AERODYNAMIC CHARACTERISTICS CALCULATION

57-001- Aerodynamic

Prepared by

Maria Antoneta COTEA

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REFERENCES

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4. Roskam, Jan Airplane Design Part VI Preliminary Calculations of Aerodynamic, Thrust and Power Characteristics - 2004
5. 57-002 4416 Profile- Aerodynamic characteristics
6. 57-003 Balancing & Maneuvering Loads
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Location of the file: ftp.aerolab.biz/Release/57-001_R0.doc

General Symbols

Simbol	Definition	Dimension
a	speed of sound	m/s
x.ac , a_c	aerodynamic center (% MAC)	
Xac	aerodynamic center location	m
A	wing aspect ratio	
b	wing span	m
CD	drag coefficient	
CD0	zero lift drag coefficient	
CDb	base drag coefficient	
cf	turbulent flat plane coefficient	
$cL\alpha$	section lift curve slope	1/°
$CL\alpha$, $CN\alpha$	lift curve slope	1/°
CLmax	maximum lift coefficient	
cmo	zero lift pitching moment	
ew	span efficiency factor	
iw	wing incidence	°
H	height	m
T	temperature	°C
L	length	m
MAC	mean aerodynamic chord	m
Re	Reynolds no.	
S	surface area	m^2
Swet	wetted area	m^2
V	speed	m/s^2
VB	fuselage volume	m^3
xcg	center of gravity location (% MAC)	
α	incidence	°
αCLmax	stall incidence	°
ε	thickness	
λB	fuselage finess ratio	
λ	taper ratio	
χ	sweep angle	°
ρ	air density	kg/m^3

SUBSCRIPTS

W	wing
H	horizontal tail
B	fuselage
WB	wing -fuselage
AC	aircraft

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1. WING AERODYNAMICS

This report is dealing with the aerodynamic characteristics of Lo Camp Aircraft, using the DATCOM (Ref. 1) methods. The definitions and the symbols are identically as in DATCOM.

The aerodynamics characteristics are calculated up to angles of attack approaching the stall.

For plan form of wing and horizontal and vertical tail the elliptical tips are not taken into consideration, the span and aspect ratio of wing and tails, are reduced.

$$\mathbf{MAC} := 1.594$$

Flight height H=0

$$\mathbf{H} := 0 \qquad \rho_0 := 1.225 \qquad \mathbf{T_0} := 288.15$$

$$\mathbf{T} := \mathbf{T_0} - 0.0065\cdot\mathbf{H} \qquad \rho := \rho_0\cdot(1 - 0.0000226\,\mathbf{H})^{4.225}$$

$$\mu := 1.82\cdot10^{-5}\cdot\left(\frac{\mathbf{T}}{288}\right)^{\frac{3}{2}}\cdot\frac{388}{110+\mathbf{T}} \qquad \mathbf{a} := 20.0519\cdot\sqrt{\mathbf{T}}$$

$$\mathbf{T} = 288.15 \qquad \rho = 1.225 \qquad \mathbf{a} = 340.381 \qquad \mu = 1.775\times10^{-5} \qquad \nu := \frac{\mu}{\rho}$$

Flight speed Vd=237.1Km/h

$$\mathbf{V} := \frac{237.1}{3.6} \qquad \mathbf{V} = 65.861$$

Mach No.

$$\mathbf{M} := \frac{\mathbf{V}}{\mathbf{a}}$$

Reynolds no.

$$\mathbf{Re} := \frac{\mathbf{V}\cdot\mathbf{MAC}\cdot\rho}{\mu} \qquad \mathbf{Re} = 7.245\times10^{6}$$

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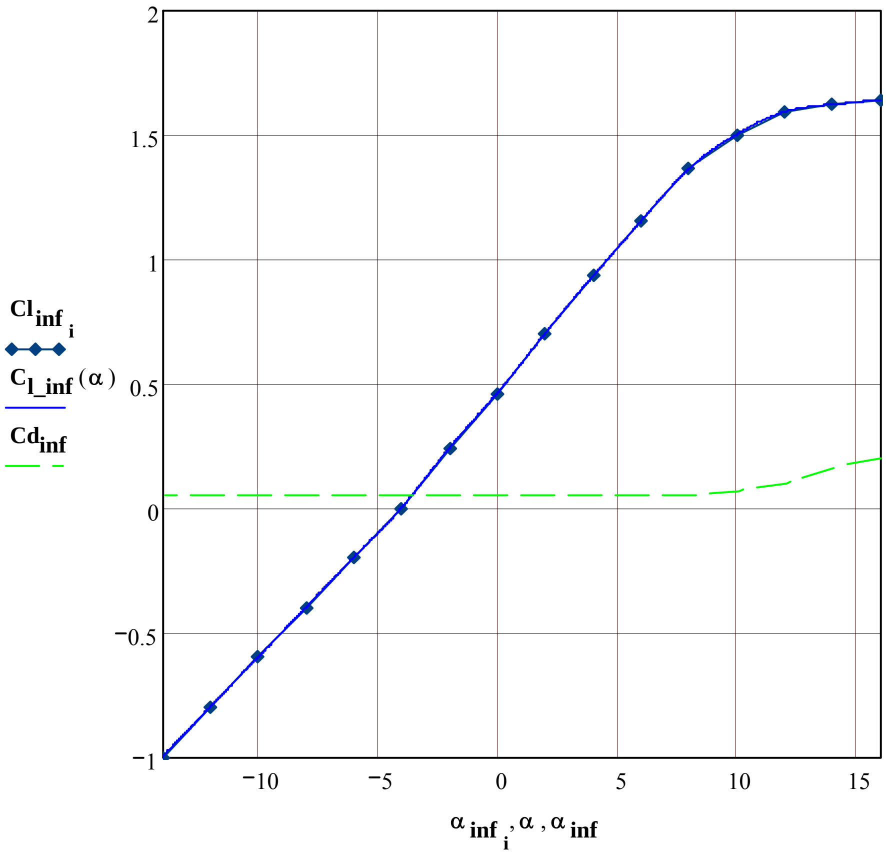
1.1 AIRFOIL CHARACTERISTICS NACA 4416 see Ref. 5

$$\alpha_{inf} := \begin{pmatrix} 16 \\ 14 \\ 12 \\ 10 \\ 8 \\ 6 \\ 4 \\ 2 \\ 0 \\ -2 \\ -4 \\ -6 \\ -8 \\ -10 \\ -12 \\ -14 \end{pmatrix} \qquad Cl_{inf} := \begin{pmatrix} 1.64 \\ 1.62 \\ 1.59 \\ 1.5 \\ 1.36 \\ 1.15 \\ .93 \\ .7 \\ .46 \\ .24 \\ 0 \\ -.2 \\ -.4 \\ -.6 \\ -.8 \\ -1 \end{pmatrix} \qquad Cd_{inf} := \begin{pmatrix} 0.2 \\ 0.16 \\ 0.1 \\ 0.07 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \\ 0.05 \end{pmatrix}$$

$$\mathbf{i} := 0..15 \qquad \alpha\mathbf{1e}_{\mathbf{i}} := \alpha_{\mathbf{inf}_{15-\mathbf{i}}} \qquad \mathbf{Cl1e}_{\mathbf{i}} := \mathbf{Cl}_{\mathbf{inf}_{15-\mathbf{i}}}$$

$$\mathbf{C_{l\_inf}(x)} := \mathbf{interp}\left(\mathbf{lspline}\left(\alpha\mathbf{1_e}, \mathbf{Cl1_e}\right), \alpha\mathbf{1_e}, \mathbf{Cl1_e}, \mathbf{x}\right)$$

PROFILE POLAR



Lift slope $$Cl_{\alpha\_inf} := \frac{C_{l\_inf}(4) - C_{l\_inf}(-4)}{8} \qquad Cl_{\alpha\_inf} = 0.116 \qquad Cz_{A\_inf} := Cl_{\alpha\_inf}\cdot\frac{180}{\pi}$$

$$\alpha_{0inf} := -4\cdot \mathbf{deg}$$

Wing geometry

Surface

$S := 13.5$

Span $b := 8.47$

Sweep Angles $\chi_0 := 0$ $\chi_{25} := 0$ $\chi_{50} := 0$

Taper ratio $\lambda := 1$

Aspect ratio $A := \frac{b^2}{S}$ $A = 5.314$

Wing zero incidence $i_w := 2$

Maximum thicness $\varepsilon := 0.16$

DATCOM table 4.1.1-A

$c_{l\alpha} := 0.105$ $a\_c\_1 := .245$ $c_{m0} := -0.093$

$\alpha_{s0} := -4.3$ $\alpha_{0lmax} := 15$ $c_{lmax} := 1.64$ $\alpha_{cr} := 8$

The profiles 4412, 4415 and 4418 have the same lift curve slope 0.105, the value of lift curve slope for profile 4416 has to be 0.105.

