Tampere University KONE.730: Aerodynamics and Flight Mechanics

Final Exam - 07.05.2024

INSTRUCTIONS TO CANDIDATES:

1. EXAM DURATION: 3h
2. This exam is divided into two parts:
   (a) A theory part that contains 3 COMPULSORY exercises.
   (b) A calculation part that involves two sub-parts, of which ONLY ONE exercise of your choice is compulsory per each subpart.
   (c) A bonus exercise (NOT COMPULSORY).
3. The total number of exercises to be performed is 5 + (bonus exercise if interested).
4. THIS EXAM PAPER MUST BE HANDED IN AT THE END OF THIS EXAMINATION.
5. STUDENTS ARE PERMITTED TO TAKE ONE APPROVED CALCULATOR INTO THIS EXAMINATION, i.e. basic calculators. Graphic scientific calculators are NOT allowed.
6. Candidates are reminded that the major steps in all arithmetical calculations are to be set out clearly, i.e. display all the steps of your problem-solving strategy in questions involving calculations.
7. Standard mathematical formulae, some flight physics formulae and tables specific to the course are attached to the exam paper.

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Part I - Theory - (50 marks)

Both Exercises 1, 2 and 3 are compulsory:

Exercise 1 - (5 marks)

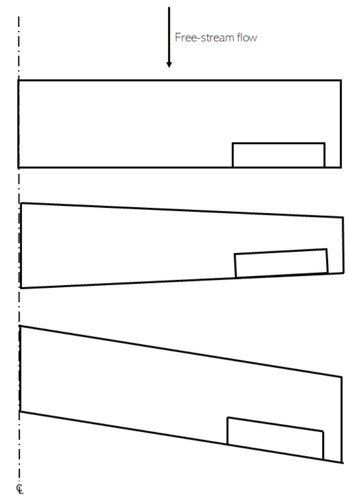
Define or explain ANY FIVE of the following terms or concepts:

1. Ackeret's rule
2. Drag-critical Mach number
3. Canard
4. Double-slotted flap
5. Downwash
6. Isentropic flow
7. Absolute ceiling
8. Zoom climb or zoom maneuver
9. Longitudinal dynamics
10. Roll damping

Exercise 2 - (25 marks)

(a) Briefly discuss the general flow physics of stall for airfoils and wings, with an emphasis on the stall characteristics. (6 marks)

(b) On the Figure below, identify (name) each of the wing type and display using equipotential lines and/or velocity vectors how stall patterns develop in each case. (9 marks)



(c) In the case of the first wing on the picture, where does the stall generally start and why is that stall pattern a desirable/advantageous behavior? (5 marks)

(d) You are a design lead within an aircraft aerodynamic design team. You are requested to reduce the stall airspeed of a newly designed commercial aircraft. What proposal(s) will you present to your design team? what will be the main advantages of reducing the stall airspeed? (5 marks)

Exercise 3 - (20 marks)

(a) In compressible flow, separately discuss the effects of wing sweep both in subsonic flow and supersonic flow. (6 marks)

(b) Fill in the table below with the right values or expressions. For the cases where the inputs are complex to remember, simply write their sign (i.e. $+$, $-$ or $\pm$, meaning the input is positive, negative or can be either positive or negative). (8 marks)

	c_{l_α}	aerodynamic center	center of pressure	$\alpha_{L=0}$
symmetrical airfoil				
cambered airfoil				

(c) What is aircraft stability? Discuss the factors affecting aircraft stability. (6 marks)

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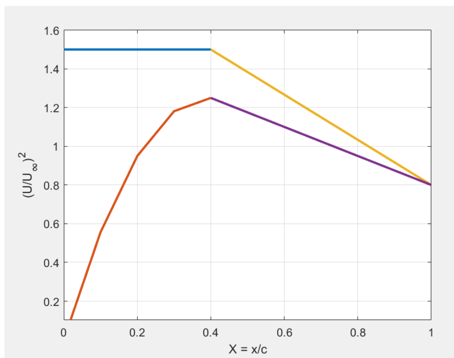
Part II - Calculations - **(50 marks)**

Subpart 1 - Aerodynamics **(25 marks)**

Freely select ONLY 1 exercise among the 4 exercises below.

