# Shock Train and Pseudo-shock Phenomena in Supersonic Internal Flows

**Kazuyasu Matsuo**
Department of Mechanical Systems and Environmental Engineering, Faculty of Environmental Engineering, University of Kitakyushu, Hibikino, Wakamatsu, Kitakyushu, 808-0135 Japan

When a normal shock wave interacts with a boundary layer along a wall surface in supersonic internal flows and the shock is strong enough to separate the boundary layer, the shock is bifurcated and a series of shocks called "shock train" is formed. The flow is decelerated from supersonic to subsonic through the whole interaction region that is referred to as "pseudo-shock". In the present paper some characteristics of the shock train and pseudo-shock and some examples of the pseudo-shocks in some flow devices are described.

**Keywords: shock wave, shock train, pseudo-shock, supersonic flow, internal flow.**

## Introduction

The interaction between a normal shock wave and a boundary layer along a wall surface appears in a variety of flow-devices including pipelines, supersonic wind tunnels, supersonic ejectors, supersonic inlets of aircraft engines, etc. In such an internal supersonic flow, when the normal shock is strong enough to separate the boundary layer, the shock is bifurcated and one or more shocks appear downstream of the bifurcated shock. A series of repeated shocks thus formed has been called in many ways by many researchers, multiple shocks[1], shock system[2], for example. We refer to such a series of repeated shocks in line as a "shock train", as in the paper by Carroll and Dutton[3]. The shock train is usually followed by an adverse pressure gradient region, if the duct is long enough[4]. The interaction region including both the shock train and the subsequent pressure recovery region is referred to as a "pseudo-shock", as Crocco[5] called it. However, the term "shock train" as well as "pseudo-shock" has been used in confusing way in most of previous papers. Furthermore, as the shock train and pseudo-shock phenomena are extremely complex, they are poorly understood in spite of decades of research. This paper describes the mechanism of the shock train and the pseudo-shock phenomena with the emphasis on the physics of the normal shock/boundary layer interaction involved in these processes.

## Definitions of Shock Train and Pseudo-shock

The definitions of the shock train and pseudo-shock have been given by the present author[4], and the difference is clearly shown in Fig.1, an illustration of a shock train and the supposedly drawn corresponding static pressure distributions at the wall and at the centerline of a long constant area duct. The horizontal axis represents the distance along the wall, and the vertical axis is the static pressure. The static pressure begins to rise at point 1 where the foot of the first shock of the shock train is located, and increases continuously at the wall, but the static pressure along the centerline rises and falls repeatedly up to the point j as a result of the successive normal shocks in the "shock train region". If the flow be fully subsonic and uniform at the point j, then the static pressure downstream of this point should decrease along the duct due to friction. However, the shock train is followed by "mixing region" as shown in the figure. In this region there exist no shocks, but the pressure increases to some extent due to the mixing of a highly non-uniform profile created by the shock train. The point j locates near the end of the shock train, and the static pressure reaches maximum at point 2. Since the static pressure rises from point 1 to point 2 and the flow is supersonic and subsonic at points 1 and 2, respectively, it may be reasonable to regard this region, that is, the "whole interaction region between the normal shock and boundary layer", as a "normal shock". However, it is not a real shock, although it contains a series of genuine shocks of the "shock train" within it. Hence we refer to this region as "pseudo-shock", as Crocco[5] called it. If no boundary layer is present, the pressure rises "discontinuously" from $p_1$ to the Rankine-Hugoniot value $p_{2n}$ by a single normal shock, as shown in Fig.1.

