Memoirs of the Faculty of Engineering, Kyushu University, Vol.68, No.1, March 2008

Experimental Study on Effects of Shock Wave Impingement on Supersonic Combustion

by

Arif Nur HAKIM*, Shigeru ASO** and Yasuhiro TANI***

(Received February 1, 2008)

Abstract

An experimental study has been conducted using reflected type of shock tunnel to investigate the effect of shock wave impingement on supersonic combustion. In the experiment, air is compressed by reflected shock wave up to total temperature of 2800 K and total pressure of 0.35 MPa. Shock heated air is used as a reservoir gas of supersonic nozzle. Hydrogen gas is injected transversely through 2mm-diameter circular sonic nozzle into free stream of Mach 2. Flow duration is around 300 microseconds. The effects of shock wave impingement on the combustion has been studied by generating three different shock waves using ramp which is located on the opposite side of the supersonic nozzle wall. The wedge angles of the ramp of 5°, 10° and 15° are selected. Schlieren method is used to visualize flow pattern and shock structures and the UV-CCD camera is used to observe region of combustion because the camera could catch the self luminescence of OH radicals which is produced by combustion. The combination of Schlieren images and UV images shows the shock structure in the flow field and its effect to the supersonic combustion. The shock wave impingement enhances the combustion and shock induced separated region, which is observed at stronger shock impingement, could work as flame-holder.

Keywords: Shock tunnel, Supersonic combustion, Shock wave, SCRAM-Jet

1. Introduction

Supersonic combustion ramjet (SCRAM-jet) engine is currently one of most promising propulsion systems for achieving hypersonic airbreathing propulsion. Most of supersonic combustion experiments in the ground-based facilities use the combustion heated tunnel in order to produce supersonic hot air flow simulating the air flow in the real flight. The high temperature air

* Graduate Student, Department of Aeronautics and Astronautics
** Professor, Department of Aeronautics and Astronautics
*** Assistant Professor, Department of Aeronautics and Astronautics

flow is achieved through hydrogen-air combustion with oxygen replenishment to obtain the test gas with the same oxygen mole fraction as in the atmospheric air1). This type of experimental facility is relatively easy to operate and has long duration of the order of several seconds or longer. However, the high temperature test air obtained by the facility contains a large concentration of H2O as well as OH radicals that increases the uncertainty of the supersonic combustion experiments[2]. Another example of the tunnel used in the scramjet experiments is arc-heated tunnel1). However, as same as the combustion heated tunnel, the arc-heated tunnel also produces some gas contaminants such as nitrogen oxides and copper contaminants due to copper electrode erosion.

The alternative facilities that can be considered to generate the high temperature air flow without the contaminants are impulse facilities. Those facilities have a very short flow duration that is only the order of several milliseconds or lower. However those types of facilities are capable to provide the high total temperature and total pressure which are enough to conduct experiments. There are some types of impulse facilities such as shock tube3), expansion tube [4] and reflected shock tunnel [5]. However, compared with shock tube and expansion tube, the reflected shock tunnel offers longer test time. In the reflected shock tunnel, high pressure and high temperature reservoir gas is produced by compression of test air by incident shock wave and reflected shock wave.

In the scramjet engine, the residence time of air flow in the combustor is only the order of 1 millisecond. Therefore, injection method, mixing and combustion process become important problems for the success of airbreathing hypersonic vehicles. To achieve efficient combustion, fast mixing and fast combustion with minimum total pressure loses are required. In the scramjet combustor, the presence of oblique shock waves causing total pressure loses is often unavoidable. However, those shock waves also can be used as the means for enhancing the combustion. With its compression effects, the shock wave can increase the static temperature, static pressure. Those increase of static temperature and pressure contribute to increase reaction rates and accelerate combustion. In the past studies the concept of shock wave used for enhancing and stabilizing combustion is introduced. Dudebout et al.6) investigated hypersonic airbreathing propulsion utilizing shock induced combustion ramjets by numerical approach. Results demonstrated that shock induced combustion can be used as a viable means of hypersonic propulsion. Using the same model with Dudebout, Sislian et al.7) presented the comparative numerical study. Also, Arai et al.[8] investigated experimentally using similar model in shock tunnel.

