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Experimental Investigation on Shock Wave and Turbulent Boundary Layer Interactions in a Square Duct at Mach 2 and 4

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ABSTRACT

In order to investigate the supersonic internal flows with shock waves, a new supersonic wind tunnel (pressure-vacuum type, Mach 2.0 and 4.0) was designed and constructed in Muroran Institute of Technology. This paper describes firstly outlines of the new Mach 4 supersonic wind tunnel, and describes secondly the location, structure and characteristics of the Mach 2 and 4 pseudo-shock waves in a square duct. The structure and characteristics of the pseudo-shock waves were clarified by color schlieren photographs and duct wall pressure measurements.

INTRODUCTION

When a supersonic or hypersonic flow in ducts which could be a flow plug, nozzle or combustion heat release interact with downstream blockage devices, "pseudo-shock wave" or "shock-train" are produced as the result of shock wave and wall turbulent boundary layer interactions. The study of pseudo-shock waves has important implications for the design and operation of new air breathing engines for space planes, that is, inlet-combustor isolators for ramjet/scramjet engines, ejector and wind tunnel supersonic diffusers, supersonic centrifugal compressor, and so on.

So far, the macroscopic structures and characteristics of pseudo-shock waves in rectangular ducts at the low Mach number (below about Mach number 2) have been studied by many researchers (Ikui et al. (1974), Sugiyama et al. (1987, 1988, 1991, 1995), and Carroll & Dutton (1990, 1992)) and the macroscopic structures and characteristics of the pseudo-shock waves were clarified. However, the locations, structures and characteristics of pseudo-shock waves in large size rectangular ducts at the high Mach number (above Mach number 3) have not been fully clarified (Sugiyama et al. (2002)).

This paper describes firstly outlines of the new Mach

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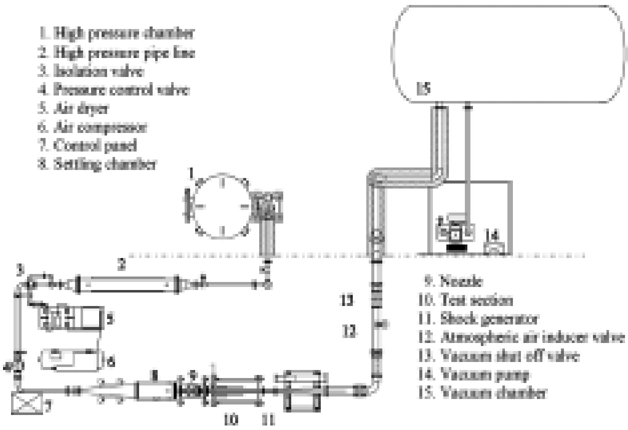


Fig.1 Pressure-vacuum type Mach 4 supersonic wind tunnel

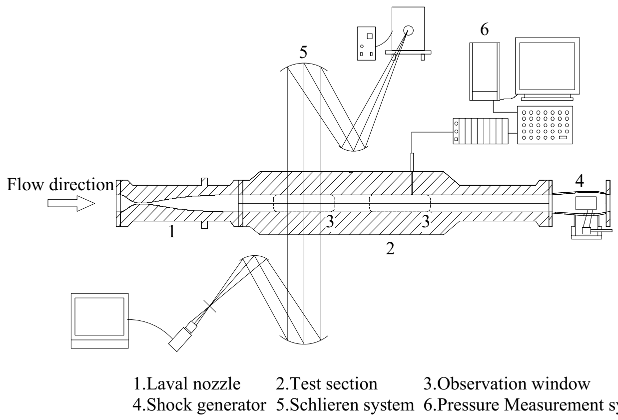
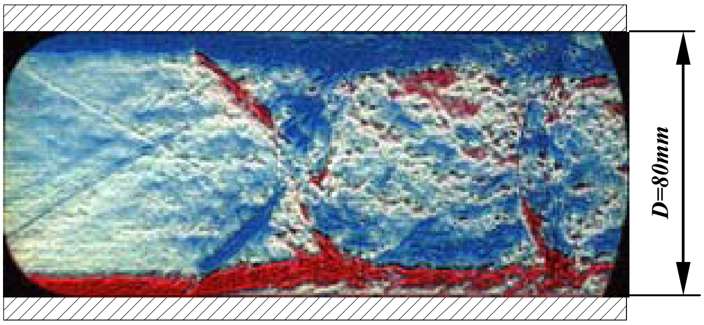


