AME 436	Assigned: Friday 4/22/2016
Problem Set #6	• "Due" Friday 4/29/2016 at 4:30 pm in drop-off box in OHE 430N (back room of the OHE 430 suite of offices, where the Xerox machine is located or at the "Final Exam Party") but no late penalty until Monday 5/2/16 at 7:00 PM. **No late assignments accepted, period!** • Hard copies are preferred but you can email your assignment to the graders at ame436usc@gmail.com if you're off campus. Emailed files **<u>must be a single .pdf file</u>**, not 20 .jpg images! Multiple files or "hybrid" paper + electronic assignments will **<u>not</u>** be accepted. • DEN students submit through the usual channels.

<u>Problem #1 (20 points)</u>

Ronney Oil & Gas Company claims to have developed a fuel, called PDR®, whose chemical formula is C_8H_{18} (octane) and has all the same thermodynamic properties, transport properties, etc. as C_8H_{18}. The **only** difference between C_8H_{18} and PDR® is that PDR has **<u>10% higher heating value</u>** than octane. If PDR® fuel were used instead of C_8H_{18}, how would each of the following be affected? In particular, state whether the property would increase, decrease or remain the same, and if there is a change, would it be by more than, less than, or equal to 10%. **<u>No credit without explanation!</u>**

a) **Burning velocity (S_L)** of a stoichiometric octane-air flame
b) **Soot concentration** in the products of a very rich premixed octane-air flame
c) **Indicated thermal efficiency** of an <u>ideal</u> diesel cycle
d) **Brake Mean Effective Pressure (BMEP)** of a premixed-charge engine operating at wide-open throttle
e) **CO emissions** from a premixed-charge engine operating at wide-open throttle
f) **Thrust Specific Fuel Consumption (TSFC)** of an afterburning turbojet with no $\tau_{\lambda,AB}$ limit in the afterburner.

<u>Problem #2 (20 points)</u>

Two engine designs, A and B, are being considered for a high-speed transport aircraft operating at a flight Mach number of 5.

Engine A produces a flow at the exit with a stagnation pressure 50 times the ambient pressure and a stagnation temperature 15 times the ambient temperature.

Engine B produces a flow at the exit with a stagnation pressure 100 times the ambient pressure and a stagnation temperature 10 times the ambient temperature.

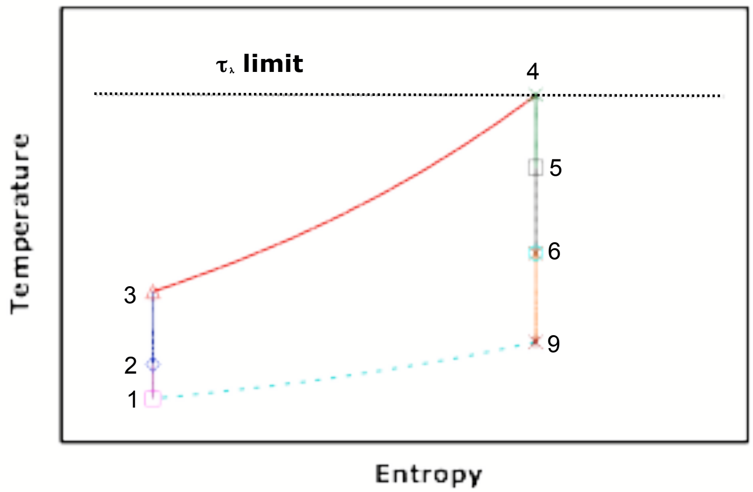
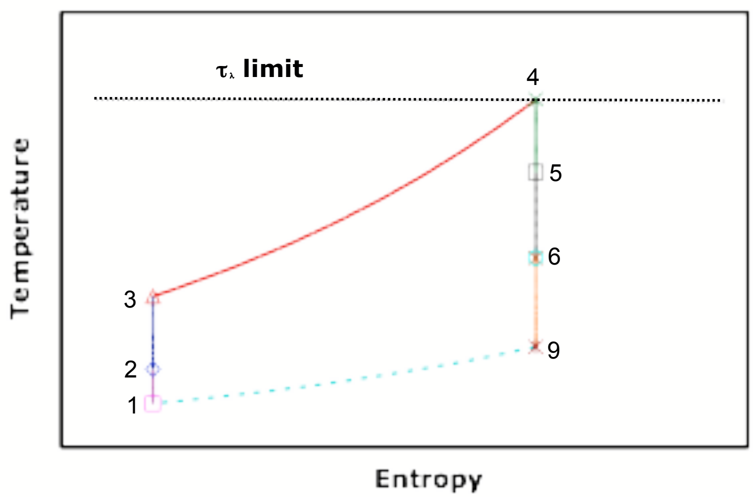
Because these two engines are made by rival companies with trade secrets, little is known about what happens inside the engines. It is not known for either engine it uses a compressor or not, whether combustion occurs at constant P or not, if the diffuser is reversible or not, nor is τ_λ known. All that is known is that for both engines (1) the same fuel is used, (2) reversible adiabatic expansion occurs in the exhaust nozzle to ambient pressure, (3) during the expansion the gas has constant specific heats with γ = 1.4, and (4) the fuel to air ratio (FAR) is much less than 1.

