WING SPANWISE LIFT DISTRIBUTION: SCHRENK APPROXIMATION.

One of the most important jobs to be carried out when designing an aircraft is to calculate the strength of the wing in bending, so that the main spar and wing attachment fittings can be sized. The spar bending loads are derived through the calculation of shear force and bending moment diagrams. The term shear force refers to the vertical loads on the wing, generated by wing lift and wing weight. The term bending moment refers to the moments generated along the spar caused by these shear forces acting at a distance from the load reaction points along the wing (wing root, strut locations, etc).

To be able to calculate the loads on the wing, an assumption has to be made about how the lift forces generated by the wing are distributed along the wing-span. The most aerodynamically efficient wing (for minimum induced drag) will have an elliptical distribution of lift along the span, with zero lift generated at the tip and maximum lift generated at the aircraft centre-line. Most practical wing geometries have a spanwise lift distribution approaching elliptical, but with relatively small variations in spanwise lift distribution due to wing planform. Wing twist, flap deployment and changes in aerofoil section along the wing will effect the lift distribution along the span, and these effects should be taken into account where appropriate.

Careful thought must be given when choosing a lift distribution approximation. The most often used method is the Schrenk approximation, where the spanwise lift distribution is assumed to be a variation on elliptical. This method will be demonstrated in this data sheet. However, before ploughing into the Schrenk distribution calculation, thought should be given to the type of structure you are looking at.

If you have a cantilever monoplane, then the Schrenk distribution method will be a satisfactory and useful technique. If the aircraft is a biplane, or a strut braced monoplane, then more thought should be given, and often a much simpler assumption of lift distribution is acceptable and much more useful from the stressing point of view. This is because the external bracing of the structure results in compression in the wing spars, which then act as 'beam columns'. That is, the spars will be under combined compression and bending and will require special techniques for finding stresses in these components. Most often, these techniques will be based on uniformly distributed loads along the compressed portion of the spars. This means that a uniform lift distribution would be the most convenient to use. A typical lift distribution for the case of a strut braced monoplane or an externally braced biplane would be a uniform distribution from wing root to the strut, then elliptical from the strut to the tip.

Figure 1 shows the two methods available. With a strut-braced wing, a good rule of thumb would be to assume a uniform lift distribution between root and strut only if the strut is attached to the wing outboard of half the wing semi-span. If the strut is attached to the wing less than halfway along, then the Schrenk distribution should be used. This is to ensure that a conservative assumption is used to approximate the lift distribution.
1. **Strut (or Wire) Braced Wing**

   - **STRUT BRACED WING**
   - \[ \text{Lift} = nW \]
   - Area of uniform lunes = \( a \times l_1 \)
   - Area of ellipse = \( \frac{\pi}{4} a l_2 \)
   - Total area = \( \frac{\pi}{4} a l_2 + a l_1 = \frac{nW}{2} \)
   - \( a = \frac{\left[ \frac{nW}{2} \right]}{\sqrt{l_1 + \frac{\pi}{4} l_2}} \)

---

2. **Cantilever Wing**

   - **CANTILEVER WING**
   - Use Schrenk Approximation

---

**Figure 1**

1. The Schrenk Approximation

   The Schrenk method relies on the fact that the distribution of lift across the span of an unswept wing does not differ much from elliptic, even for a highly non-elliptic planform. The process required is therefore to create an elliptic planform over a wing semi-span, and then modify it by considering the wing chord variation along the wing.

   The first step is to consider the wing without any washout, just looking at the wing planform. I find this the ideal application for a computer spreadsheet. Once set up, the spreadsheet will be there for all design iterations. What we are about to do is to find the lift coefficient distribution along the span assuming an aircraft lift coefficient of 1.0. Based on these results, we will eventually factor this lift coefficient distribution by whatever aircraft lift coefficient we have to obtain the true lift coefficient, and hence lift distribution.
Method

1. Construct a quarter ellipse with a length equal to the semi-span. The height of the ellipse is given by the equation:

\[ \text{Ellipse height} = \frac{4S}{\pi b} \left[1 - \left(\frac{2y}{b}\right)^2\right] \]  

(At the aircraft centre line \( \text{Height} = \frac{4S}{\pi b} \))

where \( S = \text{wing area}, \ b = \text{wing span}, \ y = \text{distance along span from aircraft centre-line} \)

2. Plot this against span location, \( y \), on a graph, and add to this a plot of the wing chord.

3. Draw a line averaging the two plots.

This will generate a curve showing the parameter \( c_{C_{a}} \), where \( c \) is the local wing chord and \( C_{a} \) is the local lift coefficient for a global lift coefficient, \( C_{L} = 1.0 \).

