Leading-Edge Vortices and Shock-Detachment Flow over Delta Wings

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A review of the pertinent literature and a discussion of various flow phenomena, how they arise, and their relevance to highly swept wings at cruise conditions, is presented. The relative significance of each of these flow phenomena, their interaction, and their variation with the wing operating conditions have been examined. Also established is an understanding of the reasons for the differences between experiment and theory with regard to the reduction in drag due to the lift of supersonic wings by the use of wing camber and twist. Failure to attain experimentally the reductions in drag due to the lift is traced to the inadequacy of the theory for predicting wing pressure distributions. The linear theory neglects such significant influences as 1) three-dimensional separation and subsequent vortex formation, 2) detachment of the leading-edge shock waves, and 3) the presence of shock waves on the wing surfaces, etc. It is suggested that an effort be undertaken to develop analytical representations of the essential features of the flow which are observed in the experiments.

Introduction

SOME form of flow separation is encountered over a wing in flight which is not accounted for in the theory and thus is responsible for large discrepancies between the theory and experiment. The linear theory is based on the assumption of small disturbances, but the wing twist and camber predicted by the optimization procedures which utilize linear theory is so extreme that it causes the flow to violate this assumption. Discrepancies between the linear theory and experimental results occur for flat wings as well as for other wings. Love[1] and Lampert[2] have independently tested families of delta wings at supersonic Mach numbers. Both indicate that the lift slope becomes low relative to the theoretical values as the Mach number normal to the leading edge approaches one.

An experimental investigation which reveals how the leading-edge vortex phenomenon influences the wing pressure distribution is reported by Drougge and Larson.[3] The results clearly indicate the existence of three-dimensional flow separation from the upper surface and the formation of relatively concentrated regions of vorticity streaming back above the wing surface. A very good insight into the various flow phenomena which can occur on the three-dimensional wings is provided by Rogers and Hall.[4] Kuchemann[5] provides additional insight into the various flow phenomena which can become significant in the flow of a supersonic freestream about three-dimensional bodies. Maskell[6] concluded that, when separation occurs, there are two basic viscous flow elements which can exist in the resulting flow. These flow elements are the free vortex layer and the separation bubbles, each of which is identifiable by characteristic flow patterns in the family of limiting streamlines on the solid surface. Lock and Rogers[7] present a semiempirical procedure for designing transonic wingbody combinations which are reasonably free from surface shocks and flow separations. The idea is to maintain quasi-two-dimensional flow over the wing with specified chordwise pressure distributions known to be attainable without developing surface shocks. Lee[8] has tested a series of delta wing models with cross sections designed to capitalize on the separated vortex flow pressure distribution by inclining forward as much as possible those portions of the lifting surface carrying the greatest loads.

Fraenkel and Watson[9] have attempted to solve the theoretical inviscid problem of detachment of the shock from the leading edge of three-dimensional wings. They consider the problem of a thin, conical wing with sonic leading edges. Solutions are obtained for 1) the entire flowfield excluding the neighborhood of the shock cone and the wing leading edges, 2) the neighborhood of the shock cone but excluding the neighborhood of the wing. However, they have been unable to solve the problem in domain—3) the neighborhood of the leading edge by analytic means even for a particular case. Maslen[10] and Fowell[11] have presented solutions of the exact nonlinear equations for a conical wing at angle of attack in a supersonic freestream. However, neither method can treat the wing with detached leading-edge shock. Clarke and Wallace[12] present an analytical method which is applicable only with an attached leading-edge shock. They have developed a uniform second-order solution for supersonic flow over delta wings through the application of reverse-flow integral method. It appears that the two-dimensional inviscid problems can be handled reasonably well. However, there appears to be no way to relate these problems to the analogous three-dimensional problem.

