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 do not 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.

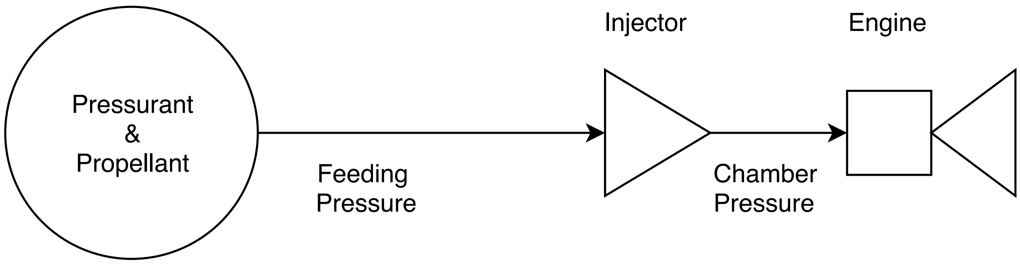

Figure 1: Simplified view of a pressure blow down system used to transport propellant to the engine. The pressurant and the propellant are separated by a diaphragm.

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 3 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 NASA CEA 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 (see Appendix B). 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 4 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 4. 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.

Additionally, 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, 356g ADN can be dissolved in 100g of Water and 86g ADN can be dissolved in 100g Methanol [LW11].

According to this data, the used 39% Methanol can dissolve 33.54% of ADN. The remaining 25.46% of ADN would need to be dissolved by the propellants 2% of Water, which can only dissolve 7.12% of ADN at a temperature of 20°C. Therefore, the found propellant composition would contain 18.34% of solid ADN rendering it unusable in real engines, because the solids are likely to cause problems in the feeding systems and would lead to inhomogenous mixtures in the combustion chamber.

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 being 20N. With thrust F being defined as

$$F = I_{sp} \cdot \dot{m}_{\text{prop}} \cdot g \overset{!}{=} 20\text{N}. \tag{4}$$

Therefore, the required 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 in task B3 determined specific impulse of $I_{sp} = 236.5s$.

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

To reach the same amount of total impulse $I_{\text{total}} = 517000\text{Ns}$ (see Table 3) 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 11 to obtain the characteristic exhaust speed $c^* = 4240.4\frac{\text{ft}}{\text{s}} = 1292.5\frac{\text{m}}{\text{s}}$.From here we are able to calculate appropriate throat area A_t to

$$A_t = \frac{c^* \cdot \dot{m}_{\text{prop}}}{8 \cdot p_c} = 6.96\text{mm}^2. \tag{5}$$

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

$$A_e = \epsilon * A_t = 4.17\text{cm}^2.$$

5 Detailed Design Blow-Down Feed System

Combining Equation 4 and Equation 5 the produced thrust of the optimized propellant can be studied for varying chamber pressures p_c as seen in Equation 6.

$$F = I_{sp} \cdot g \cdot \frac{8 \cdot A_t \cdot p_c}{c^*} \tag{6}$$

We want the engine to retain at least a quarter of its thrust, being $F_{\text{EOL}} = 5\text{N}$, at the end of its life. According to our calculations (see Figure 2) the engine will linearly loss thrust with

