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Aircraft Design in a Nutshell

Based on the Aircraft Design Lecture Notes

1 Introduction

The **task of aircraft design** in the **practical sense** is to supply the "*geometrical description of a new flight vehicle*".
To do this, the new aircraft is described by
- a three-view drawing,
- a fuselage cross-section,
- a cabin layout and
- a list of aircraft parameters.

The following **requirements** should be known when aircraft design begins:
Cruise performance:
- Payload m_{PL}
- Range R
- Mach number M_{CR}

Airport performance:
- Take-off field length s_{TO}
- Landing field length s_L
- Climb gradient γ_{CLB} (2nd segment)
- Missed approach climb gradient γ_{MA} .

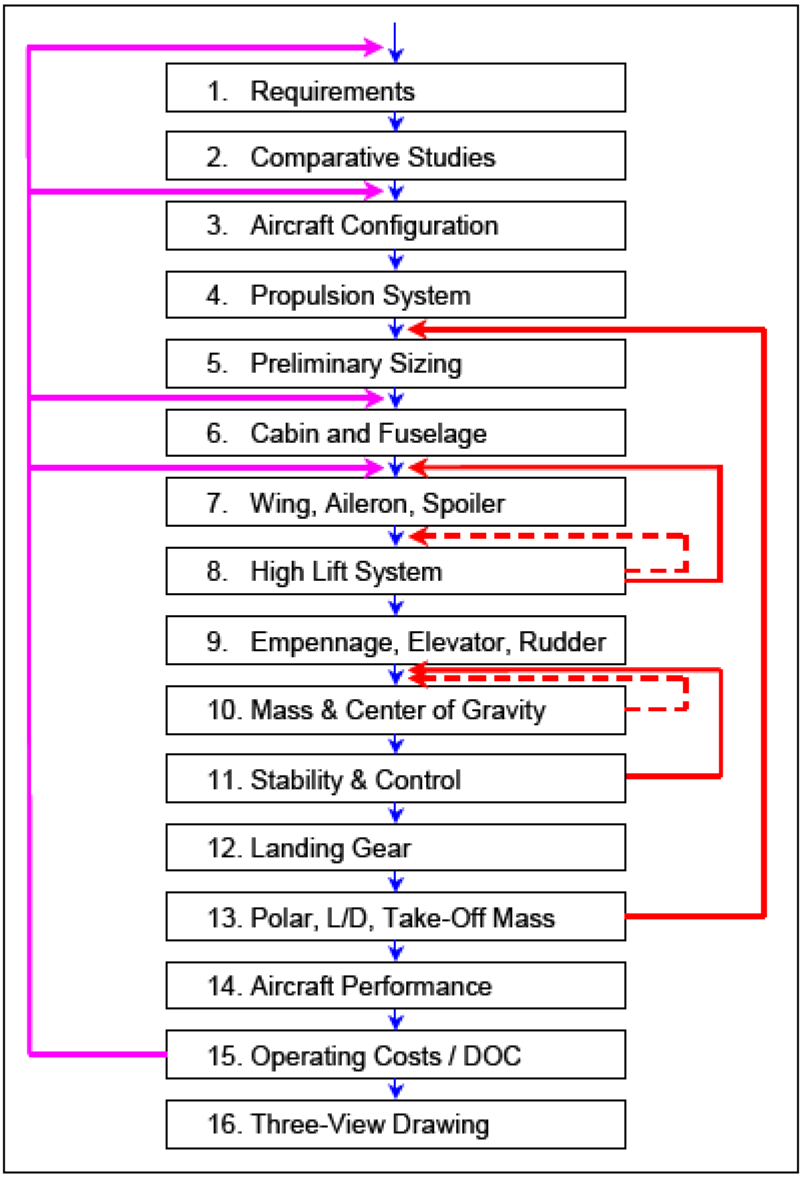
The **key design parameters** are:
- Take-off mass m_{TO}
- Fuel mass m_F
- Operating empty mass m_{OE}
- Wing area S_W
- Take-off thrust T_{TO} .

The **task of aircraft design** in an **abstract sense** is to determine the design parameters so as to ensure that
1. the *requirements* and *constraints* are met (then we have a permissible design including certification) and, furthermore,
2. the design *objectives* are optimally met (then we have an optimum design).

2 Aircraft Design Sequence

The sequence of activities during the project phase can be divided into:

1. preliminary sizing (step 1 to 5)
2. conceptual design (step 6 to 16)



3 Requirements and Certification

The **seat-range diagram** shows the aircraft of one manufacturer based on their number of seats versus their range. A big aircraft manufacturer should fill all viable areas of the diagram with aircraft on offer. Filling a seat-range diagram is possible with a limited number of aircraft families.

Aircraft should always be designed as an **aircraft family**. An aircraft family consists of several aircraft based on a standard model and additional aircraft with shortened fuselage (shrink) and aircraft with lengthend fuselage (stretch).

Aircraft families have proved to be economical if the number of seats is increased from one model to the next bigger one by 20 % ... 25%.

load factor = sold payload / offered payload

range flexibility = aircraft range / demanded flight distance

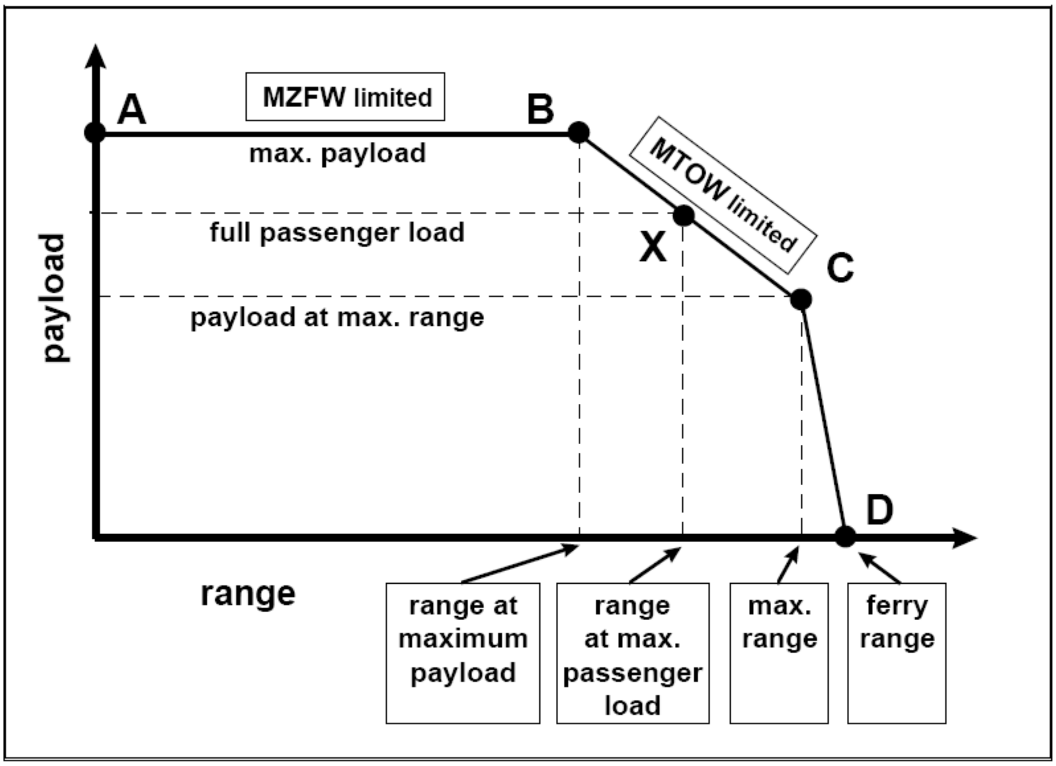
A typical average load factor is	80 %.
A typical average range flexibility is:	4 for short medium range aircraft,
	2 for long range aircraft.