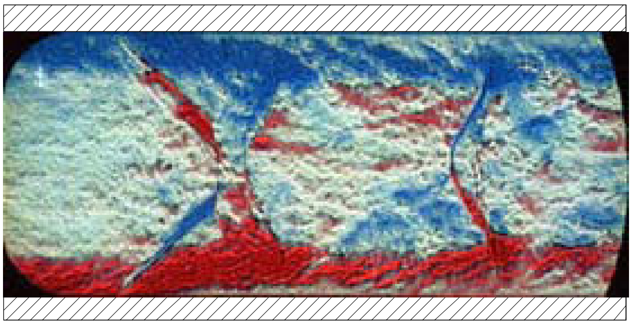
Fig.2 Test section for internal flow and measuring system

Table 1 Location of the first shock wave and flow confinement

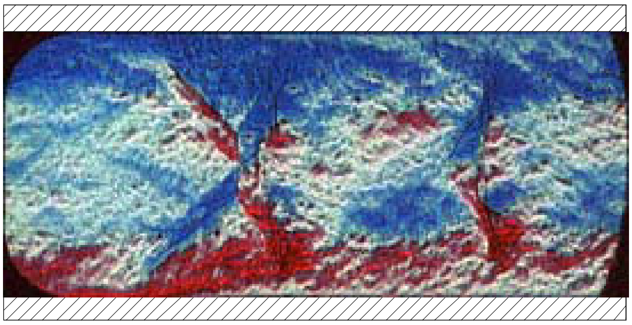
		Case A	Case B	Case C
M_∞=2	X_f/D	2.6	7.9	13.6
	δ_∞/h	0.15	0.25	0.35
M_∞=4	X_f/D	—	8.8	13.8
	δ_∞/h	0.28	0.39	0.47



(a) Case A ($X_f/D \fallingdotseq 2.6$, $\delta_\infty/h=0.15$)

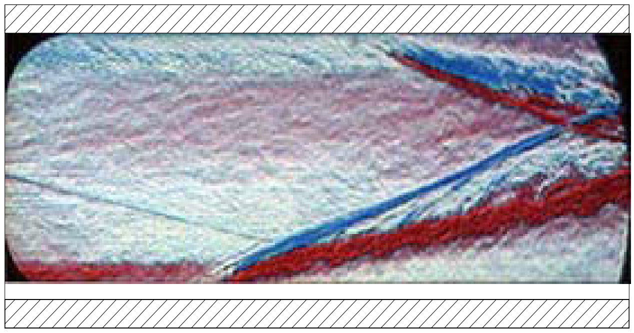


(b) Case B ($X_f/D \fallingdotseq 7.9$, $\delta_\infty/h=0.25$)

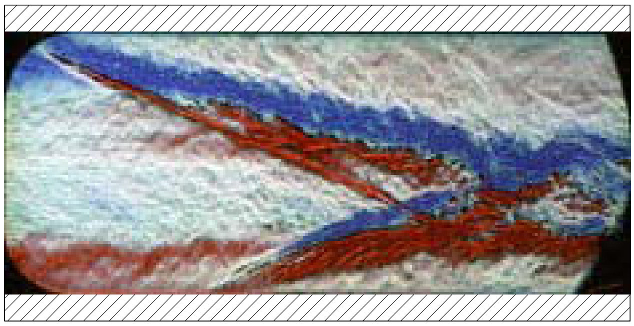


(c) Case C ($X_f/D \fallingdotseq 13.6$, $\delta_\infty/h=0.35$)

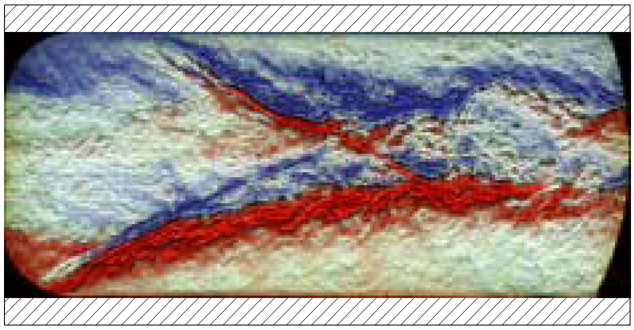
Fig.3 Schlieren photographs of the pseudo-shock wave ($M_\infty = 2$)



