Wall Interference Effects in Wind-Tunnel Testing of STOL Aircraft

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A problem associated with the wind-tunnel testing of very slow flying aircraft is the correction of observed pitching moments to free air conditions. The most significant effects of such corrections are to be found in the domain of flight between high speed and the STOL approach. The wind-tunnel walls induce interference velocities at the tail location different from those induced at the wing, and these induced velocities also alter the trajectory of the trailing vortex system. The relocated vortex system induces different velocities at the tail from those experienced in free air. A method of calculating the interference velocities is presented in which the effects of the altered wake location are included, as well as the wall-induced velocities. Results are presented comparing the computed tail interference angles, with and without including the effect of the vortex wake relocation, which show the importance of the wake shift. In some cases, the tail angle corrections are reduced to zero and may even change sign. It is concluded that, to calculate correctly the interference velocities affecting pitching moments, the effects of vortex wake relocation must be included.

Nomenclature

$\mathcal{R}$ = wing aspect ratio
b_g = wing geometric span
C = wind-tunnel cross-section area
C_L = wing lift coefficient
e = distance downstream to wake roll-up
S = wing planform area
V_x = x component of velocity, positive downstream
V_y = y component of velocity, positive upward
$\alpha_{\text{free air}}$ = $\tan^{-1}(V_y/V_x)$ based on free-air velocities
α_{tunnel} = $\tan^{-1}(V_y/V_x)$ based on wind-tunnel velocities
$\Delta\alpha$ = $\alpha_{\text{free air}} - \alpha_{\text{tunnel}}$
δ = interference factor, $(C/S\,C_L)\Delta\alpha$
δ_t = δ evaluated at tail location
δ_w = δ evaluated at wing location

Introduction

EXPERIENCE of recent years has demonstrated that the difficult stability and control problems encountered by V/STOL aircraft in transition are still with us and, indeed, show very little sign of disappearing. Linear aerodynamic theory is inadequate to predict the flowfields largely because of the assumption of small deflections. Flow separation from some parts of the lifting system is often an unavoidable part of the problem. Separation effects are, of course, configuration dependent and sensitive to Reynolds number as well as to local flow distortions caused by interference velocities from the ground or from other parts of the aircraft. Consequently, the wind tunnel continues to be a most necessary tool to insure the safety of slow flight. Unfortunately, the classical theory of wind-tunnel wall interference is also inadequate to predict the effects of testing large downwash lifting systems, since it too is based on linear, small deflection aerodynamic theory.

A number of working rules were put forward early in order to minimize the problem, notably, the widely used requirement that the model should be small enough that the angle correction computed by linear theory be less than 2°. Figure 1 shows how the use of this limit results in very small models when the approaching transition lift coefficients. The sensitivity of all the aerodynamics to Reynolds number and the difficulty of physically constructing models of sufficient fidelity at these small scales has motivated an insistent push both toward larger wind tunnels and to better theoretical treatment of the interference problems which would permit larger model-to-tunnel size ratios. Several V/STOL wind tunnels have been constructed or are under construction at this time, having test-section sizes limited only by the funding available and still not approaching the size desired by the aerodynamicist.

Outstanding in the efforts to provide adequate theoretical treatment of this problem has been the contribution of Heyson in a series of papers,[1–3] of which the principal one is Ref. 3. He has developed a method of handling the helicopter or a wing by representing the lifting system and its wake as an infinite string of point doublets extending along a straight line to the tunnel floor. The slope of the line is determined by momentum theory, modified to give a good estimate of the location of the vortex wake. The tunnel walls are made to be reflection planes by positioning an infinite set of images outside the tunnel walls, as in the classical theory, and computing the interference velocities due to these images. This treatment has been shown to yield good results at the im-

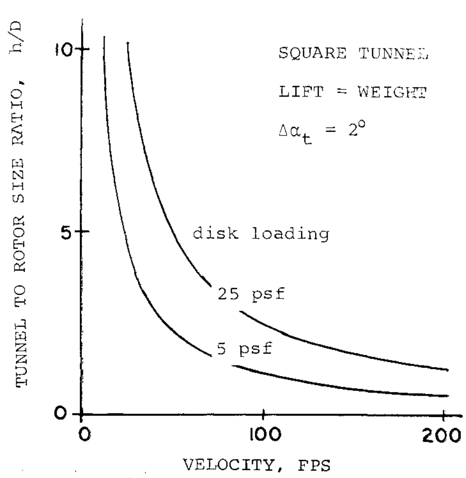

Fig. 1 Tunnel size required to limit wall interference to 2°.