The dependencies of payload and range for one aircraft are depicted in the **payload-range-diagram**. It is based on

$$m_{TO} = m_{OE} + m_F + m_{PL}$$

m_{TO} take-off mass
m_{OE} operating empty mass
m_F fuel mass
m_{PL} payload.

The **payload-range-diagram**:



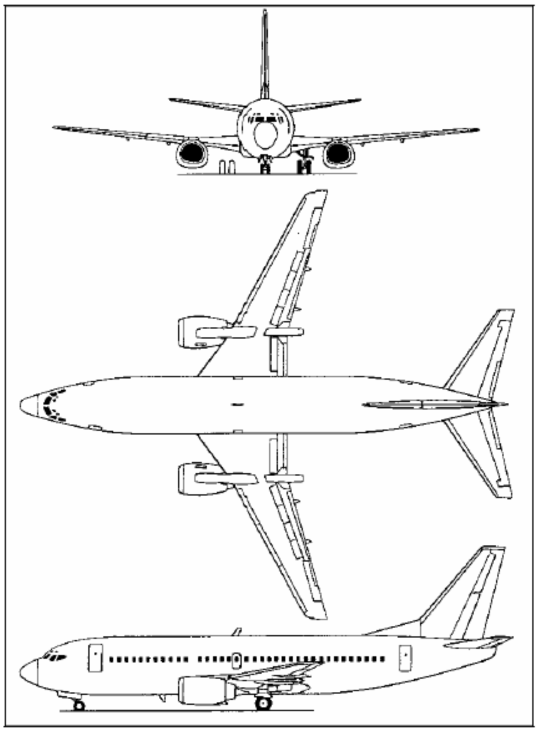
Certification requirements are important for aircraft design because an aircraft may only be operated if it is certified (i.e. has a type certificate).

Table 3.3 Selection of the certification specifications by the characteristics of the aircraft

aircraft type	normal, utility and aerobatic aeroplanes	commuter aeroplanes	large aeroplanes (JAR) transport category airplanes (FAR)
characteristics	passenger seats = 9 MTOW = 5700 kg	passenger seats = 19 MTOW = 8600 kg propeller driven twin-engined	MTOW > 5700 kg
airworthiness standard	**CS-23, FAR Part 23**		**CS-25, FAR Part 25**
interpretative material	**FAR:** Advisory Circular AC23-?		**EASA:** Advisory Circular Joint (ACJ) Advisory Material Joint (AMJ) included in CS-25 **FAR:** Advisory Circular AC25-?

4 Aircraft Configurations

A **three-view-drawing** is used to communicate the ideas about an aircraft configuration.