Separation Phenomena and Vortex Formation

The following is a brief discussion of the flow separation phenomena neglected in the linear theory which are considered to be responsible for its inability to predict the three-dimensional wing pressure distributions. For the subsonic leading-edge lifting wing, the attachment line of the flow is below the leading edge on the lower surface of the wing and thus there is a flow around the leading edge from the lower surface. The expansion of the flow in going around the leading edge results in a high negative pressure peak and a subsequent steep pressure recovery. This pressure recovery is a strong adverse pressure gradient to the three-dimensional boundary layer and can cause it to separate from the surface. When this separation occurs, the boundary layer leaves the wing surface along a swept separation line and rolls into a region of concentrated vorticity which is swept back over the surface of the wing. This separation and vortex formation (i.e., leading-edge vortex) apparently always occurs for highly

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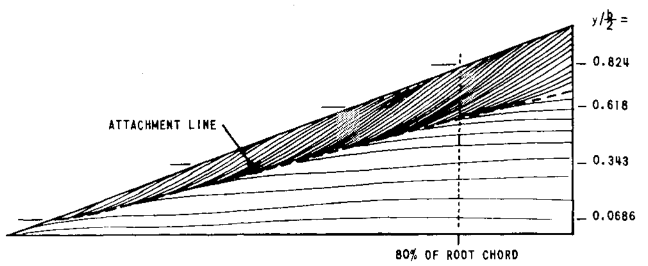
swept lifting wings with subsonic leading edges. The effect of this vorticity is to alter the velocity distribution and thus the pressure distribution over the wing.

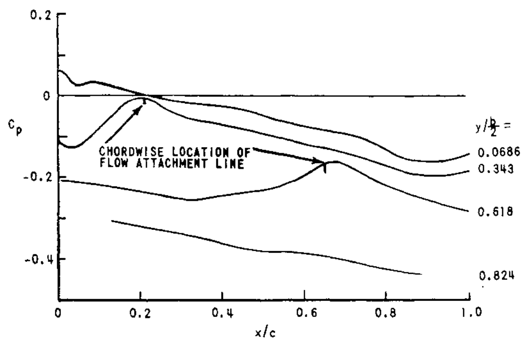
For the supersonic leading-edge wing, the inviscid theory considers an oblique compressive shock to be attached to the leading edge so that there is no flow around it from the lower to the upper surface. In reality, when the flow component normal to the leading edge is only slightly supersonic, the slope of the wing surface can exceed the turning capability of an oblique shock. Under these circumstances, the leading-edge shock becomes detached and locally the flow behind the shock negotiates part of the turn in the resulting subsonic component region which then exists in the vicinity of the leading edge. Furthermore, detachment of the leading-edge shock of a lifting wing causes the attachment line of the flow to move to the lower surface of the wing which in turn allows for flow around the leading edge, separation, and vortex formation, as in the case of the subsonic leading-edge wings. Detachment of the leading-edge shock thus permits the formation of leading-edge vortices on lifting supersonic leading-edge wings. However, the adverse pressure gradient due to the pressure rise across shock waves can have a more significant effect by separating the three-dimensional boundary layer and thereby changing the flow and pressure distribution from that predicted by linear theory. This type of separation can also result in the formation of a concentrated vortex under the proper conditions. Evidently, all that is required to precipitate this type of vortex formation is that the boundary-layer separation line be sufficiently swept with respect to the local flow. The strength of this vortex and its rate of growth depend not only on the wing geometry and freestream velocity but also on such subtle aspects as the strength and relative orientation of the boundary-layer vorticity with respect to the separation line. In each of the preceding descriptions of flow possibilities about wings with highly swept leading edges in supersonic freestreams, three-dimensional boundary-layer separation was involved. It is generally conceded that the pressure gradient required to separate three-dimensional boundary layers is considerably less than that for two-dimensional boundary layers.

Qualitative Explanation of the Nature of Influence of the Leading-Edge Vortices

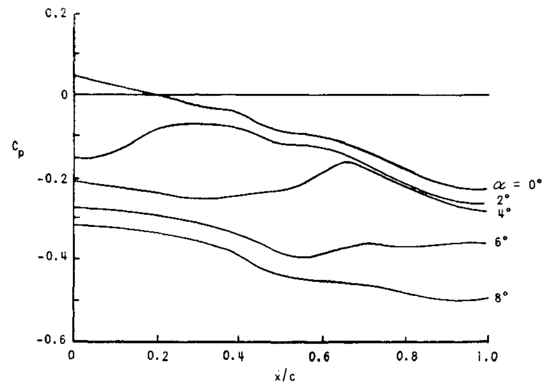
Figure 1a is a sketch of the upper-surface flow pattern on a sharp-edge wing at an angle of attack of 5°. The flow has separated at the sharp leading edge and the vortex sheet rapidly rolled up into a leading-edge vortex which reattaches



