MASS DISTRIBUTION OF HYBRID ROCKETS FOR LAUNCHING NANOSATS INTO LEO

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Abstract. *Hybrid rockets using green propellants have been considered for launching nanosats into low Earth orbit (LEO). This work describes an iterative process to determine the mass distribution of air and ground launched hybrid rockets using H_2O_2 and solid paraffin as propellants. The payload fraction, propellant mass fraction and inert mass fraction, including masses of tanks, fuel grain, nozzle, engine case and vehicle case, are calculated from given initial conditions. It is considered a mission to place a 20 kg payload into a 300 km circular equatorial orbit, by two and three stages hybrid rockets.*

Keywords: *Paraffin, H_2O_2, nanosats, low Earth orbit (LEO), mass distribution*

1. INTRODUCTION

Hybrid rocket technology is known for more than 50 years, however only in the 1960's its safety characteristics motivated a significant research. Nowadays, the need for green propellants (propellants with low toxicity and low pollutant characteristics), the requirements of safe operation and storability, low cost missions, and the interest for launching small payloads and nanosats into LEO made hybrid rockets more attractive.

Hybrid propulsion systems employ propellants in different phases, being the most usual hybrid systems with a solid fuel and a liquid oxidizer. Since they use only one liquid propelllant, they require only one liquid line and a relatively simple injection system, as compared to liquid bipropellant systems which require two separate liquid lines and a complex injection plate in order to collide and mix the fuel and oxidizer jets. The control of the oxidizer flow rate in hybrid systems allows several starts and an accurate control of the thrust level.

The safe operation of hybrid propulsion systems is related to the separation of fuel and oxidizer, differently from solid systems which mix fuel and oxidizer in the grain. Another important safety characteristic is the independence of the regression rate with respect to the chamber pressure, making hybrid systems safer than solid systems if pressure peaks do occur.

The main disadvantage of hybrid rockets is the low thrust level attainable, due to the relatively low regression rates of conventional solid fuels, making necessary the use of a large number of ports. Some methods to increase the regression rate are known, such as i) insert screens or mechanical devices in the ports to increase the turbulence level; ii) use of metallic addictives; iii) use of oxidizers mixed within the solid fuel; iv) increase the surface rugosity adding small solid particles. However, these solutions have also undesirable characteristics.

Recently, it was developed in the Stanford University and in the Ames-NASA Research Center, both in the USA, a new paraffin-based fuel whose regression rate is approximately three times higher than conventional hybrid fuels (Karabeyoglu et al., 2003a,b, 2004). Promising results were obtained by several researchers (Brown and Lydon, 2005; Karabeyoglu et al., 2004; Santos et al., 2005; McCormick et al., 2005) using paraffin with different oxidizers – liquid oxygen (LOX), gaseous oxygen (GOX), nitrous oxide (N_2O) and hydrogen peroxide (H_2O_2).

The hydrogen peroxide (H_2O_2) is a well-known oxidizer and has been used for decades in rockets, gas generators, helicopter rotors and rocket belts (Davis Jr and Keefe, 1956; Wernimont et al., 1999). It was used, for example, as an oxidizer in the British rocket Black Knight (Peroxide Propulsion, 2006). Heister et al. (1998) cites some advantages of using hydrogen peroxide as oxidizer: high density, easy of handling, non-toxicity and mono-propellant characteristics. Turbo-pumps and pressurization systems can utilize the energy released during peroxide decomposition and its products in order to simplify the tank pressurization systems.

Walter (1954) describes the decomposition and detonation characteristics of peroxide and mentions that peroxide at concentrations lower than 82 % is not detonable and that pressure does not affect the peroxide decomposition velocity. Williams et al. (2004) state that HTP (High Test Peroxide) is similar to nitroglicerin in terms of shock sensitivity and explodes with the same strength than the same quantity of TNT (Trinitrotoluen).

The propulsive characteristics of HTP and paraffin as hybrid propellants were determined by Gouvêa et al. (2006) who also specified a preliminary mass distribution of hybrid rockets using a fixed inert mass fraction (=0.15) in all stages, but did not verify the effects of mechanical characteristics of materials on the inert mass.