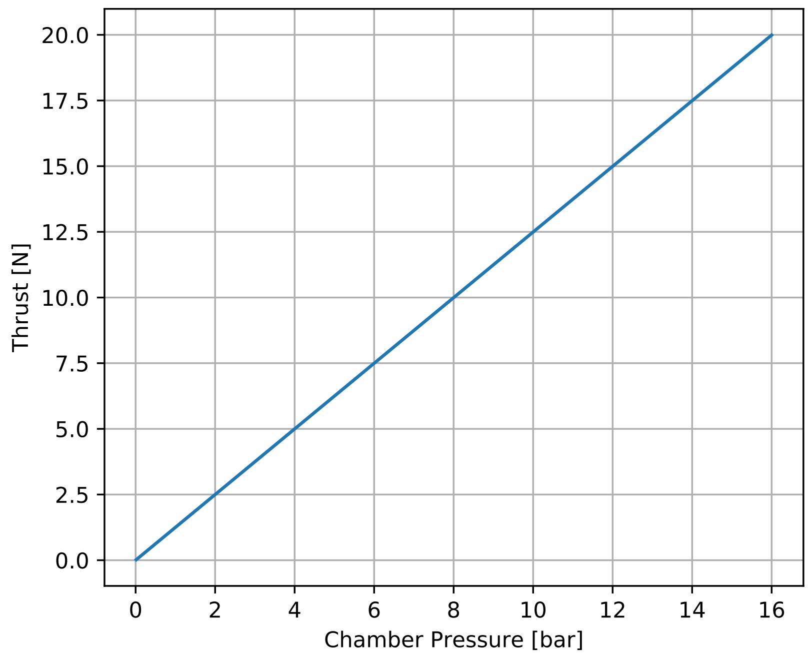

Figure 2: The optimized propellants thrust in vacuum at different chamber pressures.

decreasing chamber pressures. Therefore, an end of life chamber pressure of $p_{c,\mathrm{EOL}} = 4\mathrm{bar}$ is defined. Furthermore, the chamber pressure at mission launch can defined to be the same as in the reference case $p_{c,\mathrm{BOL}} = 16\mathrm{bar}$.

Since it is the simplest and cheapest design, a blow-down feed system is used to feed the propellant to the engine. Assuming the same pressure loss at the fuel injector $\Delta p_{\mathrm{injector}} = \frac{p_c}{2}$ as in Section 1 leads to feeding pressures of $p_{feed,\mathrm{BOL}} = 24\mathrm{bar}$ and $p_{feed,\mathrm{EOL}} = 6\mathrm{bar}$.

To achieve an even greener propulsion system, we decided to use Nitrogen instead of Helium to pressurize the propellant. Considering the global shortage of Helium this will ensure a sustainable propulsion system for years to come. The final feeding system architecture can be seen in Figure 3.

In order to Calculate the amount of needed pressurant, we will need to estimate the density of the used propellant. According to this lectures exercise 3 from the 27.06.2019, the green propellant LMP-103S has a density of $\rho_{\mathrm{LMP}} = 1250\frac{kg}{m^3}$. Its composition is quite similar to the here optimized propellant. Thus, the propellants density is estimated to be

$$\rho_{\mathrm{prop}} \approx \rho_{\mathrm{LMP}} = 1250\frac{kg}{m^3}.$$

This leads to a propellant volume of

$$V_{\mathrm{prop}} = m_{\mathrm{prop}}/\rho_{\mathrm{prop}} = 0.1696\mathrm{m}^3.$$

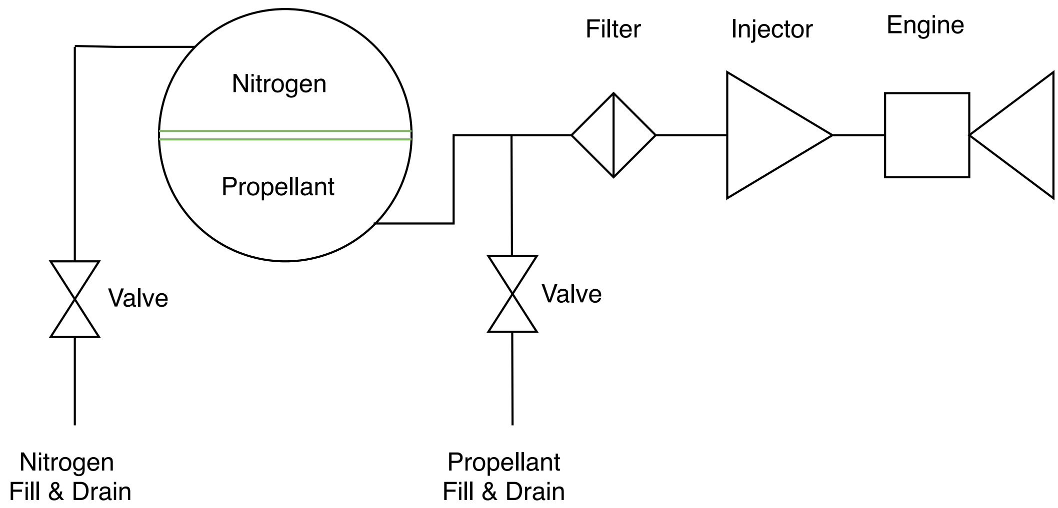

Figure 3: Final design of the blow down feeding system for the green orbital propulsion system.

