13.1) If the airplane is cruising at constant altitude and at constant airspeed, the product $q_\infty S$ is constant. Since the lift balances the weight:

$$\frac{W}{q_\infty S} = \frac{L}{q_\infty S} = C_L = C_{L\alpha}(\alpha - \alpha_{OL})$$

Thus, the angle of attack is given by:

$$\alpha = \alpha_{OL} + \frac{W}{q_\infty S C_{L,\alpha}}$$

Note that the weight of the aircraft is the only variable quantity in the right-hand side of this equation. It is clear that α decreases as fuel is consumed during the flight, since the weight of the aircraft decreases as the fuel is consumed.

13.2] The maximum lift-to-drag ratio occurs when the induced drag equals the parasite drag.

$$C_{Di} = kC_L^2 = C_{Do}$$

Thus, the total drag is $2C_{Do}$ and the L/D ratio is given by:

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{[C_{Do}/K]^{0.5}}{2C_{Do}} = \frac{1}{2\sqrt{kC_{Do}}}$$

To increase $(L/D)_{max}$, we can either decrease C_{Do} or k.

13.2 Contd. C_{D_0} includes those contributions to the drag which exist when the configuration generates zero lift. Thus, C_{D_0} includes such non-lift-related drag components, as: (1) the profile drag (i.e., that both due to skin friction and to the pressure distribution) of the wing, fuselage, tail, nacelles, etc.; (2) losses of momentum of the airstream due to power plant cooling, leakage through the skin, etc.; and (3) the drag which results when the flow fields of the various components interact. An example of this latter component (i.e., the drag due to the interactions) occurs at the wing/fuselage juncture, where the presence of the already-developed boundary layer on the fuselage reduces the local velocities of the air molecules as they flow over the wing root. Because the air molecules in the fuselage boundary layer have already been slowed, they are more likely to be separated in the presence of an adverse pressure gradient created as the flow in the boundary layer approaches the wing root. Proper design of a wing-root fillet can minimize these effects. Reductions in skin-friction drag can be accomplished by cleaning up (or smoothing) the surface of the vehicle. Thus, use is made of non-protuding fasteners (flush rivets), care is taken that the panel joints are smooth.

Referring to Chapter 5,

$$C_{Di} = kC_L^2 = \frac{1}{\pi e AR} C_L^2$$

13.2 Contd. Thus, we can reduce the induced drag (or the drag-due-to-lift) by increasing the aspect ratio or by increasing e (which is referred to as Oswald's efficiency factor). The airplane efficiency factor includes the parasite drag-due-to-lift. Recall that, as the angle of attack (and, therefore, the lift) increases, the pressure distribution changes and there are regions where the adverse pressure gradient is relatively large. Thus, separation may occur and the form drag would increase. The wing planform, the airfoil section, and twist may be used to minimize the effect.

We can increase e by using devices which produce equivalent increases in the aspect ratio, such as by adding winglets at the wing tips, which is discussed in this chapter.

13.3 A laminar flow airfoil section would be desireable for a transport aircraft that operates for long periods of time in cruise flight, since the reduction of friction drag would be an important consideration. Operating at low angles of attack, flow separation and, therefore, form drag would not be as important.

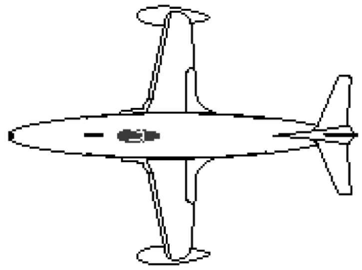
The maneuvers of a stunt airplane would be at large angles of attack, with large pressure variations in the flow field, with regions of flow separation and (as a result) with relatively high form drag. Since boundary-layer separation (or stall) is more likely to occur if the boundary layer is laminar, laminar boundary layers would be undesireable.

It should be recalled that it is difficult to maintain a laminar boundary layer at the high Reynolds numbers characteristic of flight.

