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MANIPAL INSTITUTE OF TECHNOLOGY
MANIPAL
(A constituent unit of MAHE, Manipal)

V SEMESTER B.TECH. (AERONAUTICAL ENGINEERING)

END SEMESTER EXAMINATIONS, DEC 2018

SUBJECT: AIRCRAFT DESIGN I [AAE 3104]

REVISED CREDIT SYSTEM
(01/01/2019)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ALL** the questions.
- Missing data may be suitably assumed.
- Draw suitable graphs or give plots wherever possible to illustrate final answer

1A. Figure 1 below shows mission profile drawn for a long-haul transport aircraft capturing basic segments like warm up, taxiing to run way, take off, climb, cruise, descent and landing. Other segments are as required by FAR **(04)**

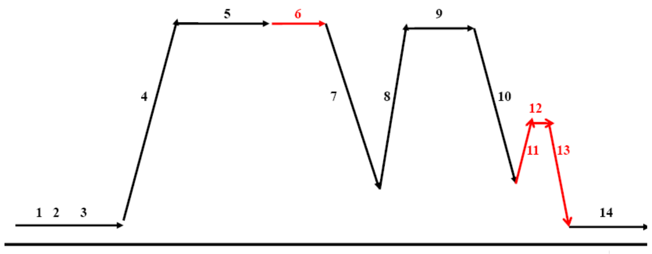

Fig. 1 Aircraft Mission Profile

The following data is compiled for mission analysis:
Cruise condition: M = 0.89 at 11km (ρ = 0.3639 kg/m^3; a = 295.07 m/s)
Design Range (5): 12000km, Additional Range (6): 10% of Design Range
Diversion to alternate airport (9): 200nm (370 km) , Hold up or loiter at 15000 ft (12) before landing: 30 min, Design value of $(L/D)_{max}$: 21 and $(L/D)_{Cruise}$ and $(L/D)_{Loiter}$: 0.92$(L/D)_{max}$, Design value of c_t engine TSFC at cruise altitude:1.52 x 10^{-4} N/Ns, Denoting weight fraction of ith segment by (W_i/W_{i-1}), we can get ratio (W_{14}/W_0) as product Π (W_i/W_{i-1}), i = 1 to 14. The fuel burnt in 10 out of 14 segments, excluding 3 cruise (5,6,9) and 1 loiter (12) segments, is a small percentage (10 to 12%) of mission fuel and product of aircraft weight fractions for 10 segments is given to be 0.94.
Using above data, calculate (W_{14}/W_0) and fuel fraction (W_{Fuel}/W_{TO}) for above mission profile. Assume constant speed cruise with range equation of $R = (V/c_t)(L/D)_{Cruise} \ln (W_i/W_f)$ and loiter equation of $E = (1/c_t)(L/D)_{Loiter} \ln (W_i/W_f)$.

1B. Aircraft in Q1A has an estimated pay load (mass) of 52 T. Correlation of empty weight (W_{Empty} in N) with take-off weight (W_{TO} in N) for metallic aircraft is $W_{Empty} = 0.911 W_{TO}^{0.947}$. Assume Technology Factor $K_S = 0.8$ to account for advanced technologies in designing and building a lighter aircraft. Obtain W_{TO}, W_{Empty} and W_{Fuel} in kN for above aircraft. Illustrate your approach with tables/plots used. W_{TO} lies between 2.5 and 3 times W_{PL} and use 2 to 3 values of W_{TO} for solving equations relating W_{TO} with W_{Entry}. **(06)**

2A. In drawing constraint diagram for an aircraft, meeting design requirements on take-off distance ($\leq s_{TO}$), cruise (M_{Cru}, h_{Cru}), landing distance ($\leq s_{Land}$) and FAR requirements on climb gradient (CGR 3% for OEI case), the following relationships between (W_{TO}/S) and (T_{SSL}/W_{TO}) are obtained: **(08)**

Take off	$(T_{SSL}/W_{TO}) = 1.44\,(W_{TO}/S)/(s_1 k_1 g \rho C_{LmaxTO}) + \mu$
Cruise	$(T_{SSL}/W_{TO}) = (C_{D0}\, q/\delta_{Cru})/(W_{TO}/S) + (k/q)\,(\omega^2_{Cru}\, \delta_{Cru})(W_{TO}/S)$
Landing	$(W_{TO}/S) = (d\, s_2\, \rho/1.69\omega_{Land})\ C_{Lmax\,Land}$
Climb	$(T_{SSL}/W_{TO}) = [N/(N-1)]\,[\omega_{Climb}/\delta_{Climb}]\,[\sin\gamma + 1/(L/D)_{Climb}]$

