More On "Simplified Wing Stress"

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Referring to the article "Simplified Wing Stress Analysis of a Strut-Braced Monoplane", (*SPORT AVIATION*, November, 1963, publication of these comments would not detract but rather add to a better understanding and use of your method.

The use of center of pressure positions for several angle of attack conditions is not only obsolete but also cumbersome. Aerodynamic information now in use for more than 25 years for loads and stress analysis provides data in terms of an "aerodynamic center" (a.c.) and a "zero lift pitching moment coefficient" about this center (C_{mac}, or also C_{mo}). Both values remain constant for a large range of angles of attack and are perfectly applicable to your type of analysis. The a.c. is the point of lift-action and is, for all practical purposes, very close to 25 percent chord. C_{mo} is a negative value for normally cambered sections and zero for symmetrical sections; flap and aileron deflections cause additional negative increments. Key references for section and wing geometry effects are: NACA Report 824, and Abbott-Doenhoff "Theory of Wing Sections" (McGraw Hill, and Dover Publications). For flap effects see NACA TN-4040. A detailed definition of a.c. and C_{mo} can be found in the book: Perkins-Hage, AIRCRAFT PERFORMANCE STABILITY AND CONTROL.

The airfoil sections mentioned on page 13 are long out of circulation owing to their inferior stall, drag and pitching moment characteristics.

The ultimate net load $V^* = (W - W_w)\ n_{ult}\ y/b$ which acts through the a.c. may be split into the proper proportions as given by the a.c. location with respect to the spar locations. The pitching moment C_{mo} is irrelevant of reference. In the case of spars taking all loads (no shear resisting skin), this moment (torque) may be taken in differential bending by the front and rear spar with the additional shear forces in lbs. at each spanwise station $- V_f = + V_r = T/X$, where the torque in in-lbs. $T = (\rho V^2)/(2 \times 144) \int_0^y C_{mo}\ C^2\ dy$.

In case of spars and shear resisting skin forming a torque box, the skin gauge in inches would result in $t = \sqrt[3]{T\ W^2/2\ A\ K_s\ E}$. In correct analysis, the elastic axis position can be established and should be used; in a very good approximation for lightplanes, the elastic axis may be assumed to go through the structure center of gravity, or in some cases even through the centroid of the box cross section. Then, we can add the torque due to the ultimate net load $T_V = \int_0^y V\ X_e\ dy$ to the above basic torque T.

The integrals are quickly obtained by plotting the variables versus y (for instance $C_{mo}\ C^2$ versus y) and calculating the area under the curve, accounting for the plotting scale and dimensions as shown in the illustration below:

Symbols other than those already explained are:

- ρ = Air density (slugs/ft.^{3} or lb. s^2/ft.^{4})
- W = Aircraft design gross weight (lbs.)
- W_w = Wing weight (lbs.)
- n_{ult} = Ultimate load factor = 1.5 n limit
- y = Running length from tip to root (in.)
- b = Span (in.)
- X = Inter spar distance (in.) @ any station
- W = Panel width (in.) @ any station
- A = Torque box cross sectional area (in.^{2}) @ any station.
- K_s = Buckling constant (Peery, Aircraft Structures)
- E = Modul of elasticity (10500000 lb./in.^{2} for 24 ST Al. alloy sheet)
- X_e = Distance between a.c. and elastic axis (in., negative if a.c. is behind the elastic axis) @ any station
- C = Chord at any station (in.)

FOOTNOTE—*Simplified as per the article. In case you do not wish to calculate aero-loads, you may use those specified in the FAA-CAM 3 (Airplane Airworthiness; Normal, Utility and Aerobatics Categories) May, 1962, sections 3.171, 3.172, 3.181 through 3.191, Appendix A, sections 1.0 through 6.42, figures 4, 5 and Table 1. These loads are conservative and based on current experience, clearly outlined and simple to apply.