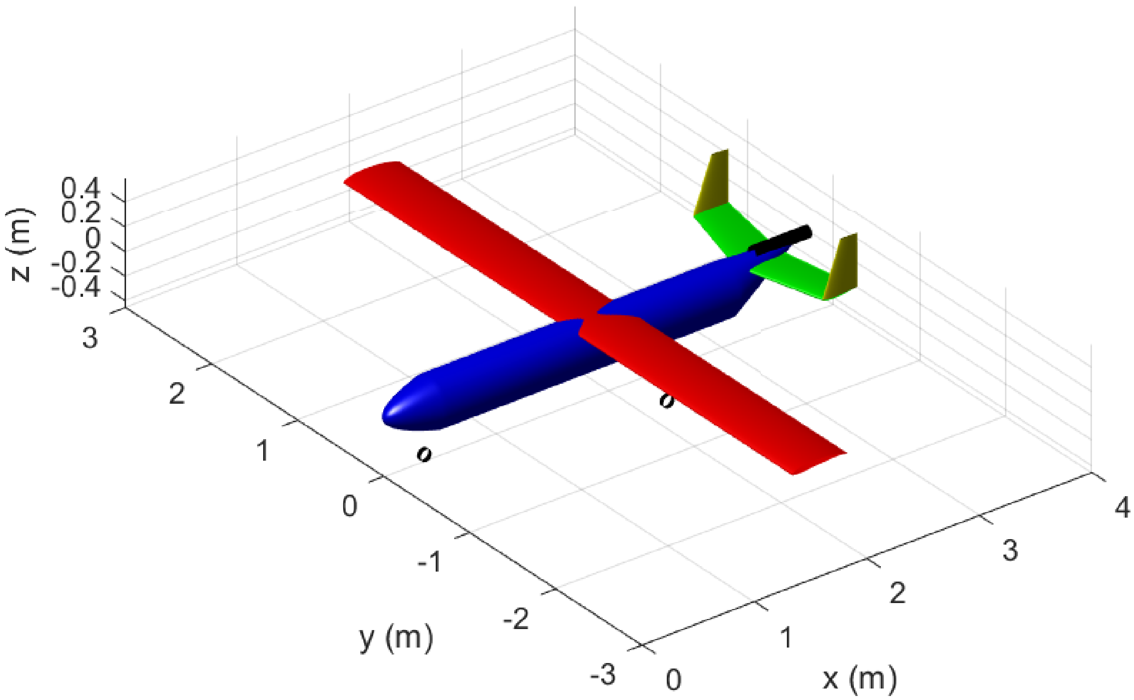
Flight Loads: DroneVLA aircraft



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Chapter 1. Introduction

This document defines the SUBPART C - Structure - Flight Loads of the:DroneVLA.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.

HERE BELOW AN EXAMPLE OF REFERENCES TO BE EDITED

Chapter 3. List of Abbreviations

- CL = lift coefficient
- CD....
- ...
- ...
- ...
- ...
- ...

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	5.2	m
S	2.589	m^2
AR	10.446	-
taper	NaN	-
sweep_first	0	deg
sweep_second	0	deg
sweep_third	0	deg
sweep_location	0	percentage
secondary_sweep_location	0	percentage
croot	0.498	m
ctip	0.498	m
xle	1.638	m
yle	0	m
zle	0.165	m
xtip_le	NaN	% fuselage length
dihedral_first	0	deg
dihedral_second	0	deg
dihedral_third	0	deg
mac	0.498	m
xmac	NaN	% fuselage length
ymac	NaN	% semispan
ypos	NaN	% semispan
zpos	NaN	% fuselage diameter
camberloc	0.15	percentage
thickchord	0.18	percentage
type	Rectangular	flag
twist_angle_first	0	deg

Wing parameters	Value	Measure unit
twist_angle_second	0	deg
twist_angle_third	0	deg
twist_angle_fourth	0	deg
sweep	0	deg
dihedral	0	deg
chord_kink_one	0.498	m
chord_kink_two	0.498	m
panel_span1	0.33	Semispan percentage
panel_span2	0.33	Semispan percentage
panel_span3	0.33	Semispan percentage
twist_angle	0	deg
mgc	0.49788	m
taper_ratio	1	Non dimensional

Table 4.2. Horizontal Tail parameters

Horizontal parameters	Value	Measure unit
S	0.529	m^2
l	1.492	m
camber	0	percentage
camberloc	NaN	percentage
thickchord	0.12	percentage
twist	0	deg
twistloc	0.25	percentage
xloc0	1.49	m
xloc	3.128	m
yloc	0	m
zloc	0.15	m
xrot	0	m
yrot	0	m
zrot	0	m
b	1.496	m
ctip	0.3136	m
croot	0.3929	m
sweep	15	deg
sweeploc	0	percentage
secsweeploc	1	percentage
dihedral	0	deg

Horizontal parameters	Value	Measure unit
location_of_camber	0.2	percentage
secondary_sweep_location	1	percentage

Table 4.3. Vertical Tail parameters

Vertical parameters	Value	Measure unit
xle	0.95	fus length percentage
croot	0.3136	m
ctip	0.15347	m
xtip_le	1	fus length percentage
b	0.4375	m
zpos	1	fus diameter percentage
S	0.1022	m^2
chord	0.3136	m
sweep	20	deg
sweeploc	0	percentage
secsweeploc	1	percentage
dihedral	0	deg
twist	0	deg
twistloc	0	percentage
MAC	0.23354	m
l_vt	1.65	m
empennage_flag	Double fin	NaN

Table 4.4. Fuselage parameters

Fuselage parameters	Value	Measure unit
length	3.64	Non dimensional
diameter	0.42	Non dimensional
Non_dim_radius_of_gyration	0.34	Non dimensional
Radius_of_gyration	NaN	m

Table 4.5. Elevator parameters

Elevator parameters	Value	Measure unit
S	0.14749	m^2
chord	0.12324	m
chord_ratio_ce_c	0.35	Non dimensional
overhang	0.12	Non dimensional
span_ratio	0.8	Non dimensional

Elevator parameters	Value	Measure unit
S_hinge	0.126	m^2
eta_inner	0.1	percentage
eta_outer	0.9	percentage
cf_c_inner	0.3	percentage
cf_c_outer	0.3	percentage
y_inner	0.0748	m
y_outer	0.6732	m
cf	0.10845	m
moment_arm	0.016021	m
ce_c_root	0.34	Non dimensional
ce_c_tip	0.36	Non dimensional

Table 4.6. Rudder parameters

Rudder parameters	Value	Measure unit
S	0.019063	m^2
chord	0.10893	m
chord_ratio_cf_c	0.35	Non dimensional
overhang	0.12	Non dimensional
span_ratio	0.8	Non dimensional
cr_c_root	0.45	Non dimensional
cr_c_tip	0.5	Non dimensional
eta_inner	0.1	Non dimensional
eta_outer	0.9	Non dimensional
croot	0.14112	m
ctip	0.076736	m
y_inner	0.021875	m
y_outer	0.19688	m
moment_arm	0.014161	m

Table 4.7. Aileron parameters

Aileron parameters	Value	Measure unit
S	0.14018	m^2
b	0.908	m
ca	0.15438	m
cb	0.019	m
y_inner	1.63	m
y_outer	2.538	m

Aileron parameters	Value	Measure unit
eta_inner	0.627	Non dimensional
eta_outer	0.976	Non dimensional
ca_c_root	0.31	Non dimensional
ca_c_tip	0.31	Non dimensional
croot	0.15438	m
ctip	0.15438	m
cf	0.13538	m
moment_arm	0.016928	m

Table 4.8. Flaps parameters

Flaps parameters	Value	Measure unit
S	0.21073	m^2
b	1.365	m
ca	NaN	m
cb	NaN	m
y_inner	0.26	m
y_outer	1.625	m
eta_inner	0.1	Non dimensional
eta_outer	0.625	Non dimensional
cf_c_root	0.31	Non dimensional
cf_c_inner	0.31	Non dimensional
croot	0.15438	m
ctip	0.15438	m
cf	0.15438	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	100	kg
W_OperativeEmpty	NaN	kg
W_Payload	NaN	kg
W_Fuel	NaN	kg
W_Crew	NaN	kg
IY	100	kg * m^2

4.3. Aerodynamic

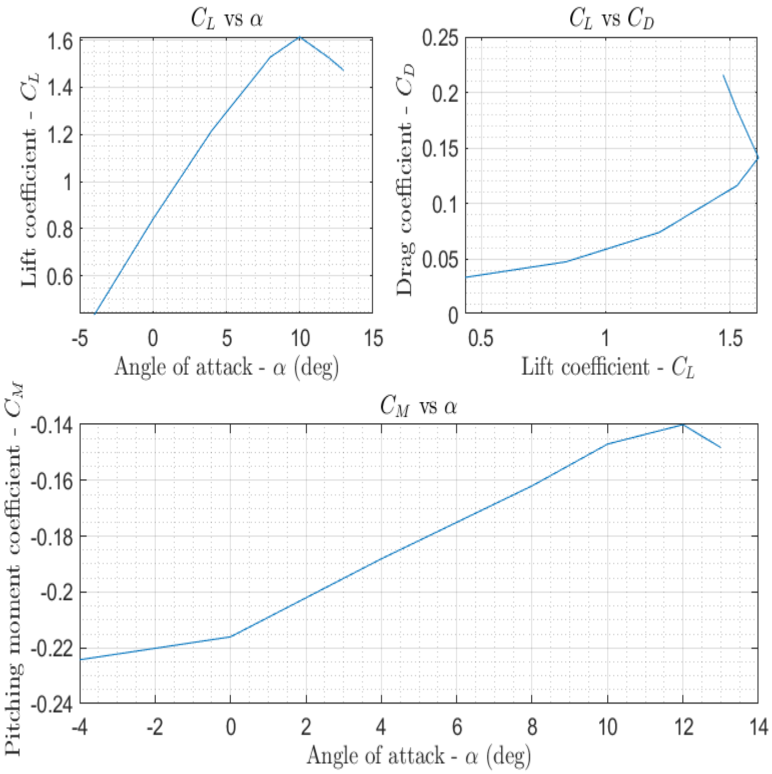

Figure 4.1. Lift, Drag and pitching moment coefficient of the 3D wing-body configuration.

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

Chapter 5. Design Airspeeds

This chapter defines the operating and design airspeed as required for certification CSVLA.

5.1. Maximum speed in level flight VH

Data not yet available...to be added Available and Required Power.

5.2. Stall speeds VS, VS0, VS1

These speeds will be verified by flight test according to certification requirements. In order to calculate the stall speed, the maximum lift coefficient of the aeroplane as a whole is determined first. The maximum lift coefficient of the aeroplane has been calculated from high fidelity CFD. In landing configuration computed with full flap, CLMAX landing =2.1 in take-off configuration leading to CLMAX takeoff =1.9, and in clean configuration, leading to CLMAX clean =1.58 , also considering the horizontal tail balancing force.

Flaps retracted(clean configuration):

$$V_S = \sqrt{\frac{2\,W_{MTOM}}{\rho_0 C_{L_{MAX_{Clean}}} S}} = \sqrt{\frac{2*981}{1.225*1.58*2.589}} = 19.7839m/s$$

Flaps extended(Landing configuration):

$$V_{S_0} = \sqrt{\frac{2\,W_{MTOM}}{\rho_0 C_{L_{MAX_{Landing}}} S}} = \sqrt{\frac{2*981}{1.225*2.1*2.589}} = 17.1606m/s$$

Flaps extended(Take-off configuration):

$$V_{S_1} = \sqrt{\frac{2\,W_{MTOM}}{\rho_0 C_{L_{MAX_{Takeoff}}} S}} = \sqrt{\frac{2*981}{1.225*1.9*2.589}} = 18.0412m/s$$

Add here comments if necessary

Note: These speeds are estimates. The methods for the estimation can be various.It is important that these estimations are as precise as possible. Flight tests will be used to validatethe stall speeds. In case the flight tests show different values, this might have an impact on the speedsused for design and ultimately might impair the compliance to the-CSVLA

5.3. Design manoeuvring speed VA

According to CSVLA 335 , the maneuvring speed VA cannot be less then:

$$V_A \geq V_S\sqrt{n_{max}} = 19.7839 * \sqrt{3.8} = 38.566m/s$$

Add here comments if necessary

5.4. Flaps maximum operating speed VF

According to CSVLA 345, such speed shall be not less than the greater of 1.4*VS and 1.8*VS0.

The speed has been selected as the greater between 1.4VS =27.6975m/s and 1.8 VSF =24.0248m/s, where VSF is the computed stalling speed with flaps fully extended at the design weight.

The flaps operating speeds is:

$V_F \ = 27.6975 m/s$

5.5. Flaps maximum extension speed VFE

On this aeroplane the maximum flap extension speed is identical to the flap operating speed VF. This speed is the maximum speed for flaps in take-off and landing configuration.

$V_{FE} \ = 27.6975 m/s$

5.6. Design cruising speed VC

According to CSVLA 335, VC (in m/s) may not be less than

$$2.4\sqrt{\frac{Mg}{S}}\left(V_C(kt) = 4.7\sqrt{\frac{Mg}{S}}\right) \rightarrow 2.4 * \sqrt{\frac{100 * 9.8066}{2.589}} \ = 46.7095 m/s$$

where M/S is the wing loading in kg/m^2 and g is the acceleration due to gravity in m/s^2.

VC need not be more than 0.9*VH at sea level.

VH must be available. Otherwise previous value is considered!!!

$V_C \ = 46.7095 m/s$

5.7. Design dive speed VD

According to CSVLA 335

(1) VD may not be less than 1.25*VC; and (2) with VCmin, the required minimum design cruising speed, VD may not be less than 1.40*VCmin.

(1) 1.25*VC =58.3869m/s

(2) 1.4VCmin =40m/s

$V_C = 1.25 * 46.7095 = 58.3869 m/s$

5.8. Demonstrated dive speed VDF

VDF is not a design airspeeds for this category.

5.9. Never exceed speed VNE

VNE is not a design airspeeds. It must be checked into sec. CSVLA 1505 Airspeed limitations.

5.10. Design Airspeeds summary

Design airspeeds summary is resumed in Table:Design airspeeds

Table 5.1. Design airspeeds

Design airspeeds	Value	Measure unit
VS	19.78	m/s
VS0	17.16	m/s
VS1	18.04	m/s
VA	38.57	m/s
VC	46.71	m/s
VD	58.39	m/s
VE	58.39	m/s
VG	30.46	m/s
VS_inv	24.87	m/s
VF	30.89	m/s

Chapter 6. Altitude

The maximum permissible operational altitude for the aircrat is 1300m. Despite the-CSVLA requirements do not require to accounts for the effects of altitude,such effects have been considered up to 1300m. In fact the gust load factor have been calculated at such altitude. This is considered acceptable since it covers the operational range within which the aeroplane will fly most of the time.

(Note: the-CSVLA requirement does not require to account for the effects of altitude.Calculating the loads at sea level would be acceptable. In this case, the choice to consider such effect up to 1300m is a decision of a designer, which would be accepted by the team.)

Chapter 7. Manoeuvring and Gust load factors n

According to CSVLA 337(a) , the positive limit moeuvring load factor n may not be less than 3.8, while according to CSVLA 337(b) , the negative limit manoeuvring load factor may not be less than -1.5.

The following value will be considered:

1. nmax =3.8
2. nmin =-1.5

7.1. Gust envelope

Gust load factors need to be considered because they can exceed the prescribed maximum load factors at different weights and altitudes. Since gust loads depend on air density and aircraft mass they will be calculated for Compliance with the flight load requirements of this subpart to show:

(1) At each critical altitude within the range in which the aeroplane may be expected to operate from sea level up to maximum operative altitude equal to: 1300 m

(2) At each practicable combination of weight and disposable load within the operating limitations specified in the Flight Manual according to requirement CSVLA 321 and fully extended (requirement CSVLA 345(b) at VF).

