CHALLENGES IN HIGH-ALPHA VEHICLE DYNAMICS

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Abstract—The evolution of aerospace vehicles towards ever-increasing maneuverability, including flight at high angles of attack and vehicle motions of large amplitudes and high angular rates, has led to the need for prediction of vehicle aerodynamics that are dominated by unsteady separated flow effects. The existing data base is reviewed to determine to what degree the following critical issues are understood: 1. Cause and effect of asymmetric forebody flow separation with associated vortices. 2. Cause of slender wing rock. 3. Effect of vehicle motion on dynamic airfoil stall. 4. Extrapolation from subscale tests to full-scale free flight. To extend the present knowledge to include the coupling existing between novel aerodynamic controls and the vehicle dynamics is the challenge facing designers of future agile aircraft operating at high angles of attack.

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NOTATION

b—wing span
c—reference length, wing chord or diameter (d) for circular cylinder and body alone
D—cylinder diameter
D'—sectional cylinder drag, coefficient $c_D = D'/q_o c$
$f$—frequency
$K_1$—constant in Eq. (3)
$K_2$—constant in Eq. (4)
l—sectional lift, coefficient $c_l = l/q_o c$
L—wing lift, coefficient $C_l = L/q_o S$
$t$—rolling moment, coefficient $C_t = t/q_o S b$
$m$—mass flow injected
$m_0$—calculated mass flow in boundary layer just upstream of injection
$m'$—sectional pitching moment, coefficient $c_m = m'/q_o c^2$
1. INTRODUCTION

High performance aerospace vehicles are subject to unsteady flow fields which generate highly nonlinear aerodynamics with strong coupling between longitudinal and lateral degrees of freedom. The complex vehicle dynamics are caused by separated-flow effects of various types and, therefore, usually cannot be predicted by theoretical means. As a consequence, heavy reliance has to be placed upon phenomenological analyses, based on classic fluid mechanics and guided by flight data and dynamic subscale tests, where dynamic support interference, coupled with wall interference, and dynamic simulation problems add to the difficulty of determining the full-scale separated flow characteristics. Experimental results for high performance aerospace vehicles are analyzed to assess our present level of understanding of the four critical issues outlined in the abstract.

2. ASYMMETRIC FOREBODY FLOW SEPARATION

2.1. INTRODUCTION

The results in Fig. 1 illustrate some aspects of the problem facing the designer of an advanced aircraft, i.e. the yawing moment at $\alpha > 40^\circ$ greatly exceeds the available control capability; and wind tunnel test results can be highly confusing. It is somewhat of a relief to find that the huge yawing moments in Fig. 1 are almost entirely caused by the loads generated on the slender forebody (at zero sideslip) by asymmetric flow separation, as has been demonstrated for a generic aircraft model in static tests and for an advanced aircraft model in dynamic tests (Fig. 2). Thus, body-alone aerodynamic results will represent the high-alpha yaw characteristics of the complete aircraft. It will be shown that
Fig. 1 Typical high-alpha wind tunnel test results for a slender-nosed aircraft.\textsuperscript{(12)}

Fig. 2. Aircraft yaw damping.\textsuperscript{(14)}
the high-alpha roll and spin characteristics of an advanced aircraft are also to a large extent
determined by the aerodynamic characteristics of a slender aircraft forebody.

2.2. YAW CHARACTERISTICS

One cause of the roll sensitivity of asymmetric forebody flow separation at zero sideslip,
shown in Fig. 3 for laminar flow conditions,\(^{15}\) is the so-called nose microasymmetry.\(^{16}\) In
turbulent flow the side-force variation is more irregular, with the sign changing more than
four (4) times per cycle.\(^{16}\) Using the equivalence between sideslip and roll for the high-
alpha characteristics of an axisymmetric forebody, the experimental results in Fig. 3 were
used in Ref. 15 to show that the high-alpha lateral characteristic of an aircraft could take the
different forms shown in Fig. 4. These unpredictable, very undesirable aerodynamic charac-
teristics would be realized if nose microasymmetry were the dominant cause of asymmetric
forebody flow separation. Fortunately, it is not, because only when it is performing
a maneuver will the aircraft enter the high-alpha region where asymmetric forebody flow
separation occurs. Consequently, moving wall effects will determine the separation asym-
metry.\(^{17}\) The moving wall effect discussed here refers to the unsteady boundary conditions
at the body surface between stagnation and separation points. The significant moving wall
effects are concentrated in the boundary layer build-up region, decreasing rapidly with
increasing boundary layer thickness away from the flow stagnation point. For this reason,
the moving wall effects are similar for rotating and translating motions as long as the
\(U_w/U_\infty\) ratio at the stagnation point is the same.

![Fig. 3. Side force characteristics of a 3.5 caliber long pointed ogive\(^{15}\)](image)

![Fig. 4. Possible variations of \(C_Y(\beta)\) and \(C_n(\beta)\) characteristics of a slender-nosed aircraft\(^{15}\)](image)
The moving wall effects are very dependent upon the crossflow Reynolds number in regard to the nature of their induced aerodynamic effects, as illustrated by Magnus lift measurements on a rotating circular cylinder\(^{(17,18)}\) (Fig. 5). For laminar flow conditions positive Magnus lift is generated, mainly as a result of the separation delay on the top side through the downstream moving wall effects. In contrast, at critical flow conditions negative Magnus lift is generated by the promotion of transition on the bottom side through upstream moving wall effects, causing a change from the subcritical to the supercritical type of flow separation, e.g. at \(Re = 0.128 \times 10^6\) and \(\frac{U_w}{U_\infty} \gg 0.3\) (curve f in Fig. 5). The negative Magnus lift reaches its maximum magnitude when the initial, static crossflow conditions are of the critical type (curves j, k, and l in Fig. 5), in which case the moving wall effect always generates negative Magnus lift.

The full-scale vehicle in free flight will only enter the high-alpha flight regime when performing maneuvers, where the associated moving wall effects\(^{(17)}\) will overpower the (static) effects of nose microasymmetry. Even a very modest roll rate has been shown to have a large effect on the developed side force. The laminar moving wall effect generated by a coning motion has been shown to be very powerful (Fig. 6). In the experiment\(^{(19)}\) only a slight push was needed to establish the coning motion in one direction or the other, in spite of the fact that the static yawing moment at \(\beta = 0\) was biased in one direction due to nose microasymmetry. The cone-cylinder body reached nearly equal steady-state coning

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**Fig. 5.** Magnus lift characteristics for initially subcritical (laminar) flow conditions\(^{(18)}\)
rates in both directions. That is, the motion-induced, wall-jet-like moving wall effect, illustrated in the inset sketch in Fig. 6, dominated over the static asymmetry, locking-in the separation asymmetry in the direction of the motion, driving it until the drag-generated yawing moment balances the moment generated by the separation asymmetry. The prediction in Fig. 6 neglects the bearing friction present in the test.

The experimental results in Fig. 6 demonstrate that, during the initial ‘spin-up’ of the coning motion, the effect of roll-angle, which is large in static tests at similar, laminar test conditions (Fig. 3), is completely overpowered by the moving wall effect. Consequently, on an advanced aircraft, performing a high-alpha maneuver at moderate sideslip angles, it is the moving wall effect, realized in the boundary layer build-up region near the crossflow stagnation point on the nose, that will determine the separation asymmetry. The agreement between prediction and experiment in Fig. 6 is very satisfying considering that the effect of the unknown bearing friction could not be included in the prediction. Thus, one has reason to be optimistic about future development of methods for prediction of the high-alpha characteristics of advanced aircraft in nose-slice or coning motions.

2.2.2. Flat Spin

The results in Fig. 6 are for laminar flow conditions. On a full-scale aircraft such flow conditions will only exist in an insignificant region near the nose tip. The significant regions of crossflow separation occur for critical or supercritical flow conditions. In the former case the separation is controlled by the coupling between crossflow boundary layer transition and vehicle motion, whereas in the latter case the coupling between crossflow separation and vehicle motion is similar to that for the laminar flow case discussed earlier (Fig. 6). Furthermore, the coning will occur in the whole \( \alpha \)-range up to 90°. Consequently, flat-spin results for axisymmetric bodies provide useful information.

