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LIQUID ROCKET ENGINES FOR LARGE THRUST: PRESENT AND FUTURE†

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Abstract—The possibility of an increase in economy and rise in efficiency of liquid-propellant rocket engines of launch vehicles and flight vehicles is considered. The characteristics of liquid-propellant rocket engines, made in the Soviet Union, U.S.A., France, Japan, China and other countries, and working on the following fuels: O_2, $+H_2$, O_2 + kerosene, $N_2O_4 + H_2N\text{-}N(CH_3)_2$, HNO_3 + dimethylhydrazine, etc. are recalled. Ways of further improvement of liquid-propellant rocket engines are outlined and the problems, arising during their realization, are discussed.

1. INTRODUCTION

The specific impulse I_{sp} is a main criterion of rocket engine economy. It is proportional to the thermal efficiency which depends on the degree of gas expansion in the jet nozzle:

$$\eta_t = 1 - (P_{on}/P_a)^{-\frac{\gamma-1}{\gamma}} \qquad I_{sp} \equiv \sqrt{\eta_t}$$

where

P_{on} = gas total pressure at the nozzle entry,
P_a = gas pressure at the nozzle exit section,
γ = adiabatic exponent.

The value of η increases with the growth of P_{on} and with the decrease of P_a.

The cruise engines of launch vehicles and other vehicles were developed in the direction of p value increase. Thus, on the early launch vehicles, Vostok, Saturn-V, Ariane-4, the pressure in the combustion chamber of liquid-propellant engines did not exceed 6–7 MPa, while at present in the launch vehicle engine: Energia, Space Shuttle, this pressure reaches 20–25 MPa [1–4]. Besides, the pressure in nozzle exit section (p) has decreased from 0.06 to 0.02 MPa. Correspondingly, the thermal efficiency of engines had increased from 0.4 to 0.7, but has not yet reached the limiting value. In the closed cycle liquid rocket engines with uncooled turbine blades, made of heat-resistant alloys, the pressure P_{on} = 30 to 40 MPa can be obtained. A further increase of P_{on} is possible in a turbo-pump unit with cooling[2,6] and in rotor feed systems [3]. The limits of attainable pressure in this system are considered in this paper.

The pressure decrease at the nozzle exit leads to the increase of nozzle and engine sizes and is accompanied by the occurrence of problems of aeroelasticity of thin-walled shell[7,8], thrust losses at great overexpansion of gas in the nozzle [2], engine shacking etc. For the successful realization of this increase it is necessary to use an optimal method of nozzle contouring at prescribed overall dimension, to develop a satisfactory elasticity theory and effective measures against the panel flutter, to use the nozzles with variable exit section and with other means preventing thrust loss due to the gas overexpansion.

It must be noted that 30–50% of the heat energy is carried out with exhaust gas from the engine and is not used for the creation of thrust. A small part of this energy is utilized in nozzles with regenerative cooling and increases the liquid-propellant rocket engine economy by $<1\%$ (SSME, liquid-propellant rocket engine of the launch vehicle Energia). There exists another possibility of heat utilization[9,10], which is connected with the appearance of a thrust turbulent component in the fluctuating flow. Its value amounts to 0.5–2%. The increase of these kinds of heat utilization will permit better engine economy.

The engine economy of liquid-propellant rocket engine can also be raised at the expense of thermal efficiency. It is possible to use the planet atmosphere matter for the improvement of engine characteristics[11,12], but in this case the engine must be supplemented by new elements: heat exchangers, ejector shrouds etc., which increase the mass of the liquid-propellant rocket engine.

2. LIMITS OF ACHIEVABLE EFFICIENCY

Thermal efficiency of the existing rocket engine varies in the range η_t = 0.4–0.7 (Fig. 1). The highest value is achieved at $P_{on}/P_a = 3 \times 10^3$. It can be shown[5] that there exists a limiting pressure ratio in the nozzle, namely $P_{on}/P_a = 1 \times 10^5$. At a subsequent increase of the expansion, thrust increase is compensated by the forces of gas friction against the wall and