The calculation is based on CSVLA 341 . To calculate the gust loads at altitudes other than at sea level the following equation is altered to include the density at any altitude.

$$n = 1 + \frac{1/2\, \rho_0\, V\, a\, K_g\, U_{de}}{Mg/S}$$

where:

- $K_g = \dfrac{0.88\mu_g}{5.3 + \mu_g}$
- $\mu_g = \dfrac{2(M/S)}{\rho \bar{C} a}$
- $U_{de} =$ derived gust velocities referred to in CSVLA 333(c) (m/s)
- $\rho_0 =$ density of air at sea level (kg/m3)
- $\rho =$ density of air (kg/m3)
- $M/S =$ wing loading (kg/m2)
- $\bar{c} =$ mean geometric chord (m); $g =$ acceleration due to gravity (m/s2);
- $a =$ slope of the aeroplane normal force coefficient curve CNA per radian

Since the gust loads on the wing and tail have been chosen to be treated together, a is the slope of the lift-curve of the aeroplane is equal to a =5.2341/rad and0.09131/deg.

The gust speed at VC is equal to: 15.24m/s

The gust speed at VD is equal to: 7.62m/s

TABLE TO BE CHECKED!!!

Table 7.1. Gust load factor, different Speeds and Altitude

ID	V(m/s)	M(kg)	M/S(kg/m^2)	Altitude(m)	rho(kg/m^3)	mug	Kg	Ude(m/s)	n
A1	45.79	100	38.62	1300	1.079	27.47	0.7377	15.24	5.357
C	46.71	100	38.62	1300	1.079	27.47	0.7377	15.24	5.444
D	58.39	100	38.62	1300	1.079	27.47	0.7377	7.62	3.778

(Note: the applicant should provide the method for the calculation of the slope of the lift-curve of the aeroplane)

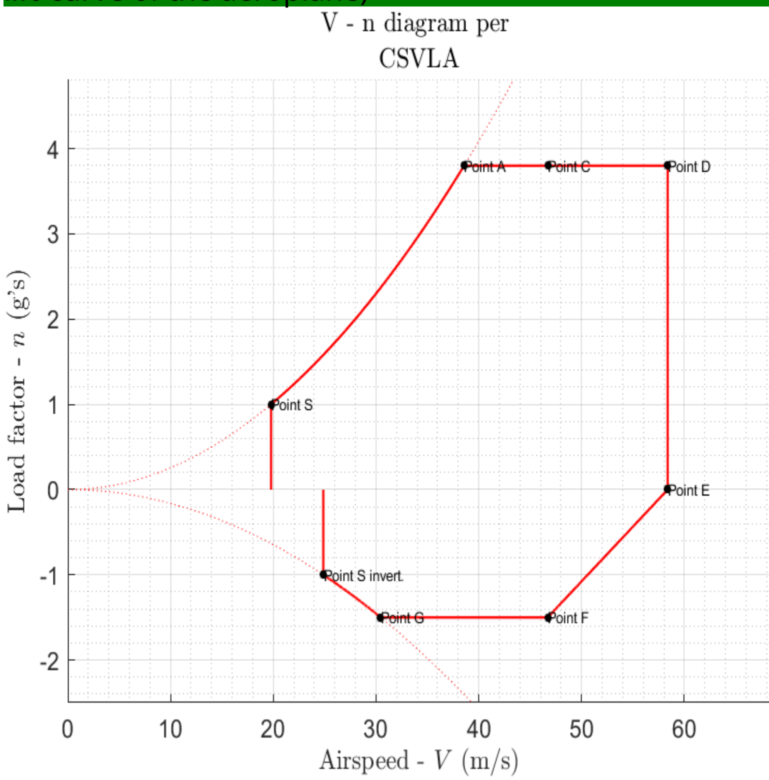

Figure 7.1. V-n diagram

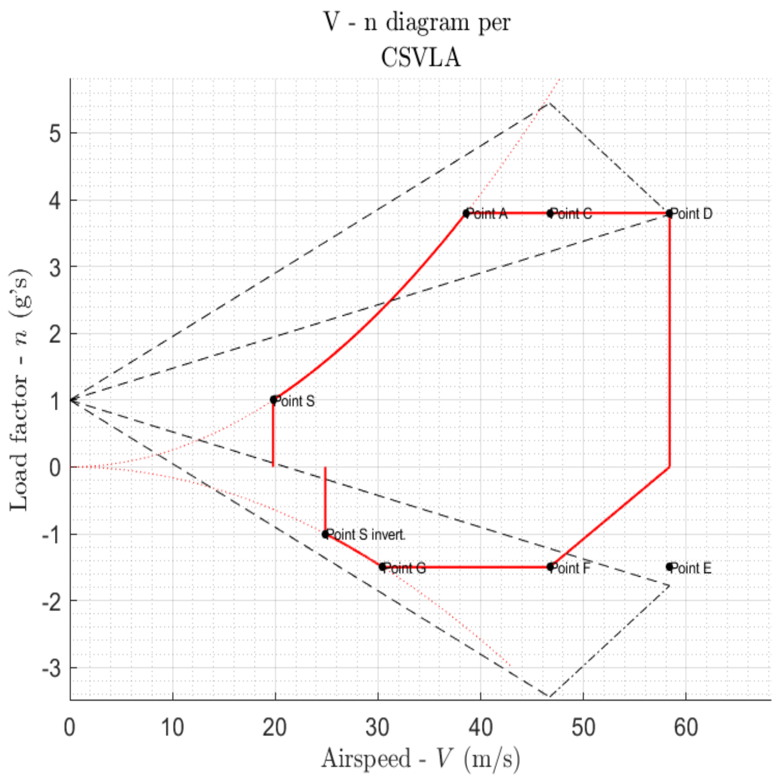

Figure 7.2. Gust diagram

Chapter 8. V-n Envelope

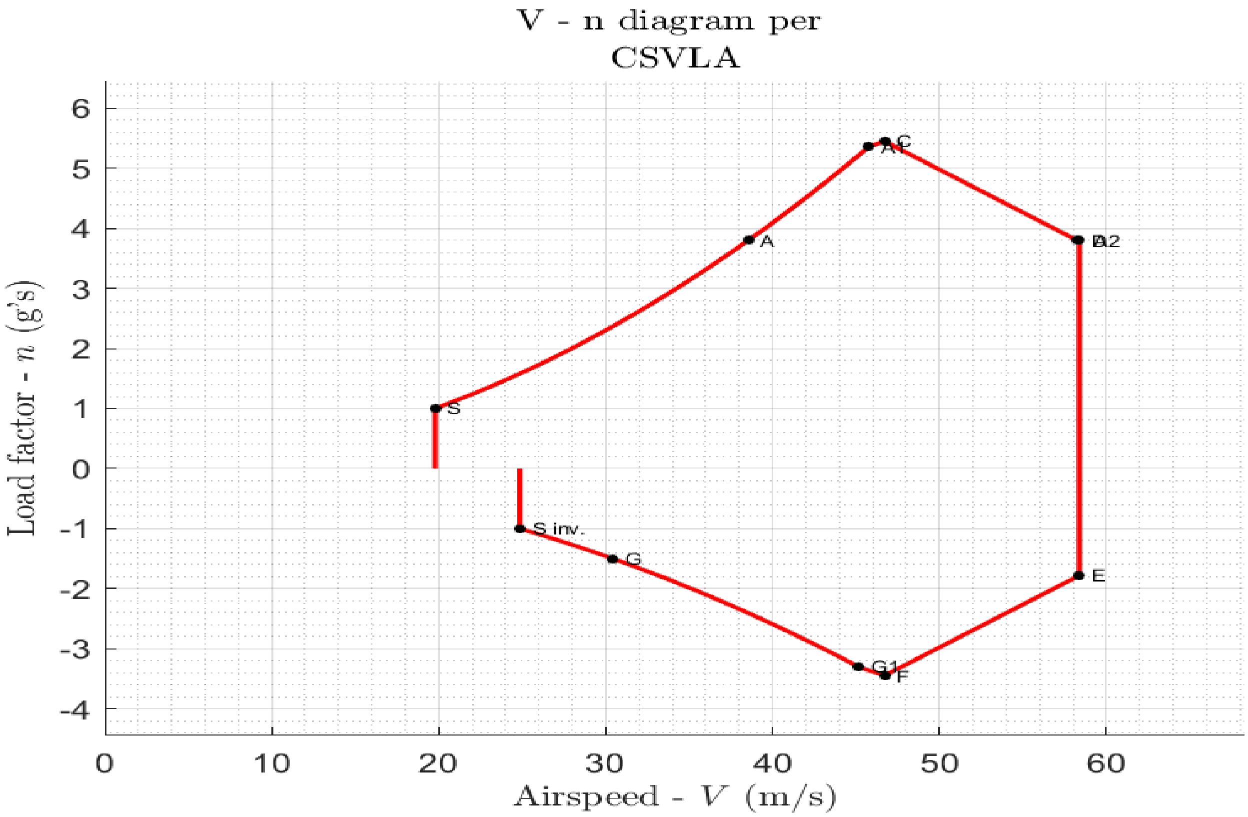

Figure 8.1. Maneuver and Gust load factors and diagram

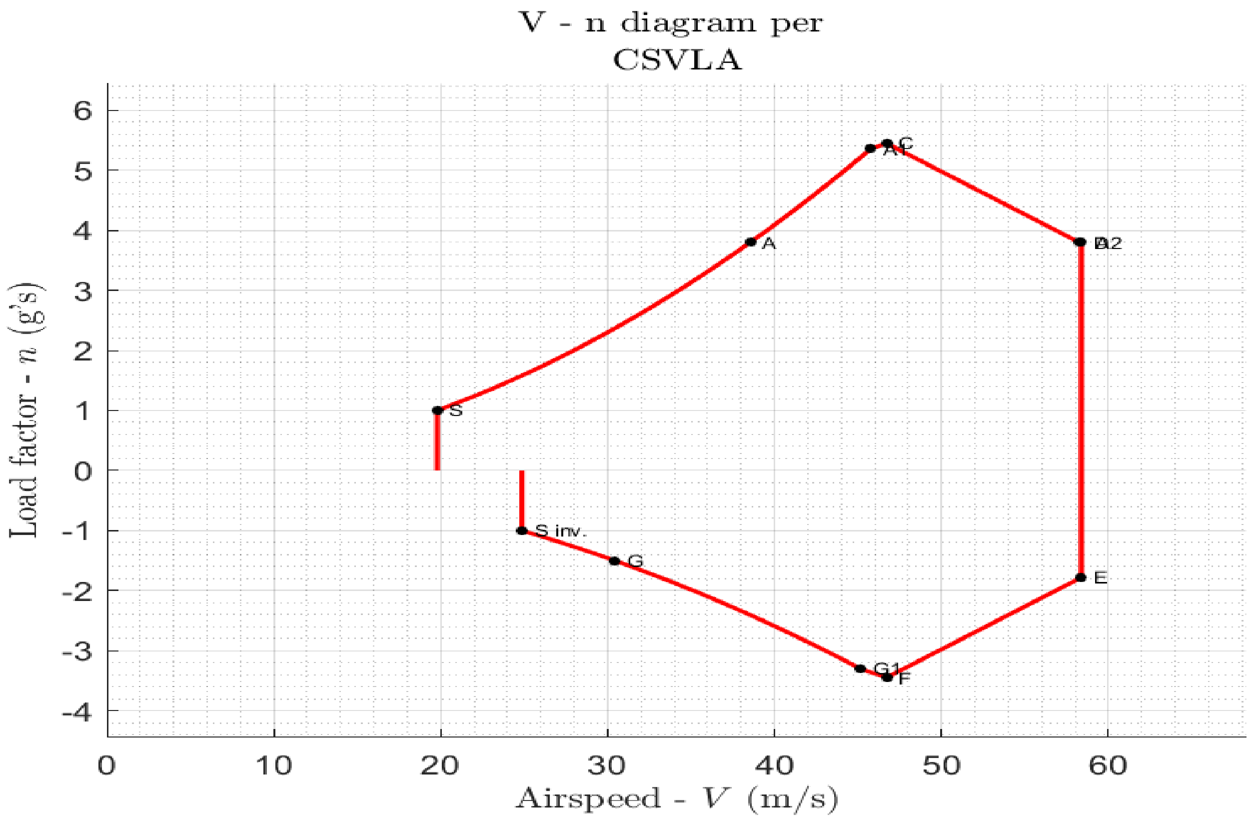

Figure 8.2. Maneuver and Gust load factors and diagram

ADD HERE V-n Envelope

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.

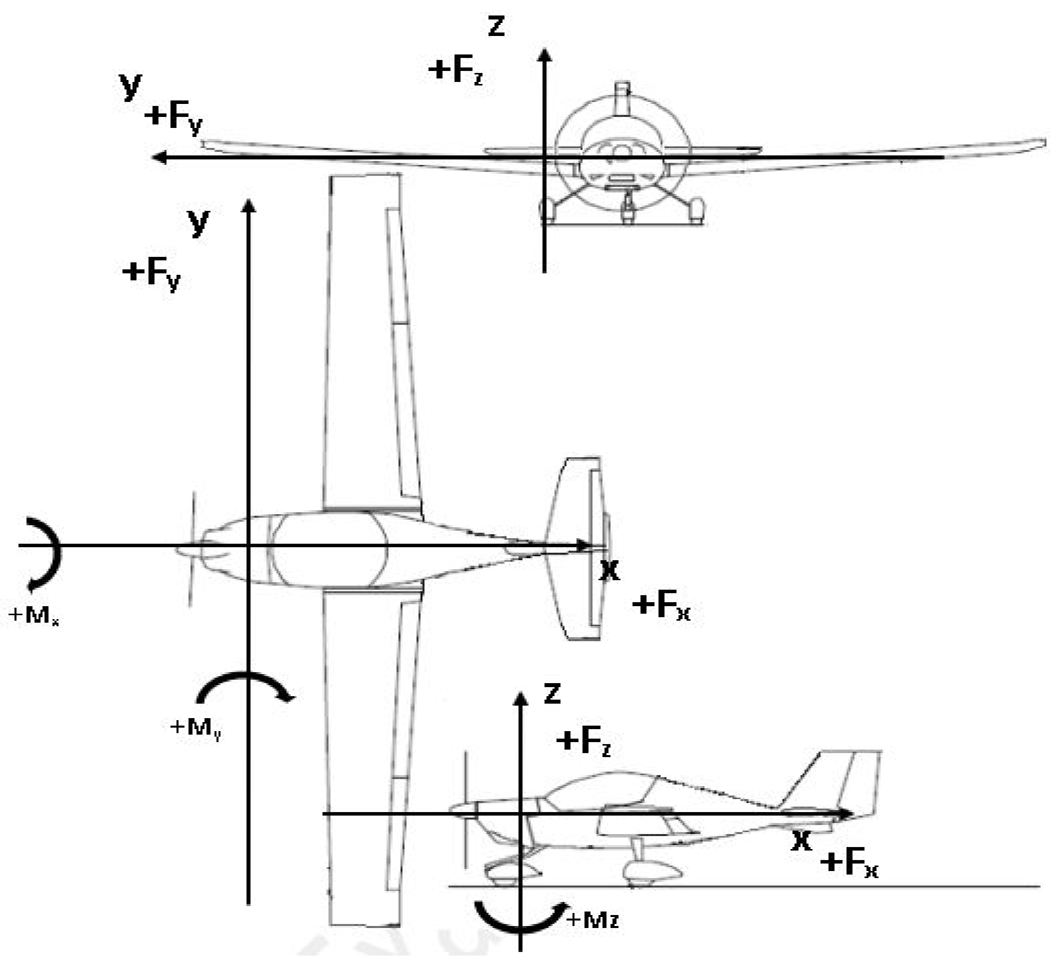

Figure 9.1. Reference axis

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:

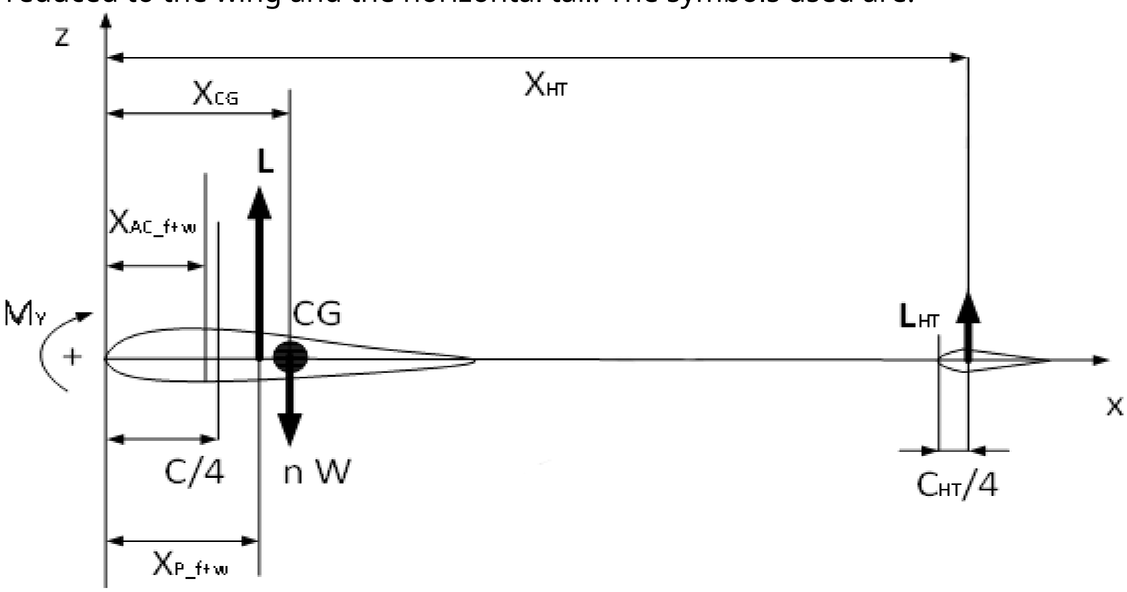

Figure 9.2. Simplified equilibrium of the aircraft

- 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 it is possible to write the following equilibrium equations:

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

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

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

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

Table 9.1. Balance parameters.