Received 2003

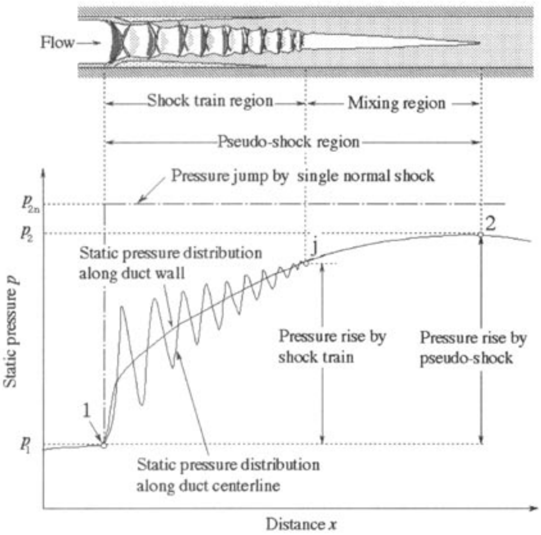



**Fig.1** Schematic static pressure distribution along duct centerline and wall surface in constant-area duct for condition wherein a “normal”" shock is present

In real flows, the pressure rise extends over a great distance rather “continuously” to the value $p_2$. The value $p_2$ is in general smaller than $p_{2n}$, which will be shown later.

The definition mentioned above may be difficult to distinguish the “pseudo-shock” from the “shock train” except for the case of long and nearly constant area ducts where the “mixing region” may appear. Hence, in the present paper, we use the term “shock train” when we pay attention to the series of successive shocks caused by the normal shock/boundary layer interactions, and, when the “whole interaction region” is considered to play a part of a single normal shock in flow passages, we use the term “pseudo-shock” in order to represent the whole interaction region. The pseudo-shock plays a very similar role as a single normal shock in the sense that the flow is decelerated from supersonic to subsonic, etc. Therefore, the normal shock in internal flows in the simplest assumption should be replaced by the pseudo-shock in more rigorous analysis, and it should be noted that, since the pseudo-shock is formed by the interaction of a normal shock with the boundary layer, the flow properties and flow pattern in the region are greatly affected by the boundary layer as well as the Mach number just upstream of the shock.

## Properties of Pseudo-shocks

In order to design and/or operate flow devices in which shock trains or pseudo-shocks are supposedly present, it will be required to estimate the pressure distribution along the flow direction, the pressure rise, and the length necessary to fully achieve the pressure rise. The following section deals with such properties of the pseudo-shocks in straight flow passages with constant area, under adiabatic wall condition.

### Static pressure distributions

As mentioned above, the shock train is followed by the mixing region, if the duct is long enough. When the duct is not long enough, the static pressure recovery region between the points j and 2 in Fig.1 becomes shorter, and the pressure recovery in this region reduces. An example of such a case is shown in Fig.2. This experiment was conducted by Waltrup and Billig[6], prompted by the design of scramjet engines. In this experiment, compressed air was fed through a contoured axisymmetric nozzle into constant area cylindrical ducts. Fig.2 shows the axial distributions of wall static pressure $p$ normalized by the plenum pressure $p_{op}$. The horizontal axis denotes the distance $x$ from the duct inlet (nozzle exit), and the duct exit is located at $x$=578 mm as shown in the figure. Evidently, the pressure rise in each case is performed only by the shock train, and there appears no subsequent pressure rise. A striking feature of these results is the similarity in shape and slope of all of the profiles. This suggests that the shock train structure for the tests with $p_{op}$ higher than the minimum $p_{op}$ represents just a proportional part of the shock train structure in the minimum case. After statistically analyzing their experimental results, Waltrup and Billig[6,7] developed an empirical equation in axisymmetric circular ducts between the pressure distribution in the shock train region and the flow parameters such as the upstream Mach number, the Reynolds number based on the boundary layer thickness, the duct diameter, etc. Modifying the Waltrup-Billig's equation, Sullins and McLafferty[8] suggested an correlation equation in

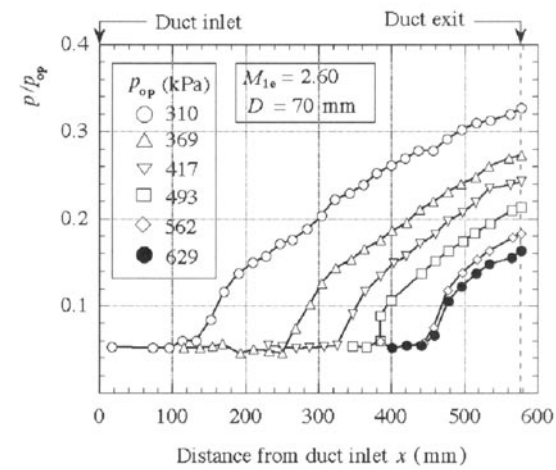



**Fig.2** Wall static pressure distributions in short cylindrical duct wherein a shock-train is included (Experimental data from Ref. [6])

rectangular ducts such as in isolators in air-breathing supersonic combustion ramjets. The flow structure including the pressure distributions in the shock train and pseudo-shock region for various duct cross section is still under discussion[9].