Ben-Yakar et al.[9] also reported the experiments investigating the interaction of oblique shock wave with the fuel jet. The results demonstrated that the jet plume is bent towards the wall along with the OH radicals, which is present along the jet/freestream interface. Huh and Driscoll10) investigated the effects of shock waves on supersonic non premixed, jet-like flame experimentally. The hydrogen has been injected parallel with the airstream. The results have been reported that shock waves enhanced the air-fuel mixing and improved the flame stability limit substantially, when optimum oblique shock waves were introduced in supersonic jet-like flame. Also Kim et al.11) investigated numerically by using the same model with Huh's model and compared with Huh's results. However, there are few studies concerning a shock wave interacting with a reacting jet.

In the present study we have studied the effects of shock wave to enhance the combustion in the Mach 2 air stream. This work is an extension of our previous study[12, 13] in our laboratory on supersonic combustion with transverse injection of hydrogen without shock waves. In the previous study, hydrogen is injected in three different pressures, but in all cases the combustion is detected at about 45 mm downstream of the injection point, indicating that the induction time of combustion is in order of 10^{-5} sec. In the present study shock wave impingement into supersonic mixing region is conducted in order to observe the effect of shock wave impingement to supersonic combustion. Through the present study the shock wave impingement is proved to be quite effective to accelerate supersonic combustion.

2. Experimental Apparatus and Experiment Description

2.1 Experimental apparatus

The experiments have been conducted using the reflected shock tunnel as shown in **Fig. 1** at Kyushu University. In the reflected shock tunnel, test air is compressed by incident shock wave and reflected shock wave up to total pressure of 0.35 MPa and total temperature of 2800 K. The shock heated air is used as reservoir gas and expanded in the test section through two-dimensional supersonic nozzle of Mach 2 that is designed using a method of characteristics and manufactured by the present authors. **Figure 2** shows the enlarged view of test section. The distance from the throat of nozzle to the location of fuel injection is 100 mm. At the location of x = 100 mm (x is measured from the throat of the nozzle) the test section has cross section with width of 33 mm and height of 44 mm. The nozzle wall has a slope of 1° to keep the flow parallel in the test section. After the location of x = 120 mm the upper wall and lower wall become parallel and keep the same geometrical cross section of width of 36 mm and height of 44 mm. The hydrogen or helium, whose total temperature is same as room temperature, is injected transversely into freestream by using solenoid valve operated by time control system from 2mm-diameter circular sonic nozzle just before the test air arrives. The total pressure of injection gas is set in 0.35 MPa. To avoid injection gas to flow into low pressure section, which is located upstream of nozzle throat, aluminium foil diaphragm is installed at the throat of nozzle.

Three different angle ramps are mounted on the opposite wall of the test section in order to generate oblique shock wave. Schlieren photographs are taken at first without injection of gas in order to find the suitable location where shock wave generated by the ramp impinges the opposite wall. Also the assumed location of shock wave impingement point of oblique shock wave is calculated by using the oblique shock wave relation. Through those procedures final position of each ramp is decided. **Table 1** shows the angle and the position of each ramp calculating after assumed the position of shock wave interaction. The first ramp is mounted at 93 mm from throat (x = 93 mm) and had an angle of 5° which is considered as a weak shock generator. The second ramp is located at x = 97 mm and had an angle of 10° which is considered as a relative strong shock generator. And the third is located at x = 102 mm and had an angle of 15°, considered as a strong shock generator. Each ramp has a compression wall length of 20 mm. **Table 2** summarizes the change of flow properties across the oblique shock waves calculated using the oblique shock wave relations.

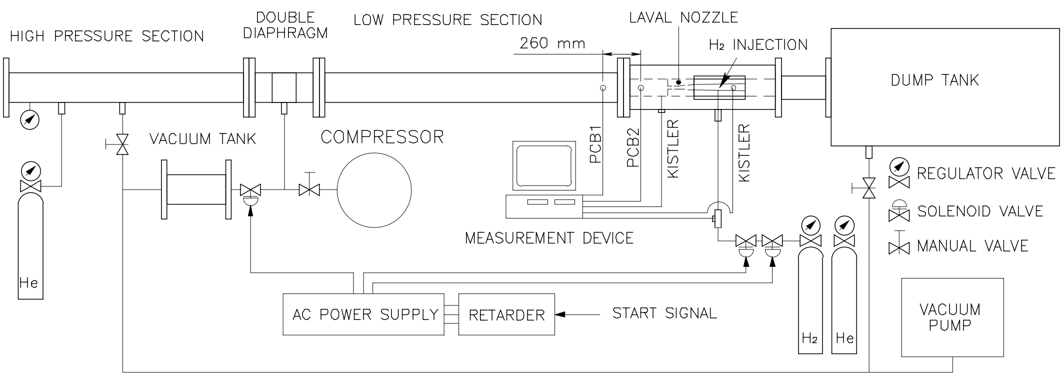


Fig. 1 Schematic Diagram of Shock Tunnel Facility.