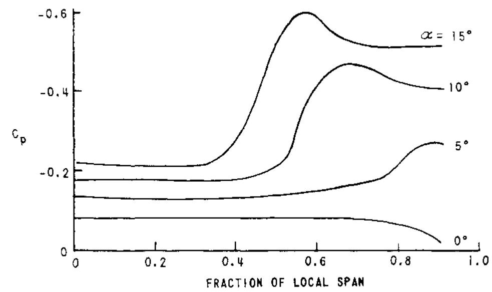
a) Upper-surface flow pattern obtained at $\alpha = 5°$, $M = 1.50$, for a delta wing with a sharp-edged biconvex airfoil and leading edges swept back 70°



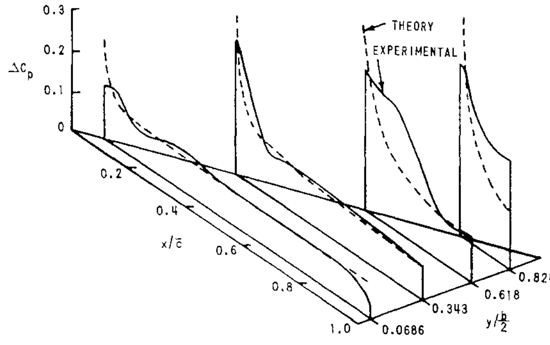
b) Chordwise distributions of upper-surface pressure coefficients at four spanwise positions for $\alpha = 4°$, $M = 1.50$



c) Chordwise distributions of upper-surface pressure coefficients at spanwise position, $y/(b/2) = 0.618$, for five angles of attack, $M = 1.50$



d) Spanwise distributions of upper-surface pressure coefficients at four angles of attack and a chordwise location of 80% of the root chord, $M = 1.50$



e) Chordwise loading distributions at four spanwise positions for $\alpha = 4°$, $M = 1.50$

Fig 1. Flow over a typical delta wing.

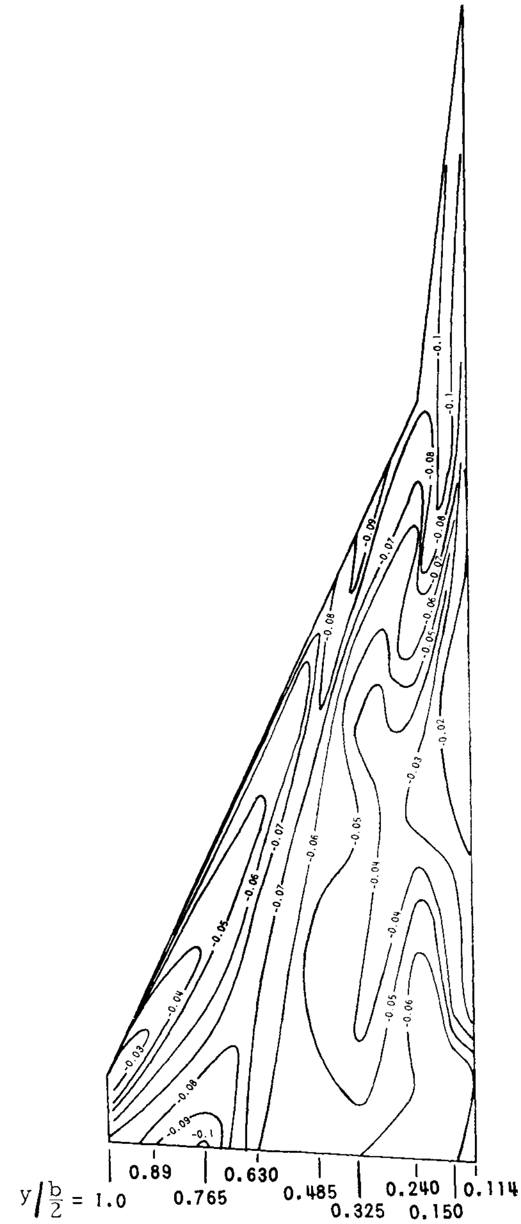


Fig. 2a) Contours of measured upper-surface pressure coefficients at $\alpha = 4.39°$, $M = 2.68$

