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Payload Evaluation of a Tripropellant Carrier Rocket

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The scheme of a tripropellant engine is examined for use on the first stage of the carrier rocket (liquid oxygen + kerosene + LH_2). The engine includes four chambers and one turbopump. The turbine drive gas is bled off from the wall boundary layer in the supersonic part of the nozzle and reinjected into the nozzle downstream. It is shown that the use of the tripropellant engine with a 30% addition of hydrogen increases the payload mass by 10% for a 200-ton two-stage vehicle. The arrangements to inject the three components into the combustion chamber, as well as bleeding and reinjecting the turbine drive gas from/to nozzle, increases the complexity of the chamber head and the nozzle, respectively.

Nomenclature

c^* = characteristic velocity, m/s
d = diameter, m
$\bar{F}$ = area ratio
g_0 = 9.81 m/s^2
I = impulse, m/s
k = oxidizer/fuel ratio
L = characteristic length, m
m = mass, kg
$\dot{m}$ = mass flow rate, kg/s
N = quantity
P = thrust, kN
p = pressure, Pa
q = specific consumption, s/m
R = gas constant, J/kg/K
$\bar{S}$ = relative surface area
T = temperature, K
t = time, s
V = velocity, m/s
α = equivalence ratio
γ = mass per unit area
ΔV = losses of velocity, m/s
ε = expansion ratio
θ = pitch angle, nozzle angle
κ = ratio of specific heats
ρ = density
φ = loss coefficient

Subscripts

aer = aerodynamical
b = booster
cav = cavitation
ch = chamber
cr = critical
cstr = construction
ctrl = control
cyl = cylindrical section
eng = engine
ent = entry
ex = exit
f = fuel
fin = finish
gg = gas generator
gp = pressurized gas
gr = gravitational
gv = gas volume
id = ideal
ker = kerosene
liq = liquid
m = mass
n = nozzle
ox = oxidizer, oxygen
p = propellant
pm = payload mass
ps = pressurization system
res = reserve
rp = reserve of propellant
sat = saturated
sp = specific
sub = subsonic section
sup = supersonic section
t = tank
th = theoretical
tp = turbopump
v = vacuum
0 = initial
I = first stage
II = second stage

Introduction

TWO desirable characteristics of rocket propellants are high-density and high-energy content. Unfortunately, these two characteristics usually do not come together in the propellants used in practice. The general trend is that high-density propellants have low-energy contents, and high-energy propellants have low density. A common practice in the design of carrier rockets is to use a high-density propellant, e.g., liquid oxygen (LOX)/kerosene (ker), in the first stage and a high-energy propellant, e.g., LOX/LH_2, for the upper stages. This

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combination provides a good compromise between overall performance, size, and liftoff mass of the rocket.

Schemes of tripropellant engines have been proposed to combine the benefits of the density properties of the hydrocarbon fuels and the energetic characteristics of hydrogen. In the U.S. these studies were directed for use in single-stage-to-orbit (SSTO) vehicles to reduce the dry mass. Salkeld[1,2] carried out calculations for a dual-fuel, dual-expander rocket engine for reusable launch vehicles. Christensen and Kent[3] discussed advanced rocket engine concepts for applications to reusable and expendable SSTO vehicles. Rocket engine schemes designed to utilize the mixed-mode principle are also discussed in Ref. 3.

This paper presents the calculations for the use of a tripropellant engine in a conventional staged carrier rocket. It is shown that the rocket payload performance is increased with the use of tripropellant engines (hydrocarbon + hydrogen + oxygen) in the first stage, although at the expense of more complex engine design. The amount of hydrogen addition depends on the mass characteristics of the tanks and propulsion units. The payload mass is used to optimize the hydrogen addition, taking into account the average density of the propellants and the specific impulse of the engine.

