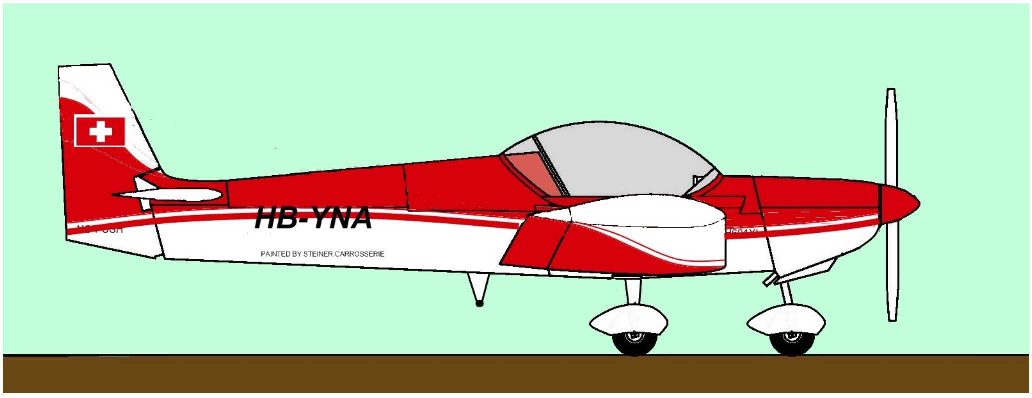
CZECH AIRCRAFT WORKS
ZENAIR CH601XL-B ZODIAC

LOAD ANALYSIS



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1 Overview

1.1 Introduction

The CH601XL-B is certified in many countries under different regulations: e.g. as an Ultralight Aircraft in Germany, as a Light Sport Aircraft in the USA or as an Experimental airplane in the UK.

Several load analysis' for the Zenair/CZAW CH601XL-B Zodiac were prepared by different aviation authorities. Chris Heintz, Zenair Aircraft and designer of the CH601XL-B, prepared a detailed load and stress analysis for the CH601XL, which should be considered as the master analysis for the Zenair CH601XL-B.

The present load analysis was specifically prepared for certification of the CZAW CH601XL-B in Switzerland, based on European regulations and the layout of the CZAW CH601XL-B. The CZAW CH601XL-B was built by Czech Aircaft Works under license-agreement with Zenair and has some minor changes compared to the original Zenair CH601XL-B:

- Angle of incidence increased by 2° (better visibility during cruise flight) [1]
- Rotax 912ULS engine (different weight than Continental O-200 or Jabiru engine)
- Composite main gear legs (instead of aluminium gear legs)

This analysis was revised by independent aviation engineers. Nevertheless it is of informal character only and the author doesn't take any responsibility if parts of the analysis are incorrect.

1.2 Regulations

TM 02.001-60

According to TM 02.001-60 issued by the Swiss FOCA (federal office of civil aviation), the following requirements must be fulfilled for engine-driven airplanes:

- Airworthiness based on CS Part 23 or CS-VLA
- Stall speed not more than v_{s0} = 61 kts
- Maximum takeoff weight less than MTOW = 1'750 kg
- Maximum occupancy (including pilot) = 4 persons

The following load analysis is based on the "Certification Specifications for Very Light Aircraft CS-VLA" issued by the European Aviation Safety Agency (EASA) [Ref].

References to CS-VLA are given throughout the entire load analysis.

CS-VLA 1

CS-VLA is valid for the following type of aircraft:

- Single engine (spark)
- Max. 2 seats
- MTOW not more than 750 kg
- Stalling speed in landing configuration of not more than 45 kts
- Day-VFR only

CS-VLA 301 (d)

Simplified structural design criteria are defined in CS-VLA, Appendix A, and are valid for aircraft with conventional configurations.

[1] Today the newer Zenair CH601XLBs (and its successors CH650) provided by Zenair use the same increased angle of incidence and composite gear (Zenair Europe).

AMC VLA 301 (d)

AMC are the "acceptable means of compliance" issued by the EASA.

In this context "aircraft with conventional configuration" means:

- Forward wing with an aft horizontal tail
- Wing untapered or continuously tapered with no more than 30° fore or aft sweep
- Trailing edge flaps may be fitted, but no winglets/tip devices, T-/V-tail, slotted flap devices.

The CH601XL-B Zodiac airplane satisfies all of these criteria. Therefore the "simplified design load criteria for conventional very light aircraft" CS-VLA, Appendix A, can be applied and are used throughout this load analysis.

2 Definitions

2.1 Aircraft Parameter

Parameters in *italic type* are based on the original drawings of Zenith and CZAW and on aviation technology publications.

Parameters in standard type are calculated values (formulary in a following sub-chapter).

Fuselage (B)

Fuselage Length:	L_B =	*6,1 m*	
Fuselage Width (Cockpit):	b_B =	*1,07 m*	

Wing (W)

Overall Wing Span:	b =	*8,23 m*	
Wing Span one Wing:	b_W =	3,58 m	
Chord at Wingtip:	c_1 =	*1,42 m*	
Chord at Wing Root:	c_{Root} =	*1,60 m*	
Chord at Fuselage Center Line:	c_0 =	1,626 m	
Overall Wing Area:	A =	12,5 m² (incl. Fuselage Part and Flaps/Ailerons)	
One Wing Area:	A_W =	5,4 m² (incl. Flaps/Ailerons)	
Mean Geometrical Chord:	c_{mean} =	1,523 m	
Mean Aerodynamical Chord:	c_{aero} =	1,525 m	
Wing Aspect Ratio:	Λ_W =	5,40	
Sweep Angle at 25% Line:	$\varphi_{25°}$ =	-0,72°	
Taper ratio:	λ_W =	0,87	
Angle of incidence:	γ_W =	3°	
Wing profile:	Riblet Wing	*GA 35-A-415*	
	Thickness t_{max} =	*15,0% at 35%* (≈ 22,8 cm)	
	Max. camber Y =	*3,3% at 43%* (≈ 5,0 cm)	
Wing Profile Lift Curve Slope:	$d(c_a)/d\alpha$ =	*6,3 rad⁻¹*	
Wing Lift Curve Slope:	$d(c_A)/d\alpha$ =	4,2 rad^{-1}	
Maximum Lift Coefficient:	$c_{Lmax,Clean}$ ≈	*1,82*	*[Ref. H. Riblett]*
	$c_{Lmax,FullFlaps}$ ≈	*2,3*	*[Ref. H. Riblett]*
Wing Profile Zero Lift Angle:	$\alpha_{L=0}$ =	-3,34°	*[Ref. R. Hiscocks]*
Wing Lift Aero Center:	at wing ¼ chord station (25%)		
Wing Pitching Moment Coeff.:	$c_{m(c/4)}$ =	*-0,0587*	*[Ref. Zenithair]*

Aircraft Aerodynamics

Aircraft Drag Polar:	C_D =	*0,033 + 0,07·C_L^2*	*[Ref. R. Hiscocks]*

Wing Flaps (F)

1 Flap Span:	b_F =	*2,03 m*	
Flap Chord:	c_F =	*0,335 m*	
1 Flap Area:	A_F =	0,68 m² (1 Flap only)	
Flap Deflection:	δ_F =	0° / -30°	
Wing Pitching Moment:	$\Delta c_{m(C/4)}$ =	*-0.18*	*[Ref. R. Hiscocks]*

Ailerons (Ail)

1 Aileron span:	b_{Ail} =	*1,50 m*
Aileron chord:	c_{Ail} =	*0,31 m*
1 Aileron area:	A_{Ail} =	0,46 m² (1 Aileron only)
Aileron deflection:	δ_{Ail} =	±11,5°
Aileron trim tab span:	$b_{AilTrim}$ =	*0,538 m*
Aileron trim tab chord:	$c_{AilTrim}$ =	*0,072 m*
Aileron trim area:	$A_{AilTrim}$ =	0,039 m²

Horizontal Tail (HT)
Elevator (Elev)

HT Span:	b_{HT} =	*2,3 m*
HT Chord:	c_{HT} =	*0,8 m*
HT Area:	A_{HT} =	1,84 m² (including Elevator)
HT Aspect ratio:	Λ_{HT} =	2,96
HT Arm (at 25% chord)	d_{HT} =	*3,25 m*
HT Profile:	NACA 0012	
	Thickness t_{max} =	*12% at 30%* (= 100 mm)
HT Lift curve slope:	$d(c_a)_{HT}$ =	*3,27* rad^{-1}
Elevator span:	b_{Elev} =	*2,2 m*
Elevator chord:	c_{Elev} =	*0,35 m*
Elevator area:	A_{Elev} =	0,77 m²
Elevator deflection:	δ_{Elev} =	+30° / -27°
Elevator trim span:	$b_{ElevTrim}$ =	*0,897 m*
Elevator trim chord:	$c_{ElevTrim}$ =	*0,062 m*
Elevator trim area:	$A_{ElevTrim}$ =	0,056 m²

Rudder (R)

Rudder span:	b_R =	*1,42 m*
Rudder chord tip:	c_{R1} =	*0,37 m*
Rudder chord bottom:	c_{R0} =	*0,98 m*
Rudder aspect ratio:	Λ_R =	*2,25*
Rudder arm (Wing -> R)	d_R =	*3,90 m*
Rudder area surface:	$A_{R,Surface}$ =	0,52 m² (moving rudder surface only)
Rudder profile:	NACA 0012	
	Thickness t_{max} =	45 (tip) - 105 mm (root)
Rudder deflection:	δ_R =	±20°

Flight Control System

Aileron – Control stick:	Pilot: 330 mm / Control: 80 or 100 mm
Aileron – Wing bell crank:	To Stick: 80 mm / To Aileron: 85 mm
Aileron – Rudder horn:	100 mm
Elevator – Control stick:	Pilot: 330 mm / Control: 120 mm
Elevator – Rudder horn:	100 mm
Rudder – Pedals:	Pilot: 190 mm / Control: 65 mm
Rudder – Rudder horn:	125 mm
Control cables tension:	110 N

Engine Mount

Attachment bolt:	AN6

Seat Belts

Attachment bolt:	AN5-5A
Thickness attachment plate:	0,040” = 1 mm

2.2 Formulary

Most of the following formulas are self-explanatory and are all based on the geometry of the Zodiac CH601XL-B.

Wing span 1 wing:	$b_W = \frac{b-b_B}{2}$	
Chord at fuselage center:	$c_0 = c_{Root} + \frac{c_{Root}-c_1}{2} \cdot \frac{b}{b_W}$	
Wing area:	$A_W = \frac{c_0+c_1}{2} \cdot b$	
Mean geometrical chord:	$c_{mean} = \frac{c_0+c_1}{2}$	
Mean Aerodynamical Chord:	$c_{aero} = \frac{2}{3} \frac{{c_0}^2+c_0 c_1+{c_0}^2}{c_{0+}c_1}$	
Wing Aspect Ratio:	$\Lambda_W = \frac{b^2}{A}$	
Sweep Angle at 25% Line:	$\varphi_{25°} = \frac{1}{4} \cdot \arctan\left(\frac{c_0-c_1}{b/_2}\right)$	*straight wing leading edge*
Wing Taper Ratio:	$\lambda_W = \frac{c_1}{c_0}$	
Wing Profile Lift Curve Slope:	$\frac{d(c_a)}{d\alpha} = 2\pi$	
Wing Lift Curve Slope:	$\frac{d(c_a)}{d\alpha} = 0{,}1 \cdot \frac{\Lambda_W}{\Lambda_W+2}$	
Wing Profile Zero Lift Angle:	$\alpha_{L=o} = -100 \cdot \frac{Y}{c} - 3{,}2 \cdot \frac{X}{c} + 1{,}4$	*max. chamber Y at pos. X*

