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Hybrid rocket engines: The benefits and prospects

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Abstract

The review of research avenues for working processes (ignition, combustion, etc.) in hybrid rocket engines (HRE), for thermal protection and power characteristics is stated. Based on conducted design and exploratory studies and design developments, the reasonable fields of application of HRE to launch systems for orbit insertion of payload are determined.

Keywords: Hybrid rocket engines; Solid fuel; Oxidizer

1. Introduction

Among the most promising systems using chemical rocket propellant and possessing practically all merits of up-to-date LREs and SRMs are hybrid rocket engines running on solid fuel and liquid or gaseous oxidizer. In their specific power and mass characteristics, they are intermediate between LRE on cryogenic propellants and SRM, have noncooled flow passages (owing to the use of present-day thermal-protective materials), permit adjustment of the value of thrust and steering of the thrust vector, multiple firing, as well as long-term stay in space conditions. The use of such oxidizers as liquid oxygen, hydrogen peroxide and combustibles (rubber, polyethylene and others) ensures ecological purity of both propellants in themselves and combustion thereof. Among HRE merits are also high stability of operational process, the absence of abnormal combustion and, as a consequence higher reliability, the capability of protection of nozzle and chamber walls through the use of propellants.

One of base peculiarities should be noted: HRE allow unification of principal assemblies and units to a greater extent than LRE and SRM do.

Having regard to a relatively lesser cost of propellants and HRE design simplicity as compared to LRE and SRM, the application of hybrid engines would lead to a lesser cost of series production of HRE and propulsion systems (PS) on its basis, than that of PS of other types.

The use of HRE allows raising reliability and diminishing specific cost of payload delivery to space orbit at the expense of:

1. Curtailment of PS development periods (as compared to development of similar PS with LRE and SRM) from 4…5 years down to 6…10 months when using refined assemblies and units.
2. Curtailment to HRE fabrication cost by a factor of one and a half to two as compared with LRE of the same class, due to the application of thermal protective carbon-containing composite materials (carbon-carbon composite materials and carbon-ceramics composite materials) to nozzle unit, of combustion chamber design (the "cocoon" type) on the basis of scientific and engineering work done in anticipation for creation of SRM.
3. Curtailment of the launching cost due to:
   - diminishing expenditures for fabrication of materiel;
   - two – to threefold decrease of a time of pre-launch procedure;
   - decreasing operating costs by 40 to 50%;
   - lowering the cost of fire-and-explosion safety systems;

and so on.

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Nomenclature

CC	combustion chamber	LV	launch vehicle
CM	composite materials	MHD	magnetohydrodynamic
DHRE	demonstrator hybrid rocket engine	NPPS	nuclear power propulsion system
HC	hydrocarbon	PPS	power propulsion system
HF	high frequency	PS	propulsion system
HRE	hybrid rocket engine	SC	spacecraft
LRE	liquid rocket engine	SRM	solid rocket motor

2. Research into basic processes in HRE

Hybrid propellants shall meet general requirements imposed upon chemical propellants and, consequently, include components that have the greatest initial density, provide the chemical interaction with the greatest heat release and yield combustion products with a possibly lesser molecular weight.

Propellants based on polymer HC fuels (synthetic rubbers) with liquid oxygen or concentrated hydrogen peroxide are most simple, highly mastered and, at the same time, efficient enough. For raising power characteristics and stabilization of combustion, metals (or metals hydrides) can be added to solid components. These may be aluminium, boron, lithium. The process of grain burning in HRE is closely related to the technique of oxidizer supply to combustion chamber (CC) and flow organization. For maintaining optimum internal ballistic parameters and ensuring stable burning at a high completeness of combustion, it is necessary to change the mixture ratio at the expense of regulation of the oxidizer flow rate.

Solving the problem of determination of optimum oxidizer flows is based on combined calculation of thermodynamic parameters of combustion products mixture and burning processes. Calculation of burning is based on models of thermochemical decomposition of propellants and diffusion burning within the front of propellants mixing (Fig. 1). The calculation procedure incorporates solving equations of the turbulent boundary layer for binary chemically-reacting mixture. Correctness of such an approach was justified by numerous comparisons of calculated results with experimental data that were obtained both for subscale HREs and on special facilities. In experiments, principal regularities of burning of polymeric materials (polymethyl methacrylate, polyethylene, rubbers, etc.) in oxygen and hydrogen peroxide were researched into. In particular, experiments and calculations reveal that, because of variations in heat exchange conditions, temperature and concentration of propellants with the channel length, the rate of propellant burning and, consequently, erosion of cylindrical channel is nonmonotone function of coordinate x. Shown in Fig. 2 is profile of cylindrical channel of the grain after tests.

