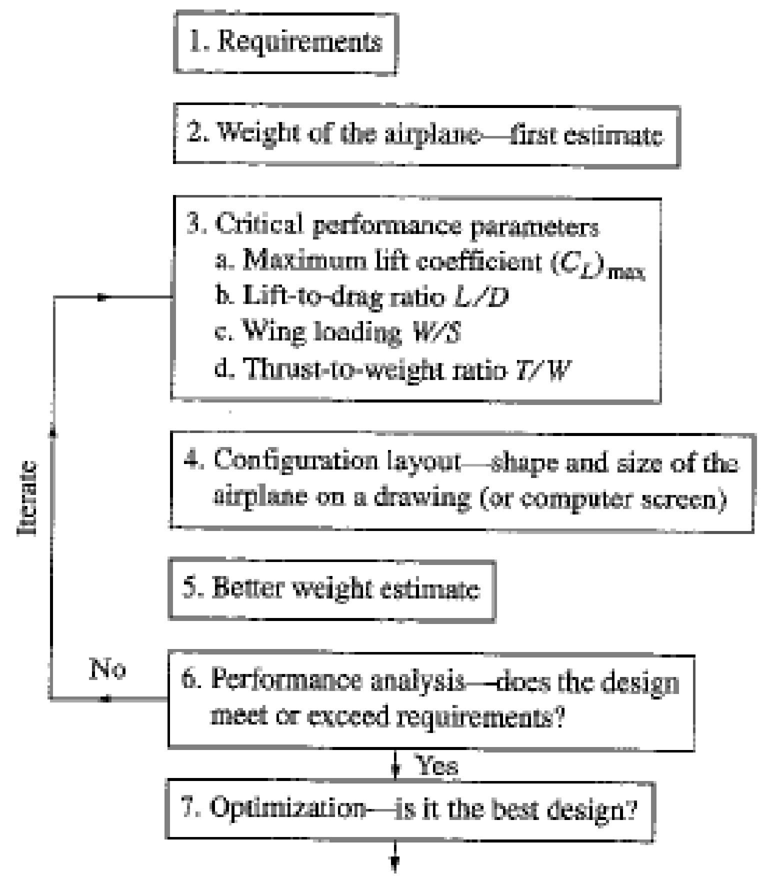
# SEVEN POINTS FOR CONCEPTUAL DESIGN

The Seven Intellectual Pivot Points for Conceptual Design



# REQUIREMENTS

Range
Take off Distance
Stall velocity
Endurance
Maximum velocity
Rate of Climb
Maximum turn rate
Maximum ceiling
Cost
Reliability

1. Maximum level speed at midcruise weight: 250 mi/h
2. Range: 1,200 mi.
3. Ceiling: 25,000 ft.
4. Rate of climb at sea level: 1,000 ft/min.
5. Stalling speed: 70 mi/h.
6. Landing distance (to clear a 50-ft obstacle): 2,200 ft.
7. Takeoff distance (to clear a 50-ft obstacle): 2,500 ft.

# WEIGHT

Lift has to be larger than Weight

$$W_0 = W_{\text{crew}} + W_{\text{payload}} + W_{\text{fuel}} + W_{\text{empty}}$$

$$W_0 = W_{\text{crew}} + W_{\text{payload}} + \frac{W_f}{W_0} W_0 + \frac{W_e}{W_0} W_0$$

$$\boxed{W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - W_f / W_0 - W_e / W_0}}$$

Corning suggests the average passenger weight of 160 lb, plus 40 lb of baggage per passenger. A more recent source is Raymer who suggests an average passenger weight of 180 lb plus 40 to 60 lb of baggage per person in the cargo hold.

