Computation of the Flow Around Wings with Rear Separation

K. Jacob*
German Aerospace Research Establishment (DFVLR), Göttingen, Federal Republic of Germany

Abstract

A METHOD for computing the flow around wings with trailing edge stall is briefly described. The method combines an inviscid three-dimensional lifting surface theory with a two-dimensional airfoil theory, which includes boundary-layer calculations and a displacement model for rear separation. The total iterative procedure allows for predicting the complete wing characteristics, including maximum lift, but is restricted to wings with moderate to high aspect ratios, and low sweep and to low Mach numbers. Some results are shown and partly compared to experiments.

Nomenclature

c_L, c_D	= total lift- and drag- coefficient of the wing
c_{ln}	= local lift coefficient of section n
c_{mn}	= local moment coefficient of section n
M	= number of wing sections
s	= half-span of the wing
x,y	= chordwise and spanwise coordinate
η	$= y/s$

The Method

In order to predict the aerodynamic forces of wings up to and beyond stall, viscosity effects, including flow separation, must be taken into account. This is extremely difficult, and a general three-dimensional method does not yet exist. But an approximate method for a limited but important class of wings has been designed. For high aspect ratio and low sweep, the flow is approximately two-dimensional over large portions of the wing, and separation mostly starts near the trailing edge in the middle of the wing. The basic idea of our method is to treat each wing section with a two-dimensional method for viscous flow with rear separation, and take account of the three-dimensional vortex system of the finite wing by computing the effective onset flow at each section with a three-dimensional lifting surface theory. The two theoretical methods are linked via the spanwise lift distribution $\gamma(y)$, which is found by an iteration process (Fig. 1), starting at a low angle of attack with the lift and moment distribution of the inviscid flow.

For the three-dimensional calculation, the linear lifting surface theory of Truckenbrodt[1] is used giving a linear relation:

$$\alpha_\nu^* = \sum_{n=1}^{M} A_{\nu n}^* \gamma_n + B_{\nu n}^* \mu_n$$

$$\alpha_\nu^{**} = \sum_{n=1}^{M} A_{\nu n}^{**} \gamma_n + B_{\nu n}^{**} \mu_n \qquad (1)$$

$$\nu = 1,2,\ldots,M$$

with α_ν^* and being α_ν^{**} being respectively the induced downwash angles at the trailing edge and the quarter-chord point of section ν. The γ_n and μ_n are essentially the local lift and moment at section n.

The A and B depend on M and on the wing geometry. With γ and μ known (from previous approximation), the angles α_ν^* and α_ν^{**} can be obtained from Eq. (1) by simple matrix-vector multiplication. Then, with the given geometric angle of attack α_g, the effective angles of attack $\alpha_{e\nu}$ for the two-dimensional viscous flow calculations can be computed.

For the two-dimensional viscous flow, the author's method[2,3] is applied. The flow model is shown in Fig. 2. With a vortex distribution all around the surface of the airfoil, a source distribution between points S and T, and a sink behind the airfoil, a potential flow is constructed that contains a displacement model of a dead air region starting at S. The separation point is found by adding a boundary-layer calculation[4] for the attached part of the flow and repeating the total computation for various positions of S until the boundary-layer separation point coincides with S. The final results are the pressure and skin friction distributions and the force and moment coefficients c_l, c_d, c_m for a given airfoil section, angle of attack α_e, and Reynolds number Re.

It would be very time-consuming to carry out the two-dimensional viscous flow computation for each $\alpha_{e\nu}$ occuring during the iteration process (Fig. 1). Instead, we compute a sufficient number of points of the full characteristics $c_l(\alpha_e)$, etc., for a sufficient number of wing sections in advance and, within the iteration loop, get the needed values simply and fast by interpolation. Thus, computing the complete characteristics of a wing $c_L(\alpha_g)$, etc., takes only about 1 h CPU-time on an IBM 370/158 computer. Some other important details of the iteration process are discussed in the full paper.

