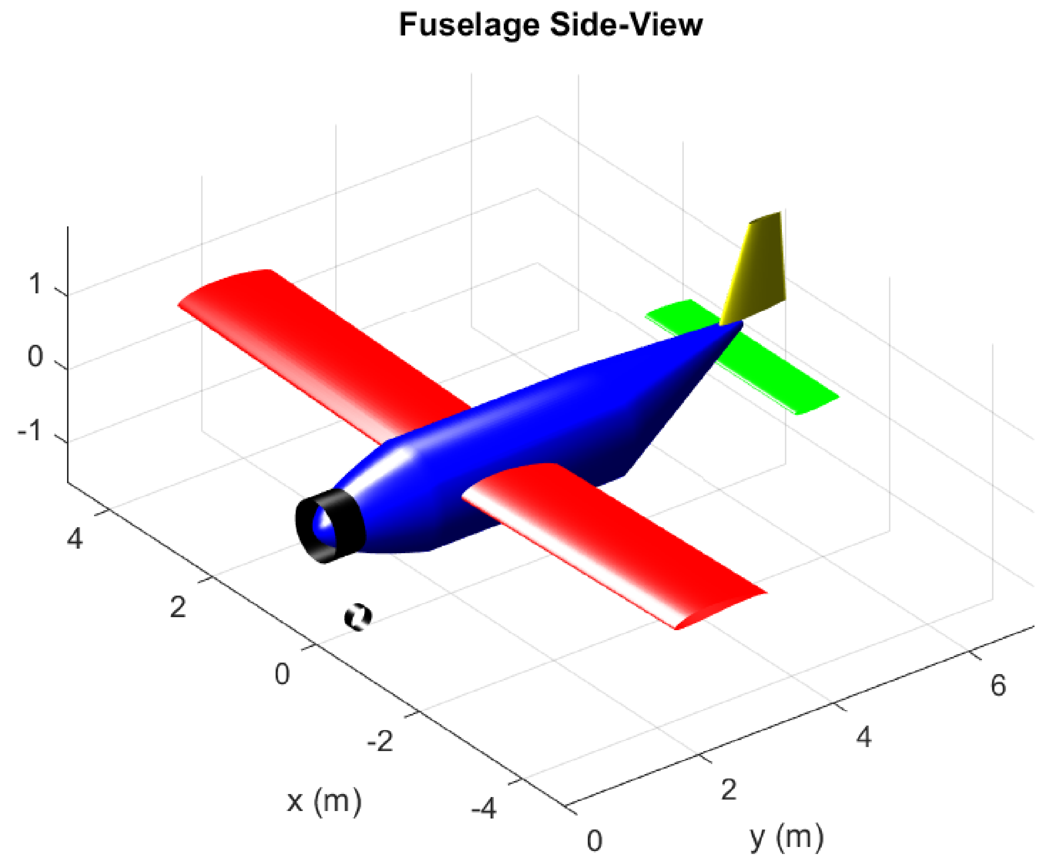
Flight Loads: TecnamP92 aircraft



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Chapter 1. Introduction

This document defines the SUBPART C - Structure - Flight Loads of the:TecnamP92. The boundaries of the flight envelope will be defined within this document. All speeds are calibrated airspeeds (CAS) (requirement 4.4 [1]) and given in knots if not stated otherwise. All other units used are metric (SI units). The weights are given in mass units (kg) but the formulas require force units as input, therefore these are calculated in place wherever they are used. Note: The speeds defined within this document should be used for the placards, speed markings, aeroplane flight manual (limitations), load calculations and need to be verified by flight test.

Chapter 2. References

1. ASTM F2245-12d,” ASTM.“ASTM F2245-12d, ASTM.
2. ABCD-FL-57-00 Wing Load Calculation, EASA.
3. ISO 2533:1975, International Standardization Organization, 1975.
4. CS-LSA Certification Specifications and Acceptable Means of Compliance, Amnd.1 29.Jul.2013, EASA, 2013.
5. “ABCD-FTR-01-00 Flight Test Report,” EASA.
6. L. Smith, “NACA technical note 1945, ‘Aerodynamic characteristics of 15 NACA airfoil sections at seven Reynolds numbers from 0.7x10E6 to 9x10E6,” 1949.
7. ABCD-WB-08-00 Weight and Balance Report, EASA.
8. Schrenk, Technical Memorandum 948 - A SIMPLE APPROXIMATION METHOD FOR OBTAINING THE SPANWISE LIFT DISTRIBUTION NACA, 1940
9. NACA Report No.824, Summary of Airfoil Data, NACA, 1945
10. Report 751 - The Mean Aerodynamic Chord and the Aerodynamic Center of a Tapered Wing, NACA, 1942.

HERE BELOW AN EXAMPLE OF REFERENCES TO BE EDITED

Chapter 3. List of Abbreviations

- **c.g.** = centre of gravity;
- **l.e.** = leading edge;
- **i.s.a.** = international standard atmosphere;
- **CSVLA** = certification specification for very light airplane;
- **MTOW** = maximum takeoff weight;
- **MTOM** = maximum takeoff mass;
- $W_{MTOM} = g * MTOM$ = weight at maximum takeoff mass;
- V_S = clean configuration stall speed;
- V_{S_0} = landing configuration stall speed;
- V_{S_1} = takeoff configuration stall speed;
- $C_{L,MAX_{clean}}$ = clean configuration maximum lift coefficient;
- $C_{L,MAX_{Landing}}$ = landing configuration maximum lift coefficient;
- $C_{L,MAX_{Takeoff}}$ = takeoff configuration maximum lift coefficient;
- ρ_0 = air density at sea level;
- S = wing area;
- V_A = manoeuvring airspeed;
- $n_{max/min}$ = maximum/minimum load factor as prescribed by regulations;
- V_F = flaps operating airspeed;
- V_{FE} = maximum flap extension airspeed;
- V_C = design cruising airspeed;
- $\frac{M}{S}$ = wing loading $\left[\frac{\mathrm{kg}}{\mathrm{m}^2}\right]$;
- V_H = maximum level flight airspeed;
- V_D = design dive airspeed;
- V_{DF} = demonstrated dive airspeed;
- $V_{C_{min}}$ = required minimum design cruising airspeed;
- V_{NE} = never exceed airspeeds;
- h = altitude;
- ρ = air density according to standard atmosphere $\left[\frac{\mathrm{kg}}{\mathrm{m}^3}\right]$;
- p = pressure according to standard atmosphere [Pa];
- T = temperature according to standard atmosphere [K];

- $a \;=\;$ speed of sound according to standard atmosphere $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$;
- $K_g \;=\;$ gust factor $\;=\; \dfrac{0.88 \cdot \; \mu_g}{5.3 \;+\; \mu_g}$
- $\mu_g \;=\;$ mass factor $\;=\; \dfrac{2 \cdot \; \frac{M}{S}}{\rho \;\cdot\; \overline{C} \;\cdot\; C_{L_\alpha}}$
- $C_{L_\alpha} \;=\;$ slope of the normal force coefficient curve $\;C_N \left[\frac{1}{\mathrm{rad}}\right]$;
- $\overline{C} \;=\;$ mean geometric chord;
- $x_{CG} \;=\;$ distance to aircraft centre of gravity;
- $x_{AC_{f+w}} \;=\;$ distance to wing – fuselage combination aerodynamic centre;
- $x_{P_{f+w}} \;=\;$ distance to wing – fuselage combination centre of pressure;
- $x_{HT} \;=\;$ distance to horizontal tail quarter chord line;
- $l_t \;=\;$ distance between wing aerod. centre and horizontal tail aerod. centre;
- $T \;=\;$ thrust vector;
- $L_{wb} \;=\;$ wing body lift vector;
- $D_{wb} \;=\;$ wing body drag vector;
- $M_{wb} \;=\;$ wing body pitching moment;
- $N_{wb} \;=\;$ wing body normal force component;
- $A_{wb} \;=\;$ wing body axial force component;
- $C_N \;=\;$ normal force coefficient;
- $C_A \;=\;$ axial force coefficient;

ADD HERE list of abbreviations as a formatted table....to be created

Chapter 4. Aircraft data

The aircraft geometrical, masses, inertial and aerodynamic data, useful for flight loads estimation are summarized in this chapter.

4.1. Geometry

The aircraft reference geometrical characteristics are summarized in the following tables.Wing parameters

Table 4.1. Wing parameters

Wing parameters	Value	Measure unit
b	9.62	m
S	13.4	m^2
AR	6.9063	Non dimensional
taper	NaN	-
sweep_first	0	deg
sweep_second	0	deg
sweep_third	0	deg
sweep_location	0	Percentage
secondary_sweep_location	0	Percentage
croot	1.4	m
ctip	1.4	m
xle	1.638	m
yle	0	m
zle	0.165	m
xtip_le	NaN	% fuselage length
dihedral_first	1.5	deg
dihedral_second	1.5	deg
dihedral_third	1.5	deg
mac	1.4	m
xmac	NaN	% fuselage length
ymac	NaN	% semispan
ypos	NaN	% semispan
zpos	NaN	% fuselage diameter
camberloc	0.4	Chord percentage
thickchord	0.12	Chord percentage
type	Rectangular	flag
twist_angle_first	-1.5	deg

Wing parameters	Value	Measure unit
twist_angle_second	-1.5	deg
twist_angle_third	-1.5	deg
twist_angle_fourth	-1.5	deg
dihedral	1.5	deg
panel_span1	0.33	Semispan percentage
panel_span2	0.33	Semispan percentage
panel_span3	0.33	Semispan percentage
twist_location	0.25	Chord percentage
chord_kink_one	1.4	m
chord_kink_two	1.4	m
camber	0.049	Chord percentage
twist_angle	-1.5	deg
mgc	1.3929	m
taper_ratio	1	Non dimensional

Table 4.2. Horizontal Tail parameters

Horizontal parameters	Value	Measure unit
S	1.97	m^2
l	3.78	m
camber	0	Chord percentage
camberloc	0.2	Chord percentage
thickchord	0.12	Chord percentage
twist	0	deg
twistloc	0.25	Chord percentage
xloc0	1.49	m
xloc	6	m
yloc	0	m
zloc	0.15	m
xrot	0	m
yrot	0	m
zrot	0	m
b	2.9	m
ctip	0.68	m
croot	0.68	m
sweep	0	deg
sweeploc	0	Chord percentage
secsweeploc	1	Chord percentage

Horizontal parameters	Value	Measure unit
dihedral	0	deg
location_of_camber	0.2	Chord percentage

Table 4.3. Vertical Tail parameters

Vertical parameters	Value	Measure unit
xle	0.95	Fuselage length percentage
croot	1	m
ctip	0.476	m
xtip_le	1	m
b	1.23	m
zpos	1	Fuselage length percentage
S	0.72	m^2
chord	0.68	m
sweep	20	deg
sweeploc	0	Chord percentage
secsweeploc	0	Chord percentage
dihedral	0	deg
twist	15	deg
twistloc	0	Chord percentage
MAC	0.23354	m
l_vt	3.2	m
empennage_flag	Single fin	NaN
xloc	0	m
yloc	0	m
zloc	0	m
xloc_percentage	0.95	Fuselage length percentage
yloc_percentage	0	Fuselage length percentage
zloc_percentage	1	Fuselage length percentage

Table 4.4. Fuselage parameters

Fuselage parameters	Value	Measure unit
length	6.3	m
diameter	1.4	m
Non_dim_radius_of_gyration	NaN	Non dimensional
Radius_of_gyration	NaN	m
width	1.1	m

Table 4.5. Elevator parameters

Elevator parameters	Value	Measure unit
S	1.972	m^2
chord	0.68	m
chord_ratio_ce_c	1	Non dimensional
overhang	0.12	Non dimensional
span_ratio	NaN	Non dimensional
S_hinge	1.4775	m^2
eta_inner	0	Non dimensional
eta_outer	1	Non dimensional
cf_c_inner	1	Non dimensional
cf_c_outer	1	Non dimensional
b	2.9	m
ca	1.4	m
cb	1.4	m
y_outer	1.45	m
y_inner	0	m
cf	0.5984	m
cf_outer	1.4	m
cf_inner	1.4	m
moment_arm	0.0884	m
ce_c_root	1	Non dimensional
ce_c_tip	1	Non dimensional

Table 4.6. Rudder parameters

Rudder parameters	Value	Measure unit
S	0.16339	m^2
chord	0.3321	m
chord_ratio_cf_c	0.48	Non dimensional
overhang	0.12	Non dimensional
span_ratio	0.8	Non dimensional
b	1	m
eta_inner	0.1	Non dimensional
eta_outer	0.9	Non dimensional
cr_c_root	0.45	Non dimensional
cr_c_tip	0.45	Non dimensional
cf	0	m

Rudder parameters	Value	Measure unit
ca	0	m
cb	0	m
y_inner	0.0615	m
y_outer	0.5535	m
croot	0.45	m
ctip	0.2142	m
moment_arm	0.043173	m

Table 4.7. Aileron parameters

Aileron parameters	Value	Measure unit
S	0.76845	m^2
b	1.97	m
ca	0.39004	m
cb	0.39004	m
y_inner	2.4199	m
y_outer	4.3901	m
eta_inner	0.5031	Non dimensional
eta_outer	0.9127	Non dimensional
ca_c_root	0.2786	Non dimensional
ca_c_tip	0.2786	Non dimensional
moment_arm	-Inf	m
ca_root	0.39004	m
ca_tip	0.39004	m
croot	0.39004	m
ctip	0.39004	m
cf	0	m

Table 4.8. Flaps parameters

Flaps parameters	Value	Measure unit
S	0.21073	m^2
b	1.365	m
ca	0.39004	m
cb	0.39004	m
y_inner	0.52	m
y_outer	4.8215	m
eta_inner	0.054054	Non dimensional
eta_outer	0.5012	Non dimensional

Flaps parameters	Value	Measure unit
cf_c_root	0.2786	Non dimensional
cf_c_tip	0.2786	Non dimensional
croot	0.39004	m
ctip	0.39004	m
cf	0.39004	m
max_deflection	25	deg
moment_arm	NaN	m

4.2. Masses and inertia

The aircraft reference masses and inertia are summarized in this subsection

The Aircraft masses and inertia are summarized in Table: Weight parameters

Table 4.9. Weight parameters

Weight	Value	Measure unit
W_maxTakeOff	450	kg
W_OperativeEmpty	NaN	kg
W_Payload	NaN	kg
W_Fuel	NaN	kg
W_Crew	NaN	kg
IY	100	kg * m^2
W_EmptyMass	281	kg
W_UsefulMass	169	kg
W_FuelMass	50.5	kg
W_CrewMass	180	kg
Oil_mass	1.85	kg

TABLE TO BE CHECKED!!!

4.3. Aerodynamic

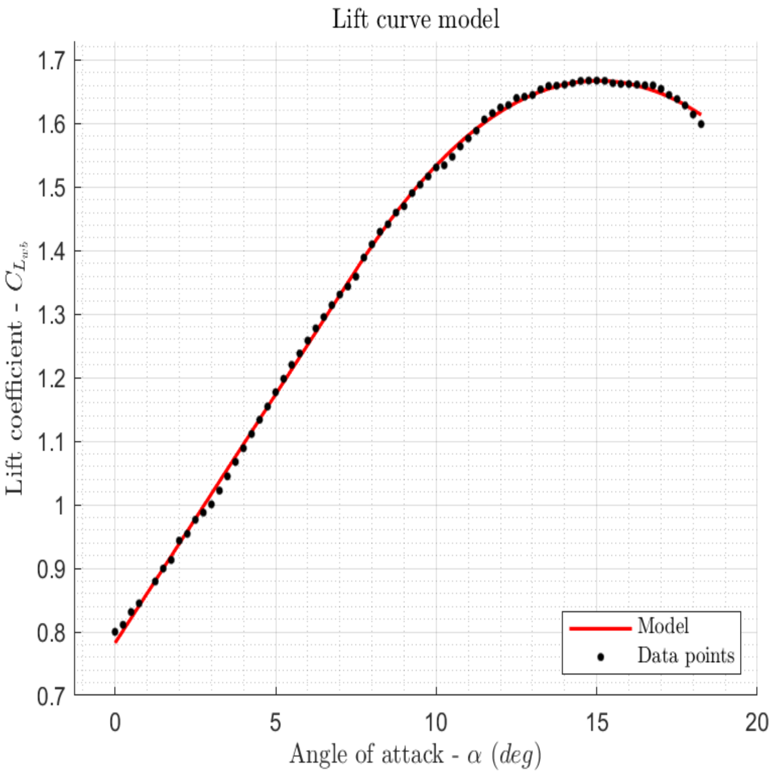

Figure 4.1. Lift coefficient of the 3D wing-body configuration.

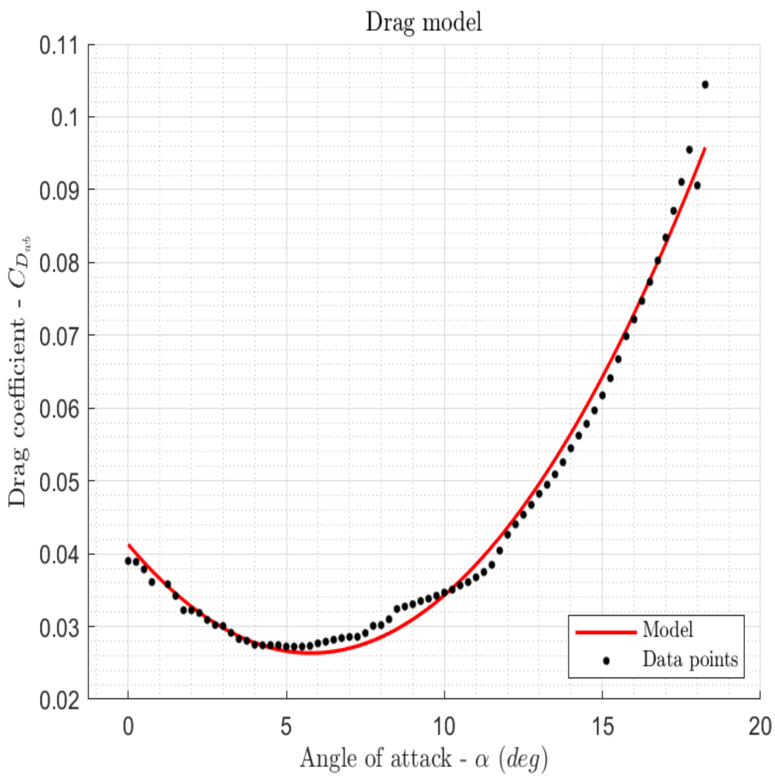

Figure 4.2. Drag coefficient of the 3D wing-body configuration.

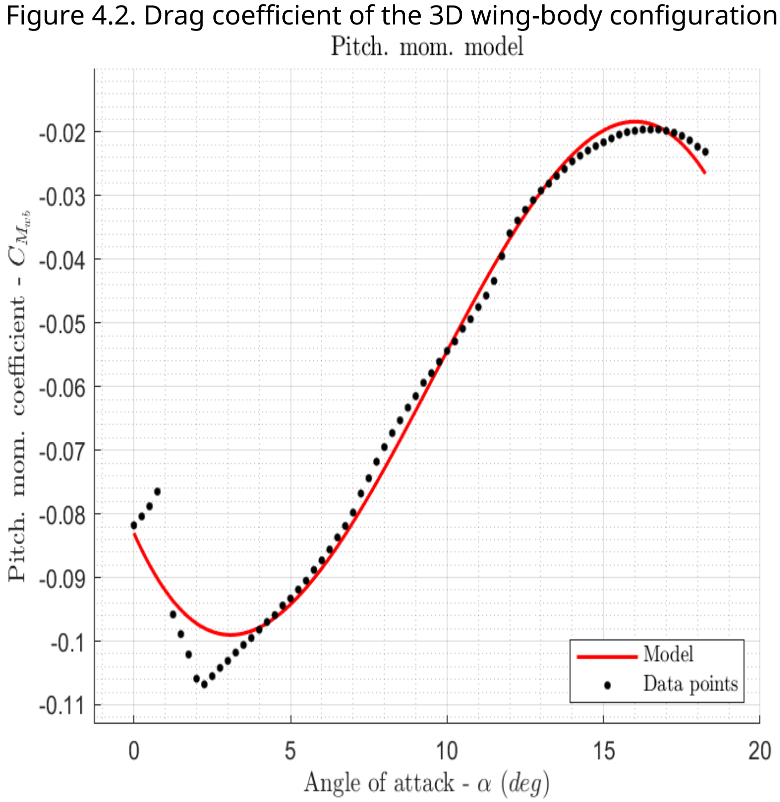

Figure 4.3. Pitch moment coefficient of the 3D wing-body configuration.

The aircraft reference aerodynamic is shown in figure: Wing-Body reference Aerodynamics

The previous figures refer to wing-body aerodynamics, obtained via computational fluidynamics simulations. Full-vehicle aerodynamics is not available for this aircraft. The calculations illustrated in this document are based on an aerodynamic reference model, obtained from wing-body aerodynamics through interpolation (red curves in figure). The interpolation procedure is in good agreement with present data. Further examination of this data through wind-tunnel or in-flight tests must be performed.

Table 4.10. Aerodynamic reference values.

