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Self-Ignition and Supersonic Reaction of Pylon-Injected Hydrogen Fuel

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The flame stabilization and reaction processes in a scramjet combustion chamber have been experimentally investigated. Hydrogen was injected into a vitiated Mach 2.15 airstream by means of pylon-like fuel injectors. The supersonic flame was stabilized in a purely kinetical way; i.e., by means of fuel self- ignition. The flame-stabilization mechanisms have been studied. The interaction between the gasdynamics and the reaction kinetics are discussed. A small wedge has been mounted into the test combustor to modify the oblique shock structure. The response of the reacting flow on this change has been observed. To assess the reacting flow, only nonintrusive, optical measurement techniques have been employed: the schlieren technique to visualize the flow structure, the Rayleigh scattering technique to study the injected mixing jets, as well as the self-fluorescence of the OH radical to determine location and intensity of the reaction zones. Additionally, the wall static pressure has been measured.

Nomenclature

$A_{entrance}$ = combustor entrance area
A_{exit} = combustor exit area
M = Mach number
$\dot{m}$ = mass flow rate
p = static pressure
p_w = wall static pressure
p_0 = total pressure
Re = Reynolds number
T = static temperature
$T_{entrance}$ = static air temperature at combustor entrance
$T_{hydrogen}$ = static temperature of hydrogen at fuel-injection orifice
$T_{self\text{-}ignite}$ = fuel self-ignition temperature
T_0 = total temperature
u = velocity
w = mass fraction
x = axial distance from pylon
σ = scattering cross section
$\tau_{self\text{-}ignite}$ = self-ignition time scale
Φ = equivalence ratio

Introduction

CONVENTIONAL burner technology usually uses recirculation areas to stabilize a flame. Local areas are fluid-dynamically created, wherein the flow velocity is in the range of the flame velocity. Mass transport, turbulent as well as diffusive, provides a combustible mixture, and, the heat transport suffices for raising the mixture temperature above its ignition limit. However, relatively large flow disturbances are necessary to create these recirculation zones. In subsonic combustors flameholders are used. Supersonic combustors often employ backward-facing steps or ramps. These flow disturbances generate unacceptably high total pressure losses.

If the high-enthalpy flow is to produce a maximum of thrust in the nozzle, total pressure losses within the supersonic combustor must be minimized.[1] Therefore, a flame stabilization method was employed that does not rely on gasdynamically disadvantageous recirculation zones in the combustor. Instead, the supersonic flame was stabilized in a purely reaction–kinetical way. The reaction of the fuel/air mixture was initiated by self-ignition. The thermal ignition conditions were partly provided by means of oblique shock waves (shock induction). This turned out to be a reliable method for the stabilization of supersonic flames. The interaction of gasdynamics, i.e., oblique shock structure, and chemical kinetics has been investigated. A small wedge (10-deg angle at the front and 15-deg angle at the rear) was mounted into the combustor channel to modify the shock system. The response of the reacting flow was observed.

Experimental Details

The experimental investigations have been performed at the supersonic test facility of the Institute A for Thermodynamics of the Technical University of Munich. A description of the test facility is given in Refs. 2 and 3. The flow conditions of the supersonic airstream at the entrance plane of the test combustor are listed next: $M = 2.15$, $T_0 = 1350$ K, $T = 760$ K, $\dot{m} = 0.33$ kg/s, $p = 1.1$ bar, $u = 1160$ m/s, $Re = 4.6 \times 10^5$, $w_{N_2} = 0.727$, $w_{O_2} = 0.193$, and $w_{H_2O} = 0.08$. It should be noted, that the airstream has been replenished with oxygen before the flow temperature is increased within the preheater by precombustion of hydrogen. The resulting composition of the test air was given in the preceding text.

The test combustor is shown in Fig. 1. The flow channel is of rectangular cross section (27.5×25 mm). The overall length is 645 mm. The lower side of the flow channel consists of individual segments allowing for the variation of the combustor area ratio. The expansion angle of 5 deg of the first segment leads to a combustor area ratio $A_{exit}/A_{entrance} = 1.48$. The increase of channel area accounts for the fuel input and the heat release of the reaction. All experiments were conducted with the same combustor area ratio. The fuel is injected at the beginning of the expansion segment. The small wedge is shown mounted about 26 mm downstream of the fuel injection.