Exercise 4 - (25 marks)
The airfoil velocity distribution presented below is valid for incompressible flow.



It is defined as

$$\left(\frac{U}{U_\infty}\right)^2_{\text{upper}} = \begin{cases} 1.5 & 0 \leq X \leq 0.4 \\ -\dfrac{7}{6}X + \dfrac{118}{60} & 0.4 \leq X \leq 1.0 \end{cases}$$

$$\left(\frac{U}{U_\infty}\right)^2_{\text{lower}} = \begin{cases} -\dfrac{65}{8}X^2 + \dfrac{51}{8}X & 0 \leq X \leq 0.4 \\ -\dfrac{3}{4}X + \dfrac{31}{20} & 0.4 \leq X \leq 1.0 \end{cases}$$

(a) Calculate the section lift coefficient c_l and the vortex distribution Γ. (15 marks)

(b) Estimate what would c_l be for the airfoil considered when the Mach number is 0.7 and the angle of attack is 2° higher than in the previous question (a). (7 marks)

(c) Has the Kutta condition been achieved with the present distribution? (3 marks)

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Exercise 5 - (25 marks)
The lift curve slope of a straight rectangular and thin wing in incompressible flow can be approximated using the following expression

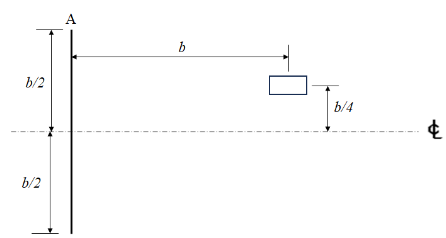
$$C_{L_\alpha} = \frac{2\pi A\!\!R}{2 + \sqrt{\frac{A\!\!R^2}{\kappa^2}\left(1 + \tan^2\Lambda_{1/2}\right) + 4}}$$

where $\kappa = c_{l_\alpha}/2\pi$.

Calculate the lift coefficient for the wing when then angle of attack $\alpha = 3°$, Mach number $M = 0.65$, aspect ratio $A\!\!R = 7$, swept angle at the half-chord $\Lambda_{1/2} = 41°$ and the lift curve slope for the profile is 6.01 1/rad (Assume the angle at zero lift to be zero for both the wing and the profile).

Exercise 6 - (25 marks)
Show that the downwash velocity, induced at the wing tip (point A, Fig. below) by a small area of the vortex sheet that forms the wake, has a magnitude (per unit area)

$$\frac{4}{(17)^{3/2}} \frac{L'_s}{\pi\rho V_\infty b^3}$$



The small area is downstream of the lifting line by a distance b and off the centerline of the wing by a distance $b/4$. Assume that the vorticity is constant over the small area and equal to the value at the center of the area. The lift distribution varies linearly from root to tip according to the equation

$$L' = L'_s\left[1 - \frac{1}{2}\left(\frac{|y|}{b/2}\right)\right]$$

Note that the prime in these formulas designate the loads per unit span and the index 's' the plane of symmetry.

Exercise 7 - (25 marks)
Air at Mach number 2.0 expands a sharp convex corner. While expanding around the corner, the flow has deflected away by an angle of 10°. If the initial pressure and temperature of air are 100 kPa and 300 K, find the final pressure, temperature and Mach number of the air. Assume the expansion to be isentropic, and that $p_{0_1} = p_{0_2}$.

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Subpart 2 - Flight Mechanics (25 marks)

Freely select ONLY 1 Exercise among the two exercises below.