In order to illustrate the effect of Reynolds number on the crossflow on an axisymmetric body in flat spin it is instructive to examine the two-dimensional flow over a circular cylinder. Four distinct flow regions occur, each with a different type of flow separation and associated drag characteristics. Achenbach has classified these regions as shown in Fig. 7. At subcritical Reynolds numbers the boundary layer is laminar and separation
occurs near the lateral meridian (at $\phi = 80 - 90$ deg). In the critical Reynolds number region laminar separation starts near the lateral ($\phi = 90$ deg) meridian, followed by transition in the lifted shear layer and turbulent reattachment. The reattached turbulent boundary layer, being more energetic than the laminar boundary layer, does not separate until well past the lateral meridian ($\phi \approx 140$ deg). The resulting reduction in the wake width causes a significant drag reduction. When the Reynolds number is increased into the supercritical range, transition moves forward of the lateral meridian and the laminar separation bubble is wiped out. As a result, the beneficial effects of the laminar separation bubble with following turbulent reattachment is lost, and separation moves forward of $\phi = 140$ deg in response to the increasing thickness of the turbulent boundary layer with increasing Reynolds number, causing increased drag. In the transcritical Reynolds number range the separation has advanced to its final, asymptotic value $\phi \approx 100$ deg, and the drag reaches a plateau.

The static experimental results for a 5.4 deg cone-cylinder (24) (Fig. 8) show that the local crossflow Reynolds number is a good indicator of when asymmetric separation of various types will occur. The figure shows that the first side-force peak, point C, is caused by subcritical/critical flow separation. In the static case the force can be positive or negative. However, in the flat-spin case the force establishes itself in a direction that produces an antispin moment (17) (Fig. 9a). At points D and E, the two-bubble, symmetric, critical/critical separation geometry is established. When perturbed by a translatory motion of the cross-section, this separation type also produces an antispin moment (Fig. 9b). When the Reynolds number has been increased to point F in Fig. 8, the prospin supercritical/critical crossflow separation is finally established (22) (Fig. 9c).

Flat spin is initiated at critical flow conditions by a static moment generated at $\alpha = 90$ deg through the effects of asymmetric body-roughness (22). The large static moment is able to accelerate the flat spin through the spin-resisting subcritical/critical (Fig. 9a) and critical/critical (Fig. 9b) regions of crossflow separation to the prospin supercritical/critical separation (Fig. 9c). The measured yawing moment on an ogive-cylinder driven in flat spin at critical flow conditions (22,35) (Fig. 10) illustrates how for $\Omega < 0.25$ the critical/critical crossflow conditions (Fig. 9b) produce an anti-spin moment, whereas the supercritical/critical crossflow conditions (Fig. 9c) existing at $\Omega > 0.50$ produce a prospin moment, as illustrated by the experimental flat-spin characteristics for a cone-cylinder (26) (Fig. 11).

In the case of a supermaneuver (27,28) the translatory motion of the circular cross-section is driven by the control-induced moment. The demanded rapid roll around the velocity vector could establish the supercritical/critical flow separation (Fig. 9c) directly, and not only at $\alpha = 90$ deg but in a substantial $\alpha$-range below 90 deg. In such a case the dynamic vehicle
response will significantly amplify the control-induced moment. In addition, an increase of
the pitch-up moment will be generated by the change to an asymmetric crossflow separ-
ation geometry. The experimental results for a 5 caliber long tangent-ogive (29,30) (Fig. 12) show how the separation asymmetry in addition to generating a side force $C_Y$ also
produces an increase of the normal force \( C_n \). Experimental results for the effect of a nose strake\(^{31}\) indicate that the normal force can be increased by as much as 33\% through the change from symmetric to asymmetric crossflow separation. This would cause a sudden increase of the pitch-up moment, resulting in an increase of the angle of attack that could cause a slender-nosed aircraft to enter into a flat spin.\(^{32}\)
2.2.3. Reversal of Coning Motion

When the cone-cylinder model in Fig. 6 was turned around 180°, similar coning results were obtained (19) (Fig. 13). However, in this case the coning motion changed direction without any external forcing, ‘spinning up’ to the limiting rates in both directions before slowing down and starting coning in the opposite directions. The fluid dynamics behind this behavior can be understood by studying Magnus lift results for a rotating circular cylinder (18) (Fig. 5). For the Reynolds number \( Re = 0.128 \times 10^6 \) (curve f in Fig. 5), at low rotation rates, \( U_w/U_\infty < 0.30 \), the downstream moving-wall effect delays separation on the top side, generating greater suction. On the opposite side, the upstream moving wall effect promotes separation. This results in a positive lift that is proportional to the wall-to-freestream velocity ratio, \( U_w/U_\infty \). At some critical value of \( U_w/U_\infty \), transition from a laminar to a turbulent boundary layer occurs on the bottom side due to the upstream moving wall effect. As a consequence, the separation type changes to turbulent separation, while laminar separation is maintained on the top side. This reverses the separation asymmetry, generating a negative Magnus lift.

A similar reversal of the separation asymmetry occurs for the coning body (19) (Fig. 13). Initially, flow asymmetry and/or minute surface irregularities set the separation asymmetry. The ensuing coning motion reinforces the effect of the asymmetry, as the laminar separation is delayed on the advancing side and promoted on the retreating side, generating a force that drives the coning (\( \dot{\psi} > 0 \)). However, the moving wall effect eventually causes boundary layer transition on the retreating side, reversing the separation asymmetry (i.e. \( \dot{\psi} > 0 \) but \( \ddot{\psi} < 0 \)). Eventually this results in accelerated coning in the opposite direction (\( \dot{\psi} > 0 \) but \( \ddot{\psi} < 0 \)). The rotation reversal moves transition back into the wake on the new advancing side, and asymmetric laminar separation is reestablished. Eventually, transition occurs on the retreating side, reversing the separation asymmetry and decelerating the coning motion (\( \dot{\psi} < 0 \) but \( \ddot{\psi} > 0 \)). The process repeats itself, resulting in a self-reversing coning motion. It will be seen in what follows that a similar process for the rolling body drives the wing rock of slender-nosed aircraft.

2.3. ROLL CHARACTERISTICS

Smoke flow visualization revealed that the violent wing rock observed in wind tunnel tests of a generic aircraft configuration (23) (Fig. 14) was probably generated by asymmetric
Fig. 13. Reversal of coning motion for a flat-faced circular cylinder.¹⁹

vortex shedding from the slender forebody. As the Reynolds number was in the critical region, \( Re = 0.26 \times 10^6 \), the reversal of the Magnus effect shown in Fig. 5 will occur very close to \( U_w/U_\infty = 0 \). This suggests that the wing rock is generated by a forebody flow phenomenon similar to that causing the reversal of the coning motion discussed earlier (Fig. 13), with the moving wall effects generated by the rolling motion rather than by coning. The following scenario can be visualized¹³⁴ (Fig. 15). At \( t = t_1 \), the adverse upstream moving wall effect on the crossflow over the forebody causes boundary layer transition to occur earlier, switching the separation toward the supercritical type. In the absence of any time lag, the vortex geometry sketched at \( t = t_1 \) would result. Due to convective time-lag effects, similar to those generating slender wing rock,¹³⁵ this vortex geometry is not realized until \( t = t_2 = t_1 + \Delta t \). For simplicity, only the vortex closest to the body is shown at \( t_2 \), as it illustrates the vortex-induced effect. It generates port-side downwash with associated statically stabilizing rolling moment. This produces the positive aerodynamic spring needed for the oscillatory wing rock motion. At \( t = t_3 \), when the roll rate reaches its maximum in the opposite direction, another switch in the forebody separation asymmetry occurs.
Because of the convective time-lag effect, the vortex geometry at the (now horizontal) wing has not changed, but is the same as at \( t = t_2 \), in agreement with flow visualization results.\(^{(33)}\) During the time lag \( \Delta t \) required for the new vortex geometry to reach the wing, the vortex-induced rolling moment generated by the previous vortex geometry drives the rolling motion, generating the observed wing rock.\(^{(36,37)}\)

