P8 – 0110 – 00014S
Design speeds and loads

Table of contents

1 SUMMARY

This report contains P8 aerodynamical data to calculate design speed and loads and defined design speeds and load factors.

Statement of conformity

This report is done in accordance to ASTM F2245-13b + CS-LSA amendment 1 additional requirements.

ASTM Appendixes;

- “X1 SIMPLIFIED DESIGN LOAD CRITERIA FOR LIGHT SPORT AIRPLANES” and
- “X3. ACCEPTABLE MEANS OF GUST LOAD FACTOR CALCULATIONS” and
- “X4. ACCEPTABLE MEANS FOR CALCULATING GUST LOADS ON STABILIZING SURFACES”

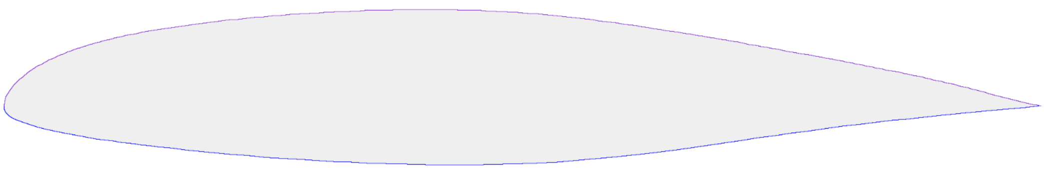
are utilised.

2 BASIC CALCULATION

2.1 Aerodynamical calculations

2.1.1 Profile data

Pik-28 has wing profile NASA NLF(1)-0115. Wing is equipped with a 18,3% c trailing edge flap.



Profile selection was based mainly on reference http://www.n56ml.com/as504x/default.htm on design of new airfoil for KR-2S wing by Ashok Gopalarathnam. The AS5045 airfoil selected for KR-2S was little bit better than NLF(1)-0115, but due to sharper leading edge the more round NLF(1)-0115 was selected for this aircraft.

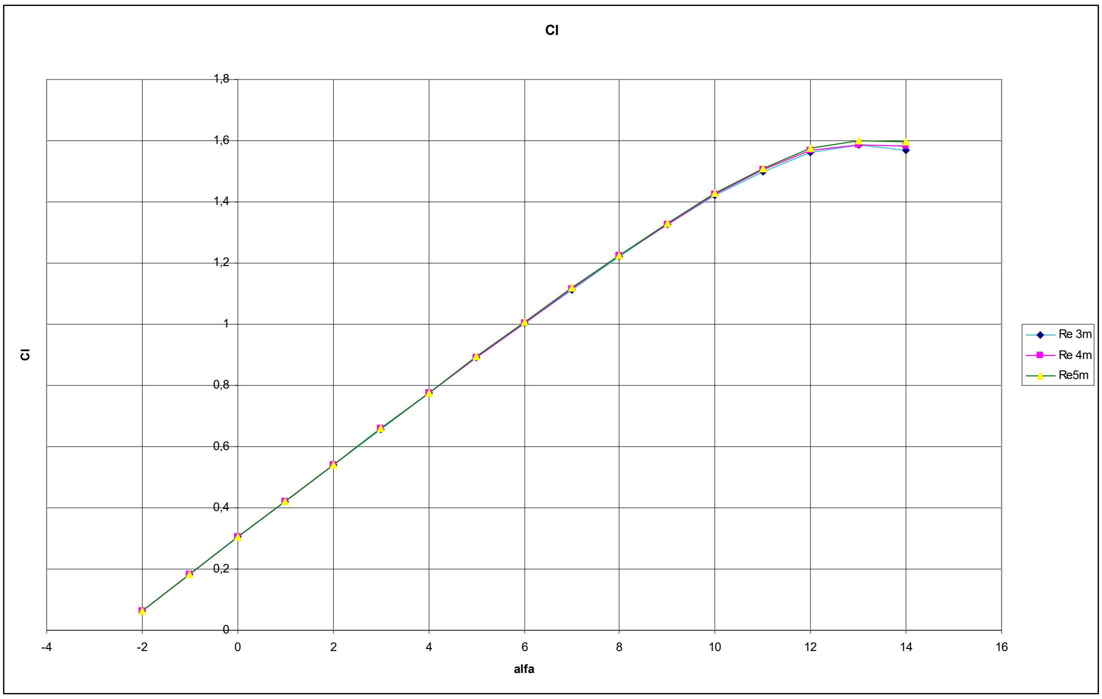
Conference paper by Selig, Michael S., Maughmer, Mark D., Somers, Dan M., “An airfoil for general aviation applications” may, 01, 1990 and Jornal of aircraft Vol 32, no 4 July-august 1995 “Natural-Laminar-Flow Airfoil for General-Aviation Applications” are used as source.

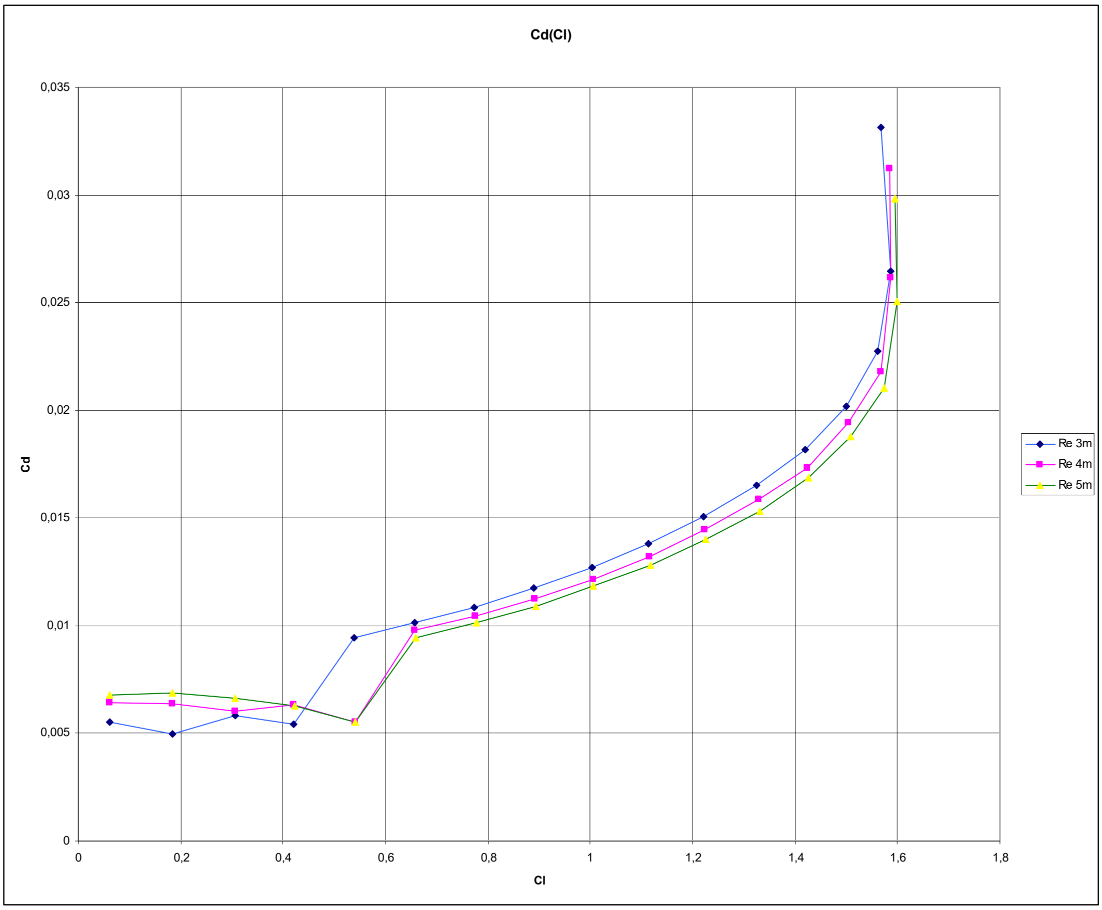
Profile is relatively thick and has large leading edge radius to aid manufacturing, while maintains low drag in cruise lift range and relatively high max lift coefficient without flaps. Also effect of surface contamination is lower than that of NACA 23015.

