the axial direction, so that the flow into the reflected shock, outside the boundary layer, also varies with time. In the previous works the first effect is neglected which is, however, of the same order of magnitude as the stream nonuniformity. Woods also postulates a model in which the flow in the boundary layer close to the wall has not sufficient stagnation pressure to be raised to the pressure behind the shock ($P_5$) when the reflected shock passes back through the oncoming flow. As a result the boundary layer as a whole separates from the wall and at the junction of the shock and the boundary layer a 'bubble' of entrained boundary layer air is carried forward at shock speed. This grows with time and should be taken into account in a treatment of the conservation equations. Considering all these effects, he predicts a general shape of $P_5$ at the end wall as shown in Fig. 26-a. A few reflected pressure histories which were measured by a transducer mounted at the side wall are shown in Fig. 26-b for comparison with the theoretical shape. It may be noted that the initial sharp jump in pressure is missing in the side wall pressure histories. Since the theoretical pressure history is at the end wall, those pressures were measured at the same conditions.
and are shown in Fig. 26-b. In these pressure histories, not only the initial sharp pressure rise appears but also the general shape of the pressure trace is very similar to that predicted. Also the initial pressure jump agrees very well with theory as shown in Fig. 25. Concluding, one can note that the end wall pressure histories agree with theory in general shape. The end wall pressure rise is in agreement with the theory and the pressure settles down to a lesser value rather quickly (see Fig. 26-b).

The reservoir temperature and enthalpy are taken from tables corresponding to measured shock Mach numbers. The measured pressure is usually used as a check, but in most of the cases the pressure at which the testing is done is less than the theoretical value due to shock boundary layer interaction and more than the theoretical value if over-tailored operation is employed. The theoretical reservoir temperature and enthalpy are corrected for these two effects by assuming isentropic expansion or compression and the gas is assumed inequilibrium at the reservoir.

4.7 Wave Diagram and Testing Time in the Shock Tube

Assuming perfect gas, a typical (x, t)-diagram for a shock Mach number $M_S = 9.5$, and 70% helium dilution is shown in Fig. 27. The useful testing time in the test section of the shock tunnel in the case of tailored operation is limited by one of the following factors depending on shock-tube dimensions and incident shock Mach number:

a) The unsteady rarefaction wave head reaching the end of the channel after reflection at the driver end.

b) The tail of the unsteady rarefaction wave travelling downstream from the diaphragm station and reaching the end of the channel.

c) Available mass flow through the nozzle.

Which of the above three is the limiting criterion depends on many factors, namely, shock Mach number, driver and channel lengths and channel size, nozzle throat size, and driver-gas speed of sound. Some theoretical considerations of these criteria are reported in Ref. 21.

In the 2 in. x 2 in. shock tube with a 7 ft. driver and a 15 ft. 5 in. channel the reflected rarefaction wave from the driver end limits the testing time in the shock Mach number range 6 to 10 and $P_1$ above 10mmHg, with 70 to 80% helium dilution. At very low initial channel pressures (less that 10mmHg) and shock Mach numbers greater than 10, the mass flow consideration overrides the other criteria. The method of estimation of testing time by mass flow consideration taking into account real gas effects is out-
lined in Appendix I. However, in this estimation, the true testing time is usually about 50% of the estimated value due to viscous effects and interface mixing.

5. SCHLIEREN PHOTOGRAPHS

5.1 Optical System

The conventional double mirror, single pass schlieren system consists of the following components mounted on separate adjustable mounting posts:

1) Spark source system
2) Two 10 inch diameter parabolic mirrors with a 72 inch focal length
3) A 2 3/4 inch diameter plane mirror
4) Cut-off knife edge
5) 40 inch focal length Achromatic lens
6) Wollensak 'Rapax synchronatic' shutter
7) Camera assembly

A schematic layout of the complete schlieren system which was previously used with the 5 in. x 7 in. UTIAS supersonic wind tunnel (Ref. 22), is shown in Fig. 28. The effective spark-source slit and the cut-off knife edge can be orientated either horizontally or vertically. The camera assembly and one parabolic mirror are shown in Fig. 29. The spark source consists of the main spark gap and a concentric ionization gap. A 2 μF, 2000V capacitor is directly connected across the main gap and is charged up to 2000V by the same power supply as used for the ignition system (see Fig. 9). The air gap between the main electrodes is ionized by using a current transformer and a thyratron control. In this way the timing of the spark can be controlled. The schlieren optical path is set up before the run by using a General Electric Microscope Illuminator (18A/T10/1P-6V) as a light source.

5.2 Schlieren Photographs

Schlieren photographs of flows over right circular cylinder model and a flat plate model are shown in Figs. 30 and 31.

The right circular cylinder of 1 1/2 in. diameter and 9 in. length was mounted in the 11 in. x 15 in. test section. The reservoir and test section conditions were $M_S = 6.7$, $p_5 = 975$ psi., $T_5 = 4200^\circ$K, $M_{po} = 7.9$. At first, only a self-luminosity picture was taken by moving the schlieren knife edge away from the field of view (Fig. 30a). The camera shutter was left open throughout the test and the photograph gives the integrated result
during the flow duration. It is possible to obtain schlieren and self-luminosity pictures simultaneously and this is shown in Fig. 30b. At this temperature the region of luminosity is very sharply defined at the front, indicating that the relaxation time to reach equilibrium after the shock wave is very short. As the flow expands around the cylinder and the temperature decreases the luminosity disappears. With a 'field stop' at the knife edge it is possible to obtain schlieren pictures with a more accurate representation of the luminous region. Two schlieren pictures with field stops of decreasing area, that is, permitting less light to go through at the knife edge are shown in Fig. 30c and 30d, respectively.

Schlieren pictures of a flat plate with a sharp leading edge and at two different angles of attack are shown in Fig. 31. The flow conditions are the same as that for cylinder pictures. Field stops were not used in this case. Luminosity is only significant at the model sting when stagnation conditions occur.

6. TEST SECTION FLOW CALIBRATION

6.1 Static Pressure Measurement

Static pressure at the wall, pitot pressure, and stagnation-point heat transfer were measured to determine the flow quantities, testing time, and the flow uniformity in the test section.

6.1.1 Wall Static Pressure Measurement

The static pressure transducers (see Appendix II), which have been designed to measure pressures as low as 0.001 psi., are very sensitive to vibration and shock, therefore special care has to be taken in shock mounting. They are rigidly mounted in a heavy rectangular steel block. This is in turn attached rigidly to another block and the whole assembly is mounted at the side wall of the tunnel test section with rubber cushions so that they are isolated from any vibrations from the tunnel. The steel block floats in the rectangular cut-out in the side wall of the test section (see Fig. 48). Soft rubber sheeting is inserted between the rectangular block and the cut-out so that there will be no direct metal-to-metal contact between the tunnel wall and the mounting. Even with this naive arrangement the transducer picks up the high-frequency flow-induced noise which is inherent in hypersonic testing. The noise was filtered out by using the built-in filter in the plug-in unit (Tektronix 3A3). With this arrangement, static pressures as low as 0.005 psi. were measured successfully. Typical static pressure traces are shown in Fig. 32.