For research into hybrid-propellant burning processes and resistance of thermal-protective materials of combustion chamber and nozzle unit, a 30 kN thrust demonstrator HRE (DHRE) and its bench variant (Figs. 3, 4) allowing a fast change of PS design according to the goals of research were developed and made. The finishing development of DHRE ignition system and experimental investigations of burning rate of HC solid fuels were conducted with using oxygen as oxidizer.

As fuel, grains of complex, "star"-like and "carpet" shape, with a cylindrical channel, fabricated out of rubber, polyethylene etc., were used (Fig. 5).

Some results of subscale experiments on determination of the burning rate of butyl rubber subject to the density of O_2 oxidizer current at the entry of solid-fuel grain channel (Fig. 6) reveal the burning rate of solid fuel increases in proportion to the oxidizer current density. For reference purposes, shown in the same figure are experimental data on the burning rate of $HTPB + O_2$ propellants (red squares) taken from [1] (for colors see the web version of this article).

One of serious problems of development of the efficient HRE that can be in successful competition with rocket engines of other types consists in the provision of high completeness

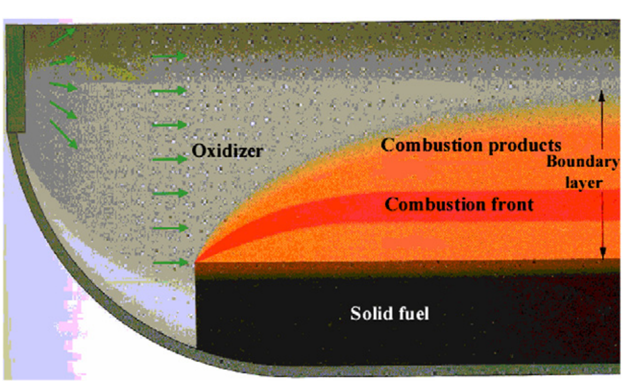

Fig. 1. The scheme of burning.

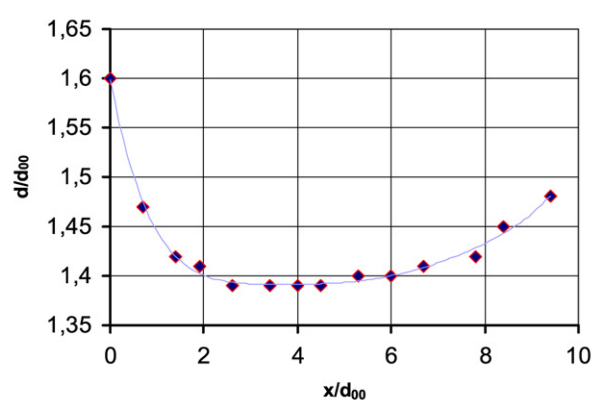

Fig. 2.

Fig. 3. DHRE.

