Green Orbital Propulsion System for a Small Satellite

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1 Reference Case Definition

Most the orbital propulsion systems used for AOCS tasks don't need to provide a very high amount of Δv to the satellite bus. Therefore, lightweight and simple blow-down feed systems (see Figure 1) are usually implemented to supply the propellant to the engines. Usually an injector is used to evenly distribute the fuel within the combustion chamber leading to a pressure loss of $\Delta p_{\text{injector}}$ along the fuel line.

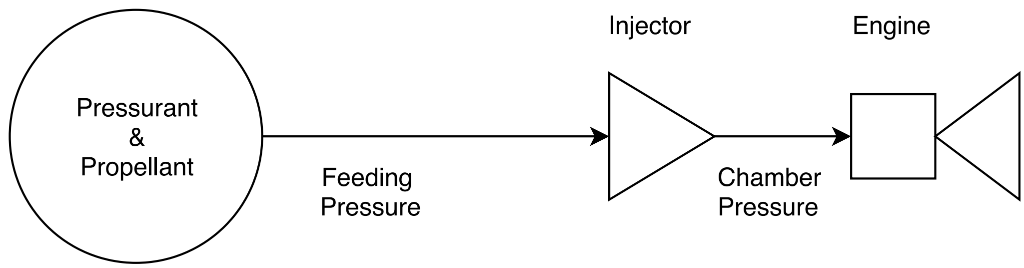

Figure 1: Simplified view of a pressure blow down system used to transport propellant to the engine. The pressurant and the propellant are separated by a diaphram.

We will assume the following injection pressure loss, since neither the chamber pressure p_c nor the pressure loss at the injector $\Delta p_{\text{injector}}$ are commonly cited in propulsion system specifications.

$$\Delta p_{\text{injector}} \approx \frac{1}{2} p_c \tag{1}$$

This will lead to an estimated chamber pressure using Equation (1) as shown in the following.

$$p_{\text{feed}} = p_c + \Delta p_{\text{injector}} \tag{2}$$

$$\rightarrow \quad p_c \approx \frac{2}{3} p_{\text{feed}} \tag{3}$$

During our internet research the propulsion systems in Table 5 were found. All systems use monopropellants and are used for AOCS tasks of the spacecraft. In the following we will use the **XMM Thruster** as a reference case using a feeding pressure of $p_{\text{feed}} = 5.5\text{bar} - 24\text{bar}$ and a nozzle expansion ratio of $\epsilon = 60$.

Case	Chamber Pressure [bar]	Expansion Ratio	Initial Propellant Temperature [K]
XMM Thruster	3.67-16	60	293.15

Table 1: Parameters needed for NasaCEA calculations based on the reference case of the XMM thruster system.

Using Equation (3) we can translate the feeding pressure of the XMM Thruster system to a chamber pressure of $p_c = 3.67\text{bar} - 16\text{bar}$. Furthermore, an initial propellant temperature of 20°C was assumed for all calculations. The reference case is summarized in Table 1.

2 Propellants Comparision

Using the parameters of the reference case in Table 1 the following green propellants in Table 2 were investigated using NASA CEA. The reaction products were set to frozen from the nozzle on outwards, to prevent further reaction in the nozzle as demanded in the task description.

Propellant	Vacuum Specific Impulse [s]	Combustion Temperature [K]
LMP-103S	253.2	1864-1865
AF-M315E	261	2102-2105
H2O2, 98%	188	1225

Table 2: Comparison of green propellants to the reference case. Each calculation is done using the minimum and maximum feeding pressure of the reference case. If only one result is displayed no difference was calculated between max. and min. chamber pressure.

3 Propellant Optimization

To optimize the propellant composition of Methanol, ADN and Water a rocketCEA script was written in Python. It performs a search on a composition grid, with a step size of 1% weight fraction. It iterates through all possible composition permutations. The input parameters were taken from the reference case in Table 1 using only the maximum chamber pressure. The results are discussed in the following.

All results The top figure in Figure 2 shows the result of the optimization with a maximum specific impulse in vacuum of $I_{sp} = 301.43s$ using 17% of Methanol, 83% ADN and 0% Water.