Parameter	Value	Unit of measure
XCG	2.884	m
XCG/MAC	0.25	% MAC
XAC/MAC	0.25	Non dimensional
XP	0.25	Non dimensional
nW	380	kg
XHT	1.492	m
ZCG	0.33	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.

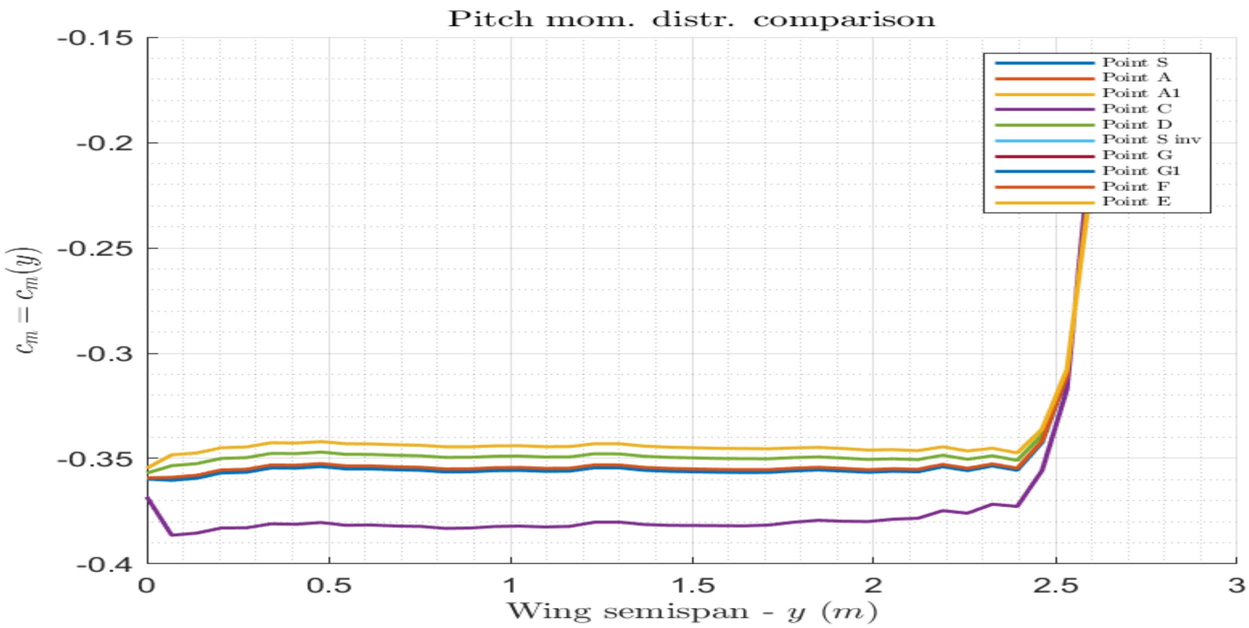

Figure 9.3. Pitching moment coefficient distribution along the wing semi-span.

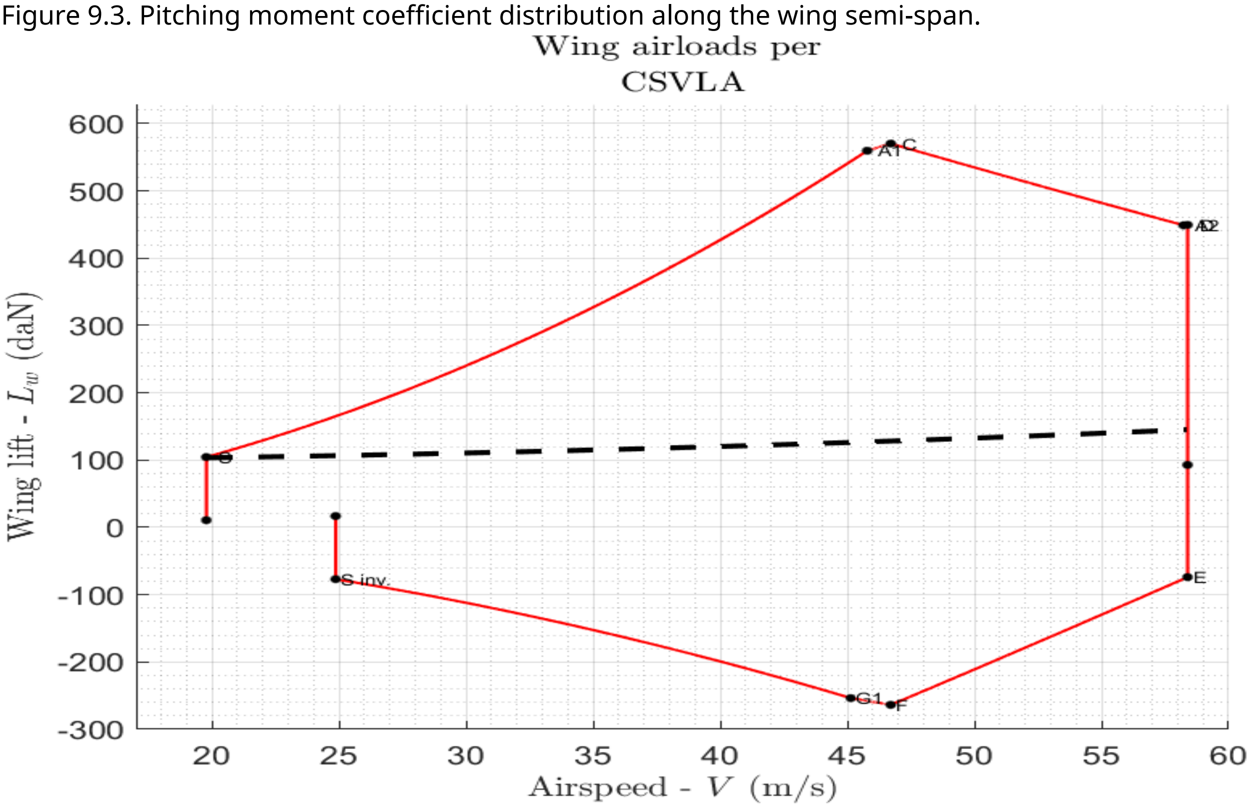

Figure 9.4. Wing airloads

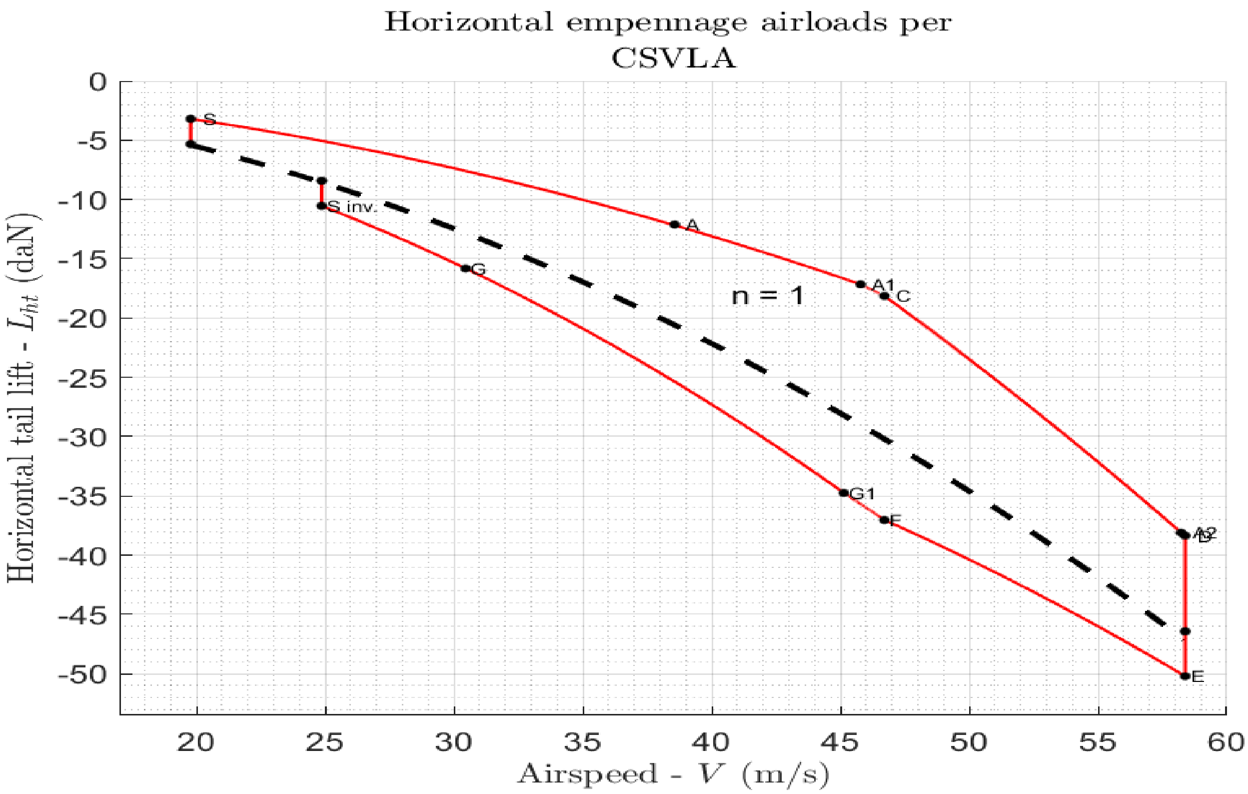

Figure 9.5. Balancing loads

9.5. Complete aircraft balancing loads

Here, the aircraft balancing loads are collected inside table.

Table 9.2. Flight envelope points.

Point	V(m/s)	n(g's)	alpha(deg)	CL	L(daN)	L tail(daN)
Point A	38.57	3.8	9.972	1.604	384.8	-12.11
Point C	46.71	5.444	9.972	1.604	570.2	-18.14
Point D	58.39	3.8	-0.4263	0.7603	449.3	-38.35
Point G	30.46	-1.5	-18.57	1	-131.3	-15.82
Point F	46.71	-3.444	-18.32	0.9762	-263.7	-37.03
Point E	58.39	-1.778	-11.29	0.3225	-73.93	-50.2

Chapter 10. Loads on the wing

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.