Quantity	Value	Units
CLMAX	1.55	Non dimensional
CLMAX inverted	-1	Non dimensional
CLMAX takeoff	1.9	Non dimensional
CLMAX landing	2.2	Non dimensional
CL0 wb	0.7825	Non dimensional
CLALFA wb (rad)	4.36	1/rad
CLALFA wb (deg)	0.07823	1/deg
CD0	0.035	Non dimensional
CD wb landing gear	0.01	Non dimensional
CM0 wb	-0.01654	Non dimensional
CM wb landing gear	0	Non dimensional
CMCL wb	-0.1697	Non dimensional
Alfa wb zero lift	-10	deg
e	0.8	Non dimensional

Chapter 5. Design Airspeeds

Chapter 6. Altitude

Chapter 7. Manoeuvring and Gust load factors n

Chapter 8. V-n Envelope

The final flight envelope diagram showed in the above figure was drawn considering limit load factor values related to the maximum wing stress condition. The determination of wing loads for structural design is important to establish structural adequacy and to assess structural weight for the aircraft. Aeroelasticity effects will be discounted; this is acceptable for relatively slow, small aircraft. From this diagram, the designer can obtain load envelopes in terms of shear, bending moment and torsion. The diagram is referred to sea level density and the airworthiness rules applied are relative to csvla. The selected operational altitude is 2000 m. According to csvla 321, each critical altitude expected during normal operations and each critical weight and centre of gravity positions has been examined. Symmetrical flight maneouvring conditions are defined as flight maneouvres for which the aircraft is to be designed that does not involve motion about the roll or yaw axis. Except where limited by maximum static lift coefficients, the airplane is assumed to be subjected to symmetrical maneouvres resulting in the limit maneouvring load factors. The aeroplane is in steady-state flight condition, where the pitching acceleration is assumed negligible.

(Note: the csvla requirement does not impose to take into account for altitude changes. Calculating the loads at sea level would be acceptable.

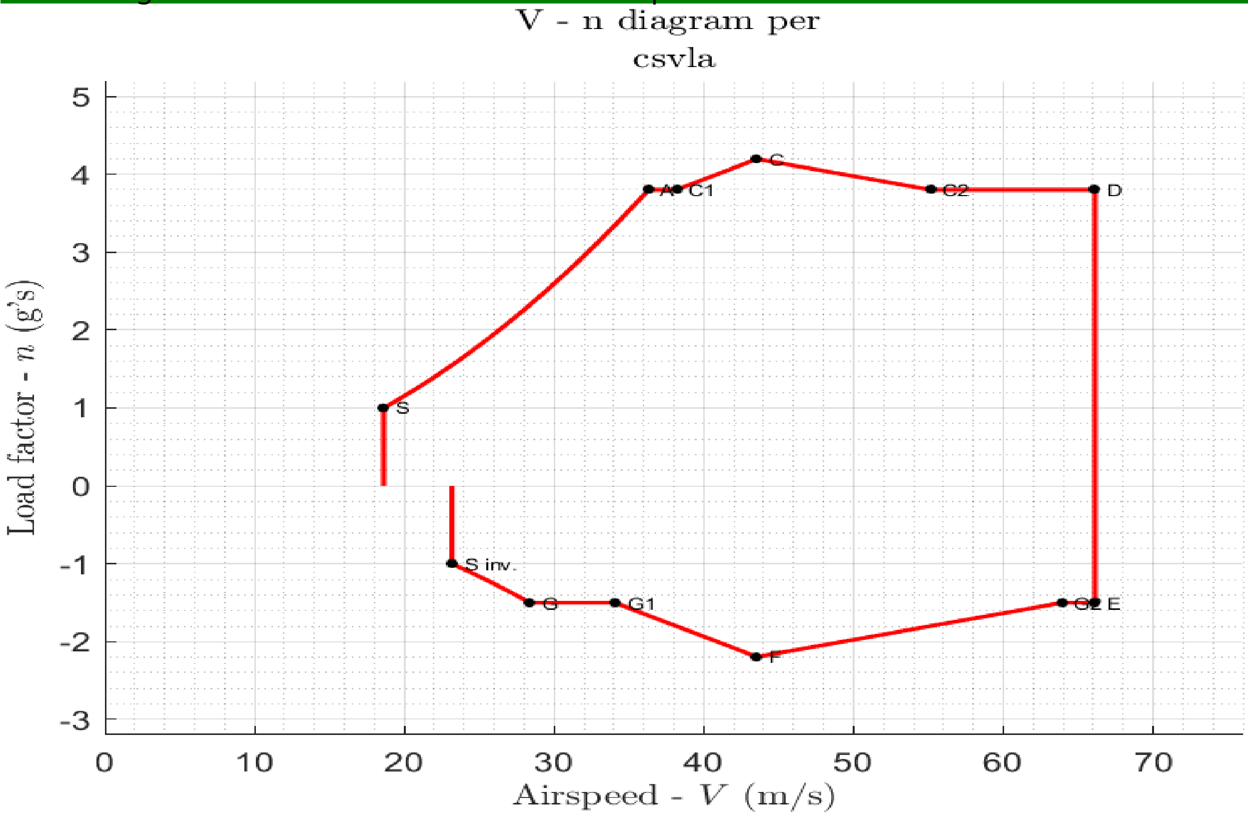

Figure 8.1. Maneuver and Gust load factors diagram

The figure shows the final flight envelope, resulting from superposition of manoeuvring and gust loads, according to csvla 333(d) , obtaining the limit combined loads diagram.

For the calculation of structural design speeds, the stalling speeds Vs0 and Vs1 should be taken to be the 1-g stalling speeds in the appropriate flap configuration.

According to csvla, the load factor n varies linearly from points at cruise speed VC (or VF for inverted flight) to points at dive speed VD (D or E for inverted flight) as shown in figure.

The gust load factors are determined from the previous equation, known as the "gust formula"; the gust alleviation factor Kg was determined as an empirical function of airplane mass ratio mug. This analysis method is considered acceptable for relatively slow, small, light airplane; more complicated gust analysis requires a statistical approach.

TABLE TO BE CHECKED!!!

Table 8.1. Final envelope points

ID	V(m/s)	n
S	18.62	1
A	36.31	3.8
C	43.55	4.195
D	66.11	3.8
S_inv	23.19	-1
G	28.4	-1.5
F	43.55	-2.195
E	66.11	-1.5

Chapter 9. Loads on the aeroplane

Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety). Unless otherwise provided, prescribed loads are limit loads. Unless otherwise provided, the air, ground, and water loads must be placed in equilibrium with inertia forces, considering each item of mass in the aeroplane. These loads must be distributed to conservatively approximate or closely represent actual conditions.

9.1. Reference axes and sign convention

In the figure is represented the reference frame used to project forces and moment acting on the aircraft structures. The origin is located at the airplane axis of symmetry (x axis) with the y axis passing through the leading edge of the mean aerodynamic chord section of the wing.

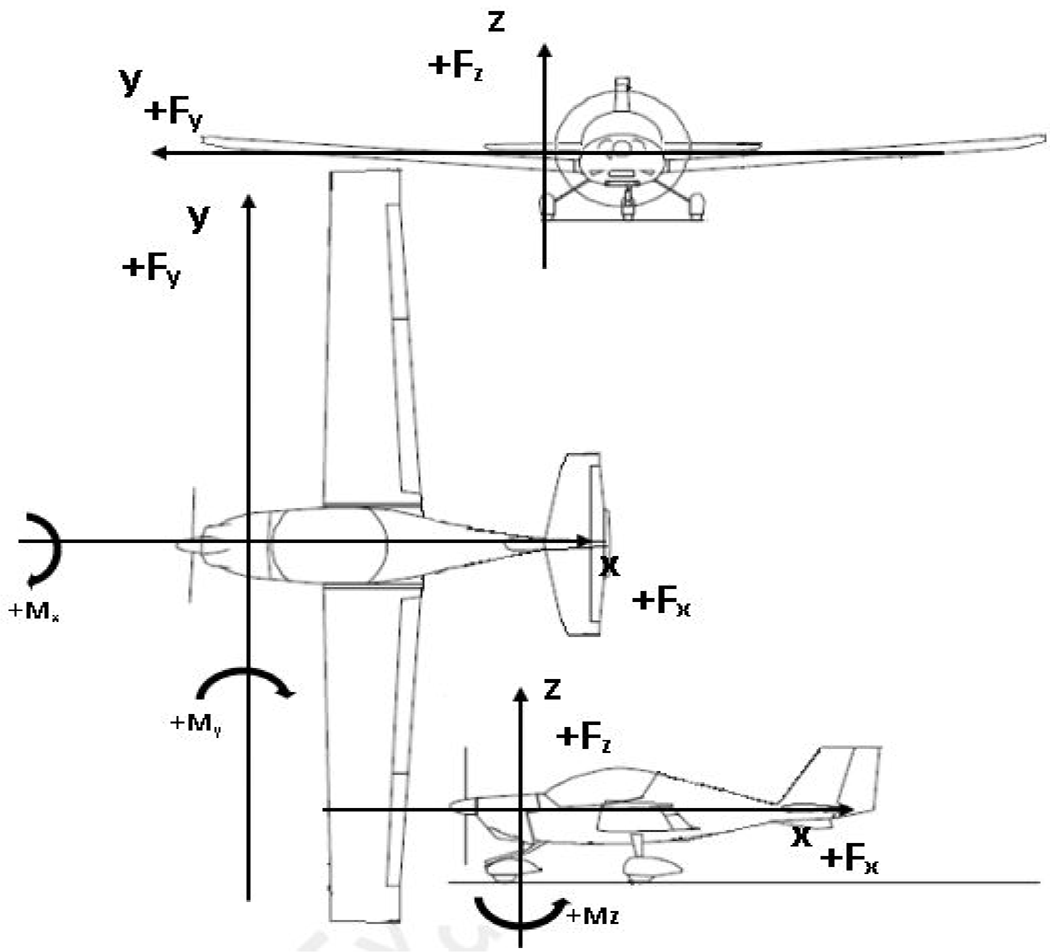

Figure 9.1. Reference axis

9.1.1. Sign conventions and symbols

Sign conventions and symbols used are summarized as follows:

- x = longitudinal axis of the aircraft;
- y = lateral axis of the aircraft;
- z = vertical axis of the aircraft;
- M_x = total rolling moment;
- M_y = total pitching moment;
- M_z = total yawing moment;
- F_x = toal axial force;
- F_y = total lateral force;
- F_z = total normal force;

9.2. Symmetrical flight conditions

The external forces and moments acting on the aeroplane in a balanced flight condition have been determined. The simplified scheme in figure is considered. The aeroplane is reduced to the wing and the horizontal tail. The symbols used are:

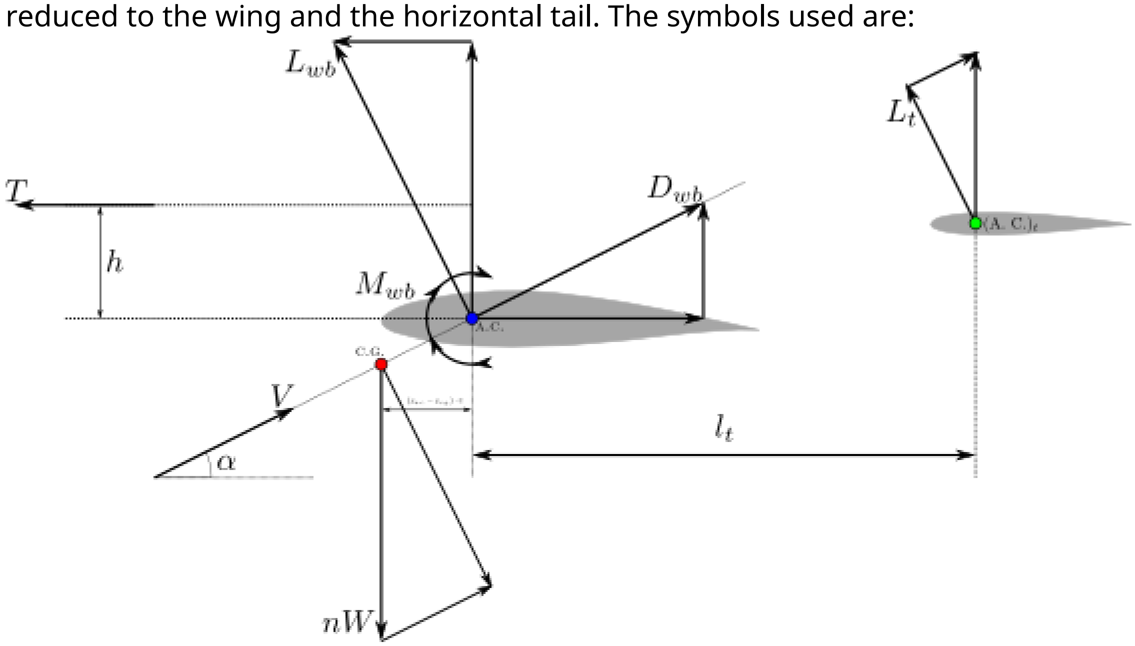

Figure 9.2. Simplified equilibrium of the aircraft

- x_{CG} = distance to aircraft centre of gravity;
- $x_{AC_{f+w}}$ = distance to wing fuselage combination aerodynamic centre;
- $x_{P_{f+w}}$ = distance to wing fuselage combination centre of pressure;
- nW = aircraft total weight force;

- $x_{HT} \; = \;$ distance to HT quarter chord line;

Assuming positive forces and moment as depicted in the figures and remembering that equilibrium along x and y axes are automatically fullfilled, it is possible to write the following equilibrium equations:

$$\text{X equilibrium}: \quad T \; = \; L_{wb} \cdot \; \sin\alpha \; - \; D_{wb} \cdot \; \cos\alpha$$

$$\text{Z equilibrium}: \quad L_{wb} \cdot \cos\alpha \; + \; D_{wb} \cdot \sin\alpha \; + \; \frac{L_t}{\cos\alpha} \; - \; nW \cdot \cos\alpha \; = \; 0$$

$$\text{Rotation about } \; Y:$$

$$[-L_{wb} \cdot \cos\alpha \cdot (x_{AC} \; - \; x_{CG}) \; - \; L_{wb} \cdot \sin\alpha \cdot \; z_{CG}] \; + \; [D_{wb} \cdot \cos\alpha \cdot \; z_{CG} \; - \; D_{wb} \cdot \sin\alpha \cdot \; mgc \cdot (x_{AC} \; - \; x_{CG})]$$

$$+ \left[-\frac{L_t}{\cos\alpha} \cdot (l_t \; + \; mgc \cdot (x_{AC} \; - \; x_{CG})) \right] \; - \; T \cdot \; h \; + \; M_{wb} \; = \; 0$$

Table 9.1. Balance parameters.

Parameter	Value	Unit of measure
xcg	NaN	m
xcg/MAC	0.25	% MAC
xac/MAC	0.25	Non dimensional
xht	3.78	m
zcg	NaN	m
h	0	m
mgc	1.393	m

9.3. Aerodynamic centre

The aerodynamic centre of the wing-body aircraft was fixed at 25% of the mean aerodynamic chord. This assumptions is use in all calculations relative to forces and moments. Also, the maximum takeoff weight of the aircraft has been assumed.

9.4. Pitching moment of the wing-body

The pitching moment coefficient of the wing-body aircraft was determined by high fidelity computational fluidynamics. The results are depicted in the following figure.

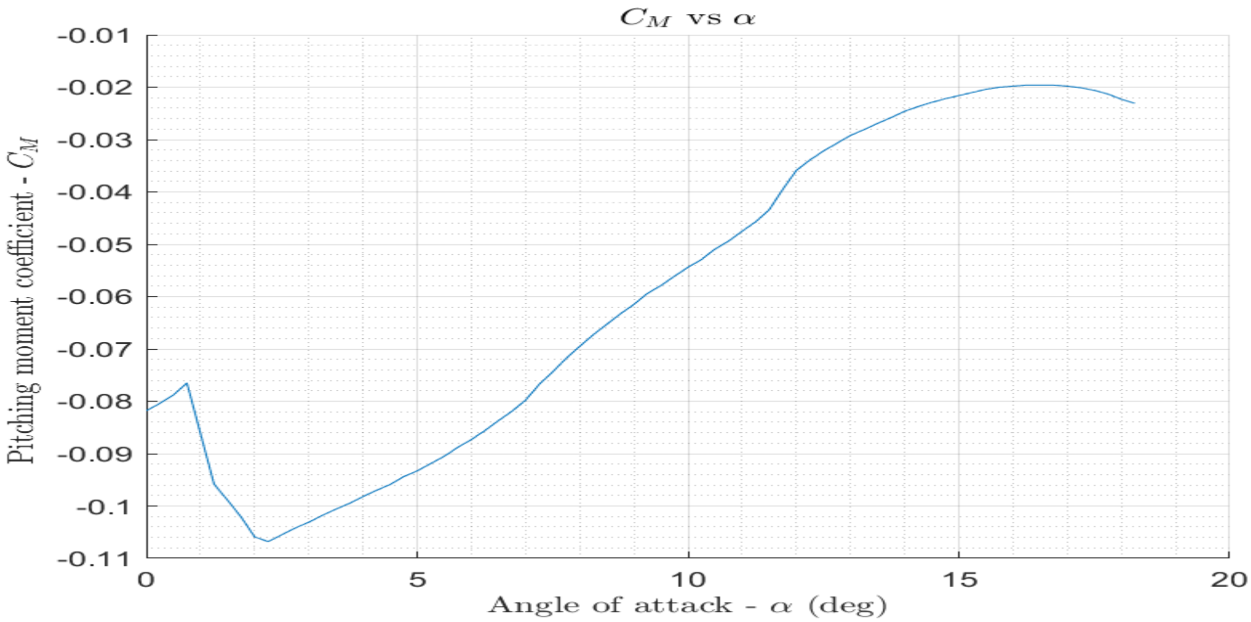

Figure 9.3. Wing-body pitching moment coefficient.

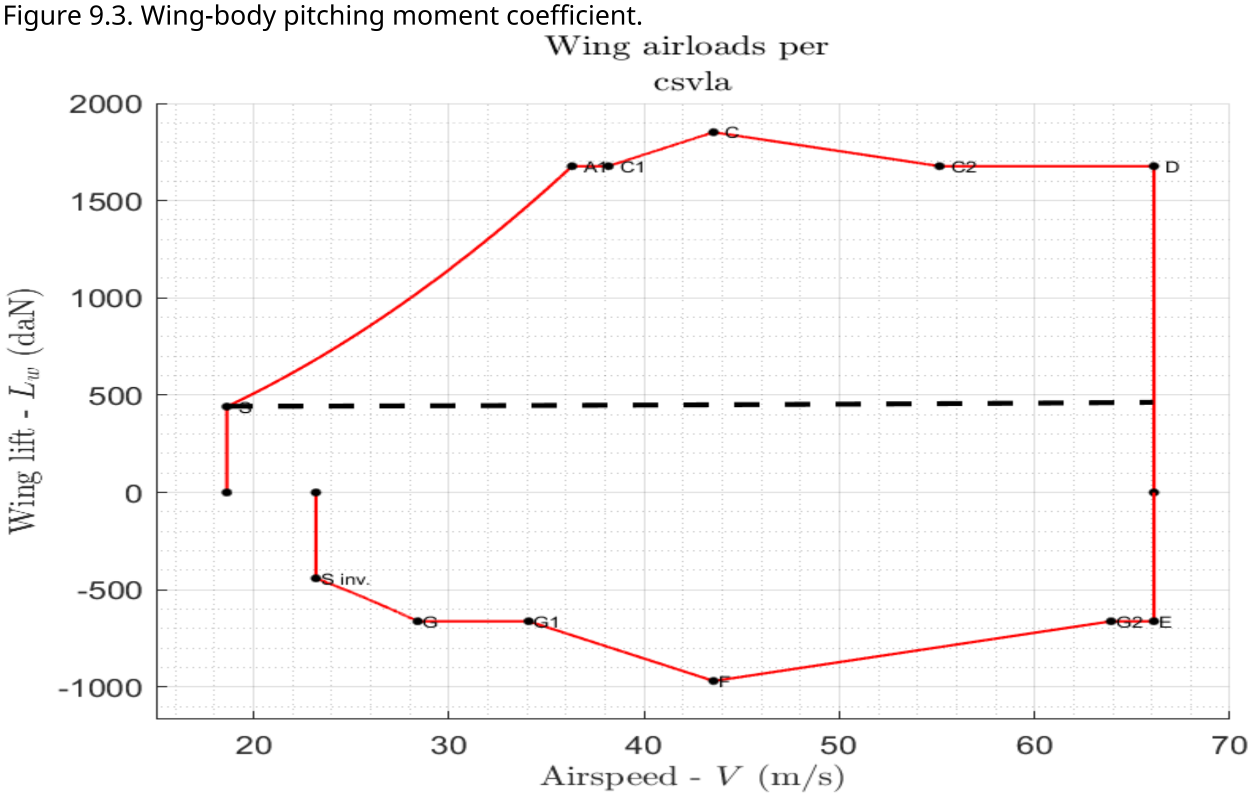

Figure 9.4. Wing airloads

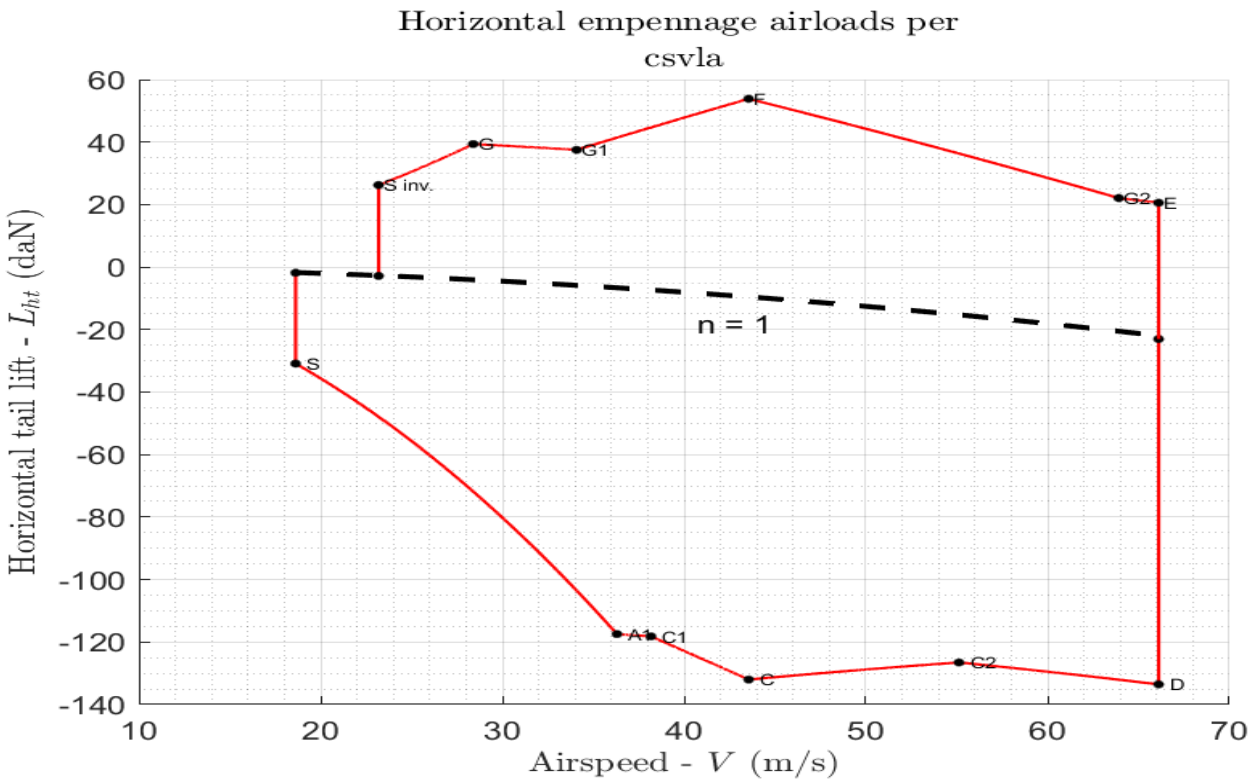

Figure 9.5. Balancing loads

9.5. Wing airloads

Wing airloads are depicted in figure; these are the airloads necessary to size wing structures and they comes from the equilibrium equations applied on the full aircraft. Mainly, these loads come from lift, during a symmetric, straight, non-accelerated flight condition, including lift that results from horizontal tail negative lift.