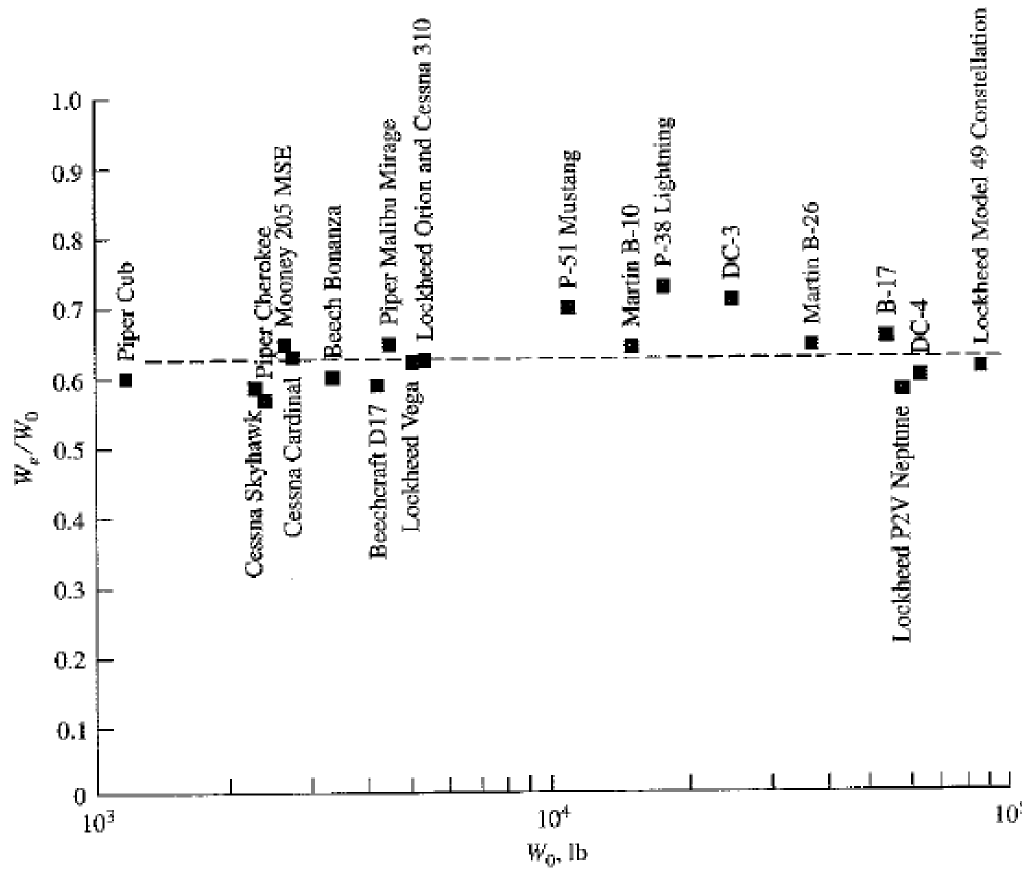
**The crew** comprises the people necessary to operate the airplane in flight. For our airplane, the crew is simply the pilot.

***Payload*** is what the airplane is intended to transport-passengers, baggage, freight. etc. If the airplane is intended for military combat use, the payload includes bombs, rockets.

***Fuel weight Wf*** is the weight of the fuel in the fuel tanks. Since fuel is consumed during the course of the flight, ***Wf*** is a variable, decreasing with time during the flight.

***Empty weight We*** This is the weight of everything else: the structure, engines, electronic equipment (including radar computers, communication devices, etc.). landing gear, fixed equipment (seats galleys, etc.), and anything else that is not crew, payload, or fuel.

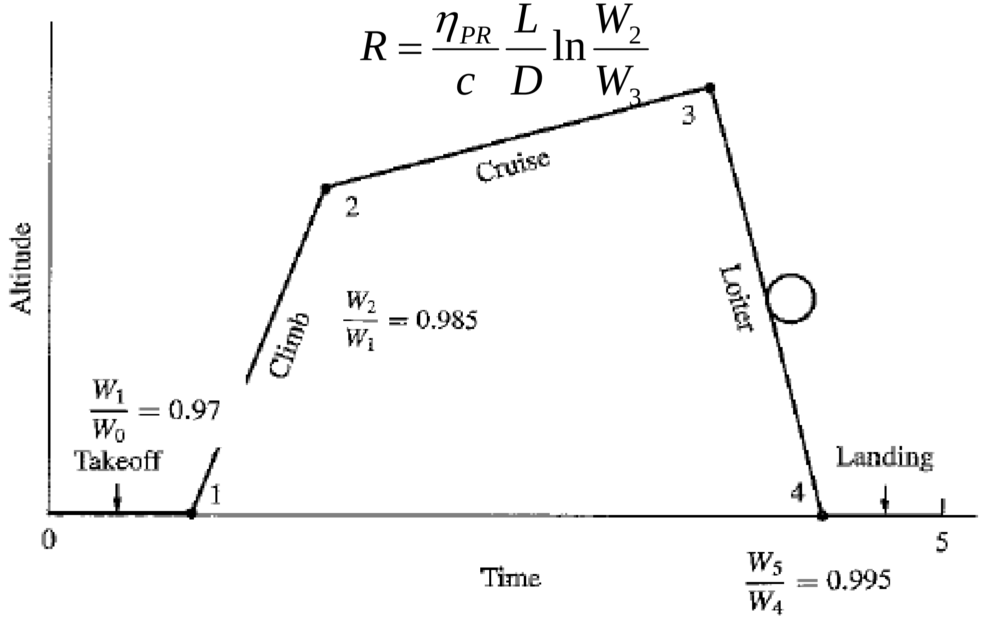
Most airplane designs are evolutionary rather than revolutionary; a new design is usually an evolutionary change from previously existing airplanes. For this reason, historical, statistical data on previous airplanes provide a starting point for the conceptual design of a new airplane. Data for 19 airplanes covering the time period from 1930 to the present are shown.




