Homework 5
Assigned Friday March 4, Due Monday March 14, 2022
10 pts total

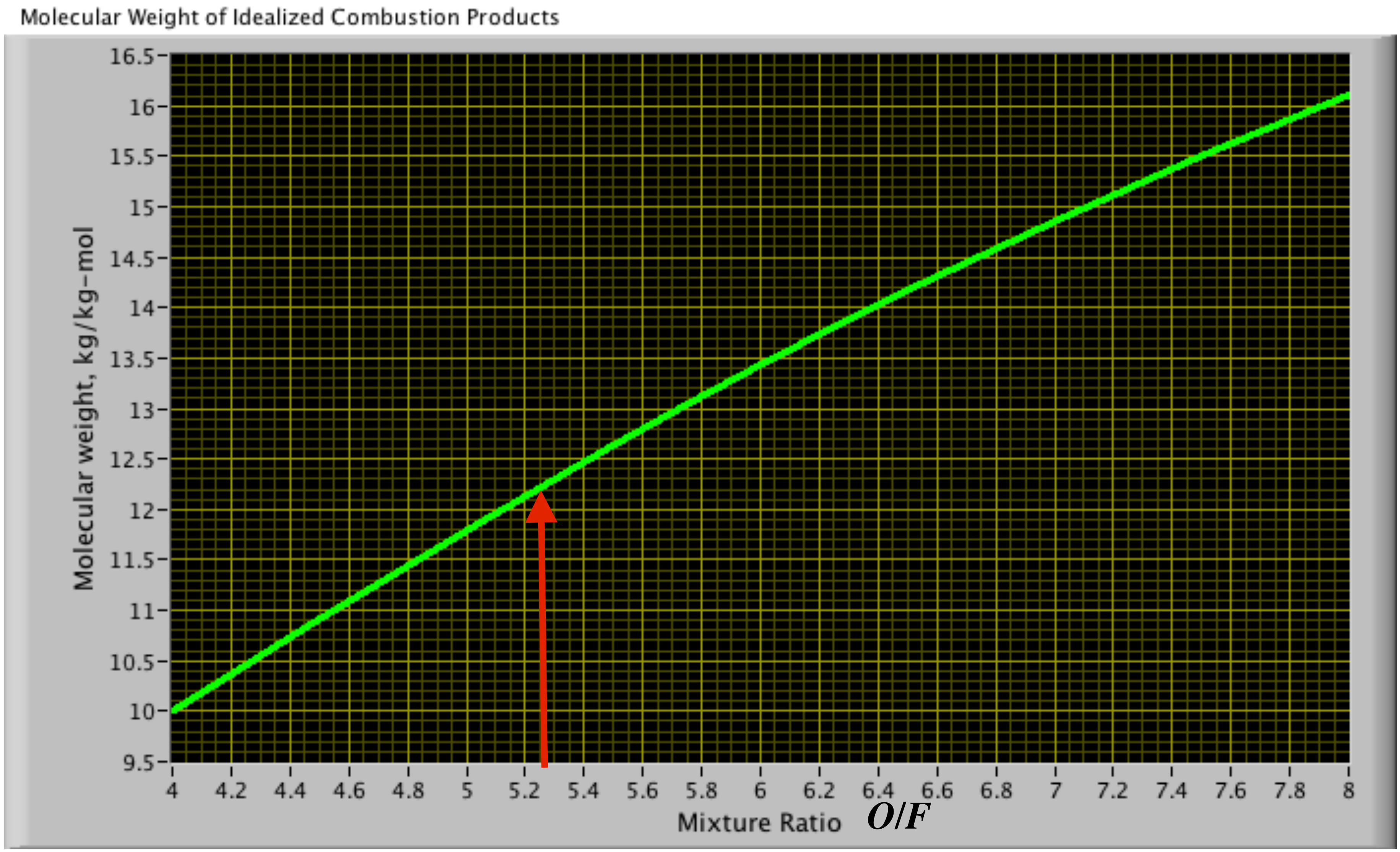
1) 1 points, Propellant Performance
2) 2 points, Combustor Cooling
3) 1 points, Combustor Geometry
4) 2 points, Combustor Pressure, Injector Design
5) 2 points, Optimal Performance
6) 2 pts Conical Nozzle Correction