### 2.3.1. Dynamic Forebody/Leading-Edge Vortex Interaction

When the wing has a highly swept Leading-Edge Extension (LEX), the forebody vortex affects the wing loading through its interaction with the LEX-vortex,\(^{(38,39)}\) as is illustrated in Fig. 16 based upon laminar experimental results for a model of the F/A-18 aircraft.\(^{(40-42)}\) Flow visualization of the wing rock\(^{(43)}\) shows that the interaction between forebody and LEX vortices suddenly undergoes a dramatic change, the lower forebody vortex ‘dipping down’ to interact strongly with the co-rotating LEX vortex. That started the wing-rock, with the forebody-LEX vortex interaction alternating between the two sides. The interaction between the co-rotating forebody and LEX vortices is likely to lift up the LEX vortex and promote breakdown of both vortices. As a result, much of the beneficial effect of the LEX vortex, i.e. the delay of flow separation on the main wing due to the vortex-induced flow field, is lost, causing a loss of wing lift and producing a statically stabilizing rolling moment. Thus, instead of acting through the generated downwash, as illustrated in Fig. 15, the forebody vortex controls the breakdown of the LEX vortex\(^{(38,39)}\) (Fig. 17). On the rolling forebody, the downstream moving wall effect for laminar flow conditions delays flow separation \( (U_w/U_\infty < 0.3 \text{ for } Re \leq 12.8 \times 10^4 \text{ in Fig. 5} ) \). As in the case of the generic aircraft model in Fig. 15, there is a time delay \( \Delta t \) before the vortex responds to the change of rotation of the forebody. This generates the rolling moment that drives the wing-rock
motion, resulting in the observed build-up to limit-cycle amplitudes of significant magnitudes (Fig. 18).

The wing rock amplitude of F/A-18, observed in flight and measured at low Reynolds numbers in wind tunnel tests (Fig. 18), shows the amplitude to increase with angle of attack, from approximately 3 deg at $\alpha = 30^\circ$ to a maximum of slightly more than 20 deg at $\alpha = 45^\circ$. When the angle of attack is increased further, the limit cycle amplitude decreases rapidly. For the slender forebody with a slenderness ratio of approximately 2.5 static asymmetric forebody vortices should start developing when $\alpha > 47.4$ deg. This is in good agreement with the measured yawing moment which started occurring at $\alpha > 50^\circ$ at zero sideslip. As discussed in Ref. 45 in connection with test results for a spinning nose tip, the moving wall effect is largest just before natural separation asymmetry occurs; at $\alpha > 50^\circ$ in Fig. 18. At that point, the symmetric flow separation is easiest to perturb to change the symmetric vortex pattern to an asymmetric one in the direction dictated by the moving wall effect. The rapid decline of the wing rock amplitude at higher angles of attack reflects the fact that the moving wall effect now has to overcome the natural, static separation asymmetry. Also contributing to this data trend is that the static vortex breakdown (for the nonrolling aircraft) occurs closer and closer to the LEX apex, leaving less room for the vortex interaction because of the presence of the cockpit (see Section B-B in Fig. 16).

The agreement between the wind tunnel test and free flight results in Fig. 18 appears at first rather surprising. Although it is well known that static side forces generated by asymmetric forebody flow separation are of similar magnitudes for laminar and turbulent flow conditions, in the dynamic case one must also consider the laminar and turbulent moving wall effects. Figure 5 shows that in the laminar, subcritical case, the Magnus lift is
Fig. 16. Flow geometry for LEX-forebody vortex interaction.\textsuperscript{38}

Fig. 17. Conceptual flow mechanism for dynamic LEX-forebody vortex interaction at laminar crossflow conditions on the F/A-18 aircraft.\textsuperscript{39,39}
generated mainly by the downstream moving wall effect on the top side, which moves the flow separation from the subcritical towards the supercritical position. On the bottom side the flow separation is occurring forward of the lateral meridian already in the static case, and the upstream moving wall effect does not have much leverage for its separation-promoting action. However, in the turbulent, supercritical case, the situation is reversed. The main effect is that of the upstream moving wall effect on the bottom side, which promotes flow separation, moving it from the supercritical towards the subcritical position. The downstream moving wall effect on the top side has very limited possibility to cause further delay of the already supercritical flow separation.

In the present case of interest the laminar flow separation is delayed from the static position $S_{st}$ to the dynamic location $S_{dyn}$ by the moving wall effect (Fig. 19a), whereas the turbulent separation is promoted from the static position $S_{st}$ to the dynamic location $S_{dyn}$ (Fig. 19b). The result is a forebody vortex pair that looks quite similar. This similarity is facilitated greatly by the process illustrated in Fig. 12 by static experimental results for a 5 caliber pointed tangent-ogive.\(^{30}\) When one vortex is lifted up as a result of asymmetric crossflow separation, the other vortex moves inboard under the lifted vortex. This occurs even under steady, symmetric crossflow conditions, when the separation asymmetry is caused by nose microasymmetry,\(^{16}\) generating a side force as well as an increase of the normal force. Thus, regardless of how the asymmetric crossflow separation is established, the resulting asymmetric vortices are similar, as is illustrated in Fig. 19 for the case of the F-18 wing rock.

Based upon earlier discussion of the results in Fig. 5, one expects the effect of forebody rotation on the flow separation and associated vortex geometry to switch if the Reynolds number is increased from the laminar flow region to fall in the critical range. In that case, the moving wall effect does not influence the forebody flow separation directly, as in the case illustrated in Fig. 17, but acts instead only indirectly through its effect on crossflow boundary layer transition, reversing the ultimate effect on crossflow separation from what it is for laminar flow conditions (Fig. 17), as illustrated in Fig. 15. Thus, the forebody-LEX vortex interaction would be statically destabilizing/dynamically stabilizing. That is, no wing rock would occur. This is exactly what was observed\(^{47}\) when the F/A-18 model was tested at roughly the same crossflow Reynolds number for the forebody as in the case of the generic aircraft model in Fig. 15. In full-scale flight this critical flow region is restricted to a minuscule region near the nose tip, and the moving wall effect on the downstream turbulent crossflow separation can generate the observed wing rock\(^{43,44}\) (Fig. 18).
2.4. SUMMARY

In general, the effects of asymmetric forebody flow separation can be summarized as follows:

1. Significant asymmetric loads occur only at subsonic crossflow conditions.\(^{16}\)
2. The asymmetric loads are dependent upon the nose slenderness ratio.\(^{16}\)
3. The moving wall effects provide a flow mechanism that can control the crossflow separation and thereby drive a slender-nosed body in a coning motion at 30 deg < \(\alpha\) < 60 deg and in flat spin in an \(\alpha\)-range around \(\alpha = 90\) deg.
4. The moving wall effects on crossflow separation and associated vortices on slender forebodies generate downwash on the downstream wing, causing the observed wing rock of a generic aircraft model in wind tunnel tests at critical forebody crossflow conditions. They can also, through forebody–LEX vortex interaction, cause wing rock of the F-18 HARV aircraft at the turbulent flow conditions of full scale flight as well as in subscale wind tunnel tests at laminar crossflow conditions.
5. The experimental results provide examples of the fact that dynamic simulation of separation-induced, self-excited oscillations in subscale tests is not possible unless the full-scale Reynolds number is simulated. One encouraging result of the present analysis is that it indicates that even in the presence of moving wall effects analytic extrapolation\(^{48,49}\) to full-scale vehicle dynamics from subscale tests may be possible, provided that the fluid mechanics involved are fully understood.

