**Design Of Structural Components - Wing, Fuselage and Tail**

- Mainplane: Airfoil cross-section shape, taper ratio selection, sweep angle selection, wing drag estimation.

  Spread sheet for wing design.

- Fuselage: Volume consideration, quantitative shapes, air inlets, wing attachments; Aerodynamic considerations and drag estimation. Spread sheets.

- Tail arrangements: Horizontal and vertical tail sizing.

  Tail planform shapes.

  Airfoil selection type.

  Tail placement.

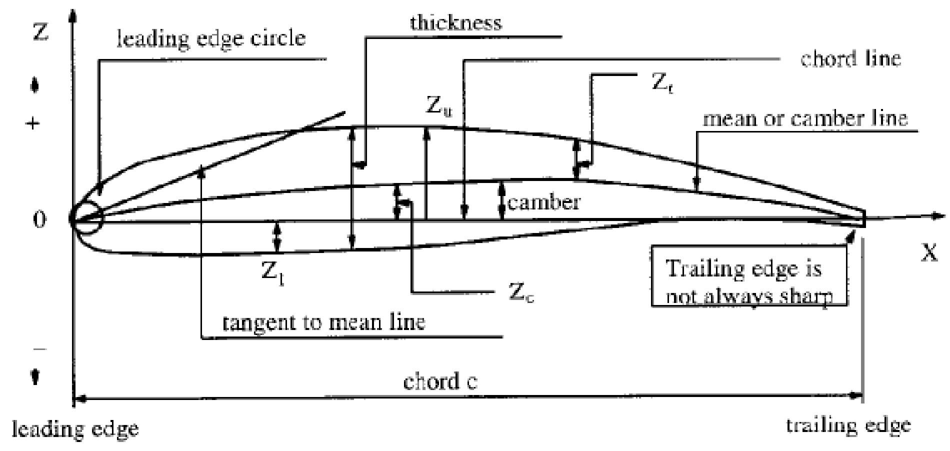
  Spread sheets for tail design

# Main Wing Design

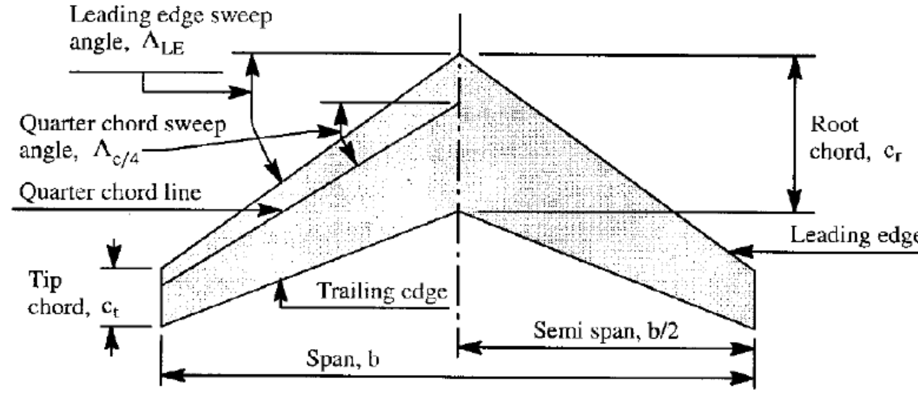
## 1. Introduction:

- Wing is the main lifting surface of the aircraft.
- Wing design is the next logical step in the conceptual design of the aircraft, after selecting the weight and the wing-loading that match the mission requirements.
- The design of the wing consists of selecting:
  i) the airfoil cross-section,
  ii) the average (mean) chord length,
  iii) the maximum thickness-to-chord ratio,
  iv) the aspect ratio,
  v) the taper ratio, and
  vi) the sweep angle which is defined for the leading edge (LE) as well as the trailing edge (TE)
- Another part of the wing design involves enhanced lift devices such as leading and trailing edge flaps.
- Experimental data is used for the selection of the airfoil cross-section shape.
- The ultimate "goals" for the wing design are based on the mission requirements.
- In some cases, these goals are in conflict and will require some compromise.

## Airfoil Geometry:




## Wing Geometry:



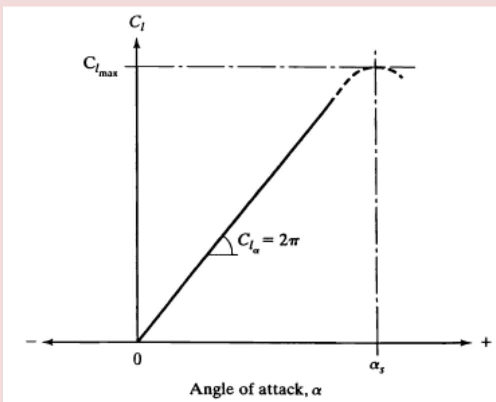
# Main Wing Design (contd)

## 2. Airfoil Cross-Section Shape:

- The shape of the wing cross-section determines the pressure distribution on the upper and lower surfaces of the wing.
- The pressure distribution integrated around the wing is the lift force (*L*).
- The lift force normalized by the wing area and dynamic pressure is the lift coefficient ($C_l$)
- Notation: $C_l$ 2-D lift coefft; $C_L$ 3-D lift coefft.
- A plot of lift coefficient versus AoA for a symmetric 2-D wing section is shown in Fig.
- Linear airfoil theory indicates that for a 2-D (infinite AR) airfoil section,

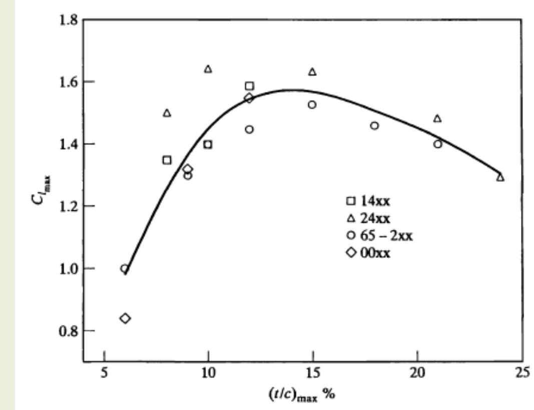
$$C_l \propto \alpha$$

$$\frac{dC_l}{d\alpha} = 2\pi / rad$$




- At stall AoA ($\alpha_s$), $C_l = C_{l_{max}}$
- The maximum lift coefficient is also affected by the maximum thickness-to-chord ratio $(t/c)_{max}$

- Effect of $(t/c)_{max}$ on $C_{l_{max}}$ for a variety of 2-D airfoil sections is shown in the Fig.




- It can be noted that $C_{l_{max}}$ occurs for

$$(t/c)_{max} \cong 14\%$$

- Beyond the optimum $(t/c)_{max}$, the decrease in $C_{l_{max}}$ is caused by a flow separation that occurs downstream of the maximum thickness point on the airfoil ("trailing-edge separation) .
- Such flow separations are more likely to occur at lower chord Reynolds numbers.
- From a structural point of view, there is a benefit to having thicker wing cross-sections (*t*).
- A small increase in the thickness in the wing has a large benefit toward increasing the bending stiffness.
- As a result, the load carrying elements in the wing can be made lighter, and the structure weight of the wing can be reduced by increasing $(t/c)_{max}$ because, $W_w \propto (t/c)_{max}^{0.5}$

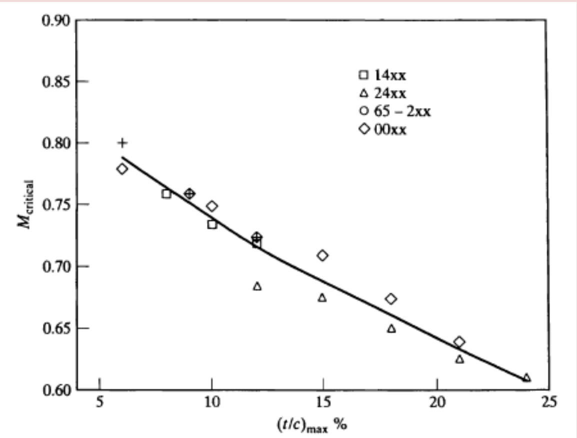
# Main Wing Design (contd)

## Airfoil Cross-Section Shape: (contd)

- Increasing $(t/c)_{max}$ also increases the internal volume of the wing and more fuel can be carried in the main wing.
- Historical data show that the weight of fuel carried in the main wing, increases as:

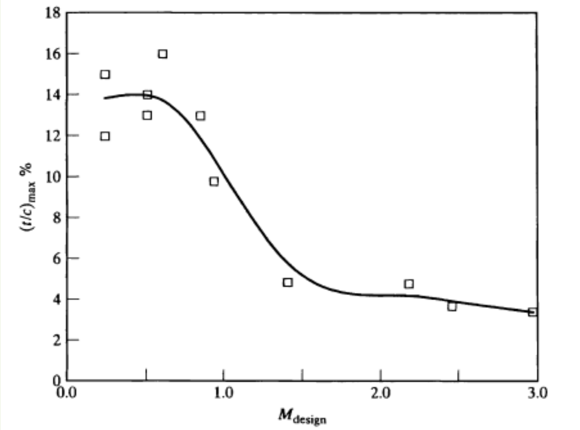
$$W_f \propto (t/c)_{max}$$

- The effect of $(t/c)_{max}$ on the critical Mach number ( $M_{critical}$ ) for a variety of 2-D airfoil sections is shown in the Fig.
- The critical Mach number for a wing section decreases with $(t/c)_{max}$




- Critical Mach No.
  - An aircraft with subsonic cruise Mach number may have conditions over the wing where the Mach number is transonic or supersonic.
  - This would result in an increase of the base drag, $C_{D_0}$ .
  - The cruise M ach number at which the local Mach number over the wing is supersonic is called the critical Mach number, $M_{critical}$

- Historical values of $(t/c)_{max}$ of aircraft as a function of cruise Mach number is shown in the Fig.

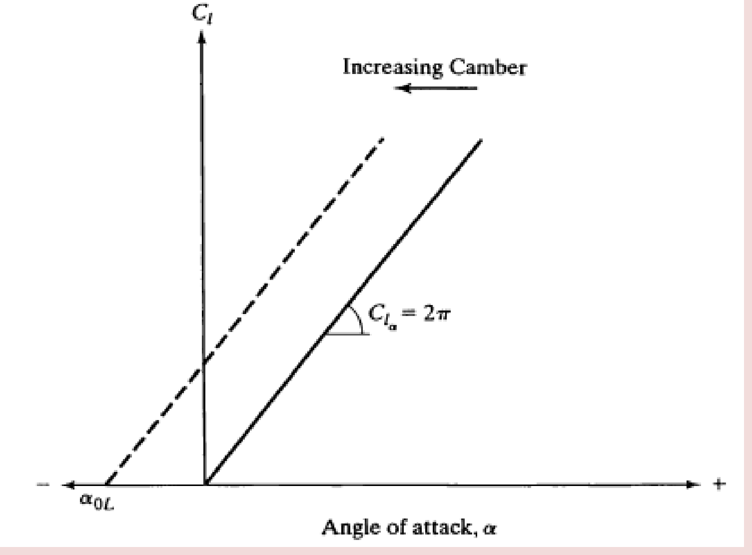


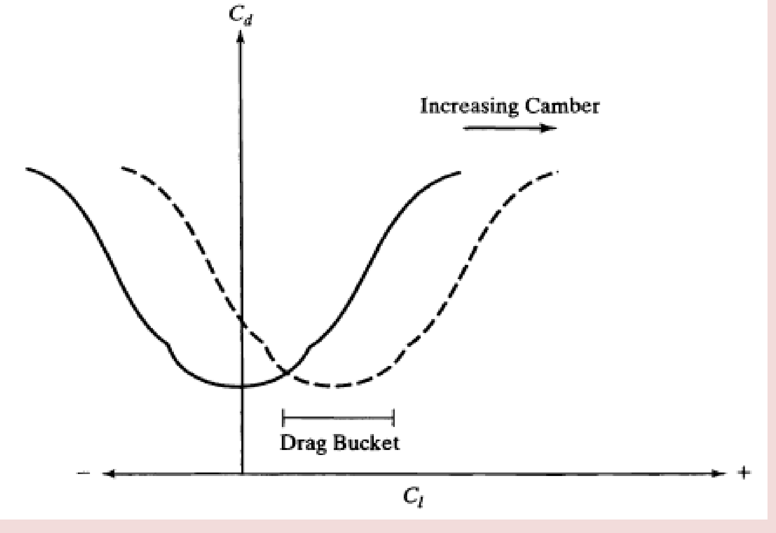

- It can be noted that $(t/c)_{max}$ decreases as $M_{design}$ increases.
- The airfoil sections on supersonic aircraft tend to be thinner and have a smaller nose radius.
- The purpose is to minimize the base and wave drag components.
- In addition, the critical Mach number for the wing is designed to be different from that of the other components (horizontal and vertical tail sections and fuselage) so that drag buildup that occurs near Mach 1 does not simultaneously occur for all the components at the same free-stream Mach number.

# Main Wing Design (contd)

## Airfoil Cross-Section Shape: (contd)

- **The effect of camber** on $C_l$ *vs* $\alpha$ and $C_d$ *vs* $C_l$ is shown in the Figs.




- Camber is a parameter corresponding to a curving of the airfoil section.
- It can be noted from the plot of $C_l$ *vs* $\alpha$ that without camber (solid line), a wing section produces zero lift at zero AoA.
- It can also be noted from the $C_d$ *vs* $C_l$ plot, that the base drag is a minimum when AoA is zero.
- Many airfoil types have a noticeable depression near the minimum drag coefficient. This depression is referred to as the "drag bucket".
- The effect of positive camber is to shift the AoA for zero lift ( $\alpha_{0_L}$) to negative values.
- As shown in the bottom figure, positive camber shifts the value of the lift coefficient for minimum drag (drag bucket) from zero to positive values.
- This is important for in the design of aircraft for efficient cruise.

# Main Wing Design (contd)

**Airfoil Shape Selection: (contd)**

i) At cruise conditions, the required lift equals the weight:

$$(C_L)_{cruise} = q\left(\frac{W}{S}\right)_{cruise}$$

- For long-range aircraft, the weight of the aircraft can change significantly from the start of the cruise to the end of cruise.
- Assuming that the cruise altitude and speed remain the same, this means that the required lift coefficient changes during cruise.
- The average of the required lift coefficient during cruise is referred to as the "design $C_L$".

ii) In order to have the maximum range during cruise, the drag needs to be a minimum.

- The airfoil shape is then selected based on two criteria:

  1) that it can provide the "design $C_L$" in level flight, and

  2) that the range of $C_L$ values from the start of cruise to end of cruise is within the "drag bucket".

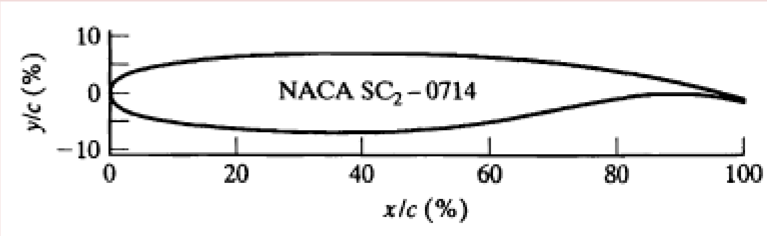
- The choice of the airfoil section type that is most suitable for a new design depends on the general type of the aircraft.
- For example, smaller low sub-sonic speed $(M_{cruise} \le 0.4)$ aircraft would likely to use the NACA 5-digit series with a thickness-to-chord ratio near the optimum 14%.

| 5-digit series | NACA 23012 |
|---|---|
| 23012: maximum camber $= 0.02c$ | |
| 23012: design $C_l = 0.15(2) = 0.30$ | |
| 23012: position of maximum camber $= 0.30/2 = 0.15c$ | |
| 23012: thickness/chord ratio $= 0.12$ | |


- High sub-sonic speed aircraft $(M_{cruise} \cong 0.8)$, such as commercial or business jets, use the NACA supercritical (SC) airfoils.
- A typical super critical airfoil is shown in the Fig.