The values of We/ W0 tend to cluster around a horizontal line at We/ W0 = 0.62. For gross weights above 10,000 Ib, We/ W0 tends to be slightly higher for some of the aircraft. However, there is no technical reason for this; rather, the higher values for the heavier airplanes are most likely an historical phenomenon.

The amount of fuel depends on the efficiency, TSFC through the Breguet equation

$$R = \frac{\eta_{PR}}{c}\frac{L}{D}\ln\frac{W_2}{W_3}$$




The total fuel consumed during the mission is that consumed from the moment the engines are turned on at the airport to the moment they are shut down at the end of the flight. Between these times, the flight of the airplane can be described by a ***mission profile.*** It starts at point 0, when the engines are first turned on. The takeoff segment is denoted by the line segment 0-1, which includes warm-up, taxiing, and takeoff.

Segment 1-2 denotes the climb to cruise altitude (the use of a straight line here is only schematic and is nor meant to imply a constant rate of climb to altitude). Segment 2-3 represents the cruise, which is by far the largest segment of the mission. Segment 2-3 shows an increase in altitude during cruise, consistent with an attempt to keep *CL* (and hence *LID)* constant as the airplane weight decreases because of the consumption of fuel. Segment 3-4 denotes the descent, which includes loiter time to account for air traffic delays; for design purposes, a loiter time of 20 min is commonly used. Segment 4-5 represents landing.

Each segment of the mission profile is associated with a ***weight fraction*** defined as the airplane weight at the end of the segment divided by the weight at the beginning of the segment.

$$\frac{W_5}{W_0} = \frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}$$

$$W_f = W_0 - W_5$$

$$\frac{W_f}{W_0} = 1 - \frac{W_5}{W_0}$$

However, at the end of the mission, the fuel tanks are not completely empty. There should be some fuel left in reserve at the end of the mission in count of weather conditions or traffic problems require that the pilot of the airplane drive to another airport, or spend a longer-than-normal time in a holding pattern. Also, geometric design of the fuel tanks and the fuel system leads to some trapped fuel which is unavailable at the end of the flight. Typically, a 6% allowance is made for reserve and trapped fuel.

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_5}{W_0}\right)$$

For the calculation of the fuel consumed in cruise we use the Braguet eq.

$$R = \frac{\eta_{PR}}{c}\frac{L}{D}\ln\frac{W_2}{W_3}$$

This parameters are usually taken for an initial guess

$$\eta_{pr} = 0.85$$

$$c = 0.4\frac{\text{lb}}{\text{hp}\cdot\text{h}}\frac{1\ \text{hp}}{550\ \text{ft}\cdot\text{lb/s}}\frac{1\ \text{h}}{3{,}600\ \text{s}}$$

$$c = 2.02 \times 10^{-7}\frac{\text{lb}}{(\text{ft}\cdot\text{lb/s})}(\text{s})$$

$$(L/D)_{max} = 14$$

# ESTIMATION OF THE CRITICAL PERFORMANCE PARAMETERS

**Maximum Lift Coefficient**

the root airfoil section is relatively thick (about 15%-17%), and the wing airfoil shape tapers to a thinner section at the tip (about 12%). Structurally the wing bending moment is greatest at the root; a thicker airfoil readily allows the design for greater structural strength at the root. Aerodynamically, an 18% airfoil will stall at a lower angle of attack than a 12% airfoil. Hence, a wing which has airfoil sections which taper from 18% thick at the root to 12% thick at the tip will tend to stall first at the wing root, with attached flow still at the tip. The resulting buffeting that occurs at stall at the root is a warning to the pilot, while at the same time the ailerons remain effective because flow is still attached at the tip. Finally, a thicker wing section at the root allows additional volume for the storage of fuel in the wing.