6.1.2 Calibration Technique

The long time constant obtained with the cathode follower circuit allows a quasi-static calibration of the transducers. Rapid application of
the calibration pressure to the transducer is accomplished by an electro-pneumatic valve which operates very quickly. Detailed description of the calibrating device is given in Ref. 6. A typical calibrating pressure trace is shown in Fig. 32 and the calibration curve is shown in Fig. 33. The sensitivity of the transducers changed with time and therefore frequent calibrations are required.

6.1.3 Determination of Test Section Flow Quantities

Usually the hot pocket of gas at the end of the shock tube will be in vibrational and dissociational equilibrium. This gas is expanded in a system of nozzles to obtain hypersonic Mach numbers. Depending upon the reservoir conditions and nozzle geometry, the flow in the test section can assume one of the following states:

a) Equilibrium
b) Nonequilibrium
c) Frozen

For any given reservoir conditions and nozzle geometry a complete nonequilibrium analysis has to be done to determine the condition of the nozzle flow. This type of analysis for this nozzle system is underway and will be reported later. However, the flow may be assumed to be in equilibrium or frozen depending upon the reservoir pressure and temperature. A determination of the flow quantities is then simplified and the method of estimation is given in the following section.

a) Equilibrium Flow

The method of calculating the flow in the equilibrium case for the nozzle is straight-forward. The flow is assumed to be one-dimensional and free of viscous effects. This means that the thermodynamic state of the expanding gas can be determined as a function of the area ratio for a given reservoir condition.

The calculation then involves satisfying the conservation of mass and energy, and the equilibrium thermodynamic state relations as follows (Ref. 31)

\[ \rho VA = \rho^* V^* A^* \]  \hspace{1cm} (1)
\[ h + \frac{1}{2} v^2 = h_0 \]  \hspace{1cm} (2)
\[ \rho = f(h_0, s_0) \]  \hspace{1cm} (3)
where, \( f \) is the density, \( V \) is the velocity, \( A \) is the cross-sectional area, \( h \) is the enthalpy, and \( S \) is the entropy. The \((\quad)^*\) quantities refer to conditions at the throat, while the subscript \((\quad)_0\) refers to initial reservoir conditions. In addition to the above equations, the throat condition of \( V^* = a^* \) when \( A = A^* \) must be satisfied, where \( a^* \) is the velocity of sound given by

\[
a^* = g(h, s_0)
\]

The functions \( f \) and \( g \) and other similar relations among the thermodynamic properties are not analytic but must be obtained from equilibrium thermodynamic charts (Ref. 16). The above relations can be solved in an iterative manner to obtain the equilibrium solution. However, in the case of air, extensive data has been computed for a range of reservoir conditions and area ratios in Ref. 23, and this data is used to compute thermodynamic properties for equilibrium air.

b) Frozen Flow

Frozen flow is defined as one where the composition is fixed at its upstream reservoir value and wherein only translational and rotational energy modes equilibrate with all other inert modes fixed at the reservoir condition. Since this definition implies that no recombination or chemical reactions take place, any change in enthalpy of the gas is directly proportional to a change in the temperature. The assumption of the above flow model permits the calculation of an effective specific heat ratio defined by (Ref. 31)

\[
\gamma_f = 1 + \left( \frac{Z R}{C_v} \right)
\]

where \( Z \) is the compressibility factor and \( C_v \) is the specific heat at constant volume and is given by

\[
C_v = \frac{R}{2m_o} \sum_{i=1}^{n} \gamma_i N_i
\]

\( N_i = \) no. of degrees of freedom in the active mode \( (t + r) \) of the \( i \)th species.
\( \gamma_i = \) mole fraction of the \( i \)th species

For a diatomic gas,

\[
C_v = \frac{R}{2} \left( \frac{s + \alpha}{2} \right)
\]
where \( \alpha \) is the mass fraction.

Therefore,

\[
\gamma_f = \left[ \frac{7 + 3\alpha}{5 + \alpha} \right]
\]

(4)

Once the isentropic index \( \gamma_f \) is known, the ratio of static pressure to reservoir pressure is related to area ratio and flow Mach number by the following relations

\[
\left( \frac{A}{A^*} \right)^2 = \left( \frac{\gamma_f - 1}{2} \right) \frac{\left( \frac{2}{\gamma_f + 1} \right) \frac{\gamma_f + 1}{\gamma_f - 1}}{\left( \frac{b}{b_0} \right)^{\gamma_f} \left[ 1 - \left( \frac{b}{b_0} \right)^{\gamma_f - 1} \right]} \]

(5)

\[
\frac{b}{b_0} = \left[ 1 + \frac{\gamma_f - 1}{2} M^2 \right]^{-\left( \frac{\gamma_f + 1}{\gamma_f - 1} \right)}
\]

(6)

If area ratio is known accurately, then static pressure can be computed from Eq. 5. The flow Mach number and other thermodynamic quantities can be obtained using Eq. 6, and similar expressions for \( T \) and \( \rho \).

Nozzle area ratio is necessary to estimate the flow quantities in equilibrium and frozen flow cases. But at hypersonic speeds, the estimated effective area ratio is not accurate due to large boundary layer growth along the nozzle walls. To avoid this difficulty, usually a semi-theoretical-experimental approach is used to estimate the test section flow quantities. For any given set of reservoir conditions, static pressure is measured at the test section. Then this measured static pressure ratio is used to obtain other flow quantities, either from equilibrium charts or from frozen flow equations given in the previous section. Such flows are still under study and an accepted accurate method of determining all of the flow quantities from limited measured data is still under development by many laboratories.

6.2 Stagnation Point Heat Transfer Measurements

During the investigation of tailoring conditions (see Sec. 4.4.1),
test section flow calibration was also done. Stagnation point heat transfer to a spherical model mounted along the tunnel line and wall static pressure were measured during each run. In all of these runs a 0.4 inch deep nozzle was used, which gives a test section flow Mach number $M_v = 11.4$ based on geometrical area ratio and perfect gas theory. However, the actual flow Mach numbers computed from measured static pressures were used in reducing the data.

6.2.1 Gauge Construction

Heat transfer gauges are prepared by baking a thin film of platinum paint (Hanovia 05-X) on pyrex (7740) spherical ends cut from test tubes. Two models of 13mm. and 17mm. dia. (nominal) were tested. The film sizes were $\frac{1}{4}$" to $\frac{1}{8}$" long and 0.5 to 1.0 mm. wide. The resistance of the films was 50 ohms to 75 ohms at room temperature. The initial voltage across the gauge was kept at 0.4 V. A photograph of the heat transfer gauge with the sting is shown in Fig. 35.

