Introduction to Aerospace Engineering

Exams

TU Delft

Answers exam AE1101

Part I Intro to Aeronautics

Problem 1 *(4 correct: 5 pts, 2 correct: 2 pts)*

A-3 (Cayley)
B-4 (Lilienthal)
C-2 (Langley)
D-1 (Plesman)

Problem 2 *(10 points = 2 + 4 + 4=2a + 2b +2c)*

2a)

Stringer (NL: verstijver), frame (NL: spant), spar (NL:ligger), rib, ….

2b)

I Tension (NL:Trekkracht/trekspanning): both girders (NL:gordingen) and web plate (NL:lijfplaat) experience tension
II Bending(NL: Buiging): the girders (NL:gordingen) experiences normal forces (one with tension and one compression); the web plate experiences shear stress (afschuiving)
Sketches with sufficiënte explanation are also acceptable

2c)

I. The deformations (NL: deformations) are: rectangle becoming an paralellogram (or in NL: wiebertje, ruit)
II The webplate fails due to buckling/wrinkling (NL: knikgedrag/plooivorming)
Sketches clearly showing deformations and failure are also acceptable

Problem 3 *(20 points, 3a: 8, 3b-3d each 4 points)*

3a)

Moments around c.g.:

$$\sum M_{cg} = M_{ac_{wb}} + L_{WB} \cdot l_{cg} - L_H \cdot (l_H - l_{cg})$$

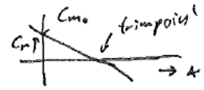
$$= M_{ac_{wb}} + (L_{WB} + L_H) \cdot l_{cg} - L_H \cdot l_H$$

$$= M_{ac_{wb}} + L \cdot l_{cg} - L_H \cdot l_H$$

For static longitudinal stability two conditions:

I) $C_{m_0} > 0$; Defines i_H

II) $C_{m_\alpha} < 0$



For condition II: ① Calculate C_{m_α} : – make dimensionless
– derive to α.

Dimensionless:

$$\frac{\sum M_{cg}}{\frac{1}{2}\rho V^2 S \bar{c}} = \frac{M_{ac_{WB}}}{\frac{1}{2}\rho V^2 S \bar{c}} + \frac{L \cdot l_{cg}}{\frac{1}{2}\rho V^2 S \bar{c}} - \frac{L_H \cdot l_H}{\frac{1}{2}\rho V^2 S \bar{c}}$$

$$C_M = C_{M_{ac_{WB}}} + C_L \cdot \frac{l_{cg}}{\bar{c}} - \frac{C_{L_H} \cdot S_H \cdot l_H}{S \cdot \bar{c}}$$

$$C_M = C_{M_{ac_{WB}}} + C_L \cdot \frac{l_{cg}}{\bar{c}} - C_{L_H} \cdot V_H \quad \text{with} \quad V_H = \frac{S_H \cdot l_H}{S \cdot \bar{c}}$$

Differentiate to α:

$$C_{M_\alpha} = \underset{\text{(because a.c.)}}{0} + C_{L_\alpha} \cdot \frac{l_{cg}}{\bar{c}} - \frac{dC_{L_H}}{d\alpha} \cdot V_H$$

$$= C_{L_\alpha} \cdot \frac{l_{cg}}{\bar{c}} - \frac{dC_{L_H}}{d\alpha_H} \cdot \frac{d\alpha_H}{d\alpha} \cdot V_H$$

$$= C_{L_\alpha} \frac{l_{cg}}{\bar{c}} - \left(C_{L_\alpha}\right)_H \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right) \cdot V_H$$

$$= a \frac{l_{cg}}{\bar{c}} - a_t \left(1 - \frac{d\varepsilon}{d\alpha}\right) V_H$$

$\alpha_H = \alpha - \varepsilon + i_H$

$\frac{d\alpha_H}{d\alpha} = 1 - \frac{d\varepsilon}{d\alpha}$

$a = C_{L_\alpha}$ aircraft

$a_t = \left(C_{L_\alpha}\right)_H$

For stability: $C_{m_\alpha} < 0$

$$a \frac{l_{cg}}{\bar{c}} - a_t \left(1 - \frac{d\varepsilon}{d\alpha}\right) V_H < 0$$

$$a \frac{l_{cg}}{\bar{c}} < a_t \left(1 - \frac{d\varepsilon}{d\alpha}\right) V_H$$

$$\frac{l_{cg}}{\bar{c}} < \underbrace{\frac{a_t}{a} V_H \left(1 - \frac{d\varepsilon}{d\alpha}\right)}$$

↳ given that this is l_{np}
(where $C_{m_\alpha} = 0$)

$$\frac{l_{cg}}{\bar{c}} < \frac{l_{np}}{\bar{c}}$$

$$l_{cg} < l_{np}$$

So from diagram: c.g. in front of n.p.