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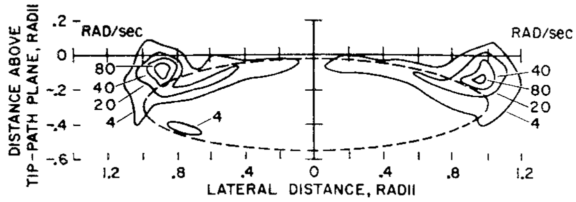

Fig. 2 Vorticity distribution at 7% of radius behind a rotor. Momentum angle is 15°.

mediate vicinity of the model when applied to lift and drag, and when the model-to-tunnel size ratio is not too large.[4–6]

Attempts to apply these or any other correction methods to pitching moments have not always fared so well. The interference velocities are rather large and are not linearly distributed along the length of a model. If some part of the model, such as a tail or rotor blade, is operating at or near its stalling angle, the changes in pitching moment may be badly misleading, and not correctable by any simple linear method. Heyson has discussed this problem in Ref. 7. Nevertheless, the calculation of pitching moment interference velocities remains an important first step which must be taken before corrections can be attempted.

The Problem

The problem which has not yet been approached is that of determining the proper trajectory of the vortex wake from the lifting system and then accounting for all of the effects that result from the presence of the wind-tunnel walls. It is evident that a vortex wake trailing from a high downwash system must follow a different trajectory in the tunnel than it would in free air because of the wall constraints. The relocated vortex wake will result in a different downwash field which will be important in the case of a long model such as a helicopter rotor or one having a tail. If the tail of the model is normally above the wake, this direct effect is in the opposite direction and, in some circumstances, may be larger than the reflection effects of the walls themselves.

In order to evaluate these effects, it is necessary to solve for the trajectory of the vortex wake of a lifting system, both in the tunnel and in free air, and to compare their velocity fields. The entire difference between these fields should be charged to wind-tunnel interference. The following sections present a method for accomplishing this calculation.

A Method of Solution

Choice of a Lifting-System Representation

The following argument suggested that changes in location of the vortex wake, due to tunnel wall constraints, would affect the pitching moments most strongly for moderate wake deflection. The induced velocity at a point nearly in the plane of a pair of trailing vortices and midway between them is only changed slightly by a moderate vertical shift of location, since the radius of action changes only by the cosine of a small angle. Likewise, if the point is of the order of the vortex span or more away from the plane of the trailing pair, the induced normal velocities are small and changes due to location change are also small. However, when the point is at some intermediate distance from the plane of the trailing pair, there is a maximum effect due to change in the location, and this corresponds to the case of the moderate wake deflection angles. The moderate deflections are typical of slow flight of STOL aircraft or of VTOL aircraft approaching transition from the high-speed side. Evidence is growing from the experimental work of Rae[8] that testing in the large deflection angle regions of transition to hover may be badly misleading due to recirculation effects. Consequently, one would expect that the most immediately useful results from predicting interference effects would lie in the moderate downwash region.

A configuration suitable for flight in this region, and one that is most amenable to analysis, is that of a simple straight wing having some sort of boundary-layer control so that it can develop large values of circulation. Such a lifting system can be represented by a single lifting line and a trailing pair of vortices. For the high-lift/low-aspect-ratio systems we are interested in, the single trailing vortex filament pair is a better representation than a vortex sheet because the sheet rolls up into a cylindrical core of vorticity very quickly. It is easily shown that a cylinder of uniformly distributed longitudinal vortex filaments has the same effect at an external point as a single filament having the same total strength and located on the axis of the cylinder.