Therefore, the objective of this work is to present an iterative process to determine the mass distribution of hybrid propulsion systems using paraffin and hydrogen peroxide as propellants, considering the effects of material characteristics on the rocket inert fraction. Two and three stage rockets, ground launched and air-launched are considered for placing a 20 kg nanosat into a low Earth circular equatorial orbit (300 km).

2. MASS DISTRIBUTION OF HYBRID ROCKETS

The optimization of a propulsion system to perform a given mission is a complex task, since there several coupled variables which depend on time and on rocket trajectory.

To place a satellite into a specified orbit around Earth, the launching vehicle must attain a characteristic velocity, ΔV, to overcome the Earth gravitational field, the drag, to make maneuvers and to attain a prescribed orbital velocity.

Humble et al. (1995) used historical data of several launching vehicles and presented typical ΔV values between 8800 and 9300 m/s, as required to place satellites into a low Earth orbit. In this work it was adopted a conservative ΔV = 9300 m/s for ground launched rockets and a ΔV = 8700 m/s for air launched vehicles, based on data from the American air launched rocket Pegasus.

Usually, a rocket must have several stages to transport a significant payload fraction, above 1 %, into an orbit around Earth. The increase in payload fraction with a larger number of stages is significant up to 3 or 4 stages, but above 4 stages, the propulsion system complexity grows considerably, with consequent reduction in reliability and no significant increase on payload fraction.

In this work, hybrid rockets with 2 or 3 stages are studied, assuming a uniform distribution of characteristic velocities among stages. Sutton (1992) shows that, for simplified cases and disregarding trajectory effects, that a uniform distribution of characteristic velocities is an optimum solution.

Initially, in order to determine the mass distribution of a rocket, it is necessary to estimate the inert mass fraction of all stages. The inert mass is the total initial mass **less** the propellant and the payload masses.

The inert mass fraction, $f_{inert,j}$, of the j-stage (j = 1, 2 or 3) is defined by

$$f_{inert,j} = m_{inert,j} \Big/ \left(m_{prop,j} + m_{inert,j} \right) \tag{1}$$

where $m_{prop,j}$ is the propellant mass and $m_{inert,j}$ is the inert mass of the j-stage.

Tables 1 and 2 show data concerning the mass distribution, in kg, of rocket engines using solid and liquid propellants, respectively.

Table 1 – Mass distribution of solid propellant engines (mass in kg).

Engine	Propellant	Insulation	Engine case	Nozzle
Castor IVA	10.101	234	749	225
GEM	11.767	312	372	242
ORBUS 21	9707	145	354	143
OBUS 6E	2721	64.1	90.9	105.2
Star 48B	2010	27.1	58.3	43.8
Star 37XFP	884	12.7	26.3	31.7
Star 63D	3250	71.4	106.3	60.8
Orion 50SAL	12.160	265.2	547.9	235.4
Orion 50	3024	75.6	133.4	118.7
Orion 38	770.7	21.9	39.4	52.8

(Continues)

Table 2.1 - Continuation

Ignition	Miscelaneous	Inert	f_{prop}	f_{inert}
10	276	1494	0.871	0.129
7.9	291	1224.9	0.906	0.094
16	7	665	0.936	0.064
9.5	5.3	275	0.908	0.092
0.0	2.2	131.4	0.939	0.061
0.0	1.3	72	0.915	0.085
1.0	11.6	251.1	0.928	0.072
9.1	21.0	1078.6	0.918	0.082
5.3	9.9	342.9	0.898	0.102
1.3	10.6	126	0.859	0.141

Source: Isakowitz (1999)

Table 2 - Mass distribution of liquid propellant engines (mass in kg).

Engine	Propellant	Inert	f_{prop}	f_{inert}
YF-40	14.200	1.000	0.93	0.07
YF-73	8.500	2.000	0.81	0.19
11D49	18.700	1.435	0.93	0.07
LE5-A	14.000	2.700	0.84	0.16
LE-5B	16.600	3.000	0.85	0.15
RL10B-2	16.820	2.457	0.87	0.13
AJ10-118K	6.004	950	0.86	0.14
RS27A	95.500	6.820	0.93	0.07
11D58M	14.600	2.720	0.84	0.16
RD-171	325.700	28.600	0.92	0.08

Source: Isakowitz (1999)

The propellant mass of the j-stage is calculated by

$$m_{prop,j} = m_{pay,j}\left(1 - f_{inert}\right)\left(e^{\Delta V_j / Isp_j g_o} - 1\right) \Big/ \left(1 - f_{inert,j} e^{\Delta V_j / Isp_j g_o}\right) \tag{2}$$

where $m_{pay,j}$ is the payload mass, Isp_j is the specific impulse and ΔV_j is the characteristic velocity of the j-stage.

