UNIVERSITI TUN HUSSEIN ONN MALAYSIA

FINAL EXAMINATION
SEMESTER II
SESI 2012/2013

COURSE NAME	:	FLIGHT MECHANICS
COURSE CODE	:	BDU 20603
PROGRAMME	:	BACHELOR OF AERONAUTICAL ENGINEERING TECHNOLOGY WITH HONOURS
EXAMINATION DATE	:	JUNE 2013
DURATION	:	3 HOURS
INSTRUCTION	:	ANSWER FOUR (4) OUT OF FIVE (5) QUESTIONS.

THIS PAPER CONSISTS OF FIVE (5) PRINTED PAGES

Q1. A Propeller driven aircraft uses a piston engine of 2000 BHP. This aircraft has aircraft data as follows:

The aircraft weight W = 60000 N
The wing area reference S = 20 m^2
Wing span = 9 m
Wing mean aerodynamics chord $C_{m.a.c}$ = 1.4 m
The drag coefficient $C_D = 0.012 + 0.064\, C_L^2$.
The maximum lift coefficient at clean configuration C_{Lmax} = 1.4
The lift coefficient C_L as function of angle of attack α is given as :

$$C_L(\alpha) = \frac{\partial C_L}{\partial \alpha}(\alpha - \alpha_{L=0}).$$

Where :

The slope of lift coefficient $\frac{\partial C_L}{\partial \alpha} = 0.098/\text{deg}$.

The zero lift angle of attack $\alpha_{L=0} = -2.4^0$

The atmospheric air properties for altitude h below 11000 m are given as:

$$T = (15.0 - 0.00649\,h)^0 C = \left[(15.0 - 0.00649\,h) + 273\right]^0 K$$

$$P = 101290\ [(\frac{T(^\circ C) + 273.1)}{288.08}]^{5.256}\ \frac{N}{m^2}$$

$$\rho = \rho_{SL}\left(\frac{T}{T_0}\right)\left(\frac{P}{P_0}\right) \qquad (kg/m^3)$$

$$\mu = 1.458\ 10^{-6}\ \frac{\sqrt{\left\{288.15\left(\frac{T}{T_{SL}}\right)\right\}^3}}{288.15\left(\frac{T}{T_{SL}}\right) + 110.4}\ \left(\frac{kg}{m.sec}\right)$$

Temperature T in degree Kelvin and h is in meter.
Pressure $p_{SL} = 1.01325\ 10^5\ N/m^2$,
Temperature $T_{SL} = 288.15^0\ K$,
Air density $\rho_{SL} = 1.225\ kg/m^3$
Air viscosity $\mu_{SL} = 1.7894\ 10^{-5}\ \frac{kg}{sec.\ m}$
Speed of sound a : $\mathbf{a} = \sqrt{\gamma RT}$
The ratio of heat coefficient γ = 1.4
The Universal gas constant $R = 287\ \frac{m^2}{sec^2\ {}^0K}$
Definition of Mach number M : $M = \frac{U}{a}$
Definition of Reynolds Number R_L : $\mathbf{R_L} = \frac{\rho\, U\, c_{mac}}{\mu}$

If the aircraft flies at altitude 6000 m. At this altitude, determines:

(i) The atmospheric properties (p,T,ρ, μ and a)
(ii) The Mach Number and Reynolds number of the airplane under level flight.
(iii) The Power required at level flight and the flight angle of attack and flight speed at the minimum power required.
(iv) If at this flight altitude h = 6000 m, the aircraft engine suddenly off, find the minimum glide angle γ_G, the flight speed and the horizontal distance can be achieved from the time engine off when the aircraft reaches the ground.