3b) Independent of the configuration, the C_{m_α} always has to negative. Because sign of α and C_m are positive in the same direction, a negative $C_{m_\alpha} = \frac{dC_m}{d\alpha}$ means a negative $\frac{\Delta C_m}{\Delta \alpha}$, so a disturbance of the angle of attack is countered by a moment in the other direction.

- For the center of gravity position of a canard use the given relation for C_{m_α}:

$$C_{m_\alpha} = a_c V_H - a \frac{l_{cg}}{\bar{c}}$$

For n.p. $C_{m_\alpha} = 0$ so $\frac{l_{np}}{\bar{c}} = \frac{a_c}{a} V_H$ and for a negative C_{m_α} l_{cg} has to be longer, so further away from the wing so in front of np.

- Alternative comments about the c.g. position relative to conventional configuration also receive points

3c) If c.g. is in neutral point we are on the edge of stability. If canard is bigger, S_c becomes larger and thus also V_H $\left(= \frac{S_c \cdot l_c}{S \, \bar{c}}\right)$. This means C_{m_α}, which was zero because of the c.g = n.p, will now become positive. So: a larger canard means an unstable aircraft.

3d) The canard should stall first, so the nose drops again and the airplane recovers from the stall which is caused by a too high angle of attack. Then the wing will never stall and keeps generating lift.

Problem 4 (15 points)

a) First calculate the lift coefficient C_L with $V_{stall} = 282/3.6 = 78.33\ m/s$

$$W = L = C_L \cdot \tfrac{1}{2} \rho V_{st.}^2 \cdot S \qquad (C_L = C_{L_{max}})$$

$$\Rightarrow C_L = \frac{W}{S} \cdot \frac{2}{\rho} \cdot \frac{1}{V_{stall}^2}$$

$$= \frac{68{,}000 * 9.81}{149} \cdot \frac{2}{1.225} \cdot \frac{1}{(78.33)^2}$$

$$= 1.191$$

8 points

$$C_D = C_{D_0} + \frac{C_L^2}{\pi A e}$$

$$= 0.0182 + \frac{1.19^2}{\pi\, 8.125 * 0.8}$$

$$= 0.0182 + 0.0695 = 0.0877$$

$$D = C_D \cdot \tfrac{1}{2} \rho V^2 S$$

$$= 0.0877 * \tfrac{1}{2} \cdot 1.225 \cdot 78.33^2 \cdot 149$$

$$= 49{,}087\ N \qquad (4.91 * 10^4\ N)$$

b $M = 0.3$

Standard atmosferic conditions

$$a = \sqrt{\gamma R T} = \sqrt{1.4 * 287 * 288.15}$$

$$a = 340.3 \ m/s$$

$$V = M * a = 0.3 * 340.3 = 102.1 \ m/s$$

7 points

$$C_L = \frac{W}{S} \cdot \frac{2}{\rho} \cdot \frac{1}{V^2}$$

$$= \frac{68,000 * 9.81}{149} \cdot \frac{2}{1.225} \cdot \frac{1}{(102.1)^2}$$

$$= 0.70$$

$$C_D = 0.0182 + \frac{0.7^2}{\pi * 8.125 * 0.8}$$

$$= 0.0182 + 0.0240 = 0.0422$$

$$D = C_D \cdot \frac{1}{2} \rho V^2 \cdot S$$

$$= 0.0422 * 0.5 * 1.225 * 102.1^2 \cdot 149$$

$$= 40,147.3 \ N \qquad (4.02 * 10^4 \ N)$$

Problem 5 (15 points)

a) **Critical mach number (2 points)**

The critical Mach number is that free-stream Mach number at which sonic flow is first obtained somewhere on the airfoil (wing) surface.

b) **Boundary layer transition (2 points)**

Boundary layer transition is the process in which the character of the boundary layer changes from laminar into turbulent.

