MULTIDIMENSIONAL SIMULATIONS AND TEST FIRES OF A HYDROGEN-FUELED RAMJET WITH AN ANNULAR DETONATIVE COMBUSTOR AT APPROACHING AIR FLOW OF MACH 2 AND 1.5

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The conceptual design of a hydrogen-fueled dual-duct detonation ramjet (DR) of a new type for a cruising flight speed of Mach 2 at sea level is developed using multivariant three-dimensional (3D) numerical simulations of the operation process. The possibility of arranging the continuous-detonation (rotating or longitudinally pulsating) combustion of hydrogen in an expanding annular combustor of the DR is proved for the first time. The calculated effective thrust of such a DR is shown to become positive at $M = 1.3$, i. e., the startup Mach number for such a DR can be lower than $M = 2.0$ which is typical for ramjets operating on continuous-deflagration combustion. A DR demonstrator is designed and manufactured. Its test fires are performed in a pulsed wind tunnel (WT) at free air jet Mach num-

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bers M = 2.0 and 1.5. The most important result of test fires is the experimental proof of the possibility of arranging stable continuous-detonation combustion of hydrogen in the DR of the developed design at both Mach numbers.

1 Introduction

Existing ramjets operating on a thermodynamic cycle with continuous-deflagration combustion of fuel at constant pressure are effective in the range of aircraft flight Mach numbers ranging from approximately 2 to 6. One of the most important characteristics of a ramjet-powered aircraft is the minimum Mach number at which the autonomous flight of aircraft is possible after the initial acceleration provided by various auxiliary boosting means. This flight Mach number is often called the startup Mach number. For existing ramjet-powered aircraft, the startup Mach number is approximately equal to 2. At a lower flight speed, the stagnation pressure of the approaching air stream is too low to ensure stable operation of the air intake and the engine thrust is insufficient for the autonomous flight of the aircraft. To accelerate aircraft to such a speed, auxiliary boosters, e. g., rocket engines, are usually used. The rocket engines have relatively low specific thrust performances which leads to an increase in the starting mass and dimensions of the aircraft. For example, according to the well-known Tsiolkovsky formula, the mass of the booster fuel required for accelerating an aircraft to Mach 1 with a rocket engine is about 10%–20% of the mass of the entire aircraft. To accelerate the aircraft to Mach 2, the mass of the booster fuel increases to about 20%–40%. It is seen that for reducing the starting mass of the aircraft, the startup Mach number must be decreased.

The thrust performances of ramjets have reached their limits and their further increase is problematic. Therefore, for obtaining a significant increase in the ramjet efficiency, the possibility of using a thermodynamic cycle with continuous-detonation (rotating or longitudinally pulsating) rather than continuous-deflagration combustion of fuel is currently considered. Such a cycle is known to have a higher efficiency [1, 2]. In addition, as compared with continuous-deflagration combustion, continuous-detonation combustion of fuel has several advantages. Thus, with continuous-deflagration combustion, the chemical energy of fuel is released over the entire cross section of a combustor creating a backpressure to the incoming air flow. This makes it difficult to organize the operation process at a relatively low flight speed of the aircraft and leads to the intake unstart. With continuous-detonation combustion, the chemical energy of fuel is released in a narrow reaction zone in one or several detonation waves (DWs), continuously rotating in an annular or hollow combustor. Since the DW at each time instant occupies only a small part of the combustor cross section, this allows one to arrange the operation process at a relatively low flight speed of the aircraft with only a partial distortion of the air flow in the intake, i. e., without intake unstart. Another important advantage of continuous-detonation combustion is the possibility of arranging the operation process with high combustion completeness in a combustor of a relatively small longitudinal dimension close to the length of a self-sustaining DW. This allows a significant reduction of viscous losses in the engine path compared to the engine operating on continuous-deflagration combustion. The disadvantage of continuous-detonation combustion is, obviously, its unsteady nature. However, due to the high propagation velocity of the DWs continuously rotating in the combustor, the pulsations of flow parameters possess high frequencies (1 kHz and higher) and the operation process appears to be quasi-steady.

There are several reviews of the physical processes in the engines utilizing continuous-detonation combustion often referred to as rotating detonation engines (RDEs) [3, 4]. The first studies of the RDE come back to [5, 6]. Currently, the research on RDEs is progressing worldwide. Most of the relevant publications deal with hydrogen as a fuel. Continuous detonation combustion of hydrogen–oxidizer mixtures was studied experimentally and computationally elsewhere [7–10] in annular and hollow combustors of different scale and design. Various modes of continuous-detonation combustion are reported including modes with several DWs rotating in the combustor in the same or opposite directions or with a single rotating DW referred to as the continuous spin detonation (CSD) mode as well as the longitudinally pulsating detonation (LPD) mode arising at certain limiting conditions of hydrogen and oxidizer supply. In the LPD mode, the detonation is reinitiated at a position close to the combustor outlet and propagates axially upstream as a supersonic reaction front covering the entire cross section of the combustor [7, 8]. The gaseous fuels other than hydrogen used so far in experiments on continuous-detonation combustion with air are ethylene, acetylene, syngas ($H_2 + CO$), etc. As for liquid fuels, the pioneering results of experiments on continuous-detonation combustion of aviation kerosene in air in a detonation afterburner are reported recently in [11, 12].

In the open literature, there are two types of publications on the DRs: those powered by the rocket-type RDEs and air-breathing RDEs. The former are beyond the scope of this study. As for the latter (the object of this study), there are only few publications on the experimental and theoretical studies of such DRs. The possibility of realizing the continuous-detonation combustion of hydrogen in air-breathing DR demonstrators is studied experimentally in [13, 14]. In our earlier works [15–18], the results of calculations and test fires of a large-scale hydrogen-fueled air-breathing DRs with an annular combustor under the conditions of an approaching supersonic air stream are reported. In [15], based on multivariant 3D numerical simulations, the feasibility of the continuous-detonation process in an annular combustor of a DR operating on hydrogen as fuel and air as oxidant in conditions of flight at a Mach number of M = 5 and an altitude of 20 km is demonstrated. In [16–18], the test fires of a hydrogen-fueled DR demonstrator are performed in the short-duration (pulsed) WTs. The design of the DR demonstrator corresponds to the conceptual design proposed in [15]. In [16, 17], two modes of continuous-detonation combustion are registered experimentally in the DR demonstrator at M = 4 to 8 while varying the overall air-to-fuel equivalence ratio at a fixed stagnation temperature of 300 K. In the first mode at M = 4, 5, and 6, a single DW is found to rotate in the annular combustor. In the second, limiting, mode at M = 5, 6, and 8, DWs were spontaneously reinitiated in the rear part of the combustor at a frequency of 0.9 kHz and propagated upstream towards the hydrogen supply nozzles. Test fires of the hydrogen-fueled air-breathing DR demonstrator are performed in [18] at the approaching air stream Mach number of 5.7 and the stagnation temperature of 1500 K. The various modes of continuous-detonation combustion are also registered. It is concluded that the transition from the conventional deflagrative combustion to the detonative combustion in DRs can potentially resolve the problems of rapid turbulent and molecular mixing of fuel with air and achieving high combustion completeness at the shortest distances from fuel injectors. A strong shock wave leading the detonation, either circulating continuously across the airflow in the annular combustor or periodically recovering and propagating upstream the airflow, induces enormous transient shear stresses in the cross streams of fuel and air and in transverse shock waves reflecting from the combustor walls causing aerodynamic breakup of fuel jets/sprays and mixing of reactants. As for chemical transformations, they proceed in a shock compressed gas in the self-ignition mode at a very high rate.

