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Rocket Engine with Continuous Film Detonation of Liquid Fuel

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Abstract—It was experimentally proven for the first time that it is possible to perform continuous detonation combustion of a film of a liquid fuel in the annular combustor of a demonstrator of a liquid-fuel detonation rocket engine. Firing tests revealed a near-limit mode of longitudinally pulsating film detonation and modes of continuous spinning film detonation with one and two detonation waves circulating within the annular gap of the combustor.

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The heterogeneous gas–film system has a number of important advantages for using in liquid-fuel rocket engines of a new type—detonation rocket engines (DRE) with continuous detonation combustors [1]. First, the gas–film system can be additionally used for active thermal protection of DRE walls by feeding the film to heat-stressed areas of the combustor. Second, in such a system, detonation can propagate at virtually any liquid film thickness [2, 3], which reduces requirements for fuel metering accuracy and increases operation process reliability. Third, in the stratified gas–film system, the interfacial area of which is relatively small (in comparison with that of the gas–drops system), the liquid preevaporation ahead of the leading shock wave of the detonation front is insignificant, and this prevents various violations of the operation process (flashbacks, etc.). As a disadvantage of the gas–film system, the necessity of using powerful sources of initiation of film detonation was mentioned in the literature. However, in our recent works [1, 4, 5], we have experimentally detected a deflagration-to-detonation transition in the gas (oxygen)–liquid-fuel (*n*-heptane [1, 4] or *n*-decane [5]) film system using a very weak ignition source, which generates a weak primary shock wave with a Mach number of 1.03–1.05.

The purpose of this work was to experimentally prove the possibility to perform continuous detonation combustion of a film of a liquid fuel in the annular combustor of a DRE demonstrator.

Figure 1 illustrates the design of a demonstrator of a DRE with a combustor shaped as an annular gap between a cylindrical-conical center body and a coaxial outer cylindrical wall. This design differs from that of DREs in which a gaseous or liquid fuel is fed to a combustor through distributed fuel nozzles as gas or drop jets, respectively [6–8]. In our case, a liquid fuel is fed to the combustor through the outer wall designed as a porous ring liner of finite length under the pressure of an expelling gas (nitrogen) and forms a thin liquid film on the surface of the wall. The oxidizer (gaseous oxygen) is supplied to the combustor through the annular gap in the axial direction, favoring the uniform spreading of the film over the wall. Film detonation is initiated by a shock wave transferred to the annular gap of the combustor, e.g., through the DRE exit. The shock wave, propagating over the liquid film, ensures rapid propellant mixing (owing to aerodynamic breakup of the film, evaporation of the formed drops, and turbulent molecular mixing of the fuel vapor with the oxidant), subsequent volume combustion of the obtained mixture (thanks to self-ignition of the shock-compressed two-phase medium), and formation of a self-sustained detonation wave capable of circulating in the annular combustor until, ahead of the wave front, the conditions necessary for the existence of the detonation wave persist.

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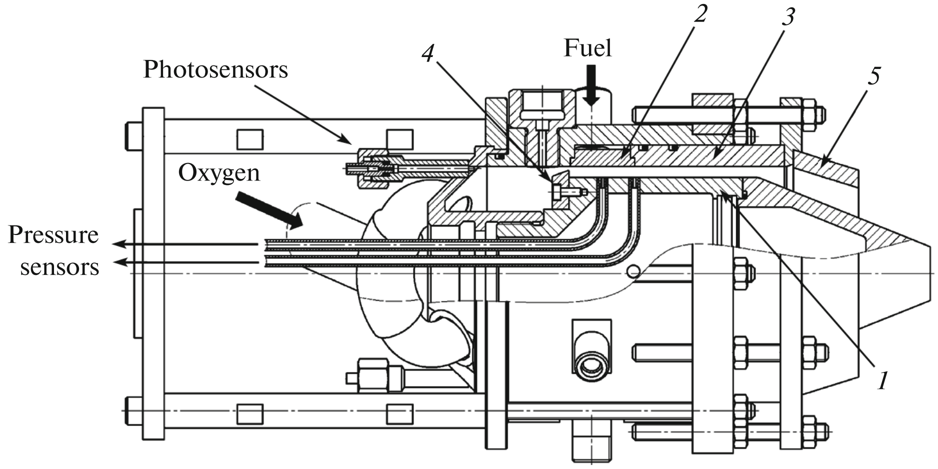

