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FEASIBILITY ANALYSIS OF CUBESAT MISSION ON MARS USING A SMALL DEDICATED LAUNCHER AND ELETRIC PROPULSION

Artur Gustavo Slongo*, Nícolas Winckler Musskopf*, Samara Herrmann*, André Luís da Silva* and João Felipe de Araújo Martos*

*Federal University of Santa Maria - UFSM, Technology Center, Santa Maria - RS, Brazil.

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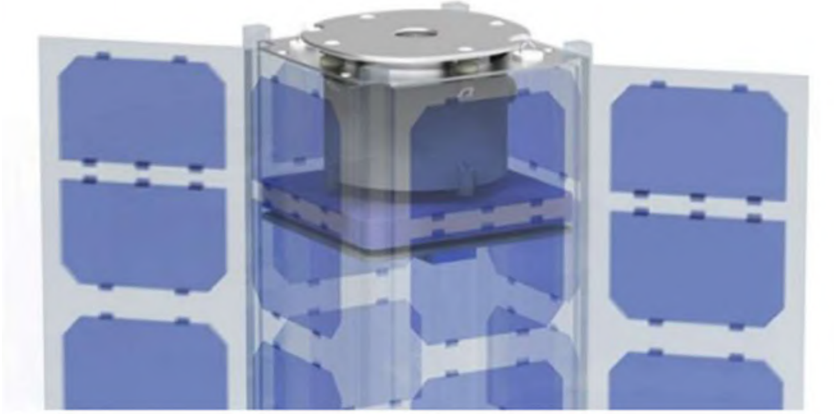
Mission Definition

- **Two main missions :**
  - Use of a Small Launcher to launch a CubeSat to LEO (Low Earth Orbit):
    - Delta-V = 9 km/s.
  - Transit from LEO to Mars Orbit from an Electric propeller coupled to CubeSat:
    - Delta-V = 6 km/s.

Figure 1. Electric thruster attached on CubeSat.



- **Payloads:**
  - Earth Mission: CubeSat + Electric Thruster;
  - Space Mission: CubeSat.

Definition of Basic Characteristics According to Market Research

Launch Vehicle	Bloostar	Electron	Neptune N1	Skyora XL	Taymyr-1A
Payload (kg)	75 - 140	150	20	320	12
Launch	Balloon	Ground	Ground	Ground	Ground
Orbit (km)	200 - 600	500	500	600	N.E.
Height (m)	N.A.	18	11	N.D.	16
Diameter(m)	N.A.	1.2	0.64	N.D.	0.32
Total Weight (kg)	5818	12500	2449	N.D.	2600
Specific Impulse (s)	355 (stage 3)	303 (stage 1) 333 (stage 2)	245 - 300 (stage 3)	290 - 306 (stage 3)	187 - 224 (stage 3)
Thrust (kN)	2	0.12	3.3 - 4.4	7	3.92
Propellant	Methane and liquid oxygen	Kerosene (RP-1) and liquid oxygen	White fuming nitric acid (WFNA) and turpentine	Hydrogen peroxide (HTP) and kerosene (RP-1)	Hydrogen peroxide (HTP)(PV-85) and aviation kerosene (TS-1)
Engine	Teide I	Rutherford	GPRE 0.75KNTA	N.E.	RLD-100S "Atar"

Table 1. Parameters of launch vehicles presents in market.

- **Specific Impulse :**
  - First stage = 303 s;
  - Second stage and higher = 333 s.
- **Propellant** = Kerosene RP-1;
- **Oxidizer** = Liquid Oxygen (LOX);
- **Payload** = 31 kg.