Exercise 8 - (25 marks)
A trainer aircraft (cf. Fig. below) has the following features:

$C_{L_{\max}} = 1.9$ $m_{\mathrm{TO}} = 4240$ kg $S = 32$ m^2 $C_{D_{0_{\mathrm{clean}}}} = 0.03$ $C_{D_{0_{\mathrm{LG}}}} = 0.01$

$C_{D_{0_{\mathrm{flap}}}} = 0.007$ $(L/D)_{\max} = 9.4$ $\Delta C_{L_{\mathrm{flap_{TO}}}} = 0.4$ $T_{\max_{\mathrm{SL}}} = 9300$ N

(a) Prove that (5 marks)

$$\left(\frac{L}{D}\right)_{\max} = \frac{1}{2\sqrt{KC_{D_0}}}$$

(b) How much (in meters) is the take-off ground roll on a dry concrete runway ($\mu = 0.04$) that is located at the elevation of 4000 ft (1219 m) ISA condition? The aircraft cruising speed at 22000 ft (6705 m) is 170 knots (315 km/h). Consider $k_{\mathrm{LO}} = 1.2$. (20 marks)

Exercise 9 - (25 marks)
The pitching moment equation, referred to the center of gravity (cg), for a canard configured combat aircraft is given by

$$C_m = C_{m_0} + (h - h_0)C_{L_{\mathrm{wb}}} + \bar{V}_{\mathrm{f}}\left(\frac{a_{1_{\mathrm{f}}}}{a_{\mathrm{wb}}}C_{L_{\mathrm{wb}}} + a_{1_{\mathrm{f}}}\delta\right)$$

where the symbols have the usual meaning and, additionally, $\bar{V}_{\mathrm{f}}$ is the foreplane volume ratio, $a_{1_{\mathrm{f}}}$ is the foreplane lift curve slope and δ is the control angle of the all moving foreplane.

(a) State the conditions required for an aeroplane to remain in longitudinal trimmed equilibrium in steady level flight. (4 marks)

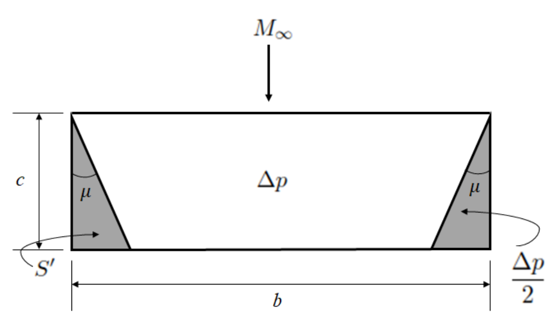
(b) Derive expressions for the controls fixed static margin K_n and for the controls fixed neutral point h_n. State any assumptions made. (10 marks)

(c) Given that the mean aerodynamic chord (mac) is 4.7 m, the wing–body aerodynamic center is located at 15% of mac, the foreplane volume ratio is 0.12 and the lift curve slope of the wing–body and foreplane are 3.5 and 4.9 1/rad respectively, calculate the aft cg limit for the aircraft to remain stable with controls fixed. (6 marks)

(d) Calculate also the cg location for the aircraft to have a controls fixed static margin of 15%. (5 marks)

Part III - Bonus exercise (Voluntary) - (20 marks)

Exercise 10 - (20 marks)
A thin rectangular wing (cf. Fig. below) has a lift coefficient $C_L = 0.10$ at Mach number 2.1 at an angle of attack $\alpha = 3°$. What is the aspect ratio of this wing?



Hint: μ on the Figure is the mach angle. The total area of the wing is S. The total lift generated by the wing is the sum of the lift generated by the shaded zones (or zones of disturbance or zones of action, each of area S') and the unshaded area (zone of silence).