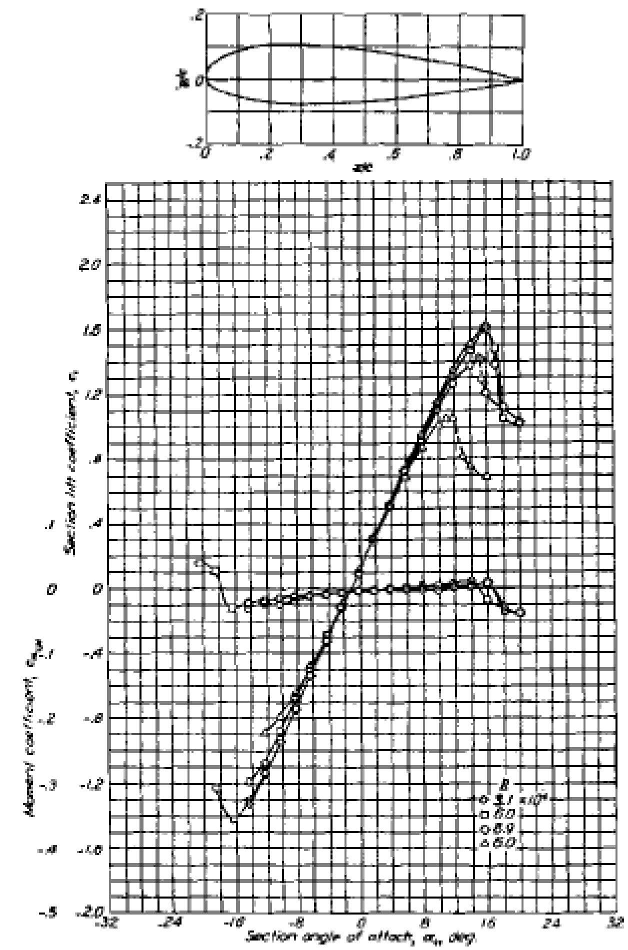



**Figure 8.3** Lift coefficient, moment coefficient, and airfoil shape for the NACA 23018 airfoil.

The resulting $C_{L,max}$ for the wing will be an average of the root and tip section values, depending on the planform taper ratio and the degree of geometric twist of the wing (if there is any). Also $C_{L,max}$ for the finite wing is less than that for the airfoil due to three-dimensional flow effects. Since we have not laid out the planform shape or twist distribution yet, we will assume that $C_{L,max}$ is a simple average of those for the airfoil sections at the root and tip, reduced by 10% for the effect of a finite aspect ratio.

In most airplane designs, wing loading is determined by considerations of $V_{stall}$ and landing distance

$$V_{stall} = \sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{(C_L)_{max}}} \longrightarrow \frac{W}{S} = \frac{1}{2}\rho_\infty V_{stall}^2 (C_L)_{max}$$

landing distance is the sum of the approach distance **sa,** the flare distance sf and the ground roll **sg.**

$$R = \frac{V_f^2}{0.2g} \qquad h_f = R(1-\cos\theta_a) \qquad s_a = \frac{50-h_f}{\operatorname{Tan}\theta_a} \qquad s_f = R\sin\theta_a$$

From the stall velocity the flare velocity, the radius, sa and sf can be calculated. Using the requirements the value of sg can be computed and using equation

$$s_g = jN\sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{(C_L)_{max}}} + \frac{j^2(W/S)}{g\rho_\infty(C_L)_{max}\mu_r}$$

***W/S can be calculated (j***= 1.15 for commercial airplanes N is the time increment for free roll immediately after touchdown, before the brakes are applied, N = 3 $\mu_r$= 0.4)

Once W/S is known, the area of the wing can be estimated as $S = \frac{W_0}{W/S}$

The value of T/W determines the takeoff distance, rate of climb, and maximum velocity

## TAKE OFF

$Cl_{max}$ is relative to flaps partially extended then we recalculate $V_{stall}$ $$V_{stall} = \sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{(C_L)_{max}}}$$

A 20 degree flap deflection can be assumed. Substituting the values in

$$s_g = \frac{1.21(W/S)}{g\rho_\infty (C_L)_{max}(T/W)}$$

To calculate the distance in airborne flight we apply the following equations

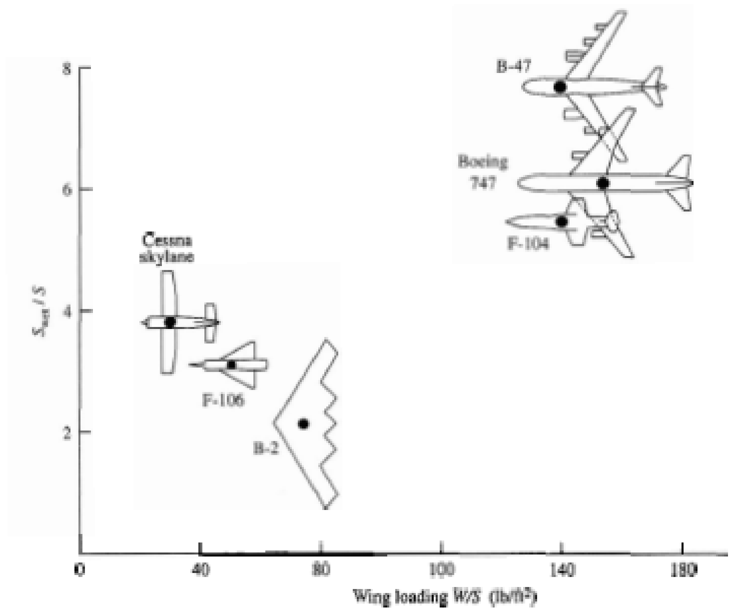
$$V_{stall} = \sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{(C_L)_{max}}} \qquad R = \frac{6.96(V_{stall})^2}{g} \qquad \theta_{OB} = \cos^{-1}\left(1 - \frac{h_{OB}}{R}\right) \qquad s_a = R\sin\theta_{OB}$$