The payload mass of a given stage is the total initial mass of all upper stages, and the last stage payload mass is the nanosat mass. The inert mass of the j-stage is calculated in terms of the assumed j-stage inert fraction:

$$m_{inert,j} = \frac{f_{inert,j}}{1 - f_{inert,j}} \tag{3}$$

and the total initial mass of each stage, $m_{0,j}$, is calculated by

$$m_{0,j} = m_{inert,j} + m_{prop,j} + m_{pay,j} \tag{4}$$

The F/W ratio relates the thrust, F, and the weight , W, of a rocket, and it is generally expressed in g-number. This ratio (acceleration) is limited to a range. It can not be high to avoid damages to the equipment, or to not harm an eventual crew, obviously it cannot be smaller than unity, but it should be small to optimize the performance. The thrust to obtain a specified j-stage thrust/weigth ratio, $(F/W)_j$, is obtained from

$$F_j = \left(F/W\right)_j m_{0,j} g_0 \tag{5}$$

The total mass flow rate of propellants of the j-stage, $\dot{m}_{prop,j}$, is related to the thrust and to the specific impulse of the j-stage by

$$\dot{m}_{prop,j} = \frac{F_j}{Isp_j g_0} \tag{6}$$

The fuel mass flow rate of j-stage, $\dot{m}_{fuel,j}$, limits the thrust levels, due to the relatively low regression rates of the fuels used in hybrid rockets. It is related to the total mass consumption rate of propellants and to the $(O/F)_j$ (oxidizer/fuel) mass ratio, by the relation:

$$\dot{m}_{fuel,j} = \frac{\dot{m}_{prop,j}}{1 + \left(O/F\right)_j} \tag{7}$$

The mass flow rate of oxidizer of the j-stage, $\dot{m}_{oxid,j}$, is calculated by

$$\dot{m}_{oxid,j} = \dot{m}_{prop,j} \frac{\left(O/F\right)_j}{1 + \left(O/F\right)_j} = \dot{m}_{prop,j} - \dot{m}_{fuel,j} \tag{8}$$

The burning time of the j-stage, $t_{b,j}$, is obtained from

$$t_{b,j} = \frac{m_{prop,j}}{\dot{m}_{prop,j}} \tag{9}$$

The oxidizer volume of the j-stage, $V_{oxid,j}$, is calculated by

$$V_{oxid,j} = \frac{m_{oxid,j}}{\rho_{oxid,j}} \tag{10}$$

where $\rho_{oxid,j}$ is the hydrogen peroxide density in the j-stage, which depends on temperature, pressure and peroxide concentration. The subscript j will be disregarded in the next equations.

The oxidizer tank internal diameter, $D_{i,tk,oxid}$, can be calculated from the oxidizer volume plus an ullage volume (e.g., 5 % of the oxidizer volume), $D_{i,tk,oxid} = \left(1.05 \times 6V_{oxid}/\pi\right)^{1/3}$.

The oxidizer pressure in the tank, P_{oxid}, equals the chamber pressure, P_c, plus the pressure losses, ΔP_{oxid}, in lines, injection and valves, i.e., $P_{oxid} = P_c + \Delta P_{oxid}$. It was assumed a total pressure loss $\Delta P_{oxid} = 5$ MPa in all stages.

The thickness of a spherical oxidizer tank, considering a 10 % overpressure, is given by

$$e_{tk,oxid} = 1.1\frac{P_{oxid} D_{i,tk,oxid}}{2\sigma_{tk,oxid}} \tag{11}$$

where $\sigma_{tk,oxid}$ is the yielding strength of the tank material. Therefore, the oxidizer tank external diameter is $D_{e,tk,oxid} = D_{i,tk,oxid} + 2e_{tk,oxid}$. The theoretical minimum thickness of a cylindrical tank is twice the thickness of a spherical tank.