1) Propellant Performance (1 Point)

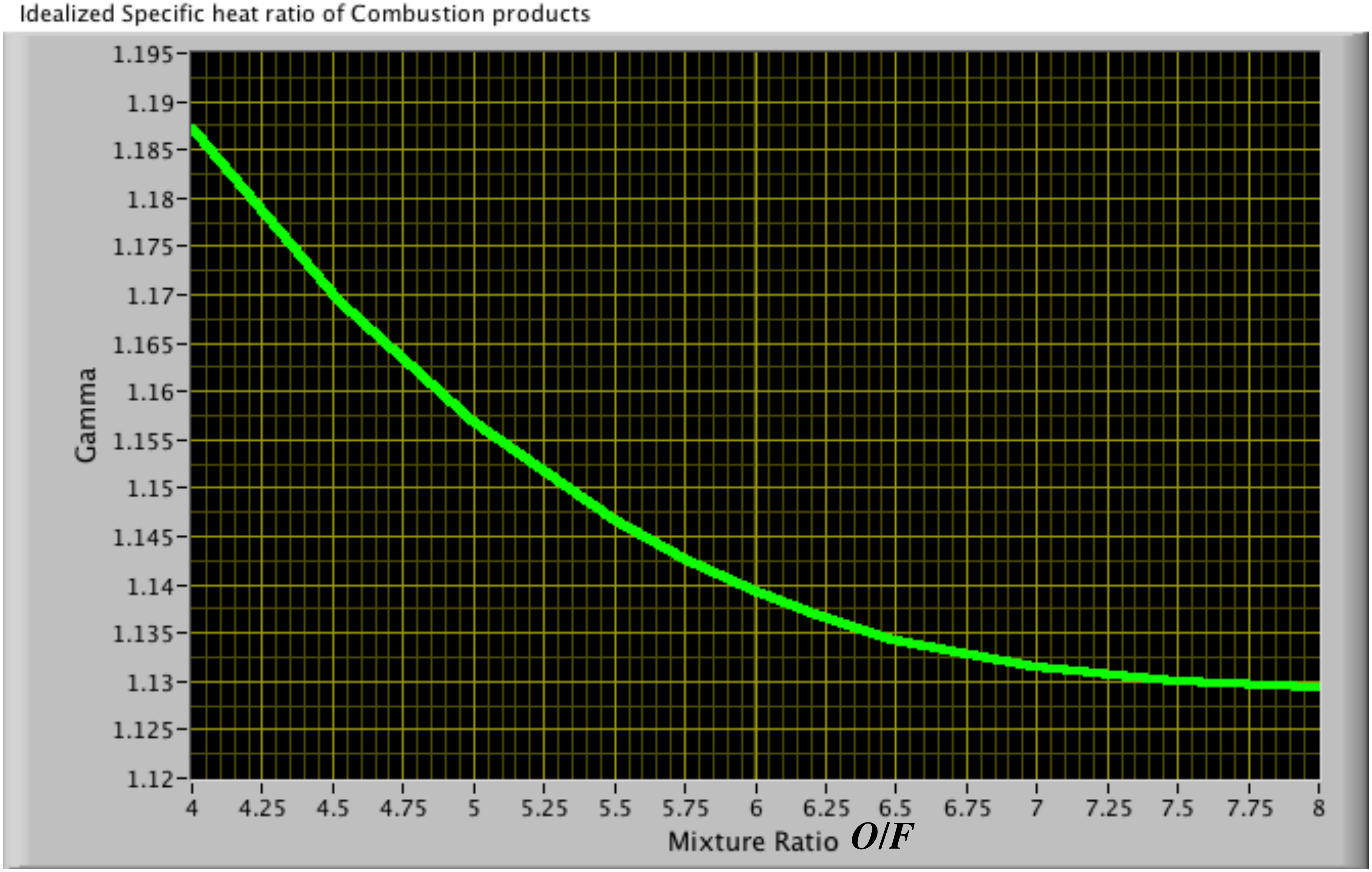
- A Bi-Propellant Rocket Engine Burns LOX/LH_2 … with an O/F ratio of 5.33333

