Hochschule für Angewandte Wissenschaften Hamburg
Hamburg University of Applied Sciences

Aircraft Design Studies Based on the ATR 72

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Two Design Steps

– **Preliminary sizing**

(Paper on RRDPAE CD)

– **Conceptual Design**

(Master Thesis on WWW)

Emphasis of this presentation

Overview

- **Preliminary sizing**

– **Gives input parameters for the conceptual design:**
  » **Maximum take-off mass,** m_{MTO}
  » **Fuel mass,** m_F
  » **Maximum operating empty mass,** m_{OE}
  » **Wing area,** S_W
  » **Take-off thrust,** T_{TO} **or take-off power,** P_{TO}

Overview

- **Conceptual design**

Fuselage → Wing → Empennage → Mass and CG location

Stability and control → Landing gear → Drag and polar → Direct Operation Costs (DOC)

- **The Fuselage**
  - **Requirements:**
    - » **Passengers comfort**
    - » **Drag**
    - » **Weight**
  - **Cross section:**
    - » **Given: Number of passengers** $n_{PAX} = 70$
    - » **Yields: Number of seats abreast** $n_{SA} = 0.45 \cdot \sqrt{n_{PAX}} = 4$

      and number of aisles $n_{SA} \leq 6 \Rightarrow 1\ Aisle$

      (CS 25.817)

Fuselage

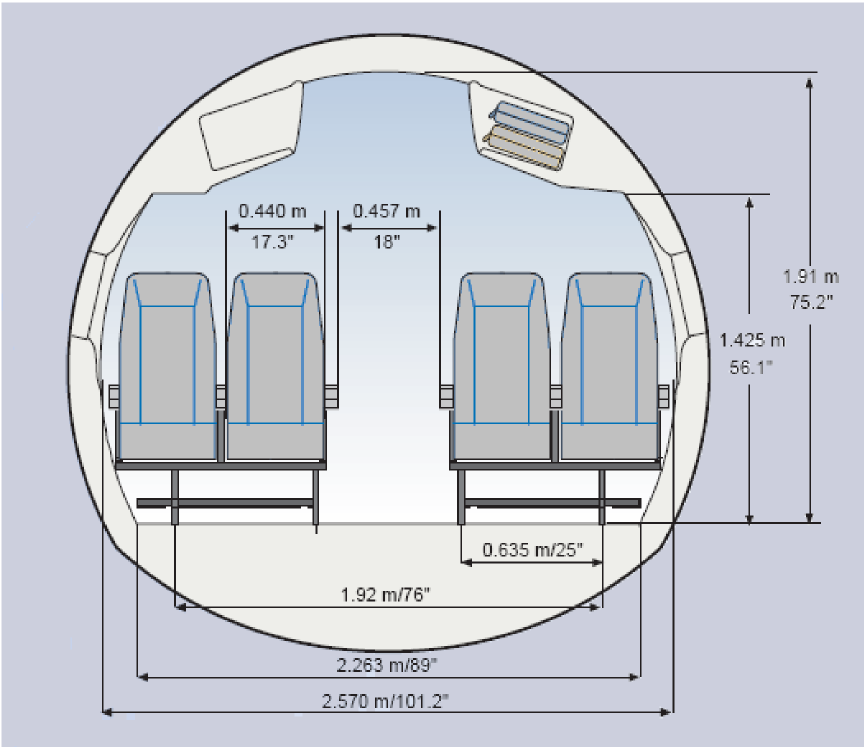
» **Interior diameter of the fuselage**

$$d_{F,I} = (2 \times Bench\ width + Aisle\ width)m + 2 \times 0.025m = 2.57m$$

Empirical Equation

» **Exterior diameter of the fuselage**

$$\Delta d = d_{F,O} - d_{F,I} = 0.084m + 0.045 \cdot d_{F,I} \Leftrightarrow d_{F,O} = 2.77m$$



