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# Final Exam

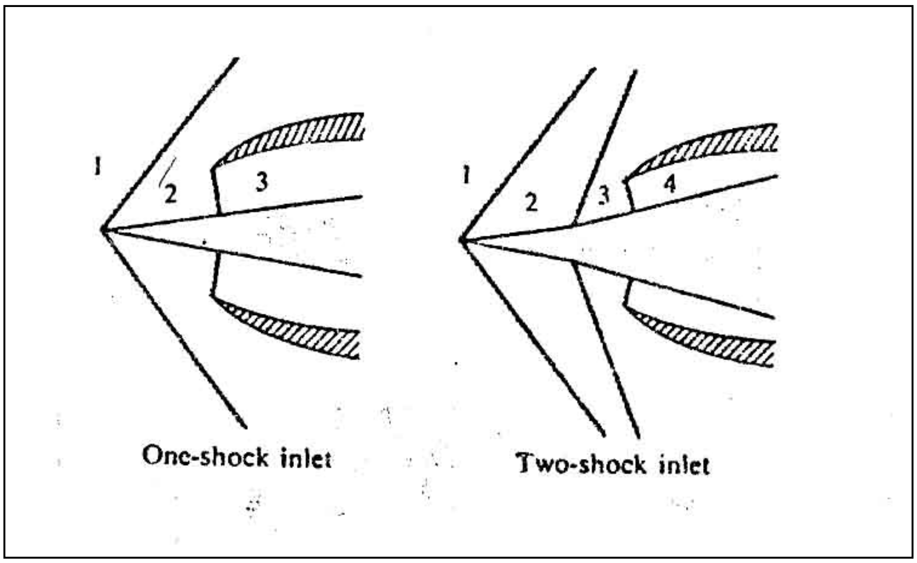
## Aerospace 410

## Due 12/16/2002 Monday 12:00 noon (take-home exam)

since this is a final exam, team work is neither encouraged nor acceptable

PLEASE PRINT THE FOUR PAGES AND ADD MORE PAPER IF NEEDED

1. (20/20) Compare the total pressure losses incurred by a one shock spike diffuser (2 dimensional) with that incurred by a two shock diffuser operating at Mach 2. Perform the same calculations for an inlet Mach number of 4. Compare "interpret" the results. Assume each oblique shock turns the flow through an angle of 10 degrees.



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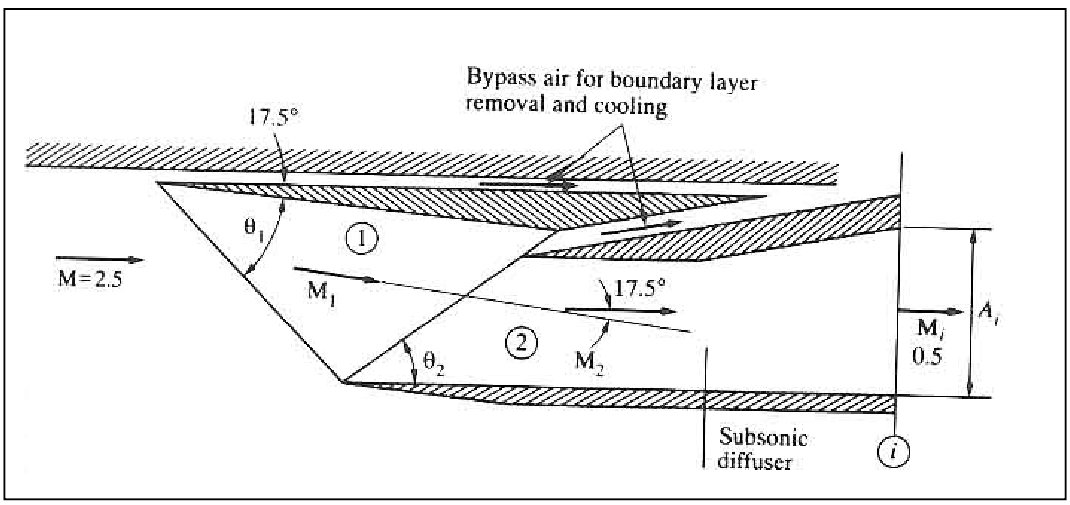
2. (30/30) A new supersonic passenger aircraft is being designed for flight Mach number 2.5 at an altitude where the ambient pressure and temperature are 9 kPa and 220 K respectively. The engine inlet configuration shown below allows for double oblique shock deceleration followed by a zone of subsonic deceleration. The Mach number is 0.5 at the engine inlet plane "*i*". Losses in the subsonic diffuser are neglected. Determine:

a. The Mach numbers $M_1$ and $M_2$ in the zones 1 and 2 shown on the sketch.
b. The wave angles $\theta_1$ and $\theta_2$, also shown on the drawing,
c. The overall stagnation pressure ratio $P_{oi}/P_{oa}$ ,
d. The overall static pressure ratio $p_i/p_a$ ,
e. The velocity ratio $c_i/c_2$ for the subsonic diffuser, and
f. The cross sectional area $A_i$ ($m^2$) at the engine inlet plane if the engine mass flow rate is 500 kg/s



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3. (20/20) Perform a simple analysis to show that the optimum Mach number at which $T/\dot{m}_a$ is maximum for an ideal ramjet is given by:

$$M^2{}_{optimal} = \left[(T_{04}/T_a)^{1/3} - 1\right] \cdot [2/(\gamma - 1)]$$

Use the nomenclature introduced in class and neglect f (fuel air ratio) in the thrust equation. Assume that the ramjet max. temperature $T_{04}$, ambient temperature $T_a$ and specific heat ratio $\gamma$ are constants. ($\gamma$=1.33)

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4. (30/30) A scramjet propulsion unit is designed for a flight Mach number of $M=7.0$ . The design flight altitude is 30,000 feet. You may assume that at the end of the supersonic combustor, a local Mach number of $M_2=1.0$ is reached. A hypothetical cycle maximum temperature of $T_{02}=3000$ K is assumed.

a. Find the static temperature $T_2$ at the end of the supersonic combustor area.
b. Find the total temperature $T_{oa}$ at the inlet of the scramjet unit.
c. Calculate the fuel air ratio f for the given conditions. ($\eta_b=0.70$, $Q_r=18{,}000$ Btu/lbm)
d. Find the local Mach number at the inlet of the scramjet combustor ($M_1$).
e. Calculate the required heat input per unit air mass flow rate in the scramjet unit for the given conditions.

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| SI unit system should be used in the solution. You are responsible to obtain reasonable values for $C_p$, R and $\gamma$. Use an altitude chart for the atmospheric static pressure and temperature. |