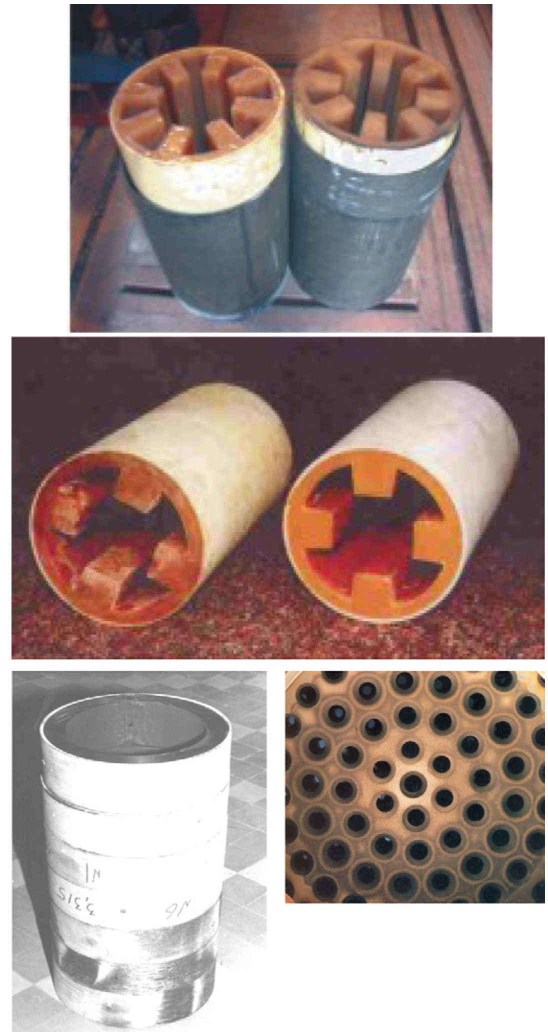

Fig. 5. “Star”-like, “carpet” and cylinder grains.

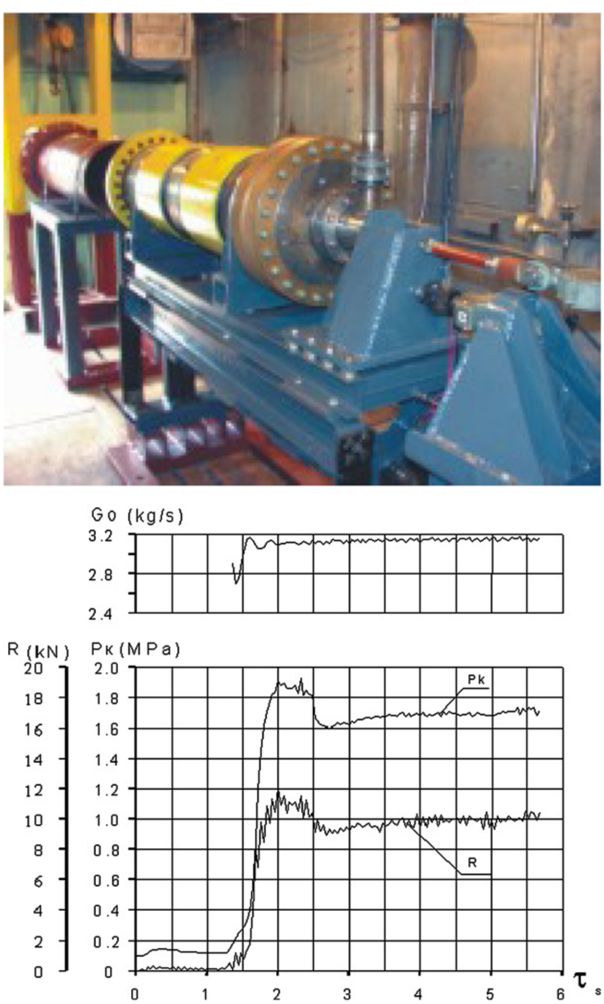

Fig. 4. The bench HRE of 10 to 30 kN thrust.

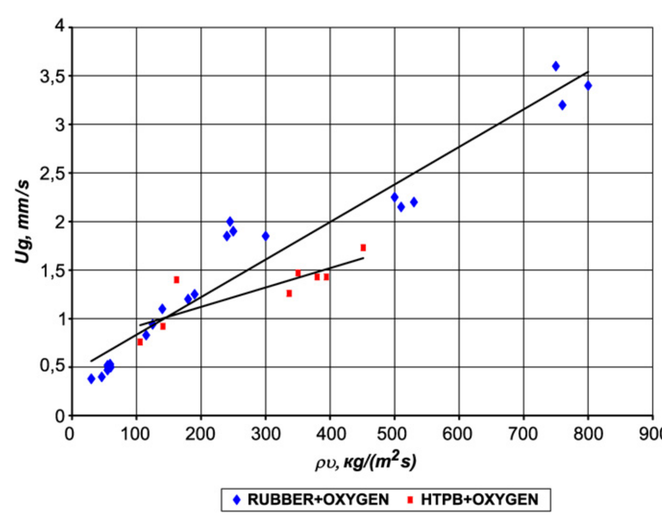

Fig. 6. Burning rate versus current density for propellant of the type “rubber”.

of propellant combustion. For this purpose, catalysts of burning are used, and propellant composition is optimized, as well as mixers and afterburning chambers are applied.