Fuselage

– **Cabin and fuselage:**

» **Seat pitch:**

$$31\,\text{in} = 0,78\,\text{m};$$

$$k_{CABIN} = 1\,\text{m}$$

An average seat pitch including galleys,lavatories

» **Cabin Length**

$$l_{CABIN} = k_{CABIN} \cdot \frac{n_{PAX}}{n_{SA}} = 19.25m$$

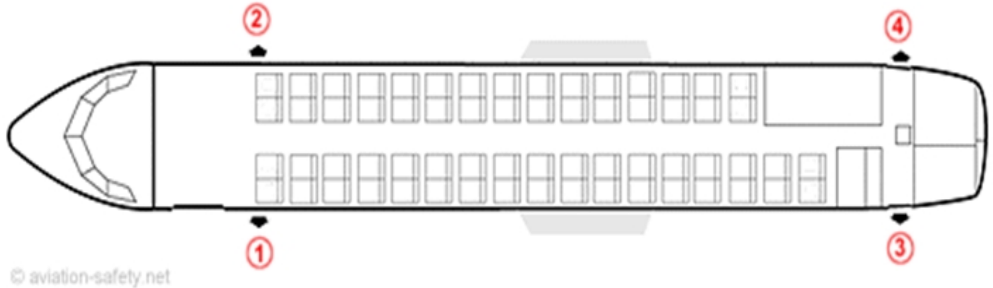
Cockpit length

» **Fuselage length**

$$l_F = l_{CABIN} + 1.4 \cdot d_F + 4m = 27.13m$$

Tail length

» **Emergency exits: 2+2 type I and III**



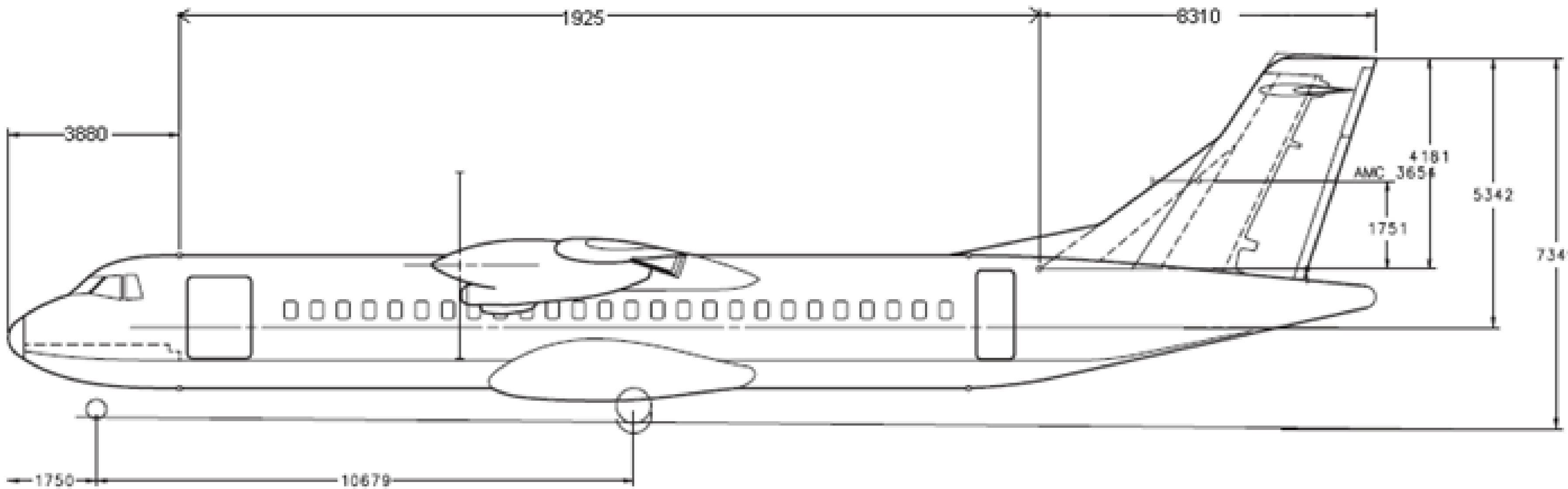
① emergency exit 0.91m x0.51m
② emergency exit 0.91m x 0.51m
③ pax door 1.75m x 0.82m
④ service door 1.22m x 0.61m

– **Other parameters:**

» **Slenderness parameter**

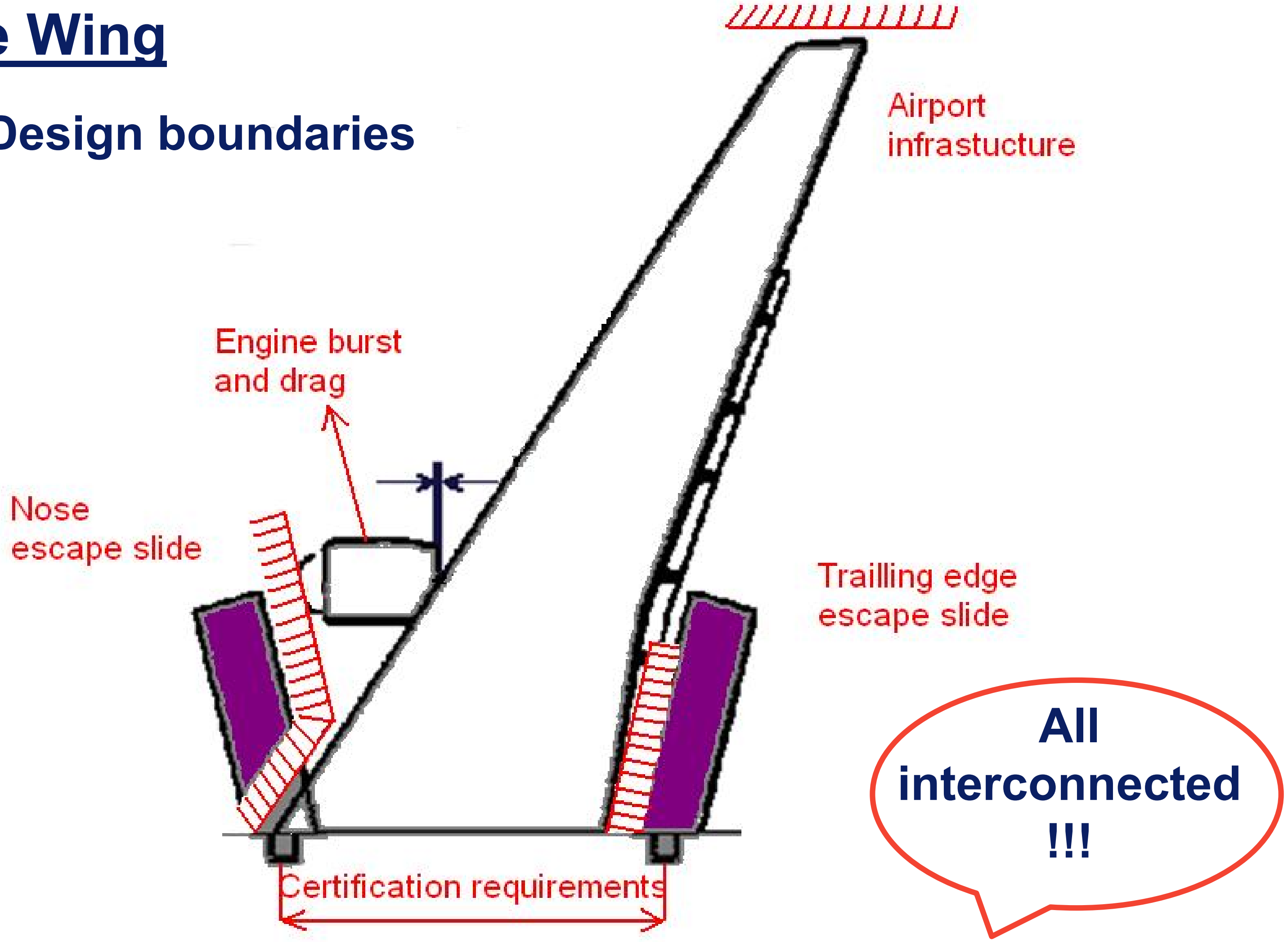
$$\lambda_F = \frac{l_F}{d_F} = 9.79 \approx 10$$