2.3 3-View Drawing

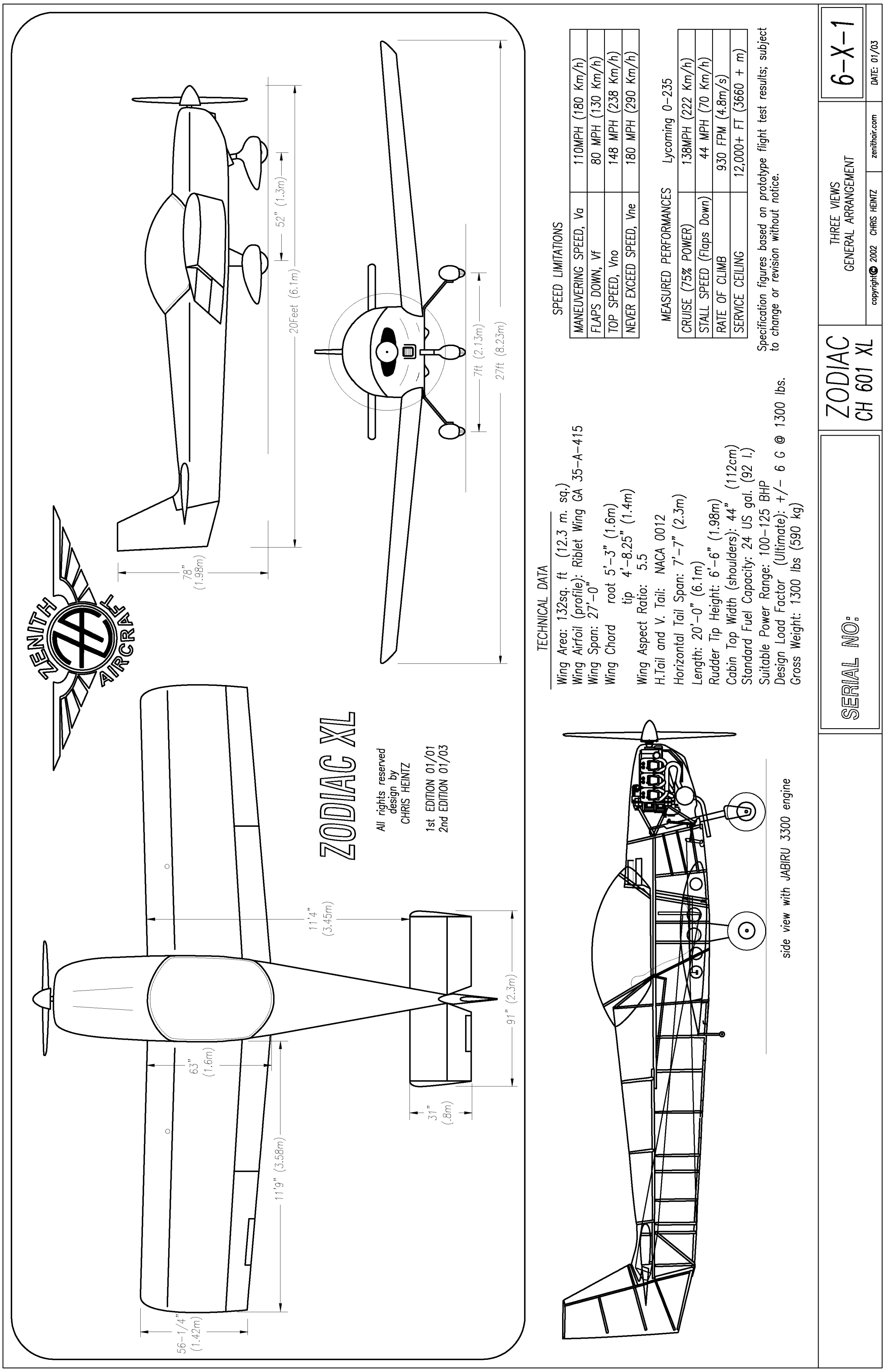


Fig. 1: Zenithair CH601XL-B 3-view drawing

2.4 Weights

CS-VLA 25 (a) (b)

Compliance with each applicable requirement (structural loading and flight requirements) of CS-VLA at both maximum and minimum weight has to be shown.

(a) Maximum weight:

Maximum weight has to be the highest of:

- Each seat occupied (2 x 86kg), at least enough fuel for 1 h of flight with max. continuous power (25 L ≅ 20 kg), whereas empty weight is W_{ZFW} = 340 kg (approx.):

$$W_{max,1} = 340 + 2 \cdot 86 + 20 = 532kg$$

- One pilot (86 kg), full fuel (180 L = 135 kg):

$$W_{max,2} = 340 + 86 + 135 = 561kg$$

- Design weight: $\boldsymbol{W_{max} = 600kg}$

(b) Minimum weight:

Minimum weight is ZFW + one light pilot (55 kg) + ½ h of flight with max. continuous power:

- Design weight: $\boldsymbol{W_{min}} = 340 + 55 + 10 = \boldsymbol{405kg}$

CS-VLA 321 (b)(2) [2]

"Compliance with the flight load requirements must be shown […] at each practicable combination of weight and disposable load within the operating limitations specified in the Flight Manual".

Although not part of Appendix A, this requirement can be taken as a guideline for how the fuel distribution in the wing has to be taken into account for the load calculations. Most critical case is at maximum weight and minimum fuel.

CS-VLA A7 (a)

Based on the simplified criteria (Appendix A) only the conditions at maximum design weight must be investigated.

2.5 Limit Load Factors

CS-VLA A3 / A7 (b)(c)

The limit flight load factors (normal category) are:

Positive maneuvering limit load factor:	n_1 = 3,8
Negative maneuvering limit load factor:	n_2 = -1,9
Positive gust limit load factor at v_C:	n_3 = 3,8
Negative gust limit load factor at v_C:	n_4 = -1,9
Positive limit load factor with flaps fully extended at v_F:	n_{flap} = 1,9

2.6 Center of Gravity

CS-VLA A7 (d)

Mean C.G.:	x_{CG} =	25% = 380 mm (based on MAC = 1,52 m)
Forward C.G.:	e_{Fwd} =	-5% = -76 mm / 304 mm
Rearward C.G.:	e_{Aft} =	+5% = 76 mm / 456 mm

[2] CS-VLA 321 is not part of and not required under CS-VLA Appendix A.

2.7 Airspeeds

CS-VLA A3 / A7 (e)(2)

Design speed	CS-VLA (min)	CH601XL-B
Minimum design flap speed:	$V_{F,min}$ = 70 kts	v_F = 70 kts
Minimum design maneuvering speed:	$V_{A,min}$ = 95 kts	v_A = 95 kts
Minimum design cruising speed:	$V_{C,min}$ = 107 kts	v_C = 107 kts
Minimum design dive speed:	$V_{D,min}$ = 151 kts	v_D = 156 kts
Stall speed clean:		v_S = 40 kts
Stall speed full flaps:		v_{S0} = 35 kts
Never exceed speed:		v_{NE} = 140 kts

2.8 V-n-Diagram

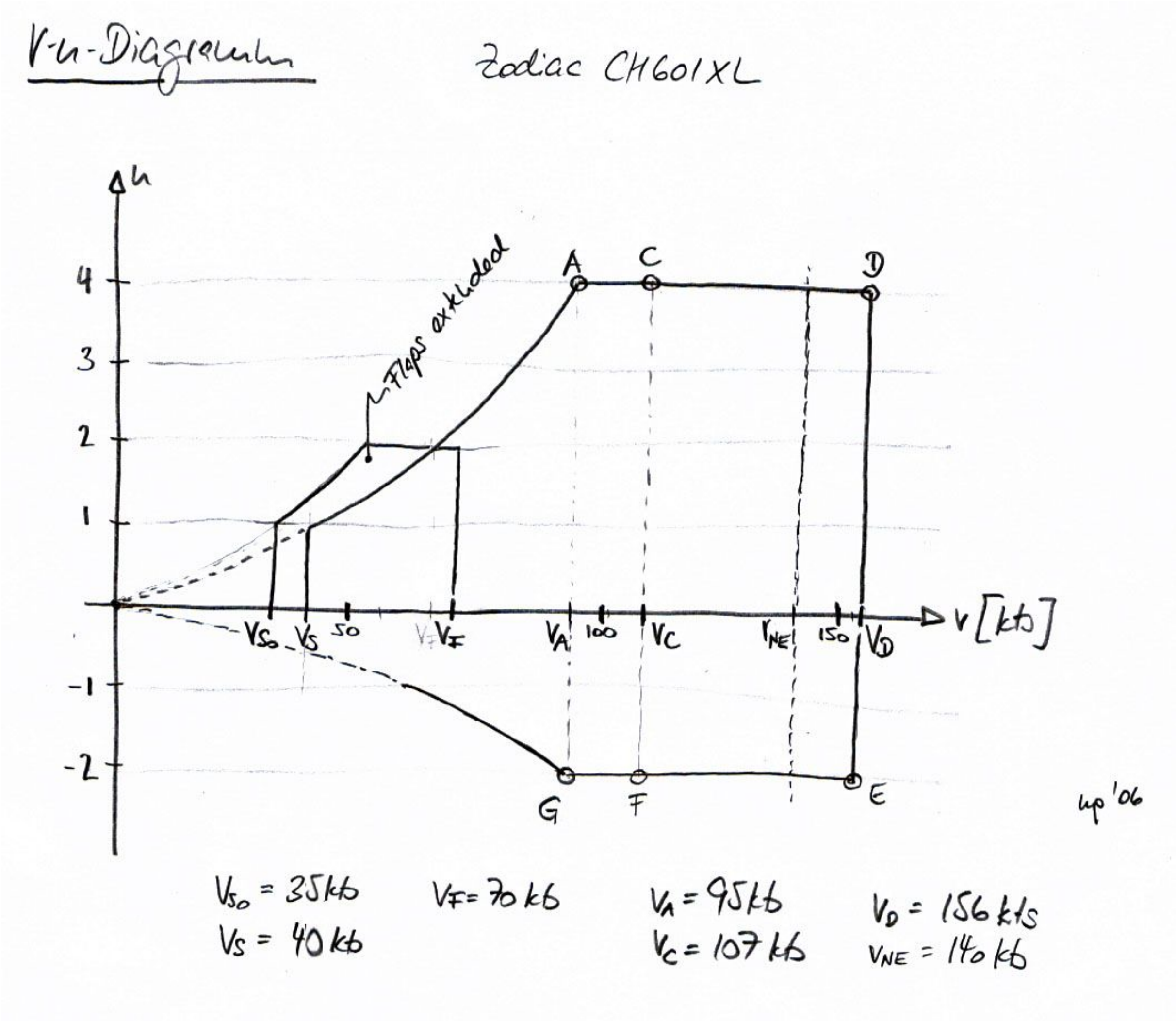


Fig. 2: V-n-diagram

3 Wing

3.1 Wing Geometry / Weights

The wing of the CH601XL-B is made up of a main spar, a rear spar and 10 wing ribs. Two 45 L (12 USG) fuel tanks are placed in front of the main spar, between nose ribs NR4/NR5 and NR5/NR7 (extended range version, the standard version has 1x 12 USG tank per wing, see Fig. 3).

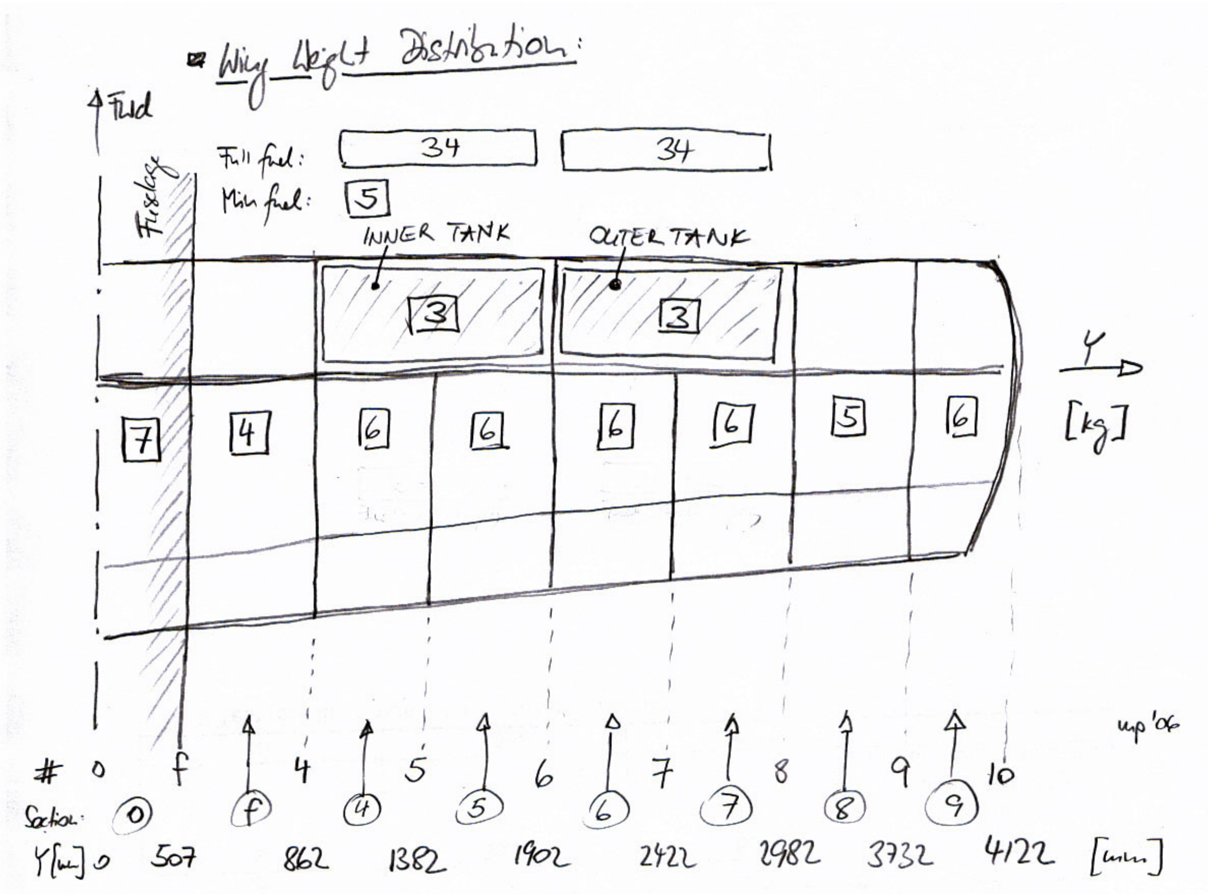


Fig. 3: Spanwise wing weight distribution

Position of Wing Ribs

Y is the distance between the rib and the airplane center line [in mm].

Rib #	#0	#f	#4	#5	#6	#7	#8	#9	#10
Position Y [mm]	0	507	862	1382	1902	2422	2982	3732	4122

Wing Dry Weight Distribution

The dry weight of the wing is split up in several sections (e.g. section 8 = between ribs #8 and #9). Each section weight includes the corresponding wing structure, primer, paint and all systems installed (e.g. strobe light transformer).