Sizing of the Launch Vehicle

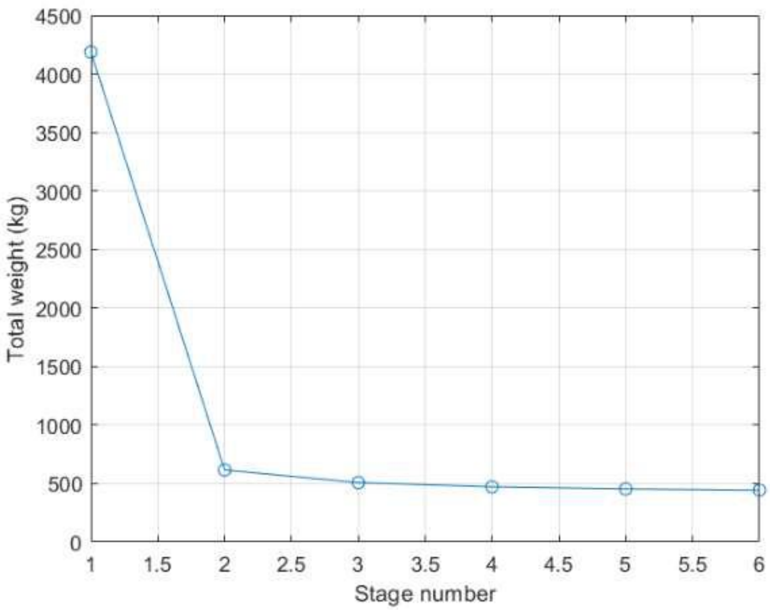


Figure 2. Relation between the stage number and total weight of the small launch vehicle

	Stage 1	Stage 2	Stage 3
Fuel weight (kg)	291.5	101.2	41.1
Structural weight (kg)	29.1	10.1	4.1
Payload weight (kg)	-	-	31
Total weight (kg)		508.1	

Table 2. Weight and stages number sizing for the small launch vehicle

Combustion Chamber Project

- **Input data for the *CEA* program *(Chemical Equilibrium with Applications):***
  - Combustion Chamber Pressure: 20 atm;
  - Propellant: Pure kerosene (RP-1);
  - Oxidizer: Pure Liquid Oxygen (LOX);
  - Oxidizing / Propellant Ratio: 2,56;
  - Expansion to infinite combustion area.
- **Requirement :**
  - First Stage : 303 s;
  - 2nd and 3rd stages: 333 s.
- **Design variable:**
  - Pc / Pe;

Combustion Chamber Project

- **1st Stage:**
  - Pc/Pe = 88;
- **Important data (CEA):**
  - Pe = 0,23 bar ⇒ 11 km Altitude;
  - Te = 2250,58 K;
  - Gammae = 1,1699;
  - Me = 3,187.
- **Performance and parameters goals for the 1st Stage:**
  - Throat isp = 1156.5 m / s or 117.93 s;
  - Exit Isp = 2977.8 m / s or 303.65 s;

Combustion Chamber Project

- **2nd and 3rd Stages:**
  - Pc/Pe = 334,5;
- **Important data (CEA) :**
  - Pe = 0,06 bar ⇨ > 40km;
  - Te = 1828,35 K;
  - Gammae = 1,2021;
  - Me = 3,834
- **Performance and parameters goals for the 2nd and 3rd Stage:**
  - Throat Isp = 1156.5 m/s or 117,88 s;
  - Exit Isp = 3266,1 m/s or 332,93 s;

Nozzle Project

NASA "MOC Nozzle Simulation"

- Internal inputs:
  - Exhaust outlet Mach;
  - Throat area;
  - Width;
  - Throat length;
  - Total pressure in the chamber;
  - Total temperature at the outlet;
  - Specific heat ratio (Gamma).
- External inputs:
  - External Mach;
  - Altitude;
  - Angle at the end of the nozzle.

Thrust Estimative

Electron

Weight:
- 1st stage: 12500 kg;
- 2st stage: 2300 kg.

Thrust:
- 1st stage: 192 kN;
- 2st stage: 22 kN.

T/W Ratio:
- 1st stage: 15,36 **N**/kg;
- 2st stage: 9,56 N/kg.

Our Rocket

Weight:
- 1st stage: 508,1 kg;
- 2st stage: 187,5 kg;
- 3rd stage: 76,2 kg.

Thrust:
- 1st stage: 7804 N;
- 2st stage: 1792,5 N;
- 3rd stage: 728,5 N.

T/W Ratio:
- 1st stage: 15,36 N/kg;
- 2st stage: 9,56 N/kg;
- 3rd stage: 9,56 N/kg.

Nozzle Project

- **Results :**

	Stage 1	Stage 2 e 3
Thrust (N)	7984.558	2130.7
Throat Area (cm^2)	25.81	6.4516
Throat Radius (cm^2)	2.8661	1.433
Exit Area (cm^2)	130.16	59.55
Exit Radius (cm^2)	6.4367	4.3539
Area Ratio	5.043	9.231
Nozzle Length (m)	0.592	1.338
Mass Flow Rate (kg/s)	4.456545	1.23649

Table 5. Estimate of nozzle design for small launch vehicle

2nd and 3rd stages are equal due to Software Limitation -> Throat Area (1 in^2).