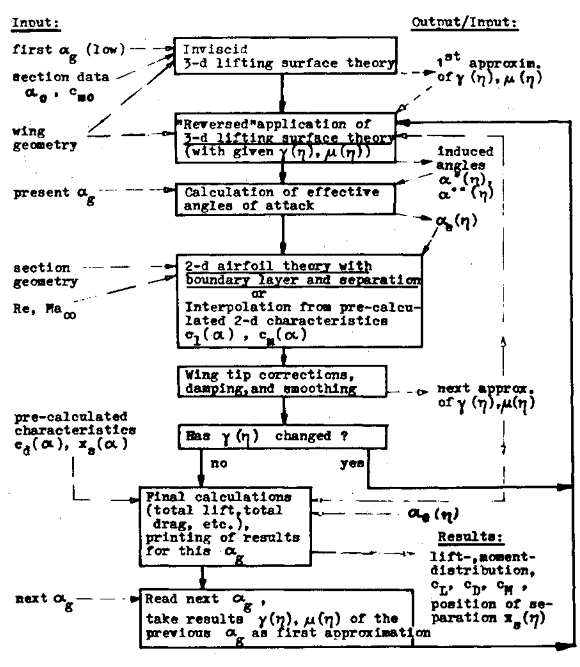

Fig. 1 Survey of the total iteration process.

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*Dr.-Ing., Institut für Theoretische Strömungsmechanik.

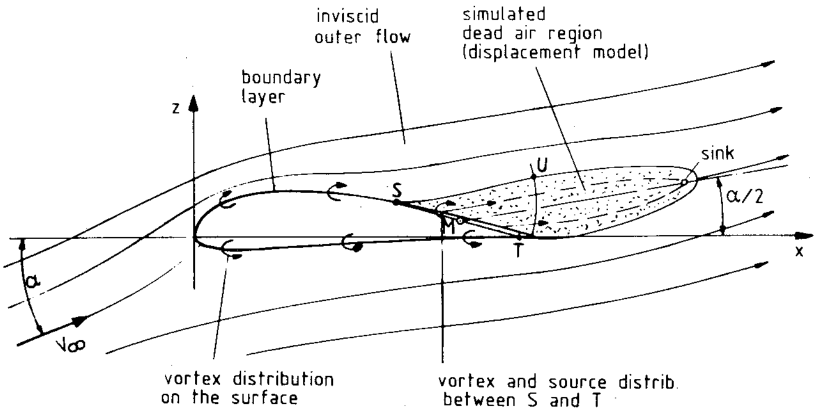

Fig. 2 Model of a two-dimensional flow around an airfoil section with rear separation.

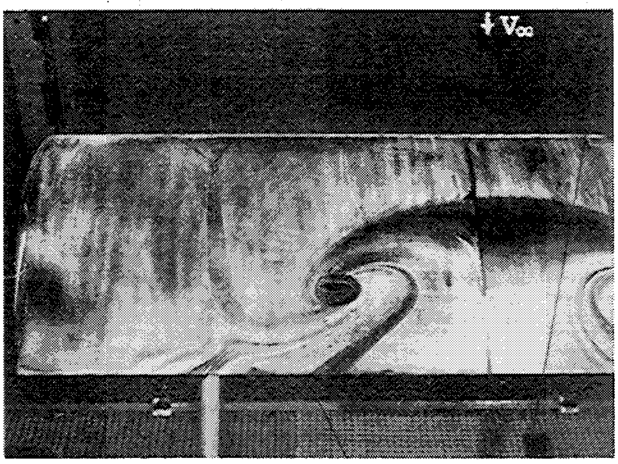

Fig. 3 Oil flow pattern on the upper surface of a rectangular wing ($\Lambda = 3.1$, $Re = 2.1 \times 10^6$, $\alpha_g = 21$ deg).