Surface-temperature and heat transfer time histories were recorded in all the runs using the analogue network described in detail in Ref. 28. Typical surface temperature and heat transfer time histories are shown in Fig. 36c. This analogue network has 1 msec. testing time and 4 \mu sec. rise time. Since in most shock tunnels the testing times exceed more than 1 msec., a four channel analogue network with 5 msec. testing time has been built and will be used for future work. The stagnation point heat transfer is computed from the analogue output using the relation (Ref. 28)

$$
\dot{q}_t = \frac{1}{K} \frac{u}{U_0} \frac{2}{\sqrt{RC}} \left( \frac{P_b C_{pb} K_b}{\alpha_p} \right)^{1/2}
$$

where,

- $K$: amplification factor of the preamplifier
- $u$: voltage output from the analogue
- $U_0$: initial voltage across the gauge
- $RC$: resistance-capacitance constant for the analogue network
- $P_b C_{pb} K_b$: calibration constant for the gauge backing material (Pyrex)
- $\alpha_p$: coefficient of resistance of platinum film

6.2.2 Calibration of Gauges

In order to compute heat transfer from Eq. 1 the calibration constants ($P_b C_{pb} K_b$) and $\alpha_p$ should be known.
\( \alpha_p \) is determined by immersing the heat transfer gauges in a controlled temperature bath and measuring the resistance of the gauge at different temperatures. Then the rate of change of resistance with temperature \( (\alpha_p) \) is determined from a plot of resistance \( V \) vs temperature. An average value of \( 2.25 \times 10^{-3} \) \( /{ }^\circ C \) is obtained.

The value of \( (\rho C_p K_b)^{1/2} \) could easily be determined by using current pulse technique as described in Ref. 29. However this method requires the surface area of the platinum film, which is difficult to measure accurately near the stagnation point. But a new technique has been developed in Ref. 30 in which there is no need to measure the surface area of the film. This technique has been used to determine \( (\rho C_p K_b)^{1/2} \) and is briefly described in the following section.

A step current is applied to the gauge (that is, the thin film on its glass backing) and the surface temperature rise is recorded. The gauge is then immersed in a water bath so that the exposed side of the film is in contact with water. The step current is again applied and heat flows both into the glass and into the water. The temperature rise is again recorded. These two temperature records have the same time dependence but different amplitudes. The amplitude ratio is simply related to the ratio of the \( (\rho C_p K_b)^{1/2} \) of the glass and water by the simple relation

\[
\left( \frac{(\rho C_p K_b)_g}{(\rho C_p K_b)_w} \right)^{1/2} = \left( \frac{P_g}{P_w} \right) \left( \frac{\rho_g}{\rho_w} \right)^{1/2} \left( \frac{K_g}{K_w} \right) \left( \frac{C_p_g}{C_p_w} \right)^{1/2}
\]

where \( P \) and \( P^* \) are the amplitudes of gauge surface temperatures when the gauge is pulsed in air and in water respectively. Subscripts \( 'g' \) and \( 'w' \) stand for gauge and water respectively. Distilled water has a value of \( (\rho C_p K_b)^{1/2} = 0.0780 \) Btu \( /{ }^\circ F \) \( / \) sec \( / \) ft \( ^2 \) at room temperature.

The above method was used to determine the gauge constant. A typical surface temperature history of gauge pulsed in air and in water is shown in Fig. 36b. An average value of \( 0.08 \) Btu \( /{ }^\circ F \) \( / \) sec \( / \) ft \( ^2 \) has been obtained for the constant \( (\rho C_p K_b)^{1/2} \) for the gauge.

6.2.3. Comparison of Heat Transfer Results with Theory

During heat transfer measurements the reservoir pressures and temperatures varied from 632 to 1330 psi. and reservoir temperatures varied from 3700\( ^0 \)K to 5150\( ^0 \)K, and air was the test gas. For these reservoir conditions the flow in the nozzle could be assumed to be in near equilibrium (Ref. 31). Therefore, in computing the test section flow quantities for these runs, equilibrium charts for air (Ref. 23) were used. The following
test section flow quantities were obtained.

Test section flow Mach numbers 9.7 to 8.60
Test section flow velocities 9,900 to 13,200 ft/sec.
Reynolds number /ft. 1.5 x 10^4 to 3.82 x 10^3 (based on conditions behind the bow shock)
Heat transfer rate. 142.5 to 475 BTU /ft^2 -sec. (0.162 to 0.540 KW/cm^2)

In Fig. 37 the experimental Stanton numbers are compared with theoretical curves from Ref. 32.

6.3 Starting Process and Tunnel Testing Time

A nozzle starting process exists in supersonic and hypersonic shock tunnels of the non-reflected as well as the reflected type. The case of the non-reflected type has been studied in some detail in Ref. 33. Physically, if there is a thin diaphragm separating the shock tube from the working section, then, when it breaks the high pressure reservoir gas moves into the low pressure test section, forming a shock wave which initially reaches fairly high velocities (typically $M_s = 10$ to $30$). As this shock wave proceeds through the nozzle it is weakened owing to wall curvature. At the same time a contact surface and an upstream facing shock wave are usually formed to match the nonstationary and stationary (nozzle) flow processes (see Ref. 34). This shock wave may become sufficiently strong that it may move upstream (relative to the gas) at a speed which is high compared with the gas velocity, thus taking a long time to be swept down the nozzle, past the test section. If this time is longer than the available testing time then steady flow will never be established. In the worst case, the shock may not even be swept downstream and actually stand in the nozzle. A photograph of the starting of a supersonic nozzle in a non-reflected shock tunnel is seen in Fig. 38 (Ref. 33).

This problem, as it affects the successful operation of a hypersonic shock tunnel was considered by Glick, Hertzberg & Smith (Ref. 35), Holder, and Schultz (Refs. 36 & 37), Henshall & Gadd (Ref. 38) and Henshall (Ref. 39).

The process may be understood by considering the flow in the pressure-velocity $(P-U)$-plane (see Fig. 39 and Ref. 9). It is pointed out by Flagg (Ref. 4) that this approach is particularly useful when considering the relationship between steady and unsteady processes. The value $P_o$ is the reservoir pressure from which the gas is to be expanded to the steady flow test section static pressure $P_m$. Let us consider the case of the initial tunnel pressure $P_i$, and that the Mylar diaphragm is at the throat of the convergent-divergent nozzle. Immediately after the rupture of the diaphragm,
the normal shock tube problem can be assumed to exist, i.e. a right-running shock and a left-running rarefaction wave are formed (Fig. 40b.). However, the steady state which the gas is attempting to reach is that at point $S_1'$, in Fig. 39, which is the pressure and velocity (or flow Mach number) obtained in the steady nozzle expansion. It is clear from the diagram that it is not possible to 'match' the two pressures $P_{s_1}$ and $P_{c_1}$, merely by means of a shock and an expansion, but that two rarefaction waves are necessary (Fig. 40c). The rarefaction wave on the right hand side will refract, then overtake the shock wave and their interaction will weaken the shock and reflect a weakened rarefaction wave. A compression could also be reflected. Then a new interaction would result. This wave will again refract and will in turn coalesce with the left hand rarefaction wave (Fig. 40d) It can be seen, therefore, that the representation given in Refs. 37 and 39 are oversimplified in that such a pressure system as shown in Fig. 40d can only exist after such time as it takes for the rarefaction wave to overtake the shock and then reflect back (neglecting the effects of second order interaction). In the particular case of initial tunnel pressure $P_{c_2}$, only one rarefaction wave exists (the other being a Mach wave in this unique case) (Fig. 40e) but this must still refract, overtake the shock, and return. This is as close as it is possible to get to the so called 'perfect or optimum start' condition (Fig. 40f) for the required flow Mach number. The only condition under which it is possible to 'match' the initial tunnel pressure with the test section static pressure solely by means of a shock wave is when the flow Mach no. required is at point 'X', that is, the cross-over point where the steady and unsteady flow properties match (Fig. 39). If $\gamma = 1.4$, this point corresponds to a flow Mach number of $M_0 = 4/(3-\gamma) = 2.5$ (see Ref. 34).