Space Thruster Design

Market Research of Electric thrusters:

Name	I_{sp}	Estructural weight	Fuel weight	Thrust
IFM NANO THRUSTER	2000 - 6000 s	900 g	230 g	10 μN - 0.4 mN
IFM NANO THRUSTER SE	2000 - 6000 s	950 g	230 g	10 μN - 0.4 mN
IFM MICRO 100	1500 - 6000 s	3.2 kg	1.3 kg	75 μN - 1.5 mN
IFM MICRO 200	1500 - 6000 s	6.4 kg	2.6 kg	75 μN - 3 mN
IFM MICRO 400	1500 - 6000 s	12.8 kg	5.2 kg	75 μN - 6 mN
IFM MICRO 600	1500 - 6000 s	19.2 kg	7.8 kg	75 μN - 9 mN

Table 6. Possible thrusters for space mission

Space Thruster Design

By the Tsiolkovsky equation:

Requirement: 6 km / s

Name	ΔV (m/s)
~~IFM NANO THRUSTER~~	~~2699,93~~
~~IFM NANO THRUSTER SE~~	~~2673,27~~
IFM MICRO 100	9769,98
IFM MICRO 200	13134,22
IFM MICRO 400	15872,39
IFM MICRO 600	17059,68

Table 7. Delta-V of space thrusters

These thrusters are based on Field Emission Electric Propulsion (FEEP) technology.

Space Thruster Design

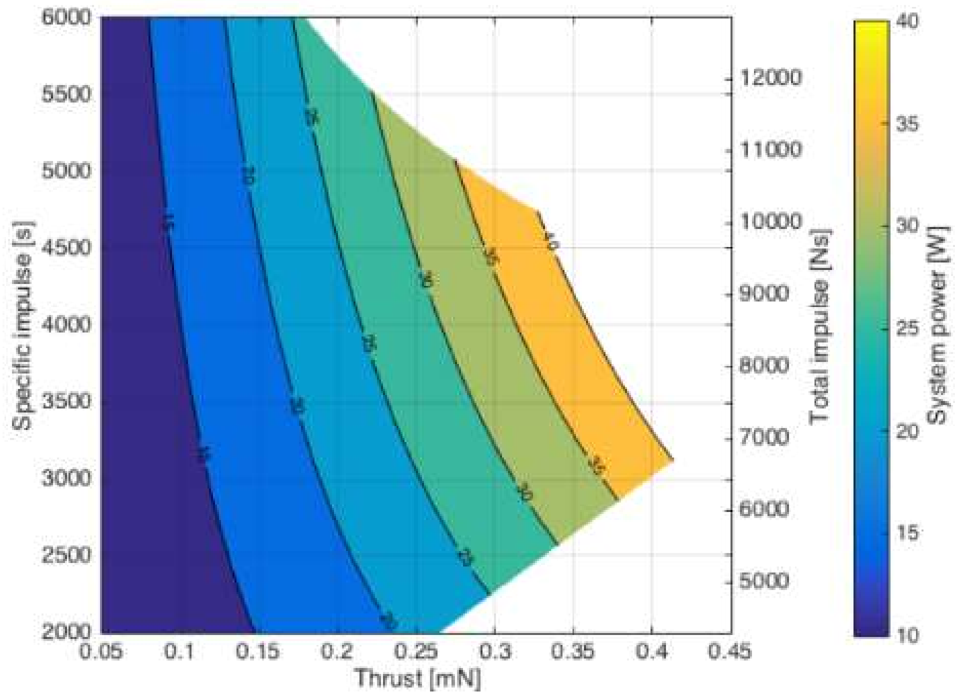


Figure 3. Thrust x Isp relation for IFM MICRO 100

Conclusion

- Option for a small dedicaded launch system, with liquid fuel and 3 stages, as well as electric thrusters for LEO-Mars path;
- Total mass for launch vehicle: 508.1 kg;
- Cost estimate for this mission: 787 Thousand USDs
- Required Isp and exit pressure were obtained successfully;
- Nozzles presented dimensional and performance values in accordance with the established requirements;
- Propulsion system design and analysis of use of electric propulsion shows feasibility in a preliminary analysis, with reduced costs and know-how required.

Obrigado!

Thank you!

Grazie!

ARTUR GUSTAVO SLONGO
arturgustavoslongo@gmail.com

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