<table>
<thead>
<tr>
<th>DESIGN EXAMPLE</th>
</tr>
</thead>
<tbody>
<tr>
<td>SPAN</td>
</tr>
<tr>
<td>WING AREA</td>
</tr>
<tr>
<td>1</td>
</tr>
<tr>
<td>STATION (IN)</td>
</tr>
<tr>
<td>WING TIP</td>
</tr>
<tr>
<td>114.0</td>
</tr>
<tr>
<td>110.0</td>
</tr>
<tr>
<td>105.0</td>
</tr>
<tr>
<td>100.0</td>
</tr>
<tr>
<td>95.0</td>
</tr>
<tr>
<td>90.0</td>
</tr>
<tr>
<td>85.0</td>
</tr>
<tr>
<td>80.0</td>
</tr>
<tr>
<td>75.0</td>
</tr>
<tr>
<td>70.0</td>
</tr>
<tr>
<td>65.0</td>
</tr>
<tr>
<td>60.0</td>
</tr>
<tr>
<td>55.0</td>
</tr>
<tr>
<td>50.0</td>
</tr>
<tr>
<td>45.0</td>
</tr>
<tr>
<td>40.0</td>
</tr>
<tr>
<td>35.0</td>
</tr>
<tr>
<td>30.0</td>
</tr>
<tr>
<td>25.0</td>
</tr>
<tr>
<td>20.0</td>
</tr>
<tr>
<td>15.0</td>
</tr>
<tr>
<td>10.0</td>
</tr>
</tbody>
</table>

Table 1. Schrenk Approximation.

Columns are numbered 1 to 6. The equations used in each column are as follows:

(1) Choose a number of spanwise locations of interest. For example, in this aircraft, the wing root joint is 15 inches outboard from the centre-line. Hence I have chosen a point at station 15. Apart from these points of interest, you may want to place a point at each rib, if you have decided where the ribs are going. This will enable you to add discrete loads at each rib later on in the analysis for flap loads, etc. On a fabric covered two spar wing, the loads will be input into the spar at each rib location. The rest of the points have been chosen to produce nice graphs!

(2) \[ = 2 \times (1)/(12 \times b) \]

(3) Wing chord at each span location

(4) \[ \text{Ellipse} = \frac{4S}{\pi b} \sqrt{1 - (2y)^2} \]

(5) \[ cC_{a} = \frac{(3) + (4)}{2} \]

(6) \[ = (5)/(3) \]

Note that the odd 12 has crept in to convert inches to feet.

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
From this table, a number of graphs can be plotted. Figure 2 shows the wing chord, the ellipse with height at the aircraft centre line = Ellipse Height = \( \frac{4S}{\pi b} \) and the line drawn between them which gives us local unit lift coefficient multiplied by local chord, \( c_{Cl} \). All these terms are plotted against wing semi-span.

![Schrenk Lift Distribution](image)

Draw Schrenk curve equi-distant between chord curve and ellipse

**Figure 2**

Dividing the resulting Schrenk distribution for the untwisted wing, \( c_{Cl} \), by the local chord, \( c \), will give the local lift coefficient along the span for an aircraft lift coefficient, \( C_l = 1.0 \), with no washout (see column 6 of table 1).