In the third case, when the initial tunnel pressure is $P_{c_2}$, the system consists of two shocks and a rarefaction wave as shown in Fig. 40g. After several interactions, two opposite facing shock waves separated by a contact surface ultimately result in this case.

It is worth noting that the same diagram may be used to analyse the operation of "straight-through" type of nozzle starting processes shown in Fig. 38. In this case the initial shock tube problem is eliminated. The state on the left (5) (Ref. 9) is joined to the state on the right (1') by an upstream facing rarefaction wave and a contact surface. The downstream shock wave $P_{c_2}/P_{c_1}$ is weakened by the increased area ratio. In the case when the initial nozzle pressure is $P_{c_3}$, then state (5') occurs and an upstream facing shock wave results. This is the case shown in Fig. 38. It is of interest to note how quickly the wave system is formed in the nozzle.

It is clear that the processes postulated in Refs. 37 & 39 are only possible after some period of time, which may not be short enough to allow the necessary adjustments to occur in any given nozzle. However, the analysis of the starting process, taking these waves into account, would
become far more complex and therefore for the purposes of a qualitative examination a simpler picture is considered at this time.

Consider the starting process as it affects the pitot and static pressure measurements in the test section. In the case of static pressure the problem is simple. At any fixed point in the test section the static pressure measured is the pressure in the different wave systems as they pass along the nozzle. The three cases, low tunnel pressure, 'optimum' tunnel pressure and high tunnel pressure are shown in Figs. 41a, b, and c, (and correspond to conditions in Fig. 40d, f, and g), for typical nozzle conditions. It is important to note that the static pressure transducer will not detect the passage of the interface or contact surface between the initial tunnel gas and the test gas as the pressure is the same on either side. The dotted line indicates the fall off of pressure due to the arrival of the reflected rarefaction wave from the driver at the nozzle entrance i.e. the end of the testing time.

In the case of pitot pressure, the time history at a fixed point has the general form shown in Figs. 42a, b, and c. The peak 'X' is the instantaneous reflected shock pressure at the model stagnation point which decays as shown to the steady pitot pressure for the flow condition behind the initial shock. The rise to point 'Y' is due to the pitot pressure change at the contact surface and that at Y due to the upstream facing expansion wave. The peak values shown are only presented as an indication of typical ones for the given condition, but do indicate the general trend. The shape shown is that which would be reproduced by a perfect pressure transducer, however the limited response capability of a typical transducer in a nosecap would modify the shape so that the expected signal from the pitot pressure transducer is shown in Fig. 43. The similarly modified shape for the static pressure is shown in Fig. 44. This figure is given only as an approximation, the times t1 and t4 shown in Fig. 42 are almost impossible to compute especially in the case of a nozzle system as complex as that used here.

If the time origin used for Figs. 41 to 44 represents the breaking of the mylar diaphragm then t1 is the time taken for the initial shock to reach the test section. On the basis of the conditions across the mylar diaphragm, the initial shock velocity can be computed to be about $M_s = 20$ to $30$. However, the shock is soon attenuated by interaction with the area changes and the average velocity from nozzle to test section, measured in practice, is less than $M_s = 8$. For this reason the strength of the initial shock used in the calculations for Fig. 42 is $M_s = 6$. The times $t_2$ and $t_3$ are almost impossible to ascertain theoretically, but it is clear that for a long nozzle, and if the velocity of the upstream moving shock is high relative to the local flow velocity, then the time $t_3$ may be very large. Similarly, if the total testing time at the nozzle entrance ($t_5$) is not much larger than $t_2$ and $t_3$, then the usable testing time in the working section
would be small. For example, the time \( t_2 \) and \( t_3 \) is typically 0.5 to 1.0 millisecond, therefore in the case of the present facility the \( t_5 \) of 1.5 msec. only allows about 0.5 msec. for \( t_4 \). However, many facilities e.g. those at Cornell Aeronautical Laboratory (Ref. 27), General Electric (Ref. 40), Douglas Aircraft (Ref. 41) have available running times of 5 to 15 milliseconds, in which case the starting process is usually not significant and is not usually visible in the published pitot pressure profiles. The reason for this is that these workers are able to choose hole sizes for the nosecaps such that the 'filling time' (see Sec. 6.4.2) is as much as 2 msecs., thus hiding the pressure fluctuations due to the starting process. An example of a visible starting system is seen in Fig. 45 which is taken from Ref. 43. In this case an attempt was being made to compare the pressure variation with that obtained from theory. By means of schlieren photographs they are able to discern and follow the motion of the initial and upstream running shocks. In case 'c' in Fig. 42 the spikes 'x' and 'y' seen in the trace correspond well in time with those times predicted from the schlieren pictures. The magnitudes, however are not predicted.

Fig. 46 shows tests made in the working section of the 11 in. x 15 in. Hypersonic Shock Tunnel at two different initial tunnel pressures. Fig 46a shows a test with \( P_c = 5 \text{ Hg} \). The process is probably that which has an upstream running rarefaction wave and the traces should be compared with Figs. 43a and 44a. In the second test, Fig. 46b, the tunnel pressure was about \( 100 \text{ Hg} \) and the starting process should be compared with Figs. 43c and 44c.

6.4 The Pitot Survey Rake

The design of the rake was especially influenced by the choice of # 603 Kistler piezoelectric pressure transducers for the measurement of pitot pressure. In order that several adjacent points in the small 11 inch wide test section may be investigated, it is necessary to use small pressure transducers; and the need for rapid response makes a piezoelectric gauge an obvious choice. Three of the # 603 transducers were available, together with their associated charge amplifiers and so the survey rake was designed around these.

The original design of the pitot rake is shown in Fig. 47. The main body is of mild steel and has mounted along its front edges seven wedge-shaped pieces of 1 1/4 in. width. Three of these hold stainless steel probes which in turn hold the pressure transducers. The transducer itself was supported from the rear by the probe body and from the front by a brass nosecap, in both cases teflon spacers were utilized, as recommended by the manufacturers.

Later the probes were modified to accomodate the #701A transducers which are a little larger (3/8 in. dia. compared to 1/4 in. dia.) Here the transducer is supported from the front by the brass nosecap by a
teflon ring and from the rear by the probe body, by a rubber '0' ring.

The rake may be mounted in the working section either directly, or at the end of 1 ft. or 2 ft. extension rods, and may be offset in order to bring the probes closer to the wall. The rake is shown in Fig. 48 mounted in the latter manner.