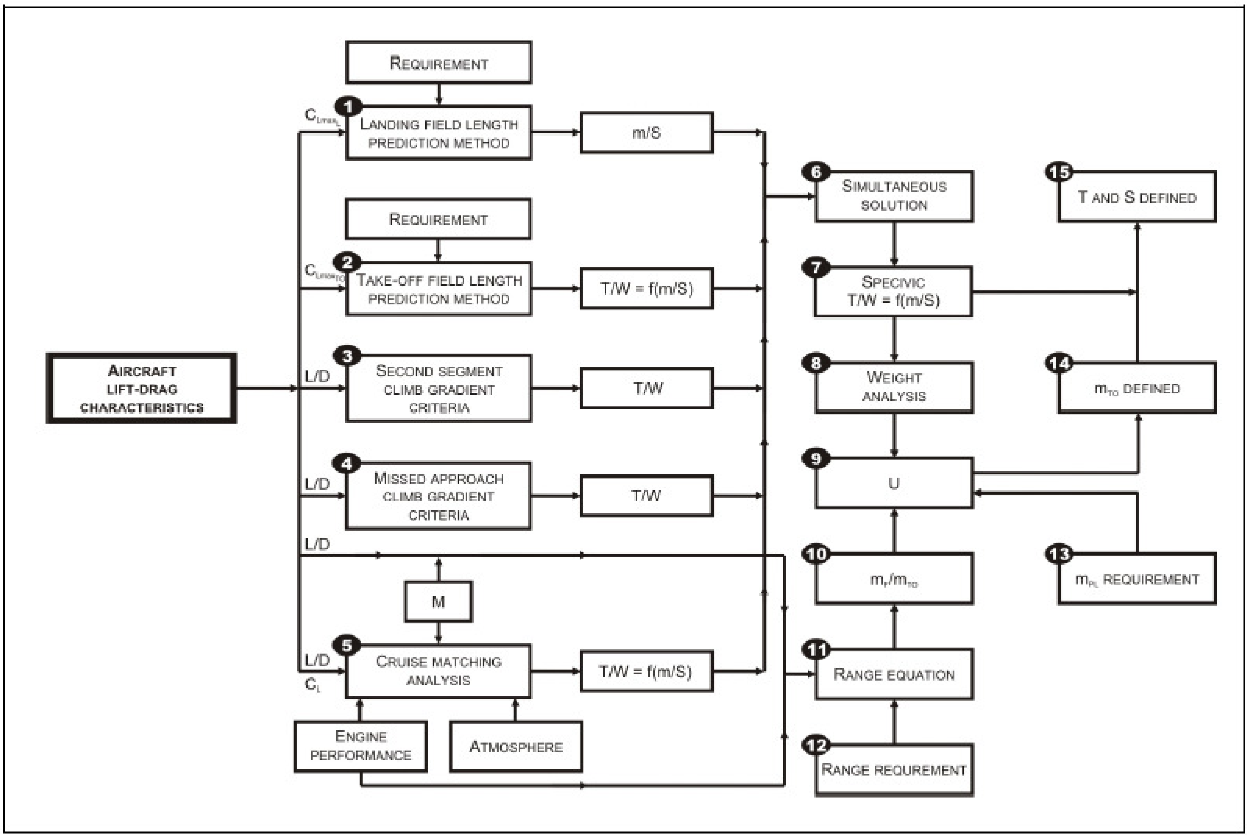


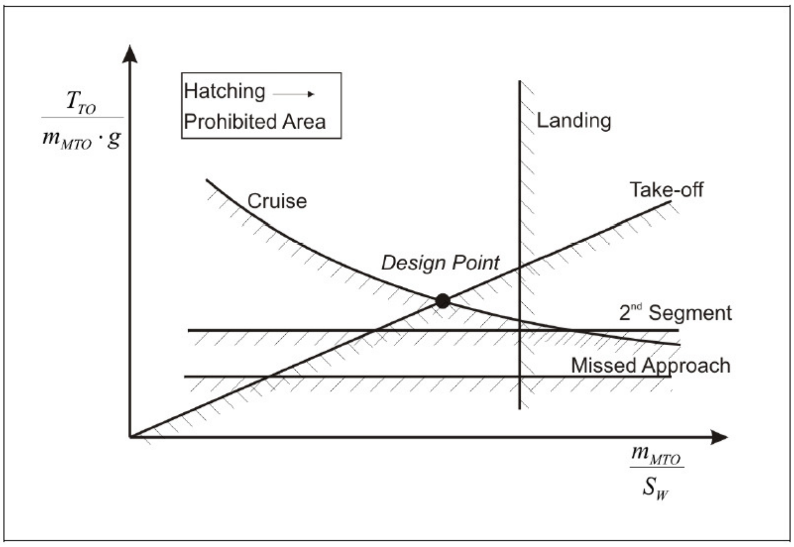
Boeing B737-300

Conventional aircraft configurations all have one fuselage, one wing, and an empennage at their rear end. This configuration is also called *tail aft aircraft*.

Unconventional aircraft configurations differ in at least one attribute from the definition of a conventional configuration.

5 Preliminary Sizing





The aim of optimization is to achieve the following:

- Priority 1: to achieve the smallest possible thrust-to-weight ratio;
- Priority 2: to achieve the highest possible wing loading.

"First law of aircraft design": Estimation of **maximum take-off mass**, MTOW

$$m_{MTO} = \frac{m_{PL}}{1 - \frac{m_F}{m_{MTO}} - \frac{m_{OE}}{m_{MTO}}} \quad . \tag{5.45}$$

Estimation of **operating empty mass**, OEW

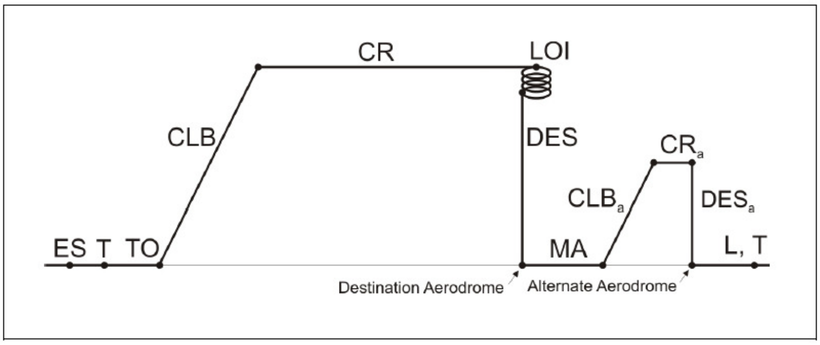
$$\frac{m_{OE}}{m_{MTO}} = 0.23 + 1.04 \cdot \frac{T_{TO}}{m_{MTO} \cdot g} \quad . \tag{5.49}$$

Estimation of **fuel mass**, m_F

$$\frac{m_F}{m_{TO}} = 1 - M_{ff} \quad . \tag{5.52}$$

Mission fuel fraction:

$$M_{ff} = \frac{m_{SO}}{m_T} \cdot \frac{m_T}{m_L} \cdot \frac{m_L}{m_{DES}} \cdot \frac{m_{DES}}{m_{CR,alt}} \cdot \frac{m_{CR,alt}}{m_{CLB}} \cdot \frac{m_{CLB}}{m_{MA}} \cdot \frac{m_{MA}}{m_{DES}} \cdot \frac{m_{DES}}{m_{LOI}} \cdot \frac{m_{LOI}}{m_{CR}} \cdot \frac{m_{CR}}{m_{CLB}} \cdot \frac{m_{CLB}}{m_{TO}} = \frac{m_{SO}}{m_{TO}} \tag{5.50}$$



SFC_T = 16 mg/(Ns)

$$B_s = \frac{L / D \cdot V}{SFC_T \cdot g} \qquad \frac{m_{LOI}}{m_{CR}} = e^{-\frac{s_{CR}}{B_s}}$$

Table 5.9 Generic mission segment mass fractions (based on **Roskam I**)

type of aircraft	engine start	taxi	take-off	climb	descent	landing
business jet	0.99	0.995	0.995	0.98	0.99	0.992
jet transport	0.99	0.99	0.995	0.98	0.99	0.992

$$T_{TO} = m_{MTO} \cdot g \cdot \left(\frac{T_{TO}}{m_{MTO} \cdot g} \right) \qquad S_W = m_{MTO} / \left(\frac{m_{MTO}}{S_W} \right)$$

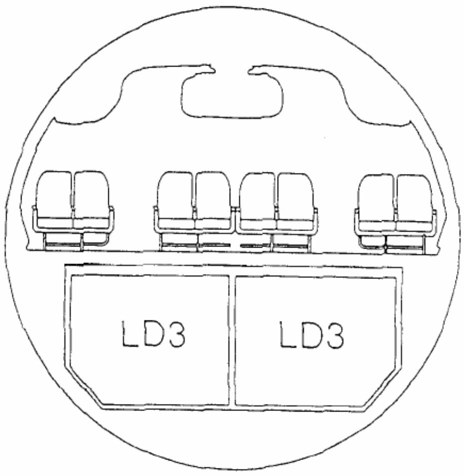
For the full set of calculations for preliminary sizing an **Excel table** is provided!