The following design inputs are compiled for above studies:
s_1 ground acceleration distance for takeoff taken as 0.75 s_{TO}; s_{TO} = 2500m
s_2 ground deceleration distance for landing taken as 0.7s_{Land}; s_{Land} = 2500m
k_1 fraction of initial acceleration in take off ground run taken to be 0.8
μ is rolling friction coefficient = 0.02
q: cruise dyn head at 11 km M_{Cru} = 0.89. (ρ = 0.3639 kg/m^3; a = 295.07 m/s)
Drag polar $C_D = C_{D0} + kC_L^2 = 0.012 + 0.035\, C_L^2$
δ_{Cru}: T_{Cruise}/T_{SSL}= 0.75σ. (σ at 11 km = 0.2971)
δ_{Climb}: T_{Climb}/T_{SSL}= 0.75σ. = σ (σ at mid climb segment 5.5 km = 0.5691)
ω_{Cru}: weight fraction at mid cruise segment W_{Cru}/W_{TO} = 0.85
ω_{Climb}: weight fraction at mid climb segment W_{Climb}/W_{TO} = 0.92
ω_{Land}: weight fraction at landing W_{Land}/W_{TO} = 0.8
C_{LMaxTO} and $C_{LMaxLand}$ are respectively 1.8 and 2.6
d: deceleration in landing ground run = 0.2g
γ: climb angle for OEI case given by FAR climb gradient (CGR) 3%(1.72 deg)
N: number of engines = 2 and $(L/D)_{Climb}$ = 18
Obtain above 4 boundaries meeting design requirements and plot them on a graph sheet to get constraint diagram in design space [(T_{SSL}/W_{TO}), (W_{TO}/S)] and show clearly feasible design space.

2B. Choose optimum design values for [(T_{SSL}/W_{TO}), (W_{TO}/S)] and give your criteria in selecting optimum design values **(02)**

3A. Table below shows mass and CG location of fully equipped fuselage, wing, empennage and engine. The CG locations given as aircraft stations, indicate distance from aircraft nose in centimetres. Similarly, mass and CG of pay load and fuel contents of IB and OB fuel tanks are also given in the table. **(04)**

Assembly/System	Mass in T	CG Location – Aircraft Station
Fuselage with Equipment	70 T	Stn 3700
Wing with Equipment (both side)	55	Stn 3800
Propulsion System (both Engines)	19	Stn 2950
Empennage	12	Stn 6800
Pay Load	56	Stn 3600

Fuel IB Tank (both sides)	87	Stn 3450
Fuel OB Tank (both sides)	58	Stn 4100

Accompanying Figure 2 shows these stations representing CG locations, passenger cabin front and rear ends (Stn 1300 and 5900) and location of main landing gear (MLG) (Stn 5900). CG distance of engine, wing assembly, fuel tanks IB and OB from plane of symmetry are also marked in the figure and tabulated. Half width of the fuselage is given to be 2.5 m.

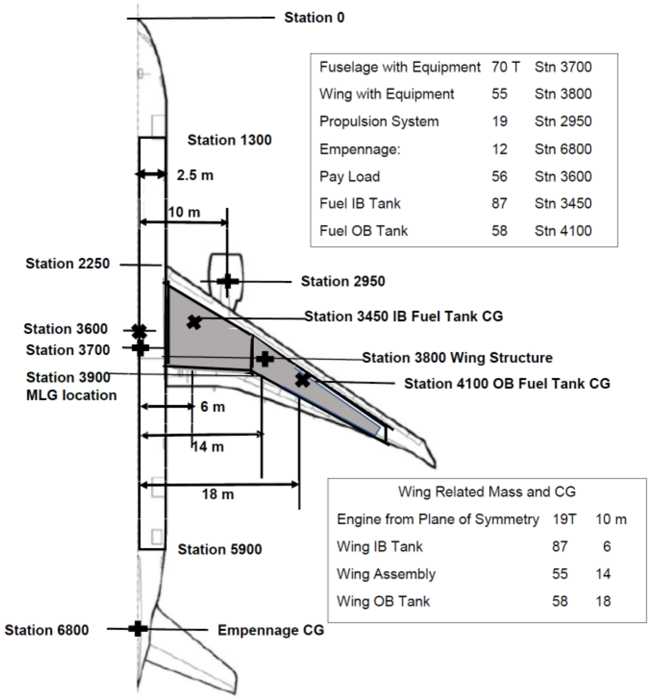

Figure 2

For the 4 items – Engine, Wing Assembly, IB and OB Fuel Tanks, associated with the right wing, obtain total mass, CG location (longitudinal and spanwise) with reference to exposed wing apex Stn 2250 and root section (2.5 m from plane of symmetry).