Measurement Techniques

To investigate the combustion processes in the supersonic flow, only nonintrusive, optical measurement techniques have been

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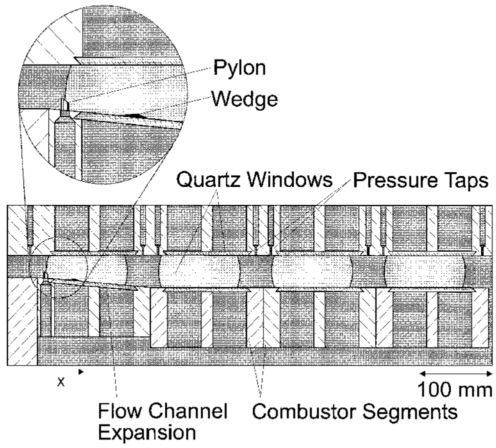

Fig. 1 Supersonic test combustion chamber.

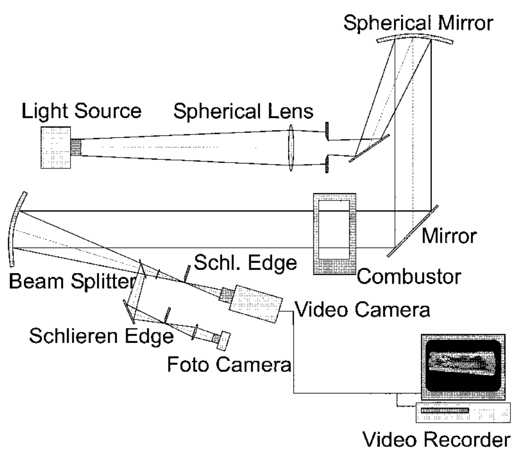

Fig. 2 Optical setup for schlieren measurements.

applied. Only a brief listing of the employed measurement techniques is given here. More detailed information is given in Refs. 2 and 3. For a more general treatment of the optical measurement techniques, the reader is referred to the literature.[4–8]

Schlieren Technique

The oblique shock structure of the supersonic flow has been visualized by means of the schlieren technique. The optical setup is shown in Fig. 2.

During the experiments it was found that in the case of the combustor operating, the sensitivity of the schlieren setup was substantially reduced. This was attributed, on one hand, to the three-dimensional nature of the shock system forming in the reacting flow. On the other hand, due to their temperature gradients during the combustor operating, the windows of the combustor were found to act as an additional lens, adversely affecting the optical path of the collimated light beam. Therefore, to obtain detailed information about the complex flow structure, schlieren images of nitrogen injection into an unpreheated airstream have been taken. By comparison with the images taken from the combustion experiments, it was found that the general flow structure was the same. Thus, the shock structure of the flow, referred to in this paper, was partly obtained by these cold experiments.

Rayleigh Scattering Technique

To examine the supersonic mixing process, the Rayleigh scattering technique has been employed. Helium has been used to simulate the fuel jets. Additionally, to further increase the contrast of the images, helium was injected into an unpreheated airflow.

By forming the exciting laser beam ($t_{\text{pulse}} = 17$ ns) into a light sheet of 25×0.3 mm, spatially and time-resolved images of the two-dimensional distribution of the scattering signal have been recorded. The signal was detected with a 14-bit intensified charged coupled device (ICCD) camera and processed by an image processing unit and a personal computer.

OH-Self-Fluorescence

The OH self-fluorescence has been recorded to determine the location and intensity of the reaction zones.

The (0, 0)-band of the $A^2\Sigma^+$-$X^2\Pi$ system with its bandhead at $\lambda = 306$ nm has been observed by spectrally filtering the flame emission with an interference filter and recording the signal with an ICCD camera. The exposure time was set to 0.1 s, which gives averaged images of the reaction zone distribution.

Wall Static Pressure

To measure the wall static pressure, seven static pressure orifices are distributed along the upper combustor wall. The distribution of the static pressure along the combustor can be regarded as a measure of the heat release into the flow and can be employed to assess the combustion process.