Evidence about the rate of rolling-up of the vortex sheet is incomplete, although the problem has been studied for a long time. Sprieter and Sachs[9] have added to the theoretical developments of Kaden[10] and Westwater[11] and report the rolling-up distance as a fraction of span to be $e/b_g = 0.28\, \mathcal{R}/C_L$. Unpublished calculations made using the method of Westwater, but in finer detail using an electronic computer, show that the rolling up is much faster than he reports. Since the developments referred to here are based on two-dimensional flow, the forecast can only be regarded as approximate. Certainly the three-dimensional nature of the roll-up, which is inherent in the very short roll-up distances, must alter the local flowfield appreciably; these effects are ignored here. Nevertheless, the order of the distance is taken as essentially correct and it is assumed for the purpose of this paper that it is adequate to let a single vortex pair represent the trailing wake. It is recognized that this will introduce certain errors, but the results will be qualitatively correct. More quantitative results will depend on more detailed analyses of a particular configuration. It is interesting to note, however, that data taken in the wake of a helicopter rotor at moderate downwash angles exhibit the same general pattern of a rapid roll-up into two distinct vortex cores. Figure 2, taken from Ref. 12 for a rotor having a momentum downwash angle of 15°, shows two clearly defined cores already well developed at a plane only 7% of the rotor diameter downstream from the rotor trailing edge.

Solution of the Wake Trajectory in Free Air

Figure 3 shows a sketch of the vortex wake representing a plane elliptical wing and indicates the induced velocity due to an element of the vortex acting at an arbitrary point. The

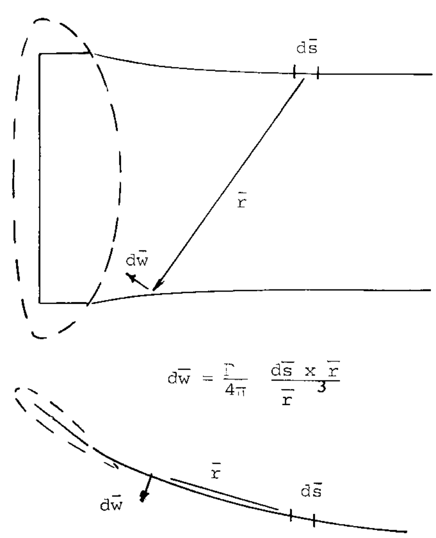

Fig. 3 Velocity induced at a point by a vorticity element.

element of induced velocity is evaluated by the Biot-Savart law, and, when integrated over the entire wake, the direction of the flow at a point can be determined. When the direction of the flow is known everywhere on the trailing vortex, a first iteration of its trajectory can be found, for it must be everywhere parallel to the local flow. More than one iteration must be made, and this is most conveniently performed using a digital computer.

To facilitate the solution, the vortex system is broken into a series of short, straight line segments. The bound vortex lies on the quarter chord line and has a span of $\pi/4$ times the geometric span, which is appropriate for representing an elliptical wing. The first trailing segment lies in the plane of the wing, extending from the bound vortex tip to the trailing edge. Downstream vortices are assumed to spring from that point and are divided into segments whose length is approximately $\frac{1}{10}$ of the vortex span. The angle of the first segment, being in the plane of the wing, is determined by adding the induced angle of attack and the effective angle of attack at the plane of symmetry. The induced angle of attack of the wing is computed at the lifting line by summing the induced velocities of all the trailing segments and adding them vectorially to the remote velocity. The effective angle of attack is determined according to the assumption of two-dimensional flow at the plane of symmetry by combining the local velocity vector with the velocity induced by the bound vortex at the three-quarter chord point.

The direction of each downstream element in turn is calculated by summing the individual velocities due to all other elements at its own upstream end. This direction is used to determine the coordinates of the downstream end of the segment, the entire string of segments downstream from that point is translated so that it stays attached, and the next segment direction is determined. Thus, the wake is moved into place by sweeping along its length from the wing aft in several iterations.

When a vortex line lies in a plane and follows a path of varying curvature, it induces on itself velocities normal to the original plane which are nonuniform. The filament which leaves the wing at a fixed location curves upward from its angle of departure, and so each downstream section experiences an inward deflection from its own upstream elements. This vanishes as the trajectory straightens out, but it must leave the final straight wake at a smaller vortex span than it had on leaving the wing. The iteration process must then allow for this lateral freedom as well as for the vertical motion of the wake. When the previously described iteration process was first attempted, simultaneously calculating both downward and inward deflections, the computation became unstable after only a few iterations. This instability was avoided by a double iteration process. First, one pass was made calculating only downward deflections, and then a second was made allowing only horizontal or inward deflections. By this stepwise process, a trajectory can be found which converges after only 3 or 4 such double passes, and which seems to be free of instability.

It should be noted that the vortex line is physically unstable in that curvature of the line causes more self-induced curvature, eventually producing vortex rings. An example may be observed in the contrails of jet aircraft where the engine exhaust is drawn into and makes visible the trailing vortex pair. This instability could be accentuated by round-off errors in the computing machine and places a limit on the number of times an iteration can be carried out.