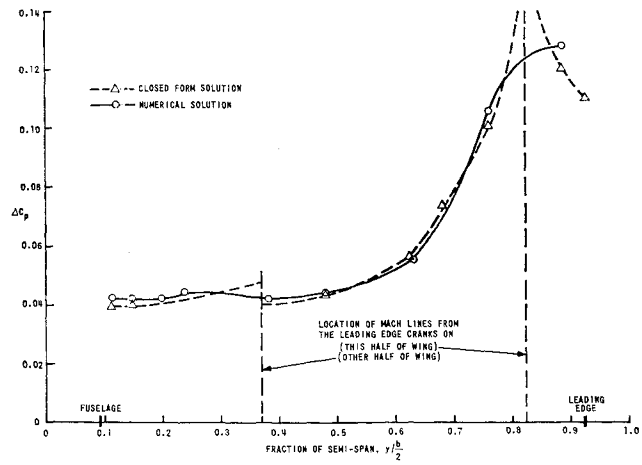


Fig. 2b) Comparison of computed spanwise loading distributions due to angle of attack at 82% of the main delta root chord for $\alpha = 4.48°$, $M = 2.68$

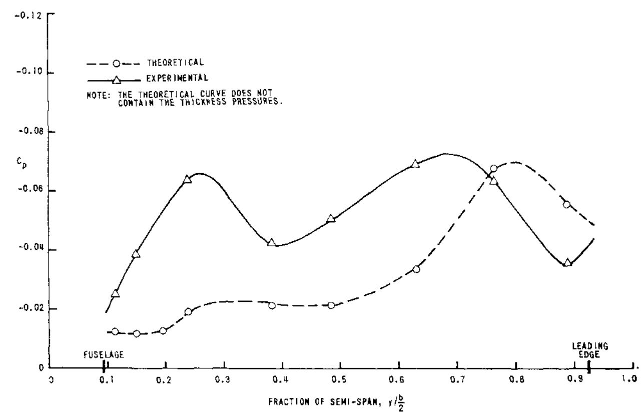


Fig. 2c) Comparison of theoretical and experimental spanwise distributions of upper-surface pressure coefficients at 82% of main delta root chord for $\alpha = 4.39°$, $M = 2.68$

Fig. 2 Flow over a typical double delta wing.

the flow approximately along the dotted line indicated. From the attachment line, the flow divides, part of it moving in a generally downstream direction and the remainder moving toward the leading edge where it must again separate. At low angles of attack, this secondary separation line effectively coalesces with the primary one at the leading edge. However, with the increasing angle of attack, the surface flow patterns indicate that this secondary separation becomes distinct by moving inward, away from the leading edge.

The effect of the leading-edge vortices is to induce low pressures in their vicinities on the upper surface of the wing. This is illustrated in Fig. 1b, which presents the chordwise upper-surface pressure distributions obtained at the four spanwise locations indicated in Fig. 1a for an angle of attack of 4°. The abscissa has been normalized by the local chord. The lower pressures over the forward part of the wing generally between the attachment line and the leading edge are clearly evident. The low-pressure region due to the leading-edge vortices occurs over proportionately larger portions of the outboard stations with the tip station being completely dominated by it. The variation of the influence of the leading-edge vortices with angle of attack illustrated in Fig. 1c, which presents the chordwise upper-surface pressure distributions at one spanwise station obtained for angles of attack, is increased from zero, the extent of the low-pressure influence of the leading-edge vortices increases and it moves back over this local chord corresponding to the inward movement of the vortices mentioned previously. This is further illustrated in Fig. 1d, which presents, for angle of attack from 0° to 15°, the spanwise upper-surface pressure distributions at the chordwise position indicated in Fig. 1a. The spanwise location of the primary vortex is evidenced by the peak load pressures it induces on the wing, and the influence of vortices is to maintain a low pressure between the leading edge and the primary vortex instead of allowing it to increase with distance from the primary vortex as would be the case if it were the only one present. The influence of these leading-edge vortices on the loading distribution at a 4° angle of attack is indicated in Fig. 1e, which presents a pictorial isometric plot of the theoretical (linear) and experimental chordwise loading distributions at the four spanwise positions indicated in Fig. 1a. There are two features of interest here in addition to the expected failure to experience experimentally the singular leading-edge loading of linear theory.