3. SLENDER WING ROCK

3.1. INTRODUCTION

Recent test results\(^{50}\) have shown that asymmetric leading-edge vortices are not generated at symmetric flow conditions (\(\phi = \beta = 0\)) on a sharp-edged delta wing. However, the vortex breakdown occurring at very high angles of attack can be asymmetric for very slender delta wings. Thick delta wings with rounded leading edges have an apex geometry similar to that of a slender forebody and could, therefore, be subject to vortex asymmetry at \(\phi = \beta = 0\). The wing rock observed in experiments with sharp-edged delta wings\(^{51,52}\) is not initiated by vortex asymmetry.\(^{53}\) However, vortex asymmetry and vortex breakdown are the flow phenomena determining the maximum limit cycle amplitude of slender wing rock.\(^{35,53,54}\)
3.2. RECENT EXPERIMENTS AND DISCUSSION

In a recent experimental investigation\(^{55}\) the effect of leading-edge sweep on wing rock was found to be of a rather complicated nature. The results for an 80° delta wing were in general agreement with earlier experiments\(^{51}\) (Fig. 20), and showed a downstream progression of vortex lift-off from apex, as described in Ref. 35. However, it was found that wing rock occurred even in the absence of asymmetric vortex lift-off, e.g. at \(\alpha = 31°\) on a 75° delta wing. A developed analytic method\(^{56}\) predicts that the roll damping of an 80° delta wing is lost, even without vortex lift-off, when the angle of attack exceeds \(\alpha = 20°\). This is in good agreement with experimental forced-oscillation results\(^{51}\) (Fig. 21). The fact that the free-oscillation results\(^{51}\) showed no wing rock to occur on the 80° delta wing until vortex lift-off took place, at a higher angle of attack (\(\alpha = 27°\)), is the likely result of the presence of bearing friction in the test. The nonlinear, more or less discontinuous increase of the undamping due to vortex lift-off\(^{53,54}\) had no difficulty in overpowering the mechanical (frictional) damping, and wing rock resulted (Fig. 20). What happens in free flight at \(20° < \alpha < 27°\), in the absence of mechanical damping?

In free flight at \(\alpha > 20°\), an 80° delta wing is undamped in roll\(^{51,56}\) (Fig. 21). If the angle of attack is large enough the wing will oscillate in roll until a critical roll angle is exceeded, at which point the asymmetric flow condition causes the lee-side wing to experience vortex lift-off. The resulting large undamping will cause a rapid build-up of the wing-rock amplitude until the damping action of the vortex breakdown on the windward wing brings

![Fig. 20. Wing rock amplitude ratio for an 80 deg delta wing.\(^{55}\)](image)

![Fig. 21. Roll damping of 80 deg delta wing.](image)
the net damping to zero, resulting in the large limit cycle oscillation amplitude observed\(^{[53,54]}\) and predicted\(^{[53,54]}\). Although in the free-to-roll tests bearing friction prevented wing rock from developing until vortex lift-off occurred in the test\(^{[53,54]}\), it did not have any significant influence on the magnitude of the final limit-cycle amplitude, which is largely determined by the magnitude of the large change of the rolling moment generated by the vortex lift-off\(^{[53,54]}\). The main effect of the bearing friction was to move the wing-rock boundary for \(\phi = \beta = 0\) to a higher angle of attack (Fig. 22). The predicted free-flight boundary\(^{[56]}\) is shown by the solid line, and the boundary determined in water tunnel tests\(^{[55]}\) is by the dashed line. The bearing friction had less of a chance to affect the results in the denser fluid (water) than in the wind tunnel test of the 80° (\(\theta_A = 10°\)) delta wing\(^{[51]}\) (the data point represented by + at \(\alpha = 27°\) in Fig. 22). Even lower friction was apparently realized in a wind tunnel test using an air bearing\(^{[57]}\), producing wing rock closer to the free-flight boundary\(^{[53]}\) (the data point indicated by x at \(\theta_A = 10°\) and \(\alpha = 22°\) in Fig. 22). The main effect of friction is to delay the occurrence of wing rock. It has little effect on the maximum amplitude, as is demonstrated by Fig. 20.

Based upon the agreement between prediction\(^{[56]}\) and experiment\(^{[51]}\) for the 80° delta wing shown in Fig. 21, the predicted large effect of leading edge sweep shown in Fig. 23 is credible. Thus, the 85° delta wing should lose its roll damping when \(\alpha > 11°\) is exceeded. In the test\(^{[55]}\), wing rock was observed at \(\alpha > 17°\), and was associated with asymmetric vortex lift-off. This supplied the extra undamping needed to overcome the bearing friction, just as in the case of the 80° delta wing. At \(\alpha = 25°\), one vortex stayed attached while the other vortex went through the vortex-lift-off/vortex-breakdown process, apparently single-handedly supplying the wing rock mechanism described in Refs 35 and 54 for the case that both vortices are working together. This indicates that slender wing rock will occur more frequently than what has been assumed\(^{[58]}\), based upon the concept of hydrodynamic instability, the 'crowding' theory\(^{[35,59]}\). When the angle of attack was increased to \(\alpha = 35°\), the rolling motion became erratic, nonoscillatory in character\(^{[55]}\).

Figure 23 shows that the roll damping is lost for a 75° delta wing when the angle of attack exceeds \(\alpha = 29°\). In the test\(^{[55]}\), wing rock was observed at \(\alpha = 31°\), without the occurrence of vortex lift-off (or vortex breakdown). The fact that the bearing friction could be overcome without resorting to the undamping contribution from vortex lift-off is the likely result of the 50 percent increase of the wing area and the associated 125 percent increase of the vortex-induced rolling moment compared to the 80° delta wing. At \(\alpha = 35°\), the boundary for vortex breakdown will be penetrated\(^{[60]}\) during the roll oscillations of a 75 deg delta wing\(^{[53]}\). The test\(^{[55]}\) showed that vortex breakdown occurred on one side of the wing, and

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Fig. 22. Boundaries for the occurrence of slender wing rock and vortex breakdown on delta wings\(^{[53]}\).
stayed there during the wing-rock motion. It could occur on either wing half, but would not switch sides during the rocking motion. Thus, the amplitude-limiting mechanism was vortex breakdown, just as in the case of the 80° delta wing. However, in this case only one side of the delta wing was generating the wing rock, with the roll oscillation taking place around a nonzero roll angle, greatly increasing the probability for slender wing rock to occur also in the presence of a fuselage.

Figure 24 shows the magnitude of the wing-rock limit-cycle amplitude measured for the three delta wings tested. It can be seen that the single-sided wing-rock mechanism can generate significant roll amplitudes, $\Delta \phi \leq 30^\circ$ for $\Lambda = 85^\circ$ and $\Delta \phi \leq 15^\circ$ for $\Lambda = 75^\circ$, although of less magnitude than for the double-sided wing rock of the 80 deg delta wing.

3.3. SUMMARY

The results of an analysis of slender wing rock in view of existing experimental results can be summarized as follows:

1. Wing rock of slender delta wings, $\Lambda \leq 80^\circ$, is initiated by the large undamping generated by attached leading-edge vortices. Thus, asymmetric lift-off of the leading edge vortices on slender delta wings does not start the wing rock observed in experiments. For very
slender delta wings, $\alpha = 80$ and $85$ deg, it is, however, a flow mechanism contributing substantially to the observed very large limit-cycle amplitudes. The magnitude of the wing rock amplitude is limited by breakdown of the leading-edge vortex on the 'dipping', windward side of the delta wing.

(2) Slender wing rock is not always dependent upon the interaction between the two leading edge vortices on a delta wing but can be generated by the vortex-induced rolling moment on one wing half.

(3) As a consequence of the characteristics outlined above, slender wing rock is a phenomenon of concern to high agility aircraft and to aerospace vehicles in general that are operating at high angles of attack.

4. DYNAMIC AIRFOIL STALL

4.1. INTRODUCTION

It was long assumed that the character of the airfoil motion did not have a significant effect on the dynamic stall process as long as the effective angular amplitude was the same. That is, the effective angle of attack for pitching and plunging motions could be expressed as follows:

\[ \alpha = \alpha_0 + \theta^* \]
\[ \theta^* = \bar{\alpha} \sin \omega t \]
\[ \bar{\alpha} = \bar{\theta} = \bar{z}/U_\infty \]

where $\bar{z}/U_\infty$ is the 'equivalent pitch'.