… compute the Ideal characteristic velocity, C*
and Maximum ideal specific impulse (infinite expansion nozzle)

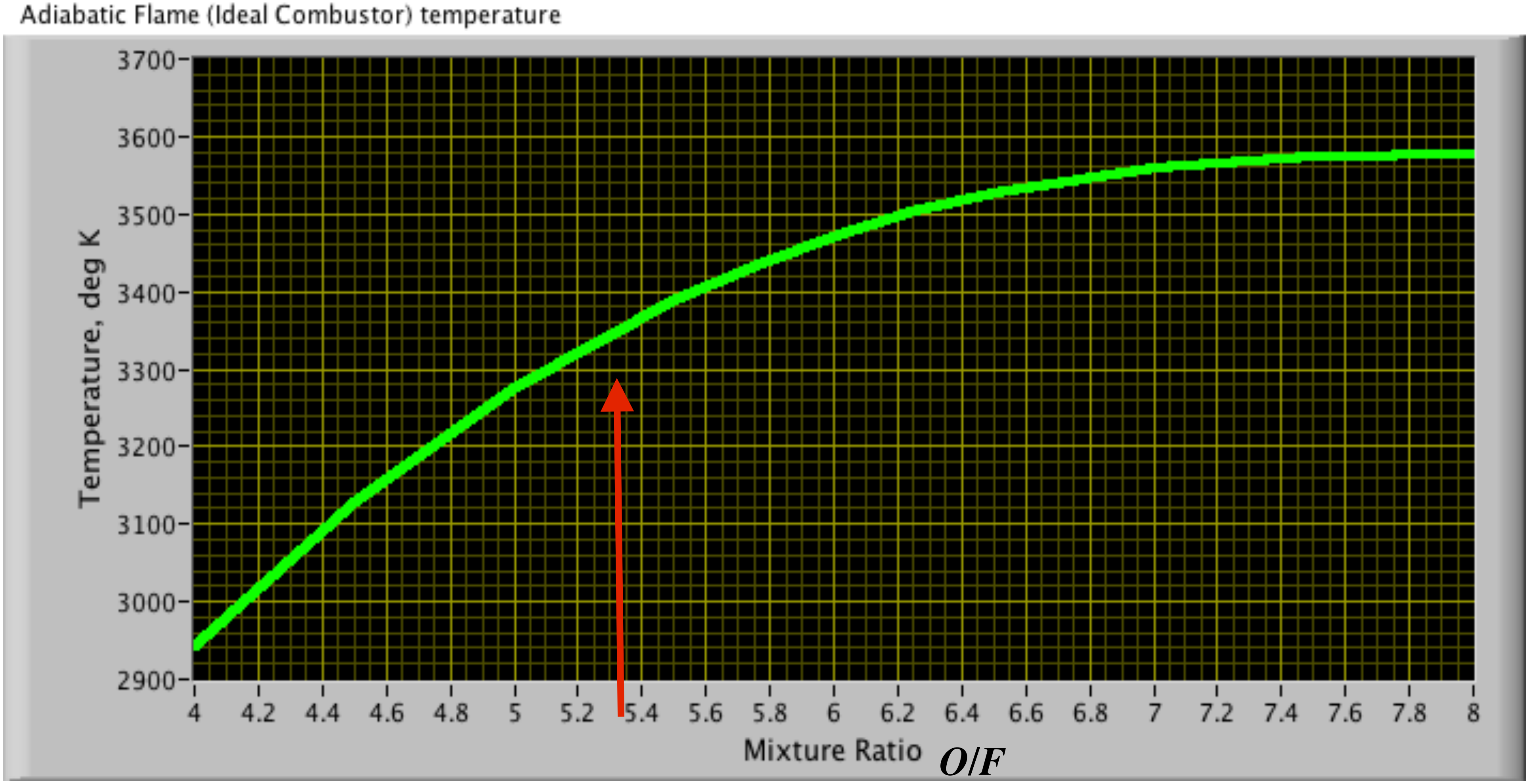
Propellant Combustion Properties



Propellant Combustion Properties



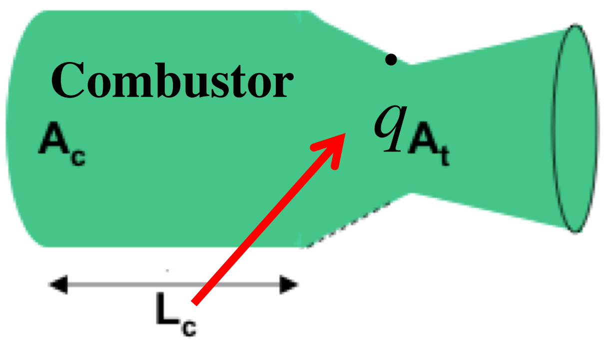
Propellant Combustion Properties



2) Non-Ideal Combustion Temperature (1 point)

- For the same Rocket in (1) …Assume that there is regenerative cooling in the combustor …..

Combustor Geometry



When operating in steady state mode
The regenerative cooling system removes
700 kWatts / kg/sec of propellant mass flow
entering the combustor

$$Assume \rightarrow h_{0_{actual}} = h_{0_{ideal}} - \left(\frac{\dot{q}}{\dot{m}}\right)_{cooling} \rightarrow h \approx c_p \cdot T$$

Calculate the actual stagnation temperature at the throat of the combustor and the efficiency of the combustion-->