Results and Discussion

Fuel Self-Ignition and Flame Stabilization

In a scramjet propulsion system the air taken up by the inlet is compressed due to the ram effect, thereby raising the air static temperature. If the static temperature suffices for fuel self-ignition, i.e., the static temperature of the fuel/air mixture is above fuel self-ignition temperature ($T_{\text{self-ignite}} = 860$ K) and the self-ignition delay $\tau_{\text{self-ignite}}$ is sufficiently short ($\tau_{\text{self-ignite}}$ of hydrogen is $\sim 10^{-4}$ s for the given temperature range), no additional flameholding devices need to be provided.

These requirements are fulfilled at the given test facility conditions. The resulting flame stabilization mechanisms are displayed in Fig. 3. The combustor equivalence ratio is $\Phi = 0.23$. The air static temperature at the combustor entrance T_{entrance} is about 760 K, and the static pressure level p_{entrance} is 1.2 bar. The measured OH self-fluorescence is shown. The most important oblique shocks are indicated. They have been taken from corresponding schlieren images.

The upper image reveals local reactions of rather modest intensity right after the fuel injection. (Note that the scale of the signal intensity is not the same for all three pictures, but has been adjusted to give the highest contrast in each image.) However, the main part of the fuel only ignites farther downstream, as can be seen in the lower image of Fig. 3. This behavior can be explained as follows.

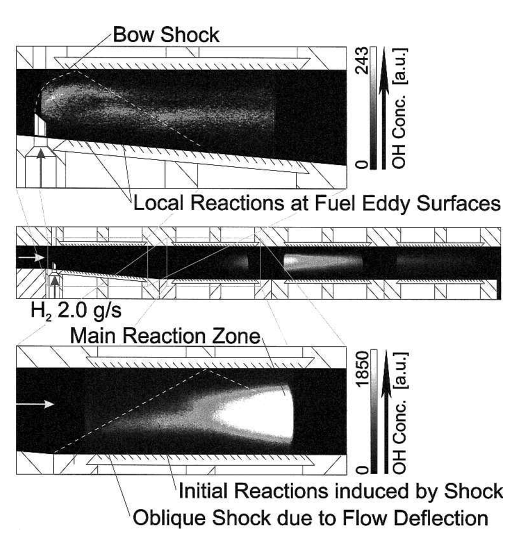

Fig. 3 Flame stabilization mechanisms: reaction intensity in the vicinity of the fuel injection and at the beginning of the main reaction zone (OH self-fluorescence measurements).

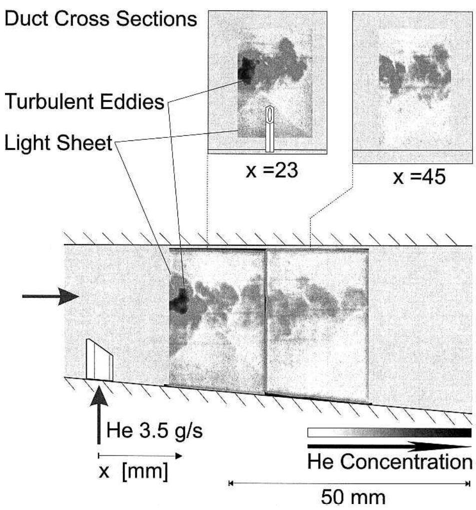

Fig. 4 Turbulent nature of the fuel-injection jet: single-shot, Rayleigh scattering measurements, He injection into cold Mach 2.15 airstream.

The ignition of supersonic turbulent diffusion flames is determined by a series of interacting physical and chemical processes; i.e., two distinct ignition–delay time scales exist. The physical delay is the time required for the mixing of the fuel with the oxidizer to establish a combustible mixture, and for the travel of the flow through oblique shocks raising static temperature and pressure above the self-ignition conditions of the mixture. The chemical delay is the time elapsed from the instant a combustible mixture has been formed until the appearance of a hot flame. It involves the kinetics of the preflame reactions that result in the formation of critical concentrations of intermediate free radicals and atoms. These two different time scales must be kept in mind when discussing the observed flame development.

Injecting hydrogen into the supersonic airflow results in large-scale fuel eddies, which can be seen in Fig. 4, where single-shot, Rayleigh scattering measurements of helium injection into the supersonic airstream are shown.