(25 Marks)

Q2 The aircraft as given in the question Q1, used an engine with the Brake horse power of 3000 Hp. This engine brake horse power varies with respect to the flight altitude h through density ratio σ defined as :

$$[\mathbf{BHP}]_{\text{at h}} = [\mathbf{BHP}]_{\text{at sea level}}\left(1.132\,\frac{\rho_h}{\rho_{SL}} - 0.132\right)$$

Where :

ρ_h : air density at altitude h
ρ_{SL} : air density at sea level

For a given $[BHP]_{\text{at h}}$ and propeller efficiency η_p, the engine power available P_a can be determined by the following relation:

$$P_a = \eta_p\,[\mathbf{BHP}]_{\text{at h}}$$

Assume the propeller efficiency η_p is constant and equal to 0.85, and use the aircraft data as given in Question Q1 and flies at the same flight altitude 6000 m. Determines

(i) The climb angle γ and the rate of climb if the aircraft flies at a constant speed of 140 m/sec.
(ii) The maximum rate of climb
(iii) The minimum flight speed
(iv) The maximum flight speed

(25-Marks)

Q3. An airplane weighing 160000 N has a wing area of 45 m^2 and drag polar given by $C_D = 0.017 + 0.055\,C_L^2$. The air properties at sea level and at altitude h = 3 km are given respectively as below:

At sea level (SL) are :

$$p_{SL} = 1.01325\ 10^5\ \text{N/m}^2\ ,\ T_{SL} = 288.15^0\ \text{K}\ ,\ \rho_{SL} = 1.225\ \text{kg/m}^3$$

$$\mu_{SL} = 1.7894\ 10^{-5}\ \frac{\text{kg}}{\text{sec. m}}$$

At an altitude 3 km is given as :

Pressure ratio $\dfrac{p}{p_{SL}} = 0.69204$

Temperature $T = 268.659^0$ K

Air density $\dfrac{\rho}{\rho_{SL}} = 0.74225$

Viscosity $\dfrac{\mu}{\mu_{SL}} = 0.94656$

Determines :

(i) The thrust required and power required for a rate of climb of 2400 m/min at a speed of 540 km/hour at 3 km altitude.

(ii) The rate of climb at sea level if the aircraft flies at flight speed U = 400 km/hour and the thrust is 45000 N.

(25-Marks)

Q4. **A propeller** driven airplane has aircraft data as follows :

Wing area $S = 40\ m^2$
Aircraft weight $W = 80000$ N.
The drag polar is given by :$C_D = 0.022 + 0.05C_L^2$.
Weight of fuel and oil = 15000 N,
BSFC = 2.67 N/kW-hr. and
The propeller efficiency $\eta_p = 85\%$.

The air properties at sea level is given as follows:

$$p_{SL} = 1.01325\ 10^5\ N/m^2\ ,\ T_{SL} = 288.15^0\ K\ ,\ \rho_{SL} = 1.225\ kg/m^3$$
$$\mu_{SL} = 1.7894\ 10^{-5}\ \frac{kg}{sec.\ m}$$

Determines :

(i) The maximum range and endurance at sea level in a steady level flight at a constant angle of attack.

(ii) The velocity at the beginning and the end of flight

(iii) The power required at the beginning and the end of flight

(25-Marks)

Q5. The aircraft weight of jet plane is 400000 N. The aircraft having a high wing configuration with the wing area reference is $100\ m^2$. The maximum lift coefficient at landing phase is 2.7 . Other aircraft data is given as follows:

The take-off speed $V_1 = 1.2$ Vs
The transition speed $V_2 = 1.1\ V_1$
The lift coefficient C_L during ground run is 1.15
The drag polar with landing gear and flaps is $C_D = 0.044 + 0.05C_L^2$
Thrust variation during take-off can be approximated as :
$T = 128{,}500 - 0.0929\ V^2$
where V is the km/hour
gravitational acceleration $g = 9.81\ m/sec^2$

Take-off takes place from a level and a dry concrete runway (μ=0.02) at sea level.

Determine :

(i) The ground run distance S_1 and the required time for ground run t_1.

(ii) The transition distance S_2 and the required time for the transition phase t_2