![Local Lift Coefficient Cls for Unit Cl vs Span](image)

**Figure 3**

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
2. Stalling Behaviour and Washout

It can be seen from figure 3 that this wing profile will almost certainly have a tip stalling problem. Ideally, we would like the distribution of local lift coefficient to be at a maximum at the aircraft centre-line. This will help to ensure that the inner wing section will stall first. The inner part of the wing to stall for our wing planform will be outboard along the span, which will probably lead to a wing drop and very poor aileron control near the root. You will find that the slab-wing (constant chord) will have desirable stall characteristics from the start, with maximum lift coefficient at the root and no washout generally required. That is one reason why they are so popular. However, a tapered wing will have a deeper spar at the root, and will probably turn out lighter with far less loading at the wing attachment joints at the wing root. The reduction of loads in the wing attachment joints will be due to lower bending moment because of the planform taper and attachment lugs which are further apart vertically because of the deeper spar. Particularly in a wooden wing, these root joints loads are critical because of low bolt-bearing strength of wood. The two solutions to the tip stalling problems for our wing will be to either add a very high-lift section to the outboard portion of the wing (introduction of slats?), or to add washout. The former solution is often unsuccessful and adds much complexity, and there really is no substitution for washout.

Washout can be achieved in two ways. The first conventional method is by introducing a trailing-edge-up twist along the wing towards the tip. The second is to change the aerofoil section, or more specifically, the zero-lift angle of the aerofoil section. For example, a typical aerofoil section such as the NACA 2412 has a zero lift angle of 2 degrees. A symmetric section (NACA 0012, for example) has a zero lift angle of zero degrees. If we use the NACA 2412 section at the inboard sections of the wing, and the symmetric section at the tip, then we will have 2 degrees of washout in the wing. Care must be taken however. The NACA 0012 may have a lower stalling angle than the NACA 2412, so although some aerodynamic washout is present, a tip stall may still occur. It is therefore up to the designer to immerse themselves in aerofoil curve data to decide on a suitable combination of aerofoil sections along the wing. Even if a good combination of sections is achieved, then washout might still be required, as is the case for our example wing in all probability.

If we were to choose a section with a high-lift capability for the tip, take care because using a more cambered section will generally result in the section having a more negative zero-lift angle than the sections inboard which will actually result in aerodynamic 'wash-in', the exact opposite effect of using a symmetrical section with a zero lift angle of 0 degrees.

We will therefore eventually extend our spreadsheet to handle both aerodynamic and geometric washout. For the purpose of this data sheet, we will ignore washout for the time being because we are interested in the bending loads on the wing, rather than the wing aerodynamic behaviour at the stall.

I calculated that 5 degrees of washout was required for a design similar to the design example to ensure a stall at the wing root. By taking this washout into effect, the ultimate aerodynamic bending moment found at the wing centre-line was 184359 lb.in. Without the washout, the bending moment came out as 194769 lb.in. This amounts to a 5% difference when ignoring the washout at high wing incidence conditions, which is a nice amount to have in reserve.

One last effect which can have a critical effect is based on Reynolds number. Reynolds number is equal to aircraft speed x air density x wing chord divided by viscosity of air (don't panic!). A spreadsheet used to calculate Reynolds number was included in John Roncz's series of issues on aircraft aerodynamic design in the EAA's Sport Aviation magazine (1990). The equations from that spreadsheet are included in Appendix B. Looking at references like 'Theory of Wing Sections' by Abbot and Von Doenhoff, you will see lift and drag data presented for a number of aerofoil sections. Curves are presented for aerofoils at three different Reynolds's numbers, usually 3 million, 6 million and 9 million. You will be able to see that the lower the Reynolds number, the higher the drag and the lower the maximum lift coefficient of the section. Notice that the Reynolds number is based on wing chord. By having a heavily tapered wing, the tip chord will be small and the Reynolds number at the tip relatively low. This may well lead to a low $C_{\text{max}}$ at the tip and a subsequent tip stalling problem. It is therefore important for the designer to

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
consider the effects of Reynolds number on the aerofoils used in the design. Do not expect a wing to perform at a Reynolds number of 1 million as predicted by aerofoil wind tunnel data taken at a Reynolds number of 9 million.

When looking at aerofoil data from Abbot and Von Doenhoff, you will be able to see that as well as aerofoil data given for three different Reynolds numbers, data is also given for each aerofoil with 'Standard Roughness' applied to it. 'Standard Roughness' is in fact far too rough. Perhaps representative of a rib-stitched fabric covered wing, but too rough for a typical homebuilt wing surface. It is better to reduce the expected maximum lift coefficient to some extent due to roughness - but not to the extent shown in the data presented.