# Main Wing Design (contd)

## Airfoil Shape Selection: (contd)




- This super critical airfoil was developed to give higher critical Mach numbers compared to other shapes with comparable thickness-to-chord (*t/c*) ratio.
- This airfoil is used on the Boeing C-17 and 777, and the Airbus A-330/340.
- Supersonic military aircraft use the NACA Series 6 airfoils since they have good characteristics at both subsonic and supersonic Mach numbers.
- In order to reduce wave drag in supersonic flight, the (t/c) ratio should be relatively small, of the order of 4%.
- Following the selection of the airfoil section shape, the conversion from the 2-D lift coefficient, $C_l$ , to the 3-D lift coefficient, $C_L$ depends on the wing planform design. (wing span, taper ratio, sweep angle)

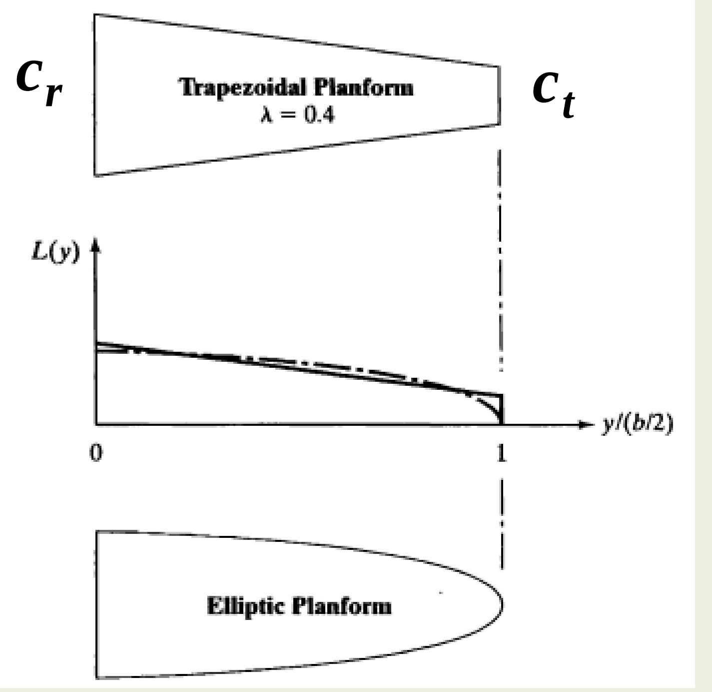
# Main Wing Design (contd)

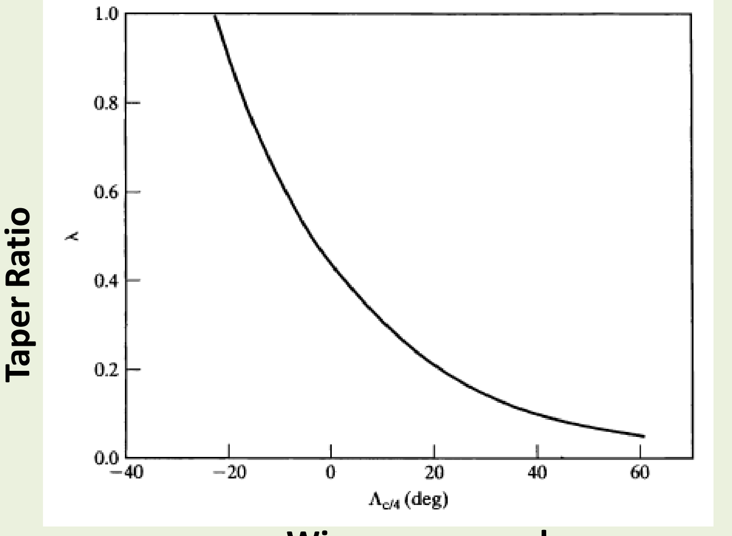
## 3. Wing Taper Ratio Selection:

- The wing taper ratio is defined as:

$$\lambda = \frac{c_t}{c_r} = \frac{tip\,chord}{root\,chord}$$

- Taper ratio should be selected such that the amount of lift-induced drag is minimum.
- The 'lifting line theory' indicates that for an unswept and untwisted wing, an ellipse-shaped wing planform gives the minimum drag.
- An ellipse-shaped wing planform gives an elliptic spanwise (*y*) variation in lift.
- This lift profile can be approximated reasonably well with a more simply constructed trapezoidal wing with a taper ratio of 0.4, as shown in Fig.
- The taper ratio for a trapezoidal wing, which gives the minimum lift-induced drag, is slightly dependent on the aspect ratio and more significantly dependent on the wing sweep angle.




Wing sweep angle

# Main Wing Design (contd)

## 4. Wing Sweep Angle Selection:

- **Wing sweep** is defined as the angle between a line perpendicular to the aircraft centerline and a line parallel to the leading edge $(\Lambda_{LE})$ .
- The primary reason for adding a sweep angle to a wing is to increase its section critical Mach number, as shown in Fig. 1.
- Sweep reduces the effective Mach number at the leading edge as $M_{\text{effective}} = M_\infty \cos(\Lambda_{LE})$.
- The critical Mach number in this case is increased as

$$M_{\text{critical}} \propto \frac{1}{\cos^m(\Lambda)}$$

where m is a function of the lift coefficient, $C_L$ .

- **Disadvantages of adding sweep:**

i) Lowers the lift through a lower effective dynamic pressure, according to $q_{\text{effective}} = q_\infty \cos^2(\Lambda)$

ii) Increases the wing weight, according to

$$W_{\text{wing}} \propto [\tan(\Lambda)]^2$$

iii) Degrades take-off and landing characteristics because enhanced lift devises are less effective.

- Fig. 2 shows the historic trend for sweep angle as a function of the cruise Mach number.

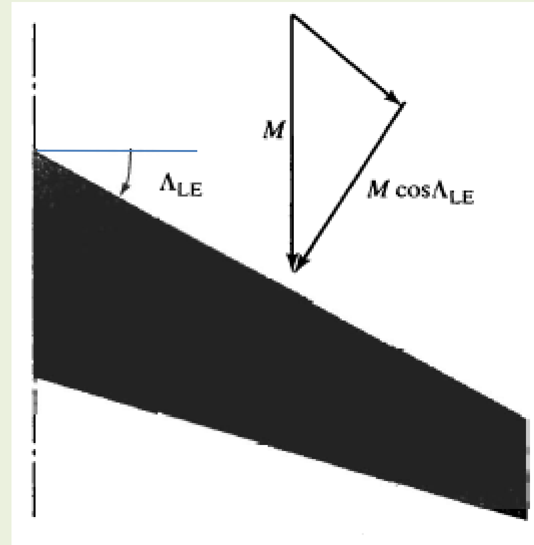



Fig. 1

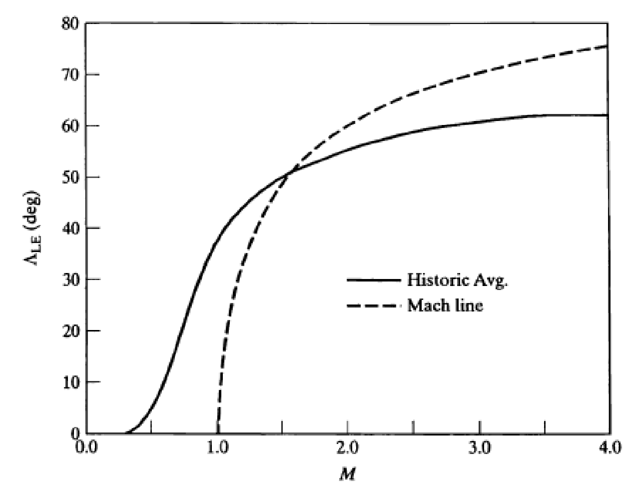
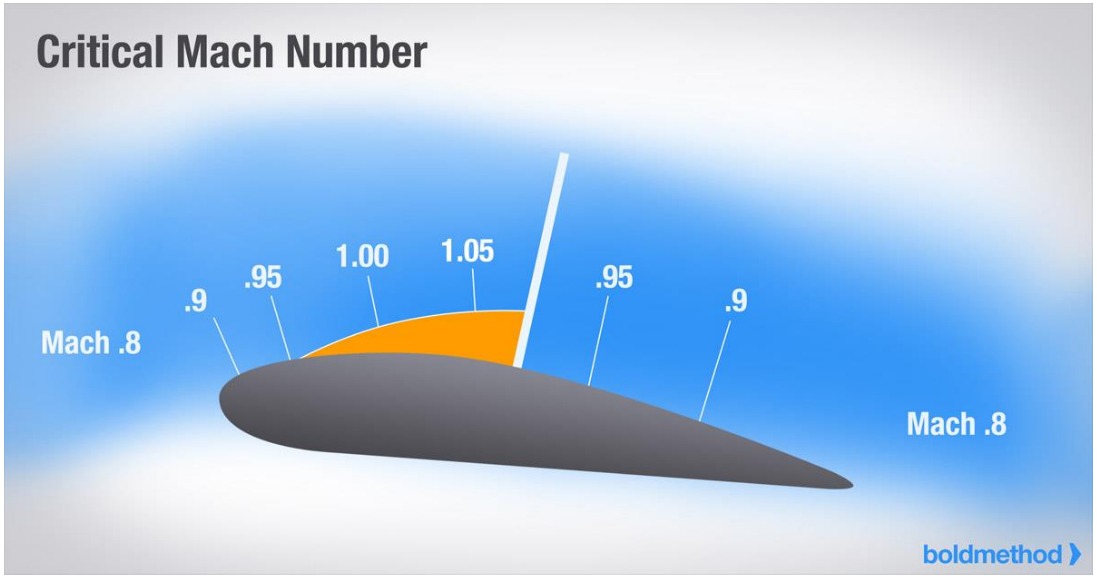



Fig. 2

- The dotted line shows the minimum sweep angle for the LE to be inside the Mach cone for supersonic flight: $\Lambda = 90° - \sin^{1}(1/M_\infty)$

In aerodynamics, the **critical Mach umber** (Mcr or M* ) of an aircraft is the lowest **Mach number** at which the airflow over some point of the aircraft reaches the speed of sound, but does not exceed it. At the lower **critical Mach number**, airflow around the entire aircraft is subsonic.



# Main Wing Design (contd)

## Wing Sweep Angle Selection - Summary:

1. For aircraft with cruise Mach number below 0.4 the wings are designed without sweep.

2. For transonic cruise numbers, sweep angles of approximately 30 degrees are used.

3. With cruise Mach numbers in the range 1.0 - 2.4, the sweep angles are close to the angle of the Mach line.

4. At Mach numbers greater than 2.4, the sweep angles are less than needed to have subsonic flow at the wing LE.

   This is primarily the result of the poor subsonic flight characteristics and the excessive structure weight that comes from having excessively large wing sweep.

# Main Wing Design (contd)

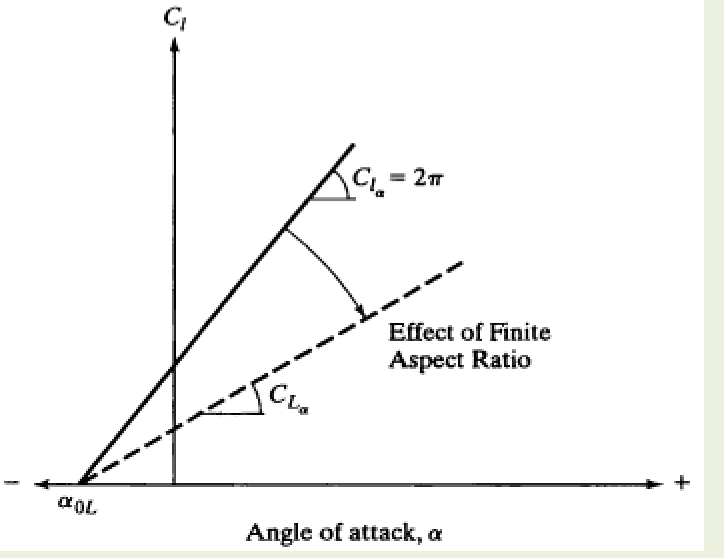
## 5) 3-D Lift Coefficient:

- Experimental airfoil data provide 2-D section coefficients.
- For aircraft design, they need to be converted to 3-D coefficients.
- The effect of finite wing span on the 2-D lift coefficient, $C_l$ , is to reduce the slope $dC_l/d\alpha$
- This is illustrated in the Fig. for a cambered airfoil
- The reduction in the slope amounts to a clockwise rotation of the $C_l$ versus $\alpha$ line about the $\alpha_{0L}$ point.
- The result is that at the same AoA, the lift generated by the 3-D wing will be less than that based the 2-D lift coefficient.
- For subsonic Mach numbers, the slope of the 3-D lift coefficient is given as,

$$\frac{dC_L}{d\alpha} = \frac{2\pi A}{2 + \sqrt{4 + (A\beta)^2\left(1 + \frac{\tan^2(\Lambda_{t/c})}{\beta^2}\right)}}$$

[ <1.0 for subsonic flows ]

where, $\beta = \sqrt{1 - M_{\mathrm{eff}}^2}$ and $M_{\mathrm{eff}} = M_\infty \cos(\Lambda_{\mathrm{LE}})$




- Using the slope, 3-D lift coefficient versus AoA is given by,

$$C_L = \frac{dC_L}{d\alpha}\alpha + C_{L_{\alpha=0}}$$ where $$C_{L_{\alpha=0}} = -\frac{dC_L}{d\alpha}\alpha_{0_L}$$

- **Note:** $\alpha_{0L}$ = 0 for uncambered airfoil, and it is negative for a positive cambered airfoil.
- If the design 2-D lift coefficient, $C_l$ , is provided at zero AoA, then the design 3-D coefficient, $C_L$, requires placing the airfoil at a slight positive AoA
- Otherwise, the design process requires selecting an airfoil section which provides a higher $C_l$ such that the lower 3-D coefficient matches the design $C_L$ at zero AoA.

# Main Wing Design (contd)

## 6. WING DRAG ESTIMATION

- The drag coefficient for the wing corresponds to the base drag, the lift-induced drag, and any additional drag that results from viscous losses such as produced by flow separations.

$$C_D = \underbrace{C_{D_0}}_{base} + \underbrace{kC_L^2}_{lift\ induced} + \underbrace{k'(C_L - C_{L_{\min_D}})}_{losses}$$

where

$$k = \frac{1}{\pi A e}$$ $$e = e'\left[1 - \left(\frac{d}{b}\right)^2\right]$$ $$e' \simeq 0.98$$

and, '*e*' is a wing efficieny factor that accounts for taper ratio and fuselage effects on the wing, and '**d/b**' is the ratio of the fuselage diameter to wing span ratio.

- The value ***k'*** is dependent on the LE radius and taper ratio and has values in the range

$$0.02 \leq k' \leq 0.16$$

- If the airfoil section was chosen so that the drag bucket encompasses the range throughout cruise, then $(C_L - C_{L_{\min_D}}) = 0$ and the drag coefficient simplifies to: $C_D = C_{D_0} + kC_L^2$

## BASE DRAG ESTIMATION

- The base drag coefficient, $C_{D_0}$ , is the 'zer0-lift' drag coefficient that corresponds to viscous skin friction and flow separations.
- In the best case, $C_{D_0} = C_{d_0}$ which is the minimum drag coefficient in the drag bucket for the 2-D wing section.
- In practice, $C_{D_0} > C_{d_0}$ as a result of flow disturbances caused by the wing attachments to the fuselage, wing-mounted external store such as fuel tanks, and surface imperfections such as hinge gaps at movable surfaces.
- The base drag coefficient, $C_{D_0}$ , is given by:

$$C_{D_0} = C_f \mathcal{F} Q \frac{S_{wet}}{S}$$

where

$C_f$ = Viscous drag coefficient (skin friction coeff)

$\mathcal{F}$ = Form factor $Q$ = Interference factor

$S_{wet}$ = Wetted surface area of the wing

$S$ = Wing area

# Main Wing Design (contd)

**BASE DRAG ESTIMATION (contd)**

- The wetted surface area $S_{wet}$ is the area of the wing exposed to outside air.

