INVESTIGATION ON ROTATING AILERONS

S. Mozumdar* and T.S. Patel**
Department of Aeronautical Engineering
Indian Institute of Technology
Bombay, India

ABSTRACT

The effectiveness of rotating ailerons in a subsonic flow has been theoretically studied here with a view to evaluate their capability in comparison to the conventional ones. The present analysis is based on the vortex lattice approach of Byelotserkovskii[8] for different wing planforms of small and large aspect ratios. The effect of such ailerons on the coefficients of lift and rolling moment has been worked out for their positions corresponding to the most suitable positions of rotating flaps..

Introduction

An efficient control surface is of particular interest to an airplane designer. The present problem of investigation about the feasibility of rotating ailerons is directed towards an engineering solution of improving rolling control of an aircraft in the subsonic flight regime.

Some aerodynamic properties of autorotating or forcibly rotated cylinders and aerofoils were investigated even prior to the beginning of aviation. The following two important properties of a profile revealed by simple earlier experiments suggested their promising application in aircraft design:

1. Ability to autorotate, and
2. Ability to create force normal to the axis of the rotation and free-stream velocity.

The rotating profile as a lift augmenting device seems to be one of the possible aids about which some work[1-7] is already available in literature. The concept of rotating profile as control surface elements arose from the suggestions made by Crabtree[1] and Neumark[2]. The present work deals with an estimation of effectiveness of rotating ailerons for different wing planforms using linear lifting surface theory of Byelotserkovskii[8].

* Senior Research Scholar (Now at National Aeronautical Laboratory, Bangalore as Scientist in Aerodynamic Division).
** Assistant Professor.
Received Feb. 1978; revision received Sep. 9, 1978.

Formulation of the Problem

Let us consider the wing as a thin lifting surface with rotating ailerons placed in an incompressible fluid flow. In general, the uniform undisturbed flow may be given by (Fig.1):

$$u = U_0 \cos \alpha ; \quad v = U_0 \sin \alpha \quad , \tag{1}$$

where u and v are the velocity components along X and Y axes respectively.

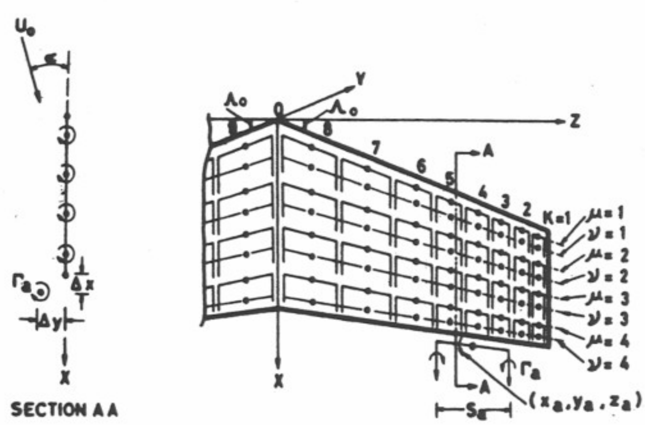

Fig.1. Vortex lattice model of the wing with rotating ailerons

In the present model the wing is replaced by a vortex surface as in the case of linear lifting surface theory of Byelotserkovskii[8]. The lifting vortex sheet is then approximated by a series of n equally spaced bound vortices along its span, the number of which could be as large as practicable. Each of these bound vortices is further divided into N elementary pieces by the sections parallel to the root chord so that the elementary vortices in the inboard region of wing are longer than those in the outboard region as was done in the Multhopp's lifting surface theory[9].

Then, the spanwise positions of those N sections on the wing may be written in the form

$$z_p = (b/2)\cos(p\pi/N) \quad , \qquad (2)$$

where

- b is wing span;
- p is an integer varying from 0 to N ($p = N/2$ indicates the root chord section).

The free vortices from the tips of these elemental bound vortices so formed are assumed to be in the plane of the wing, trailing parallel to the plane of symmetry. The rotating ailerons are replaced by a pair of horse-shoe vortices of given circulations[1,2] having the same magnitude but opposite sign. The bound part of these vortices lie along the axes of rotation of the ailerons. The trailing vortices are assumed to remain parallel to the trailing vortices of the main wing.