Tripropellant Engine in the First Stage

The scheme of the tripropellant engine for the first stage, with extraction of the drive gas for the turbine from the supersonic nozzle is shown in Fig. 1.[4] The engine contains four chambers and one main turbopump with a boost turbopump on the oxygen line. The main turbopump comprises the turbine and three pumps: single-stage pumps for oxygen and kerosene and a two-stage hydrogen pump. The active turbine works with drive gas extracted from all nozzles. The turbine exhaust gas is reinjected downstream into the nozzle, and it is used for thrust vector control. A solid-propellant cartridge is used to start and speed up the turbopump unit. Thrust throttling is provided by a controlled bypass of the turbine drive gas. The mixture ratio in the chambers is controlled by changing the mass flow of hydrogen, which is also used for cooling the chamber. All injectors are placed on an alternate concentric pattern: four rings of two-component kerosene–oxygen swirl injectors (128 bicomponent swirl injector elements, each one with an oxygen mass flow of 0.88 kg/s and kerosene mass flow of 0.128 kg/s), and three rings of single-component hydrogen injectors (96 single-component swirl injector elements, each one with a mass flow of 0.051 kg/s). Hydrogen cooling films are injected from the chamber wall at two places: near the head of the chamber (1 kg/s) and at the end of the cylindrical part of the chamber (2 kg/s). The injected hydrogen cooling sheet guarantees the reduction of the mass ratio and a sweet mixture in the wall boundary layer. Each chamber has a nozzle extension that permits the increase of the area ratio from 40 to 80 (one step change).

The combustion chamber and thrust nozzle are cooled by the hydrogen that is injected at the nozzle exit and flows upward. One part of the hydrogen is used for film cooling. The other part of the hydrogen is injected into the main flow through the injector plate along with the other propellants.

The injector plate has three labyrinth cavities from which the propellants are distributed to the injector elements. The hydrogen mass flow enters from the cavity closest to the combustion face. The kerosene enters from the top dome through a toroidal manifold, and the LOX enters from the cavity in the middle, also through a toroidal manifold.

The oxygen tank is pressurized with helium that is stored in cylinders located inside the LOX. Before being injected into

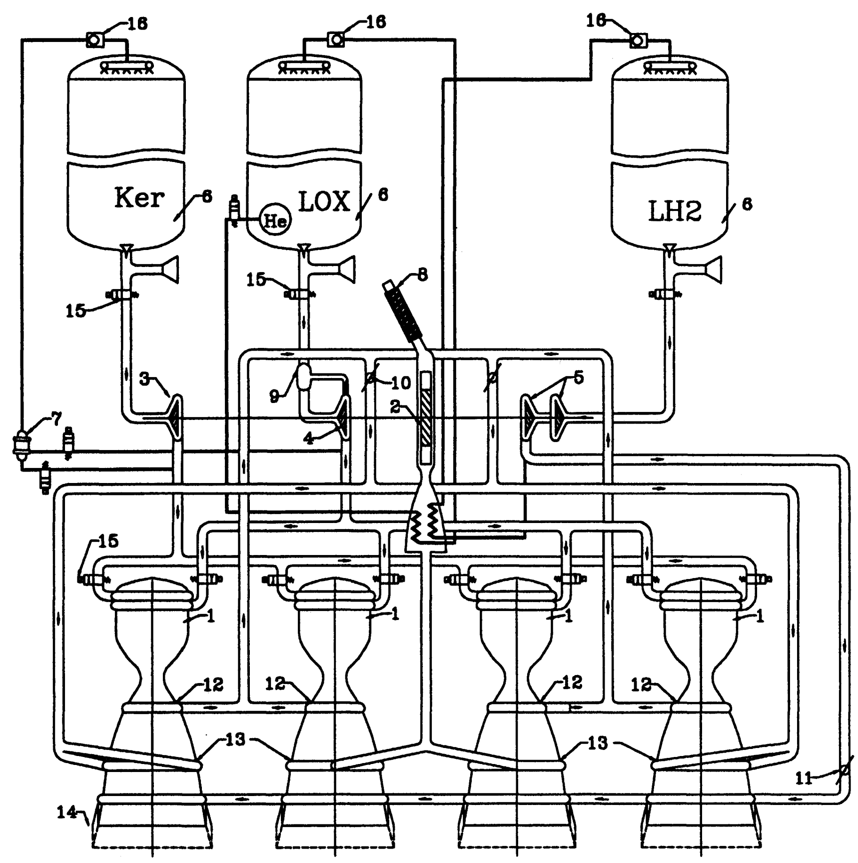

Fig. 1 Scheme of tripropellant LRE propulsion unit. 1, chamber; 2, turbine; 3, kerosene pump; 4, LOX pump; 5, hydrogen pump; 6, tank; 7, gas generator for pressurization; 8, solid propellant starter; 9, booster turbopump; 10, thrust regulator; 11, mass ratio regulator; 12, arrangement for extraction of turbine drive gas; 13, arrangement for nozzle reinjection of turbine exhaust; 14, nozzle extension; 15, pneumohydraulic valve; and 16, check valve.

