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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 $\gamma_{CLB}$ (2nd segment)
- Missed approach climb gradient $\gamma_{MA}$.

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%.

\[
\text{load factor} = \frac{\text{sold payload}}{\text{offered payload}}
\]

\[
\text{range flexibility} = \frac{\text{aircraft range}}{\text{demanded flight distance}}
\]

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>
<thead>
<tr>
<th>aircraft type</th>
<th>characteristics</th>
<th>airworthiness standard</th>
<th>interpretative material</th>
</tr>
</thead>
<tbody>
<tr>
<td>normal, utility and aerobatic</td>
<td>passenger seats = 9 MTOW = 5700 kg</td>
<td>CS-23, FAR Part 23</td>
<td>FAR: Advisory Circular</td>
</tr>
<tr>
<td>aeroplanes</td>
<td>passenger seats = 19 MTOW = 8000 kg propeller driven twin-engined</td>
<td></td>
<td>AC23-7</td>
</tr>
<tr>
<td>commuter aeroplanes</td>
<td>MTOW &gt; 5700 kg</td>
<td></td>
<td></td>
</tr>
<tr>
<td>large aeroplanes (JAR)</td>
<td>transport category airplanes (FAR)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>EASA: Advisory Circular Joint (ACJ)</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Advisory Material Joint (AMJ) included in CS-25</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>FAR: Advisory Circular</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>AC25-7</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
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
Estimation of operating empty mass, OEW

\[ \frac{m_{OEW}}{m_{MTO}} = 0.23 + 1.04 \cdot \frac{T_{TO}}{m_{MTO} \cdot g} \]  

Estimation of fuel mass, \( m_F \)

\[ \frac{m_F}{m_{TO}} = 1 - M_{ff} \]  

Mission fuel fraction:

\[ M_{ff} = \frac{m_{SO}}{m_T} \cdot \frac{m_T}{m_L} \cdot \frac{m_{DE}}{m_{DE}} \cdot \frac{m_{CR, alt}}{m_{CR, alt}} \cdot \frac{m_{CLB}}{m_{CLB}} \cdot \frac{m_{MA}}{m_{MA}} \cdot \frac{m_{DES}}{m_{DES}} \cdot \frac{m_{LOI}}{m_{LOI}} \cdot \frac{m_{CR}}{m_{CR}} \cdot \frac{m_{CLB}}{m_{CLB}} \cdot \frac{m_{TO}}{m_{TO}} \]  

\( SFC_T = 16 \text{ mg/(Ns)} \)

\[ B_z = \frac{L}{D \cdot V} \cdot \frac{g}{SFC_T} \cdot \frac{m_{LOI}}{m_{CR}} = e^{-\frac{L}{B_z}} \]  

**Table 5.9**  
Generic mission segment mass fractions (based on Roskam 1)

<table>
<thead>
<tr>
<th>type of aircraft</th>
<th>engine start</th>
<th>taxi</th>
<th>take-off</th>
<th>climb</th>
<th>descent</th>
<th>landing</th>
</tr>
</thead>
<tbody>
<tr>
<td>business jet</td>
<td>0.99</td>
<td>0.995</td>
<td>0.995</td>
<td>0.98</td>
<td>0.99</td>
<td>0.992</td>
</tr>
<tr>
<td>jet transport</td>
<td>0.99</td>
<td>0.99</td>
<td>0.995</td>
<td>0.98</td>
<td>0.99</td>
<td>0.992</td>
</tr>
</tbody>
</table>

\[ T_{TO} = m_{MTO} \cdot g \cdot \left( \frac{T_{TO}}{m_{MTO} \cdot g} \right) \]

\[ S_w = m_{MTO} / \left( \frac{m_{MTO}}{S_w} \right) \]

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), \( d_{F,I} \):

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

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), $\Delta d$:

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

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

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

**Number of rows**:

$$n_R = \frac{n_{PAX}}{n_{SA}}$$

**Cabin length**:

$$l_{cabin} = n_R \cdot 1 \, \text{m}$$

**Fuselage shape of a passenger aircraft**:

<table>
<thead>
<tr>
<th>$L_{Bug}/D_{eff}$</th>
<th>$D_{eff} = \sqrt{H \cdot D}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.7</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>$L_{Heck}/D_{eff}$</th>
<th>$L_{KabE}/D_{eff}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.5</td>
<td>1.9</td>
</tr>
</tbody>
</table>

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

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

$$\varphi_{tail} = \arctan \left( \frac{d_F}{l_{stern}} \right)$$  
$$\varphi_{tail} = \arctan \left( \frac{1}{3.5} \right) = 15.9^\circ$$