In spite of the fact that tests showed this assumption to be invalid, except for the case that the pitch axis is located close to the leading edge, attempts are still being made to justify the use of Eq. (1), which originally was necessitated by the lack of experimental results for plunging airfoils. The results for the NACA-0012 airfoil shown in Fig. 25 were recently used to illustrate that 'the normal force for equivalent pitch (due to plunging) has higher slopes than that for true pitch'. The semblance of similarity between oscillations in plunge and pitch, shown in Fig. 25, was obtained using the ordinate stretching shown in Fig. 26. This distortion of the measurements was assumed to account for the

![Fig. 25. Comparison of lift characteristics for pitching and plunging oscillations of a NACA-0012 airfoil.](image)
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Fig. 26. Lift loops for pitching and plunging oscillations of a NACA-0012 airfoil.\textsuperscript{62}

Fig. 27. 'Leading-edge-jet' effects on pitching and plunging airfoils.\textsuperscript{65}

difference in test Reynolds number, $Re \approx 0.25 \times 10^6$ for the plunging oscillations compared to $Re \approx 2.5 \times 10^6$ for the pitching oscillations. As discussed in Ref. 65, a more reasonable assumption is that the difference in Reynolds number can be accounted for by zero-shifting the static data until they essentially agree. Zero-shifting the loops to accomplish this gives the results shown in Fig. 27. The remaining difference in the attached-flow branches during the upstroke is caused by the pitch-rate-induced camber effect.\textsuperscript{66} However, the main difference between pitching and plunging loops in Fig. 27 is caused by the difference in the moving wall effects,\textsuperscript{17} the so-called leading-edge-jet effect\textsuperscript{67} illustrated in Fig. 28.

4.2. MOVING WALL EFFECT ON FLOW SEPARATION AND REATTACHMENT

It has been shown that a rotating leading edge can increase the maximum lift substantially.\textsuperscript{68} A similar 'leading-edge-jet' effect can delay stall on an airfoil in pitch-up...
Fig. 28. Leading-edge jet effect.\cite{67}

Fig. 29. Measured moment loops for pitching ($C_D = 0.25$) and for plunging oscillations of a NACA-0012 airfoil, (a) plunging\cite{62} (b) pitching.\cite{63}
motion, as is illustrated by the 'rolling leading edge' analogy in Fig. 28, adding to the separation delay due to accelerated flow effect. However, for plunging airfoil oscillations, where the effective angle of attack is increasing on the downstroke, the moving wall effect at the leading edge will promote flow separation. This is demonstrated by the different moment characteristics for the plunging and pitching oscillations (Fig. 29). In this case, no coordinate stretching will be able to make the $C_m$-loops for pitching and plunging oscillations look similar. The adverse 'leading-edge-jet' effect for the plunging oscillation causes dynamic (moment) stall to occur already at $\alpha \approx 16^\circ$, whereas the beneficial moving wall effect for the pitching airfoil delays dynamic stall to $\alpha \approx 24^\circ$. It is shown in Ref. 69 that the difference would have been even larger if the tests had been performed at the same Reynolds number. The reason for this is that the adverse moving wall effects on the plunging downstroke caused boundary layer transition to occur upstream of separation, changing the stall from the laminar to the turbulent leading-edge stall type. As a consequence, dynamic stall occurred at $\alpha \approx 16^\circ$ (Fig. 29a) rather than $\alpha < \alpha_s$, where $\alpha_s \approx 12^\circ$ (Fig. 25).

The experimental results obtained by Niven and Galbraith for down-ramp motion of a NACA-0015 airfoil from fully stalled conditions (Fig. 30) demonstrate the power of the moving wall effect, as will be discussed in detail later. A satisfactory understanding of these dynamic characteristics is critical for prediction of the wing rock of aircraft with wings of moderate sweep, such as the Gnat aircraft or the X-29A (Fig. 31). The wing rock is caused by the negative damping-in-plunge generated by the airfoil section on the rocking
wing at penetration of the static stall region,\(^{(58)}\) as is demonstrated by the experimental results presented by Liiva et al.\(^{(73)}\) (Fig. 32). During the downstroke the upstream moving-wall effect promotes flow separation. This effect is larger than the separation-delaying accelerated-flow effect. From Ref. 66 one obtains the following definition of the dynamic stall overshoot \(\Delta \alpha_s\) above the static stall angle \(\alpha_s\) during the upstroke.

\[
\Delta \alpha_s = \Delta \alpha_{s1} + \Delta \alpha_{s2}. \quad (2)
\]

The component \(\Delta \alpha_{s1}\) is generated by the accelerated-flow effect on the ambient pressure-gradient time history. It is only determined by the rate of change of the effective angle of attack and can be written as follows for pitching and plunging oscillations, respectively

\[
\Delta \alpha_{s1} = K_1 \dot{\alpha}/U_w. \quad (3a)
\]
\[
\Delta \alpha_{s1} = K_1 \dot{\alpha}_{LE}/U_w. \quad (3b)
\]
The other component, $\Delta \alpha_{s2}$, is generated by the LE-jet effect (Fig. 28). In a first approximation $\Delta \alpha_{s2}$ is proportional to the LE-plunging velocity $\dot{z}_{LE}$. Thus, for pitching and plunging oscillations $\Delta \alpha_{s2}$ becomes

$$\Delta \alpha_{s2} = K_2 \xi_{CG} \dot{\alpha}/U_\infty$$

(4a)

$$\Delta \alpha_{s2} = - K_2 \dot{z}_{LE}/U_\infty.$$  

(4b)

Combining Eqs (3) and (4) defines $\Delta \alpha_s$ as follows for pitching and plunging airfoils, respectively

$$\Delta \alpha_s = \Delta \alpha_{s1} + \Delta \alpha_{s2} = (K_1 + K_2 \xi_{CG}) \dot{\alpha}/U_\infty$$

(5a)

$$\Delta \alpha_s = \Delta \alpha_{s1} + \Delta \alpha_{s2} = (K_1 - K_2) \dot{z}/U_\infty.$$  

(5b)

Equation (5a) shows that for a pitching airfoil the dynamic stall overshoot of the static maximum lift or normal force should increase with increasing $\xi_{CG}$, in agreement with experimental results$^{(74)}$ (Fig. 33). Figure 34 shows that when plotted as a function of the
reduced frequency $\omega$, these experimental results for 6 deg amplitude oscillations of the NACA-0012 airfoil exhibit the same highly nonlinear behavior as those of Carr et al. for 10 deg amplitude oscillations.\(^{(63)}\) It is shown in Ref. 75 that the nonlinearity is in large part caused by the effect of the spilled leading-edge vortex. In earlier analysis of dynamic test results for a pitching NACA-0012 airfoil\(^{(76)}\) it was found that $K_2 \approx 2K_1$ in Eq. (5a). That is,

$$\Delta \alpha_s \approx K_1(1 + 2\xi_{CG})\dot{\alpha}c/U_\infty. \quad (6)$$

Figure 35 shows that when plotted in the normalized form $\Delta c_{n_{\text{max}}}/(1 + \xi_{CG})$, the experimental data from Fig. 34 tend to collapse. For $\xi_{CG} = 0.37$ and 0.50 the collapse is complete, including the envelopes of the data spread. Although the normalized experimental results for $\xi_{CG} = 0.25$ fall somewhat below those for $\xi_{CG} = 0.37$ and 0.50, when considering the data scatter for each $\xi_{CG}$ the collapse of the experimental results is satisfactory. Thus, one can represent the dynamic stall overshoot $\Delta \alpha_s$ during the upstroke by Eq. (6), where $K_1$ is the accelerated flow effect $K_a$. Equation (6) can also represent the dynamic undershoot $\Delta \alpha_r$ of static flow reattachment during the downstroke, in which case $K_1 = K_d$, where $K_d$ is the decelerated flow effect.

$$\Delta \alpha_r \approx -K_d(1 + 2\xi_{CG})\dot{\alpha}c/U_\infty. \quad (7)$$

Figure 36 shows that the angle of attack for flow reattachment, as determined from the experimental results in Fig. 30, is linearly dependent upon $\dot{\alpha}c/U_\infty$ in agreement with Eq. (7). The initial drop of $\alpha_r$ at $\dot{\alpha}c/U_\infty = 0$ is caused by static hysteresis. It is shown in Ref. 77 that based upon the Magnus lift results\(^{(189)}\) in Fig. 5 one obtains the ratio $K_d/K_a = 8/3$, which gives the prediction $\Delta \alpha_r \approx -15\dot{\alpha}c/U_\infty$ compared to the experimental value $\Delta \alpha_r \approx -13\dot{\alpha}c/U_\infty$ in Fig. 36.