In view of the specific features of continuous-detonation combustion mentioned above, it can be expected that the startup Mach number for air-breathing DRs will be less than for ramjets operating on continuous-deflagration combustion. This fundamental issue is addressed in our paper by considering hydrogen as a fuel.

Thus, in this work, we focus on:

(1) developing the conceptual design of the air-breathing DR which could ensure the startup Mach number significantly less than M = 2 at sea level (to reduce the starting mass of the aircraft with a booster);

(2) design of an air-breathing DR demonstrator; and

(3) manufacturing and testing the air-breathing DR demonstrator in a pulsed WT at approaching air stream Mach numbers M = 2 and 1.5.

2 Numerical Simulations

2.1 Statement of the problem

To develop the conceptual design of the DR, we follow the approach of [15] which well justified itself in our previous studies. Thus, we consider the flow of homogeneous stoichiometric hydrogen–air mixture in a cylindrical computational domain with an axisymmetric DR (Fig. 1). Both the external flow around the DR with a zero angle

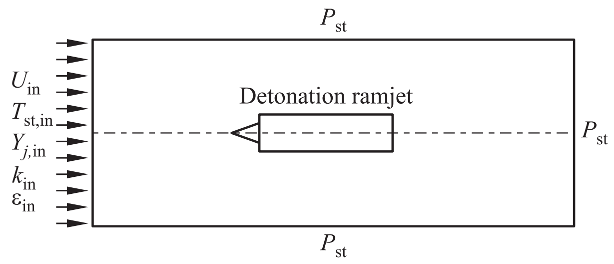


Figure 1 Schematic of the computational domain

of attack and the internal flow in it are considered. All boundaries of the computational domain are located far enough from the DR to avoid the influence of the boundaries on the flow inside the DR and in its vicinity.

At the left (inlet) boundary of the computational domain, we specify the flow velocity $U_{\rm in}$ and the static temperature of the mixture $T_{\rm st,in}$ corresponding to DR flight with Mach number M at sea level, and the mean mass fractions of the mixture components $Y_{j,\rm in}$ ($j = \rm O_2$, $\rm N_2$, and $\rm H_2$), as well as nonzero parameters of (weak) turbulence: the kinetic energy $k_{\rm in}$ and its dissipation $\varepsilon_{\rm in}$.

At the lateral boundaries and at the right (outlet) boundary of the computational domain, we specify a constant static pressure $P_{\rm st}$.

The task is to search for the configuration of the DR with continuous-detonation combustion of the hydrogen–air mixture which, firstly, ensures the cruising flight with M = 2 and, secondly, provides autonomous acceleration of the DR at a startup Mach number significantly less than M = 2. Mathematically, the possibility of autonomous acceleration of the DR means that the total force (effective thrust) acting on all solid surfaces of the DR is positive, that is, directed opposite to the approaching stream. It is understood that an excess of the effective thrust can be used to overcome the aerodynamic drag of other elements of the DR-powered aircraft and to accelerate the aircraft to the cruising speed.

Note that the replacement of the approaching air stream by the stream of homogeneous hydrogen–air mixture is very convenient for massive calculations. Firstly, the most complicated process of mixture formation in the DR combustor is excluded from consideration and, therefore, the search for DR configuration with continuous-detonation combustion at various flight Mach numbers is greatly simplified despite some differences in the thermophysical properties of the approaching stream. Secondly, chemical transformations in the gas can be artificially suppressed (reaction rates are taken equal to zero) both in the external flow and in the internal flow up to a certain cross section, the location of which in the future could determine the best placement of fuel supply holes in a DR with separate supply of air and hydrogen. Thirdly, we are mainly focused herein on the development of the conceptual design of the DR, manufacturing of DR demonstrator and its test fires. Therefore, the direct comparison of computational and experimental results is out of the scope of this study. Three-dimensional numerical simulations of the DR with separate supply of air and hydrogen will be the next step of this study, when a direct comparison of calculations with the results of test fires will be made. Also, further numerical simulations will be used to optimize the geometry of the DR flow path.

2.2 Model and calculation procedure

To solve the problem, a physical and mathematical model and calculation procedure are used which are described in detail in [15, 19]. Here, we only briefly repeat the main points.

The flow is described by the 3D Unsteady Reynolds Averaged Navier–Stokes (URANS), energy conservation and species continuity equations for the multicomponent reactive mixture. Turbulent fluxes are simulated using the k–ε turbulence model for compressible flows. The particle method (PM) is used to determine the contributions of volumetric reactions to chemical sources. The instantaneous local thermochemical states of the turbulent reacting flow are represented in the PM as a set of interacting (Lagrangian) particles. For each particle, a system of conservation equations of mass, momentum, and energy is solved and the classical models of relaxation to the mean are used to determine the flux (exchange) terms. The caloric and thermal equations of state for the mixture of ideal gases with variable specific heats are used. Variable thermophysical properties of the gas are considered.

The combined control volume – PM algorithm is applied for solving the governing equations numerically. Chemical source terms in particles are calculated implicitly with an internal integration time step. Hydrogen oxidation is described by a single-step reaction. The values of preexponential factor and activation energy in the Arrhenius-type equation for the local instantaneous reaction rate are obtained by fitting the dependences of the self-ignition induction period on pressure, temperature, and air-to-fuel equivalence ratio α with those provided by the validated detailed reaction mechanism of hydrogen oxidation.

The effective thrust, total thrust, specific impulse, specific fuel consumption, etc. of the DR are calculated according to the methods described in detail in [15]. Thus, the effective thrust F_{eff} is calculated as the integral of axial projections of pressure P and friction F_f forces over all rigid walls of the DR:

$$F_{\mathrm{eff}} = \frac{\int_S P\,dS + \int_S F_f\,dS}{S}$$

where S is the surface area of the rigid walls and dS is its infinitesimal element.