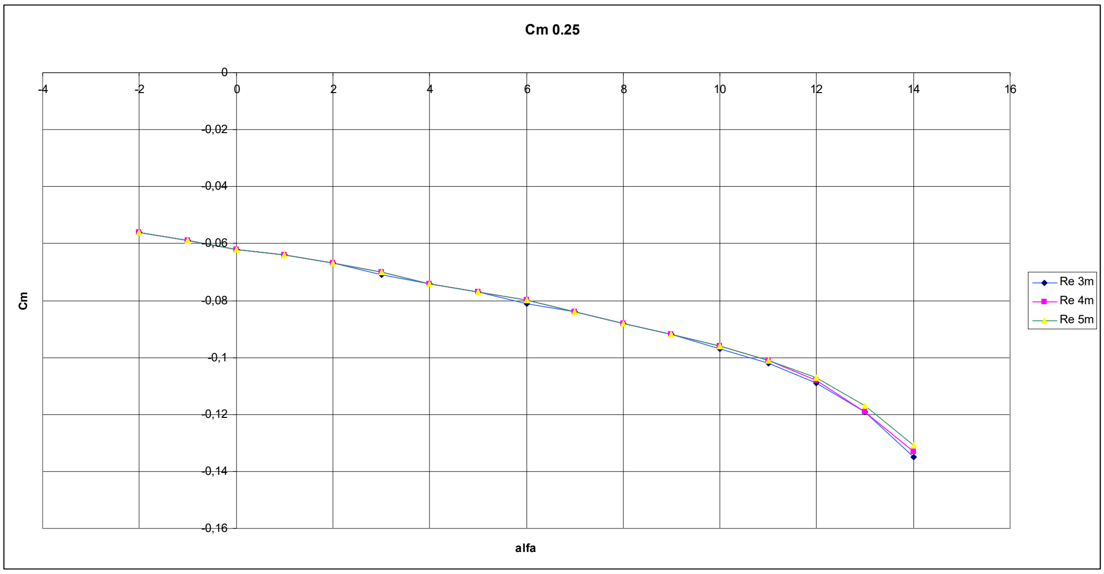
Data in previous references was used and profile lifting characteristics were calculated based on 2D profile data.

Profile data was inserted into datafiles for our calculation program to use. Following graphs are from these files.

Following data is for Reynolds number of 3E6 to 5E6 which correspond to speed of 220 km/h at sea level with wing dimensions from tip to root.



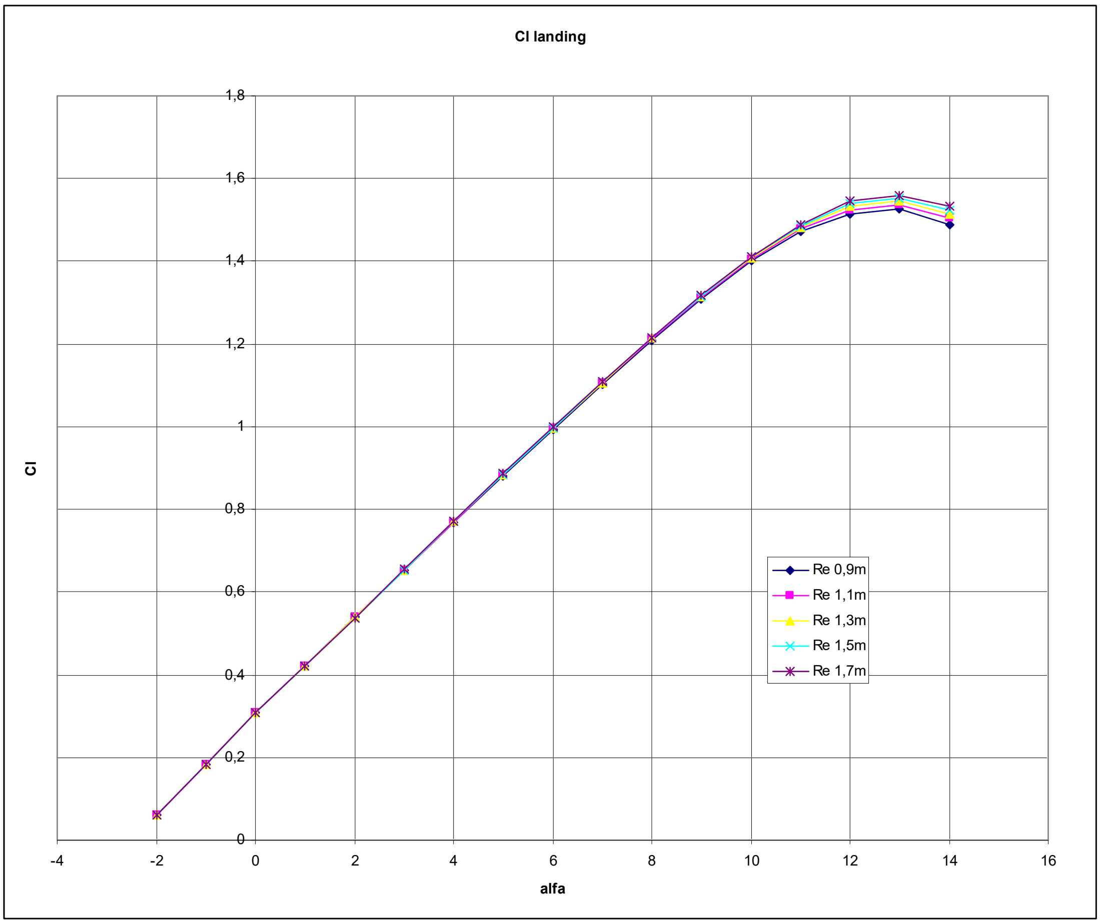




Laminar flow dip ends at about Cl 0,6, which represents air speed of 140 km/h @ 499 kg or 153 km/h @600 kg. So it is safe to say at at cruise speed aircraft wing is working in low drag angle of attack.

Landing speed Re numbers are 0,9 million (tip) to 1,8 million (root)

In a combination graph Cl curves show a typical decrease of maximum Cl when Re number decreases:



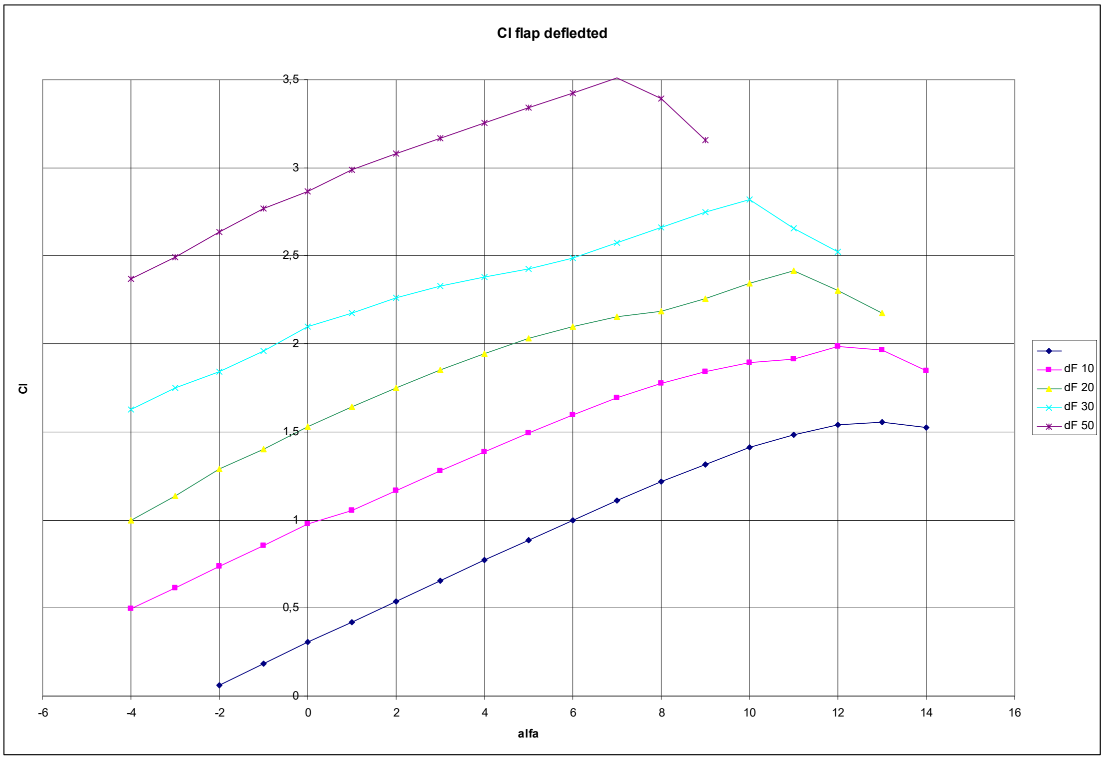
For flap deflection no wind tunnel data was available. Airfoil characteristics were estimated using calculation programs.

For verification Re 2e6 case was calculated and compared to wind tunnel data. Programs used were Martin Hepperle Javafoil and also Dr.Hanley Multisurface Aerodynamics airfoil program was use.

Results show that Javafoil Cl(alfa) is very close to windtunnel data. Hanley MSA calculates Cl(alfa) less steep. JavaFoil estimates max Cl to2 deg higher than real, Hanley has equal alfa for Cl max but CL max value is lower. As we need mainly Cl(alfa) values JavaFoil is better for this purpose. That overestimation of max Cl angle is corrected on following graphs.

Wing inner section has 20% c plain flaps with settings of 0°, 10°, 30° and 50°.

Profile Cl curve for three flap settings was calculated estimated to be following:

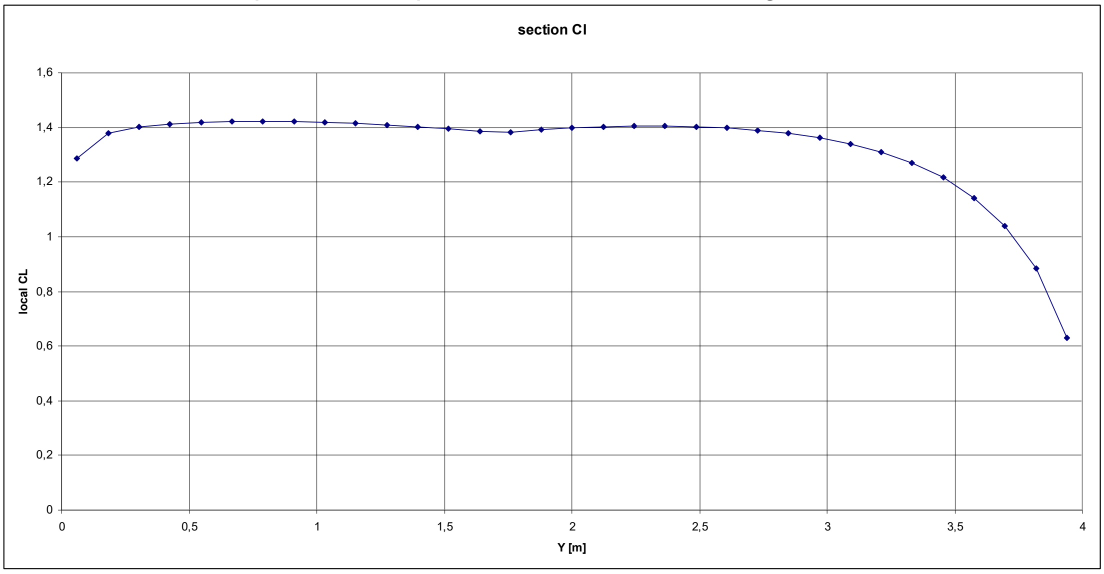


2.1.2 *Wing lifting data*

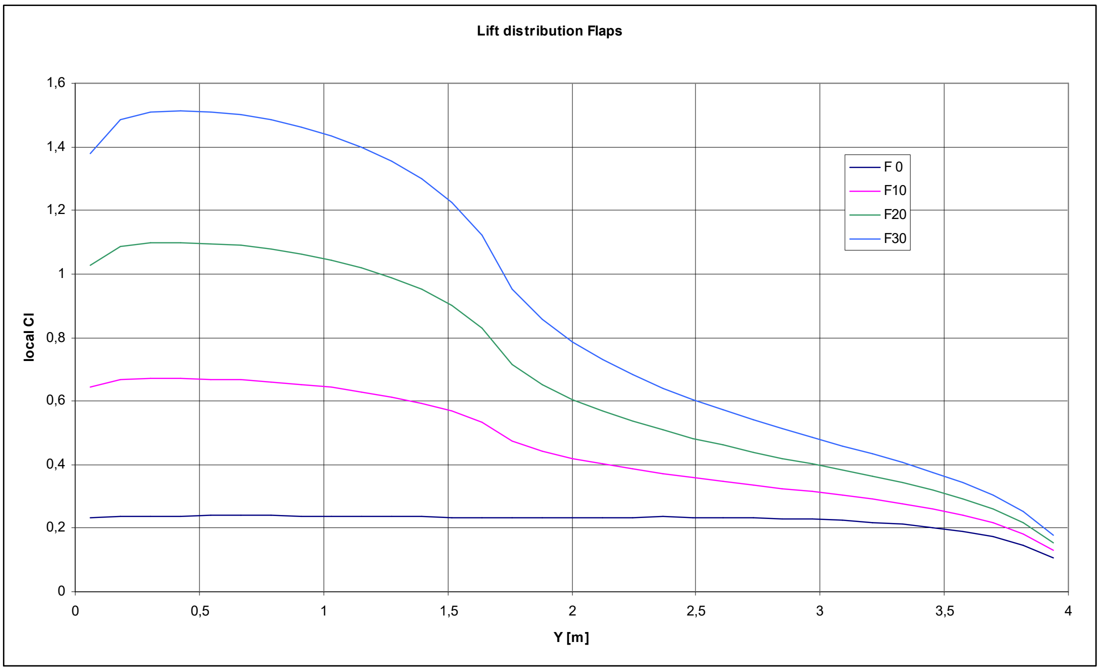
Lifting characteristics of wing are calculated using a lifting panel method computer program Hanley Multisurface aerodynamics and Desktop Aeronautics LINAIR 4. Results were very close to each other. Calculated results here are from Linair 4

Data from previous chapter is used. But flap angle of 50 deg is not used.

