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SUPERSONIC COMBUSTION AND GASDYNAMICS OF SCRAMJET

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Abstract

The mechanisms of the supersonic combustion are investigated in the following types of scramjet combustors: axisymmetric combustor with preliminary gasification of liquid kerosene in a gas generator; two - dimensional hydrogen - fuelled combustor; two - dimensional stepwise combustor on liquid kerosene. The gas generator parameters at which the maximum combustion efficiency takes place are determined. The liquid kerosene ignition and combustion stabilization system with the naturally disappearing obstacles at the exit of the chamber are investigated and explored. The new hydrogen supply technique effective from the point of view of organization of mixing is proposed and implemented; it relies upon slot struts. The results of the numerical modeling of a set of supersonic hydrogen combustion problems are presented. By means of boundary layer equations and parabolized Navier-Stokes equations the combustion of coflowing and tangential near-wall jets of hydrogen is considered. The reasons of reduction of heat release in the diverging channels are identified, the dependence of the combustion efficiency and the flame length on the flight Mach number and the oxidizer-to-fuel equivalence ratio in the scramjet combustor is established, and the analysis of the effect of the near-wall hydrogen injection on the friction drag and the heat flux is carried out.

1. Introduction

The problem on combustion of gaseous and liquid fuel in supersonic flows is of both fundamental and practical significance. Among the practical motivations of the attempted efforts on this problem the most important one is in the development of the hypersonic scramjet which differs from the commonly used ramjet in that the flow in the combustor has a supersonic speed. As to the performance characteristics the scramjet engine exceeds the ramjet when $M_\infty \geq 6$, therefore the scramjet is assumed to be included as component of the combined propulsion systems of advanced hypersonic vehicles. For acceleration of a hypersonic vehicle up to speeds at which the scramjet begins to operate, the accelerator engine is required, for example the turbojet or the liquid rocket engine. Due to a relatively low maximum flight speed with a turbojet, or a low specific impulse of a liquid rocket engine, it is expedient to ignite the scramjet at $M_\infty \geq 3$. Within the existing views the scramjet beginning to operate at such low M_∞ values must be a combined cycle engine, i.e. with subsonic combustion, when $M_\infty \leq 6$, and with supersonic combustion at higher Mach numbers. Development and design of such an engine require solution of numerous problems related to proper organization of the process in the combustor. The present paper reviews the results of the experimental and computational investigations performed by authors in this direction during last several years.

2. Experimental tests of scramjet combustor models

The combustor is one of key components of the scramjet. The process of supersonic combustion and heat release in a combustor are governed by a set of factors such as turbulent and molecular mixing, chemical kinetics, gasdynamics, etc. At certain conditions, each one of these factors can become critical for the combustion process. The investigations of the scramjet combustor have been conducted on a direct-connect test bed with a burner preheating of the air. As fuel, gaseous hydrogen and the liquid hydrocarbon fuel (T-1 kerosene) have been used. Considerable attention has been paid to searching for the ways to improve performance of the scramjet combustor at lower flight speeds ($M_\infty \simeq 5$) when the air temperature at the combustor inlet is below the fuel mixture ignition temperature (as is known, this problem is especially challenging for the liquid hydrocarbon fuel) and the Mach number in the combustor is comparatively low. Both factors influence greatly the combustor operation. The basic problem here consists in the existence of a heat addition threshold at which the thermal choking of the combustor occurs that limits the thrust of the engine and becomes a challenge in organization of scramjet operation at flight Mach numbers M_∞ within the range of 3 to 6.

To provide serviceability of the propulsion system (and obtain the required thrust) in the given range of M_∞, it is necessary - either to use two separate flow paths, in one of which subsonic combustion is implemented (the ramjet engine) at low Mach numbers, and in the other, supersonic combustion (the scramjet engine), or – to pass to a dual-mode scramjet, where both subsonic and supersonic combustion is organized in a common duct[1].

Axisymmetric combustor of a dual-mode scramjet fulled by gasified liquid hydrocarbon

The present combustion chamber is a two-duct combustor that corresponds to a possible scheme of organization of process in a combined cycle scramjet[2]. The combustor is an axisymmetric, 148 mm dia. duct with a constant section area F=const where an additional single-pass gas generator is installed coaxially; it is designed to prepare a high-temperature chemically active vapor-gas mixture to be injected into the main combustor flow path. The gas generator duct has a single-staged inlet with an entry diameter d=42mm; the cone semi-angle of the center body is θ_k=20^o and the relative area of the throat f =0.35; the inlet is designed for Mach number M =4.2. The gas generator duct is of 550mm length and 40mm diameter. On the end face of the inlet central body, at the entry of a gas generator chamber, a swirl injector is installed to additionally supply the fuel and preheat the gas ahead of the vaporization section. The primary fuel supply to the gas generator duct is performed from the walls through an annular manifold with 20 fuel orifices 0.5mm in diameter, placed at 290mm downstream of the entry of the main gas generator chamber. The gas is blown out of the gas generator chamber into the main combustor through 6 longitudinal slots of 50x4mm^2 size, located on the side surface of the gas generator shank, as well as through a face-plate nozzle whose size, d_k, may be varied between 5 and 25mm. The change of d_k makes it possible to vary both the flow velocity and the time of fuel stay in the gas generator. The length L_{cc} of the main combustor from the generator gas blowing-out site to the exit section may be changed over the range of 100 to 980mm. The non-uniform supersonic flow of the heated air ahead of the entry into the two flow paths was created by means of the conical nozzle designed for the mean Mach number M = 4.2. At the entry of the main combustor the Mach number is M_{cc} = 2.7. The output section of combustor is connected with the 1.0m divergent channel (the divergence angle is 2^o) which exhausts into the atmosphere. The relative flow rate of the air through the gas generator was 0.09. Tests were carried out at the following parameters of the gas in the heater: P_{to} ≤ 4.2MPa, T_{to} = 1150 – 1350K. These parameters correspond to the flight Mach number of 4.8 to 5.3.