6 Fuselage Design

Number of seats in one row (**number of seats abreast**) in economy class:

$$n_{SA} = 0.45 \cdot \sqrt{n_{PAX}}$$

Cabin cross section:



Internal fuselage diameter (internal cabin width), $d_{F,I}$:

$d_{F,I}$ = width of all seats + width of all aisles + 2 $\cdot$ (gap between seat and side wall)

Width of seats (economy class):

Single seat	21 in
Bench with 2 seats	40 in
Bench with 3 seats	60 in

Width of aisles:

Minimum according to certification rules	15 in
Typical short medium range	19 in

Number of aisles, CS-25.817 requires:

$n_{SA} \leq 6$: one aisle
$6 < n_{SA} \leq 12$: two aisles

Gap between seat and side wall: 1 in

Conversion to SI units: 1 in = 0,0254 m

Fuselage wall thickness (left and right), Δd:

$$\Delta d = d_{F,O} - d_{F,I} = 0.084\,\mathrm{m} + 0.045 \cdot d_{F,I}$$

Outer **fuselage diameter** (internal cabin width), $d_F = d_{F,O}$:

$$d_F = d_{F,I} + \Delta d$$

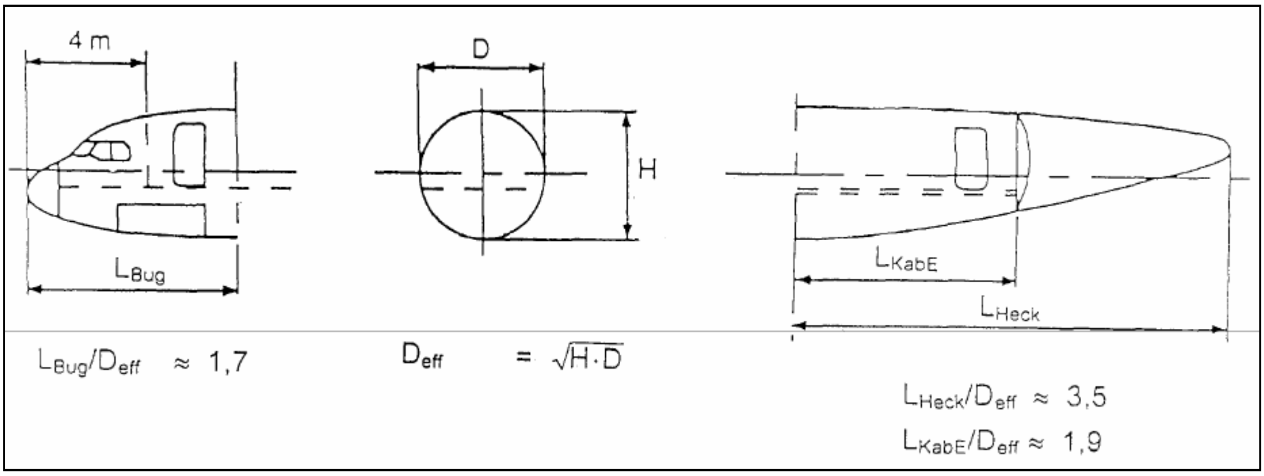
Number of rows:

$$n_R = n_{PAX} / n_{SA}$$

Cabin length:

$$l_{cabin} = n_R \cdot 1\ \mathrm{m}$$

Fuselage shape of a passenger aircraft:



Bug (German) = bow
Heck (German) = stern

Tail angle and length of stern are related**:**

$\varphi_{tail} = \arctan\,(d_F / l_{stern})$ $\qquad\qquad$ $\varphi_{tail} = \arctan\,(1/ 3.5) = 15.9°$

$l_{stern} = d_F / \tan \varphi_{tail}$

φ_{tail} is also the maximum angle for rotation at take-off.
φ_{tail} is also the aircraft's angle of attack, α at take-off.

Fuselage length:

$$\boxed{l_F = l_{cabin} + l_{cockpit} + l_{tail} = l_{cabin} + 4\,\mathrm{m} + 1.6 \cdot d_F}$$

Checking for sufficient size of cargo compartment:

$$\boxed{V_{CC} \geq V_C + \left(V_B - V_{OS}\right)}$$

V_{CC} volume of the cargo compartment,
V_C volume of cargo,
V_B volume of baggage,
V_{OS} volume of overhead stowage.

$$V_{CC} = l_F \cdot k_{CC} \cdot S_{CC}$$

k_{CC} proportion of the fuselage length used for cargo ranging from 0.35 to 0.55,
S_{CC} cross-section of the cargo compartment.

$$V_B = m_B / \rho_B$$
$$V_C = m_C / \rho_C$$
$$V_{OS} = S_{OS,tot} \cdot l_{OS}$$
$$S_{OS,tot} = n_{OS,lat} \cdot S_{OS,lat} + n_{OS,ce} \cdot S_{OS,ce}$$
$$l_{OS} = k_{OS} \cdot l_{cabin}$$

Density of baggage, ρ_B and cargo, ρ_C :

Baggage: 170 kg/m^3,
Cargo: 160 kg/m^3.

m_B	mass of baggage,
m_C	mass of cargo,
ρ_B	density of baggage,
ρ_C	density of cargo,
$S_{OS,tot}$	total cross-section of the overhead stowages calculated as a sum of the cross-sections of lateral stowages, $S_{OS,lat}$, and central stowages, $S_{OS,ce}$,
$n_{OS,lat}$	number of lateral rows of overhead stowages,
$n_{OS,ce}$	number of central rows of overhead stowages: $n_{OS,ce} = n_{aisles} - 1$,
l_{OS}	total length of the overhead stowages (lateral and central),
k_{OS}	proportion of the cabin length occupied by the overhead stowages.

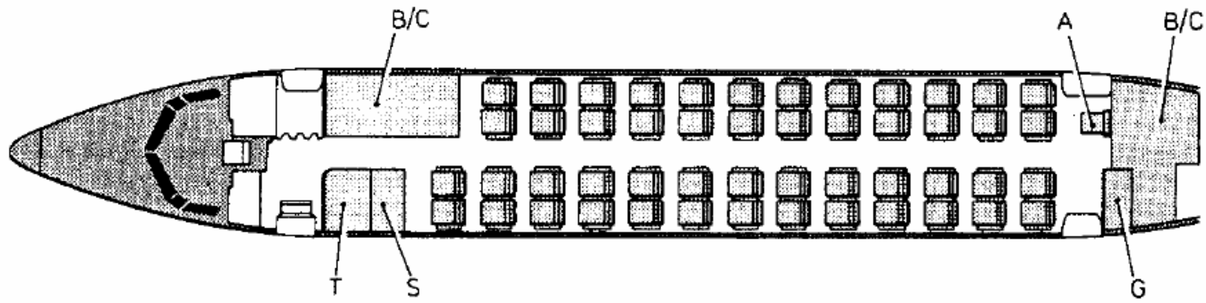
Table 3.3 Values for the $S_{OS,lat}$, $S_{OS,ce}$ and k_{OS} for selected aircraft with 1 or 2 aisles

n_{OS}	Selected Aircraft	k_{OS}	$S_{OS,lat}$	$S_{OS,ce}$	ρ_B
Single aisle:					
	Average	**0.723**	**0.201**	**-**	**180.13**
Twin aisle:					
	Average	**0.751**	**0.208**	**0.241**	**185.01**
	Overall average	**0.737**	**0.213**	**-**	**182.57**

S_{OS} given in m². ρ_B is the maximum allowed baggage density in the bin.