It takes these turbulent eddies about 100–150 mm mixing length to dissipate. As long as they exist, a large part of the fuel is not in contact with oxygen. Only at the surface of the fuel eddies do oxygen and fuel exist as a combustible mixture. If the local temperature (at the eddy surface) is sufficiently high, self-ignition can start. But the amount of fuel undergoing reaction is very small. Hence, the low self-fluorescence signal intensity in the upper image of Fig. 3.

The majority of the fuel (within the eddies) does not have contact with oxygen. Moreover, the fuel temperature (later on the fuel/air mixture temperature) is still too low for self-ignition and needs to be increased (at least at the given airstream temperature in the combustor entrance, which is below the self-ignition threshold) to compensate for 1) the difference $T_{\text{self-ignite}} - T_{\text{entrance}}$, 2) the temperature decrease due to the flow expansion along the first combustor segment, and 3) the mixture temperature decrease by the injection of cold hydrogen ($T_{\text{hydrogen}} = 245$ K). This temperature rise is produced by the oblique shocks that the mixture is traversing.

The self-ignition of the whole mixing jet is initiated by the strong oblique shock that is created by the flow deflection at the beginning of the second combustor segment (shock-induced combustion); refer to the lower image of Fig. 3. The chemical ignition delay, i.e., the time from the instance when ignition conditions are established by traversing the shock until significant radical concentrations are detectable, is in the range of 10^{-5} s. Of course, this value cannot directly be compared with calculations such as those given in Ref. 9, as the local reactions right behind the fuel injection (refer to the upper image of Fig. 3) may exert a substantial influence on the overall reaction mechanism in that they provide small quantities of free radicals, which accelerate the self-ignition reactions of the main part of fuel.[10,11]

The overall ignition delay, i.e., the time elapsed from fuel injection until the start of the main reaction, is in the range of 10^{-4} s.

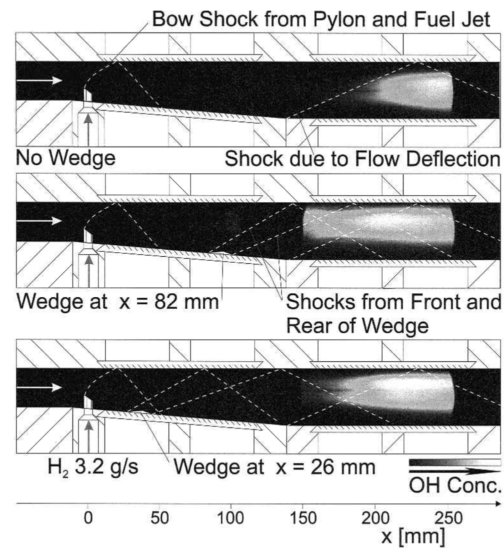

Fig. 5 Influence of additional shock waves from wedge on the flame stabilization: reaction zone distribution (OH self-fluorescence measurements), shock-wave locations taken from schlieren measurements.

In general, it can be seen that the early stages of the ignition process are limited by the physical delay (fuel/air mixing process, rise of the static temperature by shock waves), and the later stages by the chemical delay (the self-ignition delay after traversing the oblique shock at the beginning of the second combustor segment).

Influence of Additional Oblique Shocks on the Fuel Self-Ignition Process

To explore the possibility to actively improve the fuel self-ignition process by oblique shocks, a small wedge has been mounted onto the lower combustor wall (see Fig. 1). The wedge introduces additional shock waves in the flow: one by the front edge, and another by the rear edge. A test series has been conducted injecting a fuel mass flow rate $\dot{m}_{\text{fuel}} = 3.2$ g/s into the airflow ($T_0 = 1350$ K), giving a combustor equivalence ratio $\Phi = 0.23$. Figure 5 shows the measured reaction zone distributions. The indicated oblique shocks are obtained from schlieren measurements.

The upper image in Fig. 5 shows a reference case without the wedge. The pylon together with the injected fuel jet generate a strong bow shock that is reflected at the upper combustor wall. (Of course, the actual bow shock is of three-dimensional shape and, hence, is reflected by the side walls as well. But for the sake of simplicity, only the part of the shock intersected by the channel center plane will be included in the discussion. Moreover, with the employed schlieren method, only this simplified shock structure can be visualized.) The reflected shock impinges on the mixing jet. The other shock included in the image is generated by the flow deflection at the beginning of the second combustor segment. As just described, the main reaction occurs after this oblique shock wave.

