EXAMPLE MICROLIGHT AIRCRAFT LOADING CALCULATIONS

1. Introduction

This example loads report is intended to be read in conjunction with BCAR Section S and CS-VLA both of which can be downloaded from the LAA webpage, and the excellent book 'Light Aircraft Design' by Hiscocks which is available from the LAA shop. The loads report is in imperial units to be consistent with Hiscocks' book.

This loads report is for an imaginary microlight, a conventional monoplane, with the following specifications:

Max gross weight	*450 Kg (992 Lbs)*		
Engine:	*Rotax 912S*		
Wing span	*30 feet*	*Wing shear centre*	*30% chord*
Fuselage width	*3 feet*	*CG of individual wing panels*	*55% chord*
Wing area	*126 sq feet*	*Tailplane area (including elevators)*	*28 sq ft*
Wing root chord	*4.4 feet*	*Fin area (including rudder)*	*13 sq ft*
Wing tip chord	*4.0 feet*	*Wing aerofoil section:*	*23012*
Aileron span	*6 feet*	*Aileron deflection*	*+/- 30 degrees*
Flap span	*7.5 feet*	*Flap deflection*	*35 degrees*

From Section S para 331, prescribed minimum load factors of flight envelope as follows:

n1	+4g
n2	+4g
n3	-1.5g
n4	-2g

Assume CLmax flaps up = 1.35
(this value is chosen because it coincides with the assumptions of CS-VLA Appendix A, which we will use later)

Vs1 = Stall speed flaps up = $\sqrt{}$ (391 x W) / (S x CLmax) =$\sqrt{}$ (391 x 992 / 126 x 1.35)
= 47.8 mph = 47.8 / 1.15 = 41.5 kts

Therefore VA = Vs1 x $\sqrt{n1}$ = 41.1 x $\sqrt{4}$ = 83 kts

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Section S para S335 requires that Vc = max level speed at full power.

CS-VLA Appendix A calls for

VC min Minimum design cruise speed = $7.69 \sqrt{(n1 \times W/S)} = 95$ kts

As we wish to use CS-VLA Appendix A later we will use the CS-VLA Appendix A figure for Vc, providing experience suggests this is not less than the max speed which will be obtained at full throttle. We will assume Vc = Vcmin = 95 kts.

Section S para S335 calls for VDmin = 1.4 Vc 1.4 x 95 = 133 kts

CS-VLA Appendix A calls for
VD min Minimum design dive speed = 10.86 root (n1 x W/S) = 134.7 kts

134.7 ≥ 133 kts therefore as we wish to use CS-VLA Appendix A later we will use the CS-VLA Appendix A figure rounded up to 135 kts which exceeds the VDmin stated in Section S.

According to Section S para S1505, based on VD of 135 kts, Vne should be not more than 0.9 x 135 = 121 kts.

Although the designer can call for a lower value of Vne if he wishes, it cannot be reduced to less than 0.9 x VDF (Section S para S1505)

2. Wing loads

Assuming a 5% additional factor for tail load, max gross weight 992 Lbs we get a maximum wing lift at +4g at point A and D of the flight envelope of:

1.05 x n1 x W = 1.05 x 4 x 992 = 4166 Lbs (limit) at points A and D of the flight envelope.

At point G of the envelope, -2g, wing lift is (-2 / 4) x 4166 Lbs = -4166 / 2 = - 2083 Lbs

At point E of the envelope, -1.5g, wing lift is (-1.5 / 4) x 4166 Lbs = 1562 Lbs.

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2.1 Spanwise wing lift distribution

Rather than carry out a full Shrenk analysis we will use the option described in BCAR Section S AMC S337 of using a spanwise load per inch proportional to the local chord.

Wing span = 30 feet
Fuselage width = 3 feet
Span of one wing panel = (30 – 3) / 2 feet = 13.5 feet
Wing lift per wing panel at +4g = 4166 x (13.5/ 30) _= 4166 x 0.45 = 1875 Lbs
Average wing lift per inch of span = 1875 / (13.5 x 12) = 10.34 Lbs/inch
Root chord (including aileron) = 4.4 feet
Tip chord (including aileron) = 4.0 feet

Average wing chord = (4.4 + 4.0) / 2 = 4.2 feet. This is essentially the same as the MAC with a wing with little taper such as this.

Wing lift per inch at root = (4.4 / 4.2) x 10.34 = 10.83 Lbs/inch of span

Wing lift per inch at tip = (4.0 / 4.2) x 10.34 = 9.85 Lbs/inch of span

2.2 Inertia Relief

The wings are simple ladder-type frames of approximately uniform weight per inch of span from root to tip. Total weight of each wing panel, including ailerons = 45 Lbs

Therefore weight of panel per inch of span is 45 / (13.5 x 12) = 0.28 Lbs per inch

At +4g, inertia relief on wing is 4 x 0.28 = 1.12 Lbs per inch of span
Therefore at +4g the load due to lift is reduced by inertia relief to
10.83 – 1.12 = 9.71 Lbs/inch at the root, falling linearly to
9.85 – 1.12 = 8.73 Lbs/inch at the tip.

