JOURNAL OF THERMOPHYSICS AND HEAT TRANSFER
Vol. 8, No. 2, April–June 1994

Combustion Shock Tunnel and Interface Compression to Increase Reservoir Pressure and Enthalpy

M. A. S. Minucci*
Instituto de Estudos Avançados—CTA, São José dos Campos, SP 12231-970, Brazil
and
H. T. Nagamatsu† and L. N. Myrabo‡
Rensselaer Polytechnic Institute, Troy, New York 12180

This article discusses the production of hypervelocity-hypersonic flows in a combustion shock tunnel operating in the equilibrium interface mode. In this mode of operation, the additional compression provided by the approaching interface is used to obtain higher pressures and temperatures, as opposed to the reflected method. A computer code was developed to model the operation of a shock tunnel in the equilibrium interface condition. In this article, all the calculations were made for the Rensselaer Polytechnic Institute (RPI) 1.22-m-diam Combustion Driver Hypersonic Shock Tunnel. The major drawback of the interface compression technique, which is the contamination of the driven gas by the driver gas, was overcome through the utilization of a small volume region separating the two gases. Numerical results indicate that the RPI facility will be able to generate reservoir temperatures of the order of 20,000 K and reservoir pressures of the order of 30,000 psi. These reservoir conditions can be used to produce test section Mach numbers of 35.

Nomenclature

A = cross-sectional area
h = specific enthalpy
M = Mach number
p = pressure
T = absolute temperature

Subscripts
c = corrected for area change at diaphragm station
s = incident shock conditions
t = shock tube conditions
1 = driven tube initial conditions
4 = driver tube postcombustion conditions
5 = reflected conditions
∞ = freestream conditions

Superscripts
$'$ = equilibrium interface conditions
$*$ = nozzle throat conditions

Introduction

THERE has been a growing interest in hypersonic ground test facilities capable of duplicating actual flight conditions for NASP- and AOTV-type vehicles. As an example of such flight conditions, Fig. 1 shows the energy requirements[1] for the flight duplication for a selected NASP trajectory. In order to meet this challenge the present investigation examines the utilization of a 1.22-m-diam Combustion Driver Hypersonic Shock Tunnel,[2] operating at an equilibrium interface condition[3] to produce the necessary high-energy hypersonic airflow in the test section.

The interface compression technique was first suggested by Hertzberg,[3] and it is capable of considerably increasing both the pressure and temperature produced by the reflected method. This is accomplished through the additional compression provided by the approaching contact-surface which behaves as a "leaky-piston." Figure 2 shows a typical x-t diagram corresponding to this mode of operation as well as pressure and temperature history plots. These plots clearly show the advantage of the interface compression technique over the reflected technique on the basis of reservoir pressures and temperatures. Such a technique has been successfully used by several authors[4–6] to increase the shock tunnel test envelope. References 4 and 5 present the utilization of the described method for near ambient temperature helium drivers, while Ref. 6 reports the application of the technique to combustion heated helium drivers.

One of the major problems associated with the equilibrium interface technique has been the contamination of the driven gas, air, by the driver gas, helium. Such a contamination is caused by two main mechanisms: 1) diffusion through the interface and 2) the so called "jetting" phenomenon.[6] As will be seen later, it was experimentally observed that this con-

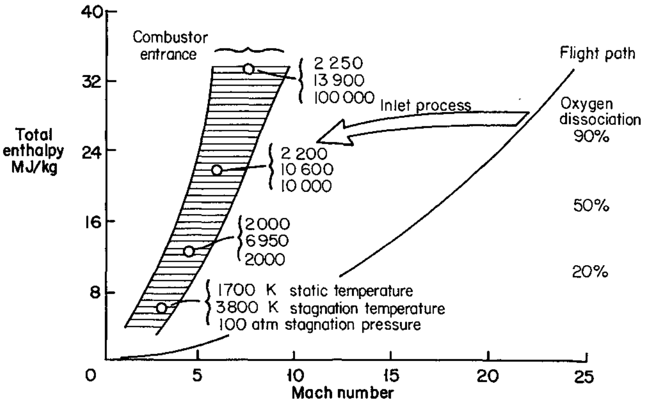

Fig. 1 Enthalpy requirements for flight conditions duplication in ground test facilities.

Presented as Paper 92-0569 at the AIAA 30th Aerospace Sciences Meeting, Reno, NV, Jan. 6–9, 1992; received Sept. 3, 1992; revision received June 22, 1993; accepted for publication July 9, 1993.

*Research Engineer, Captain in the Brazilian Air Force. Member AIAA.

†Active Professor Emeritus of Aeronautical Engineering. Fellow AIAA.

‡Professor of Aeronautical Engineering. Member AIAA.

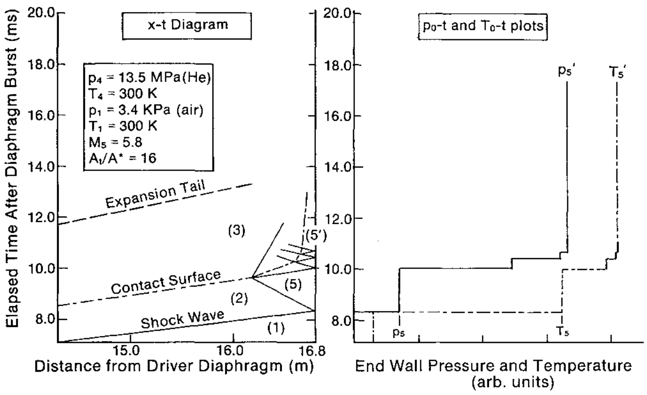

Fig. 2 Wave diagram and end-wall equilibrium interface pressure and temperature histories (T_4 = 25°C).

tamination can be minimized by using a high molecular weight gas separating the driver gas from the driven gas. The use of this separating gas will also improve the final equilibrium condition achieved by this technique.

