The problem of estimating wing weights is a very complex one, for no other major structural item of an airplane has so many varying factors that affect weight. There is a definite need for an estimating method, based on these varying factors, that can be applied to any type of wing with reasonably accurate results. The attack of the problem must be based, however, on logic, and be as simple as possible within the limits of reasonable accuracy. In other words, a compromise should be made between the simplest method of wing estimation, an estimate based on a similar model already constructed, and the most complex method, a complete stress analysis of the wing.

The purpose of this article is to present a method of wing weight estimation that is logical, practical, accurate, and, although complex in its derivation, simple to apply. Basically, the method was evolved from an equation of the bending material required to sustain the airloads. It is applicable to all types of airplane, and it accounts for weight variations due to wing loading, span, thickness, taper ratio, load factor, and material. This method is also adaptable for use with particular types of structure instead of particular types of airplanes. Essentially, this means that a broader scope of estimates can be made; for most modern airplanes employ similar types of wing structure.

The three steps that comprise the solution to this method of wing weight estimation are as follows:

1. Derivation of a wing weight formula based on the bending material required for bending loads.
2. Tabular form of the formula used to solve for "N" (Bending Material Weight) which is the ratio of Wing Weight (Less Ail. and Flaps) to Total Wing Weight.
3. "N" and bending material weight found in (2) for known ships plotted against basic wing dimensions and design criteria.

**Derivation of Wing Weight Formula**

Several assumptions have been made in arriving at this method for estimating wing weights:

1. Bending loads are the design criteria. They are based on air loads, tail loads, and dead weight loads.
2. An average allowable stress of 35,000 lb./in.² has been used in determining the material required for bending.
3. All-metal, box-beam, uniformly tapered wings form the basis for this analysis.

The following is a detailed derivation of the formula:

1. The running load on any element, X distance from the wing tip, is equal to the chord at Sta. X multiplied by the wing loading in lb./in.². The equation for the running load at X is as follows:

\[ L_x = K \left( C_T + \frac{X}{b} (C_T - C_R) \right) \]

Where:
- \( K = \) Wing Loading in lb./in.²
- \( C_T = \) Tip Chord in inches
- \( C_R = \) Root Chord in inches
- \( b = \) Span in inches
- \( X = \) Span from tip in inches.

2. Integrating the load formula will give the shear:

\[ S_x = K \left[ \frac{C_T X + \frac{X^2}{2}}{2} (C_T - C_R) \right] \]

3. Subsequently, integrating the shear formula will give the moment equation:

\[ M_x = K \left[ \frac{C_T X^2}{2} + \frac{X^3}{3} (C_T - C_R) \right] \]

4. The bending material area (A) may be evaluated by substituting \( f = \frac{1}{M} \) in the moment equation when \( f \) is written in terms of \( A \), as follows:

\[ f = \frac{2M}{A} \text{ or } A = \frac{M}{f} \]

5. Since \( Y \) is the average depth of bending material between centroids of the upper and lower surfaces, its equation may be written in terms of wing span, bending stresses, and chord:

\[ Y = T_x \left( \frac{T_x - T_{x-1}}{b} \right) \]

Where:
- \( T_x = \) Avg. thickness of tip chord.
- \( T_{x-1} = \) Avg. thickness of root chord.

6. Hence, substituting \( A_x \) and \( Y_x \) in equation 3 will give the bending material area at \( X \) at 1C load factor:

\[ A_x = K \left[ \frac{C_T X^2}{2} + \frac{X^3}{6b} (C_T - C_R) \right] \]

7. Multiplying \( A_x \) by the ultimate load factor (L.F.), a factor \( Z \) to correct the wing load for tail load and wing weight, the aluminium weight constant .10, and dividing by a factor \( N \) representing the ratio of bending material weight to total wing weight, will result in the ultimate formula for the total wing weight in lb./in. at any station \( X \):

\[ \text{Wing weight} = \left( \frac{L.F. \times Z \times .10 K}{\frac{5N}{f}} \right) \left( \frac{C_T X^2 + \frac{X^3}{6b} (C_T - C_R)}{T_x + \left( \frac{T_x - T_{x-1}}{b} \right) X} \right) \]