USEFUL FORMULAE

General thin airfoil theory equations

$$\Gamma = \int_0^c \gamma(x)dx = \frac{c}{2}\int_0^\pi \gamma(\theta)\sin\theta d\theta$$

$$\frac{1}{2\pi}\int_0^c \frac{\gamma(x)dx}{x_0 - x} = V_\infty\left[\alpha - \left(\frac{dz}{dx}\right)_0\right] \quad (0 \leq x_0 \leq c)$$

$$x = \frac{1}{2}c\left(1-\cos\theta\right)$$

$$X = \frac{x}{c}$$

Symmetric airfoil equations

$$\gamma(\pi) = 0$$

$$\frac{1}{2\pi}\int_0^\pi \frac{\gamma(\theta)\sin\theta d\theta}{\cos\theta - \cos\theta_0} = V_\infty\alpha \quad \text{for } 0 \leq \theta_0 \leq \pi$$

$$\gamma(\theta) = 2\alpha V_\infty \frac{1+\cos\theta}{\sin\theta}$$

$$\Gamma = \pi\alpha c V_\infty$$

Cambered airfoil equations

$$\gamma(\pi) = 0$$

$$\frac{1}{2\pi}\int_0^\pi \frac{\gamma(\theta)\sin\theta d\theta}{\cos\theta - \cos\theta_0} = V_\infty\left[\alpha - \left(\frac{dz}{dx}\right)_0\right] \quad \text{for } 0 \leq \theta_0 \leq \pi$$

$$\gamma(\theta) = 2V_\infty\left[A_0\frac{1+\cos\theta}{\sin\theta} + \sum_{n=1}^\infty A_n \sin n\theta\right]$$

$$A_0 = \alpha - \frac{1}{\pi}\int_0^\pi \frac{dz}{dx}d\theta \qquad A_n = \frac{2}{\pi}\int_0^\pi \frac{dz}{dx}\cos n\theta d\theta \quad (n \geq 1)$$

$$\Gamma = cV_\infty\left(\pi A_0 + \frac{\pi}{2}A_1\right)$$

$$c_l = \pi\left(2A_0 + A_1\right)$$

General airfoil aerodynamic equations

$$L' = \int_0^c \Delta p dx = \int_0^c \rho_\infty V_\infty \gamma(x)dx = \rho_\infty V_\infty \Gamma$$

$$c_l = \frac{1}{c}\int_0^c \Delta c_p(x)\ dx = \int_0^1 \left(C_{p,l} - C_{p,u}\right)\ d\left(\frac{x}{c}\right)$$

$$C_p = \frac{q - q_\infty}{q}$$

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Wing aerodynamic equations

Wing lift : $L = q_\infty S C_L$

Wing drag : $D = q_\infty S C_D$

Dynamic pressure : $q_\infty = \frac{1}{2}\rho_\infty V_\infty^2$

Kutta-Joukowsky Theorem : $L = \rho_\infty V_\infty \Gamma b$

Elliptic wing circulation distribution : $\Gamma(y) = \Gamma_0 \sqrt{1 - \left(\frac{2y}{b}\right)^2}$

Mach number : $M = \frac{V_\infty}{a}$

Wing aspect ratio : $AR = \frac{b^2}{S}$

Induced drag : $D'_i = L'_i \alpha_i \rightarrow D_i = \int_{-b/2}^{b/2} L'(y)\alpha_i(y)$

Total drag (finite wing) : $C_D = c_d + C_{D_i} = C_{D_0} + K C_L^2$

Induced angle of attack (general lift distribution) : $\alpha_i = \frac{C_L}{\pi e AR}$

Induced angle of attack (elliptical lift distribution) : $\alpha_i = \frac{C_L}{\pi AR}$

Induced drag coefficient (elliptical lift distribution) : $C_{D_i} = \frac{C_L^2}{\pi AR}$

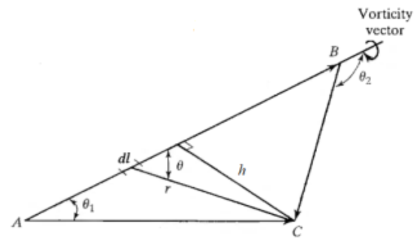
Wing lift coefficient slope (elliptic finite wing) : $C_{L_\alpha} = \frac{c_{l_\alpha}}{1 + c_{l_\alpha}/\pi AR}$