In general, the flow field in the DR and the predicted thrust performance of the DR depend on the selected value of N_p and on the computational mesh. Preliminary calculations showed that for $N_p > 10$, the dependence of the predicted flow field on N_p becomes weak. The influence of the computational mesh is studied by comparing the results of calculations on different structured meshes condensed towards the DR rigid walls and containing from 500 thousand to 2 million cells with a mean cell size at a level of 0.2 mm in the key regions of the combustor. Despite the meshes used did not allow the resolution of fine structures (transverse waves, etc.) in the DW, they allowed adequately catching the overall flow field and performing massive multivariate calculations while searching for the proper configuration of the DR. As for the DR thrust performance, the base meshes containing about 1 million cells provided the values of the effective thrust which were within 2 percent accuracy as compared with the simulations using the finest meshes.

The calculation starts with purging the DR with a free flow of the homogeneous stoichiometric hydrogen–air mixture during a time interval of 0.1 ms to form the active layer of reactive mixture of a certain thickness in the combustor. Next, the procedure of detonation initiation is launched. This procedure involves rapid depletion of fuel in all particles in a preset initiation region in the active layer. Fuel depletion in the particles leads to a rapid increase in local pressure and the formation of the initiating shock wave propagating in the annular gap of the combustor in all directions. To ensure the propagation of the DW in the selected direction, a layer of temporarily inert particles is provided in the initiation region.

The predictive capability of the model and the computational code were previously tested against experimental data on the continuous-detonation combustion of a hydrogen–air mixture obtained by a research group from the M. A. Lavrentiev Institute of Hydrodynamics [20] and our group [21, 22].

2.3 Conceptual design of detonation ramjet

First, we search for the conceptual design of DR with continuous-detonation combustion of hydrogen–air mixture which will ensure a flight with a cruising Mach number M = 2 at sea level with a zero angle of attack. The conceptual designs of the DR with the front position of air intake and combustor like those studied in [15] failed to provide a positive effective thrust at such flight conditions. As a result of multivariate gasdynamic calculations (25 DR configurations all in all), the exterior and interior appearance of the DR shown in Fig. 2 with the rear position of the combustor is formed.

The DR consists of a center body with a front cone, cylindrical part, and rear cone, an annular intake with the gasdynamic isolator, and the flow splitter dividing the incoming flow into two annular channels: outer combustor and inner bypass. The dimensions of the DR are as follows. The external diameters of the combustor and the cylindrical part of the center body are 120 and 90 mm, respectively. The width of the annular gap at the combustor entrance is 7 mm. The internal and external diameters of the annular gap at the entrance to the bypass channel are 80 and 102 mm, respectively. The combustor inlet is displaced downstream from the annular intake by 60 mm. The length of the combustor and bypass channel is 140 mm. The inner and outer diameters of the combustor exit are 82 and 120 mm,

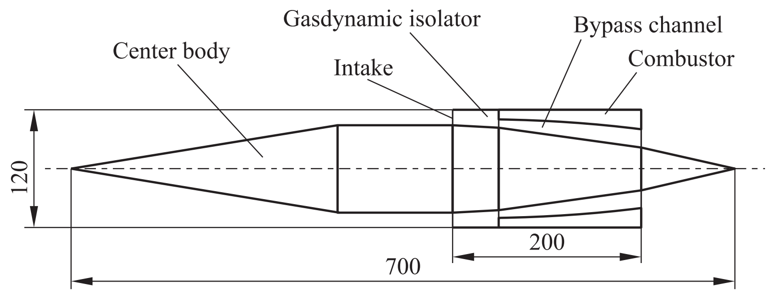
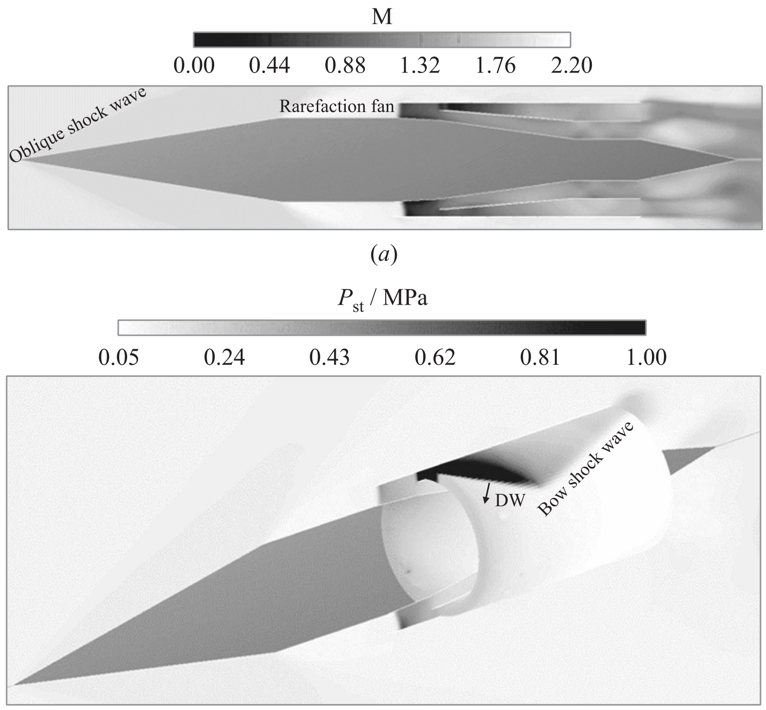


Figure 2 Dual-duct conceptual design of the detonation ramjet. Dimensions are in millimeters

respectively. The inner and outer diameters of the bypass channel exit are 42 and 80 mm, respectively. It is implied that hydrogen could be injected from the outer combustor wall through a belt of radial holes located at a certain distance downstream from the combustor entrance.

The transverse geometrical dimensions of the DR and the front cone angle (9°) are chosen based on the limitations imposed by the nozzles of the WT (see below). The length of the DR combustor is chosen based on the implication that it should be about twice longer than the length of the rotating DW. The length of the cylindrical part of the center body is selected so that the flow entering the combustor is uniform except for the near-wall boundary layer, whereas the boundary layer thickness at the entrance to the bypass channel does not exceed its height. From now on, this conceptual design will be referred to as the dual-duct design of the DR. This conceptual design is patented by us [23] and discussed with the expert community at the 10th International Workshop on Detonation for Propulsion (St. Petersburg, 2019) [24].

