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Advanced approach to the design of the propulsion systems propellant tanks hot pressurization systems

Primary focus is on propulsion systems propellant tanks hot pressurization systems. New approaches to pressurization working medium heating are investigated. The purpose of implementation is a pressurization system (PS) mass reduction, engine design and start-up simplification, system reliability improvement by further elimination of helium from the propellant tanks PS.

Introduction.

Nowadays, high-boiling-point fuels of kerosene type (РГ-1, T-1, T-6, sintine) and cryogenic oxidizer - liquid oxygen are obtaining the wide spread (and are planned to be used) as modern launch vehicles (LV) propulsion systems propellant, especially for the first stages. Apart from that, at the time being, the hypergolic propellants NT and UDMH are widely used in liquid rocket engines (LRE) (LRE "Proton", "Cyclone", LV Vega IV stage and others).

Problem description.

During 75 years of rocket technology history, various pressurization systems (PS) have been successfully used: evaporative (oxygen, nitrogen), flowing air (F-2), liquid and solid fuel generator, chemical, self-pressurizing (the oxidizer upper layer boiling in a tank) and others [1]. However, in the propellant supply systems of modern LVs, due to various reasons, the hot helium gas cylinders of PS were most widely used [2]. Suffice to mention the launch vehicles Zenith, Antares, Angara, Falcon 9.

Hot helium PS originate from the first stage of LV Saturn-V [3]. There they were used for pressurization of fuel tank. Helium was stored in cylinders placed in the lower part of the tank with liquid oxygen. At the start-up of the F-1 liquid-propellant engine, helium was fed to a regenerative heat exchanger located in the aft bay. He used regenerative generator gas after the turbine as a coolant. Its temperature was about 900K. It should be noted that until the shutdown, each of F-1 liquid rocket engines was not throttled. The propulsion system was throttled - at first four peripheral liquid rocket engines were switched off, then the central engine was turned off. In other words, the gas temperature after the turbine was high and constant during the LRE operation time, as well as the turbine gas flow rate itself, which allowed high PS characteristics obtaining.

In the mid-60s of the last century, a qualitative leap was made in the world engine development. The LREs (RD-107, RD-111, F-1, etc.), operated without turbine generator gas afterburning, were replaced by engines with generator gas afterburning (stage combustion cycle) - 11D58M, 11D33, NK-9, NK-33. This allowed noticeable improvement of their specific impulse [4]. However, at the same time, the design of the propulsion systems became much more complicated, their mass, time and scope of experimental test increased. It is not surprising that the propulsion systems of the new cycle became the most expensive part of the LV [5].

LRE designed in compliance with oxidizer-rich stage combustion cycle required a sufficient drop (almost twice) of the generator gas temperature in front of the turbine (from ~ 1400 K to ~ 840 K [4]). The reason for that is the limited materials resistance in high-temperature and high pressure oxidizing medium. Namely, the generator gas is a heat carrier for heating of the propellant components in tanks. Naturally, it results in a hot gas-cylinder PS performance deterioration. However, the existing methodology for PS development during transition to oxidizer-rich stage combustion cycle LRE has not undergone serious scientific comprehension in terms of new realities, opportunities and risks.

The situation in terms of helium heating has aggravated with a deep and long throttling of the liquid rocket engine. This is the imperative of the time being. Thus, the modern RD-191 is throttled to 30%. In this case, not only the generator gas (heat carrier) flow rate decreases, but its temperature as well. In order to raise somehow the helium temperature after the heat exchanger in the heating circuit, an oxidizing generator gas is used upstream of the turbine [6]. At the same time, the whole problem of materials ignition in a high-temperature medium and under high pressure is transferred to a heat exchanger. To exclude its ignition, a special system for its cooling (!) was introduced. It is clear that this is the limit of the old-fashion approach to helium heating.

Statement of the main material of the study.

Consider a fundamentally different coolant - reducing (neutral) combustion products of modern solid fuels. The reducing combustion products have a significantly higher heat capacity, and the problem of materials ignition is eliminated. In this case, the coolant flow rate does not depend on the liquid rocket engine operating conditions. It can be provided as it is required [7]. After the heat exchanger, this coolant should be directed to the pressurization of the fuel tank and to the booster pre-pump drive [8]. It should be especially noted that the use of a high-temperature neutral coolant requires use of the simplest contact heat exchangers.

At the time being, a pressurization system for propulsion system propellant tanks [8] is proposed, which includes fuel and oxidizer tanks, a fuel tank safety valve, compressed helium cylinders located in the upper tank of the oxidizer and connected by pressurization pipelines through automation units with a heat exchanger and a free volume of the oxidizer upper tank. Herewith, helium heat exchanger is located in the intertank bay and connected to a solid-fuel gas generator along the heat carrier, which produces it, and the output of said heat exchanger is connected to the free volume of the fuel tank. The undeniable advantages of this solution are as follows: any desired temperature of the coolant, its independence on the propulsion system operation modes, and the reduction of helium cylinders number. Last one results from the higher temperature of helium for oxidizer tank pressurization and reduction of the helium final pressure in the cylinders due to the resistance decrease of the shorter pressurization lines.