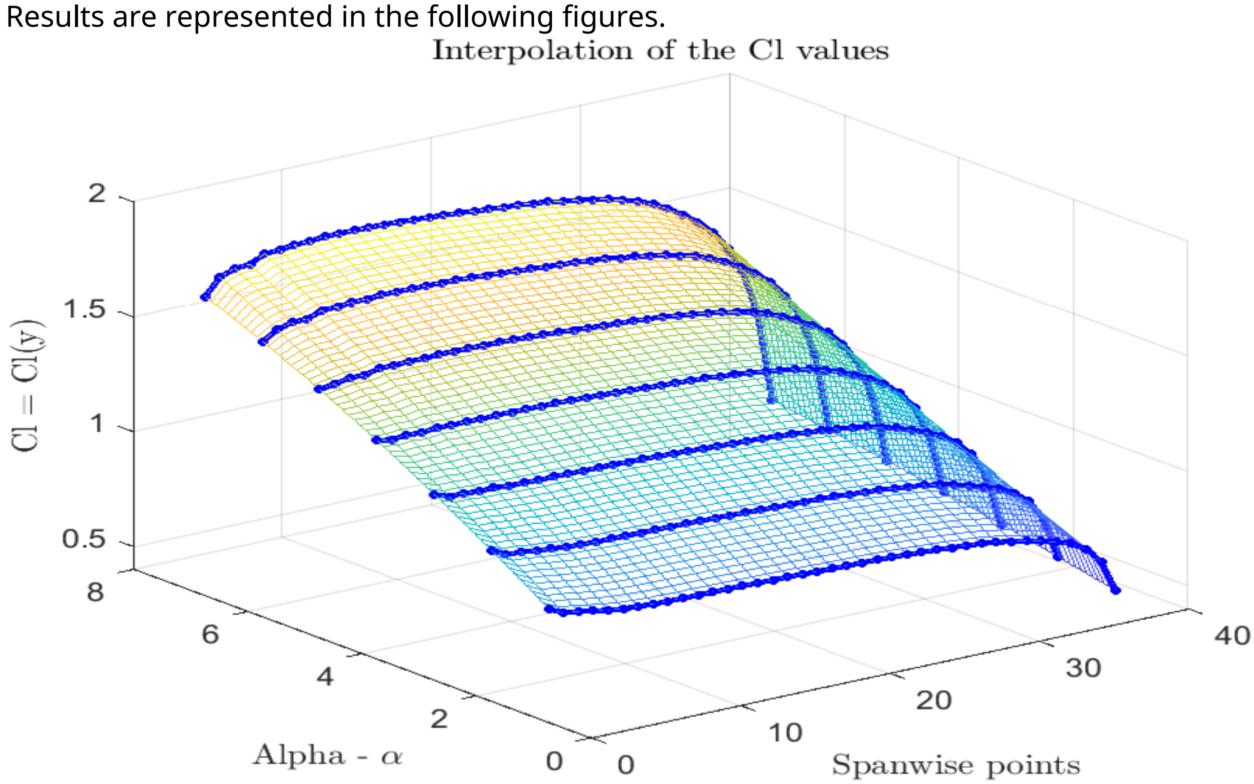

Figure 10.1. Wing lift coefficient spanwise distribution

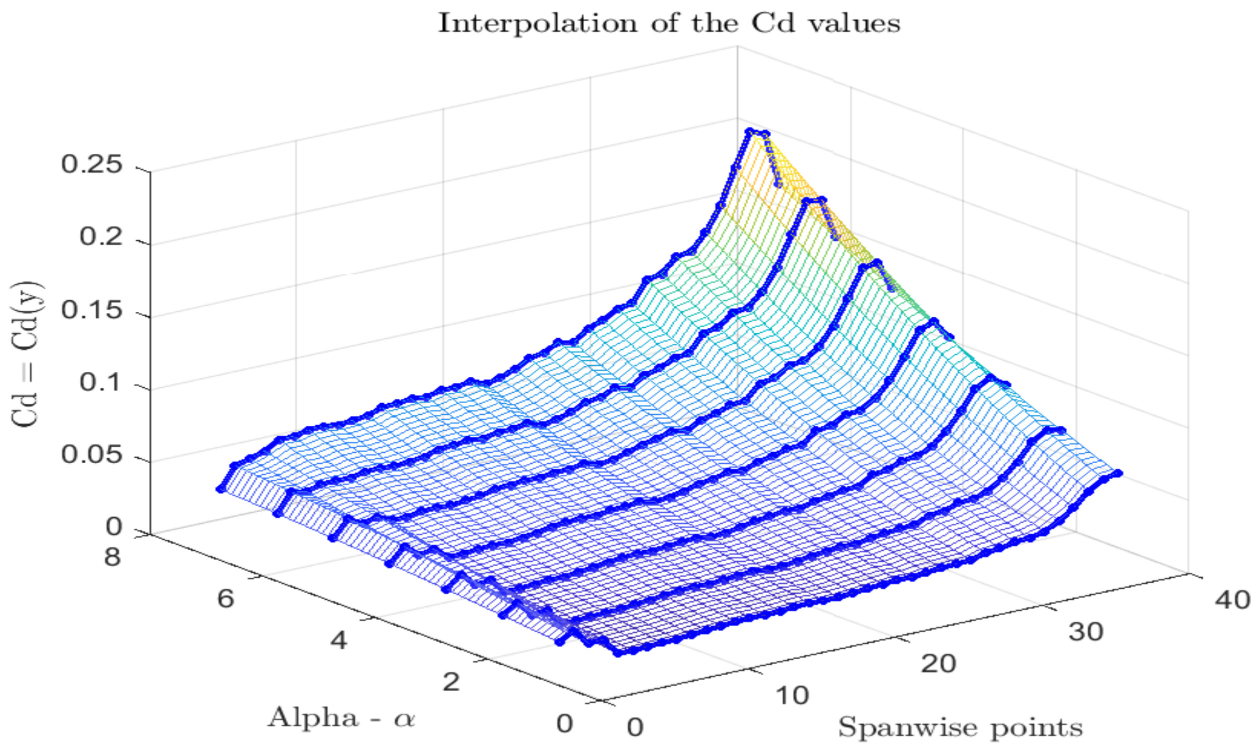

Figure 10.2. Wing drag coefficient spanwise distribution

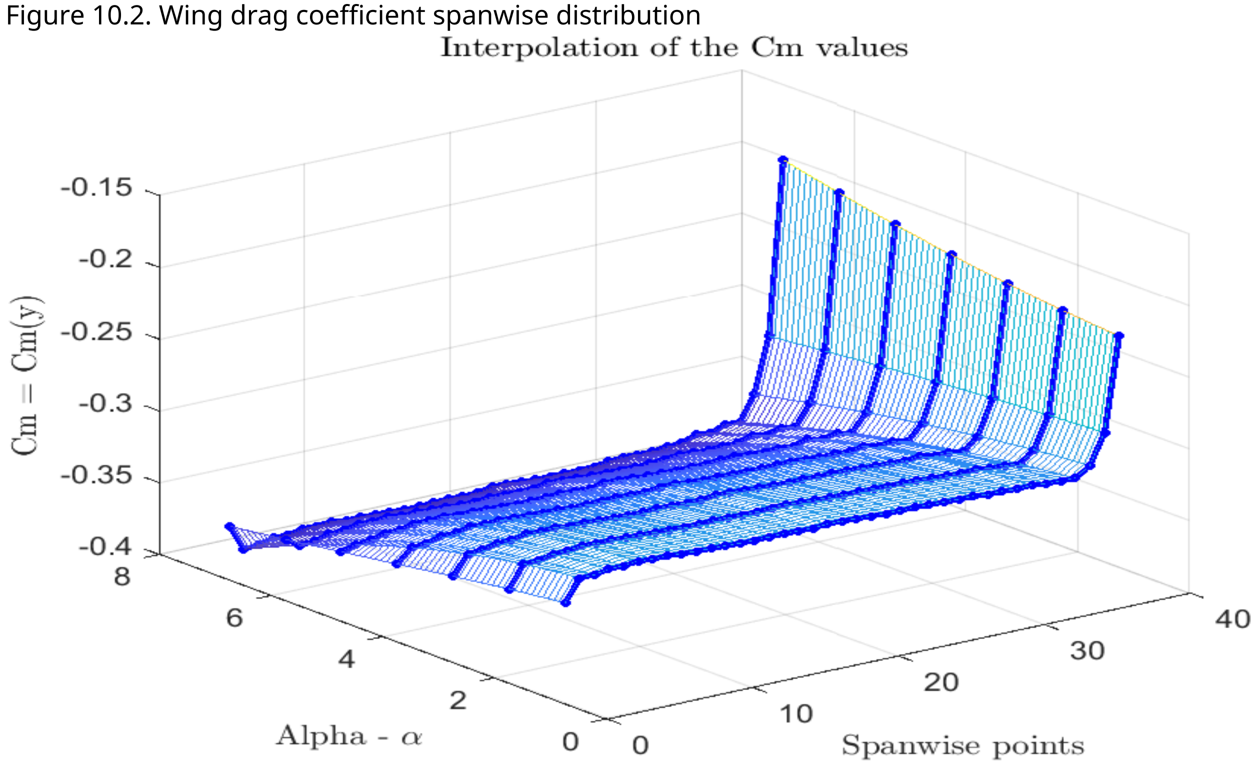

Figure 10.3. Wing pitching moment coefficient (0.25mac) spanwise distribution

10.2.2. Normal and parallel component

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.

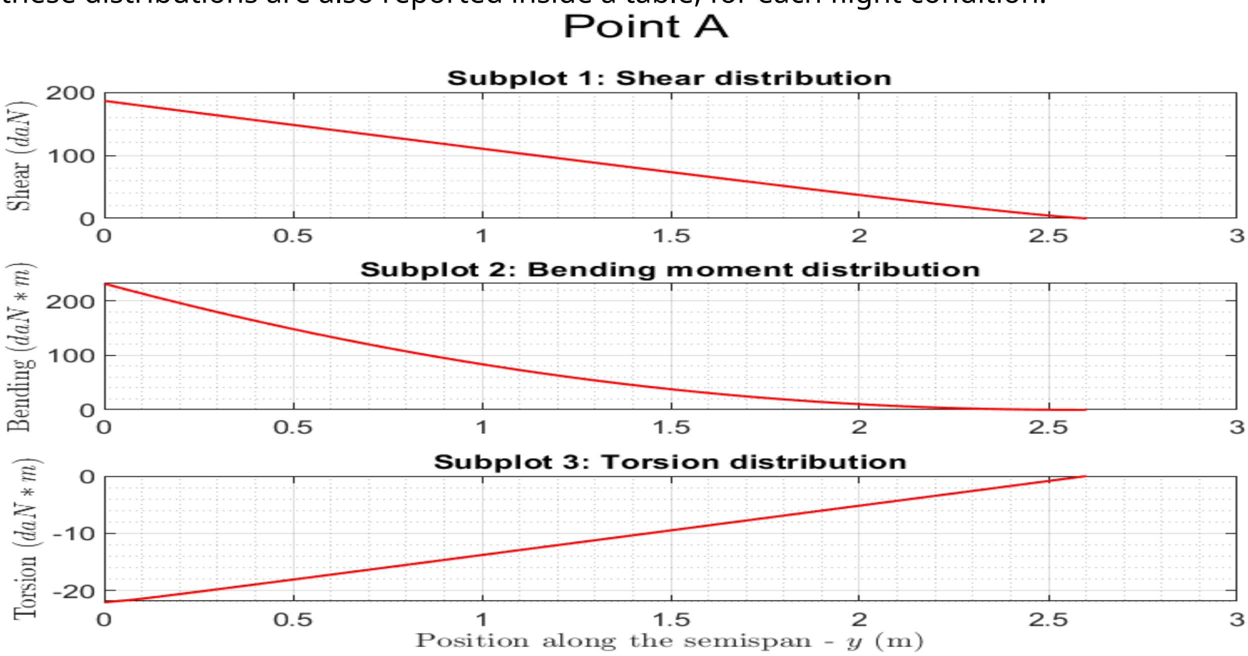

Figure 10.4. Shear, Bending and Torsion due to airloads - POINT A

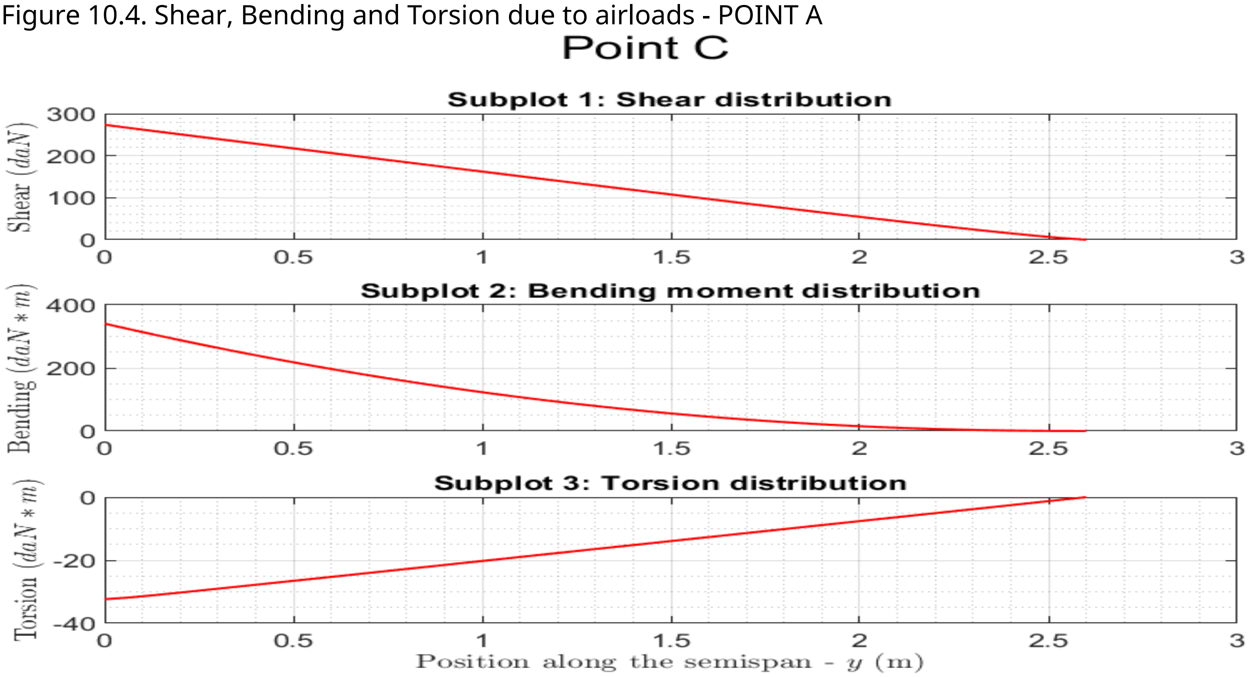

Figure 10.5. Shear, Bending and Torsion due to airloads - POINT C

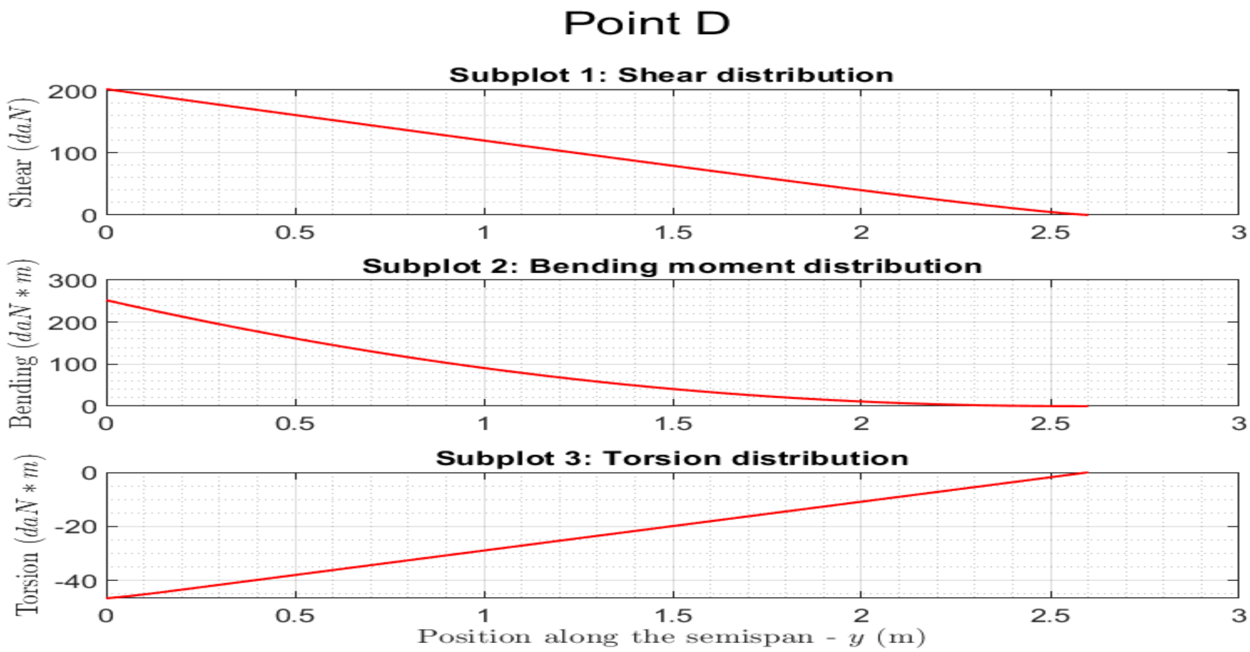

Figure 10.6. Shear, Bending and Torsion due to airloads - POINT D

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 A	38.57	3.8	186.4	232.2	-22.06
Point C	46.71	5.444	273.4	340.6	-32.36
Point D	58.39	3.8	203.3	252.4	-46.61
Point G	30.46	-1.5	-70.69	-88.37	12.91
Point F	46.71	-3.444	-162.5	-203.1	30.24
Point E	58.39	-1.778	-87.7	-110	46.01