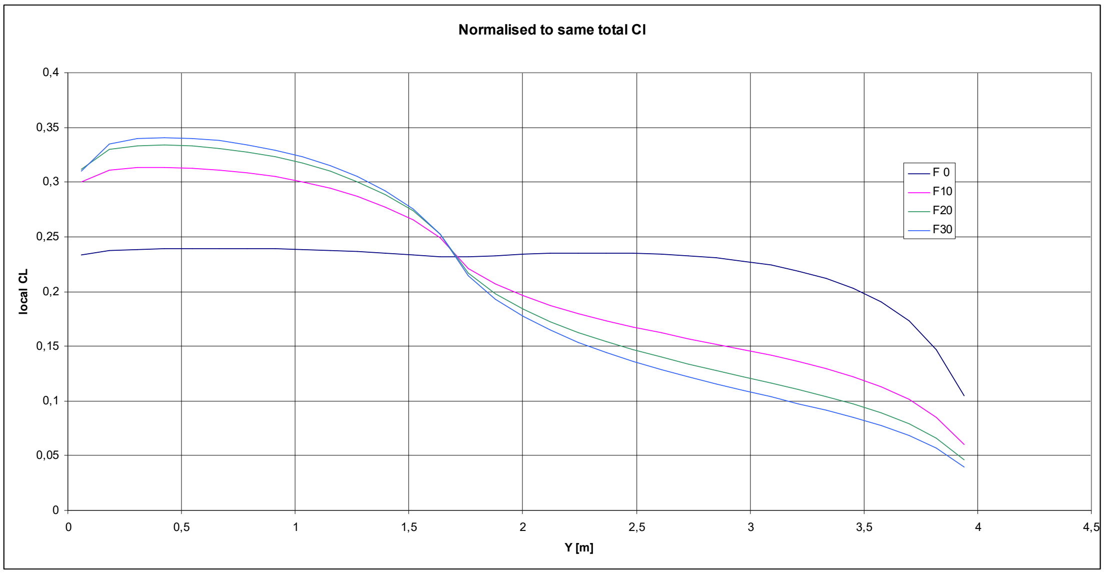
Graph for alfa 16 deg with flaps at 0 deg corresponding to Wing Cl of 1,475 which correspond to air speed of 93,8 km/h at 499 kg.



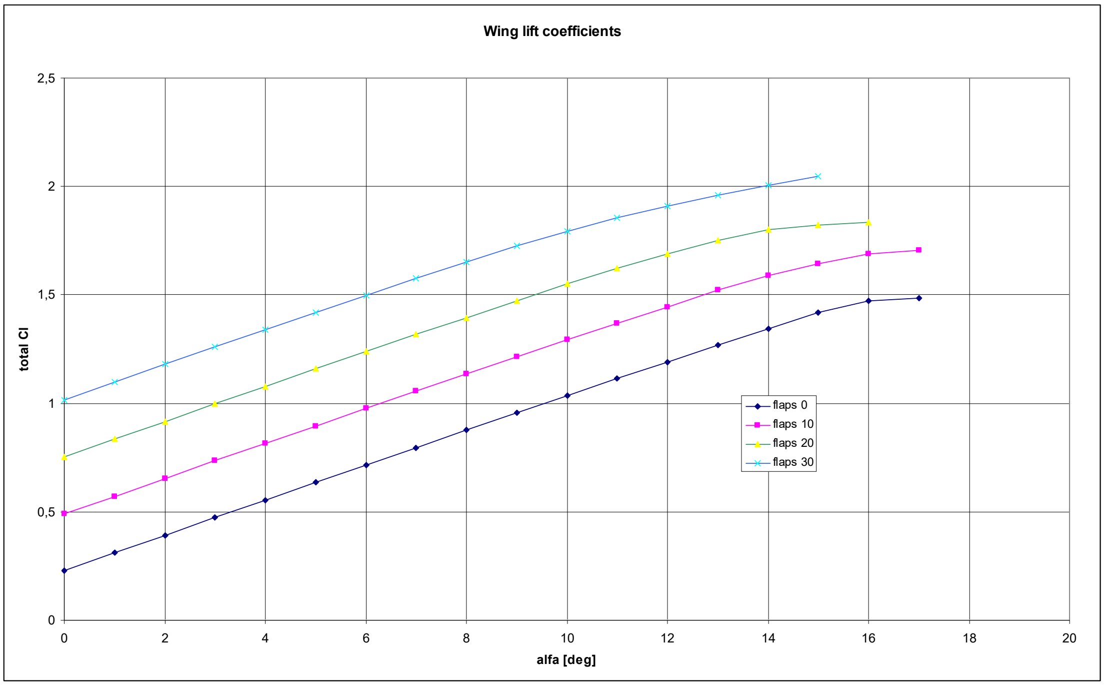
With flaps extended lift distribution is:



And these with normalised CL. So that all have same Cl of 0,2286 (same as Flaps 0 @ alfa 0 deg).



Lift curves for wing was calculated using wing-loading of 560,7 N/m². This represents flying weight of 499 kg. Difference (in distribution) in results is negligible for other weights.



Maximum lift coefficients and corresponding stall speeds @499 kg. Stall speed at forward c.g.:

flap [deg]	Cl max	Vs [m/s]	Vs [km/h]
0	1,47	25,83	93,0
10	1,70	22,94	82,6
20	1,82	22,16	79,8
30	2,00	21,12	76,0
50	2,28	19,76	71,2

Same for 600 kg mass:

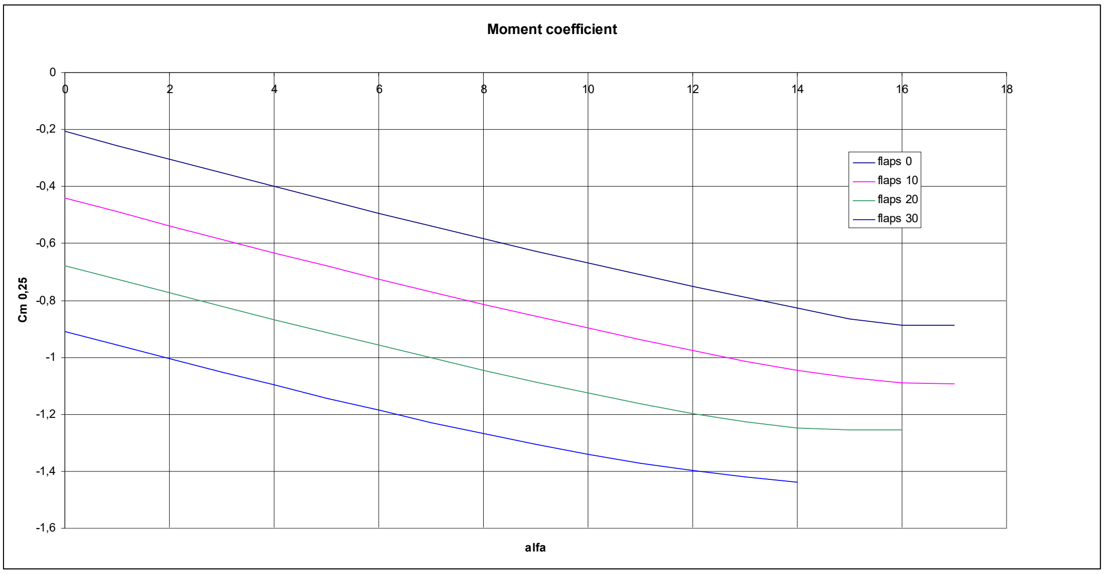
flap [deg]	Cl max	Vs [m/s]	Vs [km/h]
0	1,47	27,09	97,5
10	1,70	25,15	90,6
20	1,82	24,30	87,5
30	2,00	23,16	83,4
50	2,28	21,67	78,0

Wing lift curve slope was determined to be (linear portion):

flap 0 deg	flap 20 deg	
0,0815	0,0814	[1/deg]
4,669	4,664	[1/rad]

2.1.3 Wing moment data

Wing moment data was calculated along lift data with Linair program. Here is graphic summary of this data:



ASTM 5.2.2.4 states to use moment coefficient of at least +/- 0,025, the calculated values are more, so they are used.

2.1.4 Mean aerodynamical chord

Wing is double tapered.