Section	0	f	4	5	6	7	8	9	Total
Wing	(7 kg)	4 kg	6 kg	6 kg	6 kg	6 kg	5 kg	6 kg	
+ Fuel Tanks	-	-	1,5 kg	1,5 kg	1,5 kg	1,5 kg	-	-	45 kg
Fuel Min Fuel	-	-	5 kg	0 kg	0 kg	0 kg	-	-	50 kg
Fuel Max Fuel	-	-	17 kg	17 kg	17 kg	17 kg	-	-	113 kg

3.2 Spanwise Lift Distribution

In general the spanwise lift distribution can be divided into:

- Base lift distribution
- Lift distribution due to wing twist
- Additional lift distribution due to flaps and/or ailerons

Base Lift Distribution

According to *Schrenk [Ref]* the base lift load at any selected spanwise station is the arithmetical mean between the load proportional to the chord of the real wing and the load proportional to the chord of an elliptical wing with equal wing area.

It can be assumed that the lift distribution is continuative over the entire wingspan, including the fuselage *[Schlichting/Truckenbrodt + Peery, Ref]*. Therefore the lift of the fuselage can be substituted by the lift of the (fictive) wing centerpiece.

The formulas' numbering [in brackets] corresponds to the numbers in the results-table (chapter 3.4).

Relative spanwise position: $\bar{y} = \frac{y}{b/2}$ [1]

Chord tapered wing: $c_{wing}(y) = c_0 - \frac{c_0 - c_1}{b/2} \cdot y$ [2]

$$= 1{,}626 - \frac{1{,}626 - 1{,}42}{8{,}23/2} = 1{,}626 - 0{,}0503 \cdot y$$

Chord elliptical wing: $c_{ell}(y) = c_{0,ell} \cdot \sqrt{1 - \left(\frac{y}{b/2}\right)^2}$ [3]

with $c_{0,ell} = \frac{c_0 + c_1}{\pi} = \frac{1{,}626 + 1{,}42}{\pi} = 0{,}970$

$$c_{ell}(y) = 0{,}970 \cdot \sqrt{1 - \left(\frac{y}{4{,}115}\right)^2}$$

Mean chord: $c_{mean}(y) = \frac{c_{wing}(y) + c_{ell}(y)}{2}$ [4]

The drawing in Fig. 4 shows the chord of the CH601XL-B wing ***c(wing)***, the chord of the surrogate elliptical wing with identical wing area ***c(0,ell)*** and the mean value of the two chord lines ***c(mean)***.

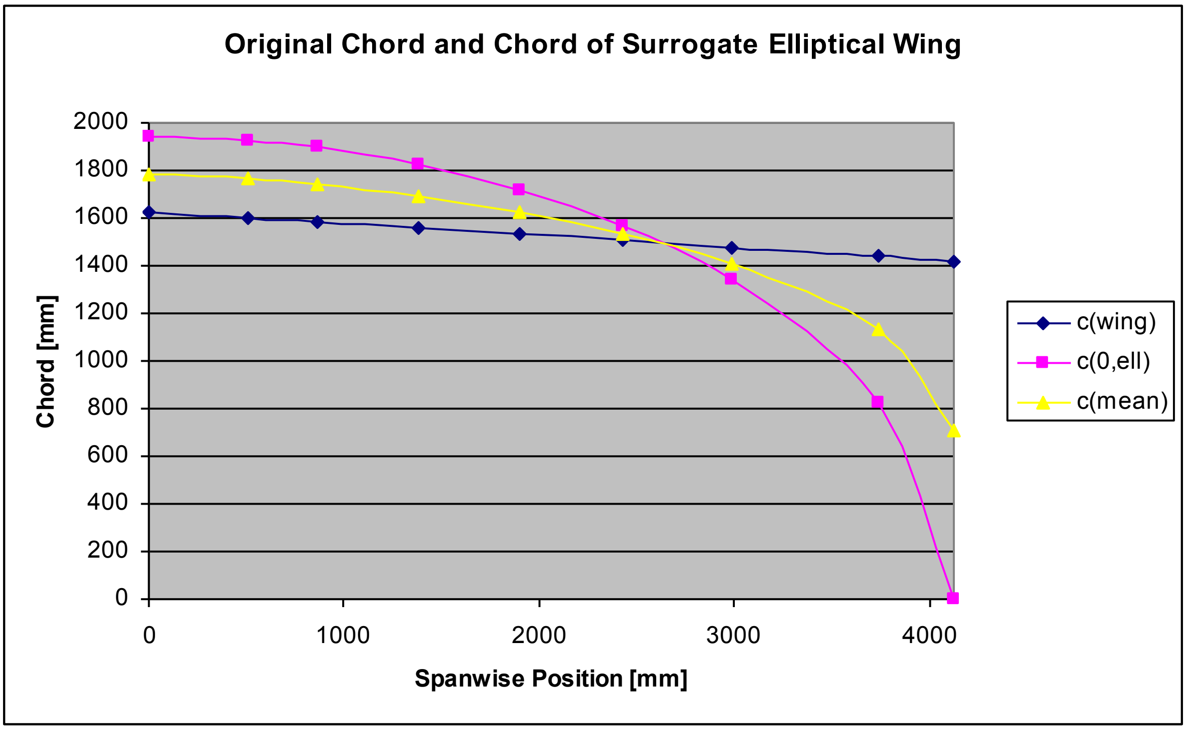


Fig. 4: Spanwise distribution for original, elliptical and mean wing chord

For the sake of convenience, the wing is split up in several sections to calculate the spanwise lift distribution (similar to the wing dry weight distribution).

Mean chord of one section: $c_i = \frac{c_{mean}(y_i)+c_{mean}(y_{i+1})}{2}$ [6]

Width of one section: $\Delta y_i = y_{i+1} - y_i$ [7]

The wing lift for one specific section can be calculated by multiplying the total lift required with the ratio between the section area and the total wing area:

Wing lift of one section: $\Delta L_i = \frac{c_i \cdot \Delta y_i}{A_W} \cdot L_{total,required}$ [8]

Lift due to Wing Twist

The ailerons are twisted 2,5° up along the trailing end, which corresponds to a wing twist of closely 1,25° over the aileron span. It is therefore conservative to consider a wing without twist for calculation of the maximum wing bending moment.

Additional Lift with Flaps Extended

CS-VLA A9 (b)(2)

With flaps extended the lift coefficient at the corresponding wing section is increased by approx. 1,0.

However it is obvious that the shear and bending moment on each wing is considerably lower because of the reduced load factor of n = +1,9 / 0,0 with flaps extended. Therefore the case with flaps extended will not be further investigated regarding shear and bending moment.

3.3 Shear and Bending Moment

Shear and bending moment of the wing are again calculated for the same discrete sections of the wing, starting from wing tip to wing root. It is obvious that the higher the lift forces the higher the stress on the wing (Fig. 5). On the contrary the inertia force of the wing (masses) act as a relieving factor and unload the stress on the wing.

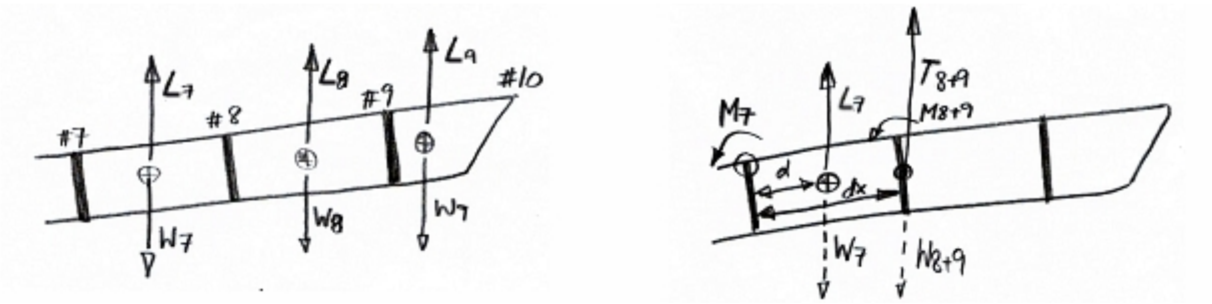


Fig. 5: Discrete wing weight, lift and bending moment

Wing Lift

Shear due to lift at rib: $T_i = T_{i+1} + \Delta L_i$ [9]

Bending moment due to lift at rib: $M_i = M_{i+1} + \Delta y_i \cdot T_{i+1} + \frac{\Delta y_i}{2} \cdot \Delta L_i$ [10]

Wing Inertia Relief

Wing inertial relief force of one section: $\Delta W_i = \Delta W_{i,Wing} + \Delta W_{i,Fuel}$ [13]

Inertia relief at rib: $T^-{}_i = T^-{}_{i+1} + \Delta W_i$ [14]

Inertia relief bending moment at rib: $M^-{}_i = M^-{}_{i+1} + \Delta y_i \cdot T^-{}_{i+1} + \frac{\Delta y_i}{2} \cdot \Delta W_i$ [15]

Total Shear and Bending Moment

Shear at rib: $T_{i,limit} = T_i + T^-{}_i$ [16]

Bending moment at rib: $M_{i,limit} = M_i + M^-{}_i$ [17]

Ultimate Loads

Ultimate shear: $T_{i,ult} = 1{,}5 \cdot T_{i,limit}$ [18]

Ultimate bending moment: $M_{i,ult} = 1{,}5 \cdot M_{i,limit}$ [19]

3.4 Symmetrical Flight Conditions

CS-VLA A9 (b)(1)(i) (ii)

The calculation of wing lift, wing inertia relief, shear and bending moment for the symmetrical flight condition is performed by using an excel calculation sheet. The results for MTOW = 600 kg and 20 L of fuel (most critical/conservative loading with ½ hour of fuel and 5 L unusable fuel) are shown below. The down force of the horizontal tail is assumed to be 5% of the total wing lift (*CS-VLA Appendix A*).

Input Parameters Zodiac CH601XLB									
c0	[mm]	1626	*Yellow fields are input parameters!*						
c1	[mm]	1420							
c(0,ell)	[mm]	1939							
b/2	[mm]	4122							
A(w,total)	[m²]	12,6							
Load Factor	[-]	3,8				*5% of wing lift*			
L(req,total)	[N]	23476	*TOW*	*600 kg*	*22358*	*HT*	*1118*	***TOTAL***	***23476***
f(corr.)	[-]	1,00715							

	Rib/Section	**#**	**0**	**f**	**4**	**5**	**6**	**7**	**8**	**9**	**10**
	Wing Geometry										
1	**Y/(b/2)**	[-]	0,00	0,12	0,21	0,34	0,46	0,59	0,72	0,91	1,00
2	**Y**	[mm]	0	507	862	1382	1902	2422	2982	3732	4122
3	**c(wing)**	[mm]	1626	1601	1583	1557	1531	1505	1477	1439	1420
4	**c(ell)**	[mm]	1939	1924	1896	1827	1720	1569	1339	823	0
5	**c(mean)**	[mm]	1783	1763	1740	1692	1626	1537	1408	1131	710
	Wing Lift (Flaps up)										
6	**c(i)**	[mm]	1773	1751	1716	1659	1581	1472	1270	921	
7	**dy(i)**	[mm]	507	355	520	520	520	560	750	390	
8	**dL(i)**	[N]	1692	1171	1680	1624	1548	1553	1793	676	
9	**T(i)**	[N]	11738	10046	8875	7195	5571	4022	2469	676	
10	**M(i)**	[Nm]	22001	16479	13120	8942	5623	3129	1311	132	
	Wing Weight										
11	**Wing**	[kg]	7	4	7,5	7,5	7,5	7,5	5	6	
12	**Fuel**	[L]			10	0	0	0			
13	**dW(i)**	[N]	-261	-149	-559	-279	-279	-279	-186	-224	
14	**T-(i)**	[N]	-2217	-1956	-1807	-1248	-969	-689	-410	-224	
15	**M-(i)**	[Nm]	-4117	-3059	-2391	-1597	-1020	-589	-281	-44	
	Total Shear / Bending Moment										
16	**T(limit)**	[N]	9521	8089	7068	5947	4602	3333	2059	453	
17	**M(limit)**	[Nm]	17884	13420	10729	7346	4603	2540	1030	88	
18	**T(ultimate)**	[N]	14281	12134	10602	8920	6903	4999	3089	679	
19	**M(ultimate)**	[Nm]	26826	20129	16094	11018	6905	3810	1545	132	

The resulting maximum shear and bending moment at the wing root are highlighted in amber (limit) and red (ultimate) color.

For comparison the results for different MTOW and fuel quantities are summarized in the following table. The considered cases are:

1. MTOW = 600 kg, minimum fuel (1/2 h + unusable fuel):
   Load inside the fuselage = 260 kg (useful load) – 15 kg (fuel) = 245 kg.
2. MTOW = 600 kg, full tanks (180 L):
   Remaining dry load = 260 kg (useful load) – 135 kg (full fuel) = 125 kg.
3. Two standard persons aboard (2 x 86 kg) + full inner tanks (90 L):
   TOW = 580 kg.
4. One person aboard (86 kg) + minimum fuel (20 L):
   TOW = 440 kg.

Case		1	2	3	4
TOW	[kg]	600	600	580	440
Fuel Quantity	[L]	20	180	90	20
T(limit)	[N]	8089	5853	6776	5410
M(limit)	[Nm]	13420	10070	11942	9025
T(ultimate)	[N]	12134	8780	10164	8116
M(ultimate)	[Nm]	20129	15105	17913	13538

Case 1 is critical (MTOW = 600 kg, minimum fuel = 20 L).

3.5 Lift + Drag Components

For a structural analysis of the airplane, it is important to determine all forces acting on the wing. The wing lift is balancing the weight/inertia forces of the airplane, whereas (in horizontal, steady flight) the drag is overcome by the thrust of the engine.

In order to be able to properly analyze the structure of the wing, the lift L and drag D are normally converted into their resulting force R. In addition the tangential force acting on the wing T is calculated, which is the component of R along the wing axis (Fig. 6).