Table 1 Propellant efficiency for different contents of hydrogen in the fuel[a]

% H_2	$k_{m,0}$	k_m (α = 0.9)	ρ_p (α = 0.9), kg/m^3	I_{sp}, m/s	ρ_p^c	$I_{sp}\rho_p^c$
0	3.442	3.098	1033	3650	56.54	206,371
20	4.343	3.903	990	3970	55.16	218,985
30	4.8	4.32	966	4080	54.38	221,870
50	5.714	5.143	902	4210	52.25	219,972

[a] $k_{m,0}$ is the stoichiometric ratio.

the tank, the helium pressurant is heated in a heat exchanger located at the turbine exhaust. The hydrogen tank is pressurized with gaseous hydrogen that is bled from the main pressurized feed line and heated in a heat exchanger also located in the turbine exhaust. The kerosene tank is pressurized with the combustion products of a dedicated LOX/ker gas generator.

The hydrogen mass ratio in the fuel was calculated, using as criterion for optimization, the maximization of the product of specific impulse and a power function of the density[5] $I_{sp}\rho_p^c$, where c is given by the formula:

$$c = \frac{m_p/m_0}{\ell n[1/(1 - m_p/m_0)]} \tag{1}$$

This criterion is very useful for the payload evaluation of the propellants in the early design stages of a rocket, when the trajectory parameters and mass characteristics are not yet well known. It takes into account not only the specific impulse of the engines but also the density of the propellants that determine, to a great extent, the size of the rocket. The maximum of this parameter is a smooth function of mass characteristics and trajectory parameters of the rocket; therefore, its value will not change appreciably as the calculation of these data is refined.

A sample calculation was done for a rocket using a tripropellant engine [O_2 + (ker) + H_2] with initial mass, m_0 = 200 ton, propellant mass m_p = 140 ton, and c = 0.5814. The results are shown in the Table 1.

A thermodynamic and transport properties code, named *Astra*, described in Ref. 6, was used to calculate the specific impulse of the engine. The propellant density is given by the following formula:

$$\rho_p = \frac{k_m + 1}{1/\rho_f + k_m/\rho_{ox}} \tag{2}$$

From the results shown in Table 1, it can be seen that the maximum value of the product $I_{sp}\rho_p^c$ is reached for a fuel ratio of 30% hydrogen to kerosene. This composition of the propellant will be used in the calculations shown later in this paper. The main advantages of the liquid rocket engine (LRE) scheme, shown in Fig. 1, are the high-energy effectiveness and the lower number of combustion units (no gas generator), which increases the engine reliability. The disadvantage of this scheme is the structural complexity of the chamber head and the arrangement to bleed the drive gas for the turbine from the main combustion chamber. The quantitative evaluation of the tripropellant propulsion unit efficiency for a two-stage carrier rocket is the main purpose of this work. This evaluation plays an important role in the choice of the engine scheme. The efficiency calculation of this scheme using the payload mass as the criterion for optimization is shown next.

Calculation of Tripropellant Propulsion Unit Efficiency for a Two-Stage Carrier Rocket

For comparison purposes, the baseline mission considered was the launch into low-Earth orbit with a two-stage carrier rocket with a liftoff mass of 200 ton. The reference rocket has a LOX/ker staged-combustion cycle engine in the first stage and an LOX/LH_2 staged-combustion cycle engine in the second stage. The maximization of the payload mass was used as optimization criterion. The mission velocity requirement was set at 8000 m/s. The propellant load was fixed at 140 ton in both cases. In the case considered here, the first-stage tripropellant engine has four thrust chambers, each with a sea-level thrust of 600 kN. A single turbopump unit feeds the four thrust chambers. The turbine drive gas for the turbopump unit is extracted from the four chambers and it is also reinjected in the four chambers. The second-stage engine is a staged-combustion-cycle type with a vacuum thrust of 440 kN. For the following calculation of LRE propulsion units it is necessary to know the mass characteristics of the different parts in the final design.