Table 3.3 from: NIȚĂ, Mihaela Florentina: Contributions to Aircraft Preliminary Design and Optimization. München : Verlag Dr. Hut, 2013. - ISBN 978-3-8439-1163-4, Dissertation, Download: http://OPerA.ProfScholz.de

Cabin layout:



Cabin layout of the Fokker 50:
Baggage and cargo are also accommodated in the cabin of this aircraft.
A: attendant seat
B: baggage, C: cargo
G: galley
S: stowage, wardrobe
T: toilet.

7 Wing Design

Sweep angle of the wing**:**

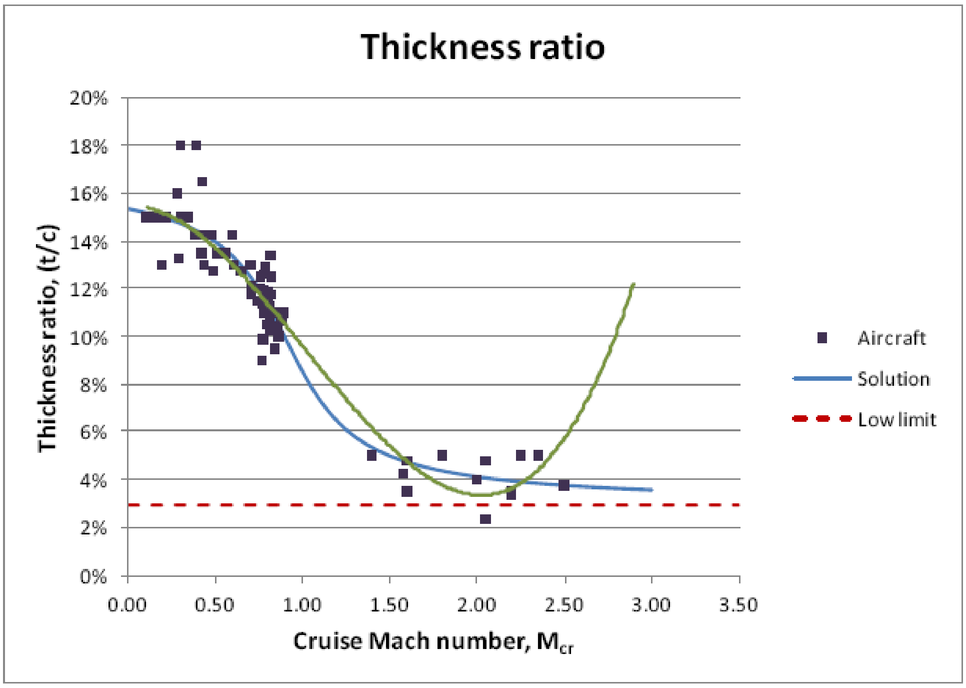
$$\varphi_{25} = 39{,}3° \; (M_{CR})^2$$

This first method can be used for a wide range of cruise Mach numbers (see chart). It is a simple method just based on statistics and considers only one input parameter.

Relative thickness of the wing, *t/c* from cruise Mach number only:

$$\left(\frac{t}{c}\right) = -0{,}0439 \cdot \tan^{-1}(3.3450 \cdot M_{CR} + -3.0231) \;\; +0{,}0986$$

Note: The arctan function delivers an angle in rad in this equation.



Relative thickness of the wing, *t/c* from cruise Mach and wing sweep:

This second method accounts for more parameters, but is only valid for subsonic aircraft cruising above their critical Mach number. These aircraft will typically show a cruise Mach number of more than 0.7 and less that 0.9. The aircraft will typically have a wing sweep of more than 20°.

Airbus and **Boeing** use the following design Mach number $M_{design} = M_{CR}$:

$$M_{DD} = M_{CR}$$

$$M_{DD,eff} = M_{DD} \cdot \sqrt{\cos\varphi_{25}}$$

$$(t/c) = 0.3 \cdot \cos\varphi_{25} \cdot \left(\left[1 - \left(\frac{5 + M_{DD,eff}^{\ 2}}{5 + \left(k_M - 0.25 \cdot C_L\right)^2} \right)^{3.5} \right] \cdot \frac{\sqrt{1 - M_{DD,eff}^{\ 2}}}{M_{DD,eff}^{\ 2}} \right)^{\frac{2}{3}}$$

$k_M = 1.00$	conventional airfoils; maximum t/c at about 0.30c,
$k_M = 1.05$	high-speed (peaky) airfoils, 1960-1970 technology,
$k_M = 1.12$ to 1.15	supercritical airfoils.
C_L :	the design lift coefficient (for cruise) chosen in Section 5

$$t/c = \frac{3(t/c)_t + (t/c)_r}{4}$$

$$(t/c)_r \,/\, (t/c)_t = r \approx 1.3$$

$$(t/c)_t = 4/(3+\mathrm{r})\ t/c$$

$$(t/c)_r = r\ (t/c)_t$$

Optimum **taper ratio**, λ_{opt} :

$$\lambda_{opt} = 0.45 \cdot e^{-0.036 \varphi_{25}}$$

φ_{25} in degree

λ should not be smaller than 0.2 otherwise aileron integration will be too difficult and the wing
tips will have a tendency to stall.

Wing twist:

$$\varepsilon_t = i_{w,tip} - i_{w,root}$$

$\varepsilon_t = -3°$ (wash out). However the A310 (see Subsection 7.5) shows $\varepsilon_t = -8°$.

Incident angle of the wing (at the wing root):

$$i_w = \frac{C_{L,CR}}{C_{L_\alpha}} + \alpha_0 - 0.4 \cdot \varepsilon_t \ .$$

In this equation:

C_{L_α}	the lift curve slope according to equation (7.24),
$C_{L,CR}$	the necessary lift coefficient in cruise flight,
α_0	the angle of attack at zero wing lift or a characteristic profile of the wing,
ε_t	the twist (see above).

$$C_{L,\alpha} = \frac{2 \cdot \pi \cdot A}{2 + \sqrt{A^2 \cdot \left(1 + \tan^2 \varphi_{50} - M^2\right) + 4}}$$

Dihedral angle of the wing:

$$\Gamma = \frac{\partial \Gamma}{\partial k_{Z,W}} \cdot k_{Z,W} + \frac{\partial \Gamma}{\partial \varphi_{25}} \cdot \varphi_{25} + \Gamma_0$$

$k_{Z,W} = 0.0$, for low wing aircraft
$k_{Z,W} = 0.5$, for mid-wing aircraft
$k_{Z,W} = 1.0$ for high-wing aircraft

$\frac{\partial \Gamma}{\partial k_{Z,W}}$	- 7.46°
$\frac{\partial \Gamma}{\partial \varphi_{25}}$	- 0.115
Γ_0	6.91°

NIȚĂ, Mihaela Florentina: Contributions to Aircraft Preliminary Design and Optimization. München : Verlag Dr. Hut, 2013. - ISBN 978-3-8439-1163-4, Dissertation, Download: http://OPerA.ProfScholz.de