It is not obvious from the very first in which direction the tangential force T is pointing to. A discussion of results at different airspeeds and load factors is therefore of high importance.

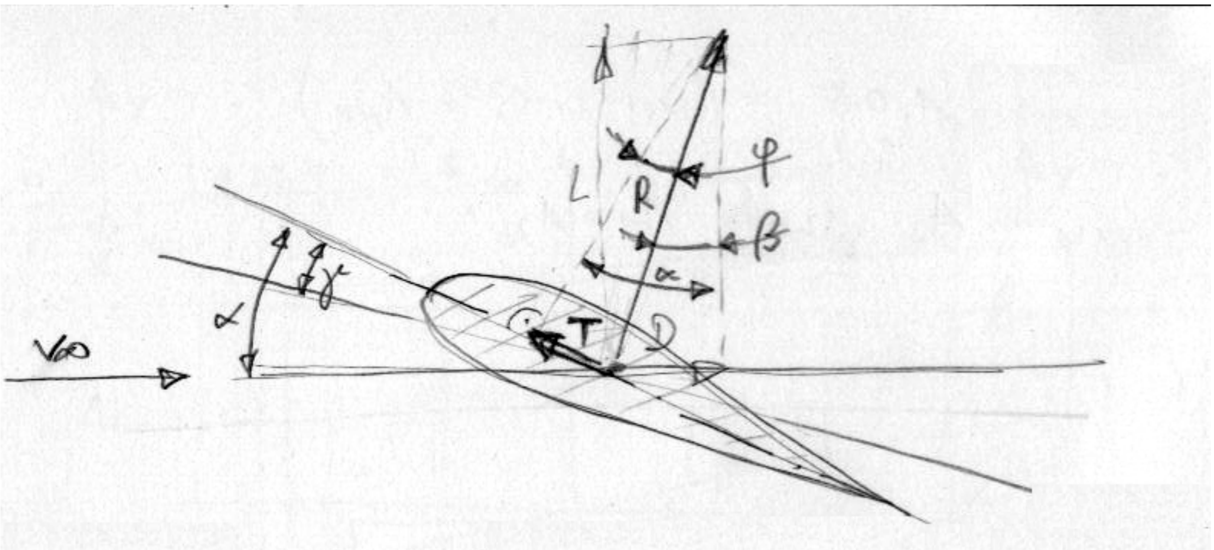


Fig. 6: Wing lift and drag components

Lift, drag and the corresponding resulting force as well as the tangential force acting on the wing are calculated by using the formulas below. The wing forces are all a function of airspeed and load factor. Therefore different cases from the v-n-diagram are considered, i.e. at the points A, D, G and E.

Wing Lift

Wing lift curve slope: $\frac{d(c_L)}{d\alpha} = 0{,}1 \cdot \frac{\Lambda}{\Lambda+2}$ [1]

Total lift $L_{total} = 1{,}05 \cdot n \cdot W$ [2]

The total lift includes an assumed 5% additional lift for counteracting the horizontal tail down force.

Lift 1 wing: $L_{1Wing} = L_{Total} \cdot \frac{A_{1Wing}}{A}$ [3]

Lift coefficient: $c_L = \frac{L_{1Wing}}{\frac{\rho}{2} \cdot v^2 \cdot A_{1Wing}}$ [4]

Angle of attack: $\alpha = \frac{c_L}{\left(\frac{d(c_L)}{d\alpha}\right)}$ [5]

The wing weight (i.e. the inertia forces of the wing) can be subtracted from the wing lift:

Inertia relief 1 wing: $I_{1Wing} = -n \cdot W_{1Wing}$ [6]

Net shear load 1 wing: $T_{1Wing} = L_{1Wing} + I_{1Wing}$ [7]

Wing Drag

The inertia forces in the direction of the wing axis are small compared to the wing drag. Therefore they are neglected in this calculation.

Drag coefficient: $c_D = 0{,}01 + \frac{{c_L}^2}{\pi \cdot \Lambda}$ [8]

Drag 1 wing: $D = c_D \cdot \frac{\rho}{2} v^2 \cdot A_{1Wing}$ [9]

Resulting Force

Resulting total force: $R = \sqrt{L^2 + D^2}$ [10]

Tangential Force

Angle between L and R: $\beta = \arctan\left(\frac{D}{L}\right)$ [12]

Angle between R and perpendicular of wing: $\varphi = \alpha - \beta$ [13]

Forward tangential force on 1 wing: $T_{1Wing} = R \cdot \sin(\varphi)$ [14]

Ultimate tangential force on 1 wing: $T_{1Wing,ult} = 1{,}5 \cdot T_{1Wing}$ [15]

The results for different airspeeds and load factors according to the V-n-diagram are summarized in an excel-table on the next page. The formula's numbering [in brackets] corresponds to the numbers in the table.

LIFT + DRAG FORCES

Aspect Ratio	Λ		5,4
Total Wing Area	A	[m²]	12,5
Wing Area 1 Wing	Aw	[m²]	5,4
MTOW	W	[kg]	600
Weight 1 Wing	Ww	[kg]	44

Airspeeds / Load Factors								
	Speed				vA			vD	
		v	[kts]	95	95	95	156	156	156
			[m/s]	48,9	48,9	48,9	80,3	80,3	80,3
	Load Factor	n	[-]	1,0	3,8	-1,9	1,0	3,8	-1,9
Wing Lift									
1	Lift curve slope	d(cL)/dα		0,073	0,073	0,073	0,073	0,073	0,073
2	Total Lift (incl. 5% HT-Load)	L	[N]	6178	23476	-11738	6178	23476	-11738
3	Lift 1 Wing	Lw	[N]	2669	10141	-5071	2669	10141	-5071
4	Lift coefficient	cL		0,338	1,284	-0,642	0,125	0,476	-0,238
5	Angle of attack	α	[°]	4,6	17,6	-8,8	1,7	6,5	-3,3
6	Inertia Relief 1 Wing	Iw	[N]	-431	-1640	820	-431	-1640	820
7	Net Shear Load 1 Wing	Tw	[N]	2237	8502	-4251	2237	8502	-4251
Wing Drag									
8	Drag coefficient	cD		0,017	0,107	0,034	0,011	0,023	0,013
9	Drag 1 Wing	Dw	[N]	132	846	271	233	498	284
Resulting Force									
10	Resulting Force	R	[N]	2241	8544	4260	2249	8516	4260
11	% of L			100,2%	100,5%	-100,2%	100,5%	100,2%	-100,2%
Tangential Force									
12	Angle between L and R	β	[°]	3,4	5,7	-3,6	5,9	3,3	-3,8
13	Angle between R and n_Wing	φ	[°]	1,2	11,9	-5,2	-4,2	3,2	0,6
14	Fwd Tangential Force on 1 Wing	T	[N]	49	1763	-382	-166	472	42
15	Ultimate Tangential Force 1 Wing	T,ult	[N]	73	2644	-574	-248	707	63

The maximum ultimate forward tangential force F = 2’644 N occurs at v_A and n = 3.8.

The maximum ultimate rearward tangential force F = -574 N occurs at v_A and n = -1.9.

3.6 Wing Torsion

The wing torsion, which acts at each wing section and which is computed relative to the wing shear center (defined at 23% chord), consists of the following components (Fig. 7):

- Aerodynamic wing moment
- Moment due to wing lift force
- Moment due to wing structure weight
- Moment due to fuel weight.

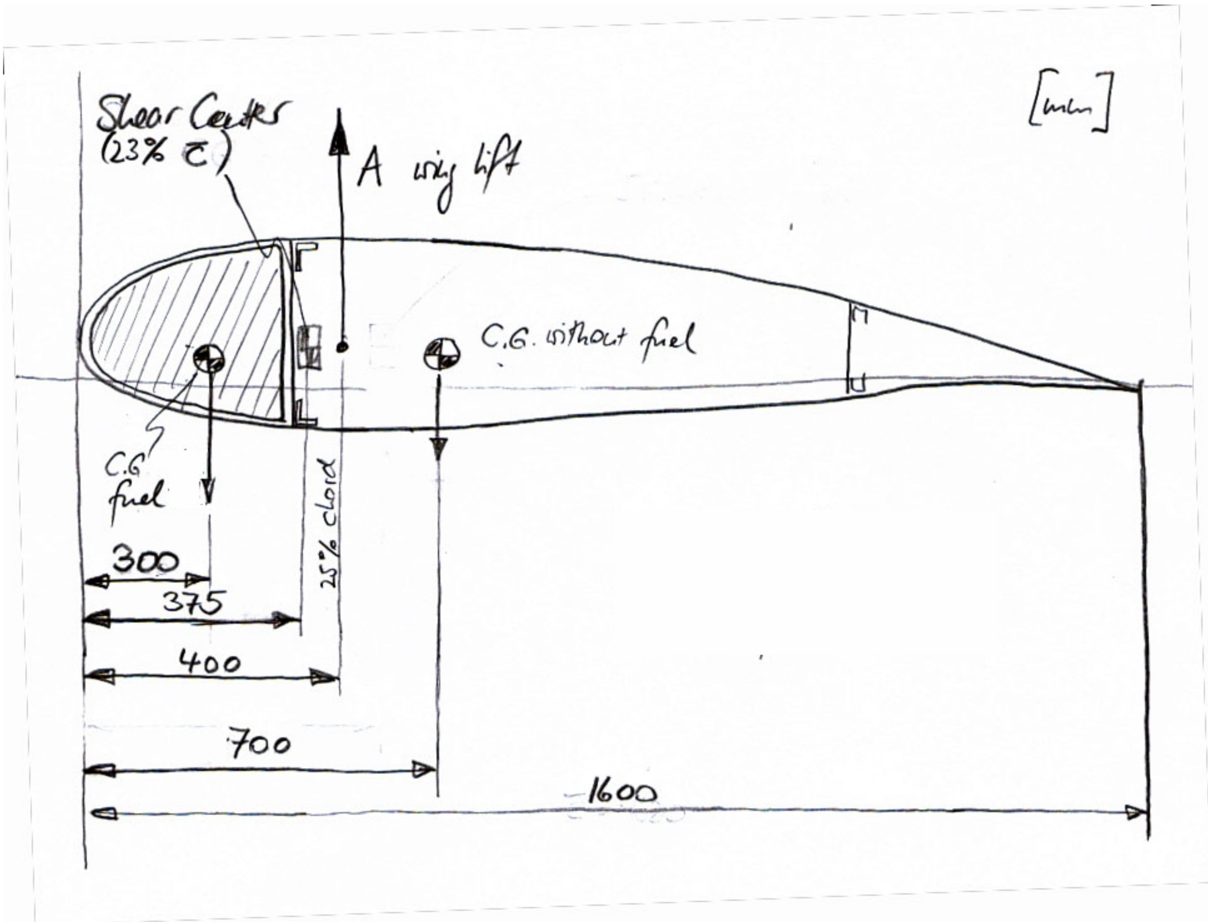


Fig. 7: Forces acting on wing cross section

Aerodynamic wing moment: $$M_{c/4} = c_{m,c/4} \cdot \frac{\rho}{2} v^2 \cdot A_{1Wing} \cdot c_{mean}$$

with $c_{m,c/4} = -0{,}0587$ and $c_{m,c/4,flaps} = -0{,}25$
$A_{1Wing} = 5{,}4\ m^2$, $c_{mean} = 1{,}52\ m$

Wing torsion moment: $$M_{Torsion} = M_{c/4} - \Delta_{Lift} \cdot L_{Wing} + \Delta_{Wing} \cdot W_{Wing} - \Delta_{Fuel} \cdot W_{Fuel}$$

with $\Delta_{Lift} = 0{,}4 - 0{,}375 = 0{,}025\ m$
$\Delta_{Wing} = 0{,}7 - 0{,}375 = 0{,}325\ m$
$\Delta_{Fuel} = 0{,}375 - 0{,}3 = 0{,}075\ m$

It is obvious that the wing torsion depends on airspeed and load factor. Therefore calculations for different points of the flight envelope (A, D, E and G) have to be performed.

The results are summarized in the following table:

Wing Torsion			Speed		n	M(T,wing)	M(T,total)
Lift(1 wing)	259,2	kg		[m/s]	[-]	[Nm]	[Nm]
W(1 wing)	44	kg	vF	36,0	1,0	-1630	-1603
Fuel(1 wing)	90	Liter		36,0	1,9	-1630	-1579
c(M,c/4)	-0,0587		vA	48,9	1,0	-705	-678
c(M,c/4,flaps)	-0,25			48,9	3,8	-705	-602
			vD	80,3	1,0	-1901	-1874
				80,3	3,8	-1901	-1798

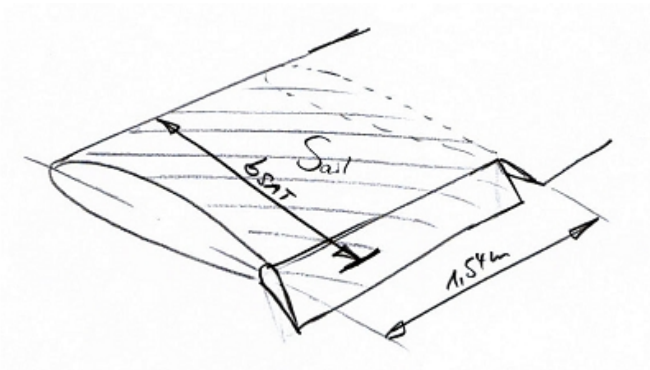
The critical case is at v_D and n = 1,0 (highlighted in red). The ultimate torsion moment is:

Ultimate torsion moment: $M_{Torsion,ult} = 1{,}5 \cdot M_{Torsion,\lim} = 1{,}5 \cdot -1'874\,Nm = -2'811\,Nm$

3.7 Unsymmetrical Flight Conditions

CS-VLA A9 (c)(3)

According to regulations (CS-VLA Appendix A) the wing has to withstand a combination of 75% of the positive maneuvering wing loading on both sides and the maximum wing torsion resulting from aileron input (Fig. 8).