Tests showed that ignition and combustion in the combustor occurs only if thermal destruction of kerosene in the gas generator is provided. An efficient process of thermal destruction holds at a certain ratio of the input and output cross-sections and at the total temperature of the air inflow T_{to} ≥1250K. It is found that the best conditions for the process in the main combustor are provided at the oxidizer-to-fuel ratio in the generator over the range α_{gg}=0.3-0.4. Fig.1 shows longitudinal distributions of the static pressure for some tests carried out. From Fig.1 it can be seen that reliable stabilization and high-intensity combustion are implemented, with formation of a developed pseudoshock upstream of the generator gas supply site and its propagation up to the entry of the duct. In this case, combustion takes place downstream of the pseudo-shock, in a subsonic flow; this is evidenced by with pressure istribution with negative longitudinal gradient at the sector ofcombustor with F = const. The pressure level at the end of the duct F = const is P/P_{to} = 0.042, that corresponds to the flow with sonic speed. In the divergent duct the pressure drop observed is typical of an accelerating supersonic flow. The tests showed also that, under the present conditions in the combustor F=cons, it is possible to provide subsonic combustion only. Supersonic combustion for acceptable values of α_{gg} may occur only in the divergent duct. The combustion efficiency η at the exit section of combustor F = const of length L_{cc} = 370mm is η = 0.74 (calculated according to the one-dimensional theory without regard for the wall heat losses). Increasing the combustor length up to L_{cc} = 650mm with thermal chocking (α=4.1-5.3) may increase the combustion efficiency up to 0.95. Along with the pure gas generator based concept for supplying fuel the tests were conducted with the supply of most liquid kerosene (about 80%) immediately into the basic duct of the combustor through the annular manifold installed 300mm upstream of the gas generator supply pipe-line. It has been possible to reach the high combustion efficiency (η > 0.9) in these tests but over essentially greater lengths, L_{cc} ≥ 1.0m, at the values of total oxidizer-to-fuel ratio similar to those considered above, and T_{to} ≥ 1200K. The latter becomes quite clear if we take into account that the generator acts only as the pilot device in this case and does not participate in the preparation of most combustible mixture.

Two-dimensional combustor of scramjet on liquid hydrocarbon fuel

The present combustor is the duct with the rectangular cross-section. It is composed of several sequentially arranged compartments: the 150 mm long compartment 1 with the constant cross-section (H x B =30x100mm), the 150mm long compartment 2 with the divergence angle of the upper wall of 3 degrees, the 300 mm long compartment 3 with the 2.3^{o} divergence angle of the upper wall, and the 148 mm long compartment 4 with the constant cross-section of 50x100mm size. The exit section of the compartment 4 is connected with a 500 mm long exhaust diffuser; its exit size is 150 by 100 mm. The input section of the combustor is connected with a two-dimensional, 30 x 100 mm, aerodynamic nozzle designed for the Mach number M=2.5. The combustor cross-section is rectangular, with rounded corners. Kerosene was supplied normal to the flow through 5 struts (of 1.2 mm thickness) spaced uniformly in one row across the duct, 68 mm downstream of the entry into the combustor. The diameter of orifices of jet injectors on struts is equal to 0.34mm or 0.3mm at the total number of 18 and 26, respectively. Saturation of kerosene with the air is envisaged in the supply system to improve fuel atomization, mixing and combustion. The flowrate of the air for saturation was 0.02 of the fuel consumption. Forced ignition of kerosene was provided by choking the flow by means of air jets (their mass flow was 0.2 of the combustor flow rate) supplied at the end of the combustor. Flow conditions at the entry of the combustor and at the heater of the wind tunnel are: M_{cc} = 2.5, T_{to} = 1300 - 1700K, P_{to} = 2.6MPa.

In the tests carried out, ignition of the kerosene saturated with the air occured at T_{to} > 1600K and the oxidizer-to-fuel ratio α = 1.5 - 2.3. The typical distributions of the static pressure along the duct for the case α = 2.3 are presented in Fig. 2. The injection of air jets for choking the chamber started before supplying kerosene and stopped immediately after igniting the fuel. It can be seen from Fig.2 that supplying the kerosene simultaneously with the pneumatic ignition system operation results in subsonic flow with combustion in the pseudoshock. The pressure rise propagates upstream up to the nozzle where the supersonic flow of the air is formed. With choking terminated, combustion is maintained throughout the chamber including the diverging part. The pressure at the end of the chamber is P/P_{to} = 0.15 that corresponds to the subsonic combustion. Impossibility of the kerosene ignition in the combustor at T_{to}<1600K in the tests carried out seems to be caused by insufficient choking of the flow by means of the air injection.