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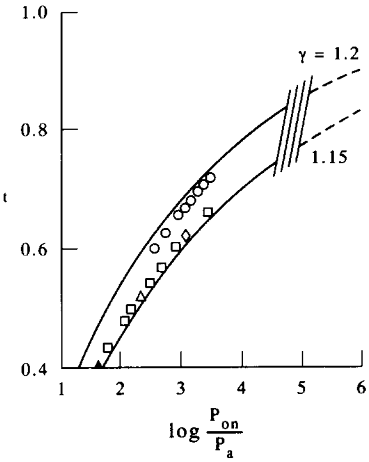

Fig. 1. Thermal efficiency of rocket engine. ///, Limiting pressure ratio.

by forces of nozzle case inertia. Therefore, in the liquid rocket engines (with Bryton cycle), the thermal efficiency cannot exceed 0.75–0.85. To reach this limit, it is necessary to increase the pressure differential by not less than one order of magnitude as compared with up-to-date liquid-propellant rocket engines.

The high values of the pressure P_{∞} are ensured by the fuel component feed system—a turbo-pump unit.

The value of P_{on} is proportional to the turbine specific power. It can be increased by increasing gas temperature at the turbine entry. In the modern liquid rocket engine, the gas temperature, T_{to} at the uncooled turbine entry amounts to 800–1000 K and at these conditions, the P_{on} reaches P_{on} = 20–30 MPa. Further increase of gas temperature requires turbine blade cooling. In such a case, the highest specific power can be obtained at single stage combustion of the whole fuel with an oxidizer-fuel ratio of $\alpha = 0.7–1$ [5].

The structure of such a liquid-propellant rocket engine is shown in Fig. 2. The cooled nozzles of the combustion chamber create a twisted annular gas flow as a stator of the turbine. The flow over the rotating blades (also cooled) has the total temperature T_{wo} which is less than T_{to} [13] because, at the relative motion with transfer tip speed U, only the gas relative speed is decelerated. The gasdynamic parameters of flow passage can be optimized to decrease T by 20–30% relative to T_{to}. This makes the problem of rotor blade cooling easier. In such a liquid-propellant rocket engine, working on O + kerosene, it is possible to attain the pressure P_{on} = 250–300 MPa (Fig. 3) at the optimum degree of gas expansion in the turbine and increase specific impulse by 15%. But with this, the very high pressures in the cycle will occur at the fuel and oxidizer pump exit (800–1000 MPa). At some reduction of π_{to} and decrease of P_{on} to 200–250 MPa, these maximum pressures in the engine feed system will be halved.

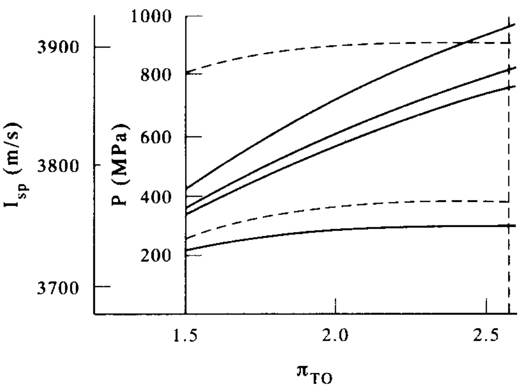

Fig. 3. Characteristics of liquid rocket engine with single-stage combustion. O_2 + Kerosene, $P_{on}/P_a = 5952$, $\alpha = 0.95$. 1, P_{on}; 2, I_{sp} ($P_a = P_H$); 3, P_c; 4, P_{po}; 5, P_{pf}; 6, I_{sp}.

The conclusion is as follows: the turbo-pump system of fuel feed in a limiting case of closed cycle liquid-propellant rocket engine with cooled blades can increase the gas total pressure approximately by one order of magnitude as compared with rocket engine without cooled blades.

Consider a fuel-feed system with a rotor unit (Fig. 4). It contains three elements: a combustion chamber, a pump, and a turbine. A prototype of such a system was tested in 1929 [3]. The combustion chambers are placed at the periphery of the moving rotor with an inclination to the meridian plane. The fuel is fed through the rotor channels which perform the functions of a centrifugal pump.