Dynamic test results for the NACA-0012 airfoil\(^{(73)}\) gave a value $K = K_1 + K_2\xi_{CG} = 3.0$ for $\xi_{CG} = 0.25$. That is, $K_1 \approx 2$ and $K_2 \approx 4$, giving $\Delta \alpha_s = -2\dot{\alpha}c/U_\infty$ in Eq. (5b). Thus, the moving wall effect generates undamping and is the flow mechanism responsible for the measured undamping\(^{(73)}\) in Fig. 32. As a consequence, plunging oscillations become undamped when stall occurs, leading to a loss of roll damping and the initiation of wing rock as experienced by the Gnat and X-29A aircraft\(^{(71,72)}\) (Fig. 31). For the rolling wing, the effective angle of attack varies as follows with the roll angle $\phi$, for the roll-axis inclination, $\alpha_r = \alpha(0)$

$$\alpha(\phi) = \tan^{-1}(\tan \sigma \cos \phi). \quad (8)$$

![Fig. 34. Variation of the overshoot $\Delta c_{n_{\text{max}}}$ with $\dot{\omega}$ for the NACA-0012 airfoil.\(^{(74)}\)](image-url)
When $\alpha(\phi) < (\alpha_\text{dyn})$ during the roll oscillation, the plunging wing section generates positive damping. At the limit-cycle amplitude, the net damping is zero. Equation (8) shows how $\alpha(\phi)$ increases with increasing inclination $\sigma$ of the roll axis, requiring higher and higher roll angle $\phi$ to reach the condition $\alpha(\phi) < (\alpha_\text{dyn})$. The fast increase of the wing-rock amplitude with increasing angle of attack for X-29A (Fig. 31b) reflects this effect of increasing inclination of the roll axis.

It is obvious that one must be able to predict $(\alpha_\text{dy})$ in order to predict the high-alpha vehicle dynamics of agile aircraft.\(^{(27,28)}\) This requires a thorough understanding of the pitch-down characteristics in Fig. 30. Based upon the satisfactory prediction of the experimental results\(^{(70)}\) in Fig. 36, the analysis in Ref. 77 indicates that reliable means for the prediction of these pitch-down characteristics can be developed.
The conclusions of a review of dynamic stall characteristics can be summarized as follows:

1. The dynamic delay of flow separation is caused by time lag and boundary layer improvement effects; the former only causes the static loads to lag the instantaneous flow environment, e.g. the angle of attack, whereas the latter produce an overshoot of the static stall characteristics, e.g. the effective angle of attack and associated lift maximum. Likewise, the dynamic flow reattachment is delayed by time lag and boundary layer impairment effects; the former only causing the static loads to lag the instantaneous angle of attack, whereas the latter produce an undershoot of the static stall characteristics, in some cases delaying flow reattachment to negative angles of attack.

2. The boundary layer improvement/impairment is caused by accelerated/decelerated flow and moving wall effects.

3. The moving wall effects usually dominate over the accelerated/decelerated flow effects, except when $\zeta_{CG}$ is small, causing the experimentally observed negative damping-in-plunge and being to a large extent responsible for the extreme delay of flow reattachment observed in pitch-down tests.

5. EXTRAPOLATION FROM SUBSCALE TESTS

To correct for the effects of compressibility, subscale Reynolds number, and the always present ground facility interference are major problems one faces when trying to use subscale ground test results to predict full-scale flight characteristics.

5.1. COMPRESSIBILITY EFFECTS

When trying to apply results such as those in Figs 30 and 32 to full-scale flight, one has to consider the large effect of compressibility on stalling airfoils. The wing rock will start at lower and lower angle of attack for increasing subsonic Mach number, in agreement with the decreasing stall angle with increasing Mach number illustrated by the decreasing maximum lift ($\lambda_{Ta}$) (Fig. 37). It is shown in Ref. 78 that the data trend in Fig. 37 is the result of the decrease of the effective nose radius $\rho_N = r_N/c$ with increasing Mach number through the effect of compressibility. This results in the predicted data trend shown by the solid line in Fig. 37.

5.2. SCALING PROBLEMS IN 2-D FLOW

The moving wall effect on boundary layer transition presents a severe problem in subscale dynamic tests, as is illustrated by the measured damping in plunge shown in Fig. 38. The tests of the NACA-0012 airfoil performed at low Reynolds number, by Carta and Rainey, experienced transition-induced stall delay on the plunging downstroke. This caused an increase of dynamic lift, similar to moving from a low to a higher Reynolds number, from $Re = 0.33 \times 10^6$ to $1.34 \times 10^6$ in Fig. 39 for example. As a result, the measured damping in plunge was larger in the stall region than for attached flow (compare $\alpha > 10^\circ$ with $\alpha < 10^\circ$ in Fig. 38a). In contrast, the tests performed by Liiva (Fig. 38b), at an order of magnitude higher Reynolds number, showed the plunging oscillations to become undamped in the stall region, $\alpha > 12^\circ$. In this case the adverse moving wall effect on the downstroke promotes flow separation, not transition, as was the case in the low Reynolds number tests. The resulting lift loss drives the downstroke motion, causing the measured undamping.

The example of the scaling problems illustrated in Fig. 38, showing how the moving wall effect on boundary layer transition can reverse the dynamic stall effect expected at the
higher Reynolds number of full-scale flight, is similar to the reversal observed in the case of the Magnus lift of a rotating circular cylinder\textsuperscript{(18)} (Fig. 5), discussed earlier. For the modest magnitude of $U_w$ of interest in the case of dynamic airfoil stall, $U_w/U_\infty \ll 0.1$ in Fig. 5, the moving wall effect is of significant magnitude only in the region near the flow stagnation point, where the boundary layer is thin and, therefore, very sensitive to this wall jet-like action.

As illustrated in Fig. 28, plunging and pitching airfoils have opposite moving wall effects for increasing effective angle of attack, during the plunging downstroke $\dot{z}/U_\infty$ and the pitching upstroke $\theta$, respectively. Carta's hot film data\textsuperscript{(61)} (Fig. 40) show how the adverse (upstream) moving wall effect $\dot{z}(t)$ will promote transition and cause the plunging airfoil to
have a longer run of attached, turbulent flow prior to stall. As a result, the flow stays attached past 7.5% chord, whereas flow separation occurs forward of 5% chord on the pitching airfoil, which has a shorter turbulent run before stall due to the opposite, transition-delaying moving wall effect. In addition to showing the opposite moving wall effects for pitching and plunging oscillations, a fact illustrated by Eq. (4), Fig. 40 also demonstrates that the moving wall effect completely dominates over the accelerated flow effect, which effect is the same for pitching and plunging oscillations. This dominance is found in numerous flow situations, both in two-dimensional and three-dimensional flows.

Experimental results have been presented that vividly demonstrate the strong coupling existing between vehicle motion and boundary layer transition. In an aeroelastic test at $\alpha = 0$ of a 25° swept wing, with a symmetric airfoil section, violent oscillations in the first bending mode occurred if the location of boundary layer transition was not fixed (Fig. 41). The oscillations were of the limit cycle type, the typical result of nonlinear aerodynamics with negative total (aerodynamics plus structural) damping at oscillations below the limit cycle amplitude. Similar results have been obtained by others, when transition was not fixed.