6.4.1 Operational Problems of the Kistler Type Pressure Transducers

Several problems arose due to the choice of Kistler type pressure transducers. Basically, the quartz element utilized in these transducers offer several advantages such as: high stability, high natural frequency (and hence low risetimes) and static calibration capability. In particular, those manufactured by the Kistler Instrument Corporation offer exceptional qualities of ruggedness and performance. However, care must be exercised, particularly when applications such as pitot pressure measurement are concerned. In this application, the stagnation temperature is high, there is no cold boundary layer such as in the case of a static pressure measurement, and the pressure measured is low.

The high stagnation temperature affects the transducer because internal strains are set up, causing spurious outputs. Such effects are relatively slow to take effect and are alleviated by mounting the transducer in a large heat sink. The major problem is that crystals such as quartz exhibit a pyroelectric effect analogous to the piezoelectric effect, that is, a charge output is caused by a temperature rise (Ref. 41 and 42).

An early attempt at a pitot pressure measurement using a #603 transducer is shown in Fig. 49a. The upper trace is of static pressure, showing that the flow is established. However, the pitot pressure transducer output is negative, remaining substantially constant for 0.7 millisecond. This is not caused by inadvertant reversal of sign of the output. The transducer is effectively acting as a heat transfer gauge, the pressure signal being overwhelmed by the pyroelectric signal. The slight lag between flow arrival and the negative signal can be seen.

A further problem encountered with this #603 transducer was that the output was not linear with pressure down to very low pressures. As the pressure tends to zero the sensitivity of this particular transducer was 0.06 pico-coulombs/psi. (pcb/psi.) compared with 0.2 pcb./psi. at 40 psi. The manufacturers have confirmed that this was a very early model of this series, in which they were attempting to achieve very fast rise-times. This model incorporated several new features, not all of which were successful and it is possible for nonlinearities to occur.

The #605 transducers exhibited much more satisfactory characteristics at low pressure. Typical sensitivities were 0.37 pcb./psi., being substantially the same up to above 10 psi. These transducers also exhibit the pyro-electric effect and the pitot pressure trace obtained initially
with these transducers is shown in Fig. 49b. The slight rise due to pressure is clearly seen, followed shortly by the rapid fall due to the thermal effect. The lower trace is that for a pitot probe with the transducer face blanked off from the flow, demonstrating that effects due to cable motion, probe and transducer vibrations, electrical interference etc. are not responsible for the effect. In the case of this transducer, the signal is not negative at any time because the piezoelectric output is greater and thus more nearly able to overcome the pyroelectric one.

The problem is solved by protecting the face of the transducer by means of a smear of silicon grease or a coating of room temperature vulcanising (RTV) rubber such as General Electric RTV-108. A test with two probes, one with grease and one without is shown in Fig. 50. It can be seen that the upper trace rises and then remains substantially constant and that the temperature effect is eliminated.

Typical pitot pressure traces taken with these transducers are shown in Fig. 51. The trace of \( \beta_F \) for this run is also seen. The flow Mach number in the test section was computed to be about 7.2. (This run was used for taking a schlieren photograph of the flow about the pitot pressure rake. The blanking of the signal indicated is due to interference from the spark source for the schlieren system. This acts as a convenient marker to indicate the time during the flow at which the photograph was taken.) The #605 transducer, however has insufficient signal/noise ratio if pressures as low as 1 psi. are to be measured. As this capability was felt to be necessary, the pitot survey rake was modified to accept the slightly larger #701A transducer. Provided that the same precautions outlined above are followed, these transducers (#701A) are emminently suitable for such pressures.

6.4.2 Pitot Probe Hole Size

The effect of the size of hole in the nosecap of the pitot pressure probe has been considered. This problem was investigated empirically by Kamimoto (Ref. 44), who concluded that the size of hole did not effect the final value of pitot pressure measured. However, a smaller hole was found to increase the response time and reduce the signal to noise due to gauge ringing and flow turbulence. Harris and Kaegi (Ref. 45) have considered this problem theoretically by analogy with an R-C electrical circuit and computing the response to a step input of pressure. This analogy suggests that the response time is independant of the applied pressure.

An investigation was undertaken in the 11 in. x 15 in. hypersonic shock tunnel to find the effect of hole size on a qualitative basis. It was found that the available range of hole sizes was extremely small as too slow a rise did not allow the peak pressure to be registered by the time the tunnel testing time was over. An example of a test of three different hole
sizes is seen in Fig. 52. It can be seen that the smallest hole gave substantially the same final pressure reading, at the cost of sensitivity to the finer detail, particularly of the starting process (this aspect of the flow investigation is discussed more fully in Sec. 6.3). A hole of 1/8 in. dia. and 1/8 in. length was found to give substantially the same response as a transducer whose face is completely exposed to the flow. It was decided to use this nosecap in order to give the transducer some protection and so that the pitot pressure is measured over a smaller area. The theory of Harris and Kaegi would suggest a time to 99% of peak pressure (for a step increase of pressure and with this hole) of about 80 μsec. or 16 μsec. to 65%. This is less than the risetime of the plug-in unit used in the oscilloscope which has a cutoff filter to reduce high frequency noise.

6.5 Pitot Pressure Surveys

A survey has been made of the pitot pressures across the vertical and horizontal planes of the test section of the shock tunnel. This survey has been made in detail only at one operating condition as follows,

<table>
<thead>
<tr>
<th>Test gas</th>
<th>Oxygen</th>
</tr>
</thead>
<tbody>
<tr>
<td>Incident shock Mach No.</td>
<td>9.5 ± 0.3</td>
</tr>
<tr>
<td>Initial channel Pressure (p_i)</td>
<td>25 Hg.</td>
</tr>
<tr>
<td>Reservoir pressure (p_o)</td>
<td>500 psi. (± 50)</td>
</tr>
<tr>
<td>Reservoir temperature (T_o)</td>
<td>4,550°K (± 100°K)</td>
</tr>
<tr>
<td>Nozzle evacuation pressure</td>
<td>≈ 10 μHg</td>
</tr>
</tbody>
</table>

The low stagnation pressure, high temperature and use of oxygen was chosen in order to simulate the conditions required for future work in dissociated oxygen flows. The nozzle used was that giving the highest flow Mach number and which could be started satisfactorily. These conditions are also those under which the greatest nozzle boundary layer growth was to be expected and also gives very low static and pitot pressures, thus pressing the instrumentation to its limit. It was to be expected, therefore, that the testing time, flow uniformity and uniform core size would be all at their worse.

A typical test with the rake in the horizontal plane is shown in Fig. 53. In this case the pitot pressure is measured at the centreline and at two points 1½ in. either side of the centreline. A typical test with the rake in the vertical plane is shown in Fig. 54. In this case the centreline pressure history is seen to be similar to Fig. 53, whereas, the other pressures at 2½ in. above and below the centreline are of a different form.
It is quite clear that the flow is not uniform in the vertical plane and that flow is not established at points a distance away from the centreline. The rise to point 'A₁', as indicated, is due to the starting process and is therefore similar in all three cases. The centreline pressure then rises to the value 'B₁', whereas those away from the centreline fall as shown. The reason that no steady flow is seen in these latter cases is thought to be that the boundary layer, which causes the lower pressure, is not completely established when the stagnation pressure drops ('C₁'). The reason for the 'spikes' 'D₁' in trace number a is not known, but they do not appear in all runs. In those runs with the rake in the horizontal position, the pressures are easy to determine from the traces, however, those from the vertical traverses are clearly more difficult to determine. In the work done so far the assumed pressures are those seen typically in Fig. 54.