Root chord at aircraft centerline: 1,306 m

Mid chord, Y=1,698 m: 1,227 m

Tip chord, Y = 3,984 m: 0,718 m

Beyond tip chord, a fairing is designed. With it total span is 8,136 m.

Root chord X position: 0,370 m

mid chord X position: 0,38918 m

tip chord X position: 0,51572 m.

Using adaptation of PDAS program form http://www.pdas.com/, used spreadsheet is available from tools section of our website.

Results:

Wing area: 8,75 m²

MAC: 1,129 m

Y mac at 1,807 m

X of mac: 0,414 m

2.2 Tail aerodynamical values

2.2.1 Radius of gyration

Radius of gyration in yaw is needed for vertical tail gust load calculations.

Values for this is estimated using statistical data method taken from “Airplane Design Part V by Jan Roskam. ISBN 1-884885-50-0 published by DARcorporation, Lawrence, Kansas” . This estimates moment of gyration in yaw as

$$I_z = 27{,}26 \text{ kg m}^2$$

So

K = 1,326 m

2.2.2 Horizontal tail

ASTM X4

Estimate of downwash factor is needed for horizontal tail gust estimates. Downwash is calculated in document P8-0110-00044S

Downwash factor is:

0,8203

Similar to wing calculations:

Tail profile NACA 0010

Horizontal tail lift curve slope was determined to be (linear portion):

Elevon 0 deg	
0,081521	[1/deg]
4,6708	[1/rad]

2.2.3 Vertical tail

ASTM X4

S_{VT} = 1,22 m²

C_{VT} = 0,91 m

L_{VT} = 2,90 m

K = 1,34 m

Similar to wing calculations:

Tail profile NACA 0010

Vertical tail lift curve slope was determined to be (linear portion):

Rudder 0 deg	
0,04769	[1/deg]
2,732621	[1/rad]

2.3 Level speed

Data from PIK-11 indicated that top level speed with Continental A65 engine is 223 km/h.

From flight manual data PIK-11 drag coefficient for wing formula is:

Cd = 0,047945 – 0,00117*v + 9,29e-06 * v^2

Cd = 0,04418 – 0,000888*v + 5,51e-06 * v^2

Pik-28 fuselage drag should be less as the form along stremline is continuous compared to stepped form of PIK11. But for conservatism same fuselage drag is used. Tail drags should be quite similar.

Using this as baseline and using same method for wing drag estimate for PIK-11 and for this Pik-28 design (which has slightly less drag) we end up with

Engine	Max power [kW]	VH [km/h]
Continental A65	48,5	237
Rotax 912 UL	59,6	256
Rotax 912 ULS	73,5	275
UL power UL260i	72,3	274
UL power UL260iS	79,8	283

2.4 Trim condition

Tail trim force is calculated from simple force moment equation. Aircraft mass is acting at center of gravity position. Wing moment coefficient is acting and mass moment is mass*g*distance from ¼ point of mac.

Tail arm (from wing mac ¼ point to tail mac ¼ point) is 3,366 m.

Flying mass 499 kg.

Cg 695 mm, n= 1

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		0,294	0,213	0,108
Wing CM		-0,0611	-0,0573	-0,05959
Moment (mass &cm)		-1142,0	-1478,2	-3030,1
Tail lift	[N]	-339,3	-439,2	-900,2

cg 695 mm, n= 4,4

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		1,293	0,937	0,475
Wing CM		-0,0919	-0,0654	-0,07863
Moment (mass &cm)		-1721,1	-1691,1	-4002,1
Tail lift		-511,3	-502,4	-1189,0

cg 695 mm, n= -2,0

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		-,588	-,426	-,216
Wing CM		-,046	-,0507	-,04877
Moment (mass &cm)		-856,4	-1304,4	-2476,3
Tail lift		254,4	387,5	735,7

Flying mass 499 kg.

Cg 807 mm, n= 1

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		0,294	0,213	0,108
Wing CM		-0,0611	-0,0573	-0,05959
Moment (mass &cm)		-593,7	-929,9	-2481,9
Tail lift	[N]	-176,4	-276,3	-737,3

cg 807 mm, n= 4,4

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		1,293	0,937	0,475
Wing CM		-0,0919	-0,0654	-0,07863
Moment (mass &cm)		691,2	721,2	-1589,7

Tail lift		205,4	214,3	-472,3

cg 807 mm, n= -2,0

		VA	VC	VD
Speed [km/h]		200	235	330
Total CL		-,588	-,426	-,216
Wing CM		-,046	-,0507	-,04877
Moment (mass &cm)		-1952,9	-2400,9	-3572,8
Tail lift		580,2	713,3	1061,4

Wing moment coefficient was calculated separately as it is not dependent on tail loads nor is it dependent of center of gravity. Moment coefficient is calculated to reference of wing MAC ¼ point, which is at 0,69525 m point.

	VC	VA	VNE	VD
CM				
Moment [Nm]				

2.5 Summary of values needed

W = 499 kg, 4895,2 N
600 kg, 5886 N
S = 8,75 m²

Estimated data.

Vs0 = 77 km/h
84 km/h
VH = 283 km/h

3 DESIGN SPEEDS

3.1 Design values general

The selected design airspeeds are calibrated airspeeds (CAS).

For manouvering limits, CS-23 utility category values for positive loads are used, allowing gentle aerobatic manouvers like barrel roll, etc. Negative limit is taken form ASTM (higher than CS23 U cat limit).

Maneuverign limits are chosen to be + 4,4 g and -2,0 g.

For landing conditions maneuvering limits are chosen to be + 2,0 g and -0,0 g.

3.2 Design maneuvering speed VA

ASTM X1.1

For minimum design maneuvering speed V_A the following applies:

$$V_{Amin}=2{,}17\times\sqrt{\frac{n_1\times W}{S}}\cdot kts$$

V_{Amin} = 2,17 * sqr(4,4*499*g/8,76) = 107,6 kts = 199,3 km/h

V_{Amin} = 2,17 * sqr(4,4*600*g/8,76) = 117,7 kts = 217,9 km/h

We select our design maneuvering speed VA 218 km/h

3.3 Design flap speed in landing configuration VF

ASTM X1.1

For design flap speed in landing configuration V_F following applies:

$$V_{Fmin}=1.59\times\sqrt{\frac{n_1\times W}{S}}\cdot kts$$

V_{Fmin} = 1,59 * sqr(4,4*499*g/8,76) = 78,9 kts = 146,1 km/h

V_{Fmin} = 1,59 * sqr(4,4*600*g/8,76) = 86,2 kts = 159,7 km/h

We select our design flap speed VF 160 km/h

3.4 Design Cruise Speed VC

ASTM X1.1

$$V_{Cmin}=2{,}46\times\sqrt{\frac{n_1\times W}{S}}\cdot kts$$

V_{Cmin} = 2,46 * sqr(4,4*499*g/8,76) = 122,0 kts = 226,0 km/h

V_{Cmin} = 2,46 * sqr(4,4*600*g/8,76) = 133,4 kts = 247,0 km/h

V_H is estimated to be 283 km/h.

ASTM X1.2.5.2 defines that V_C need not be more that 0.9 V_H (=0,9 * 283 km/h = 254,7 km/h).

V_{Cmin} = 247,0 km/h = 133,4 kts.

We select our design cruising speed to be VC 255 km/h.