All the reservoir and test condition calculations were carried out for the Rensselaer Polytechnic Institute 1.22-m-diam Combustion Driver Hypersonic Shock Tunnel. This facility was designed and used by the senior author at the General Electric Research Laboratory in the early 1960s for the development of the ICBMs and Scramjet inlet studies.[2]

The above calculations were performed using the HSTR1 computer code[5,7] which will be discussed later. This code actually models the reflected shock wave-contact surface interactions that take place during the approach to the equilibrium. The capacity of the HSTR1 algorithm to predict reservoir pressures and temperatures has been verified experimentally as discussed in Ref. 5. This reference presents numerical and experimental results for the RPI 0.61-m-diam Hypersonic Shock Tunnel operating at an equilibrium interface condition of 4100 K. The shock tunnel conditions used to produce this equilibrium interface are indicated in Fig. 2. As can be observed from this figure, the driver gas used was helium at near ambient temperature. This relatively low initial temperature, T_4 = 300 K, in conjunction with a high Mach number shock wave, M_s = 5.8, were able to produce "hard" contact surfaces. These contact surfaces, in turn, provided efficient compression of the test gas slug, raising its temperature and pressure to 4100 K and 850 psi, respectively, for 5–6 ms of useful test time.

Unfortunately, it is well known that in order to generate reservoir pressures and temperatures representative of NASP flight envelope, e.g., Fig. 1, it is necessary to use incident shock wave Mach numbers much higher than 5.8. It is also equally known that cold gas drivers can only produce incident shock waves with limited intensities. Such a limitation can be overcome by heating the driver gas. This procedure, however, has the inconvenience of producing "soft" interfaces. This characteristic will not permit the achievement of the interface compression for low-speed incident shock waves. Therefore, an optimum combination of shock wave strength and initial driver gas temperature T_4 must be found. As will be seen in the following sections, such a combination corresponds to overtailoring[3] the interface. The interest in combining the advantages from the use of the available Combustion Driver Hypersonic Shock Tunnel with those from the equilibrium interface operation, as well as determining the theoretical performance of this combination, motivated the present study.

RPI 1.22-m-Diam Combustion Driver Hypersonic Shock Tunnel

Since the present investigation deals with the operation of the RPI Combustion Driver Hypersonic Shock Tunnel in the equilibrium interface mode, the authors felt that a brief description of this facility would be of interest. A much more detailed description, however, can be found in either Ref. 2 or 8.

The Rensselaer Polytechnic Institute 1.22-m Combustion Driver Hypersonic Shock Tunnel, presently in storage, is shown schematically in Fig. 3. Figures 4 and 5 show actual photographs of the facility when in operation at the General Electric Research Laboratory, Schenectady, NY. The driver section is 15.24 cm in diameter and 6.10 m long, while the driven tube is 30.48 m long with an i.d. of 10.16 cm. A thick scored stainless steel diaphragm separates the driver gas from the driven gas. The driver tube, designed for a maximum operating pressure of 690 MPa (100,000 psi), is fitted with 18 special heavy-duty spark plugs.[8] A 30-deg conical nozzle with an exit diameter of 1.22 m is mounted at the end of the driven tube. Separating the nozzle section from the driven section, there is a thin scored aluminum diaphragm. This diaphragm bursts with the arrival of the incident shock wave. The hypersonic/hypervelocity airflow exiting the nozzle is exhausted into a 5.66-m^3 (200-ft^3) dump tank/test section. The dump tank can be pumped down to pressures below 0.133 Pa (1 μm of Hg) to accelerate the flow establisment in the nozzle. An electrically driven 15.24-cm-o.d. stainless steel sting supports the test model inside the test section.

The desired final driver temperature T_4, and pressure p_4, can be obtained by adjusting the initial driver gas composition and pressure.[2,8] This gas mixture consists of stoichiometric proportions of hydrogen and oxygen diluted by helium.

An area change at the diaphragm section, A_4/A_1 = 2.25, helps to decrease the required driver gas pressure ratio, p_4/p_1, necessary to produce the desired shock wave Mach number at a given driver gas temperature (after combustion). A double diaphragm section, to be installed immediately downstream of the area change, will contain a separating gas to minimize the spreading of the interface region and enhance the interface compression process. Preliminary experimental results, discussed in Refs. 5 and 7 and to be addressed later in the present work, do indicate that the proper choice of this separating gas can indeed improve the final equilibrium conditions. The double diaphragm section will also eliminate the necessity of a single, extremely thick, diaphragm required to withstand the driver gas elevated pressures.

Computer Program HSTR1

The HSTR1[5,7] algorithm actually models the reflected shock wave contact surface interactions, as seen in Fig. 2, that take place during the approach to the equilibrium condition. The airflow is assumed to be one-dimensional, inviscid, and in thermodynamic equilibrium. In this code, helium at a specified (by the user) temperature is the driver gas and it is modeled as a calorically perfect gas. The equilibrium air thermodynamic properties used were taken from Ref. 9 and are valid for temperatures up to 25,000 K.

The incident shock wave Mach number M_s, the initial driven temperature and pressure, T_1 and p_1, and both the driven and nozzle throat cross-sectional areas, A_1 and A^*, enables the HSTR1 code to model the partial reflections of the shock wave (incident or reflected off the interface) off the nozzle entrance. These partial reflections are due to the presence of the nozzle throat in the driven tube end wall.

The interactions of the reflected shocks with the helium interface are numerically modeled by simultaneously solving two sets of one-dimensional Euler equations. One for the transmitted shock (in helium) and the second one for the re-reflected shock (in equilibrium air). The computation is terminated when a Mach wave is produced from either the shock wave-nozzle reflection or from the shock wave-interface reflection.

The x-t diagram and the end-wall pressure and temperature histories, presented in Fig. 2, were computed using HSTR1 for the conditions indicated in this figure. Figure 6 shows a comparison between the end-wall pressure calculated using

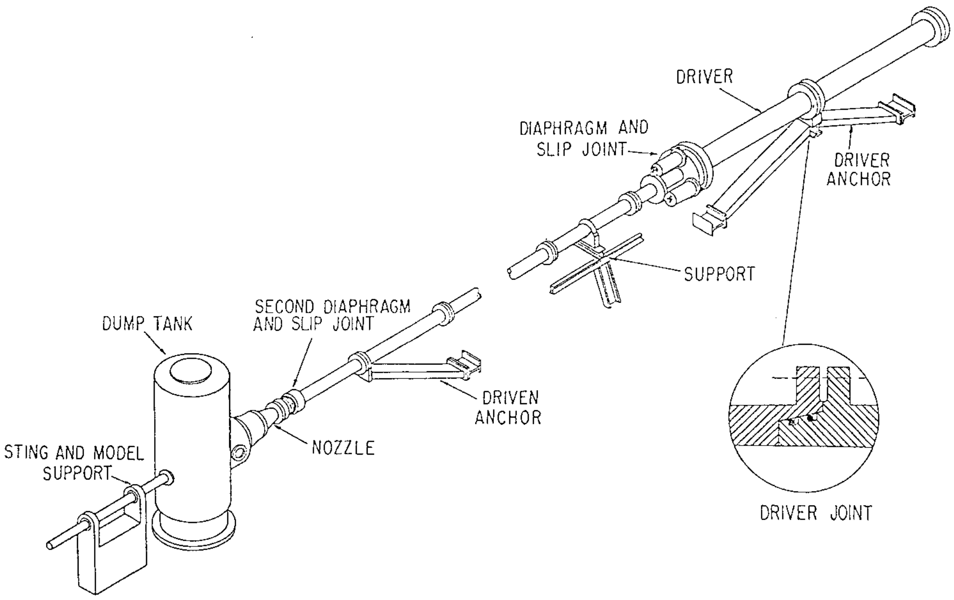

Fig. 3 RPI 1.22-m Combustion Hypersonic Shock Tunnel—overall schematic.