It is interesting to note that the liquid-propellant rocket engine with rotor unit and the hypersonic ramjet have similar combustion processes. In this process, the heat is added to the working fluid moving

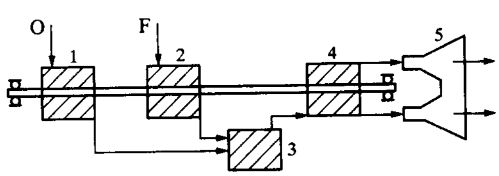

Fig. 2. Liquid rocket engine with single-stage combustion. 1, Oxidizer pump; 2, fuel pump; 3, combustion chamber; 4, turbine; 5, annular nozzle.

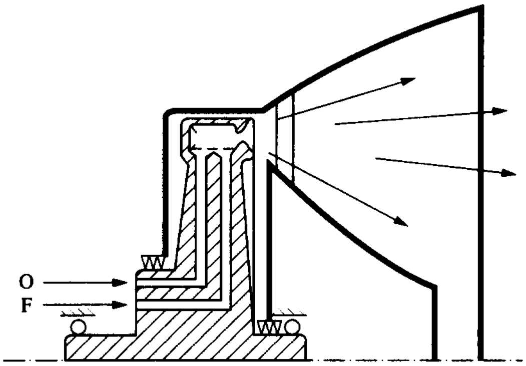

Fig. 4. Scheme of rotor liquid rocket engine.

with great velocity and having static pressure differing greatly from the total pressure. In this case, the specific heat fluxes in thermally stressed engine elements decrease at the expense of hot gas pressure and density decrease but not at the expense of temperature decrease. Pressure P_{on}, P_o maximum attainable, in rotor liquid rocket engine, depends on the fuel component pair density, tip speed and Mach number *Ma* at the rotating chamber nozzle exit (Fig. 5). At axial gas flow in engine annular exhaust nozzle $P_{on} \leqslant 300$ MPa.

The possibility of increasing η_t at the expense of decreasing P_a is connected with allowable engine overall dimensions. The dimensions of modern advanced liquid-propellant rocket engine are determined basically by the nozzle overall dimensions. Strict methods of optimal nozzle contouring[14–21] allow the most effective use of the overall dimensions and the decrease of engine mass. An unexpected result is obtained in a number of cases, namely: optimum nozzles with maximum thrust must have, internally, a shock wave at a given nozzle length.

The nozzles with a greater expansion degree must have a light thin shell. During the work of such nozzles, self-oscillations can occur (the shell panel flutter) leading to some loss of thrust and decrease in engine reliability. For the present, we possess no reliable methods of preventing this phenomenon and have no satisfactory theory of panel flutter[7].

The thermal efficiency can grow at a corresponding increase of the adiabatic exponent as it is in the nozzle of three-component liquid rocket engine. Such an engine begins to work on O_2 + kerosene(+H_2) and finish on $O_2 + H_2$. The effectiveness of a launch vehicle with a three-component liquid rocket engine is due to the improvement of mass/overall dimension characteristics of the rocket. The perfection of engine

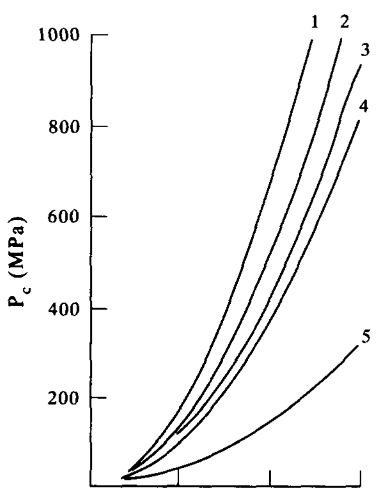

Fig. 5. Pressure in rotor combustion chamber. 1, H_2O_2 + Kerosene ($K_m = 7.39$); 2, O_2 + Kerosene (2.72); 3, $O_2 + CH_4$ (3.5); 4, $F_2 + H_2$ (13.0); 5, $O_2 + H_2$ (6.0).

internal processes is connected with thermal efficiency. At the change of fuel in the engine with fa = const at a constant P_{on} value, the value of γ increases and the pressure in the nozzle exit section decreases. In sum, this leads to the increase of η_t by 10% (Fig. 1).