Two tank configurations were considered: i) a spherical tank, and ii) a cylindrical tank, with a cylinder section of length $L_{c,tq,oxid}$ and two hemispherical domes, with total length $L_{tk,oxid}$, as shown by Eq. 12. If the spherical tank external diameter were larger than the external diameter of the paraffin chamber case than a cylindrical configuration was adopted, with an internal diameter equal to the external diameter of the paraffin chamber case.

The total length of an oxidizer cylindrical tank with spherical domes is given by

$$L_{tk,oxid} = L_{c,tk,oxid} + D_{e,tk,oxid} = 4\left(\frac{1.05V_{oxid} - \pi/6D^3_{i,tk,oxid}}{\pi D^2_{i,tk,oxid}}\right) + D_{e,tk,oxid} \tag{12}$$

and the oxidizer cylindrical tank mass, $m_{tk,oxid}$, is

$$m_{tk,oxid} = \rho_{tk,oxid}\pi\left[\frac{1}{6}\left(D^3_{e,tk,oxid} - D^3_{i,tk,oxid}\right) + \frac{1}{4}\left(D^2_{e,tk,oxid} - D^2_{i,tk,oxid}\right)L_{c,tk,oxid}\right] \tag{13}$$

If the oxidizer tank is spherical its mass is

$$m_{tk,oxid} = \rho_{tk,oxid}\left(\pi/6\right)\left(D^3_{e,tk,oxid} - D^3_{i,tk,oxid}\right) \tag{14}$$

A pressurizing system with Helium was chosen to empty the oxidizer tank. The pressurant mass required is given by:

$$m_{press} = \frac{P_{oxid}V_{oxid}}{R_{press}T_{oxid}}\left(\frac{\gamma_{He}}{1 - P_{oxid}/P_{in,press}}\right) \tag{15}$$

where γ_{He} is the specific heat ratio of helium, $\gamma_{He} = 1.666$, P_{oxid} is the pressure in the oxidizer tank and $P_{in,press}$ is the initial pressure in the pressurant tank, and $R_{press} = 8314/4$ Nm/kgK.

Assuming the pressurized Helium is an ideal gas, its volume, V_{press}, can be obtained from the perfect gas equation with pressure $P_{press} = 200$ atm. Then, the internal diameter of the pressurant spherical tank is obtained from

$$D_{i,tk,press} = \left(6V_{press}/\pi\right)^{1/3} \tag{16}$$

The pressurant tank thickness, $e_{tq,press}$, and its external diameter, $D_{e,tq,press}$, are obtained, respectively, from:

$$e_{tk,press} = 1.1\frac{P_{press}D_{i,tk,press}}{2\sigma_{tk,press}} \tag{17}$$

$$D_{e,tk,press} = D_{i,tk,press} + 2e_{tk,press} = \left(\frac{6V_{press}}{\pi}\right)^{1/3} + 1.1\frac{P_{press}D_{i,tk,press}}{\sigma_{tk,press}} \tag{18}$$

and the mass of the spherical tank of pressurant is given by:

$$m_{tk,press} = \rho_{tk,press}\left(\pi/6\right)\left(D^3_{e,tk,press} - D^3_{i,tk,press}\right) \tag{19}$$

where $\rho_{tk,press}$ is the density of the pressurant tank material.

The fuel chamber is composed by the injection plate, case, insulation and nozzle. The chamber mass depends on the paraffin grain geometry. The fuel grain internal diameter, D_{ig}, is calculated by

$$D_{i,g} = \left(4\dot{m}_{oxid}/\pi G_{oxid}\right)^{1/2} \tag{20}$$

where G_{oxid} is the mass flow rate of oxidant per unit area in the fuel chamber.

The fuel regression rate, $\dot{r}$, is adjusted by equation $\dot{r} = aG^n_{oxid}$ (mm/s) where a and n are experimental constants. Brown and Lydon (2005) obtained $a = 0.0344(mm/s)\left(m^2s/kg\right)^n$ and $n = 0.9593$ (non-dimensional) for paraffin burning with 84 % hydrogen peroxide.