Now the needed amount of pressurant can be calculated using the ideal gas equation and taking into account that the tank volume is constant. Moreover, we assume that the propellant is only released in short bursts which will keep the propellant temperature constant at the initial temperature of $T_0 = 20°\text{C}$.

$$V_{\text{tank}} = \frac{m_{N2} R_{N2} T_0}{p_{feed,\text{BOL}}} + V_{\text{prop}} = \frac{m_{N2} R_{N2} T_0}{p_{feed,\text{EOL}}} \tag{7}$$

Using the molecular mass of Nitrogen $M_{N2} = 28.01340 \frac{\text{g}}{\text{mol}}$, its specific ideal gas constant can be determined to be $R_{N2} = 296.78 \frac{\text{J}}{\text{kg K}}$. Solving Equation 7 results in a pressurant mass of

$$m_{N2} = 1.56kg.$$

In comparison, using Helium with a specific ideal gas constant of $R_{He} = 2078.5 \frac{\text{J}}{\text{kg K}}$, we would only need $m_{He} = 0.223kg$ of pressurant.

Then, we can calculate the dimensions of the combined pressurant and propellant tank using Equation 3 to be

$$V_{\text{tank}} = 0.226\text{m}^3.$$

Which leads to a tank radius r, assuming the use of a spherical tank, of

$$r = \sqrt[3]{\frac{3}{4\pi} V_{\text{tank}}} = 0.377m$$

Using Titanium with a maximum allowed operating stress of $S_{Ti} = 6894.75\text{bar}$ we can estimate the tank thickness t with a safety factor of 2 to

$$t = \frac{p_{feed,\text{BOL}} \cdot r}{4 \cdot S_{Ti}} = 0.329\text{mm}$$

Finally the empty tank mass is calculated using a Titanium density of $\rho = 4430\frac{\text{kg}}{\text{m}^3}$

$$m_{\text{tank}} = \rho_{Ti} \cdot \frac{4}{3}\pi((r+t)^3 - r^3) = 2.605kg$$

The complete system mass will further need to include several engines, valves, fuel lines and filters. Estimating a mass of the residual parts of $m_{\text{res}} \approx 15\text{kg}$, the complete system mass adds up to

$$m_{\text{system}} = m_{\text{tank}} + m_{\text{prop}} + m_{\text{N2}} + m_{\text{res}} = 231.135\text{kg}.$$

Since the system has a rather high amount of propellant it should be able to propell the spacecraft for longer mission durations in orbit.

6 Conclusion

As shown in Section 4 and 5 we were able to obtain a preliminary design for a green orbital propulsion system. The used propellant has a higher specific impulse in vacuum of $I_{sp} = 236.6\text{s}$ and most probably a higher density of $\rho \approx 1250\frac{kg}{s}$, although this is only an estimation. In comparison, the reference case uses Hydrazine, with a density of $\rho_{\text{N2H4}} = 1020\frac{kg}{s}$ at $T_0 = 20°\text{C}$, achieving around $I_{sp} = 222\text{s} - 230\text{s}$ of specific impulse in vacuum [19d].

The designed system needs less propellant saving around 78kg in propellant mass. Therefore, we could allow for a heavier pressurant, using Nitrogen instead of Helium, since the target was to create a sustainable green propulsion system, which made a difference of around $1.3kg$.

Furthermore, the designed thruster is only half as wide in diameter compared to the reference case thruster (comparing Section 4 to Table 3). This combined with the higher propellant density possibly result in a smaller overall build volume.

Overall, the new system is void of difficult to handle Hydrazine, performs better at a lower mass and volume and comes with a similar combustion temperature as a common Hydrazine thruster. Yet, it shall be noted that the proposed propellant composition might not be usable in real world conditions due to the unresolved solubility issues with the ADN in the used liquids. Therefore, the designed thruster might fail in operation.

