equation of velocity reduces to

$$V_{\max} \cong I_s \ln(1/\lambda) \quad (9)$$

Equation (9) is identical, except for the "approximately equal" notation, to the well-known formula that gives the ideal velocity attained by a conventional rocket in a gravitationless vacuum,† the derivation of which assumes that the exhaust velocity and mass flow rate remain constant. Both of these items, however, are varying in the case of the cold-gas thruster.

### Accuracy of Approximation

By comparing Eqs. (1) and (5), it may be seen that the terminal velocity, according to the approximate formula, is slightly overestimated. The maximum percentage error occurs with thrusters using the monatomic gases ($k=3$ or $\gamma=1.667$), and amounts to about 5% in the extreme case of $\lambda=½$. As indicated in Fig. 1, the difference between the two equations decreases as $k$ or $\lambda$ grows large, such that the error in any situation does not exceed 1% for propellant mass fractions ranging up to 10%.

†See, for example, Ref. 5.

It may be concluded that for the class of cold-gas thrusters containing small propellant mass, the approximate velocity equation closely represents the integral equation of ideal velocity. The abbreviated formula permits the effect of various design factors to be readily determined.

### Acknowledgment

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# Technical Comments

## Comment on "A Re-Entry Control Effectiveness Methodology for the Space Shuttle Orbiter"

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IKAWA, in his strip-analysis of the control characteristics of the Space Shuttle Orbiter,[1] draws conclusions that easily can be interpreted to mean that there are no significant three-dimensional flow effects in the separated flow region generated by windward side control deflection. This is a gross misinterpretation of the experimental results.

Even when the inviscid flow is axisymmetric, large-scale three-dimensional flow characteristics are observed in the separated flow region caused by a compression corner.[2] Consequently, the "nearly two-dimensional" attached flow characteristics existing on the windward side of the Orbiter, deduced from the flow picture in Fig. 2 of Ref. 1, and also demonstrated by the theoretical and experimental investigations by Adams et al.,[3] does not imply that the control-induced separated flow region also will be "nearly two-dimensional" in character. Ikawa admits as much but concludes that because the control hinge line is unswept in his case the gross effects on the Space Shuttle Orbiter control surface effectiveness can be analyzed by his strip-theory approach with reasonable accuracy "even though the microscopic flow field detail may be highly three-dimensional." If the three-dimensional effects truely were microscopic, this would be true. However, Whitehead and Keyes[4] tested a control with straight unswept hinge line on a delta wing and found the separated flow region to be highly three-dimensional, containing large-scale vortices. These separated flow characteristics were obtained with natural boundary-layer transition occurring well inboard of the leading edge. When roughness was applied near the leading edge, causing the transition to move closer to the leading edge, a more regular separated flow region was obtained. Even in the more regular separated flow region obtained with roughness, the effects of spanwise flow were found to be large. The effect of this spanwise flow is to shrink the separated flow extent[4] and to decrease the pressure in the separated flow region.[5] The results presented by Marvin et al.[6] show such three-dimensional flow effects even for control-induced turbulent flow separation.‡

Ikawa[1] acknowledges that spanwise flow tends to reduce the separated flow region. This "shrinking" of the separated flow region due to spanwise flow will by itself, without considering any pressure changes, change the separation-induced loads. The sketch in Fig. 1 illustrates how the decrease of the extent of the pressure plateau causes the separation-induced normal force to lose the forward component $\Delta C_{N1}$. The earlier reattachment on the flap adds the aft normal force component $\Delta C_{N2}$. For a certain control geometry it is possible that the two force components could cancel each other, i.e., $\Delta C_{N2}-\Delta C_{N1}=0$. However, in this case the pitching moment would be decreased by the stabilizing component $\Delta C_m = -\Delta\xi \cdot \Delta C_N$, where $\Delta C_N = \Delta C_{N1} = \Delta C_{N2}$. Ikawa's data display precisely these characteristics (Fig. 8 of Ref. 1). That is, the measured control-induced normal force is essentially in agreement with prediction, whereas the measured control-induced pitching moment is underpredicted.§ Thus, a comparison between strip-theory and experiment based upon gross normal force

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‡The 22 deg swept hinge line may have contributed somewhat to the three-dimensionality in their case.