The plunging airfoil section of the bending wing (Fig. 41) will experience a transition-promoting moving wall effect on the top side during the ‘downstroke’ of the bending oscillation. On the bottom side, the moving wall effect is the opposite, delaying transition. As a result, a negative lift component is generated, which drives the oscillation (see top left inset in Fig. 42). How this coupling of transition to the bending oscillation via the moving wall effect can generate the observed bending oscillation can be described using the experimental results shown in Fig. 42, involving normal blowing through a series of small holes at 5% chord on top and bottom surfaces. It is obvious that when transition occurs aft of mid-chord, the plunging-induced moving wall effect is overmatched by the large, adverse, static pressure gradient. However, just forward of mid-chord the pressure gradients are much milder (see top right inset in Fig. 42). Thus, the moving-wall effect can influence transition sufficiently to generate the observed large bending oscillations only when the blowing has been increased to the range $0.020 < \dot{m}/\dot{m}_b < 0.025$. This apparently is sufficient to move transition on the top side forward of mid-chord, $\xi_T < 0.5$. When the blowing rate is increased further, $\dot{m}/\dot{m}_b > 0.025$, transition moves farther forward, and the moving wall effect is overmatched again, this time by the favorable static pressure
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Fig. 40. Hot film response for plunging and pitching oscillations of the NACA-0012 airfoil. (The amplitudes $\dot{\alpha}/U_\infty$ and $\dot{\alpha}_{\text{ref}}/U_\infty$ are of the same magnitude.) (a) Plunging; (b) pitching.

gradients (see top right inset in Fig. 42). Consequently, the bending oscillations rapidly decrease in amplitude towards the level obtained for fixed transition (compare Figs 41 and 42).

The experimental results in Figs 41 and 42 illustrate a sinister aspect of the Reynolds number-scaling problem. In full-scale flight transition is likely to occur forward of 25% chord, resulting in the insignificant oscillation amplitudes shown in Fig. 42 for $\dot{m}/\dot{m}_0 > 0.03$. In a small-scale facility with limited Reynolds number capability, such as the ones used at most universities, transition could occur aft of mid-chord $\xi_{\text{ref}} > 0.5$. This would also result in modest oscillation amplitudes, typified by the results for $m/m_0 < 0.02$ in Fig. 42. The irony is
that a better testing facility, e.g. such as is used in the aircraft and aerospace industry, due to its higher Reynolds number capability may manage to place transition somewhere between 25 and 50% chord, resulting in the catastrophic type of oscillations shown in Fig. 41 and illustrated for $0.02 < \dot{m}/\dot{m}_b < 0.025$ in Fig. 42. Thus, the high Reynolds number wind tunnel test could indicate a potentially catastrophic problem that never will be encountered in full-scale flight. The only thing worse than this is the reverse, i.e. the test fails to point out a problem that the full-scale vehicle will encounter. That can also happen
through the coupling between vehicle motion and boundary layer transition, as will be illustrated.

Figure 43b illustrates how the plunging-induced moving wall effect generates a negative force component in the case of laminar separation, similar to the case for attached-flow boundary layer transition (Fig. 43a), resulting in wing bending oscillations in both cases. If the Reynolds number is increased, the situation sketched in Fig. 43c may result. In this case, the moving wall effect promotes transition, thereby delaying leeside flow separation, as was demonstrated by Carta's results (61) (Fig. 40). Thus, in this case the combined effect of transition and flow separation is to generate damping for the plunging oscillations. If the Reynolds number is increased further, until transition moves far forward of the flow separation, the situation becomes similar to that illustrated in Fig. 44b for laminar separation.

The implication of the possible flow situations sketched in Fig. 43 is that in subscale tests one can obtain dangerously misleading results. For example, in full-scale flight the flow separation will usually be turbulent, and the coupling with the bending wing amplifies the

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Fig. 43. Dynamic transition and flow separation characteristics on a symmetric airfoil: (a) transition; (b) laminar separation; (c) transition and separation.
separation-induced negative damping, resulting in an increase of the oscillation amplitude. If the subscale test is performed at laminar flow conditions, the wing bending response will be similar, at least qualitatively, to the full-scale response, exhibiting divergent bending oscillations until the limit cycle amplitude is reached. At that amplitude the attached-flow lift-damping, together with any significant structural damping, balances the separation-induced negative damping. It is not unlikely, however, that in his desire to get as close as
possible to the full-scale Reynolds number range, the test engineer ends up with the situation sketched in Fig. 43c, in which case he may never become aware of the fact that the full-scale wing could experience possibly unacceptable bending oscillations.

A solution to the problem illustrated above would appear to be to fix transition near the leading edge, where it is expected to occur in full scale. That this is no panacea is demonstrated by the results\(^{(88)}\) in Fig. 44. In the case of bending oscillations, the trip may increase or reduce the driving lift-potential by moving the shock-induced separation forward. For small amplitude perturbations, the pressure distribution in Fig. 44 indicates that the shock-induced pressure jump would be decreased, producing unconservative dynamic results.

5.2.1. Ground Facility Interference

The presence of side plates or wind tunnel walls in so called two-dimensional dynamic stall tests has always been of some concern as regards possible interference effects on the separated flow region.\(^{(89-91)}\) The large effect of compressibility on dynamic stall discussed earlier (Fig. 37) makes it difficult to obtain a true measurement of the two-dimensional wall interference.\(^{(92,93)}\) It cannot be emphasized enough that incompressible dynamic stall will rarely be of practical importance. The 'incompressible' limit \(M_{p} = 0.4\) is exceeded already at \(M_\infty \geq 0.1\) on a stalling airfoil\(^{(94,95)}\) (Fig. 45). Consequently, there is a need for tests in ground facilities in which the Reynolds number can be varied while keeping the subsonic Mach number constant, and vice versa. Even if only static testing would be possible in such facilities,\(^{(96)}\) valuable information could be obtained.

5.3. SCALING PROBLEMS IN 3-D FLOW

Figures 14–19 illustrate the scaling dilemma one faces when planning high-alpha wind tunnel tests of advanced aircraft. In one case, when transitional crossflow separation drives the wing rock\(^{(33,34)}\) (Fig. 14), laminar tests will not produce any wing rock at all, whereas

![Graph](image)

Fig. 45. Peak local Mach number just before stall, \(\alpha = 15^\circ\), \(\alpha = 10^\circ\), \(\omega = 0.20\).
Fig. 46. Effect of dummy sting on experimental lateral characteristics of the HIRM 2 model at $M = 0.2$ and $Re = 0.9 \times 10^5$. (a) $\alpha = 40^\circ$; (b) $\alpha = 50^\circ$; (c) $\alpha = 60^\circ$. 
the full-scale vehicle will experience wing rock, e.g. as in the case of the X-29A aircraft. In another case, the laminar test will show wing rock very similar to that experienced by the full-scale vehicle, e.g. as in the case of the F-18 HARV aircraft (Fig. 18). Consequently, tests in an industrial facility at fairly high Reynolds numbers would give a conservative estimate of the wing rock to be experienced in full-scale flight for the case illustrated in Fig. 14, whereas such tests would be completely misleading in the other case, illustrated in Fig. 18, giving no warning at all about the wing rock to be experienced by the full-scale vehicle, as in the case of the F-18 HARV aircraft.

5.3.1. Ground Facility Interference

The problem of aerodynamic ground facility interference increases both in severity and complexity when performing tests at high angles of attack, where body and wing vortices can interact with the support structure. The support becomes massive in the case of tests with a rotary rig, because of the high stiffness needed in order to avoid excessive deflections and vibrations in the presence of the large, unbalanced, rotating, inertial forces generated by the coning motion.

Recent rotary tests of the HIRM 2 model (Fig. 46) show that, when using a top-mounted dummy strut in addition to the aft sting support, in the case of $\alpha = 50^\circ$, for example, the effect of the dummy strut was insignificant, whereas the results for $\alpha = 40^\circ$ and $\alpha = 60^\circ$ not only show an effect of the dummy strut, but through the different effects on roll and yaw characteristics illustrate the complexity of the support interference. Similar test results have been obtained at AerMacchi. A true evaluation of the support interference is possible through the use of the orbital-platform testing concept.