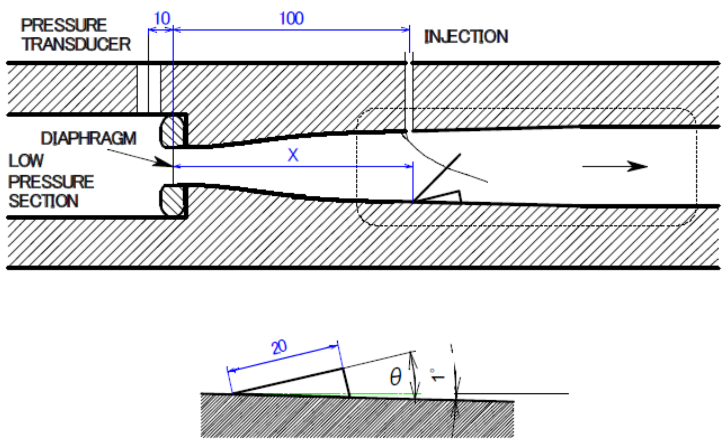


Fig. 2 Test Section (nozzle wall is inclined by 1 degree from horizontal axis x).

Table 1 Shock Generator (M_1=2).

Ramp	Angle (θ)	Position (x)
1	5°	93 mm
2	10°	97 mm
3	15°	102 mm

Table 2 Change of Flow Properties across the Oblique Shock (M_1=2).

Ramp Angle (θ)	Shock Deflection Angle (β)	$\frac{p_2}{p_1}$	$\frac{T_2}{T_1}$
5°	34.3°	1.21	1.07
10°	39.3°	1.58	1.14
15°	45.3°	1.92	1.24

2.2 Measurement of test condition

The total pressure is obtained in the low pressure section by measuring pressure of test gas at the stagnation condition after the incident shock wave is reflected. Measurement of this pressure is made by using Kistler pressure transducer located 10 mm before nozzle throat. On the other hand, the measurement of the stagnation heat flux of semi-sphere cylinder model, which is located at the center of test section and at the same position with injection point, has been conducted without injection of gas in order to obtain the total temperature of test air. The measurement is conducted by measuring temporal surface temperature rise at the stagnation point of semi-sphere model. For the measurement of surface temperature thin film platinum heat transfer gauge is used. Assuming an isentropic expansion of test air from the driven section to the test section and using combined relationship between total pressure of reservoir and the heat flux in the test section, the total temperature of test air in the test section can be obtained.

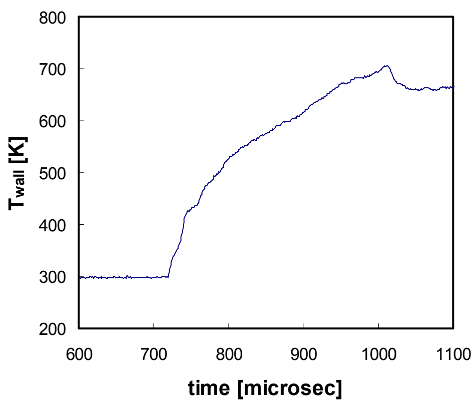


Fig. 3 Time history of wall temperature of platinum thin film heat transfer gauge in the test section.

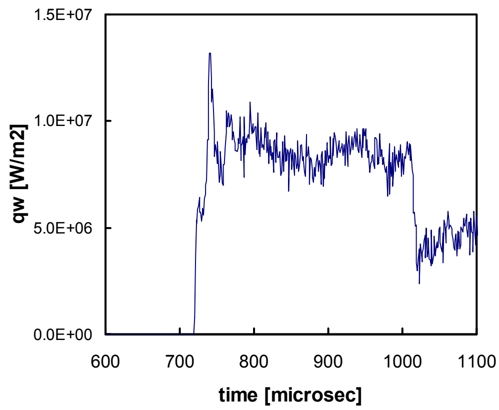


Fig. 4 Time history of heat flux in the test section.