§This is clearly the case for $\alpha=40$ deg, whereas the experimental data scatter extends across predictions for $\alpha=30$ and $\alpha=20$ deg.

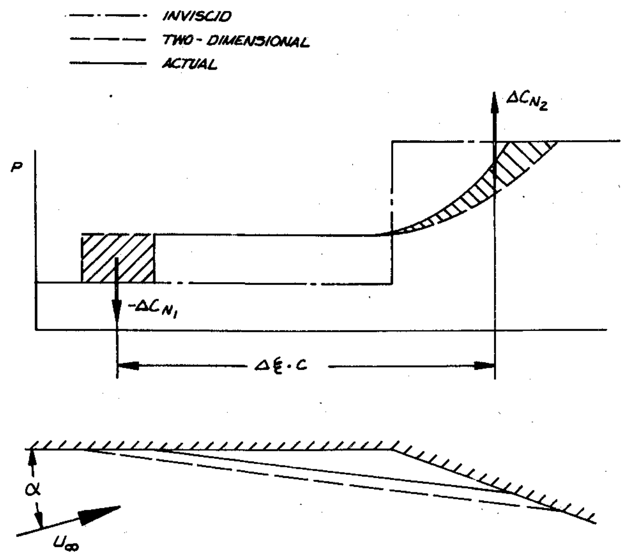


**Fig. 1 Sketch of spanwise flow effects on control-induced flow separation.**

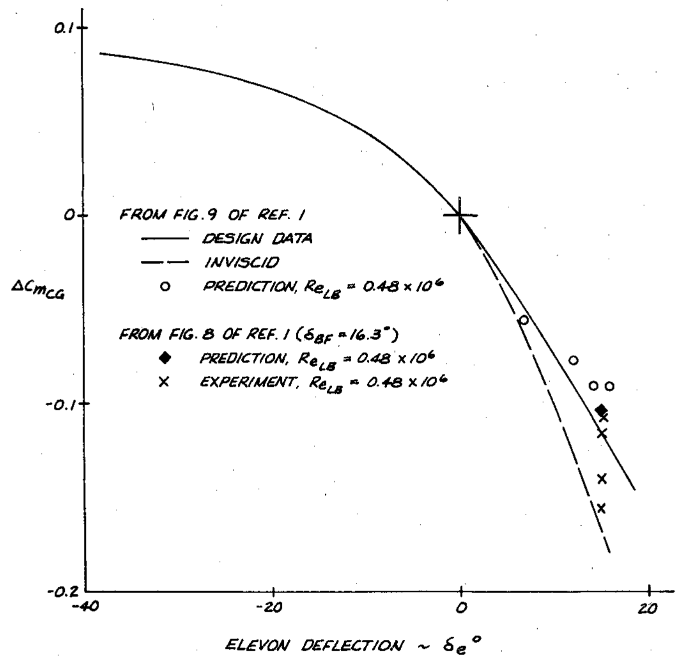


**Fig. 2 Control effectiveness at $M_\infty = 12.2$ and $\alpha = 40$ deg.[1]**

and moment characteristics is generally inconclusive. If anything, the comparison in Fig. 8 of Ref. 1 shows the presence of spanwise flow effects. Although we agree with Ikawa that it is unfortunate that experimental pressure data were not available for a more conclusive comparison, we cannot understand why a comparison was not made between predicted separation geometry and experimental flow visualization results.