Portion of wing with aileron:
$A_{w,ail}$ = 2,3 m²
b_{ail} = 1,47 m

Portion of wing without aileron:
$A_{w.clear}$ = 3,1 m²
b_{clear} = 1,57 m

Aileron deflections:
δ_{up} = + 11,5°
δ_{down} = - 11,5°

Fig. 8: Wing with deflected aileron

The method of calculation for the effect of aileron displacement on wing torsion is described in CS-VLA Appendix A.

Step 1: Determination of critical airspeed / aileron deflection

Total aileron deflection at v_A: $\Delta_A = \delta_{up} + \delta_{down} = 11{,}5° + 11{,}5° = 23°$

Total aileron deflection at v_C: $\Delta_C = \frac{v_A}{v_C} \cdot \Delta_A = \frac{48{,}9}{55{,}0} \cdot 23° = 20{,}0°$

Total aileron deflection at v_D: $\Delta_D = 0{,}5 \cdot \frac{v_A}{v_D} \cdot \Delta_A = 0{,}5 \cdot \frac{48{,}9}{80{,}3} \cdot 23° = 7{,}0°$

K-factor:

$$K = \frac{\left(c_{m0} - 0{,}01 \cdot \frac{\Delta_D}{2}\right) \cdot v_D^2}{\left(c_{m0} - 0{,}01 \cdot \frac{\Delta_C}{2}\right) \cdot v_C^2} = \frac{\left(-0{,}0587 - 0{,}01 \cdot \frac{7{,}0}{2}\right)}{\left(-0{,}0587 - 0{,}01 \cdot \frac{20{,}0}{2}\right)} \cdot \frac{80{,}3^2}{55{,}0^2} = \frac{-0{,}0973}{-0{,}1587} \cdot \frac{80{,}3^2}{55{,}0^2} = 1{,}30$$

Step 2: Calculation of aerodynamic torsion moment at v_D:

<u>K > 1</u>, therefore aileron deflection Δ_D at v_D is critical and must be used in computing wing torsion loads over the aileron span.

Modified c_m, aileron up: $c_{m,up} = c_{m0} + 0{,}01 \cdot \delta_{up} = -0{,}0587 + 0{,}01 \cdot 3{,}5 = -0{,}0237$

Modified c_m, aileron down: $c_{m,down} = c_{m0} - 0{,}01 \cdot \delta_{down} = -0{,}0587 - 0{,}01 \cdot 3{,}5 = -0{,}0937$

The torsion moment of the wing is calculated for the inner section of the wing without aileron (M_{clear}) and the outer section of the wing with deflected aileron (M_{ail}).

Torsion moment clear: $M_{clear} = c_{m0} \cdot \frac{\rho}{2} v_D{}^2 \cdot A_{w,clear} \cdot b_{clear}$

$$M_{clear} = -0{,}0587 \cdot \frac{1{,}225}{2} 80{,}3^2 \cdot 3{,}1 \cdot 1{,}57 = -1'128 Nm$$

Torsion moment aileron: $M_{ail} = c_{m,up/down} \cdot \frac{\rho}{2} v_D{}^2 \cdot A_{w,ail} \cdot b_{ail}$

$$M_{ail,up} = -0{,}0237 \cdot \frac{1{,}225}{2} 80{,}3^2 \cdot 2{,}3 \cdot 1{,}47 = -316 Nm$$

$$M_{ail,down} = -0{,}0937 \cdot \frac{1{,}225}{2} 80{,}3^2 \cdot 2{,}3 \cdot 1{,}47 = -1'251 Nm$$

Total aerodynamic moment: $M = M_{clear} + M_{ail}$

$$M_{up} = -1'128 - 316 = -1'444 Nm$$

$$M_{down} = -1'128 - 1'251 = -2'379 Nm$$

Step 3: Calculation of total torsion moment at v_D and 75% positive normal load (n=3,8):

Wing lift at 75% normal load (1 wing): $L_{75\%} = 75\% \cdot L_{1wing} = 0{,}75 \cdot 10'046 N = 7'535 N$

Wing inertia relief at 75% normal load: $W_{75\%} = 75\% \cdot W_{Wing} = 0{,}75 \cdot (-1'640 N) = -1'230 N$

Fuel inertia relief at 75% normal load: $F_{75\%} = 75\% \cdot W_{Fuel} = 0{,}75 \cdot (-2'513 N) = -1'885 N$

Total torsion moment: $M_T = M - \Delta_{Lift} \cdot L_{75\%} + \Delta_{Wing} \cdot W_{75\%} - \Delta_{Fuel} \cdot F_{75\%}$

$$M_{T,down} = -2'379 N - 0{,}025 m \cdot 7'535 N + 0{,}325 m \cdot 1'230 N - 0{,}075 m \cdot 1'885 N = -2'309 Nm$$

$$M_{T,up} = -1'444 N - 0{,}025 m \cdot 7'535 N + 0{,}325 m \cdot 1'230 N - 0{,}075 m \cdot 1'885 N = -1'374 Nm$$

Step 4: Ultimate loads:

Ultimate asymmetric torsion moment: $M_{T,up,ult} = 1{,}5 \cdot M_{T,up} = -3'463 Nm$

$$M_{T,down,ult} = 1{,}5 \cdot M_{T,down} = -2'061 Nm$$

3.8 Gust Loading

CS-VLA 333 (not required for Appendix A)

The gust loading of the wing can be calculated according to CS-VLA 333 (however, not required for CS-VLA Appendix A). Gust loads are considered as follows:

- at V_C: gusts of U_{de} = 15.24 m/s
- at V_D: gusts of U_{de} = 7.62 m/s.

Critical aircraft weights are MTOW (W_{max} = 600 kg) and minimum weight (W_{min} = 405 kg).

Gust load calculation (CS-VLA 333): $$n = 1 + \frac{\frac{\rho_0}{2} \cdot v \cdot \frac{d(c_L)}{d\alpha} \cdot K_g \cdot U_{de}}{W \cdot g/S}$$

$$K_g = \frac{0.88 \cdot \mu_g}{5.3 + \mu_g}$$

$$\mu_g = \frac{2 \cdot W/S}{\rho \cdot c_{mean} \cdot \frac{d(c_L)}{d\alpha}}$$

The results are summarized in the following table:

Acft Weight		*Airspeed*		*Gust*		μ_g	K_g	n_g*(pos)*	n_g*(neg)*
	[kg]		[m/s]		[m/s]				
MTOW	600	v_C	55	U_{de}	15,24	12,18	0,613	**3,83**	**-1,83**
Wmin	405	v_C	55	U_{de}	15,24	8,22	0,535	**4,66**	**-2,66**
MTOW	600	V_D	80	U_{de}	7,62	12,18	0,613	**3,06**	**-1,06**
Wmin	405	V_D	80	U_{de}	7,62	8,22	0,535	**3,66**	**-1,66**

The resulting gust loads remain below the design load of +3.8/-1.9, except for the case 2 (405kg/v_C).

Remarks for case 2 (W_{min} = 405 kg, v_C = 55 m/s)

In case 2 the load limit of the flight envelope is exceeded. The calculation of the wing shear and bending moment at W_{min} = 405 kg and n = +4,66 gives the following result:

Limit shear load: 6’754 N
Ultimate shear load: 10’132 N

Limit bending moment: 10’851 Nm
Ultimate bending moment: 16’276 Nm

The loads at W_{min} = 405 kg and n = +4,66 are much lower than at MTOW = 600 kg and n = +3,8. However the local supporting structure for dead weight items needs to withstand the limit load of n = +4,66.

4 Fuselage

CS-VLA A9

The fuselage has to be load tested according to CS-VLA, similar to the wing load tests. The required loads on the fuselage (Fig. 9) are equal to the ultimate loads calculated for engine mount, wing, horizontal tail and vertical tail and the inertial forces of pilot/passenger, baggage, ballistic recovery system and fuselage structure itself (including weight of instrumentation, avionics, airplane systems) at n = 3.8 and a safety factor of 1.5 (ultimate loads).

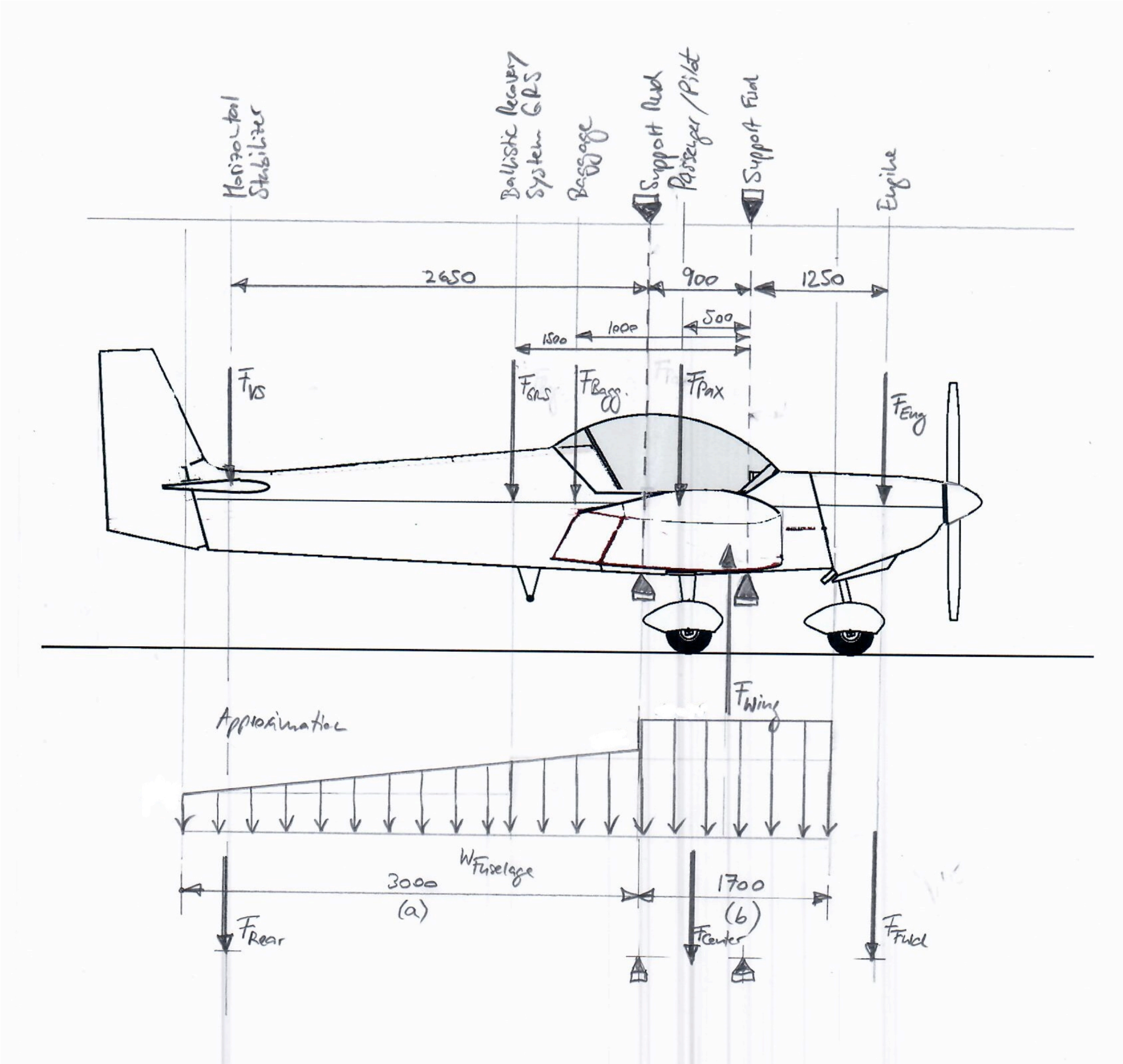


Fig. 9: Load distribution on fuselage

The ultimate loads acting vertically on the fuselage are:

Engine: $F_{Eng} = 4'305N$ *(chapter 9 Engine Mount)*

Pilot/passenger: $F_{Pax} = 1.5 \cdot 3.8 \cdot (2 \cdot 86kg) \cdot 9,806 \frac{m}{s^2} = 9'607N$

Baggage: $F_{Bag} = 1.5 \cdot 3.8 \cdot 10kg \cdot 9,806 \frac{m}{s^2} = 559N$

Ballistic recovery system: $F_{GRS} = 1.5 \cdot 3.8 \cdot 12.3kg \cdot 9{,}806 \frac{m}{s^2} = 687N$

Horizontal stabilizer: $F_{HT} = 3'270N$ *(chapter 5 Horizontal Tail)*

The fuselage weight including weight of instrumentation, avionics and airplane systems is:

Weight of fuselage: $m_{Fuselage} = ZFW - m_{Eng} - m_{Wings} - m_{GRS}$

$$= 340 - 90 - 83 - 12 = 155kg$$

The total weight of the fuselage is distributed along the longitudinal axis of the airplane. A simplified distribution $w_{Fuselage}$ is shown in Fig. 9. It takes into account, that most of the weight (including gear and instruments) is located in the forward part of the fuselage, whereas the weight of the rear fuselage decreases rearward due to taper of the fuselage.