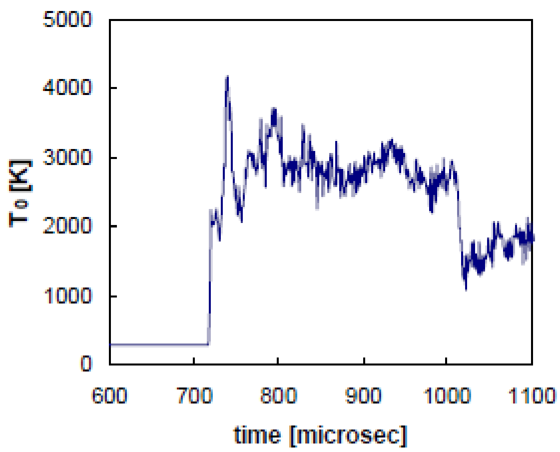


Fig. 5 Time history of total temperature of freestream calculated from heat flux and total pressure.

Figure 3 shows time history of wall temperature of semi-sphere cylinder model. The temperature begins to increase when starting shock wave arrives to body surface. **Figures 4** and **5** show the time history of heat flux calculated from time history of wall temperature and the calculated total temperature, respectively. The total temperature of test air begins to increase rapidly just after arrival of the starting shock wave and become relatively constant at about 2800 K. In the supersonic combustion, if the static temperature is high enough for fuel to initiate self ignition, no additional heating devices is needed. In the present investigation, the air static temperature at the location of injection is about 1550 K, which is quite enough for the self ignition.

Test time in this experiment, as shown in **Fig. 5**, is defined as the time in which the total temperature keeps constant, that is about 300 microseconds and this interval is enough to conduct supersonic combustion experiments. **Table 3** summarizes the experimental conditions for the present investigation.

Table 3 Experimental Conditions.

	Freestream	Injectant
Gas	Air	He, H_2
Total Pressure	0.350 MPa	0.350 MPa
Total Temperature	2800 K	290 K
Mach Number	2	1

2.3 Flow visualization

To visualize the shock structure of the supersonic flow field, Schlieren method has been conducted by using the instantaneous light source that have less than 1 microsecond exposure time. Helium and hydrogen were used as injection gas to investigate shock structure in both non-reacting and reacting cases respectively. Also Ultra Violet CCD camera is used to observe the OH self-luminescence which is generated in combustion flame front.

3. Results and Discussion

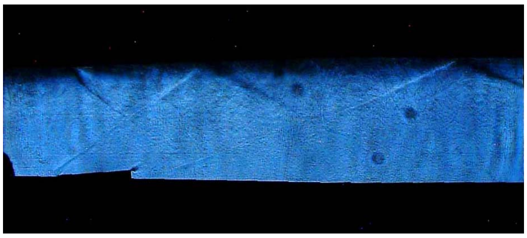
3.1 Flow characteristics

Figures 6 and **7** show Schlieren photographs of the flow field in reacting case (hydrogen injection) and non-reacting case (helium injection), respectively. In the **Fig. 6** and **7,** (a), (b) and (c) show the photographs in case of ramp angle 5°, 10° and 15°, respectively. The three-dimensional bow shock wave, which is formed due to interaction between fuel injection and free stream, can be seen clearly upstream of the injector in the figure. In the non-reacting case, shock wave appears more distinctly than in the reacting case because of the difference of molecular weight between helium and hydrogen. This also can be seen as the result of the turbulence due to the reaction between fuel and air in the recirculation zone. The combustion reaction process in the reacting case also caused the change of gas composition and gas density. Those phenomena can be seen in the figure. In the reacting case small white curve appears near the injection distinctly and the contrast of color is higher in downstream of the injection point compared with non-reacting case.

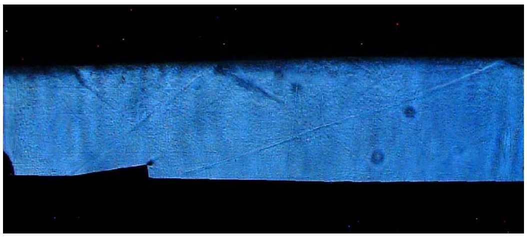
Planar oblique shock waves are generated by ramp wall mounted at the bottom wall of the test section and interact with the air-fuel mixture near upper wall and then reflected by upper wall. The angles of shock wave generated by each ramp both in reacting case and non-reacting cases are listed in **Table 4**. As seen in the table, the angle of shock wave generated by the ramp slightly deviates from its theoretical value. This change of shock angle causes the shift of the impingement point of the oblique shock.

Table 4 Comparison of the theoretical and experimental shock wave angle.