$$\eta^* = \frac{1}{C^*_{ideal}} \cdot \left(\frac{P_0 \cdot A^*}{\dot{m}}\right) = \frac{C^*_{actual}}{C^*_{ideal}}$$

Assume that heat transfer has no effect on combustion products and that Flow is "frozen" with species mix defined at O/F = 5.333

3) Combustor Geometry

- For the same Rocket in (1), (2), and (3) …

Ranges of Combustor Characteristic Length

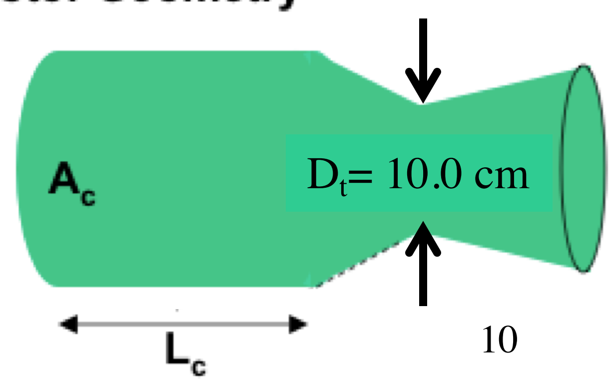
Propellants	Characteristic Length L^*	
	Low(m)	High(m)
Liquid fluorine / hydrazine	0.61	0.71
Liquid fluorine / gaseous H_2	0.56	0.66
Liquid fluorine / liquid H_2	0.64	0.76
Nitric acid/hydrazine	0.76	0.89
N_2O_4 / hydrazine	0.60	0.89
Liquid O_2/ ammonia	0.76	1.02
Liquid O_2 / gaseous H_2	0.56	0.71
Liquid O_2 / liquid H_2	0.76	1.02
Liquid O_2 / RP-1	1.02	1.27
H_2O_2 / RP-1 (incl. catalyst)	1.52	1.78