Representation of the Wind-Tunnel Walls

In the classical theory, and in more recent work, the walls of a wind tunnel are represented by a system of vortex images outside the walls such that each wall, like a mirror, is a plane of symmetry and is, therefore, parallel to the local flow. This approach assumes that the walls are infinitely long and, furthermore, is restricted to cases for which such image systems can be found, such as the rectangular test section. In an effort to remove these restrictions, the method reported in Ref. 13 was developed.

In this method, the image concept was abandoned and the walls were represented by a vortex lattice. The strength of each element of the lattice is found by simultaneously requiring that the normal component of the resulting flow vanish at control points in the center of each such lattice element. The method was tested against classical theory and found to give excellent agreement. Interference factors have also been developed for some irregularly shaped tunnels, and the method has proven to be convenient to use with any wake trajectory since it avoids the problem of changing the image system for each different trajectory. Consequently, the vortex lattice method was chosen to represent the tunnel walls for the present problem.

The Combined Solution

The solution for the wake trajectory in the wind tunnel is an iterative combination of the free-air trajectory solution and the wind-tunnel vortex lattice method. The model, represented by a horseshoe vortex, is placed inside a vortex lattice tube representing the tunnel and is given an initial vortex strength and an undeflected wake. A solution is found for the wall vorticity required to satisfy the boundary conditions exactly as described in Ref. 13. The wake location is iterated several times to find its equilibrium trajectory under its own influence and the influence of the tunnel-wall vorticity appropriate for the undeflected wake. A second solution is then made for the tunnel-wall vorticity required by the model in its new location, followed by a second iteration of the wake trajectory. Usually only 3 or 4 cycles are necessary since the two systems do not interact strongly for moderate deflections.

Determination of the Interference Factors

In order to find the total interference, one must compare the flow patterns of the same system operating at the same circulation and remote velocity, both inside the tunnel and in free air. When the solutions for the two systems are complete, the program yields the velocity field at any desired point in the tunnel, as well as the separate contributions to that field by the wall vortex lattice and by the lifting system.

The interference velocities then are defined correctly by stating that the difference between the velocity at a point in the tunnel and the velocity at the same point in free air is the total interference velocity. This interference velocity may be expressed as a change in the direction of flow and a change in magnitude. For the examples presented here, it was found that the magnitude of speed changes was small (about 3% in the extreme cases) and of much less importance than the angle changes. Consequently, only the angle changes are presented in this report. They were arrived at by computing the flow angles at a point, both in the tunnel and in free air, and comparing them:

$$\Delta\alpha = \alpha_{\text{tunnel}} - \alpha_{\text{free air}}$$

These interference angles are then presented in a familiar form in terms of the factor δ used in the classical wind-tunnel interference calculations. The equation is

$$\Delta\alpha = \delta(S/C)C_L$$

where the $\Delta\alpha$ is the difference computed previously. The angles are not small enough to permit use of the small angle approximation, so they are defined using the computed local velocity components. Thus the interference factor presented

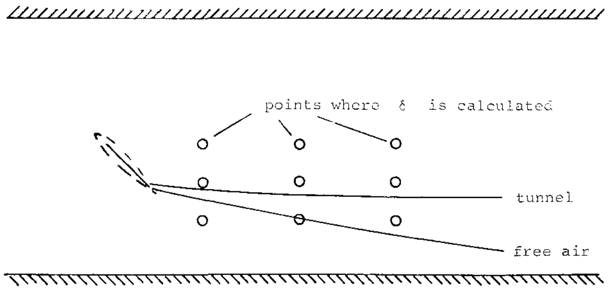

Fig. 4 Effect of tunnel walls on vortex wake trajectory in a 1:1.5 closed tunnel. Vortex span = ½ tunnel width. $C_L/\mathcal{R} = 0.9$.

is given as

$$\delta = (C/SC_L)[\tan^{-1}(V_y/V_x)_{\text{tun}} - \tan^{-1}(V_y/V_x)_{\text{f.a.}}]$$

Results

Calculations are presented for a plane wing at lift coefficients approaching the maximum theoretically possible for an unpowered system. In order to achieve the highest wake deflection angles, the aspect ratio of sample calculations was taken at 3.0 so that high $C_L/\mathcal{R}$ values could be attained. The wing vortex span was taken as one half the tunnel width in a rectangular tunnel of height-to-width ratio 1:1.5.