LRE Mass

The correlations given in Ref. 7 are used to calculate the mass of a staged-combustion engine. In all correlations that follow, the dimensional units indicated in the Nomenclature are used. The total mass of the engine is four times the mass of one engine unit:

$$\begin{gathered} m_{eng} = 4(m_{ch} + m_{tp} + 0.235P + 57) \\ 14.7 \text{ kN} \leq P \leq 981 \text{ kN} \end{gathered} \tag{3}$$

The mass of the engine chamber is further subdivided into the mass of the cylindrical part of the chamber, the subsonic and supersonic parts of the nozzle, the mass of the injector head, and the mass of the inlet manifold:

$$\begin{aligned} m_{ch} = F_{cr} \Bigg[& \gamma_{ch}(\bar{S}_{cyl} + \bar{S}_{sub}) + \gamma_n \bar{S}_{sup} + \frac{3.962 \times 10^6}{(p_{ch}\dot{m}_{ch}c^*)^{1/4}} \\ & + 17.58(p_{ch}\dot{m}_{ch}c^*)^{1/8} - \frac{13.3}{F_{cr}} \Bigg] \end{aligned} \tag{4}$$

In Eq. (4), F_{cr} is the area of the nozzle throat and $\bar{S}$ is the area divided by the critical area. The last constant term is a correlation factor.

We use the following correlation for the mass of the turbopump:

$$m_{tp} = 19 + 0.232 \times 10^{-3} D \tag{5}$$

where D is a hydromechanical parameter given by

$$D = \sum_i \frac{\dot{m}_i}{\omega}\left(\frac{\Delta p_i}{\rho_i}\right)^{1.5}, \qquad i = \text{ox}, f - \text{number of stages} \tag{6}$$

and the rotation speed of the pump, ω, is obtained from the following expression:

$$\omega = c_{cav}(\Delta p^*)^{0.75}/298\sqrt{\dot{m}_{ox}/\rho_{ox}} \tag{7}$$

$\Delta p_i = p_{ex,i} - p_{ent,i}$ is the pressure gain across the pump, and c_{cav} is the cavitating specific speed coefficient. The following expressions were used to calculate the fuel and oxidizer pressure at the exit of the pump (staged-combustion cycle):

$$\begin{aligned} p_{ex,f} &= 120 \times 10^5 + 1.802 p_{ch} \\ p_{ex,ox} &= 80 \times 10^5 + 1.667 p_{ch} \end{aligned} \tag{8}$$

Table 2 **Mass characteristics of the engine**

Engine	Propellant	P, kN	p_{ch}, MPa	$\bar{F}$	k_m	I_{sp}^{th}, m/s	φ_{ch}	φ_n	m_{eng}, kg
Tripropellant, stage I	LOX + (ker + H_2)	2400	15	80	4.3	4080	0.99	0.98	1682
Bipropellant, stage I	LOX + ker	2400	15	80	3.098	3650	0.99	0.98	1739
Bipropellant, stage II	LOX + H_2	432	15	80	6	4700	0.99	0.98	574

The values of the entrance pressure of the fuel and oxidizer were set at $p_{ent,f} = 2.5 \times 10^5$ Pa and $p_{ent,ox} = 4.5 \times 10^5$ Pa, respectively.

The head available for suction is obtained by subtracting the propellant saturation pressure, $p_s(T)$, and a reserve pressure drop from the entrance pressure, Δp_{res}:

$$\Delta p^* = p_{ent} - p_{sat}(T) - \Delta p_{res}$$
$$\Delta p_{res} = 0.2 \times 10^5 \text{ Pa}, \qquad c_{cav} = 4200 \tag{9}$$

The mass of the tripropellant engine is calculated from the formula for a bipropellant open-cycle engine without a gas generator, because in this case, the turbine drive gas is bled off from the chamber nozzle[6]:

$$m_{eng} = N_{ch} \times m_{ch} + N_{tp} \times m_{tp} + 0.217P + 57.5 - m_{gg}$$
$$14.7 \text{ kN} \le P \le 981 \text{ kN} \tag{10}$$