! **Important parameter that determines drag and structural weight**



Lateral view and dimensions

- **The Wing**
  - **Design boundaries**



– Design method

- Cruise Mach number, M_{CR}
- Wing area, S_W
- Aspect ratio, A

From known parameters

Calculate new parameters

- 1) Sweep φ_{25}
- 2) Airfoil family
- 3) Airfoil thickness
- 4) Incidence angle, i_w
- 5) Wing twist, ε_w
- 6) Dihedral, v_w
- 9)Taper ratio, λ
- 7) Span, b
- 8) Root and tip chord, c_t, c_r
- 9) Lift curve slope, $dC_L/d\alpha$
- 9) Tank volume, V_{tank}

- Fuel fits in the tank volume?

And verify results

– Results

$S_W = 61.3 m^2$, $A = 12$ → $b = 27.13 m$

$M_{CR} = 0.41$ → $\varphi_{25} = 3^0$

Chosen airfoil: NACA 430xx

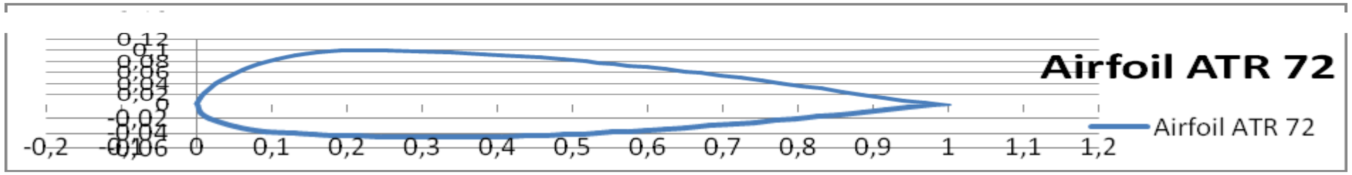
$$(t/c) = k_t \cdot M_{DD}{}^t \cdot \cos\varphi_{25}{}^u \cdot C_L{}^v \cdot k_M{}^w = 0.141$$

Non linear regression

→ $(t/c)_r = 18\%$

$(t/c)_t = 13\%$

NACA 43018
NACA 43013



$\varepsilon_t = -3^0$

Abbot, Pankhurst

$$i_w = \frac{C_{L,CR}}{C_{L_\alpha}} + \alpha_0 - 0.4 \cdot \varepsilon_t = 4^0$$