Due to specific peculiarities of hybrid-propellants burning, the conventional cycle of HRE cannot result in a complete propellant combustion, because fuel and oxidizer have no time to mix with each other in the channel. The combustion completeness η depends on geometric characteristics of the channel: length, normalized extension, diameter, as well as on the kind of propellant, combustion products current density ρv in CC, the oxidizer-to-fuel ratio α, etc.

The analysis of the available experimental data reveals that when using hydrogen peroxide and oxygen as oxidizers, a combustion completeness higher than 0.9 and 0.95, respectively, cannot be obtained. The availability of the minimum of the combustion completeness η near the stoichiometric mixture ratio α being optimum as to the impulse, leads to the necessity to search in this area for ways to increase the combustion completeness. When using special devices, power characteristics of the engine, the combustion completeness η can be increased for propellants on the basis of hydrogen peroxide up to 0.93, for those based on oxygen – up to 0.98.

At present, a sufficiently large scope of experimental data has been accumulated on burning rates of different hybrid propellants. Each propellant has its own chemical composition, so comparison of burning rates of different propellants allows one to make some conclusions on capability of adjustment and increase of the burning rate by chemical methods.

One of straightforward methods of chemical increase of the burning rate is the use of propellant systems with a higher combustion temperature. The changeover from "polymer-hydrogen peroxide" propellants to "polymer-oxygen" propellants results in the increase of burning rates. A more impressive effect was obtained, for example, when using the $LiH + ClH_3$ propellants. If one and the same oxidizer is used, the changeover from sublimating fuels to melting ones (naphthalene, decaborane, paraffines) turns to be effective affording the burning rate to be improved practically by an order.

The agreement between the rate of gasification and the total heat of phase transformation is verified in experiments on polymer fuels. Reduction of the amount of energy, which is necessary to gasify a solid fuel, can be attained using materials with a low specific heat capacity, latent heat of phase change and by introduction of combustible admixtures to the fuel composition which decompose in the condensed phase releasing heat, catalysts, etc.

Investigation into the effect resulting from addition of solid oxidizer to the fuel composition was made. The oxidizers, which were studied, are arranged in the increasing order of the effectiveness of influence on the burning rate: $C_7H_3NClO_4$, NH_4ClO_4, $KClO_4$, $NaClO_4$, $KClO_3$.

Investigations were made into capability of increasing burning rates by addition of organic perchlorates to solid fuel. Dipropylamine perchlorate was found to be the most effective. If 20% of it are added, the burning rate can be twice as high, with a minimum deterioration of propellant thermodynamic characteristics.

Addition of metals – Al, Mg and their alloys to the fuel results in 20 ... 30% increase of burning rate.

The increase of burning rates through addition of catalysts ($CuCl_2$, $K_2Cr_2O_7$, ferrocene) amounts to 5 ... 25%.

There are also known investigations into increasing the burning rate of hybrid propellants by adding heat-conducting elements and granules of fulminating to the composition of solid fuel.

Among the chemical methods of increasing burning rates covered above, addition of solid oxidizers and metals found the most extensive practical applications at present.

In spite of a large scope of experimental data, the difference in experimental conditions and a great scatter allow only qualitative conclusions to be made which are reduced to the following: chemical methods can yield a maximum of 2 ... 10-fold increase of burning rate.

All the above results on burning of hybrid propellants apply to stable burning, for calculation of which, as experimental results show, the model of diffusion burning can be successfully used.

As compared with the case of stable burning, the process of grain ignition is believed to be more complicated, because it depends, in many respects, on the way of ignition and igniter design. Experiments reveal that with forced ignition of grain, a delay in ignition is evident. The delay time is dependent on the initiator thermal power and oxidizer current density.

The pressure influence on HRE power characteristics may be exemplified by an estimated ballistic efficiency of the application of the "O_2 + rubber" HRE to the makeup of a light-weight LV: at a launching mass of ∼65 t, a payload mass comprises 1.5 t ($H_{cir} = 200$ km – the orbital altitude, $i = 51°$ – the orbital inclination). Suggesting that the influence of HRE combustion chamber pressure on HRE power and mass characteristics may be essential, three levels of pressure, $P_k = 3.0$; 6.0 and 10.0 MPa were used for performing calculations. In so doing, the values of thrust of rocket stages' PS were accepted as 80 and 20 tf, and the operational time – 120 and 350 s, accordingly, for the first and second stages (Table 1).