SECTION LIFT CURVE SLOPE

The effect of the NASA standard roughness on section data as indicated in fig. 4.1.1.2-7

$\Delta c_{l\alpha} := -0.0025$ $c_{l\alpha} := c_{l\alpha} + \Delta c_{l\alpha}$ $c_{l\alpha} = 0.103$

SECTION LIFT VARIATION WITH ANGLE OF ATTACK NEAR MAXIMUM LIFT

$c_{lmax\_base} := 1.55$ Fig. 4.1.1.4-5 Δy=4 position of max thickness 30%

$\Delta_{1clmax} := 0.08$ Fig. 4.1.1.4-6 Δy=4 position of max camber 40%

$\Delta_{2clmax} := 0.0$ Fig. 4.1.1.4-7a Δy=4 position of max thickness 30%

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Fig. 4.1.1.4-7a Δy=4 position of max thickness 30%

$\Delta_{4clmax} := -0.2$ Fig. 4.1.1.4-8a Δy=4 the minimum value

$$c_{lmax\_c} := c_{lmax\_base} + \Delta_{1clmax} + \Delta_{2clmax} + \Delta_{4clmax}$$

$$c_{lmax\_c} = 1.43$$

SECTION PITCHING MOMENT

SECTION ZERO LIFT PITCHING MOMENT

taken from Table 4.1.1-A $c_{m0} = -0.093$

SECTION PITCHING MOMENT VARIATION WITH LIFT

NACA4415 $a\_c\_15 := 0.245$ **NACA4418** $a\_c\_18 := .242$ **NACA4416** $a\_c := 0.244$

1.2 WING LIFT

Wing zero lift angle $\alpha_0 := \alpha_{s0}$ $\alpha_0 = -4.3$

1.2.1 WING LIFT CURVE SLOPE

$$c_{l\alpha} := c_{l\alpha} \cdot 57.3$$

$$C_{L\alpha} := \frac{2\pi \cdot A}{2 + \left[4 + \left[\frac{2\pi \cdot A \cdot \left(1 - M^2\right)}{c_{l\alpha}}\right]^2\right]^{0.5}}$$

$C_{L\alpha} = 4.266$ 1/rad

1.2.2 WING LIFT IN THE NONLINEAR ANGLE OF ATTACK

$$C_L(\alpha_r) := \frac{C_{L\alpha}}{57.3}\cdot(\alpha_r - \alpha_0)$$

1.2.3 WING MAXIMUM LIFT

The wing maximum lift of high aspect ratio wings is directly related to the maximum lift of the wing airfoil section with plan form geometry of secondary importance

$C_1 := 0$ $\qquad\qquad \frac{4}{C_1 + 1}\cos(\chi_0) = 4$

because $A = 5.314$ is > 4 method 2 deal with the Wing maximum lift

$C_{L1max} := 0.9$ $\qquad \Delta\alpha_{CLmax} := 2.25$ grad

$C_{Lmax} := C_{L1max}\cdot c_{lmax}$ $\qquad C_{Lmax} = 1.476$

$\alpha_{CLmax} := \frac{C_{Lmax}}{C_{L\alpha}}\cdot 57.3 + \alpha_0 + \Delta\alpha_{CLmax}$ $\qquad \alpha_{CLmax} = 17.774$ grad

$\alpha_{rc} := 16$ $\qquad C_{L1}(\alpha_{rc}) := C_{Lmax} - 0.07$ $\qquad C_{L1}(\alpha_{rc}) = 1.406$

$\alpha_{rc} := 17.77$ $\qquad C_{L1}(\alpha_{rc}) := C_{Lmax}$ $\qquad C_{L1}(\alpha_{rc}) = 1.476$

$\alpha_{rc} := 18$ $\qquad C_{L1}(\alpha_{rc}) := C_{Lmax} - 0.02$ $\qquad C_{L1}(\alpha_{rc}) = 1.456$

1.3 WING DRAG

1.3.1 WING ZERO LIFT DRAG

$S_{wet} := 2 \cdot b \cdot MAC \cdot \left(1 + \varepsilon \cdot \frac{MAC}{4}\right)$ $\quad S_{wet} = 28.724$ $\quad$ Ref. 4 page 533

$R_{LS} := 1.07 \quad L_1 := 1.2 \quad k := 0.4 \cdot 10^{-3}$

$l := MAC \cdot \frac{1000}{25.4} \quad \frac{l}{k} = 1.569 \times 10^5 \quad R_l := 10^7 \quad C_f := 0.003$

$C_{D0} := C_f \cdot \left(1 + L_1 \cdot \varepsilon + 100 \cdot \varepsilon^4\right) \cdot R_{LS} \cdot \frac{S_{wet}}{S} \quad C_{D0} = 8.589 \times 10^{-3}$

1.3.2 WING DRAG AT ANGLE OF ATTACK

$R_{LE} := 0.029 \cdot MAC \quad R_{LE} = 0.046 \quad R_{ILER} := \frac{V \cdot R_{LE} \cdot \rho}{\mu} \cdot \left(1 - M^2\right)^{0.5}$

$R_{ILER} = 2.061 \times 10^5 \quad R1 := 0.96$

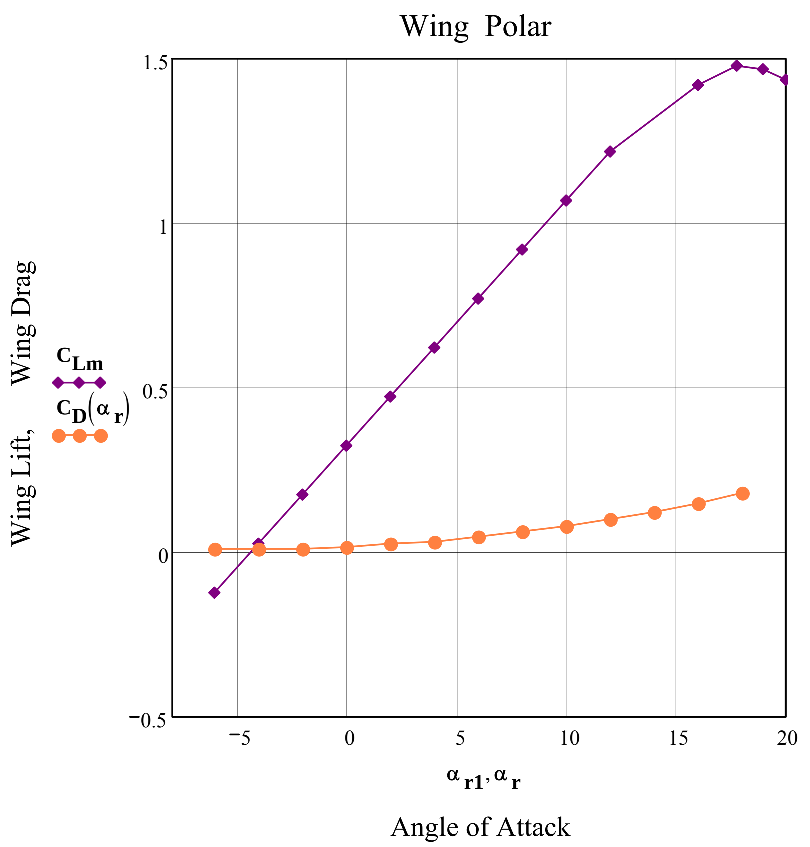
$$e_w := \frac{\frac{1.1 C_{L\alpha}}{A}}{R1 \cdot \frac{C_{L\alpha}}{A} + \pi \cdot (1 - R1)} \qquad e_w = 0.985 \qquad C_{DL}(\alpha_r) := \frac{C_L(\alpha_r)^2}{\pi \cdot A \cdot e_w}$$

For straight wing with wing twist zero: $\quad C_D(\alpha_r) := C_{D0} + C_{DL}(\alpha_r)$

$\alpha_r := -6, -4 .. 18$

$$\alpha_{r1} := \begin{pmatrix} -6 \\ -4 \\ -2 \\ 0 \\ 2 \\ 4 \\ 6 \\ 8 \\ 10 \\ 12 \\ 16 \\ 17.77 \\ 19 \\ 20 \end{pmatrix} \qquad C_{Lm} := \begin{pmatrix} -0.127 \\ 0.022 \\ 0.171 \\ 0.32 \\ 0.469 \\ 0.618 \\ 0.767 \\ 0.916 \\ 1.065 \\ 1.214 \\ 1.42 \\ 1.476 \\ 1.466 \\ 1.436 \end{pmatrix}$$

$\alpha_r =$	$C_D(\alpha_r) =$
-6	9.563·10-3
-4	8.619·10-3
-2	0.01
0	0.015
2	0.022
4	0.032
6	0.044
8	0.06
10	0.078
12	0.098
14	0.121
16	0.147
18	0.176



1.4 WING LIFT PITCHING MOMENT

1.4.1 WING ZERO LIFT PITCHING MOMENT

$$C_{m0} := \frac{A}{A+2} \cdot c_{m0} \qquad C_{m0} = -0.068$$

1.4.2 WING PITCHING MOMENT CURVE SLOPE

The characteristics of high aspect ratio wings are primarily determined by the wing two dimensional section characteristics.

$$x_{cg} := 0. \qquad x_{a.c} := a\_c \qquad x_{a.c} = 0.244$$

$$C_{m\alpha} := \left(x_{cg} - x_{a.c}\right) \cdot C_{L\alpha} \qquad C_{m\alpha} = -1.041$$

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2. HORIZONTAL TAIL

GEOMETRY

Root chord $c_{Hr} := 1.427$

Tip chord $c_{Ht} := 0.776$

1/ Taper ratio $r_{H1} := \frac{c_{Hr}}{c_{Ht}}$ $\quad r_{H1} = 1.839$

Taper ratio $r_H := \frac{1}{r_{H1}}$ $\quad r_H = 0.544$

Span $b_H := 2.58$

Area $S_H := (c_{Hr} + c_{Ht}) \cdot \frac{b_H}{2}$ $\quad S_H = 2.842$

Aspect ratio $A_H := \frac{b_H^{\,2}}{S_H}$ $\quad A_H = 2.342$

Maximum thickness $\varepsilon_H := 0.06$ Stabilizer profile NACA 0006

Sweep angles $\chi_{H0} := 20 \cdot \mathbf{deg}$

$$\chi_{H25} := \mathrm{atan}\left[\tan(\chi_{H0}) + \frac{4}{A_H} \cdot \left[(.25 - 0) \cdot \frac{1 - r_{H1}}{1 + r_{H1}}\right]\right] \qquad \chi_{H25} = 13.377\ \mathbf{deg}$$

$$\chi_{H50} := \mathrm{atan}\left[\tan(\chi_{H0}) + \frac{4}{A_H} \cdot \left[(.5 - 0) \cdot \frac{1 - r_{H1}}{1 + r_{H1}}\right]\right] \qquad \chi_{H50} = 6.37\ \mathbf{deg}$$

$$\chi_{H100} := \mathrm{atan}\left[\tan(\chi_{H0}) + \frac{4}{A_H} \cdot \left[(1 - 0) \cdot \frac{1 - r_{H1}}{1 + r_{H1}}\right]\right] \qquad \chi_{H100} = -8.008\ \mathbf{deg}$$