The mass of the gas generator is obtained from the correlation $m_{gg} = 3 + 0.0417P$. Substituting this expression into the previous correlation, we obtain

$$m_{eng} = N_{ch} \times m_{ch} + N_{tp} + 0.1753P + 54.3 \tag{11}$$

where $N_{ch} = 4$ is the number of thrust chambers, and $N_{tp} = 1$ is the number of turbopumps. The mass of the turbopump is given by

$$m_{tp} = 21 + 0.54 \times 10^{-3} \text{ D} \tag{12}$$

The fuel and oxidizer pressure at the exit of the pumps is obtained from the chamber pressure in the following way (open cycle):

$$p_{ex,f} = 1.714 p_{ch}, \qquad p_{ex,ox} = 1.428 p_{ch} \tag{13}$$

The following correlation is used to calculate the mass of the chamber:

$$m_{ch} = F_{cr}\left[\gamma_{ch}(\bar{S}_{cyl} + \bar{S}_{sub}) + \gamma_n \bar{S}_{sup} + \frac{1.63 \times 10^6}{(p_{ch}\dot{m}_{ch}c^*)^{1/4}} - \frac{8.5}{F_{cr}}\right] \tag{14}$$

The relative areas of the cylindrical part of the chamber, $\bar{S}_{cyl}$, subsonic part of the nozzle, $\bar{S}_{sub}$, and supersonic part, $\bar{S}_{sup}$, were calculated using the following correlations:

$$\bar{S}_{cyl} = 3.544L\sqrt{(p_{ch}\bar{q}_{ch}/\dot{m}_{ch})} - (2/\sqrt{\bar{q}_{ch}c^*}) + \sqrt{\bar{q}_{ch}c^*} + 1 \tag{15}$$

$$\bar{S}_{sub} = (2/\bar{q}_{ch}c^*) + (0.818/\sqrt{\bar{q}_{ch}c^*}) - 0.974 \tag{16}$$

$$\bar{S}_{sup} = S_0\{1 - [1.415 - (0.274/\sqrt[4]{\bar{F}_{ex}})]f(z)\} \tag{17}$$

where

$$S_0 = (32 - 10\kappa)(2.1 + 1.6\kappa^4)(\sqrt{\bar{F}_{ex}} - 1)^{2.25} \tag{18}$$

$$f(z) = 1 - \exp[-(1 - z)^{1/3}] \tag{19}$$

$$z = 1 - \left[\frac{\sin\theta_{ex}}{0.6 - (0.018\kappa - 0.0175)(\sqrt{\bar{F}_{ex}} + 24)}\right]^{4/3} \tag{20}$$

$$\sqrt{\bar{F}_{ex}} = \frac{[2/(\kappa + 1)]^{0.5/(\kappa - 1)}[(\kappa - 1)/(\kappa + 1)]^{0.25}}{[\varepsilon^{-2/\kappa} - \varepsilon^{-(\kappa+1)/\kappa}]^{0.25}} \tag{21}$$

The mass per unit of the chamber surface, γ_{ch}, and the mass per unit area of the nozzle, γ_n, were calculated with the following correlations:

$$\gamma_{ch} = 3.03 \times 10^{-6}(p_{ch}/\sqrt{d_{cr}}) - 17$$
$$1.581 \times 10^3 \le (p_{ch}/\sqrt{d_{cr}}) \le 5.85 \times 10^3 \tag{22}$$

$$\gamma_n = 4.722 \times 10^{-2}(p_{ch}/\sqrt{\varepsilon d_{cr}})^{0.475} - 23.58$$
$$63 \le (p_{ch}/\sqrt{\varepsilon d_{cr}}) \le 316 \tag{23}$$

The mass of the boost turbopump for the tripropellant engine was calculated with the following formula:

$$m_{b,tp} = 19.7 + 1.469 \times 10^6(\dot{m}_{ox}/\omega g_0 \rho_{ox}) \tag{24}$$

All coefficients were obtained with help of statistical data of real constructions. The initial data for the calculation of the engine masses and the results are shown in Table 2.