In spite of an increase in specific impulse with growing pressure in the combustion chamber P_k, the development of engines operating at $P_k > 10$ MPa requires additional analysis. So, if the payload mass launched into orbit is taken to be the efficiency criterion of the engine as part of LV, then, in terms of ballistic efficiency and increasing the engine weight (due to an increase in "dry" mass) with growing P_k, the optimum

Table 1
Energy characteristics of engines

Stage	I			II		
Thrust (tf)	80			20		
Operational time (s)	120			350		
Pressure at nozzle exit (MPa)	0.4			0.08		
Combustion chamber pressure (MPa)	3	6	10	3	6	10
Propellant consumption (kg/s)	308	281	266	56	54	53

Table 2
Losses of HRE specific impulse

Propellant pairs: fuel + oxidizer	Thermodynamic specific impulse, J_{sp}^T	Total losses of specific impulse, ξ (%)
Rubber + LO_2	362 ... 364	2.36 ... 3.68
Rubber + H_2O_2	307 ... 345	1.87 ... 3.3
6% rubber + 94% $AlH_3 + H_2O_2$	348 ... 402	7 ... 10

values of pressure turn out to fall within the range between 3 and 6 MPa. It has been shown that for initial data taken (Table 1), an increase of P_k up to 10 MPa results in a decrease in the payload mass by $\sim$8% as compared to the maximum value (at $P_k \sim 6$ MPa). It should be noted that the attainment of the maximum ballistic efficiency of LV with HRE at relatively moderate pressures renders the HREs very attractive from the viewpoint of the construction reliability, because the range of pressures P_k between 3 and 6 MPa has been successfully worked through with SRMs whose cases and nozzle units are the same as in HREs. The HRE actual specific impulse depends on total losses which, for the case of metal-free propellants, exceed mere insignificantly corresponding values inherent in LRE, being thereat essentially lower than in SRM having identical mass-and-dimension characteristics.

In the case of metallized propellants, levels of total losses increase because of additional losses by two-phase flow. However, if particles are available, the propellant combustion completeness, η, has to grow, and if an additive Al ($Z = 10 \dots 15\%$) raises η from 0.9 to 0.95, this may be advantageous, as, in doing so, losses by two-phase flow decrease by $1 \dots 2\%$.

Besides, at the level of condensed phase $2 \dots 10\%$, there are no problems of instability, no acoustic oscillations, HF oscillations in CC, especially at large extensions, damp out. Presented in Table 2 are results of parametric calculations of HRE specific impulse losses within the range of parameters variations:

$R_{sp} = 4 \dots 120$ tf – the specific thrust;

$P_k = 3 \dots 10$ MPa – the pressure in combustion chamber;

$D_a = 3.5 \dots 10$ – the area expansion ratio;

($d_a = d_a/d_k$, where d_a – the nozzle exit diameter, d_k – the nozzle throat diameter).

3. Thermal-protective materials in HRE

One of the ways to essentially raise power-and-mass characteristics of HRE is the application of composite materials (CM) for manufacture of CC and nozzle units. In doing so, designs and technologies approved widely during designing and finishing development of SRMs can be successfully used. However it should be considered the operating conditions of thermal-protective materials in HRE differ from those in SRM. At comparable heat fluxes, temperatures and loads, a mean distinctive feature of HRE operating conditions is a great concentration of oxygen-containing combustion products. So, for the provision of nozzle operability and minimization of ablations of thermal protection, enhanced requirements to oxidizing resistance of the latter are imposed upon. To satisfy these requirements, along with conventional carbon-graphite composite materials and carbon-phenolics, carbon-ceramics composite materials, including siliconized ones, as well as various methods of active thermal protection are considered for the use thereof in HRE.

For determination of resistance and ablation rate of composite materials in oxidizer flows, design and experimental investigations were carried out, fire tests of various material samples were conducted for subscale HREs too. Test results allow performing more precise definition of mechanisms of the interaction between oxidizer flows and materials of different structure and composition, determination of necessary empiric constants describing physical and chemical processes at surface and within CM. Among these are the constants of the Arrhenius law and the constant of reaction order, that describe kinetics of endothermic oxidation of carbon frame and carbon or carbide matrix of material, heat effects of reactions or phase transitions, including formation and destruction (evaporation) of oxide films arising with heating of carbon-ceramics materials.

