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MKWS Project

Thrust determination for simple model of a liquid rocket engine powered by propane - oxygen mixture

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Contents

1. Introduction

The purpose of this project is to create a simple program in Python using Cantera environment to calculate thrust of a liquid rocket engine propelled with propane and oxygen. Simulations were done for different initial conditions in propellant tanks, but engine's elements geometry were treated as fixed (dimensions of nozzle throat and area of injectors was the same for each case).

2. Model

2.1. General background

Chemical rocket propulsion systems are generally divided into three subgroups: solid, hybrid and liquid. Although liquid engines are definitely the most complicated ones in this group, they utilizing simpler and more common substances as a fuel and oxidizer. For instance as a fuel hydrogen can be used (for a years the most popular propellant in space applications) as well as hydrocarbons and LOX as an oxidizer agent, which is much simpler from chemical point of view than complicated mixtures based on ammonium perchlorate and HTPB in solid motors. In this project propane was chosen as a fuel and oxygen as an oxidizer. Propane is not the most popular hydrocarbon to be used in liquid rocket engine, but it has been used in some applications in the past. It also has some advantages that can make it attractive option in some future. In general in liquid rocket engine, fuel and oxidizer are stored separately in liquid form in tanks and they are injected into combustion at certain pressure where the mixing and burning are taking place (at least in bipropellant engine with no pre-mix chamber).

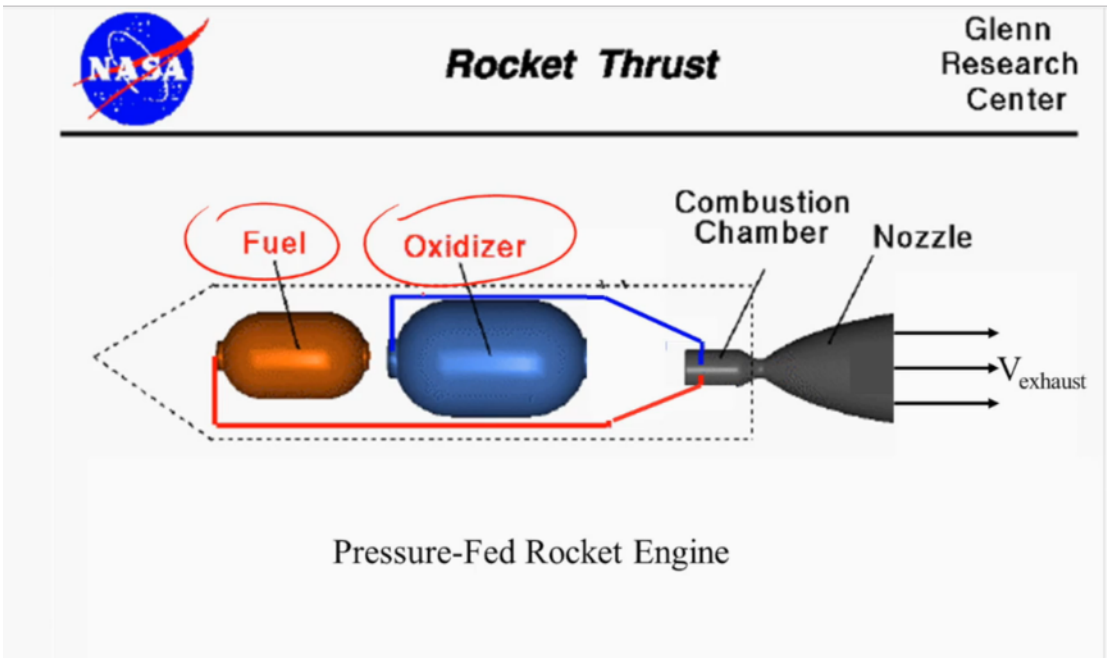

Figure 2.1. Liquid rocket engine

2.2. Physical model

To create first simple but useful model of combustion to implement it in Cantera following assumption was made:

- mass flow rate from tanks to combustion chamber is constant
- flow of exhaust gases is isentropic
- pressure at the exhaust is equal to atmospheric
- all properties of combustion products are the same in whole combustion chamber
- mixture is stechiometric