Two-dimensional combustorr of hydrogen - fuelled scramjet

The combustor consists of two compartments . The first one is the square (100x100mm) duct of 700mm length. The cross section area of the second compartment grows due to deflection of the upper wall at the angle of 2 degrees, the compartment length is also 700mm ; the cross-sectional area ratio is 1.24. The diverging portion of the combustor comes to the 400 mm long exhaust diffuser. The combustor is large-size and the problem of organizing the effective mixing of the fuel with the air flow is acute here. The gaseous hydrogen was supplied into the combustor from the upper wall through two specially developed slot pylons with internal flow. The slot pylon is comprised of two panels with sharp leading edges installed along the flow (parallel to each other) at the distance of 8mm, the length of the pylon is 160mm, the height is 50mm. The slot portion of 80mm length is closed on the top by a cap. Hydrogen is supplied into the slot between the panels from the wall of the combustor through the 5mm dia. orifice placed at the distance of 20mm from the leading edges of panels. The distance between two pylons is equal to 30mm, the blockage ratio of the chamber is 0.08. Configuration of the pylon is given in Fig. 3. Test conditions are: Mach number at the entry of the combustor M_{cc}=2.3; α = 3.15-4.9; heater parameters, P_{to} = 4.0-4.2MPa; T_{to} > 1200K.

The tests showed that blowing-out the hydrogen jet into the slot passage between the panels makes it possible to provide a greater penetration height in the supersonic flow, more uniform distribution of the fuel across the combustor, better mixing with the air, and reliable flame holding in comparison with the case of hydrogen supply in the free stream. As an example the longitudinal distribution of static pressures for the case of struts installed in a divergent compartment at the distance of 170mm from the entry is represented in Fig. 3. The pressure distribution along the slot passage of the strut is also shown in this figure. It can be seen that at T_{to}=1200K and the variation of α over the range of 3.15-4.2 the ignition occurs and a stable combustion of hydrogen is established in the divergent channel. At α = 3.15 both the level and shape of the pressure profile indicate both the thermal choking of the flow in the divergent channel and formation of a weak pseudoshock upstream of the fuel supply location. The pressure rise in this case propagates 500mm upstream of the hydrogen supply location. At $\alpha = 4.2$ the pure supersonic combustion without the disturbance propagation upstream takes place in the divergent channel. The combustion efficiency determined by the one-dimensional theory (without allowance for heat losses into the wall) is $\eta = 0.75-0.9$; in this case, smaller values of η correspond to supersonic combustion . To implement higher heat release than those in the present tests (i.e. to implement combustion at smaller values of α), it is necessary to increase the combustor divergence ratio.

__Two-dimensional stepwise combustor of dual mode scramjet engine on liquid hydrocarbon fuel__

As one of alternative schemes of the combustor for a dual mode scramjet engine the combustor with stepwise expansion is considered at present. Such a combustor is expected, first, to provide essential reduction of the combustible mixture ignition temperature due to the stabilizing effect of the recirculation zone and, secondly, to fix the combustion zone on the step, thus preventing propagation of combustion disturbances to the engine inlet. The stepwise combustor has a number of other favorable features in terms of its use over the wide range of operation/mission conditions.

The flow passage of the combustor involved comprises two sections in series. The first section is a slightly divergent duct (the divergence angle is 30') with the entry cross-section size $h \times b = 30\times100$ mm. The cross-section size at the second location is 58x100 mm. The length of the forward portion is 790 mm, and that of the second one, 510 mm or 910 mm. The combustor cross-section is rectangular with rounded corners. The supersonic flow at the combustor entry is produced by the planar aerodynamic nozzle designed for the Mach number 2.5, with the cross-section of 30x100 mm. The combustion products are discharged into the atmosphere through an exhaust diffuser. The test conditions were: $M_{cc} = 2.5$, $P_{to} = 2.5 - 3$ MPa, $T_{to} = 1000 - 1400$K. The kerosene is injected by three diamond-shaped pylons which could be installed at different intervals from the combustor entry: 90 mm, 180 mm or 690 mm. Each pylon had six injector ports of 0.4 mm diameter, with axes being normal to lateral pylon surfaces. When testing, kerosene was preliminary saturated with the air. The fuel mixture was ignited at $T_{to} \geq 1200$K by choking the flow with 2 mm thick aluminium panels installed at the combustor exit (on both upper and lower walls). In this case, the blockage of the duct was 40%. After the fuel ignition the panels burn away, and, if $\alpha \leq 2$, in the duct there occurs a stable subsonic combustion accompanied by thermal choking at the exit of the combustor (in the fore duct there occurs a pseudoshock occupying the area of 300 to 450 mm length), whereas at any $\alpha \geq 2$, the flame is out. The tests showed that propagation of pressure disturbances upstream of the combustor can be avoided, if the fuel delivery pylons are placed just near the step (at the distance of 690 mm from the combustor entry). The abovesaid is illustrated in Fig.4 for $\alpha = 1.26$ at $T_{to} = 1330$K. In this case, the combustion efficiency determined by a one-dimensional theory, is $\eta \geq 0.9$. With the combustor ignited, stable combustion is maintained at reduction of the total temperature to $T_{to} \simeq 1000$K and $T_{to} \simeq 1200$K at the length of the combustor second portion of 510 mm and 910 mm, respectively, for α of 1.4 to 1.5. The presaturation of kerosene to be injected into the combustor does not result in notable change of the minimum flow temperature at which ignition/quenching of flame would occur in the combustor.