3. Symmetric Loads On An Aircraft Wing

In this data sheet we will initially look at aerodynamic loads. We will then expand our spreadsheet (or hand calculated tables) to enable us to consider the inertia loads acting on the wing. The analysis will be simplified to some extent, in that the only thing we are trying to find using this data sheet is the normal (vertical) bending loads on the main spar of a single-spar wing. Other data sheets or references will deal with two-spar wings. By a single spar wing, I mean a wing where one main spar is considered to carry all bending loads, with a small auxiliary rear spar pin jointed to the fuselage to react torsion loads into the fuselage. A typical two-spar wing is as used in Austers and Piper Cubs, where both front and rear spars carry bending loads which are reacted by lift struts as well as at the fuselage root fittings. Many homebuilts fall into the two-spar wing category, biplanes included. The basic lift distribution calculation will be the same, but you will need to resolve the lift and inertia loads between the two spars based on the lift centre of pressure and the wing centre of gravity respectively.

We are neglecting drag loads - which will generate loads in the internal drag bracing or skin of the wing, together with producing end-loads in the spars and at the wing attachment fittings. A later spreadsheet will show how to resolve wing lift and drag loads into normal and chordwise directions which will give both vertical and chordwise shear and bending loads on the wing panel.

We are also neglecting wing torque generated by the lift and inertia loads acting at a distance from the wing shear centre. This torque will generate shear loads in the wing skins.

As stated before, the purpose of this design sheet is purely to determine the shear forces and bending moments along the main spar of a single spar wing. This will enable us to size the spar webs and booms, together with the root wing attachment fittings.
Aerodynamic Loads

What we know from table 1 is the variation in local wing lift coefficient along the span, assuming an over-all wing lift coefficient of 1.0. We can use these results to develop the normal shear force and bending moment diagrams along the semi-span. Along the way, we will be able to determine the mean aerodynamic chord of the wing, and the wing area based on wing span and chord. We can use this to update our assumed wing area at the start, or to alter the chord or span to arrive at our required wing area.

### Table 2. Shear Forces and Bending Moments Along The Semi-Span

Extra terms which need to be calculated are as follows:

- Wing Lift, \( L = 1.05 \times nW - L_t \)
  - \( n \) is the positive limit manoeuvre load factor (from the flight envelope)
  - \( W \) is the aircraft weight.
  - \( L_t \) is the likely tail load (+ve upwards).
  - 1.05 is a factor required by JAR VLA to ensure a conservative answer (and to take into account tail loads if you have assumed \( L_t = 0 \)).

\[ L = qSC_L \]

where
- \( q \) is the aircraft dynamic pressure \( (q = 1/2 \rho V^2) \)
- \( S \) is the wing area
- \( C_L \) is the over-all wing lift coefficient.

Therefore
\[ qC_L = L/S \]

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
What we have done so far is to find the chord and local unit lift coefficient at various stations along the wing. We now need to find the average unit lift coefficient between each station. Knowing this, we can calculate the average lift force between each station based on the area of each spanwise element (see figure 4).

Starting at the aircraft centre-line (station 0), we therefore average the chord and lift coefficient values between two adjacent stations. We also find the distance between each station, which we call the spanwise element width. From this we can calculate the area of each spanwise element, $\Delta S$. Then we can find the lift force generated by each spanwise element from the equation:

$$ \text{Element lift} = qC_L \Delta S C_A $$

Hence:

| Column (7) | Width of spanwise element (inches) | (see fig (4)) |
| Column (8) | Average chord over spanwise element (ft) | (see fig (4)) |
| Column (9) | Area of spanwise element $= \Delta S = (7)/12 \times (8)$ (sq ft) (see fig (4)) |

We will eventually sum up this column and multiply by 2 to find the wing area based on chord and span. You can use this to check your initial assumption for wing area.

$$ C_{AV}^2 = (7)/12 \times (8)^2 $$

We eventually sum up this column and divide the result by the wing area. This will give us the mean aerodynamic chord of the wing.

| Column (10) | $= \Delta Y C_{AV}^2 = (7)/12 \times (8)^2$ |

At the wing root (station 0), the shear force must equal approximately half the lift. It is sensible to perform a check to establish that this is the case. In the example case, a 0.125% error was found, which is acceptable.