Final result The task demanded a combustion temperature below 1000°C (1273.15K). Hence, all results with a higher combustion temperature were removed from the lower figure of Figure 2. This lead to the final result with a maximum specific impulse in vacuum of $I_{sp} = 236.50s$ using 39% of Methanol, 59% ADN and 2% Water.

Furthermore, we need to consider that solid ADN will need to be dissolved in the other two components of the propellant. At a temperature of $20°C$, which is coherent with our initial propellant temperature, $s_{\text{ADN,H2O}} = 3.56$ times more Water than ADN is needed to completely dissolve the ADN. Methanol can dissolve much more ADN at a rate of only $s_{\text{ADN,Meth}} = 0.86$ [LW11]. The solubility s is defined as

$$s_{a,b} = \frac{m_b}{m_a}. \tag{4}$$

The used 39% Methanol can dissolve 45.35% of ADN. The remaining 13.65% of ADN would need to be dissolved by the propellants 2% of Water, which can only dissolve 0.56% of ADN at a temperature of 20°C. Therefore, the found propellant composition would contain 13.09% of solid ADN, which would make it unusable under real conditions.

Optimizing the propellant composition to comply with the mentioned solubility constraint is beyond the scope of this task and shall only be mentioned here.

4 Thruster Preliminary Design

The new propulsion system should provide the same amount of thrust as the reference case beeing 20N. With thrust F beeing defined as

$$F = I_{sp} \cdot \dot{m}_{\text{prop}} \cdot g \stackrel{!}{=} 20\text{N}.$$

Therefore, the needed mass flow rate $\dot{m}_{\text{prop}}$ is calculated using the earth's gravitational acceleration of $g = 9.81\frac{m}{s^2}$ and the determined specific impulse of task B3 of $I_{sp} = 236.5s$.

$$\dot{m}_{\text{prop}} = 0.00862\frac{\text{kg}}{\text{s}} = 8.62\frac{\text{g}}{\text{s}}$$

Furthermore, to reach the same amount of total impulse $I_{\text{total}} = 517000\text{Ns}$ (see Table 5) the propellant mass m_{prop} can be calculated to be

$$m_{\text{prop}} = \frac{I_{\text{Total}}}{F} \cdot \dot{m}_{\text{prop}} = 211.97\text{kg}$$

Using NASA CEA the complete output is computed for the propellant composition found in Section 3. It can be found on Page 7 to obtain the characteristic exhaust speed $c^* = 4240.4\frac{\text{ft}}{\text{s}} = 1292.5\frac{\text{m}}{\text{s}}$.From here we can calculate the needed throat area A_t to

$$A_t = \frac{c^* \cdot \dot{m}_{\text{prop}}}{p_c} = 6.96 \cdot 10^{-6}\text{m}^2.$$

Using the expansion ratio $\epsilon = 60$ this leads to a nozzle exit area A_e of