At -2g, inertia relief on the wing is 2 x 0.28 = 0.56 Lbs per inch. Therefore at -2g the load due to lift is reduced by inertia relief to (10.83 x 2 / 4) – 0.56 = 4.85 Lbs per inch at root
reducing linearly to (9.85 x 2 / 4) -0.56 = 4.36 Lbs/inch at the tip.

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At -1.5g, inertia relief on the wing is 1.5 x 0.28 = 0.42 Lbs per inch. Therefore at -1.5g the load due to lift is reduced by inertia relief to (10.83 x 1.5 / 4) – 0.42 = 3.64 Lbs/inch at root reducing linearly to (9.85 x 1.5 / 4) - 0.42 = 3.27 Lbs/inch at the tip.

2.3 Chordwise load distribution

We could simply assume a wing centre of pressure travel of 20% to 60% chord per BCAR Section S AMC 337 under positive g, and 0% to 25 % chord centre of pressure under negative g.

Alternatively we can calculate the centre of pressure travel as follows:

At point A on the flight envelope: Wing lift W = 4166 Lbs
Airspeed V = VA = 83 kts = 95 mph
Wing area S = 126 square feet
Dynamic pressure q = V^2 / 391 = 24.0 Lbs/sq ft

Therefore wing lift coefficient = W / qS = 4166 / 24.0 x 126
= 1.38
Moment coefficient Cm = -0.025
(Section S para S331 doesn't permit lesser values of pitching moment coefficient than –0.025 even when aerofoil data suggest lesser values should apply, as with the 23012 section)

Position of centre of pressure = 100 x (0.25 – (Cm/CL))
(as percentage chord) = 100 x (0.25 - (-0.025 / 1.38))
= 26.8 % chord

At point D on the flight envelope: Wing lift W = 4166 Lbs
Airspeed V = VD = 135 kts = 155mph
Wing area S = 126 square feet
Dynamic pressure q = 155^2 / 391 = 61.4 Lbs/sq ft

Therefore wing lift coefficient = W / qS = 4166 / 61.4 x 126
= 0.54
Moment coefficient Cm = -0.025

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Percent chord position of centre of pressure = 100 x (0.25 – (Cm/CL))

= 100 x (0.25 - (-0.025 / 0.54)
= 29.6 % chord

At point G on the flight envelope: Wing lift W = - 2083 Lbs
We will assume airspeed V = VA = 83 kts = 95 mph
Wing area S = 126 square feet
Dynamic pressure q = V^2 / 391 = 24.0 Lbs/sq ft

Therefore wing lift coefficient = W / qS = -2083 / (24.0 x 126) = -0.69

Moment coefficient Cm = -0.025

Percent chord position of centre of pressure = 100 x (0.25 – (Cm/CL))
= 100 x (0.25 - (-0.025 / - 0.69))

= 21.3% chord

At point E on the flight envelope: Wing lift W = -1562 Lbs
Airspeed V = VD = 135 kts = 155mph
Wing area S = 126 square feet
Dynamic pressure q = 155^2 / 391 = 61.4 Lbs / sq ft

Therefore wing lift coefficient = W / qS = -1562 / 61.4 x 126
= -0.20

Moment coefficient Cm = -0.025

Percent chord position of centre of pressure = 100 x (0.25 – (Cm/CL))

= 100 x (0.25 - (-0.025 /-0.20))
= 12.5 % chord

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2.4 Anti-drag loads

In the absence of aerofoil data to derive the anti-drag loads it will be acceptable to mount the wing with a 14 degree nose-down angle between the mean wing chord line and the horizontal when carrying out the forward centre of pressure tests (points A and G), which will simulate a wing anti-drag of 25% of the lift, which is likely to be conservative.

2.5 Wing Load Testing

If the wing is to be proven by load testing, the dead weight of the test wing at 1 g can also be subtracted from the loads calculated in 2.2 above when calculating the weight of sand bags (test load) that must be applied in the tests.

Hence limit test load at +4g = 9.71 - 0.28 = 9.43 Lbs/inch at the root, falling to
8.73 – 0.28 = 8.45 Lbs/inch at the tip.

Ultimate test load at +4g = (9.71 x 1.5) – 0.28 = 14.25 Lbs/inch at root, falling to
(8.73 x 1.5) -0.28 = 12.81 Lbs/inch at the tip

Note that the test load (ie weight of sand bags) at ultimate is not simply 1.5 times the test load at limit, because an additional test load of half the wing dead weight has to be included.

Limit test load at -2g = 4.85 - 0.28 = 4.57 Lbs/inch at root
falling to 4.36 – 0.28 = 4.08 Lbs/inch at the tip.

Ultimate test load at -2g = (4.85 x 1.5) - 0.28 = 7.00 Lbs/inch at root
falling to (4.36 x 1.5) – 0.28 = 6.26 Lbs per inch at tip.