**Static pressure ratio across pseudo-shock**

We consider a duct with a constant area long enough for a maximum point such as the point 2 in Fig.1 appearing in the static pressure distributions. In this case, the static pressure ratio across the pseudo-shock can be defined by the ratio of the static pressure at point 2 to that at point 1. In general the measured static pressure ratio across the pseudo-shock depends on the freestream Mach number just upstream of the shock and the flow confinement effect of the boundary layer. In Fig.3, the measured static pressure ratio across the pseudo-shock[3,6, 10~14], $(p_2/p_1)_p$, divided by the theoretical pressure ratio across the normal shock for the same Mach number, $(p_2/p_1)_n$, is plotted against the freestream Mach number just upstream of the shock, $M_{1e}$. This figure contains the data obtained from sufficiently long ducts. The shaded area between two dashed lines denotes the range of scattering data. The experimental data show a great deal of scattering. This is considered mainly due to the fact that the flow confinement effect of the boundary layer is different from experiment to experiment. Anyway, it can be said from Fig.3 that the pressure ratio across the pseudo-shock is lower than that of an inviscid normal shock. This tendency becomes more remarkable as the Mach number increases. The reduction in the static pressure rise in comparison with that of the normal shock may be result from the existence of upstream boundary layer, wall friction, turbulence mixing loss occurring inside the pseudo-shock, etc. The effects of these parameters on the pseudo-shock flows are not fully understood so far. Further work is highly needed to obtain full understanding on this ambiguous problem.

**Length of pseudo-shock**

The length of the "shock train" can be defined clearly by the length of region where a series of shocks can be visible by optical observations. This definition can be applicable to any internal flow. On the other hand, in case of a long duct with a constant area, the length of "pseudo-shock" can be defined as the distance from point 1 to point 2 in Fig.1, that is, the distance from the onset of pressure rise to the point where the pressure is maximum. This definition is invalid for flows in various devices such as supersonic diffusers, ejectors, etc., because the flow passages in these devices change from place to place along the flow direction and so, due to the prevailing streamwise pressure gradient, the end point of the pseudo-shock cannot be identified. From the viewpoint of many engineering problems, it may be rather practical to diagnose what is the length necessary to fully achieve the pressure recovery in such a flow.

Fig.4 shows the relationship between the length of pseudo-shock, $L_p$, in the case of a long straight duct with a constant area divided by the equivalent diameter of the duct, $L_p/D$, and the freestream Mach number $M_{1e}$ upstream of the shock. These data are from Refs. [6, 10~12, 14] and [15]. The solid line represents the value calculated by the diffusion model[16] for pseudo-shocks. Similar to Fig.3, there is a great deal of scattering in the data in Fig.4, probably mainly due to different flow confinement effects of the boundary layers. To reduce this scattering, the effect of the flow properties associated with the wall boundary layer on the pseudo-shock length should be known. This problem awaits further work. In spite of this scattering, a qualitative explanation still can be made from the figure.

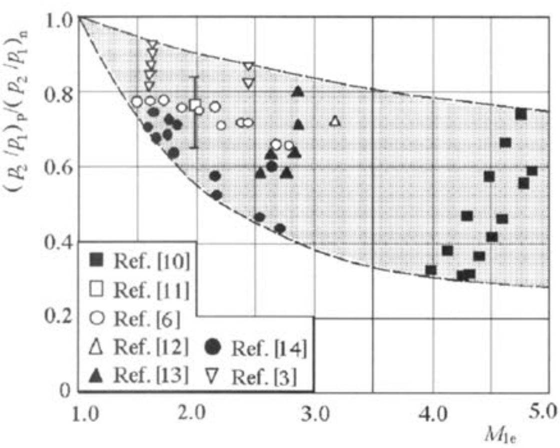



**Fig.3** Static pressure ratio across pseudo-shock normalized by that across normal shock versus freestream Mach number upstream of shock