(a) Case A ($X_f/D \fallingdotseq 4.0$, $\delta_\infty/h=0.28$)



(b) Case B ($X_f/D \fallingdotseq 8.8$, $\delta_\infty/h=0.39$)



(c) Case C ($X_f/D \fallingdotseq 13.8$, $\delta_\infty/h=0.47$)

Fig.4 Schlieren photographs of the pseudo-shock wave ($M_\infty = 4$)

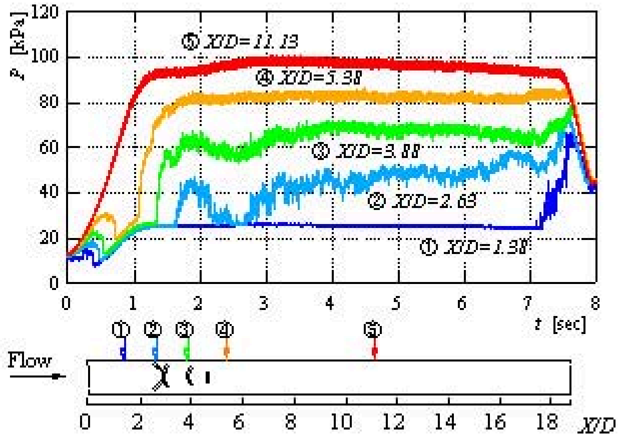
4 supersonic wind tunnel, and describes secondly the locations, structures and characteristics of the Mach 2 and 4 pseudo-shock waves in a square duct with 80×80 mm^2 cross section. The macroscopic structures and characteristics of the pseudo-shock waves were investigated by color schlieren photographs and duct wall pressure fluctuation measurements.

EXPERIMENTAL APPARATUS AND METHOD

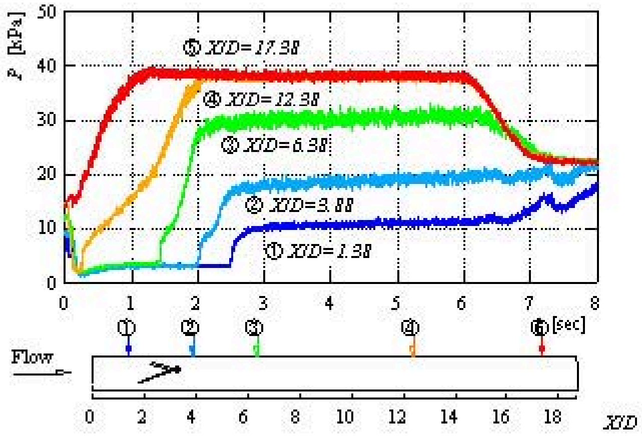
A new supersonic wind tunnel (pressure-vacuum type, Mach number 2 and 4) of Muroran Inst. of Technology was used in this experiment. Figure 1 shows the schematic diagram of the supersonic wind tunnel, and Fig.2 shows that of the test section and measuring systems. The test section is a square duct of width D = 80 mm and length L = 1,500 mm. Two-dimensional supersonic nozzles of design Mach number M_∞= 2.0 and 4.0 were used. The working times of the wind tunnel are about 15 and 20 seconds for Mach 2 and 4 conditions.

The structures of the pseudo-shock waves were visualized by color schlieren photography. A nanospark flash light with flash time 30 ns was used. Duct wall pressure fluctuations were measured simultaneously at 5 measuring points along the duct using semi-conductor pressure transducers (Kulite CT-190).