→ $\nu_W = 0^0$ →

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

$\lambda = c_t / c_r = 0.59$

↓

$$c_r = \frac{2b}{A[(1-\lambda)\eta_k + \lambda_i + \lambda]} = 2.6m$$

$$c_t = \lambda c_r = 1.5m$$

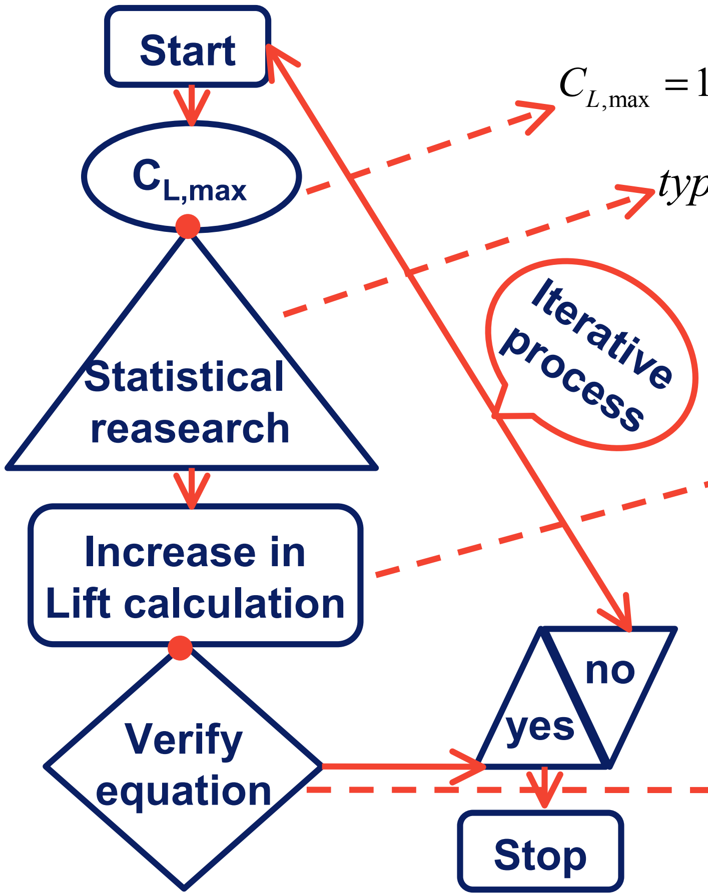
↓

$$V_{\text{tank}} = 0.54 \cdot S_W^{\,1.5} \cdot (t/c)_r \cdot \frac{1}{\sqrt{A}} \cdot \frac{1+\lambda \cdot \sqrt{\tau} + \lambda^2 \cdot \tau}{(1+\lambda)^2} = 9.3m^3 > V_{\text{tank,nec}} = 4.5m^3$$

where $\tau = \frac{(t/c)_t}{(t/c)_r} = 0.72$

From preliminary sizing

- **The high lift system**
  - **Design method**



$$C_{L,\max} = 1.1 \cdot C_{L,\max,INITIAL\ SIZING} = 2.2$$

$$type\ of\ flap: double\ slotted\ flap\ and\ slats$$

$$\Delta c_{L,\max,f} = k_1 k_2 k_3 (\Delta c_{L,\max})_{base} = 1.089$$

$$\Delta C_{L,\max,f} = \Delta c_{L,\max,f} \cdot \frac{S_{W,f}}{S_W} \cdot K_\varphi = 0.679$$

$$\Delta c_{L,\max,s} = c_{l,\delta,\max} \eta_{\max} \eta_\delta \delta_f \frac{c'}{c} = 0.766$$

$$\Delta C_{L,\max,s} = \Delta c_{L,\max,s} \cdot \frac{S_{W,s}}{S_W} \cdot \cos\varphi_{H,L} = 0.613$$

DATCOM

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

$$1.3 \geq 0.9$$

- **Estimating the empennage area from statistics**

  - Horizontal tail, vertical tail
  - Configuration: T-tail

    (engine location on a high wing)
  - Surface area from statistical approach

    - Tail volume coefficient
    - Lever arm → $l_V = l_H = 50\% \cdot l_F = 13.565 m$

Tail volume

$$C_H = \frac{S_H \cdot l_H}{S_W \cdot c_{MAC}} = 1.05 \qquad C_V = \frac{S_V \cdot l_V}{S_W \cdot b} = 0.119$$

Tail volume coefficient

$$S_H = \frac{C_H \cdot S_W \cdot c_{MAC}}{l_H} = 10.756 m^2$$

$$S_V = \frac{C_V \cdot S_W \cdot b}{l_V} = 14.904 m^2$$

Results

- Other parameters
  - **Aspect ratio and taper ratio:**

    $A_H = 0.5 A_w = 6$ $\lambda_H = 0.6$

    $A_V = 1.6$ $\lambda_V = 0.6$

  - **Dihedral and sweep:**

    $V_H = 80^0$ $\varphi_{25,H} = 8^0$

    $V_V = 0^0$ $\varphi_{25,V} = 25^0$

  - **Airfoil:** NACA 0012 **for the vertical tailplane**

    NACA 0009 **for the horizontal tailplane**

- **Mass estimation and CG location**
  - **Estimation per each component using a Class II method (Torenbeek)**
  - **Example calculation: wing mass**

$$\frac{m_W}{m_{MZF}} = 6.67 \cdot 10^{-3} \cdot b_s^{\ 0.75} \cdot \left(1 + \sqrt{\frac{b_{ref}}{b_s}}\right) \cdot n_{ult}^{\ 0.55} \cdot \left(\frac{b_s / t_r}{m_{MZF} / S_W}\right)^{0.30} = 0.17$$