Tank volume within the wing (without center tank):

$$V_{tank} = 0.54 \cdot S_W^{1.5} \cdot (t/c)_r \cdot \frac{1}{\sqrt{A}} \cdot \frac{1 + \lambda \cdot \sqrt{\tau} + \lambda^2 \cdot \tau}{(1 + \lambda)^2}$$

$$\tau = \frac{(t/c)_t}{(t/c)_r} \quad .$$

8 Design of Highlift Systems

$$C_{L,max} = 1.1 \cdot C_{L,max,INITIAL\ SIZING}$$

Factor **1.1** ensures that the aircraft can still stay in the air if the empennage creates negative lift to trim the aircraft.

$$0.95 \cdot \Delta C_{L,max,f} + \Delta C_{L,max,s} \geq C_{L,max} - C_{L,max,clean}$$

Factor **0.95** takes into account the following interrelationship: the use of landing flaps creates a moment around the pitch axis. This moment must be compensated for by using trim. The negative lift created by the trim has to be balanced out by an additional lift of the wing.

9 Tail Sizing

$$S_H = \frac{C_H \; S_W \; c_{MAC}}{l_H} \qquad\qquad S_V = \frac{C_V \; S_W \; b}{l_V}$$

l_H the lever arm of the horizontal tailplane is the distance between the aerodynamic centers of wing and horizontal tailplane,

l_V the lever arm of the vertical tailplane is the distance between the aerodynamic centers of wing and vertical tailplane.

As a good approximation the 25 % - point on the mean aerodynamic chord can also be referred to instead of the distances between the aerodynamic centers.

Table 9.4 Conventional tail volume coefficients of horizontal and vertical tails (**Raymer 1989**)

type	**horizontal C_H**	**vertical C_V**
General Aviation - Twin Engine	0.80	0.07
Transport Jets	1.00	0.08

Table 9.5: Conventional tail lever arms of horizontal and vertical tails (**Raymer 1998**)

aircraft configuration	**average of l_H and l_V**
propeller in front of fuselage	60% of fuselage length
engines on the wing	50 ... 55% of fuselage length
engines on the tail	45 ... 50% of fuselage length

- The tail volume coefficients can be reduced by 10% to 15% in the case of **trimmable horizontal stabilizers**.

- In the case of a **T-tail,** the tail volume coefficients can be reduced by 5% for horizontal and vertical tailplane due to the end plate effect and the improved flow.

10 Mass and Center of Gravity

Table 10.1: Mass groups of a very simple mass breakdown for the design based on the mass breakdowns according to [DIN 9020] and [ATA 100]

			Flügel (wing) m_W ,
		+	Rumpf (fuselage) m_F ,
		+	Höhenleitwerk (horizontal tail) m_H ,
		+	Seitenleitwerk (vertical tail) m_V ,
		+	Bugfahrwerk (nose landing gear) $m_{LG,N}$,
		+	Hauptfahrwerk (main landing gear) $m_{LG,M}$,
		+	Triebwerksgondel (nacelle) m_N
	=	Struktur (structure)	
	+	Triebwerk, installiert (power plant, installed) $m_{E,inst}$	
	+	Flugzeugsysteme (aircraft systems) m_{SYS}	
=	Hersteller-Leermasse (⇒ manufacturer's empty weight, MEW) m_{ME}		
	+	Ausrüstung und Besatzung (⇒ standard and operational items)	
=	Betriebsleermasse (⇒ operational empty weight, OEW) m_{OE}		

Iterative calculation of masses

For individual or group masses the percentage of the total mass (MTOW, OEW, MEW or MZFW) is partly calculated. However, the total mass is itself only an estimated figure, as the **masses** only arise **from an iteration**:

Step 1: Calculate the **individual masses** with the equations (10.1) to (10.16).

Step 2: **Add together** all the individual masses for operating empty mass m_{OE} .

Step 3: Calculate the **maximum take-off mass** m_{MTO} with M_{ff} according to Step 5.9.2:

$$m_{MTO} = \frac{m_{MPL} + m_{OE}}{M_{ff}} \quad . \tag{10.17}$$

Step 4: **Continue the (inner) iteration** by returning to Step 1. Keep going through Steps 1 to 3 until the maximum take-off mass m_{MTO} changes by no more than 0.5% from one iteration step to another.

Step 5: If, after the (inner) iteration, the maximum take-off mass m_{MTO} differs by more than 5% from the value m_{MTO} , with which the wing area S_W and take-off thrust T_{TO} were calculated, S_W **and** T_{TO} must be **recalculated**. This is carried out on the basis of the wing loading or the thrust-to-weight ratio determined in Section 5.

Question 4: How far must the wing be moved to find a suitable position for the aircraft's center of gravity?

Step 1: The aircraft is split into 2 main groups:

- The **fuselage group**, *FG,* consists of: horizontal tailplane, vertical tailplane, fuselage, sum of all the systems, and rear engines, if fitted.
- **Wing group**, *WG,* consists of: wing, landing gear, and engines, if wing-mounted.

The mass and center of gravity is determined for both groups (see Question 1).

Step 2: The moment around the leading edge, LE, of the mean aerodynamic chord MAC: LEMAC is established.

$$\left(m_{WG} + m_{FG}\right) \cdot x_{CG,LEMAC} = m_{WG} \cdot x_{WG,LEMAC} + m_{FG} \cdot \left(x_{FG} - x_{LEMAC}\right) \qquad \textbf{(10.23)}$$

$x_{CG,LEMAC}$ The distance from the LE on the MAC (LEMAC) of the entire aircraft up to the CG. $x_{CG,LEMAC}$ is predefined, as required, e.g. $x_{CG,LEMAC} = 0.25 \cdot c_{MAC}$,