__3. The computational analysis of supersonic combustion and scramjet gasdynamics__

The computational fluid dynamics is an effective means of investigating complex physicochemical/gasdynamic processes in a scramjet . The current successful and perspective applications of numerical methods to the solution of the scramjet problems are considered in[3] . The scramjet combustor computation is mentioned there to be the most challenging one.

The full Navier--Stokes equations are to be solved in strict formulation, which is rather a complicated task for up-to-date computers. Therefore, use would be made of various simplifications of these equations. Most studies were accomplished in the framework of boundary layer approximation and parabolized Navier- Stokes equations (PNS).

This part of the survey presents certain results obtained through such approximations. In[4] , computation was carried out on the basis of PNS for a planar combustor of a scramjet, where gaseous hydrogen is injected tangentially in the form of non-isobaric near-wall jets. To determine the turbulent viscosity, the Cebeci-Smith algebraic model was employed. The hydrogen-air combustion was described by thirteen elementary reactions. When the maximum static temperature exceeds 3000K, nitrogen and its oxides are introduced into the finite-rate chemistry model. The combustor entry airflow parameters corresponding to the vehicle flight at the altitude of 26 km ($M_\infty = 7$) were assumed to be homogeneous over the cross-section area : T_{cc} = 1010K, M_{cc} = 2.63, P_{cc} = 0.174MPa. The hydrogen preheated up to 800K at the pressure P_j = 1.04 MPa (P_j / P_{cc} = 6) was injected at the sonic speed from two symmetrically arranged slots of 0.5 cm height into the duct of 30 cm height (see Fig.5). These flow parameters correspond to $\alpha \cong 3.8$.

From Fig.5 (which shows the pressure profiles along the wall and the duct symmetry plane as obtained with a frozen flow) it follows that injecting underexpanded jets results in formation of a periodic shock wave pattern in the duct. The computation of a flow without combustion has also made it possible to conclude that the combustion length considered (of 3 m) would be insufficient for complete mixing of hydrogen with air.

Preliminary estimates using the one-dimensional theory showed that the combustion under these conditions causes the thermal choking in the combustor . In two-dimensional formulation, ignition begins on the outer jet boundary 12 cm apart from the nozzle exit, and is accompanied by a rapid temperature growth decreasing the flow Mach number. As a result of the interaction of this zone with the shock wave reflected from the symmetry plane and augmented due to the ignition, there appears a comparatively extensive 'suspended' subsonic zone occupying up to 1/4 the section area. With this, further computation in the framework of PNS is not possible. Diverging the duct by 2.3^o downstream of the section x = 50 cm made it possible to attain the supersonic combustion throughout the combustor (Fig.6).

The computation showed that the near-wall hydrogen injection results in a very low combustion efficiency. In the example considered it reaches $\eta \simeq 0.4$ over the length of 3.0m. The effect of the near-wall hydrogen injection on the friction and heat fluxes in both a scramjet combustor and a nozzle has also been analyzed in [4]. Here, consider the results for a nozzle. The computation was performed for the following combustor entry conditions: M_{cc} = 1.5, T_{cc} = 2400 K, P_{cc} = 0.287 MPa, α = 1.1. The external flow composition was assumed to be frozen. The problem was being solved for three versions. In the first two, hydrogen was injected into the boundary layer out of a single slot of height of 1.5mm and 3mm, respectively, located at the nozzle entry. The relative hydrogen flowrate (referenced to the combustion product flowrate) was equal to 2.7×10^{-3} in the first case, and 4.8×10^{-3} in the second one. In the third case, hydrogen was injected out of two sequentially located slots of 1.5 mm height. The second slot was separated from the first one by 84 cm. In all cases cold hydrogen (T_j = 300K) was injected at the sonic speed, the outflow being isobaric. The wall was considered adiabatic as long as its temperature is less than T_w = 1400 K. The distribution of the local friction coefficients C_f for the first case, shown in Fig.7, can serve as an example of the results obtained. It can be seen that the hydrogen injection does result in notable reduction of the friction drag - by 34% over the length of 12 m (with heat flux decreasing by 71% over the same length). A much greater effect can be achieved by injecting hydrogen through the slot of 3mm height : the friction drag is reduced by 42% (and the heat flux eliminates practically at all). The distributed injection (the third case, Fig.7) will be valued as more effective in terms of the hydrogen savings.

In [5], the numerical modelling of supersonic combustion of hydrogen in a coflowing jet was carried out on the basis of PNS. Use has been made of the same combustion kinetics as in[4]. The turbulent viscosity was obtained using the semi-empirical equation taken from[6].

The results of the computation indicate that in the course of the combustion there occur the shock waves formation which cause considerable irregularity of static pressure across the duct. The low-size zones with elevated temperature and increased pressure contribute to the hydrogen ignition. From this, the conclusion should be drawn that the shock wave structure will influence greatly the process of combustion.

In [6], two important problems of supersonic combustion of unmixed gases in ducts were analyzed numerically within the framework of boundary layer approximation. These problems are: 1) drastic reduction of heat release in divergent ducts; 2) dependence of fuel burning process on the flight Mach number of advanced hypersonic vehicle. It is shown that, over the wide range of combustor entry flow conditions, the reduction of heat release in a divergent duct is caused by moderation of turbulent mixing. This can be judged, specifically, from the profile of the turbulent viscosity along the axis of an axisymmetric combustor with coaxial supply of hydrogen (Fig.8). The computation conditions were: M_j = 1.0; M_e = 2.6; T_j = 1000 K; T_e = 1500K, the mass concentration of hydrogen in a jet (diluted by nitrogen) g_{H2} = 0.3; r/R = 0.3; R = 5 cm; α = 4.3; $P_i = P_j = P_e$ = 0.1MPa, the turbulent viscosity at the inout location is $\nu_{ti}/u_e R = 3.5 \cdot 10^{-3}$; the index j refers to the parameters of a jet, the index e refers to the parameters of outer airflow at the inout section.