Limit test load at -1.5g = 3.64 - 0.28 = 3.36 Lbs/inch at root
falling to 3.27 – 0.28 = 2.99 Lbs/inch at the tip.

Ultimate test load at -1.5g = (3.64 x 1.5) - 0.28 = 5.18 Lbs/inch at root
falling to (3.27 x 1.5) – 0.28 = 4.62 Lbs/inch at tip.

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These results are summarised in the table below

	Limit Root Lbs/inch	Ultimate Root Lbs/inch	Limit Tip Lbs/inch	Ultimate Tip Lbs/inch	Test angle	Centre of pressure
A	9.43	14.25	8.45	12.81	14 deg	26.8%
D	9.43	14.25	8.45	12.81	0 deg	29.6%
G	4.57	7.00	4.08	6.26	14 deg	21.3 %
E	3.36	5.18	2.99	4.62	0 deg	12.5 %

To produce a loading plan, divide the span of the wing into a convenient number of strips of even width and then calculate the amount of load to apply to each strip.

Dividing the 13.5 foot long wing panel into ten strips, the width of each strip is
13.5 / 10 = 1.35 feet wide (16.2 inches).

The load per inch can be expressed as a function of the spanwise measurement from the wing root, z (in feet), as follows:

Load per inch at 'z' feet from root

= load per inch at root – z (load per inch at root - load per inch at tip) / span of panel (in feet)

= 9.43 – z (9.43 – 8.45)/ 13.5 Lbs per inch

= 9.43 - 0.0726 z Lbs per inch

Taking the 4th strip outboard from the root, for example:

Distance from root to centre of fourth strip (z) = (4 - 0.5) x 1.35 feet

= 3.5 x 1.35 feet = 4.725 feet

Average load per inch in the 4th strip is 9.43 – (0.0726 x 4.725) Lbs/inch

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= 9.086 Lbs/inch

Load in fourth strip is therefore

Average load per inch x width of strip in inches = 9.086 x 16.2 = 147.2 Lbs

In this way we can derive the load in each strip, as below:

Strip	Z (in feet)	Average Load Lbs/inch in strip	Load in strip (Lbs)
1	0.675	9.381	152.0
2	2.025	9.283	150.4
3	3.375	9.184	148.8
4	4.725	9.086	147.2
5	6.075	8.989	145.6
6	7.425	8.891	144.0
7	8.775	8.793	142.4
8	10.125	8.694	140.9
9	11.475	8.597	139.3
10	12.825	8.499	137.7

Total 1448.3 Lbs

As check, compare total with 13.5 x 12 x (9.43 + 8.45) / 2 = 1448.3, hence OK.

2.6 Flap Down case

Section S para S345 is unusual in requiring the flaps to be able to be deflected while manoeuvring at up to n1 ie up to +4g. This seems a very demanding case. Also the flap limiting speed in Section S para S335 is no less than twice the flaps-down stall speed or 1.4 times the flaps up stall speed.

The effect of lowering the flap is

1. to introduce an additional nose-down pitching moment and

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2. to transfer a proportion of the wing lift inboard to that portion influenced by the flap. The inboard transfer of lift is unlikely to be critical in a cantilever wing but might be in a strut-braced high wing where the beam column strength of the wing spars may be critical between the root and the strut attachment.

Vs1 = 42 kts (from above)

Vs0 = 35 kts = 40 mph This is max allowable flaps-down stall speed for a microlight defined by BCAR Section S para S2.

This would require a CLmax with flap of: CLmax = (391 x 992) / (126 x $40^{2)}$ = 1.92 *(this is a bit high for comfort, but we will carry on!)*

1.4 x VS1 = 1.4 x 42 kts = 59 kts   2.0 x VS0 = 2 x 35 kts = 70 kts

Therefore we make VF the greater of the two which is VF = 70 kts

Wing pitching moment at VF = 70 kts

q = $(70 \times 1.15)^2$ / 391 = 16.6 Lbs/sq ft

Pitching moment of wing with undeflected flap = Cmo x q x panel area x mean chord

= -0.025 x 16.6 x (13.5 x 4.2) x 4.2 = 98.8 ft Lbs

Wing chord at outboard end of flap = 4.4 – ((7.5/13.5) x 0.4) = 4.18 ft

Area of wing affected by one flap = 7.5 x (4.4 + 4.18) / 2 = 32.17 sq ft
Average chord of this portion = (4.4 + 4.18 / 2) = 4.29 ft
Moment coefficient with 35 degrees of flap Cm = -0.025 + (35 x -0.01)
= -0.025 - 0.35 = -0.375

Area of wing panel outboard of flap = (13.5 x 4.2) – 32.17 = 24.53 sq ft
Average chord of this portion = (4.0 + 4.18) / 2 = 4.09 ft

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Pitching moment of outer portion of wing = Cmo x q x outboard area x outer mean chord

= -0.025 x 16.6 x 24.53 x 4.09 ft Lbs
= 41.6 ft Lbs

Pitching moment of inboard portion of wing = Cm x q x inboard area x inner mean chord

= -0.375 x 16.6 x 32.17 x 4.29 = -859 ft Lbs

Total pitching moment at wing root = -859 + -41.6 ft Lbs
= -900.6 ft Lbs (limit)

If flap chord = 15% of wing chord, Cf/ C = 0.15. from fig 2.8(b) of Hiscocks, with 35 degree flap chord, flap effectiveness K = 0.28.