Mean aerodynamic chord.

$$MAC_H := \frac{2}{S_H} \cdot \int_0^{\frac{b_H}{2}} \left[c_{Hr} - y \cdot \left(\tan\left(\chi_{H0}\right) - \tan\left(\chi_{H100}\right)\right)\right]^2 dy \qquad MAC_H = 1.134$$

$$y := 0 \qquad y_{macH} := \mathrm{root}\left[c_{Hr} - y \cdot \left(\tan\left(\chi_{H0}\right) - \tan\left(\chi_{H100}\right)\right) - MAC_H, y\right]$$

MAC. y location $\qquad y_{macH} = 0.581$

Reynolds No. $\qquad Re_H := \frac{V \cdot MAC_H \cdot \rho}{\mu} \qquad Re_H = 5.152 \times 10^6$

2.1 AIRFOIL SECTION CHARACTERISTICS NACA 0006

$$\alpha_{sH0} := 0 \qquad c_{m0\_H} := 0 \qquad c_{l\alpha\_H} := .108 \qquad a\_c_H := .25 \qquad \alpha 0_{max\_H} := 9$$

$$\alpha_{cr\_H} := 9 \qquad c_{lmax\_H} := .92$$

The effect of NACA standard roughness on section data : $\quad \Delta c_{l\alpha\_H} := 0$

2.2 HORIZONTAL TAIL LIFT

2.2.1 HORIZONTAL TAIL LIFT CURVE SLOPE

HT zero lift angle $\qquad \alpha_{H0} := \alpha_{sH0} \qquad c_{l\alpha\_H} := c_{l\alpha\_H} \cdot 57.3$

$$C_{L\alpha\_H} := \frac{2\pi \cdot A_H}{2 + \sqrt{4 + \left(1 + \frac{\tan\left(\chi_{H50}\right)^2}{1 - M^2}\right) \cdot \left[\frac{2 \cdot \pi\left(1 - M^2\right) \cdot A_H}{c_{l\alpha\_H}}\right]^2}} \qquad C_{L\alpha\_H} = 2.914 \quad 1\ /\mathrm{rad}$$

2.2.2 HORIZONTAL TAIL LIFT IN THE NONLINEAR ANGLE OF ATTACK

$$C_{L\_H}\left(\alpha_r\right) := \frac{C_{L\alpha\_H}}{57.3} \cdot \left(\alpha_r - \alpha_{H0}\right)$$

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2.2.3 HORIZONTAL TAIL LIFT MAXIMUM LIFT

Method 3 $\quad C_{1H} := 0.292 \quad C_{2H} := 1.062$

$$\frac{3}{(C_{1H}+1)\cdot\cos(\chi_{H0})} = 2.471$$

$$\frac{4}{(C_{1H}+1)\cdot\cos(\chi_{H0})} = 3.295$$

$A_H = 2.342$ is < 2.471 we can use method 3 -low aspect ratio procedure to find out the HT maximum lift

$(C_{2H}+1)\cdot A_H\cdot\tan(\chi_{H0}) = 1.758 \qquad \Delta C_{Lmax\_H} := -0.08$

$A_H\cdot\cos(\chi_{H0})\cdot\left[1+(2\cdot r_H)^2\right] = 4.804 \qquad \Delta\alpha_{CLmax\_H} := -3.5$

$(C_{1H}+1)\cdot A_H\cdot\cos(\chi_{H0}) = 2.844 \qquad C_{Lmax\_base\_H} := 0.83 \qquad \alpha_{CLmax\_base\_H} := 22.5$

$C_{Lmax\_H} := C_{Lmax\_base\_H} + \Delta C_{Lmax\_H} \qquad C_{Lmax\_H} = 0.75$

$\alpha_{CLmax\_H} := \alpha_{CLmax\_base\_H} + \Delta\alpha_{CLmax\_H} \qquad \alpha_{CLmax\_H} = 19$

$\alpha_{rc} := 17 \qquad C_{L\_H1}(\alpha_{rc}) := C_{Lmax\_H} - 0.07 \qquad C_{L\_H1}(\alpha_{rc}) = 0.68$

$\alpha_{rc} := 19 \qquad C_{L\_H1}(\alpha_{rc}) := C_{Lmax\_H} \qquad C_{L\_H1}(\alpha_{rc}) = 0.75$

$\alpha_{rc} := 20 \qquad C_{L\_H1}(\alpha_{rc}) := C_{Lmax\_H} - 0.02 \qquad C_{L\_H1}(\alpha_{rc}) = 0.73$

2.3 HORIZONTAL TAIL DRAG

2.3.1 HORIZONTAL TAIL ZERO LIFT DRAG

$S_{wet\_H} := 2\cdot S_H \qquad L_{1\_H} := 1.2$

$$\chi_{H30} := \operatorname{atan}\left[\tan(\chi_{H0}) + \frac{4}{A_H}\cdot\left[(.30-0)\cdot\frac{1-r_{H1}}{1+r_{H1}}\right]\right] \qquad \chi_{H30} = 12.001\ \mathbf{deg}$$

$\cos(\chi_{H30}) = 0.978$ $\quad R_{LS\_H} := 1.07$ $\quad l_H := MAC_H \cdot \frac{1000}{25.4}$ $\quad l_H = 44.628$

$\frac{l_H}{k} = 1.116 \times 10^5$ $\quad R_{l\_H} := 6 \cdot 10^6$ $\quad C_{f\_H} := 0.0033$

$$C_{D0\_H} := C_{f\_H} \cdot \left(1 + L_{1\_H} \cdot \varepsilon_H + 100 \cdot \varepsilon_H^{\,4}\right) \cdot R_{LS\_H} \cdot \frac{S_{wet\_H}}{S_H} \qquad C_{D0\_H} = 7.58 \times 10^{-3}$$

2.3.2 HORIZONTAL TAIL DRAG AT ANGLE OF ATTACK

$R_{LE\_H} := 0.011$ $\quad R_{ILER\_H} := \frac{V \cdot R_{LE\_H} \cdot \rho}{\mu}$ $\quad R_{ILER\_H} = 5 \times 10^4$

$R_{ILER\_H} \cdot \cot(\chi_{H0}) = 1.374 \times 10^5$ $\quad \frac{A_H \cdot r_H}{\cos(\chi_{H0})} = 1.355$ $\quad R_{1\_H} := 0.924$

$$e_H := 1.1 \cdot \frac{\dfrac{C_{L\alpha\_H}}{A_H}}{R_{1\_H} \cdot \dfrac{C_{L\alpha\_H}}{A_H} + \left(1 - R_{1\_H}\right) \cdot \pi} \qquad e_H = 0.986$$

For HT with HT twist zero : $\quad C_{DL\_H}(\alpha_r) := \frac{C_{L\_H}(\alpha_r)^2}{\pi \cdot A_H \cdot e_H}$

$$C_{D\_H}(\alpha_r) := C_{D0\_H} + C_{DL\_H}(\alpha_r)$$

2.4 HORIZONTAL TAIL PITCHING MOMENT

2.4.1 HORIZONTAL TAIL ZERO LIFT PITCHING MOMENT

$$C_{m0\_H} := \frac{A_H \cdot \cos(\chi_{H25})^2}{A_H + 2 \cdot \cos(\chi_{H25})} \cdot c_{m0\_H} \qquad C_{m0\_H} = 0$$

2.4.2 HORIZONTAL TAIL PITCHING MOMENT CURVE SLOPE

Method 2

Step 1 $$a\_c_{Hcalc} := \frac{\frac{2(1-r_H)}{3} + 0.5\cdot\left(1 - \frac{r_H^2}{1+r_H}\right)\cdot\pi\cdot\log\left(1+\frac{A_H}{5}\right)}{1+\pi\cdot\log\left(1+\frac{A_H}{5}\right)}$$ $a\_c_{Hcalc} = 0.339$

Step 2 $C_{m\alpha\_H\_th} := -a\_c_{Hcalc}\cdot C_{L\alpha\_H}$ $C_{m\alpha\_H\_th} = -0.986$

Step 3 $\varepsilon_H = 0.06$ $C_{m\alpha\_H\_Cm\ \alpha\_th} := 1$

$C_{m\alpha\_H} := C_{m\alpha\_H\_th}$ $C_{m\alpha\_H} = -0.986$ $X_{a\_c\_H} := \frac{-C_{m\alpha\_H}}{C_{L\alpha\_H}}\cdot c_{Hr}$ $X_{a\_c\_H} = 0.483$