Fig. 1. Schematic of DRE: (*1*) center body, (*2*) porous ring liner, (*3*) hold-down impermeable ring liner, (*4*) hot side plate, and (*5*) convergent nozzle.

The annular combustor of the DRE demonstrator (Fig. 1) consists of four components: center body *1* (90 mm in diameter, comprising a 90-mm-long cylindrical section and 101-mm-long conical section, both made of copper), porous ring liner *2* (98 mm i.d., 30 mm long, and 9 mm thick, made of a permeable material), hold-down impermeable ring liner *3* (98 mm i.d., 70 mm long, and 11 mm thick, made of copper), and hot side plate *4* (a thin steel disk with a sharpened edge, which covers a part of the annular section at the combustor inlet, leaving a 1.2-mm-thick annular gap. All the components of the annular combustor are mounted in a cylindrical stainless-steel housing with one end flange. In the housing and the flange, there are orifices for feeding a liquid fuel (*n*-pentane) and a gaseous oxidizer (oxygen), respectively. The fuel was chosen for reasons of high vapor volatility (the boiling point of *n*-pentane at atmospheric pressure is 36°C); in our further investigations, *n*-pentane will be replaced by less volatile liquid fuels. In a number of tests, to the open outlet of the combustor, convergent nozzle *5* (with a length of 34 mm and a convergence angle of 35°) was attached.

The main component of the combustor is the porous ring liner. The liner is produced by powder metallurgy from a coarse nickel powder (PNK grade, GOST (State Standard) 9722-79) consisting of particles 70 to 100 μm in size. The component is formed in an elastic mold representing the shape of the original item by cold isostatic pressing at a pressure of 200 MPa in a CIP 62330 hydrostatic press. A required permeability of the sintered material is created using a pore-forming agent, ammonium bicarbonate $(NH_4)_2CO_3$ (10 vol %), which volatilizes at sufficiently low temperature to form through channels. A necessary strength of the material is reached without sacrificing the required permeability by selecting the sintering conditions. The optimum conditions turned out to be sintering at 900°C for 2 h in a hydrogen atmosphere.

The permeability of the porous liner is preliminarily measured with a special device, which allows to construct calibration curves of liquid-fuel flow rate versus expelling-gas pressure.

The system of diagnostics of the operation process in the combustor includes (Fig. 1) two photosensors with the −3dB cutoff frequency $F_{-3dB} > 2$ MHz (based on a BPW34 photodiode and an AD8066 operational amplifier) in the end flange and in the middle of the annular gap of the combustor, and also pressure sensors attached to waveguide tubes introduced to the center body: a low-frequency static pressure sensor (Kurant-DA, 2.5 MPa) measuring the average static pressure in the combustor 15 mm downstream of the hot side plate, and three high-frequency pressure sensors, PS1, PS2, and PS3 (Kistler 211B3) measuring pressure pulsations at three points located at an angle of 120° from each other within the same cross section of the combustor at a distance of 30 mm from the hot side plate. The temperatures of the center body and the porous liner are measured with chromel–alumel and chromel–copel thermocouples. The oxygen flow rate is calculated from the pressure decay rate in the oxygen receiver. The liquid-fuel flow rate is measured with a turbine velocity flowmeter.

The measurement errors are the following: operation process frequency (measured with the pressure pulsation sensors), within 3%; average static pressure in the combustor, within 1%; temperature of the walls, within 10%; and flow rates of the propellants, within 10%.