With this assumptions model, implemented in code was created in following way:
Firstly, we are defining all needed reactants and their initial parameters: pressure, temperature and chemical composition. Initial pressure and temperature are variables that changes with analysed cases. Then, we are defining geometric parameters of an engine, that specify mass flow rates, so injectors area and throat area of a nozzle. We are also defining volume of combustion chamber. Combustion chamber is set as ideal gas reactor which is filled with an oxygen at the beginning. Defined functions are calculating k coefficient, necessary to calculate exit gases velocity, as well as critical flow (mass flow rate). Simulation is set for relatively short time as such parameter as thrust will be constant after short time, as mass flow rates from tanks (defined as reservoirs) are constant - as well as nozzle geometry.
Values of constant geometric parameter are specified below (values are more or less based on the student class experimental rocket engines:

- $A_{throat} = 5 * 10^{-4} m^2$
- $A_{infuel} = 4 * 10^{-5} m^2$
- $A_{inox} = 4 * 10^{-5} m^2$

Thrust is calculated using standard rocket thrust equation:

$$T = \dot{m} * V_{exit} + (p_{exit} - p_{atm}) * A_{exit}$$

Since we assumed that full decompression of gases in the nozzle (that would be achieved by designing appropriate outer diameter of the nozzle) the second component in formula presented above in irrelevant.
Thrust was calculated then in the following way:

$$T = \dot{m} * V_{exit}$$

Whee mass flow rate is calculated by multiplying density, exhaust gasses velocity and appropriate nozzle area.

3. Results

As the result thrust, mas flow rate and velocity of exhaust gases will be plotted for 6 different combinations of initial parameters in propellant tanks:

1. oxidizer: 350K, 75atm
   fuel: 500K, 60atm

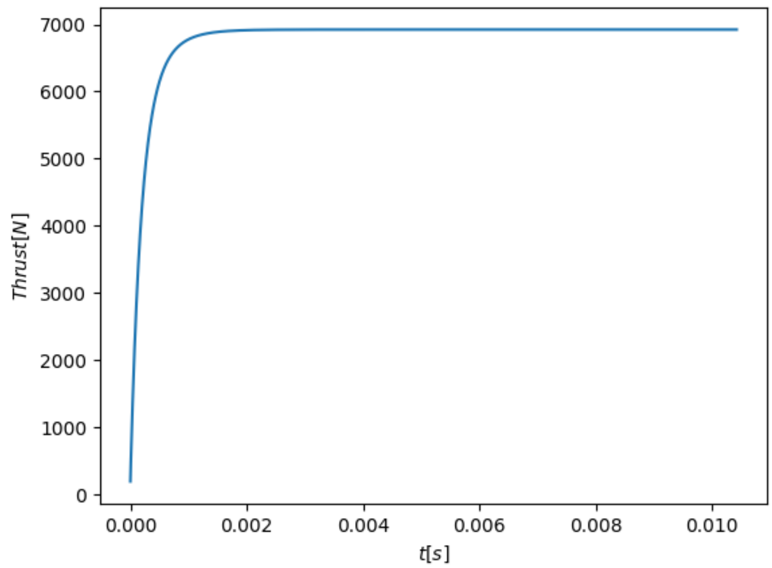

Figure 3.1. Thrust - 1st case

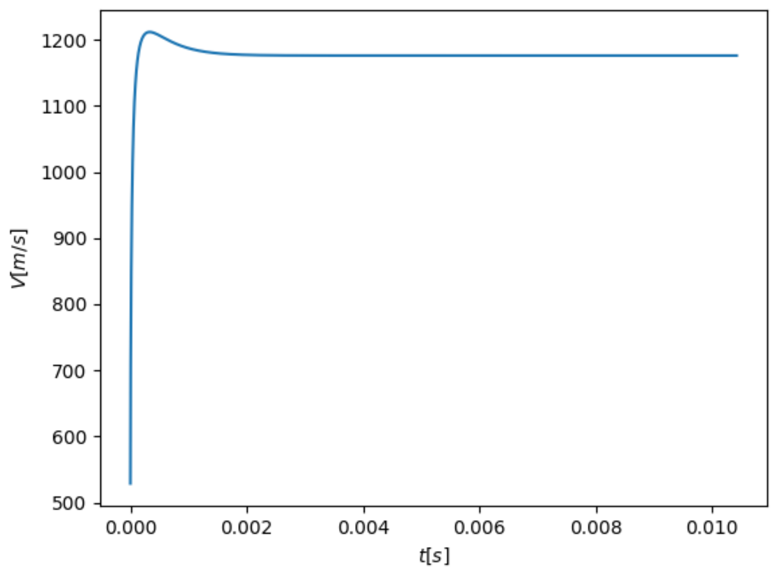

Figure 3.2. Exhaust gases velocity - 1st case

2. oxidizer: 300K, 50atm
   fuel: 300K, 50atm

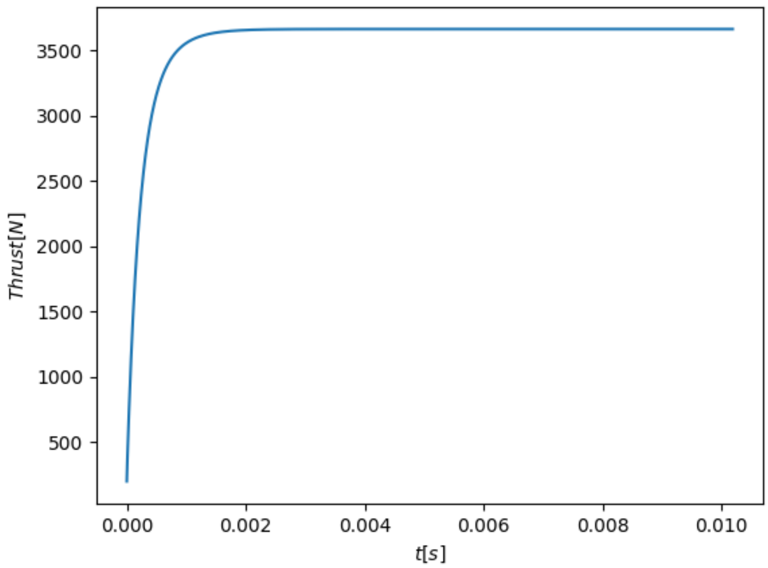

Figure 3.3. Thrust - 2nd case

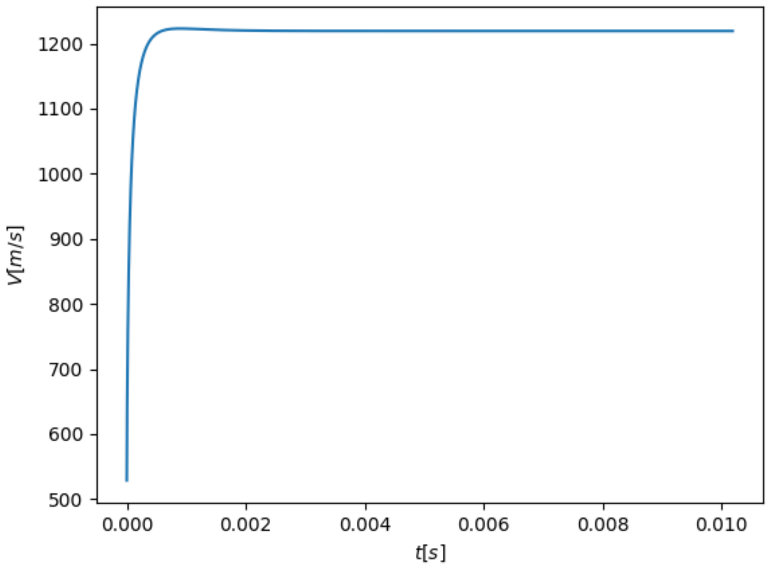

Figure 3.4. Exhaust gases velocity - 2nd case

6

3. oxidizer: 400K, 60atm
   fuel: 400K 20atm

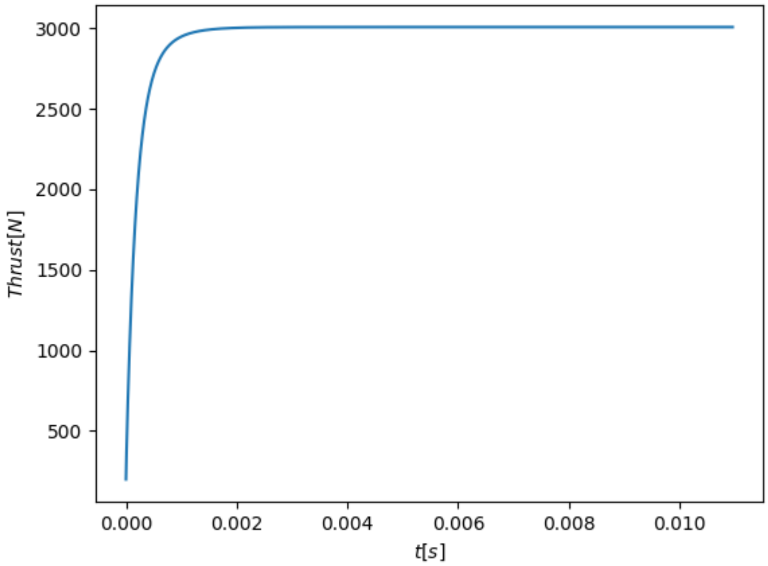

Figure 3.5. Thrust - 3rd case

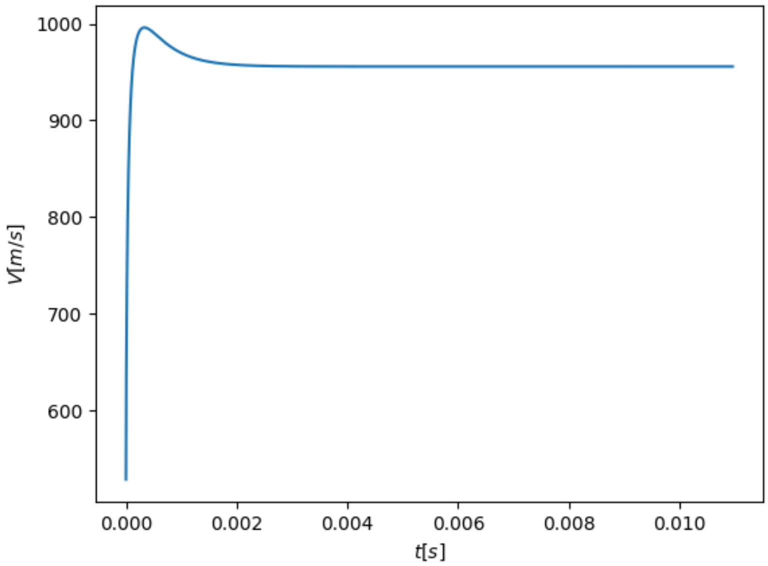

Figure 3.6. Exhaust gases velocity - 3rd case

7

4. oxidizer: 400K, 20atm
   fuel: 400K, 60atm

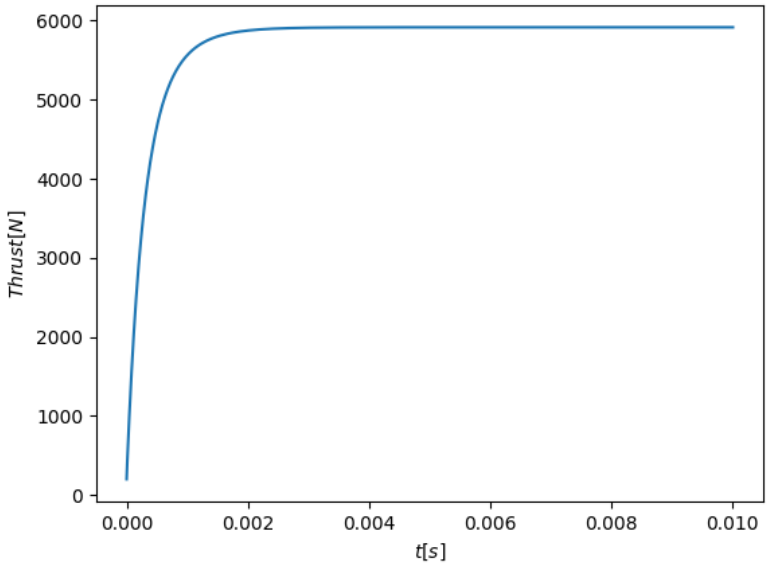