The distributions of pitot pressure obtained in these tests are seen in Figs. 55 and 56. It can be seen that the boundary layer along the vertical walls is very thin and that the 'core' in the horizontal plane is relatively uniform and of a width of about 8 in. The core in the vertical plane, however, is seen to be very small; i.e. less than 1.5 in.

It has been stated by Hertzberg (Ref. 46) that this distribution is similar to that which was found in this same test section when it was used at Cornell Aeronautical Laboratory and that this is probably due to the final nozzles not completely skimming away the thick boundary layer which grows in the primary nozzle. However, this does not explain the difference in distribution shown in Figs. 55 and 56.

7. CONCLUSIONS

Many aspects of hypersonic testing using a shock tunnel are considered. Different types of ignition systems have been discussed and their relative advantages and disadvantages have been pointed out. It has been demonstrated that very repeatable and smooth combustion can be achieved by using an impulsively heated tungsten-wire technique.

This technique was used to obtain tailored shock Mach numbers up to $M_{st} = 9.45$, by using both constant volume and constant-volume, constant-pressure combustion. Under these operating conditions repeatable detonation-free combustion and repeatable diaphragm bursting are the deciding factors. The combustion method has advantages of simplicity and low cost well within the capabilities of most research institutions.

A test section flow calibration has been done by measuring wall static pressures, pitot pressures, and stagnation point heat transfer rates. Static pressures as low as 0.005 psi, and pitot pressures as low as 0.5 psi.
we're successfully measured. At these extreme conditions proper care must be exercised to eliminate the heating (pyro-electric) effects on Kistler pressure transducers. Also vibration isolation techniques and filters are indispensable.

If short driver and driven sections are used to operate a shock tunnel, care must be taken in estimating the testing time, since under these conditions most of the running time is used up to establish the flow in the nozzle. Theoretically predicted shapes (Sec. 6.3) of static and pitot-pressure time histories during the nozzle starting processes should be used in conjunction with the experimental traces to predict and interpret the testing time.

The available core of uniform flow in the test section at low reservoir pressures (20 to 30 atmos.) and high reservoir temperatures (4000°K to 6000°K) is indeed very small, due to large boundary layer growth along the nozzle walls at these extreme conditions.

Consequently, large test sections and long channels and drivers are used in modern shock-tunnel facilities.
<table>
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<th>Reference</th>
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<th>Title and Source</th>
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<td>Title</td>
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<td>----------------------------------------------------------------------</td>
</tr>
</tbody>
</table>
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# TABLE I

<table>
<thead>
<tr>
<th>Method</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>(a) Crimped Wire</td>
<td>1) Theoretically good, gives multi-point ignition.</td>
<td>1) Difficulty in handling wire without breakage.</td>
</tr>
<tr>
<td>(Ref. 6, 48)</td>
<td></td>
<td>2) High accuracy required in crimping.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3) Energy used to explode wire must be exact.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>4) Wire breaks into pieces and is carried down the tube as projectiles. Any failure in the first three requirements assists detonation.</td>
</tr>
<tr>
<td>(b) Exploding Wire</td>
<td>1) Line ignition</td>
<td>1) Energy used is very critical.</td>
</tr>
<tr>
<td>(Ref. 3)</td>
<td></td>
<td>2) Wire vaporizes and contaminates the tube.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3) Particles can act as projectiles.</td>
</tr>
<tr>
<td>(c) Aluminum Foil</td>
<td>1) Multi-point ignition</td>
<td>1) Difficulties in manufacture and handling.</td>
</tr>
<tr>
<td>(Ref. 49)</td>
<td></td>
<td>2) Foil breaks causing contamination.</td>
</tr>
<tr>
<td>(d) Multiple Spark Systems</td>
<td>1) Multi-point ignition 2) No contamination</td>
<td>1) Spark plugs must be checked and maintained frequently.</td>
</tr>
<tr>
<td>(Ref. 5)</td>
<td></td>
<td>2) Large number of plug holes required in driver which causes problems in driver design.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3) Failure of single plug causes</td>
</tr>
<tr>
<td>Method</td>
<td>Advantages</td>
<td>Disadvantages</td>
</tr>
<tr>
<td>-------------------------------</td>
<td>----------------------------------------------------------------------------</td>
<td>-------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>(d) Multiple Spark Systems</td>
<td>1) Wire stays in place, hence no contamination.</td>
<td>hazards.</td>
</tr>
<tr>
<td>(continued)</td>
<td>2) Even and line ignition.</td>
<td>4) Not all workers have had success with this method. It appears to be a function of tube geometry.</td>
</tr>
<tr>
<td>(e) Heated Tungsten Wire</td>
<td>3) Low energy requirement.</td>
<td></td>
</tr>
<tr>
<td>(Ref. 3)</td>
<td>4) Easy handling, wire is tough.</td>
<td></td>
</tr>
<tr>
<td></td>
<td>5) Energy requirement is not too critical.</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6) Wire can be used many times.</td>
<td></td>
</tr>
</tbody>
</table>
Testing Time By Mass Flow Consideration With Real Gas Effects

The point \( t \) (see Fig. 27), from the entrance to the nozzle in the driven tube where the reflected shock meets the contact surface determines the mass \( m \) of heated gas available for expansion through the nozzle.

The rate of mass flow is \( \dot{m} = \frac{\rho^* A_t a^*}{\rho \cdot a^*} \) where \( \rho^* \) and \( a^* \) are the density and speed of sound at the nozzle throat respectively. Then the duration of test flow, limited by the arrival of the contact surface is

\[
t_c = \frac{\rho_s l A_s}{\rho^* a^*} \text{ (Secs)}
\]

\( l \) can be calculated for any given geometry and shock Mach number or can be obtained from \((x, t)\)-diagram as shown in Fig. 27. \( \rho_s \) is taken from real-gas, shock-tube performance charts, and \( \rho^* \) and \( a^* \) have to be computed for any given conditions using equilibrium charts. However, for air, the mass flow can be easily computed by using the plot of the normalized mass flow parameter

\[
\frac{H}{P_0 A^*} \left( \frac{1b \cdot \text{sec}}{\text{atm} \cdot \text{ft}^2} \right) \text{ vs } h_0 (\text{Btu/lb})
\]

given in Ref. 23.