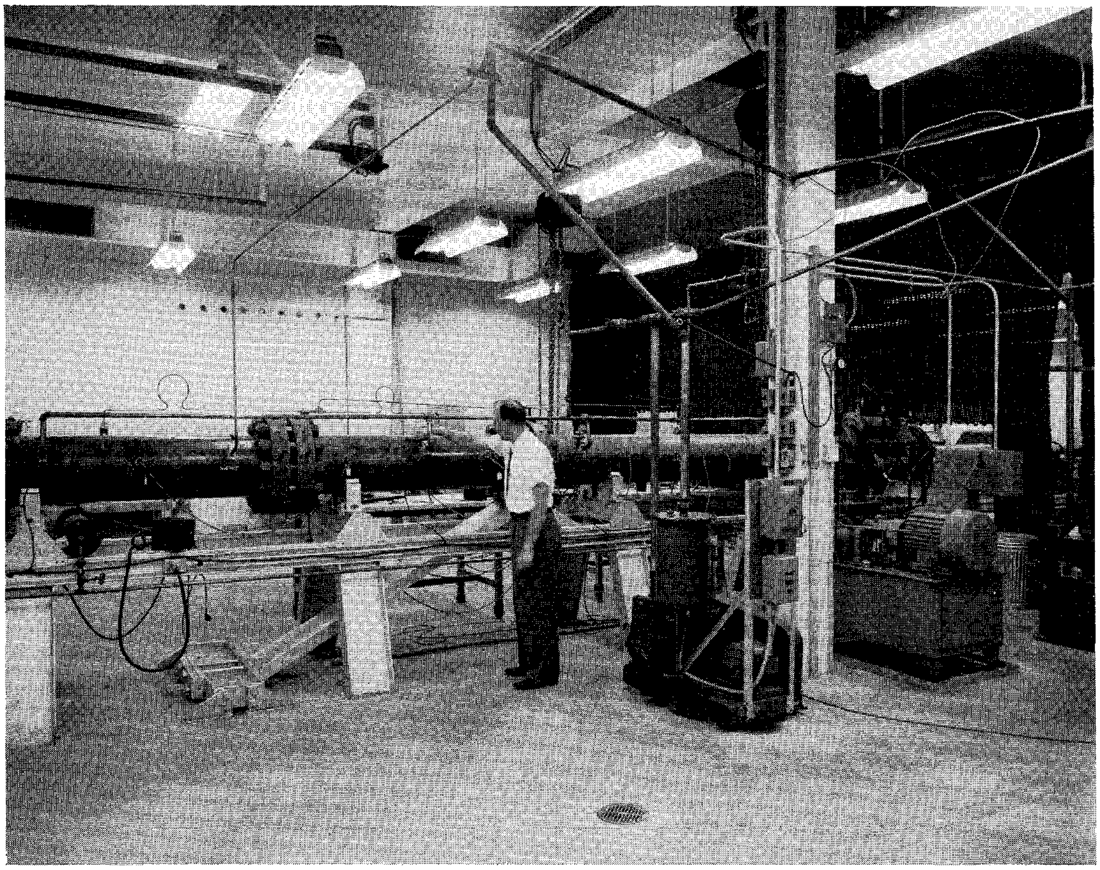

Fig. 4 RPI 1.22-m Combustion Hypersonic Shock Tunnel—driver end view.

the code HSTR1 and the experimentally measured for the conditions indicated. The experiment was carried out in the RPI 0.61-m Hypersonic Shock Tunnel and it is described in Ref. 5. The agreement between theory and experiment is quite encouraging. The underestimated theoretical pressure level indicated in Fig. 6 is believed to be caused mainly by viscous and shock overtaking effects.[10] These effects were not taken into account by the relatively simple theoretical model[5] used in the analysis. For more information about the HSTR1 algorithm architecture, the reader is referred to Ref. 7.

Numerical Results

For the present numerical investigation, the driver gas pressure after combustion, p_4, was kept constant at 207 MPa (30,000 psi). This procedure was followed in order to make the calculations simpler and to ensure that the driver gas pressure (after combustion) is within the structural limitations of the facility. Also, for the sake of simplicity, the presence of water vapor (steam) in the driver gas composition after combustion was neglected.