The unknown distribution of the strength of the elementary vortices representing the wing surface is determined by imposing the boundary condition that the resultant velocity component normal to the wing surface doe to the wing vortex system, ailerons and the freestream should be equal to zero. This condition is satisfied at selected pivotal points on the wing. The position of the mid-point of each of the bound vortex elements of the wing and the pivotal points are shown in Fig.1. The notation used is as follows:

- k serial number of the strips parallel to root chord ($1 \leqslant k \leqslant N$);
- μ serial number of bound vortex lines distributed along chord ($1 \leqslant \mu \leqslant n$);
- ν serial number of the lines on which the boundary condition is satisfied at selected pivotal points ($1 \leqslant \nu \leqslant n$);
- i serial number of the oblique horse-shoe vortex ($1 \leqslant i \leqslant Nn$);
- j serial number of the pivotal points ($1 \leqslant j \leqslant Nn$).

Let k_i, k_j be the values of k corresponding to the values of i and j. We have then

$$i = k_i + (\mu - 1)N; \quad j = k_j + (\nu - 1)N \quad . \qquad (3)$$

Here the chord is divided into n equal segments. Each of the elemental surface formed by spanwise and chordwise sections is represented by the Weissinger's model namely, the elemental bound vortex placed along the quarter-chord line of each elemental surface and the pivotal point for each elemental surface is located at the mid-point of its three-quarter-chord line. With sufficiently large value of n this distribution of vortices and pivotal points practically satisfies Kutta-Joukowsky condition at trailing edge.

The coordinates of the mid-point of the i-th elementary bound vortex (x_i, y_i, z_i) and those of the j-th pivotal point (x_j, y_j, z_j) on the starboard wing will then be given by the following relations:

$$x_i = x_{oi} + \frac{1}{4}\frac{c_{zi}}{n} + (\mu - 1)\frac{c_{zi}}{n} \quad ,$$

$$x_j = x_{oj} + \frac{3}{4}\frac{c_{zj}}{n} + (\nu - 1)\frac{c_{zj}}{n} \quad ,$$

$$y_i = y_j = 0 \quad ,$$

$$z_i = \frac{b}{4}\left[\cos\left\{i - (\mu - 1)N\right\}\frac{\pi}{N} + \cos\left\{i - 1 - (\mu - 1)\right\}\frac{\pi}{N}\right] \quad ,$$

$$z_j = \frac{b}{4}\left[\cos\left\{j - (\nu - 1)\right\}\frac{\pi}{N} + \cos\left\{j - 1 - (\nu - 1)\right\}\frac{\pi}{N}\right] \quad , \qquad (4)$$

where

- c_{zi}, c_{zj} wing chords on which the points i and j are located respectively;
- x_{oi}, x_{oj} projections of the leading edges of the sections with the chords c_{zi} and c_{zj} on X axis respectively.

The values of z_i and z_j on the port side wing will be the same in magnitude but different in sign. The coordinates x_i, x_j, y_i, y_j will be the same as those for the starboard wing. The semi-span of i-th elementary bound vortex is:

$$s_i = \frac{b}{4}\left[\cos\left\{i - (\mu - 1)N\right\}\frac{\pi}{N} - \cos\left\{i - 1(\mu - 1)N\right\}\frac{\pi}{N}\right] \qquad (5)$$

Knowing these geometrical details the induced velocity components at the pivotal points due to all the wing vortex elements together with those of rotating ailerons could be obtained using Biot-Savart law for induced velocity or from the corresponding relations for an oblique horse-shoe vortex[10].

The boundary condition to be satisfied over the wing surface at selected j points will be given by

$$U_o \sin\alpha + v_{jy} = 0 \quad , \qquad (6)$$

where v_{jy} is the velocity component at a point j, normal to the wing surface, induced by the system of wing vortices and rotating ailerons. The value of $\sin\alpha$ may be taken equal to α in radian if the angle of attack considered is very small.

If v_{ijy} is the Y-component of the induced velocity at a point j due to the i-th elemental vortex then, evidently,

$$v_{jy} = \sum_{i=1}^{Nn} v_{ijy} \quad . \qquad (6)$$

Solution of the Problem

It is obvious that we are dealing with anti-symmetric load distribution on the wing arising out of operation of ailerons. Therefore, the boundary condition (6) is satisfied at all Nn pivotal points on the wing surface. This condition then results into a system of Nn linear algebraic equations which are solved simultaneously using Gauss Sideal method and all Nn unknown vortex strengths on the wing are determined.