c) **Mean camber line (3 points)**

The mean camber line is the locus of points halfway the upper and lower surfaces of an airfoil (as measured perpendicular to the mean camber line itself).

d) **Aerodynamic center (2 points)**

The aerodynamic center is that point of an airfoil or wing around which the moment coefficient is independent of the angle-of-attack. ($dCm/d\alpha=0$)

e) **Center of pressure (3 points)**

The center of pressure is the centroid of the pressure distribution.
Or:
The center of pressure is that point on the airplane through which the resultant aerodynamic force effectively acts (so $Cm_{c.p.}=0$)

f) **Induced drag (3 points)**

Explanation 1:
The wing tip vortices change the flow field and as a result the pressure distribution in such a way that an extra drag component is induced.

Explanation 2:
Due to the downwash induced by the wing tip vortices the lift vector is tilted backward giving a force component in drag direction

Explanation 3:
The wing tip vortices contain a certain amount of (rotational) kinetic energy. This extra energy has to come from the airplane's propulsion system: the extra power is needed to overcome the induced drag.

Problem 6 (20 points)

a) The triple S – shape, strength, stiffness.

b) For a phase-A it is an appropriate start design. The Marsian atmosphere is sparse. The gravity differs but not too much. During further phases of the design changes will appear because of different descent and thrust requirements etc.

c) See next page – either method 1 or 2

d) See next page – either methos 1 or 2

e) 117.7 : 9.81 = 11.99

<table>
<tr><td>Parameter</td><td>Method 1
Approximate solution as dealt during lectures
$V2 = (gm - a)\cdot t + V1$
$h1 - h2 = \frac{1}{2}\cdot(gm - a)\cdot t^2 + V1\cdot t$
$Mp := M0\cdot\left(1 - e^{\frac{-\Delta V}{Ve}}\right)$
$T=M_0a$ at t=0</td><td>Method 2
Approximate solution taking instantaneous exhaust at t=0 with exhaust mass $Mp=Mp1$
$V2 = (gm - a)\cdot t + V1$
$h1 - h2 = \frac{1}{2}\cdot(gm - a)\cdot t^2 + V1\cdot t$
$Mp1\cdot Ve = (M0 - Mp1)\cdot -\Delta V$
$T=M_0a$ at t=0</td></tr>
<tr><td>a [m/sec^2]</td><td>117.7</td><td>117.7</td></tr>
<tr><td>t [sec]</td><td>1.305</td><td>1.305</td></tr>
<tr><td>T [kN]</td><td>1707</td><td>1707</td></tr>
<tr><td>Mp [kg]</td><td>701.3</td><td>684.9</td></tr>
</table>

Problem 7 (10 points)

Q:

a) Mention the 6 Kepler parameters that are commonly used to characterize the orbit of an arbitrary satellite. Discuss each element briefly.
b) What is the assumption that has to be made to describe the orbits in such a way?

A:

a) Kepler elements:
   - a -> semi-major axis -> a measure of the absolute scale of the orbit [km].
   - e -> eccentricity -> a measure of the flattening, non-roundness of the orbit. $e = 0$ corresponds to a circular orbit, $0 < e < 1$ corresponds to a closed orbit, $e = 1$ corresponds to a parabolic orbit (i.e. escape from central body), and $e > 1$ corresponds to a hyperbolic orbit (also escape).
   - i -> inclination: the angle between the orbital plane and the equator (for Earth orbits, that is), measured at the ascending node (where the satellite moves from the southern hemisphere to the northern hemisphere). $0 \leq i < 180$ deg.
   - Ω -> right ascension of the ascending node -> the angle in the equatorial plane between a reference direction and the ascending node, the location where the satellite goes from southern to northern hemisphere. $0 \leq \Omega \leq 360$ deg.
   - ω -> argument of pericenter -> the location of the pericenter in the orbit, measured from the ascending node. $0 \leq \omega \leq 360$ deg.
   - θ_0 or M_0 : the position of the satellite at the reference epoch. Satellite position as time follows from this. An alternative to fix the satellite position as a function of time is to specify the moment of pericenter passage, by a value t_0 or τ (tau).
b) The motion of the orbit is fully determined by a perfectly round, homogeneous Earth (i.e. no perturbations from whatever source).