$$m_W = 0.17 \cdot m_{MZF} = 3045 kg$$

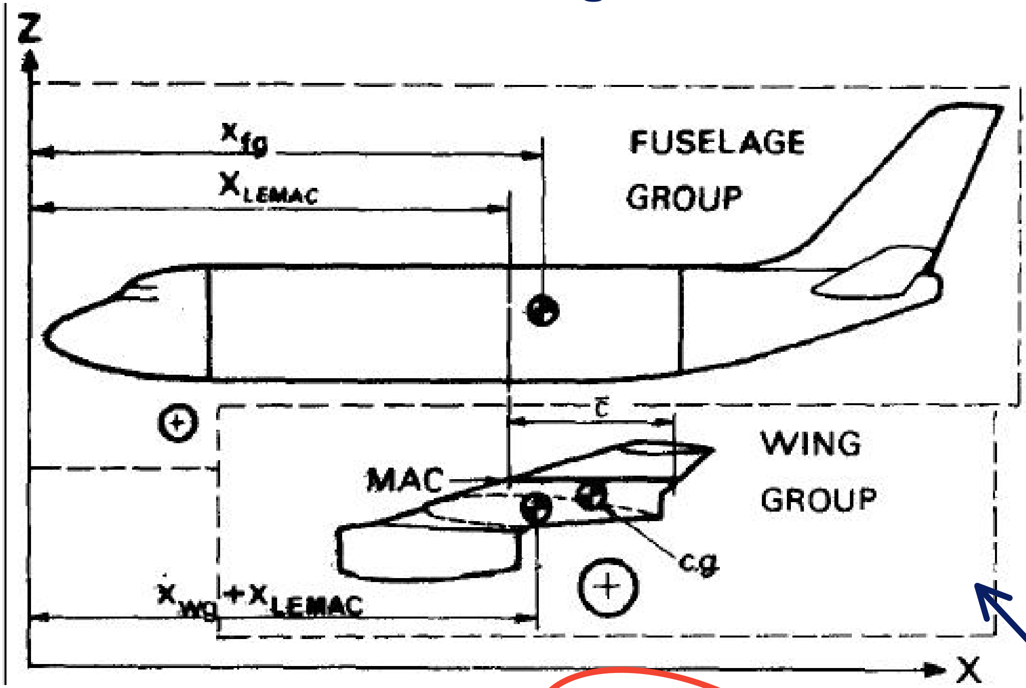
  - **The approximations are made by taking into account variations with specific parameters, as it is shown in the next table**

Mass and CG

	Parameters used for the mass estimation	Results [kg]
Wing	B_{ref}/b_s; m_{MZF}/S_W;n_{ult}	3045
Fuselage	$S_{wet,F}$; l_H; V_D; d_F	2323
Horizontal Tailplane	S_H; V_D	124
Vertical Tailplane	S_V; V_D	179
Landing gear	m_{MTO} and coefficients	961
Engine nacelle	T, respectively P,η,V	242
Installed engine	n_E; m_E	1533
Systems	m_{MTO}	3114
Supplemental mass	n_{Seat}; n_{Pax}	1050
Operating empty mass	Sum of components	12834

Mass and CG

– CG position and position of the wing towards the fuselage



- CG position of the wing

$$x_{WG,LEMAC} = \frac{\sum m_i \cdot x_i}{\sum m_i} = 11.625m$$

-CG position of the fuselage

$$x_{FG} = \frac{\sum m_i \cdot x_i}{\sum m_i} = 11.392m$$

TORENBEEK, E.:
"Synthesis of Subsonic Airplane Design"
Delft University Press, 1988

Position of the wing:

Equilibrium of moments

$$x_{LEMAC} = x_{FG} - x_{CG,LEMAC} + \frac{m_{WG}}{m_{FG}}\left(x_{WG,LEMAC} - x_{CG,LEMAC}\right) = 11m$$

25%C_{MAC}

- **Sizing the empennage according to stability and control requirements**
  - **Horizontal Tail**
    - **Sizing after control requirements** $S_H / S_W = a \cdot \overline{x_{CG-AC}} + b$

$$a = \frac{C_L}{C_{L,H} \cdot \eta_H \cdot \frac{l_H}{c_{MAC}}} = -0.4887 \qquad b = \frac{C_{M,W} + C_{M,E}}{C_{L,H} \cdot \eta_H \cdot \frac{l_H}{c_{MAC}}} = 0.20768$$