- For infinitely thin wing sections, the wetted area would be twice the wing planform area (*S*)

- Therefore, as an approximation:

For thin airfoils

$$S_{wet} \simeq 2.003S \quad \text{for} \quad (t/c)_{max} \leq 0.05.$$

For thicker airfoils

$$S_{wet} \simeq S[1.977 + 0.52(t/c)_{max})] \quad \text{for} \quad (t/c)_{max} > 0.05.$$

- The Skin Friction Coefficient ( $C_f$ ) is given by

$$C_f = \frac{1.328}{\sqrt{Re_x}} \quad \text{:Laminar,} \qquad Re_x = V_o x/\nu,$$

$$C_f = \frac{0.455}{(\log_{10} Re_x)^{2.58}(1 + 0.144\ M^2)^{0.65}} \quad \text{:Turbulent.} \quad \sqrt{Re_x} \geq 1000.$$

- The Form Factor ( $\mathcal{F}$ ) is given by:

$$\mathcal{F} = \left[1 + \frac{0.6}{(x/c)_m}\left[\frac{t}{c}\right] + 100\left[\frac{t}{c}\right]^4\right]\left[1.34\ M^{0.18}(\cos(\Lambda_{t/c_{max}}))^{0.28}\right]$$

$(x/c)_m$ location of maximum thickness point

- The Interference Factor $Q$ is intended to estimate the increase in base drag due to interference effects caused by the fuselage or wing attachments.

- These include nacelles or external stores mounted on n or near the wing, and filleted wing attachments.

Values of Interference Factor, $Q$, for different arrangements.

| | $Q$ |
|---|---|
| Wing-Mounted Nacelle or Store | 1.5 |
| Wing-Tip Missile | 1.25 |
| High-Wing, Mid-Wing | 1 |
| Well Filleted Low-Wing | 1 |
| Unfilletted Low-Wing | 1.25 |

- Once $C_L$ and $C_D$ are determined, the lift-to-drag is given by $\frac{L}{D} = \frac{C_L}{C_D}$

# Main Wing Design (contd)

## 7. Wing Planform Geometric Relations for Trapezoidal Shape :

- Wing span = $b = \sqrt{SA}$ , S = Wing area
- Aspect ratio = $A = \dfrac{2b}{c_r(1+\lambda)}$
- Root chord = $c_r = \dfrac{2b}{A(1+\lambda)}$ , Taper ratio= $\lambda = \dfrac{c_t}{c_r}$
- Tip chord = $c_t = \lambda c_r$
- Mean Aerodynamic Chord (mac)

$$\bar{c} = \frac{2c_r}{3}\frac{(1+\lambda+\lambda^2)}{(1+\lambda)}$$

- The normalized spanwise location of the 'mac' from the center span of the wing

$$\frac{\bar{y}}{b} = \frac{1}{6}\frac{1+2\lambda}{1+\lambda}$$

- Starting with the LE sweep angle, the sweep angle at any ( *x/c* ) location on the wing

$$\Lambda_{x/c} = \tan^{-1}\left[\tan\Lambda_{\mathrm{LE}} - \frac{x}{c}\frac{2c_r}{b}(1-\lambda)\right]$$

- The sweep angle at the quarter chord line

$$\Lambda_{c/4} = \tan^{-1}\left[\tan\Lambda_{\mathrm{LE}} - \frac{1}{4}\frac{2c_r}{b}(1-\lambda)\right]$$

# Main Wing Design (contd)

## 8. Spreadsheet for Wing Design

- A sample spreadsheet for wing design for the Super Sonic Business Jet (SSBJ) is shown in Fig.
- In the spreadsheet the input parameters are placed at two areas marked, "Design Parameters" and "Airfoil Data".
- The important airfoil data that are required for wing design calculations are:

1. $C_{l_{\max}}$;
2. $C_{l_\alpha} = d_{C_l}/d_\alpha$;
3. location of the aerodynamic center;
4. zero-lift angle of attack, $\alpha_{0L}$;
5. base (minimum) drag coefficient, $C_{d_0}$;
6. leading-edge radius, $r_{\mathrm{LE}}$;
7. extent of the drag bucket, $C_{l_{\min D}}$.

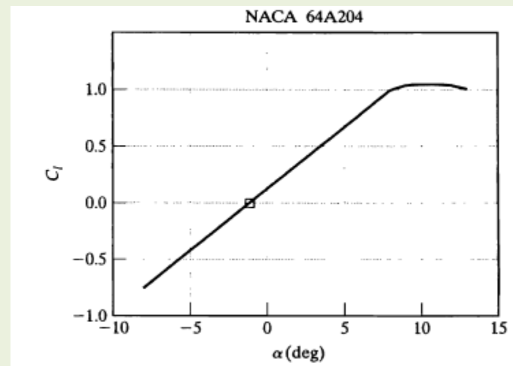
- The other design parameters and the Flight Condition data include:

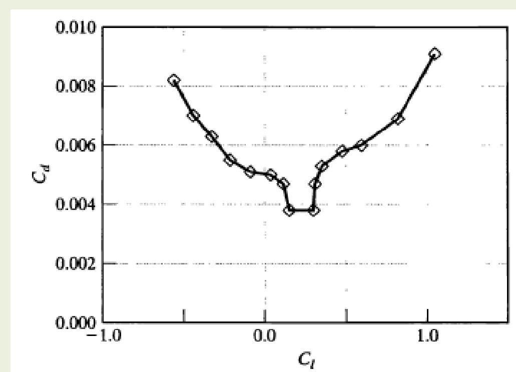
1. wing area, $S$;
2. wing aspect ratio, $A$;
3. leading-edge sweep angle, $\Lambda_{\mathrm{LE}}$;
4. maximum $t/c$;
5. taper ratio, $\lambda$.

1. cruise Mach number, $M$;
2. cruise altitude, $H$;
3. weight at the start and end of cruise;
4. dynamic pressure, $q$, at the start and end of cruise.

- The design calculations and plots are done as shown in the spreadsheet

### Case Study: SSBJ Wing Design



**Design Parameters**

| | | |
|---|---|---|
| M | 2.10 | |
| S | 519 | ft$^2$ |
| A | 2.0 | |
| $\Lambda_{LE}$ | 62 | deg |
| t/c | 0.04 | |
| λ | 0.00 | |
| W c-start | 88,817 | lbf/f^2 |
| W c-end | 53,976 | lbf/f^2 |
| q c-start | 531.07 | lbf/f^2 |
| q c-end | 347.41 | lbf/f^2 |
| Cl c-start | 0.32 | |
| Cl c-end | 0.20 | |

**Calculations**

| | | |
|---|---|---|
| b | 32.2 | ft |
| $M_{eff}$ | 0.99 | |
| $c_r$ | 32.2 | ft |
| $c_t$ | 0.0 | ft |
| m.a.c. | **21.5** | ft |
| | | |
| β | 0.17 | |
| $C_{L\alpha}$ | **0.044** | 1/deg |
| $C_{L\alpha}$ | 0.06 | |
| $\alpha_{trim}$ | **6.0** | deg |
| $C_{Ltrim}$ | 0.322 | |
| k | 0.2 | |
| $C_D$ | **0.028** | |
| L/D | **11.48** | |
| | | |
| | | |
| Total Drag | **7739.81** | lbf |

**Plotting:**

**Spanwise View**

| x | y |
|---|---|
| 0 | 0 |
| 32.2 | 0 |
| 30.3 | 16.11 |
| 30.3 | 16.11 |
| 0 | 0 |

**Lift Curves**

| α | Cl | α | CL |
|---|---|---|---|
| −1.33 | 0 | −1.33 | 0 |
| 8.03 | 1.03 | 8.03 | 0.41 |

**Airfoil Data**

| | | |
|---|---|---|
| Name | *NACA 64A204* | |
| $Cl_{max}$ | 1.03 | |
| $Cl_\alpha$ | 0.11 | 1/deg |
| a.c. | 0.25 | c |
| $\alpha_{0L}$ | −1.33 | deg |
| Cd0 | 0 | |
| $r_{le}$ | 0 | c |
| $Cl_{minD}$ | 0.1 − 0.3 | |
| (t/c)max | 0.40 | c |

**Sweep Angles**

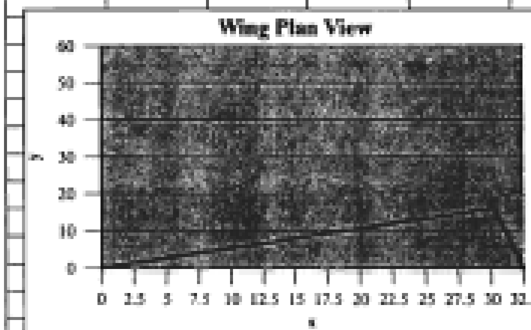
| | x/c | $\Lambda_{x/c}$ (deg) |
|---|---|---|
| LE | 0.00 | 62.0 |
| 1/4C | 0.25 | 54.1 |
| a.c | 0.25 | 54.0 |
| (t/c)max | 0.40 | 47.2 |
| TE | 1.00 | −6.8 |

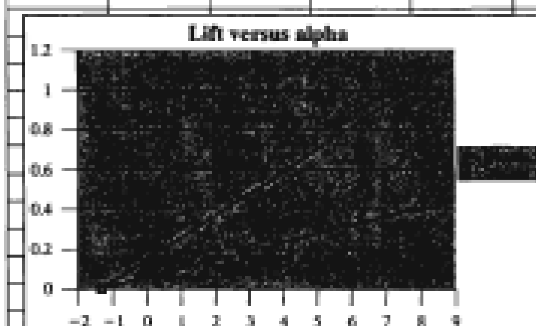
**Viscous Drag**

| | | |
|---|---|---|
| V_eff | 904.06 | f/s |
| q_eff | 117.05 | lbf/f^2 |
| Re_mac | 1.67E+07 | |
| sqrt(Re) | 4091.11 | |
| Cf | 2.54E−03 | |
| S_wet | 1039.56 | ft$^2$ |
| *F* | 1.46 | |
| *Q* | 1 | |
| $C_{D0}$ | **0.0074** | |

**Air Properties**

| | | |
|---|---|---|
| Cruise Alt. | 55,000 | ft |
| V | 1,925.30 | f/s |
| ρ | 0.01 | lbm/f^3 |
| q | 531.07 | lbf/f^2 |
| μ | 107.0E−7 | lbm/(f−s) |
| ν (cruise) | 0.00118 | f^2/s |



# Fuselage Design

## 1. Introduction

- The fuselage has a number of functions that vary depending on the type and mission of the aircraft.
- These include accommodating the crew, passengers, baggage or other payload, and possibly internal engines.

  Other considerations are fuel storage, the structure for wing attachments, and accommodations for retractable landing gear.
- **The main considerations in the design of the fuselage are:**

  i) Volume considerations,

  ii) Aerodynamic considerations, and

  iii) Drag estimation.
- **B-377PG “Pregnant Guppy”**
- Was designed to transport outsized cargo for the NASA Apollo Program.
- The upper fuselage was more than 20 feet in diameter to accommodate portions of the Saturn V rocket.

Aero Spacelines B-377PG “Pregnant Guppy”

# Fuselage Design (contd)

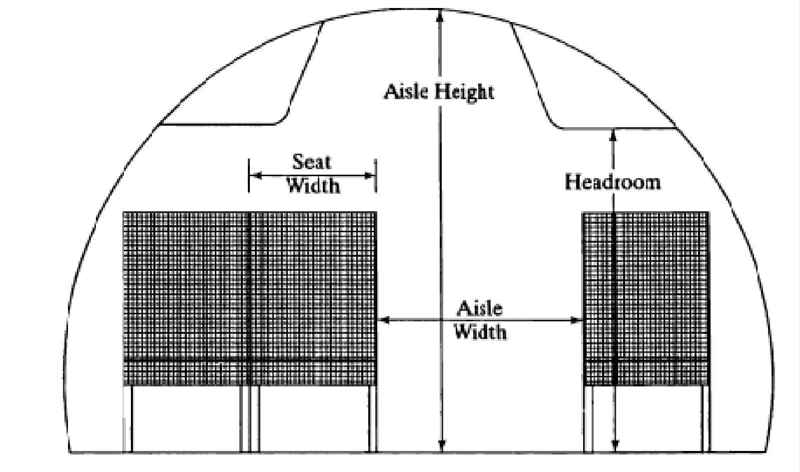
## 2. Volume Considerations

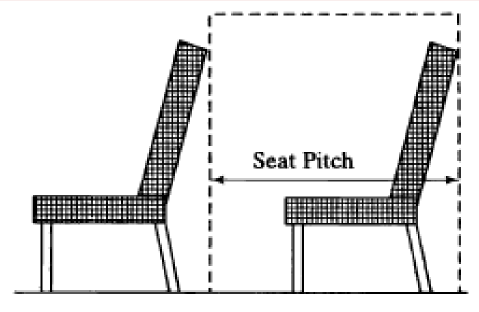
### 2.1 Passenger/Cargo Requirements

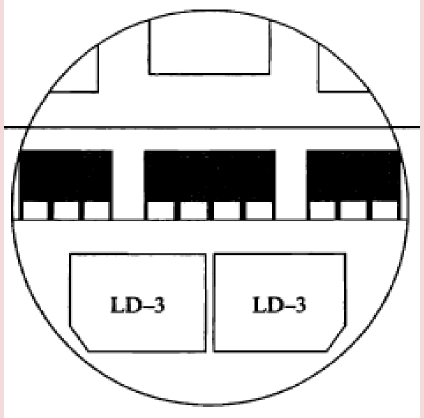
- The size and shape of subsonic commercial aircraft are generally determined by the number of passengers, seating arrangements and cargo requirements.
- Typical dimensions for the passenger compartments are shown in Table.
- These dimensions are generally based on the assumption that an average passenger weighs 180 lbs.

Passenger compartment requirements.

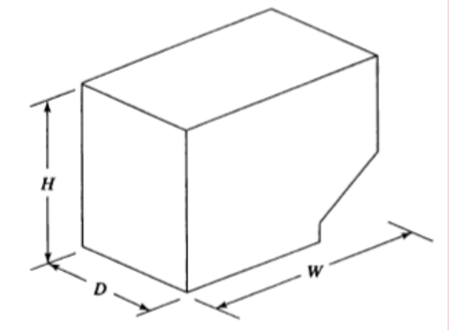
| | Long-Range | Short-Range |
|---|---|---|
| Seat Width (in) | 17–28 | 16–18 |
| Seat Pitch (in) | 34–40 | 30–32 |
| Headroom (in) | >65 | — |
| Aisle Width (in) | 20–28 | >15 |
| Aisle Height (in) | >76 | >60 |
| Passengers/Cabin | 10–36 | ≤50 |
| Lavatories/Passenger | 1/(10–20) | 1/(40–50) |
| Galley Volume/Passenger ($f^3$) | 1–8 | 0–1 |
| Baggage/Passenger (lbs) | 40–60 | 40 |



Boeing 747



LD–2, LD–3

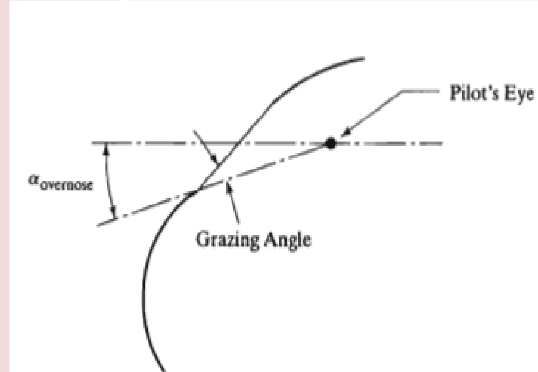
Cargo container

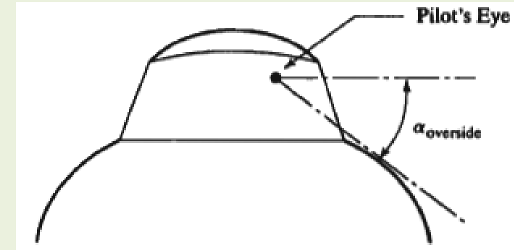
# Fuselage Design (contd)

## 2.2 Crew Requirements

- The size of the crew compartment will vary depending on the aircraft.
- For long rage civil/military aircraft, the crew compartment should be designed to accommodate from 2 to 4 crew members.
- Recommended length for crew compartment:
  - 150 inches – 4 crew
  - 130 inches – 3 crew
  - 100 inches – 2 crew
- An important factor that impacts the shape of the forward section of the fuselage is the requirement that the pilot must have an unobstructed forward view.
- This is especially important for the landing phase for all aircraft, and during the combat phase of fighter aircraft. (Concorde and Tu-144)






- The over-nose angle ($\alpha_{overnose}$) is defined as the angle between a horizontal line through the pilot's eye, down to the point of the highest visual obstruction.

$$\alpha_{overnose} = \gamma_{approach} + 0.07 V_{50}$$

$$\gamma_{approach} = \sin^{-1}\left(\frac{-D}{W}\right) \qquad V_{50} = 1.3 V_s$$

Values of over-nose and over-side angles for different aircraft.

| | $\alpha_{overnose}$ | $\alpha_{overside}$ |
|---|---|---|
| Military Transports/Bombers | 17° | 35° |
| Military Fighter | 11°–15° | 40° |
| General Aviation | 5°–10° | 35° |
| Commercial Transport | 11°–20° | 35° |

# Fuselage Design (contd)

## 2.3. Fuel Storage Requirements

- In long-range aircraft, a large percentage of the weight at take-off is due to the weight of the fuel.
- The volume required to hold this fuel can be allocated to the fuselage or wing or to both.
- The decision on where to store the fuel depends on a number of factors:

i) Location of the center of mass with respect to the center of lift, thus affecting the static stability

- ➢ In order to maintain static stability in the pitch direction, the center of mass must always be forward of the center of lift.
- ➢ As a result, if any fuel is stored in the fuselage, it should be located at or slightly forward of the wing attachment point.

ii) The vulnerability of crew and passengers in the event of an uncontrolled landing, and

iii) The vulnerability of the fuel in combat aircraft caused by enemy fire.