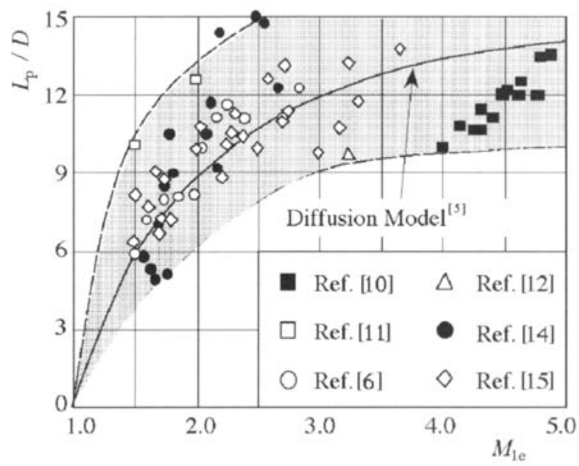



**Fig.4** Non-dimensional pseudo-shock length in constant area duct versus freestream Mach number upstream of shock

The experimental data have a tendency qualitatively similar to the solid curve, that is, the length of pseudo-shock increases with the increase in upstream Mach number, but it is very likely to approach a nearly constant value for further increase in Mach number. For the upstream Mach numbers larger than 2, it should be noted that the length of the pseudo-shock extends over 6 to 15 times the equivalent diameter of the flow passages. The pseudo-shock length is a strong function of the upstream Mach number and a weak function of the Reynolds number based on the upstream boundary layer thickness and the equivalent diameter of the duct.

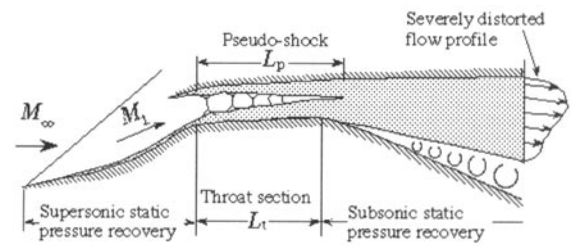
## Pseudo-Shocks in Flow Devices

The shock train and pseudo-shock phenomena commonly occur in confined, internal, supersonic flows where normal shocks are supposed to stand and interact with wall boundary layers, for example, in supersonic wind tunnel diffusers, supersonic ramjet or scramjet inlets, supersonic gas ejectors, etc. Some examples are shown in the following in such flow devices. Generally the cross-sectional area of flow in various devices changes along the flow direction. The variation of cross-sectional area would have no effect on a normal shock structure because of its extremely thin thickness. On the other hand, the pseudo-shock structure is significantly affected by the area change, because its length extends over a great distance along the passage. Hence the description in this section may be rather qualitative.

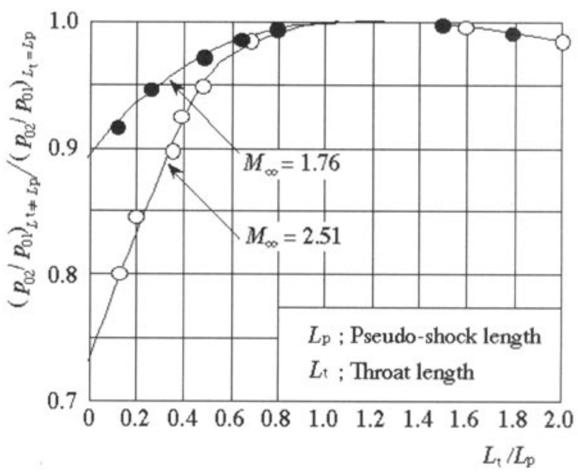
### Supersonic inlet diffusers

During operation in the supersonic velocity regime, air-breathing jet engines require an inlet which reduces the supersonic flow velocity to low subsonic and provides for the highest pressure recovery with a minimum stagnation pressure loss. Under the situation of a started inlet, the overall compression process in the supersonic inlet diffuser is made in three stages, that is, supersonic compression by compression waves and oblique shocks, deceleration with a "normal shock" from supersonic to subsonic, and the subsequent subsonic compression. The "normal shock" mentioned above in the supersonic inlet diffuser is usually called a terminal shock. The performance and stability of the supersonic inlet is strongly affected by the structure of the terminal shock and its location. The simplest realization of the terminal shock is a single normal shock. But such a simple pattern rarely occurs in reality. The boundary layer on the wall surface of the inlet interacts with the shock, creating a shock train and a pseudo-shock for a high Mach number.