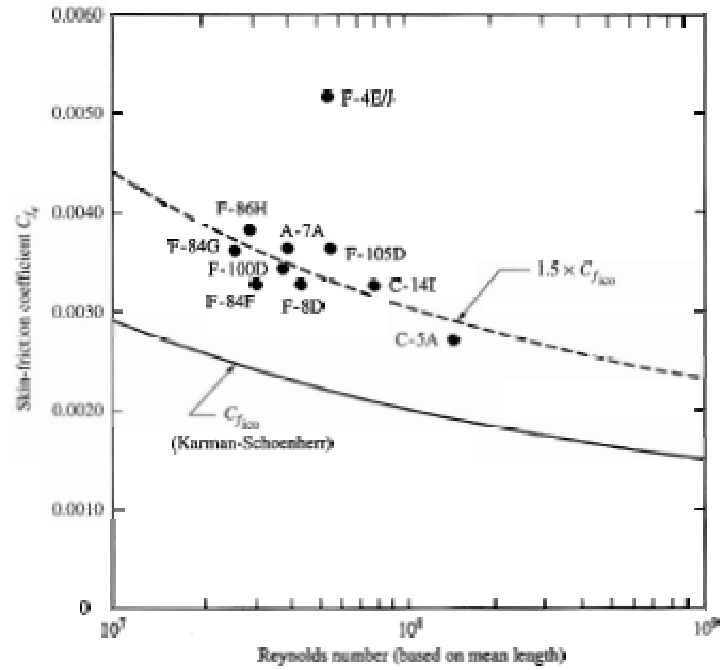
To match the requirements sa+sg=f(T/W). With this we can compute a first value of T/W

If we solve a propeller airplane, $$P_R = TV_\infty = \frac{T}{W}W_0V_\infty \qquad P = \frac{P_A}{\eta_{pr}}$$

# RATE OF CLIMB

We need to estimate $$C_{D,0} = \frac{S_{wet}}{S} C_{fe}$$




To estimate k, we use

$$\left(\frac{L}{D}\right)_{max} = \sqrt{\frac{1}{4C_{D,0}K}} \longrightarrow K = \frac{1}{4C_{D,0}(L/D)^2_{max}}$$

$$(R/C)_{max} = \frac{\eta_{pr} P}{W} - \left(\frac{2}{\rho_\infty}\sqrt{\frac{K}{3C_{D,0}}}\frac{W}{S}\right)^{1/2} \frac{1.155}{(L/D)_{max}}$$

From this I can determine T/W or P/W

# MAXIMUM VELOCITY

$$T = D = \frac{1}{2}\rho_\infty V_\infty^2 S C_{D,0} + \frac{2KS}{\rho_\infty V_\infty^2}\left(\frac{W}{S}\right)^2$$

$$\frac{T}{W} = \frac{1}{2}\rho_\infty V_\infty^2 \frac{C_{D,0}}{W/S} + \frac{2K}{\rho_\infty V_\infty^2}\frac{W}{S}$$

$$\eta_{pr} P = T V_\infty$$

Be careful to use the right weight of the plane for the required maximum velocity.

**Tractor Configuration** Advantages:

1. The heavy engine is at the front, which helps to move the center of gravity forward and therefore allows a smaller tail for stability considerations.

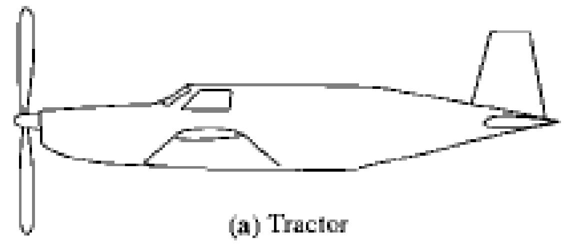
**2.** The propeller is working in an undisturbed free stream.

*3.* There **is** a more effective flow of cooling air for the engine.