The middle image in Fig. 5 shows the case with the wedge mounted at $x = 82$ mm. The main reaction starts after the oblique shock induced by the wedge front. At the beginning of the second segment, the reaction is fully established. This indicates that a combustible mixture exists already about 100 mm after fuel injection. It only needs to be ignited, which is being done here by the oblique shock from the wedge front.

The lower image in Fig. 5 shows the wedge being mounted even farther upstream, at $x = 26$ m. In this case the induction length is not reduced any further. The additional shock waves from the wedge impinge on the mixing jet and may enhance the mixing process.[12–16] However, the mixing has not yet reached a degree that would allow the main fuel to ignite. The mixing continues behind the additional shocks from the wedge; but this time at a higher static temperature level. The main reaction starts as soon as a combustible mixture exists. It can be seen that this is the case well before the shock induced by the second combustor segment.

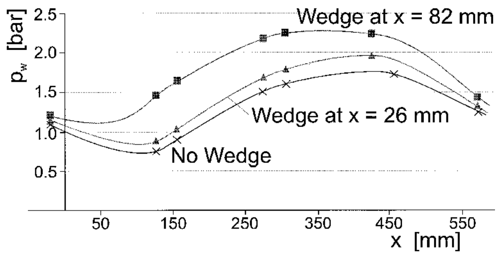

Fig. 6 Influence of additional shock waves from wedge on the flame stabilization: wall static pressure distribution.

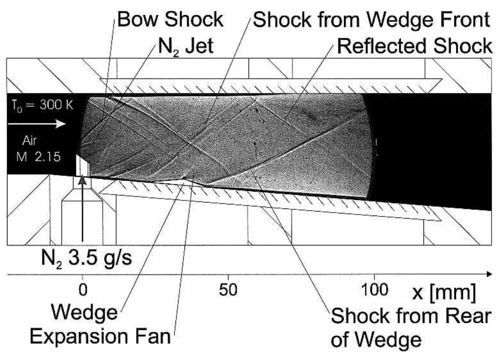

Fig. 7 Flow structure (schlieren image) in the vicinity of the fuel injection with the wedge mounted at $x = 26$ mm.

The observed reaction zone distributions are consistent with the measured wall static pressures. They are given in Fig. 6.

For the wedge mounted at $x = 82$ mm, the pressure in the reacting flow rises earlier and reaches a higher level than that for the reference case. Hence, the heat release takes place earlier and is more intense. This corresponds to the observed reaction zone shown in Fig. 5. For the wedge mounted at $x = 26$ mm, the pressure rise and the final pressure level only slightly exceed that for the reference case. This too is consistent with the reaction zone distribution.

Interaction Gas Dynamics–Chemical Kinetics

To clarify the interaction between the gas dynamics and the chemical kinetics, the flow structure and the OH self-fluorescence in the vicinity of the fuel injection in the lower image of Fig. 5, i.e., the case with the wedge mounted at $x = 26$ mm, is discussed hereafter. Figure 7 shows a schlieren photograph of the region.

To obtain a sharp schlieren image, nitrogen was injected into an unpreheated airstream as explained earlier. Nevertheless, the main features of the flow structure correspond to the combustion experiment. The bow shock from pylon and injected jet as well as its reflection can clearly be seen. The nitrogen jet itself can also be seen. A weaker shock emanates from the pylon base. The wedge introduces two shock waves into the flow. The 10-deg flow deflection at the front edge results in a shock that crosses the reflected bow shock. At the top of the wedge the streamlines are deflected back toward the combustor wall. This creates the Prandtl–Meyer expansion fan, which can also be discerned. At the rear of the wedge another strong shock is produced (flow deflection 15 deg). The image also shows several Mach lines stemming from small boundary-layer disturbances due to unevenness of the combustor walls; e.g., quartz window inserts.