3.5 Design dive speed V_D

ASTM X1.1

$$V_{Dmin} = 3{,}47 \times \sqrt{\frac{n_1 \times W}{S}} \cdot kts$$

V_{Dmin} = 3,47 * sqr(4,4*499 g/8,76) = 172,1 kts = 318,8 km/h

V_{Dmin} = 3,47 * sqr(4,4*600 g/8,76) = 188,1 kts = 348,4 km/h

But need not to exceed

$$1{,}4 \times V_{Cmin} \times \sqrt{\frac{n_1 \times W}{S}} \quad \text{kts}$$

V_{Cmin} = 226,0 km/h = 122,0 kts, so the value V_{Dmin} need not exceed is

V_{Dmin} = 1,4 * 122,0 * sqr(4,4*499 g/8,75) = 340,4 km/h = 183,8 kts

for 600 kg;

V_{Cmin} = 247,0 km/h = 133,4 kts, so the value V_{Dmin} need not exceed is

V_{Dmin} = 1,4 * 133,4 * sqr(4,4*600 g/8,75) = 372,1 km/h = 200,9 kts

We select our design dive speed to be VD 360 km/h.

3.6 High lift devices

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 V_F, must have limit manoeuvring load factor for this condition. These limits must be determined.

Limit manoeuvring load factor with flaps extended is selected to be +2,0 g.

3.7 Never exceed speed V_{NE}

ASTM 4.1.1.2

V_{NE} must be less than or equal to $0.9V_{DF}$ and greater than or equal to $1.1V_C$. In addition, V_{NE} must be greater than or equal to V_H.

V_{DF} may be less than or equal to V_D (ASTM 4.1.1.1).

Lower limits for V_{NE} are:

V_H = 283 km/h (lower limit)

1,1 V_C = 235 km/h*1,1 = 258,5 km/h (lower limit)

0.9 V_{DF} <= 0,9 VD = 297 km/h (upper limit)

For 600 kg lower limits for V_{NE} are:

V_H = 283 km/h (lower limit)

1,1 V_C = 255,0 km/h*1,1 = 280,5 km/h (lower limit)

0.9 V_{DF} <= 0,9 VD = 324 km/h (upper limit)

So VNE must be between 283 and 324 km/h

We select our never exceed speed to be VNE 310 km/h.

4 LOAD FACTORS

4.1 Design limit flight load factors

ASTM table X1.1

Values of table X1.1 are used. But see 3.1.

The positive limit manoeuvring load factor n_1 is selected to be 4,4.

The negative limit manoeuvring load factor n_2 is defined to be -2,0.

For manoeuvring flaps down limit load factor is defined to be + 2,0 g, negative load flap open is 0,0 g.

Table X1.2:

Factor K is V_{Csel}/V_{Cmin} = 235/226 = 1,040

n_1*W/S = 51,4 lb/sqft

so: n_3 = 4,4

Table X1.3

Factor K is V_{Csel}/V_{Cmin} = 235/226 = 1,040

n_1*W/S = 51,4 lb/sqft

so: n_4 = -2,0

For 600 kg;

Table X1.2:

Factor K is V_{Csel}/V_{Cmin} = 255/247 = 1,032

n_1*W/S = 61,4 lb/sqft

so: n_3 = 4,4

Table X1.3

Factor K is V_{Csel}/V_{Cmin} = 255/247 = 1,032

n_1*W/S = 61,4 lb/sqft

so: n_4 = -2,2

Notation of X1.1:

n_1	4,4
n_2	-2,0
n_3	4,4
n_4	-2,2
n_f	2,0
n_{f-}	0,0

4.2 Gust load factors (wing)

ASTM X3

The gust load factors may be computed as follows:

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

or

$$n = 1 - \frac{\frac{1}{2} \times \rho_0 \times V \times a \times K_g \times U_{de}}{\frac{Mg}{S}}$$

where;

Kg = $\frac{0{,}88\mu_g}{5{,}3+\mu_g}$ = gust alleviation factor;

μ_g = $\frac{2\times(\frac{M}{S})}{\rho\times\bar{C}\times a}$ = aeroplane mass ratio;

Ude = derived gust velocities referred to (m/s) 15 or 7,5 m/s;

ρ_0 = density of air at sea level (1,225 kg/m³);

M/S = wing loading (kg/m²); 499/8,76 = 57,0 kg/m²

$\bar{C}$ = mean geometric chord (m); 1,117 m

g = acceleration due to gravity (9,81 m/s²);

V = aeroplane equivalent speed (m/s); and

a = slope of the aeroplane normal force coefficient curve C_{NA} per radian. Value of 4,669 $^{1}/_{rad}$ is calculated for our wing at 0 deg flap setting.

Note: Formula for gust alleviation factor, see appendix 1 of this report.

With these values we get:

aeroplane mass ratio = 17,8354
gust alleviation factor = 0,6784

And with gusts of 15 m/s at Vc and 7,5 m/s at Vd we get;

Vc	n+	4,40	g
Vc	n -	-2,40	g
Vd	n+	3,39	g
Vd	n -	-1,39	g

For 600 kg

With these values we get:

aeroplane mass ratio = 21,1125
gust alleviation factor = 0,70342

And with gusts of 15 m/s at Vc and 7,5 m/s at Vd we get;

Vc	n+	4,23	g
Vc	n -	-2,23	g
Vd	n+	3,28	g
Vd	n -	-1,28	g

4.3 Engine mount

ASTM 5.10.1.2, 5.2.9

Engine emergency landing condition load factors are:

Up n = 3

Forward n = 10

Lateral n = 1,5

From engine torque conditions;

Limit takeoff torque and power simultaneously with 75% of n1. For calculations this is calculated at VY speed. Engine produces during takeoff an estimated of 1500 N of thrust at climb speed.

Limit continuous torque and power simultaneously with 100% of n1. For calculations this is calculated at VA speed. Engine produces during takeoff estimated of 870 N of thrust at VA speed.

5 V-N DIAGRAM

5.1 Combined

Combining previous paragraphs we get aircraft's V-n diagram.

5.1.1 Gust loads

For 499 kg

Selected speed are:

V s0	77	km/h	1 g
V s	90	km/h	1 g
V a	200	km/h	4,4 g
V c	235	km/h	4,4 g
V ne	297	km/h	4.4 g
V d	330	km/h	
V f	147	km/h	2,0 g

Calculated gust envelope is:

	v [km/h]	n
V c	235	4,40
V d	330	3,39
V d	330	-1,39
V c	235	-2,40

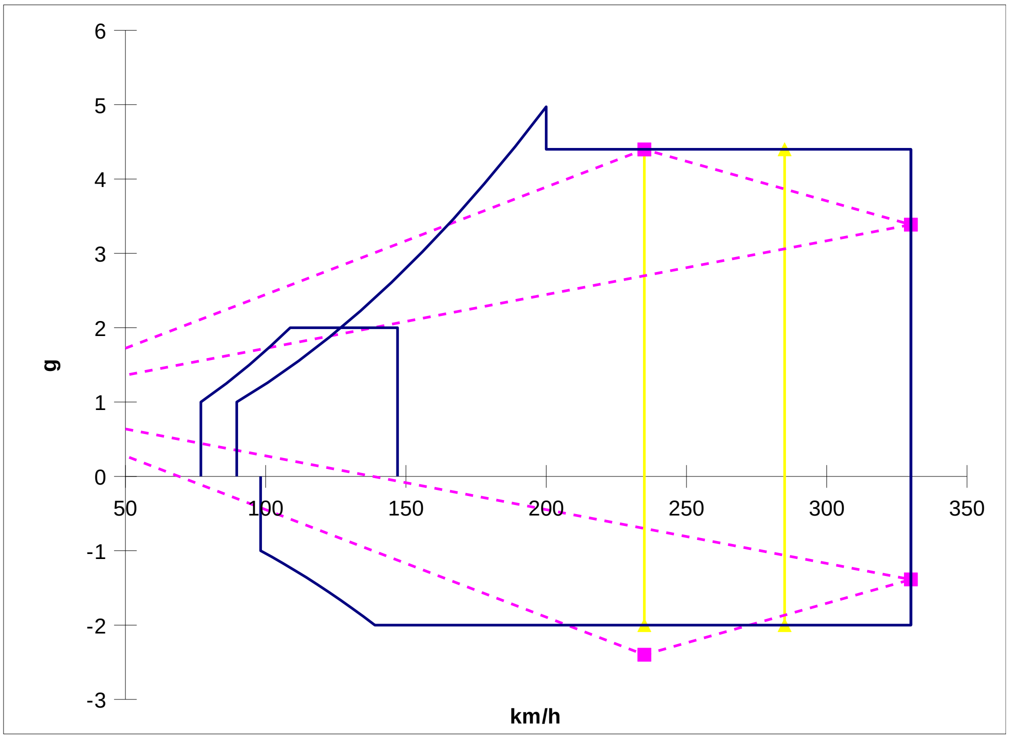
V-n diagram is:

Gust envelope is presented with dashed lines.