Mass of Tanks

For the calculation of the mass of the propellant tanks, it is necessary to take into account all active forces during the flight (the weight of the upper part, the bending moment from the aerodynamical forces, pressures in calculated sections of the tank, etc.). In this calculation, specific masses of tank prototypes were used (for the closely spaced volumes of the tanks). The mass of the oxygen and hydrogen tanks were calculated using the relationships for the bipropellant engine II stage:

$$m_{t,ox} = \frac{m_{p,ox}\gamma_{t,ox}}{\rho_{ox}}, \qquad m_{p,ox} = \frac{m_{p,\Sigma}k_m}{k_m + 1}$$
$$m_{t,hyd} = \frac{m_{p,hyd}\gamma_{t,hyd}}{\rho_{hyd}}, \qquad m_{p,hyd} = \frac{m_{p,\Sigma}}{k_m + 1} \tag{25}$$

(the prototype liquid-hydrogen tank of the first-stage carrier rocket H-II). For the tripropellant engine of the first stage, accordingly,

$$m_{p,hyd} = 0.3[m_{p,\Sigma}/(k_m + 1)]$$
$$m_{p,ker} = 0.7[m_{p,\Sigma}/(k_m + 1)] \tag{26}$$

The initial data and results of the calculations are shown in Table 3.

Mass of the Tank Pressurizing System

The oxygen tank pressurant system uses helium stored in cylinders with initial pressure of 300 atm. The cylinders are placed inside the LOX tank to increase helium density (inside the storage cylinder). Before being injected into the oxygen tank, the helium is heated in a heat exchanger that is placed

Table 3 Mass characteristics of the tanks

Stage	Component	m, kg	ρ, kg/m^3	γ_t, kg/m^3	m_t, kg
I stage, tripropellant,	M_Σ	140,000	——	——	8877.5
O_2 + ker + H_2	O_2	113,684	1140	0.04	3989
	Ker	18,421	800	0.036	663
	H_2	7,895	71	0.038	4225.5
II stage, bipropellant,	M_Σ	36,988	——	——	3940.5
O_2 + H_2	O_2	31,704	1140	0.04	1112.43
	H_2	5,284	71	0.038	2828
I stage, bipropellant,	M_Σ	140,000	——	——	4943.6
O_2 + ker	O_2	120,000	1140	0.04	3714.6
	Ker	20,000	71	0.036	1230
II stage, bipropellant,	M_Σ	41,049	——	——	4374
O_2 + H_2	O_2	35,185	1140	0.04	1235
	H_2	5,864	71	0.038	3138.6

Table 4 Initial data and final results of calculations[a]

Description	Bipropellant	Tripropellant
Initial data		
Liftoff mass, ton	200	200
Propellant load of first stage, ton	140	140
Vacuum thrust of first-stage engine, kN	2,400	2,400
Propellant mass ratio	3.098	4.32
Chamber pressure, MPa	15	15
Vacuum specific impulse, m/s	3,650	4,080
Chamber loss coefficient	0.99	0.99
Nozzle loss coefficient	0.98	0.98
Nozzle area ratio I	40	40/80
Average pitch angle of first stage, deg	60	60
Average pitch angle of second stage, deg	15	15
Aerodynamical velocity losses, m/s	150	150
Control velocity losses, m/s	70	80
High-altitude velocity losses, m/s	237	237
Results of calculations		
Burn time of first stage, s	205.5	229.75
Gravitational losses, m/s	1,746	1,952
Total velocity losses of first stage, m/s	2,203	2,419
Effective velocity at first-stage burnout, m/s	2,060	2,345
Dry mass of first stage, kg	8,076.5	12,039
Initial mass of second stage, kg	51,923.5	47,961
Propellant mass ratio of second stage	6	6
Theoretical vacuum specific impulse, m/s	4,700	4,700
Vacuum thrust of second-stage engine, kN	440	440
Burn time of second stage, s	433.7	391
Gravitational velocity losses, m/s	1,099	993
Control velocity losses, m/s	90	80
Mass of second stage at burnout, kg	10,874	10,972
Propellant mass of second stage, kg	41,049	36,988.5
Dry mass of second stage, kg	5,419	4,939.5
Payload mass, kg	5,455	6,033

[a] $\Delta m_{pm} = 578$ kg.

at the exit of the turbine. The average temperature and pressure of the pressurant helium inside the oxygen tank are 170 K and 4.5×10^5 Pa, respectively. In this condition, the average density of the pressurant helium is 1.25 kg/m^3.

The mass of the pressurization system for the oxygen tank was calculated using the formula:

$$m_{ps,ox} = (m_{ox}K_{gv}/\rho_{ox})\rho_{gp}\gamma_c \tag{27}$$

where $\gamma_c = m_c/m_{gp} = 4.5$ is the ratio of reservoir mass to pressurized gas mass. $K_{gv} = 1.05$ − the relative initial gas volume in the tank.