However, even in this case, the helium pressurization systems have significant disadvantages:

- low efficiency, which consists of the required large mass of the working medium filling system and its storage system - compressed helium for pressurization the oxidizer tank. For example, the best titanium cylinder from the alloy BT-6 with an internal volume of 132 l weighs about 45 kg, takes a volume of 140 l in an oxidizer tank, from which only up to 8.5 kg helium is used;

- a significant complication and rise in price of the launch pad. For example, the helium supply system for Zenith LV launch contains 1266 high pressure cylinders (37.5- 40.5 MPa) 500 l each, more than 3000 units of pneumatic instruments, and the length of pipelines is tens of kilometers [9].

- at the time being, helium pneumohydraulic systems are the most expensive systems of the launch vehicles after the LRE [5].

Due to the wide use of helium all over the world, one may have an impression of increased reliability of helium systems. However, this is not the case. Here are examples of helium systems unreliability only from the very recent past [10]:

- August 7, 2012, abnormal completion of the Proton-M LV mission on satellites injection due to the clogging of the "Breeze-M" booster fuel tank helium pressurization pipeline. The payload was lost;

- October 26, 2012, the launch team had to postpone the launch of the KSLV-1 due to the leakage of "cold" helium during filling of the pressurization system cylinders of the first stage, made in Russia (analogue of the Angara LV I stage). LV was removed from the launch pad for troubleshooting;

- April 17, 2014, the Dragon spacecraft launch to the ISS was abolished. According to the press service of NASA, the launch date postponement was caused by helium leak in the propulsion system of the Falcon 9 LV first stage;

- June 27, 2014, the first launch of the light class Angara LV, which has been under development for more than 19 years, was abolished. The reason for that was a drop of helium pressure in the gas cylinder, installed in the discharge line of liquid oxygen. To fix the problem at the Plyesetsk (Russia), it was necessary to remove Angara from launch pad;

- June 28, 2015, a helium cylinder was torn off from oxidizer tank of the second stage of the Falcon 9 LV during 1st stage propulsion system operation. Useful hardships for the ISS, the multiple Dragon ship were lost;

- September 1, 2016, 2 days prior to launch, in preparation for traditional static fire, an explosion at the launch site SLC-40 at Cape Canaveral abruptly destroyed both the launch vehicle and the payload, the telecommunication satellite Amos-6. In the course of the investigation, the immediate location of the incident was identified - one of three cylinders for compressed helium (composite overwrapped pressure vessel, COPV), which is used to provide a working pressure in a fuel tank with liquid oxygen of the second stage.

The question arises: can we abandon helium in modern pressurization systems? Here it should be noted at once that the last generations of intercontinental ballistic missiles of Yuzhnoye SDO were perfectly dispensed with helium pressurization and automation units [11]. Let's consider the pressurization system of propellant tanks, which allows to decrease the mass of PS, increase the reliability of the system by complete elimination of helium from the pressurization system of propellant tanks, and also to achieve simplification of the propulsion system and the start-up process [12].

The task is solved by the fact that along the oxidizer tank pressurization the heat exchanger is connected at its inlet to the oxidizer pump outlet, and at its outlet to the free volume of the oxidizer tank. Another feature is that along the oxidizer tank pressurization, there is a fuel gas generator, installed between the heat exchanger and the free volume of the oxidizer tank, and having a neutral balance, for example, based on sodium azide, or a reducing balance, for example, based on the starting pallets ПГ-2.

The most simple and developed (and, therefore, cheap) solid fuel compositions burn in a sustained way at temperatures of 1270 ÷ 1800 K, with that, due to the presence of hydrogen, water vapor, methane in their combustion products, they have a very good working capacity (~ 500 J / kg · deg). With a huge temperature difference between the heat carrier and the oxygen at the outlet of the heat exchanger (not 100 K, as in modern engines with oxidizer-rich generator gas afterburning), the pressure loss along the path of the heat exchanger (its mass reduction) can be minimized with a simple design.

Conclusions.

The proposed hot pressurization systems provide an opportunity to conduct autonomous development test of the pressurization system under substantially normal conditions with an available source of normal coolant, which allows accurate and timely corrections to be made in the parameters of the pressurization system.

The trade-off analysis carried out by the authors in relation to the proposed solutions, according to the proven method of the system parameters calculation, shows the following: an increase in the average mass oxygen temperature at the inlet to the oxidizer tank up to 1300 K, ~ 2 times reduces the mass of pressurization system comparatively to gas cylinder. For the two-staged Zenith LV, the transition to the proposed pressurization system only at the 1st stage leads to the possibility of payload mass increasing (satellite mass, for example) by ~ 115 kg. With that, conditions are created for the abandonment of the cold helium systems at the launch pad.

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