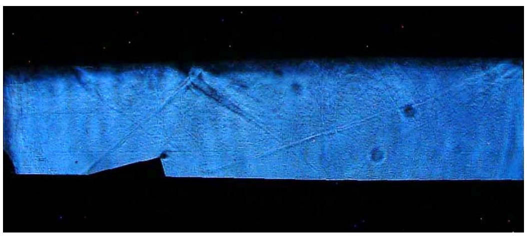
Ramp angle (θ)	Shock wave angle (β)		
	theory	Experiment (hydrogen injection)	Experiment (helium injection)
5°	34.3°	34°	34°
10°	39.3°	37°	38°
15°	45.3°	42°	42°



(a)

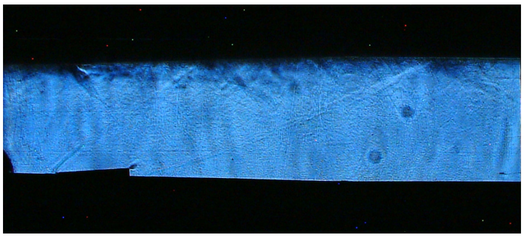


(b)

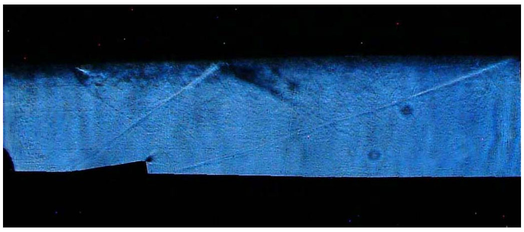


(c)

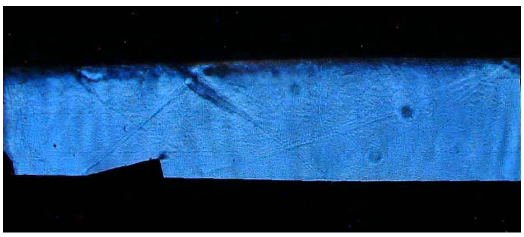
Fig. 6 Schlieren photograph in Non-reacting case (helium injection); (a) Ramp angle 5°; (b) Ramp angle 10°; (c) Ramp angle 15°.



(a)



(b)



(c)

Fig. 7 Schlieren photograph in Reacting case (hydrogen injection); (a) Ramp angle 5°; (b) Ramp angle 10°; (c) Ramp angle 15°.

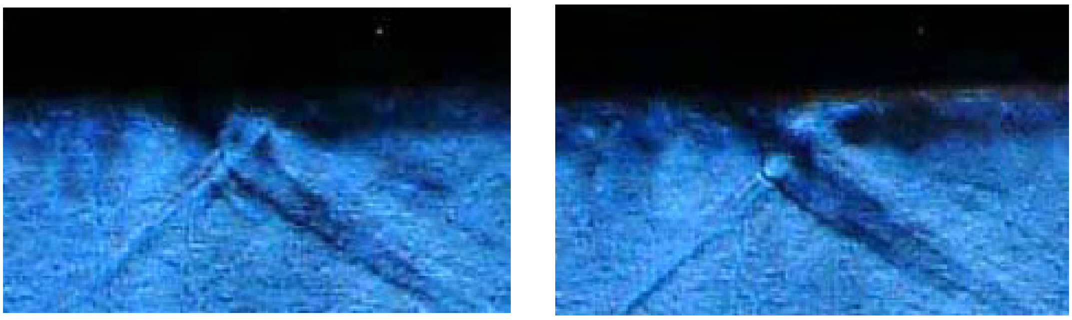


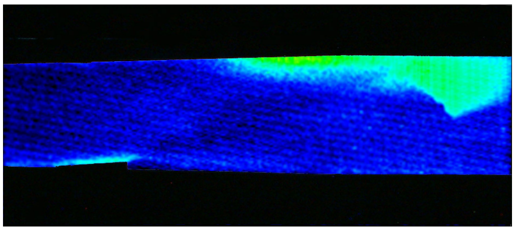
Fig. 8 Enlarged image of shock structure around interference point (Ramp angle 15°); Helium injection (left) and Hydrogen injection (right).

Figure 8 shows the enlarged image of shock structure around interference point in reacting case and non-reacting case of ramp angle 15°. As seen in the figure, incident shock wave interacts with the boundary layer on the upper wall and generates Mach stem and reflected shock wave. Also at the foot of Mach stem the boundary layer separates and the recirculation region is formed. In the reacting case, the combustion reaction produced turbulence more than in the non-reacting case, which can be seen as the difference of image contrasts. Also the pressure increase due to combustion makes the front shock wave stronger.