In [6], the mechanisms of fuel burn-out at the variation of M_∞ and α were investigated, a planar combustor with coflowing hydrogen slot injection from vertical pylons (Fig.9) considered as an example. The height of combustor h = 15cm, the distance between adjacent pylons ℓ = 5 cm, the turbulent viscosity at the jet injection section $\nu_{ti} = 1.5 \times 10^{-3} u_e \cdot h$, the total temperature of hydrogen jets is 1000K. The air parameters at the combustor entry are given in the table in Fig.9. They correspond to the inlet throat conditions of scramjet engine hypersonic vehicle flying the trajectory with the dynamic pressure q_∞ = 75kPa. The results of computing the flame length L_c (the length, throughout which the combustion efficiency is 98% of the maximum) are given in Fig.9. It is clear, that L_c grows substantially as an air-fuel mixture becomes leaner and the Mach number increased. Hence, the problem of effective mixing of fuel components seems to be one of the key problems of developing scramjet .

Conclusion

The major results of the present experimental/analytical investigations into gasdynamics and supersonic combustion in scramjet can be formulated as follows:

1. The gas-generator based concept for injecting/burning kerosene in the model axisymmetric combustor of a dual mode scramjet engine has been implemented and tested in the direct-connect facility with the burner preheating. The gas generator parameters to provide the maximum combustion efficiency have been determined. The effective kerosene combustion (the combustion efficiency $\eta \geq 0.9$) within the operating conditions of M_∞ = 4.8 to 5.3 (T_{to}=1150-1350K) has been accomplished.
2. A new method of gaseous hydrogen injection (through slot pylons) effective in terms of mixing in a large-scale combustor has been suggested and implemented.
3. The system of ignition/flameholding of kerosene by means of naturally disappearing obstacles at the end of a dual mode scramjet engine combustor has been studied and upgraded; the effective kerosene combustion ($\eta \geq$ 0.9) at operation wiht M_∞ = 4 - 5.3 (T_{to} = 1000 - 1400 K) has been carried out.
4. Making use of PNS solution, the near-wall hydrogen injection effect on the friction and heat flux in the combustor/nozzle of a scramjet engine has been analyzed; it is shown that the hydrogen injection can be the effective means of reducing both the friction drag and heat flux.
5. The causes of natable reduction of heat release in the divergent ducts with supersonic flow, as well as the mechanisms of supersonic combustion at various flight Mach numbers, M_∞ , and the oxidizer-to-fuel ratio, α , have been numerically analyzed in the framework of boundary layer approximation. It is shown that: a) over a wide range of combustor entry conditions, the heat release reduction will be caused by moderation of turbulent mixing of reagents; b) the flame length will essentially grow at greater M_∞ and with leaner air-fuel mixtures.

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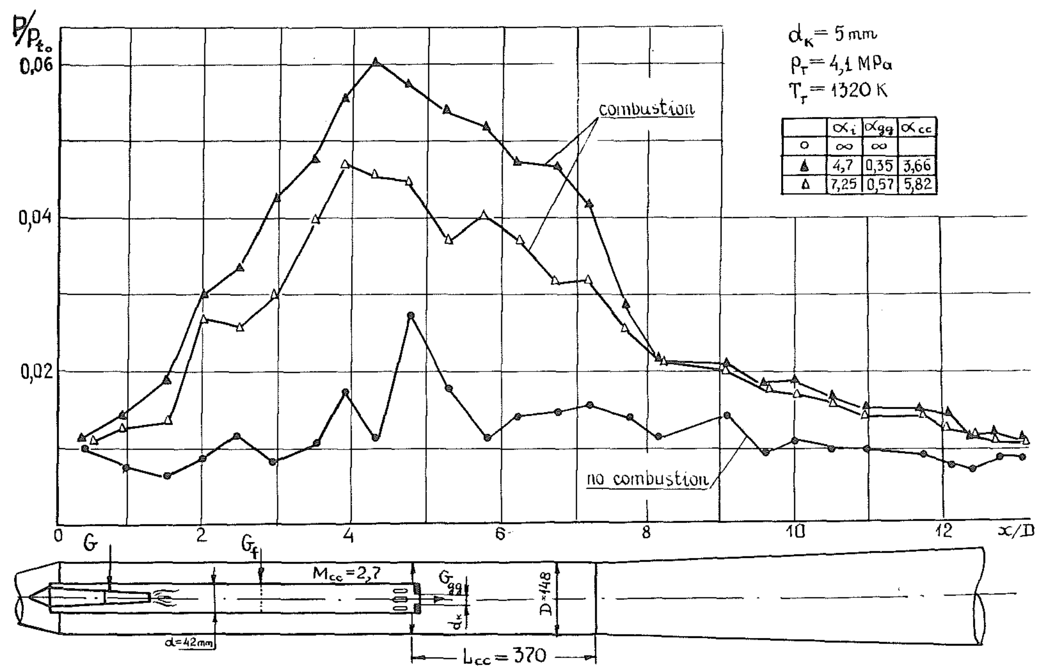