A typical firing test of the DRE lasts 1 s. Along with the time of the operation process involving combustion of the propellant mixture, the firing test time also includes the times of opening and closing of quick-acting fuel and oxidizer valves. All the firing tests were

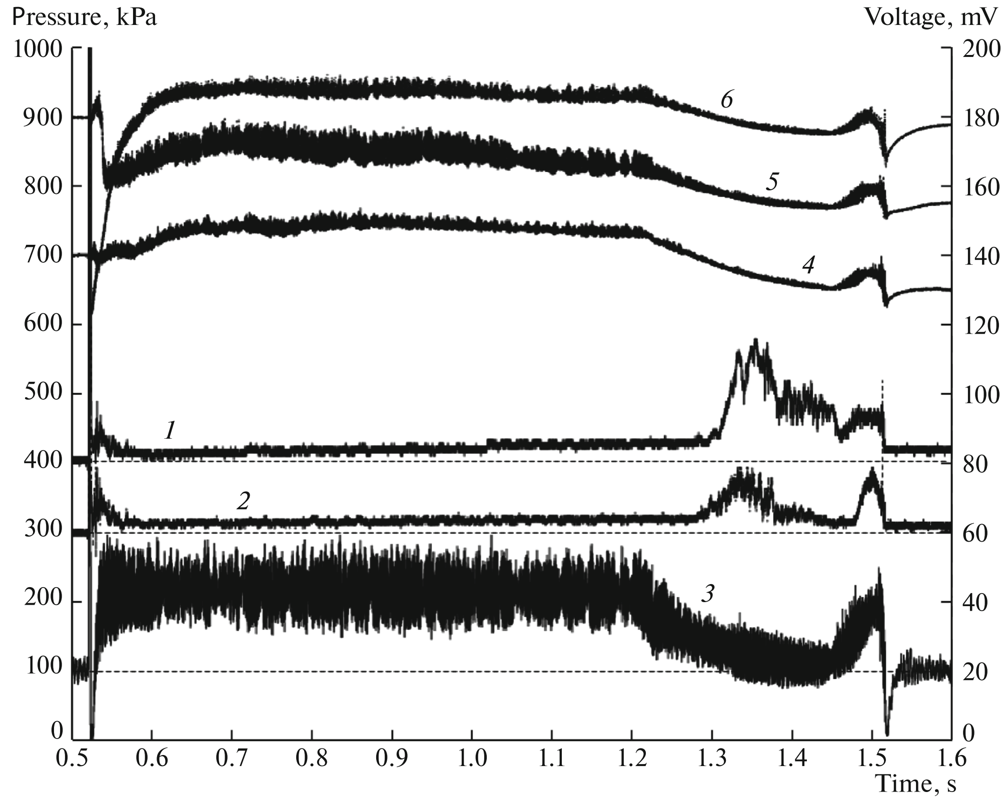

Fig. 2. Readings of (*1*, *2*) the photosensors, (*3*) the static pressure sensor in the combustor, and (*4–6*) the pressure pulsation sensors in one of the firing tests.

carried out at an ambient temperature from –3 to +3°C.

The most important result of the tests is an experimental proof of the possibility to perform continuous detonation combustion of a film of a liquid fuel in the annular combustor of a DRE demonstrator. The firing tests revealed a near-limit mode of longitudinally pulsating film detonation and modes of continuous spinning film detonation with one and two detonation waves circulating within the annular gap of the combustor.

Figure 2 presents the primary readings of two photosensors (curves *1*, *2*, right-hand axis), the static pressure sensor in the combustor (curve *3*, left-hand axis), and three pressure pulsation sensors (curves *4–6*, left-hand axis) in one of the firing tests at oxygen and *n*-pentane flow rates of 160 and 40 g/s, respectively, which corresponded to a total fuel–air equivalence ratio of ~0.8. Within the time range from 700 to 1100 ms, an approximately constant glow intensity and a constant value of the average absolute static pressure in the combustor of 0.22 MPa were detected. The temperature of the uncooled center body for the time of the test became much higher (about 100°C) than the temperature of the hold-down impermeable ring liner (approximately 50°C), although the total weight of the center body including the cone section (2.7 kg) exceeded the weight of the hold-down ring liner (2.1 kg). The temperature of the porous liner, which is cooled by the fuel, did not exceed 10°C.