Figure 4 shows the trajectory of the wake in free air and in the wind tunnel for the sample wing. The difference in location of the wake in the tunnel is evident. In Fig. 5, the value of the interference factor δ is shown as a function of $C_L/\mathcal{R}$ at the location of the wing and for three downstream locations assumed to be on the tunnel centerline.

The tail interference angle is taken as the difference between the angle at the wing and at the tail, and presented as the difference between the values of δ at these two points. Figure 5 also shows the tail interference factor $(\delta_t - \delta_w)$.

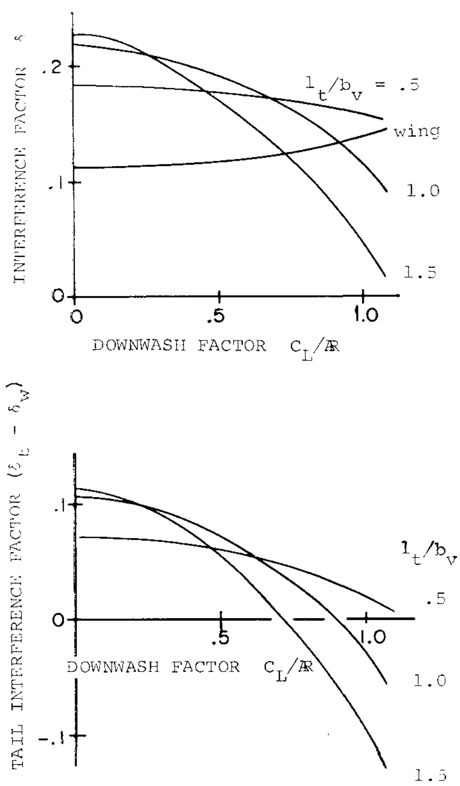

Fig. 5 Interference factors at wing and tail including wake relocation effects. Tail on tunnel centerline.

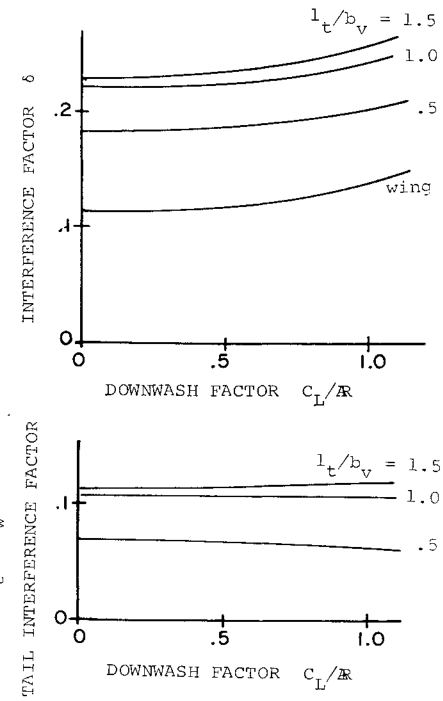

Fig. 6 Interference factors at wing and tail using only wall-induced effects. Tail on tunnel centerline.

This curve shows that, for the geometry chosen, the pitching-moment corrections may become small or even negative at the higher lift coefficients.

In order to demonstrate the effect of the wake shift, Fig. 6 was prepared for comparison with Fig. 5. The same factors

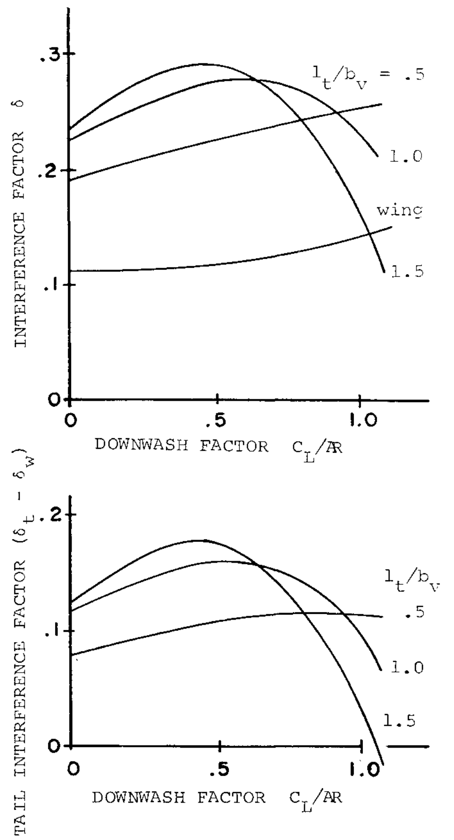

Fig. 7 Interference factors at wing and tail at 0.2 b_v below tunnel centerline.