The fuel grain external diameter, D_{eg}, and the grain length, L_g, are given, respectively, by:

$$D_{e,g} = \left[(a/1000)(4n+2)(4\dot{m}_{oxid}/\pi)^n t_b + D^{2n+1}_{i,g}\right]^{\frac{1}{2n+1}} \tag{21}$$

$$L_g = \frac{4m_{fuel}}{\rho_{fuel}\pi\left(D^2_{e,g} - D^2_{i,g}\right)} \tag{22}$$

The nozzle throat area, A_t, and its diameter, D_t, are calculated from combustion characteristics and from the chosen chamber pressure, P_c:

$$A_t = \frac{\dot{m}_{prop}\sqrt{T_c R}}{\Gamma P_c}\ ,\quad \Gamma = \gamma^{1/2}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \tag{23}$$

$$D_t = \left(4A_t/\pi\right)^{1/2} \tag{24}$$

where T_c is the temperature of the gaseous products in the fuel chamber, R is the gas constant, and γ is the specific heat ratio of the gaseous products.

The nozzle exit area ratio, ε, depends on combustion characteristics and exit/chamber pressure ratio:

$$\frac{1}{\varepsilon} = \frac{A_t}{A_e} = \left((\gamma+1)/2\right)^{\frac{1}{\gamma-1}}\left(P_e/P_c\right)^{\frac{1}{\gamma}}\left((\gamma+1)/(\gamma-1)\left[1-\left(P_e/P_c\right)^{\frac{\gamma-1}{\gamma}}\right]\right)^{1/2} \tag{25}$$

where A_e is nozzle exit area and P_e is the nozzle exit pressure.

The nozzle mass, m_{noz}, is calculated from an empirical relation obtained by Humble et al. (1995) for hybrid engine nozzles:

$$m_{noz} = 125\left(\frac{m_{prop}}{5400}\right)^{2/3}\left(\frac{\varepsilon}{10}\right)^{1/4} \quad (26)$$

The nozzle length is calculated by:

$$L_{noz} = \frac{D_{e,g} - D_t}{2tg\theta_c} + \frac{D_e - D_t}{2tg\theta_D} \quad (27)$$

where θ_C and θ_D are, respectively, the convergence and divergence semi-angles of the nozzle, and D_e is the nozzle exit diameter. It is assumed that the nozzle convergent section has initial diameter equal to the fuel grain external diameter.

The vehicle case length is approximately equal to the sum of chamber, nozzle and tanks lengths plus the lengths of control devices, valves and feeding lines. It was considered an extra 10 % increase on vehicle case length for stage coupling and other devices. Therefore, the total vehicle case length was calculated by

$$L_{case} = 1.1\left(L_g + L_{noz} + L_{tk,oxid} + L_{tk,press}\right) \quad (28)$$

The internal diameter of the vehicle case, $D_{i,,case}$, is assumed equal to the external diameter of the engine case (solid fuel chamber). The thickness of the vehicle case and, thus, the external diameter of the vehicle case, $D_{e,case}$, both depend on material yielding strength and on the applied compression force:

$$D_{e,case} = \left[D_{i,case}^2 + \frac{4(F + W_o)}{\pi\sigma_c}\right]^{1/2} \quad (29)$$

where is the case internal diameter and σ_c is the compression strength of the case material. This formula leads to a very small thickness, and then a minimum case thickness of 2 mm was considered for all stages.

The vehicle case mass is calculated by

$$m_{case} = \rho_{case} L_{case} \left(\pi/4\right)\left(D_{e,case}^2 - D_{i,case}^2\right) \quad (30)$$

The fins and engine insulation weights are assumed incorporated into the vehicle case weight.

3. RESULTS AND COMMENTS

Tables 3, 4, 5 and 6 show the initial conditions considered and Table 7 shows the materials, and their properties, for the iterative process for mass distribution analysis.

The propellants chosen were 90 % H_2O_2 and $C_{20}H_{42}$ paraffin. They present maximum theoretical specific impulse when O/F = 7, for P_c = 30 atm and ε = 20. Therefore, the paraffin grain was designed to attain this maximum O/F ratio when the fuel single perforated grain reached its average diameter.

Using data from Tables 3 to 7, the masses and the geometry of all components and stages were calculated. Then a new inert fraction was calculated for each stage. If the new one was approximately equal to the previous one (< 0.1 %) the calculation was stopped, and if not a new iteration was made.