Under flight conditions, the approaching supersonic gas stream is partly decelerated in the oblique shock wave attached to the front cone of the center body and in the near-wall boundary layer and then accelerates in the fan of rarefaction waves with partial recovery of flow parameters and enters the annular intake (Fig. 3*a*). The length of the cylindrical part of the center body is selected so that the fan of rarefac-



(b)

Figure 3 Calculated distributions of local flow Mach number M (a) and static pressure P_{st} (b) in the longitudinal section and at the combustor surfaces under conditions of detonation ramjet flight with Mach 2

tion waves does not enter the intake. At the entrance to the annular intake, the maximum local flow Mach number is M ≈ 2.2, the mean static pressure is approximately $0.9P_{\mathrm{st}}$, and the total mass flow rate of gas reaches 94% of the value calculated based on the velocity and density in the approaching gas stream. After entering the annular intake and passing the gasdynamic isolator, the gas flow is split into two parts: one part enters the annular combustor and the other enters the annular bypass channel. Upon ignition, a continuous-detonation combustion of the reactive mixture is established in the annular combustor which provides acceleration of the detonation products downstream with the formation of a quasi-steady exhaust jet and the creation of thrust. The DW is inclined to the combustor axis and is terminated downstream by the bow shock wave (Fig. 3*b*). The maximum pressure in the DW is about 3.5 MPa (scaling in Fig. 3*b* is changed to better illustrate the flow structure). Continuous-detonation combustion in the annular combustor is organized so that neither the combustible mixture nor the detonation products penetrate upstream the intake. In the calculations for flight conditions with Mach 2, a steady-state operation process is obtained with one DW, continuously rotating in the annular combustor. The tangential and axial components of the DW velocity vector are about 1900 and 520 m/s, respectively, which gives a normal detonation velocity of ≈ 1970 m/s. The length of the DW in the axial direction is about 70 mm, i. e., nearly half the length of the combustor.

The gas flow entering the bypass channel includes a near-wall boundary layer formed on the center body. This eliminates the negative effect of the boundary layer on filling the combustor with a reactive mixture. On the one hand, the gas flow directed to the bypass channel provides cooling of the internal wall of the combustor and, on the other hand, prevents the gasdynamic effect of continuous-detonation combustion of the reactive mixture in the combustor on the gas flow at the entrance to the intake. The continuous-detonation operation process in a combustor is accompanied by generation of gasdynamic disturbances in the form of shock waves traveling upstream toward the intake which cause an increase in pressure at the entrance to the intake. This can lead to the intake unstart and to the failure of continuous-detonation combustion in the combustor. Since the front edge of the wall separating the combustor from the bypass channel is displaced deep into the gasdynamic isolator, such shock waves, when they exit the combustor, are effectively attenuated and transformed into weak pressure waves that do not affect much the intake operation but contribute to the creation of additional thrust when the waves and jets are directed to the bypass channel.

The cross sections of the intake, combustor, and bypass channel are selected to ensure a steady-state operation process with at least one DW and on-design operation of the intake without unstart. The outlet cross sections are selected to ensure as complete expansion

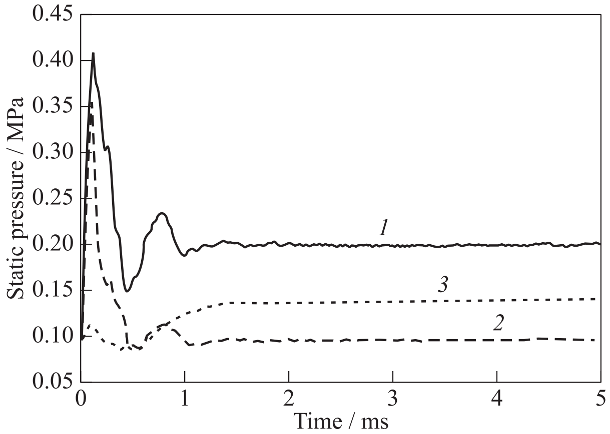


Figure 4 Calculated time histories of the mean static pressure in the combustor volume $\overline{P_C}$ (*1*) and in the exit cross sections of the combustor (*2*) and bypass channel (*3*) under conditions of DR flight with Mach 2

of the detonation products and the gas passing through the bypass channel to the atmospheric pressure as possible. Figure 4 shows the calculated time histories of the static pressure $\overline{P_C}$ averaged over the combustor volume as well as the static pressures averaged over the outlet sections of the combustor and bypass channel of the DR. The steady-state flow in the indicated elements of the DR is seen to be established in 2–3 ms after the beginning of calculation.

Table 1 shows the steady-state values of the calculated thrust characteristics of the DR in flight conditions with Mach 2.0. The effective thrust F_{eff} is defined as the integral of pressure and friction forces over all solid surfaces of the DR during the continuous-detonation operation process. The total thrust R is defined as the difference between the effective thrust F_{eff} and the aerodynamic drag force F_d of the DR in the absence of combustion.

Table 1 Calculated thrust performances of DR under flight conditions with M = 2

F_{eff}, N	R, N	G, kg/s	G_C, kg/s
510	1040	3.51	1.24

It should be noted that the thrust of the DR is created only by 35% of the gas entering the intake, while 65% of the gas expels through the bypass channel.

2.4 Calculated startup Mach number

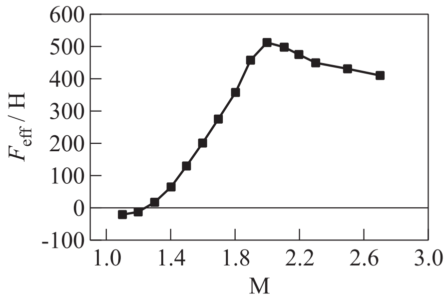


Figure 5 Calculated dependence of the DR effective thrust on the flight Mach number

Next, we determine the startup Mach number for the DR configuration discussed above. To do this, a series of calculations of the operation process in the combustor with a change in the Mach number of the approaching gas stream other conditions being equal is performed.

Figure 5 shows the calculated dependence of the effective thrust of the DR on the flight Mach number. As expected, the maximum value of the effective thrust (510 N) is attained at M = 2, since the DR is designed precisely for such a cruising flight speed. Thus, for such a flight speed, the outlet sections of the combustor and bypass channel are optimized to ensure as full expansion of the detonation products and gas passing through the bypass channel as possible. At M $<$ 2 and M $>$ 2, there is an overexpansion and underexpansion of gases in the combustor, respectively, and, therefore, a decrease in the effectiveness of the DR. With an increase in the flight Mach number, the axial length of the DW increases and at M $\approx$ 2.7 it becomes comparable with the combustor length. At M $\geq$ 2.8, the continuous-detonation operation process is getting unstable: it is blown off irreversibly without residual combustion of hydrogen. It follows from Fig. 5 that the calculated effective thrust of the DR becomes positive at M $\geq$ 1.3, i. e., the minimum calculated value of the startup Mach number is 1.3.

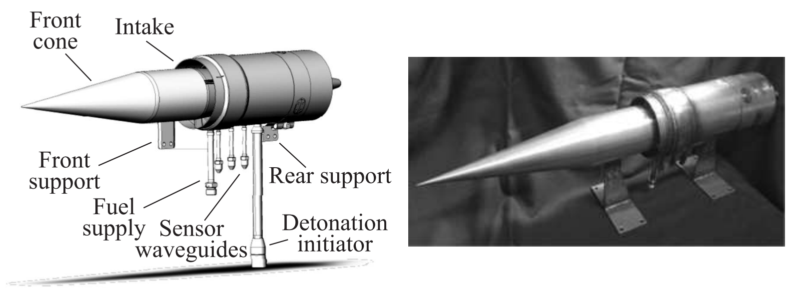
3 Test Fires

3.1 Detonation ramjet demonstrator and test rig

Based on the conceptual design of the DR obtained computationally, we developed design documentation for the dual-duct DR demonstrator. Figure 6 shows a general view and a longitudinal section of a 3D model of the DR 730 mm long with an annular combustor 120 mm in outer diameter. The walls of the combustor and the bypass channel made of stainless-steel sheet 1 mm thick are attached to the center body by three pylons in the front and rear parts of the structure.