Aerodynamic Forces

The lift L and rolling moment M_x produced by the main wing and rotating ailerons are computed using Joukowsky's theorem. The lift and rolling moment produced by main wing alone will be given by the following relations respectively:

$$L_w = \rho \sum_{i=1}^{Nn} 2s_i\Gamma_i(U_o \cos a + w_{xi})\cos a \quad ,$$

$$M_{xw} = -\rho \sum_{i=1}^{Nn} 2s_i\Gamma_i z_i(U_o \cos a + w_{xi}) \quad , \qquad (7)$$

where ρ is density of air and w_{xi} is the x-component of velocity at a point (x_i, y_i, z_i) on the wing induced by the wing vortex system and ailerons.

The resultant lift and rolling moment for the complete systemof wing and ailerons are given by

$$L = L_w + L_a \; ; \quad M_x = M_{xw} + M_{xa} \quad , \qquad (8)$$

where for ailerons we have

$$L_a = 2\rho\Gamma_a s_a[(u_{a+} - u_{a-})\cos a + (v_{a+} - v_{a-})\sin a] \quad ,$$

$$M_{xa} = -2\rho\Gamma_a s_a(u_{a+} + u_{a-})z_a \quad , \qquad (9)$$

where (u_{a+}, u_{a-}) and (v_{a+}, v_{a-}) are the velocity components chordwise and normal to the wing surface at the mid-points of the axes of up-going and down-going ailerons respectively; s_a is aileron semi-span; z_a is the distance of the mid-points of ailerons from rolling axis. The corresponding aerodynamic coefficients may be expressed in the following form:

$$C_L = \frac{2L}{\rho U_o^2 S} \; ; \quad C_{mx} = \frac{4M_x}{U_o^2 Sb} \quad , \qquad (10)$$

where S is the planform area of wing.

Numerical Calculations

Calculations of the above aerodynamic coefficients for various wing planforms at different values of angles of attack and rotational speed of ailerons have been carried out on EC-1030 computer at Indian Institute of Technology, Bombay.

The vortex distribution over the wing has been carried out for N = 16 and n = 4. The aileron chord c_a is 25% of the wing chord c_{ar} corresponding to p = 4 in equation (2). The ailerons extend in spanwise direction from strip number 5 to strip number 3 on the starboard part of the wing and from strip number 12 to strip number 14 on the port side of the wing. The position of axes of rotation of ailerons is given by Δx = $0.05\,c_{ar}$, $-\Delta y = 0.13\,c_{ar}$ (see Fig.1) along X and Y axes which are reported to be the best position of flap axis[1]. The circulation Γ_a of the rotating ailerons is represented by[10]:

$$\frac{\Gamma_a}{U_o} = \frac{1}{4}\pi c_{ar}\, E \quad , \qquad (11)$$

where

E = $K(U/U_o)$;
U = peripheral speed of the rotating ailerons;
K = effectiveness coefficient depends on the geometrical parameters of rotating profile, taken equal to 0.25 based on experimental results[10] for conventional profiles.

The calculations of aerodynamic coefficients C_L and C_{mx} using relations (10) are carried out for the values of a varying from 3° to 15° and those of the ratio U/U_o = 1, 2, 3, & 4 for the wing planforms of aspect ratios 7 and 3 with taper ratios of 1 and 0.5 for each wing planform. The results so obtained are shown in Figs. 2, 3, 4 & 5.

Discussion of the Results

The rotating ailerons seem to be promising rolling control surfaces. The curves in Figs. 2-5 suggest that the rolling moment coefficient obtained with the rotating ailerons of the span equal to 22% of the wing span is higher than that