Hence change in zero lift angle for flapped portion of wing = flap deflection angle x K
= 35 x 0.28 = 10 degrees.

Zero lift angle of 23012 section, from fig 2.4.2in Hiscocks is -1.3 degrees. Therefore zero lift angle for flapped portion of wing = -1.3 – 10 = -11.3 degrees.

Assuming a lift curve slope for both portions of the wing of 0.05, (fig 3.2(b) in Hiscocks)

(11.3 – i) x 0.05 x 32.17 x q = (i – 1.3) x 0.05 x 24.53 x q

11.3 – i = (i – 1.3) x 24.53 / 32.17 = (i – 1.3) x 0.762

11.3 = i + 0.762 i – (0.762 x 1.3) = i + 0.762i – 0.99 = (1 + 0.762) i - 0.99

i = (11.3 +0 .99) / (1 + 0.762) = 12.29 / 1.762 = 7.0 degrees

Therefore individual angles of attack measured with respect to zero lift line are:

Inboard portion 11.3 – 7 = 4.3 degrees for the flapped portion

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Outboard portion 1.3 – 7 = -5.7 degrees

Basic lift over flapped portion is alpha x lift curve slope x q x flapped area

= 4.3 x 0.05 x 16.6 x 32.17 = 116 Lbs

Basic lift over outboard section = -5.7 x 0.05 x 16.6 x 24.53 = - 116 Lbs

With flap up, lift on inboard section at point A = (32.17 / 126) x 4166 Lbs = 1064 Lbs
With flap up, lift on outboard section at point A = (24.53 / 126) x 4166 Lbs = 811 Lbs

With flap at 35 degrees, lift on inboard section of wing = 1064 + 116 Lbs = 1180 Lbs

Lift coefficient in inboard portion = L/q S = 1180 / (16.6 x 32.17) = 2.21

Centre of pressure position in inboard section = 100 x (0.25 - Cm/CL)
= 100 x (0.25 –(-0.375 / 2.21)) = 50.0%

With flap at 35 degrees, lift on outboard section of wing = 811 – 116 Lbs = 695 Lbs

Lift coefficient in outboard section = L/qS = 695 / (16.6 x 24.53) = 1.71
Centre of pressure position in outboard section = 100 x (0.25 – (Cm/CL))
= 100 (0.25 –(-0.025 /1.71)) = 26.5%

This is well within the Section S default Cp travel of 20-60 percent chord.

(Note that at 1.71 and 2.2, the lift coefficients in the outboard and inboard portions of the wing have turned out unrealistically high for a simple flap wing, this is another indicator that the aeroplane has not got enough flap area or wing area to achieve the required 35 knot stall speed at this weight, ie our original assumed CL max with flap of 1.92 is too high. If this was a real design, the next step would be to go through the process again with a bigger wing or longer flaps. A better starting point would be to size the wing based on a flapped max lift coefficient not more than 1.7 or 1.8)

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Particularly with a cantilever or single-strut braced wing, it is also necessary to check a condition with full flap at VF and zero g. This is because the shear centre of the wing is typically behind the ¼ chord position therefore the structural torque (ie torque about the shear centre) is alleviated by the nose-up moment due to wing lift under positive g.

Therefore in the example above, at point A with full flap:

Wing panel pitching moment = -900.6 ft Lbs
Wing panel lift = 1875 Lbs
If the shear centre is at 30% chord, offset between shear centre and ¼ chord is
30 – 25 =5% mean chord, which is 0.05 x 4.2 = 0.21 feet

Therefore structural torque = -900.6 + (1875 x 0.21) ft Lbs = -900.6 + 394
= -506.6 ft Lbs

By comparison, at VA and zero g, L= 0 therefore lift moment goes to zero, hence
Structural torque = wing pitching moment = -900 ft Lbs

The chordwise inertia relief is commonly ignored on microlight aircraft which is generally conservative as the wing's chordwise cg is usually aft of the shear centre therefore the effect of wing inertia is to reduce the nose-down wing structural torque under positive g. However, if the cg of the wing panel is forward of the shear centre, (eg with full leading edge fuel tanks), the incremental torque due to wing inertia will be in a nose-down sense under positive g and hence increase the total torque, in which case the wing panel inertia must be taken into account in the torque calculations.