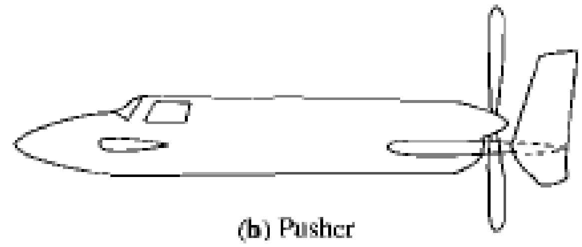
Disadvantages:

1. The propeller slipstream disturbs the quality of the airflow over the fuselage and wing root.

***2.*** The increased velocity and flow turbulence over the fuselage due to the propeller slipstream increase the local skin friction on the fuselage.



(a) Tractor



(b) Pusher

**Pusher Configuration** Advantages:

**1.** Higher-quality (clean) airflow prevails over the wing and fuselage.

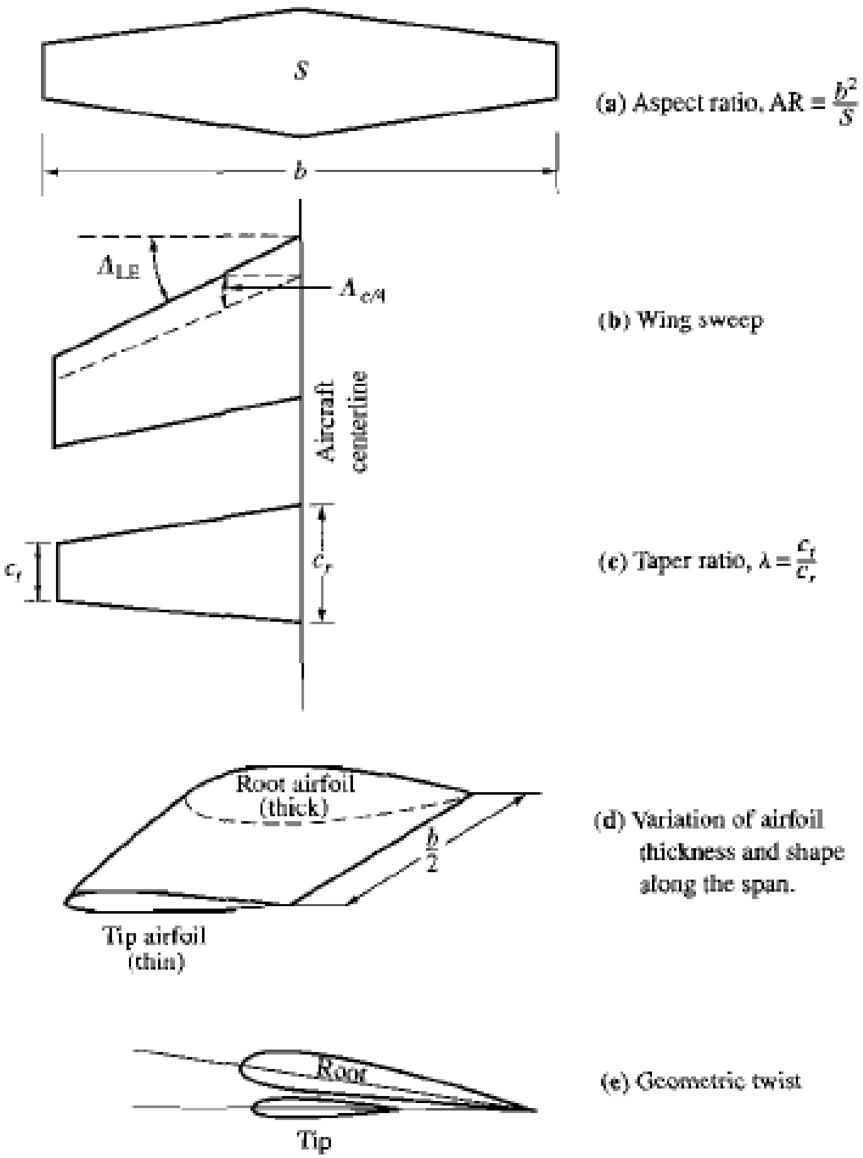
**2.** The inflow to the rear propeller induces a favorable pressure gradient at the rear of the fuselage, allowing the fuselage to close at a steeper angle without flow separation. This in turn allows a shorter fuselage, hence smaller wetted surface area.