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

$\varphi_{tail}$ is also the maximum angle for rotation at take-off.  
$\varphi_{tail}$ is also the aircraft’s angle of attack, $\alpha$ at take-off.
Fuselage length:

\[ l_F = l_{cabin} + l_{cockpit} + l_{tail} = l_{cabin} + 4\, m + 1.6 \cdot d_F \]

Checking for sufficient size of cargo compartment:

\[ V_{CC} \geq V_C + (V_B - V_{OS}) \]

\( V_{CC} \)  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.

\[ V_B = m_B / \rho_B \]
\[ V_C = m_C / \rho_C \]
\[ V_{OS} = S_{OS,iat} \cdot l_{OS} \]
\[ S_{OS,iat} = n_{OS,iat} \cdot S_{OS,iat} + n_{OS,ce} \cdot S_{OS,ce} \]
\[ l_{OS} = k_{OS} \cdot l_{cabin} \]

Density of baggage, \( \rho_B \) and cargo, \( \rho_C \):

**Baggage:** 170 kg/m³,
**Cargo:** 160 kg/m³.
\[ m_B \] mass of baggage,
\[ m_C \] mass of cargo,
\[ \rho_B \] density of baggage,
\[ \rho_C \] 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

<table>
<thead>
<tr>
<th>( n_{OS} )</th>
<th>Values for the ( S_{OS,tot}, S_{OS,ce} ), and ( k_{OS} ) for selected aircraft with 1 or 2 aisles</th>
<th>( k_{OS} )</th>
<th>( S_{OS,lat} )</th>
<th>( S_{OS,ce} )</th>
<th>( \rho_B )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Single aisle:</td>
<td>Average</td>
<td>0.723</td>
<td>0.201</td>
<td>-</td>
<td>180.13</td>
</tr>
<tr>
<td>Twin aisle:</td>
<td>Average</td>
<td>0.751</td>
<td>0.208</td>
<td>0.241</td>
<td>185.01</td>
</tr>
<tr>
<td></td>
<td>Overall average</td>
<td>0.737</td>
<td>0.213</td>
<td>-</td>
<td>182.57</td>
</tr>
</tbody>
</table>

\( S_{OS} \) given in m², \( \rho_B \) is the maximum allowed baggage density in the bin.


**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:

\[ \varphi_{25} = 39.3^\circ \left( M_{CR} \right)^2 \]

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

\[ \left( \frac{t}{c} \right) = -0.0439 \cdot \tan^{-1} \left( 3.3450 \cdot M_{CR} - 3.0231 \right) + 0.0986 \]

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

Airbus and Boeing use the following design Mach number

\[ M_{\text{design}} = M_{CR} \]

\[ M_{DD} = M_{CR} \]

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

\[k_M = 1.00\]  
conventional airfoils; maximum \( t/c \) at about 0.30c,

\[k_M = 1.05\]  
high-speed (peaky) airfoils, 1960-1970 technology,

\[k_M = 1.12\text{ to } 1.15\]  
supercritical airfoils.

\[C_L:\]  
the design lift coefficient (for cruise) chosen in Section 5

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

\[(t/c)_r / (t/c)_t = r \approx 1.3\]

\[(t/c)_t = 4/(3+r) \ t/c\]

\[(t/c)_r = r (t/c)_t\]

Optimum taper ratio, \( \lambda_{opt} \):

\[\lambda_{opt} = 0.45 \cdot e^{-0.036 \phi_{25}}\]

\( \phi_{25} \) in degree

\( \lambda \) 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:

\[\varepsilon_t = i_{w,tip} - i_{w,root}\]

\( \varepsilon_t = -3^\circ \) (wash out). However the A310 (see Subsection 7.5) shows \( \varepsilon_t = -8^\circ \).
**Incident angle** of the wing (at the wing root):

\[ i_w = \frac{C_{L,CR}}{C_{\tau_w}} + \alpha_0 - 0.4 \cdot \varepsilon_1. \]

In this equation:
- \( C_{\tau_w} \) the lift curve slope according to equation (7.24),
- \( C_{L,CR} \) the necessary lift coefficient in cruise flight,
- \( \alpha_0 \) the angle of attack at zero wing lift or a characteristic profile of the wing,
- \( \varepsilon_1 \) the twist (see above).

\[ C_{L,\alpha} = \frac{2 \cdot \pi \cdot A}{2 + \sqrt{A^2 \cdot (1 + \tan^2 \psi_{s0} - M^2) + 4}} \]

**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

\[ \frac{\partial \Gamma}{\partial k_{Z,W}} = -7.46^\circ \]
\[ \frac{\partial \Gamma}{\partial \varphi_{25}} = -0.115 \]
\[ \Gamma_0 = 6.91^\circ \]