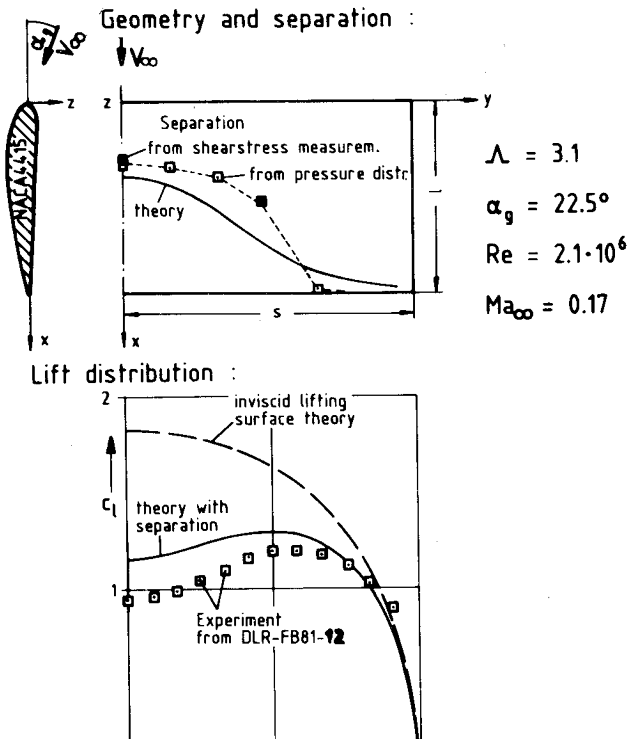

Fig. 4 Separation line and lift distribution of a rectangular wing at a high angle of attack.

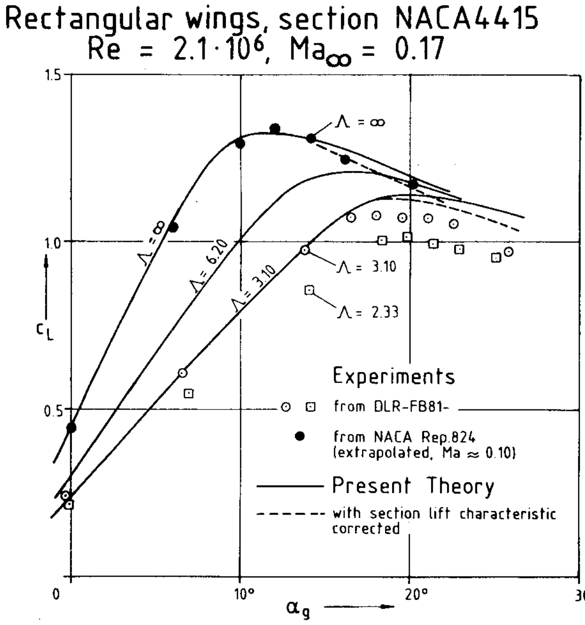

Fig. 5 Total lift characteristics of rectangular wings with different aspect ratios Λ.

Some Results

Figures 3 – 5 show experimental and theoretical results for a rectangular wing with an aspect ratio $\Lambda = 3.1$ and a constant section NACA 4415. The oil flow pattern (Fig. 3) indicates a large region of rear separation in the middle of the wing for high α_g. Figure 4 shows separation lines lines and spanwise lift distributions from theory and experiment. In Fig. 5, computed and measured total lift characteristics for different aspect ratios are compared. Even though the aspect ratio $\Lambda = 3.1$ is not high, the theoretical results still compare reasonably well with the experiments. Better results can be expected for higher aspect ratios.

References

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[3]Jacob, K., "Advancement of a Method for Calculating Separated Flows Around Airfoils with Special Consideration of Profile Drag," German Aerospace Research Establishment (DFVLR), Köln 90, FRG. DLR-FB 76-36, 1976; see also ESA TT373, N 77-26091, Sept. 1977 (English translation).

[4]Rotta, J. C., "Computer Program for Boundary Layers in Compressible Two-Dimensional and Axisymmetric Flow," German Aerospace Research Establishment (DFVLR), Köln 90, FRG, DLR-FB 71-51, 1971.