The main cause of ground facility interference on high-alpha measurements is the interaction of the vortices shed from the model with the downstream wall-mounted support structure, as in the case illustrated in Fig. 46. The interaction can vary all the way from a minor deflection of the downstream path of the vortices to a complete change of the vortex structure through support-induced vortex breakdown. In the latter case, the vortex-induced loads on the model can be changed drastically, as is demonstrated by Hummel's classic experiment (Fig. 47). Adding to the complexity of the high-alpha support interference is the frequent coupling with wind tunnel wall interference.
5.4. SUMMARY

(1) Compressibility effects are large in the case of two-dimensional flow, making incompressible flow concepts inapplicable in most cases of airfoil stall. In three-dimensional flow compressibility effects are usually less severe, except for at very large angles of attack where the body crossflow will experience compressibility effects almost as severe as in the case of airfoil stall.

(2) Dynamic simulation in subscale tests requires simulation of the full-scale Reynolds number, a task made very difficult because of the large compressibility effects discussed above.

(3) Support and wall interference can have significant influence on the high-alpha experimental results. The correction for such interference effects is complicated by the coupling with Reynolds number and moving wall effects. Thus, the support interference has to be investigated for the full range of flow conditions, from subscale to full-scale Reynolds numbers, before confident extrapolation from subscale high-alpha data (such as rotary-balance data) to full-scale flight will be possible. International cooperation, started 1986 under the auspices of AGARD and presently continuing through AGARD FDP Working Group 16, will hopefully produce the understanding needed to minimize the effect of ground facility interference on the high-alpha data accuracy.

6. CONTROL/VEHICLE DYNAMICS COUPLING

When the lateral loads induced by asymmetric forebody flow separation exceeded the available control capability, the initial effort was to try to eliminate or greatly reduce the flow asymmetry. Presently, however, the more ambitious goal is to be able to control the asymmetry thereby providing the desired supermaneuverability. The main effort has, so far, almost exclusively been focused on the achievable control capability, whereas the mutual interaction between control application and vehicle dynamics has received very little attention. One obvious problem is to account for the large effect of roll angle on the control effectiveness, as illustrated by recent experimental results (Fig. 48). On the maneuvering aircraft the crossflow stagnation point will continually change its position relative to the controls. In extreme cases, as for the dual, rotatable

![Fig. 48. Effect of dual, rotatable tip strakes on the yawing moment of an F/A-18 model.](image-url)
tip-strakes in Fig. 48, a 10 deg rotation of the crossflow stagnation point would at \( \alpha = 55 \) deg change the yawing moment from approximately 0.06 at \( \phi = 0 \) to \( -0.06 \) at \( \phi = 10 \) deg. Of course, the vehicle response to this 10 deg rotation of the tip strakes will be associated with convective flow time lag effects. In addition, moving wall effects generated by the control-induced vehicle motion have to be considered.

One example of control/vehicle dynamics coupling is provided by the F-14 flight data\(^{32}\) discussed earlier, which indicated that the aircraft entered flat spin as a result of a control-induced rapid roll around the velocity vector. Another example of such control/vehicle dynamics interaction is provided by the investigation reported in Ref. 108. A model of a very slender aircraft or space plane was tested in a water tunnel. The forebody vortex system interacted with the leading-edge vortices from the delta wing to produce wing rock of the type described earlier in connection with Fig. 17. The resulting limit cycle amplitude is shown in Fig. 49 as a function of angle of attack. The separation asymmetry, and thereby the vortex asymmetry, could be controlled by blowing jets near the apex. The test showed steady blowing to be able to eliminate the wing rock by generating a steady, nonzero roll trim. By using alternating left and right side blowing at a sufficiently high frequency the time-averaged effect of the flow asymmetry could be made negligibly small, eliminating both the wing rock and the nonzero roll-trim. At lower frequencies, however, the time-averaged effect was not zero, illustrating the potential for undesirable coupling between control and vehicle dynamics.

The forebody-generated vortex system is complicated because of the presence of large moving wall effects. However, interaction between vortices generated by wing leading edges can also provide strong control/vehicle dynamics coupling. The interaction between a LEX vortex and the leading-edge vortex from the main wing can have adverse dynamic effects, as was demonstrated in the case of the wing buffet problem of a variable sweep aircraft.\(^{109,110}\)

Recent test results\(^{111}\) show that when the breakdown of a leading-edge vortex reaches the apex of the windward, 'dipping' side of a rolling delta wing, something akin to dynamic airfoil stall\(^{66}\) occurs. The suction generated by the leading-edge vortex downstream of

![Figure 49](https://example.com/figure49.png)

**Fig. 49.** Peak-to-peak wing-rock amplitude of a slender delta wing-body model.\(^{108}\) (NT denotes no tail.)
a spiral type vortex breakdown is wiped out ($\alpha = 39.1^\circ$ in Fig. 50). To quote the authors of Ref. 111: “For $\alpha = 39.1^\circ$ the pressure distribution on the upper surface of the wing becomes more or less constant in spanwise and chordwise direction.” That is, the ‘spiral leading-edge vortex’ is ‘spilled’ downstream, much as in the case of dynamic airfoil stall.\textsuperscript{(75)} In the presence of a fuselage, this ‘spilled leading-edge vortex’ phenomenon can for non-zero effective angle of sideslip occur on one side of the delta wing at a time. This generates the potential for strong coupling between control application and vehicle dynamics.

This sampling of vortex/vortex interactions demonstrates the need for improved understanding of the control/vehicle dynamics coupling before safe use can be made of ‘supercontrols’ to achieve ‘supermaneuverability’.

7. CONCLUSIONS

Our present knowledge of the critical issues in high-alpha vehicle dynamics can be summarized as follows:

1. \textit{Cause and effect of asymmetric forebody flow separation with associated vortices:} Asymmetric flow separation on the slender nose of an aircraft, maneuvering at high angles of attack, is controlled by the moving wall effects generated by the vehicle motion, which have a dominating influence on the boundary layer development between cross-flow stagnation and separation points. This coupling between vehicle motion and flow separation is the cause of experimentally-observed forebody-induced nose-slice and coning motions, as well as of the forebody-induced wing rock phenomenon.

2. \textit{Case of slender wing rock:} Wing rock of slender delta wings, $\alpha \leq 80^\circ$, is initiated by the large undamping generated by attached leading-edge vortices. Thus, asymmetric lift-off of the leading edge vortices on slender delta wings does not start the wing rock observed in experiments. For very slender delta wings, $\alpha = 80$ and $85^\circ$, it is, however, a flow mechanism contributing substantially to the observed very large limit-cycle amplitudes. The magnitude of the wing rock amplitude is limited by breakdown of the leading-edge vortex on the ‘dipping’, windward side of the delta wing.
3 Effect of vehicle motion on dynamic airfoil stall: The observed large difference for $\xi_{CG} > 0$ in dynamic stall characteristics between pitching and plunging airfoil oscillations is caused by the opposite moving wall effects. The moving wall effect is extremely large when it is acting near the flow stagnation point, in which case it completely dominates over accelerated/decelerated flow effects.

4 Extrapolation from subscale tests to full-scale free flight: A strong coupling exists between boundary layer transition and vehicle motion, which can have a significant effect on the vehicle dynamics. When the vehicle motion affects flow separation via boundary layer transition, the penalty for testing at subscale Reynolds number increases, and the subscale test can in some cases show viscous flow effects opposite to those experienced in full-scale flight. The dynamic scaling problem is complicated by the often large effects of compressibility. The problem of ground facility interference is aggravated in tests of aircraft at high angles of attack because of the capability of the model support not only to deflect the vortex wake but also, in many cases, to cause premature vortex breakdown.

The immediate challenge is to expand our present understanding of the high-alpha flight dynamics to include the coupling existing between control application and vehicle dynamics in order to assure safe use of 'supercontrols' to achieve 'supermaneuverability'.

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