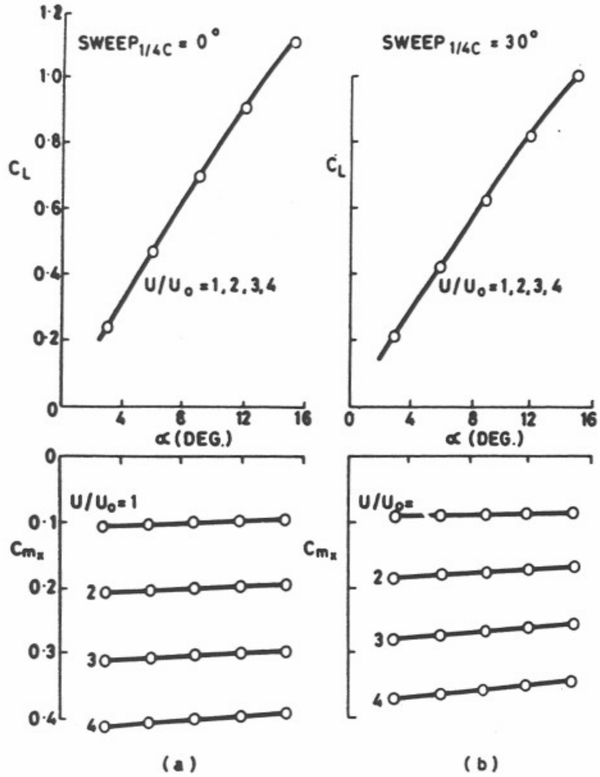

Fig.2. Curves of C_L Vs α and C_{m_x} Vs α with $U/U_o = 1, 2, 3, 4$ for the wing of AR = 7, taper ratio = 1.

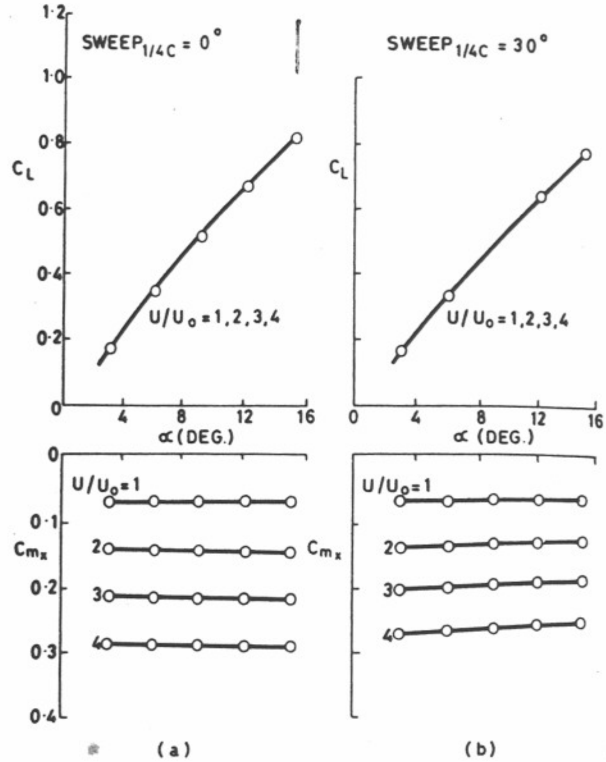

Fig.3. Curves of C_L Vs α C_{m_x} Vs α with $U/U_o = 1, 2, 3, 4$ for the wing of AR = 7, taper ratio = 0.5.

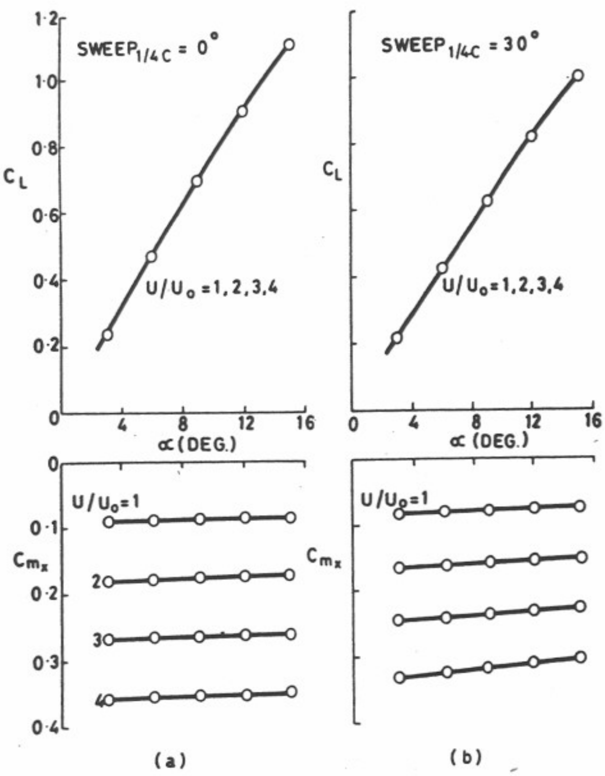

Fig.4. Curves of C_L Vs α and C_{m_x} Vs α with $U/U_o = 1, 2, 3, 4$ for the wing of AR = 3, taper ratio = 1.