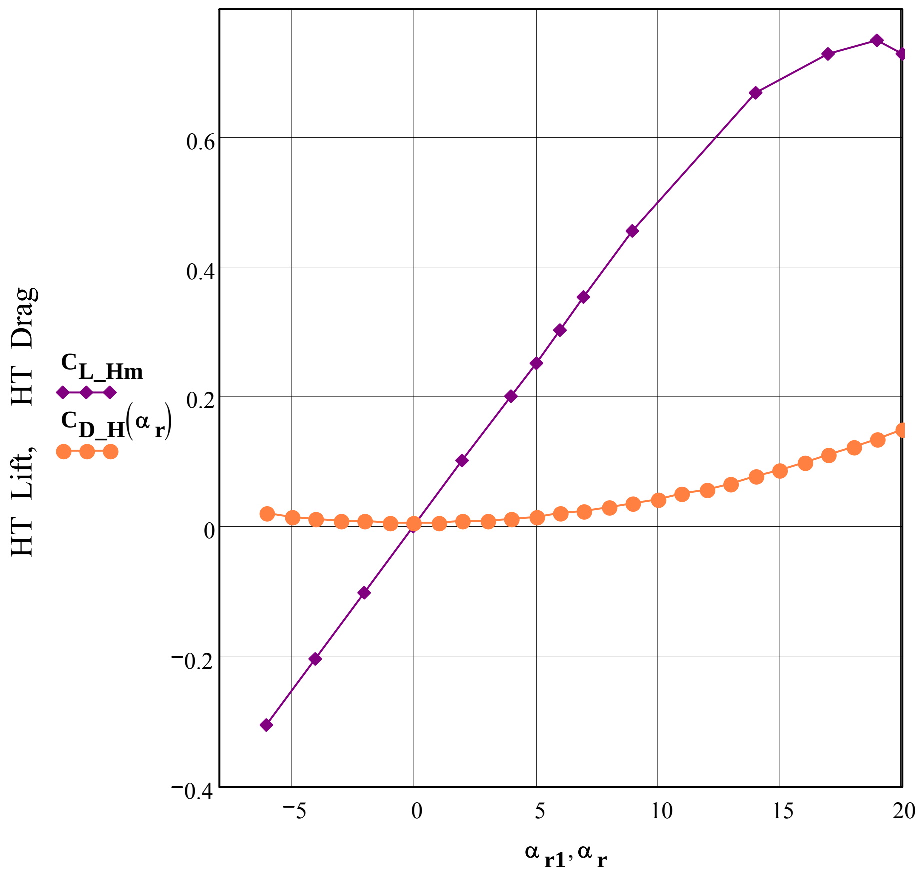
m

$\alpha_r := -6 .. 20$

$$\alpha_{r1} := \begin{pmatrix} -6 \\ -4 \\ -2 \\ 0 \\ 2 \\ 4 \\ 5 \\ 6 \\ 7 \\ 9 \\ 14 \\ 17 \\ 19 \\ 20 \end{pmatrix} \qquad C_{L\_Hm} := \begin{pmatrix} -.305 \\ -.203 \\ -.102 \\ 0. \\ 0.102 \\ 0.203 \\ 0.254 \\ 0.305 \\ .356 \\ .458 \\ .67 \\ .73 \\ .75 \\ .73 \end{pmatrix}$$

$\alpha_r =$	$C_{D\_H}(\alpha_r) =$
-6	0.02
-5	0.016
-4	0.013
-3	0.011
-2	9.006·10-3
-1	7.936·10-3
0	7.58·10-3
1	7.936·10-3
2	9.006·10-3
3	0.011
4	0.013
5	0.016
6	0.02
7	0.025
8	0.03
9	0.036

Horizontal Tail Polar



Angle of Attack

3. FUSELAGE AERODYNAMICS

GEOMETRY

$\mathbf{Rmin} := .33 \quad \mathbf{Rmax} := 0.36$

Equiv. radius	$R_B := 0.38$	
Reference surface	$S_B := \pi \cdot R_B^2$	$S_B = 0.454$
Nose lenght	$L_N := 0.9 + 0.723$	
Cilindrical lenght	$L_C := 2.4 - 0.723$	
Aft lenght:	$L_A := 2.8$	
Overall lenght:	$L_B := L_N + L_C + L_A$	$L_B = 6.1$
Fuselage Finess ratio:	$\lambda_B := \dfrac{L_B}{2 \cdot R_B}$	$\lambda_B = 8.026$
Nose Finess ratio	$\lambda_N := \dfrac{L_N}{2 \cdot R_B}$	$\lambda_N = 2.136$
Cilinder Finess ratio	$\lambda_C := \dfrac{L_C}{2 \cdot R_B}$	$\lambda_C = 2.207$
Side surface area:	$S_{lat} := 2.85 \cdot L_B \cdot \sqrt{S_B}$	$S_{lat} = 11.709$
End surface of fuselage	$S_{pB} := 0.002$	
Critical No. for fuselage:	$M_{cB} := \lambda_B \cdot (0.17 - 0.0076 \cdot \lambda_B)$	$M_{cB} = 0.875$
Fuselage volume	$V_B := S_B \cdot \left(L_N + L_C + \dfrac{L_A}{3}\right)$	$V_B = 1.92$

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3.1 FUSELAGE LIFT

3.1.1 FUSELAGE LIFT CURVE SLOPE

$$x_1 := L_N + L_C \qquad x_1 = 3.3 \qquad x_0 := 0.378 \cdot L_B + 0.527 \cdot x_1 \qquad x_0 = 4.045$$

Finess ratio $\dfrac{x_0}{2 \cdot R_B} = 5.322 \qquad k_{2\_k1} := 0.84 \qquad S_0 := S_B \cdot \dfrac{L_B - x_0}{L_B - x_1}$

$$S_0 = 0.333$$

$$C_{L\alpha\_B} := 2 \cdot \frac{k_{2\_k1} \cdot S_0}{V_B^{\frac{2}{3}}} \qquad C_{L\alpha\_B} = 0.362 \qquad \text{1/ rad}$$

3.1.2 FUSELAGE LIFT IN THE NONLINEAR ANGLE OF ATTACK RANGE

$$\eta := 0.6$$

$$c_{dc} := 1.2 \qquad R_x(x) := 0.14 + \frac{1.4}{x \cdot (.707 - .28)}$$

$$\alpha := -10, -8 .. 18$$

$$C_{L\_B}(\alpha) := C_{L\alpha\_B} \cdot \frac{\alpha}{57.3} + 2 \cdot \frac{\left(\frac{\alpha}{57.3}\right)^2}{V_B^{\frac{2}{3}}} \cdot \int_{x_0}^{L_B} \eta \cdot c_{dc} \cdot R_x(x)\, dx$$

3.2 FUSELAGE DRAG

3.2.1 FUSELAGE ZERO LIFT DRAG

$$\frac{L_B}{2 \cdot R_B} = 8.026 \qquad d_b := 0.28 \qquad \frac{d_b}{2 \cdot R_B} = 0.368 \qquad S_S := 23 \cdot S_B \qquad S_S = 10.434$$

$$k := 0.4 \cdot 10^{-3} \qquad l_B := L_B \cdot \frac{1000}{25.4} \qquad \frac{l_B}{k} = 6.004 \times 10^5 \qquad R_{lB} := (6 \cdot 10)^7 \qquad C_{f\_B} := 0.0023$$

$$C_{D0B\_b} := C_{f\_B} \cdot \left[1 + \frac{60}{\left(\frac{L_B}{2 \cdot R_B}\right)^3} + 0.0025 \left(\frac{L_B}{2 \cdot R_B}\right)\right] \cdot \frac{S_S}{S_B} \qquad C_{DB\_b} := 0.029 \cdot \frac{\left(\frac{d_b}{2 \cdot R_B}\right)^3}{C_{D0B\_b}^{\,0.5}}$$

$$C_{DB\_b} = 5.916 \times 10^{-3}$$

$$C_{D0B} := C_{f\_B} \cdot \left[1 + \frac{60}{\left(\frac{L_B}{2 \cdot R_B}\right)^3} + 0.0025\left(\frac{L_B}{2 \cdot R_B}\right)\right] \cdot \frac{S_S}{S_B} + 0.029 \cdot \frac{\left(\frac{d_b}{2 \cdot R_B}\right)^3}{C_{D0B\_b}^{0.5}} \qquad C_{D0B} = 0.066$$

3.2.2 FUSELAGE DRAG AT AOA

$$C_{DB\alpha}(\alpha) := C_{L\alpha\_B} \cdot \left(\frac{\alpha}{57.3}\right)^2 + 2 \cdot \frac{\left(\frac{\alpha}{57.3}\right)^3}{V_B^{\frac{2}{3}}} \cdot \int_{x_0}^{L_B} \eta \cdot R_x(x) \cdot \frac{(L_B - x_0) \cdot c_{dc}}{L_B - x_1} \, dx$$

$$C_{D\_B}(\alpha) := C_{D0B} + C_{DB\alpha}(\alpha)$$

3.3 FUSELAGE PITCHING MOMENT

3.3.1 FUSELAGE PITCHING MOMENT CURVE SLOPE

$C_{Im\alpha\_B} := 0.0031$

1/ grd see Ref. 6

Cm,α,WB - UPWASH - DATCOM: § 4.2.2.1

Strip	x1 [m]	x1/Cr	Δx [m]	w [m]	(δε/δα)m	(δε/δα)1	w^2.((δε/δα)1 +1).Δx
1	0.2500	0.1568	0.5000	0.6600	-	3.0000	0.8667
2	0.7500	0.4705	0.5000	0.6600	1.4200	-	0.5250
3	1.2500	0.7842	0.5000	0.6600	0.8500	-	0.4017
4	1.6230	1.0182	0.2460	0.6600	0.7500	-	0.1870