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

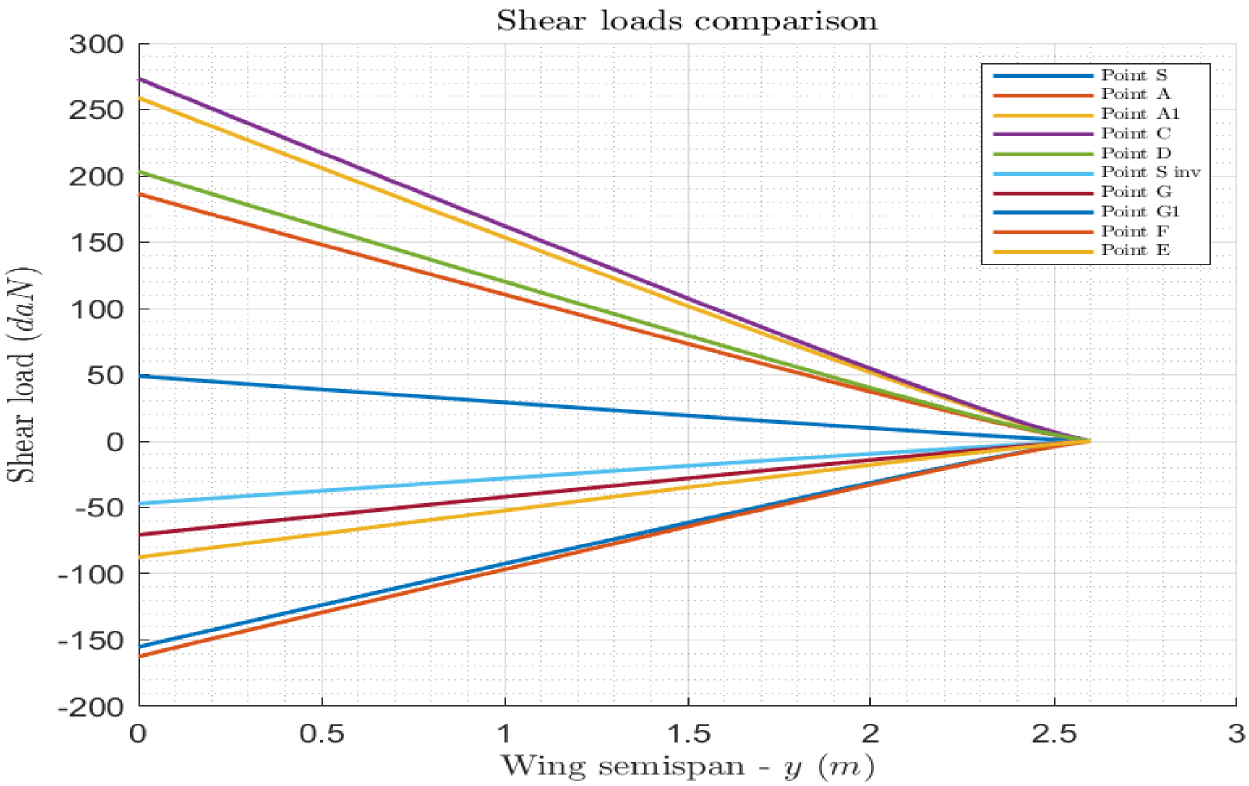

Figure 10.7. Shear comparison

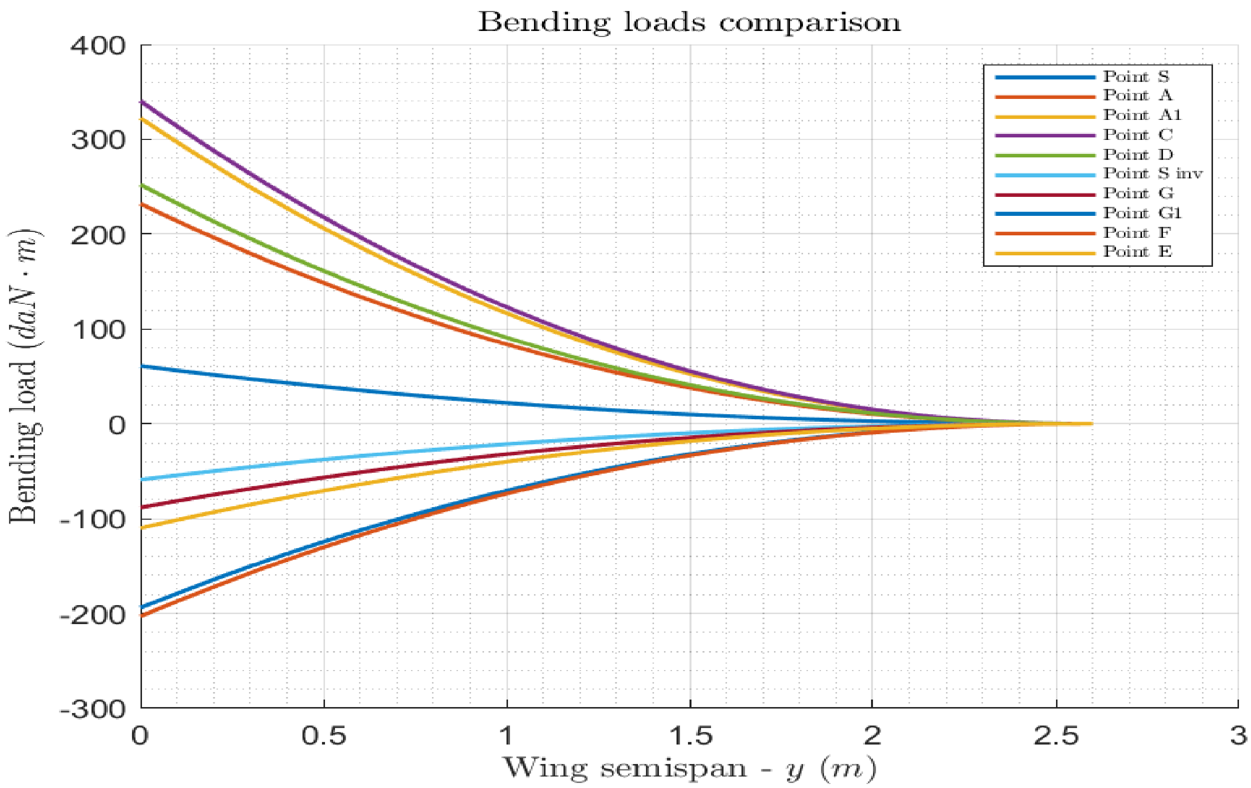

Figure 10.8. Bending comparison

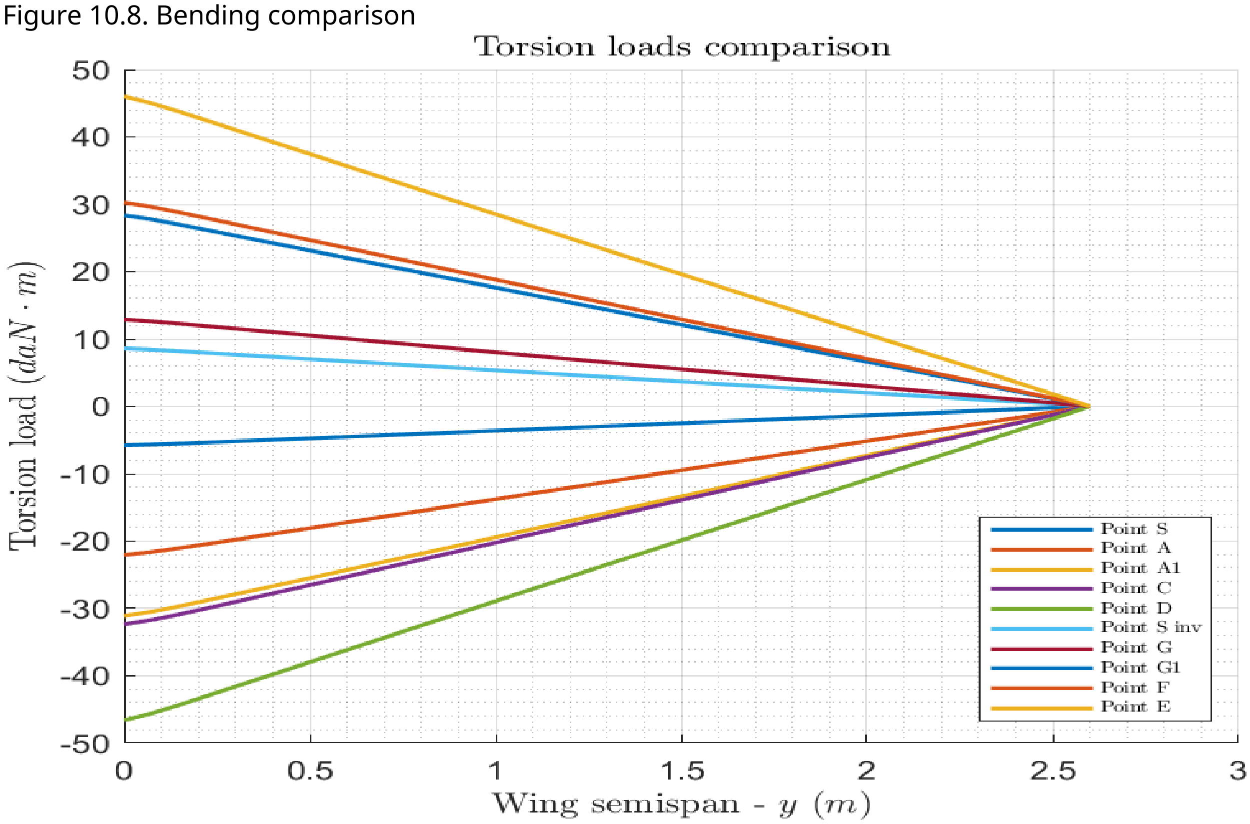

Figure 10.9. Torsion comparison

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 10.2. Unsymmetrical flight conditions. Pitching moment at various flight conditions.

y(m)	cmA	cmC	cmD
0	-0.3682	-0.3682	-0.3569
0.2053	-0.3829	-0.3829	-0.3499
0.4105	-0.3812	-0.3812	-0.3477
0.6158	-0.3815	-0.3815	-0.348
0.8211	-0.3831	-0.3831	-0.3494

y(m)	cmA	cmC	cmD
1.026	-0.382	-0.382	-0.3488
1.232	-0.3802	-0.3802	-0.3477
1.437	-0.3817	-0.3817	-0.3494
1.642	-0.3819	-0.3819	-0.35
1.847	-0.5293	-0.5031	-0.3822
2.053	-0.5288	-0.5026	-0.3831
2.258	-0.5259	-0.4998	-0.3834
2.463	-0.5057	-0.4795	-0.3718

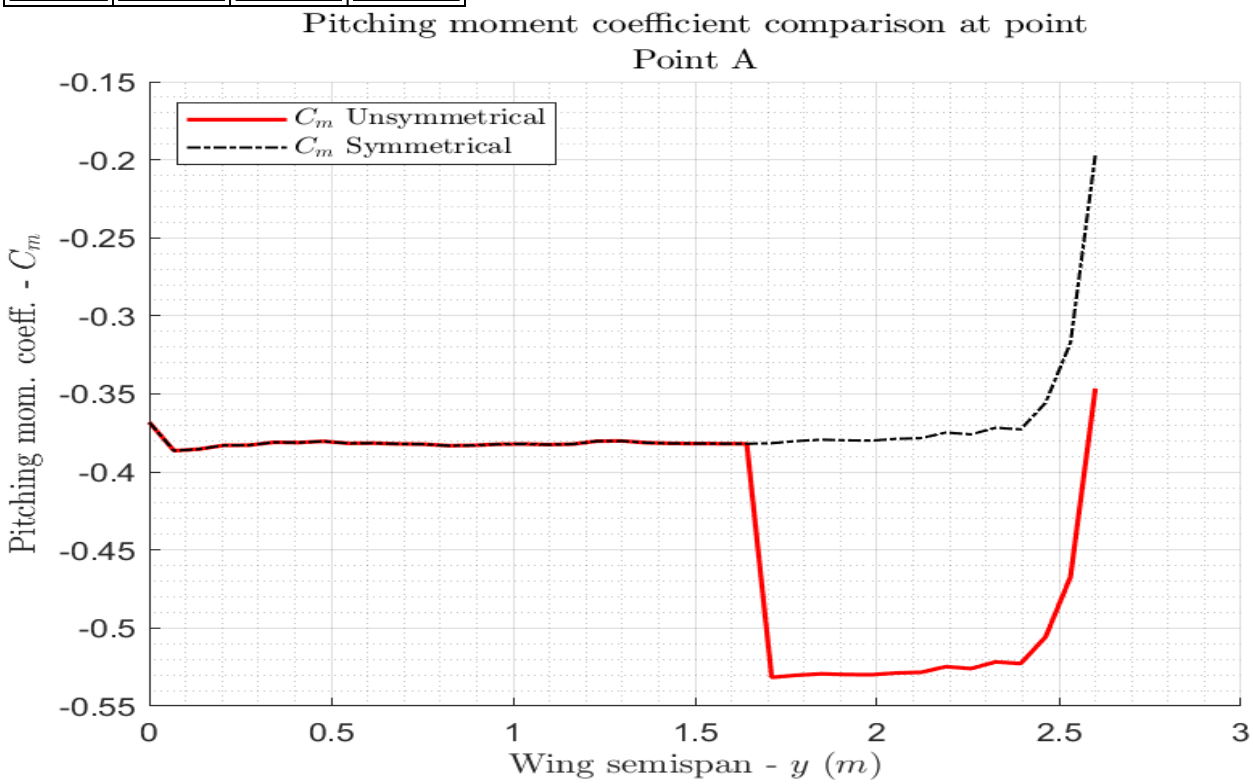

Figure 10.10. Pithcing moment coefficient - POINT A

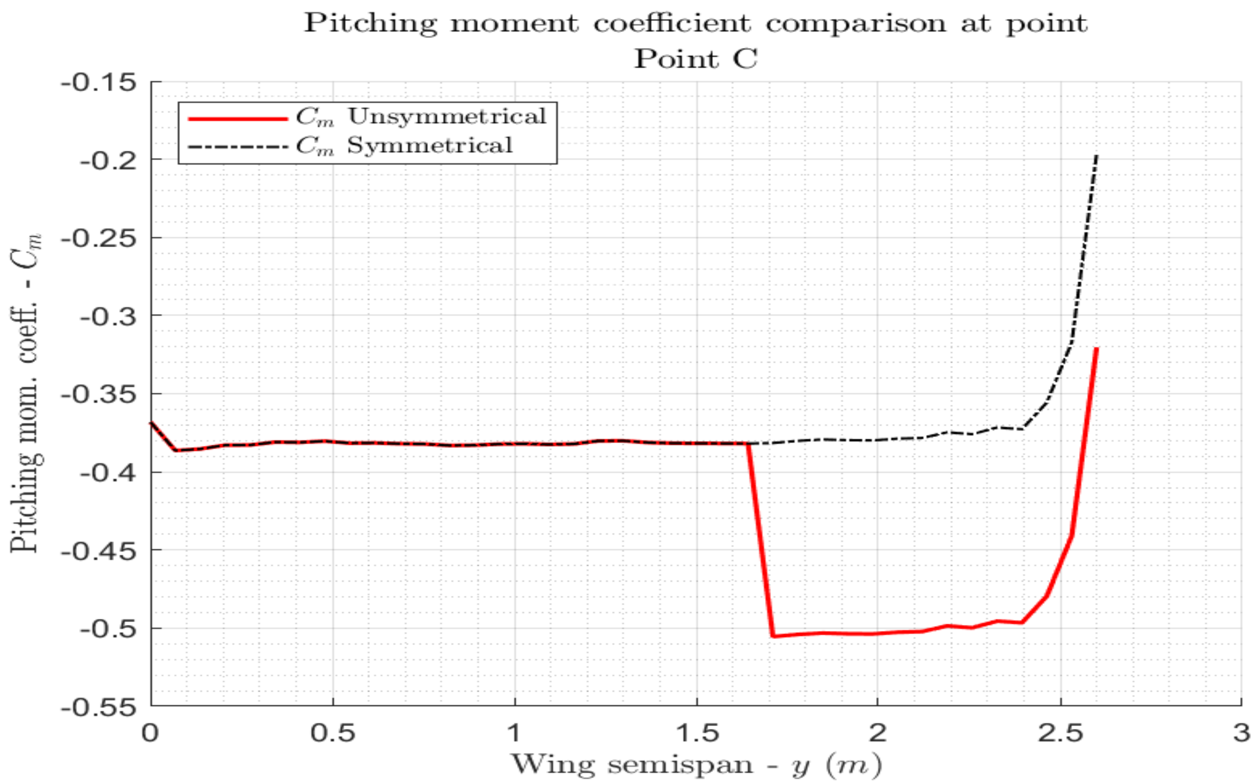

Figure 10.11. Pithcing moment coefficient - POINT C

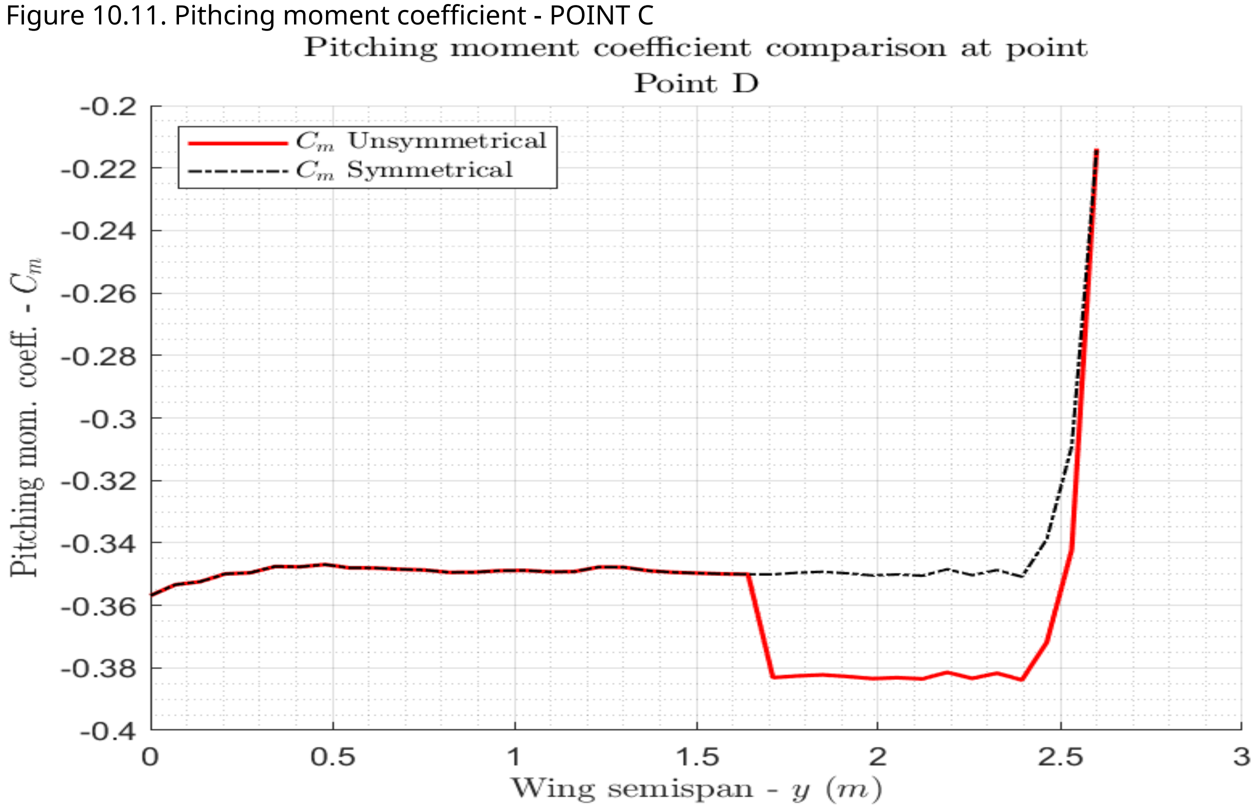

Figure 10.12. Pithcing moment coefficient - POINT D

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 10.3. 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
0.2053	-2.175	-1.522	-3.012	-2.109	-3.587	-2.511
0.4105	-4.603	-3.222	-6.396	-4.477	-7.651	-5.356
0.6158	-7.051	-4.935	-9.809	-6.866	-11.72	-8.205
0.8211	-9.506	-6.654	-13.23	-9.263	-15.79	-11.05
1.026	-11.62	-8.135	-16.25	-11.37	-19.68	-13.78
1.232	-13.39	-9.374	-18.84	-13.19	-23.39	-16.38
1.437	-15.16	-10.61	-21.43	-15	-27.09	-18.97
1.642	-16.93	-11.85	-24.03	-16.82	-30.8	-21.56
1.847	-18.7	-13.09	-26.63	-18.64	-34.51	-24.16
2.053	-20.47	-14.33	-29.23	-20.46	-38.21	-26.75
2.258	-22.24	-15.57	-31.82	-22.27	-41.9	-29.33
2.463	-24.01	-16.81	-34.42	-24.1	-45.62	-31.93