Tables 8 and 9 show the mass distributions for all stages.

In Tables 5 and 6 initial values for two stage vehicles were proposed, however the iterated results showed that with those conditions it is not possible to design two stage vehicles to reach the desired orbit. This may be circumvented by:

- propellants with higher specific impulses (higher concentration peroxide and additives in paraffin).
- change in characteristic velocities among stages.
- utilization of lighter and stronger materials.

To verify the effects of using a lighter and stronger material, Table 10 shows the mass distribution considering titanium oxidant tanks. It can be seen on Table 10 a significant reduction on masses of the oxidizer tanks and, consequently, the inert fractions of all stages are lower than the inert fractions found on Table 8.

It is well known that tanks, engine case, insulation and vehicle case of high performance rockets are not made of just a single material, but combinations of materials, such as carbon fibers and metal alloys. For each mission there is an optimum set of materials that yield the best relation among mass, compatibility, strength and safety and, consequently, the lowest possible inert fraction.

In all cases considered, it was chosen an initial inert fraction f_{inert} = 0.15, however the final inert fractions were, in general, larger. The inert fraction is strongly affected by the choice of materials, especially for tanks, vehicle case and engine case. The present results show that the materials used in the calculations are not as efficient as the ones used in advanced launching systems, for example, the materials used in the American rockets Taurus and Titan which have inert fractions varying from 0.03 to 0.1. Nevertheless, the inert fractions for the hybrid rockets considered in this analysis were similar to the inert fractions of the Indian rocket PSLV or the Brazilian rocket VLS, which vary from 0.13 to 0.28, mainly due to the relatively low specific impulse propellants adopted.

It should be noted that many rocket performance parameters are coupled and vary with time, such as specific impulses, ambient pressure and drag coefficient, making necessary also to optimize the rocket performance considering its trajectory.

Table 3 - Initial conditions for a ground launched vehicle with three stages.

Number de stages	3		
Payload	20 kg		
ΔV_{total} (m/s)	9300		
Stage	**1st**	**2nd**	**3rd**
ΔV (m/s)	3100	3100	3100
Expansion rate (ε)	10	40	60
Isp (s)	262	291	297
Inert fraction (f_{inerte})	0.15	0.15	0.15
F/W (g's)	2.5	2.5	2.5
O/F	7	7	7
Chamber pressure (MPA)	3	3	3
Case	Carbon Fiber		
Case	Stainless steel		
Oxidizer tank	Stainless steel		
Pressurant tank	Titanium		

Table 4 - Initial conditions for an air launched vehicle with three stages.

Number de stages	3		
Payload	20 kg		
ΔV_{total} (m/s)	8700		
Stage	**1st**	**2nd**	**3rd**
ΔV (m/s)	2900	2900	2900
Expansion rate (ε)	10	40	60
Isp (s)	262	291	297
Case	Carbon Fiber		
Case	Stainless steel		
Oxidizer tank	Stainless steel		
Pressurant tank	Titanium		

Table 5 - Initial conditions for a ground launched vehicle with two stages.

Number of stages	2	
Payload	20 kg	
ΔV_{total} (m/s)	9300	
Stage	**1st**	**2nd**
ΔV (m/s)	4650	4650
Expansion rate (ε)	10	40
Isp (s)	262	291
Fuselage	Carbon Fiber	
Case	Stainless steel	
Oxidant tank	Stainless steel	
Pressurant tank	Titanium	

Table 6 - Initial conditions for an air launched vehicle with two stages.

Number of stages	2	
Payload	20 kg	
ΔV_{total} (m/s)	8700	
Stage	**1st**	**2nd**
ΔV (m/s)	4350	4350
Expansion rate (ε)	10	40
Isp (s)	262	291
Fuselage	Carbon Fiber	
Case	Stainless steel	
Oxidant tank	Stainless steel	
Pressurant tank	Titanium	

Table 7 - Materials and mechanical properties.