The rear fuselage part (a) weighs approximately 35 kg and the forward fuselage part (b) 120 kg.

Rear fuselage load (ultimate): $F_{RearFuselage} = 35kg \cdot 9{,}806 \frac{m}{s^2} \cdot 1{,}5 \cdot 3{,}8 = 1'956N$ (a)

Forward fuselage load (ultimate): $F_{FwdFuselage} = 120kg \cdot 9{,}806 \frac{m}{s^2} \cdot 1{,}5 \cdot 3{,}8 = 6'707N$ (b)

A simplified setup for a fuselage load test is shown in Fig. 10. The fuselage is supported upside down at the horizontal tail attachment points and at the engine mount, while a (wing lift) force is introduced at the wing attachment points.

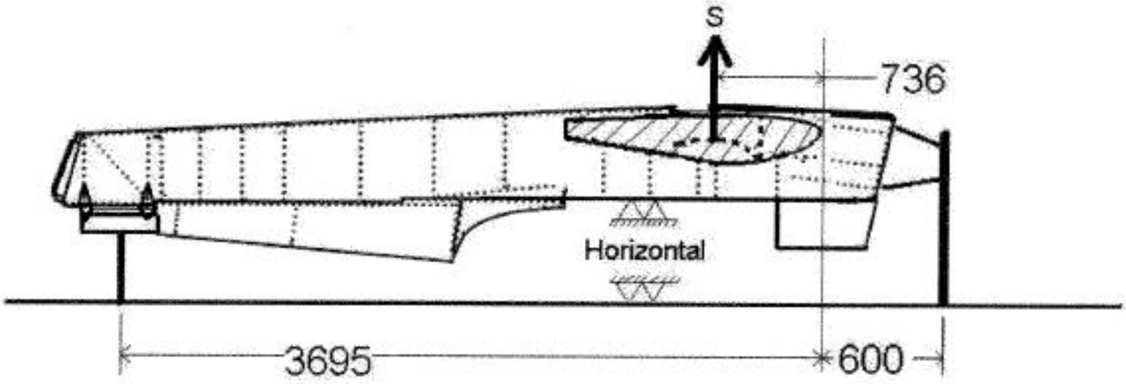


Fig. 10: Simplified load test on fuselage

The total load, that has to be tested and verified at the engine mount and at the horizontal tail attachment points, is:

Load at engine mount: $F_{MOT} = F_{Eng} + \frac{400}{1250} \cdot \frac{800}{1700} \cdot F_{FwdFuselage}$

$$F_{MOT} = 4'305N +\cdot 0{,}15 \cdot 6'707N = 5'311N$$

Load at horizontal tail attachment: $F_{VOP} = F_{HT} +\cdot 0{,}42 \cdot F_{RearFuselage} + 0{,}04 \cdot F_{Bag} + 0{,}23 \cdot F_{GRS}$

$$= 3'270N +\cdot 0{,}42 \cdot 1'956N + 0{,}04 \cdot 559N + 0{,}23 \cdot 687N$$

$$= 4'271N$$

The LAA engineering states in the document *TL 1.17 “Aircraft Loads and Load Testing” [Ref]*:

“The airloads acting on the fuselage itself are usually ignored, and with light aircraft it is also common to neglect the inertia relief provided by the mass of the fuselage structure.”

Therefore the loads calculated above can be considered conservative.

5 Horizontal Tail

5.1 Surface Loading Condition

CS-VLA A11 (c)(1)

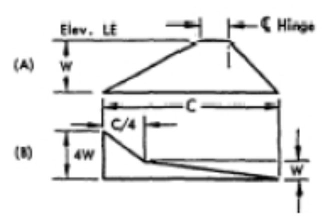
The average limit loading of the horizontal tail can be calculated according to CS-VLA Appendix A, Table 2 and Figure A4:

Simplified limit surface distributions: $w_{HT} = 4{,}8 + 0{,}109 \cdot n_1 \cdot \frac{W}{S} = 24{,}68 \frac{lb}{ft^2} = 120{,}8 \frac{kg}{m^2}$

Simplified limit surface loading: $L_{HT,\lim} = A_{HT} \cdot w_{HT} \cdot g = 1{,}84 \cdot 120{,}8 \cdot 9{,}806 = 2'180\,N$

Ultimate surface loading: $L_{HT,ult} = 1{,}5 \cdot L_{HT,\lim} = 3'270\,N$

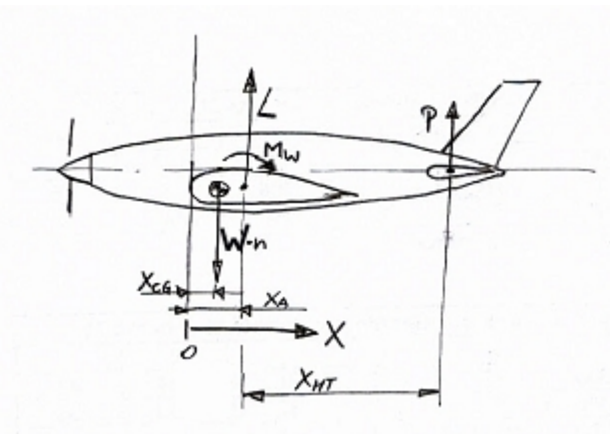
The load must be distributed on the horizontal tail as follows:

HORIZONTAL TAIL I	(a)	Up and Down	Figure A4 Curve (2)	**3'270 N** 
	(b)	Unsymmetrical loading (Up and Down)	100% $\overline{w}$ on one side aeroplane ℄ 65% $\overline{w}$ on other side aeroplane ℄ for normal and utility categories. For aerobatic c... A11(c) **2'126 N**	

5.2 Balancing Load

For comparison/confirmation of the simplified criteria, a detailed calculation for the balancing load is performed.

The horizontal tail acts with a downward force against the forward nick moment and keeps the airplane in balance. Instead of using the simplified criteria of CS-VLA Appendix A (Chapter 3) the following, more detailed analysis may be used:



L .... Wing Lift

M_W .... Wing Nick Moment

P.............. Horizontal Tail Lift

W·n.......... Inertial Weight Force

x_{CG} .... Center of Gravity (behind wing L.E.)

xA .... Center of Lift (behind wing L.E.)

xHT.......... Arm to H.T. Center of Lift

Equilibrium of moment at wing L.E.: $0 = M_W + x_A \cdot L - x_{CG} \cdot n \cdot W + x_{HT} \cdot P$

Equilibrium of forces: $0 = -n \cdot W + L + P$

Zero Lift Moment: $M_W = c_{m,c/4} \cdot c_{mean} \cdot \frac{\rho}{2} v^2 \cdot A_{2Wing}$

$c_{m,c/4} = -0{,}0587$, $c_{mean} = 1{,}523\ m$, $\rho = 1{,}225\ kg/m^3$, $A_2 = 12{,}5\ m^2$

Force on horizontal tail: $P = \frac{M_W + (x_{CG} - x_A) \cdot n \cdot G}{x_{HT}}$

The results for the balancing loads on the HT are summarized in the following table:

v	CG	n	Pb [N]
vA	fwd	1,0	-588
vA	fwd	3,8	-997
vA	fwd	-1,9	-164
vA	aft	1,0	-316
vA	aft	3,8	35
vA	aft	-1,9	-680

v	CG	n	Pb [N]
vD		0,0	-1190
vD	fwd	1,0	-1336
vD	fwd	3,8	-1746
vD	fwd	-1,9	-912
vD	aft	1,0	-1065
vD	aft	3,8	-714
vD	aft	-1,9	-1428

v	CG	n	Pb [N]
vC	fwd	1,0	-706
vC	fwd	3,8	-1115
vC	fwd	-1,9	-282
vC	aft	1,0	-435
vC	aft	3,8	-84
vC	aft	-1,9	-798

v	CG	n	Pb [N]
vF	fwd	1,0	-1167
vF	fwd	1,9	-1298
vF	fwd	0	-1021
vF	aft	1,0	-895
vF	aft	1,9	-782
vF	aft	0,0	-1021

The maximum balancing load on the horizontal tail appears to be at v_D, forward C.G. and n = 3,8.

Ultimate HT balancing load: $P_{b,ult} = 1{,}5 \cdot P_b = 1{,}5 \cdot -1'746\,N = -2'619\,N$

The resulting ultimate load is 20% lower than the result from the simplified calculation according to CS-VLA. The CS-VLA approximation is therefore conservative.

6 Vertical Tail

6.1 Surface Loading Condition

CS-VLA A11 (c)(1)

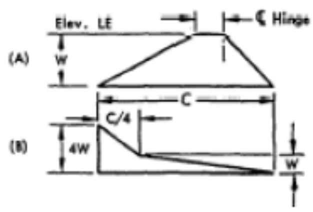
The average limit loading of the vertical tail can be calculated according to CS-VLA Appendix A, Table 2 and Figure A4:

Simplified limit surface distributions: $w_{VT} = 1{,}656 \cdot \sqrt{n_1 \cdot \frac{W}{S}} = 22{,}37 \frac{lb}{ft^2} = 109{,}4 \frac{kg}{m^2}$

Simplified limit surface loading: $L_{VT,\lim} = A_{VT} \cdot w_{VT} \cdot g = 0{,}52 \cdot 109{,}4 \cdot 9{,}806 = 558N$

Ultimate surface loading: $L_{VT,ult} = 1{,}5 \cdot L_{VT,\lim} = 837N$

The load must be distributed on the horizontal tail as follows:

~~HORIZONTAL TAIL I~~	~~(a)~~	~~Up and Down~~	~~Figure A4 Curve (2)~~	
	~~(b)~~	~~Unsymmetrical loading (Up and Down)~~	~~100% $\overline{w}$ on one side aeroplane ℄~~ ~~65% $\overline{w}$ on other side aeroplane ℄ for normal and utility categories.~~ ~~For aerobatic category see A11(c)~~	
VERTICAL TAIL II	(a)	Right and Left	Figure A4 Curve (1)	Same as (A) above
	(b)	Right and Left	Figure A4 Curve (1)	Same as (B) above

7 Control Surfaces

CS-VLA A11 (c)(1)

The average limit loading of the control surfaces can be calculated according to CS-VLA Appendix A, Table 2 and Figure A5.

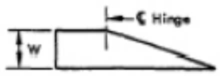
7.1 Aileron

Simplified limit surface distributions: $w_{Ail} = 0{,}095 \cdot n_1 \cdot \frac{W}{S} = 17{,}33 \frac{lb}{ft^2} = 84{,}8 \frac{kg}{m^2}$

Simplified limit surface loading: $L_{Ail,\lim} = A_{Ail} \cdot w_{Ail} \cdot g = 0{,}46 \cdot 84{,}8 \cdot 9{,}806 = 383N$

Ultimate surface loading: $L_{Ail,ult} = 1{,}5 \cdot L_{Ail,\lim} = 575N$

The load must be distributed on the aileron as follows:

AILERON III	(a)	Up and Down	Figure A5 Curve (5)	(C) 

7.2 Wing Flap

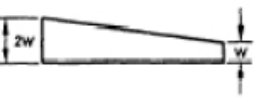
Simplified limit surface distributions: $w_{Flap} = 0{,}131 \cdot n_1 \cdot \frac{W}{S} \cdot \frac{c_{n,flap}}{1{,}6} = 17{,}92 \frac{lb}{ft^2} = 87{,}7 \frac{kg}{m^2}$

with $c_{n,flap} = 1{,}2$

Simplified limit surface loading: $L_{Flap,\lim} = A_{Flap} \cdot w_{Flap} \cdot g = 0{,}68 \cdot 87{,}7 \cdot 9{,}806 = 585N$

Ultimate surface loading: $L_{Flap,ult} = 1{,}5 \cdot L_{Flap,\lim} = 878N$

The load must be distributed on the wing flap as follows:

WING FLAP IV	(a)	Up	Figure A5 Curve (4)	(D) 
	(b)	Down	0·25 x Up load (a)	

7.3 Elevator

The elevator limit surface loading can be calculated by assuming that the limit load on the horizontal tail (calculated in chapter 5) is distributed pro-rata on the elevator surface.

Limit surface loading: $L_{Elev,lim} = \frac{c_{Elev}}{c_{HT}} \cdot L_{HT,lim} = \frac{0{,}35m}{0{,}8m} \cdot 2'180N = 763N$

Ultimate surface loading: $L_{Elev,ult} = 1{,}5 \cdot L_{Elev,lim} = 1'145N$

7.4 Rudder

The CH601XL-B has an all-moving rudder. Therefore the control forces of the rudder can be considered equal to the calculated vertical tail limit surface loading (chapter 6).

Limit rudder surface loading: $L_{Rud,lim} = L_{VT,lim} = 558N$

Ultimate rudder surface loading: $L_{Rud,lim} = 1{,}5 \cdot L_{Rud,lim} = 837N$

7.5 Aileron + Elevator Trim Tab

Simplified limit surface distributions: $$w_{Tab} = 0{,}16 \cdot n_1 \cdot \frac{W}{S} \cdot \frac{c_{n,tab}}{0{,}8} = 29{,}18 \frac{lb}{ft^2} = 142{,}8 \frac{kg}{m^2}$$

with $c_{n,tab} = 0{,}8$

Simplified limit surface loading: $$L_{AilTab,\lim} = A_{AilTab} \cdot w_{Tab} \cdot g = 0{,}039 \cdot 142{,}8 \cdot 9{,}806 = 49{,}6N$$
$$L_{ElevTab,\lim} = A_{ElevTab} \cdot w_{Tab} \cdot g = 0{,}056 \cdot 142{,}8 \cdot 9{,}806 = 78{,}4N$$

Ultimate surface loading: $$L_{AilTab,ult} = 1{,}5 \cdot L_{TailTab,\lim} = 74N$$
$$L_{ElevTab,ult} = 1{,}5 \cdot L_{ElevTab,\lim} = 118N$$

The load must be distributed on the trim tabs as follows:

TRIM TAB V	(a)	Up and Down	Figure A5 Curve (3)	Same as (D) above

See „wing flap loading“

8 Control System

CS-VLA A13 (a)(2)

The acceptable limit pilot forces can be used as requirement for the control system strength.