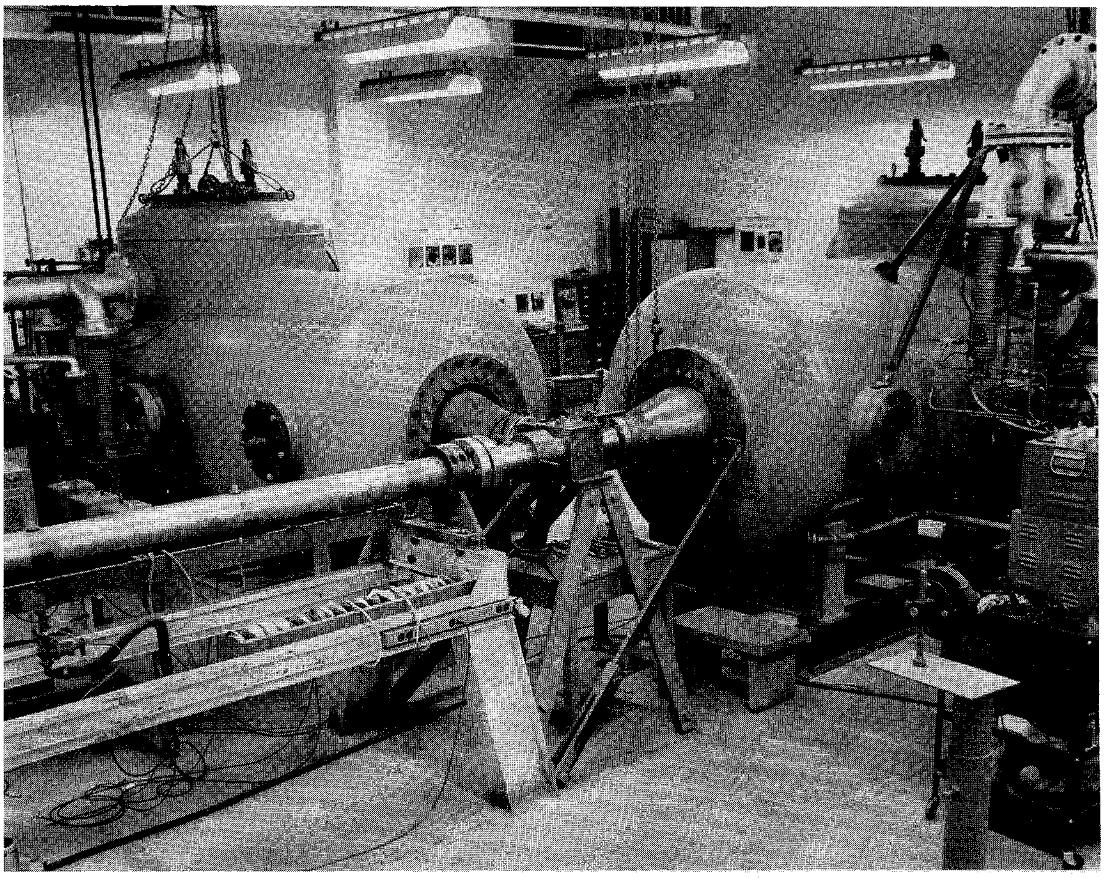

Fig. 5 RPI 1.22-m Combustion Hypersonic Shock Tunnel—nozzles and test sections.

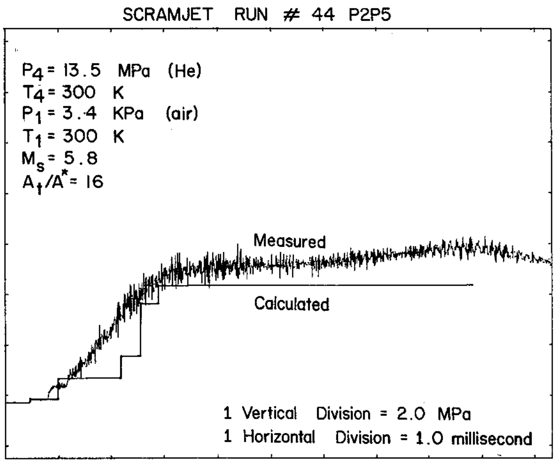

Fig. 6 Comparison between the measured and the calculated equilibrium interface pressures 20 cm upstream of the nozzle entrance.

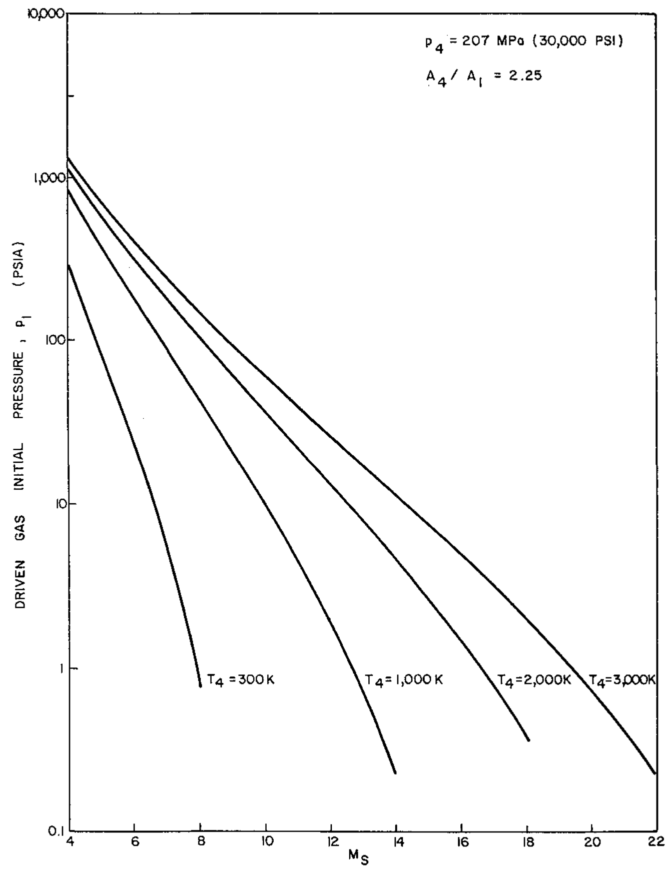

Fig. 7 Required driven gas initial pressure p_1 to produce a given shock Mach number for different driver gas initial temperatures T_4.

The desired incident shock wave Mach number is obtained by selecting p_1 and T_4 after combustion. This is shown in Fig. 7. It is of interest to point out that the described method of increasing the incident shock strength can be applied experimentally and it is not just a numerical "trick."

Since there is an area change at the diaphragm section, the initial (after combustion) driver gas pressure and temperature must be corrected by using the equivalent straight shock tunnel theory.[11] These corrections are