Many experimental and computational investigations have been reported so far on the terminal shocks. Mahoney[17] showed very clearly the effect of the pseudo-shock length on the total pressure recovery in the supersonic inlet. As shown in Fig.5(a), the freestream Mach number $M_\infty$ is reduced to Mach number $M_1$ through the supersonic static pressure recovery region, and then the transition from supersonic to subsonic flow occurs across a pseudo-shock that stands in the minimum area, or throat, under normal design operating conditions. The result is a substantial reduction in the total pressure recovery and a severely distorted flow profile that can degrade combustion efficiency. A solution to the problem is that the throat of the inlet is stretched to contain the entire pseudo-shock. The effect of changes in throat length $L_t$ relative to pseudo-shock length $L_p$ is shown in Fig.5(b). If $L_t$ is equal to $L_p$, then the total pressure recovery, $p_{02}/p_{01}$, of the compression process is the maximum attainable with the inlet configuration. If $L_t$ is longer than $L_p$ there are additional viscous losses, but the overall total pressure recovery decreases only gradually with increasing throat length. If $L_t$ is shorter than $L_p$, the total pressure recovery decreases with decreasing throat length, and the recovery loss is much greater at a higher Mach number of $M_\infty$=2.51 than at a lower Mach number




(a) Schematic of shock train with separation in supersonic inlet diffuser




(b) Effect of throat length on inlet total pressure recovery

**Fig.5** Pseudo-shock in supersonic inlet diffuser and relation between throat length and total pressure recovery ( Original figures from Ref. [17] )

of $M_\infty$=1.76. From the results mentioned above, Mahoney concluded as follows: At any design Mach number, the total pressure recovery of a supersonic inlet is sensitive to throat section design. For throat lengths equal to or slightly greater than the pseudo-shock length, the recovery is the maximum attainable. In order to incorporate the proper length throat in any inlet, the designer must know the length of the pseudo-shock and parameters that control the pseudo-shock length.

### Inlet/combustor isolators

During the past two decades, a significant progress has been brought toward the realization of air-breathing supersonic combustion ramjets (called scramjets, briefly). Any air-breathing flight vehicle operating at hypersonic speeds will require a combined cycle engine which operates efficiently from low subsonic to supersonic speed range. From this point, the dual-mode engine concept was proposed by Curran and Stull[18]. When operating in the ramjet mode, the air leaves the inlet at supersonic speeds in the case of absence of combustion. However, the combustion in the combustor creates a back pressure which requires a shock to form ahead of the combustor. The shock is required to be of sufficient strength to decelerate the supersonic flow from the inlet to subsonic at the combustor entry for a conventional subsonic burning. This shock is usually referred to as the precombustion shock[19]. Stabilizing the precombustion shock and isolating it from the inlet is imperative to avoid an inlet unstart and a significant loss in thrust. For this purpose, a constant area duct is placed between the inlet and the combustor . This duct is called an isolator duct and its key function is to prevent the static pressure rise associated with combustor operation from unstarting the inlet. The precombustion shock in the isolator duct interacts with the wall boundary layers to form a shock train and pseudo-shock, and it has been investigated by many researchers[20].