- The volume needed to accommodate the fuel is based on the maximum fuel at take-off and the density of the fuel. The specific volumes for different aviation fuels are given in Table.

Specific volumes for different aviation fuels ($f^3$/lb).

| | 0°F | 100°F | Mil-spec |
|---|---|---|---|
| AV-gas | 0.0219 | 0.0235 | 0.0223 |
| JP-4 | 0.0199 | 0.0208 | 0.0206 |
| JP-5 | 0.0186 | 0.0196 | 0.0197 |
| JP-8 | — | — | 0.0199 |

❑ Three types of fuel tanks:

i) Discrete Type:

- ➢ Used for small general aviation aircraft.
- ➢ They are fuel containers that mount in the aircraft.
- ➢ In the wing, these are mounted at the inboard span portion, near the leading edge.
- ➢ In the fuselage, they are placed behind the engine and above the pilot’s feet.

# Fuselage Design (contd)

## Fuel Storage Requirements (contd)

- **Bladder Type Fuel Tanks:**

➢ These are thick rubber bags that are placed into cavities in the wing or the fuselage.

➢ An advantage is that they can be made of self-healing rubber. This improves aircraft survivability in the case of uncontrolled landing or enemy fire.

➢ The disadvantage is the thickness of the rubber bladder walls reduces the available volume of the cavity.

➢ As a general rule, 77% of the cavity volume in the wing, and 83% of the cavity volume in the fuselage, is available with bladder tanks.

❑ **Integral Tanks:**

➢ These are cavities inside the airframe structure that are sealed to form fuel tanks.

➢ Examples are the wing box areas formed between wing spars and the area between bulkheads in the fuselage.

➢ Integral tanks are more prone to leaking compared to other two types, they should not be located near air inlet ducts or engines.

- As a general rule, 85% of the volume measured to the external skin of the wing, and 92% measured to the external skin of the fuselage, is available with integral tanks.

# Fuselage Design (contd)

## 2.4. Internal Engines and Air Inlets

- Engines can be mounted internal to the fuselage
- This is a common practice with combat aircraft and general aviation aircraft.
- Some long-range commercial passenger aircraft also have internal engines (B-727, L-1011)
- To accommodate internal engines, the volume to enclose them must be accounted for in the fuselage design.
- In the conceptual design stage, the best approach is to rely on suitable comparison aircraft.
- For internally mounted jet engines, the air delivery system is an integral element. The type and geometry of the inlet will determine the pressure loss and uniformity of the air supplied to the engine.

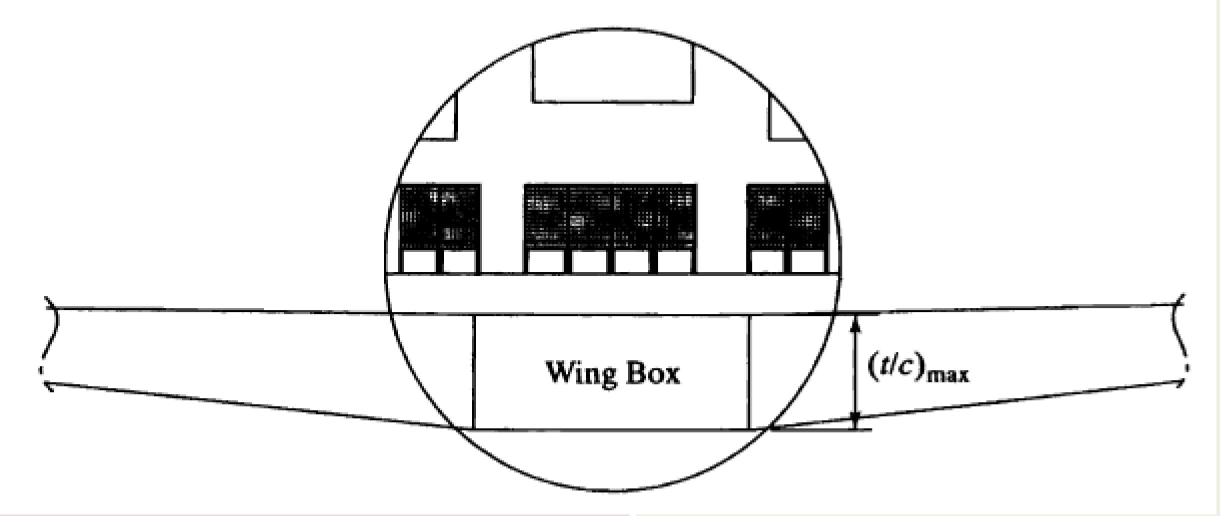
❑ The types of Air Inlets depend on the operating Mach number.

- The objective of the air inlet system for turbojet and turbofan engines is to reduce the Mach number of the air at the compressor face to between 0.4 and 0.5.
- In subsonic aircraft this is accomplished using a subsonic diffuser.
- In supersonic aircraft, this is done through area changes at the inlet that result in the formation of one or more compressive shocks.
- At the sizing or conceptual design stage, it can be assumed, based on empirical data, that the diameter of the air inlet can be the same as that of the engine compressor face.
- Further, the length of the air inlet can be 60% of the engine length.

# Fuselage Design (contd)

## 2.5 WING ATTACHMENTS

- The manner in which the main wing attaches to the fuselage is an important element in the fuselage design.
- For structural reasons, the wing is constructed as as integral unit.
- The portion of the wing that passes through the fuselage is referred to as the wing carry-through.
- The root-span portion of the wing has the largest thickness in order to the large bending moment in the wing.
- As a result, the wing carry-through occupies a large volume where it passes through the fuselage.
- A sketch of a typical fuselage wing carry-through is shown in the Fig.
- Since the details of the main wing are known at this stage of design, the volume requirements for the carry-through structure can be directly applied to the design of the fuselage.



# Fuselage Design (contd)

## 2.6 LANDING GEAR PLACEMENT

- In most aircraft, the fuselage needs to accommodate all or some parts of the landing gear when it is retracted.
- Therefore, the placement and volume requirements of the landing gear need to be considered in the design of the fuselage.
- The size and location of the landing gear will vary depending on the type of the aircraft.
- A good first estimate can come by examining suitable comparison aircraft that have a comparable take-off weight.
- Fig. shows an illustration of some of different landing gear arrangements.
- The largest portion of the landing gear for which space has to be allotted in the fuselage is the landing gear wheels.
- Typically, the tires on the main landing gear carry approximately 90% of the aircraft weight. The other 10% is carried by the nose gear.

- The size of the landing gear wheels can be estimated as follows:

Main Wheel Diameter or Width = $AW_{main}^{B}$

Weight on each of the main landing gear ,

$$W_{main} = \frac{0.9W_{TO}}{N_{wheels}}$$

- The size of the nose wheel can be assumed to be 60% of the main wheel.

Main landing gear wheel sizing coefficients

| | Diameter | | Width | |
|---|---|---|---|---|
| | *A* | *B* | *A* | *B* |
| General Aviation | 1.510 | 0.349 | 0.715 | 0.312 |
| Business Twin jet | 2.690 | 0.251 | 1.170 | 0.216 |
| Transport/Bomber | 1.630 | 0.315 | 0.104 | 0.480 |
| Jet Fighter/Trainer | 1.590 | 0.302 | 0.098 | 0.467 |

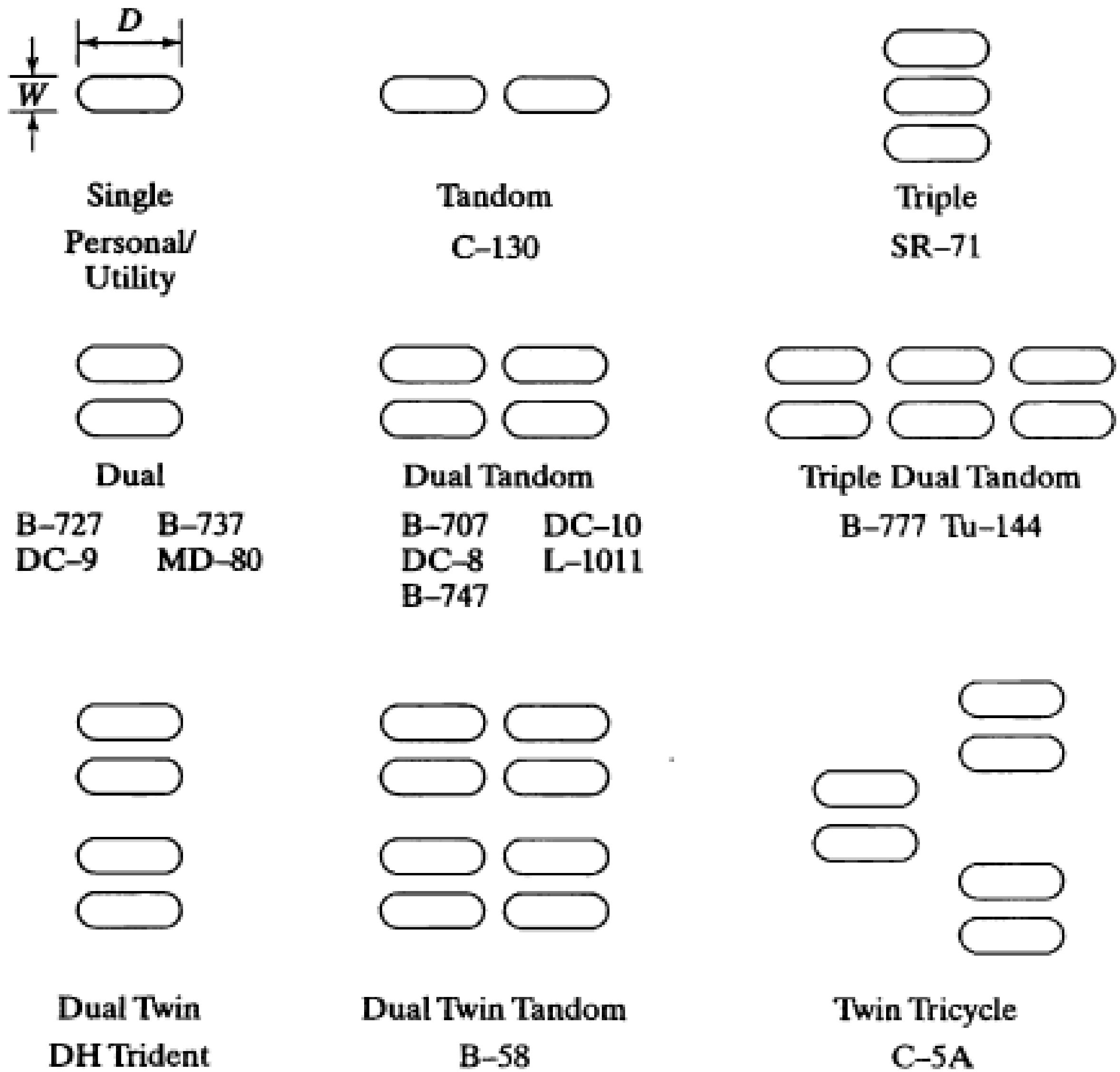



Illustration of different main landing gear footprints.

# Fuselage Design (contd)

## 2.6 ARMAMENT PLACEMENT

- With combat aircraft, the number and size of bombs and armament are generally decided in the initial design proposal when the mission requirements are set.
- Therefore, while designing the fuselage, the arrangements for storage, positioning and release of weapons need to be examined.
- If the weapons are carried externally, they add considerable amount of aerodynamic drag.
- So, the weapons are partially or fully recessed in the underside of the wing or fuselage.
- In some cases, the weapons may be located inside the fuselage, in a weapons bay. In such cases, provisions need to be made in the design of the fuselage to have the necessary volume and exterior access.

# Fuselage Design (contd)

**FUSELAGE FINENESS RATIO** (contd)

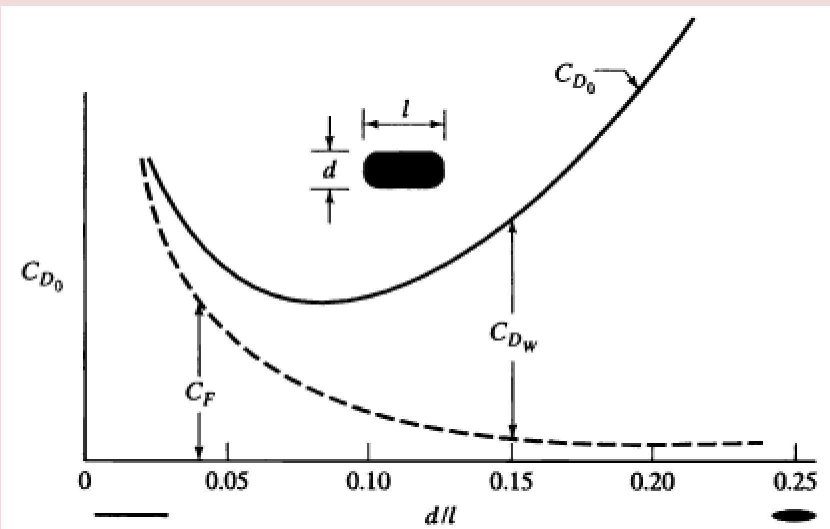
**❑ SUPERSONIC AIRCRAFT**

- For a supersonic aircraft.

overall drag = viscous drag + supersonic wave drag

$$C_{D_0} = C_F + C_{D_W}$$

- The percentage that each contributes to the total drag , as a function of fineness ratio, is shown in Fig.




- In general for supersonic flight, the overall drag coefficient on a slender body is 2-3 times higher than for subsonic flight.

- The rise in wave drag as ( $d/l$ ) increases is particularly severe.

- For a blunt body approaching a sphere ( $d/l = 1$), the overall drag is predominantly wave (bow shock) drag.

- For supersonic flight the optimum fineness ratio value is, $d/l = 0.07$ or $(l/d = 14)$.

- In contrast to subsonic aircraft, minimizing the aerodynamic drag is the design driver for long-range supersonic aircraft, and their fuselage designs use the optimum fineness ratio.

- In practice, fineness ratios in the range

$$0.1 \le (d/l) \le 0.125 \qquad [\,8 \le (l/d) \le 10\,]$$

are used since supersonic aircraft may sometimes need to operate at subsonic speeds also.

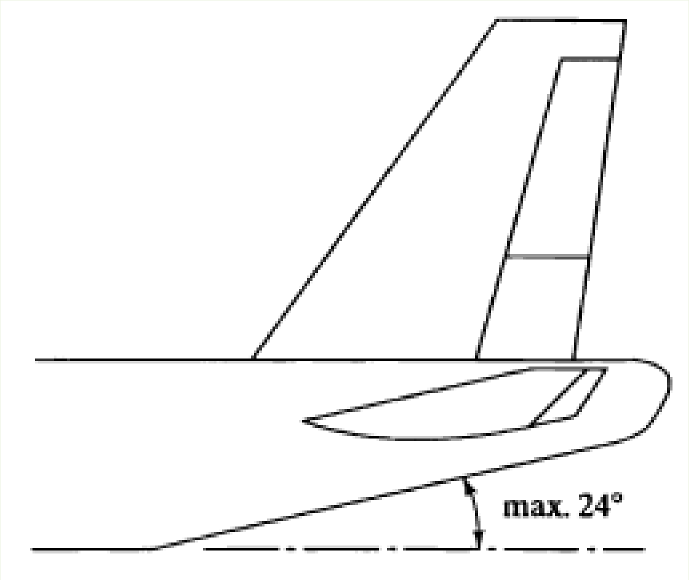
- Fineness ratios for typical supersonic passenger aircraft are shown in the Table.