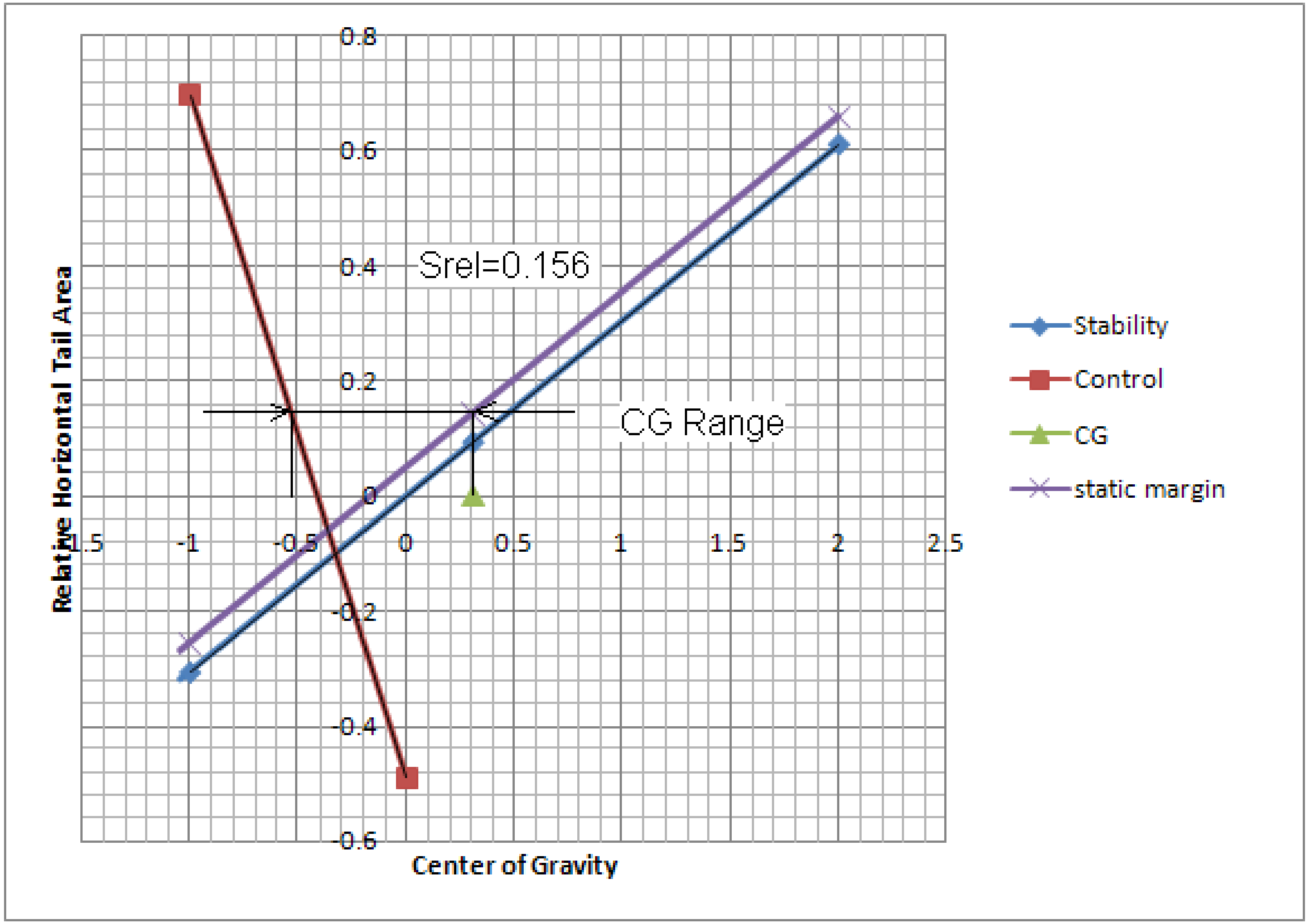
    - **Sizing after stability requirements** $S_H / S_W = a \cdot \overline{x_{CG-AC}}$

$$a = \frac{C_{L,\alpha,W}}{C_{L,\alpha,H} \cdot \eta_H \cdot \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right) \cdot \left(\frac{l_H}{c_{MAC}}\right)} = 0.305$$

    - **Intersection of requirements**

$$\frac{S_H}{S_W} = 0.156 \Rightarrow S_H = 9.701 m^2$$

      - **Following the introduction of the stability margin, according to the next graph**


0.8
0.6
0.4
0.2
0
-0.2
-0.4
-0.6
-1.5
-1
-0.5
0
0.5
1
1.5
2
2.5
Srel=0.156
CG Range
Relative Horizontal Tail Area
Center of Gravity
Stability
Control
CG
static margin

– Vertical Tail

- **Sizing after control requirements**

$$S_V = \frac{N_E + N_D}{\frac{1}{2}\rho V_{MC}^{\ 2} \cdot \delta_F \left[\frac{c_{L,\delta}}{(c_{L,\delta})_{theory}} \right] \cdot (c_{L,\delta})_{theory} \cdot K' \cdot K_\Lambda \cdot l_V} = 14.085 m^2$$

- **Sizing after stability requirements**

$$\frac{S_V}{S_W} = \frac{C_{N,\beta} - C_{N,\beta,F}}{-C_{Y,\beta,V}} \cdot \frac{b_W}{l_V} = 0.1539$$

$$\Rightarrow S_V = 9.57 m^2$$

– Evaluation of the results

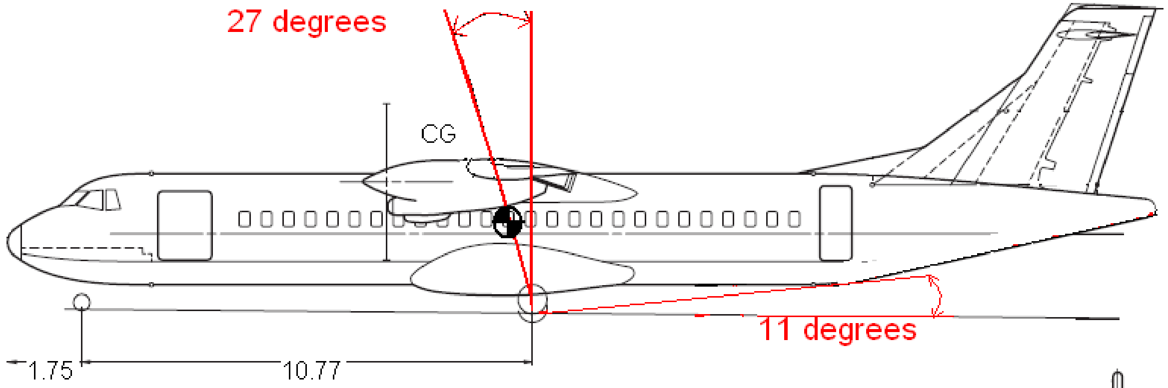
!