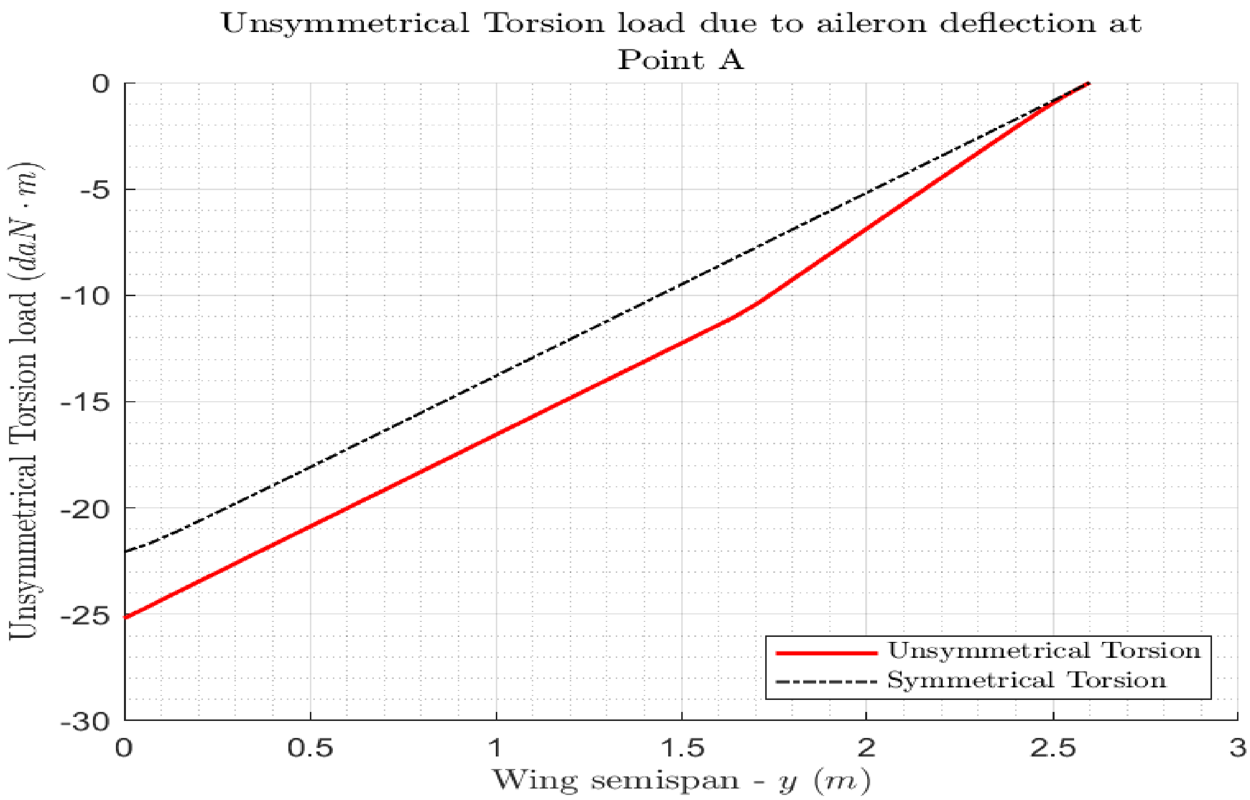

Figure 10.13. Torsion distribution full loads - POINT A

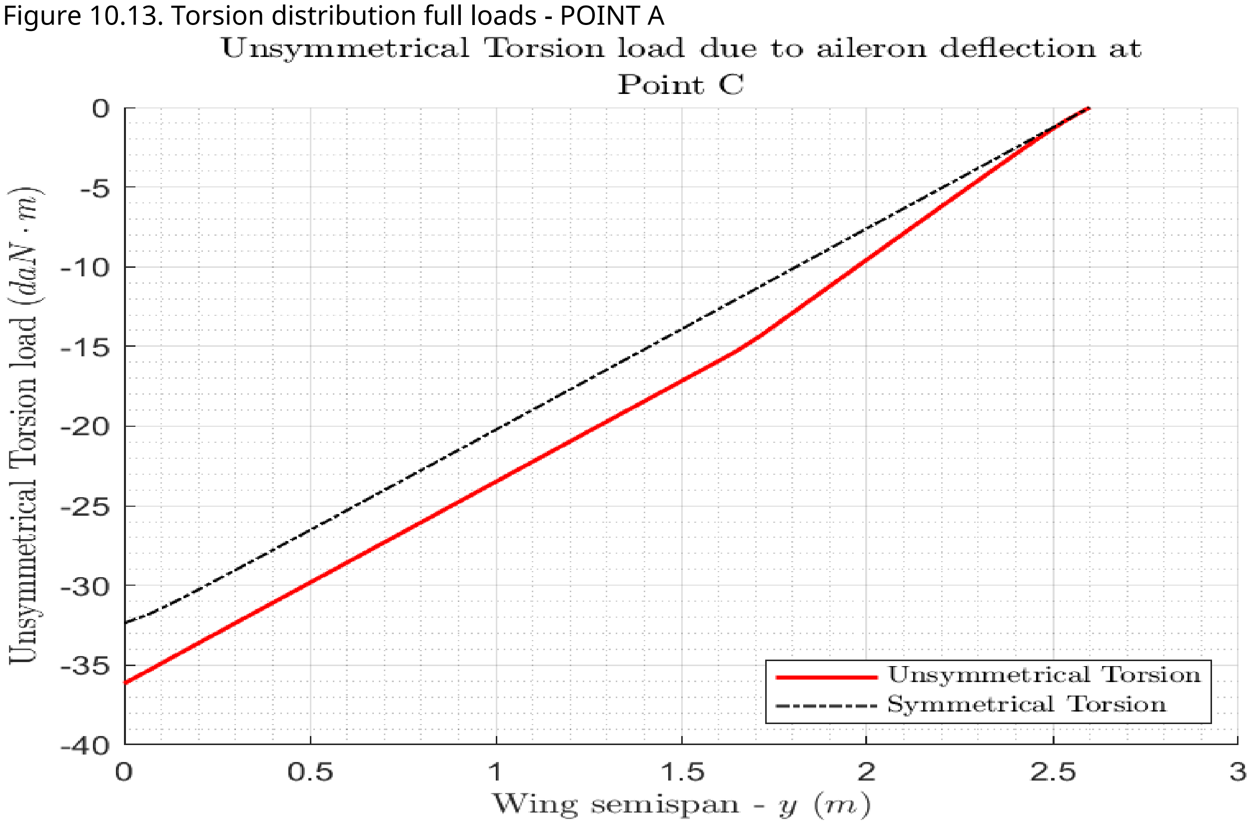

Figure 10.14. Torsion distribution full loads - POINT C

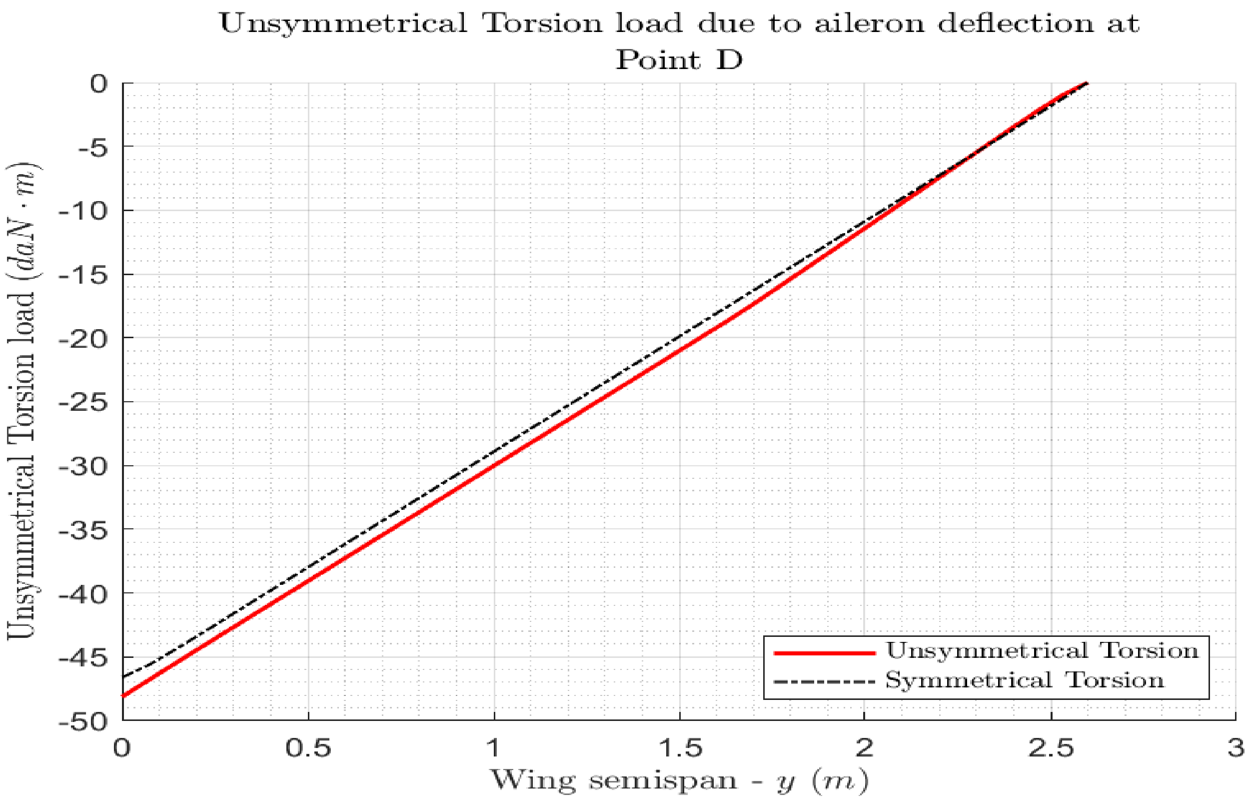

Figure 10.15. Torsion distribution full loads - POINT D

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.

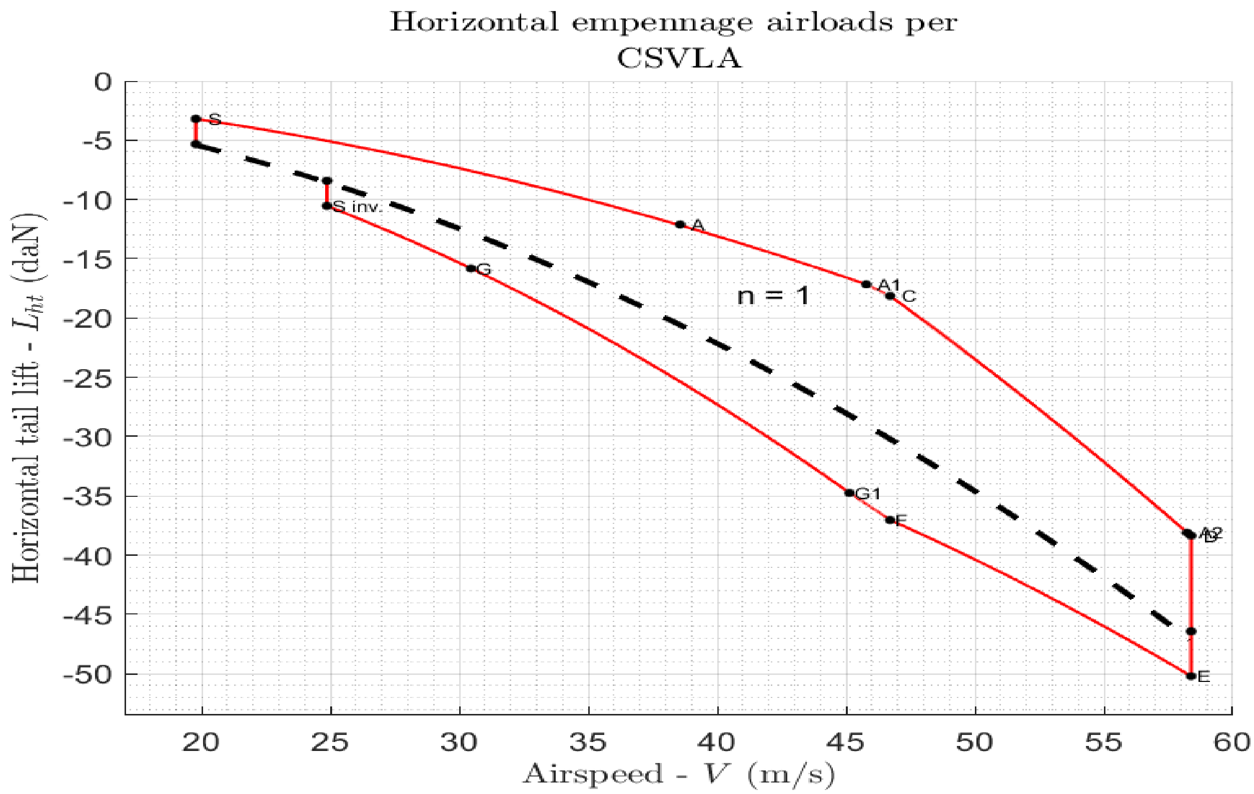

Figure 11.1. Balancing loads

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:

- $\theta = \text{rotation pitching angle}$
- $q = \text{dynamic pressure (Pa)}$
- $S_{tail} = \text{horizontal tail area } (\mathrm{m}^2)$
- $d = \text{C.G.} - \text{tail} - \text{A.C. distance (m) with } x_{C.G.} = 0.25 * MAC$
- $a_{tail} = \text{tail lift curve slope } (1/\text{deg});$
- $I_y = \text{airplane pitching inertia moment } (\text{kg} * \mathrm{m}^2)$
- $\omega = \dfrac{\delta_{e,max} * \tau}{t_{deflection}} = \text{control angular speed of plane deflection (rad/sec)}$
- $\tau = 0.5 = \text{elevator efficiency factor;}$
- $t_{deflection} = \text{total time required to full stop;}$
- $\delta_{e,max} = \text{maximum elevator deflection;}$
- $\dfrac{\Delta\, v}{V_A} = \text{damping angle}$
- $DF = \text{damping effect reduction factor} = 0.3$

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 -9.437 deg and a corresponding limit tail load of -37.75 daN;
2. a pitch down case where the resultant tailplane angle of attack is 11.8 deg and a corresponding limit tail load of 47.19 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
0.7583	0.01139	1.276	0.0001322	0.017	1.268
1.508	0.04541	2.551	0.000527	0.06775	2.521
2.248	0.1018	3.827	0.001181	0.1518	3.759
2.98	0.1802	5.102	0.002092	0.2689	4.982
3.703	0.2805	6.378	0.003255	0.4185	6.191
4.418	0.4023	7.653	0.00467	0.6003	7.386
5.124	0.5455	8.929	0.006331	0.8139	8.566

d2theta(rad/sec^2)	dtheta(rad/s)	alfa ht(deg)	delta theta(rad)	delta v(m/s)	alfa new ht(deg)
5.822	0.7097	10.2	0.008237	1.059	9.732
6.511	0.8947	11.48	0.01038	1.335	10.88

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
0.7583	0.01139	1.276	0.0001322	0.017	1.268
1.508	0.04541	2.551	0.000527	0.06775	2.521
2.248	0.1018	3.827	0.001181	0.1518	3.759
2.98	0.1802	5.102	0.002092	0.2689	4.982
3.703	0.2805	6.378	0.003255	0.4185	6.191
4.418	0.4023	7.653	0.00467	0.6003	7.386
5.124	0.5455	8.929	0.006331	0.8139	8.566
5.822	0.7097	10.2	0.008237	1.059	9.732
6.511	0.8947	11.48	0.01038	1.335	10.88

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 -91.38 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);
- $\left(1 \;-\; \frac{d\,\epsilon}{d\,\alpha}\right) \;=\;$ downwash factor.