Hydrogen is supplied to the combustor from the manifold on the outer wall through a belt of 120 equidistant holes 0.8 mm in diameter at a distance of 10 mm downstream from the combustor entrance. The hydrogen supply pressure in test fires does not exceed 1.1 MPa.

To initiate the continuous-detonation operation process, a detonation initiator fueled by a hydrogen–oxygen mixture is used. The initiator consists of a prechamber and detonation tube. The prechamber is a round tube 20 mm in diameter and 30 mm long. The detonation tube is a straight round tube 10 mm in diameter and 200 mm long. A standard automobile spark plug is used to ignite the mixture. The detonator is mounted on the outer wall of the DR combustor 80 mm downstream from the combustor entrance. Hydrogen



(*a*) (*b*)

Figure 6 Dual-duct DR: (*a*) 3D model; and (*b*) demonstrator and oxygen are fed into the prechamber through the tubes 4 mm in diameter. After the control signal is applied to turn on the detonator, the detonation tube is filled with the hydrogen–oxygen mixture during about 200 ms and then the mixture is ignited, the arising flame accelerates providing the deflagration-to-detonation transition in the detonation tube, and the generated DW enters the annular gap of the combustor. The characteristic time during which the detonation pulse created by the detonator influences the operation process in the DR combustor does not exceed 10 ms. A similar initiator was used in experiments [7, 16–18] and proved its effectiveness.

Test fires of the DR demonstrator are performed in a pulsed WT of the N. N. Semenov Federal Research Center for Chemical Physics initially designed for testing a kerosene-fueled pulse-detonation ramjet in free air jets with Mach numbers up to M = 0.9 [25]. The volume and the maximum air pressure in the WT receiver are 10.4 m^3 and 1.6 MPa, respectively. To obtain a supersonic free air jet, the WT is modified for installing replaceable converging-diverging supersonic nozzles with design Mach numbers M = 1.5 and 2.0. The converging parts and the regions of the critical section of the nozzles are shaped to ensure the air flow without separation. The diverging parts of the nozzles are shaped for isentropic expansion of the air.

Additionally, calculations of the supersonic air flow around the DR demonstrator placed in free air jets of the WT nozzles are conducted. The calculated values of the aerodynamic drag force of the DR F_d in free air jets at M = 2.0 and 1.5 are −700 and −450 N, respectively. Note that the drag of the fairing of the DR demonstrator used to hide all supports and fittings is not included in these values.

Hydrogen is supplied to the DR demonstrator from a fuel receiver with a volume of 0.64 m^3 along a tube 32 mm in diameter through a system of fast-response valves with an opening time of $\sim$ 50 ms.

The DR demonstrator is mounted on the thrust table along the axis of a supersonic nozzle with a zero angle of attack. Using preliminary 3D gasdynamic calculations, the best position of the DR demonstrator with respect to the nozzle exit is determined, in which the flow field over the DR demonstrator in a free supersonic air jet is as close as possible to the flow field in a uniform supersonic approaching air stream.

The static pressure P_C in the combustor is measured with a CourantDA 1.6 MPa low-frequency static pressure sensor. Pressure pulsations P'_C in the combustor are measured by a PCB 113B24 high-frequency pressure pulsation sensor. Both pressure sensors are installed remotely using the waveguide tubes. The static pressure sensor is installed at the end of a 2-meter long waveguide tube in a gasdynamic damper 180 cm^3 in volume. The pressure pulsation sensor is installed in a waveguide tube 4 mm in diameter at a distance of 0.8 m from its fastening to the external wall of the combustor. To avoid the interference of incident and reflected pressure waves during pressure pulsation measurements, a damping volume in the form of the same tube 25 m long is installed behind the sensor. Other parameters measured in the test fires include the force (F) acting on the DR demonstrator measured by the load cell, the pressure in the air receiver (P_r), the pressure in the WT high-pressure chamber (P_0), and the pressure at the exit of the supersonic nozzle ($P_{\mathrm{ST,NOZ}}$). The force F acting on the DR demonstrator is measured by a Tenzo-M T2 load cell designed for ± 2000 N. The pressures P_r, P_0, and $P_{\mathrm{ST,NOZ}}$ are measured by CourantDA 2.0 MPa, 1.6 MPa, and 250 kPa low-frequency pressure sensors. The maximum estimated errors of static

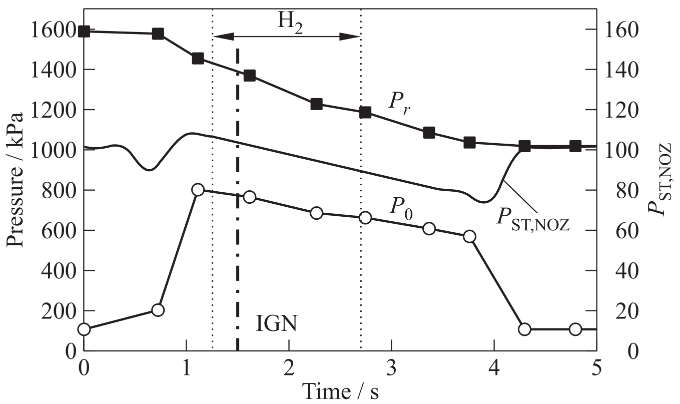


Figure 7 Measured time histories of pressure in the air receiver (P_r), in the high-pressure chamber (P_0), and at the nozzle exit ($P_{\mathrm{ST,NOZ}}$) in the test with M = 2.0

pressure and force measurements are 5%. Figure 7 shows examples of the measured time histories of P_r, P_0, and $P_{\mathrm{ST,NOZ}}$ when launching the WT with a Mach 2.0 nozzle. The operation time of the WT from ON to OFF is about 4 s.

The vertical dotted lines in Fig. 7 mark the period during which hydrogen is supplied to the combustor, and the vertical dash-dotted line marks the instant of triggering the detonation initiator (IGN). The time delay between the start of hydrogen supply and the initiation of the operation process in the combustor is required to fill the fuel lines and manifold. It follows from Fig. 7 that in the period from detonation initiation to hydrogen cutoff, the static pressure at the exit of the WT supersonic nozzle changes from 105 to 90 kPa which ensures the on-design mode of operation of the DR intake.