Points of this envelope, and corresponding lift coefficients are:

	km/h	g	Cl
A	200	4,4	1,140
C	235	4,4	0,9709
D	330	4,4	0,6914
VD	330	0	0
H	139	-2,0	-1,1073
F	235	-2,0	-0,6546
E	330	-2,0	-0,4661

These Cl values represent maximum take-off weight.



5.1.2 *600 kg*

For 600 kg

Selected speed are:

V s0	78	km/h	1,0 g
V s	98	km/h	1,0 g
V a	218	km/h	4,4 g
V c	255	km/h	4,4 g
V ne	310	km/h	4,4 g
V d	360	km/h	4,4 g
V f	160	km/h	2,0 g

Calculated gust envelope is:

	v [km/h]	n
V c	255	4,23
V d	360	3,28
V d	360	-1,28
V c	255	-2,23

V-n diagram is:

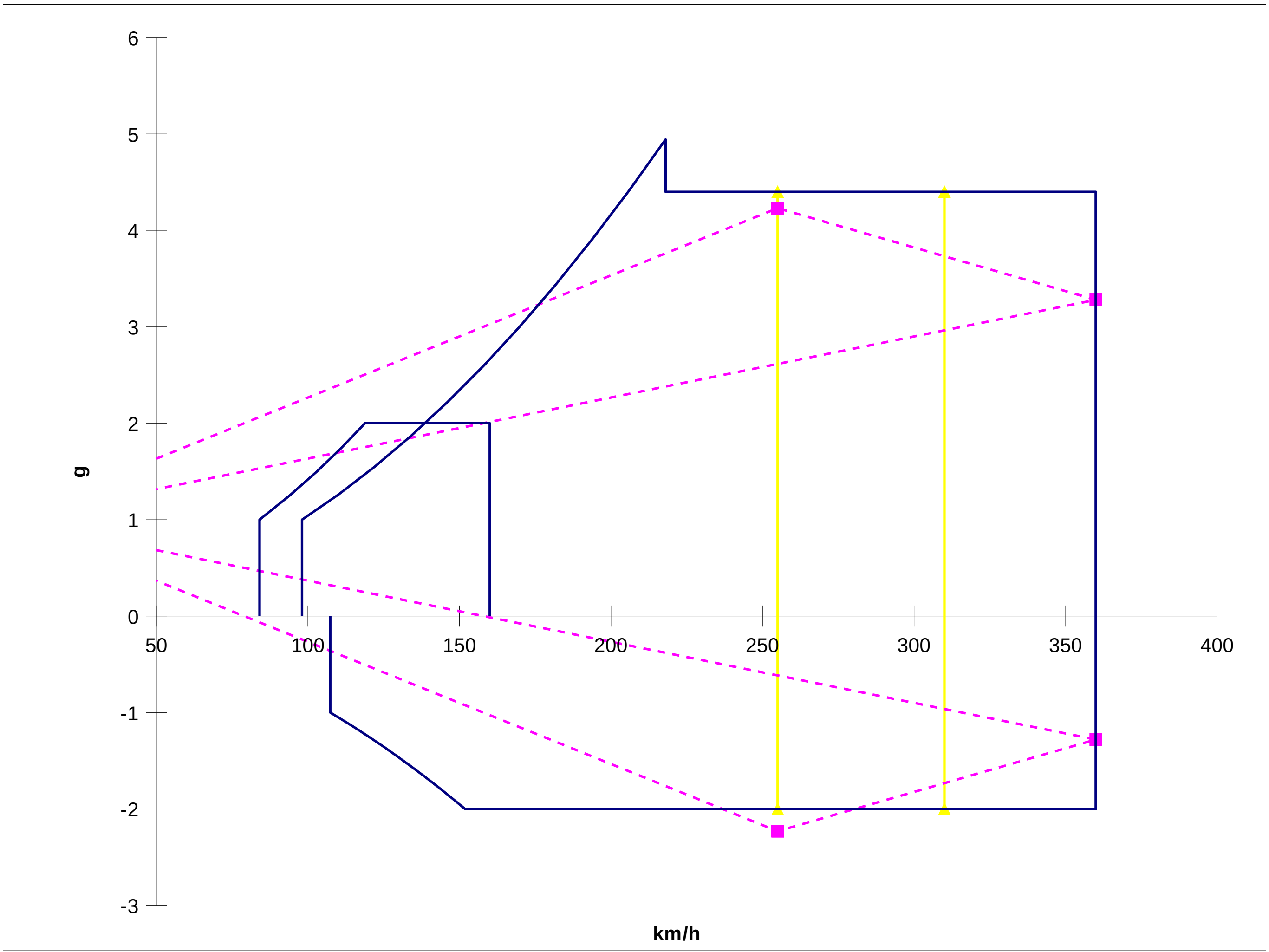
Gust envelope is presented with dashed lines.

Points of this envelope, and corresponding lift coefficients are:

	km/h	g	Cl
A	218	4,4	1,309
C	255	4,4	0,957
D	360	4,4	0,480
VD	360	0	0

H	160	-2,0	-1,104
F	255	-2,0	-0,435
E	360	-2,0	-0,218

These Cl values represent maximum take-off weight.



5.1.3 *Load cases*

Load case numbering for calculation is

Load case	km/h	g	
1	218	4,4	VA
2	255	4,4	VC +gust
3	360	4,4	VD

5.2 Horizontal tail

5.2.1 Gust loads

ASTM X4

Dimensions and other values see, 2.2.2.

From X4.1

ΔL_{HT} = 2117,6 N at V_C

ΔL_{HT} = 1486,8 N at V_D

At trim condition, cg at forward limit, max mass, tail loads are:

VC (235 km/h), g = 1

CL total: 0,213

Tail lift = -439,2 N (down)

At VD, (330 km/h) g=1

CL total: 0,108

Tail lift = -900,2 N (down)

So gust conditions are:

VC

Up gust

Tail load = trimmed force + ΔL_{HT}

Tail load = -439,2 +2117,6 N = 1678,4 N

Down gust

Tail load = trimmed force - ΔL_{HT}

Tail load = -439,2 – 2117,6 N = -2556,8 N

VD

Up gust

Tail load = trimmed force + ΔL_{HT}

Tail load = -900,2 + 1486,8 N = 586,6 N

Down gust

Tail load = trimmed force - ΔL_{HT}

Tail load = -900,2 – 1486,8 N = -2387,0 N

####

5.2.2 Manouvering loads

ASTM X1.4.3

For horizontal tail with span of 2,56 m and chord of 0,711 m.

Design maneuvering wing loading:

n1*W/S [N/m²]

= 4,4 * 4895,19 / 8,76

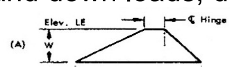
= 2460 N/m²

From figure X1.4;

Average surface loading, w [N/m²]

w = 1545 N/m²

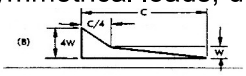
For up and down loads, distribution is Table X1.2 (A).