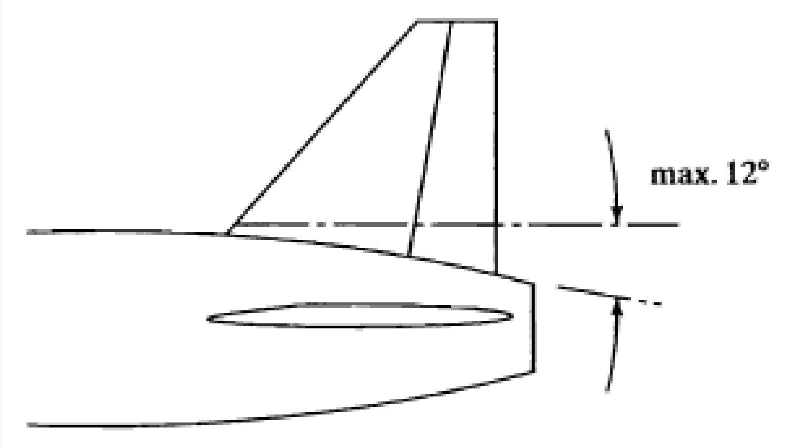
| Supersonic | $d/l$ |
|---|---|
| Concorde | 0.05 |
| Tu-144 | 0.05 |

# Fuselage Design (contd)

**FUSELAGE SHAPES**

- Within the design constraints imposed by the volume requirements, the fuselage shape should be aerodynamic, with smooth and gradual dimension changes and blended curves.
- Large divergence angles should be avoided as they can cause the flow over the fuselage to separate.
- This would lead to a higher base drag of the fuselage and a reduction in cruise efficiency.
- Particular care should be taken in the design of the aft-part of the fuselage.
- The aft body usually has an upward slope to allow ground clearance during pitch-up in the "rotation" portion of take-off.
- As a general rule, the total divergence angle should be less than 24 degrees.
- If this is divided around the fuselage, the local angle should be less than 12 degrees, as shown in Fig.




Schematic drawings showing the divergence angle limits for different types of aft-fuselage designs

# Fuselage Design (contd)

### DRAG ESTIMATION

- The drag force due to viscous drag is given by

$$F_f = qSC_f$$

where $q$ is dynamic pressure (based on cruise condition), $S$ is the surface area and $C_f$ is the friction coefficient.

- On subsonic aircraft, minimizing the surface or wetted area is one of the most powerful considerations in reducing the drag.

- For elliptic cross sections, the total surface area is given by

$$S = \int_0^L P(x)\,dx$$

where,

$$P(x) \cong \pi \left[ \frac{h(x) + w(x)}{2} \right]$$

$$h(x) = local\ height,\ w(x) = local\ width$$

- The area integral can be approximated by dividing the fuselage shape into $N$ piecewise-linear $X$-portions with constant dimensions.

$$S = \sum_{i=1}^{N} P_i\ x_i$$

- For elliptic cross-section with constant dimensions ( $h$ and $w$ ) and length $L$, the total surface area is given as

$$S \cong \pi \left[ \frac{(hL) + (wL)}{2} \right] \quad \text{or} \quad S \cong \pi \left[ \frac{A_{side} + A_{top}}{2} \right]$$

where, $A_{side}$ and $A_{top}$ are the areas of the respective side and top projected views of the fuselage.

- The friction coefficient, $C_f$, for different flow types is given as:

$$C_f = \frac{1.328}{\sqrt{\mathrm{Re}_x}} \quad \text{:Laminar} \qquad \mathrm{Re}_x = U_o x / \nu$$

where, $U_0$ = velocity at the outer edge of the boundary layer, and $\nu$ =kinematic viscosity, at a given flight condition.

$$C_f = \frac{0.455}{(\log_{10} \mathrm{Re}_x)^{2.58} (1 + 0.144 M^2)^{0.65}} \quad \text{:Turbulent}$$

$$\sqrt{\mathrm{Re}_x} \geq 1000$$

- The term $(1 + 0.144 M^2)^{0.65}$ is a Mach number correction that approaches 1.0 for low Mach numbers.

# Fuselage Design (contd)

**DRAG ESTIMATION (contd)**

## ❑ Effective Reynolds Number:

- Surface roughness affects the Reynolds number at which the flow becomes turbulent and increases the friction coefficient ($C_f$).

- The effect of roughness on boundary layers can be expressed in terms of an effective Reynolds number, which is a function of roughness height ( $k$ ) with respect to the boundary layer thickness.

- Using empirical data from flat plate experiments, it has been shown that,

$$\mathrm{Re}_{\text{effective}} = 38.21 \left(\frac{x}{k}\right)^{1.053} \quad (M < 1)$$

$$\mathrm{Re}_{\text{effective}} = 44.62 \left(\frac{x}{k}\right)^{1.053} M^{1.16} \quad : (M \geq 1)$$

- Values of roughness height, $k$ , for different aircraft surface conditions are given in the Table:

| Surface Type | $k$ ($10^{-5}$ ft) |
|---|---|
| Smooth Molded Composite | 0.17 |
| Polished Sheet Metal | 0.50 |
| Production Sheet Metal | 1.33 |
| Smooth Paint | 2.08 |

- The effective Reynolds number should be used only if $Re_{effective} > Re_x$

- The friction force on an elemental streamwise segment of the fuselage is a function of the streamwise location, $x$ , and is given by

$$F(x) = q\,P(x)C_f(x)$$

- The total viscous drag force can be estimated using the approximation

$$F = q \sum_{1}^{N} P_i x_i C_f^i$$

($N$ = No. of fuselage segments)

## ❑ Friction Coefficient at Nose ($C_{f_{nose}}$):

- In the area of the nose of the aircraft up to the near-constant diameter fuselage section, the favourable pressure gradient results in a slightly higher friction coefficient compared to the flat plate equivalent, and an estimate of is given by:

$$C_{f_{nose}} \simeq 1.16 C_{f_{flatplate}}$$

- The pressure gradient on the nose virtually assures that the boundary layer is laminar. Therefore,

$$C_{f_{flatplate}} = C_{f_{laminar}}$$

# Fuselage Design (contd)

**DRAG ESTIMATION (contd)**

❑ **FORM FACTOR**

- As with the estimation of the base drag coefficient on the main wing, the estimate of the friction drag coefficient of the fuselage makes use of a form factor.
- The form factor for the fuselage is given by:

$$\mathcal{F} = 1 + \frac{60}{f^3} + \frac{f}{400}$$

where, f = l/d ( inverse fineness ratio )

❑ **INTERFERENCE FACTOR**

- In most cases, the fuselage has a negligible interference factor.
- Therefore, $Q = 1$ is appropriate for the fuselage.

❑ **Finally, the total viscous drag on the fuselage is given by:**

$$F_f = qSC_f\mathcal{F}Q$$

where *S* is the fuselage surface (wetted) area.

-

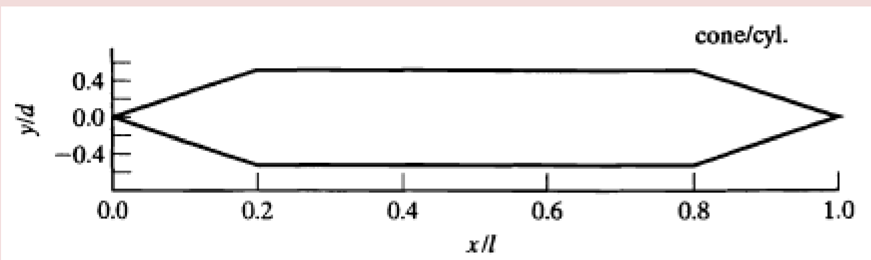
# Fuselage Design (contd)

**QUANTITATIVE FUSELAGE SHAPES**

- There are a number of quantitative fuselage shapes for which drag data are available.
- These are useful as a starting point in laying out the fuselage since their analytical form allows direct integration for determining the surface area and volume. Some of these shapes are discussed.

**1. Cone – Cylinder:**

- In this shape, the nose of the fuselage is a right circular cone whose base matches onto a constant diameter cylindrical fuselage section.




- With this shape:

i) At subsonic Mach numbers, the total drag force on the fuselage is primarily due to viscous drag. The viscous drag coefficients can be determined using the relations presented in previous section.

ii) At supersonic Mach numbers, the wave drag coefficient, $C_{D_W}$, is equal to the pressure coefficient, $C_P$, due to the conical shock wave that will form at the leading nose.

- The $C_P$ values are easily determined from conical shock charts available in compressible flow handbooks.

**2. Power Series – Cylinder:**

- This shape is similar to the cone-cylinder except that the nose section is derived from the following relation,

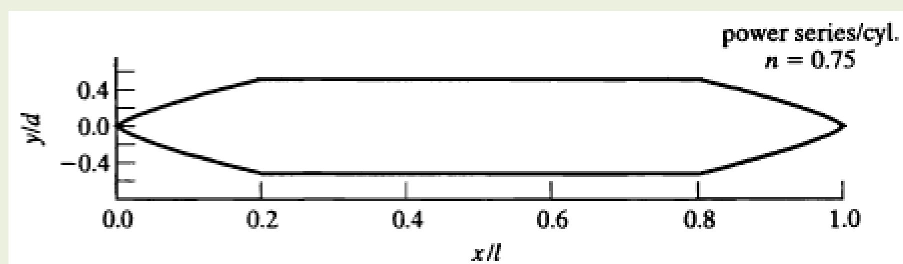
$$\frac{r(x)}{r(l_c)} = \left(\frac{x}{l_c}\right)^n$$

where, $l_c$ = length of the nose section

$r(l_c)$ = radius at the base ( $x = l_c$ )

$r(x)$ = local radius ( at x )

- When n=1, the above equation describes a right-circular cone.
- A good amount of drag data are available for a variety of values of *'n'*.
- For this family of shapes, n = 3/4 gives the minimum wave drag.

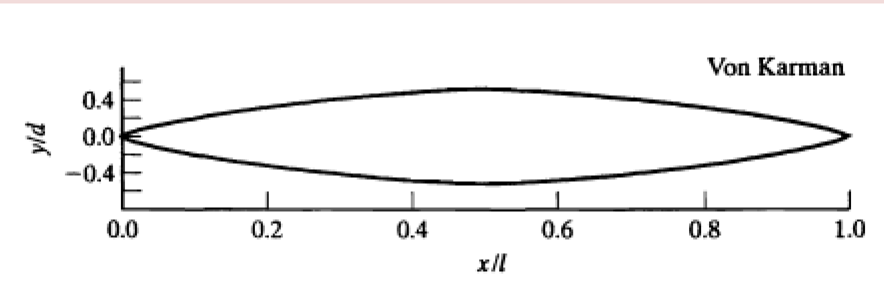


# Fuselage Design (contd)

**QUANTITATIVE FUSELAGE SHAPES** (contd)

**3. Von Karman Ogive:**

- A schematic drawing of Von Karman fuselage shape is shown below:




•This is a symmetric body of revolution that is described by the following relation:

$$\left[\frac{r(x)}{r(0)}\right]^2 = \frac{1}{\pi}\left[\frac{2x}{l}\sqrt{1-\left(\frac{2x}{l}\right)^2} + \cos^{-1}\left(\frac{-2x}{l}\right)\right]$$

for $(-l/2 \leq x \leq l/2)$

- For this shape, $r(0)$ is the maximum radius, which occurs at $x=0$ and 'l' is the overall length.
- In the fuselage design, the above equation will form the leading half, namely, from the leading point to the point of largest diameter.

- Thus, if fuselage length = $L$
  maximum diameter = $D$
  then in the above equation,

$$l = L/4, \quad r(0) = D/2, \quad x = x' - (L/4)$$

where, $x'$ = streamwise position along the fuselage starting from the most leading point $(0 \leq x' \leq L)$

- The downstream half of the fuselage can be made as a mirror image of the leading half.
- The overall volume of the body is

$$\text{Volume} = \frac{l}{2}A_{max} = \frac{l}{2}\pi r(0)^2$$

- This shape is primarily suited to supersonic aircraft where its drag coefficient is the lowest of the group of shapes listed:

$$C_{D_w} = \frac{4A_{max}}{\pi l^2} = 4\left[\frac{r(0)}{l}\right]^2$$

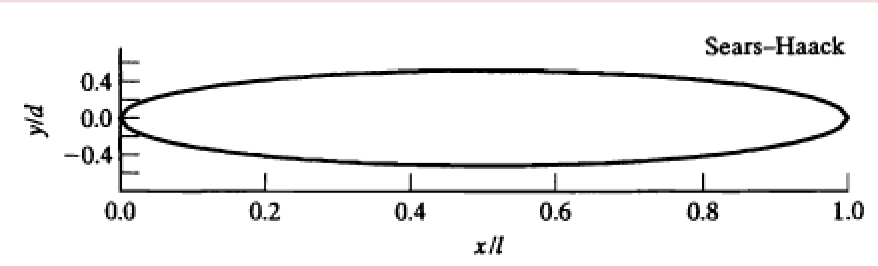
•An example of Von Karman fuselage is presented for the case study supersonic business jet (SSBJ).

# Fuselage Design (contd)

**QUANTITATIVE FUSELAGE SHAPES (contd)**

**4. Sears - Haack:**

- This is a symmetric body of revolution and also has a relatively low wave drag compared to the other shapes.




- The profile is described by the following relation:

$$\left[\frac{r(x)}{r(0)}\right]^2 = \left[1 - \left(\frac{2x}{l}\right)^2\right]^{3/2} \qquad (-l/2 \le x \le l/2)$$

- The above equation describes the complete fuselage , from leading to trailing points.

(NOTE: In the case of Von Karman fuselage the form equation describes only the leading half of the fuselage)

- The over all volume of the fuselage is

$$\text{Volume} = \frac{3}{16}\pi l A_{\max} = \frac{3}{16} l [\pi r(0)]^2$$

- The surface (wetted area) is

$$S = 1.8667[(\text{Volume})(l)]^{1/2} = 0.8083\pi l r(0)$$

- The wave drag coefficient is given as

$$C_{D_W} = \frac{9}{2}\frac{\pi}{l^2} A_{\max} = \frac{9}{2}\left[\frac{\pi r(0)}{l}\right]^2$$

- **Area Ruling the Fuselage:**
- The wavedrag coefficients ( $C_{D_W}$ ) in the case of Von Karman and Sears-Haack fuselages show that the wave drag depends on the cross-sectional area.
- This applies not only to the fuselage, but also to the fuselage and wing together.
- As a result, the cross-section of the fuselage is often indented in the vicinity of the wing attachment location in order to keep a nearly constant and smooth wing-fuselage cross-section area distribution along the length of the aircraft.
- This process is called "area ruling". Area ruling reduces the drag by 50% over a non-area ruled design.

# Spreadsheet for Fuselage Design

- A sample of the spreadsheet for fuselage design (aerodynamic) is shown in Fig.
- This contains the parameters for a conceptual Super Sonic Business Jet (SSBJ) aircraft.
- In the spreadsheet, there are two areas where the input parameters are placed.
  These correspond to the flight regime data, and the dimension data.
- The flight regime input data comprises of:

  i) Cruise Mach number (M),
  ii) Cruise Altitude (H),
  iii) Velocity (V),
  iv) Air density ($\rho$),
  v) Dynamic pressure (q),
  vi) Viscosity ($\mu$), and
  vii) Kinematic viscosity ($\nu = \mu/\rho$).

- The dimension data come from the design requirements for volume to enclose crew, payload, etc., and it comprises of:

i) Maximum diameter ( $D_{max}$ ) for circular shapes
[ Equivalent diameter for non-circular shapes ]

ii) Fineness ratio (d/l), iii) Fuselage length (l)

iii) Wing surface area (S), iv) Form factor (F), and

iv) Interference factor (Q).