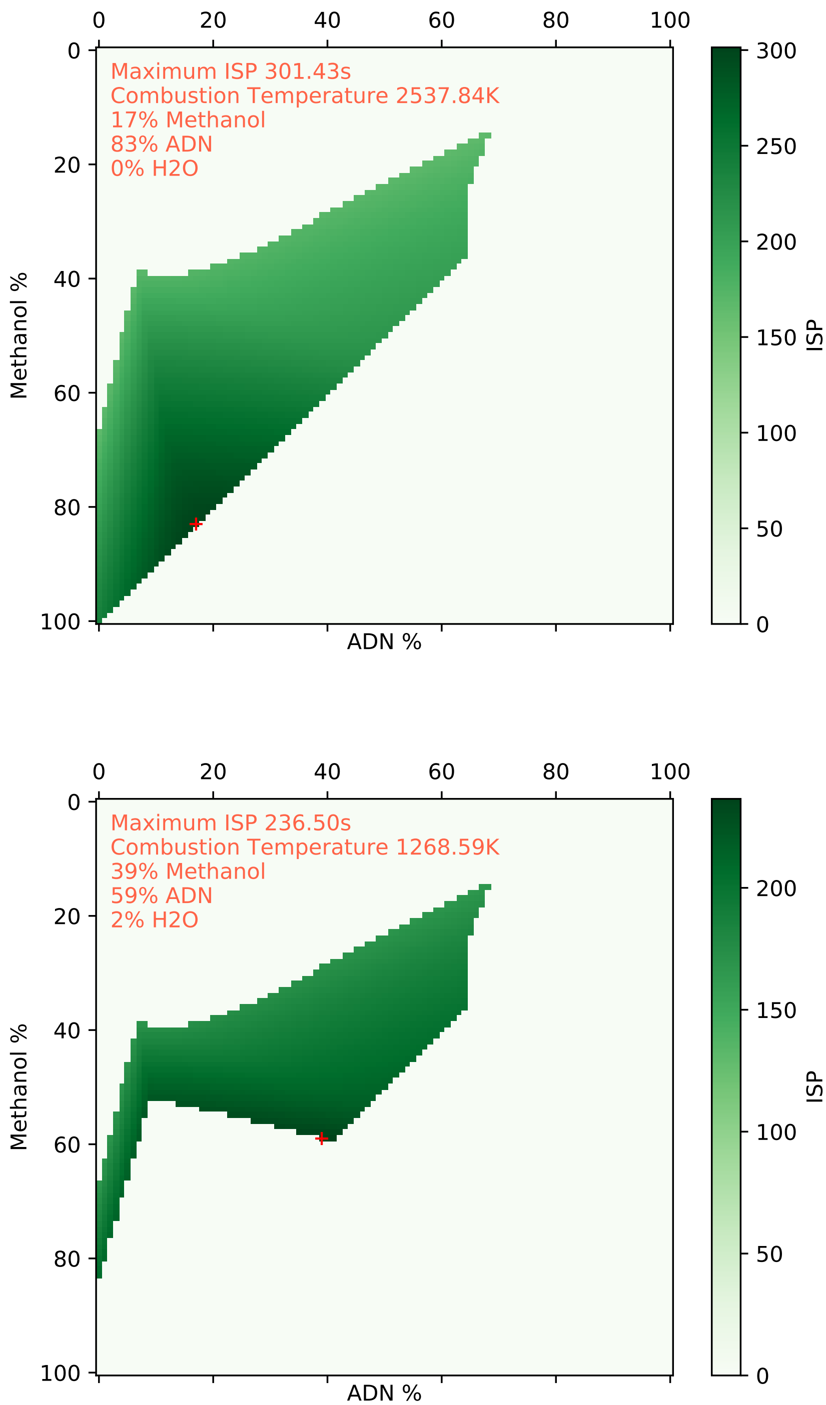

Figure 4: Figure showing the fuel optimization using Nasa CEA to compute the highest possible ISP (top). The best results are highlighted with a red cross. The second figure shows the best result after all reactions with a combustion temperature above 1000°C (1273.15K) are removed (bottom).

Table 3: Overview of small propulsion systems using monopropellants.