9.6. Complete aircraft balancing loads

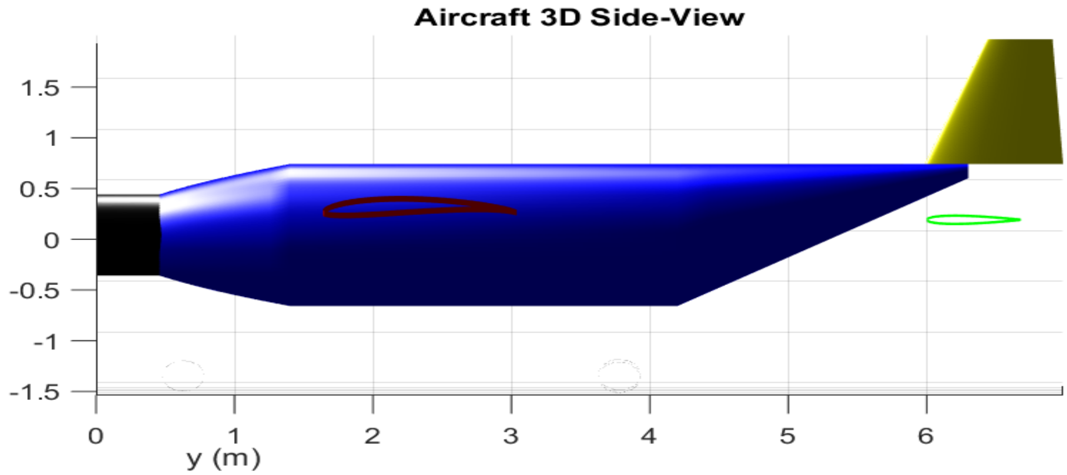

Figure 9.6. Aircraft 3D sideview.

Here, the aircraft balancing loads are collected inside a table. A horizontal surface balancing load is a load necessary to maintain equilibrium in any condition with no pitching acceleration. Determination of the balancing loads is necessary both for the calculation of the wing loads and of the balancing loads acting on the horizontal tail surface itself.

Table 9.2. Flight envelope points.

Point	V (m/s)	n (g's)	alfa (deg)	CL	CL wb	CL tail	L (daN)	L wb (daN)	L tail (daN)
Point S	18.62	1	13.75	1.55	1.659	-0.1086	441.3	472.2	-30.91
Point A	36.31	3.8	13.75	1.55	1.659	-0.1086	1677	1794	-117.4
Point C	43.55	4.195	6.279	1.189	1.274	-0.08478	1851	1983	-132
Point D	66.11	3.8	-3.552	0.4675	0.5047	-0.03723	1677	1810	-133.5
Point Sinv	23.19	-1	-14.9	-1	-1.059	0.05947	-441.3	-467.5	26.24
Point G	28.4	-1.5	-15.66	-1	-1.059	0.05947	-661.9	-701.3	39.37
Point F	43.55	-2.195	-10.51	-0.6222	-0.6568	0.03458	-968.8	-1023	53.84
Point E	66.11	-1.5	-4.55	-0.1845	-0.1903	0.005737	-661.9	-682.5	20.58

Chapter 10. Loads on the wing

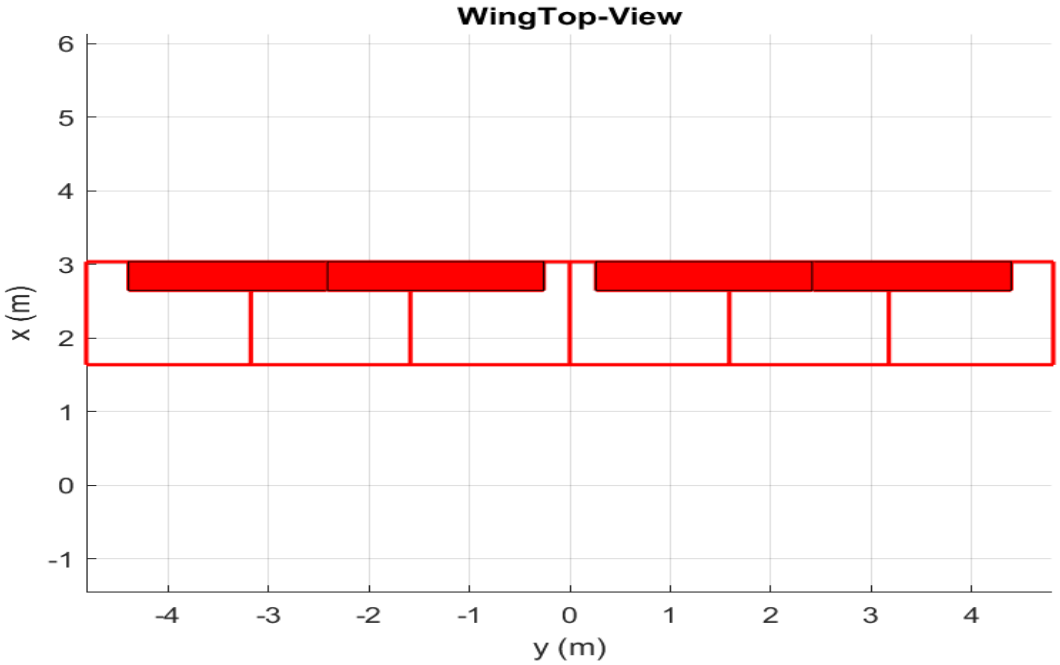

Figure 10.1. Wing, top view.

In this section will be shown all the resulting internal forces acting on the wing structural elements; having calculated lift, drag and pitching moment coefficient distribution on the wing with a panel method and the geometrical chord distribution, it is possible to evaluate normal and shear forces and pitching moment distributions along the wing span.

10.1. Influence of the fuselage

The effects of the fuselage on the wing span lift distribution cause a reduction of lift at stations near the wing root; this lift reduction can be discounted because is often negligible, leading to a more conservative design loads. On the other hand, its influence on the aeroplane equilibrium is accounted for, in particular on the pitching moment distribution.

10.2. Forces and moments acting on the wings

Numerical and graphical results from internal forces and moments calculations will be shown in this section.

10.2.1. SpanWise Airloads Distribution

Spanwise airloads distributions along the wing semi-span are obtained from a panel method; then, an interpolation through all the values of the angle of attack is performed. Results are represented in the following figures.

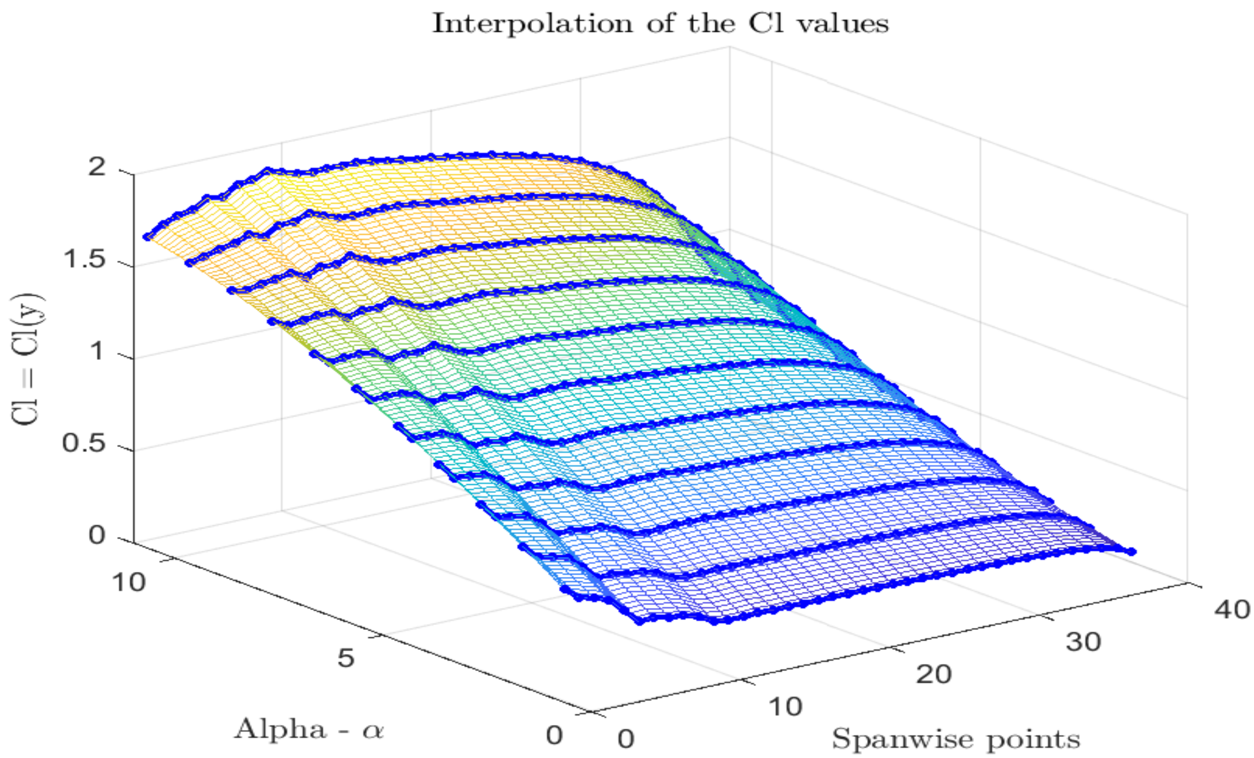

Figure 10.2. Wing lift coefficient spanwise distribution

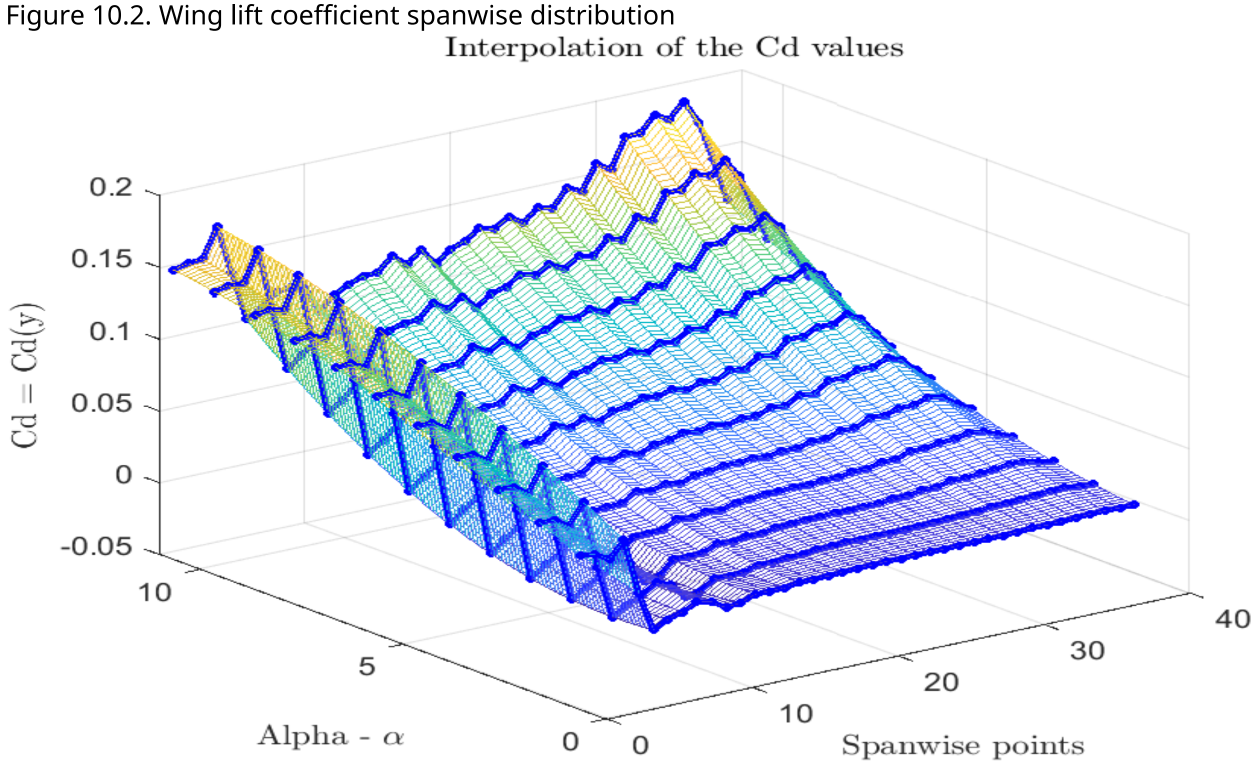

Figure 10.3. Wing drag coefficient spanwise distribution

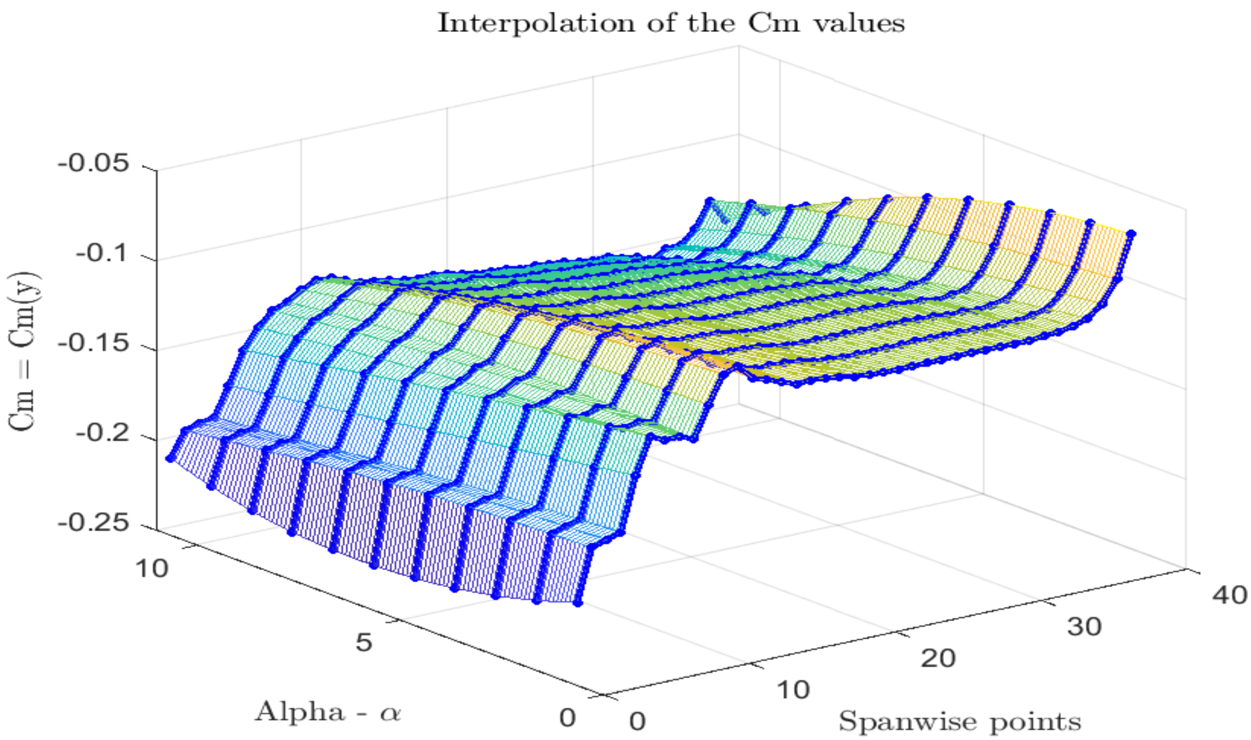

Figure 10.4. Wing pitching moment coefficient (0.25mac) spanwise distribution

10.2.2. Normal and parallel component

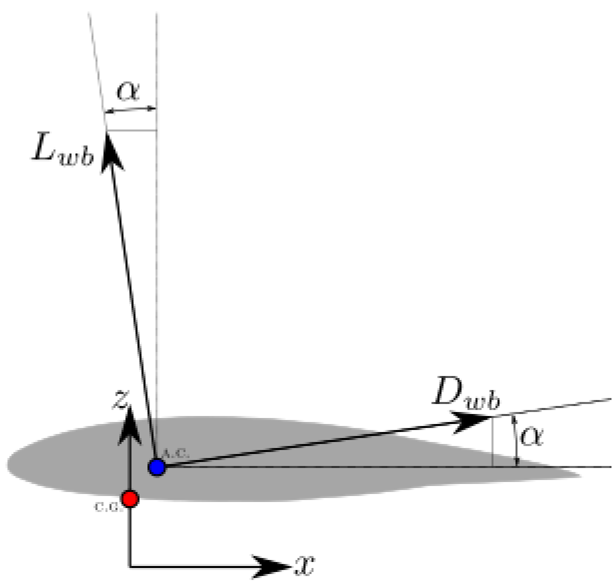

Figure 10.5. Normal and axial forces decomposition.

In order to determine the wing structural loads, the aerodynamic forces, Lift and Drag, are divided into their Normal and Parallel components in respect of reference longitudinal axis, as depicted in the figure. The wing spars and all structural component are normal or parallel to this axes which the wing angle of attack is referred too.

Normal and Axial decomposition :

$$N_{wb} = L \cdot \cos\alpha + D \cdot \sin\alpha$$

$$A_{wb} = D \cdot \cos\alpha - L \cdot \sin\alpha$$

In non dimensionalized form, the decomposition become the following:

$$C_N = C_L \cdot \cos\alpha + C_D \cdot \sin\alpha$$

$$C_A = C_D \cdot \cos\alpha - C_L \cdot \sin\alpha$$

10.2.3. Shear, Bending and Torsion

Shear, bending and torsion along the wing semi-span are shown in the following figures; these distributions are also reported inside a table, for each flight condition.