Figure 8 displays the chemical reactions taking place, visualized by recording the OH self-fluorescence. As explained earlier, the fuel eddies have not yet dissipated; chemical reactions can take place at the fuel eddy surfaces only. The amount of fuel participating is small. The recorded signal intensity in this flow region is comparatively low. (The exposure time for the OH self-fluorescence image is 0.1 s, which is too long to time-resolve the flow. Furthermore, the self-fluorescence signal measured is integrated over the channel depth. Therefore, the OH self-fluorescence image is averaged with respect to both time and the channel depth.)

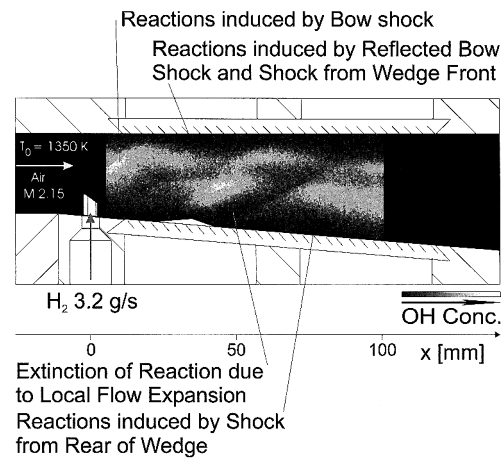

Fig. 8 Local reaction intensity (OH self-fluorescence measurements) in the vicinity of the fuel injection with the wedge mounted at $x = 26$ mm.

For the surface reactions to take place, the local temperature must be sufficiently high. This obviously is the case right behind the bow shock. Another region of sufficient temperature can be seen to exist behind the crossing of the reflected bow shock and the shock from the wedge front. Within the expansion fan above the top of the wedge, the temperature decreases, extinguishing the reaction. Only after the flow temperature is raised again by the shock from the rear of the wedge, does the reaction again appear.

These observations show the close coupling of the gasdynamics and the chemical kinetics in a reacting supersonic flow. In the same way, in which the temperature and pressure variations due to the flow structure influence the local start reactions, they will also affect the local reaction rates within the main combustion zone.

Formation of a Combustion Shock Train in the Reacting Supersonic Flow

It is known that shock waves are created not only if a supersonic flow is turned into itself, but also, if a sufficiently high back-pressure is created downstream of a supersonic flow. In a reacting flow, this back-pressure builds up because of the combustion heat release. The resulting shock-train structure can move upstream of the fuel injection, thereby significantly changing the initial conditions for the combustion.[17–19] The strength of the shock train is determined by the amount of heat release and the effective combustor area ratio $A_{exit}/A_{entrance}$ (Ref. 17).

This combustion shock train was the subject of a number of experiments. The schlieren method was employed to detect and visualize the shock train. The combustion experiments were performed under the following conditions. The airstream static temperature at the combustor entrance was $T_0 = 1380$ K. The combustor equivalence ratios were chosen to be $\Phi = 0.23, 0.33$, and 0.5. It was found that the shock-train structure only appeared for $\Phi = 0.5$. In Fig. 9, schlieren images of the shock train are shown together with the corresponding reaction zone distribution.

For $\Phi = 0.23$ and 0.33, the combustion shock train did not appear. Obviously, for the given combustor area ratio $A_{exit}/A_{entrance} = 1.48$, the heat release was not enough to create a back pressure, which would bring about the shock train.

The schlieren images show the shock train to exist within the reaction zone as well as between fuel injection and reaction zone. With the schlieren measurements no information could be obtained whether or not the shock train extended back into the combustor entrance, as there is no optical access at the combustor entrance. However, this seems unlikely because the measured static pressure just before the fuel injection was the same with and without the combustion, whereas the shock-train structure only existed as long as the combustion took place.

Nevertheless, the fact that the shock-train structure extends upstream of the combustion zone is interestingly enough. Because of the back pressure produced by the heat release within the reaction