Item	**Material**	σ_{el} **(MPa)**	σ_{yield} **(MPa)**	**Density (kg/m³)**
Vehicle case	Carbon fiber	228000	3800	1810
Engine Case	Stainless steel	-	550	7850
Oxidant tank	Stainless steel	-	550	7850
Oxidant tank	Titanium	115000	790	4480
Pressurizer tank	Titanium	115000	790	4480

σ_{el} = bulk modulus; σ_{yield} = tensile yield strentgh.

Table 8 - Ground launched rocket with three stages.

Item	**unit**	**1st stage**	**2nd stage**	**3rd stage**
m_{prop}	kg	3620.8	550.8	106.3
m_{fuel}	kg	452	68.8	13.3
m_{oxid}	kg	3168.8	482	93
m_{pay}	kg	831	162	20
m_{inert}	kg	715.49	118.46	36
$\dot{m}_{prop}$	kg/s	49.33	7.14	1.36
$\dot{m}_{fuel}$	kg/s	6.16	0.89	0.17
$\dot{m}_{oxid}$	kg/s	43.17	6.25	1.19
t_b	s	73.4	77.1	77.8
m_0	kg	5168	831	162
Engine case	kg	140	17.35	3
Nozzle	kg	138.8	38.6	13.2
Oxidizer tank	kg	278	49.6	16.56
Pressurant tank	kg	97	1.4	0.05
Pressurant	kg	25.8	3.9	0.76
Vehicle case	kg	33.9	8.8	2.85
f_{inert}	**-**	**0.165**	**0.1**77	**0.25**

Table 9 - Air launched rocket with three stages.

Item	**unit**	**1st stage**	**2nd stage**	**3rd stage**
m_{prop}	kg	2306.2	423.5	89.32
m_{fuel}	kg	288.3	52.9	11.16
m_{oxid}	kg	2017.9	370.6	78.16
m_{pay}	kg	663	141.7	20
m_{inert}	kg	439	98.7	32.37
$\dot{m}_{prop}$	kg/s	32.16	5.7	1.19
$\dot{m}_{fuel}$	kg/s	4	0.7	0.15
$\dot{m}_{oxid}$	kg/s	28.16	5	1.04
t_b	s	70.9	74.3	74.9
m_0	kg	3409	663	141.7
Engine case	kg	85	13.2	2.56
Nozzle	kg	102	32.	11.36
Oxidizer tank	kg	176.5	25	14.77
Pressurant tank	kg	34.7	0.8	0.037
Pressurant	kg	16.48	3	0.35
Vehicle case	kg	24.46	7.36	2.53
f_{inert}	**-**	**0.16**	**0.189**	**0.266**

Table 10 - Ground launched rocket with three stages and titanium oxidant tanks.

Item	**unit**	**1st stage**	**2nd stage**	**3rd stage**
m_{prop}	kg	2197.8	410.3	85.7
m_{fuel}	kg	274.7	51.3	10.7
m_{oxid}	kg	1923.1	359	75
m_{pay}	kg	619.3	130.9	20
m_{inert}	kg	319	78.13	25.18
$\dot{m}_{prop}$	kg/s	29.9	5.32	1.10
$\dot{m}_{fuel}$	kg/s	3.7	0.66	0.14
$\dot{m}_{oxid}$	kg/s	26.2	4.66	0.96
t_b	s	73.4	77.1	77.8
m_0	kg	3136.8	619.3	130.9
Engine case	kg	79.8	12.7	2.45
Nozzle	kg	99.49	31.7	11.44
Oxidizer tank	kg	69.2	23.4	8.2
Pressurant tank	kg	31	0.75	0.034
Pressurant	kg	15.7	2.93	0.61
Vehicle case	kg	23.6	7.19	2.5
f_{inert}	**-**	**0.127**	**0.16**	**0.23**

4. CONCLUSIONS

The mass distribution of hybrid propulsion systems using paraffin and hydrogen peroxide as propellants was obtained by an iterative process. Two and three stage rockets, ground or air launched were considered for placing a 20 kg nanosat into a low Earth circular equatorial orbit (300 km). It was verified that two stage hybrid rockets do not allow to perform the specified mission with the materials considered, however three stage hybrid rockets are capable of performing the mission with the assumptions made. Ground launched rockets present lower inert fractions but higher total initial masses than air launched rockets. The use of lighter and stronger materials and propellants with higher specific impulses can reduce the inert mass fractions in all cases considered.

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