The kerosene tank is pressurized with the combustion products from a dedicated gas generator. The mass of the kerosene tank pressurization system was calculated using the following formula:

$$m_{ps,ker} = \frac{m_{ker}K_{gv}}{\rho_{ker}}\rho_{gp}K_{cstr} \tag{28}$$

here $K_{cstr} = 1.15$, the coefficient of construction (takes in account the mass of gas generator, pipelines, inlet arrangements, etc.).

The hydrogen tank is pressurized using gaseous hydrogen, heated in the heat exchanger. The mass of the hydrogen tank pressurization system was calculated by the formula:

$$m_{ps,hyd} = \frac{m_{hyd}}{\rho_{hyd,liq}}K_{gv}\rho_{gp}K_{cstr} \tag{29}$$

The average density of hydrogen-pressurant gas (inside the hydrogen tank) is 0.0842 kg/m^3 (average temperature = 70 K, pressure = 2.5×10^5 Pa). The propellant reserve mass is 0.5% of the total propellant load.

Payload Mass for the Two Rockets

The tripropellant engine scheme for the first stage of the rocket is shown in Fig. 1. The second stage uses a bipropellant rocket engine with the same scheme and performance as the second stage of the reference rocket. The payload mass was obtained by integrating the equations of motion of the carrier rocket along its flight path:

$$\Delta V_{\mathrm{I}} = I^{v}_{\mathrm{sp,I}}\, \ell n \frac{m_{\mathrm{I},0}}{m_{\mathrm{I,fin}}} - \Delta V_{\mathrm{I,gr}} - \Delta V_{\mathrm{I,eng}} - \Delta V_{\mathrm{I,ctrl}} - \Delta V_{\mathrm{I,aer}} \tag{30}$$

$$\Delta V_{\mathrm{II}} = I^{v}_{\mathrm{sp,II}}\, \ell n \frac{m_{\mathrm{II},0}}{m_{\mathrm{II,fin}}} - \Delta V_{\mathrm{II,gr}} - \Delta V_{\mathrm{II,ctrl}} \tag{31}$$

$$\Delta V_{\mathrm{I}} + \Delta V_{\mathrm{II}} = 8000$$

$\Delta V_{\mathrm{gr}} = g t_i \sin\theta$; $t_i = m_{p,i}/\dot{m}_i$; $\Delta V_{\mathrm{I,eng}} = \Delta V_{\mathrm{id}}(1 - \tilde{\gamma}\tilde{P}_{\mathrm{I}})$; $\Delta V_{\mathrm{id}} = I_{\mathrm{sp}}\, \ell n\, (m_0/m_{\mathrm{fin}})$; $\Delta V_{\mathrm{aer,I}} = 150$ m/s; $\dot{m}_{\Sigma,i} = P_i/(I^{v}_{\mathrm{sp},i}\varphi_{\mathrm{ch}}\varphi_n)$; $i = \mathrm{I}, \mathrm{II}$: $\Delta V_{\mathrm{ctrl,I}} = 70 \div 80$ m/s; $\tilde{\gamma} = (P_v - P_0)/P_v$; $\tilde{P}_{\mathrm{I}} = 0.2$; $m_{\mathrm{I},f} = m_{\mathrm{I},0} - m_{\mathrm{I},p}$; $m_{\mathrm{cstr},\Sigma} = m_{t,\Sigma} + m_{\mathrm{ps},\Sigma} + m_{\mathrm{eng}} + m_{\mathrm{rp}}$; and $m_{\mathrm{pm}} = m_{\mathrm{II,fin}} - m_{\mathrm{cstr},\Sigma,\mathrm{II}}$.

The initial data and results of the calculation are shown in Table 4.

The use of a tripropellant engine (O_2 + ker + H_2) on the first stage results in a 10% increase in payload mass. The specific impulse increase compensates for the higher value of the stage structural coefficient (an additional hydrogen tank is added).

Conclusions

1) The use of a tripropellant engine for the first stage of a two-stage carrier rocket with a takeoff mass of 200 ton increases the payload mass by 10% for a fuel ratio (hydrogen to kerosene) of 30%.

2) The development of the tripropellant engine must be directed in the first phase at the design of an injector head with high-combustion efficiency.

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