Fourier analysis of the readings of the pressure pulsation sensors determined a dominating frequency of the operation process of 2.85 kHz (Fig. 3); i.e., the characteristic time of the quasi-steady-state operation process in the combustor is ~350 μs. Estimates showed that, for this time, the expulsion system feeding the liquid fuel to the combustor ensures the formation of a liquid-fuel film about 5 μm thick on the surface of the porous liner. Analysis of the phases of the pressure pulsations demonstrated that, in this firing test, there occurs a near-limit mode of longitudinally pulsating detonation, which is similar to the modes observed before [8–10] in tests with gaseous propellants. Indeed, Fig. 4 presents 2-ms-long fragments of the readings of three pressure pulsation sensors very early in the considered time range 700–1100 ms. The readings exhibit regular pressure pulsations with steep fronts, and the phases of the pressure pulsations at all the three sensors virtually (to within ~100 μs) coincide; i.e., the pressure waves arrive at these sensors almost simultaneously. Such a situation is possible if a detonation wave periodically (at a frequency of ~2.85 kHz) originates within the annular gap and

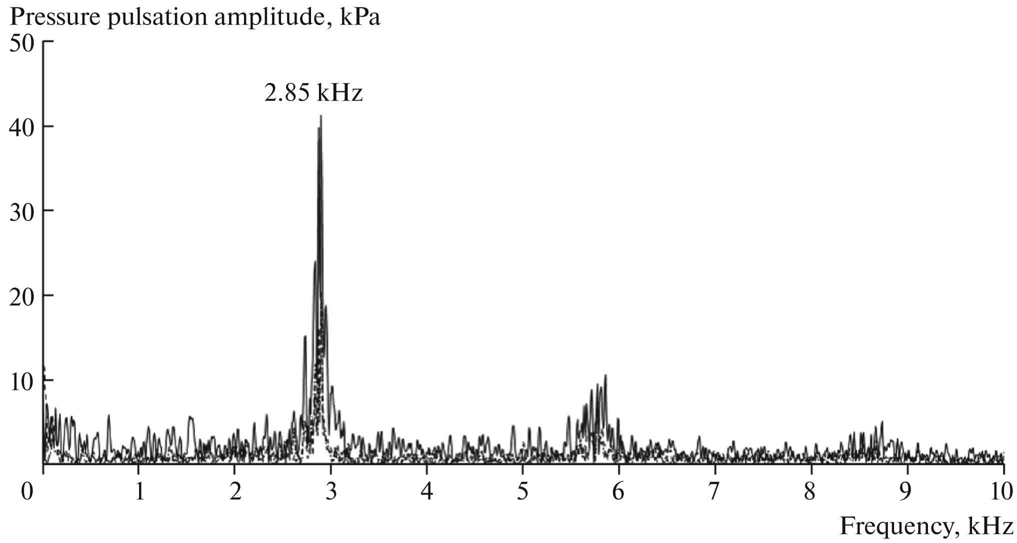

Fig. 3. Fourier analysis of a fragment of the readings of the pressure pulsation sensors.