Also:

- **Wing Area** = $2 \times \text{Sum of column (9)}$
- **Mean aerodynamic chord, MAC** = $\text{Sum of column (10)}/\text{Wing Area}$
Figure 4. Averaging wing chord, local lift coefficient and finding area of spanwise elements

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
AT STATION 85

\[ M = \frac{370.2 + 5(34.9 + 44.6)}{2} = \frac{5688}{2} = 8177 \text{ lb ft} \]

Figure 5. Calculation of bending moments from the shear-force distribution

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
Inertia Loads

The wing bending loads are elevated to some extent by the wing weight. A similar type of process can be used to calculate the shear force and bending moments due to wing weight as used above for the aerodynamic loads. You can perform the weight calculation using several approximations, or if you have got far enough down the design path, you could use accurate design data. Obviously, any point loads such as tip tanks, rocket launchers, etc., will have to be taken into account. Several books give estimation methods to calculate wing weight. From Pazmany, I calculated that the wing weight of the example aeroplane would be approximately 74 lb per side. As well as this, the aircraft will be fitted with a retractable undercarriage which will weigh around 25 lb per side. The centre of gravity of the retracted undercarriage will be, say, at station 30. Care will have to be taken to account for local load inputs caused by the undercarriage when both retracted and extended. Because the wing is tapered in both planform and thickness on the example aeroplane, I will assumed that the wing weight varies from root to tip in the ratio 4:1.

\[
\frac{w + 1/4w}{2} \times \text{span}/2 = 74 \text{ lb where } w \text{ is the inertia loading at the wing root (lb/ft).}
\]

The wing is joined to the fuselage at station 15. Hence the wing weight only acts outboard of this station. Because we have a centre-section which is 30 inches across, which does not contribute to the wing weight, the root inertia loading becomes:

\[
w = 74 \times 2/((1.25 \times (114 - 15)) = 1.196 \text{ lb/inch at the root and}
\]

\[
w = 1.196/4 = 0.299 \text{ lb/inch at the tip.}
\]

To check, \((0.299 + 1.196)/2 \times (114 - 15) = 74 \text{ lb.}\)

We will have to modify the existing spreadsheet a little to cater for the point mass caused by the undercarriage. In this case, we insert an extra line at wing station 30 which is where the inertia of the undercarriage acts.

Wing Root Reaction

It is also clear that with the wings joining the fuselage at station 15, the total vertical shear in the spar will be transferred to the fuselage at this point. Assuming for the moment a symmetric load on each wing panel about the aircraft centre-line, the shear at the aircraft centre-line must be zero (see figure 6). Hence at station 15, the total shear reaction must equal the shear at the aircraft centre-line. An extra line at station 15 should be inserted in the spreadsheet to allow for this root reaction. This will result in zero shear at the aircraft centre-line and maximum bending at the wing root fuselage attachment. The root reactions should be inserted into the shear-force distribution for both the aerodynamic and inertia load cases. Notice in table 3, showing the modified spreadsheet, that the bending moment reduces slightly between the wing root and the aircraft centre-line. This is due to air loading assuming to act on the centre-section region. In fact, the fuselage, rather than the wing centre-section, carries most of the air loading inboard of the wing root. Therefore:

Use the maximum wing root moment to stress the centre-section spar booms.

Use the total root reaction found to size the joint between the centre-section and the fuselage.

Hence in the example case, the constant ultimate bending moment across the centre-section can be assumed to be 116 956 lb in. The root shear reaction can be assumed to be \((2575 - -731) = 3306 \text{ lb per side. This is the load seen by the pins holding the centre-section to the fuselage at station 15. A first approximation for maximum shear in the spar would be 2575 lb.}\)

Of course, you could neglect the root reactions and assumed the wing is cantilevered from the aircraft centre-line. This would give a conservative estimate for root bending moment and the maximum shear force in the spar webs.