Results can be summarized as follow:

1. at VF, the incremental tail load is 29.37 daN and the resulting total load is -7.664 daN when the incremental tail load is summed and -66.4 daN when the incremental tail load is subtracted;
2. at VC, the incremental tail load is 58.58 daN and the resulting total load is 40.44 daN when the incremental tail load is summed and -76.72 daN when the incremental tail load is subtracted;
3. at VF, the incremental tail load is 36.71 daN and the resulting total load is -1.638 daN when the incremental tail load is summed and -75.05 daN when the incremental tail load is subtracted.

11.3. Horizontal tail loads summary

In this section all the maximum limit load are summarized

1. Case (a): 35.07 daN;
2. Case (b): -91.38 daN;
3. Case (a) + (b): -49.86 daN;
4. Case (c): -82.19 daN;
5. Case (d): -92.2 daN.

11.4. Unsysmmetrical loads

According to CSVLA 427 , in the absence 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 -92.2 daN; the unsymmetrical loads will be:

- load on one side: -92.2*0.5 = -46.1 daN;
- load on the opposite side: -92.2*0.72 = -66.38 daN.

Chapter 12. Loads on the vertical tail

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;
- $\dfrac{d\,C_Y}{d\,\delta_r}$ = 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:

$$Y = \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * (C_Y)_{\delta_r\,=\,30} = \frac{1}{10} * 911 * 0.2044 * \frac{2.589}{0.2044} * 0.01936 = 9.131\ daN$$

The lateral force acting on a single fin of the vertical tail plain is 9.131/2 = 4.565 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:

$$Y = \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * C_Y = \frac{1}{10} * 911 * 0.2044 * \frac{2.589}{0.2044} * 0.0245 = 5.778\ daN$$

The lateral force acting on a single fin of the vertical tail plain is 5.778/2 = 2.889 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:

$$Y = \frac{1}{10} * q_A * S_{vertical} * \frac{S_{wing}}{S_{vertical}} * C_Y = \frac{1}{10} * 911 * 0.2044 * \frac{2.589}{0.2044} * 0.0233 = 5.495\ daN$$

The lateral force acting on a single fin of the vertical tail plain is 5.495/2 = 2.748 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:

- $U_{de} = \text{derived gust velocity (m/s)};$
- $L_{vt} = \text{vertical tail load (daN)}$
- $K_{gt} = \dfrac{0.88 * \mu_{gt}}{5.3 + \mu_{gt}} = \text{gust alleviation factor};$
- $\mu_{gt} = \dfrac{2 * M}{\rho * \bar{c}_t * g * a_{vt} * S_{vt}} * \dfrac{K^2}{l_t^2} = \text{lateral mass ratio};$
- $M = \text{aeroplane mass (kg)};$
- $rho = \text{air density } (kg/m^3);$
- $l_t = \text{aeroplane c.g.} - \text{to} - \text{lift} - \text{centre of vertical surface distance } (m);$
- $S_{vt} = \text{area of vertical tail } (m^2);$
- $a_{vt} = \text{lift curve slope of vertical tail } (1/rad);$
- $V = \text{aeroplane equivalent speed } (m/s);$
- $K = \text{radius of gyration in yaw } (m);$
- $g = \text{acceleration due to gravity } (m/s^2);$

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

1. Gust load at VC: 8.8395 daN
2. Gust load at VD: 5.5247 daN

The critical gust load is 8.8395 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) : 4.565 daN;
2. CSVLA 441(a)(2) : 2.889 daN;
3. CSVLA 441(a)(3) : 2.748 daN.

12.3.2. Critical gust loads

The critical gust load is 8.8395 daN at VC.

12.4. Combined loads

According to CSVLA 447 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 20.02 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 69.39 daN.

The critical combined load is 69.3881 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 18.04 m/s.

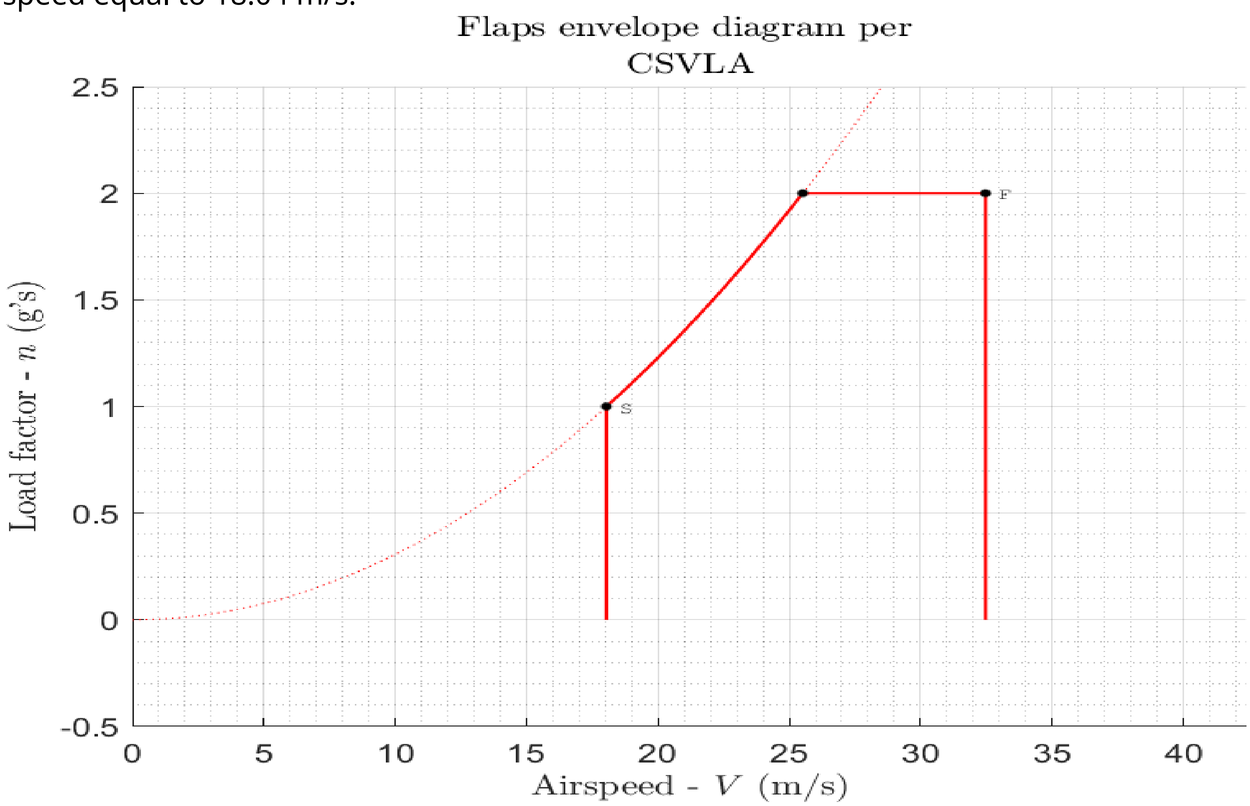

Figure 13.1. Flaps V-n diagram. Takeoff.

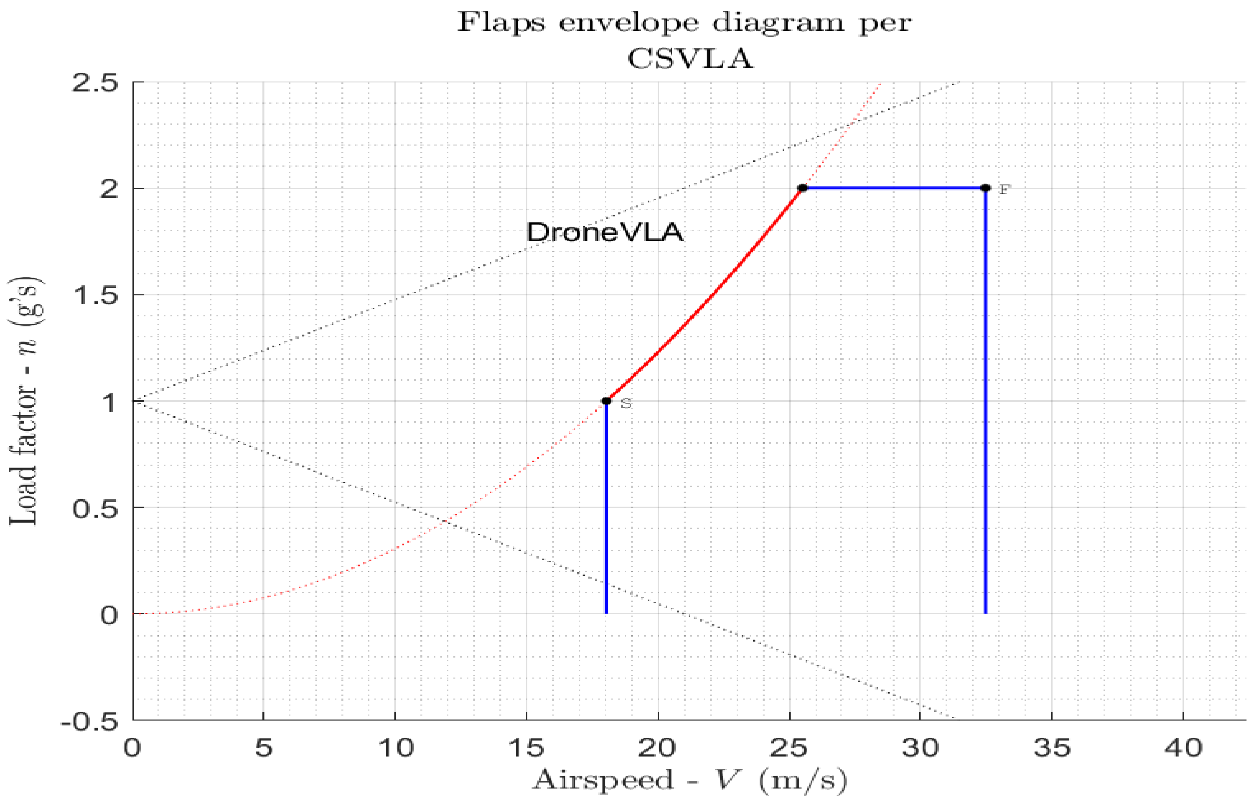

Figure 13.2. Flaps gust envelope diagram. Takeoff.

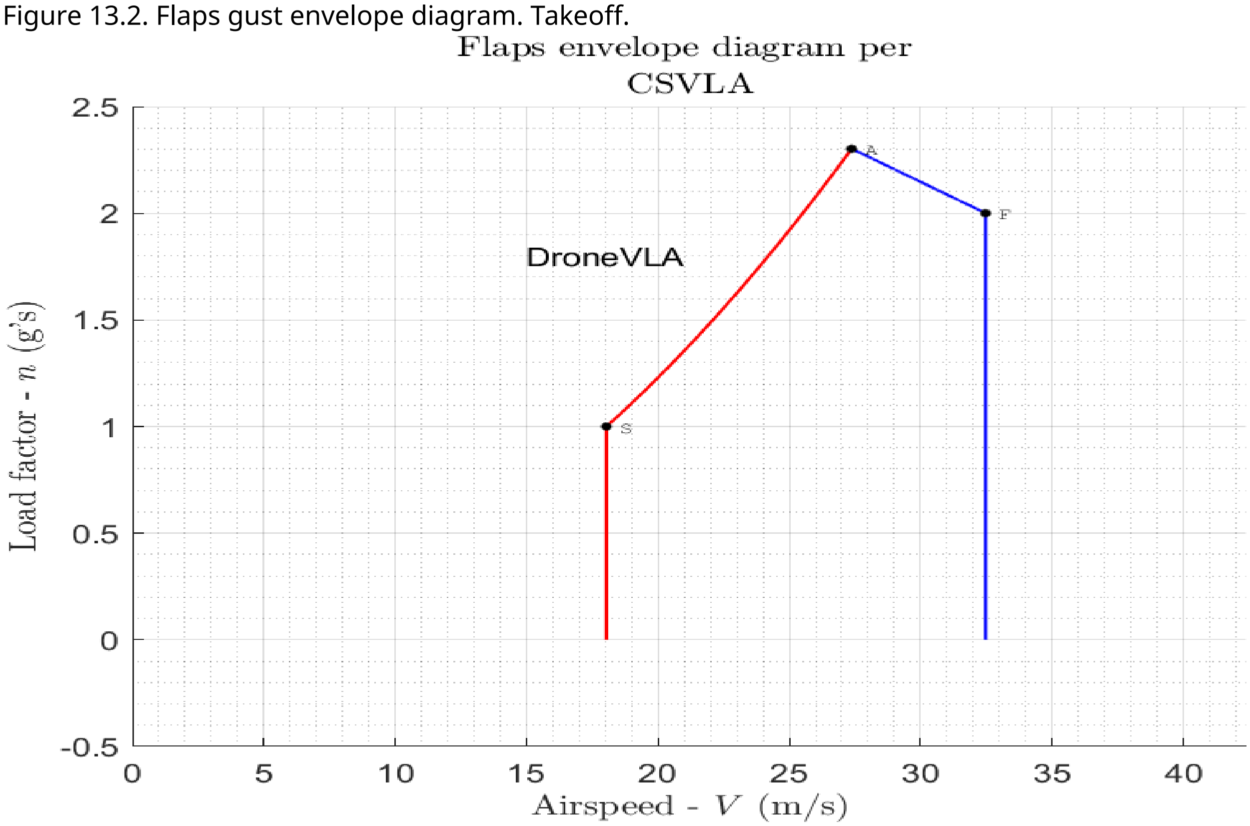

Figure 13.3. Flaps final envelope diagram. Takeoff.

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 18.04 m/s.

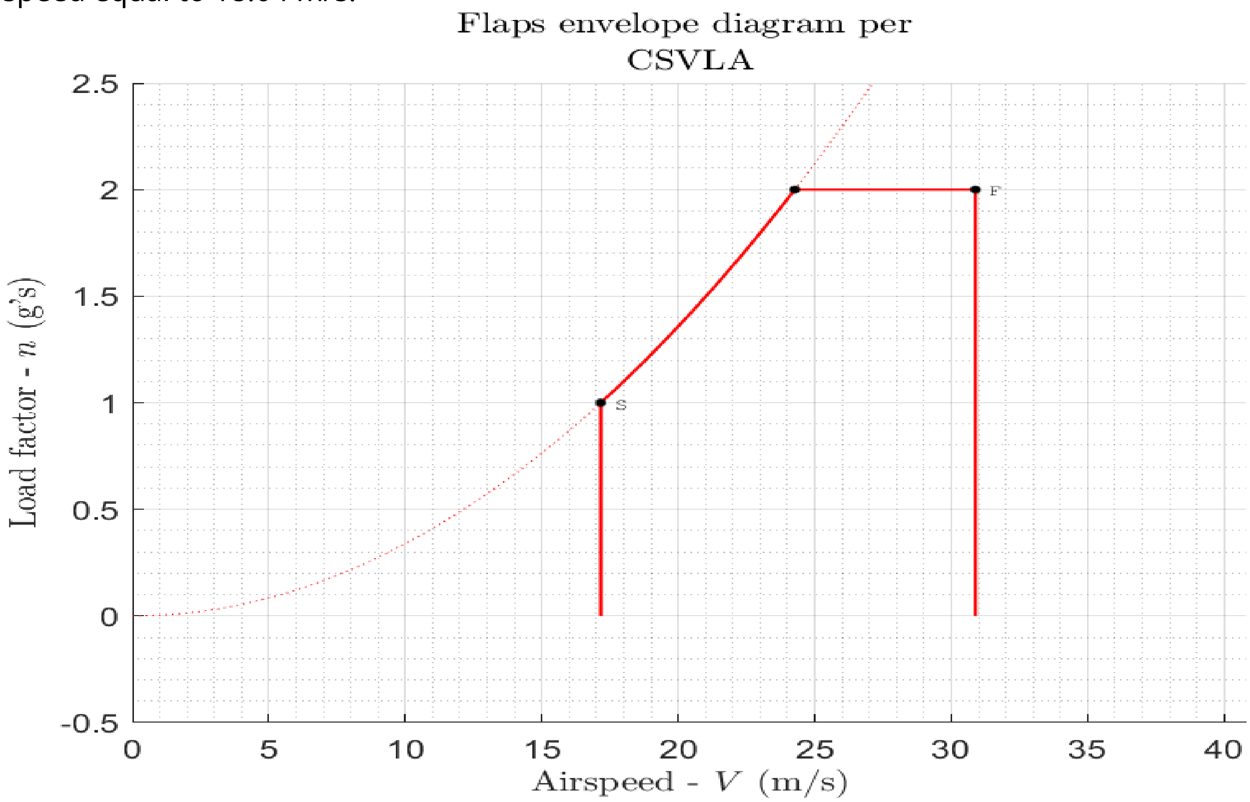

Figure 13.4. Flaps V-n diagram. Landing.