$$A_e = \epsilon * A_t = 4.18 \cdot 10^{-4}\text{m}^2.$$

5 Detailed Design Blow-Down Feed System

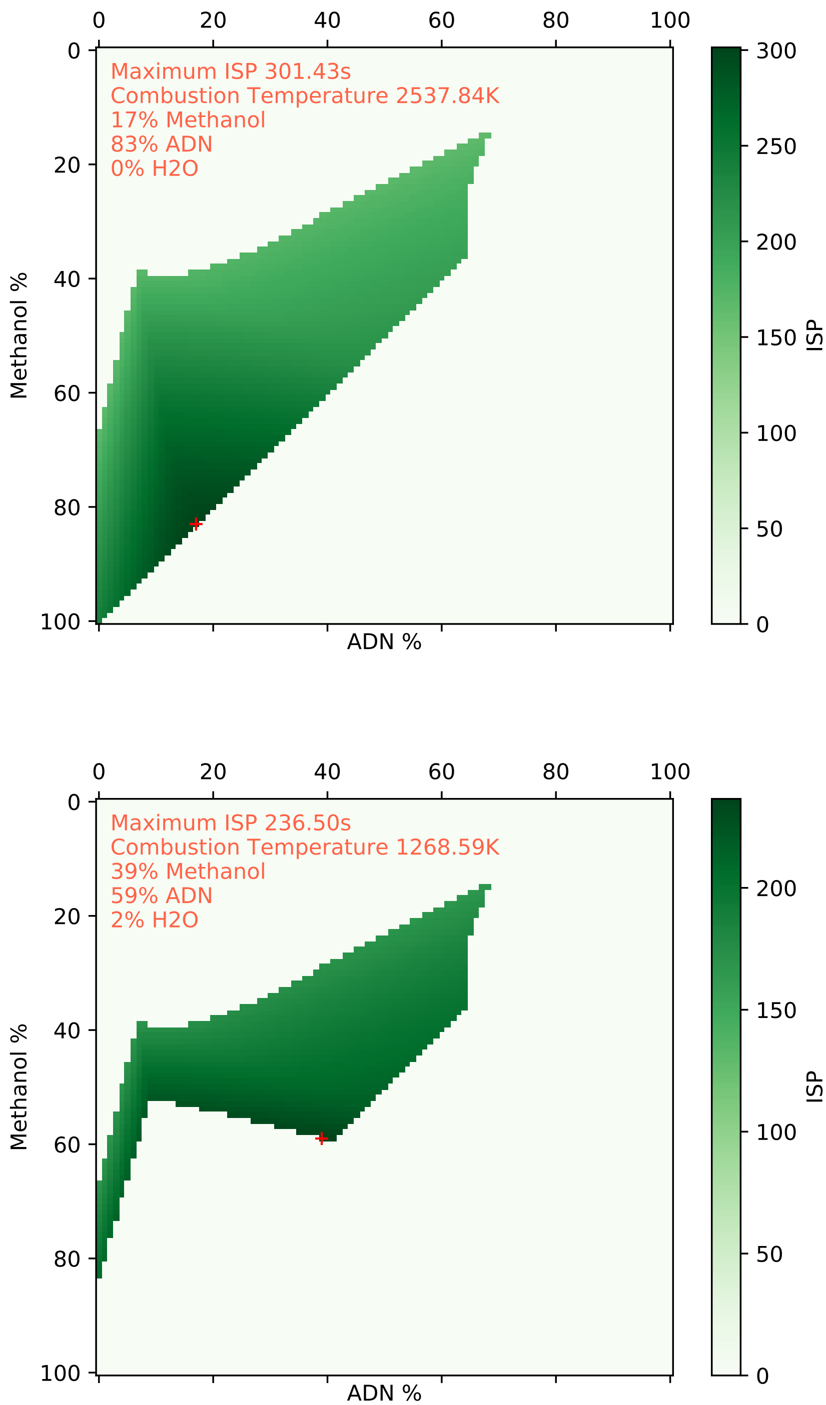

Figure 2: Figure showing the fuel optimization using Nasa CEA to compute the highest possible ISP (top). The best results are highlighted with a red cross. The second figure shows the best result after all reactions with a combustion temperature above 1000°C (1273.15K) are removed (bottom).

Name	Propellants	Number of Thrusters	Thruster Class [N]	Total Impulse [Ns]	Other	References
XMM Thruster	N2H4	8	20	> 517000	$I_{sp} = 222 - 230s$, $p_{\text{feed}} = 5.5 - 24\text{bar}$, $\epsilon = 60$	[19d] [19c]
TanDEM-X Thruster	N2H4	4	1	> 135000	$I_{sp} = 200 - 223s$, $p_{\text{feed}} = 5.5 - 22\text{bar}$, $\epsilon = 80$	[19e] [19b]
Prisma Thruster	LMP-103S	2	1	≈ 108773	$I_{sp} = 204 - 231s$, $p_{\text{feed}} = 4.5 - 22\text{bar}$, $\epsilon = 100$	[19a]