In calculations of thermal-protection parameters, the application package is used. The package is based on procedures of conjugate calculation of parameters of heat and mass exchange, instable thermal state of members and chemical kinetics processes of thermochemical transformations in composites and of their surface oxidation. At relatively moderate operating times of HRE (up to 100 s), as thermal-protective coatings, carbon-carbon CM may be used for coatings in the region of throat sections, carbon- and glass-phenolics – for nozzles of large area expansion ratios. For HRE at a long-term or cyclic operation, carbon-ceramics CM can be recommended.

If stringent requirements are placed upon stability of HRE passage sections and nozzle contour, active methods of thermal protection, including those on the basis of low-temperature films out of neutral or weakly acid products of decomposition of dual-propellant grains may be considered as alternative. At a consumption of decomposition products equal to $0.5 \dots 1.5\%$ of engine total consumption in terms of their afterburning in the near-wall layer, a decrease in the oxidizing potential of combustion products by a factor of three to five and in an equilibrium wall temperature at the throat section region by 300 to 500 K is possible. Both of factors allow it to essentially reduce the ablation rate of carbon-graphite materials from $0.1 \dots 0.2$ mm/s (with the absence of film) down to $0.01 \dots 0.015$ mm/s. A certain increase in specific impulse losses, conditioned by the influence of film is fully compensated for by the absence of losses by erosion of the nozzle throat and an associated decrease in the effective area expansion ratio. For nozzles of large area expansion ratio, the losses by film are less by a factor of three to five than losses by nozzle contour distortion.

4. The fields of application of HRE

HRE is one of most advanced systems using chemical rocket propellant that, in addition to the possibility to use simple, cheap, safe and reliable engines in launch vehicles of various classes, can be suited for fulfillment of other problems facing rocket space technology:

- orbital transfer of satellite;
- alteration of orbital eccentricity;
- closing manoeuvre in orbit;
- motion and control of space vehicles and spacecraft (including periods from 7 to 15 years);
- long-term flight of interplanetary space vehicles;
- atmospheric reentry and braking of space vehicle.

Now creation of hybrid rocket engines with high power and dimension characteristics does not result in great difficulties, and, practically, came to the stage of research and development work and flight tests, for example, at leading design firms of USA.

Project and design studies and comparative estimates of efficiency of using hybrid engines integral with PS of various launching systems revealed that the following areas of application of HRE may be promising:

- mobile rocket space land-or air-based complexes of light-weight class (for delivery of payload 0.1 to 1.5 t into 180...300 km near-earth orbits);
- special booster providing a delivery of SC with radiation waste into radiation-safe orbit around the Sun;
- transport and technical maintenance of space objects with using reusable interorbital towing vehicles, including PS for lunar manned rocket, operating on propellants out of regolith; PS of acceleration/deceleration unit – on propellants produced from atmosphere and surface soil of Mars; for transfer of cargo and crew from near-Mars orbit to surface and back:
- power propulsion system (PPS) of SC on the basis of MHD generator on hybrid propellant for cleaning of near-earth space from “space junk” (objects of 1 to 10 cm dimensions) in 300–800 km orbits, where, for radiation safety reasons, application of NPPS is not permitted;
- launching boosters;
- hybrid boosters.

Application of HRE is also effective for suborbital flights in systems with air start.

Integration of HRE in LV with cruise oxygen LREs is believed to be very promising. On realization of such design and arrangement schemes of zero stages and boosters, the problems of the launching power-to-weight ratio of LV can be efficiently solved (in particular, at the initial time of I stage operation at LV launching).

For solving interorbital transportation problems, when a long-term stay of high-energy cryogenic LREs in space is highly conjectural, HREs on hydrogen peroxide + rubber can be successfully used for towing vehicles. Estimates of technological state of these HREs and calculations of stability of hybrid propellants characteristics reveal a time of their stay in space may amount to a few years without degradation of power and operating characteristics.

5. Conclusions

1. Experimental and designed and theoretical investigations into main operating processes and thermal protection in hybrid engines reveal the feasibility to make HRE with high power and mass and operating characteristics.
2. Application of HRE to cruise stages of light-weight launch vehicles, boosters and interorbital towing vehicles, as well as to launching boosters of LV with cruise oxygen LREs of various classes is economically justified.

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