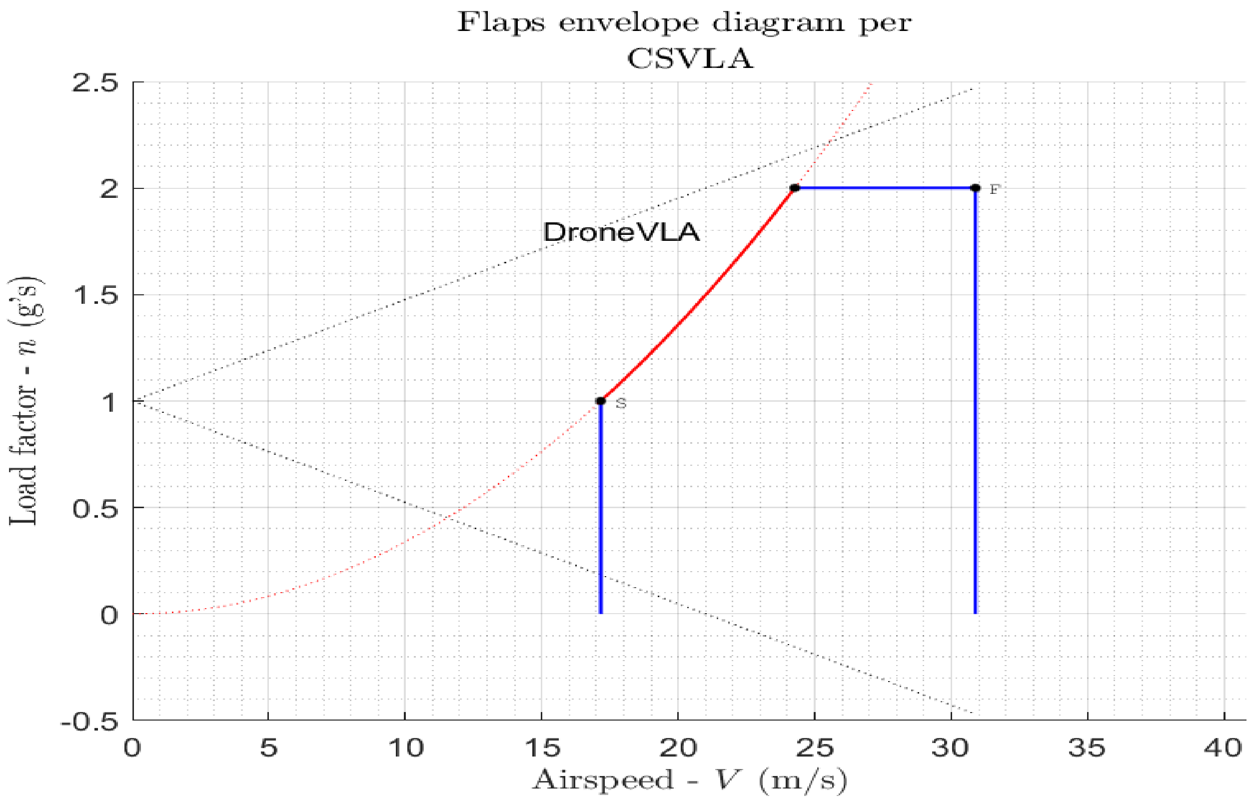

Figure 13.5. Flaps gust envelope diagram. Landing.

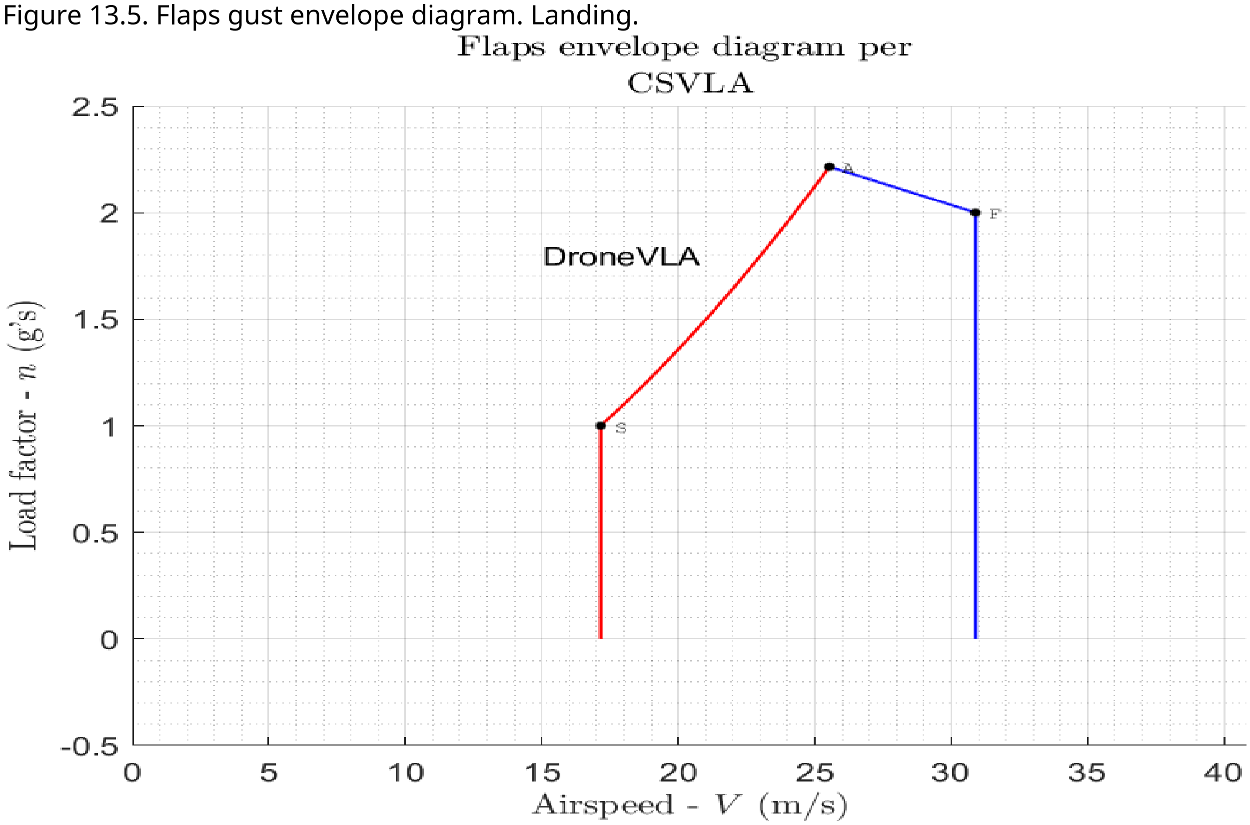

Figure 13.6. Flaps final envelope diagram. Landing.

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.

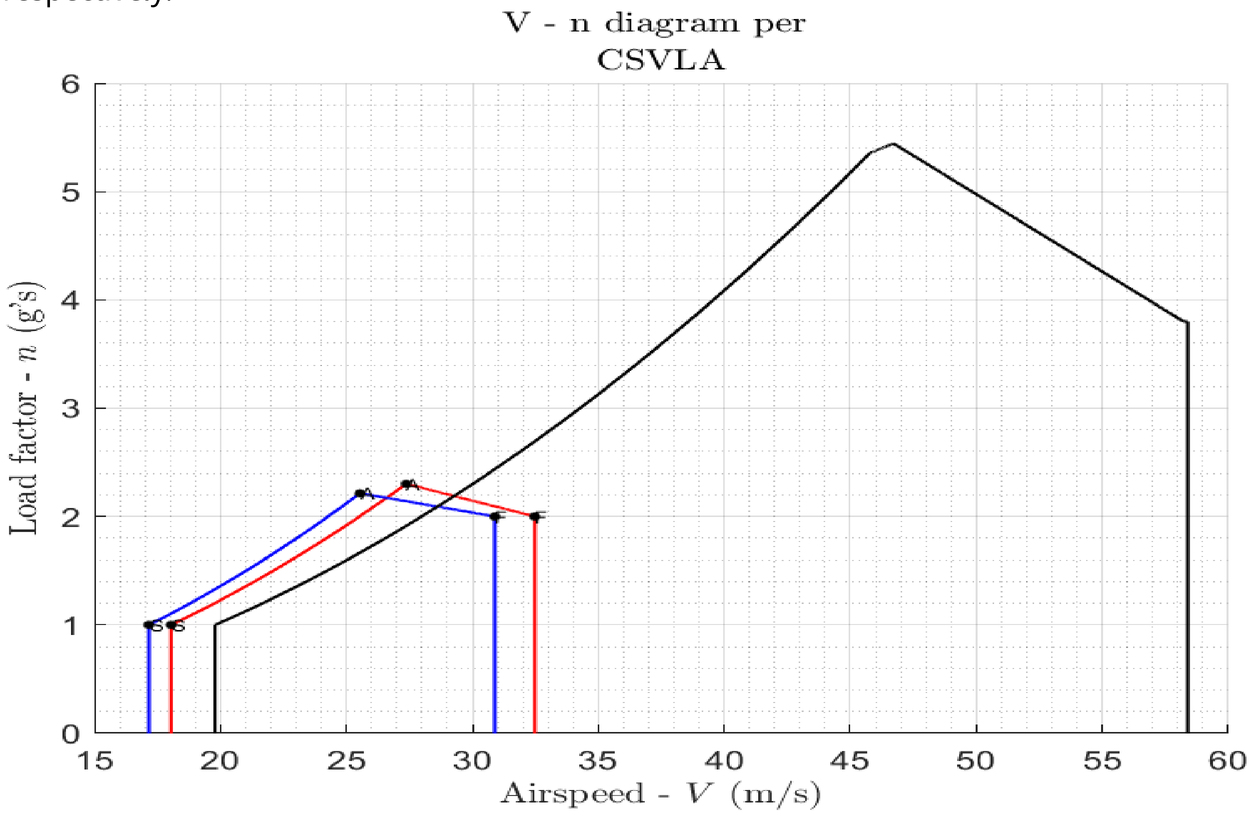

Figure 13.7. Summary of flaps flight load.

TABLE TO BE CHECKED!!!

Table 13.1. Flaps envelope points summary at takeoff.

Point	V(m/s)	n(g's)	mug	Kg	Ude(m/s)	WS(kg/m^2)	rho(kg/m^3)	altitude(m)
Point S	18.04	1	27.47	0.7377	7.62	38.62	1.079	1300
Point A	27.37	2.302	27.47	0.7377	7.62	38.62	1.079	1300
Point F	32.47	2	27.47	0.7377	7.62	38.62	1.079	1300

TABLE TO BE CHECKED!!!

Table 13.2. Flaps envelope points summary at landing.

Point	V(m/s)	n(g's)	mug	Kg	Ude(m/s)	WS(kg/m^2)	rho(kg/m^3)	altitude(m)
Point S	17.16	1	27.47	0.7377	7.62	38.62	1.079	1300
Point A	25.54	2.215	27.47	0.7377	7.62	38.62	1.079	1300

Point	V(m/s)	n(g's)	mug	Kg	Ude(m/s)	WS(kg/m^2)	rho(kg/m^3)	altitude(m)
Point F	30.89	2	27.47	0.7377	7.62	38.62	1.079	1300

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.

14.1. Ailerons

According to CSVLA 395 , the total aileron load is equal to -991.5 N . The hinge moment is calculated by the following equation

$$H_{aileron} = q * S_{aileron} * c_f * C_{h_{total}} = 911 * 0.1402 * 0.1354 * -0.3883 = -6.714\ N * m$$

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:

$$H_{aileron_{total}} = 2 * 1.25 * H_{aileron} = 2 * 1.25 * (-6.714) = -16.78\ N * m$$

14.2. Elevator

According to CSVLA 395 , the total elevator load is equal to -531.9 N. The hinge moment is

$$H_{elevator} = q * S_{elevator} * c_f * C_{h_{total}} = 911 * 0.1475 * 0.1085 * -0.2339 = -3.409\ N * m$$

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

$$H_{elevator_{total}} = 2 * 1.25 * H_{elevator} = 2 * 1.25 * (-3.409) = -8.522\ N * m$$

14.3. Rudder

According to CSVLA 395 , the total rudder load is equal to -66.13 N. The hinge moment is

$$H_{rudder} = q * (2 * S_{rudder}) * c_f * C_{h_{total}} = 911 * (2 * 0.01906) * 0.1085 * -0.00792 = -0.7491\ N * m$$

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:

$$H_{rudder_{total}} = 1.25 * H_{rudder} = 1.25 * (-0.7491) = -0.9364\ N * m$$

Chapter 15. Power plant

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 11.19 kW at 5800 RPM. The rotational speed of the propeller is 5800/2.429 = 2388 RPM. The maximum continuous power is 9.321 kW. The mean engine torque is 44.73 N * m. Using a factor of 2 for a four cylinder engine, the limit torque will be 89.45 N * m. This limit torque acts simultaneously with the 75 % of the inertia limit load. The mean engine torque at max continuous power is 39.3 N * m. Using a factor of 2 for a four cylinder engine, the limit torque will be 78.61 N * m which acts simultaneously with the 100 % of the inertia limit load.

$$MT_{continuous} = P_{continuous} * \frac{1000}{\frac{2\pi * RPM_{prop}}{60}} = 9.321 * \frac{1000}{\frac{2 * 3.14 * 2388}{60}} = 39.3\ N * m$$

$$LT_{continuous} = RR_{prop} * MT_{continuous} = 2.429 * 39.3 = 78.61 N * m$$

$$MT_{takeoff} = P_{takeoff} * \frac{1000}{\frac{2\pi * RPM_{prop}}{60}} = 11.19 * \frac{1000}{\frac{2 * 3.14 * 2388}{60}} = 44.73\ N * m$$

$$LT_{takeoff} = RR_{prop} * MT_{takeoff} = 2.429 * 44.73 = 89.45 N * m$$

- $MT_{takeoff}$ = mean torque at takeoff power (N * m); and

- $LT_{takeoff}$ = limit torque at takeoff power (N * m); and
- $MT_{continuous}$ = mean torque at max continuous power (N * m); and
- $LT_{continuous}$ = limit torque at max continuous power (N * 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 24.4 kg. The side load results is 1.33*24.4*9.807*(1/10) = 31.82 daN

15.3. Intertia load on engine mount

The inertia load is equal to the maximum limit load factor times the engine group weight: 5.357*24.4*9.807*(1/10) = 170.5 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 = polar moment of inertia of the propeller (kg * m^2); and
- ω_1 = propeller rotation speed (rad / sec); and
- ω_2 = 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 0.37 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*24.4 = 61 daN. The rotation of the propeller at maximum continuous power is 5500/2.429 = 2265. The gyroscopic couple is:

1. Yaw case: 2*0.37*2.5*(2*3.14/60)*2265 = 69.83 daN * m
2. Pitch case: 2*0.37*1.0*(2*3.14/60)*2265 = 27.93 daN * m