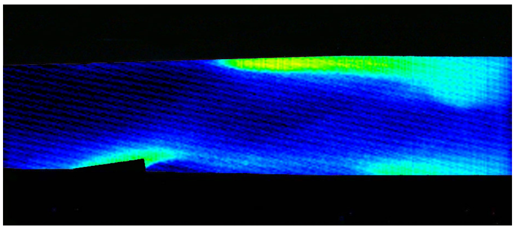
3.2 Effects of shock waves on combustion

In the process of hydrogen combustion, many OH radicals are formed as intermediate products of reaction and playing an important role in the combustion reaction. Furthermore, the OH radical emits the ultraviolet radiation with wavelength of near 310 nm. Thus, the condition of hydrogen combustion can be investigated by observing the presence of OH radicals by catching ultraviolet self-luminescence of wavelength of 310 nm. In the present experiments, UV-CCD camera was used in all cases of ramp angle to record the ultraviolet self-luminescence. **Figure 9** shows the images of ultraviolet self-luminescence in reacting case. When the shock wave becomes stronger, the temperature and pressure of air-fuel mixture increase and the induction time of combustion becomes shorter. As shown by the color of UV images it can be seen clearly that the combustion is also becomes stronger. In the 5° ramp case, as shown in the **Fig. 9 (a)**, the combustion started in about 39 mm downstream from the injector. On the other hand, when ramp angle is 10°, the temperature and pressure becomes higher compared with the case of ramp angle of 5°. Therefore, the combustion is observed just behind the shock wave. Furthermore in the case of ramp angle of 15° the strong combustion can be seen and also combustion zone becomes larger and spreads towards to the center of test section.

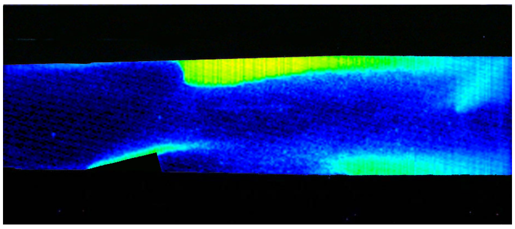
Figure 10 shows the ultraviolet self-luminescence image combined with Schlieren photograph at each ramp angle case. The images reveal that the location of the OH signals observed by UV-CCD camera is in very good agreement with the shock structure. In **Fig. 9 (a)**, for ramp angle of 5°, the combustion starts at 9 mm downstream of the impingement point of the incident shock wave. In this case the mixture of air and fuel is compressed by incident shock wave and the temperature and pressure of mixture becomes sufficiently high for combustion. The reaction becomes weaker downstream and become strong again after the impingement of the second oblique shock wave, which is formed at the bottom wall just after the ramp.



(a)

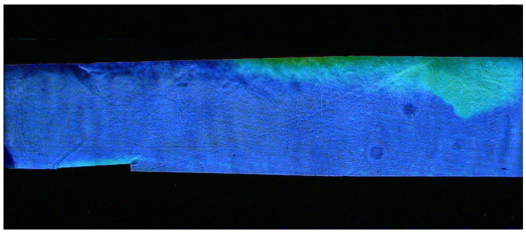


(b)

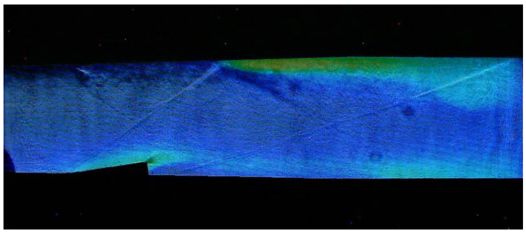


(c)

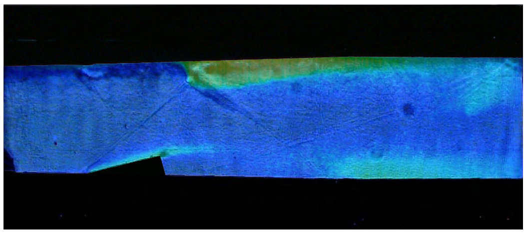
Fig. 9 Ultra violet image of combustion; (a) Ramp angle 5°; (b) Ramp angle 10°; (c) Ramp angle 15°.