a) Which engine, A or B, has the higher exhaust velocity?
b) Which engine, A or B, has the higher mass flow per unit throat area?

c) Which engine, A or B, has the higher specific thrust?
d) Which engine, A or B, has the higher thrust specific fuel consumption?

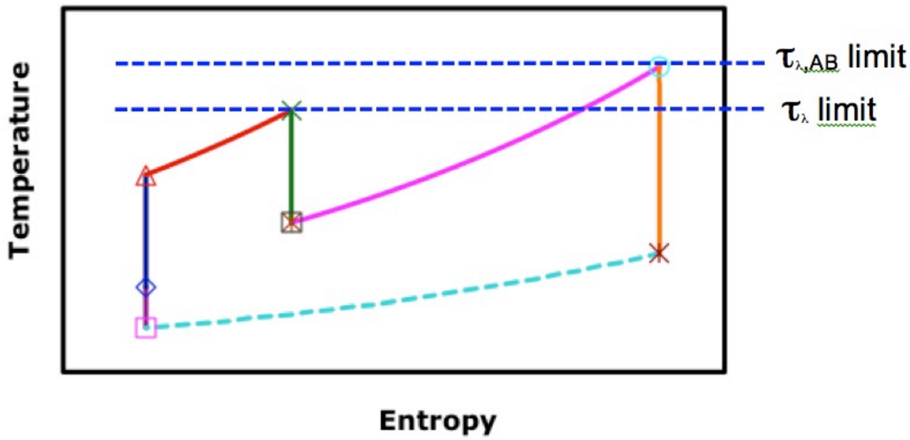
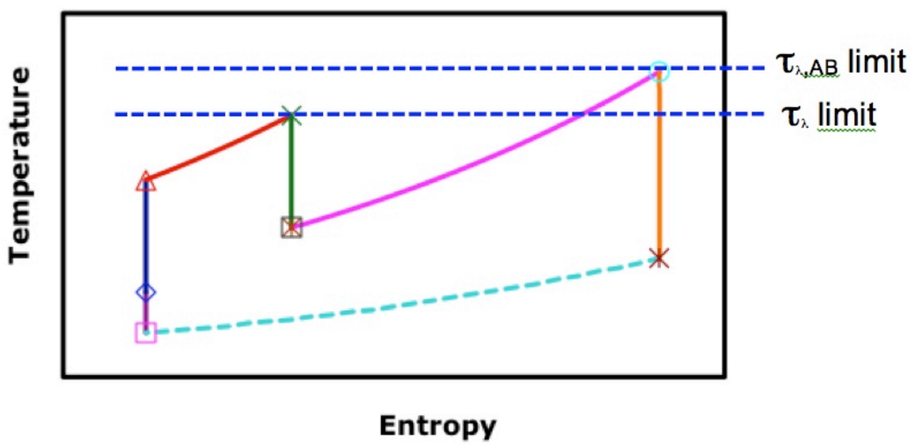
Problem #3 (10 points)