m_{WG} Mass of wing group,

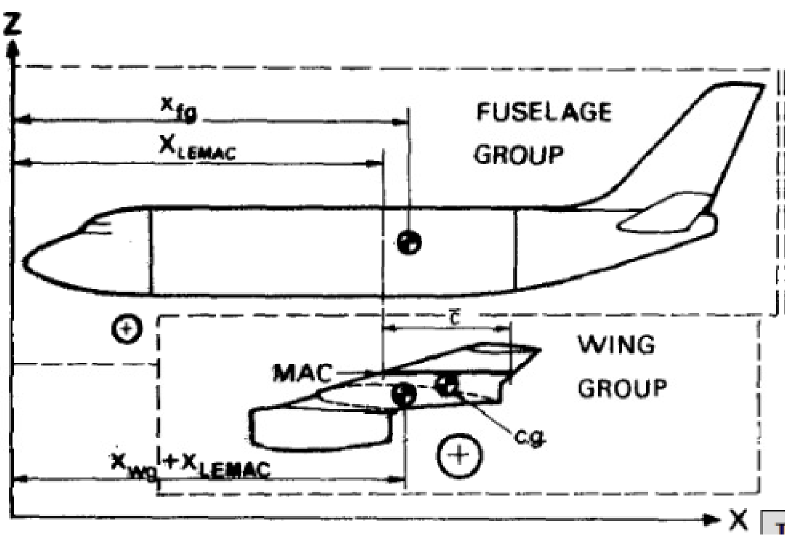
m_{FG} Mass of fuselage group,

$x_{WG,LEMAC}$ Distance from LEMAC to CG of the wing group,

x_{FG} Distance from zero point to CG of fuselage group,

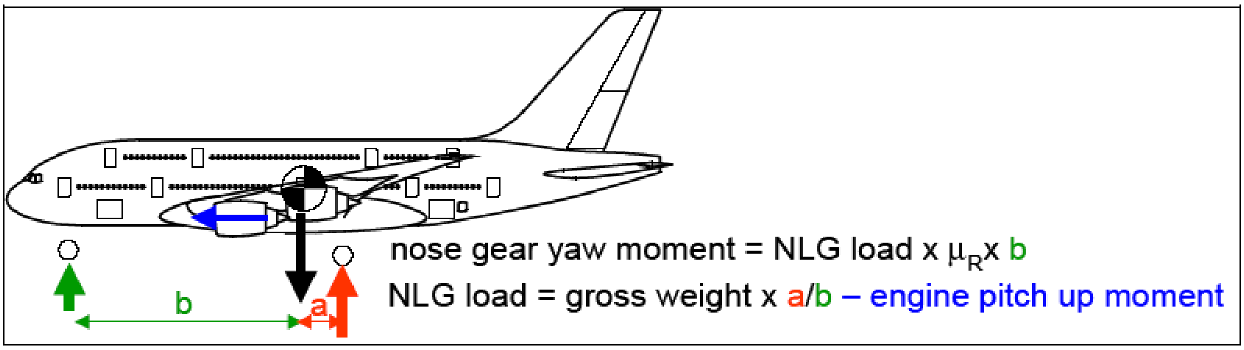
x_{LEMAC} Distance from zero point to LEMAC.

$$x_{LEMAC} = x_{FG} - x_{CG,LEMAC} + \frac{m_{WG}}{m_{FG}}\left(x_{WG,LEMAC} - x_{CG,LEMAC}\right) \ . \qquad \textbf{(10.24)}$$

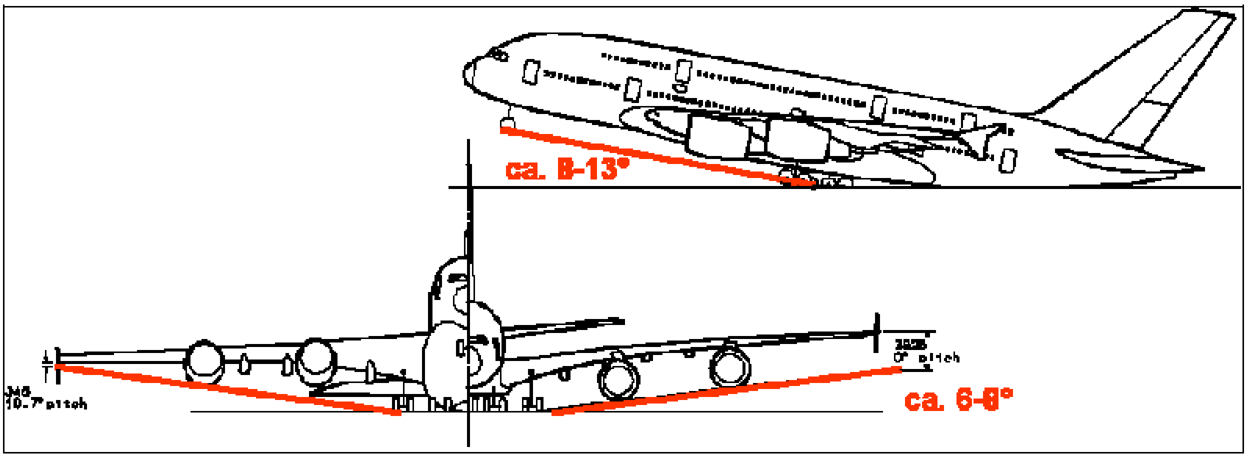


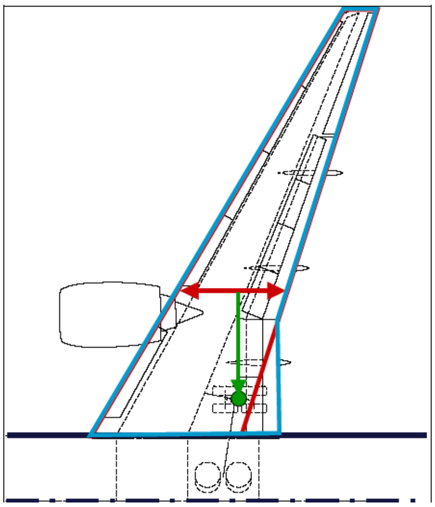
12 Landing Gear Integration

Nose gear loads need to be sufficiently high (5% to 10 % of total aircraft weight) and determine the **position of the main landing gear**:



These clearance angles determine the **length of the landing gear**:





A **kinked wing** trailing edge provides space for the integration of a wing mounted landing gear in case of an aft swept wing. The main landing gear is positioned aft of the center of gravity. Also the wing is positioned with respect of the center of gravity.

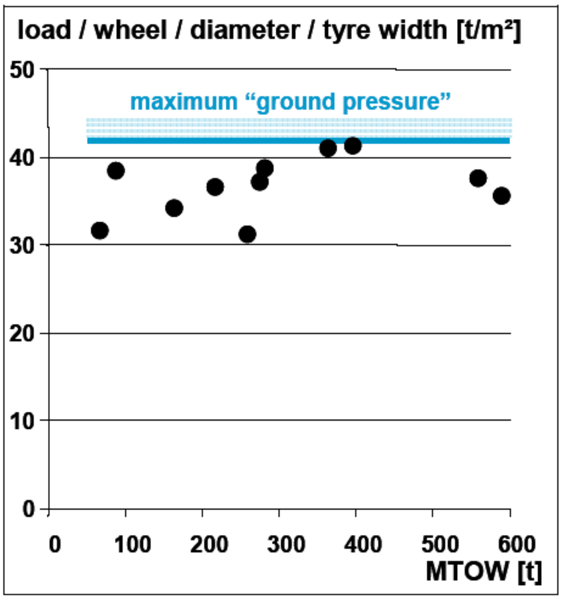
Number of main landing gear wheels:

$n_{MLG} = m_{MTO}$ / 30 t for large long range aircraft

$n_{MLG} = m_{MTO}$ / 20 t for smaller aircraft operating from smaller airports

Tire pressure:
Aircraft tires are filled with nitrogen gas at pressure up to 15bar.