3.2 Results of test fires at Mach 2

Below, we present the results of test fires of the DR demonstrator at M = 2.0, a stagnation pressure of 700–800 kPa, and a stagnation temperature of about 270 K. In the test fires, the air-to-fuel equivalence ratio in the combustor, α_C, is varied from 0.8 to 2.1 (the maximum estimated error in α_C is 10%). The tasks of the test fires are: (*i*) to confirm the possibility of arranging continuous-detonation combustion in the DR demonstrator; and (*ii*) to determine the thrust performance of the DR demonstrator.

As an example, Fig. 8 shows the primary records of the measured parameters in one of the test fires: (*a*) P_{H_2} (pressure in the hydrogen supply manifold); (*b*) $P_{\mathrm{ST,NOZ}}$; (*c*) F; (*d*) $\overline{P_C}$; and (*e*) P'_C. It can be seen from Fig. 8*c* that by the time the hydrogen supply starts ($\mathrm{H}_{2,\mathrm{ON}}$ = 1.2 s), the force acting on the DR demonstrator is negative and amounts to −1460 N and after ignition (IGN = 1.4 ms), it increases up to −600 N. In this case, the average static pressure in the combustor $\overline{P_C}$ (see Fig. 8*d*) increases from 50 to 177 kPa and regular pressure pulsations of large amplitude with a frequency $f \approx 824$ Hz appear in the combustor (see Fig. 8*e*). The operation process in the combustor continues until the hydrogen supply is cut off ($\mathrm{H}_{2,\mathrm{OFF}}$ = 2.6 s) and then the combustor is purged with air for about 1.5 s. Defining the total thrust R as the difference between the readings of the force sensor after ignition and before ignition, one ob-

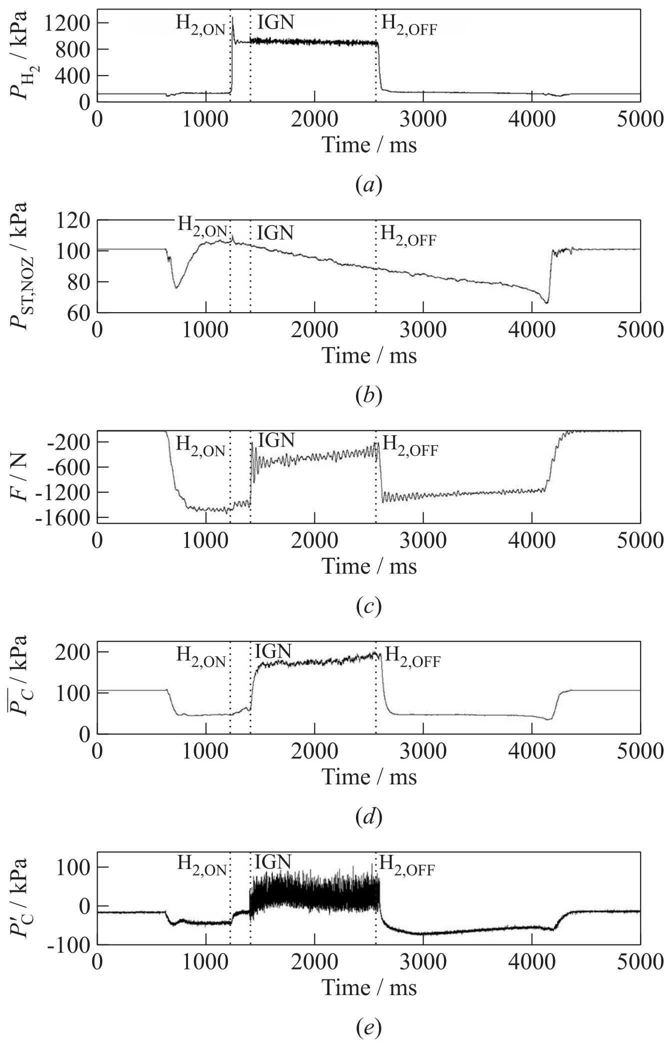


Figure 8 Examples of primary records of all sensors measuring flow parameters in one of the test fires: (*a*) P_{H_2}; (*b*) $P_{\mathrm{ST,NOZ}}$; (*c*) F; (*d*) $\overline{P_C}$; and (*e*) P'_C

Table 2 Main parameters and results of DR demonstrator test fires at M = 2.0

No.	P_{H_2}, kPa	G_{H_2}, g/s	α_C	$\overline{P_C}$, kPa	f, Hz	F, N	R, N	I_{sp}, s	F_{eff}, N
1	1065	75	0.80	160	664	−600	860	1160	160
2	903	63	0.97	177	824	−590	850	1350	150
3	735	51	1.19	176	1080/2635*	−705	755	1490	55
4	574	37	1.65	185	847/1850*	−855	605	1625	−95
5	438	29	2.10			Fuel-lean limit			

*Two pronounced pressure pulsation frequencies were observed.

tains $R \approx 850$ N. The effective thrust F_{eff}, defined as the sum of the total thrust R and the calculated aerodynamic drag force of the DR in the uniform approaching air stream ($F_d \approx -700$ N, see above), is then equal to $F_{eff} \approx 150$ N. The fuel-based specific impulse I_{sp} defined as the ratio of the total thrust R to the hydrogen weight flow rate $g\,G_{H_2}$ (g is the acceleration of gravity and G_{H_2} is the hydrogen mass flow rate measured by the pressure drop in the hydrogen receiver), is then equal to $I_{sp} \approx 1350$ s.

Table 2 shows the main parameters and the results of several representative test fires. Records of sensors in Fig. 8 correspond to the test fire No. 2 in Table 2. When α_C changes from 0.8 to 1.65 (with a decrease in hydrogen mass flow rate from 75 to 29 g/s), the total thrust R decreases from 860 to 605 N and the fuel-based specific impulse I_{sp} increases from 1160 to 1625 s. At $\alpha_C \approx 2.1$, hydrogen combustion is not registered.

The most important result of test fires at M = 2.0 is experimental proof of the possibility of arranging the steady-state continuous-detonation combustion of hydrogen in the DR demonstrator of the developed design. In the test fires, the near-limit LPD mode, previously observed in [7, 8, 11, 12], and the CSD mode are registered. In the LPD mode, detonation periodically (with a frequency f) is reinitiated in a fresh mixture in the vicinity of the combustor outlet and the generated DW propagates upstream toward the belt of hydrogen supply holes causing longitudinal pressure pulsations with a frequency f.