$$p_{4,c} = p_4 \times 1.42$$

$$T_{4,c} = T_4 \times (1.42)^{0.4}$$

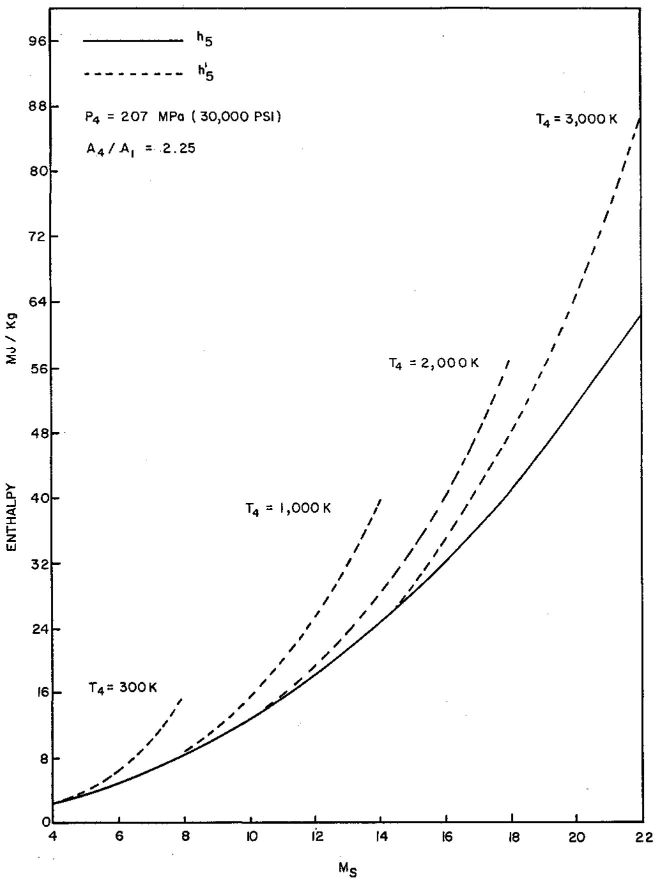

Fig. 8 Variation of the reflected enthalpy h_5 and the equilibrium interface enthalpy h_5' with the incident shock Mach number M_s.

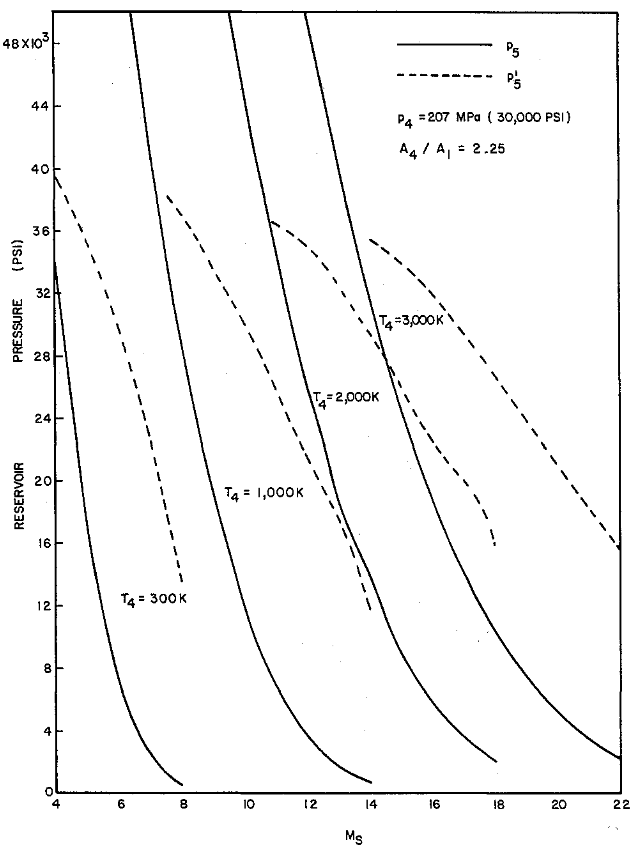

Fig. 9 Variation of the reflected pressure p_5 and the equilibrium interface pressure p_5' with the incident shock Mach number M_s.

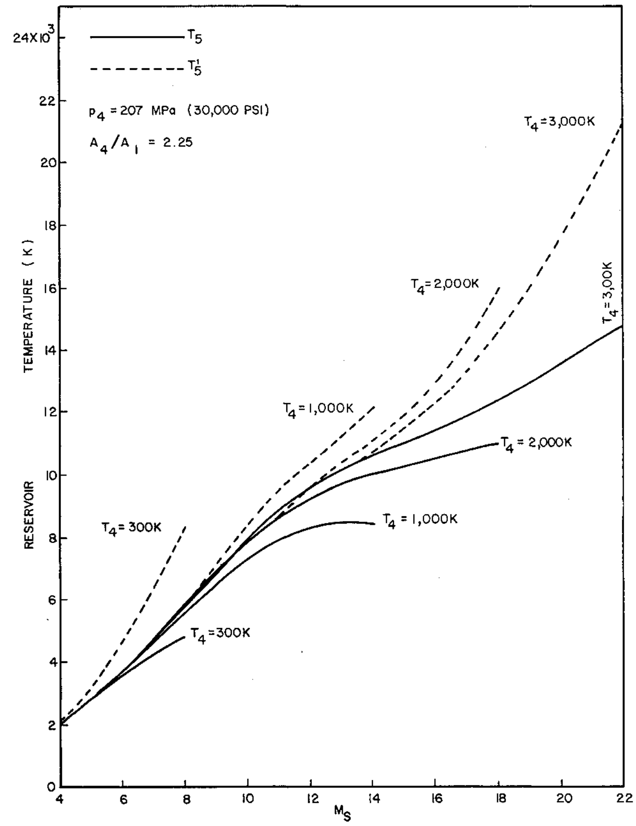

Fig. 10 Variation of the reflected temperature T_5 and the equilibrium interface temperature T_5' with the incident shock Mach number M_s.

Both the reflected and equilibrium pressures and temperatures for selected shock tunnel conditions are then calculated using the HSTR1 computer code.

Figures 8–10 show the results of the numerical simulation. In these figures, the maximum incident shock wave Mach number is 22, at $T_4 = 3000$ K. This Mach number value corresponds to the theoretical maximum value that can be achieved with an initial driver pressure of 207 MPa and temperature of 3000 K for a minimum initial driven pressure of 847 Pa (0.25 in. of Hg). A doubt on whether or not such a Mach number of 22 could be achieved, in practice, due to the presence of water vapor in the driver gas is plausible. However, it is of interest to recall that Nagamatsu[12,13] was able to generate shock Mach numbers as high as 30 with a combustion heated driver.