(a)



(b)



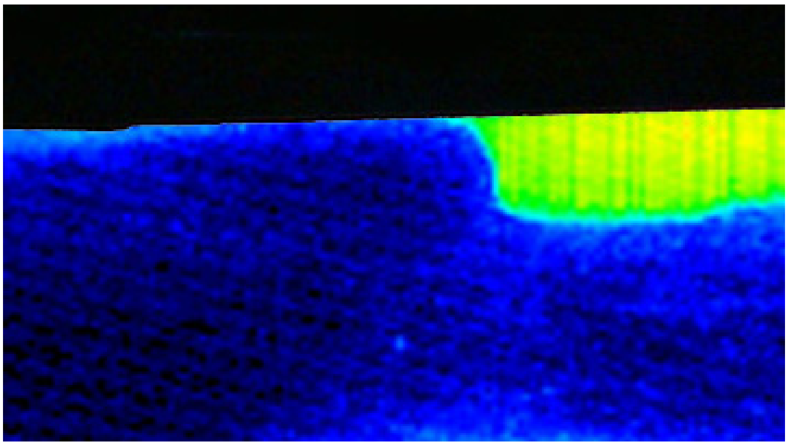
(c)

Fig. 10 Overlaid ultraviolet and schlieren image in reacting case; (a) Ramp angle 5°; (b) Ramp angle 10°; (c) Ramp angle 15°.

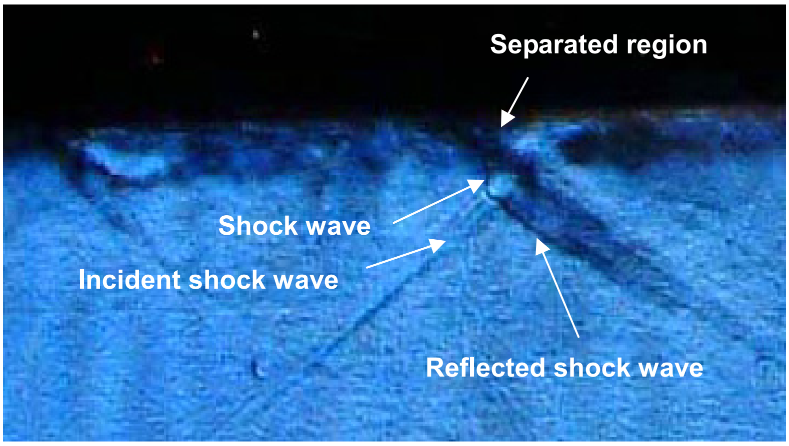
In the case of ramp angle of 10° (**Fig. 9 (b)**) the reaction starts just behind the oblique shock wave. The reaction becomes stronger after the impingement of the second oblique shock wave, which is formed at the bottom wall just after the ramp. In the case of ramp angle of 10° (**Fig. 9 (c)**) very strong combustion occurs just behind the oblique shock wave. In this case, the region of combustion is wider than the other two cases. The incident shock wave interacts with the boundary layer on the upper wall and generates Mach stem and reflected shock wave. Also at the foot of Mach stem the boundary layer separates and the recirculation region is formed. As shown in the Schlieren photograph and UV image, just after the front shock wave hydrogen begins to burn and combustion zone becomes wider.

3.3 Formation of separated region as flame holder

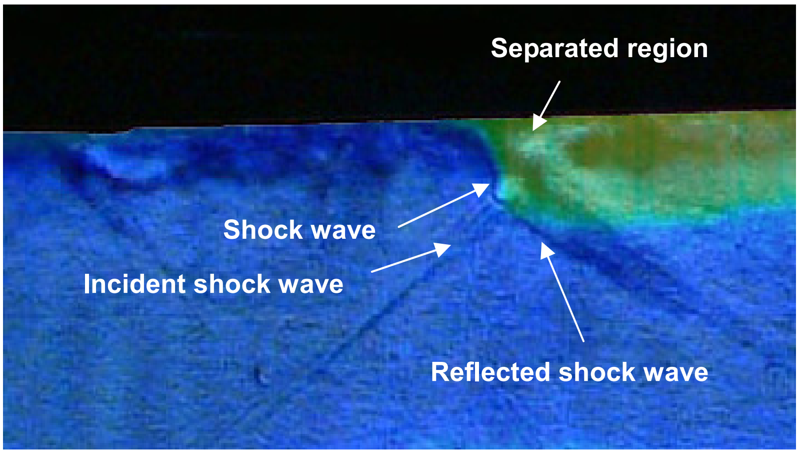
Figure 11 shows the Schlieren picture and ultraviolet image near the injector in case of ramp angle of 15°. The incident shock wave interacts with the boundary layer and forms Mach stem. The interaction between the Mach stem and boundary layer causes the boundary layer separation. The strongest combustion can be seen just behind the Mach stem due to compression effect of the normal shock. The results show that the separated region could work as flame holder in supersonic combustion.