In the example microlight, wing panel weight = 45 lbs. Centre of gravity of wing panel (including flaps and ailerons) is at 55% chord. Average wing panel chord = 4.2 feet therefore offset between shear centre and centre of gravity of wing panel is

(0.55 – 0.30) x 4.2 feet = 0.25 x 4.2 = 1.05 feet

Structural torque at point A, including inertia relief, = -506.6 + (1.05 x (45 x 4)) ft Lbs

= -506.6 + 189 = -317.6 ft Lbs (limit)

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Structural torque at VA, zero g, including inertia relief, = -900.6 + (1.05 x (45 x 0)) ft Lbs

= -900.6 + 0 = -900.6 ft Lbs

Note that in this example the structural torque with full flap at zero g is more than twice that at point A, showing how important it is to include this case.

To test the 'zero g ' case by load testing would be impossible without some kind of whiffle tree arrangement to apply torque without lift load. Normally this case would be covered by applying a simulated lift load with a centre of pressure as far rearward as practical, with sufficient (load x offset) to give the worst-case torque, accepting that the vertical shear and bending introduced by virtue of the test method will make the test, if anything, conservative. In the above example, if the test load were applied uniformly spanwise but in the inboard portion of the wing only, centred chordwise at an arbitrary 2 feet aft of the shear centre of the wing, a test load of 900.6 / 2 = 450 Lbs (limit) would be required. Spread over the 7.5 foot long inboard portion this would need a load of

450 / (7.5 x 12) = 5.0 Lbs / inch (limit)
and 5.0 x 1.5 = 7.5 Lbs/inch (ultimate)

This would produce representative torsion in the inboard portion of the wing. If the wing is strut-braced however, it would also be sensible to apply an arbitrary balancing load to the outboard part of the wing, distributed along the shear centre, not to influence torque but to balance the shear inboard of the strut attachment, and hence avoid the inboard wing and its attachments possibly being overloaded in shear or bending. In this case from the above, the outer wing symmetric case at point A calls for a loading of between 8 and 9 Lbs / inch (limit) therefore it would be satisfactory to continue the uniform load of 5.0 Lbs / inch (limit) and 7.5 Lbs / inch (ultimate) to the tip during the max torque tests, to act as a balancing load about the wing strut attachments and 'balance the see saw'. The balancing load outboard of the flap would need to be centred chordwise on the shear centre of the wing so as to prevent the balancing load influencing the structural torque.

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2.7 Asymmetric loads

The asymmetric wing cases called for by BCAR Section S paras S349, S455 and S337 are:

a. Maximum aileron deflection at VA, n = 2 / 3 x n1 = 2 /3 x 4 = 2.66g

b. 1/3 maximum aileron deflection at VD, n = 2 / 3 x n1 = 2.66g

The effect of deflecting the ailerons is:

1. To modify the individual wing torques

2. To modify the spanwise lift distribution which causes a rolling moment which is reacted mainly by wing rolling inertia. The effect on wing shear and bending is not usually critical however as the manoeuvre load factor is only 2/3 n1.

3. To introduce a residual rolling moment into the fuselage. The rolling moment due to aileron deflection can be calculated per Hiscocks methods alternatively CS-VLA Appendix A calls for an asymmetric wing lift case given by 100% of the 'point A' wing loading on one wing panel combined with only 70% of that load on the other panel. In the absence of calculated rolling moments this may be an acceptable alternative.

Wing Pitching moments:

Full aileron at VA case – aileron down:

VF = 70 kts q = 16.6 Lbs / sq ft Cmo = -0.025

Max aileron deflection = 30 degrees therefore

Cm aileron down = -0.025 +(30 x -0.01) = -0.025 + -0.30 = -0.325

Pitching moment from outer wing = Cm x q x area outer wing x mean chord outer wing

= -0.325 x 16.6 x 24.53 x 4.09 ft Lbs = -541 ft Lbs

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Pitching moment from inboard wing = Cm x q x area of inner wing x mean chord of inner wing
= -0.025 x 16.6 x 32.17 x 4.29 = -57.3 ft Lbs

Total pitching moment at root, aileron down at VA = -541 – 57.3 = - 598.3 ft Lbs (limit)

Full aileron up at VA

Cm aileron up = -0.025 + (-30 x -0.01) = +0.275

Pitching moment from outer wing = CM x q x area outer wing x mean chord outer wing
= + 0.275 x 16.6 x 24.53 x 4.09 ft Lbs = + 458.0 ft Lbs

Pitching moment from inner wing = Cm x q x area inner wing x mean chord inner wing
= -0.025 x 16.6 x 32.17 x 4.29 = -57.3 ft Lbs

Total pitching moment at root, aileron up at VA = +458.0 – 57.3 = +400.7 ft Lbs (limit)

1/3 aileron down at VD

VD = 135 kts = 155mph
Dynamic pressure q = 155^2 / 391 = 61.4 Lbs / sq ft
Max aileron deflection = 30 degrees therefore 1/3 aileron = 10 degrees deflection

Cm aileron down = -0.025 +(10 x -0.01) = -0.025 + -0.10 = -0.125

Pitching moment from outer wing = Cm x q x area outer wing x mean chord outer wing
= -0.125 x 61.4 x 24.53 x 4.09 ft Lbs = - 770 ft Lbs
Pitching moment from inner wing = Cm x q x area inner wing x mean chord inner wing
= -0.025 x 61.4 x 32.17 x 4.29 = -211.8 ft Lbs