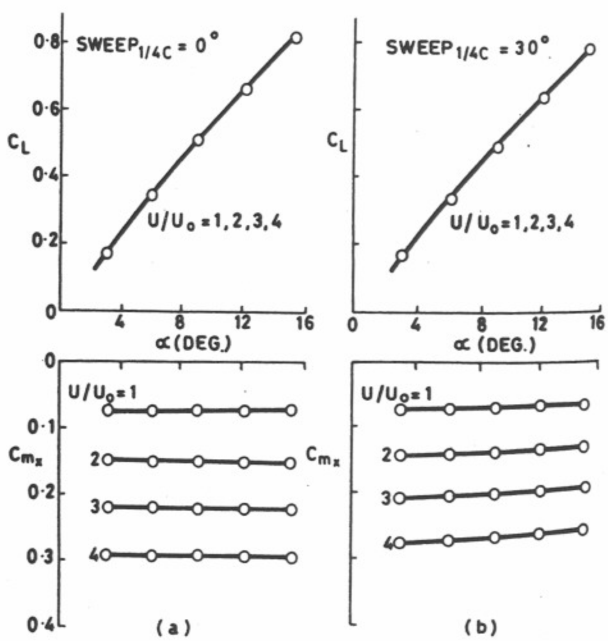

Fig.5. Curves of C_L Vs α and C_{m_x} Vs α with $U/U_o = 1, 2, 3, 4$ for the wing of AR = 3, taper ratio = 0.5.

for the conventional ailerons of almost double the span. The conventional design value of rolling moment coefficient for most of the subsonic airplanes is about 0.05 to 0.6[11]. It is possible to achieve this value of C_{mx} with rotational speed of ailerons less than the flight speed and aileron span of about 20%.

It is seen from the results that the effectiveness of such ailerons decreases with reduction in the wing aspect ratio. The value of C_{mx} produced by these ailerons also reduces with increase in wing sweep. However, this effect seems to be milder for rotating ailerons than that for the conventional ones for which C_{mx} drops to about 82% at sweep of 30° from its value for zero sweep[11]. These results indicate that there can be significant improvement in rolling capability of aircraft for a given wing planform and size of ailerons. In other words, it is possible to reduce the aileron size for required rolling capability.

The results of calculations presented in Figs. 2-5 shows that lift remains practically constant during operation of ailerons. Experimental evidences indicate that the drag of down going part of wing is considerably high compared to that of other half of wing if rotating ailerons are used[1]. Thus, it gives pro-yaw characteristics.

The work about effectiveness of this type of ailerons with non-linearity effect, structural deformation of wing influencing aerodynamic characteristics, compressibility effect etc. is being carried out.

References

1. Crabtree, L. F., "The rotating flap as a high lift device", ARC CP No. 480, Great Britain, April 1957.
2. Neumark, S., "Rotating aerofoils and flaps", J. Roy. Aero. Soc., Vol. 67, No. 1, 1963, pp. 47.
3. Ryabushinskii, M. D., "Investigation of a rotating flap as a device for increasing lift of wing", Compt. Rend. Acad. of Sci., 1957, 224, No. 13, pp. 1687-91 (French).
4. Patel, T. S., "Calculation of lift force on the wing with a rotating flap", Aviational Technique, Izvestia VUZ, USSR, Kazan, No. 4, 1970, pp. 10-15 (Russian).
5. Patel, T. S., "Ground effect on the thin aerofoil with rotating flap", VINITI, Academy of Sc., USSR, No. 4993-72, 1972 (Russian).
6. Patel, T. S., "Wing profile with a rotating flap in non-uniform flow near wall", Mechanics and Mathematics, MSU, Moscow, No. 1, 1973, pp. 16-23 (Russian).
7. Patel, T. S., "Wing profile with a rotating flap in shear flow", Vestnik-Mechanics and Mathematics, MSU, Moscow, No. 2, 1974, pp. 97-105 (Russian).
8. Byelotserkovskii, S. M., "Thin lifting surface in subsonic flow of gas", Nauka Publication, Moscow, USSR, 1965 (Russian).
9. Multhopp, H., "Methods for calculating the lift distributionof wings", ARC R & M No. 2884, Great Britain, 1955.
10. Patel, T. S., "About the effectiveness of a rotating flap", Ph. D. Thesis, Department of Aeromechanics and Gasdynamics, Moscow State University, USSR, 1973 (Russian).
11. Badyagin, A. A., et al., "Design of Airplanes", Mashinostroyenie Publication, Moscow, USSR, 1972 (Russian).