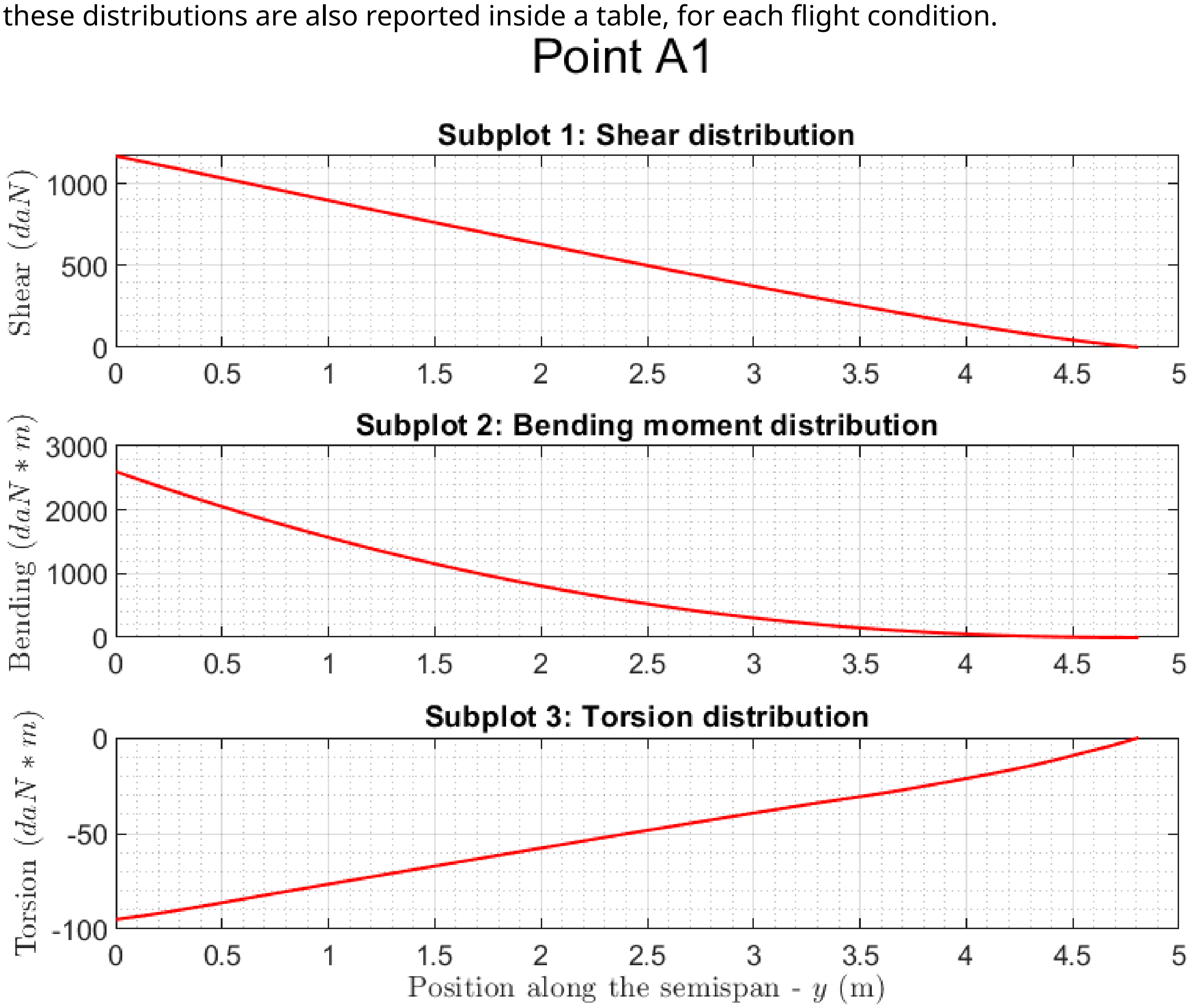

Figure 10.6. Shear, Bending and Torsion due to airloads - POINT A1

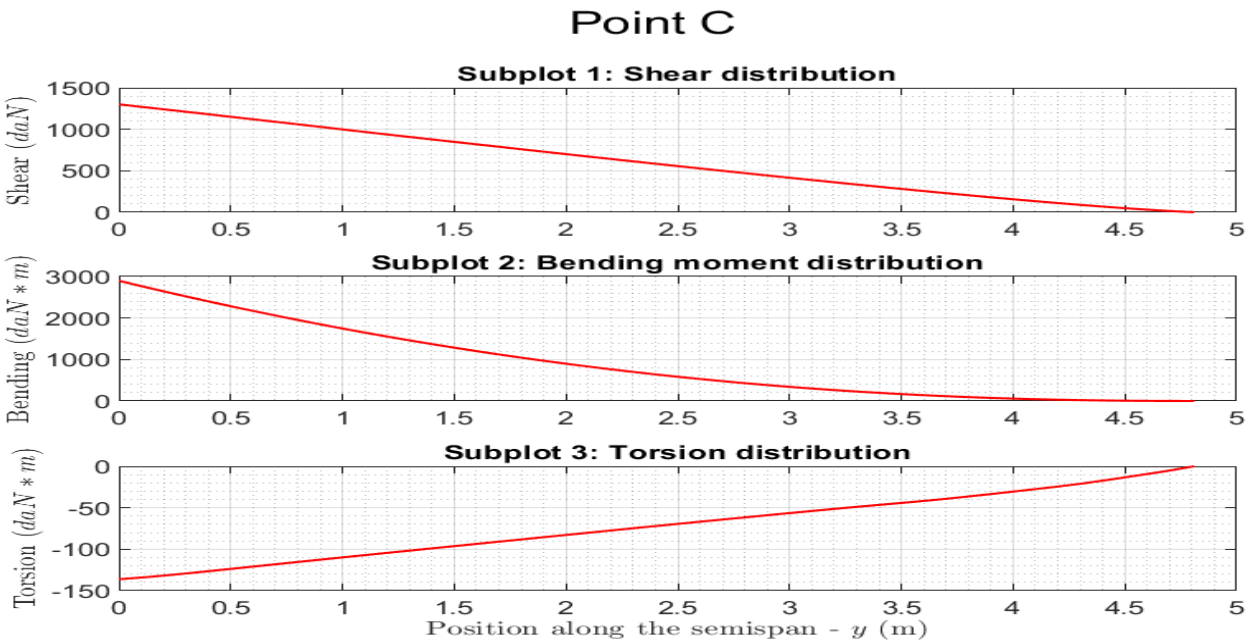

Figure 10.7. Shear, Bending and Torsion due to airloads - POINT C

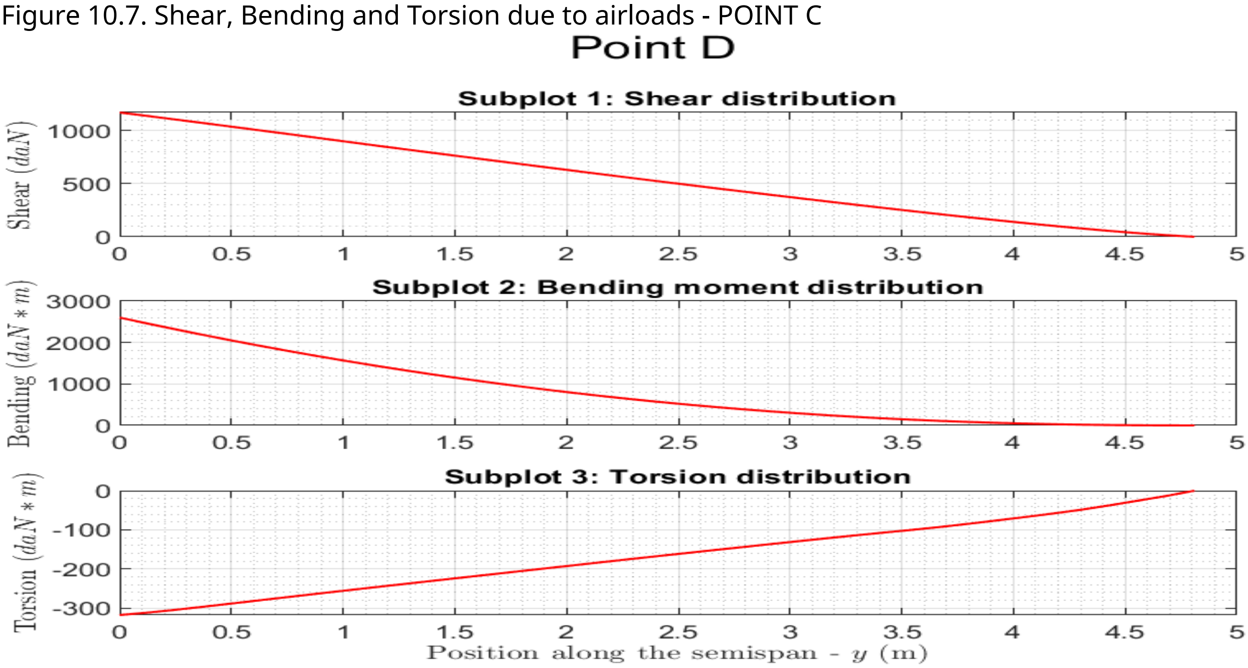

Figure 10.8. Shear, Bending and Torsion due to airloads - POINT D

To evaluate the internal forces needed to the structural sizing of the aircraft wing, a summation along the wing semi span is performed. First, axial and normal components of the aerodynamic forces are calculated from the semi-span distributions of lift and drag coefficients, which are used to evaluate shear, bending and torsion dimensional distributions.

$$S_i = S_{i-1} + s$$

where

$$s = \frac{1}{2} \cdot (N_{i-1} + N_i) \cdot (y_{i-1} - y_i)$$

where N is the normal force acting on the wing at the span station y. For the bending moment distribution, the calculations are very similar: Torsion is calculated via the same summation, but with torsion couple along the wing semi-span:

$$BM_i = BM_{i-1} + b$$

$$b = \frac{1}{2} \cdot (S_{i-1} + S_i) \cdot (y_{i-1} - y_i)$$

Torsion is calculated via the same summation, but with torsion couple along the wing semi-span:

$$M_{y_i} = C_{M_i} \cdot q \cdot (c_{y_i})^2$$

where the chords are evaluated at every semi-span station. The summation is performed in the same way as did with shear and bending distributions.

Table 10.1. Shear, bending and torsion distribution along the semi-span.

Point	V (m/s)	n (g's)	S (daN)	M (daN*m)	T (daN*m)
Point S	18.62	1	307.2	684.4	-24.97
Point A	36.31	3.8	1167	2601	-94.89
Point C	43.55	4.195	1300	2897	-136.1
Point D	66.11	3.8	1168	2601	-318
Point Sinv	23.19	-1	-300.6	-669.9	39.95
Point G	28.4	-1.5	-465.5	-1037	58.16
Point F	43.55	-2.195	-675.1	-1505	141.6
Point E	66.11	-1.5	-469.5	-1046	318.6

From the table is evident that Point A is critical for shear and bending, while torsion is critical at points D and E.

10.2.4. Critical load condition

Critical values of internal forces and moments are relative to the root station of the wing.

- The critical shear is at point C and is equal to 1300 daN.
- The critical bending is at point C and is equal to 2897 daN*m.
- The critical torsion is at point E and is equal to 318.6 daN*m.

Table 10.2. Critical shear, bending and torsion.

Point	Value	Units	Load
Point C	1300	daN	Shear
Point C	2897	daN*m	Bending
Point E	318.6	daN*m	Torsion

In the table these values are summarized, for convenience.

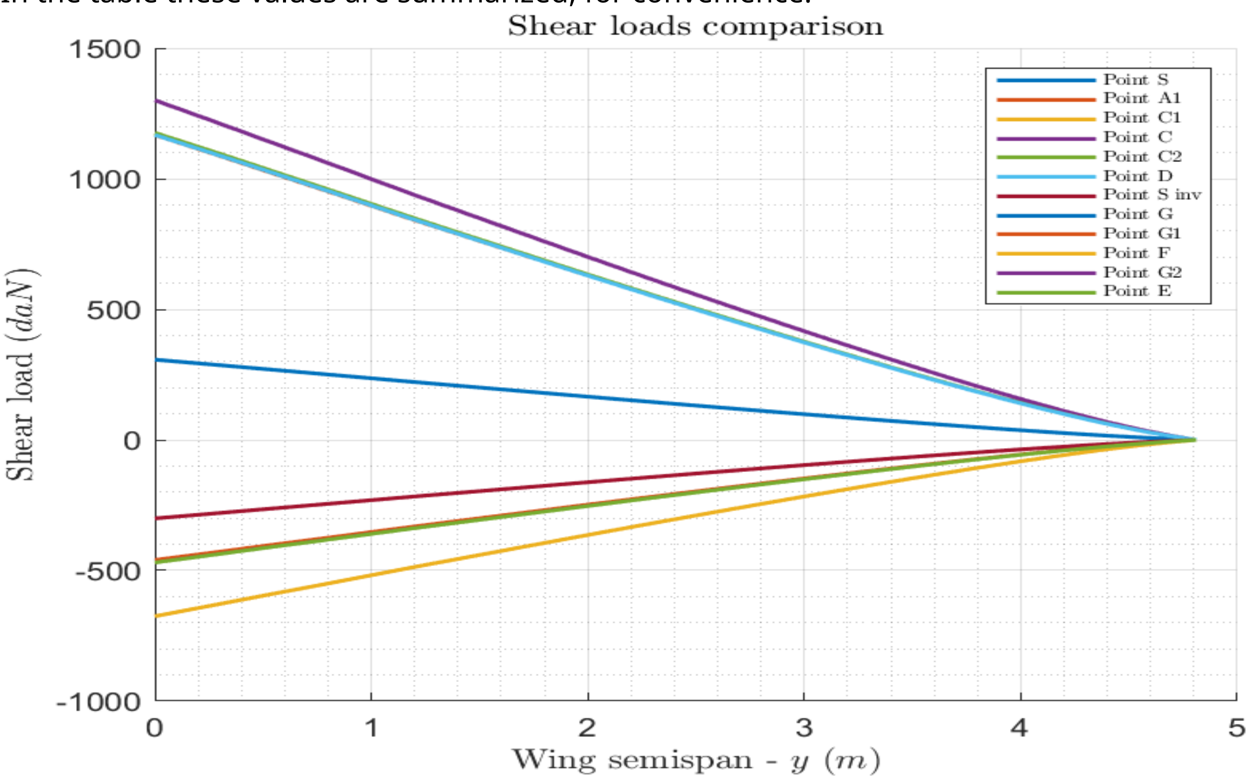

Figure 10.9. Shear comparison

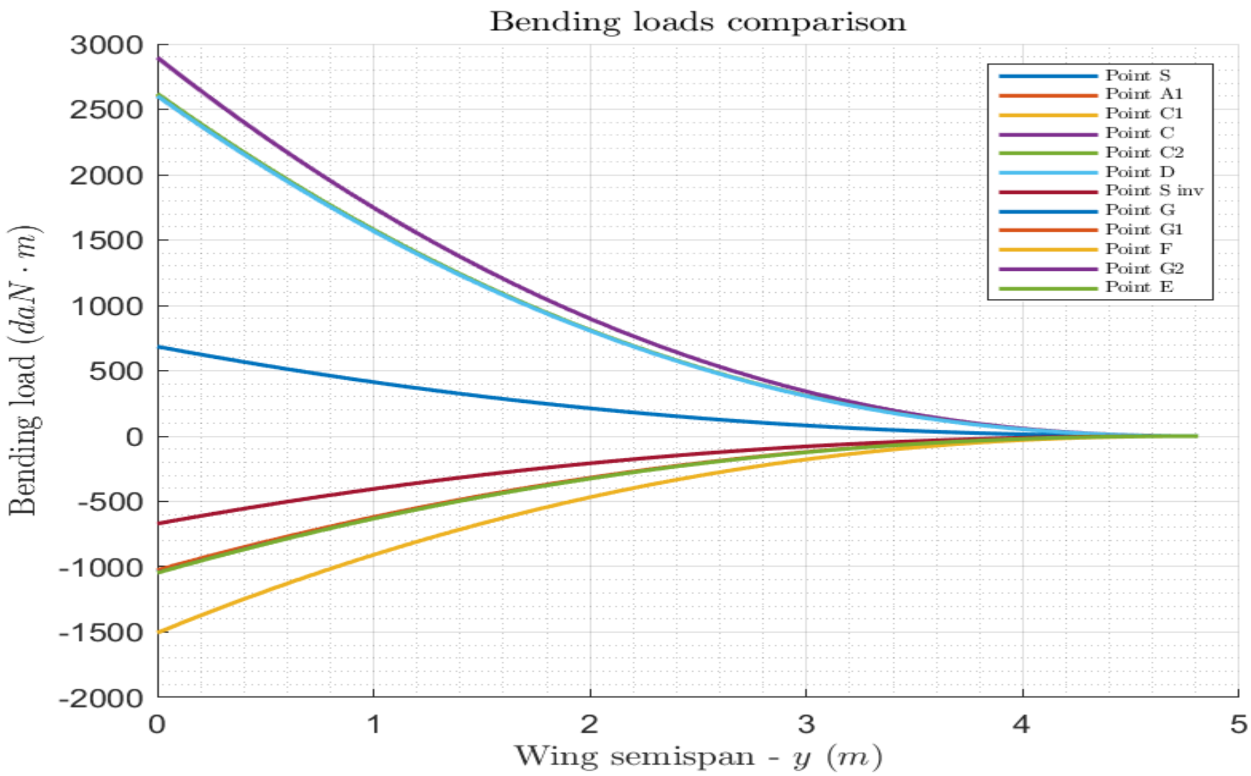

Figure 10.10. Bending comparison

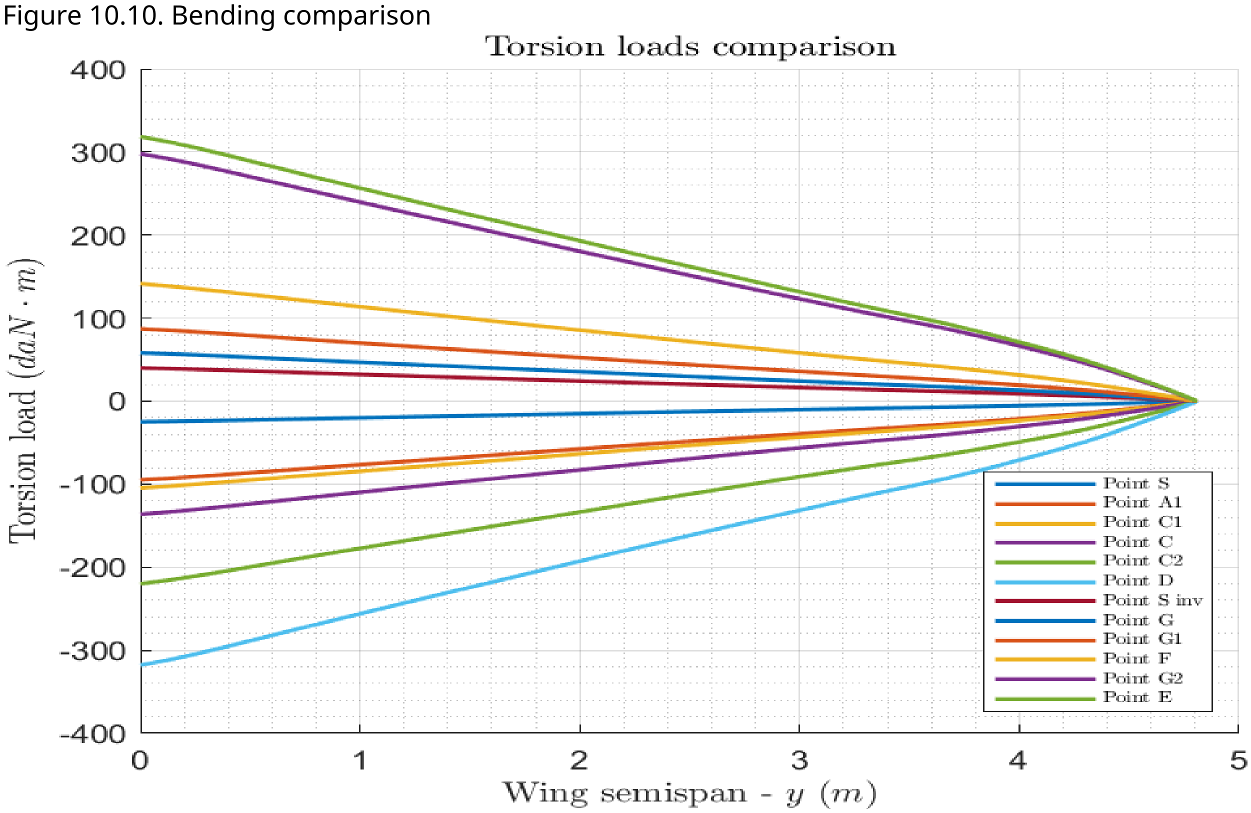

Figure 10.11. Torsion comparison

10.3. Unsymmetrical loads

According to csvla 455 , the wing and wing bracing must be designed for the following loading conditions:

1. Unsymmetrical wing loads. Unless the following values result in unrealistic loads, the rolling accelerations may be obtained by modifying the symmetrical flight conditions in csvla 333(d) follows: in condition A, assume that 100% of the semispan wing airload acts on one side of the aeroplane and 70% of this load acts on the other side.
2. Aileron deflection. The loads resulting from the aileron deflections and speeds specified in csvla 455 , in combination with an aeroplane load factor of at least two thirds of the positive manoeuvring load factor used for design. Unless the following values result in unrealistic loads, the effect of aileron displacement on wing torsion may be accounted for by adding the following increment to the basic aerofoil moment coefficient over the aileron portion of the span in the critical condition determined in csvla 333(d) .

10.3.1. Rolling condition

According to csvla 349 , the aileron displacement cause a significant change of pitching moment distribution along the wing span; these changes are shown in the following diagrams for different fligh condition in red. Unless the following values result in unrealistic loads, the effect of aileron displacement on wing torsion may be accounted for by adding the following increment to the basic aerofoil moment coefficient over the aileron portion of the span in the critical condition determined in csvla 333(d) :

$$\Delta\, C_m \;=\; (-0.01) * \delta_{aileron}$$

where:

- $\Delta\, C_m \;=\;$ pitching moment coefficient increment;
- $\delta_{aileron} \;=\;$ down aileron deflection in degrees at critical condition.

The aileron deflection at critical condition must be given in degrees.