**Tank volume** within the wing (without center tank):

\[ V_{\text{tank}} = 0.54 \cdot S_W^{1.5} \cdot (t / c)_{r} \cdot \frac{1}{\sqrt{A}} \cdot \frac{1 + \lambda \cdot \sqrt{\tau} + \lambda^2 \cdot \tau}{(1 + \lambda)^2} \]

\[ \tau = \frac{(t / c)_{r}}{(t / c)_{t}} \]

8 Design of Highlift Systems

\[ C_{L,\text{max}} = 1.1 \cdot C_{L,\text{max, INITIAL SIZING}} \]

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

\[ 0.95 \cdot \Delta C_{L,\text{max,f}} + \Delta C_{L,\text{max,e}} \geq C_{L,\text{max}} - C_{L,\text{max,clean}} \]

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

\[ S_H = \frac{C_H S_W c_{MAC}}{l_H} \quad S_V = \frac{C_V S_W b}{l_V} \]

- \( l_H \): 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>
<thead>
<tr>
<th>Table 9.4</th>
<th>Conventional tail volume coefficients of horizontal and vertical tails (Raymer 1989)</th>
</tr>
</thead>
<tbody>
<tr>
<td>type</td>
<td>( vertical \ C_V )</td>
</tr>
<tr>
<td>General Aviation - Twin Engine</td>
<td>0.07</td>
</tr>
<tr>
<td>Transport Jets</td>
<td>0.08</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Table 9.5:</th>
<th>Conventional tail lever arms of horizontal and vertical tails (Raymer 1998)</th>
</tr>
</thead>
<tbody>
<tr>
<td>aircraft configuration</td>
<td>average of ( l_H ) and ( l_V )</td>
</tr>
<tr>
<td>propeller in front of fuselage</td>
<td>60% of fuselage length</td>
</tr>
<tr>
<td>engines on the wing</td>
<td>50 ... 55% of fuselage length</td>
</tr>
<tr>
<td>engines on the tail</td>
<td>45 ... 50% of fuselage length</td>
</tr>
</tbody>
</table>

- 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

| Flügel (wing) \( m_F \) & + Rumpf (fuselage) \( m_r \) & + Höhenleitwerk (horizontal tail) \( m_H \) & + Seitenleitwerk (vertical tail) \( m_v \) & + Bugfahrwerk (nose landing gear) \( m_{L_G,N} \) & + Hauptfahrwerk (main landing gear) \( m_{L_G,H} \) & + Triebwerksgondel (nacelle) \( m_N \) |
|-----------------|-----------------|-----------------|-----------------|-----------------|-----------------|-----------------|
| Struktur (structure) & + Triebwerk, installiert (power plant, installed) \( m_{E,\text{inst}} \) & + Flugzeugsysteme (aircraft systems) \( m_{SS} \) |
| Hersteller-Leermasse (\( \Rightarrow \) manufacturer’s empty weight, MEW) \( m_{ME} \) & + Ausrüstung und Besatzung (\( \Rightarrow \) standard and operational items) |
| Betriebsleermasse (\( \Rightarrow \) 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 \( M_g \) according to Step 5.9.2:

\[
m_{MTO} = \frac{m_{ME} + m_{OE}}{M_g} .
\]

**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 \( m_{MTO} \) from which the wing area \( S_F \) and take-off thrust \( T_{TO} \) were calculated, \( S_F \) 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.
- The 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.

\[
(m_{WG} + m_{FG}) \cdot x_{CG,LEMAC} = m_{WG} \cdot x_{WG,LEMAC} + m_{FG} \left( x_{FG} - x_{LEMAC} \right) \tag{10.23}
\]

\( x_{CG,LEMAC} \) 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.

\[
x_{LEMAC} = x_{FG} - x_{CG,LEMAC} + \frac{m_{WG}}{m_{FG}} \left( x_{WG,LEMAC} - x_{CG,LEMAC} \right) \tag{10.24}
\]
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:

\[ \text{nose gear yaw moment} = \text{NLG load} \times \mu_R \times b \]
\[ \text{NLG load} = \text{gross weight} \times \frac{a}{b} - \text{engine pitch up moment} \]

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:

\[ n_{MLG} = \frac{m_{MTO}}{30 \, t} \quad \text{for large long range aircraft} \]

\[ n_{MLG} = \frac{m_{MTO}}{20 \, t} \quad \text{for smaller aircraft operating from smaller airports} \]

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 its width and diameter and is between 32 \( t/m^2 \) and 42 \( t/m^2 \)

\[ p^* = \frac{m_{MTO}}{n_W \, d \, w} \]

\[ d \cdot w = \frac{m_{MTO}}{n_W \, p^*} \]

\[
\begin{align*}
d &= \sqrt[3]{\frac{m_{MTO}}{n_W \cdot p^* \cdot w/d}} \\
w &= w/d \cdot d
\end{align*}
\]
13 Drag Prediction