***3.*** Engine noise in the cabin area is reduced.

4. The pilot's front field of view is improved.

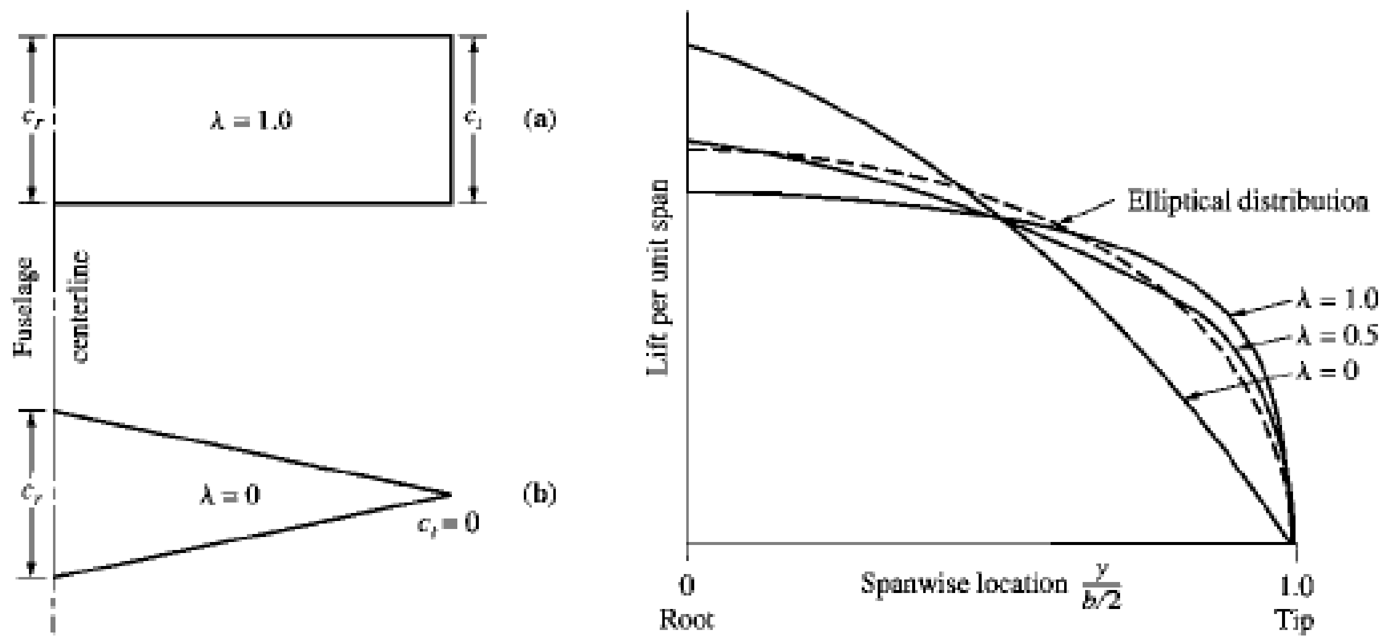
Disadvantages:

1. The heavy engine is at the back, which shifts the center of gravity rearward, hence reducing longitudinal stability.
2. Propeller is more likely to be damaged by flying debris at landing.
3. Engine cooling problems are more severe.

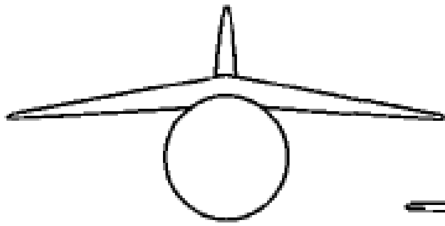



There are two sweep angles of importance, the leading-edge sweep angle and the sweep angle of the quarter chord. The leading-edge sweep angle is most relevant to supersonic airplanes because to reduce wave drag, the leading edge should be swept behind the Mach cone. The sweep angle of the quarter-chord is of relevance to high-speed subsonic airplanes near the speed of sound. The taper ratio is the ratio of the tip chord to the root chord.

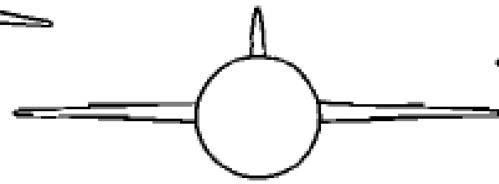
Shown in Fig. is the case when the tip chord incidence angle is smaller than that of the root chord; this configuration is called ***washout.*** The opposite case, when the tip is at a higher incidence angle than the root, is called ***wash-in.*** For minimum induced drag, we want to have a spanwise elliptical lift distribution, which for an untwisted wing implies an elliptical planform shape. However, the higher production costs associated with a wing with curved leading and trailing edges in the planform view are usually not justified in view of the cheaper costs of manufacturing wings with straight leading and trailing edges. Moreover, by choosing the correct taper ratio, the elliptical lift distribution can be closely approximated.