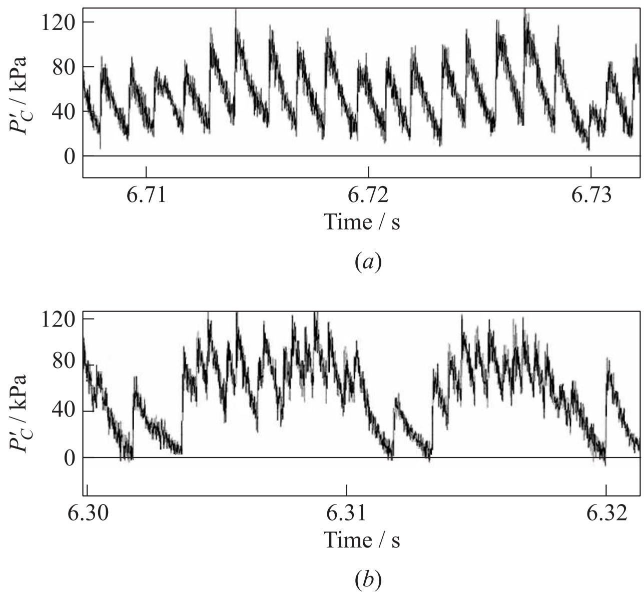


Figure 9 Fragments of records of a pressure pulsation sensor in the combustor for the LPD mode in test fire No. 2 (*a*) and for the combined LPD/CSD mode in test fire No. 4 (*b*)

Figure 9 shows fragments of records of the pressure pulsation sensor in the combustor for the LPD mode (Fig. 9*a*) and for the mixed LPD/CSD mode (Fig. 9*b*). The LPD mode is characterized by a signal with regular triangular pressure pulsations having a steep front and a constant frequency ($\sim$ 800 Hz). In the mixed LPD/CSD mode, the signal has a more complicated shape: on the background of a frequency of about 850 Hz, low-frequency ($\sim$ 100 Hz) pressure pulsations with pronounced periods of high-frequency pulsations (2 kHz) are observed. The mixed LPD/CSD mode occurs when the hydrogen mass flow rate in the DR demonstrator is below a certain value corresponding to $\alpha_C \geq 1.19$. In this case, due to a decrease in the depth of penetration of hydrogen jets, the mixture formation zone could be shifted to the outer wall of the combustor where more favorable conditions are apparently created for the propagation of CSD.

Another possible reason could be the effect of pylons supporting the flow splitter.

In all test fires, the measured value of force F acting on the DR demonstrator is negative (see Table 2). This is, firstly, due to the high aerodynamic drag of the mounting system of the DR demonstrator on the thrust table which is estimated at -700 N (in addition to the aerodynamic drag of the DR demonstrator itself). Secondly, during high-speed video recording of test fires, it is found that the intake of the DR demonstrator operates in off-design mode.

As a matter of fact, Fig. 10 shows the video frames of test fires Nos. 1 to 4 where hydrogen combustion zones are clearly visible in front of the entrance to the DR intake. All frames correspond to

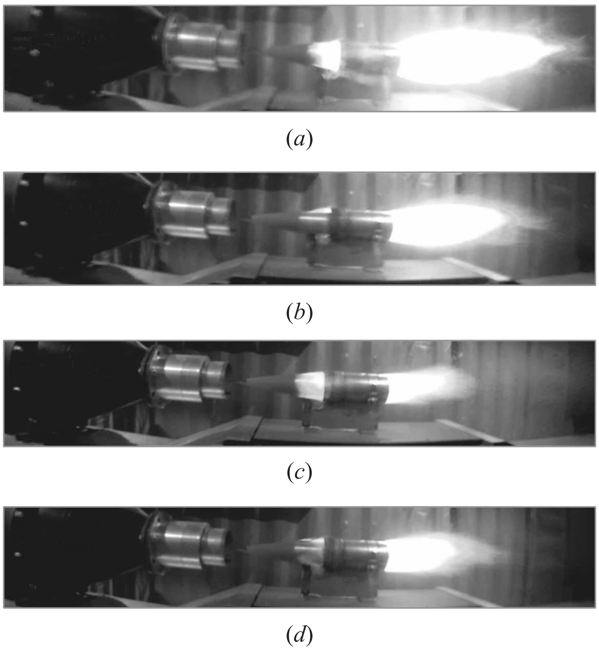


Figure 10 Frames of video records of test fires Nos. 1 to 4 at M = 2.0: (*a*) $\alpha_C = 0.80$; (*b*) 0.97; (*c*) 1.19; and (*d*) $\alpha_C = 1.65$

the quasi-steady-state operation of the DR. Presumably for the same reasons, the fuel-based specific impulse in these test fires attains a relatively low values of $I_{sp} = 1160$–1625 s.

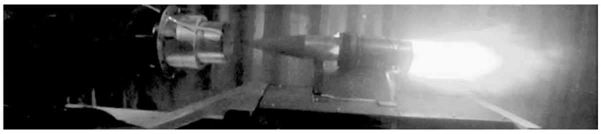
3.3 Results of test fires at Mach 1.5

Below, we present the results of test fires of the DR demonstrator at M = 1.5, a stagnation pressure of 300–400 kPa, and a stagnation temperature of about 270 K. In the test fires, the air-to-fuel equivalence ratio in the combustor α_C varies from 0.77 to 1.93. The tasks of the test fires are (i) to confirm the possibility of arranging continuous-detonation combustion in the DR demonstrator of the developed design at an off-design flight Mach number of M = 1.5; and (ii) to determine the thrust performance of the DR demonstrator at such a speed.

Table 3 shows the main parameters and results of test fires. When α_C changes from 0.77 to 1.6 (with a decrease in hydrogen mass flow rate from 60 to 29 g/s), the total thrust R changes from 605 N at $\alpha_C = 0.77$ to 440 N at $\alpha_C = 1.6$ with a maximum value of 650 N at $\alpha_C = 0.97$, and the fuel-based specific impulse I_{sp} varies from 1040 s at $\alpha_C = 0.77$ to 1560 s at $\alpha_C = 1.6$ with a maximum value of 1610 s at $\alpha_C = 1.41$. The absolute values of the total thrust R of the DR demonstrator decreases by approximately ~ 200 N as compared to

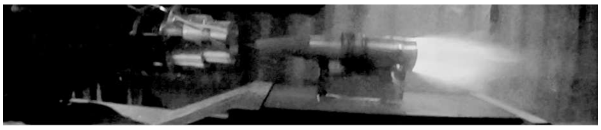
Table 3 Main parameters and results of DR demonstrator test fires at M = 1.5

No.	P_{H_2}, kPa	G_{H_2}, g/s	α_C	$\overline{P_{KC}}$, kPa	f, Hz	F, N	R, N	I_{sp}, s	F_{eff}, N
1	886	60	0.77	130	706	−395	605	1040	155
2	847	58	0.80	130	732	−385	615	1080	165
3	803	56	0.83	129	732	−355	645	1180	195
4	714	48	0.97	139	706	−350	650	1385	200
5	655	44	1.05	140	732	−370	630	1465	180
6	589	39	1.19	146	601	−410	590	1530	140
7	513	33	1.41	141	549	−480	520	1610	70
8	455	29	1.60	134	549	−560	440	1560	−10
9	387	24	1.93			Fuel-lean limit			



(*a*)

(*b*)



(*c*)

(*d*)

(*e*)

(*f*)

Figure 11 Frames of video recording of test fires Nos. 1, 3, 5, 6, 7, and 8 at M = 1.5: (*a*) $\alpha_C = 0.77$; (*b*) 0.83; (*c*) 1.05; (*d*) 1.19; (*e*) 1.41; and (*f*) $\alpha_C = 1.60$

the test fires at M = 2.0. At M = 1.5, the effective thrust F_{eff} is positive in almost the entire range of α_C. Thus, it is experimentally proved that the startup Mach number for the DR can be on the level of and even less than M = 1.5, that is, the calculation is confirmed by experiment (see Fig. 5). At $\alpha_C = 1.93$, hydrogen combustion is not registered.