The lower limit imposed on p_1 is tentative and minimizes the viscous effects which tend to decelerate the shock wave and increase the contamination of the test gas. The discussed lower limit in p_1 also limits the maximum attainable shock speed at lower postcombustion driver gas temperatures, namely $T_4 = 300$, 1000, and 2000 K. Of course, $T_4 = 300$ K corresponds to a no-combustion situation. On the other hand, the upper temperature limit for T_4, namely $T_4 = 3000$ K, corresponds to the maximum achievable combustion temperature. As could be expected, for a given driver gas temperature T_4, the interface compression becomes more efficient with the increasing incident shock Mach number. This is due to the fact that the interface becomes "harder."[5]

Figure 8 shows the reflected, h_5, and the equilibrium, h_5', enthalpies at the end of the driven tube, nozzle entrance, as a function of the shock strength having the postcombustion driver gas temperature as a parameter. One can easily see the considerable gain in the produced reservoir enthalpy that can be made through the equilibrium interface mode of operation. By recalling Fig. 1, it is possible to see that the required Mach 25 flight total enthalpy, 37 MJ/kg, can be obtained by producing an incident shock wave of Mach 14.6 in the described mode of operation. On the other hand, it would take a Mach 16.3 shock wave to produce the same enthalpy if the reflected mode was to be used. Figure 8 also indicates that enthalpies of the order of 88 MJ/kg, much higher than that required for the NASP testing, can be achieved through the equilibrium interface technique. In Fig. 8, as well as in Figs. 9 and 10, the points where the dashed lines, equilibrium interface conditions, cross the continuous lines, reflected conditions, correspond to the so-called tailored conditions. These conditions vary, of course, with driver initial, postcombustion, temperature. The critical shock Mach numbers, beyond which the interface compression process becomes efficient, can also be found in these figures. They correspond to the "crossing points" mentioned previously in this paragraph. The critical Mach number value increases with the increasing driver gas temperature. As an important conclusion, combustion heated drivers can be used to produce the interface compression if the generated incident shock wave Mach number is greater than the critical Mach number.

The reflected p_5 and equilibrium interface p_5' pressures are shown in Fig. 9. As one may realize from this figure, the final equilibrium interface pressures are many times higher than the original reflected pressures.

The features presented in Fig. 9 can be better appreciated with the aid of Fig. 10. The latter shows the variations of the reflected T_5 and equilibrium T_5' temperatures with respect to the incident shock wave Mach number having the initial driver gas temperature (after combustion) as a parameter. Again, reasonable gain in the produced stagnation temperature can be achieved by using the interface compression. Together, Figs. 9 and 10 indicate that the common problem of producing high reflected temperatures at the cost of producing lower reflected pressures is alleviated by the equilibrium interface technique. These figures show indeed that high temperatures can be produced at much higher pressures than the reflected ones through the discussed method. This is a very desirable feature as it not only enables the full flight simulation at high Mach numbers up to 15, but it also helps to minimize non-equilibrium effects in the nozzle flow.

The produced reservoir conditions shown in Figs. 8–10 are then used to generate a hypersonic/hypervelocity airflow in the shock tunnel test section. Due to the fact that the nozzle in the RPI facility is conical, the freestream Mach number can be changed by simply changing the nozzle throat insert. Therefore, provided the adequate reservoir conditions and test section background pressures, freestream Mach numbers ranging from 5 to 35, assuming an equilibrium expansion, can be produced.

Figure 11 shows the variation of both the test section flow Mach number and the corresponding flow velocity for a reservoir pressure p_5' of 20,000 psi, and a freestream temperature, T_∞ of 240 K. For the same reservoir pressure and test section air temperature, Fig. 12 shows the test section pressure and density variation with the reservoir temperature. Both Figs. 11 and 12 assume that air is in equilibrium. These figures indicate that extremely high airflow velocities and Mach numbers can be simulated with the RPI Combustion Driver Hypersonic Shock Tunnel operating in the equilibrium interface condition.

Figure 13 shows the test envelope that can be provided by the RPI 1.22-m-diam nozzle exit Combustion Hypersonic Shock Tunnel. For completeness, this figure also includes the testing capability of the RPI 0.61-m Hypersonic Shock Tunnel. The latter is already fully operational and, as indicated, is capable of duplicating the NASP flight conditions up to Mach 10. This facility has been used to experimentally investigate, among others, the equilibrium interface technique and the use of separating gases.[5,7]

As indicated in Fig. 13, the RPI Combustion Shock Tunnel can duplicate NASP flight conditions up to Mach 15. Beyond this Mach number, the flight conditions can be only partially duplicated. However, this facility does permit the combustion section entrance conditions duplication up to Mach 25. This

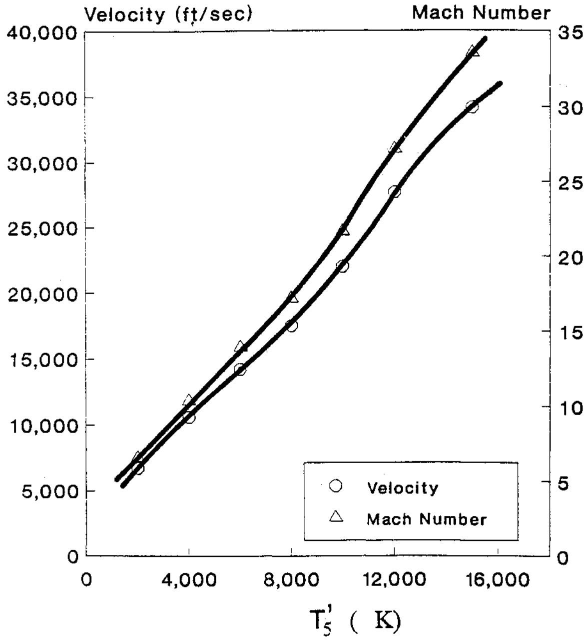

Fig. 11 Test section conditions—freestream velocity and Mach number. $p_5' = 20{,}000$ psi, $T_\infty = 240$ K.

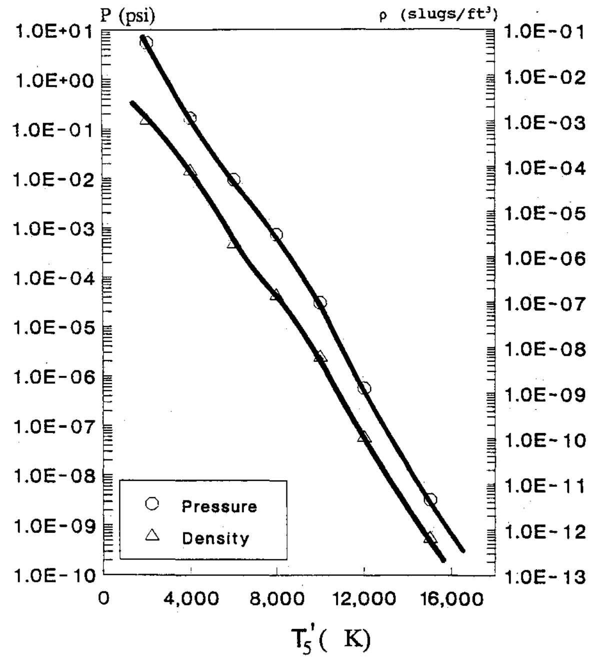

Fig. 12 Test section conditions—freestream pressure and density. $p_5' = 20{,}000$ psi, $T_\infty = 240$ K.

is possible due to lower stagnation pressures required at that section.