Figure 1

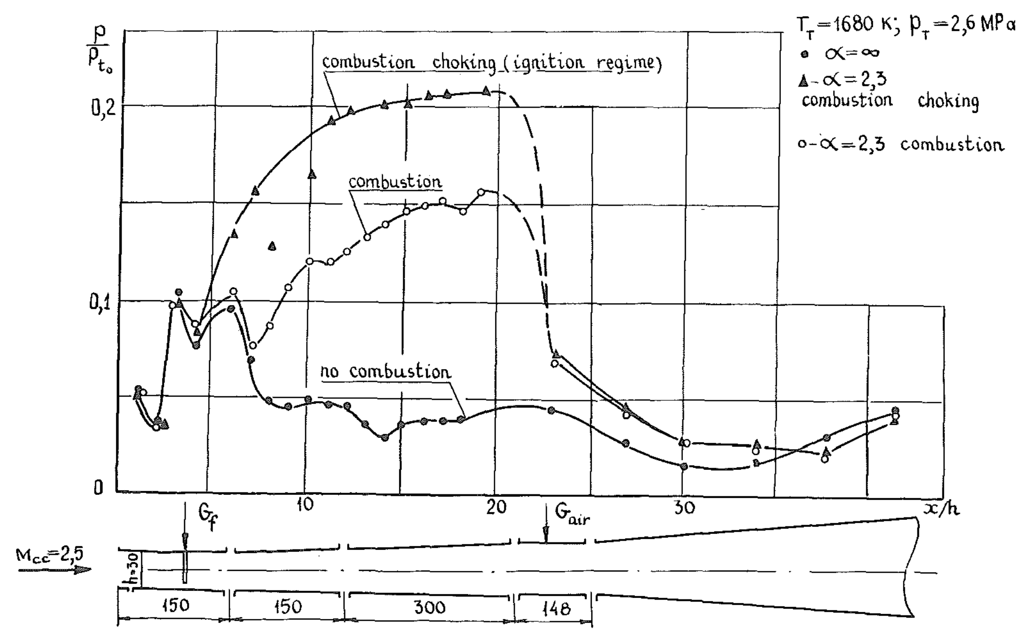


Figure 2

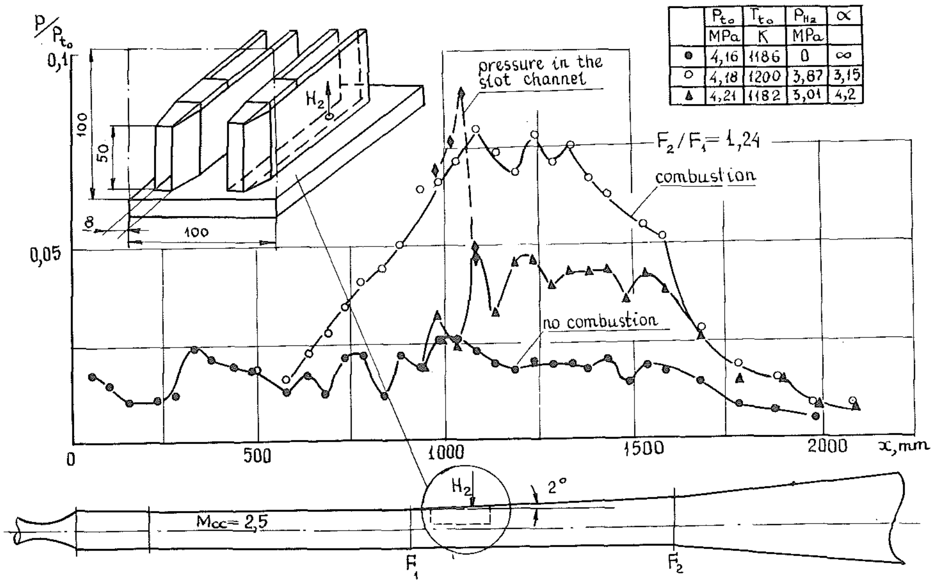
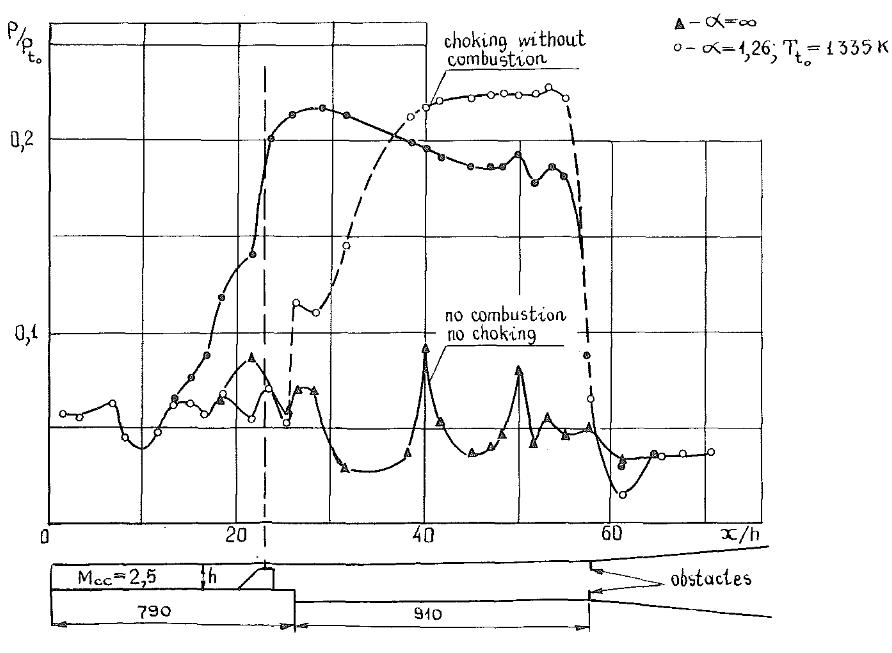