Therefore total torque at root, aileron 1/3 down at VD = -211.8 + –770
= - 981.8 ft Lbs (limit)

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1/3 aileron up at VD

Cm aileron up = -0.025 + (-10 x -0.01) = +0.075

Pitching moment from outer wing = Cm x q x area outer wing x mean chord outer wing
= + 0.075 x 61.4 x 24.53 x 4.09 ft Lbs = + 462 ft Lbs

Pitching moment inner wing = Cm x q x area of inner wing x mean chord inner wing
= -0.025 x 61.4 x 32.17 x 4.29 = -211.8 ft Lbs

Total pitching moment at root, aileron 1/3 up at VD = +462 + -211.8 ft Lbs

= +250.2 ft Lbs (limit)

2.8 Load tests

A minimum of four symmetrical wing tests will be required for the corner points of the manoeuvring envelope, A, D, G and E .The spanwise loading for points A and D can be use per the table above. The loading tables for points G and E on the flight envelope can be derived from first principles as above or more quickly calculated from the already-calculated loads for points A and D on the flight envelope by adding the 'dead weight' element, multiplying by the ratio of the load factors and then subtracting the dead weight element again. For example for the fourth spanwise strip at point E:

n3 / n1 = 1.5 / 4 = 0.375
'dead weight' element on each strip = weight of wing panel / number of strips = 45/10 = 4.5 Lbs

Strip 4 Test load for 'point A' test = 147.2 Lbs (from table above)

Therefore Strip 4 test load for 'point E' test = ((147.2 + 4.5) x 0.375) – 4.5 = 52.4 Lbs

Each test to be taken to limit load initially, then unloaded and inspected to check for permanent deformation before taking to ultimate to check for failure.

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In addition, depending on the type of structure and magnitude of the loads calculated as above it may be necessary to test the worst case asymmetric wing conditions (aileron cases above) and flap cases as above.

3. Fin loads

S441 and S351 call for fin loads to be accommodated with prescribed rudder deflections. Appendix A of CS-VLA provides an acceptable means of calculating the fin loads.

We will use the method of CS-VLA Appendix A to derive fin, tail, aileron and flap loads. Note however these methods can only be used with conventional configurations and prescribed aspect ratios of the surfaces.

Manoeuvring wing loading n1 x W/S = 4 x 992 / 126 = 31.5 Lbs/sq ft
= 153.8 Kg/ sq m

From CS-VLA Appendix A:

VA min = Minimum design manoeuvring speed = 6.79√ (n1 x W/S)
= 6.79 √ 153.8 = 83 kts

Vc min Minimum design cruise speed = 7.69 √ (n1 x W/S) = 95 kts

These values are consistent with the Va and Vc selected previously.

With (n1 W/S) = 31.5 Lbs/sq ft, and Vselected = Vmin for Vc and VA (see notes in table 2), figure A4) produces a limit fin sideload of 20 Lbs/sq ft of fin area (including the area of the rudder). Therefore ultimate fin side load is 20 x 1.5 = 30 Lbs / sq ft foot of combined fin and rudder area.

Combined fin and rudder area = 13 square feet

Therefore limit fin load = 13 x 20 = 260 Lbs Ultimate fin load = 260 x 1.5 = 390 Lbs

EXAMPLE MICROLIGHT AIRCRAFT LOADING CALCULATIONS

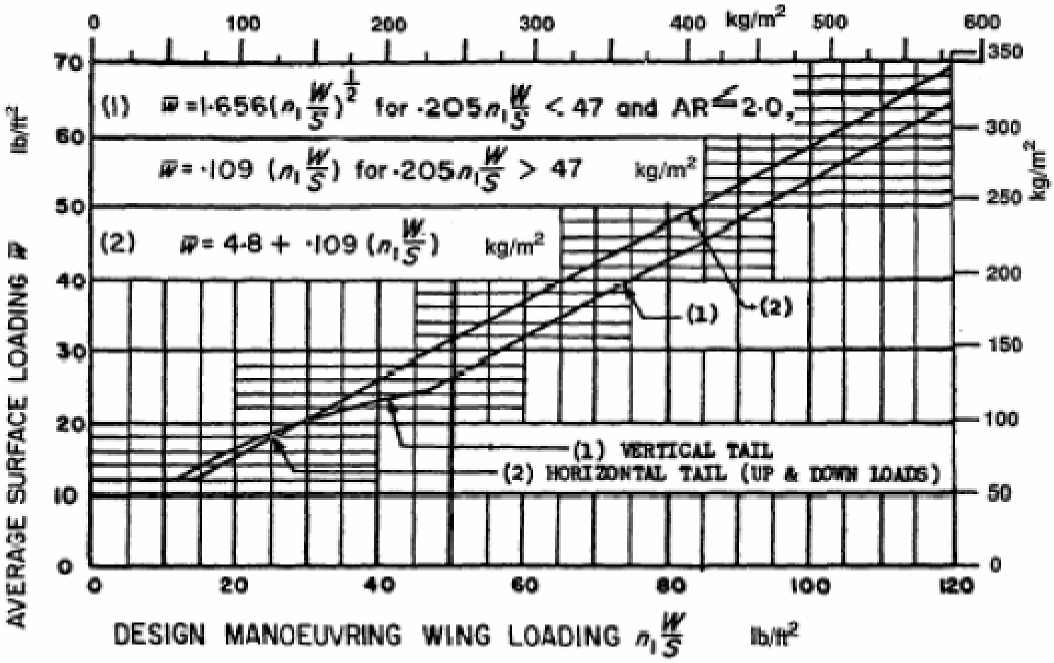
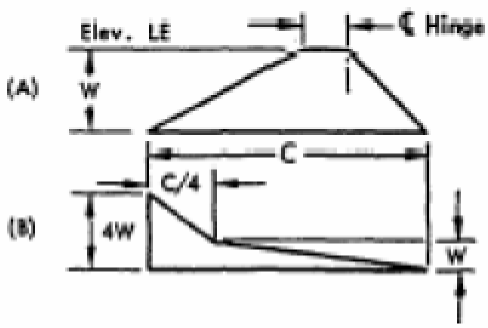