Biot–Savart law & downwash

Induced velocity (downwash) - straight vortex segment (cf. Fig below) :

$$w = \frac{\Gamma}{4\pi h} \int_{\theta_1}^{\theta_2} \sin\theta \, d\theta = \frac{\Gamma}{4\pi h} \left(\cos\theta_1 - \cos\theta_2\right)$$

For 2D case (infinite vortex length), $\theta_1 = 0,\ \theta_2 = \pi$
For the semi-infinite vortex line that starts at point, $\theta_1 = \pi/2,\ \theta_2 = \pi$



Prandtl-Glauert:

$$x' = \frac{x}{\sqrt{1-M_\infty^2}} = \frac{x}{\beta} \qquad y' = y \qquad z' = z$$

$$A\!\!\!/\!R_{\text{incomp}} = \beta A\!\!\!/\!R_{\text{comp}} \qquad C_{L_{\text{comp}}} = \frac{C_{L_{\text{inc}}}}{\beta^2}$$

$$\alpha_{\text{inc}} = \beta\alpha_{\text{comp}} \qquad C_{p_{\text{comp}}} = \frac{C_{p_{\text{inc}}}}{\beta^2}$$

Wing in supersonic flow

Pressure coefficient: $C_p \pm \dfrac{2\alpha}{\sqrt{M_\infty^2 - 1}}$ '−' ≡ suction side, '+' ≡ pressure side

Mach angle $\mu = \sin^{-1}\left(\dfrac{1}{M_\infty}\right) = \arcsin\left(\dfrac{1}{M_\infty}\right)$

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Aircraft performance equations

$V_R = k_{\mathrm{LO}} V_s$

Piston engines $\quad P = P_{\max}\left(\dfrac{\rho}{\rho_0}\right)^{1.2}$ (troposphere)

Turbojet engines $\quad T = T_{\max}\left(\dfrac{\rho}{\rho_0}\right)$ (troposphere) $\quad T = T_{11000\mathrm{ft}}\left(\dfrac{\rho}{\rho_{11000\mathrm{ft}}}\right)$ (stratosphere)

Turbofan engines $\quad T = T_{\max}\left(\dfrac{\rho}{\rho_0}\right)$ (troposphere) $\quad T = T_{11000\mathrm{ft}}\left(\dfrac{\rho}{\rho_{11000\mathrm{ft}}}\right)$ (stratosphere)

Turboprop engines $\quad T = T_{\max}\left(\dfrac{\rho}{\rho_0}\right)$ (troposphere) $\quad T = T_{11000\mathrm{ft}}\left(\dfrac{\rho}{\rho_{11000\mathrm{ft}}}\right)$ (stratosphere)

Turbojet engines $\quad T_{\mathrm{TO}} \approx 90\%\ T$

Prop-driven aircraft $\quad T_{\mathrm{TO}} = \dfrac{0.5 P_{\max}}{V_R}$ (fixed pitch) $\quad T_{\mathrm{TO}} = \dfrac{0.6 P_{\max}}{V_R}$ (variable pitch)

$1\ \mathrm{slug/ft^3} = 515.37882\ \mathrm{kg/m^3}$

$1\ \mathrm{knot} = 0.51444\ \mathrm{m/s}$

Ground roll

$$S_G = \frac{m}{\rho S (C_{D_{\mathrm{TO}}} - \mu C_{L_{\mathrm{TO}}})} \ln\left[\frac{(T_{\mathrm{TO}}/mg) - \mu}{(T_{\mathrm{TO}}/mg) - \mu - \dfrac{k_{\mathrm{LO}}^2 (C_{D_{\mathrm{TO}}} - \mu C_{L_{\mathrm{TO}}})}{C_{L_{\max}}}}\right]$$