Table 10.3. Unsymmetrical flight conditions. Pitching moment at various flight conditions.

y(m)	cmA	cmC	cmD
0	-0.2275	-0.227	-0.2288
0.3797	-0.1799	-0.1796	-0.181
0.7595	-0.1349	-0.1345	-0.136
1.139	-0.1171	-0.1169	-0.118
1.519	-0.1086	-0.1082	-0.1099

y(m)	cmA	cmC	cmD
1.899	-0.1153	-0.115	-0.1163
2.278	-0.1161	-0.1158	-0.1171
2.658	-0.3696	-0.3277	-0.1665
3.038	-0.37	-0.3281	-0.167
3.418	-0.3703	-0.3284	-0.1675
3.797	-0.3717	-0.3296	-0.1689
4.177	-0.3755	-0.3332	-0.1734
4.557	-0.1158	-0.1152	-0.1178

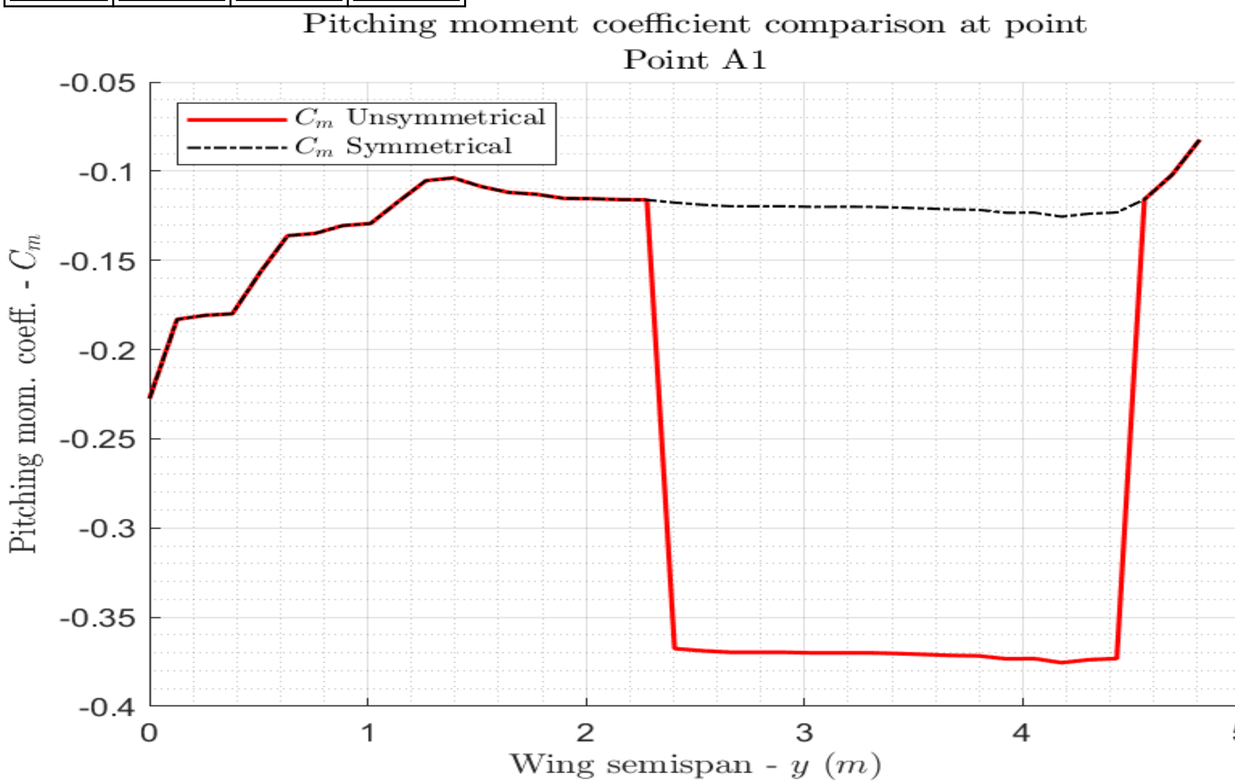

Figure 10.12. Pithcing moment coefficient - POINT A1

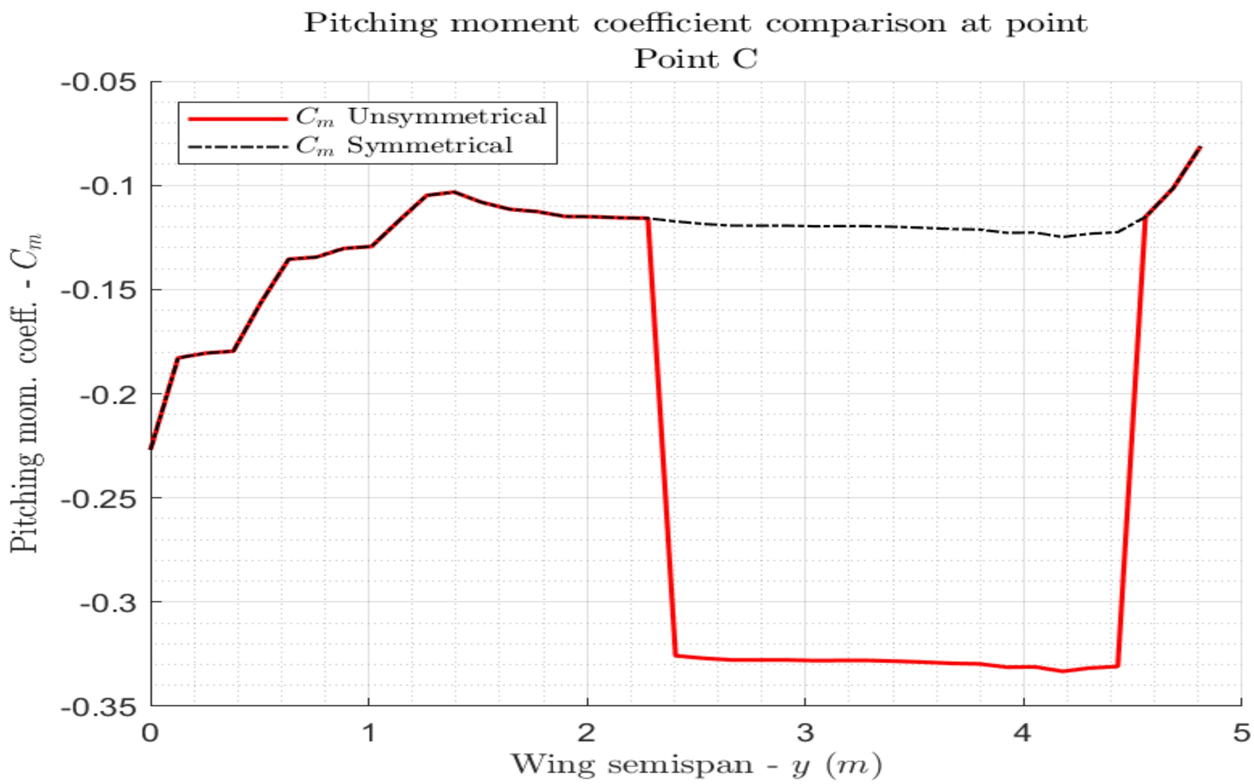

Figure 10.13. Pithcing moment coefficient - POINT C

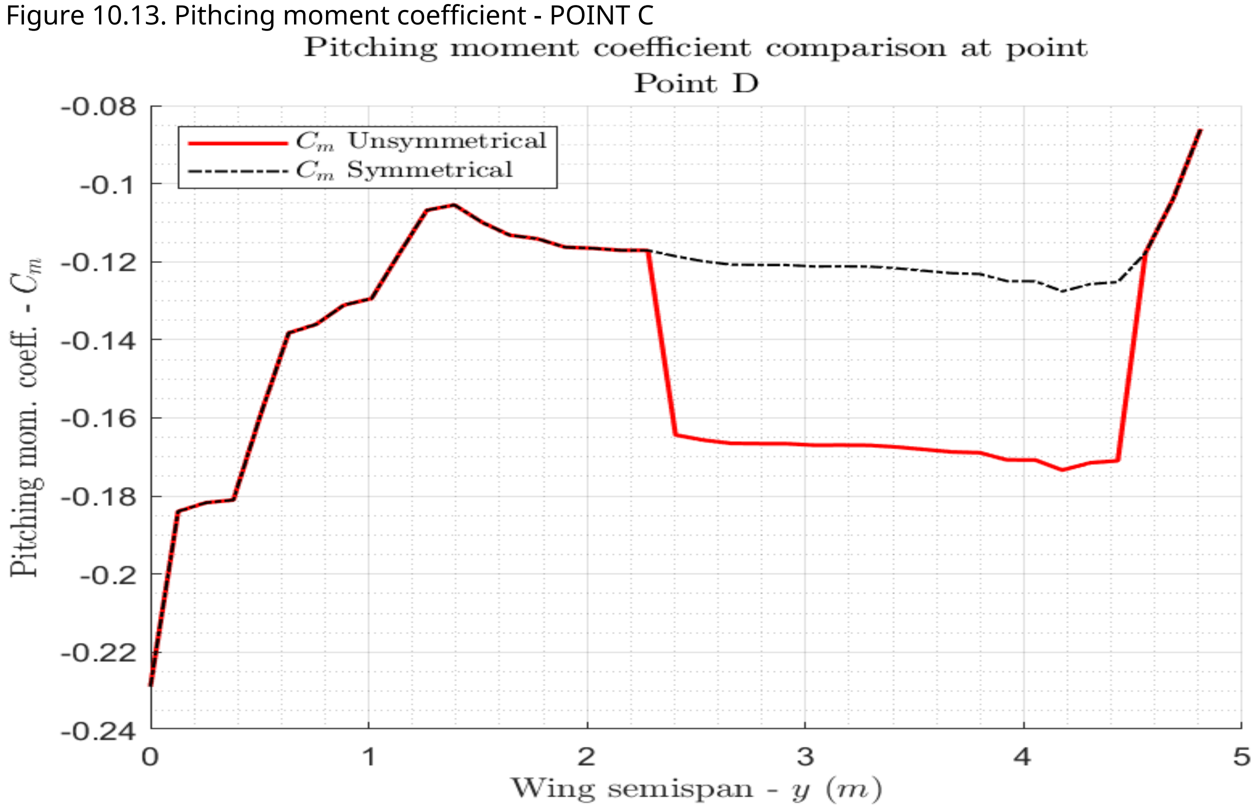

Figure 10.14. Pithcing moment coefficient - POINT D

10.3.2. Effect of aileron displacement on the wing torsion

According to csvla 349 , the aileron displacement cause a significant change of applied wing torsion. The following diagram show this increment in red.

Table 10.4. Unsymmetrical flight conditions. Full and partial torsion at various flight conditions.

y(m)	A_100(daN*m)	A_70(daN*m)	C_100(daN*m)	C_70(daN*m)	D_100(daN*m)	D_70(daN*m)
0	0	0	0	0	0	0
0.3797	-8.837	-6.186	-12.06	-8.444	-23.02	-16.12
0.7595	-31.1	-21.77	-40.48	-28.34	-56.93	-39.85
1.139	-53.25	-37.28	-68.76	-48.13	-90.42	-63.3
1.519	-75.3	-52.71	-96.89	-67.83	-123.5	-86.46
1.899	-97.31	-68.11	-125	-87.48	-156.4	-109.5
2.278	-119.3	-83.5	-153	-107.1	-189.3	-132.5
2.658	-133.7	-93.58	-171.9	-120.4	-217.1	-151.9
3.038	-140.5	-98.37	-181.8	-127.2	-240	-168
3.418	-147.1	-102.9	-191.1	-133.8	-261.9	-183.3
3.797	-153.8	-107.6	-200.7	-140.5	-284.4	-199
4.177	-161.7	-113.2	-212.1	-148.5	-310.7	-217.5
4.557	-171.5	-120	-226.2	-158.3	-343.6	-240.5

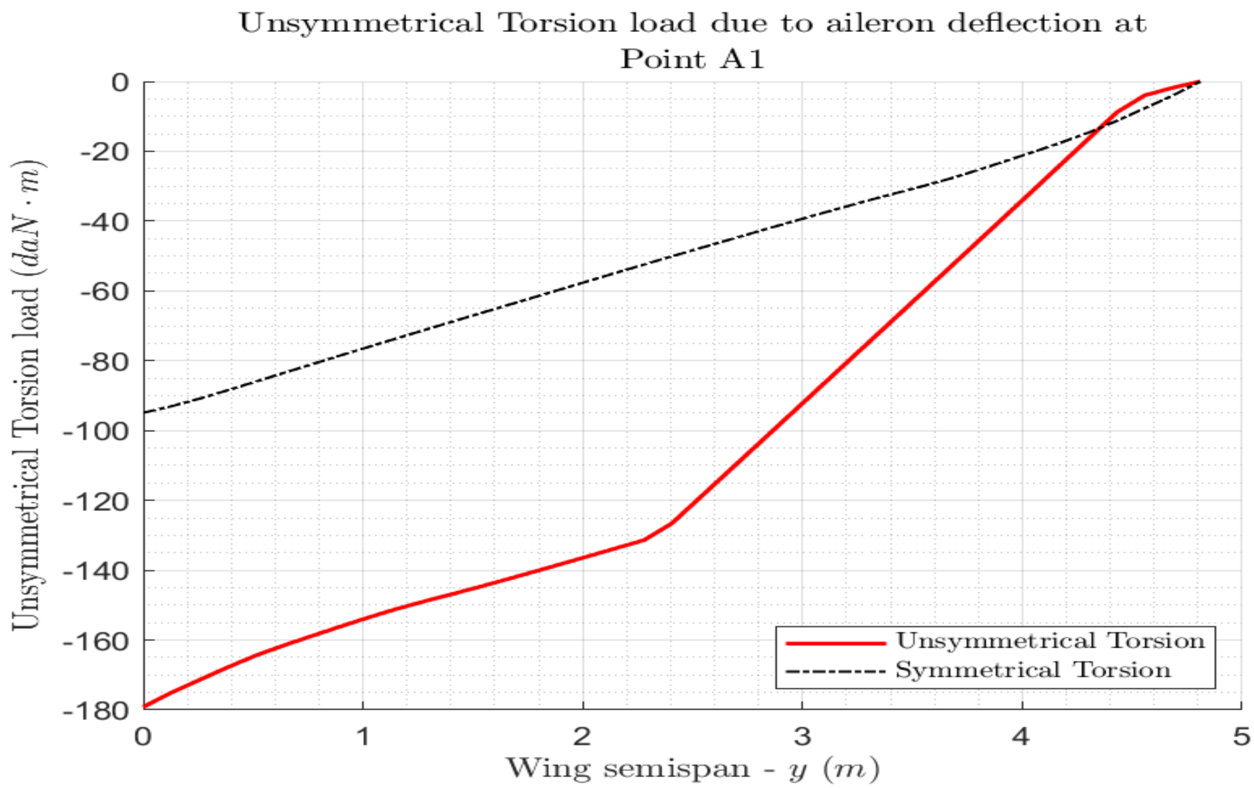

Figure 10.15. Torsion distribution full loads - POINT A1

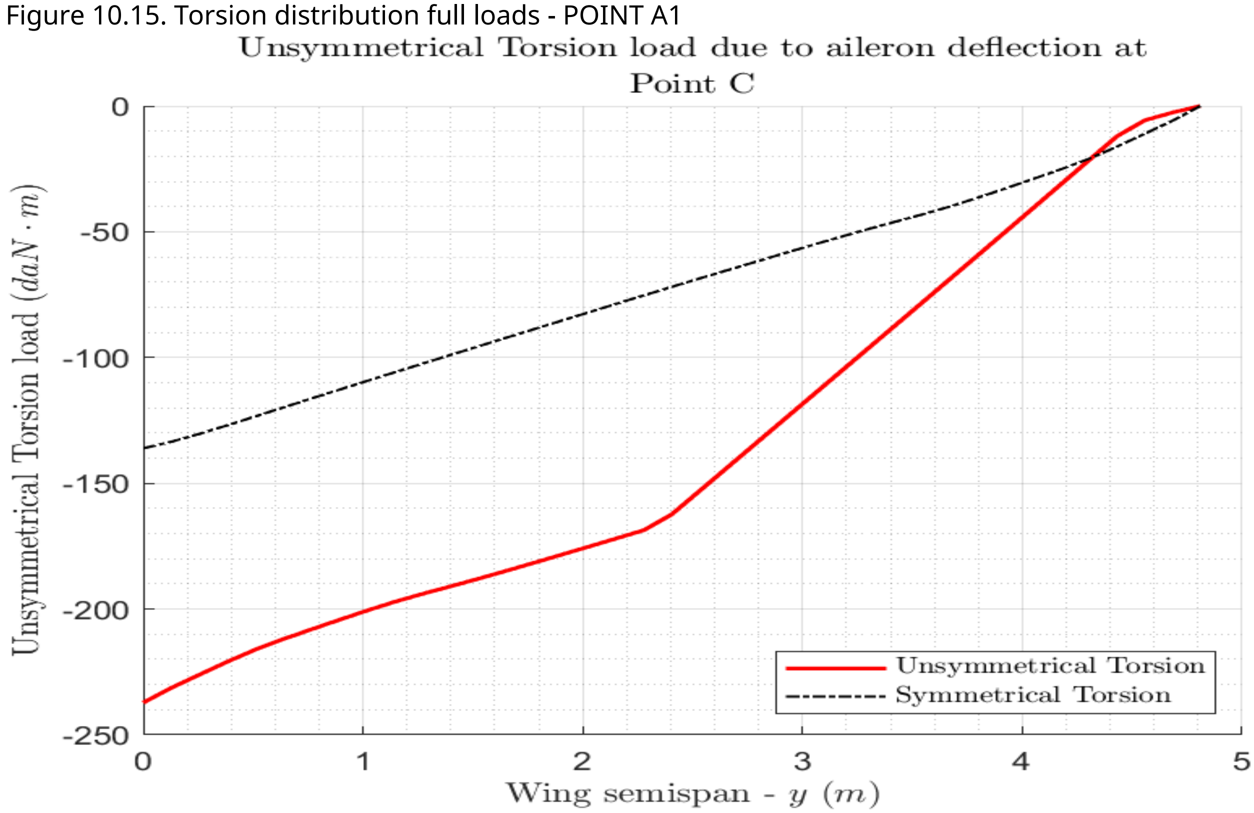

Figure 10.16. Torsion distribution full loads - POINT C

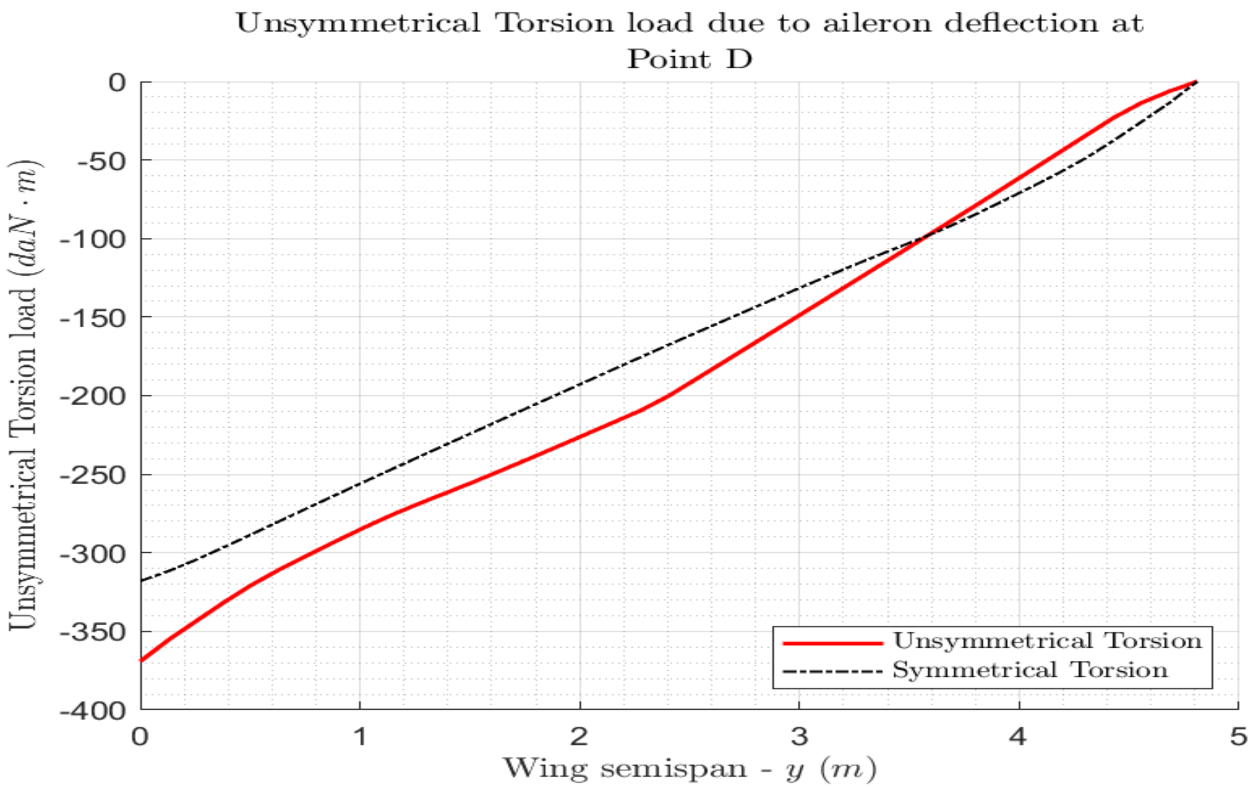

Figure 10.17. Torsion distribution full loads - POINT D

Chapter 11. Loads on the horizontal tail

According to csvla 421 , a horizontal tail balancing load is a load necessary to maintain equilibrium in any specified flight condition with no pitching acceleration. Horizontal tail surfaces must also be designed for the balancing loads occuring at any point on the limit manoeuvring envelope and in the flap conditions specified in csvla 345. The distribution in figure B6 of Appendix B may be used.

11.1. Balancing loads

The following forces are considered and placed in equilibrium:

- lift on the wing;
- horizontal tail balancing load;
- weight of the aircraft;
- for the calculation of the equilibrium, the z axis of aircraft is assumed aligned with the direction of the gravity. In a second stage, once the forces are calculated, the corresponding angle of attack will be considered for the calculation of the correct direction of the forces on the wing;
- influence of thrust and drag (of the total airplane) are considered negligible at this stage of calculation of the vertical forces. The effect of the drag will be considered in a second stage on the wing only;
- the wing lift is assumed to act on the aerodynamic centre of the wing as a starting point. The contribution of the fuselage is accounted for as a shift of the point of aerodynamic centre;
- the horizontal tail lift is applied at 25% ofthe chord;
- no aeroelastic deformation will be considered;

These approximation are by no means mandatory and can be discounted if more reliable data about the aircraft aerodynamics or more accurate calculations are necessary.