Ikawa correctly emphasizes the need for analytic means whereby one can extrapolate from wind-tunnel test data to full-scale flight conditions. However, the strip-theory of Ref. 1 can serve as such a tool only when spanwise flow effects are negligible, which in our estimation is the case only when the effect of flow separation is itself negligible. It is clear that this situation may be approached for the highest angle of attack, $\alpha = 40$ deg. In Fig. 2 the results for $\alpha = 40$ deg from Fig. 9 of Ref. 1 have been replotted. Also shown are the data from Fig. 8 of Ref. 1 for $\alpha = 40$ deg, $\delta_e = 15$ deg and $\delta_{BF} = 16.3$ deg. The predicted effect $\Delta Cm_{BF} = -0.015$ in Fig. 2 is very small. This is, however, in basic agreement with the body-flap-alone effect ($\Delta Cm_{BF} = -0.030$ from Fig. 9 of Ref. 1) when one considers the effect of the elevon-induced flow separation, extending inboard forward of the flap. In any case the experimental data scatter should remain the same regardless of body flap deflection. Consequently, Fig. 2 shows that the effect of flow separation is to reduce the control effectiveness by 6-40% according to experimental data and by 40-50% according to Ikawa's strip-theory prediction. Thus, it appears that spanwise flow effects will be important even when the true three-dimensional separated flow effect is small. This implies that there are no flight conditions of practical interest for which realistic "prediction of re-entry aerodynamic trim and control for a winged vehicle, such as the Space Shuttle Orbiter, can be made successfully with application of strip analysis of the vehicle planform, using the presently developed hypersonic laminar flow separation computer program," the claim made by Ikawa in his conclusions. Three-dimensional flow effects will always be significant!

Thus, even for the relatively simple case where the flow over the full-scale vehicle is completely laminar, one will encounter problems when trying to develop reliable means for extrapolation of subscale wind-tunnel data to full-scale flight. When boundary-layer transition occurs on the full-scale vehicle, the problem is compounded.[7,8] It is especially difficult in regard to dynamic test data because the coupling existing in full-scale flight between boundary-layer transition and vehicle motion prohibits one from using boundary-layer tripping devices in subscale dynamic tests.[7,8] When one considers the effects of leeside control deflections, e.g., the region $-40 \leq \delta_e \leq 0$ deg in Fig. 9 of Ref. 1, the separated flow patterns become a great deal more complicated and the scaling problems, as a consequence, are much more difficult. The control deflection can, on slender delta wings, cause breakdown of the leading-edge vortex, with associated large discontinuous load changes.[9,10] In the case of the Space Shuttle Orbiter, the OMS pods complicate the flow situation further. For certain flight conditions, drastic changes in the separated flow pattern occur which generate highly nonlinear, often discontinuous, control characteristics which often are associated with angle of attack and control deflection hysteresis effects.[11]

There is little doubt that much valuable information has been and can still be obtained through theoretical work done under the assumption of axisymmetric or two-dimensional separated flow. However, it is by now a well-established fact that all separated flow is three-dimensional in charcter. It is clearly a disservice to the vehicle designer to assert that strip-theory can predict separated flow effects on control effectiveness, as is done in Ref. 1. This can lead to very embarassing surprises late in the design period when changes in the vehicle design tend to be very expensive.

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# Reply by Author to L.E. Ericsson and J.P. Reding

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ERICSSON and Reding[1] commented that application of the quasi-three-dimensional methodology (QTDM) to predict the control surface effectiveness[2] without considering three-dimensional effects is a disservice to vehicle designers. It appeared that Ericsson and Reding drew these conclusions without careful examination of the paper[2] and the comparison data. The author was well aware of the importance of the three-dimensional separated flow phenomena generated by surface deflections prior to undertaking this task. However, sound engineering judgement was exercised to determine that a quasi-three-dimensional approach is valid for the Space Shuttle Orbiter (SSO) control surface effectiveness analysis when the flap/elevon is deflected into the windward flowstream.

The paper[2] did not claim that the QTDM is universally applicable to any re-entry vehicle control surface analysis. On the contrary, the application of the strip theory is limited to "hypersonic winged vehicles" as described in the paper. Furthermore, it stated that "the effects of flow spillage and cross flow components normally associated with three-dimensional viscous interaction problems are *minimized* (did not imply nonexistent) by the deflected surfaces having *unswept hinge lines*." The flow spillage due to flap end effects is minimized when the aspect ratio of elevon/flap is large. The application of this QTDM subsequently showed that the effects of the three-dimensional flow phenomena can be approximately simulated for the SSO configuration.