## Concluding Remarks

The strong interaction of normal shocks with boundary layers occurring in internal compressible flows gives rise to shock trains or pseudo-shocks. Some characteristics of the shock trains and pseudo-shocks in constant area ducts were described, and examples of shock trains and pseudo-shocks in some flow devices were presented. The deceleration from supersonic to subsonic velocity through the pseudo-shock is a very complicated three-dimensional process. It is affected by the Mach number, the boundary layer thickness and the velocity distribution within the layer, the turbulence intensity of flow, the geometry of flow passage including its aspect ratio, the upstream and downstream flow conditions, particularly, the overall pressure ratio, etc. The deceleration process affects greatly on the performance and efficiency of flow devices. It almost always entail an increase in total pressure loss, hence deterioration of the performance of flow devices, and sometimes unsteadiness of the flow field. From the engineering and industrial point of view, it is desirable to develop proper control techniques to prevent such negative consequences. However, our current understanding of the mechanism and characteristics of the pseudo-shocks remains insufficient. There is a considerable amount of uncertainty still requiring a good deal of elucidation. A more thorough investigation is highly needed for fully understanding of the shock train and pseudo-shock phenomena.

## References

[1] Om, D, Childs, M E. Multiple Transonic Shock-wave/turbulent Boundary-layer Interaction in a Circular Duct. AIAA J., 1985, 23(5): 1506—1511
[2] McCormick, D C. Shock/boundary-layer Interaction Control with Vortex Generators and Passive Cavity. AIAA J., 1993, 31(1): 91—96
[3] Carroll, B F, Dutton, J C. Characteristics of Multiple Shock Wave/turbulent Boundary-layer Interactions in Rectangular Ducts. J. Propulsion and Power, 1990, 6(2): 186—193
[4] Matsuo, K, Miyazato, Y, Kim, H D. Shock Train and Pseudo-shock Phenomena in Internal Gas Flows. Progress in Aerospace Sciences, 1999, 35(1): 33—100
[5] Crocco, L. One-dimensional Treatment of Steady Gas dynamics. In Fundamentals of Gas Dynamics. Edited by H.W. Emmons. Princeton, U.S.A., 1958. 110—130
[6] Waltrup, P J, Billig, F S. Structure of Shock Waves in Cylindrical Ducts. AIAA J., 1973, 11(10): 1404—1408
[7] Waltrup, P J, Billig, F S. Prediction of Precombustion Wall Pressure Distributions in Scramjet Engines. J. of Spacecraft and Rockets, 1973, 10(9): 620—622
[8] Sullins, G, McLafferty, G. Experimental Results of Shock Trains in Rectangular Ducts. AIAA Paper No.92-5103, 1992
[9] Matsuo, K, Miyazato, Y, Kwon, J K, et al. Structure of Shock Trains in Constant Area Ducts. Experimental Heat Transfer, Fluid Mechnics and Thermodynmics, 2001, 3: 1859—1864
[10] Neumann, E P, Lustwerk, F. Supersonic Diffusers for Wind Tunnels. J. Appl. Mech., 1949, 16(2): 195—202
[11] Tamaki, T, Tomita, Y, Yamane, R. A Study of Pseudo-shock, 2nd Report, X-type Pseudo-shock. Bulletin of the JSME, 1971, 14(74): 807—817
[12] Ostras, V N, Penzin, V I. Experimental Study of Friction in a Channel with a Pseudo-shock. Fluid Mechanics-Soviet Research, 1975, 4(6): 32—38

(*continued on page* 197)

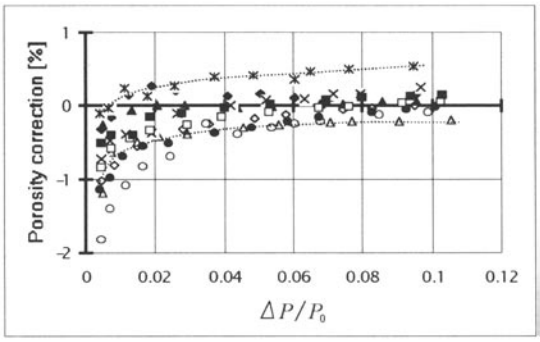



**Fig.7** Porositv correction for $A$ = 1.05 and $B$

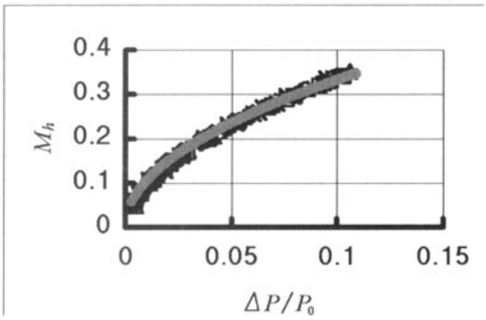



**Fig.8** Experiments with corrected porosity for $A$=1.05 and $B$=0.5

## Conclusions

A broad experimental program has allowed for the formulating of a correlation function, coupling mass flow rate with pressure drop over a perforated plate.