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
Take care when you modify the spreadsheet that you re-copy the rows under these stations to ensure that the spreadsheet does not skip over the inserted rows during calculation. Take care not to copy over anything valuable!

**Table 3** Modified spreadsheet including extra lines for point inertias (U/C) and wing root reaction

The resulting aerodynamic shear force diagram looks as follows:

---

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
AERODYNAMIC ULTIMATE SHEAR FORCE DIAGRAM

Figure 6. Aerodynamic shear force distribution with root reaction
### Table 4. Final spreadsheet including inertia effects and root reactions.

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
The columns (17) to (27) are calculated at follows:

Column (17)  Linear interpolation of wing inertia loading between the known values between station 15 (the wing root) (-1.196 lb/in) and station 114 (the wing tip) (-1.196/4 = -0.299). For example at station 86, (17) = -1.196 - (0.702-0.132)/(1.0-0.132) x (-1.196 - -0.299) = -0.607 lb/in

Column (18)  = (17) x n. n = 6 in this case

Column (19)  Average inertia loading across spanwise element. For example, between station 80 and station 85 Average inertia loading = (-3.642 + -3.371)/2 = -3.506 lb/in

Column (20)  Limit shear force = cumulative sum of column (19), starting at the tip, where the shear force is always zero unless you have a tip-tank or something at the end of the wing.

Column (21)  Limit bending moment is calculated in the same way as column 14. For example, at station 80 Limit bending moment = -978 + ((-75 + -92)/2 x (85 - 80) = -1357 lb in.

Column (22)  = 1.5 x (20)

Column (23)  = 1.5 x (21)

Column (24)  = (13) + (20)

Column (25)  = (14) + (21)

Column (26)  = (15) + (22)

Column (27)  = (16) + (23)

The following Shear Force Diagram can now be plotted

![Ultimate Shear Force Diagram](image)

Figure 7. Shear Force Diagram

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
4. Asymmetric Load Cases

Eventually, you will have to consider asymmetric load cases. These will generally size the wing/fuselage attachment pins and the centre-section shear-web. Asymmetric load cases are given in JAR VLA or other relevant airworthiness requirements.

5. Further Work

You can use the results from this spreadsheet to size a typical main-spar, if you assume that the spar carries all bending loads. Later data sheets will explain how to calculate drag loads on the wing, and how to stress a simple box spar using the results obtained from this data sheet. The effect of washout and changes in wing sections on wing bending will also be explained. The effects of torsion will also have to be established. As stated before, however, this data sheet should give you a good guide to finding the loads required to size your main wing spar for a conventional single-spar cantilever wing. Two spar wings can be handled in the same way, but the total loads must be resolved into the front and rear spars, based on wing centre-of-pressure for aerodynamic loads and centre-of-gravity for inertia loads.

6. References

Pazmany Light Aircraft Design Available from PFA Headquarters.

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.
Appendix B. Calculation of Reynolds Number

From Sport Aviation magazine, John Ronz article. June 1990 page 45.

<table>
<thead>
<tr>
<th>Spreadsheet #6</th>
<th>From Sport Aviation, June, 1990</th>
</tr>
</thead>
<tbody>
<tr>
<td>Altitude:</td>
<td>0</td>
</tr>
<tr>
<td>Wing Chord, inches:</td>
<td>23</td>
</tr>
<tr>
<td>Speed, MPH:</td>
<td>130</td>
</tr>
<tr>
<td>Rho:</td>
<td>0.00237689 Slugs per Cubic Foot</td>
</tr>
<tr>
<td>Mu:</td>
<td>3.7448E-07 LB-SEC/SQ. FT.</td>
</tr>
<tr>
<td>Temperature:</td>
<td>59 Degrees F</td>
</tr>
<tr>
<td>Reynolds Number:</td>
<td>2.320040175 Million</td>
</tr>
</tbody>
</table>

Draft Report. PFA Ulair Ltd accepts no liability what-so-ever from use of these data sheets or the data held within them. In particular, this data sheet is in draft form only, and may need correction.