The most important result of the test fires at M = 1.5 is the experimental proof of the possibility of arranging the steady-state continuous-detonation combustion of hydrogen in the DR of the developed design at an off-design flight speed. In the test fires with $0.77 \leq \alpha_C \leq 1.6$, the near-limit LPD mode is detected, whereas the maximum values of the frequency of longitudinal pressure pulsations (f = 700–730 Hz) are obtained at $0.77 \leq \alpha_C \leq 1.05$. At $\alpha_C > 1.05$, the pressure pulsation frequency first decreases to 550–600 Hz and then the operation process with the LPD mode fails to be initiated.

Apparently, one of the reasons that the effective thrust of the DR demonstrator at M = 1.5 turns out to be higher than at M = 2.0 is the more stable operation of the DR intake. Figure 11 shows the frames of video recording of test fires Nos. 1, 3, 5, 6, 7, and 8 at M = 1.5. If with $\alpha_C = 0.77$ the glow of combustion products is clearly visible at the entrance to the DR intake, then with $\alpha_C \geq 0.83$ this effect is absent. All frames in Fig. 11 correspond to the quasi-steady-state operation of the DR.

3.4 Discussion of experimental results

Comparison of the data in Tables 2 and 3 shows that with an increase in the Mach number of the approaching air stream, the pressure in the combustor increases from 130–145 kPa at M = 1.5 to 160–180 kPa at M = 2.0. For fuel-rich and near-stoichiometric mixtures with $0.77 \leq \alpha_C \leq 1.0$, the frequency of longitudinal pressure pulsations in the near-limit LPD mode is 700–800 Hz both at M = 1.5 and at 2.0. At M = 2.0 and $\alpha_C \geq 1.2$, due to the appearance of the mixed LPD/CSD mode in the combustor, the frequency of longitudinal pressure pulsations in the LPD mode increases to 800–900 Hz and a pressure pulsation mode with a frequency of 2600–1800 Hz, corresponding to the CSD mode, is added. In this case, a strange feature is manifested: despite an increase in α_C (decrease in fuel supply) and a decrease in thrust, the pressure in the combustor increases. This effect could be attributed to nonlinear pressure piling in the waveguide tube, at the end of which a static pressure sensor is installed [26] and will be addressed in future studies.

In general, the results obtained demonstrate the possibility of decreasing the Mach number of DR operation to at least M = 1.5. This possibility is associated with a combination of two factors: (*i*) the use of a dual-duct flow path; and (*ii*) continuous-detonation combustion in the DR. As for the first factor, even with $\alpha_C = 1$, the total air-to-fuel equivalence ratio at the entrance to the ramjet intake is close to 3, i. e., most of the air passes through the bypass channel without participating in the combustion and the heat release in the combustor does not lead to intake unstart even at a relatively low flight speed. As for the second factor, the heat release zone in the DW at each instant of time occupies only a small part of the combustor cross section which contributes to the on-design operation of the DR intake.

4 Concluding Remarks

The conceptual design of a hydrogen-fueled dual-duct DR for the cruising flight speed of Mach 2 at sea level is developed. The fundamental possibility of arranging continuous-detonation combustion of hydrogen in an expanding annular combustor of the DR is demonstrated computationally. The calculated effective thrust of such a DR is shown to become positive at M = 1.3, i. e., its startup Mach number can be lower than M = 2.0 which is typical for ramjets with continuous-deflagration combustion.

A DR demonstrator for cruising flight speed of Mach 2 at sea level is designed and manufactured. Its test fires are performed in a pulsed WT at approaching air stream Mach numbers M = 2 and 1.5. The most important result of the test fires at M = 2 is the experimental proof of the possibility of arranging steady-state continuous-detonation combustion of hydrogen in the DR of the developed design. In the test fires, the near-limit LPD mode and the CSD mode are registered.

The most important result of the test fires at M = 1.5 is the experimental proof of the possibility of arranging steady-state continuous-detonation combustion of hydrogen in the DR of the developed design at off-design flight speed. Thus, it is experimentally proved that the startup Mach number of the DR can be at the level of and even less than M = 1.5 which confirms the calculation qualitatively. In the test fires at M = 1.5, the near-limit LPD mode is registered.

For both Mach numbers, the thrust performance and economic characteristics of the DR are obtained. The maximum measured values of the effective thrust as well as the estimated values of the total thrust and fuel-based specific impulse are 160 N, 860 N, and 1630 s for test fires at M = 2 and 200 N, 650 N, and 1610 s for test fires at M = 1.5, respectively. The effective thrust of the DR demonstrator is shown to be positive for the air-to-fuel equivalence ratio in the combustor $\alpha_C < 1.3$ at M = 2.0 and $\alpha_C < 1.6$ at M = 1.5.

The experimental results obtained demonstrate the possibility of decreasing the Mach number of DR operation to at least M = 1.5. This possibility is associated with a combination of two factors: (*i*) the use of a dual-duct design; and (*ii*) continuous-detonation combustion in the DR. As for the first factor, even with $\alpha_C = 1$, the overall air-to-fuel equivalence ratio at the entrance to the DR intake is close to 3, i. e., most of the air passes through the bypass channel of the DR without participating in the combustion process and the heat release in the combustor does not lead to intake unstart even at a relatively low flight speed. As for the second factor, the heat release zone in the DW at each instant of time occupies only a small part of the combustor cross section which contributes to the on-design operation of the DR intake. Another important advantage of continuous-detonation combustion is the possibility of arranging an operation process with a high completeness of combustion in a combustor with a relatively small longitudinal dimension corresponding to the length of a self-sustaining DW. This allows one to significantly reduce viscous losses in the engine path compared to a ramjet with continuous-deflagration combustion.

This work was mainly focused on the development of the conceptual design of the DR, manufacturing of DR demonstrator and its test fires. Direct comparison of computational and experimental results was out of the scope of this study because the DR was designed based on the simplified 3D numerical simulations involving the DR flight in the homogeneous hydrogen–air mixture to exclude the process of mixture formation from consideration. Further work will be focused on the 3D numerical simulations of the DR with separate supply of air and hydrogen to provide a direct comparison of calculations with the results of test fires. Also, further numerical simulations will be used to optimize the geometry of the DR flow path. As for further experimental work, we intend to conduct additional test fires at the approaching air stream Mach numbers of less than M = 1.5.

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