CS-VLA 397 (b)

The limit control loads must not be higher than 125% of the computed hinge moments according to chapter 7 “Control Surfaces”.

In addition the minimum and maximum limit pilot forces are as follows:

- Aileron limit force (control stick): 178 .. 300 N
- Elevator limit force (control stick): 445 .. 740 N
- Rudder limit force (pedals): 580 .. 890 N

Elevator

The geometry for the elevator control is shown in Fig. 11.

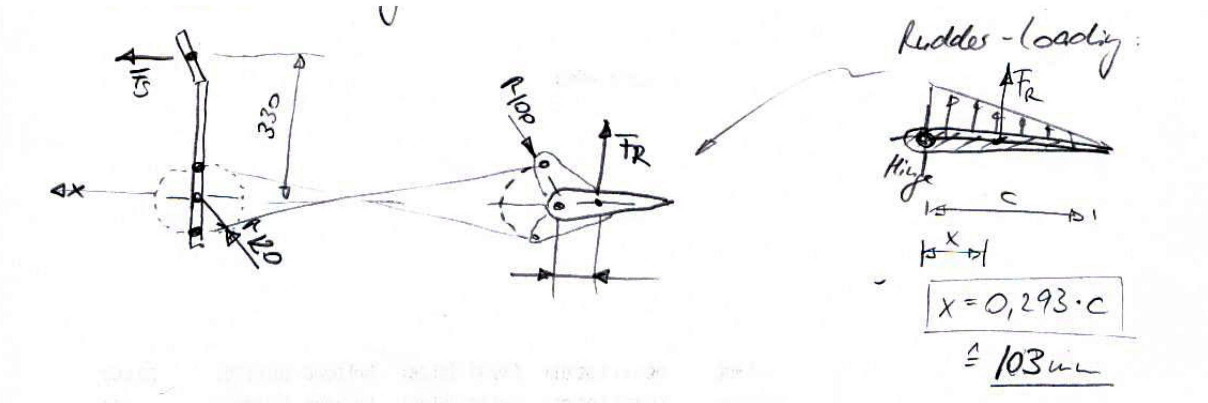


Fig. 11: Elevator control geometry

Elevator control force at stick: $F_{S,Elev} = 1{,}25 \cdot L_{Elev,lim} \cdot \frac{120}{330} \cdot \frac{100}{103} = 336N$

The force of the elevator control at the control stick due to the limit load of the elevator is lower than the minimum pilot force, therefore:

Elevator stick limit force: $F_{SLoadTest,Elev} = 445N$

Aileron

The geometry for the aileron control is shown in Fig. 12:

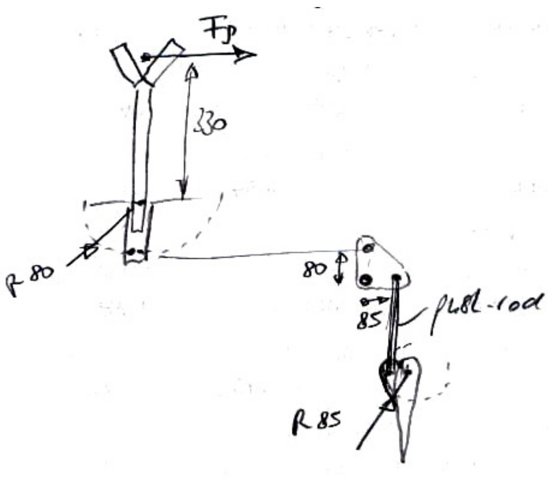


Fig. 12: Aileron control geometry

Aileron control force at stick: $F_{S,Aileron} = 1{,}25 \cdot 2 \cdot L_{Ail,lim} \cdot \frac{80}{330} \cdot \frac{85}{80} \cdot \frac{91}{85} = 264N$

The force of the aileron control at the control stick due to the limit load of the aileron is higher than the minimum pilot force and lower than the maximum pilot force, therefore:

Aileron stick limit force: $F_{SLoadTest,Ail} = 264N$

Rudder

The geometry of the rudder control is shown in Fig. 13.

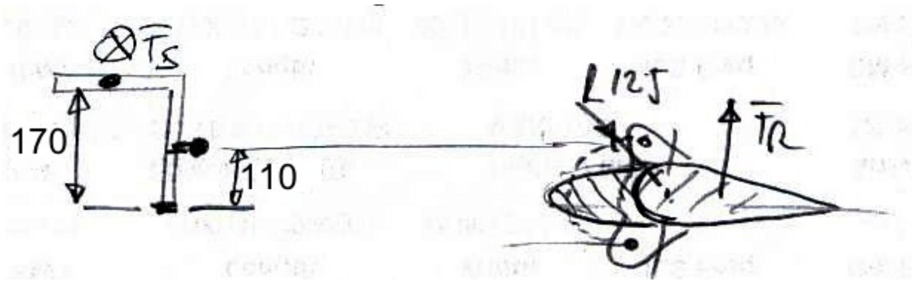


Fig. 13: Rudder control geometry

Rudder control force at pedals: $F_{S,Rudder} = 1{,}25 \cdot L_{Rud,lim} \cdot \frac{110}{170} \cdot \frac{200}{125} = 722N$

The force of the control at the rudder pedals due to the limit load of the rudder is higher than the minimum pilot force and lower than the maximum pilot force, therefore:

Rudder pedal force for load test: $F_{SLoadTest,Rudder} = 722N$

CS-VLA 397 (c)

In addition the rudder control system must withstand a simultaneous forward force of 1’000 N on both pedals.

9 Engine Mount

9.1 Loads on engine mount

Each of the following two conditions must be investigated:

A. Torque + positive maneuvering flight load

CS-VLA 361

Maximum engine torque: $M_{eng} = 121\ Nm$ [3]

Maximum propeller torque: $M_{prop} = k \cdot M_{eng} = 2{,}43 \cdot 121\ Nm = \ 294 Nm$

Limit torque for 4-stroke/4-cylinder: $M_{lim} = 2{,}0 \cdot M_{prop} = 588\ Nm$

Ultimate torque: $M_{ult} = 1{,}5 \cdot M_{lim} = 882\ Nm$

Limit loads resulting from the maximum positive maneuvering flight load factor n_1:

Limit vertical load: $T_{vert} = n_1 \cdot \left(m_{eng} + m_{prop}\right) \cdot g$

$T_{vert,lim} = 3{,}8 \cdot (65 + 12) \cdot 9{,}806 = 2'869N$

Ultimate vertical load: $T_{vert,ult} = 1{,}5 \cdot T_{vert,lim} = 4'304N$

B. Lateral side load

CS-VLA 363

Lateral (side) limit load: $T_{side,lim} = 1{,}33 \cdot \left(m_{eng} + m_{prop}\right) \cdot g$

Limit side load: $T_{side,lim} = 1{,}33 \cdot (65 + 12) \cdot 9{,}806 = 1'004N$

Ultimate side load: $T_{side,ult} = 1{,}5 \cdot T_{side,lim} = 1'506N$

[3] Specification by Rotax Engines

10 Landing Gear

The main landing gear of the CH601XL-B consists of two separate GRP-gear legs attached to a single steel beam at the fuselage bottom. The nose gear consists of a steel axle attached to the firewall and an aluminium U-profile for the wheel support.

10.1 Ground Load Conditions

The requirements for ground loads are specified in CS-VLA 471 – 499.

CS-VLA 473 (b)

Descent velocity: $v_{vertical} = 0{,}51 \cdot \left(\frac{m_{MTOW} \cdot g}{S_{Wing}}\right)^{1/4}$

$$v_{vertical} = 0{,}51 \cdot \left(\frac{600kg \cdot 9{,}806\frac{m}{s^2}}{12.3m^2}\right)^{1/4} = 2{,}39\,\frac{m}{s}$$

According to CS-VLA the descent velocity must not be more than 3.05 m/s and may not be less than 2.13 m/s.

CS-VLA 473 (c)

Remaining wing lift at landing impact: $L_{T/D} = \frac{2}{3} \cdot L = \frac{2}{3} \cdot 600kg \cdot 9{,}806\,\frac{m}{s^2} = 3'922N$

CS-VLA 473 (d)

Energy absorption tests (drop tests) are made to determine the limit load factor corresponding to the required limit descent velocities. These tests are made under CS-VLA 725 (chapter 10.7).

CS-VLA 473 (e)

Kinetic energy

In a steady descent at vertical velocity $v_{vertical}$ the kinetic energy of the motion is:

Kinetic energy: $E_{kin} = \frac{1}{2} \cdot \frac{W}{g} \cdot v_{vertical}{}^2 = \frac{1}{2} \cdot 600 \cdot 2{,}39^2 = 1'707J$

Work energy

Under CS-VLA it is assumed that the lift *L* is suddenly reduced by 1/3 at wheel ground contact. During the landing an unbalanced force *W/3* does work over a stroke equal to the vertical travel of the landing gear (d_h) plus the tire compression (d_t).

Work energy (deflection): $E_d = \frac{W}{3} \cdot (d_h + d_t)$

The sum of these two equations represents the total energy that must be absorbed by the tires and the gear legs. The maximum ground reaction *R* that results is the “Design Limit Load” for the landing gear.

With a wing lift of *2/3 W* in effect during the landing the total external load is *R + 2/3 W* and the load factor at the aircraft C.G. is given by:

Design limit load factor: $n = R/W + 2/3$

Tire energy capacity

A formula from British Ministry of Supply S&T Memo #10/52 provides:

$$R_{lbs} = 2{,}25 \cdot \sqrt{D \cdot w} \cdot \left(d_{t,in} - 0{,}03 \cdot w\right) \cdot (p + p_c)$$

*R_{lbs} = *** lbs*	*load on tire*
D = 13 in	*overall tire diameter*
w = 7.5 in	*overall tire width*
*$d_{t,in}$ = *** in*	*tire deflection*
p = 58 psi	*tire inflation pressure*
p_c = 2 psi	*tire cover rigidity*

With a ground load in [N] the resulting tire deflection in [mm] is:

$$d_t = \frac{R}{234} + 5{,}71$$

The energy absorbed by one tire for any load *R* and corresponding tire deflection d_t is:

$$E_t = 0{,}47 \cdot R \cdot d_t = R\left(\frac{R}{234} + 5{,}71\right) \qquad \textit{with } d_t \textit{ in mm}$$

Gear leg energy capacity

The springing behavior of the gear leg is determined by a drop test [*Ref: CZAW SportCruiser*] in tail-down landing position.

Drop height: $h_{drop} = 0.658m$

Effective drop weight: $m_e = 436kg$

Energy at ground impact on one gear leg: $E_{pot} = \frac{1}{2} \cdot m_e \cdot g \cdot h_{drop} = 1'407J$

The drop test resulted in a maximum horizontal gear deflection on ground (including wheel) of 370 mm (Fig. 14) at an inertia load factor of n = 6. The horizontal deflection of the composite gear leg equals approx. 300 mm.

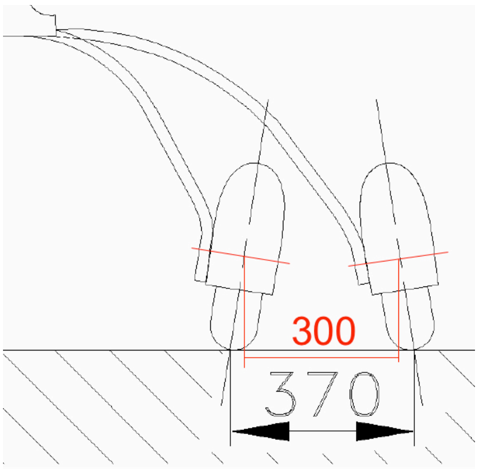


Fig. 14: Gear leg deflection during ultimate drop test

In addition a 2-dimensional finite element analysis using the LISA FEA package was created. The resulting deflection of the gear leg during the ultimate drop test is shown in Fig. 15 (E-modulus = 18'000 N/mm2, load at wheel axle = 11'300 N). The coloring represents the local normal stress in N/mm^2.