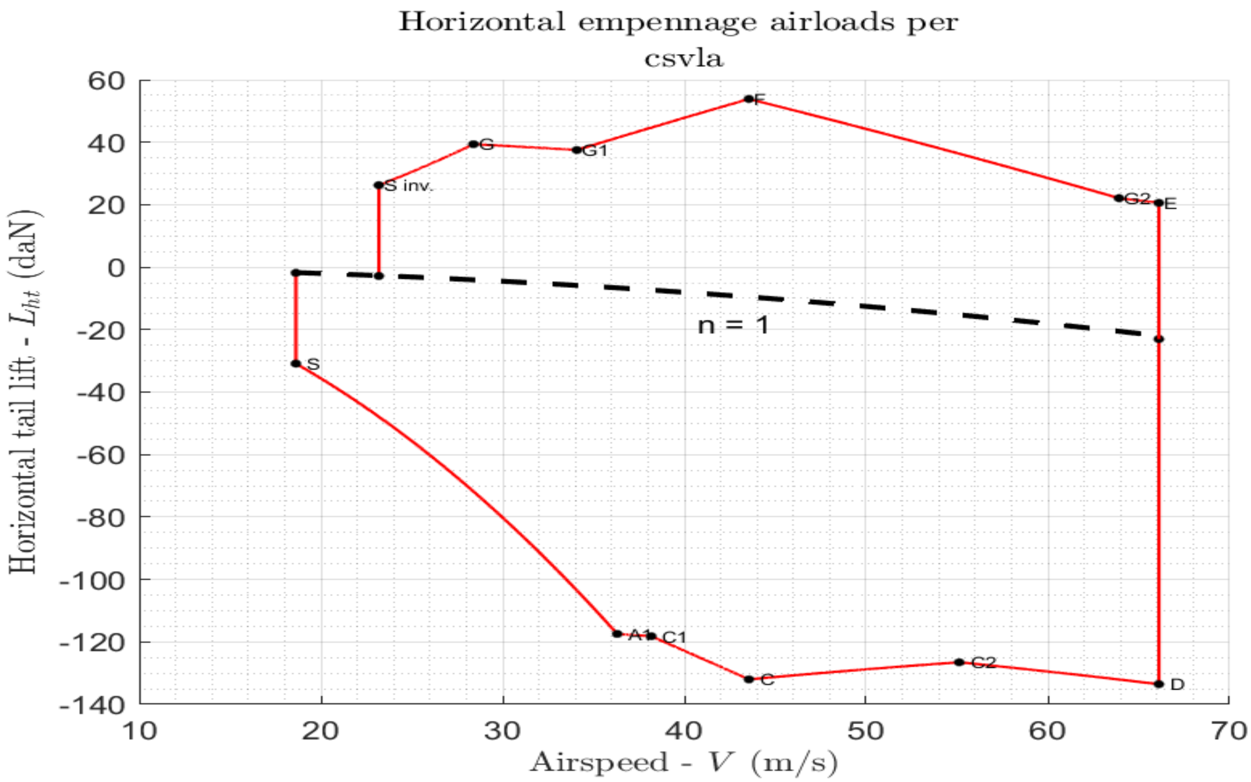

Figure 11.1. Balancing loads

11.2. Manouevring loads

According to csvla 423, each horizontal tail surface must be designed for manoeuvring loads imposed by one of the cited load conditions.

11.2.1. Unchecked manoeuvre

At speed VA the pitching control is suddenly displaced to the maximum deflection as limited by the control stops. The control stops are 25 deg pitch up and 25 deg pitch down. Assuming a linear increment of deflection angle, the tail lift and its moment about the center of gravity pitching axis grow accordingly. The aircraft angular pitching acceleration is the consequence, which, at the tail station, leads to a tangential acceleration nearly normal to tail plane. In the time interval delta t a relative speed delta v develops, which, in composition with the aircraft speed VA causes a decrement of the tail incidence angle equal to the delta v divided by VA. This damping effect is the major relevant fact of the control finite time and its consequence is a less unchecked manoeuvring load. Taking into account the drag forces, which are opposed to the body rotation, and other minus occurrencies, a conservative damping reduction factor of about 0.3 is introduced. This is a standard assumption for the sudden manoeuvring deflection from neutral position to stops. Assuming the direction and the intensity of airspeed at the center of gravity constant during the control time, the differential equation representing the motion is:

$$\frac{d^2\theta}{dt} = \frac{q * S_{tail} * a_{tail} * d}{I_y} * \left(\omega * dt - \frac{\Delta\, v}{V_A} * DF\right)$$

where:

- $\theta \ = \ \text{rotation pitching angle}$
- $q \ = \ \text{dynamic pressure (Pa)}$
- $S_{tail} \ = \ \text{horizontal tail area } (\mathrm{m}^2)$
- $d \ = \ \text{C.G.} - \text{tail} - \text{A.C. distance (m) with } \ x_{C.G.} \ = \ 0.25 * MAC$
- $a_{tail} \ = \ \text{tail lift curve slope (1/deg);}$
- $I_y \ = \ \text{airplane pitching inertia moment } (\mathrm{kg} * \mathrm{m}^2)$
- $\omega \ = \ \dfrac{\delta_{e,max} * \tau}{t_{deflection}} \ = \ \text{control angular speed of plane deflection (rad/sec)}$
- $\tau \ = \ 1.0 \ = \ \text{elevator efficiency factor;}$
- $t_{deflection} \ = \ \text{total time required to full stop;}$
- $\delta_{e,max} \ = \ \text{maximum elevator deflection;}$
- $\dfrac{\Delta \ v}{V_A} \ = \ \text{damping angle}$
- $DF \ = \ \text{damping effect reduction factor} \ = \ 0.3$

The angular speed is defined from total elevator deflection, elevator efficiency factor and total deflection time.

It is possible to solve this equation using simple and reliable numerical methods. According to csvla 423(a) , the following results are presented:

1. a pitch up case where the resultant tailplane angle of attack is -6.762 deg and a corresponding limit tail load of -89.27 daN;
2. a pitch down case where the resultant tailplane angle of attack is 8.452 deg and a corresponding limit tail load of 111.6 daN.

TABLE TO BE CHECKED!!!

Table 11.1. Unchecked manoeuvre. Pitch down case calculations.

d2theta(rad/sec^2)	dtheta(rad/s)	alfa ht(deg)	delta theta(rad)	delta v(m/s)	alfa new ht(deg)
0	0	0	0	0	0
11.67	0.1821	2.551	0.005689	0.6884	2.225
20.52	0.6711	5.102	0.02096	2.537	3.901
27.19	1.391	7.653	0.04346	5.259	5.163
32.21	2.286	10.2	0.07139	8.64	6.114
35.99	3.311	12.76	0.1034	12.52	6.829
38.84	4.436	15.31	0.1385	16.77	7.368
40.98	5.634	17.86	0.176	21.3	7.774

d2theta(rad/sec^2)	dtheta(rad/s)	alfa ht(deg)	delta theta(rad)	delta v(m/s)	alfa new ht(deg)
42.59	6.889	20.41	0.2152	26.04	8.08
43.81	8.186	22.96	0.2557	30.94	8.31

TABLE TO BE CHECKED!!!

Table 11.2. Unchecked manoeuvre. Pitch up case calculations.

d2theta(rad/sec^2)	dtheta(rad/s)	alfa ht(deg)	delta theta(rad)	delta v(m/s)	alfa new ht(deg)
0	0	0	0	0	0
11.67	0.1821	2.551	0.005689	0.6884	2.225
20.52	0.6711	5.102	0.02096	2.537	3.901
27.19	1.391	7.653	0.04346	5.259	5.163
32.21	2.286	10.2	0.07139	8.64	6.114
35.99	3.311	12.76	0.1034	12.52	6.829
38.84	4.436	15.31	0.1385	16.77	7.368
40.98	5.634	17.86	0.176	21.3	7.774
42.59	6.889	20.41	0.2152	26.04	8.08
43.81	8.186	22.96	0.2557	30.94	8.31

11.2.2. Checked manoeuvre

According to csvla 423(b) a sudden upward deflection of the elevator must be studied, at speeds abobe VA, followed by a downward deflection of the elevator, resulting in specified combinations of normal and angular acceleration. The airplane pitching inertia moment is estimated equal to 100 kg * m^2 at maximum takeoff weight and center of gravity at 25% of the mean aerodynamic chord. The maximum limit load in the checked manoeuvre is -28.97 daN.

11.2.3. Gust loads

According to csvla 425 , each horizontal tail surface must be designed for loads resulting during steady, unacclereted flight at different speeds (VF, VC, VD). The incremental tail load resulting from the gusts must be added to the initial balancing tailload to obtain the total tail load. According to csvla 425(d) , in the abscence of a more rational analysis, the incremental tail load due to the gust, must be computed as follows:

$$\Delta\, L_{ht} \;=\; \frac{1}{2} * \left[\, K_g * \rho_0 * U_{de} * V * a_{ht} * S_{ht} \left(1 \;-\; \frac{d\,\epsilon}{d\,\alpha}\right)\right]$$

where:

- $\rho_0 \;=\;$ density of air at sea level $(\mathrm{kg/m^3})$;
- $K_g \;=\;$ gust alleviation factor at MTOW;
- $U_{de} \;=\;$ derived gust speed $(\mathrm{m/s})$;

- $V \;=\;$ aircraft equivalent speed (m/s);
- $a_{ht} \;=\;$ tail lift curve slope (1/rad);;
- $S_{ht} \;=\;$ horizontal tail area (m^2);
- $\left(1 \;-\; \dfrac{d\,\epsilon}{d\,\alpha}\right) \;=\;$ downwash factor.

Results can be summarized as follow:

1. at VF, the incremental tail load is 89.54 daN and the resulting total load is 143.4 daN when the incremental tail load is summed and -35.7 daN when the incremental tail load is subtracted;
2. at VC, the incremental tail load is 178.6 daN and the resulting total load is 46.61 daN when the incremental tail load is summed and -310.6 daN when the incremental tail load is subtracted;
3. at VF, the incremental tail load is 135.9 daN and the resulting total load is 2.373 daN when the incremental tail load is summed and -269.5 daN when the incremental tail load is subtracted.

11.3. Horizontal tail loads summary

In this section all the maximum manoeuvring limit loads are summarized

1. Case (a): -5.865 daN;
2. Case (b): -28.97 daN;
3. Case (a) + (b): -117.3 daN;
4. Case (c): -95.92 daN;
5. Case (d): -315.7 daN.

11.4. Unsysmmetrical loads

According to csvla 427 , in the abscence of more rational data for conventional aircraft must be applied (1) 100% of the maximum loading from the symmetrical flight conditions on the surface on one side of the plane of symmetry and (2) the following percentage of that loading to the opposite side:

$$\% \;=\; 100 \;-\; 10 * (n-1) \;=\; \frac{1}{100} * [100 \;-\; 10 * (3.8 \;-\; 1)] \;= 0.72$$

The critical manoeuvre load is -315.7 daN; the unsymmetrical loads will be:

- load on one side: -315.7*0.5 = -157.9 daN;
- load on the opposite side: -315.7*0.72 = -227.3 daN.

Chapter 12. Loads on the vertical tail

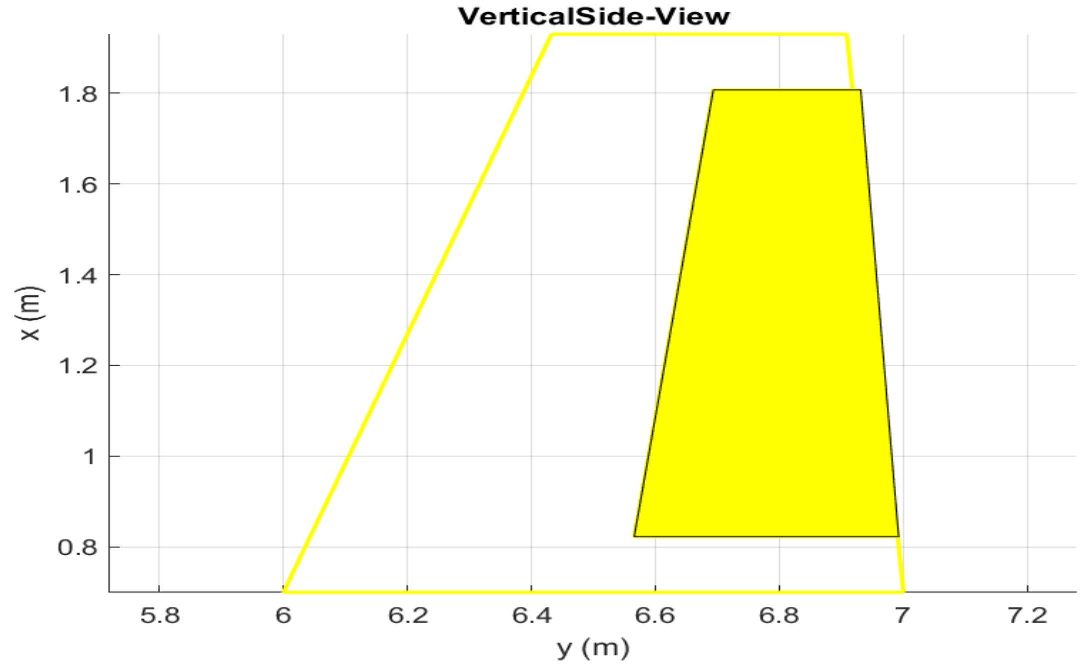

Figure 12.1. Vertical fin, side view.

According to csvla the vertical tail must withstand several manoeuvring loads. In this chapter, all these load case will be illustrated.

12.1. Manouevring load

At speeds up to VA, the vertical tail surfaces must be designed to withstand the following condition. In computing the tail loads, the yawing velocity may be assumed to be zero.

12.1.1. csvla 441(a)(1)

With the aeroplane in unaccelerated flight at zero yaw, it is assumed that the rudder control is suddenly displaced to the maximum deflection, as limited by the control stops or by limit pilot forces.The control stops are +/- 30 deg. The lateral force coefficient acting on the rudder when at maximum deflection angle is given by the following simple equation:

$$C_Y = C_{Y,0} + \frac{d\,C_Y}{d\,\delta_r} * \delta_{r,max}$$

where:

- C_Y = lateral force coefficient;
- $C_{Y,\,0}$ = lateral force coefficient at $\beta = \delta = 0$, equal to zero for symmetrical airfoil;
- $\frac{d\,C_Y}{d\,\delta_r}$ = lateral force curve slope per deg of rudder deflection;

- $\delta_{r,\,max} \;=\;$ rudder control stop.

Assuming no deflection of the control cable, the maximum value of the lateral force coefficient is:

$$(C_Y)_{\delta_r\,=\,30} \;=\; 0.000644 \;*\; 30 \;=\; 0.019356$$

The lateral force is calculated as follow:

$$Y \;=\; \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * (C_Y)_{\delta_r\,=\,30} \;=\; \frac{1}{10} * 807.4 * 1.44 * \frac{13.4}{1.44} * 0.01936 \;=\; 20.94\; daN$$

The lateral force acting on a single fin of the vertical tail plain is 20.94/2 = 10.47 daN.

12.1.2. csvla 441(a)(2)

With the rudder deflected as specified in sub-paragraph csvla 441(a)(1) of this paragraph, it is assumed that the aeroplane yaws to the resulting sideslip angle. In lieu of a rational analysis, an overswing angle equal to 1.3 times the static sideslip angle of sub-paragraph csvla 441(a)(3) of this paragraph may be assumed. The overswing sideslip angle is 1.3 * 15 = 19.5 deg. The total lateral force acting on the vertical tail in this case is:

$$Y \;=\; \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * C_Y \;=\; \frac{1}{10} * 807.4 * 1.44 * \frac{13.4}{1.44} * 0.0245 \;=\; 13.25\; daN$$

The lateral force acting on a single fin of the vertical tail plain is 13.25/2 = 6.627 daN.

12.1.3. csvla 441(a)(3)

A yaw angle of 15 degrees with the rudder control maintained in the neutral position (except as limited by pilot strength). The total lateral force in this case is:

$$Y \;=\; \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * C_Y \;=\; \frac{1}{10} * 807.4 * 1.44 * \frac{13.4}{1.44} * 0.0233 \;=\; 12.6\; daN$$

12.2. Manouevring and gust envelope

According to csvla 443 in the abscence of a more rational analysis, the gust load must be computed as follows:

$$L_{vt} \;=\; \frac{K_{gt} \;*\; U_{de} \;*\; V \;*\; a_{vt} \;*\; S_{vt}}{16.3}$$

where:

- $U_{de} \;=\;$ derived gust velocity (m/s);
- $L_{vt} \;=\;$ vertical tail load (daN)
- $K_{gt} \;=\; \dfrac{0.88 \;*\; \mu_{gt}}{5.3 \;+\; \mu_{gt}} \;=\;$ gust alleviation factor;

- $\mu_{gt} = \dfrac{2 * M}{\rho * \bar{c}_t * g * a_{vt} * S_{vt}} * \dfrac{K^2}{l_t^2} = \text{lateral mass ratio;}$
- $M = \text{aeroplane mass (kg);}$
- $rho = \text{air density } (kg/m^3);$
- $l_t = \text{aeroplane c.g.} - \text{to} - \text{lift} - \text{centre of vertical surface distance } (m);$
- $S_{vt} = \text{area of vertical tail } (m^2);$
- $a_{vt} = \text{lift curve slope of vertical tail } (1/rad);$
- $V = \text{aeroplane equivalent speed } (m/s);$
- $K = \text{radius of gyration in yaw } (m);$
- $g = \text{acceleration due to gravity } (m/s^2);$

These calculations must be performed at VC and VD; the results are the following:

1. Gust load at VC: 20.1856 daN
2. Gust load at VD: 15.3201 daN

The critical gust load is 20.1856 daN at VC.

12.3. Vertical tail loads summary

12.3.1. Critical manouevring loads

The critical manouevring loads are summarized here.

1. csvla 441(a)(1) : 10.47 daN;
2. csvla 441(a)(2) : 6.627 daN;
3. csvla 441(a)(3) : 6.302 daN.

12.3.2. Critical gust loads

The critical gust load is 20.1856 daN at VC.

12.4. Combined loads

According to csvla 447(b) the following two additional condition must be verified:

1. With the aeroplane in a loading condition correspondint to point A or point D in the V - n diagram (whichever condition leads to the higher balance load) the loads on the horizontal tail must be combined with those on the vertical tail as specified in csvla 441 ; this prescription results in a combined load equal to 68.07 daN;
2. 75 % of the loads according to csvla 423 for the horizontal tail and csvla 441 for the vertical tail must be assumed acting simultaneously; this prescription results in a combined load equal to 237.1 daN.

The critical combined load is 237.1487 daN.

Chapter 13. Loads on the wing flaps

According to csvla 345(a) , if flaps or similar high lift devices to be used for take-off, approach or landing are installed, the aeroplane, with the flaps fully deflected at VF, is assumed to be subjected to symmetrical manoeuvres and gusts resulting in limit load factors within the range determined by (1) Manoeuvring to a positive limit load factor of 2.0 (2) Positive and negative gust of 7.62 m/s acting normal to the flight path in level flight.

13.1. Manouevring load

According to csvla 347 , the wing flpas, their operating mechanisms and their supporting structure will be analyzed at two deflection angle, namely takeoff and landing position.

13.1.1. Takeoff

The flaps V-n diagram relative to the takeoff configuration is shown in this subsection; the estimated maximum lift coefficient in this configuration is 1.9 corresponding to a stall speed equal to 16.82 m/s.

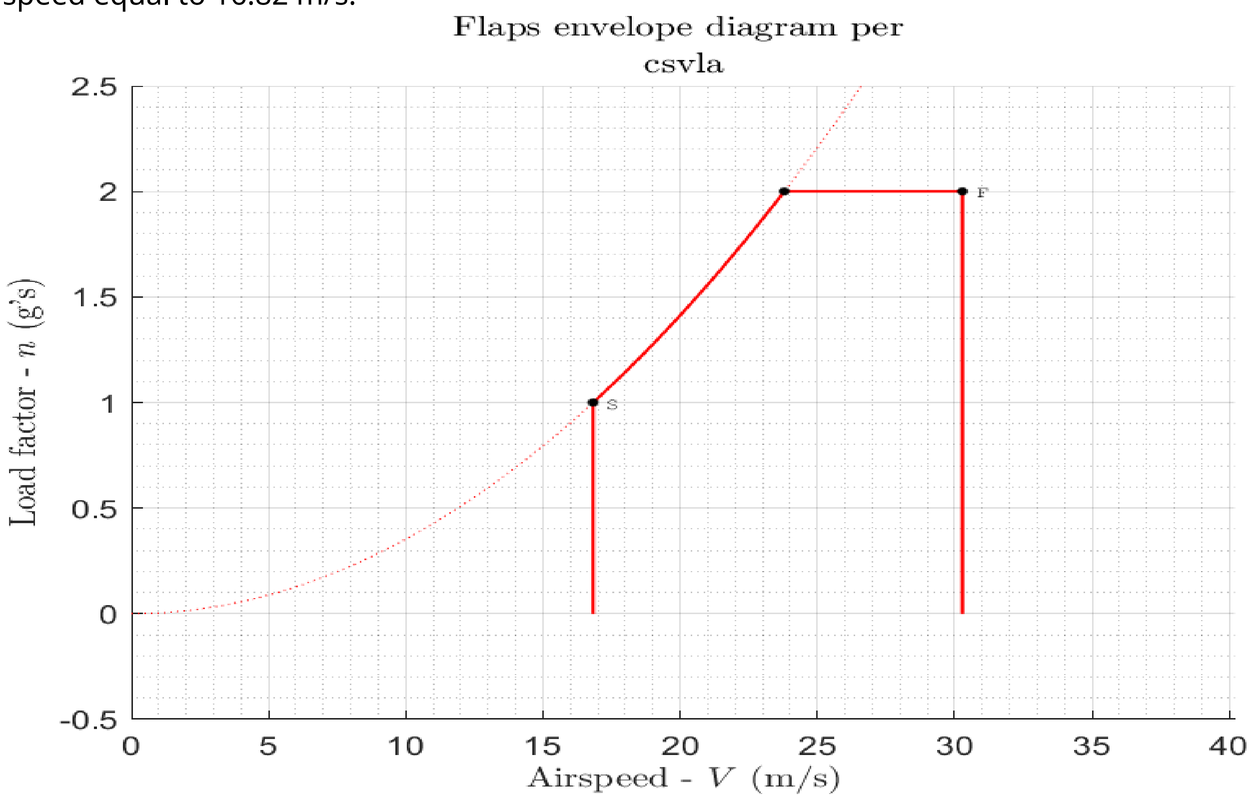

Figure 13.1. Flaps V-n diagram. Takeoff.