- The drag calculations consider viscous drag and wave drag (in the case of super sonic aircraft)

❑ **Wave Drag Calculations:**

- First, the fuselage length ( *l* ) is divided into 10 equal elements.
- The first column shows ( *x/l* ) in 10% elements.
- The equivalent locations along the fuselage ( x ) are given in the second column.
- Next, the parameters $(x - L/4)$ , *D, P* and $S_W$ are calculated as shown in Columns 3 to 6.
- Here, *P* is the local parameter given by

$$P(x) \cong \pi \left[ \frac{h(x) + w(x)}{2} \right]$$

$$h(x) = local\ height, \ \ w(x) = local\ \ width$$

and, $S_W$, is the wetted surface area given by

$$S \cong \pi \left[ \frac{A_{side} + A_{top}}{2} \right]$$

where, $A_{side}$ and $A_{top}$ are the areas of the respective side and top projected views of the fuselage.

# Spreadsheet for Fuselage Design (contd)

- The geometrical calculations are made for the leading-half of the fuselage as the down-stream half is a mirror image of the leading half.
- The local Reynolds number, the friction coefficient and the viscous drag ( $F_f$ ) are calculated, as shown in columns 7 to 9:

$$C_f = \frac{0.455}{(\log_{10} \mathrm{Re}_x)^{2.58}\,(1+0.144M^2)^{0.65}} \quad \text{:Turbulent}$$

$$\sqrt{\mathrm{Re}_x} \geq 1000$$

$$\mathcal{F} = 1 + \frac{60}{f^3} + \frac{f}{400} \qquad Q = 1 \qquad F_f = qSC_f\mathcal{F}Q$$

## ❑ Wave Drag Calculation

- For the Von Karman fuselage shape, the wave drag coefficient, and the wave drag $F_W$ are calculated using the equations

$$C_{D_W} = \frac{4A_{\max}}{\pi l^2} \qquad F_W = qA_{max}C_{D_W}$$

## ❑ Total Drag

- The total drag is given by $(F_f + F_W)$

## ❑ Equivalent Drag Coefficient

- Finally a drag coefficient that is equivalent to the drag coefficient of the main wing is calculated using the equation

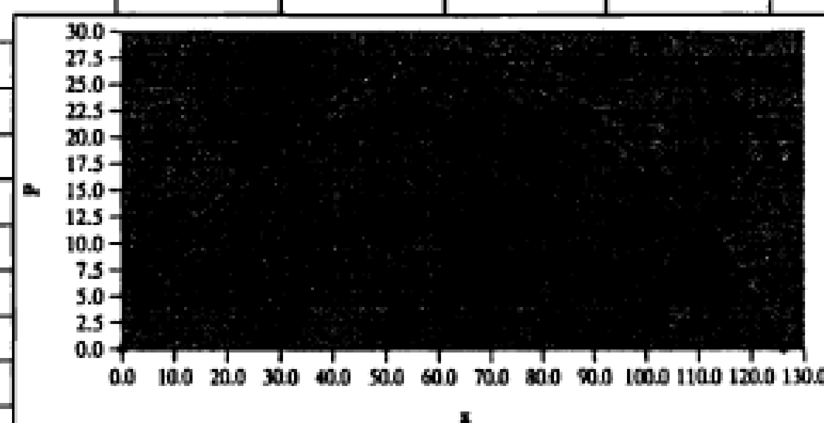
$$C_{D_0} = \frac{F_f + F_W}{qS}$$

where, S is the area of the main wing.

- This equivalent drag coefficient is then compared with the drag coefficient calculated for the main wing, in order to see the relative contributions of the wing and the fuselage to the overall drag on the aircraft.

| | | | | |
|---|---|---|---|---|
| | | | **Fuselage Design** | |
| **Flight Regime Data:** | | | | |
| Cruise Mach | 2.1 | | | |
| Cruise Alt. (ft) | 55,000 | | | |
| V (f/s) | 1,925.70 | | | |
| ρ (lbm/f^3) | 0.01 | | | |
| q (lbf/f^2) | 531.07 | | | |
| μ (lbm/(f–s)) | 0 | | | |
| ν (cruise) (f^2/s) | 0 | | | |
| **Dimension Data:** | | | **Form Factors:** | |
| D–max (ft) | 9 | | *F* | 1.06 |
| L/D | 14 | | *Q* | 1 |
| L (ft) | 126 | | *F*Q* | 1.06 |
| S (f^2) | 519 | | | |

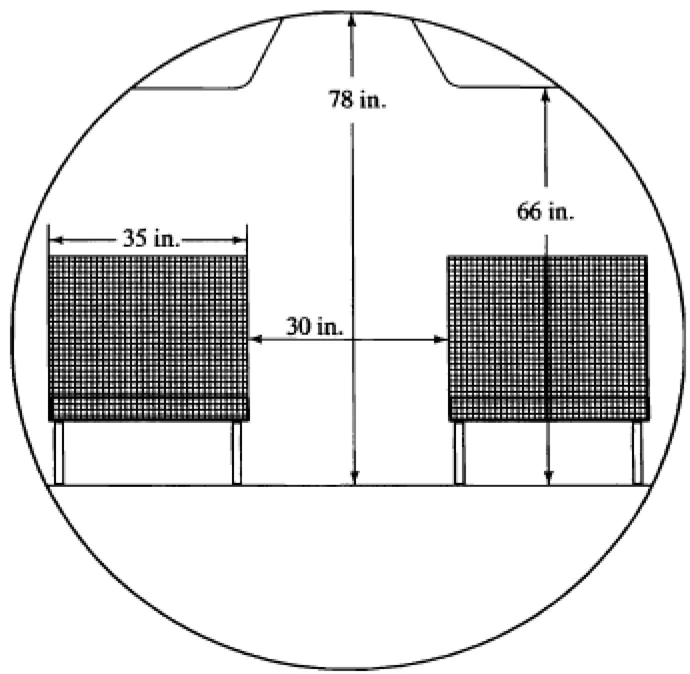
| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | | | |
| **Viscous Drag Calculations:** | | | *Von–Karman Ogive Fuselage Shape* | | | | | |
| x/L | x (ft) | x−L/4 (ft) | D (ft) | P (ft) | Sw(ft^2) | $Re_x$ | $C_F$ | Drag (lbf) |
| 0.00 | 0.00 | −31.50 | 0 | 0.0 | | | | |
| 0.10 | 12.60 | −18.90 | 3.4 | 10.7 | 134.4 | 2.1E+07 | 1.94E−03 | 147 |
| 0.20 | 25.20 | −6.30 | 5.5 | 17.3 | 217.7 | 4.2E+07 | 1.75E−03 | 214 |
| 0.30 | 37.80 | 6.30 | 7.12 | 22.4 | 282.0 | 6.3E+07 | 1.65E−03 | 261 |
| 0.40 | 50.40 | 18.90 | 8.33 | 26.2 | 329.9 | 8.4E+07 | 1.59E−03 | 294 |
| 0.50 | 63.00 | 31.50 | 9 | 28.3 | 356.3 | 1.0E+08 | 1.54E−03 | 307 |
| 0.60 | 75.60 | – | 8.33 | 26.2 | 329.9 | 1.3E+08 | 1.50E−03 | 277 |
| 0.70 | 88.20 | – | 7.12 | 22.4 | 282.0 | 1.5E+08 | 1.47E−03 | 232 |
| 0.80 | 100.80 | – | 5.5 | 17.3 | 217.7 | 1.7E+08 | 1.44E−03 | 176 |
| 0.90 | 113.40 | – | 3.4 | 10.7 | 134.4 | 1.9E+08 | 1.42E−03 | 107 |
| 1 | 126.00 | – | 0 | 0.0 | 0.0 | 2.1E+08 | 1.40E−03 | 0 |
| **Totals:** | | | | | **2284.4** | | | **2015** |
| | | | | | | | | |
| **Wave Drag Calculations:** | | | | | | | | |
| A_max | 63.62 | | | | | | | |
| CDW | 0.02 | | | | | | | |
| Drag (lbf) | **689.5** | | | | | | | |
| | | | | | | | | |
| | | | | | | | | |
| **Total Drag:** | **2705** | | | | | | | |
| **(lbf)** | | | | | | | | |
| | | | | | | | | |
| Equiv. CD | 0.0098 | | | | | | | |
| | | | | | | | | |



# Case Study: Wing Design for SSBJ

- The passenger compartment was designed to seat from 12 to 15 passengers.
- The conceptual SSBJ passenger compartment data is given in the Table.
- The diameter (d) of the fuselage was based on having two seats that are separated by a center aisle.
  d = (Widths of seat s) + fuselage wall thickness (4 inches)
  d = 9 ft.
- The length of the fuselage was chosen to be, l= 126 ft, to obtain an optimum fineness ratio of d/l=0.07, or the inverse ratio l/d=14.
- Von Karman Ogive shape was selected for the fuselage because it has the lowest wave drag among the well-known and documented shapes.
- The cross-section of the fuselage is circular. So, the perimeter is given by $P = \pi D$
- All the local Reynolds numbers $\sqrt{Re_x} > 1000$
  So, the flow is assumed to be turbulent everywhere, and the friction coefficient $C_f$ was calculated accordingly.
- Viscous drag force = 2015 lbs; Form factor F=1.056.
  The form factor adds 108 lbs (5%) to viscous drag force.
- Wave drag force = 689 lbs; Total drag force = 2705 lbs.
- Equivalent drag coefficient = 0.0061. This is 4 times smaller than the main-wing drag coefficient.

| | |
|---|---|
| Passengers/Cabin | 12–15 |
| Seats Across | 2 |
| Number of Aisles | 1 |
| Seat Width (in.) | 35 |
| Seat Pitch (in.) | 40 |
| Headroom (in.) | 66 |
| Aisle Width (in.) | 30 |
| Aisle Height (in.) | 78 |
| Lavatories | 1 |
| Galley Volume ($f^3$) | 160 |
| Baggage/Passenger (lbs) | 40 |



# Horizontal and Vertical Tail Design

**1. TAIL ARRANGEMENTS**

- A large variety of horizontal and vertical tail designs have been used on past aircraft.
- Suitable tail configurations are selected based on mission requirements. Some of these designs are discussed below.

**1. Conventional Tail**

- A majority of commercial and general purpose aircraft use the conventional tail design. An example is the Boeing 777 aircraft.
- This design places the horizontal stabilizer at or near the fuselage vertical centerline.


**Advantages:**

- Provides sufficient stability and control.
- Has the lowest tail weight

**Disadvantages:**

- Static stability requires that the CG be forward of the center of lift.
- A relatively heavy weight of this type of tail can force a redistribution of other weight or a change in the position of the main wing, which sometimes can be difficult.

# Horizontal and Vertical Tail Design (contd)

**2. T - Tail**

- The T-tail is also a relatively popular design (Boeing 727, Douglas YC-15, C-141 transport)
- This design places a horizontal tail high on the end of the vertical tail.

**Main Advantages:**

- The vertical tail can be smaller than on a conventional tail because the placement of the horizontal stabilizer acts as a winglet and increases the effective aspect ratio.
- The horizontal stabilizer can also be made smaller because it is placed high, out of the wake of the main wing.

**Main Disadvantage:**

- T-tail is heavier than the conventional tail design, since the vertical tail structure needs to be made stronger in order to carry the load of the horizontal tail.

Boeing 727

# Horizontal and Vertical Tail Design (contd)

## 3. Cruciform Tail

- The Cruciform Tail is a compromise between the conventional and T-tail designs.
- In this design, the horizontal tail is at the approximate mid-span of the vertical tail.
- An example is the JetStar aircraft.

### Advantages:

- It raises the horizontal stabilizer out of the wake of the main wing, with less of a weight penalty compared to the T-tail.

### Disadvantages:

- Because the horizontal stabilizer is not at the end of the vertical stabilizer, there is no reduction in the vertical tail aspect-ratio requirement that comes with the T-tail.

# Horizontal and Vertical Tail Design (contd)

## 4. H - Tail

- The H-tail is a popular design for some combat aircraft. An example is YA-10 aircraft.

**Advantages:**

- The H-tail design positions the vertical stabilizers in the air, which is not disturbed by the fuselage.
- It reduces the required size of the horizontal stabilizer because of the winglet effect of the vertical tail surfaces.
- Another particular advantage is that it lowers the required height of the vertical tail.

  This is particularly important on aircraft that must have a low clearance height or on combat aircraft where it reduces the projected area of this vulnerable component.

**Disadvantage:**

- The required added strength of the horizontal stabilizer makes the H-tail heavier than the conventional tail.

# Horizontal and Vertical Tail Design (contd)

**5. V - Tail**

- **A V-tail is designed to reduce the (wetted) surface area by combining the vertical and horizontal tail surfaces.**
- **Control in this case is through "ruddervators".**
- **In a ruddervator control, a downward deflection of the right elevator and an upward deflection of the left elevator will push the tail to the left, and thereby the nose to the right.**
- **Unfortunately, the same maneuver produces a roll moment toward the left, which opposes the turn.**
- **This effect is called an "adverse yaw".**
- **The solution to this is an inverted V-tail.**

# Horizontal and Vertical Tail Design (contd)

## 6. Inverted V - Tail

**Advantages:**

- An inverted V-tail avoids the adverse yaw-roll coupling of the V-tail.
- In this case, the elevator deflections produce a complimentary roll moment, which enhances a coordinated turn maneuver.
- This design also reduces spiral tendencies in the aircraft.

**Disadvantages:**

- The only disadvantage of the inverted V-tail is the need for extra ground clearance.

# Horizontal and Vertical Tail Design (contd)

**7. Y - Tail**

- The Y-tail is similar to the V-tail except that a vertical tail surface and vertical rudder are used for directional control.
- The Y-tail eliminates the complexity of the "ruddervators" on the V-tail, but still retains a lower surface area compared to the conventional tail design.

**Inverted Y-Tail**

- An inverted Y-tail was used on the F-4 aircraft as a means of keeping the horizontal surfaces out of the wake of the main wing at high AoAs.
- Another example of an inverted Y-tail is on the Altus I high-altitude surveillance drone.

# Horizontal and Vertical Tail Design (contd)

## 8. Twin - Tail

- The twin-tail is a common design on highly maneuverable combat aircraft.
  ( Examples: F-14, F-15, F-18, F-22, MiG-25 )
- The purpose of the twin-tail is to position the vertical tail surfaces and rudders away from the fuselage centerline, where they can be affected by the fuselage wake at high angles of attack.

# Horizontal and Vertical Tail Design (contd)

## 9. CANARD

- Canard is a horizontal stabilizer that is located forward of the main wing, on the fuselage.

### Types of Canard:

### (a) Control Canard:

- This type of canard is designed to produce very little lift.
- The control canard provides the same function as the aft horizontal stabilizer by introducing a moment that changes the AoA of the fuselage and main wing.
  ( Examples of Control Canard: Concorde, Tu-144 )

### (b) Lifting Canard:

- This type of canard is designed to produce considerable lift - up to 15-25 % of the total lift.
- As a result, it reduces the lift (burden) , and lift-induced drag on the main wing.
- The lifting canard is designed to stall at a lower AoA than the main wing.

  As a result, the nose of the aircraft will drop before the main wing can stall and, therefore, make it statically stable.

- In contrast to an aft tail, a canard uses a positive (downward) elevator to offset the moment produced by the main wing in level flight.

  This produces an upward lift component, which augments the main wing and further reduces the lift and lift-induced drag on the main wing.

- An example of the aircraft that uses a lifting canard is the Q-200 "Quickie".

  In this case, the canard has a plain elevator and also doubles as the main landing gear spring.

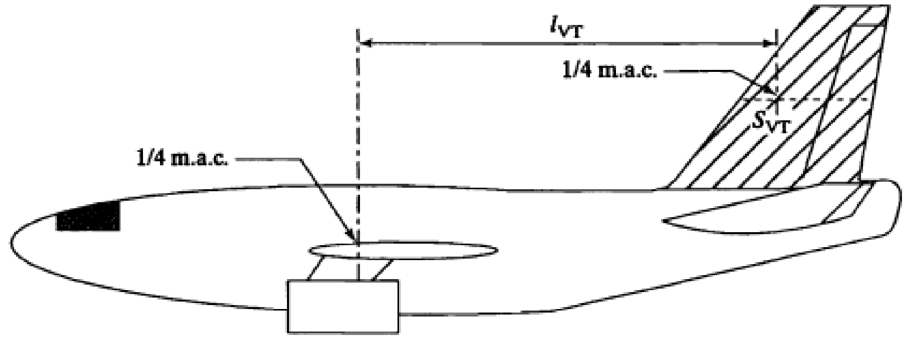
# Vertical Tail Sizing

- In the conceptual design, the sizing of the vertical and horizontal tail surfaces is based on the design of the past aircraft.
- The area of the vertical stabilizer is given by,

$$S_{VT} = C_{VT} \frac{b_W S_W}{l_{VT}}$$

where, $b_W$ and $S_W$ are the span and area of the main wing, respectively, and $l_{VT}$ is the distance between the quarter-chord locations of the mean-aerodynamic-chords (m.a.c) of the main wing and vertical stabilizer, as show in Fig.