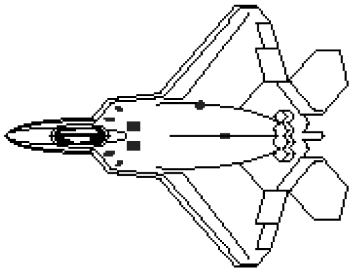
13.4)

The results for Mach number and aspect ratio come from Jane's All The World's Aircraft, while the lift-to-drag ratios are estimated from p. 244.

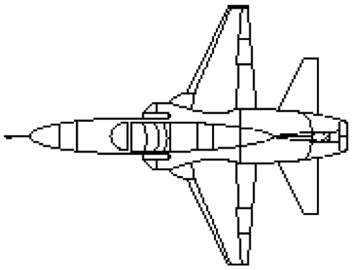
M_∞ :0.78 at 25 kft
AR: 6.38
$(L/D)_{max}$: 12
Name: P-80 (F-80) Shooting Star
Original company: Lockheed



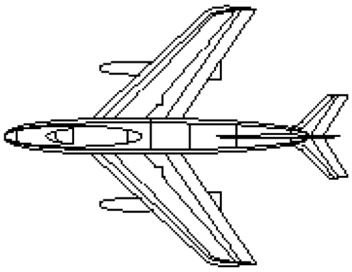
M_∞ : 1.58 at 30 kft (w/o afterburner)
AR: 2.4
$(L/D)_{max}$: 8
Name: F-22 Raptor
Original company: Lockheed Martin



M_∞ : 1.4 at 36 kft
AR: 3.75
$(L/D)_{max}$: 10
Name: F-5 Freedom Fighter
Original company: Northrop

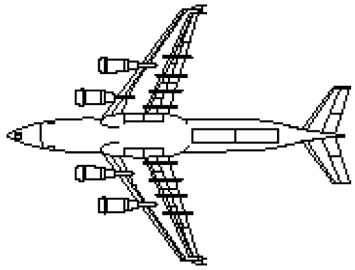


M_∞ : 0.88
AR: 5.02
$(L/D)_{max}$: 12
Name: F-86 Sabre
Original company: North American

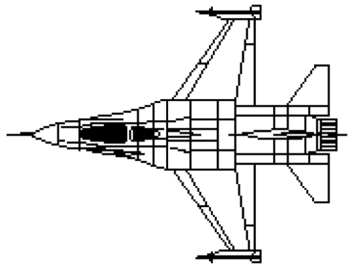


13.4) contd.

M_∞: 0.75 at 28 kft
AR: 7.2
$(L/D)_{\max}$: 18
Name: C-17 Globemaster III
Original company: McDonnell-Douglas



M_∞: 2.04 at 40 kft
AR: 3.2
$(L/D)_{\max}$: 12
Name: F-16 Fighting Falcon
Original company: General Dynamics



13.5)

Given:

T-41 with the following characteristics:

$U_\infty = 125 kts = 74.0 ft/s$
$h = 10{,}000 ft$
$b = 35 ft$
$c = 7 ft$
rectangular wing planform

The boundary layer transition location for subsonic flow is assumed to occur for:

$$\mathrm{Re}_{x_{tr}} = 500{,}000 = \frac{\rho_\infty U_\infty x_{tr}}{\mu_\infty}$$

so the transition location is found by:

$$x_{tr} = \frac{500{,}000 \mu_\infty}{\rho_\infty U_\infty}$$

13.5) contd.

For a standard day at 10,000 ft:

$$\rho_\infty = (0.73859)(0.002377) = 0.001756 \text{ slug/ft}^3$$

$$\mu_\infty = (0.94569)(3.740\times10^{-7}) = 3.537\times10^{-7} \text{ lb-s/ft}^2$$

Yielding a transition location of:

$$x_{tr} = \frac{500{,}000(3.537\times10^{-7})}{(0.001756)(74.0)} = 1.36 \text{ ft}$$

The boundary layer does not remain laminar for the entire airfoil section.
Since the wing has a chord of 7 ft, transition occurs at 19.4% of the wing chord, which means the laminar boundary layer is not having a large impact on the wing.