Cm,α,WB - DOWNWASH - DATCOM: § 4.2.2.1

Strip	x2 [m]	x2/lH_1	Δx [m]	w [m]	(δε/δα)m	(δε/δα)2	w^2.((δευ/δα)2 +1).Dx
1	0.2500	0.0688	0.5000	0.6325	0.4207	0.0399	0.2080
2	0.7500	0.2064	0.5000	0.5060		0.1196	0.1433
3	1.2500	0.3441	0.5000	0.3795		0.1993	0.0864
4	1.7500	0.4817	0.5000	0.2530		0.2791	0.0409
5	2.2500	0.6193	0.5000	0.1265		0.3588	0.0109
6	2.5940	0.7140	0.1880	0.0000		0.4137	0.0000
						0.0000	0.0000
						0.0000	0.0000
						0.0000	0.0000
						0.0000	0.0000

$\Sigma = 0.4895$

Fuselage pitching moment curve slope
(Cm,α)F [1/deg] 0.0031

Method 2

$$x_m := L_N + 0.065 \qquad S_x = S_B \cdot \frac{L_B - x}{L_B - x_1} \qquad dS_x := \frac{-S_B}{L_B - x_1}$$

$$C_{m\alpha\_B} := 2 \cdot \frac{k_{2\_k1}}{V_B} \cdot \int_0^{x_0} dS_x \cdot (x_m - x)\, dx \qquad C_{m\alpha\_B} = 0.192 \text{ 1/ rad} \qquad \frac{C_{m\alpha\_B}}{57.3} = 3.346 \times 10^{-3}$$

3.3.2 FUSELAGE PITCHING MOMENT IN THE NONLINEAR AOA RANGE

Method 1 for α less eq. 12 °

$$C_{m\_B}(\alpha) := \frac{\alpha}{57.3}\cdot C_{m\alpha\_B} + 2\cdot\frac{\left(\frac{\alpha}{57.3}\right)^2}{V_B}\cdot\int_{x_0}^{L_B}\eta\cdot c_{dc}\cdot R_x(x)\cdot(x_m - x)\,dx$$

Method 2

$$x_c := \frac{L_N\cdot\frac{L_N}{2} + L_C\cdot\left(\frac{L_C}{2} + L_N\right) + \frac{L_A}{2}\cdot\left(\frac{2}{3}\cdot L_A + L_N + L_C\right)}{L_N + L_C + \frac{L_A}{3}} \qquad x_c = 2.995$$

$$S_B = 0.454$$

$$S_p := 2\cdot Rmax\cdot(L_N + L_C) + \frac{2\cdot Rmax + 0.28}{2}\cdot L_A \qquad S_p = 3.776$$

$$C_{1m\_B}(\alpha) := \frac{V_B - S_B\cdot(L_B - x_m)}{S_B\cdot 2\cdot R_B}\cdot\sin\left(\frac{2\cdot\alpha}{57.3}\right)\cdot\cos\left(\frac{\alpha}{2\cdot 57.3}\right) + 1.33\cdot\eta\cdot c_{dc}\cdot\frac{S_p}{S_B}\cdot\frac{(x_m - x_c)\cdot\sin\left(\frac{\alpha}{57.3}\right)^2}{2\cdot R_B}$$

4. WING BODY AERODYNAMICS

4.1. WING BODY LIFT

4.1 WING BODY LIFT CURVE SLOPE

$S_e := S - 2MAC \cdot Rmin$ $\quad S_e = 12.448$ $\quad b_e := b - 2Rmin$ $\quad b_e = 7.81$ $\quad A_e := \frac{b_e^2}{S_e}$

$$C_{L\alpha\_e} := \frac{2 \cdot \pi \cdot A_e}{2 + \left[4 + \left[\frac{2 \cdot \pi \left(1 - M^2\right) \cdot A_e}{c_{l\alpha}}\right]^2\right]^{0.5}}$$

$A_e = 4.9$

$C_{L\alpha\_e} = 4.145$ $\quad$ 1 / rad

$\frac{L_N}{2 \cdot Rmin} = 2.459$ $\quad k_{2\_k1N} := 0.7$ $\quad V_{BN} := S_B \cdot L_N$

$$C_{N\alpha} := \frac{2 \cdot k_{2\_k1N} \cdot S_B}{V_{BN}^{\frac{2}{3}}}$$

$C_{N\alpha} = 0.779$ $\quad$ 1 / rad $\quad 2 \cdot \frac{Rmin}{b} = 0.078$

$K_N := \frac{C_{N\alpha} \cdot S_B}{C_{L\alpha\_e} \cdot S_e}$ $\quad K_N = 6.848 \times 10^{-3}$ $\quad K_{W\_B\_KB\_W} := \left(2 \cdot \frac{Rmin}{b} + 1\right)^2$

$C_{L\alpha\_WB} := \left(K_N + K_{W\_B\_KB\_W}\right) \cdot C_{L\alpha\_e} \cdot \frac{S_e}{S}$ $\quad C_{L\alpha\_WB} = 4.467$ $\quad$ 1 / rad

4.1.2. WING BODY LIFT IN THE NONLINEAR AOA RANGE

For normal airplane type wing-body configuration the contribution of the body vortices can be ignored

$$C_{L\_WB}(\alpha) := \frac{C_{L\alpha\_WB}}{57.3} \cdot \left(\alpha - \alpha_0\right)$$

4.1.3 WING BODY MAXIMUM LIFT

$K_{CLmax\_WB} := 1 \qquad K_{\alpha\,max\_WB} := 1$

$C_{Lmax\_WB} := K_{CLmax\_WB} \cdot C_{Lmax} \qquad C_{Lmax\_WB} = 1.476$

$\alpha_{CLmax\_WB} := K_{\alpha\,max\_WB} \cdot \alpha_{CLmax} \qquad \alpha_{CLmax\_WB} = 17.774$

$\alpha_{rc} := 16 \qquad C_{L\_WB1}(\alpha_{rc}) := C_{Lmax} - 0.07 \qquad C_{L\_WB1}(\alpha_{rc}) = 1.406$

$\alpha_{rc} := 17.77 \qquad C_{L\_WB1}(\alpha_{rc}) := C_{Lmax} \qquad C_{L\_WB1}(\alpha_{rc}) = 1.476$

$\alpha_{rc} := 18 \qquad C_{L\_WB1}(\alpha_{rc}) := C_{Lmax} - 0.02 \qquad C_{L\_WB1}(\alpha_{rc}) = 1.456$

4.2 WING BODY DRAG

4.2.1 WING BODY ZERO LIFT DRAG

$S_{wet\_e} := 2 \cdot S_e \qquad Re_B := \dfrac{V \cdot L_B \cdot \rho}{\mu} \qquad Re_B = 2.773 \times 10^7$

$S_{S\_e} := S_S - MAC^2 \cdot 0.16 \quad S_{S\_e} = 10.027 \qquad R_{WB} := 0.92 \qquad coef := 1 + L_1 \cdot \varepsilon + 100 \cdot \varepsilon^4$

$$C_{D0\_WB} := \left[C_f \cdot coef \cdot R_{LS} \cdot \frac{S_{wet\_e}}{S} + C_{f\_B} \cdot \left[1 + \frac{60}{\left(\dfrac{L_B}{2 \cdot R_B}\right)^3} + 0.0025 \cdot \frac{L_B}{2 \cdot R_B} \right] \cdot \frac{S_{S\_e}}{S} \right] \cdot R_{WB} + C_{DB\_b} \cdot \frac{S_B}{S}$$

$C_{D0\_WB} = 8.833 \times 10^{-3}$

4.2.2 WING BODY DRAG AT AOA

$$C_{DL\_WB}(\alpha) := C_{DL}(\alpha) + C_{D\_B}(\alpha)\cdot\frac{S_B}{S}$$

$$C_{D\_WB}(\alpha) := C_{D0\_WB} + C_{DL\_WB}(\alpha)$$

4.3 WING BODY PITCHING MOMENT

4.3.1 WING BODY ZERO LIFT PITCHING MOMENT

$C_{m0\_B} := -0.0003$ see Ref. 6

$C_{m0\_WB} := C_{m0} + C_{m0\_B}$ $C_{m0\_WB} = -0.068$

(Cm,0) Fuselage - DATCOM: § 4.3.2.1					
Strip	w [m]	α□□$_{0W}$ □[°]	$(i_{CL})_B$ [°]	Δx [m]	$w_F^2.(\alpha_{0,W}+(i_{CL})_B).\Delta x$
1	0.6600	-2.3321	8	0.2460	0.6074
2	0.6600		8	0.5000	1.2345
3	0.6600		8	0.5000	1.2345
4	0.6600		8	0.5000	1.2345
5	0.6600		0	0.5000	-0.5079
6	0.6600		0	0.5940	-0.6034
7	0.6600		0	0.5000	-0.5079
8	0.6325		-5	0.4600	-1.3493
9	0.5060		-5	0.4600	-0.8635
10	0.3795		-5	0.4600	-0.4857
11	0.2530		-5	0.4600	-0.2159
12	0.1265		-5	0.4600	-0.0540
13	0.0000		-5	0.4600	0.0000
				$\Sigma =$	-0.2769

Fuselage length [m]	6.1
Fuselage max width [m]	0.66
Fuselage max height [m]	0.71
Equivalent diameter [m]	0.7724
Body fineness ratio	7.8972
Apparent mass factor (k2-k1)	0.9100

Fuselage zero lift pitching moment		
Datcom Fig. 4.22.1.1-20a	$(C_{m,0})_F$	-0.0003