- Values of the coefficient, $C_{VT}$, for different types of aircraft are listed in the Table.
- At the conceptual design stage, the coefficient $C_{VT}$ for the aircraft being designed, should be taken from aircraft with similar mission requirements.
- The distance, $l_{VT}$ , is in effect the moment arm upon which the aerodynamic force generated by the vertical stabilizer acts on the fuselage.
- The equation for the area of the vertical stabilizer indicates that a larger distance requires a smaller vertical tail area.
- Therefore, the length parameter, $l_{VT}$ , is an useful parameter in the design of the tail.




Vertical and aft-horizontal tail coefficients.

| | $C_{VT}$ | $C_{HT}$ |
|---|---|---|
| Sail Plane | 0.02 | 0.50 |
| Homebuilt | 0.04 | 0.50 |
| General Aviation (single engine) | 0.04 | 0.70 |
| General Aviation (twin engine) | 0.07 | 0.80 |
| Twin Turboprop | 0.08 | 0.90 |
| Combat Jet Trainer | 0.06 | 0.70 |
| Combat Jet Fighter | 0.07 | 0.40 |
| Military Transport/Bomber | 0.08 | 1.00 |
| Commercial Jet Transport | 0.09 | 1.00 |

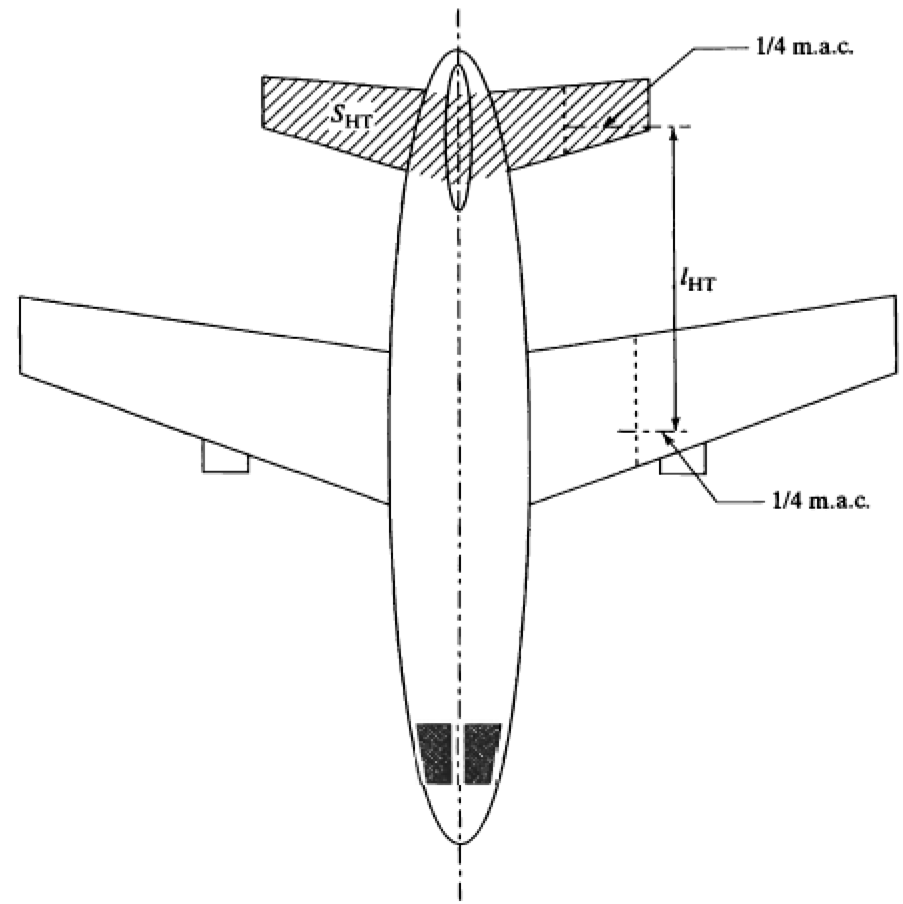
# Aft-Horizontal Tail Sizing

- The area of the aft horizontal stabilizer is given by

$$S_{HT} = C_{HT} \frac{\bar{c}_W S_W}{l_{HT}}$$

where, $\bar{c}_W$ , is the m.a.c of the main wing, and $l_{HT}$ Is the distance between the quarter-chord points of the main-wing and the horizontal stabilizer, as shown in Fig.

- The coefficient $C_{HT}$ is used in scaling the aft-horizontal stabilizer. Its values for different types of aircraft are given in Table.
- At the stage of conceptual design, $C_{HT}$ , should be taken from aircraft with similar mission requirements.
- It is to be noted that in contrast to the vertical stabilizer, $S_{HT}$ , includes the portion that runs through the fuselage.



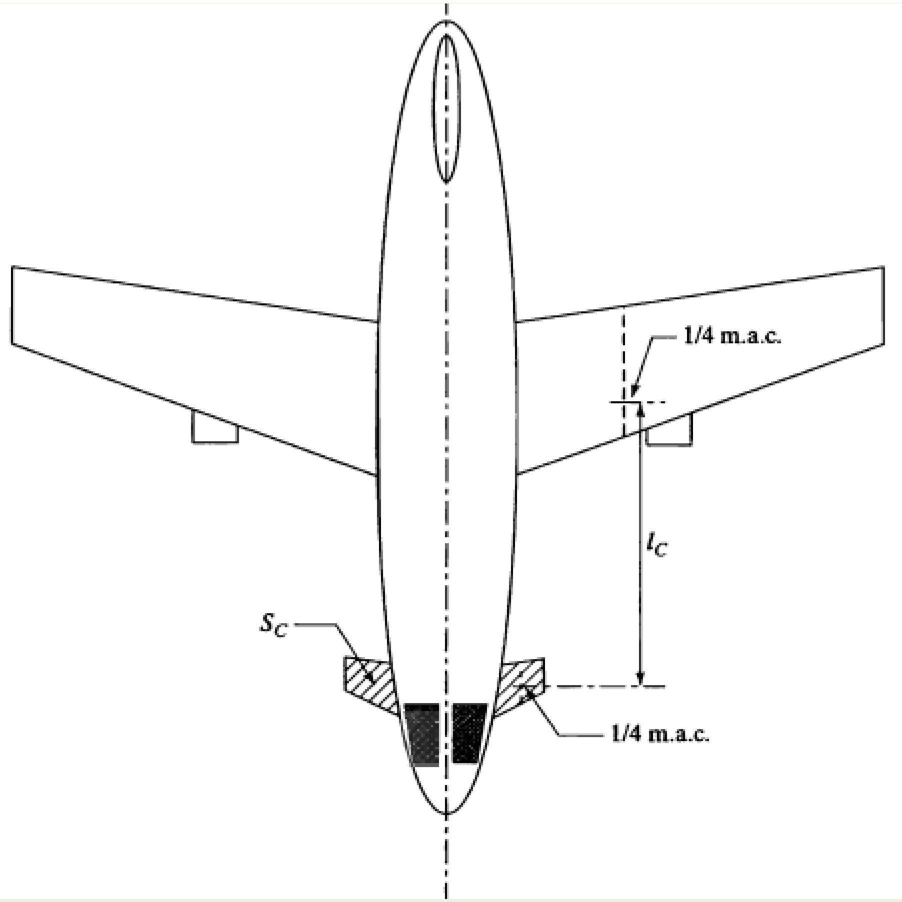
# Canard Sizing

- The coefficient used in scaling the canard is $C_C$.
- The area of the canard is given by

$$S_C = C_C \frac{\bar{c}_W S_W}{l_C}$$

where, $l_C$, is the distance between the quarter-chord locations of the mean-aerodynamic-chords (m.a.c) of the main wing and canard, as shown in Fig.

- The values of $C_C$ for different aircraft are given in Table.
- *The values of the sizing coefficient given in Table are relevant only for the control canard.*
- For lifting canards, the area is primarily based on the percentage of the total lift that the canard is designed to produce.




Control-canard sizing coefficient.

| | $C_C$ | Cruise Mach No. |
|---|---|---|
| B-70 | 0.104 | 2+ |
| CL-408 | 0.12 | 3 |
| NAA-M3.0 | 0.10 | 3 |
| F-108 | 0.11 | 2+ |

# Scaling for Different Tail Types

•

**T-Tail Design:**

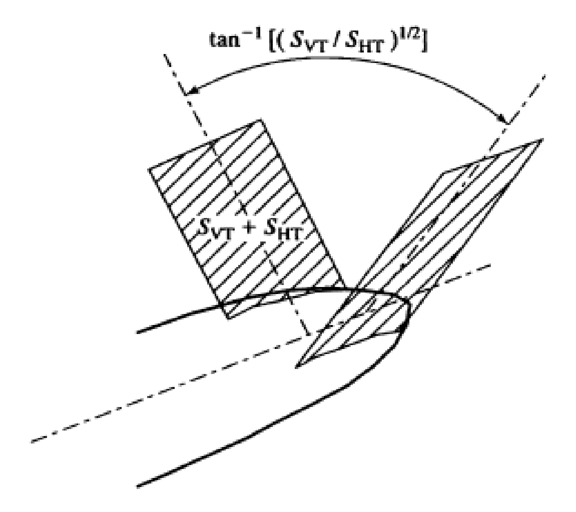
- As a general rule, for T-tail designs, the vertical and horizontal tail coefficients can be reduced by 5% compared to a conventional tail.

**H-Tail Design:**

- For an H-tail design, the horizontal tail coefficient can be reduced by 5%. Also, the vertical tail area on each side will be one-half of the required total area corresponding to a conventional tail.

**V-Tail Design:**

- For a V-tail design, the area should be the same as the combined horizontal and vertical surface areas of an equivalent conventional tail design.
- In addition, the dihedral angle of the two surfaces should be the arc-tangent of the square-root of the ratio between the required vertical and horizontal tail areas, as shown in Fig.
- The dihedral angle should be approximately 45 deg.
- The lengths, $l_{VT}$ and $l_{HT}$, will vary depending on the type of aircraft.




- For an aircraft with a front-mounted propeller engine, these lengths are approximately 60% of the fuselage length.
- With an aircraft with wing-mounted engines, these lengths are approximately 50-55 % of the fuselage length.
- For engines that are mounted on the aft portion of the fuselage, these lengths are from 45-50 % of the fuselage length.

**Control Canards:**

- With control canards, $l_C$ , varies from 30 to 50% of the fuselage length.

# Scaling for Different Tail Types (contd)

## Coefficient Scaling for Different Tail Types

| Type | Equivalent $C_{VT}$ | Equivalent $C_{HT}$ |
|---|---|---|
| T-Tail | 0.95 | — |
| H-Tail | 0.50 | 0.95 |
| V-Tail | 1.00 | 1.00 |

## Typical lengths for $l_{VT}$ , $l_{HT}$ and $l_C$

| Type | $l_{Tail} / l_{Fuselage}$ |
|---|---|
| Front-Mounted Prop. | 0.60 |
| Wing-Mounted Engines | 0.50–0.55 |
| Fuselage-Mounted Engines | 0.45–0.50 |
| Canard | 0.30–0.50 |

# Tail Planform Shape

- Once the required areas of the horizontal and vertical tail surfaces are found, the planform shapes are determined next.
- As with the main wing design, the planform shape is defined by the aspect ratio, *A*, and the taper ratio, $\lambda$

$$A = \frac{b^2}{S}$$ where *S* = surface area and, *b* = span.

- The taper ratio then defines the root-chord as

$$C_r = \frac{2S}{b(1+\lambda)}$$

- The tip-chord is then given by, $C_t = \lambda C_r$
- Historical values of the aspect ratio and taper ratio of aft-horizontal and vertical tail surfaces are given in the table.
- The leading edge sweep angle, $\Lambda_{LE}$ , of the aft-horizontal stabilizer is typically set to be a few degrees more than the sweep angle of the main wing. $(\Lambda_{LE})_{HT} > (\Lambda_{LE})_{Wing}$

Aft-horizontal and vertical tail aspect and taper ratios based on historic aircraft.

| | Aft-horizontal | | Vertical | |
|---|---|---|---|---|
| | *A* | λ | *A* | λ |
| Combat | 3–4 | 0.2–0.4 | 0.6–1.4 | 0.2–0.4 |
| Sail Plane | 6–10 | 0.3–0.5 | 1.5–2.0 | 0.4–0.6 |
| Other | 3–5 | 0.3–0.6 | 1.3–2.0 | 0.3–0.6 |
| T-Tail | — | — | 0.7–1.2 | 0.6–1.0 |

- This gives the aft-horizontal stabilizer a higher critical Mach number than the main wing. This also helps to avoid the loss of elevator effectiveness due to shock formation.
- If it is desired that $(\Lambda_{LE})_{HT} = (\Lambda_{LE})_{Wing}$ , then the above mentioned benefits can be obtained by reducing the $(t/c)_{\max}$ of horizontal stabilizer compared to the main wing.
- The sweep angle of the vertical stabilizer, $(\Lambda_{LE})_{VT}$ generally varies between 35 and 55 deg. For supersonic aircraft, higher sweep angles may be used if the leading-edge Mach number is intended to be subsonic.

# Airfoil Section Type
## (for Horizontal and Vertical Tails)

- The airfoil section type selected for the horizontal and vertical stabilizers should be:
  1) symmetric airfoil section, and
  2) having a low base drag coefficient, $C_{D_0}$
- The tail surfaces do not produce lift except with the deflection of control surfaces, which are the elevator and rudder for the horizontal and vertical stabilizers, respectively.
- As a result, the stabilizers should be symmetric airfoils that are not placed at an angle of attack.

  When the control surfaces are deflected, the effect is equivalent to adding camber to the section shape.
- Because the stabilizers are symmetric sections, at $0^o$ angle of attack, they do not produce lift or lift-induced drag.

  Therefore, the only drag component is the base drag, $C_{D_0}$. As a result, wing sections that have a lower base drag are preferable for cruise efficiency.

- As with the main wing, $C_{D_0}$, is the sum of viscous skin friction drag, $C_f$, flow separations.
  Again, the friction coefficient will be multiplied by the form and interference factors.
  The form factor for the main wing must be increased by 10% and applied to the tail.
  Values of interference factor for different tail arrangements is shown in the Table.
- The horizontal and vertical stabilizers should have a higher critical Mach number than the main wing.
- This can be achieved by choosing a slightly smaller section $(t/c)_{\max}$ than the wing.
- However, care should be taken to ensure that the stall angle, $\alpha_s$, of the tail surfaces are not reduced too much by reducing $(t/c)_{\max}$

Values of interference factor, $Q$, for different tail arrangements.

| | $Q$ |
|---|---|
| Conventional Tail | 1.05 |
| V-Tail | 1.03 |
| H-Tail | 1.08 |

# Tail Placement

- The placement of the aft-horizontal and vertical stabilizers affects the stall and spin characteristics of an aircraft.