Name	Propellants	Number of Thrusters	Thruster Class [N]	Total Impulse [Ns]	Other	References
XMM Thruster	N2H4	8	20	> 517000	$I_{sp} = 222 - 230s$, $p_{\text{feed}} = 5.5 - 24\text{bar}$, $\epsilon = 60$ $m_{\text{prop}} = 290\text{kg}$ $A_t = 14.25\text{mm}^2$ $A_e = 8.55\text{cm}^2$	[19e] [19c]
TanDEM-X Thruster	N2H4	4	1	> 135000	$I_{sp} = 200 - 223s$, $p_{\text{feed}} = 5.5 - 22\text{bar}$, $\epsilon = 80$	[19f] [19b]
Prisma Thruster	LMP-103S	2	1	≈ 108773	$I_{sp} = 204 - 231s$, $p_{\text{feed}} = 4.5 - 22\text{bar}$, $\epsilon = 100$	[19a]

A NASA CEA Output for Optimized Propellant

```
 *******************************************************************************

        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

 *******************************************************************************



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
 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=
    0.000000

                 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 OXIDANTS
```

results/b4.out

B rocketCEA Code for Fuel Optimization

```python
import matplotlib as mpl
mpl.use("Agg")
import matplotlib.pyplot as plt
from rocketcea.cea_obj import CEA_Obj, add_new_propellant
import numpy as np


""" define functions """
def kilojoule2cal(kilojoule):
  return kilojoule/4.184*1000

def bar2psi(bar):
  return bar*14.504

def createPropellant(wt1,wt2,wt3):
```

```
  card_str = """
  name H2O   H 2 O 1    wt%={:.1f}
  h,kj/mol=-285.8        t(k)=293.15
  name Methanol   C 1 H 4 O 1    wt%={:.1f}
  h,kj/mol=-239.2        t(k)=293.15
  name ADN   H 4 N 4 O 4    wt%={:.1f}
  h,kj/mol=-134.6        t(k)=293.15
  """.format(wt1,wt2,wt3)
  add_new_propellant("WaterMethanolADN_Mix",card_str)

def calculatePerformance(chamber_pressure,expansion_ratio):
  # chamber pressure in bar
  temp = CEA_Obj(propName="WaterMethanolADN_Mix")
  (isp,cstr,tc) = temp.getFrozen_IvacCstrTc(Pc=bar2psi(chamber_pressure),eps=expansion_ratio,frozenAtThroat=1)
  tc /= 1.8 # convert from rankine to kelvin
  return (isp, tc)

def cost(tc,isp):
  """ Target: Maximize ISP, stay below 1000C chamber temperature """
  if tc >= 1273.15:
    # greatly increase cost if temperature is to high
    return tc/isp
  else:
    # if below 1000C, invert ISP because optimizer will minimize
    return 1/isp



""" search for good composition """
chamber_pressure = 16 # bar
expansion_ratio = 60

# init results
results = []

# grid search
for wt1 in range(0,101,1):
  for wt2 in range(0,101-wt1,1):
    wt3 = 100 - wt1 - wt2
    try:
      # create new mixture
      createPropellant(wt1,wt2,wt3)
      # calc performance
      isp, tc = calculatePerformance(chamber_pressure,expansion_ratio)
      if isp != 0.0:
        # save in to results list
        results.append((wt1,wt2,wt3,isp,tc))
        # print
        print('Calculating Mixture of ',wt1,wt2,wt3,' WATER/METHANOL/ADN, ISP @',isp)
    except:
      print('Invalid mixture of ',wt1,wt2,wt3)