Allowances should be made to the testing times computed by the above method to take into account viscous effects on the motion of the contact surface and its mixing. References 24, 25 and 26 should be consulted when accounting for these effects.
APPENDIX B

Pressure Transducer and Cathode Follower Circuit Used to Measure Test Section Static Pressure

1. Pressure Transducer

These transducers are type PZT-50-12-AC made of lead zirconium titanate crystals. The specifications are:
1) Size: ½" dia x 1/8" thick
2) Pressure sensitivity: 2.0V/psi (nominal)
3) Acceleration Sensitivity: 0.0005 Equiv. psi/g
4) Frequency Response: 99% in 1 millisecond for step Input
5) Linear to 1 psi.
6) Will withstand 30 psi overload
7) Orifice diameter: 0.100"

A photograph of the pressure transducer is shown in Fig. 32. Complete construction details are given in Ref. 27.

2. Cathode Follower

The cathode follower circuit designed to meet the requirements of the crystal transducers is shown in Fig. 34.

The active element is a miniature pentode tube type 5840. The bias level is established at 50 volts and the gain is constant at 0.98. The input impedance is about $5 \times 10^{11}$ ohms, a value sufficiently high to produce less than 0.1% drop in 10 milliseconds in response to a square wave. Due to the high input impedance, cleanliness and high insulation resistance are required in the input circuit. The current meter should read at least 450 μA at all times. If not it indicates that input impedance is shunted, probably by dirt, moisture and a low resistance device. The press-button and voltage divider, consisting of 1 MΩ and 2 MΩ resistors, are not really necessary, but help obviate waiting time for the input to charge up to grid operating potential as indicated on the plate meter. The circuit is susceptible to 60Hz pick-up, so adequate shielding should be employed. The calibration signal is fed in series with the transducer, since it is desirable to maintain equal input impedance during test and calibration.

3. Twin Tee Filter

The Cathode follower has a built-in filter shown in Fig. 34. This 'twin tee' filter consists of variable capacitors which should be ganged together and tuned for a null at the frequency at which rejection is desired. There is interaction between the variable capacitor and the 500KΩ potentiometer; both should be tuned so that maximum rejection is obtained. Most of
the time the filter is not necessary and is used only when the transducer is excited into resonance by the shock wave impact under certain conditions. Due to insertion losses the filter should be included in any pretest transducer calibration. These insertion losses become comparable to pressure signals at low pressures.

For this reason the filter has not been found as satisfactory as that built into the Tektronik #3A3 plug-in unit and is not used.
FIG. 1. GENERAL LAYOUT OF 11in x 15in HYPERSONIC SHOCK TUNNEL.
FIG. 2 UTIAS 11 in. x 15 in. HYPERSONIC SHOCK TUNNEL AND INSTRUMENTATION
Brazed on circumference

Weld

O' ring groove

Woods metal filler

Nut for height adjustment.

FIG. 4  TYPICAL SHOCK TUBE SECTION (FULL SIZE)
FIG. 5    AERODYNAMIC OUTLINE OF PRIMARY NOZZLE, CORNER EXPANSION PLATE, AND FINAL NOZZLE OF THE UTIAS 11 in. x 15 in. HYPERS"
FIG. 6  MECHANICAL AND DIFFUSION PUMPS LAYOUT.
FIG. 7. VALVE SYSTEM LAYOUT.
FIG. 8. INTERLOCKING SYSTEM LAYOUT.
FIG. 9. CIRCUIT DIAGRAM OF IGNITION CONTROL.
FIG. 10a 2" x 2" SHOCK TUBE INSTRUMENTATION
FIG. 11  STATIC CALIBRATION OF KISTLER#603 PRESSURE TRANSUDER.
**FIG. 12** COMBUSTION PROCESSES IN THE 2in. x 2in. DRIVER USING THE CRIMPED ALUMINIUM WIRE

- Initial Pressure: 300 psi.
- Helium Dilution: 80%
- Horizontal: 1 div. = 5 milliseconds
- Vertical: 1 div. = 330 psi.

**FIG. 13** COMBUSTION PROCESSES IN THE 2in. x 2in. DRIVER USING A HEATED TUNGSTEN WIRE

- Initial Pressure: 100 psi.
- Helium Dilution: 70%
- Horizontal: 1 div. = 1 msec.
- Vertical: 1 div. = 160 psi.

(Note: There are two successive tests on this trace)

- Initial Pressure: 200 psi.
- Helium Dilution: 65%
- Horizontal: 1 div. = 1 msec.
- Vertical: 1 div. = 330 psi.
Fig. 14 Variation of peak pressure with percentage helium dilution. (Constant volume combustion)
FIG. 15 TIME TO PEAK VERSUS INITIAL CHARGING PRESSURE ($P_{4i}$)
FIG. 16 VARIATION OF DIAPHRAGM RUPTURE PRESSURE WITH THICKNESS (t)

- Diamond symbol: in shock tube
- Square symbol: in test rig

BRASS

Diaphragm thickness
0.035 in.
FIG. 17 VARIATION OF DIAPHRAGM RUPTURE PRESSURE WITH THICKNESS ($t$)
FIG. 18  DIAPHRAGMS BEFORE AND AFTER USE IN THE 2in.x2in. DRIVER

FIG. 19  DIAPHRAGMS AFTER MILD DETONATIONS, SHOWING SLIGHT LOSS OF METAL FROM TIPS
FIG. 20  REFLECTED PRESSURE (TAILORED) HISTORIES AT DIFFERENT HELIUM DILUTIONS (Constant Volume Combustion)

a) 80% Helium; $M_s=6.12$
Horiz: 1div=200$\mu$sec.
Vert: 1div=330 psi.
(both traces)

b) 75% Helium; $M_s=6.64$
Horiz: 1div=200$\mu$sec.
Vert: 1div=330 psi.
(both traces)

c) 70% Helium; $M_s=7.42$
Horiz: 1div=200$\mu$sec.
Vert: 1div=330 psi.
(both traces)
FIG. 21 COMPARISON OF TAILORED SHOCK MACH NUMBERS WITH THEORY.
FIG. 22 REFLECTED PRESSURE HISTORIES WITH DIFFERENT "OVERPRESSURE COMBUSTION RATIOS" (CONSTANT VOLUME-CONSTANT PRESSURE COMBUSTION)
Helium Dilution: 70%
Test Gas : Oxygen
FIG. 23 REFLECTED PRESSURE RATIO VERSUS SHOCK MACH NUMBER.
FIG. 24 DIAPHRAGM PRESSURE RATIO VERSUS SHOCK MACH NUMBER.
FIG. 25 REFLECTED PRESSURE RATIO VERSUS SHOCK MACH NUMBER (in oxygen).
FIG. 26-a  Schmatic Pressure History At End Wall. (From Ref. 20).

a) Transducer at side wall
   $M_a = 8.18, \ P_1 = 25 \text{ mm.}$
   Horiz: 1 div = 500 $\mu$s.
   Vert: 1 div = 200 psi.

b) Transducer at end wall.
   $M_a = 8.10, \ P_1 = 25 \text{ mm.}$
   Horiz: 1 div = 500 $\mu$s.
   Vert: 1 div = 200 psi.

c) Transducer at side wall.
   $M_a = 11.25, \ P_1 = 10 \text{ mm.}$
   Horiz: 1 div = 500 $\mu$s.
   Vert: 1 div = 200 psi.

d) Transducer at end wall.
   $M_a = 10.75, \ P_1 = 10 \text{ mm.}$
   Upper trace: Horiz: 1 div = 100 $\mu$s.
   Vert: 1 div = 100 psi.
   Lower trace: Horiz: 1 div = 500 $\mu$s.
   Vert: 1 div = 200 psi.