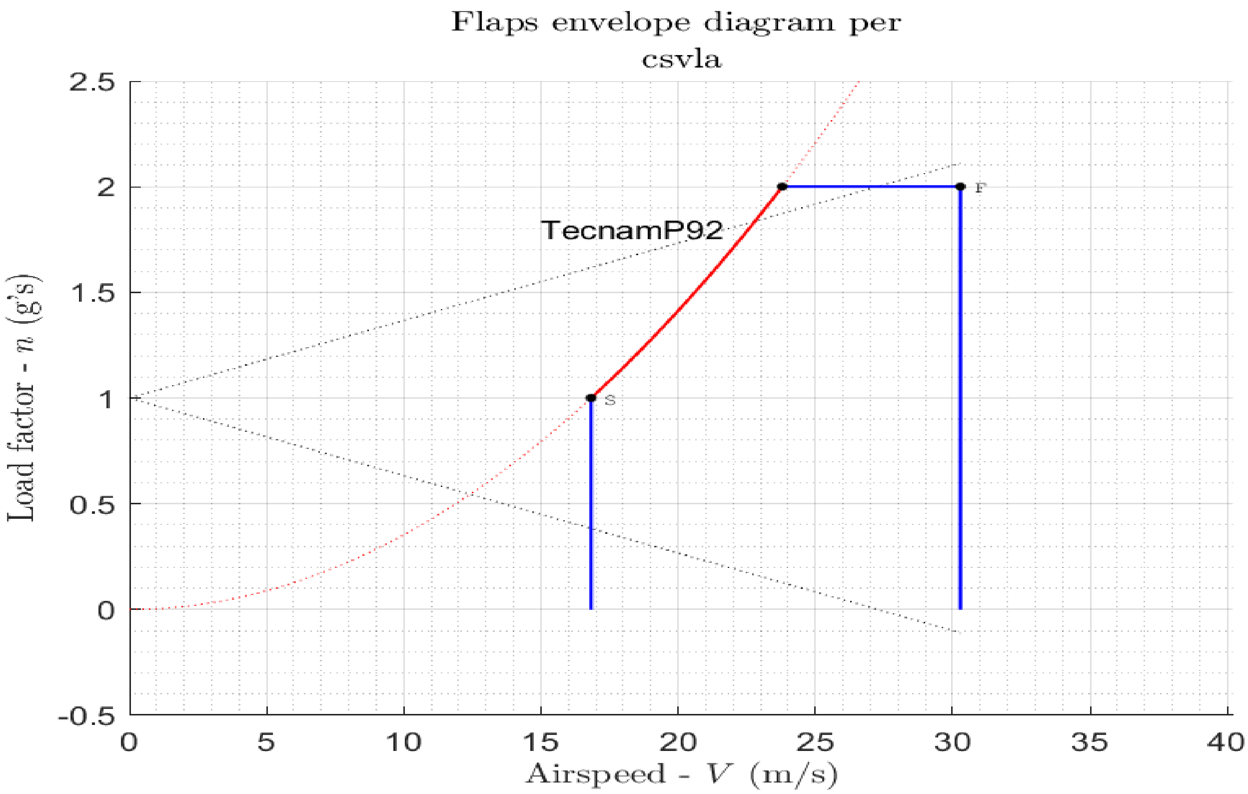

Figure 13.2. Flaps gust envelope diagram. Takeoff.

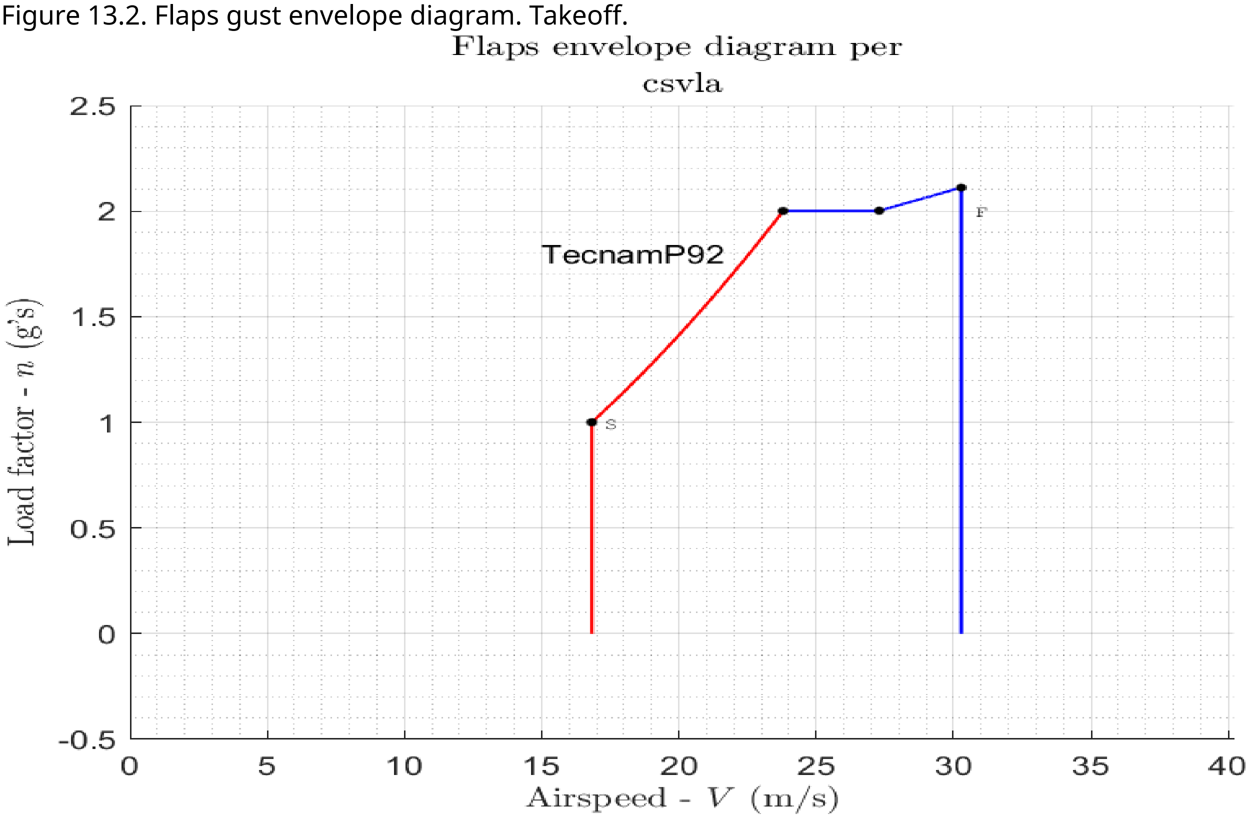

Figure 13.3. Flaps final envelope diagram. Takeoff.

13.1.2. Landing

The flaps V-n diagram relative to the landing configuration is shown in this subsection; the estimated maximum lift coefficient in this configuration is 1.9 corresponding to a stall speed equal to 16.82 m/s.

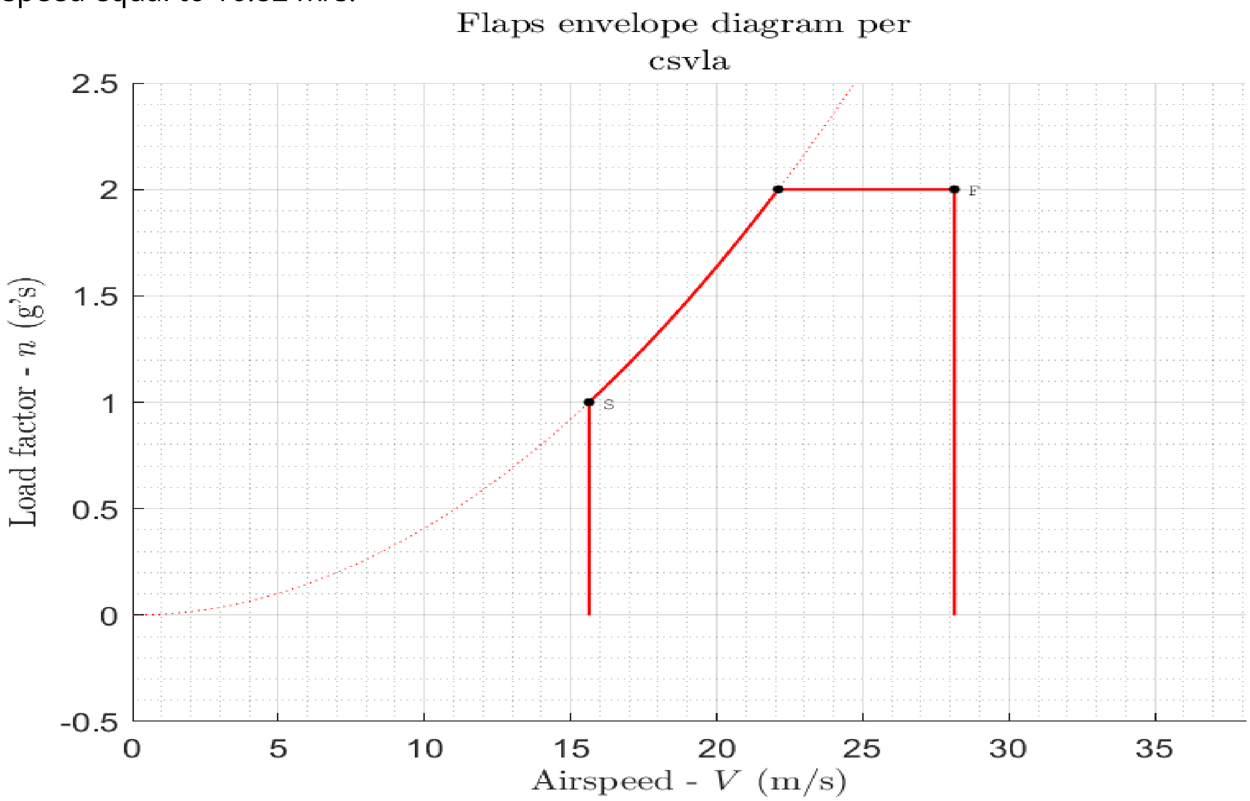

Figure 13.4. Flaps V-n diagram. Landing.

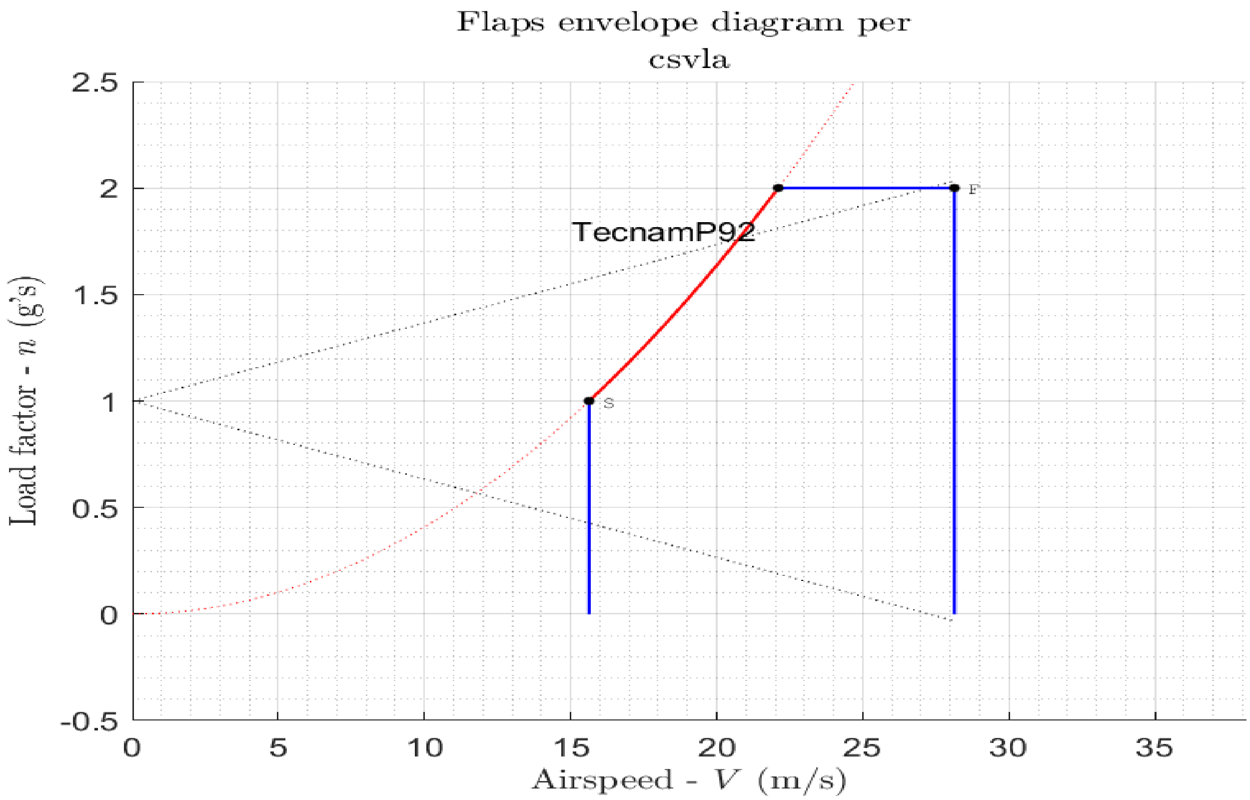

Figure 13.5. Flaps gust envelope diagram. Landing.

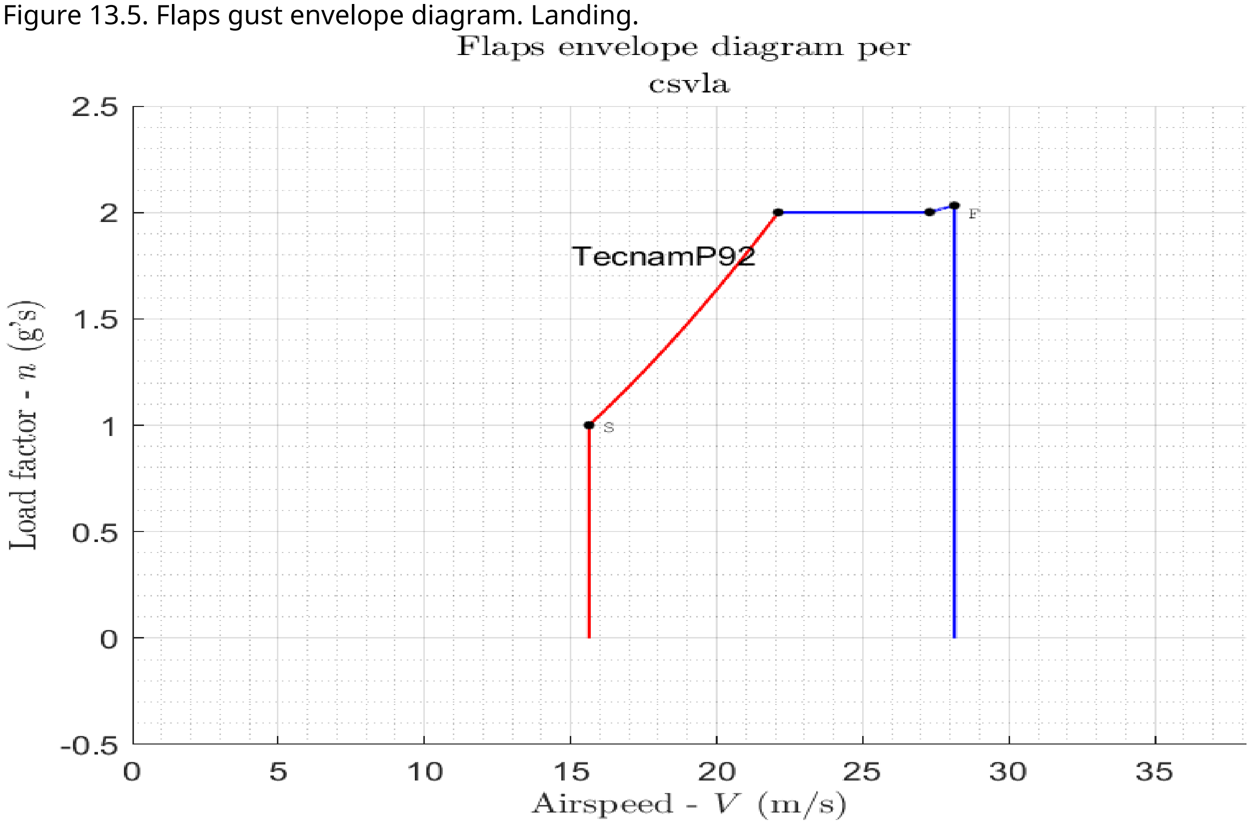

Figure 13.6. Flaps final envelope diagram. Landing.

13.1.3. Summary of flaps load

A diagram of the flaps load calculations is shown. The clean positive attitude flight envelope of the aircraft is also represented for reference. The data relative to these calculations are summarized in table form, for takeoff and landing configurations, respectively.

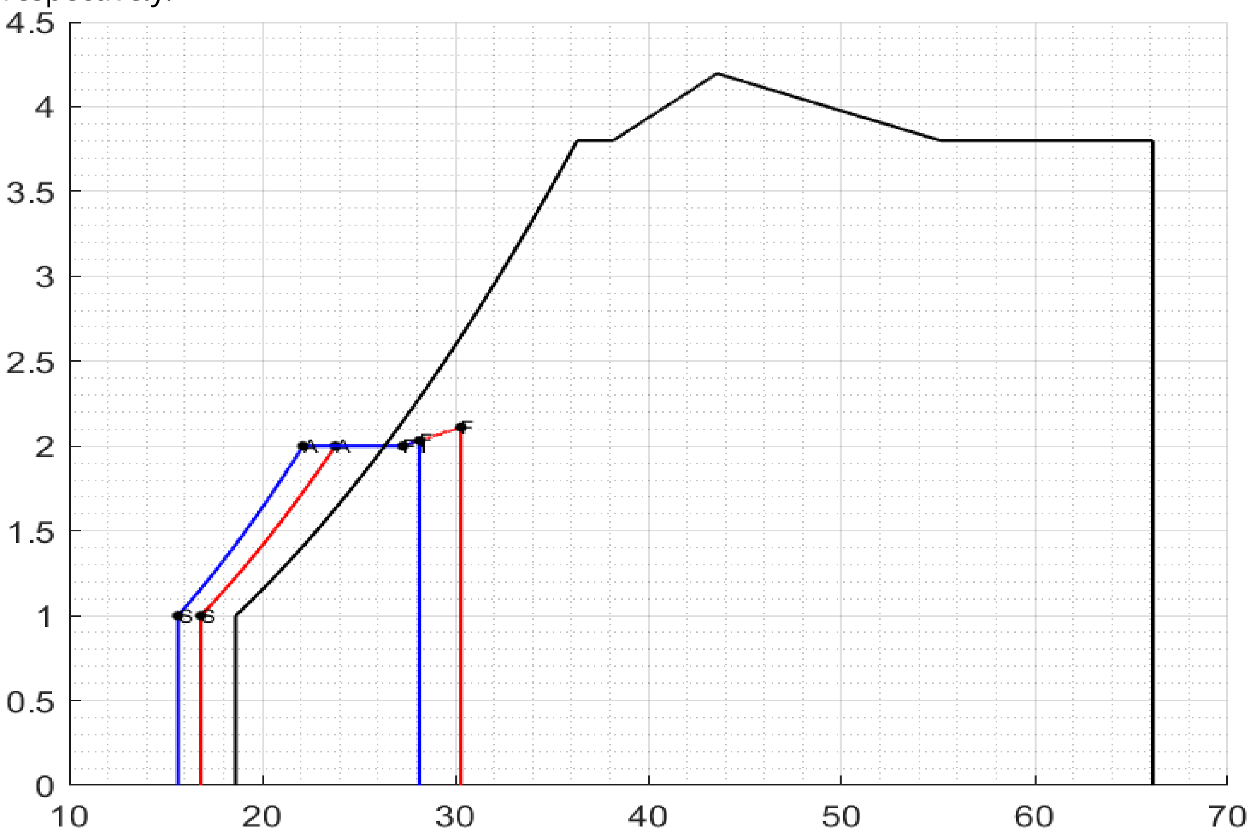

Figure 13.7. Summary of flaps flight load.

TABLE TO BE CHECKED!!!

Table 13.1. Flaps envelope points summary at takeoff.

ID	V(m/s)	M(kg)	M/S(kg/m^2)	Altitude(m)	rho(kg/m^3)	mug	Kg	Ude(m/s)	n
Point S	16.82	450	33.58	2000	1.006	10.99	0.5937	7.62	1
Point A	23.79	450	33.58	2000	1.006	10.99	0.5937	7.62	2
Point F	27.29	450	33.58	2000	1.006	10.99	0.5937	7.62	2

TABLE TO BE CHECKED!!!

Table 13.2. Flaps envelope points summary at landing.