3. PROBLEMS OF WORKING PROCESS IN THE HIGH-PRESSURE COMBUSTION CHAMBER

It can be shown that the reduced length of a combustion chamber depends on the pressure expressed in the form of $L_{red} = P_c^{-0.5}$. Under this condition, the number of pair collisions per one molecule of the combustion products during the time of its residence in the combustion chamber is constant. Therefore, at large values of P, a problem of mixture formation arises: because of the combustion chamber dimensions, the mass flow per surface unit increases and the fuel must be introduced through the whole surface of combustion chamber or through a greater part of it.

At the temperature $T = 3500$–4000 K and pressure $P_o > 100$ MPa, the molecular mean free path ($l = P_c^{-1}$) will be comparable with the size of molecules (3–5×10^{-10} m). In this case, a problem of achieving high combustion efficiency can occur due to the worsening of mixing and afterburning.

A great problem arises from the growth of heat fluxes. For their decrease, a method of boundary layer laminarization must be developed[16,17]. For the heat protection of the nozzle throat, the transpiration cooling seems to be indispensable.

4. REGENERATION AND TURBULENCE

Limiting differential pressure (see Fig. 1) restricts the possibility of liquid rocket engine effectiveness increase. This restriction can be removed by means of the exhaust gas heat regeneration.

In the liquid rocket engines, the combustion chamber walls are usually cooled by propellant components, which increase the enthalpy of the working fluid. This partially compensates the thrust friction losses. The compensation depends on the ratio of friction coefficient C_f and Stanton number St. According to the Reynolds analogy, this ratio depends only on the Prandtl number Pr:

$$\mathrm{St}/C_f = A \cdot \mathrm{Pr}^{-0.6} \qquad \mathrm{Pr} = \left(\frac{\mu C p}{\lambda}\right)_w .$$

For a turbulent boundary layer on the smooth and rough wall we have $A \approx 0.5$. For this value, the thrust losses are not compensated at the expense of heat regeneration. The compensation can be full or even excessive at $A > 0.75$–1.0. It is not known at present how to achieve this. The investigations with flow structure change in the wall and boundary layers and with the change at wall surface structure in the supersonic part of the nozzle are necessary.

In the experiment, the pressure increase on the nozzle wall at turbulent boundary layer is revealed (see [10] and other works) as compared with the calculated results for an ideal, non fluctuating flow. This is due to the normal (to the wall) component of turbulent stresses and leads to a thrust increase of up to 0.5–2% [9]. Let this increase be called the "thrust turbulent component". The possibility of increasing the turbulence level in the flow at the boundary between the flow core and the boundary layer with the purpose of thrust turbulent component increase must be studied. Solving this and similar problems opens a prospect for the increase of thermal efficiency without raising the expansion degree and overall dimensions of nozzle-only at the expense of exhaust gas thermal energy utilization.

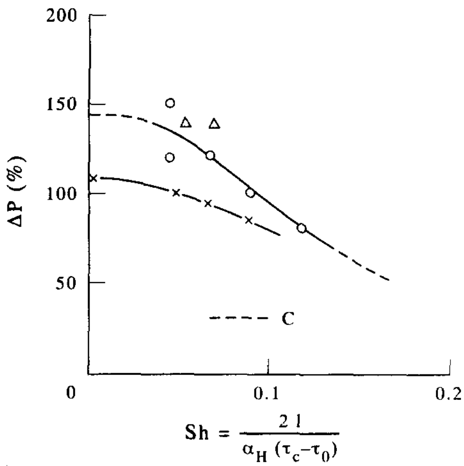

Fig. 6. Thrust increase in pulse ejector. c, Stationary ejector.

5. USE OF THE ENVIRONMENTAL MATTER FOR THE ENGINE ECONOMY INCREASE

The economy of a liquid-propellant engine can be raised using the environmental matter. Oxygen can be utilized as an oxidizer. For this, the air is liquefied in a heat exchanger at the expense of cryogenic fuel cooling capacity, and with the help of a pump is fed in the combustion chamber. This obtains a high value of η and I_{sp} (1200 s) for a liquid hydrogen engine. At high flight velocities ($M_H > 6$), the economy of such an engine is equal to that of an ordinary liquid-propellant rocket engine. The efficiency of engines with air liquefaction, and the prospects of their use, depend on the perfection of air condensation process and mass-overall dimension characteristics of the heat exchanger.