**STALL CONTROL:**

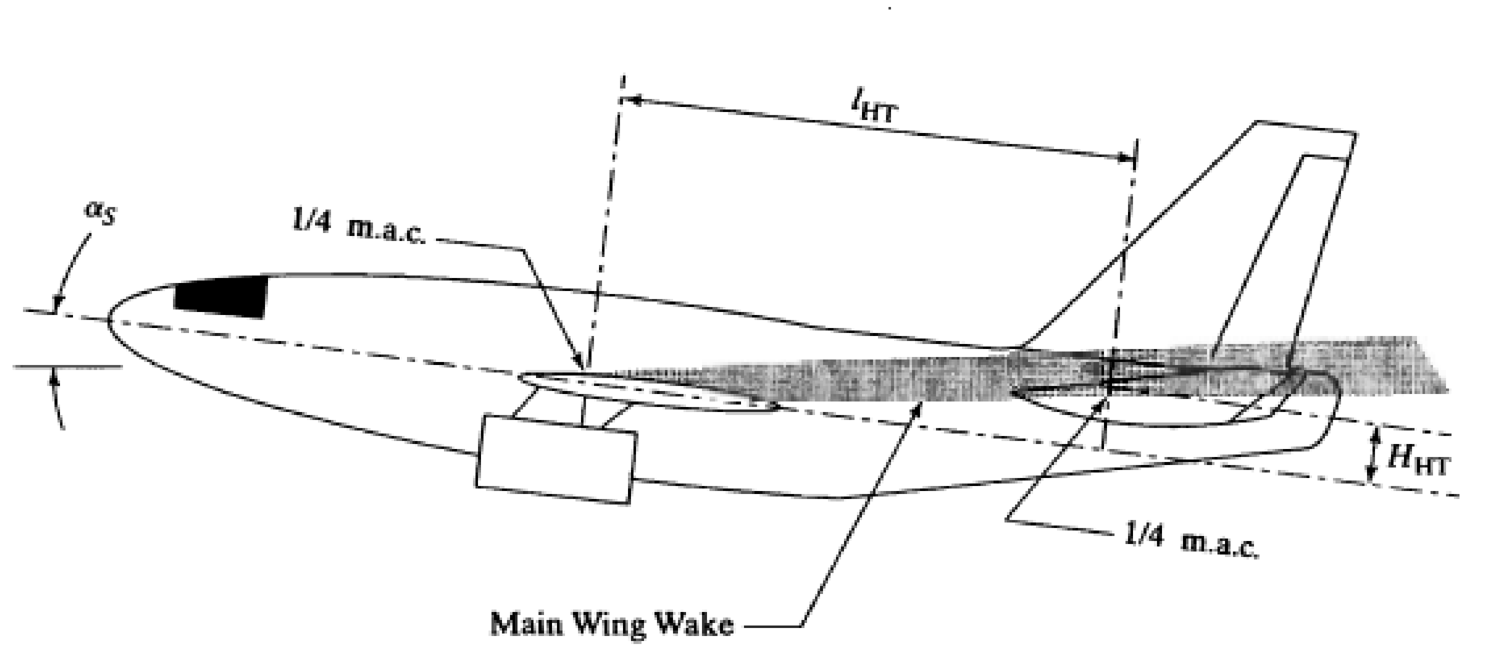
- Stall characteristics are affected by the location of the horizontal stabilizer with respect to the main wing.
- If the horizontal stabilizer is in the wake of the main wing at the stall angle of attack, $\alpha_s$ , elevator control will be lost, and further pitch up may occur.

- The solution to this potential problem is to locate the horizontal stabilizer in one of two regions:

1. Near the mean chord line of the main wing or
2. Above the wake of the main wing at the stall angle of attack.

When the main wing stalls, the airflow will separate from the leading and trailing edges.

The wake of the main wing will spread with a total angle of approximately 30 deg.

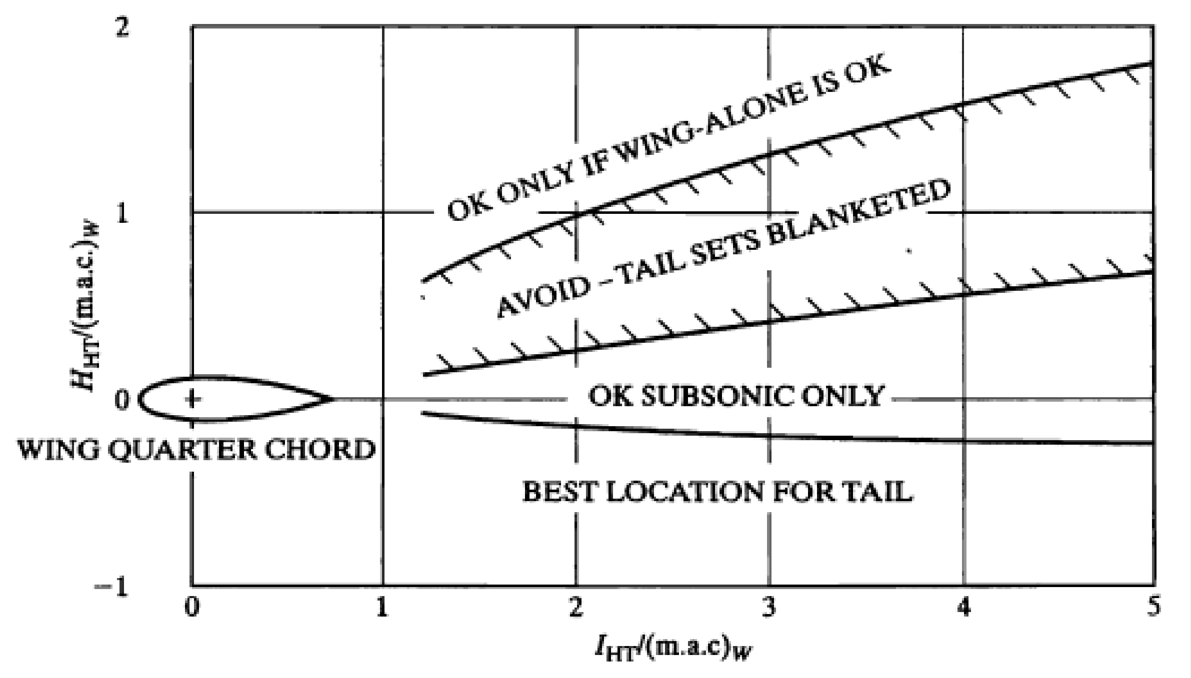


# Tail Placement (contd)

- Fig. illustrates a safe region for the vertical placement of a horizontal stabilizer in a conventional tail.
- Both the downstream location, $l_{HT}$ , and height above the main wing centerline, $H_{HT}$, are normalized by the main wing m.a.c.
- This indicates that the best location for the horizontal tail is below the wing centerline.

  However, a higher position, such as with Cruciform or T-tails, is possible if they are set high enough above the wing.

- For a T-tail, all but the trailing edge of the elevator needs to be outside the wake of the main wing.
- Having the elevator just inside the wake produces an unsteady buffeting on the pitch control that signals the pilot of an imminent stall.

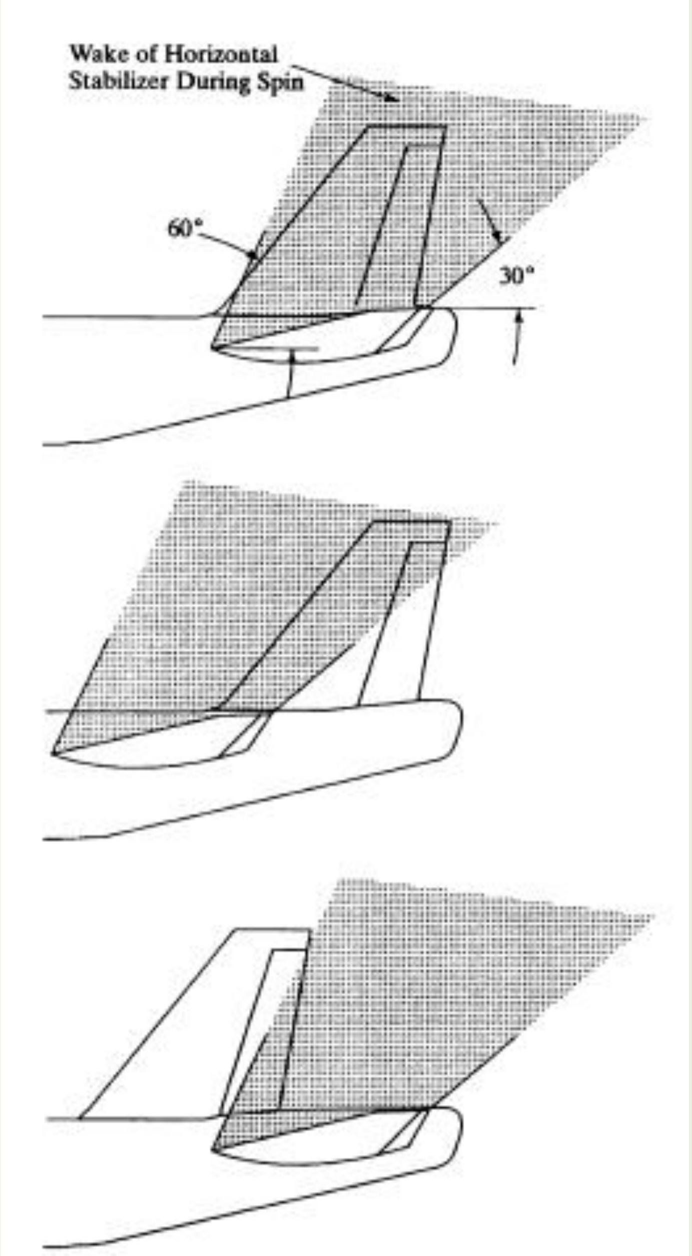


# Tail Placement (contd)

**SPIN CONTROL:**

- Spin characteristics are affected by the vertical tail.
- During an uncontrolled spin, the aircraft is falling vertically and rotating about its vertical axis.

  Recovery from the spin requires having a sufficient amount of rudder control.
- As shown in Fig., for a conventional tail, the vertical stabilizer is caught in the wake of the horizontal stabilizer during an uncontrolled spin. This makes the rudder ineffective.
- The solution for a conventional tail design is to move the horizontal stabilizer either forward or aft of the vertical stabilizer position. This is evident in the tail design of the Boeing 777.
- Alternatively, the horizontal stabilizer can be positioned higher on the vertical stabilizer. This is an advantage of the Cruciform or T-tail designs.
- In either, approach, a good design should have approximately 30% of the rudder outside of the wake of the horizontal stabilizer during a spin for proper recovery.



# Spreadsheet for Tail Design

**INPUT DATA:**

- The top of the spreadsheet contains input that corresponds to the main wing.
- Speed, altitude and air properties are also input

**DESIGN:**

- The tail design spreadsheet is divided into two parts.
- The top part deals with the vertical stabilizer. It ends with a graphical representation of the plan view of the total stabilizer.
- For the vertical tail design, there are two sets of input parameters.
  The left set includes
  i) the vertical tail coefficient
  ii) the distance between the quarter-chord locations on the m.a.c. of the main wing and vertical stabilizer
- The bottom part deals with the horizontal stabilizer. At the end, the plan view of one-half of the surface is shown. The other half is a mirror image.

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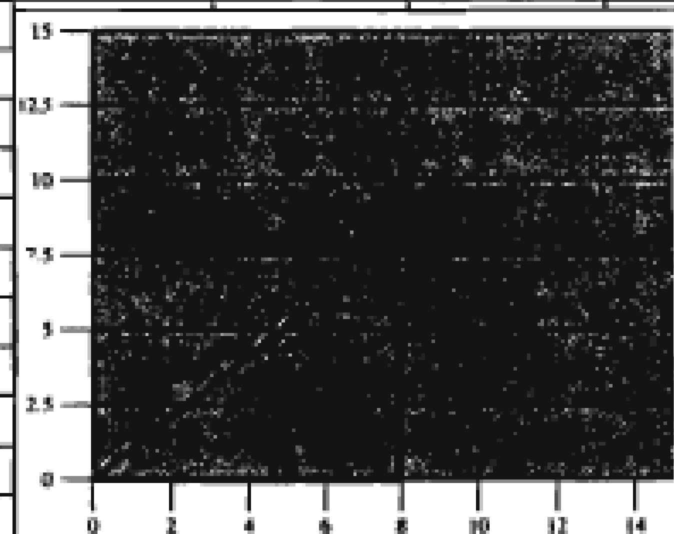
| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| **Main Wing Reference** | | | | **Air Properties** | | | | |
| b | 32.2 | ft | Cruise Alt. | 55,000 | ft | | | |
| m.a.c. | 21.5 | ft | V | 1,925.70 | f/s | | | |
| S | 519 | ft$^2$ | $\rho$ | 0.00922 | lbm/f^3 | | | |
| M | 2.10 | | q | 531.07 | lbf/f^2 | | | |
| $\Lambda_{LE}$ | 62 | deg | $\mu$ | 107.0E-7 | lbm/(f-s) | | | |
| t/c | 0.04 | | $\nu$ (cruise) | 116.0E-5 | f^2/s | | | |
| $\lambda$ | 0.00 | | | | | | | |
| | | | | | | | | |
| ***Vertical Tail*** | | | | | | | | |
| **Design Parameters** | | | **Airfoil Data** | | | | | |
| Cvt | *0.07* | | Name | *NACA 64-* | *004* | | | |
| Lvt | *40.0* | ft | $Cl_{max}$ | *0.8* | | | | |
| $\Lambda_{LE}$ | *63* | deg | $Cl_{\alpha}$ | *0.11* | 1/deg | | | |
| t/c | *0.04* | | a.c. | *0.26* | c | | | |
| $\lambda$ | *0.30* | | $\alpha_{0L}$ | *0* | deg | | | |
| Avt | *1.10* | | Cd | *0* | | | | |
| **Calculations** | | | **Sweep Angles** | | | **Viscous Drag** | | |
| Svt | 29 | ft$^2$ | | x/c | $\Lambda_{x/c}$(deg) | V_eff | 874.25 | f/s |
| b | 5.7 | ft | LE | 0.00 | 63.0 | q_eff | 109.46 | lbf/f^2 |
| $c_r$ | 7.9 | ft | 1/4 chord | 0.25 | 55.8 | M_eff | 0.95 | |
| $c_t$ | 2.4 | ft | (t/c)max | 0.35 | 51.9 | Re_mac | 426.1E+4 | |
| m.a.c. | 5.7 | ft | TE | 1.00 | 0.3 | sqrt(Re) | 2064.22 | |
| $\beta$ | 1.85 | | | | | Cf | 3.19E−03 | |
| $C_{L\alpha}$ | **0.023** | 1/deg | | | | S_wet | 58.58 | ft$^2$ |
| | | | | | | *F* | 1.57 | |
| | | | | | | Q | 1.05 | |
| Total Drag | **163.880** | lbf | | | | $C_{D0}$ | **0.0106** | |
| | | | | | | | | |

**Spanwise View**

| x | y |
|---|---|
| 0 | 0 |
| 7.9 | 0 |
| 7.95 | 5.7 |
| 5.57 | 5.7 |
| 0 | 0 |



***Horizontal Tail***

**Design Parameters**

| | | |
|---|---|---|
| Cht | 0.11 | |
| Lht | 50.0 | ft |
| $\Lambda_{LE}$ | 63 | deg |
| t/c | 0.04 | |
| $\lambda$ | 0.35 | |
| Aht | 2.00 | |

**Airfoil Data**

| | | |
|---|---|---|
| Name | *NACA 64-* | *004* |
| $Cl_{max}$ | *0.8* | |
| $Cl_{\alpha}$ | *0.11* | 1/deg |
| a.c. | *0.26* | c |
| $\alpha_{0L}$ | *0* | deg |
| Cd | *0.0040* | |

**Calculations**

| | | |
|---|---|---|
| Sht | 25 | $ft^2$ |
| b | 7.0 | ft |
| $c_r$ | 5.2 | ft |
| $c_t$ | 1.8 | ft |
| m.a.c. | 3.8 | ft |
| $\beta$ | 1.85 | |
| $C_{L\alpha}$ | **0.030** | 1/deg |
| | | |
| | | |
| Total Drag | **140.864** | lbf |

**Sweep Angles**

| | x/c | $\Lambda_{x/c}$(deg) |
|---|---|---|
| LE | 0.00 | 63.0 |
| 1/4 chord | 0.25 | 59.9 |
| (t/c)max | 0.35 | 58.4 |
| TE | 1.00 | 45.0 |

**Viscous Drag**

| | | |
|---|---|---|
| V_eff | 874.25 | f/s |
| q_eff | 109.46 | lbf/f^2 |
| M_eff | 0.95 | |
| Re_mac | 284.4E+4 | |
| sqrt(Re) | 1686.43 | |
| Cf | 3.42E−03 | |
| S_wet | 49.17 | $ft^2$ |
| *F* | 1.5 | |
| *Q* | 1.05 | |
| $C_{D0}$ | **0.0108** | |

**Spanwise View**

| x | y |
|---|---|
| 0 | 0 |
| 5.2 | 0 |
| 8.69 | 3.5 |
| 6.88 | 3.5 |
| 0 | 0 |