were calculated, but the contribution of the deflected wake was left out. The interference angle was calculated using only the velocities induced by the wall vortex lattice. The wake location as computed in the tunnel was used, so these results accurately represent interference velocities based upon only the wall-induced effects. Figure 6 also shows the tail interference factors calculated using only the wall induced velocities. The importance of including the direct effects of the wake relocation is shown when Fig. 6 is compared with Fig. 5.

Tail location is an important parameter, for, if the tail is initially below the vortex wake in free air, then the wake shift upward in the tunnel will accentuate the wall induced upwash. Figures 7 and 8 show this effect for tail heights of 0.2 and 0.4 times the vortex span below the wing, as well as the reversal which takes place when the wake shifts past the tail location.

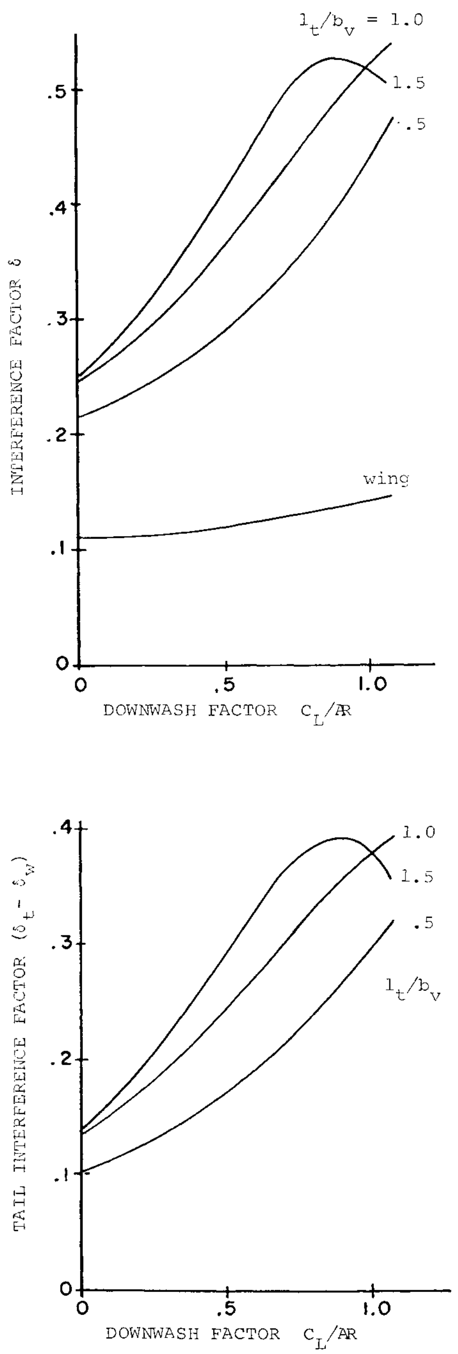

Fig. 8 Interference factors at wing and tail at 0.4 b_v below tunnel centerline.

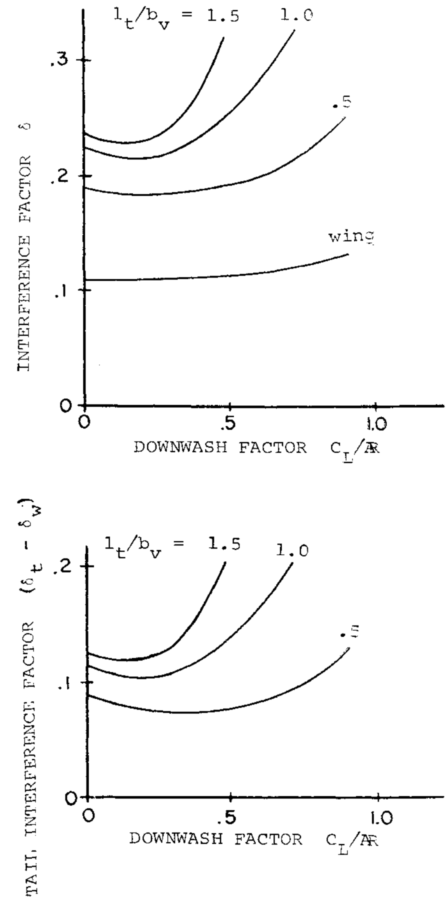

Fig. 9 Interference factors at wing and tail. Displacement included. Tail height 0.2 b_v above wing plane.