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

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

\[ e = 0.7 \quad \text{due to extended flaps and slats} \]

\[ e = 0.85 \quad \text{or} \]

\[ e \quad \text{more precisely calculated from} \]

\[ \text{Appendix A of Aircraft Design Lecture Notes} \]

\[ \Delta C_{D,slat} \quad \text{negligible} \]

\[ \Delta C_{D,gear} \quad 0.015 \quad \text{in case landing gear is extended} \]

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

\[ V / V_s = 1.2 \quad \text{for take-off and initial climb} \]

\[ V / V_s = 1.3 \quad \text{for approach and landing} \]

\[ \Delta C_{D,flap} = 0.05 \, C_L - 0.055 \]

\[ \text{for } C_L \geq 1.1 \]

**Estimating \( C_{D0} \) from \( E_{\text{max}} \)**

\[ E_{\text{max}} = k_E \sqrt{\frac{A}{S_{\text{wet}} / S_W}} \]

\[ S_{\text{wet}} / S_W = 6.0 \ldots 6.2 \]

\[ C_{D,0} = \frac{\pi \cdot A \cdot e}{4 \cdot E_{\text{max}}^2} \]
\[ k_E \text{ calculated:} \]
\[ k_E = \frac{1}{2} \sqrt{\frac{\pi e}{c_f}} \]

\[ k_E \text{ given:} \]
\[ k_E = 14.9 \text{ when calculated for standard parameters (e = 0.85, } c_f = 0.003) \]
\[ k_E = 15.8 \text{ according to data in Raymer’s book} \]
\[ k_E = 15.15 \text{ short range aircraft} \]
\[ k_E = 16.19 \text{ medium range aircraft} \]
\[ k_E = 17.25 \text{ long range aircraft} \]

**Estimating } C_{D0} \text{ from wetted area}**

\[ C_{D,0} = C_{f_0} \cdot \frac{S_{\text{wet}}}{S_W} \]

\[ S_{\text{wet}} = S_{\text{wet,F}} + S_{\text{wet,W}} + S_{\text{wet,H}} + S_{\text{wet,N}} + n_E \cdot S_{\text{wet,N}} + n_E \cdot S_{\text{wet,pylons}} \]

**Table 13.1** The equivalent skin-friction drag coefficient } C_{f_0} \text{ on the basis of general experience}

<table>
<thead>
<tr>
<th>aircraft type</th>
<th>( C_{f_0} \cdot \text{subsonic} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>jets</td>
<td>0.003 ... 0.004</td>
</tr>
<tr>
<td>twins</td>
<td>0.004 ... 0.007</td>
</tr>
<tr>
<td>singles</td>
<td>0.005 ... 0.007</td>
</tr>
</tbody>
</table>

**Estimating } C_{D0} \text{ from drag built up}**

\[ C_{D0} = \sum_{c=1}^{n} C_{f,c} \cdot F_{F,c} \cdot Q_c \cdot \frac{S_{\text{wet,c}}}{S_{\text{ref}}} \]

=> see: *Aircraft Design Lecture Notes, Section 13*
Calculating the glide ratio, $E$

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

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

$$\frac{C_L}{C_{L,\text{md}}} = \frac{1}{(V / V_{\text{md}})^2}$$
\( C_{DOC} = C_{DEP} + C_{INT} + C_{INS} + C_F + C_M + C_C + C_{FEE} \)

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

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

<table>
<thead>
<tr>
<th>correction factor</th>
<th>type of freight</th>
<th>DLH 1982</th>
</tr>
</thead>
<tbody>
<tr>
<td>( k_{\text{cargo,CMD}} )</td>
<td>containerized, main deck</td>
<td>1.0</td>
</tr>
<tr>
<td>( k_{\text{cargo,CLD}} )</td>
<td>containerized, lower deck</td>
<td>0.8</td>
</tr>
<tr>
<td>( k_{\text{cargo,B}} )</td>
<td>bulk</td>
<td>0.5</td>
</tr>
</tbody>
</table>
\[ U_{a,f} = t_f \frac{k_{U_1}}{t_f + k_{U_2}} \]

\[ U_{a,f} = t_f \ n_{t,a} \]

<table>
<thead>
<tr>
<th>source</th>
<th>( k_{U_1} )</th>
<th>( k_{U_2} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>AA 1980 / NASA 77</td>
<td>3205</td>
<td>0.327</td>
</tr>
<tr>
<td>AEA 1989a</td>
<td>3750</td>
<td>0.750</td>
</tr>
<tr>
<td>AEA 1989b</td>
<td>4800</td>
<td>0.420</td>
</tr>
</tbody>
</table>