Experimental Results Obtained with a Shock Tunnel Using a Separating Gas

As stated previously in this work, one of the major drawbacks associated with the equilibrium interface technique is the contamination of the test gas by the driver gas. Experiments[5,7] performed in the RPI 0.61-m Hypersonic Shock Tunnel revealed that such a contamination can be minimized through the use of a interface region separating the driver gas from the driven gas. More important yet, the length of this separating region is quite small, 6.67 cm, with a corresponding internal volume of 540.76 cm^3. It has also been verified that by filling this buffer region with a high molecular weight gas,

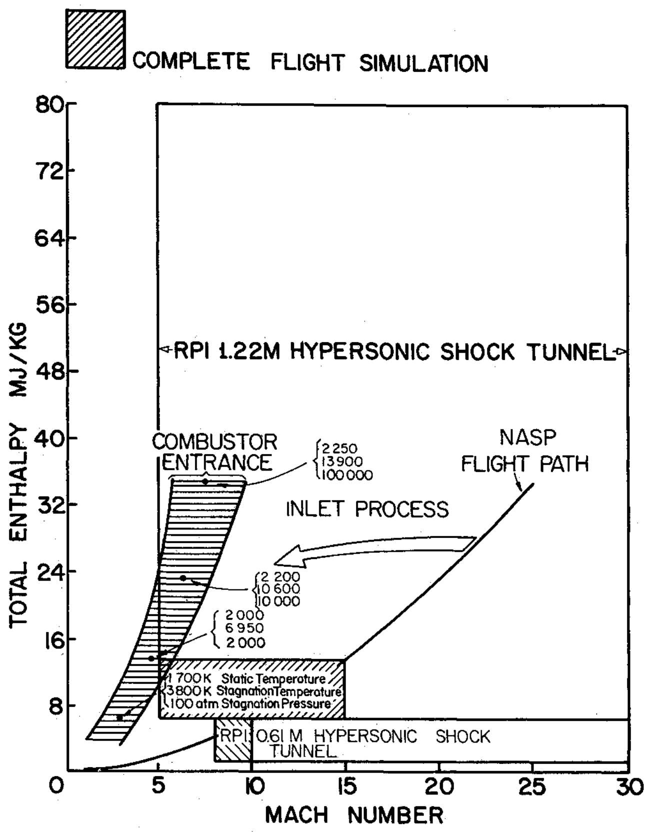

Fig. 13 Test envelope provided by the RPI 1.22-m Combustion Hypersonic Shock Tunnel operating in the equilibrium interface mode.

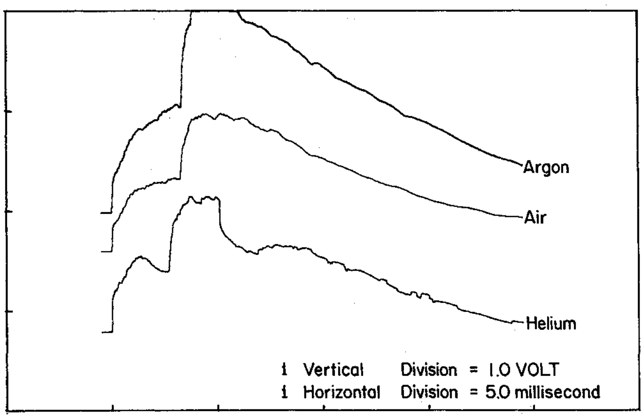

Fig. 14 Influence of the separating gas on the heat transfer rate to the shock tunnel wall measured at 66 cm upstream of the nozzle entrance.

dramatic improvements can be achieved in minimizing mixing between driver and driven gases.

Figures 14 and 15 show the influence of three different separating gases on the performance of the RPI 0.61-m Hypersonic Shock Tunnel operating in the equilibrium interface condition. Figure 14 demonstrates the influence of the separating gas on the spreading of the contact surface. This figure shows three different heat transfer traces, each one for a different separating gas. The heat transfer gauge was mounted in the shock tunnel wall 66 cm upstream of nozzle entrance. The lower trace in Fig. 14 corresponds to that obtained when helium is used as a separating gas. The heat transfer drop, located between the first and second jumps, indicates the arrival of the leading edge of the constant pressure mixing region (contact surface). The middle trace was obtained when air was used as a separating gas. In this particular case, the heat transfer drop cannot be observed. Instead, only a change in slope can be detected. Finally, when argon was used (upper trace) the heat transfer rate between the first and second jump (trace) stays practically unchanged. This is a good indication that the interface spreading and, therefore, the test gas contamination have been minimized.

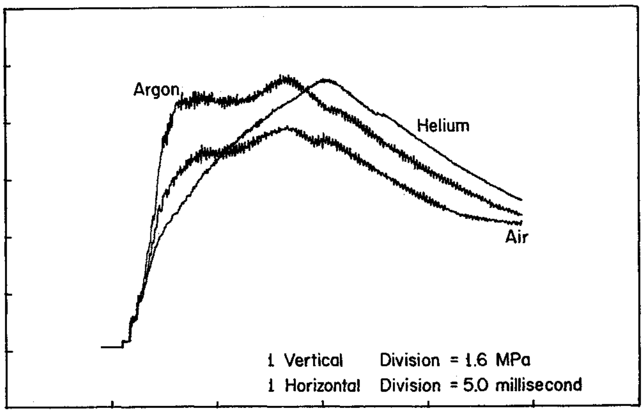

Fig. 15 Influence of the separating gas on the equilibrium interface pressure measured at 20 cm upstream of the nozzle entrance.

Figure 15 shows the effect of the same separating gases discussed in the previous paragraph on the equilibrium interface pressure. The piezoelectric pressure transducer was mounted in the driven tube wall and is located 20 cm upstream of the nozzle entrance. Here again, the best performance is obtained when argon is used in the separating region. It produces a higher and more uniform reservoir pressure level when compared to those produced by either helium or air.