Tire sizing for width and diameter:
Aircraft tires have a ratio between width, w and diameter, d of about $w/d = 0.35 \ldots 0.40$.



The equivalent ground pressure p^* a tire can carry depends on it's with and diameter and is between 32 t/m² and 42 t/m2

$$p^* = m_{MTO} / (n_W \, d \, w)$$

$$d \cdot w = m_{MTO} / (n_W \, p^*)$$

$$d = \sqrt{\frac{m_{MTO}}{n_W \;\; p^* \, w/d}} \qquad w = w/d \cdot d$$

13 Drag Prediction

$$C_D = C_{D0} + \frac{C_L{}^2}{\pi \cdot A \cdot e}$$

$$C_D = C_{D,0} + \Delta C_{D,flap} + \Delta C_{D,slat} + \Delta C_{D,gear} + \Delta C_{D,wave} + \frac{C_L{}^2}{\pi \cdot A \cdot e}$$

e	0.7	due to extended flaps and slats

e	0.85 or
e	more precisely calculated from *Appendix A of Aircraft Design Lecture Notes*

$\Delta C_{D,slat}$	negligible	
$\Delta C_{D,gear}$	0.015	in case landing gear is extended

$$C_L = C_{L,max} \left(\frac{V_s}{V} \right)^2$$

V / Vs = 1.2 for take-off and initial climb
V / Vs = 1.3 for approach and landing

$$\Delta C_{D,flap} = 0.05\ C_L - 0.055$$
$$\text{for } C_L \geq 1.1 \quad .$$

Estimating C_{D0} from E_{max}

$$E_{max} = k_E \sqrt{\frac{A}{S_{wet} / S_W}}$$

$$S_{wet} / S_W = 6.0 \ldots 6.2$$

$$\boxed{C_{D,0} = \frac{\pi \cdot A \cdot e}{4 \cdot E_{\max}^2}}$$

k_E calculated:

$$k_E = \frac{1}{2}\sqrt{\frac{\pi\ e}{\overline{\overline{c_f}}}}$$

k_E given:

k_E 14.9 when calculated for standard parameters (e = 0.85, C_f = 0.003)
k_E 15.8 according to data in Raymer's book
k_E 15.15 short range aircraft
k_E 16.19 medium range aircraft
k_E 17.25 long range aircraft

Estimating C_{D0} from wetted area

$$C_{D,0} = C_{fe} \cdot \frac{S_{wet}}{S_W}$$

$$S_{wet} = S_{wet,F} + S_{wet,W} + S_{wet,H} + S_{wet,V} + n_E \cdot S_{wet,N} + n_E \cdot S_{wet,pylons}$$

Table 13.1 The equivalent skin-friction drag coefficient C_{fe} on the basis of general experience (**Roskam I**)

aircraft type	C_{fe} **- subsonic**
jets	0.003 ... 0.004
twins	0.004 ... 0.007
singles	0.005 ... 0.007

Estimating C_{D0} from drag built up

=> see: *Aircraft Design Lecture Notes, Section 13*

$$C_{D0} = \sum_{c=1}^{n} C_{f,c} \cdot FF_c \cdot Q_c \cdot \frac{S_{wet,c}}{S_{ref}}$$

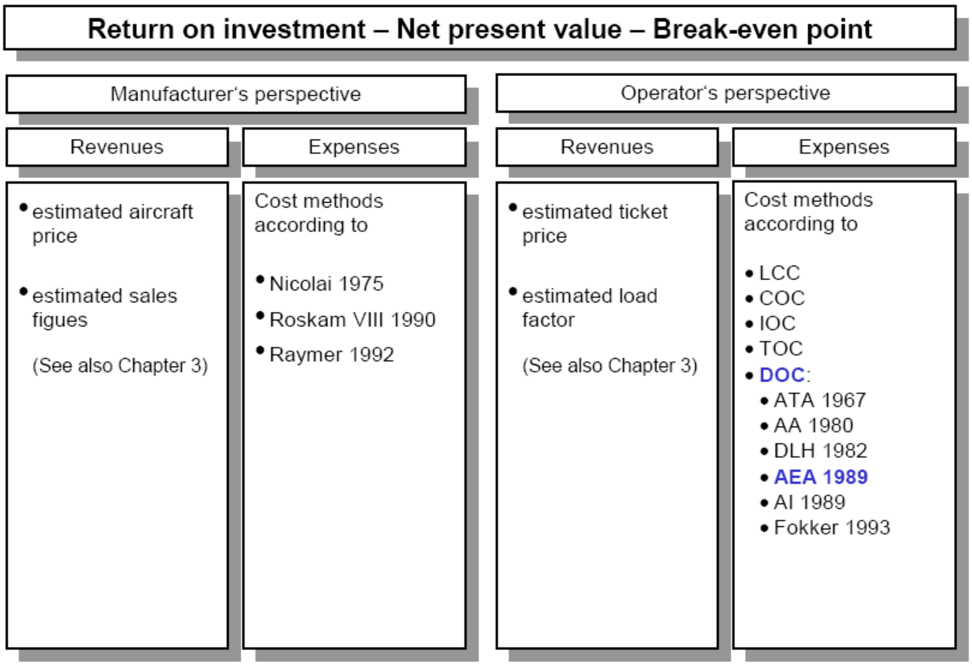
Calculating the glide ratio, *E*

$$E = \frac{2\,E_{max}}{\dfrac{1}{\left(\dfrac{C_L}{C_{L,md}}\right)} + \left(\dfrac{C_L}{C_{L,md}}\right)}$$

$$C_{L,md} = \frac{\pi\,A\,e}{2E_{max}}$$

$$C_L / C_{L,md} = 1/\left(V/V_{md}\right)^2$$

14 Design Evaluation / DOC



$$C_{DOC} = C_{DEP} + C_{INT} + C_{INS} + C_F + C_M + C_C + C_{FEE}$$

For details see *Aircraft Design Lecture Notes*, Section 14.

$$C_{equiv,t,m} = \frac{C_{a/c,t}}{(m_{pax} + m_{baggage} + k_{cargo,CMD}\; m_{cargo,CMD} + k_{cargo,CLD}\; m_{cargo,CLD} + k_{cargo,B}\; m_{cargo,B})\; R}$$

correction factor	type of freight	**DLH 1982**
$k_{cargo,CMD}$	containerized, main deck	1.0
$k_{cargo,CLD}$	containerized, lower deck	0.8
$k_{cargo,B}$	bulk	0.5

$$U_{a,f} = t_f \frac{k_{U1}}{t_f + k_{U2}}$$

$$U_{a,f} = t_f \; n_{t,a}$$

source	k_{U1} h	k_{U2} h
AA 1980 / NASA 77	3205	0.327
AEA 1989a	3750	0.750
AEA 1989b	4800	0.420