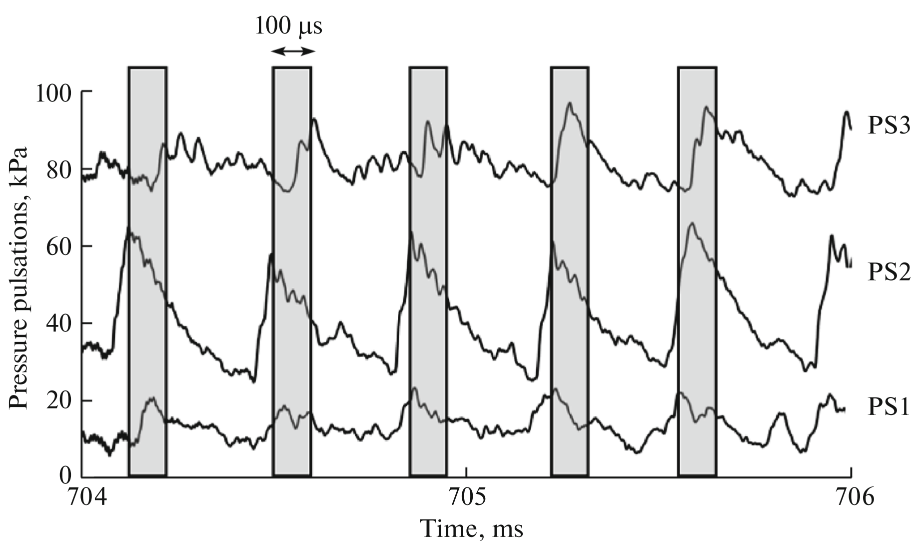

Fig. 4. Readings of the pressure pulsation sensors.

propagates upstream at high axial velocity and low tangential velocity.

An indirect proof of this assumption can be the results of the measurements [8–10], in which the spatiotemporal wave dynamics of origination, propagation, and decay of longitudinally pulsating detonation was studied and pressure readings similar to those in Fig. 4 were obtained. Because the maximum rate of filling of the annular combustor with oxygen is approximately equal to the speed of sound (~300 m/s), and the minimum (at the propagation limit) detonation velocity in the gas–film system is ~1000 m/s [1–5], the place of origination of longitudinally pulsating detonation should be at a distance of (1000–300) × 0.00035 = 0.25 m from the hot side plate. It can be assumed that detonation periodically originates in the neighborhood of the outlet cross section of the combustor, as in the recent experiments [8–10]: detonation explosion occurs if there is either localized self-ignition of a fresh propellant mixture at the developed interface with the hot products of the previous detonation wave, or shock compression of a fresh propellant mixture in the trailing-edge shock entering the combustor as the previous detonation wave decays. After the onset, the detonation wave travels toward the hot side plate, encompassing the entire volume of the annular gap. In this case, the distance traveled by the wave is comparable to the above estimate.

Along with the near-limit mode of longitudinally pulsating detonation, some of the firing tests revealed modes of continuous spinning film detonation with one and two detonation waves circulating within the annular gap of the combustor. For example, in one of the tests with an attached nozzle at oxygen and *n*-pentane flow rates of 150 and 80 g/s, respectively, which corresponded to a total fuel–air equivalence ratio of ~2.0, the average absolute static pressure in the combustor was 0.25 MPa, and the operation process frequency determined by Fourier analysis of the readings of the pressure pulsation sensors was ~4.7 kHz. Analysis of the pressure pulsation phases showed that, in this test, the operation process was accompanied by two detonation waves rotating continuously within the annular gap at a visible velocity of ~730 m/s. If the rate of filling the combustor with a fresh propellant mixture is taken into account, then the true normal detonation wave velocity is somewhat higher (~800 m/s) because the detonation wave front is inclined to the combustor axis. Such low velocities of reaction front propagation are possible only if the ignition of the propellant mixture is not determined by the temperature behind the leading detonation wave: this temperature is too low for the fuel vapor to self-ignite. According to our studies [8], under these conditions, the steady-state propagation of a detonation wave within the annular gap can be ensured only by ignition of the propellant mixture behind the shock wave reflected from the outer, compressive wall with subsequent energy release in the turbulent flame. Note that, in an annular combustor, the reflection of a shock wave from the outer, compressive, wall is an inherent feature of continuous detonation combustion [11, 12], and the turbulence intensity in the recirculation zone over the hot side plate is very high [11].

Thus, we for the first time experimentally proved the possibility to perform continuous detonation combustion of a film of a liquid fuel in the annular combustor of a demonstrator of a liquid-fuel detonation rocket engine of new type. In such a detonation rocket engine, the liquid-fuel film is used to ensure both the stable operation process and the active thermal protection of the combustor walls.

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