(a)



(b)



(c)

Fig. 11 Structure of shock wave and condition of the combustion near injector; (a) Ultraviolet image of the combustion; (b) Schlieren image; (c) Combined image of ultraviolet and Schlieren image.

4. Conclusions

The study of supersonic combustion of hydrogen has been conducted using reflected type shock tunnel which generated a stable supersonic air flow of Mach number of 2 with the total temperature of 2800 K and the total pressure of 0.35 MPa. The static temperature in the test section is about 1550 K which enables the supersonic combustion with self ignition.

1. The combustion started after the air/fuel mixed gas passed the oblique shock wave generated from the ramp wall. Due to compression effect, the static temperature behind oblique shock wave increases and combustion induction time becomes shorter. The increases of static temperature and static pressure enhance the supersonic combustion. The compression of shock wave becomes large as the angle of shock wave is increased. And the induction time of combustion becomes shorter and the combustion becomes stronger.
2. It can be seen clearly that the area of combustion near upper wall spreads to the center of test section as the strength of shock wave is increased.
3. In the case of ramp angle of 15° Mach stem and reflected shock wave have been observed. The strong combustion can be seen in the separated region formed by interaction between shock wave and boundary layer. This separated region can be used as flame-holder in supersonic combustion.

Acknowledgements

We would like to express our appreciation to Mr. Yuzo Inokuchi of Dept. of Aeronautics and Astronautics, Kyushu University and Professor Kazuhiko Toshimitsu of Ooita National College of Technology, for their valuable supports in the experiments.

References

1) R. W. Guy, R. C. Rogers, et al.; The NASA Langley Scramjet Test Complex, AIAA Paper 96-3243 (1996).
2) Y. Inokuchi, T. Kanyama, et al.; Effects of Combustion Products on the Flowfield in the Vitiated-Type Supersonic Combustion Wind Tunnel, Technology Reports of Kyushu University, Vol. 70, No.3 (1997).
3) G. Smeets, C. Quenett, et al.; Shock tube investigation of H2 combustion in a high temperature supersonic air flow. (Scramjet), Proceeding of IUTAM Symposium on Combustion in Supersonic Flows, Netherlands, pp. 173-178 (1997).
4) A. Ben-Yakar, R. K. Hanson, et al.; Experimental Investigation of Flame-Holding Capability of Hydrogen Transverse Jet in Supersonic Cross-Flow, 27th Symp. Combustion, pp. 2173-2180 (1998).
5) R. R. Boyce, A. Paull, et al.; Compasrison of Supersonic Combustion Between Impulse and Vitiation-Heated Facilities, J. Propulsion and Power, 16, pp. 709-717 (2000).
6) R. Dudebout, J. P. Sislian, et al.; Numerical Simulation of Hypersonic Shock-Induced Combustion Ramjets, J. Propulsion and Power, 14, pp. 869-879 (1998).
7) J. P. Sislian, H. Schirmer, et al.; Propulsive Performance of Hypersonic Oblique Detonation wave and Shock-Induced Combustion Ramjets, J. Propulsion and Power, 17, pp 599-604, (2001).
8) T. Arai, J. Kasahara, et al.; Experiments of Pre-Mixed Shock-Induced Combustion Scramjet

with Forebody-Wall Fuel Injection, AIAA Paper, 2002-5243 (2002).
9) A. Ben-Yakar, M. Kamel, et al.; Experimental Investigation of H2 Transverse Jet Combustion in Hypervelocity Flows, AIAA Paper (1996).
10) H. Huh, J. Driscoll, et al.; Measured Effects of Shock Waves on Supersonic Hydrogen-Air Flames, AIAA Paper 96-3035 (1996).
11) Kim, J.-H., Sim. J. et al.; Mixing Enhancement of Hydrogen Diffusion Flames in Supersonic Air using Shock Waves, AIAA Paper 99-2785 (1999).
12) A. N. Hakim, S. Aso, S. Miyamoto and K. Toshimitsu; An Experimental Study of Supersonic Combustion with Incoming High Temperature Pure Air Stream Obtained by Shock Tunnel, Proc. of 24th. International Symposium on Shock Waves, Beijing, China on July 11-16, 2004.
13) S. Aso, A. N. Hakim, S. Miyamoto, K. Inoue and Y. Tani; Fundamental Study of Supersonic in Pure Air Flow with Use of Shock Tunnel, Acta Astronautica, 57 (2005).