Moreover, a method of porosity determination has been proposed which relates its value to the aerodynamic performance.

It has been shown that transpiration flow through a perforated plate may be well described by the relation obtained from considerations of a compressible pipe flow with friction. It is an essential step forward in respect to earlier derived isentropic relations.

The theoretical analysis presented here is more general and concerns subsonic as well as supersonic flows in perforation holes. In the low Mach number limit the coincidence of derived relation with experimentally obtained flow characteristic is not bad. The application of $B$ = 0.5 deteriorates the coincidence with experimental results, but it could be used for the draft analysis of transpiration flow. It turned out that the loss coefficient of transpiration flow was $\lambda$ = 0.1296 for all perforated plates used in our experiments.

## References

[1] Bur, R, Corbel, B, Delery, J. Study of Passive Control in a Transonic Shock Wave/boundary Layer Interaction. AIAA Journal, 1998, 36(3): 394—400

[2] Frössel, W. Strömungen in Glatten Geraden Rohren mit Über- und Unterschallgeschwindigkeit. Forsch. Ingenieur wes, 1936, 7: 75—84

[3] Koppe, M. Der Reibungseinfluß auf stationäre Rohrströmungen bei hohen Geschwindigkeiten. Ber. Kaiser – Wilhelm – Ges. für Strömungsforschung, 1944

[4] Oswatitsch, K. Grundlagen der Gasdynamik. Wien: Springer, 1976. 107—112

[5] Poll, D I A, Danks, M, Humpreys, B E. The Aerodynamic Performance of Laser Drilled Sheets. Proc. of the 1st European Forum on Laminar Flow Technology. Hamburg, Germany, 1992

[6] Zierep, J, Bühler, K. Strömungsmechanik. Wien: Springer, 1991. 171—175

[7] Zierep, J, Bohning, R, Doerffer, P. Continuation of Modelling of Perforated Plate Aerodynamics Performance. Aerospace Science & Technology, accepted for printing

[8] Doerffer, P.P, Bohning, R. Modelling of Perforated Plate Aerodynamics Performance. Aerosp. Sci. Technol., 2000, 4: 525—534

(*continued from page* 208)

[13] Merkli, P E. Pressure Recovery in Rectangular Constant Area Supersonic Diffusers. AIAA J., 1976, 14(2): 168—172

[14] Ikui, T, Matsuo, K, Sasaguchi, K. Modified Diffusion Model of Pseudo-shock Waves Considering Upstream Boundary Layers. Bulletin of the JSME, 1981, 24(197): 1920—1927

[15] Handbook of Supersonic Aerodynamics Sec. 17. Navweps Rep. 1488, 1964, .6 (Ed. I.D.V. Faro)

[16] Ikui, T, Matsuo, K, Nagai, M. The Mechanism of Pseudo-shock Waves. Bulletin of JSME, 1974, 17(108): 731—739

[17] Mahoney, J J. Inlets for Supersonic Missiles. American Institute of Aeronautics and Astronautics, Inc., 1990. 55—66

[18] Curran, E T, Heiser, W H, Pratt, D T. Fluid Phenomena in Scramjet Combustion Systems. Annual Rev. Fluid Mech., 1996, 28: 323—360

[19] Sajben, M, Donovan, J F, Morris, M J. Experimental Investigation of Terminal Shock Sensors for Mixed-compression Inlets. J. Propulsion & Power, 1992, 8(1): 168—174

[20] Waltrup, P J, Billig, F S. Prediction of Precombustion Wall Pressure Distributions in Scramjet Engines. J. Spacecraft & Rockets, 1973, 10(9): 620—622