First, there is the noticeable increase of the loading forward of the reattachment line and behind the leading edge which corresponds to the direct low-pressure influence of the leading-edge vortices. Second, this region of increased loading is followed by a region, in the vicinity of the reattachment line, where experimentally the loading is slightly lower than the theoretical loading. This reduction in loading is apparently due to the higher upper-surface pressures arising in the vicinity of the reattachment line along which the flow has been somewhat decelerated and compressed.

Because of the extreme leading-edge sweep and relatively high angle of attack of forward delta of the double delta wing, there is little doubt that the separation from its leading edge generates a relatively strong vortex which is swept back over the inboard region of the wing (i.e., at a Mach number of 2.68, the leading edge of the forward delta is subsonic by a considerable margin whereas the leading edge of the main delta is supersonic by a small margin) upper surface. The presence of this vortex is clearly evident in the contour plot of the wing upper-surface pressure coefficients measured at $M = 2.68$ and presented in Fig. 2a. Because of the detachment of the main delta leading-edge shock wave, it is certain that the flow reattaches to the upper surface. In the upper-surface pressure distribution presented in Fig. 2a, there is evidence of a region of relative low pressure very close to the leading edge. For this Mach number, it is clear that the separated flow from the main delta leading edge reattaches very close to the leading edge and thus confines the vortex and its low-pressure influence to the very narrow region between the reattachment line and the leading edge.

It is also observed that there is a low-pressure ridge running aft from the leading-edge crank. The linear theory predicts a singularity, at $y/(b/2) = 0.825$ in Fig. 2b, which is propagated aft along the outward running Mach line from the leading-edge crank, and it can be argued that this singularity is the theoretical representation of the observed low-pressure ridge. In Fig. 2b, the loading is compared with the corresponding loading obtained from the computed results. The agreement is observed to be very good except in the vicinity of the Mach line from the leading-edge crank where the closed form solution predicts a singularity which is smoothed over by the numerical computer solution. This smoothing of the loading peaks by the numerical solution is similar to that reported by Middleton and Carlson.[13] However, it is observed in Fig. 2c that the measured low-pressure ridge is inboard of that predicted by linear theory. This discrepancy can be rationalized. In reality, disturbances are propagated along the local Mach lines whereas in the theory they are propagated along the freestream Mach lines. In the region in front of the low-pressure ridge, the flow is expanding and thus it is expected that the flow and the local Mach lines would be turned inward. The observed disturbances would, therefore, be aft and inboard of predicted positions.

The strength of the vortex sheet, which arises due to the separation of the lower-surface flow at the leading edge and rolls up to form the leading-edge vortex, is proportional to the velocity at the outer edge of the boundary layer at the separation point. This velocity varies in the same manner as the velocity of the flow around the leading edge and thus decreases with increasing Mach number. The strength of the vortex sheet and resulting leading-edge vortex will, therefore, also decrease with increasing Mach number. As the strength of the leading-edge vortex decreases, the upper-surface reattachment line of the separated flow moves forward until it coalesces with the lower-surface attachment line at the leading edge when the bow wave becomes attached. The direct low-pressure influence of the leading-edge vortex generated by the main delta may be relatively restricted at the cruise Mach number, but this influence for the vortex from the forward delta is evident over a relatively large portion of the wing. Furthermore, the upwash due to this relatively strong inboard vortex will certainly increase the local angle of attack of the main delta, the standoff distance of its leading-edge shock, and the strength of its leading-edge vortex over that which would occur in the absence of the forward delta. It is, therefore, clear that the leading-edge vortex due to the forward delta has a significant influence on the over-all pressure distribution of the wing.

Conclusions

The present investigation has concluded that there are several flow phenomena neglected by the linear theory which are primarily responsible for its inadequacy. Although there seems to be evidence to indicate that these phenomena are present on highly swept wings in a supersonic freestream, there is not sufficient experimental information to define their relative significance, interaction, and variation with the wing operating conditions. There are two flow phenomena of primary interest. They are 1) leading-edge separation with subsequent vortex formation and 2) detachment of the leading-edge shock wave. Thus, wing testing should be done for which the influence of these phenomena is not complicated by other effects.