FIGURE A4
AVERAGE LIMIT CONTROL SURFACE LOADING.

Note – the equations on figure A4 are incorrect in CS-VLA, so just use the graphical presentation.

Two tests are required, one to check the manoeuvre case with a chordwise distribution per diagram A of Table A2 in CS-VLA, the other to test the gust case with chordwise distribution per diagram B of Table 2 in Appendix A of CS-VLA. The spanwise load distribution to be proportional to the local fin chord. For each test, limit and ultimate loads must be checked separately.



If the fin structure and fuselage structure are symmetrical it may not be necessary to test both port and starboard loads. In that case, the load direction chosen should be such as to create the worst case conditions in the vicinity of the rudder hinge and rudder pushrod attachment.

If the fin structure is not symmetrical then it will be necessary to check both port and starboard load cases separately ie four tests rather than two tests, or it may be possible to assume which will be the weaker load orientation and just test for that load direction.

4. Tailplane loads

BCAR Section S paragraphs S421 and S423 calls for tail balancing and manoeuvring loads to be accommodated. Appendix A of CS-VLA provides an acceptable means of calculating the tail loads. With n1 W/S = 31.5 Lbs/sq ft and Vselected = Vmin for Vc and VA (see notes in table 2 of Appendix A), figure A4 produces a limit tailplane up and download of 21Lbs/sq ft of tailplane area (including the area of the elevators). Therefore ultimate tailplane up and down load is 21 x 1.5 = 31.5 Lbs/sq ft of combined tailplane and elevator area. *(it is only a coincidence the this is the same numerical value as(n1 W/S) above).*

Tailplane area = 28 square feet (including elevators) therefore

Limit tailplane load = 28 x 21 = 588 Lbs
Ultimate tailplane load = 28 x 31.5 = 882 Lbs
Two tests are required, one to check the manoeuvre case with a chordwise distribution per diagram A of Table A2 in CS-VLA, the other to test the gust case with chordwise distribution per diagram B of Table 2 in Appendix A of CS-VLA. The spanwise load distribution to be proportional to the local tailplane chord. For each test, limit and ultimate loads must be checked separately.

4.1 Asymmetric tail load

According to BCAR Section S para S427 an asymmetric tail load case must be tested involving 100 % of the maximum tailplane load on one tailplane half and 70% on the other, ie in this case 588 / 2 = 294 Lbs (441 Lbs ultimate) on one tailplane half and

EXAMPLE MICROLIGHT AIRCRAFT LOADING CALCULATIONS

294 x 0.7 = 206 Lbs limit (309 Lbs ultimate) on the other. This test should be carried out with the manoeuvre case chordwise distribution per diagram A of table 2.

(CS-VLA Appendix A calls for 65% / 100 % rather than 70%/ 100 % and if using the Appendix A method of deriving the tail loads, it would be appropriate to use the more stringent CS-VLA asymmetric case too, and so avoid the risk of being accused of 'cherry picking' the less demanding requirements from the two codes. In this case the asymmetric cases become 294 Lbs and 294 x 0.65 = 191 Lbs (limit), 441 Lbs and 441 x 0.65 = 286Lbs (ultimate).