In an ideal τ_λ-limited turbofan, how would the T-s diagrams be affected if the following changes were made? In all cases, the compressor and fan pressure ratios are the same for the baseline and modified cycle. When useful, add statements like "this ΔT = that ΔT," "this area = that area," etc. In some cases there may be no change. *Please make your modifications clear; cycles that look like random scribbles and have no explanations don't get much credit!*

a)	
	Both the compressor and fan are irreversible but the turbine is still ideal (reversible).
b)	
	The diffuser is terrible and has significant stagnation pressure losses. All other components operate normally.

Problem #4 (10 points)

In an ideal τ_λ-limited afterburning turbojet, how would the T-s diagrams be affected if the following changes were made. In all cases, the compressor pressure ratio is the same for the baseline and modified cycle. When useful, add statements like "this ΔT = that ΔT," "this area = that area," etc. *Please make your modifications clear; cycles that look like random scribbles and have no explanations don't get much credit!*

a)	
	There are pressure losses in the main burner but not the afterburner (all other components are still ideal)
b)	
	The compressor and turbine are removed (i.e., the afterburning turbojet is converted into a ramjet) and the temperature limit of this ramjet is equal to the $\tau_{\lambda,AB}$ limit of the afterburning turbojet

Problem #5 (20 points)

For turbofan example at the end of Lectures 13 and 14, using aircycles4propulsion.xls, determine what combination of bypass ratio (α) and fan pressure ratio (π_c') (changing **nothing else)** gives the minimum thrust specific fuel consumption under the following 3 conditions:

a) Component efficiencies as in Lecture 14 with drag coefficient = 0
b) Component efficiencies as in Lecture 14 with drag coefficient = 0.1

You don't have to show any calculations, just use the spreadsheet to find the optima under these conditions, but answer the following question:

1) Why was the optimal α smaller for part (b) than in problem 6(b) on HW #5?

Problem #6 (20 points)

Repeat the example numerical problem at the end of Lecture 14 (non-ideal propulsion cycle analysis) for the following case:

- Nonideal afterburning turbojet (not turbofan as in the Lecture 14 example)
- $\gamma = 1.35$
- Compressor pressure ratio (π_c) = 20
- Flight Mach number (M_1) = 1.5
- Turbine inlet temperature ($\tau_\lambda T_1$) = 1800K
- Afterburner outlet temperature ($\tau_{\lambda,AB} T_1$) = 2400K
- Ambient pressure (P_1) = 0.25 atm
- Ambient temperature (T_1) = 225 K
- $P_9 = P_1$ and FAR << 1
- Heat loss coefficient (h) = 0
- Component efficiencies: diffuser (η_d) = 0.93, compressor (η_c) = 0.93, combustor (π_b) = 0.96, afterburner combustor (π_{ab}) = 0.96, turbine (η_t) = 0.95, nozzle (η_d) = 0.97.

In particular compute:

(a) T, P and M after the diffuser (station 2)
(b) T, P and M after the compressor (station 3)
(c) T, P and M after the combustor (station 4)
(d) T, P and M after the turbine (station 5).
(e) T, P and M after the afterburner (station 7)
(f) T, P and M after the nozzle (station 9)
(g) Specific thrust (ST) (recall FAR << 1 and $P_9 = P_1$ by assumption)
(h) TSFC and overall efficiency
(i) Propulsive efficiency (note bypass ratio $\alpha = 0$)
(j) Thermal efficiency
(k) Specific impulse