The thrust and specific impulse can be increased considerably with the help of an ejector working in a pulsed regime[11]. In the pulsed ejector process, a wave process of kinetic energy transfers from the high-speed portion of gas from the engine to the mass of environmental matter. This process can be realized using pulsed liquid-propellant rocket engine and a fixed ejector or using a stationary working engine and rotating ejector. Theoretically, the engine economy and thrust can be increased by a factor of 7 or 8, but at the present time the increase obtained is equal to 100–140% (Fig. 6).

In nozzles with great expansion degree, the thrust losses are great when working in the atmosphere of planets. To decrease the losses, the portion of nozzle, where the pressure is less than the atmospheric one, makes communicating with environmental matter through a channel or a ring slot[18] which results in the nozzle pressure equalling the atmospheric pressure and in the decrease of thrust losses due to the gas overexpansion (Fig. 7). The experimentally obtained decrease of these thrust losses is equal to 50%. The thrust loss decrease due to the overexpansion of gasses can compete with the one achieved with compound nozzles where the nozzle expansion degree varies stepwise.

Annular nozzles are shorter than round ones. But the four-chamber construction length of the RD-170 engine (Soviet Union) and the YE-73 engine (China) is also equal to half the single-chamber construction with the same thrust.

Some types of annular nozzles, plug nozzles and plate nozzles, possess good thrust characteristics when working in a medium with increased back pressure at take-off or landing. But in the flight—when the dynamic pressure of external flow $\rho_A V_A^2/2$ and exhaust jet $\rho_a W_a^2/2$ become equal—this quality is lost. The annular nozzle then works as a round nozzle with great overexpansion of flow. Therefore, annular nozzles will be applied in other cases, e.g. in liquid-propellant rotor rocket engines, retroengines and engines of other types where they naturally blend with the construction.

6. CONCLUSION

A possibility exists for an increase in the liquid rocket engine thermal efficiency up to 80–85% and specific impulse by 5–15% at the expense of an increase (up to 250 MPa for O_2 + kerosene) of total

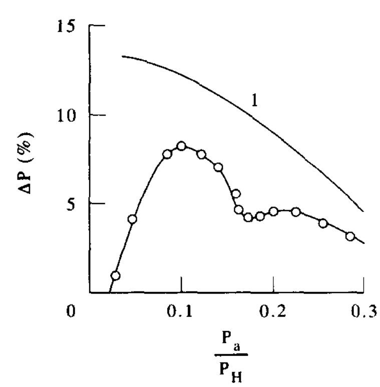

Fig. 7. Increase of thrust in a slotted nozzle. 1, Maximum increase of thrust (calc.).

pressure of the gas expanding in the jet nozzle. This can be realized in the liquid rocket engines of closed cycle with single-stage combustion and with expansion of the whole fuel in the turbine with cooled blades. The range of pressure from $P_{on} = 25$ to 250 MPa can be realized in the closed-cycle liquid rocket engine with a cooled turbine at increasing fraction of heat release in the gas generator.

In the rotor liquid-propellant engine the pressure can be raised too. For this, the rotor strength must be ensured at high rotation speed, a good combustion of fuel must be organized in the combustion chambers, rotating with great speed, and the heat protection of walls must be provided.

A further increase beyond the limiting values (0.8–0.85) is possible by means of the intensification of exhaust gas regeneration and the development of flow turbulization methods in the expanding portion of nozzle with the purpose to increase the thrust turbulent component.

The increase of η_t at the expense of pressure decrease at the nozzle exit section is limited by the growth of engine dimensions, the problem of thin-walled shell aeroelasticity and the thrust losses on the atmospheric leg of the flight. There are ways to overcome these problems, namely: the choice and contouring of optimum nozzles; the development of methods of preventing panel flutter; the use of slotted nozzles of variable geometry. All this requires the carrying out of additional investigations.

The utilization of planet atmosphere for the increase of thrust and I_{sp} using ejector devices is limited by the comparatively small flight speed ($M_H < 1$) and not more than $M_H = 6$–7 for the liquid-propellant rocket engine with the liquefaction of air by flow components.

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