The trend indicated by both the heat transfer and pressure traces leads to the use of high molecular weight gases in the separating region. Heavier gases seem to act as a barrier separating the driver gas from the driven gas. Furthermore, they act as a sort of a "gaseous piston" making the interface compression more efficient. Actually, under the microscopic point of view, such a behavior was to be expected. Heavy molecules have low mean thermal velocities and, therefore, tend to diffuse less.[14] This characteristic inhibits the diffusion that would take place through the helium/air interface in a conventional shock tunnel. The results reported in this session are indeed very promising for the equilibrium interface technique. The experimental work in this area continues[15] so that an analytical modeling of the phenomenon will be made possible in the near future.

Conclusions and Future Research

A numerical investigation has been conducted on the operation of the RPI 1.22-m Combustion Driver Hypersonic Shock Tunnel in the equilibrium interface mode. For this purpose, a computer code, HSTR1, was developed to model the shock wave-shock wave interactions, as well as the shock wave-contact surface interactions. The results indicate that extremely high reservoir enthalpies, 88 MJ/kg, and pressures, up to 30,000 psi, can be produced through this mode of operation if the generated incident shock wave Mach number is higher than a critical value. This critical value is the tailoring value, and for heated helium drivers, it varies only with the driver gas initial temperature. The numerical results also indicate that the Combustion Driver Shock Tunnel can fully duplicate NASP flight conditions up to Mach 15 and partially duplicate these conditions beyond this Mach number.

Experimental results obtained with a shock tunnel using a separating gas indicate that a small volume region, separating the driver gas from the driven gas, dramatically affects the spreading of the contact surface. It has been verified that such a spreading can be minimized by filling the separating region with a high molecular weight gas, e.g., argon.

Future directions of the present study include the upgrade of the computer code HSTR1 as well as the analytical modeling of the buffered shock tunnel. The HSTR1 code will be modified so that viscous effects will be taken into account and an estimate of the useful test time will be calculated.

In addition to the above directions, more tests are being performed in the RPI 0.61-m Hypersonic Shock Tunnel operating at an equilibrium interface condition of 5.8 MPa pressure, and 4100 K temperature. The goal is to provide experimental data to further validate the developed computer code HSTR1 and increase the understanding of the physics of the interface compression mechanism.

Also, the possibility of developing a magnetohydrodynamic accelerator-assisted wind tunnel[16] based on the RPI 1.22-m Combustion Driver Hypersonic Shock Tunnel is being investigated. Such a combination, MHD accelerator and shock tunnel, would further increase the testing capability of the RPI facility.

Acknowledgments

The authors wish to express their gratitude to R. A. Jones, F. Tesman, and C. Vannier for their help during the experiments conducted in the RPI 0.61-m Hypersonic Shock Tunnel. Acknowledgment is also due to D. Messitt for the calculation of the test section conditions. The technical support and equipment donations from the Watervliet Arsenal, Tektronix Inc., PCB Piezotronics, and RCA—General Electric are also gratefully acknowledged.

References

[1]Anderson, G., Kumar, A., and Erdos, J., "Progress in Hypersonic Combustion Technology with Computation and Experiment," AIAA Paper 90-5254, Oct. 1990.

[2]Nagamatsu, H. T., Geiger, R. E., and Sheer, R. E., Jr., "Hypersonic Shock Tunnel," *American Rocket Society Journal*, Vol. 29, No. 5, 1959, pp. 332–340.

[3]Hertzberg, A., Smith, W. E., Glick, H. S., and Squire, W., "Modifications of the Shock Tube for the Generation of Hypersonic Flow," Arnold Engineering Development Center TN 55-15, March 1955.

[4]Copper, J. A., "Experimental Investigation of the Equilibrium Interface Technique," *Physics of Fluids*, Vol. 5, No. 7, 1962, pp. 844–849.

[5]Minucci, M. A. S., and Nagamatsu, H. T., "Hypersonic Shock-Tunnel Testing at an Equilibrium Interface Condition of 4100 K," *Journal of Thermophysics and Heat Transfer*, Vol. 7, No. 2, 1993, pp. 251–260.

[6]Neumann, R. D., "Requirements in the 1990's for High Enthalpy, Ground Test Facilities for CFD Validation," AIAA Paper 90-1401, June 1990.

[7]Minucci, M. A. S., "An Experimental Investigation of a 2-D Scramjet Inlet at Flow Mach Numbers of 8 to 25 and Stagnation Temperatures of 800 to 4,100 K," Ph.D. Dissertation, Rensselaer Polytechnic Inst., Troy, NY, Aug. 1991.

[8]Nagamatsu, H. T., Sheer, R. E., Jr., Osburg, L. A., and Cary, K. H., "Design Features of the General Electric Research Laboratory Hypersonic Shock Tunnel," General Electric Research Lab. Rept. 61-RL-2711C, May 1961.

[9]Tannehill, J. C., and Mugge, P. H., "Improved Curve Fits for the Thermodynamic Properties of Equilibrium Air Suitable for Numerical Computation Using Time-Dependent or Shock-Capturing Methods," NASA CR-2470, Oct. 1974.

[10]Glass, I. I., and Gordon Hall, J., "Handbook of Supersonic Aerodynamics, Section 18 Shock Tubes," U.S. Naval Ordnance Lab. Rept. 1488, Vol. 6, Dec. 1959.

[11]Alpher, R. A., and White, D. R., "Ideal Theory of Shock Tubes with Area Change near Diaphragm," General Electric Research Lab. Rept. 57-RL-1664, Jan. 1957.

[12]Nagamatsu, H. T., and Sheer, R. E., Jr., "Magnetohydrodynamic Results for Highly Dissociated and Ionized Air Plasma," *Physics of Fluids*, Vol. 4, No. 6, 1961, pp. 1073–1084.

[13]Nagamatsu, H. T., Sheer, R. E., Jr., and Weil, J. A., "Non-Linear Electrical Conductivity of Plasma for Magnetohydrodynamic Power Generation," American Rocket Society Paper 2632-62, Nov. 1962.

[14]Kennard, E. H., *Kinetic Theory of Gases*, McGraw-Hill, New York, 1938.

[15]Minucci, M. A. S., Nascimento, M. A. C., and Nagamatsu, H. T., "An Investigation on a New Technique to Improve the Performance of the Shock Tube/Tunnel Testing in the Equilibrium Interface Condition," *Proceedings of the 33rd Israel Annual Conference on Aviation and Astronautics*, Israel, 1993, pp. 103–113.

[16]Lineberry, J. T., and Crawford, T. A., "The MHD Accelerator," *Mechanical Engineering*, Vol. 113, No. 9, 1991, pp. 70–74.