```

```python
# stack list
results = np.vstack(results)

# delete zero isp mixtures
nonzero_row_indices =[i for i in range(results.shape[0]) if not results[i,3]==0]
data = results[nonzero_row_indices,:]

# save results
np.savetxt('b3/gridsearch.txt',results, header='#wt1\twt2\twt3\tisp[s]\ttc[K]')

""" plotting """
# create data matrices
isp_mat = np.zeros([101,101])
tc_mat = np.zeros([101,101])
tc_mask = np.zeros([101,101])
for i in range(results.shape[0]):
  x = int(results[i,2])
  y = int(results[i,1])
  isp_mat[x,y] = results[i,3]
  tc_mat[x,y] = results[i,4] - 273.15
  if tc_mat[x,y] < 1273.15:
    tc_mask[x,y] = 1

# FIRST PLOT
# init plot
fig = plt.figure()
ax = fig.add_subplot(111)

# plot isp
im = ax.matshow(isp_mat, cmap='Greens')
cbar = ax.figure.colorbar(im, ax=ax)
cbar.ax.set_ylabel("ISP", rotation=90, va="top")

# add labels
ax.set_xlabel('ADN %')
ax.set_ylabel('Methanol %')

# add max
max_isp = np.max(isp_mat)
arg_max_isp = np.unravel_index(np.argmax(isp_mat, axis=None), isp_mat.shape)
t_at_max_isp = tc_mat[arg_max_isp]
ax.text(2,22,
  'Maximum ISP {:.2f}s\nCombustion Temperature {:.2f}K\n{:d}% Methanol\n{:d}% ADN\n{:d}% H2O'.format(max_isp,t_at_max_isp,arg_max_isp[1],arg_max_isp[0],100-np.sum(arg_max_isp)),
  color="tomato")
ax.plot(arg_max_isp[1],arg_max_isp[0],'r+')

# adjust plots
plt.subplots_adjust(hspace=0.2)

# save fig
```

```python
plt.savefig('b3/results.pdf')

# SECOND PLOT

isp_mat = np.multiply(tc_mask,isp_mat)

# init plot
fig = plt.figure()
ax = fig.add_subplot(111)

# plot isp
im = ax.matshow(np.multiply(isp_mat,tc_mask), cmap='Greens')
cbar = ax.figure.colorbar(im, ax=ax)
cbar.ax.set_ylabel("ISP", rotation=90, va="top")

# add labels
ax.set_xlabel('ADN %')
ax.set_ylabel('Methanol %')

# add max
max_isp = np.max(isp_mat)
arg_max_isp = np.unravel_index(np.argmax(isp_mat, axis=None), isp_mat.shape)
t_at_max_isp = tc_mat[arg_max_isp]

ax.text(2,22,
    'Maximum ISP {:.2f}s\nCombustion Temperature {:.2f}K\n{:d}% Methanol\n{:d}% ADN\n{:d}% H2O'.format(max_isp,t_at_max_isp,arg_max_isp[1],arg_max_isp[0],100-np.sum(arg_max_isp)),
    color="tomato")
ax.plot(arg_max_isp[1],arg_max_isp[0],'r+')

# adjust plots
plt.subplots_adjust(hspace=0.2)

# save fig
plt.savefig('b3/results_masked.pdf')
```

code/semester_exercise_b3.py

C rocketCEA Code for Thrust vs Chamber Pressure Plot

```python
import matplotlib as mpl
mpl.use("Agg")
import matplotlib.pyplot as plt
from rocketcea.cea_obj import CEA_Obj, add_new_propellant
import numpy as np

def createPropellant(wt1,wt2,wt3):
    card_str = """
    name H2O   H 2 O 1    wt%={:.1f}
```

```
  h,kj/mol=-285.8        t(k)=293.15
  name Methanol   C 1 H 4 O 1    wt%={:.1f}
  h,kj/mol=-239.2        t(k)=293.15
  name ADN   H 4 N 4 O 4    wt%={:.1f}
  h,kj/mol=-134.6        t(k)=293.15
  """.format(wt1,wt2,wt3)
  add_new_propellant("WaterMethanolADN_Mix",card_str)

def bar2psi(bar):
  return bar*14.504

""" prepare fuel """
meth = 39
adn = 59
h2o = 2
createPropellant(h2o,meth,adn)
temp = CEA_Obj(propName="WaterMethanolADN_Mix")

""" investigate falling chamber pressure """
chamber_pressure = np.linspace(0,16,1000)
chamber_pressure = chamber_pressure[1:]
expansion_ratio = 60
throat_area = 6.96e-6 #m^2
results = []

for pc in chamber_pressure:
  (isp, cstr, tc) = temp.getFrozen_IvacCstrTc(Pc=bar2psi(pc),eps=expansion_ratio,frozenAtThroat=1)
  tc  /= 1.8 # rankine to kelvin
  thrust = isp * 9.81 * pc * 1e5 * throat_area / (cstr*0.3048)
  results.append((pc,isp,tc,cstr,thrust))
  print(pc,isp)

results = np.vstack(results)



""" plot """
plt.figure()
plt.plot(results[:,0],results[:,4])
plt.xlabel('Chamber Pressure [bar]')
plt.ylabel('Thrust [N]')
plt.subplots_adjust(left=0.2)
plt.grid()
plt.savefig('b5/thrust_vs_pc.pdf')
```

code/semester_exercise_b5.py

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