Figure 3



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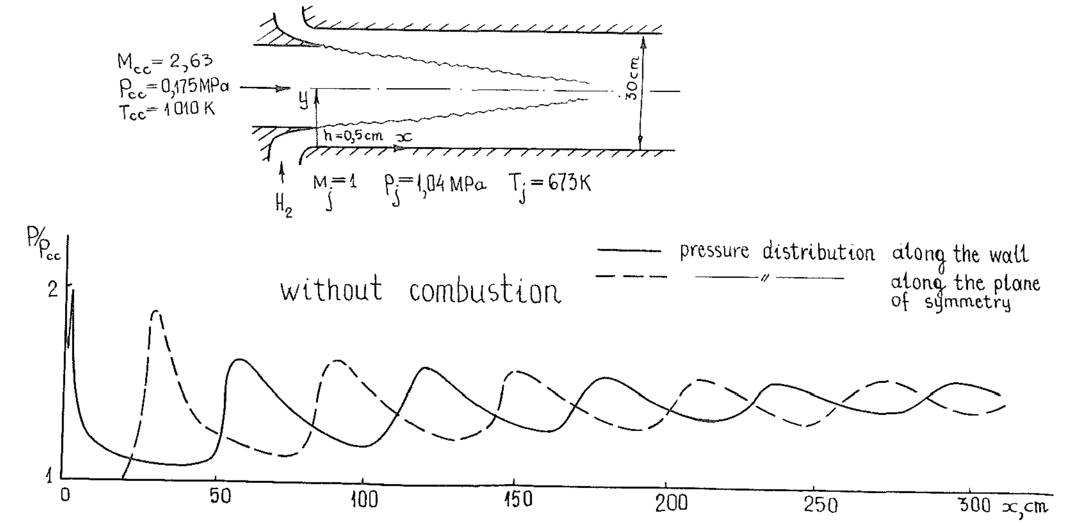


Figure 5

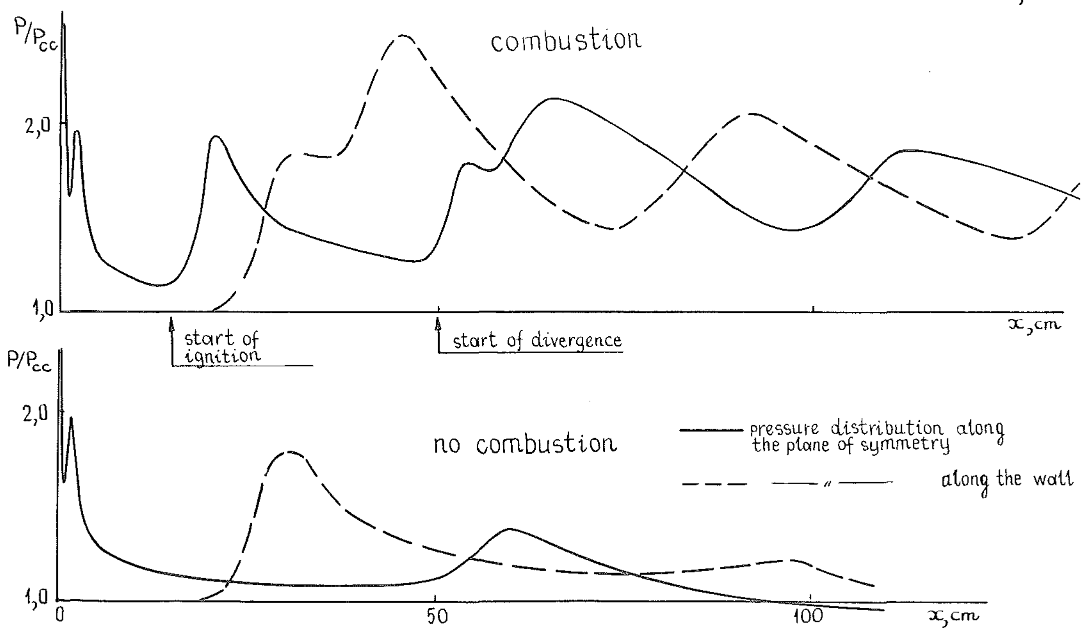
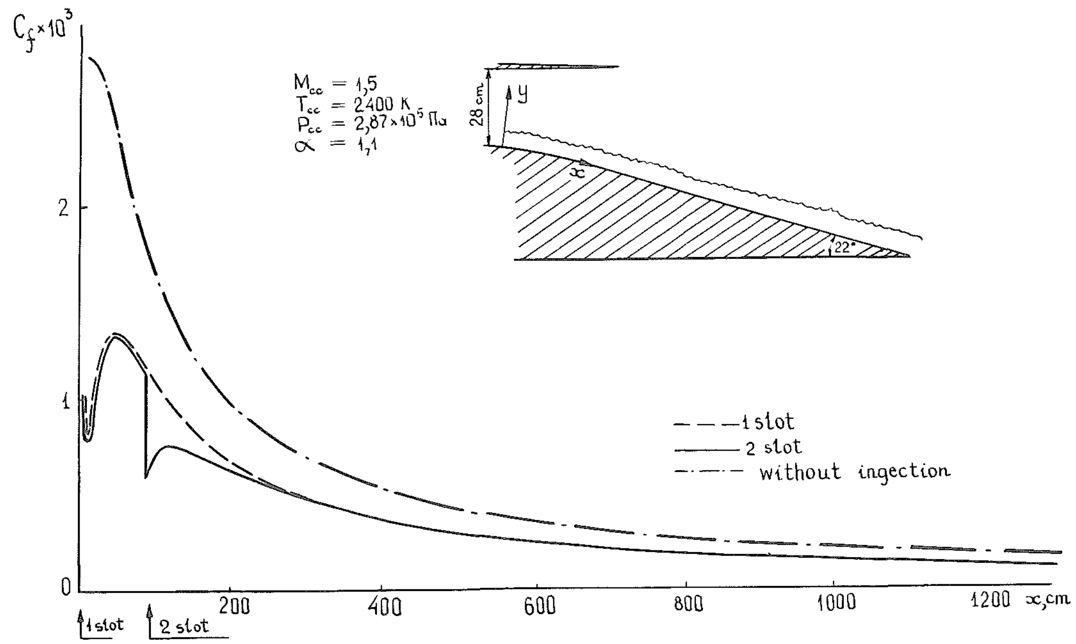