- Given the following Characteristic length Values,

Calculate a mid range value for The combustor Volume based on

$$L^*_{mean} = \frac{L^*_{high} + L^*_{low}}{2}$$

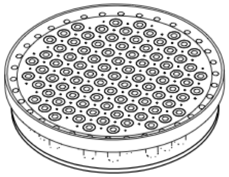
Combustor Geometry



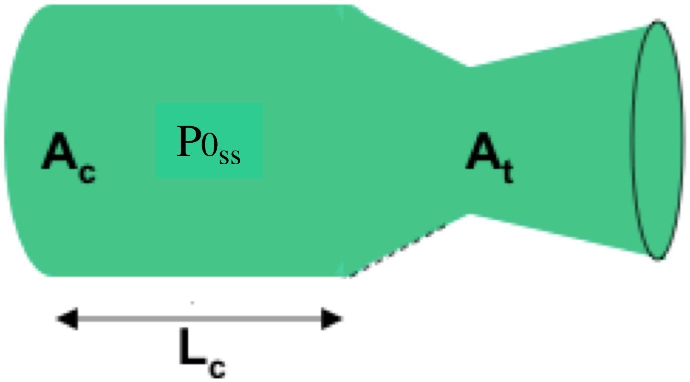
4) Steady Combustor Pressure

- For the same Rocket in (1), (2), and (3), and the combustor geometry from (4) …Use the true stagnation temperature calculated in (3), and assume that the combustion products are frozen as in (1) … for the following injector properties .. ***Calculate the steady state combustor pressure (assume incompressible propellants)***
- ***What LOX Injector feed pressure is require to give O/F 5.33333***

- **LOX Injector**
  - Port diameter:0.1841 cm
  - 100 injector ports
  - LOX density: *1.140 g/m³*
  - Cd (discharge coeff.): 0.75
- **LH_2 Injector**
  - Port diameter:0.1596 cm
  - 100 injector ports
  - LH_2 density: *0.071 g/cm³*
  - **Injector pressure 3700 kPa**
  - Cd (discharge coeff.): 0.75



Combustor



Base calculations on actual combustor temperature

5) Design Specific impulse

- Given the results of problem (1), (2), (3), (4), (5)

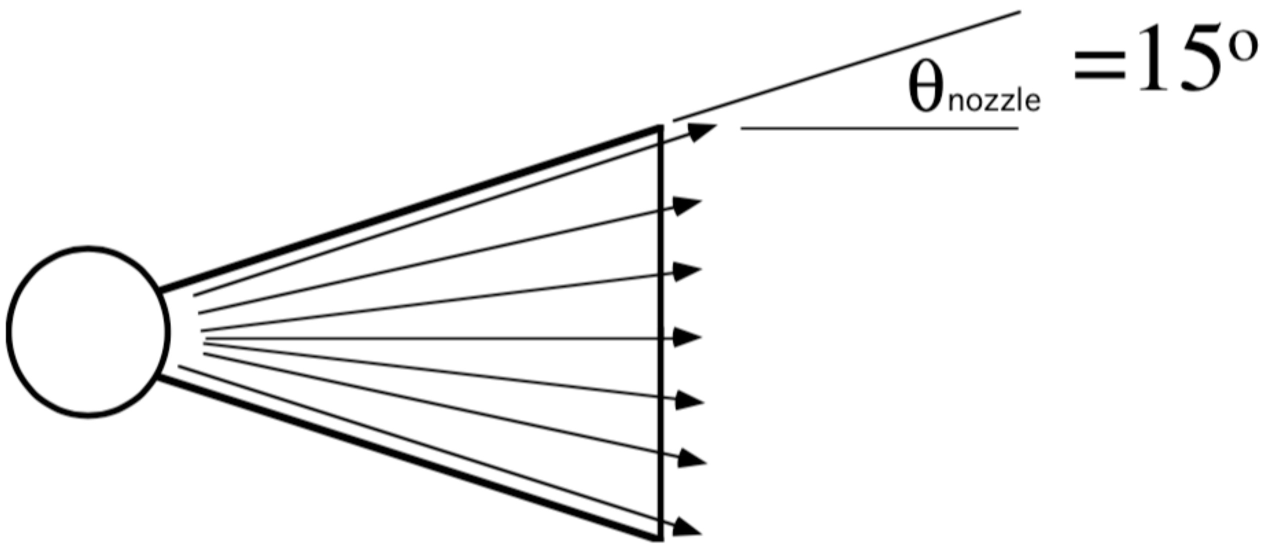
calculate the Optimal (design) Specific Impulse, Thrust Coefficient, and Total Thrust when the nozzle has a design altitude of 20 km (p_{amb} = 5.4748 kPa)

…. Assume nozzle exit divergence angle is zero, *be sure to include effects of combustor heat loss*

Calculate the expansion ratio of this optimal nozzle

6) Nozzle Exit Angle Correction

• Now Consider a conical nozzle of same expansion ratio, but with a 15° exit angle,



• How does this exit condition ***qualitatively and quantitatively*** affect the engine Isp, CF and total Thrust the design operating altitude?