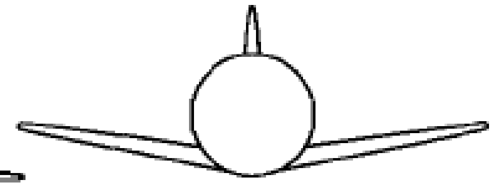
$c_r$
$\lambda = 1.0$
$c_t$
(a)
Fuselage centerline
$c_r$
$\lambda = 0$
$c_t = 0$
(b)
Lift per unit span
Elliptical distribution
$\lambda = 1.0$
$\lambda = 0.5$
$\lambda = 0$
0
Root
Spanwise location $\frac{y}{b/2}$
1.0
Tip



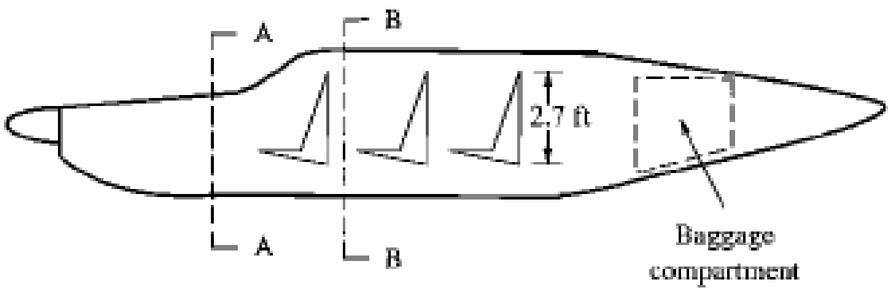
(a) High wing



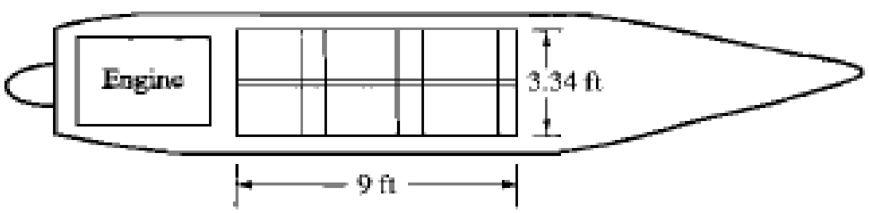
(b) Mid wing



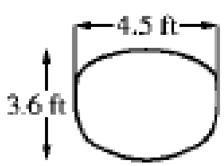
(c) Low wing



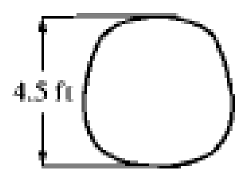

(a) Side view



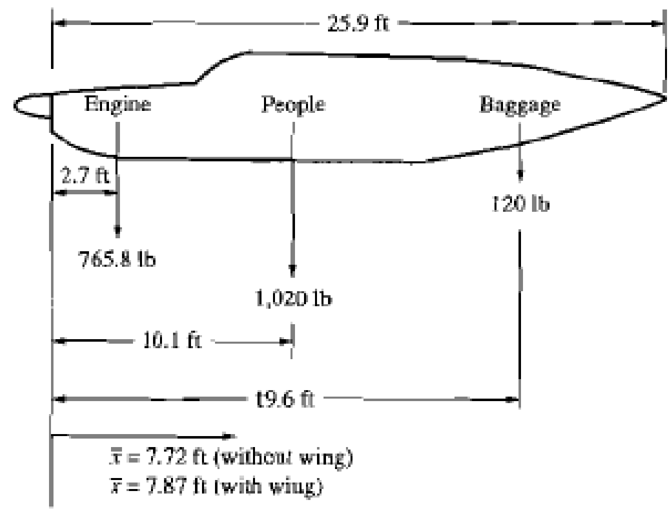

(b) Top view

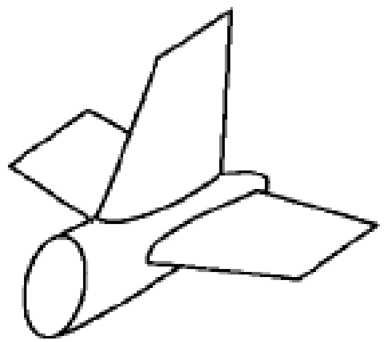



(c) Section A-A

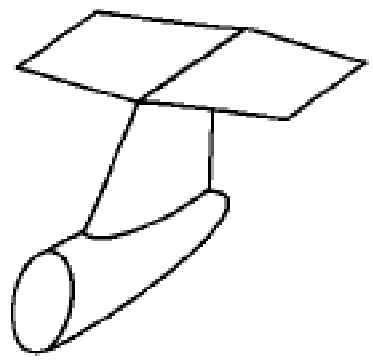



(d) Section B-B

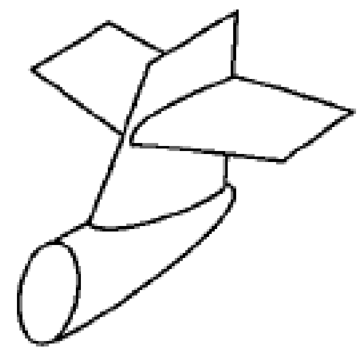




Conventional



T-tail



Cruciform