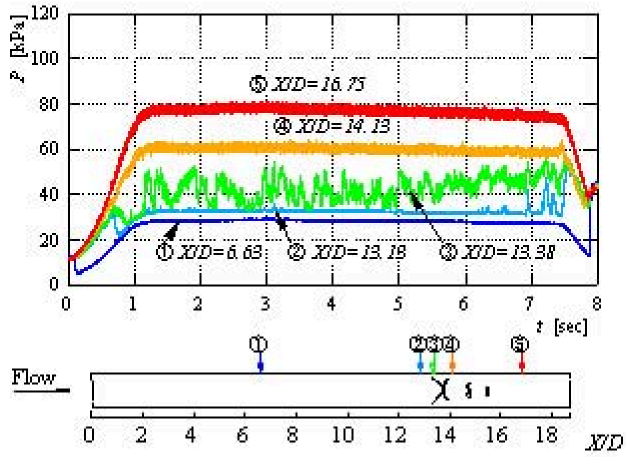
We produced pseudo-shock waves at upstream, middle stream and downstream locations of the duct by using the shock generator shown in Fig.2, that is, the present experiments were conducted for the three cases, i.e. cases A, B and C for free stream Mach number M_∞= 2 and 4. In case A, the location of the pseudo-shock wave was X_f/D = 2.6 for M_∞ = 2.0, however, for M_∞ = 4 the pseudo-shock wave was not exist in a defined location, because in this



(a) Case A (X$_f$/D ≒ 2.6)

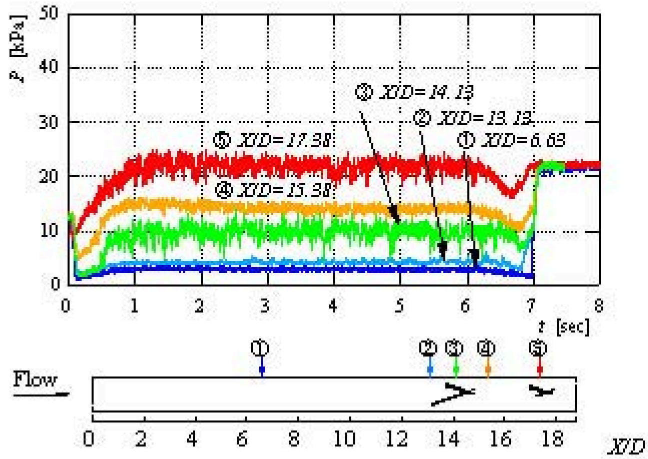


(a) Case A



(b) Case C (X$_f$/D ≒ 13.6)

Fig.5 Wall static pressure variations at 5 measuring points (M_∞=2)



(b) Case C (X$_f$/D ≒ 13.8)

Fig.6 Wall static pressure variations at 5 measuring points (M_∞=4)

case the pseudo-shock wave moves gradually from downstream to upstream, and it does not stay at a defined location. Table 1 shows the location of the pseudo-shock wave X_f/D and flow confinement δ_∞/h. Here X_f is the location of the first shock wave of pseudo-shock waves, δ_∞ the boundary layer thickness just ahead of the pseudo-shock wave and h the half width of the duct. The boundary layer thickness was estimated from schlieren photographs for M_∞ = 2 and LDV velocity profile measurements for M_∞ = 4. In the case of Mach 4, the agreement between the boundary layer thickness obtained from the LDV measurement and the one obtained from the Schlieren photograph is good. Unit Reynolds numbers just upstream of the pseudo-shock wave were Re = 2.53×10^7 /m, and 2.36×10^7 for Mach 2 and 4 flows, respectively.