Mathematical hints

$$\int \cos\theta = \sin\theta \qquad \int \cos^2\theta = \frac{1}{4}\sin 2\theta + \frac{1}{2}\theta$$

$$\int \cos^3\theta = \sin\theta - \frac{1}{3}\sin^3\theta$$

$$\int \cos^4\theta = \frac{3}{8}\theta + \frac{1}{4}\sin 2\theta + \frac{1}{32}\sin 4\theta$$

$$\cos 2\theta = \cos^2\theta - \sin^2\theta = 2\cos^2\theta - 1$$

Prandtl-Meyer Function and Mach Angle

M	ν	μ	M	ν	μ
0.1000 + 01	0.0000	0.9000 + 02	0.1600 + 01	0.1486 + 02	0.3868 + 02
0.1020 + 01	0.1257 + 00	0.7864 + 02	0.1620 + 01	0.1545 + 02	0.3812 + 02
0.1040 + 01	0.3510 + 00	0.7406 + 02	0.1640 + 01	0.1604 + 02	0.3757 + 02
0.1060 + 01	0.6367 + 00	0.7063 + 02	0.1660 + 01	0.1663 + 02	0.3704 + 02
0.1080 + 01	0.9680 + 00	0.6781 + 02	0.1680 + 01	0.1722 + 02	0.3653 + 02
0.1100 + 01	0.1336 + 01	0.6538 + 02	0.1700 + 01	0.1781 + 02	0.3603 + 02
0.1120 + 01	0.1735 + 01	0.6323 + 02	0.1720 + 01	0.1840 + 02	0.3555 + 02
0.1140 + 01	0.2160 + 01	0.6131 + 02	0.1740 + 01	0.1898 + 02	0.3508 + 02
0.1160 + 01	0.2607 + 01	0.5955 + 02	0.1760 + 01	0.1956 + 02	0.3462 + 02
0.1180 + 01	0.3074 + 01	0.5794 + 02	0.1780 + 01	0.2015 + 02	0.3418 + 02
0.1200 + 01	0.3558 + 01	0.5644 + 02	0.1800 + 01	0.2073 + 02	0.3375 + 02
0.1220 + 01	0.4057 + 01	0.5505 + 02	0.1820 + 01	0.2130 + 02	0.3333 + 02
0.1240 + 01	0.4569 + 01	0.5375 + 02	0.1840 + 01	0.2188 + 02	0.3292 + 02
0.1260 + 01	0.5093 + 01	0.5253 + 02	0.1860 + 01	0.2245 + 02	0.3252 + 02
0.1280 + 01	0.5627 + 01	0.5138 + 02	0.1880 + 01	0.2302 + 02	0.3213 + 02
0.1300 + 01	0.6170 + 01	0.5028 + 02	0.1900 + 01	0.2359 + 02	0.3176 + 02
0.1320 + 01	0.6721 + 01	0.4925 + 02	0.1920 + 01	0.2415 + 02	0.3139 + 02
0.1340 + 01	0.7279 + 01	0.4827 + 02	0.1940 + 01	0.2471 + 02	0.3103 + 02
0.1360 + 01	0.7844 + 01	0.4733 + 02	0.1960 + 01	0.2527 + 02	0.3068 + 02
0.1380 + 01	0.8413 + 01	0.4644 + 02	0.1980 + 01	0.2583 + 02	0.3033 + 02
0.1400 + 01	0.8987 + 01	0.4558 + 02	0.2000 + 01	0.2638 + 02	0.3000 + 02
0.1420 + 01	0.9565 + 01	0.4477 + 02	0.2050 + 01	0.2775 + 02	0.2920 + 02
0.1440 + 01	0.1015 + 02	0.4398 + 02	0.2100 + 01	0.2910 + 02	0.2844 + 02
0.1460 + 01	0.1073 + 02	0.4323 + 02	0.2150 + 01	0.3043 + 02	0.2772 + 02
0.1480 + 01	0.1132 + 02	0.4251 + 02	0.2200 + 01	0.3173 + 02	0.2704 + 02
0.1500 + 01	0.1191 + 02	0.4181 + 02	0.2250 + 01	0.3302 + 02	0.2639 + 02
0.1520 + 01	0.1249 + 02	0.4114 + 02	0.2300 + 01	0.3428 + 02	0.2577 + 02
0.1540 + 01	0.1309 + 02	0.4049 + 02	0.2350 + 01	0.3553 + 02	0.2518 + 02
0.1560 + 01	0.1368 + 02	0.3987 + 02	0.2400 + 01	0.3675 + 02	0.2462 + 02
0.1580 + 01	0.1427 + 02	0.3927 + 02	0.2450 + 01	0.3795 + 02	0.2409 + 02