- **If the area S_H does not match *Empennage* results then:**
  - **m_H would need to be re-evaluated**
  - **and wing position adjusted**
- **For the vertical tail the *larger* area of the two was chosen**

- <u>Landing gear</u>
  - Position: corelated with the CG aft position $x_{LG,N-LG,M} = 10.77m$
  - Turn over angle in the x direction: min. 15°
  - Distance between wheels of the main LG $y_{track} = 4.10m$
  - Tail clearence: 11°

To prevent tail tipping

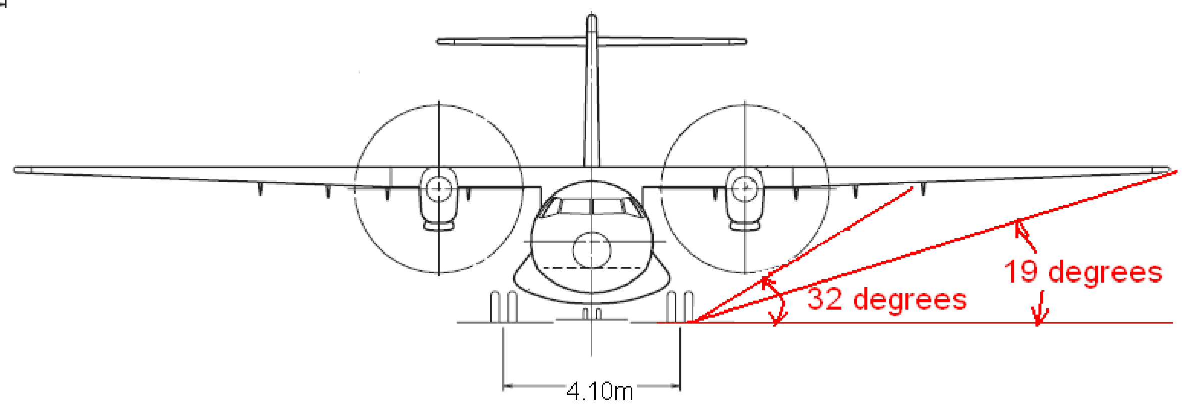
To prevent side tipping



  - Lateral clearence:

    min. 7°

    required



- **Drag estimation and polar**
  - **Three major components:**
    - **Zero lift drag – it is being estimated for each component, according to the formula:** $C_{D,0} = \sum C_f \cdot FF_c \cdot Q_c \cdot S_{wett} / S_{ref}$
    - **Lift dependent drag**
    - **Mach drag – we neglect this from the beginning, as the aircraft flies at lower speed**

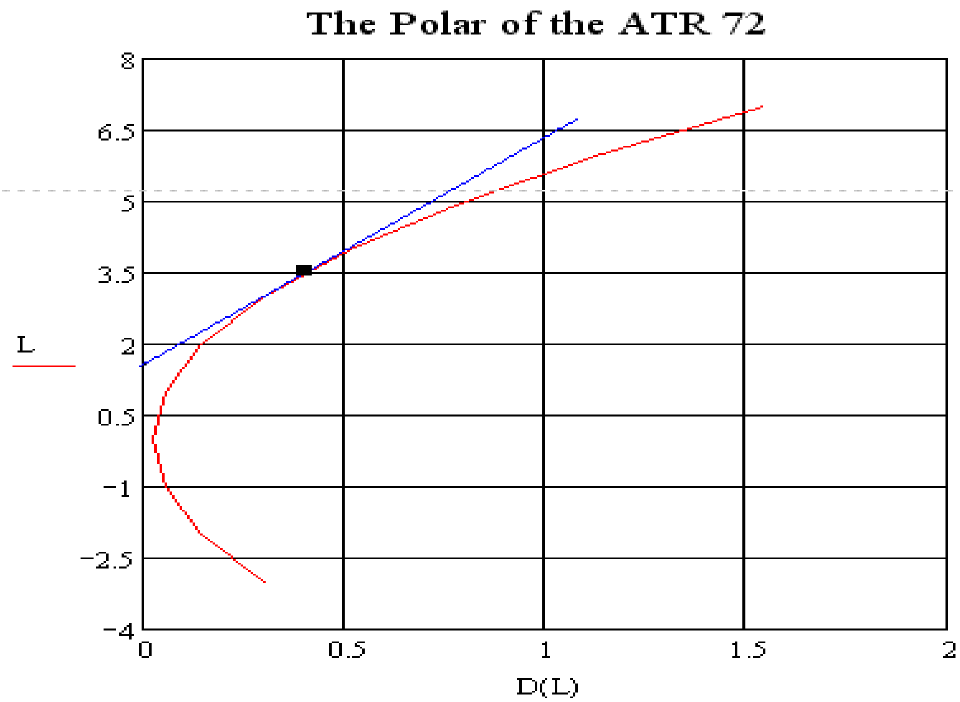
	C_f	FF_c	Q_c	S_{wett}/S_{ref}	$C_{D,0}$
Fuselage	$2.24 \cdot 10^{-3}$	1.088	1	3.3	$8.053 \cdot 10^{-3}$
Wing	$3.56 \cdot 10^{-3}$	1.84	1	2.08	$14 \cdot 10^{-3}$
Horizontal Tailplane	$3.392 \cdot 10^{-3}$	1.368	1.04	0.17	$0.8347 \cdot 10^{-3}$
Vertical Tailplane	$3.933 \cdot 10^{-3}$	1.419	1.04	0.22	$1.315 \cdot 10^{-3}$
Nacelle	$3.292 \cdot 10^{-3}$	1.072	1.5	0.3	$2 \cdot 1.6 \cdot 10^{-3}$
Total					$27.4 \cdot 10^{-3}$