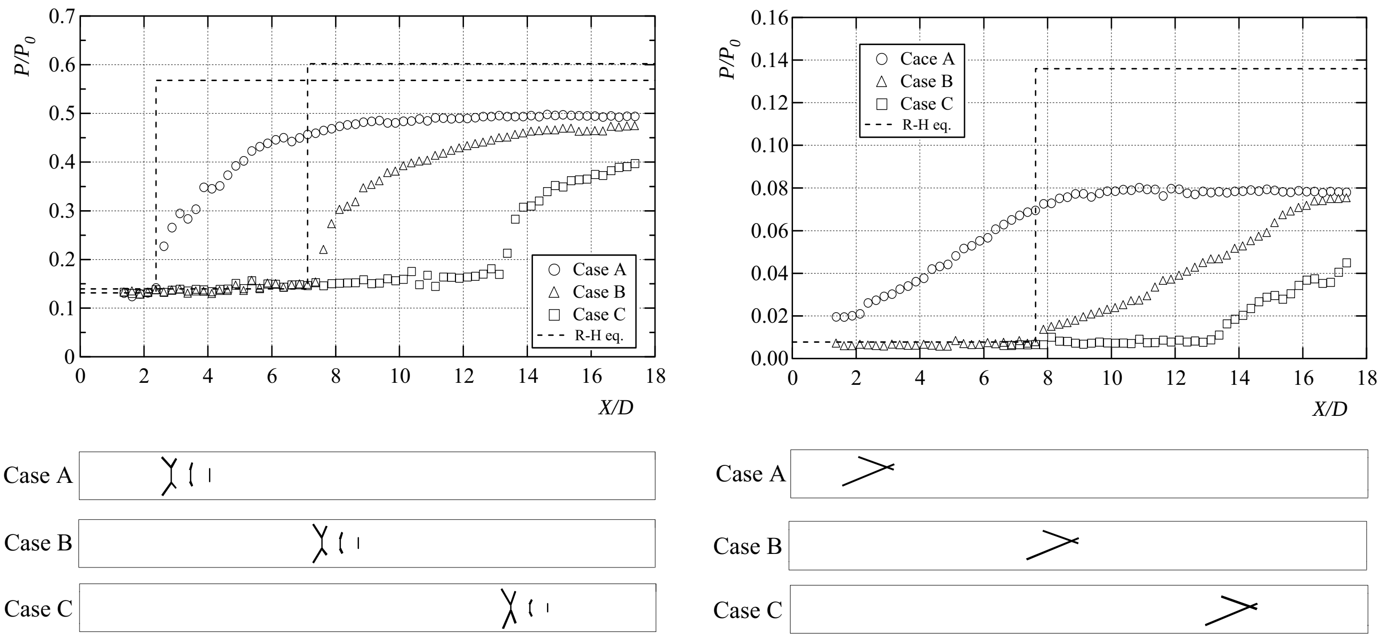
EXPERIMENTAL RESULTS AND DISCUSSION

Visualization of the pseudo-shock waves

Figures 3 (a), (b) and (c) show the schlieren photographs of the pseudo-shock waves for cases A, B and C, respectively. Flow direction is left to right, and color slit was set horizontally. The free-stream flow Mach number is M_∞= 2.0. From Figs.3 (a)～(c), we can observe clearly the λ-shape first shock wave and second shock wave, and the turbulent boundary layer along the top and bottom walls of the duct. The first shock is bifurcated while the following second and third shocks are not bifurcated. A large thickening of the boundary layer is observed through the interaction, and it may be considered that the boundary layer is separated under the first shock.

Figures 4 (a), (b) and (c) show the schlieren photographs of the pseudo-shock waves for the case A, B and C. The flow Mach number is M_∞ = 4. From Figs.4 (a)～(c), we can observe clearly the X-shape asymmetry first shock wave, and the turbulent boundary layer along the top and bottom walls of the duct, and large scale separation of the turbulent boundary layer. By the way, Figs.4 (a) and (c) show the case that boundary layer of the bottom wall side separates early, while Fig.4 (b) shows the case that boundary layer of the top wall side separates early. At present, we cannot predict which side wall boundary layer separates early.

From the point of view of flow separation, the Mach 4 supersonic internal flows with pseudo-shock waves



(a) $M_\infty=2$ **(b) $M_\infty=4$**

Fig.7 Wall static pressure distribution along the duct

change dramatically from those of Mach 2 flows.

Wall static pressure fluctuation of pseudo-shock waves

Figures 5 (a) and (b) show the wall pressure variations in the square duct with Mach 2 pseudo-shock waves at five measuring points indicated in the lower part of the figures. The ordinate is the wall surface static pressure fluctuation P and the abscissa is the time lapse after the wind tunnel starting. We call the measured wall pressure variation as wall pressure variation ① when the pressure fluctuation was measured at the measuring point①, and so on.

Figure 5 (a) shows the duct wall pressure variations in the case of A (the location of the pseudo-shock wave is X_f /D = 2.6). From the wall pressure variation ①, it is seen that the Mach 2 supersonic flow is established at t = 1.2 second and the flow continues about 7 second. From the wall pressure variations ②, ③ and ④ measured at under the leading shock of the first shock, second shock and downstream of the third shock, it is seen that the pseudo-shock wave oscillates in the main flow direction. It is also seen that at downstream of the shock train the duct wall pressure ⑤ is almost constant. These pressure variation tendencies are similar those of previous paper (Sugiyama et al. (1988)).