Based on the aforementioned assumptions, the conclusion drawn by Ericsson and Reding based on application of the present methodology to axisymmetric data[3] (cylinder-frustum configuration of Apollo/Saturn) and the older Shuttle vehicle having control surfaces with a highly swept-back hinge line[4] are unjustified. In these configurations, strong three-dimensional flow exists and the contribution arising from the crossflow component of the interacting flowfield cannot be neglected as stated by Ericsson and Reding.

The current SSO configuration consists of a flat wing lower surface, and large-aspect-ratio elevon and flap (4.5 and 2.2, respectively) with unswept hinge lines. Therefore, three-dimensional flow separation effects are minimized, meeting the aforementioned criteria.

Ericsson and Reding also discounted the effectiveness of the QTDM by quoting the experimental results of delta wings with unswept hinge lines obtained by Whitehead and Keyes. Laminar-turbulent transitional flow (mixed)[5] separation data were used to construct their argument. However, Ericsson and Reding must be well aware of the fact that the mixed separated flow phenomena cannot be predicted by either two- or three-dimensional theory unless proper turbulent transitional criteria are incorporated. For fully developed turbulent flow separation, Whitehead and Keyes stated: "For the spanwise stations on the flap of 70 degrees swept wing data, the centerline pressure is lowest and the pressure increases as the distance from the centerline increases. This behavior is just the reverse of that shown for the pressure distribution on the flap of the mixed flow separation case." The predicted spanwise pressure distribution for the reattachment region of laminar flow separation on the SSO elevon follows precisely the same qualitative three-dimensional trend observed in the turbulent experiment[5] (see Fig. 7 and discussion of Spanwise Distribution of Peak and Plateau Pressures, Ref. 2). This is to be expected since the observed phenomena in which separation length increases with Reynolds number follows the same trend for both laminar and turbulent flow separations.[5] Therefore, this observation gives credence to the QTDM for predicting the SSO control surface effectiveness, provided the viscous interaction does not occur in the laminar turbulent transitional regime and the criteria for the applicability of the present QTDM are not violated.

It should be noted that the criticism is centered on the fact that the four test data points shown on the $\alpha = 40$ deg data (Fig. 8 of Ref. 2) did not agree with the prediction. This apparent discrepancy was allegedly credited to the flow spillage at the wingtip, where in fact the observed disagreement is due to wall cooling effects. The predicted values were omitted in Fig. 8 for clarity, as discussed later. As a result of this misunderstanding, the overall merit of the QTDM is discredited by Ericsson and Reding.

Although Ericsson and Reding have qualitatively interpreted the correctness of the separated flow phenomena, they are unable to predict the exact separation and reattachment points of their arguments. Ericsson and Reding failed to recognize that the separation-reattachment points and pressure recovery are influenced by many factors other than the three-dimensional nature of separated flows, e.g., Reynolds number, Mach number, finite flap chord length, wall cooling, etc. The rigorous arguments which led to the results of Fig. 1 of Ref. 1 is simply demonstrated in Figs. 11 and 12 of Ref. 2 for the different reasons.

Due to the page limitation imposed on the paper a detailed discussion of the experimental data (Fig. 8, Ref. 2) was omitted, leading to Ericsson and Reding's misinterpretation of the results. The experimental data shown in Fig. 8 of Ref. 2 were obtained from the two experimental sources. The data points shown in open triangle and square symbols were obtained from AEDC wind tunnel tests, in which the model was heated to produce an adiabatic wall test condition ($S_w = 0$). Hence, the agreement of test and predicted data made with $S_w = 0$ is good. The open circle data points were gathered from the short-duration test conducted at CALSPAN which produced nonadiabatic wall test conditions ($S_w = -.8$). These short-duration tests were subjected to model vibration, which may account for the data scatter. The separated zone is reduced by wall cooling and control becomes more effective as compared with the adiabatic case shown in Fig. 12 of Ref. 2. Figure 2 shown by Ericsson and Reding[1] is a reproduction of the $\alpha = 40$ deg data, Fig. 9 of Ref. 2 with four data points taken from Fig. 8 of Ref. 2. However, Ericsson and Reding failed to include the predicted nonadiabatic data points ($S_w = -.8$ data also shown in black circle symbols which were

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