The relative significance of each of these two flow phenomena will vary with the freestream Mach number and the wing angle of attack. At relatively low supersonic Mach numbers, only the leading-edge vortex should be influencing the flow. For a given angle of attack, as the Mach number is increased, the strength of the leading-edge vortex is expected to decrease, and it must vanish when the Mach number has increased sufficiently to attach the bow shock wave to the wing leading edges. This Mach number dependence of the leading-edge vorticity is related to the proximity of the bow wave to the wing leading edges. At the lower supersonic Mach numbers, the rate of change of the bow-wave angle with freestream Mach number is expected to follow closely that for the Mach angle until the bow wave is near the leading edge. Then, as the bow wave approaches the leading edge, the influence of the leading edge is expected to increase and dominate the rate of approach until the bow wave becomes attached to the leading edge. At zero angle of attack, there should be no leading-edge vortex and thus, as the Mach number is increased, only the influence of the approaching bow wave should be evident. At a given Mach number, increasing the angle of attack is expected to increase the strength of the leading-edge vortex and the distance of the bow wave from the leading edge.

The nature of the detached shock wave and flow near a sonic leading-edge would be expected to have an important effect on the boundary-layer separation and leading-edge vortex formation which influence the drag on the entire wing. In order to evaluate the shortcomings of the linear theory in predicting the phenomena (i.e., the shock detachment), it is suggested that comparative computations for the inviscid flow about a conical wing be undertaken using both a nonlinear and a linear theory.

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Airplane Flying Qualities Specification Revision

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The current military flying qualities specification, MIL-F-8785(ASG), "Flying Qualities of Piloted Airplanes," was adopted in 1954, most recently revised in 1959, and is in need of further revision to enhance its applicability to modern weapon systems. As part of a three-year program to update this specification, an interim revision has been prepared which will be proposed for formal adoption. The changes that have been made are extensive, and include those affecting the organization and framework for stating the requirements as well as changes to the individual requirements for longitudinal short-period characteristics, stick force gradients, Dutch roll, and roll control. New requirements govern the sideslip and roll-sideslip coupling responses to lateral control inputs. This paper discusses the rationale for these revisions and demonstrates how they are supported by experimental data and the characteristics of existing airplanes.

Nomenclature

F_s = elevator stick force
$j\omega$, $\zeta\omega_n$ = imaginary axis and real axis of s plane
K_ϕ = gain constant in roll-aileron transfer function
n_z = normal acceleration at center of gravity
p, q, r = roll, pitch, and yaw rates
p_n = amplitude of roll-rate response at Dutch roll peaks for step aileron input
$\sphericalangle p/\beta$ = phase angle by which Dutch roll oscillation in sideslip leads Dutch roll oscillation in roll rate
s = Laplace operator
T_d = period of damped Dutch roll oscillation
$1/T_{h_1}$ = numerator factor of altitude-elevator transfer function
$1/T_{\theta_2}$ = numerator factor of attitude-elevator transfer function
V = velocity
α = angle of attack
β = angle of sideslip
γ = flight-path angle
δ_{AS}, δ_e = aileron stick deflection and elevator deflection
ζ_d = Dutch roll damping ratio
ζ_ϕ = numerator damping ratio in roll-aileron transfer function
θ, ϕ = pitch attitude and roll attitude
λ_R, λ_S = roll-mode root and spiral-mode root
τ_R = roll-mode time constant
$|\phi/\beta|_d$ = ratio of bank angle to sideslip in Dutch roll oscillation
ϕ/v_e = ratio of bank angle to equivalent side velocity in Dutch roll mode
ψ_n = angular coordinate of vector in s plane
ψ_β = phase angle of Dutch roll oscillation in sideslip response to step aileron input
ω_d = Dutch roll undamped natural frequency
ω_{SP} = undamped natural frequency of short-period mode
ω_ϕ = numerator frequency in roll-aileron transfer function

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Introduction

THE current military flying qualities specification is MIL-F-8785(ASG), "Flying Qualities of Piloted Airplanes."[1] It was adopted in 1954 and revised in 1959, and is now in need of further revision to enhance its applicability to modern weapon systems. The Flight Dynamics Laboratory contracted with Cornell Aeronautical Laboratory (CAL) in 1966 for a three-year program to help prepare an interim revision of this specification. The interim revision[2] that resulted from this effort will be proposed for formal adoption in the fall of 1968.