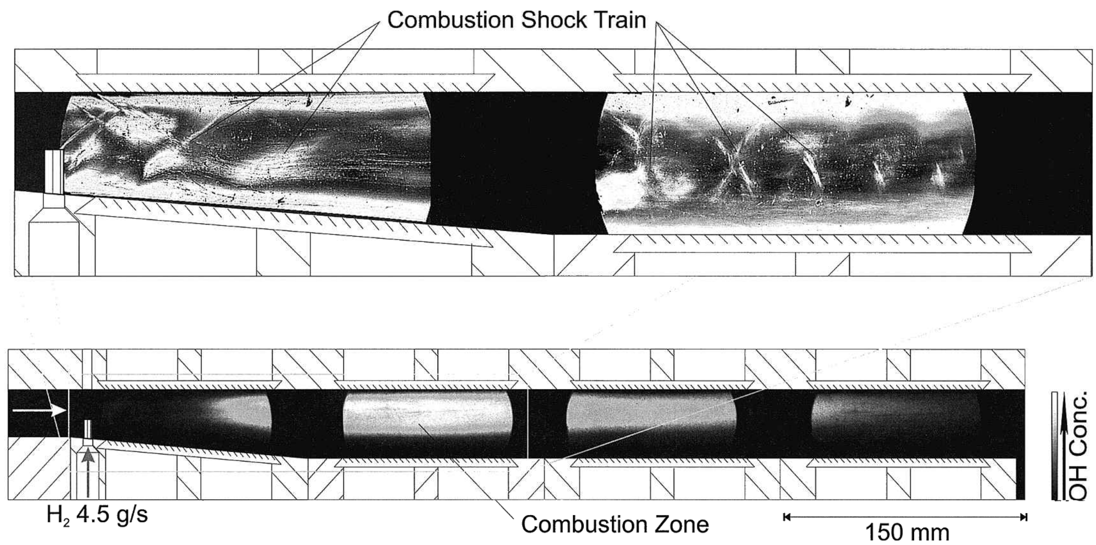

Fig. 9 Combustion shock-train structure (schlieren measurements) and corresponding combustion zone (OH self-fluorescence measurements).

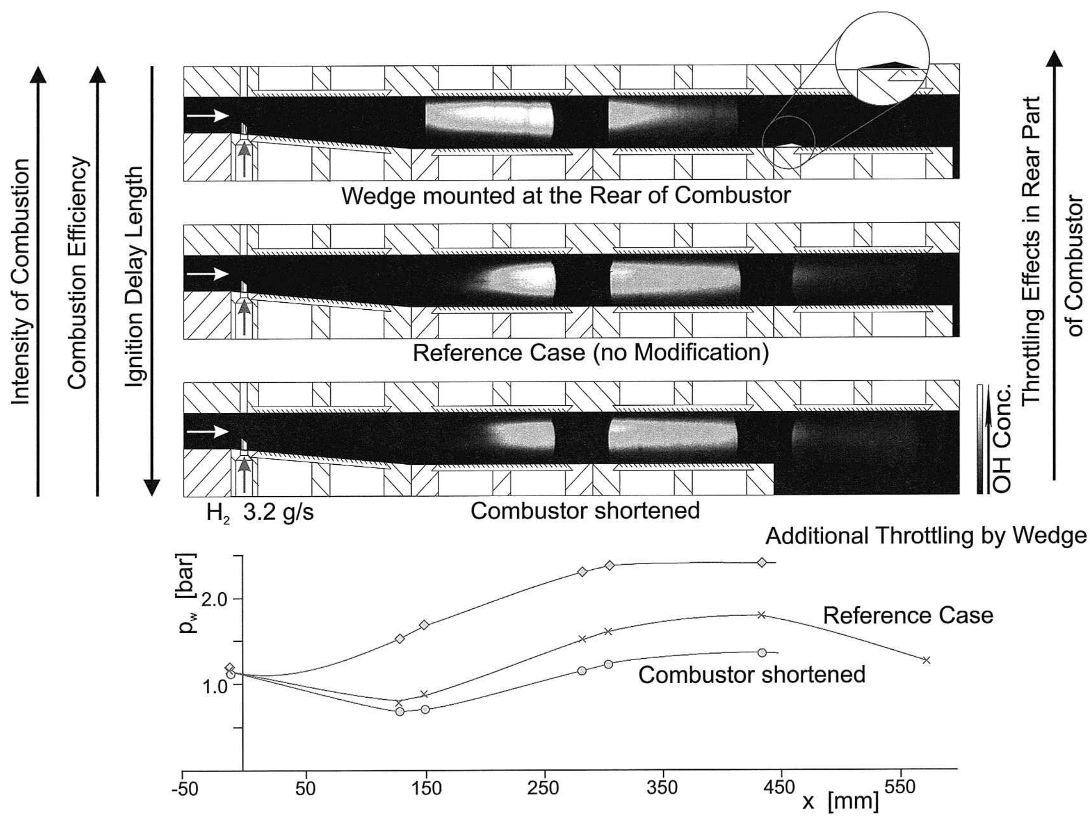

Fig. 10 Response of the reacting flow to changes downstream the combustion zone: combustion zone (OH self-fluorescence measurements) and wall static pressure distributions for three cases (wedge mounted at last combustor segment, reference case, last combustor segment removed).