Isentropic Flow Properties

M	$\frac{p_0}{p}$	$\frac{\rho_0}{\rho}$	$\frac{T_0}{T}$	$\frac{A}{A^*}$
0.1620 + 01	0.4378 + 01	0.2871 + 01	0.1525 + 01	0.1267 + 01
0.1640 + 01	0.4511 + 01	0.2933 + 01	0.1538 + 01	0.1284 + 01
0.1660 + 01	0.4648 + 01	0.2996 + 01	0.1551 + 01	0.1301 + 01
0.1680 + 01	0.4790 + 01	0.3061 + 01	0.1564 + 01	0.1319 + 01
0.1700 + 01	0.4936 + 01	0.3128 + 01	0.1578 + 01	0.1338 + 01
0.1720 + 01	0.5087 + 01	0.3196 + 01	0.1592 + 01	0.1357 + 01
0.1740 + 01	0.5244 + 01	0.3266 + 01	0.1606 + 01	0.1376 + 01
0.1760 + 01	0.5406 + 01	0.3338 + 01	0.1620 + 01	0.1397 + 01
0.1780 + 01	0.5573 + 01	0.3411 + 01	0.1634 + 01	0.1418 + 01
0.1800 + 01	0.5746 + 01	0.3487 + 01	0.1648 + 01	0.1439 + 01
0.1820 + 01	0.5924 + 01	0.3564 + 01	0.1662 + 01	0.1461 + 01
0.1840 + 01	0.6109 + 01	0.3643 + 01	0.1677 + 01	0.1484 + 01
0.1860 + 01	0.6300 + 01	0.3723 + 01	0.1692 + 01	0.1507 + 01
0.1880 + 01	0.6497 + 01	0.3806 + 01	0.1707 + 01	0.1531 + 01
0.1900 + 01	0.6701 + 01	0.3891 + 01	0.1722 + 01	0.1555 + 01
0.1920 + 01	0.6911 + 01	0.3978 + 01	0.1737 + 01	0.1580 + 01
0.1940 + 01	0.7128 + 01	0.4067 + 01	0.1753 + 01	0.1606 + 01
0.1960 + 01	0.7353 + 01	0.4158 + 01	0.1768 + 01	0.1633 + 01
0.1980 + 01	0.7585 + 01	0.4251 + 01	0.1784 + 01	0.1660 + 01
0.2000 + 01	0.7824 + 01	0.4347 + 01	0.1800 + 01	0.1687 + 01
0.2050 + 01	0.8458 + 01	0.4596 + 01	0.1840 + 01	0.1760 + 01
0.2100 + 01	0.9145 + 01	0.4859 + 01	0.1882 + 01	0.1837 + 01
0.2150 + 01	0.9888 + 01	0.5138 + 01	0.1924 + 01	0.1919 + 01
0.2200 + 01	0.1069 + 02	0.5433 + 01	0.1968 + 01	0.2005 + 01
0.2250 + 01	0.1156 + 02	0.5746 + 01	0.2012 + 01	0.2096 + 01
0.2300 + 01	0.1250 + 02	0.6076 + 01	0.2058 + 01	0.2193 + 01
0.2350 + 01	0.1352 + 02	0.6425 + 01	0.2104 + 01	0.2295 + 01
0.2400 + 01	0.1462 + 02	0.6794 + 01	0.2152 + 01	0.2403 + 01
0.2450 + 01	0.1581 + 02	0.7183 + 01	0.2200 + 01	0.2517 + 01
0.2500 + 01	0.1709 + 02	0.7594 + 01	0.2250 + 01	0.2637 + 01