FIG. 26-b  PRESSURES AFTER SHOCK REFLECTION WITH TRANSDUCERS AT SIDE WALL AND ENDWALL.
FIG. 27. x - t DIAGRAM

\[ M_s = 9.5, \text{ 70\% He, } p_1 = 25 \text{ mmHg.} \]
FIG. 28. SCHLIEREN SYSTEM FOR 11" x 15" HYPersonic SHOCK TUNNEL
FIG. 30 FLOW OVER A CYLINDER IN THE UTIAS 11 in. x 15 in. HYPERSONIC SHOCK TUNNEL (Cylinder 1.5 in. dia., $M_s = 6.7$, $p_5 = 974$ psi, $T_5 = 4200$°K, $M_1 = 7.9$, a - self luminousity, b - schlieren, c - schlieren with field stop at knife edge to decrease self luminousity, d - as for c but even smaller stop)
FIG. 31  SCHLIEREN PHOTOGRAPHS OF FLOW OVER A FLAT PLATE
(Flow conditions as for FIG. 30)
FIG. 32  TYPICAL STATIC PRESSURE HISTORIES AND PRESSURE TRANSDUCER.
FIG. 33 STATIC PRESSURE TRANSDUCER CALIBRATION CURVE.
FIG. 34 CIRCUIT DIAGRAM CATHODE FOLLOWER.
FIG. 35  HEAT TRANSFER GAUGE PHOTOGRAPH.

b) Heat Transfer Gauge Calibration Trace
Horiz. Scale : 1div=50μsec.
1: T. with gauge in water.
2: q. with gauge in water.
3: q. with gauge in air.
4: T. with gauge in air.

FIG. 36  HEAT TRANSFER AND SURFACE TEMPERATURE HISTORIES.

c) Typical Heat Transfer and Surface Temperature History
Horiz. Scale : 1div=0.5 msec.
FIG. 37  COMPARISON OF STAGNATION POINT HEAT TRANSFER MEASUREMENTS WITH THEORY

\[
\frac{q}{\rho_\infty u_{\infty}} = \frac{H_0 - H_W}{R_b}
\]

\[
Re_b = \frac{\rho_\infty u_{\infty} R_b}{M_e}
\]

\(\xi = 0.1\)

\(\xi = 0.1667\)
FIG. 38(a) Early Stages in the Development of Flow in a Divergent Nozzle. $M = 3.80$.

FIG. 38(b) Development of Flow in the Divergent Nozzle. The Prandtl-Meyer Expansion, $S_2$, the Entropy Discontinuity and $S_3$ are Visible. $M = 3.00$. 
FIG. 39 THE STARTING PROCESS IN THE PRESSURE VELOCITY ($P-U$) PLANE
FIG. 40 a, b, c, d PRESSURE PROFILES IN NOZZLE STARTING PROCESSES
FIG. 40 e,f,g PRESSURE PROFILES IN NOZZLE STARTING PROCESSES
**Fig. 41 Static Pressures at a Point in a Nozzle During Starting Processes**

- **a)** $P_o = 5 \mu \text{Hg.}$
- **b)** $P_o = 7.5 \mu \text{Hg.}$
- **c)** $P_o = 100 \mu \text{Hg.}$

$t_s$: starting time
$t_R$: testing time

**Fig. 42 Pitot Pressures at a Point in a Nozzle During Starting Processes**
FIG. 43 PITOT PRESSURE DURING STARTING PROCESSES INDICATED BY A REAL PITOT TRANSDUCER

FIG. 44 STATIC PRESSURE DURING STARTING PROCESSES INDICATED BY A REAL TRANSDUCER
FIG. 45 TYPICAL PITOT PRESSURE HISTORY FROM REF. 39
FIG. 46  STATIC AND PITOT PRESSURES IN THE 11 in. x 15 in. HYPersonic
Shock Tunnel with Different Initial Tunnel Pressures
FIG. 47 THE ORIGINAL PITOT SURVEY RAKE, AS DESIGNED FOR THE KISTLER #603 TRANSUCER

FIG. 48 THE FINAL RAKE, AS MODIFIED FOR #701A TRANSUCERS, SHOWN IN AN OFFSET POSITION
a) #603 Transducer

1 div = 0.5 msec.

Upper trace: Static Pressure

Lower trace: Pitot Pressure

1 div = 20 mv.

b) #605 Transducer

Gauge open (1/16 dia. hole)

Gauge blanked off

1 div = 0.2 msec

FIG. 49 PYROELECTRIC EFFECTS IN PRESSURE TRANSUCERS
FIG. 50 THE EFFECT OF SILICONE GREASE ON THE FACE OF THE TRANSDUCER

FIG. 51 PITOT PRESSURE TRACES USING #605 TRANSDUCER
Reservoir Pressure
1 div = 240 psi.

$M_s = 9.55$

1 div = 0.5 msec

Static Pressure: 1 div. = 0.005 psi.

Pitot Pressure: 1 div = 0.35 psi.
(Open face)

1 div = 0.5 msec

Pitot Pressure: 1 div = 0.35 psi.
(1/8 in. dia. hole)

Pitot Pressure: 1 div = 0.35 psi.
(1/16 in. dia. hole)

1 div = 0.5 msec

FIG. 52 THE EFFECT OF NOSECAP HOLE SIZE ON THE PITOT PRESSURE TRANSDUCER RESPONSE
upper: pitot 1.25 in. either side
lower: pitot of centre line
scale: 1 div = 0.35 psi.

upper: static
1 div = 0.005 psi.
lower: pitot (centre line)
1 div = 0.35 psi.

Test gas: Oxygen
Initial pressure (P_i): 25 mm. Hg.
Shock Mach No. (M_s): 9.5

Reservoir Pressure
1 div = 240 psi.

FIG. 53 PRESSURE TRACES FROM A PITOT TRAVERSE RUN (RAKE HORIZONTAL)
upper : pitot, 2.5 ins. above centre line
a) lower : pitot, 2.5 ins. below centre line
scale : 1 div = 0.35 psi.

b) upper : Static, 1 div = 0.005 psi.
lower : pitot (centre line)
1 div = 0.35 psi.

c) Reservoir Pressure
1 div = 240 psi.

Test Gas : Oxygen
Initial Pressure ($p_1$) : 25 mmHg.
$M_s$ : 9.5

All traces : 1 horiz. div = 0.5 msec.

FIG. 54 PRESSURE TRACES FROM A PITOT TRAVERSE RUN (RAKE VERTICAL)
FIG. 55 PITOT PRESSURE DISTRIBUTION IN THE TEST SECTION
(HORIZONTAL TRAVERSE)
FIG. 56 PITOT PRESSURE DISTRIBUTION IN THE TEST SECTION (VERTICAL TRAVERSE)