Figure 5 (b) shows the duct wall pressure variations in the case of C (the location of pseudo-shock wave is X_f /D = 13.6). In this case, the wall pressure variations are small compared with those of case A except the pressure variation ③. These tendencies are due to large flow confinement and the pressure increase by the shock train.

Figures 6 (a) and (b) show the duct wall pressure variations for the Mach 4 pseudo-shock wave flow. From the duct wall pressure variations ④, ③, ② and ① of Fig.6 (a), it is seen that the location of the Mach 4 pseudo-shock wave moves upstream with time lapse and the pseudo-shock wave does not stay at a defined location. Figure 6 (b) shows the duct wall pressure variations in the case of C (the location of the pseudo-shock wave is X_f /D = 13.8). From the wall pressure variations ① and ②, it is seen that the wall pressure variation at upstream location X/D = 6.63 is small and the wall pressure variation at X/D = 13.13 (just before the pseudo shock wave) is not so small because of the fully developed wall turbulent boundary layer effect. From the wall pressure variations ③, ⑤ and ④, it is seen that the wall pressure fluctuations under the first and second shock of the pseudo-shock wave is large, and wall pressure fluctuation between the first and second shocks is small.

Wall static pressure distribution of pseudo-shock waves

Figure 7 (a) shows the time-mean wall static pressure distributions along the duct with the Mach 2 pseudo-shock wave for the cases A, B and C, respectively. The ordinate is the non-dimensional pressure P/P_0, where P_0 is the upstream stagnation pressure, and the abscissa is the non-dimensional distance X/D, where X is the distance from the duct entrance and D the duct width. It is seen that the pressure rise by the pseudo-shock wave increases as the location of the pseudo-shock wave moves upstream and that the turbulent boundary layer in the pseudo-shock wave is applied by moderate adverse pressure gradient by the pseudo-shock wave. These tendencies are similar to those of previous paper (Sugiyama et al. (1988)). Wall static pressure rises by the Mach 2 pseudo-shock wave are $\Delta P/P_0$ = 0.50 and 0.47, that is, 88 % and 78% of the pressure rise by an ideal normal shock wave (the values obtained from the Rankine-Hugoniot equation) for the cases A and B.

Figure 7 (b) shows the time-mean wall static pressure distributions along the duct with the Mach 4 pseudo-shock wave. In the case of B, the pressure rise by the pseudo-shock wave is about 55% of the value obtained from the Rankine-Hugoniot equation for a Mach 4 normal shock wave.

From Figures 7 (a) and (b), it is seen that the pressure rise by the pseudo-shock wave increases with decreasing Mach number.

CONCLUSION

In this paper, the outlines of the new Mach 4 supersonic wind tunnel was firstly described, and the location, structure and characteristics of the Mach 2 and 4 pseudo-shock waves in a square duct was secondly described. The structure and characteristics of the pseudo-shock waves were clarified by color schlieren photographs and duct wall pressure fluctuation measurements. The experimental results are summarized as follows.

(1) In the case of uniform flow Mach number $M_\infty = 2$, a λ-type pseudo shock wave with symmetric geometry occurs in the square duct. In this case, the turbulent boundary layer along the square duct wall slightly separates under the first shock wave.

(2) In the case of $M_\infty = 4$, an X-type pseudo shock wave occurs in the duct, and the turbulent boundary layer separates in large under the first shock wave. In this case, an asymmetric flow which deviates greatly to top or bottom duct wall appears.

(3) From the point of view of flow separation, the flow structure of the Mach 4 pseudo-shock wave changes dramatically from the flow structure of Mach 2 pseudo-shock wave.

(4) Duct wall static pressure rises by the pseudo-shock waves depend on the locations of the pseudo-shock waves, namely flow-confinement.

(5) In the case of $M_\infty = 2$, large wall static pressure fluctuations due to the pseudo-shock wave oscillations occur under the first shock waves of the pseudo-shock waves, while in the case of $M_\infty = 4$, the large wall pressure fluctuations occur under the first and second shock waves of the pseudo-shock wave.

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