3B. Obtain gravity relief (root bending moment kN m) due to above 4 items on the wing on one side. Given elastic centre of wing root section is at Stn 2600, calculate torque on wing root section in (kN m) due to 4 items on right wing **(03)**

3C. Obtain mass and longitudinal CG location for i) Empty aircraft and ii) Aircraft with full fuel and full pay load (max take off mass) **(03)**

4A. For estimation of balanced field length of a multi engine aircraft the following basic aircraft data is given for take-off condition: **(02)**

Wing loading (W_{TO}/S) = 6500 N
Thrust to weight ratio (T_{SSL}/W_{TO}) = 0.275
C_{Lmax} = 1.8
Runway rolling friction coefficient μ = 0.02
Climb angle γ achievable for AEO case = 10 deg
Sea Level air density ρ = 1.225 kg/m^3

For AEO condition the ground acceleration is 90% of initial acceleration at brake release point given by $[(T_{SSL}/W_{TO}) - \mu]g$. The acceleration and climb angle γ following one engine failure during take off (OEI case), are 50% of corresponding values for AEO case respectively. Calculate take off distance for AEO case making suitable assumptions.

4B. Calculate take off distance required TODR and accelerate-stop distance required ASDR, following one engine failure, for two failure speeds (V_F) - 0.60 V_R and 0.80V_R, where V_R is rotation speed (=1.2 V_{Stall})). Pilot takes 3 seconds time to decide aborting the take-off, following the engine failure. After lapse of 3s, pilot cuts off working engine/s and applies wheel brakes to achieve average deceleration of 0.25 g to bring the aircraft to rest. Plot TODR and ASDR (x axis) at two failure speeds V_F (y axis) on graph sheet and obtain approximately decision speed $V_{Decision}$ and BFL. **(06)**

4C. Determine applicable aircraft take off distance based on AEO take off distance and balanced field length for OEI, as per FAR **(02)**

5A. Aircraft's capability to handle longer range segments is studied in Pay Load Vs Range Trade off studies. Aircraft data set generated for such a study, is given below: **(07)**

Aircraft Empty Mass M_{Empty} = 168T
Max Pay Load M_{PL}= 67 T
Mission Fuel on board $M_{MisFuel}$= 120 T
Max Take off Mass M_{TO} = 355 T
Drag Polar of aircraft $C_D = 0.0125 + 0.035C_L^2$
Density at cruise altitude of 12 km = 0.3108 kg/m^3
Engine TSFC, c_t at cruise altitude = 1.52×10^{-4} N/Ns
Wing area = 425 m^2
Aircraft Lift curve $C_L = 0.085\ (\alpha + 1.2)$ (α in deg.)

Draw Pay Load Vs Range trade off diagram considering the following three cases of cruise condition:

i) Design Cruise Segment: Take off with design pay load M_{PL}, mission fuel $M_{MisFuel}$ and max take off mass of M_{TO} (355 T)
ii) Cruise segment with enhanced range of 125% of design cruise segment obtained in i) above, by increasing fuel onboard and reducing pay load, retaining take off mass M_{TO} (355 T).
iii) Take off with zero payload, fuel as in ii) above and reduced take off mass

For calculations in all the three cases above, assume the following fuel availability in the beginning and at end of all the three cruise segments: 0.05$M_{MisFuel}$ utilised in reaching cruise altitude of 12 km and 0.09$M_{MisFuel}$ to be retained as unused fuel for meeting any flight exigencies before landing.

You may use the following range formula for maximum range

$R = \frac{2}{c_t}\sqrt{\frac{2}{\rho_\infty S}}\frac{C_L^{1/2}}{C_D}\left(W_0^{1/2} - W_1^{1/2}\right)$ where C_L (= $\sqrt{[C_{D0}/3k]}$ and $C_D = (4/3)C_{D0}$ giving

$$\left(\frac{C_L^{1/2}}{C_D}\right)_{\max} = \frac{3}{4}\left(\frac{1}{3KC_{D_0}^3}\right)^{1/4}$$

5B. Calculate optimum C_L for maximum range cruise and wing setting angle. **(03)**