Using this distribution (dimensions from Specification 4.5), total symmetrical load is:

2557 N

For unsymmetrical loads, distribution is Table X1.2 (B).



Using this distribution (dimensions from Specification 4.5), total unsymmetrical load is:

On one side 1409 N

On other side 916 N

5.3 Vertical tail

5.3.1 Gust loads

ASTM X4

Dimensions and other values see, 2.2.3.

From X4.2

ΔL_{VT} = 1613 N at V_C

ΔL_{VT} = 1132 N at V_D

5.4 Calculation load cases

LC1
VC down gust
Horizontal tail
tail load -2556,8 N (down) total

LC2
VC up gust
Horizontal tail
tail load 1678,4 N (Up) total

LC3
unsymmetrical manouvering load VA
Horizontal tail
On one side 1409 N
On other side 916 N

LC4
gust load vertical tail VC
1613 N

6 OTHER LOADS

6.1 Engine mount

Rotax OM for 912ULS, 5.10.1

Engine mount load are defined in ASTM 5.2.9.

Engine (including propeller) installed mass is 90 kg, acting from engine center of gravity. Mount weights 10 kg.

Engine power and torque

Takeoff	73,5 kW
5800 rpm	
Max continuous	69 kW
5500 rpm	
Gearbox ratio	1:2,43
Number of cylinders	4

Max takeoff power case yields calculation moment of 596,1 Nm.

Max continuous power case yield moment of 582,2 Nm.

Limit takeoff torque and power simultaneously with 75% of n1. For calculations this is calculated at VY speed. Engine produces during takeoff estimated of 1500 N of thrust at climb speed. As this can achieved in pull-out pitch-up rate is 0,608 rad/s. Propeller (+ engine crankshaft) moment is 25,6 Nm. Limit continuous torque and power simultaneously with 100% of n1. For calculations this is calculated at VA speed. Engine produces estimated of 870 N of thrust at VA speed.

As this can achieved in pull-out pitch-up rate is 0,602 rad/s. Propeller (+ engine crankshaft) moment is 23,3 Nm.

Emergency landing cases (ASTM 5.10). Emergency landing case load factors are:

n = 3 up,

n = 10 for engines and ESD(s)) forward, and

n = 1.5 lateral.

Max power case

down	2647,8	N
torque X	588,1	Nm
torque Z	43,8	Nm
forwards	1500	N

Max continuous case

down	3530,4	N
torque X	582,2	Nm
torque Z	23,3	Nm
forwards	870	N

Emergency landing case 1

forwards	8826	N

Emergency landing case 2

Up	2648	N

Emergency landing case 3

sidewards	1324	N

6.2 Wheel landing gear

ASTM 5.8

Required drop height is

$$dropHeight[cm]=1{,}32\times\sqrt{\frac{W}{S}}$$ = 28,8 cm

7 STABILITY

As a baseline longitudinal stability is calculated using book ‘Piero Morelli; Static stability and Control of sailplanes, 1976’ as reference.

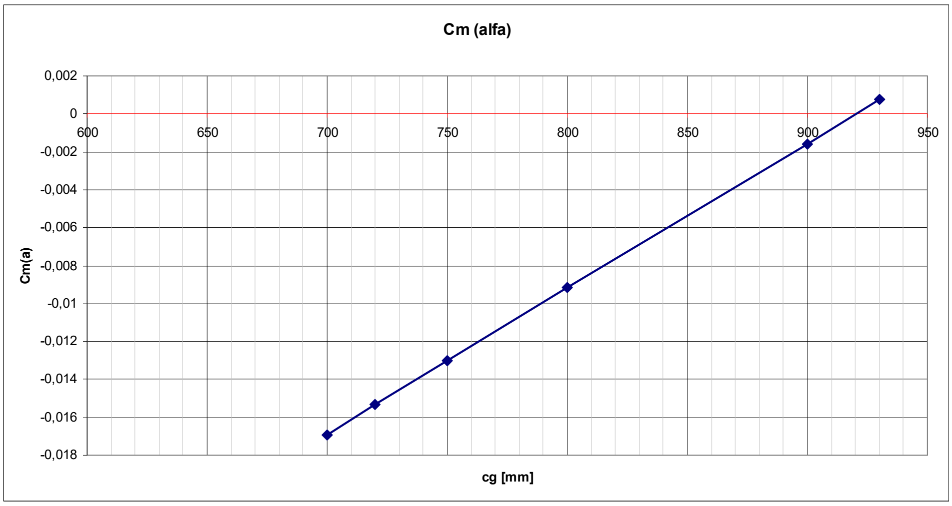
Following values for different limits were calculated

Pull up, Clmax n=1	-1 %	forward limits
Pull up, Clmax n=1, ground effect	7 %	
Pull up, Va, n max	-17 %	
Stick free neutral point	39,3 %	rear limits
dP/dn=-1(kg/g) pull, SL	43,8 %	
stick free manouvring point, high altitude	41,8 %	
stick fixed neutral point	52,4 %	
dP/dn=-0,5(kg/g) pull, SL	45,1 %	
stick fixed manouvring point, at high altitude	53,9 %	
stick free manouvring point, SL	45,0 %	
stick fixed manouvring point, SL	57,3 %	

Preliminary c.g limits are taken as 25% - 35% MAC, meaning 695 mm to 807 mm from datum.

Using Linair 4 program with model during trim condition calculation neutral point was calculated. Calculation result is stick fixed neutral poin. Linair 4 is a multisurface program taking into effect wing/tail interactions.

Flying mass used is 499 kg. Calculation is made at speed of 235 km/h (Cl 0,463).



This yields neutral point of 922 mm (45,2% MAC). Which is in close proximity of analytical results (above). Analytical result for neutral point is 5% aft of the stick free neutral point value.

Front limit will be limited by taxing/takeoff/landing characteristics, which are hard to estimate on paper. Also the front limit is where real life loading is hardest to do. So it will be estimated with prototype, what cg is feasible to reach in real life.

Preliminary center of gravity limits are:

forward limit 25 % mac equalling 695 mm from datum

rear limit 35% mac equalling 807 mm from datum.

---- END ----

Appendix 1, A hidden mistake in CS-LSA / ASTM F2245

The formula which are used to calculate gust load factors.
The calculation order how formula operations are performed is different in rules of different origin.

In ASTM F2245/16c it is found in X 4.1
Gust alleviation factor is given as [0.88 ug/5.3 +ug]

In CS-VLA 341 this same factor is given as [0.88 ug / (5.3+ug)]
Same as in CS.23.341 [0.88 ug / (5.3+ug)]

In FAR-23 this is found as paragraph 23.341. And in the current electrical form it is given as:
Kg=0.88μg/5.3+μg=gust alleviation factor;

but in historical FAR-23 achieve it is different!
Amdt. 23-34, Eff. 02/17/87
Kg = 0.88 ug / (5.3 +ug) = gust alleviation factor;
Amdt. 23-42, Eff. 02/04/91
Kg = 0.88 ug / (5.3 +ug) = gust alleviation factor;
Amdt. 23-48, Eff. 03/11/96
K g=0.88μg/5.3+μg=gust alleviation factor;

So something happened in 1996 for FAR 23 when the layout of rule was changed.
This same formula was then (probably) transferred to ASTM F2245.

Taking a real life value for mass ratio ug e.g. 12.17
ASTM/FAR formula gives Kg = 14.2
EASA CS and older FAR formula gives Kg = 0.613

Difference is large and that ASTM/FAR formula yields gust load factor which is unrealistically high.
In this design that ASTM/FAR formula gives gust load factors of about 55 g (fifty five) which is pretty high!
That older formula (and CS-VLA) gives gust load factors of 3.33 g, which is about what to expect.