PROPERTIES OF THE STANDARD ATMOSPHERE (ENGLISH UNITS)

h (ft)	T (˚F)	a (ft/sec)	p $(\mathrm{lb/ft^2})$	ρ $(\mathrm{slugs/ft^3})$	$\mu \times 10^7$ (slugs/ ftsec)
0	59.00	1117	2116.2	0.002378	3.719
1,000	57.44	1113	2040.9	.002310	3.699
2,000	51.87	1109	1967.7	.002242	3.679
3,000	48.31	1105	1896.7	.002177	3.659
4.000	44.74	1102	1827.7	.002112	3.639
5,000	41.18	1098	1760.8	.002049	3.618
6,000	37.62	1094	1696.0	.001988	3.598
7,000	34.05	1090	1633.0	.009128	3.577
8,000	30.49	1086	1571.9	.001869	3.557
9,000	26.92	1082	1512.9	.001812	3.536
10,000	23.36	1078	1455.4	.001756	3.515
11,000	19.80	1074	1399.8	.001702	3.495
12,000	16.23	1070	1345.9	.001649	3.474
13,000	12.67	1066	1293.7	.001597	3.453
14,000	9.10	1062	1243.2	.001546	3.432
15,000	5.54	1058	1194.3	.001497	3.411
16,000	1.98	1054	1147.0	.001448	3.390
17,000	-1.59	1050	1101.1	.001401	3.369
18,000	-5.15	1046	1056.9	.001355	3.347
19,000	-8.72	1041	1014.0	.001311	3.326
20,000	-12.28	1037	972.6	.001267	3.305
21,000	-15.84	1033	932.5	.001225	3.283
22,000	-19.41	1029	893.8	.001183	3.262
23,000	-22.97	1025	856.4	.001143	3.240
24,000	-26.54	1021	820.3	.001104	3.218
25.000	-30.10	1017	785.3	.001066	3.196
26,000	-33.66	1012	751.7	.001029	3.174
27,000	-37.23	1008	719.2	.000993	3.153
28,000	-40.79	1004	687.9	.000957	3.130
29,000	-44.36	999	657.6	.000923	3.108
30,000	-47.92	995	628.5	.000890	3.086
31,000	-51.48	991	600.4	.000858	3.064
32,000	-55.05	987	573.3	.000826	3.041
33,000	-58.61	982	547.3	.000796	3.019
34,000	-62.18	978	522.2	.000766	2.997
35,000	-65.74	973	498.0	.000737	2.974
40,000	-67.6	971	391.8	.0005857	2.961
45,000	-67.6	971	308.0	.0004605	2.961
50,000	-67.6	971	242.2	.0003622	2.961
60,000	-67.6	971	150.9	.0002240	2.961
70,000	-67.6	971	93.5	.0001389	2.961
80,000	-67.6	971	58.0	.0000861	2.961
90,000	-67.6	971	36.0	.0000535	2.961
100,000	-67.6	971	22.4	.0000331	2.961
150,000	113.5	1174	3.003	.00000305	4.032
200,000	159.4	1220	.6645	.00000062	4.277
250,000	-8.2	1042	.1139	.00000015	3.333

Data taken from NACA TN 1428. Courtesy of the National Advisory Committee for Aeronautics