Figure 3.7. Thrust - 4th case

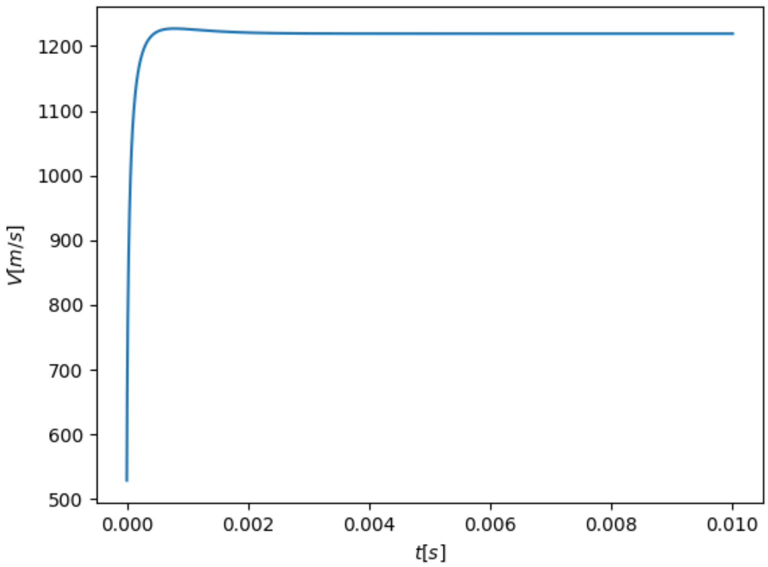

Figure 3.8. Exhaust gases velocity - 4th case

8

5. oxidizer: 2730K, 50atm
   fuel: 273K, 50atm

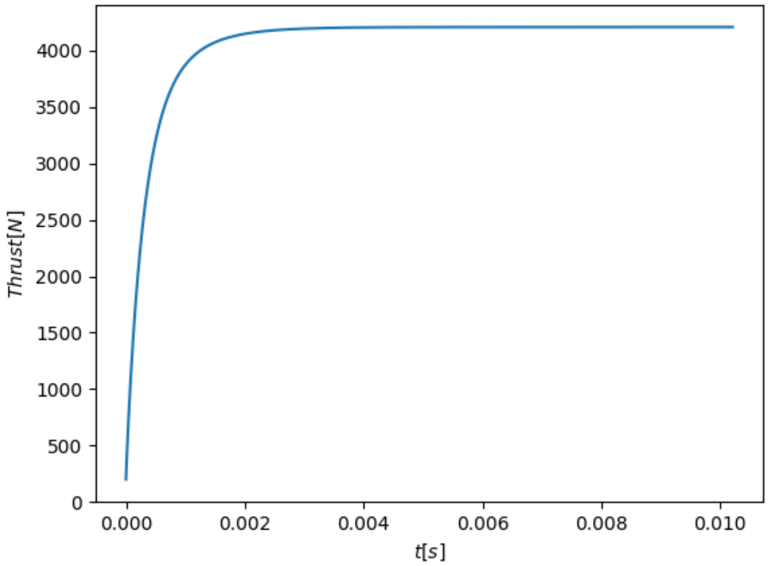

Figure 3.9. Thrust - 5th case

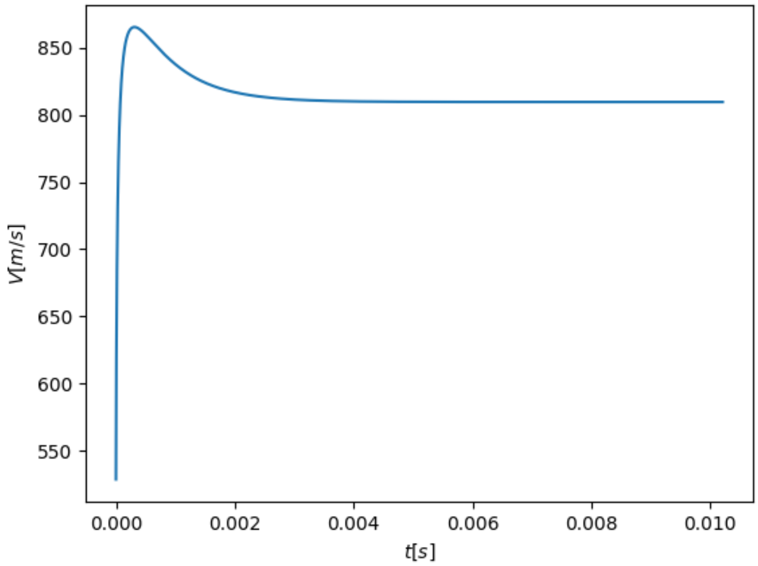

Figure 3.10. Exhaust gases velocity - 5th case

9

6. oxidizer: 273K, 50atm
   fuel: 273K, 50atm

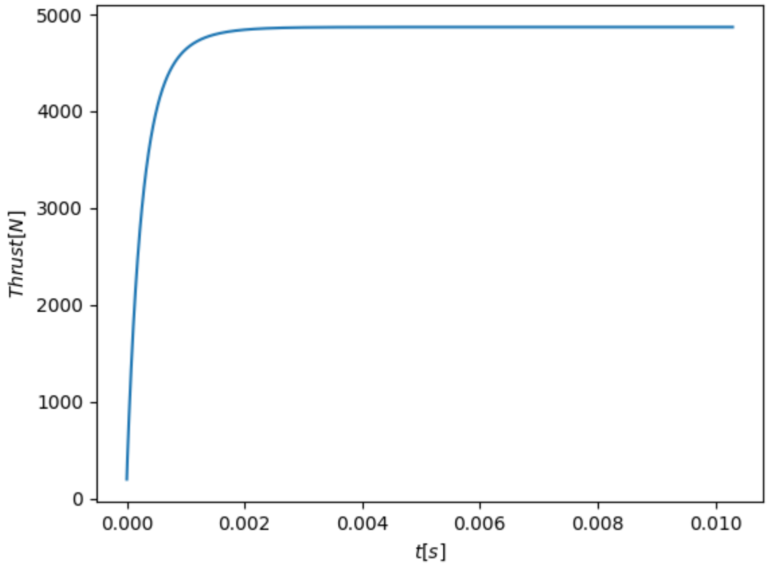

Figure 3.11. Thrust - 6th case

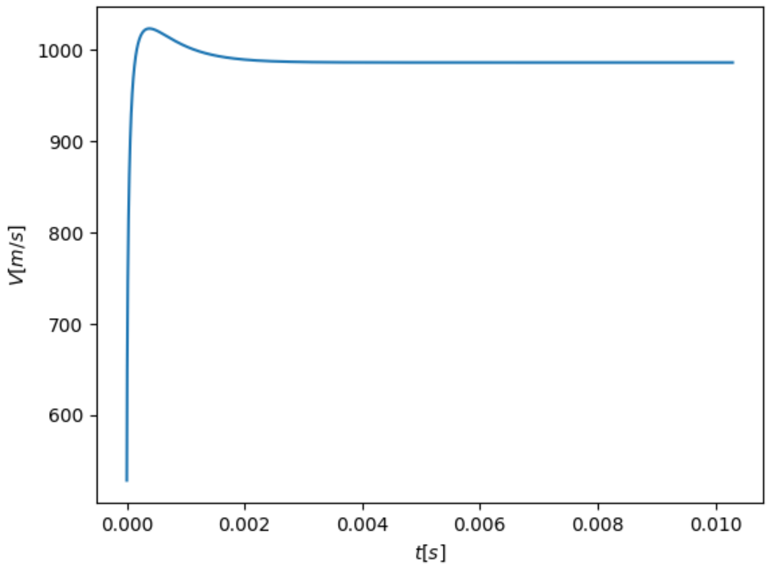

Figure 3.12. Exhaust gases velocity - 6th case

10

4. Conclusion

As we can see on plots presented in the report, the basic initial parameters in the propellant tanks make significant difference in overall performance of a liquid rocket engine. All values that was taken into account in simulations were more or less close to values of parameters that can be real initial parameters in the liquid rocket engine tanks.
In general, increasing the pressure changes mass flow rate through injector which results in higher thrust. Interesting comparison can be made between case 3 and 4, were initial temperature in both tanks were the same, but in one case pressure in one of the tanks were 3 times higher than in the other one. We can observe that in the case were pressure was higher in fuel tank, thrust is almost twice higher that in the other case. Difference is also visible on the related parameter - velocity of the gases.
Comparison of 2 cases with the same initial pressure in tanks but different temperatures (2 last cases) shows also that higher initial temperature of fuel and oxidizer in tanks correlate with higher thrust that can be obtained. Obviously to prove that a lot more simulation should be performed with much more precise model.

References

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[4] ***CANTERA Tutorials*** *A series of tutorials to get started with the python interface of Cantera version 2.1.1 by Anne Felden*

[5] Maciej Wójtowski, Combustion of Hydrogen-Oxygen mixture in rocket engine at various initial conditions

List of Symbols and Abbreviations

List of Figures

List of Tables

List of Appendices