ID	V(m/s)	M(kg)	M/S(kg/m^2)	Altitude(m)	rho(kg/m^3)	mug	Kg	Ude(m/s)	n
Point S	15.63	450	33.58	2000	1.006	10.99	0.5937	7.62	1

ID	V(m/s)	M(kg)	M/S(kg/m^2)	Altitude(m)	rho(kg/m^3)	mug	Kg	Ude(m/s)	n
Point A	22.11	450	33.58	2000	1.006	10.99	0.5937	7.62	2
Point F	27.29	450	33.58	2000	1.006	10.99	0.5937	7.62	2

Chapter 14. Loads on the control surfaces

According to csvla 395 , the flight control system and its supporting structure must be designed for loads corresponding to 125 % of the computed hinge moments of the movable control surface.

TABLE TO BE CHECKED!!!

14.1. Ailerons

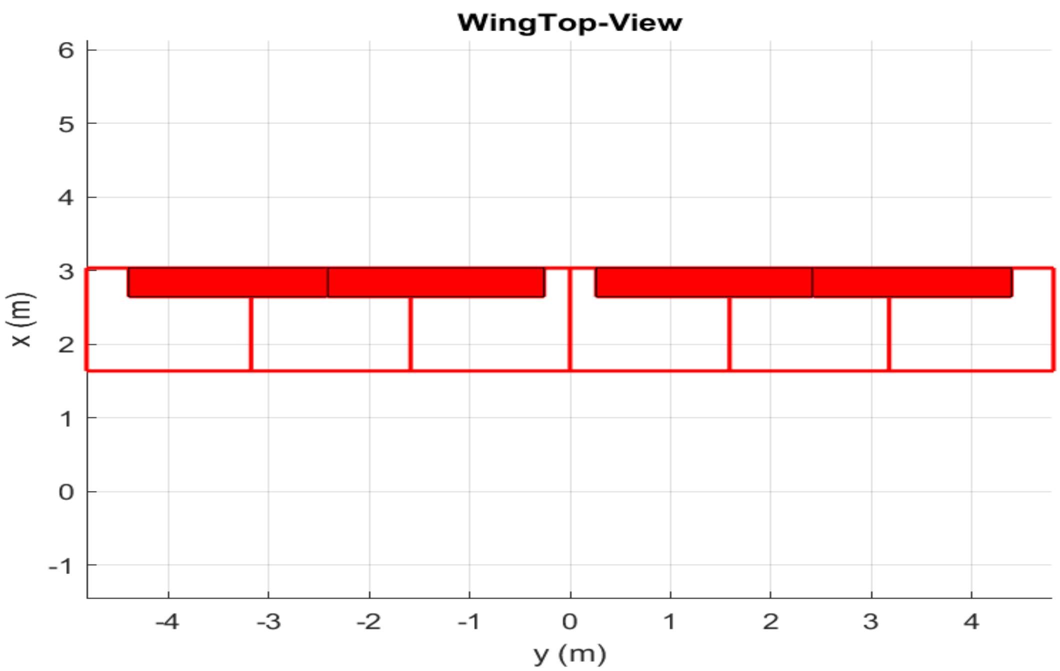

Figure 14.1. Aileron, top view.

Table 14.1. Aileron. Hinge moments coefficients.

Quantity	Value	Units
CH delta	-0.061	1/rad
CH delta	-0.001065	1/deg
CH alfa	-0.41	1/rad
CH alfa	-0.007156	1/deg
CH total	-0.002217	1/rad
CH total	-0.1271	1/deg

According to csvla 395 , the total aileron load is equal to 0 N . The hinge moment is calculated by the following equation

$$H_{aileron} = q * S_{aileron} * c_f * C_{h_{total}} = 807.4 * 0.7684 * 0 * -0.1271 = 0 \ N * m$$

where:

- $H_{aileron} = $ aileron hinge moment $(N * m)$;
- $q = $ dynamic pressure at point A (Pa) ;
- $S_{aileron} = $ aileron surface (m^2) ;
- $c_f = $ reference chord (m) ;
- $C_{h_{total}} = $ total hinge moment coefficient.

This is the formula used in all the following calculations. The total hinge moment that must be considered in structural calculations is the following:

$$H_{aileron_{total}} = 2 * 1.25 * H_{aileron} = 2 * 1.25 * (0) = 0 \ N * m$$

TABLE TO BE CHECKED!!!

14.2. Elevator

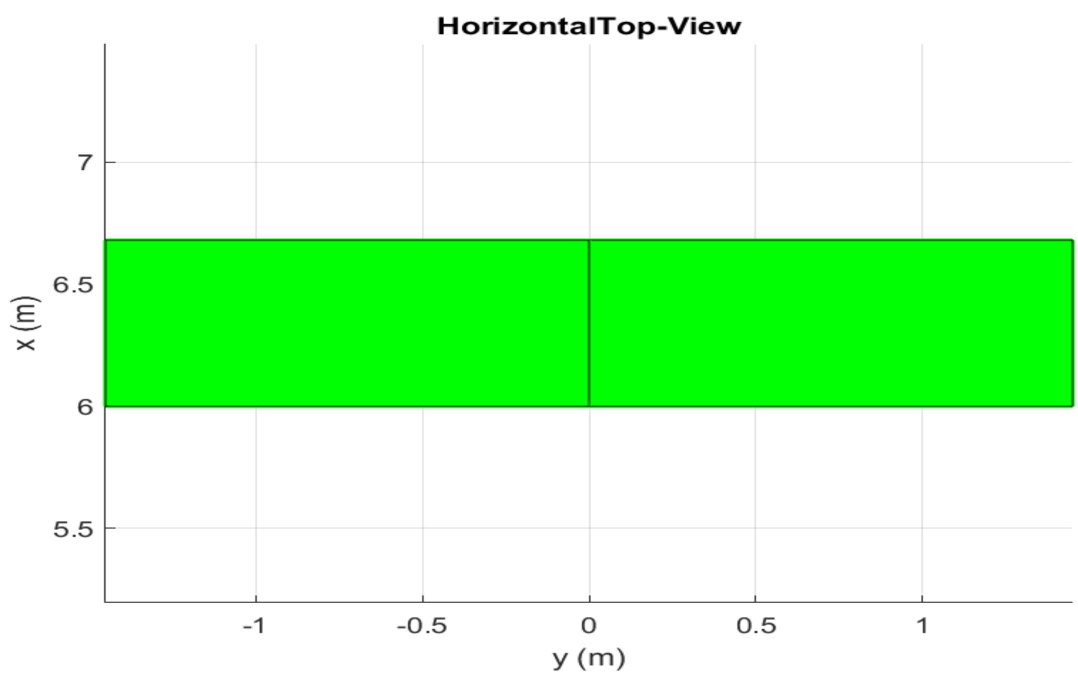

Figure 14.2. Elevator, top view.

Table 14.2. Elevator. Hinge moments coefficients.

Quantity	Value	Units
CH delta	-0.461	1/rad
CH delta	-0.008046	1/deg
CH alfa	-0.269	1/rad
CH alfa	-0.004695	1/deg

Quantity	Value	Units
CH total	-0.004102	1/rad
CH total	-0.2351	1/deg

According to csvla 395 , the total elevator load is equal to -6333 N. The hinge moment is

$$H_{elevator} = q * S_{elevator} * c_f * C_{h_{total}} = 807.4 * 1.972 * 0.5984 * -0.2351 = -223.9\ N * m$$

The total hinge moment that must be considered in structural calculations is the following:

$$H_{elevator_{total}} = 2 * 1.25 * H_{elevator} = 2 * 1.25 * (-223.9) = -559.9\ N * m$$

TABLE TO BE CHECKED!!!

14.3. Rudder

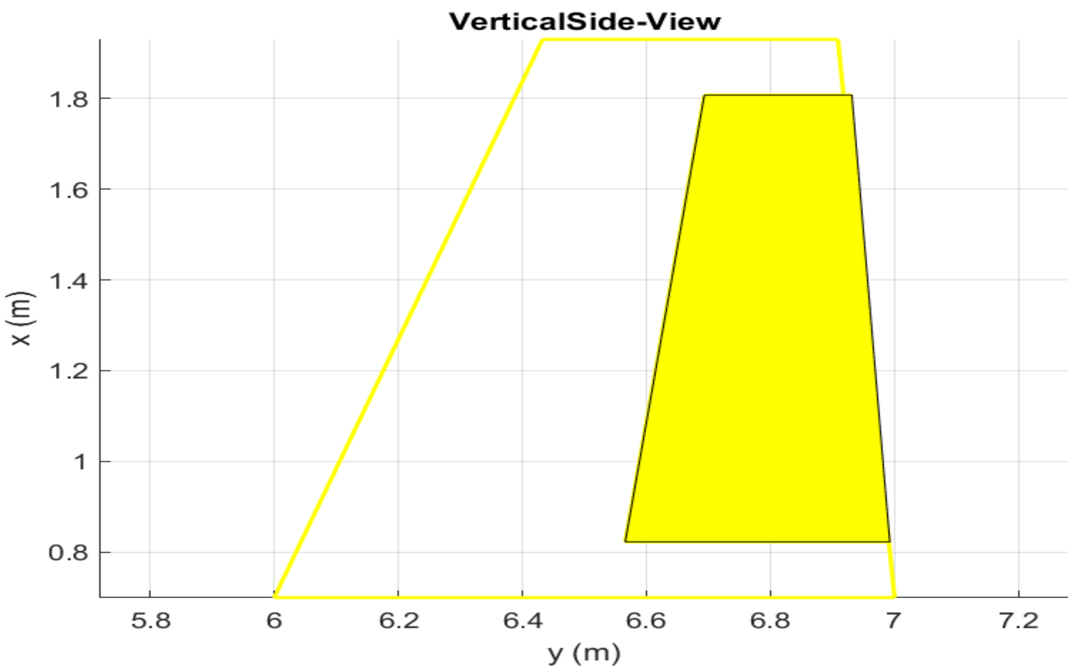

Figure 14.3. Vertical fin, side view.

Table 14.3. Rudder. Hinge moments coefficients.

Quantity	Value	Units
CH delta	-0.45	1/rad
CH delta	-0.007854	1/deg
CH alfa	-0.0024	1/rad
CH alfa	-4.189e-05	1/deg
CH total	-0.004112	1/rad
CH total	-0.2356	1/deg

According to csvla 395 , the total rudder load is equal to -298.9 N. The hinge moment is

$$H_{rudder} \ = \ q * S_{rudder} * c_f * C_{h_{total}} \ = 807.4 * 0.1634 * 0.5984 * -0.007854 \ = -10.32 \ N \ * \ m$$

where the surface is related to the double fin geometrical arrangement. The total hinge moment that must be considered in structural calculations is the following:

$$H_{rudder_{total}} \ = \ 1.25 \ * \ H_{rudder} \ = 1.25 * (-10.32) \ = -12.9 \ N \ * \ m$$

Chapter 15. Power plant

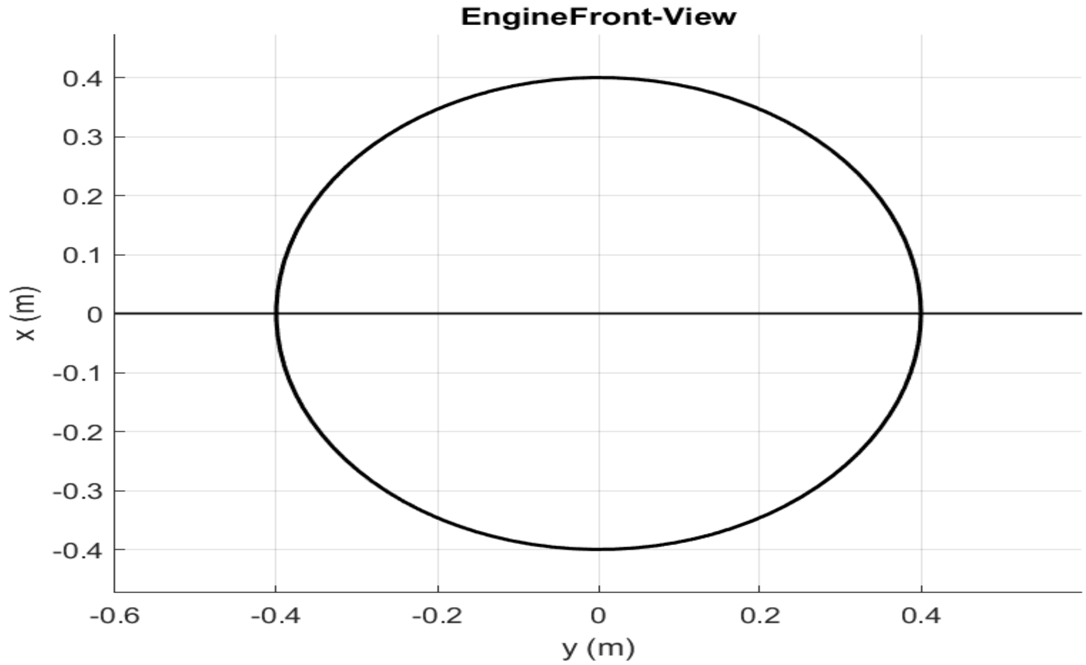

Figure 15.1. Engine, front view.

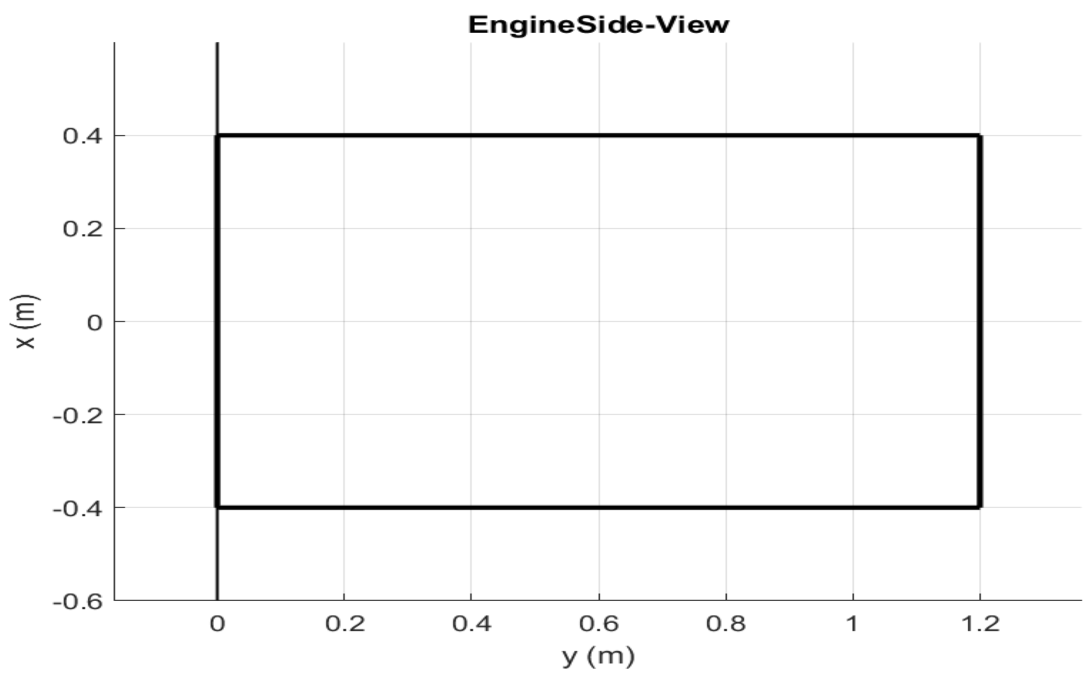

Figure 15.2. Engine, side view.

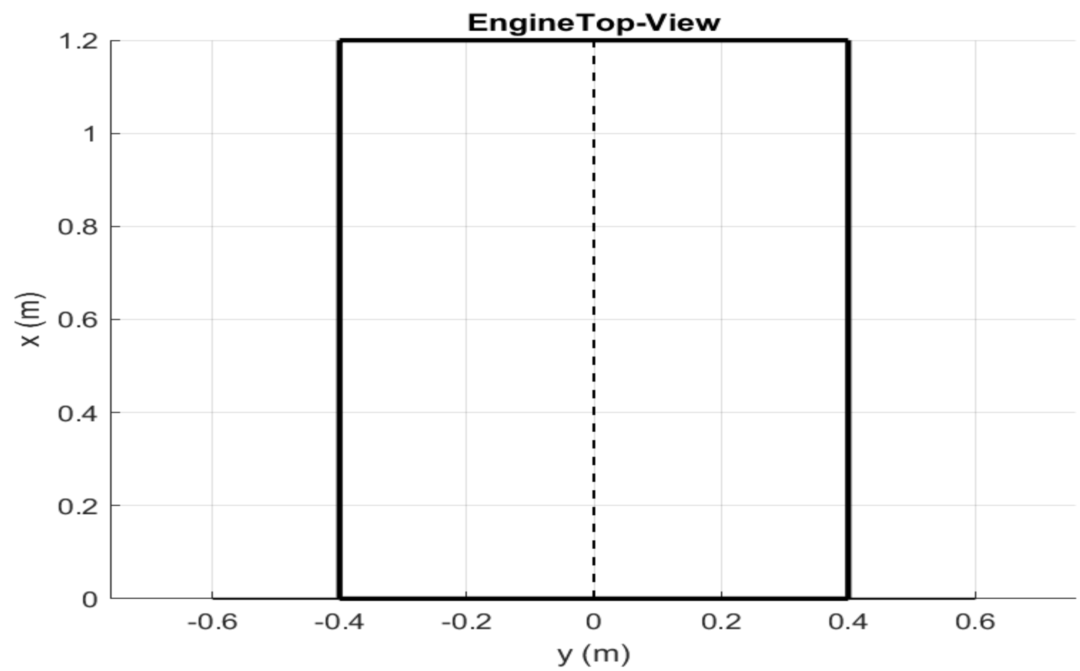

Figure 15.3. Engine, top view.

The engine mount and its supporting structure must be designed for the effects of:

1. a limit engine torque corresponding to takeoff power and propeller speed acting simultaneously with 75% of the limit loads from flight condition A of csvla 333(d) , according to csvla 361(a)(1) ;
2. the limit engine torque as specified in csvla 361(b) acting simultaneously with the limit loads from flight condition A of csvla 333(d) , according to csvla 361(a)(2) .

The limit engine torque to be considered under csvla 361(a)(2) must be obtained by multiplying the mean torque for maximum continuous power by a factor determined as follows:

1. for a four-stroke engines, (i) 1.33 for engines with five or more cylinders, (ii) 2, 3, 4 or 8 for engines with four, three, two or one cylinders, respectively;
2. for a two-stroke engines, (i) 2 for engines with three or more cylinders, (ii) 3 or 6 for engines with two or one cylinders, respectively.

The engine mount and its supporting structure must be designed for a limit load factor in a lateral direction, for the side load on the engine mount not less than 1.33 and this side load may be assumed to be independent of other flight conditions, according to csvla 363 .

15.1. Engine torque

The engine takeoff power is 74.57 kW at 5800 rpm. The rotational speed of the propeller is 5800/2.429 = 2388 RPM. The maximum continuous power is 67.11 kW. The mean engine torque is 298.2 N * m. Using a factor of 1.33 for a four cylinder engine, the limit torque will be 396.6 N * m. This limit torque acts simultaneously with the 75 % of the inertia limit load. The mean engine torque at max continuous power is 283 N * m. Using a factor of 1.33 for a four cylinder engine, the limit torque will be 376.4 N * m which acts simultaneously with the 100 % of the inertia limit load.

$$MT_{continuous} = P_{continuous} * \frac{1000}{\frac{2\pi * RPM_{prop}}{60}} = 67.11 * \frac{1000}{\frac{2 * 3.14 * 2388}{60}} = 283\ N * m$$

$$LT_{continuous} = RR_{prop} * MT_{continuous} = 2.429 * 283 = 376.4 N * m$$

$$MT_{takeoff} = P_{takeoff} * \frac{1000}{\frac{2\pi * RPM_{prop}}{60}} = 74.57 * \frac{1000}{\frac{2 * 3.14 * 2388}{60}} = 298.2\ N * m$$

$$LT_{takeoff} = RR_{prop} * MT_{takeoff} = 2.429 * 298.2 = 396.6 N * m$$

- $MT_{takeoff} = \text{mean torque at takeoff power (N} * \text{m); and}$
- $LT_{takeoff} = \text{limit torque at takeoff power (N} * \text{m); and}$
- $MT_{continuous} = \text{mean torque at max continuous power (N} * \text{m); and}$
- $LT_{continuous} = \text{limit torque at max continuous power (N} * \text{m).}$

15.2. Side load on engine mount

The limit load factor in a lateral direction is 1.33. The mass of the engine group is 90.66 kg. The side load results is 1.33*90.66*9.807*(1/10) = 118.2 daN

15.3. Inertia load on engine mount

The inertia load is equal to the maximum limit load factor times the engine group weight: 3.8*90.66*9.807*(1/10) = 449.3 daN

15.4. Gyroscopic loads

According to AMC 23.371(a) , for a two blade propeller, the maximum gyroscopic couple is given by:

$$2 * I_p * \omega_1 * \omega_2$$

Where:

- $I_p = \text{polar moment of inertia of the propeller (kg} * \text{m}^2\text{); and}$
- $\omega_1 = \text{propeller rotation speed (rad / sec); and}$
- $\omega_2 = \text{rate of pitch or yaw (rad / sec).}$

The asymmetric flow through the propeller disc is discounted because the propeller diameter is less than 2.74 m as established by AMC 23.371(a) . The polar moment of inertia of the propeller is 1 kg*m^2. The rate of pitch or yaw is established as 1.0 rad/sec and 2.5 rad/sec respectively , the load factor is 2.5 and the power condition is max continuous power as prescribed in AMC 23.371(a) . Therefore, the inertial load is equal to 2.5*90.66 = 226.6 daN. The rotation of the propeller at maximum continuous power is 5500/2.429 = 2265. The gyroscopic couple is:

1. Yaw case: 2*1*2.5*(2*3.14/60)*2265 = 188.7 daN * m
2. Pitch case: 2*1*1.0*(2*3.14/60)*2265 = 75.49 daN * m