4.3.2 WING BODY PITCHING MOMENT CURBE SLOPE

$K_{BW} := 0.1$ $\quad$ $K_{WB} := K_{W\_B\_KB\_W} - K_{BW}$ $\quad$ $A_e = 4.9$

$$C_{L\alpha WB} := K_{WB} \cdot C_{L\alpha\_e} \cdot \frac{S_e}{S} \qquad C_{L\alpha WB} = 4.059 \qquad K_{WB} = 1.062$$

$$C_{L\alpha BW} := K_{BW} \cdot C_{L\alpha\_e} \cdot \frac{S_e}{S} \qquad C_{L\alpha BW} = 0.382$$

$$x_{a.c\_N} := -C_{m\alpha\_B} \cdot \frac{V_B^{\frac{1}{3}}}{MAC \cdot C_{L\alpha\_e}} \qquad x_{a.c\_N} = -0.036$$

$$x_{a.c\_WB} := a\_c \qquad x_{a.c\_BW} := \frac{1}{4}$$

step 7

$$x_{a.c\_WB} := \frac{x_{a.c\_N} \cdot C_{N\alpha} + x_{a.c\_WB} \cdot C_{L\alpha WB} + x_{a.c\_BW} \cdot C_{L\alpha BW}}{C_{N\alpha} + C_{L\alpha WB} + C_{L\alpha BW}}$$

$$x_{a.c\_WB} = 0.203$$

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$\alpha_r := -6, -4 .. 20$

$$\alpha_r := \begin{pmatrix} -6 \\ -4 \\ -2 \\ 0 \\ 2 \\ 4 \\ 6 \\ 8 \\ 10 \\ 12 \\ 14 \\ 16 \\ 17.77 \\ 19 \end{pmatrix} \qquad C_{L\_WBm} := \begin{pmatrix} -.133 \\ 0.023 \\ 0.179 \\ 0.335 \\ 0.491 \\ 0.647 \\ 0.803 \\ 0.956 \\ 1.115 \\ 1.271 \\ 1.4 \\ 1.456 \\ 1.476 \\ 1.456 \end{pmatrix}$$

$C_{D\_WB}(\alpha_r)$
0.012
0.011
0.013
0.017
0.024
0.034
0.047
0.062
0.081
0.101
0.125
0.152
0.181
0.213

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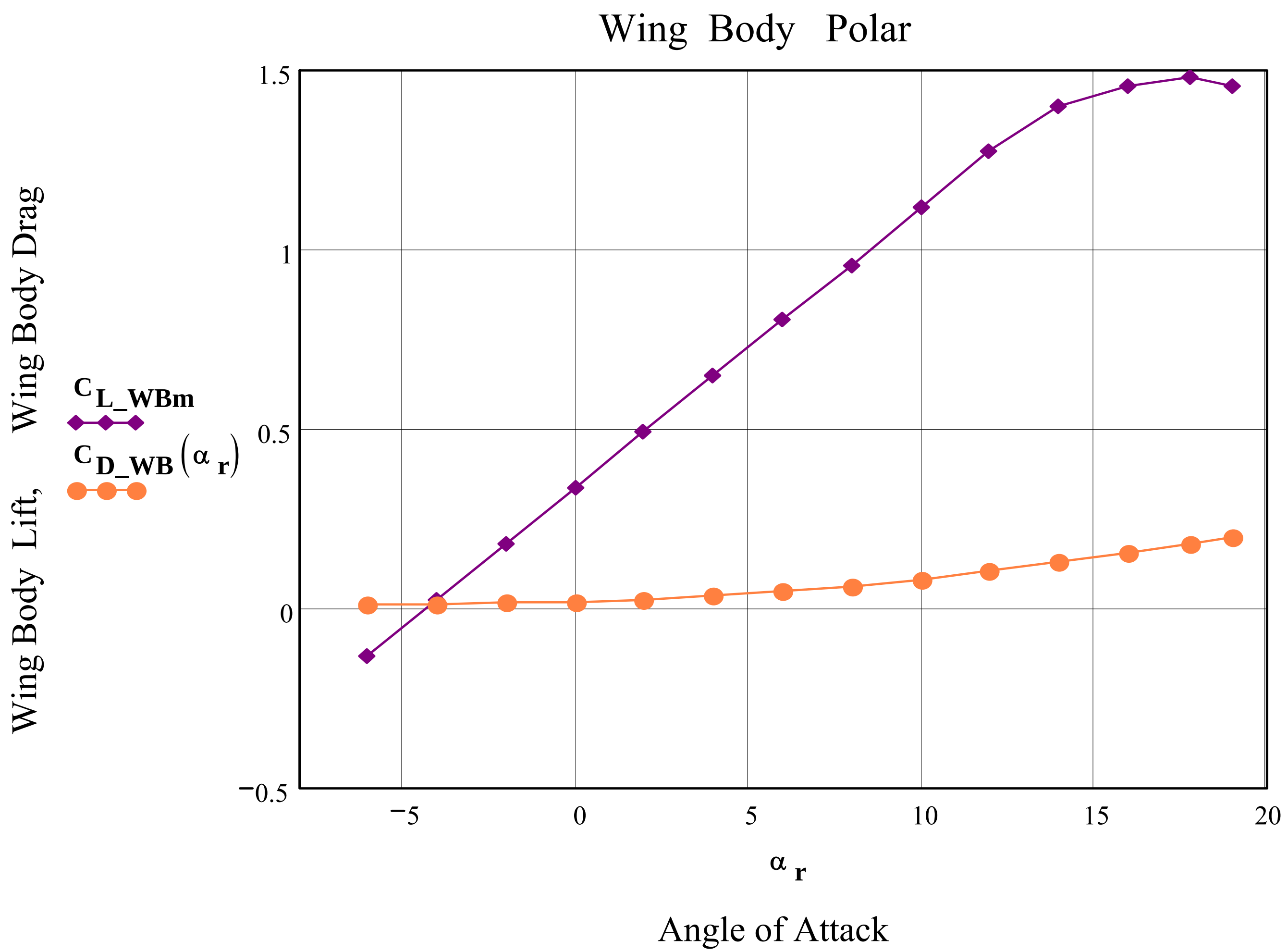
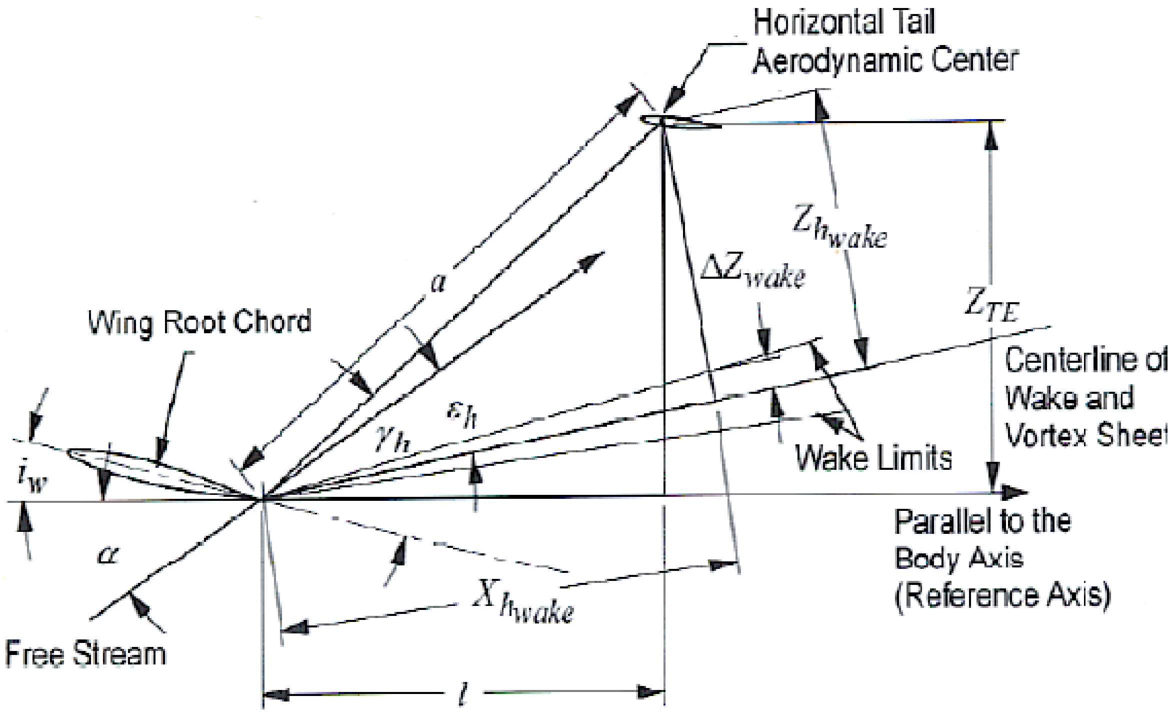

Wing Body Polar
Wing Body Lift, Wing Body Drag
C_{L_WBm}
$C_{D\_WB}(\alpha_r)$
1.5
1
0.5
0
−0.5
−5
0
5
10
15
20
α_r
Angle of Attack