6 NASA CEA Output for Optimized Propellant

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        NASA-GLENN CHEMICAL EQUILIBRIUM PROGRAM CEA, OCTOBER 18, 2002
                   BY  BONNIE MCBRIDE AND SANFORD GORDON
      REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996

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reac
 name H2O    H 2 O 1     wt%=2.0
 h,kj/mol=-285.8          t(k)=293.15
 name Methanol    C 1 H 4 O 1    wt%=39.0
 h,kj/mol=-239.2          t(k)=293.15
 name ADN    H 4 N 4 O 4    wt%=59.0
 h,kj/mol=-134.6          t(k)=293.15

prob  case= WaterMethanolADN_Mix
 rocket  frozen  nfz=2    p,psia=232.064000,  supar=60.000000,


outp    calories  short

end




          THEORETICAL ROCKET PERFORMANCE ASSUMING FROZEN COMPOSITION
                               AFTER POINT 2

 Pinj =    232.1 PSIA
 CASE = WaterMethanolAD

              REACTANT                          WT FRACTION      ENERGY
TEMP
                                                 (SEE NOTE)      CAL/MOL
K
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name H2O 2.0000000 −68307.839
293.150
name Methanol 39.0000000 −57170.172
293.150
name ADN 59.0000000 −32170.172
293.150

O/F= 0.00000 %FUEL=100.000000 R,EQ.RATIO= 1.835864 PHI,EQ.RATIO= 0.000

	CHAMBER	THROAT	EXIT
Pinf/P	1.0000	1.8063	1440.10
P, ATM	15.791	8.7422	0.01097
T, K	1541.74	1366.53	259.05
RHO, G/CC	2.2036−3	1.3764−3	9.1073−6
H, CAL/G	−924.68	−1021.35	−1538.01
U, CAL/G	−1098.23	−1175.16	−1567.17
G, CAL/G	−5489.22	−5067.15	−2304.96
S, CAL/(G)(K)	2.9606	2.9606	2.9606
M, (1/n)	17.654	17.655	17.655
Cp, CAL/(G)(K)	0.5536	0.5521	0.4138
GAMMAs	1.2555	1.2569	1.3736
SON VEL,M/SEC	954.8	899.4	409.4
MACH NUMBER	0.000	1.000	5.534

PERFORMANCE PARAMETERS

Ae/At		1.0000	60.000
CSTAR, FT/SEC		4240.4	4240.4
CF		0.6959	1.7528
Ivac,LB−SEC/LB		164.7	236.5
Isp, LB−SEC/LB		91.7	231.0

MOLE FRACTIONS

CH4 0.00004 *CO 0.14957 *CO2
0.06529
*H2 0.32692 H2O 0.29022 NH3
0.00007
*N2 0.16790

* THERMODYNAMIC PROPERTIES FITTED TO 20000.K

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANT

References

[19a] "1 N HPGP Thruster". In: Available at http://ecaps.space/products-1n.php. June 2019.

[19b] "1N Monopropellant Hydrazine Thruster". In: Available at http://www.space-propulsion.com/spacecraft-propulsion/hydrazine-thrusters/1n-hydrazine-thruster.html. June 2019.

[19c] "20N Monopropellant Hydrazine Thruster". In: Available at http://www.space-propulsion.com/spacecraft-propulsion/hydrazine-thrusters/20n-hydrazine-thruster.html. June 2019.

[19d] "ESA Achievements BR 250". In: Available at http://www.esa.int/esapub/br/br250/br250.pdf. June 2019.

[19e] "Flight Dynamics Operations of the TanDEM-X Formation". In: Available at https://www.researchgate.net/profile/Ralph_Kahle/publication/259897316_Flight_Dynamics_Operations_of_the_TanDEM-X_Formation/links/0a85e53c3b917d441a000Flight-Dynamics-Operations-of-the-TanDEM-X-Formation.pdf. June 2019.

[LW11] Anders Larsson and Niklas Wingborg. "Green Propellants Based on Ammonium Dinitramide (ADN)". In: *Advances in Spacecraft Technologies*. InTech, Feb. 2011. DOI: 10.5772/13640. URL: https://doi.org/10.5772/13640.