5. <u>Flap loads</u>

Section S does not prescribe specific flap strength requirements other than S345 which calls for manoeuvring up to +4g at up to VF with full flap. Using the CS-VLA Appendix A flap load estimate:

VFmin Minimum design flap limiting speed = $4.98 \sqrt{(n1 \times W/S)}$ = $4.98 \sqrt{153.8}$
= 62 kts

Section S calls for VFmin no less than the greater of 2 x Vs0 = 70 kts
or 1.4 x Vs1 = 1.4 x 41.5 = 58 kts

We will have to apply the greater of the Section S figures ie VF = 70 kts

Therefore VFmin Section S / VFmin CS-VLA = 70 / 62 = 1.13

With n1 W/S = 31.5 Lbs/sq ft, and VFselected = 1.13 VFmin (see notes in table 2 of Appendix A), figure A5 produces a limit flap upward load of 1.13^2 x 18 lbs/sq ft of flap area = 23 Lb/sq ft Therefore ultimate flap upward load is 23 x 1.5 = 34.5 Lbs / sq ft of flap area.

EXAMPLE MICROLIGHT AIRCRAFT LOADING CALCULATIONS

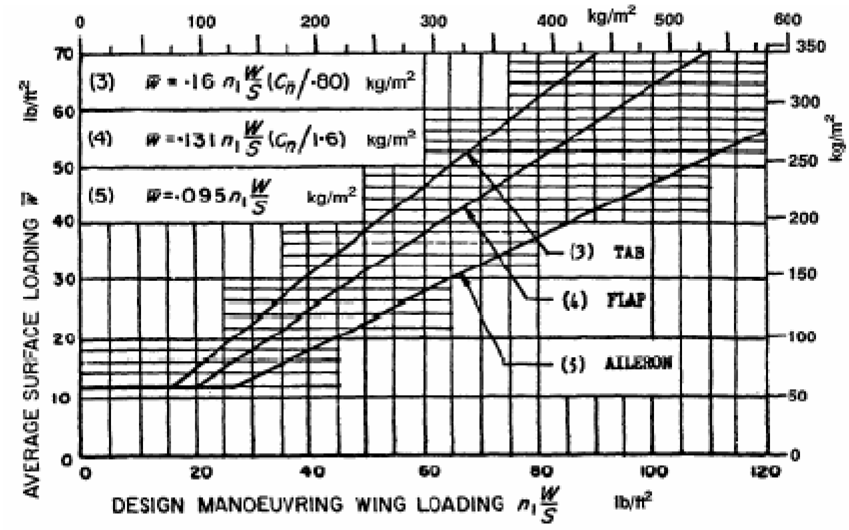


FIGURE A5
AVERAGE LIMIT CONTROL SURFACE LOADING.

Flap downward load per Table 2 of Appendix A is half the upward load figure ie

23 / 2 = 11.5 Lbs / sq ft limit, 34.5 / 2 = 17.2 Lbs / sq ft ultimate.

Two tests are required, one to check the upward load with a chordwise distribution per diagram D of Table 2 in Appendix A of CS-VLA, the other to test the downward load case using the same chordwise distribution but half the load magnitudes. The spanwise load distribution to be proportional to the local flap chord. For each test, limit and ultimate loads must be checked separately.

Appendix A Table 2 specifies the following chordwise distribution.

note – the equations on figure A5 are incorrect in CS-VLA, so just use the graphical presentation.

6. <u>Aileron Loads</u>

S455 and S349 call for aileron loads to be accommodated from prescribed aileron deflections. Appendix A of CS-VLA provides an acceptable means of calculating the aileron loads.

EXAMPLE MICROLIGHT AIRCRAFT LOADING CALCULATIONS

Referring to CS-VLA Appendix A, with n1 W/S = 31.5 Lbs/sq ft, and Vselected = Vmin for Vc and VA (see notes in table 2), figure A5 produces a limit aileron up and down load of 13 Lbs/sq ft of aileron area. Therefore ultimate aileron up and down load is 13 x 1.5 = 19.5 Lbs / sq ft of aileron area. If aileron is symmetrical in section, one test is required, to check the manoeuvring load with a chordwise distribution per diagram C of Table 2 in Appendix A of CS-VLA.

If the aileron is non-symmetrical in design it may be necessary to test both upward and download directions, using the same loads and distributions. The hinges and control connections should be tested at the same time. The spanwise load distribution to be proportional to the local aileron chord. For each test, limit and ultimate loads must be checked separately.

7 Engine Loads

BCAR Section S para S361 and S363 prescribe engine torque and inertia load cases.

If weight of engine installation (including prop and exhaust) = We

Rotax 912S engine is a four cylinder four stroke so peak torque factor = 2.

It is necessary to test the engine mountings and forward fuselage under the following conditions:

a. A download of 4 x We (limit) combined with 2 x torque at max cont power
b. A download of 6 x We (ultimate) with 2 x 1.5 x torque at max continuous power
c. A download of 3 x We (limit) combined with 2 x max torque
d. A download of 4.5 x We (ultimate) with 2 x 1.5 x max torque
e. A side load of 4/3 x We (limit) = 1.33 x We
f. A side load of 1.5 x 4/3 x We (ultimate) = 2 We

Note that the engine torques as above are the torque at the reduction drive output not at the crankshaft PTO.

Side loads usually need testing to both port and starboard unless engine mount and forward fuselage are symmetrical.

------------------------ end ---------------------