zone, a shock structure comes into existence, which affects the upstream conditions. This indicates a very complex behavior of the reacting supersonic flow confined in a duct. It would seem that the flow is able to respond to changes not only at the combustor entrance, but also within or at the rear of the combustor. To explore this, another test series has been conducted.

Response of the Reacting Supersonic Flow to Changes Downstream of the Combustion Zone

In this combustion test series the reacting flow was influenced at the rear of the combustor. The response of the flow has been observed.

On the one hand, additional throttling in the rear part of the combustor was realized by mounting the wedge at the beginning of the last combustor segment. The wedge creates oblique shocks that give an additional static pressure rise. This pressure rise could have also been realized by additional heat release in the rear part of the combustor. Therefore, by introducing the wedge, the same throttling effect can be simulated.

On the other hand, the last combustor segment was removed, thereby releasing the flow from its confinement. In this way any heat release in the rear part of the combustor cannot influence the reacting flow within the combustor in any way.

The incoming flow conditions were the same for all cases. The static temperature at the combustor entrance was $T_0 = 1360$ K. The combustor equivalence ratio was $\Phi = 0.33$. In Fig. 10 the resulting supersonic combustion processes are shown. The OH self-fluorescence shows the reaction zones. The wall static pressure distributions indicates the amount of the heat release.

A reference case, i.e., no modification, is shown in the middle image. In the upper case, where additional throttling effects are introduced, the whole reaction zone is moved upstream. The overall induction length is reduced. At the same time the reaction proceeds more intensely (a higher level of OH self-fluorescence signal). The heat release into the supersonic flow starts earlier and is stronger than in the reference case (higher level of wall static pressure). In the lower case, where the combustor has been shortened, the reaction is weakened (lower level of OH self-fluorescence signal and wall static pressure).

This demonstrates that a supersonic reacting flow confined in a duct responds to flow condition changes in the rear part or at the exit of the combustor. In real flight, these changes could be due to changes in the freestream conditions, e.g., flight altitude, or to changes in the nozzle flowfield, e.g., geometry changes. The same behavior can be expected when the fuel mass flow rate, i.e., the amount of heat release into the flow, is changed. Information about changes in conditions in the rear part of the combustor is transmitted upstream so that the reacting flow can adjust itself accordingly. It is felt that the shock-train structure plays a central role in that mechanism. However, with the measurements performed, this cannot be investigated in detail and is therefore left for research efforts in the future.

Conclusions

From the experimental work that was undertaken, the following conclusions can be drawn:

1) If the combustor entrance temperature is sufficiently high, fuel self-ignition is an appropriate means to stabilize the supersonic flame. The fuel self-ignition delay and the reaction rate depend on two factors: a combustible fuel/air mixture must be established and the thermal self-ignition conditions that must be ensured. In the lower flight Mach number range, i.e., at relatively low combustor entrance temperatures, oblique shock waves play an essential role in that they raise the temperature of the fuel/air mixture. Low-intensity reactions at the fuel eddy surfaces may accelerate the self-ignition reactions by providing free radicals and atoms.

2) In the reacting supersonic flow a close coupling between gasdynamics and chemical kinetics exists. It has been shown that temperature and pressure variations due to the oblique shock structure influence the local start reactions at the fuel eddy surfaces. The same influence can be expected with respect to the local reaction rates within the main combustion zone. On the other hand, the combustion heat release affects the flow structure. A shock-train structure may be created that reaches back upstream of the combustion zones.

3) A reacting supersonic flow that is confined in a duct not only responds to upstream parameter changes, but also to flow condition changes that take place at the rear or even at the exit of the combustor. Information is transmitted upstream so that the reacting flow can adjust itself accordingly.

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