5. WING BODY TAIL AERODYNAMICS

5.1 DOWNWASH



Intermediate Parameters for Dynamic Pressure Ratio Calculation

Method 2 $\quad g_2 := y_{macH} \cdot \tan\left(\frac{20}{57.3}\right) \qquad g_2 = 0.212$

$$K_A := \frac{1}{A} - \frac{1}{1 + A^{1.7}} \qquad K_A = 0.133$$

$$K_\lambda := \frac{10 - 3 \cdot \lambda}{7} \qquad K_\lambda = 1$$

$$h_H := 0.707 \qquad l_{H1} := 4.395 - x_{a.c} \cdot MAC - .83 + X_{a\_c\_H}$$

$$K_H := \frac{1 - \frac{h_H}{b}}{\left(2 \cdot \frac{l_{H1}}{b}\right)^{\frac{1}{3}}} \qquad K_H = 0.962$$

$$\delta\varepsilon_\alpha := 4.4 \cdot \left(K_A \cdot K_\lambda \cdot K_H\right)^{1.19} \qquad \delta\varepsilon_\alpha = 0.381$$

lH1 Distance between wing aerodynamic center and HT aerodynamic center

SUBSONIC DYNAMIC PRESSURE RATIO

l.H2 Distance between wing trailing edge and horizontal tail aerodynamic center
hH vertical location of horizontal tail aerodynamic center

$$l_{H2} := 4.395 - MAC - 0.83 + X_{a\_c\_H} \qquad l_{H2} = 2.454 \qquad a_{hw} := \left(l_{H2}^2 + h_H^2\right)^{0.5}$$

$$\varepsilon_w(\alpha) := \frac{1.62 \cdot C_L(\alpha)}{\pi \cdot A} \qquad \gamma := 57.3 \cdot \mathrm{atan}\left(\frac{h_H}{l_{H2}}\right) + i_w \qquad \gamma = 18.072$$

$$x_{hw}(\alpha) := a_{hw} \cdot \cos\left(\frac{\gamma - \alpha - i_w}{57.3} + \varepsilon_w(\alpha)\right)$$

$$z_{hw}(\alpha) := x_{hw}(\alpha) \cdot \tan\left(\frac{\gamma - \alpha - i_w}{57.3} + \varepsilon_w(\alpha)\right) \qquad z_w(\alpha) := 0.68 \cdot MAC \cdot \left[C_{D0} \cdot \left(\frac{x_{hw}(\alpha)}{MAC} + 0.15\right)\right]^{0.5}$$

$$\Delta q\_q0(\alpha) := \frac{2.42 \cdot C_{D0}^{0.5}}{x_{hw}(\alpha) + 0.3} \qquad \Delta q\_q(\alpha) := \Delta q\_q0(\alpha) \cdot \cos\left(\frac{\pi \cdot z_{hw}(\alpha)}{2 \cdot z_w(\alpha)}\right)^2$$

$$q\_q_{inf}(\alpha) := 1 - \Delta q\_q(\alpha) \qquad \alpha := 0 \qquad q\_q_{inf}(0) = 0.918$$

5.2 WING BODY TAIL LIFT

5.2.1 WING BODY TAIL LIFT CURVE SLOPE

In the area of HT assume the fuselage diameter zero. So the interference factors are :

$$K_{WB\_T} := 1 \qquad K_{BW\_T} := 0$$

$$C_{L\alpha\_AC} := C_{L\alpha\_WB} + C_{L\alpha\_H} \cdot K_{WB\_T} \cdot \left(1 - \delta\varepsilon_\alpha\right) \cdot q\_q_{inf}(0) \cdot \frac{S_H}{S} \qquad C_{L\alpha\_AC} = 4.816$$

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5.2.2 WING BODY TAIL LIFT IN THE NONLINEAR AOA

For high aspect ratio, unswept subsonic configuration, the flow remains atached over the lifting panels up to angles of attack approaching the stall. For this angle of attack , the aerodynamic lift characteristics are linear.

$$\alpha := -6, -4 .. 20 \qquad C_{L\_AC}(\alpha) := \frac{C_{L\alpha\_AC}}{57.3}\cdot(\alpha - \alpha_0)$$

5.2.3 WING BODY TAIL MAXIMUM LIFT

The trimmed maximum lift coefficient for a wing body tail configuration with no high lift devices. The wing body tail maximum lift is assumed to occur at the wing body angle of attack for maximum lift.

$$\alpha_{CLmax\_AC} := \alpha_{CLmax\_WB} \qquad \alpha_{CLmax\_AC} = 17.774$$

$$n := 0.25 \qquad \alpha := \alpha_{CLmax\_WB} - i_w \qquad \varepsilon_1 := \varepsilon_w(\alpha)$$

$$\alpha_3 := \alpha - \varepsilon_1\cdot 57.3$$

$$C_{m\_AC}(\alpha_3) := (n - x_{a.c})\left(C_{L\_WB}(\alpha_3)\cdot\cos\left(\frac{\alpha}{57.3} - \varepsilon_1\right) + C_{D\_WB}(\alpha_3)\cdot\sin\left(\frac{\alpha}{57.3} - \varepsilon_1\right)\right)\cdot MAC + C_{m0\_WB}$$

$$C_{L\_H\_WBV} := \frac{C_{m\_AC}(\alpha_{CLmax\_WB} - \varepsilon_1\cdot 57.3)}{\cos\left(\frac{\alpha}{57.3} - \varepsilon_1\right)\cdot\frac{l_{H1}}{MAC} + \sin\left(\frac{\alpha}{57.3} - \varepsilon_1\right)\cdot\frac{h_H}{MAC}}$$

$$C_{L\_H\_WBV} = -0.025$$

with respect to MAC/4

$$C_{L\_ACmax} := C_{Lmax\_WB} + C_{L\_H\_WBV} \qquad C_{L\_ACmax} = 1.451$$

$$\alpha_2 := \frac{\alpha}{57.3} - \varepsilon_1$$

$$C_{L\_H\_WBV2} := \frac{C_{m\_AC}(\alpha_2) + C_{D\_H}(\alpha_2)\cdot\cos(\alpha_2)\cdot\frac{h_H}{MAC} - C_{D\_H}(\alpha_2)\cdot\sin(\alpha_2)\cdot\frac{l_{H1}}{MAC}}{\cos(\alpha_2)\cdot\frac{l_{H1}}{MAC} + \sin(\alpha_2)\cdot\frac{h_H}{MAC}}$$

$$C_{L\_ACmax2} := C_{Lmax\_WB} + C_{L\_H\_WBV2}$$

$C_{L\_H\_WBV2} = -0.027$ $\qquad$ $C_{L\_ACmax2} = 1.449$

$\alpha_{rc} := 16$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) := C_{Lmax} - 0.07$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) = 1.406$

$\alpha_{rc} := 17.77$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) := C_{Lmax}$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) = 1.476$

$\alpha_{rc} := 18$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) := C_{Lmax} - 0.02$ $\qquad$ $C_{L\_AC1}(\alpha_{rc}) = 1.456$

5.3 WING BODY TAIL DRAG

5.3.1 WING BODY TAIL ZERO LIFT DRAG

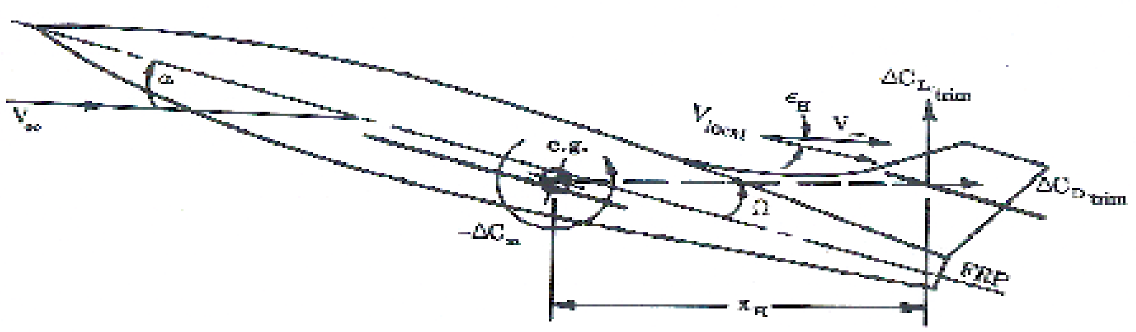
The zero lift drag of Vertical Tail is 0.5 from zero lift drag as Horizontal Tail.

$C_{D0\_AC} := C_{D0\_WB} + 0.5\cdot C_{D0\_H}$ $\qquad$ $C_{D0\_AC} = 0.013$

5.3.2 WING BODY TAIL DRAG AT ANGLE OF ATTACK

$l_t := 3.6139$ Distance from A/C center of gravity to aerodynamic center of HT

$h_H = 0.707$ The vertical location of aerodynamic center of HT



VIEW IN PLANE OF SYMMETRY

α = Airplane angle of attack (positive as shown)

x_H = Distance from c.g. to quarter-chord point of horizontal-stabilizer MAC

Ω = Angle defined by intersection of x_H with FRP (positive as shown with horizontal stabilizer above c.g.)

$$\Omega := -\alpha + 57.3\ \text{atan}\left(\frac{h_H}{l_t}\right) \qquad \Omega 1 := \frac{\Omega - \alpha + \varepsilon_1}{57.3}$$

$$S_{HW} := \frac{S_H}{S}\cdot q\_q_{inf}(0)\cdot\frac{l_t}{MAC} \qquad C_{mWB}(\alpha) := C_{m\_AC}(\alpha)$$

$$C_{L\_Htrim}(\alpha) := 2\cdot\frac{\dfrac{C_{mWB}(\alpha-\alpha_0)}{\cos(\Omega 1)} + C_{D0\_H}\cdot\tan(\Omega 1)}{1+\left[1-4\left(\dfrac{\tan(\Omega 1)}{\pi\cdot A_H\cdot e_H}\right)\left(\dfrac{C_{mWB}(\alpha-\alpha_0)}{S_{HW}\cdot\cos(\Omega 1)} + C_{D0\_H}\cdot\tan(\Omega 1)\right)\right]^{0.5}}$$

$$\Delta C_{Dtrim}(\alpha) := \left(C_{D\_H}(\alpha)\cdot\cos(\varepsilon_1) + C_{L\_Htrim}(\alpha)\cdot\sin(\varepsilon_1)\right)\cdot\frac{S_H}{S}\cdot q\_q_{inf}(0)$$

$$C_{D\_AC}(\alpha) := C_{D0\_AC} + C_{D\_WB}(\alpha) + \Delta C_{Dtrim}(\alpha)$$

$$\Delta C_{Ltrim}(\alpha) := \left(C_{L\_Htrim}(\alpha)\cdot\cos\left(\varepsilon_1\right) - C_{D\_H}(\alpha)\cdot\sin\left(\varepsilon_1\right)\right)\cdot\frac{S_H}{S}\cdot q\_q_{inf}(0)$$

$$\Delta C_{mtrim}(\alpha) := \Delta C_{Dtrim}(\alpha)\cdot\frac{l_t}{MAC}\cdot\sin\left(\frac{\Omega - \alpha}{57.3}\right) - C_{L\_Htrim}(\alpha)\cdot\frac{l_t}{MAC}\cdot\cos\left(\frac{\Omega - \alpha}{57.3}\right)$$

$$\alpha := -6, -4 .. 18$$

$$\alpha_r := \begin{pmatrix} -6 \\ -4 \\ -2 \\ 0 \\ 2 \\ 3 \\ 4 \\ 5 \\ 6 \\ 8 \\ 12 \\ 16 \\ 17.77 \\ 19 \end{pmatrix} \qquad C_{L\_ACm} := \begin{pmatrix} -.143 \\ 0.025 \\ 0.193 \\ 0.361 \\ 0.529 \\ 0.614 \\ .698 \\ 0.782 \\ .866 \\ 1.0 \\ 1.25 \\ 1.42 \\ 1.45 \\ 1.41 \end{pmatrix}$$