In the preceding examples the interference angle factors were calculated at fixed locations in the tunnel, and do not necessarily represent a physically realizable vehicle. The results can be interpreted to represent a tilt-wing type vehicle in which the body is constrained to a constant angle of attack.

For the case where body attitude changes it is necessary to calculate and compare flow angles at the tail, in free air, and in the tunnel, at angles of attack appropriate for the same wing circulation. An example is presented in Fig. 9 for a case where wing and tail are fixed to a body and rotate as a unit. The tail is located above the plane of the wing (0.2 of the vortex span) and three tail lengths are shown. The interference factor shows a minimum where the tail passes through the height of the vortex wake. The large variations of the factor indicate the importance of accounting for the wake shift and for actual tail position.

Concluding Remarks

The results presented demonstrate that the direct effects of wake relocation are of primary importance in the determination of pitching-moment interference, particularly for the moderate downwash angles produced by high lift wings or similar lifting systems at speeds above transition from hovering flight. Since the wake shift effect is dependent on the initial wake location, the results are not likely to be better than the computation of free-air downwash, itself a formidable problem. The effects are certainly very much configuration-dependent. For example, blown flaps or a rotor may be used to generate the lift while remaining at near zero angle of attack, and this results in a different starting point for the trailing wake. Work is continuing in an effort to accommodate other configurations.

References

[1] Heyson, H. H., "Wind Tunnel Wall Interference and Ground Effect for VTOL-STOL Aircraft," *Journal of the American Helicopter Society*, Vol. 6, No. 1, Jan. 1961.

[2] Heyson, H. H., "Jet-Boundary Corrections for Lifting Rotors Centered in Rectangular Wind Tunnels," TR R-71, 1960, NASA.

[3] Heyson, H. H., "Linearized Theory of Wind Tunnel Jet Boundary Corrections and Ground Effect for VTOL-STOL Models," TR R-124, 1962, NASA.

[4] Grunwald, K. J., "Experimental Study of Wall Effects and Wall Corrections for a General-Research V/STOL Tilt-Wing Model with Flap," TN D-2887, 1965, NASA.

[5] Heyson, H. H. and Grunwald, K. J., "Wind Tunnel Boundary Interference for V/STOL Testing," *Conference on V/STOL and STOL Aircraft*, NASA SP-116, 1966, pp. 409–434.

[6] Staff of Powered-Lift Aerodynamics Section, NASA Langley Research Center, "Wall Effects and Scale Effects in V/STOL Model Testing," *AIAA Aerodynamic Testing Conference*, AIAA, New York, March 1964, pp. 8–16.

[7] Heyson, Harry H., "Wind Tunnel Wall Effects at Extreme Force Coefficients," *Annals of the New York Academy of Sciences*, Vol. 154, Article 2, Nov. 1968, pp. 1074–1093.

[8] Rae, Wililam H., "Limits on Minimum-Speed V/STOL Wind Tunnel Tests," *Journal of Aircraft*, Vol. 4, No. 3, May–June 1967, pp. 249–254.

[9] Spreiter, J. R. and Sacks, A. H., "The Rolling Up of the Trailing Vortex Sheet and Its Effect on the Downwash Behind Wings," *Journal of the Aeronautical Sciences*, Vol. 18, No. 1, Jan. 1951.

[10] Kaden, H., "Aufwicklung Einer Unstabilen Unstetigkeitsflache," *Ingenieur-Archiv*, II Band, 1931, pp. 140–168.

[11] Westwater, F. L., "Rolling Up of the Surface of Discontinuity Behind an Aerofoil of Finite Span," R and M 1692, 1935, British Aeronautical Research Council.

[12] Heyson, H. H. and Katzoff, S., "Induced Velocities Near A Lifting Rotor with Non-Uniform Disk Loading," Rept. 1319, 1957, NASA.

[13] Joppa, R. G., "A Method of Calculating Wind Tunnel Interference Factors for Tunnels of Arbitrary Cross-Section," CR-845, July 1967, NASA.