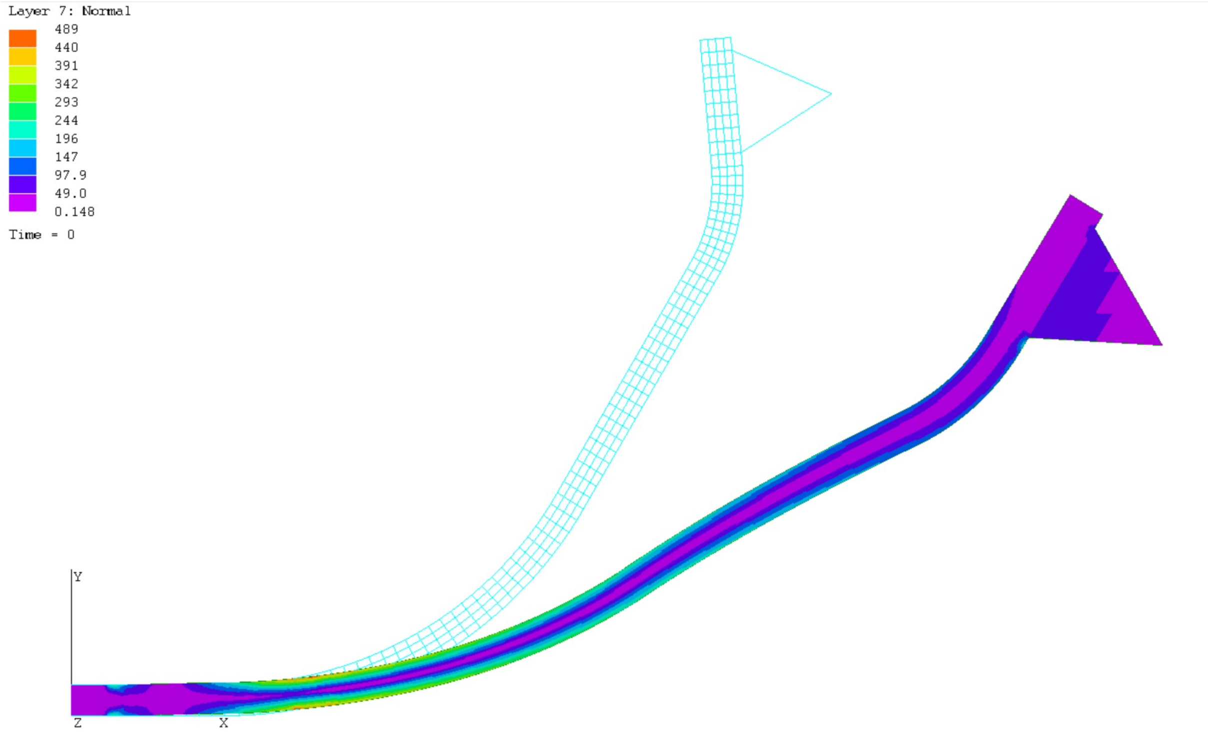


Fig. 15: FEA analysis of composite gear leg

Springing behavior of gear leg: $R = 57'000 \frac{N}{m} \cdot d_g$ *with d_g in vertical (y) direction only*

The energy absorbed by one gear leg for any load R is:

$$E_g = \frac{1}{2} \cdot 57'000 \frac{N}{m} \cdot {d_g}^2$$

Energy balance

The energy balance for one gear leg in the tail down landing condition (landing on main wheels only) is:

$$\frac{1}{2}(E_{kin} + E_d) = E_t + E_g$$

The results of the energy balance calculation are tabulated on the next page (MTOW, $v_{vertical}$, 2/3 L at touchdown).

ENERGY ABSORPTION COMPOSITE GEAR-LEG CH601XL-B

No	AIRCRAFT SPECIFICATIONS				
1	Aircraft weight (MTOW)	m	600	kg	
2	Wing area	A	12.3	m²	
	GRP MATERIAL SPECIFICATIONS				
3	Modulus of elasticity (drop test CZAW SportCruiser)	E	18000	N/mm²	
4	Max. tensile strength [Ref]	σ	720	N/mm²	
	FORCE ON 1 MAIN GEAR LEG				
5	Force on 1 gear leg	**Fg**	**9850**	**N**	
6	Fraction of total load		0.5		
7	Weight on 1 gear leg	mg	300	kg	
	LOAD IN GEAR				
8	Gear load factor	**ng**	**3.35**		
9	Max. tensile stress in gear leg (FEA results)	**e**	**490**	**N/mm²**	
10	Safety factor	**SF**	**1.47**		
	ENERGY PER MAIN GEAR LEG				**TOTAL**
	Kinetic energy				
11	Vertical speed	v	2.39	m/s	[CS-VLA 473 (b)]
12	Kinetic energy	Ekin	853.3	J	
	Work energy				
13	1/3 Weight (2/3 lift)	W/3	980.6	N	[CS-VLA 473 (c)]
14	Work energy (deflection)	Ed	216.3	J	**1069.6 J**
					Λ
	Energy gear leg deformation				=
	Gear leg springing behavior	D	57	N/mm	=
15	Gear leg deflection	**dg**	**172.8**	**mm**	=
16	Gear leg deformation energy	Epot	851.1	J	=
	Energy tire deformation				=
17	Tire deflection	**dt**	**47.8**	**mm**	V
18	Tire deformation energy	Et	221.3	J	**1072.4 J**

Total deflection (gear leg and tire): $\boldsymbol{d} = d_t + d_g = 172{,}8 + 47{,}8 = \mathbf{220{,}6mm}$

Design limit load factor: $\boldsymbol{n} = {}^{R}/_{W} + {}^{2}/_{3} = \frac{2\cdot 9'850}{600\cdot 9{,}806} + 0{,}67 = 3{,}35 + 0{,}67 = \mathbf{4{,}02}$

Safety factor (tensile stress): $\boldsymbol{S.F.} = {}^{\sigma_{t,max}}/_{\sigma_t} = {}^{720}/_{490} = \mathbf{1{,}47}$ .. with $\sigma_t = 490\ {}^{N}/_{mm^2}$

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For a cross check, the limit landing load factor n_j on the wheels for the basic landing conditions can be also computed according to ASTM-requirements:

Drop height: $$h_{drop} = 0{,}0132\sqrt{\frac{m\cdot g}{S}} = 0{,}0132\sqrt{\frac{600kg\cdot 9{,}806\frac{m}{s^2}}{12{,}3m^2}} = 28{,}9cm$$

Load factor on the wheels: $$n_j = \frac{h_{drop} + \frac{d}{3}}{R\cdot d} = \frac{28{,}9 + \frac{22}{3}}{0{,}5\cdot 22} = 3{,}29$$

Limit landing load factor: $$n = n_j + L = 3{,}29 + \frac{2}{3} = 3{,}96$$

10.2 Static Ground Load Conditions

The static ground load reactions are calculated for the most forward and most rearward center of gravity C.G. (gear locations measured by author at aircraft with composite main gear).

Location of nose wheel: *l_N = 630 mm fwd wing leading edge*
Location of main wheels: *l_M = 620 mm aft wing leading edge*

Most forward C.G. (20% = 304 mm)

1 main gear: $R_M = \frac{W}{2} \cdot \frac{l_N + C.G.}{l_N + l_M} = \frac{600 \cdot 9{,}806}{2} \cdot \frac{630+304}{1250} = 2'198N$

Nose gear: $R_N = W - 2 \cdot R_M = 600 \cdot 9{,}806 - 2 \cdot 2'198 = 1'487N$

Distance nose gear – C.G.: $a = l_N + C.G. = 630 + 304 = 934mm$

Distance main gear – C.G.: $b = l_M - C.G. = 620 - 304 = 316mm$

Most rearward C.G. (30% = 456 mm)

1 main gear: $R_M = \frac{W}{2} \cdot \frac{l_N + C.G.}{l_N + l_M} = \frac{600 \cdot 9{,}806}{2} \cdot \frac{630+456}{1250} = 2'556N$

Nose gear: $R_N = W - 2 \cdot R_M = 600 \cdot 9{,}806 - 2 \cdot 2'556 = 772N$

Distance nose gear – C.G.: $a = l_N + C.G. = 630 + 456 = 1'086mm$

Distance main gear – C.G.: $b = l_M - C.G. = 620 - 456 = 164mm$

10.3 Level Landing Conditions
Tail-down Landing Conditions

CS-VLA 479 / 481

The requirements of CS-VLA 479 (level landing conditions) and 481 (tail-down landing conditions) can be confirmed by drop tests according to CS-VLA 725ff. The required airplane weight and drop height are calculated in the following subchapter 10.7.

CS-VLA Appendix C

A table with reactions on the undercarriage for all landing conditions can be found in CS-VLA Appendix C. The following calculations are based on this table.

The strength of the main gear was proven by limit drop tests (refer to requirements of chapter 10.7). Subsequently only the reactions on the nose wheel are calculated for the level landing with inclined reactions.

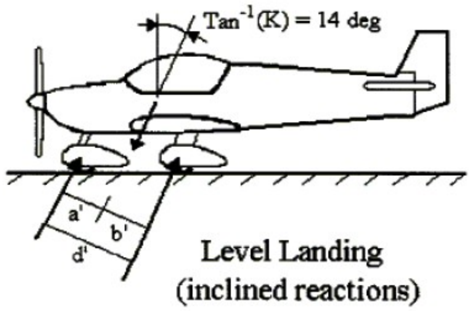
(1) Level landing with inclined reactions (nose gear only, forward C.G. [4])

Geometry for inclined reactions:

Inclination: $\phi = \tan^{-1}(0.25) = 14°$

Variables (at forward C.G.):

$$h = 450mm$$
$$a' = \cos(14°) \cdot (a - \tan(14°) \cdot h) = 797mm$$
$$d' = \mathrm{d} \cdot \cos(14°) = 1'213mm$$
$$b' = \mathrm{d'} - \mathrm{a'} = 416mm$$



Vertical load at nose wheel: $V_R = (n - L) \cdot W \cdot \frac{b'}{d'} = (4{,}0 - 0{,}67) \cdot 600 \cdot 9{,}806 \cdot \frac{416}{1213} = 6'719N$

Drag load at nose wheel: $D_R = 0{,}25 \cdot n \cdot W \cdot \frac{b'}{d'} = (4{,}0 - 0{,}67) \cdot 600 \cdot 9{,}806 \cdot \frac{416}{1213} = 1'680N$

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[4] Forward C.G. is critical for nose gear.

10.4 Side Load Conditions

CS-VLA 485

Vertical load at each main gear leg: $F_{vertical} = \frac{1}{2} \cdot 1{,}33 \cdot m \cdot g = 3'913N/5'869N$ *(Limit/Ultimate)*

Inboard side load: $F_{inboard} = 0{,}5 \cdot m \cdot g = 2'941N/4'413N$

Outboard side load: $F_{outboard} = 0{,}33 \cdot m \cdot g = 1'942N/2'912N$

10.5 Braked Roll Conditions

CS-VLA 493

Vertical load at each main gear leg: $F_{vertical} = \frac{1}{2} \cdot 1{,}33 \cdot m \cdot g = 3'913N/5'869N$ *(Limit/Ultimate)*

Lateral rearward braking force
at each main gear leg: $F_{breaking} = 0{,}8 \cdot m \cdot g = 4'707N/7'060N$

10.6 Supplementary Conditions for Nose Wheel

CS-VLA 499

Critical static load is at forward C.G.: $R_N = 1'487N$ *(see subchapter 10.2)*

Vertical load at nose gear
for all 3 cases below: $F_v = 2{,}25 \cdot R_N = 3'346N/5'019N$

For aft loads (drag loads)

Rearward drag load at nose gear: $F_{aft} = 0{,}8 \cdot F_v = 2'677N/4'015N$

For forward loads

Forward load at nose gear: $F_{fwd} = 0{,}4 \cdot F_v = 1'338N/2'008N$

For side loads

Side load at nose gear: $F_{side} = 0{,}7 \cdot F_v = 2'342N/3'513N$

10.7 Limit Drop Tests

CS-VLA 725 (a)

Minimum drop height: $h_{drop} = 0{,}0132 \cdot \sqrt{\frac{m \cdot g}{S}} = 0{,}0132 \cdot \sqrt{\frac{600 \cdot 9{,}806}{12{,}3}} = 0{,}289m$

CS-VLA 725 (b)

Effective drop weight: $m_e = m \cdot \left[\frac{h_{drop} + (1-L) \cdot d}{h_{drop} + d}\right] = 600 \cdot \left[\frac{289 + 0{,}33 \cdot 221}{289 + 221}\right] = 426kg$

CS-VLA 725 (e)

Limit inertia load factor: $n = n_j \cdot \frac{m_e}{m} + L$ *with n_j = load factor at drop test + 1*

A load test with a drop weight of 436 kg and a drop height of 0,289 m resulted in a maximum load factor of n = 4,9.

$$n = 4{,}9 \cdot \frac{436}{600} + 0{,}67 = 4{,}1$$

CS-VLA 725 (f)

This limit inertia load factor may not be more than the design limit load factor in CS-VLA 473 (e).

10.8 Ground Load Dynamic Test

CS-VLA 726

If compliance with the ground load requirements of CS-VLA 479 to 483 is shown dynamically by drop tests, one drop test must be conducted that meets CS-VLA 725 (same effective drop weight) except that the drop height must be:

Ultimate drop height: $h_{drop,ult} = 2{,}25 \cdot h_{drop} = 0{,}650m$

10.9 Reserve Energy Absorption

CS-VLA 727

Reserve energy drop height: $h_{drop,reserve} = 1{,}44 \cdot h_{drop} = 0{,}416m$

Reserve energy drop weight: $m_{e,reserve} = m \cdot \left[\frac{h_{drop}}{h_{drop}+d}\right] = 600 \cdot \left[\frac{289}{289+221}\right] = 340kg$

It is obvious that the ground load dynamic test also covers the reserve energy requirement.

11 Revisions

30.3.2010	Version 1.0	
22.4.2010	Version 1.1	Several minor corrections
6.5.2011	Version 1.2	Aircraft designator XL changed to XL-B (upgrade installed)
9.2.2014	Version 1.3	Correction of stall speeds vs/vs0, according to (a) computed stall speed formula and (b) flight tests by PFA
22.1.2016	Version 2.0	Chapter 2.2: Correction of wing aspect ratio formula Chapter 3.8: Correction of gust loads at vC and vD Chapter 4: Fuselage weight distribution added Chapter 8: Additional calculations for control loads Chapter 10: Adjustment of ground load conditions
10.2.2016	Version 2.1	Correction of loads on engine mount (→propeller torque)

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