Drag and Polar

– **The polar is given by**

$$C_D = C_{D,0} + \frac{C_L^{\ 2}}{\pi \cdot A \cdot e} \Leftrightarrow C_D = 0.027403 + 0.031 \cdot C_L^{\ 2}$$

» **In the preliminary sizing calculation the value e = 0.85 was used**



» **The resulting L/D is E = 15.8**

DOC Estimation

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- **Design evaluation**
  - **AEA method (Association of European Airliners) for estimating the direct operating costs (DOC)**

	Parameters used for the estimation	Results [mil$/year]
Depreciation	Service life, residual value	0.99
Interest	Average interest rate, total price of the aircraft	0.73
Insurance	% of aircrafts price	0.07
Fuel	Price and mass fuel, no. of flights per year	2.37
Maintenance	Labor and material, inflation factor	1.44
Crew	No. of crew members	2.07
Fees: – Landing	Maximum take-off mass, no. of flights/year, inflation factor	0.39
– Navigation	Maximum take-off mass, inflation factor	0.93
– Handling	Maximum payload, inflation factor	1.45

DOC Estimation

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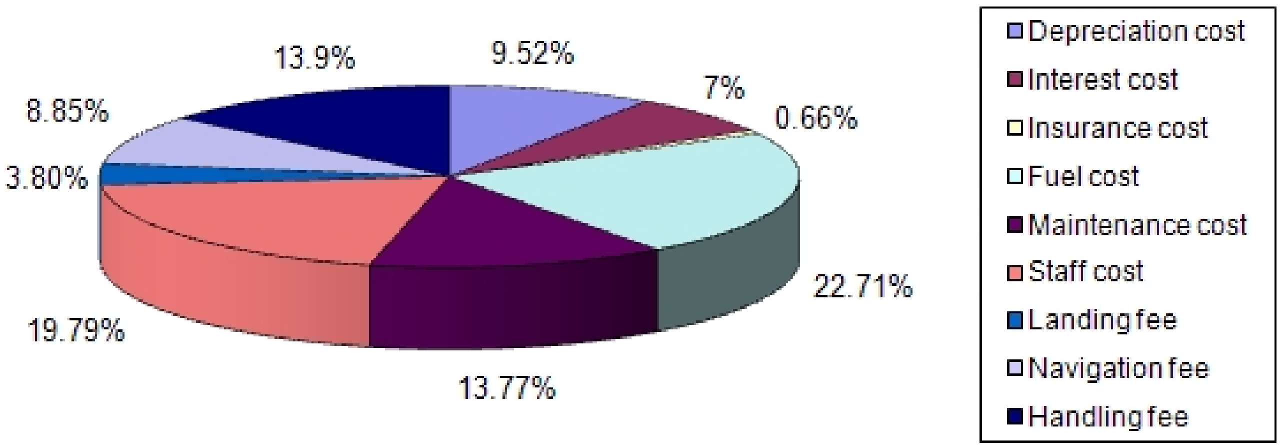
– **Total DOC = the sum of the costs of each of the following elements:**

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

C_{DOC}=10.5 mil US$/year



Summary

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Components	*Redesign*	*Original*	*Deviation*
Fuselage			
Length	27.13 m	27.17 m	0.1%
Diameter	2.77 m	2.57 m	-2.0%
Cabin Length	19.25 m	19.21 m	0.1%
Wing			
Wing Span	27.13 m	27.05 m	0.3 %
Wing Surface	61.3 m^2	61.0 m^2	0.5 %
Wing Loading	373.7 kg/m^2	373.8 kg/m^2	0.0 %
High Lift Device	Double sloted flaps and slats	Double sloted flaps	
Power Plant			
Power Loading	179.8 W/kg	179.9 W/kg	-0.1 %
Horizontal Tail			
Surface	9.7 m^2	11.7 m^2	-17.1 %
Vertical Tail			
Surface	14.1 m^2	12.5 m^2	12.8 %
Mass			
Maximum Take-Off Mass	22925 kg	22800 kg	0.5%
Operating Empty Mass	12834 kg	12950 kg	0.9%

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