$C_{D\_AC}(\alpha_r) =$

	0
0	0.027
1	0.024
2	0.025
3	0.029
4	0.037
5	0.042
6	0.048
7	0.054
8	0.062
9	0.079
10	0.124
11	0.182
12	0.212
13	0.234

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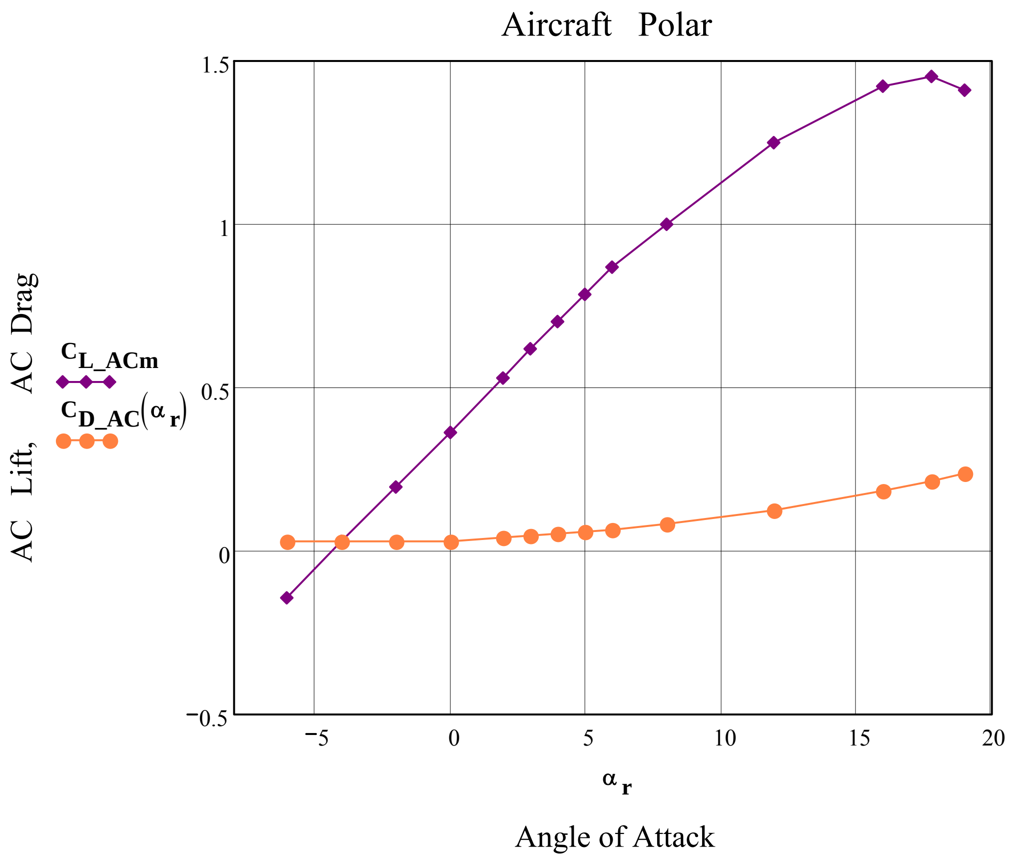

5.4 WING BODY TAIL PITCHING MOMENT CURVE SLOPE

$$C_{m\alpha AC} := -x_{a.c\_WB} \cdot MAC \cdot C_{L\alpha\_WB} - \left(l_{H1} + x_{a.c\_WB} \cdot MAC\right) \cdot C_{L\alpha\_H} \cdot \left(1 - \delta\varepsilon_{\alpha}\right) \cdot q\_q_{inf}(0) \cdot \frac{S_H}{S}$$

$C_{m\alpha AC} = -2.831$ $\qquad X_{a.c\_AC} := \frac{-C_{m\alpha AC}}{C_{L\alpha\_AC}}$ $\qquad X_{a.c\_AC} = 0.588$ m with respect LE of wing

$x_{a.c\_AC} := \frac{X_{a.c\_AC}}{MAC}$ $\qquad x_{a.c\_AC} = 0.369$ $\qquad$ % MAC

6. SINGLE SLOTTED FLAP AERODYNAMICS

Method 1 $\quad c_{l\alpha} := 0.105 \cdot 57.3$

$\delta_{f1} := 30 \qquad \alpha_{\delta 1} := -0.5 \qquad \delta_{f2} := 15 \qquad \alpha_{\delta 2} := -0.54$

$$\Delta c_{l1} := -c_{l\alpha} \cdot \alpha_{\delta 1} \cdot \frac{\delta_{f1}}{180} \cdot \pi \qquad \Delta c_{l1} = 1.575$$

$$\Delta c_{l2} := -c_{l\alpha} \cdot \alpha_{\delta 2} \cdot \frac{\delta_{f2}}{180} \cdot \pi \qquad \Delta c_{l2} = 0.851$$

MAXIMUM LIFT INCREMENT $\quad \Delta c_{lmax\_base} := \frac{30}{25} \cdot 1.67$

$k_1 := 0.83 \qquad k_{2\_30} := 0.875 \qquad k_{2\_15} := 0.6 \qquad k_3 := 1$

$$\Delta c_{lmax\_30} := k_1 \cdot k_{2\_30} \cdot k_3 \cdot \Delta c_{lmax\_base} \qquad \Delta c_{lmax\_30} = 1.455$$

$$\Delta c_{lmax\_15} := k_1 \cdot k_{2\_15} \cdot k_3 \cdot \Delta c_{lmax\_base} \qquad \Delta c_{lmax\_15} = 0.998$$

Flap effectiveness factor c.f/c 0.31 See Ref. 7

Gap open $\quad \alpha_{\delta o\_cl} := 0.54 \qquad$ Gap sealed $\quad \alpha_{\delta s\_cl} := 0.61$

$$\Delta_{gap} := \frac{\alpha_{\delta s\_cl} - \alpha_{\delta o\_cl}}{\alpha_{\delta o\_cl}} \qquad \Delta_{gap} = 0.13$$

The lift increment developed by deflection of a control surface is given by:

$\alpha_{\delta\_cl} := 0.65$ $\alpha_{\delta\_CL\_cl} := 1.055$ $b_f := 1.96$

$\eta_i := \frac{Rmin \cdot 2}{b}$ $\eta_i = 0.078$ $K_{B1} := 0.1$

$\eta_f := \eta_i + \frac{b_f \cdot 2}{b}$ $\eta_f = 0.541$ $K_{B2} := 0.64$

$K_B := K_{B2} - K_{B1}$ $K_B = 0.54$ $C_{L\alpha} = 4.266$

$\Delta C_{L1} := \Delta c_{l1} \cdot \frac{C_{L\alpha}}{c_{l\alpha}} \cdot \alpha_{\delta\_CL\_cl} \cdot K_B$ $\Delta C_{L1} = 0.636$ at 30°

$\Delta C_{L1g} := \Delta C_{L1} \cdot (1 - \Delta_{gap})$ $\Delta C_{L1g} = 0.554$

$\Delta C_{L2} := \Delta c_{l2} \cdot \frac{C_{L\alpha}}{c_{l\alpha}} \cdot \alpha_{\delta\_CL\_cl} \cdot K_B$ $\Delta C_{L2} = 0.344$ at 15°

$\Delta C_{L2g} := \Delta C_{L2} \cdot (1 - \Delta_{gap})$ $\Delta C_{L2g} = 0.299$

Maximum lift coefficient with 30 ° flap deflection is :

$C_{Lmax\_flap} := C_{L\_ACmax} + \Delta C_{L1g}$ $C_{Lmax\_flap} = 2.005$

AEROLAB SRL
Via Lazzaretto, 6/A
21013 Gallarate (VA) – ITALY
www.aerolab.it

7. CONCLUSIONS

The purpose of this report is to calculate the aerodynamic data needed to find out the Flight Envelope, the Balancing and Maneuvering Loads . The main data calculated here are:

$C_{L\alpha\_WB} = 4.467$ Wing body lift curve slope

$C_{L\alpha\_H} = 2.914$ Horizontal tail lift curve slope

$x_{a.c} = 0.244$ Wing aerodynamic center location (%MAC)

$x_{a.c\_AC} = 0.369$ Aircraft aerodynamic center location (%MAC)

$C_{L\_ACmax} = 1.451$ Aircraft maximum lift (no flap)

$\alpha_{CLmax} = 17.774$ Stall incidence °

$C_{Lmax\_flap} = 2.005$ Aircraft maximum lift with 30° flap deflection