Figure 6



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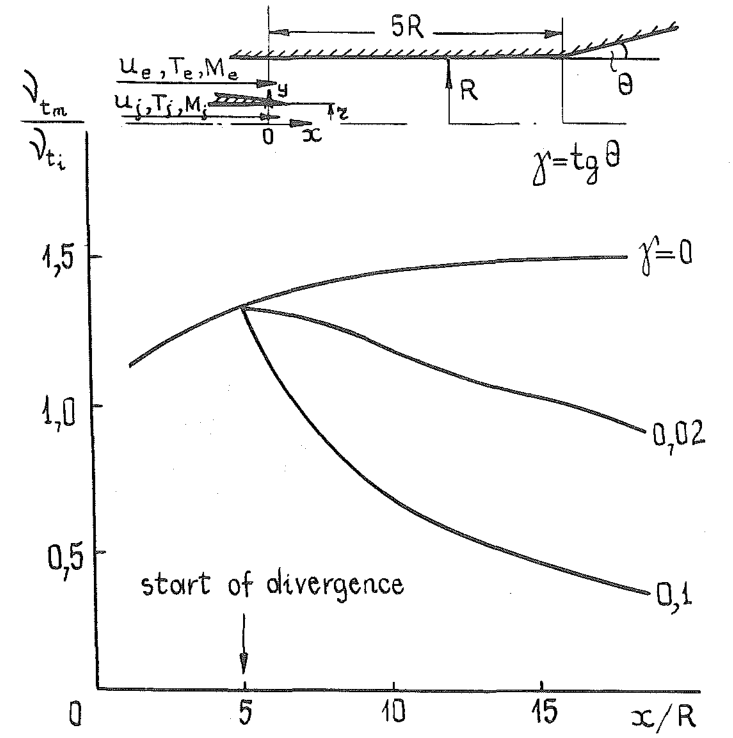
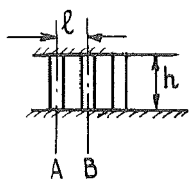


Figure 8



M_∞	10	12	14	16
P_i, MPa	0,126	0,102	0,08	0,065
T_e, K	1250	1490	1775	2020
M_e	3,7	4,3	4,9	5,5

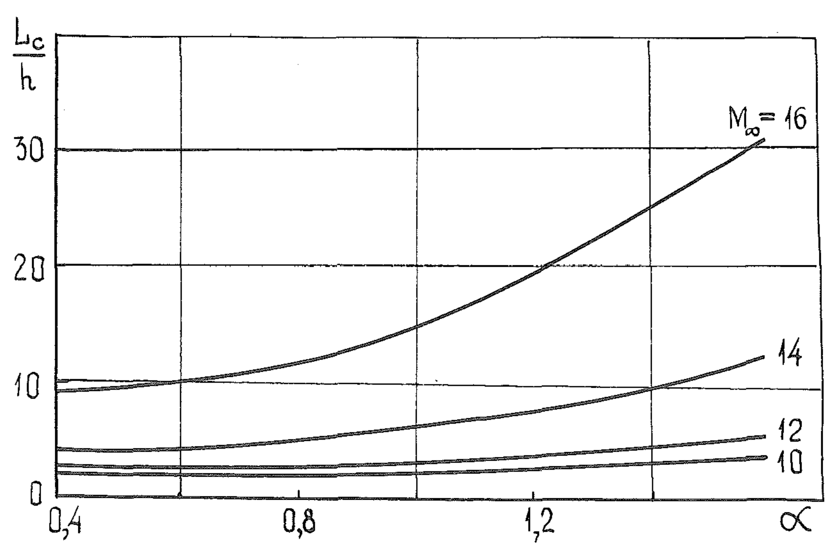


Figure 9